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FINAL REPORT

SPACE STATION ONBOARD PROPULSION SYSTEM

TECHNOLOGY STUDY

CONTRACT NO. NASA3-23893

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FOREWORD

This report was prepared by Martin Marietta Denver Aerospace, under contract NAS3-23893. The contract was administered by the Lewis Research Center of the National Aeronautics and Space Administration, Cleveland, Ohio, and the Marshall Space Flight Center of the National Aeronautics and Space Administration, Huntsville, Alabama. Mr. G. Paul Richter and Mr. Lee Jones were the NASA Project Managers. The contract technical period of performance was from 10 January 1985 to 10 January 1987.

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SUMMARY

As part of NASA's development of the Space Station, it is necessary to examine and evaluate various propulsion systems for the Station. The initial objective of this study was to define a space station onboard propulsion system that uses hydrogen and oxygen as propellants. The defined system was to be used in guiding the various component technology programs associated with the space station. The program also included a study of the evolvability of the propulsion system from nearer term to more far term systems. The specific propulsion system initially examined incorporated both high and low thrust systems to perform the onboard propulsion functions for the space station in low earth orbit. Resistojets using gaseous hydrogen as the propellant were used to provide the low thrust. High thrust was provided by chemical rockets using gaseous hydrogen and gaseous oxygen for propellants. The propulsion systems studied were limited to storing the propellants as cryogenic supercritical fluids.

The initial tasks of the program were to define the propulsion requirements and candidate systems that would meet these requirements. The candidates were then evaluated and the system that would best meet the requirements for the space station was selected. This selection was based upon consideration of technology readiness; design, development, test, and evaluation costs; life cycle costs; maintainability, reliability, and evolvability. A concept selection plan was developed and used to perform this evaluation. When the system design had been selected, an interim program review was held to review the results of these first two tasks.

Upon approval of the selection by NASA, it had been planned to develop a preliminary system level design to determine operating conditions and system level component requirements. Finally, a computer simulation of the selected onboard propulsion system was to be developed. This computer program was to be capable of simulating design point, planned emergency operation and off-design operations of the propulsion system. The program was to be developed in a "breadboard" fashion to allow easy modification of the program.

However, at this point, the basic contract was modified. The contract was transferred from Lewis Research Center to George C. Marshall Space Flight Center and a modified Statement of Work issued. The program was completed under the new statement of work with the following tasks being accomplished.

The evaluation of propulsion systems was widened to include other elements of the space station including the OMV, OTV, and the platforms. Also, the propellants considered for all of the propulsion systems was broadened to include storable mono and bipropellants and water elecrolysis for the supply of gaseous hydrogen and oxygen propellants. Propulsion requirements for these elements were identified, candidate systems defined and the most attractive concepts for each element were selected. A primary consideration in this selection was the benefit of commonality between systems.

The results of the evaluation of other propellants indicated that the cryogenic supercritical storage system for the space station was not competitive with the electrolysis hydrogen/oxygen system. Therefore, the work effort to develop a preliminary design of the cryogenic supercritical storage system for the space station was eliminated. Also, the computer simulation was redirected to provide a computer simulation of the water electrolysis system. This simulation was developed in such a manner that it can be easily modified to simulate other propellant systems as well.

The plan for conducting the study consists of four tasks: Task I - Propulsion system Concepts, Task II - Propulsion Systems Concept Selection, Task III - Preliminary Design, and Task IV - Computer Simulation. The original Propulsion System Concepts included nearly 3000 configurations. The total number was reduced to approximity 34 before entering into Task II.

I. INTRODUCTION

As the plans for the Space Station progressed, it became necessary to identify technologies that will be required for the successful completion of the space station mission. One of these areas requiring definition of technologies is the space station onboard propulsion system. Of particular interest is the area of propulsion systems that use hydrogen and oxygen as propellant. The objective of this program was to define oxygen/hydrogen space station propulsion systems in sufficient depth to be used in guiding various component technology programs. In addition, a major effort of this program was to develop a computer simulation of the onboard propulsion system. At the midpoint of the program the Statement of Work was revised to include the evaluation of other space station element propulsion systems. The effort was divided into four technical tasks with the following objectives.

TASK I - PROPULSION SYSTEM CONCEPTS

Task I - Propulsion System Concepts were defined and schematics were developed for the most viable concepts. In conformance with a revised statement of work, the study was not limited to the space station onboard propulsion system, but was expanded to include other elements of the space station architecture including the OTV, OMV, and platforms. The basic propulsion requirements of each element were identified. Concepts that could meet these requirements were then identified. Simple steady state models of the propulsion systems were developed in order to evaluate the power and logistic requirements of the concepts. An initial screening, based on engineering judgment and rudimentary costing, was then carried out to narrow the field to the most attractive candidates.

TASK II- PROPULSION SYSTEMS CONCEPT SELECTION

A concept selection plan was developed for comparing the concepts identified in Task I and selecting a single candidate to be further described in Task III, Preliminary Design. The Plan, which was reviewed and approved by NASA, included selection criteria of technology readiness, DDT&E, costs, life cycle costs, maintainability, reliability, and evolvability. Upon approval of the selection plan the candidate concepts were reviewed and the most attractive concept selected.

TASK III - PRELIMINARY DESIGN

This task originally was to have been the development of a preliminary design of the selected space station propulsion system. However, it was deleted by the revised Statement of Work.

TASK IV - COMPUTER SIMULATION

A computer simulation was developed for the onboard propulsion system. Although this model was originally intended to simulate a water electrolysis system it was expanded to give it more versatility. The model can simulate several types of propulsion systems including cold gas, monopropellant and bipropellant storable systems. It is also written in "breadboard fashion" so that subroutines can be easily modified. A user friendly front end was provided to allow trade studies of propulsion systems to be made in an expeditious manner. A complete users manual was also provided as part of this task.

II. TASK I - PROPULSION SYSTEM CONCEPTS

Task I had several objectives. The first objectives of Task I was to identify space station propulsion systems concepts and define schematics for those concepts. To accomplish this it was necessary to define the propulsion requirements for the space station. Steady state models were developed of the concepts as they were identified. These models allowed power balances and logistics requirements of the concepts to be determined. Finally, at the midpoint of the program an effort was added to define the propulsion requirements for space station elements including platforms, OMV and OTV. Propulsion concepts that met these requirements for each element were then identified, screened, and the better concepts selected for each element.

A. PROPULSION REQUIREMENTS

The propulsion system performance requirements derived during Task I and refined during the period of performance of Tasks I & II are summarized in Table II-1. The requirements are based primarily upon two sources; 1) the JSC Reference Requirements Document JSC 19989 published prior to the Phase B competition, and 2) the internal MMC preproposal and Phase B effort. The 90 day total impulse requirement was found to vary from 70,000 lb_f-sec to 700,000 lb_f-sec over the 11 year solar cycle. An attitude control impulse requirement of 50,000 lb_f-sec was identified to cover shuttle docking disturbances. The Space Station reference Attitude Control System (ACS), consisting of momentum storage and magnetic torque devices, has primary responsibility for attitude control with propulsion as the backup system. Approximately 100,000 lb_f-sec per 90 day period will be required if purely coupled torques are used to back up an ACS failure. Propellant consumption can be minimized by using single thrusters in a noncoupled mode. This results in both a torque and reboost translation to the Space Station when -x thrusters are used for either pitch or yaw torques. This propellant would come from the reboost propellant budget requiring that no additional propellant be carried for attitude control. This could change if future analysis or requirements disallow the use of noncoupled pair torques for attitude correction. An impulse requirement for an additional 500,000 to 1,000,000 lb_f-sec is specified in the reference document to cover an altitude change of 20 nautical miles for either emergency or customer

Table II-1 - Propulsion Requirements Summary

Requirements	Value	Source
SPACE STATION PROGRAM-DISCRIMINATORS		
Total Impulse/90 days - OA	.07-.7 x10^6 lb_f-sec	Range for 90 days (From MMC Ø B)
Total Impulse/90 days - RCS	25,000 lb_f-sec	RCS for docking & disturbances
RCS during OA	0 lb_f-sec	Use off-pulsed +X thrusters
90 day CMG Contingency	15-75 x10^3 lb_f-sec	Range from MMC ØB proposal effort
20 nM Contingency	.5-1x10^5 lb_f-sec	From MMC ØB effort
Thrust Range - OA	0.5-100 lb_f total	Nominal range set by control rqmts
Thrust Range - RCS	25 lb_f each	Nom. figure set by control rqmts
Emergency ΔV	5 fps ΔV (w/ 20nM)	From NASA Ref Rqmts Document
g-Level Limits (axial&rot)	<= 10^{-5} g's	From NASA Ref Rqmts Document
Electric Power Avail.		
-Maximum Steady State	2 Kwe	From MMC Ø B proposal effort
-Maximum Transient	50 Kwe for 4 hrs	From MMC Ø B proposal effort
-Duty Cycle	0-100%	From MMC Ø B proposal effort
Thermal Power Avail.		
-Maximum Steady State	20 Kwt	From MMC Ø B Proposal
-Max/Min Transient	45 Kwt for 4 hrs	From MMC Ø B Proposal
-Duty Cycle	0-100%	From MMC Ø B Proposal

accommodation reasons. This requirement does not appear to be strictly necessary. Advanced planning and proper STS resupply manifesting can greatly reduce the impact of meeting this requirement. Finally, the reference document specified that the propulsion system provide capability for a 5 fps collision avoidance delta V. As with the RCS requirements, this does not require additional propellant, merely early usage of reboost propellant.

Thruster location, number and thrust levels were not strictly system drivers but are included as part of the system design. The 36 thrusters of the reference system were reduced to 12 because the increased life available with GO_2/GH_2 thrusters reduces the need to backup failed or degraded thrusters. A thrust level of approximately 25 lb_f is adequate for attitude control. Additionally, a reboost thrust level of 100 lb_f resulting from using four thrusters is within the dynamic response tolerances of Space Station. For these reasons the reference document's thrust level specifications were used.

Cryogenic O_2/H_2 systems all require some thermal heat input to condition propellants from the storage conditions up to those required by the thrusters. This thermal power can become quite large if the propellants are conditioned in real time for a high total thrust level. The supercritical system of this study uses this power for two purposes, one to expel the propellant from the supercritical storage tanks, and the other to condition the propellants prior to storing them in an accumulator. The combined sum of this power varies with time as the propellants are conditioned at a steady mass flow rate. The total available power from the Space Station thermal and electrical subsystems is shown in Figure II-1 plotted with the initial power required to expel and condition the propellants at various thrust levels. As can be seen, for a 100 lb_f burn, the total power required exceeds the available power. For this reason, the propellant expulsion and conditioning operations must be separated from the thrusters by an accumulator.

B. Concept Definitions

SPACE STATION PROPULSION

An iterative process was used to identify all viable Space Station Propulsion system options and select those that most closely matched the requirements for this study. The process was iterative for two reasons.

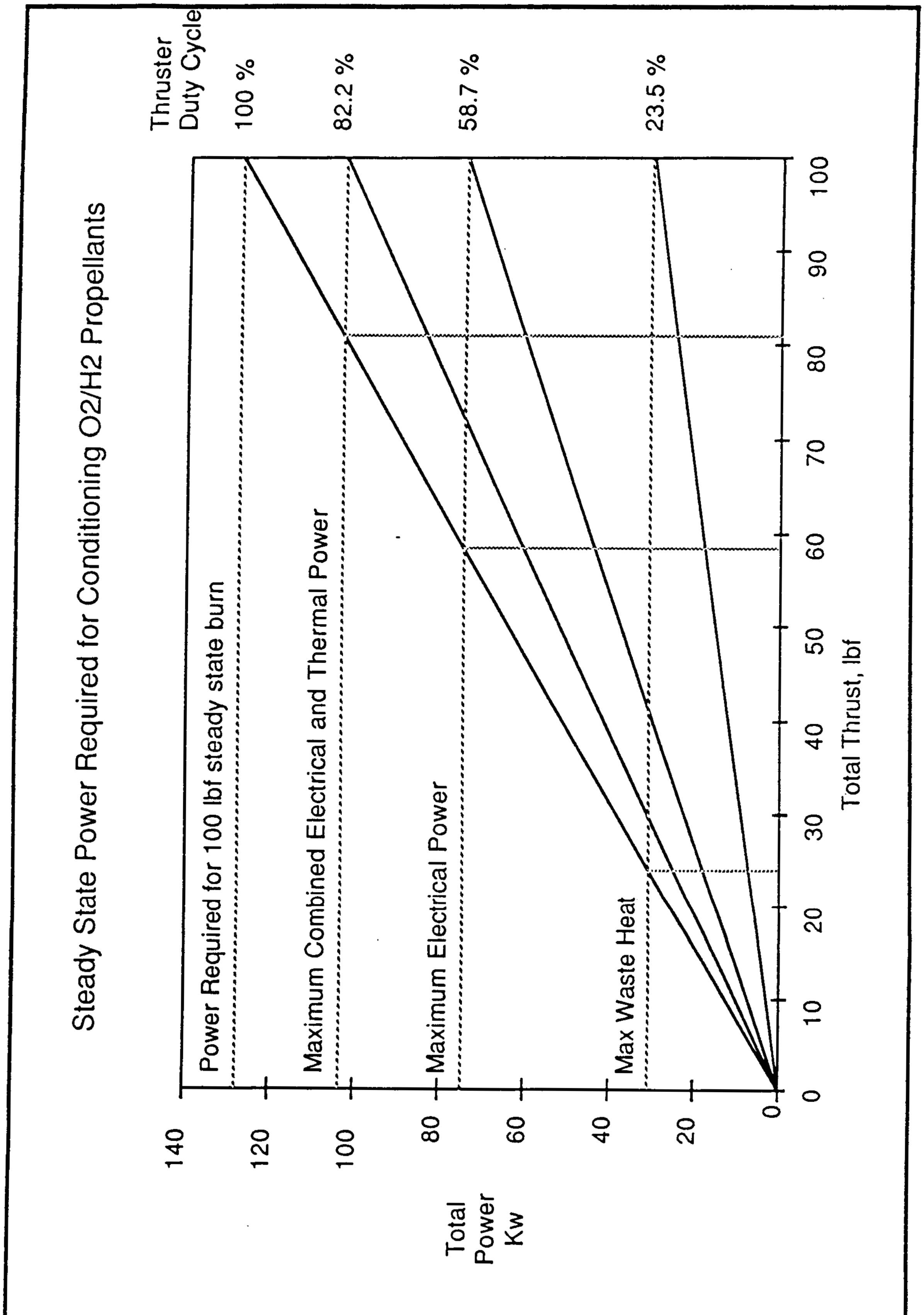


Figure II-1 - Steady State Initial Power

First, the level of detail was increased as higher level options were rejected, and secondly, additional options were identified as the process progressed. The proposal identified some 3000 possible concepts. Many of these were eliminated by inspection. However, the vast majority seemed feasible. A simple numerical sorting routine was then written which applied go-nogo rules to each individual concept and computed a rank for remaining concepts based on the sum of ranks given to each option within the concept. Using this routine, numerous passes were made through the whole process to identify options that should be retained for further study. As a bonus, sensitivities could be obtained for the ranks and rules used in the selection process. The resulting "concept" is shown in Figure II-2. This "concept" represents the options remaining. Additional screening was then applied to further narrow the concepts, as is discussed in the next section.

C. Preliminary Concept Selection

The selected concepts were combinations of the major propulsion system components that could provide the necessary functions and still meet the ground rules of the study. The minimum system that could be conceived consists of propellant storage tanks with heat exchangers/circulating systems to maintain the tank pressure during tank outflow. With sufficient power this system could supply propellant directly to the thrusters. Power restrictions on the Space Station preclude this as a viable option. This prompted the inclusion of accumulators. This represents the "minimum" concept considered for this study. The most complex system adds a propellant conditioning section which consists of additional heat exchangers and pumps to increase the temperature and pressure of the propellants prior to their introduction into the accumulators. These two systems are shown schematically in Figure II-3. Twenty-eight concepts were identified between the two extremes represented by the figure. Table II-2 details the concepts and the numbering code used. Essentially, the propulsion system was divided into two portions, the fluid storage and conditioning portion and the accumulators through thruster portion. Within these two portions various options were identified as described in the table. A hybrid set of concepts was also included whereby the oxygen system operates in a blowdown mode while the hydrogen system utilizes a pump to raise the pressure to that of the oxygen.

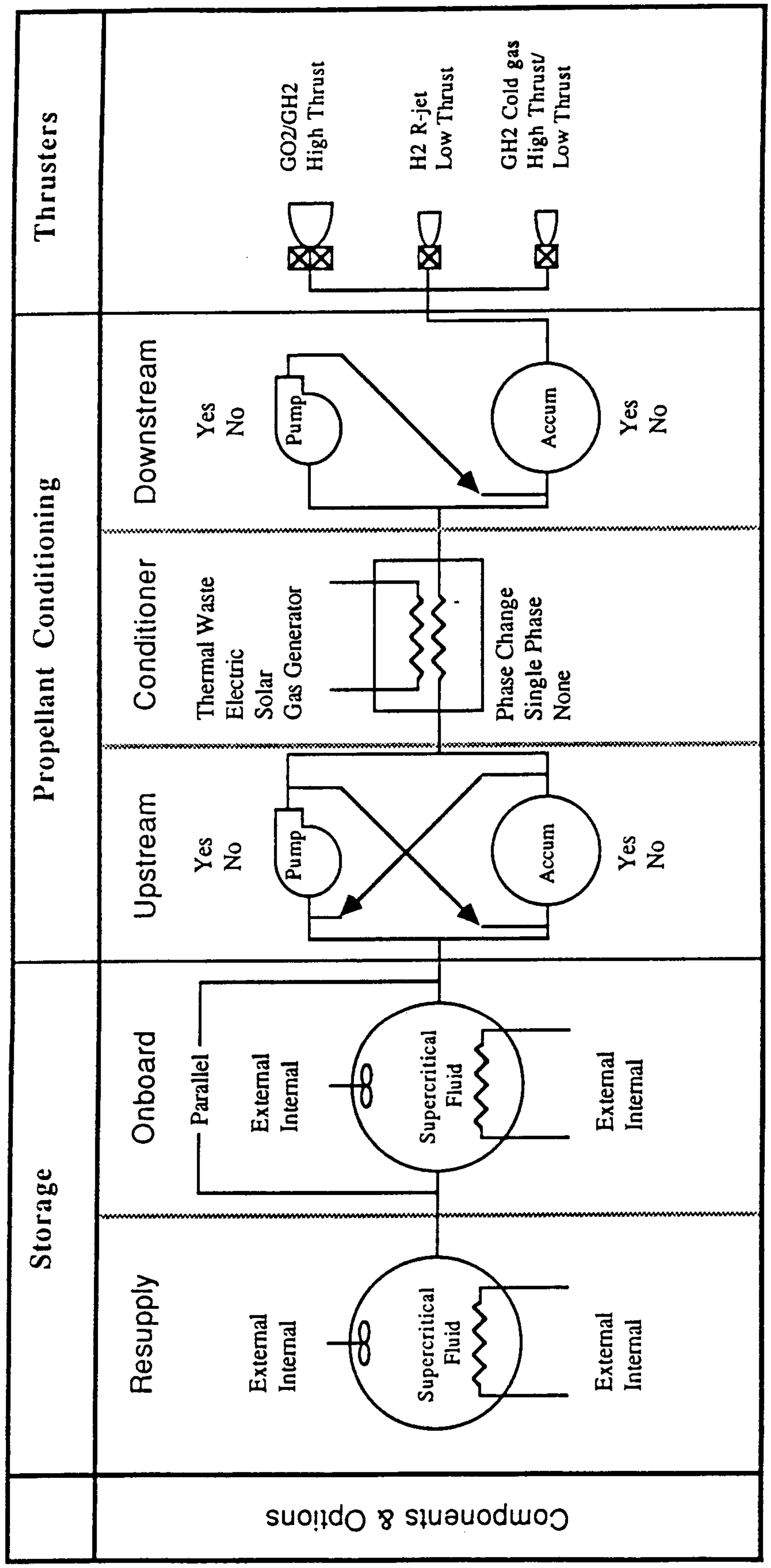
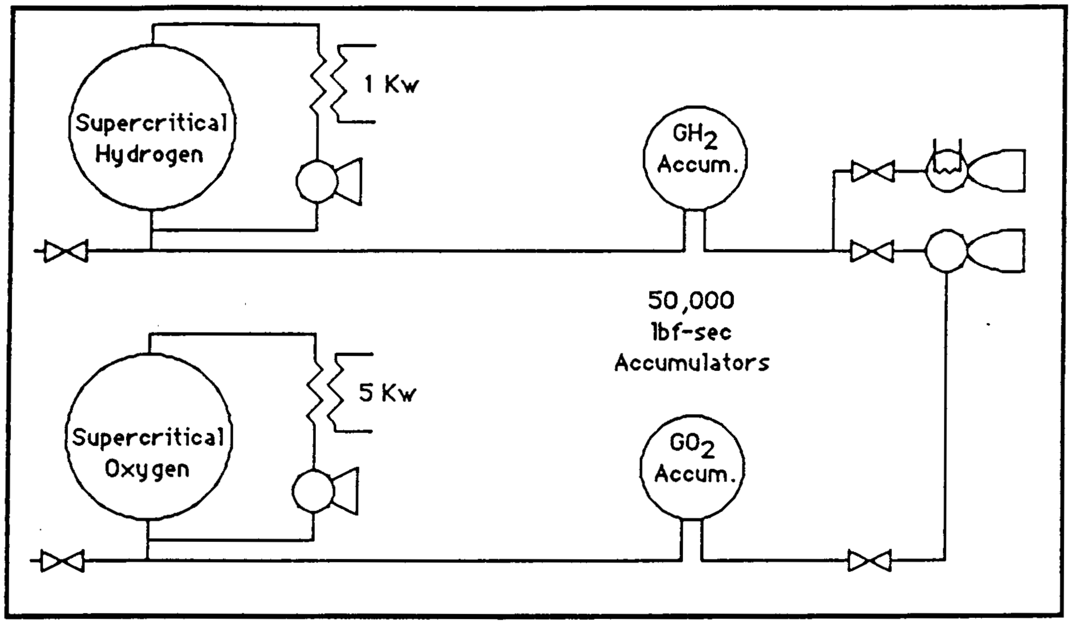
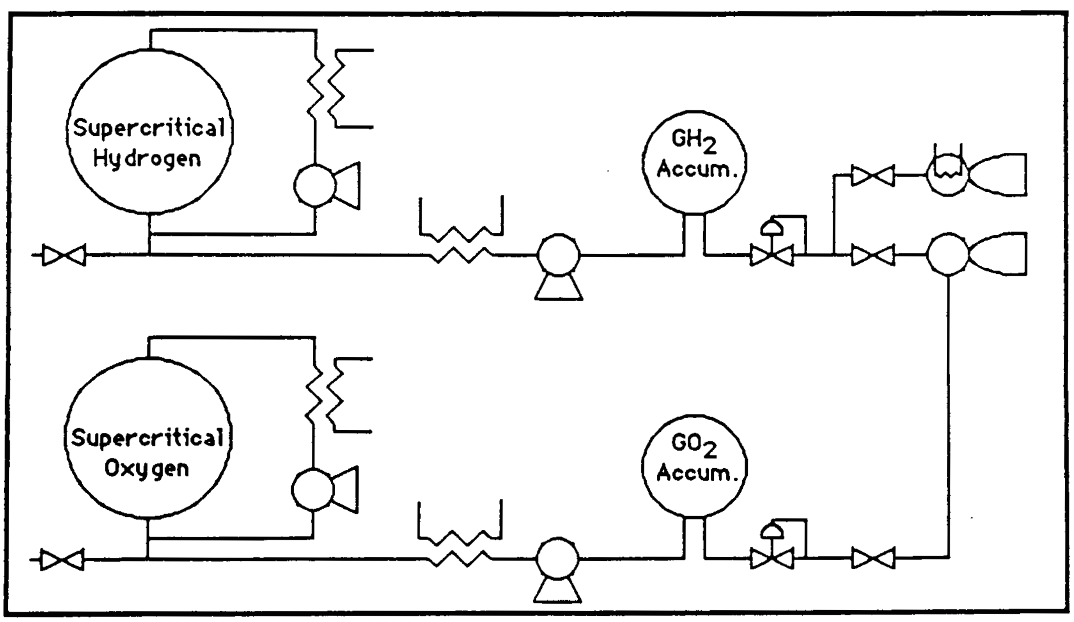


Figure II-2 - Preliminary Concepts



Simplest Concept



Most Complex Concept

Figure II-3 - Concepts Schematics

Table II-2 - Preliminary Concept Descriptions

CONDITIONING SUBSYSTEM OPTIONS - (Accumulators and propellant conditioning heat exchanger)

A	Blowdown without Heat Exchangers
B	Blowdown with Heat Exchangers
C	Regulated without Heat Exchangers
D	Regulated with Heat Exchangers

SUPPLY SUBSYSTEM OPTIONS - (Storage tank and heat exchanger)

System	Description
1 A,B,C,D	Blowdown propellant supply with condensing heat exchanger, small accumulators
2 A,B,C,D	Propellant supply pumped into accumulators with condensing heat exchanger. Small accumulator
3 A,B,C,D	Blowdown propellant supply utilizing thermal storage heat exchangers. Small accumulator
4 A,B,C,D	Blowdown propellant supply with condensing heat exchangers. Large accumulator
5 A,B,C,D	Pumped propellant supply with condensing heat exchangers. Large accumulator
6A,B,C,D	Pumped hydrogen, blowdown oxygen both using condensing heat exchangers. Small accumulators
7 A,B,C,D	Pumped hydrogen, blowdown oxygen both using condensing heat exchangers. Medium accumulators
8 A,B,C,D	Blowdown propellant supply, high power condensing heat exchangers, small accumulators
9 A,B,C,D	Pumped propellant supply, high power condensing heat exchangers, small accumulators

D. Component Models

An empirical atmosphere model was prepared using in-house data generated in support of our Space Station Phase B effort. This data is based upon the 1973 Goddard document NASA SP-8021. For this data base the empirical relationship shown in Figure II-4 was developed. The inaccuracies inherent in the fit (5%) are well within the accuracies of the data used (10%) and are thus felt to be adequate. This "brute force" method was chosen because it provided an easily evaluated expression which represented the predictable trends in the "average" global density variations and was suitable for incorporation into a variety of analyses used to predict space station propellant requirements. It is currently being used to estimate the impacts of various reboost strategies upon the propellant storage requirements. System sensitivities to reboost strategy can easily be quantified with this model.

To investigate the system sensitivities to thruster design parameters a curve fit to Marquardt GO_2/GH_2 thruster data was developed. This is shown in Figure II-5. Thruster performance is seen to improve with decreasing mixture ratio. System level sensitivities to chamber pressure favor higher pressures up to about 150 psia where further increases in pressure no longer improve performance significantly. Higher thrust levels will improve thruster performance and decrease burn time resulting in larger accumulators and higher disturbance levels. The net effect is to increase system dry weight since the improved performance does not offset the increased accumulator volume required. The major system sensitivity was found to be to the thruster mixture ratio. Trading resupply system launch mass against mixture ratio influences resulted in an "optimum" mixture ratio of about 6. This was discussed in detail in the March, 1985 report and is repeated as part of Figure II-6.

Two types of heat exchangers have been identified as applicable for the O_2/H_2 propulsion system. These are a conventional two fluid heat exchanger and a three fluid type where the third fluid used is a phase change material used to store thermal energy. The reference Space Station thermal control system is a three level two phase system. The middle temperature loop uses ammonia and is designed to operate at 70°F. This loop handles 70% of the Space Station thermal load and was selected as the waste heat source for the propulsion system. For sizing purposes this is modeled as a length of

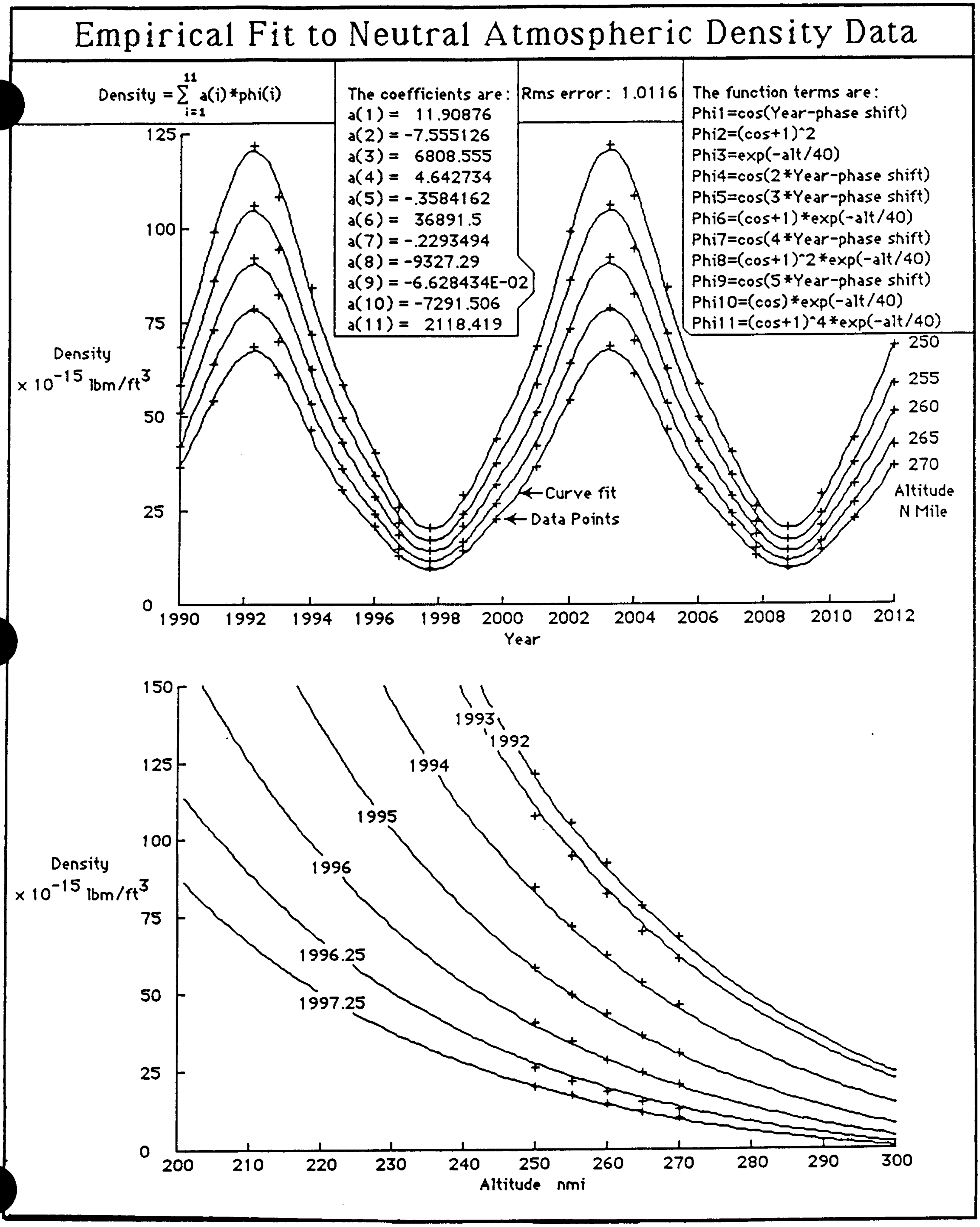


Figure II-4 - Atmosphere Curve Fit

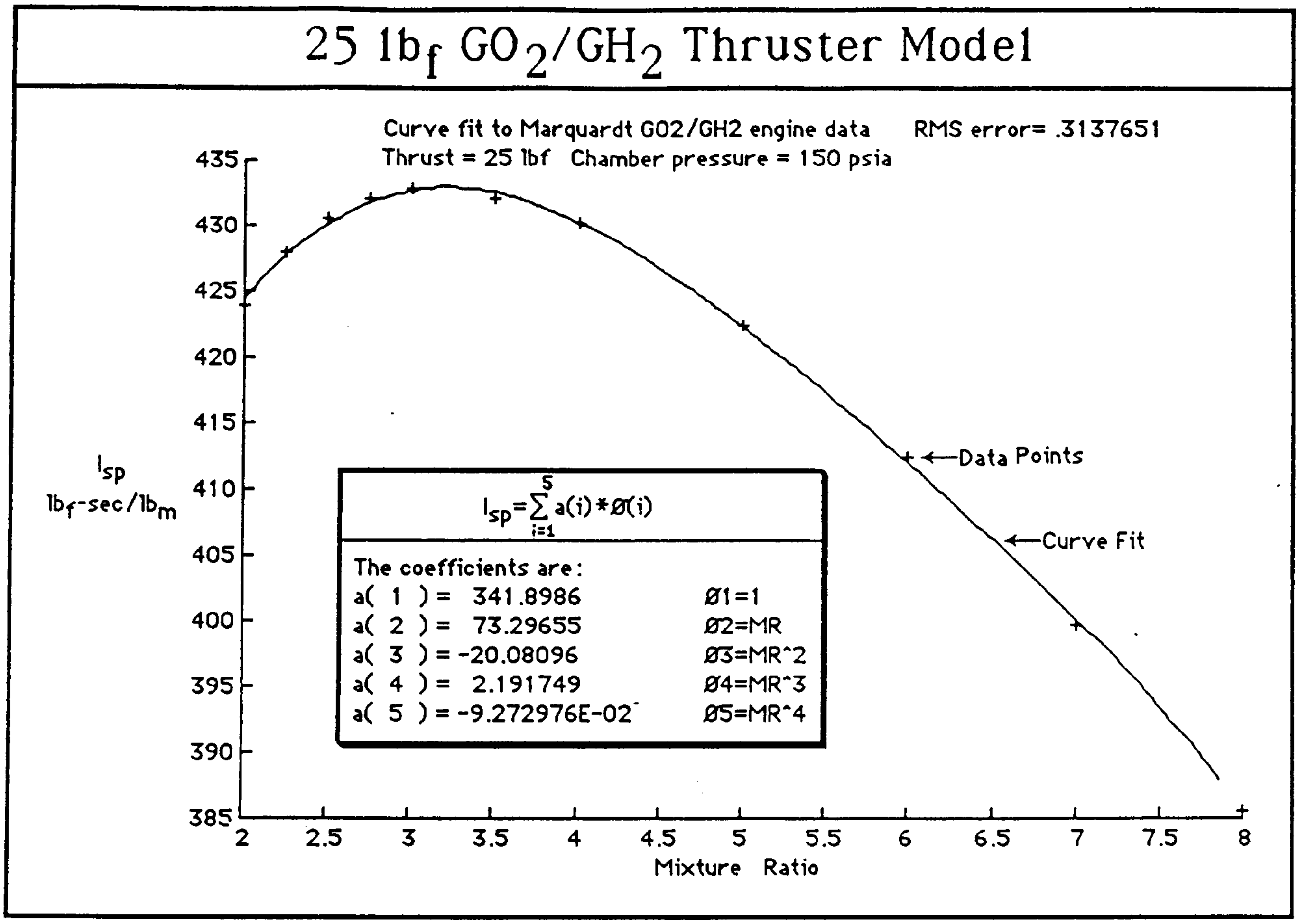
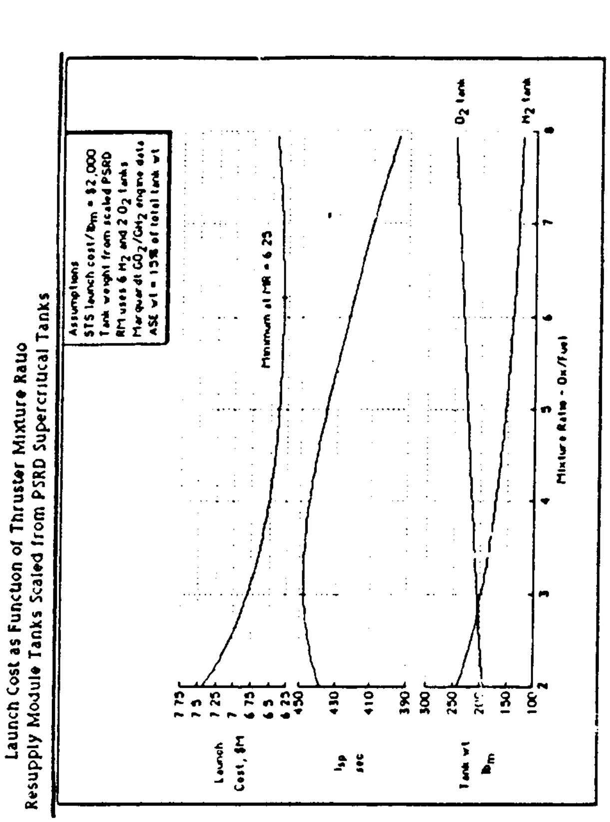
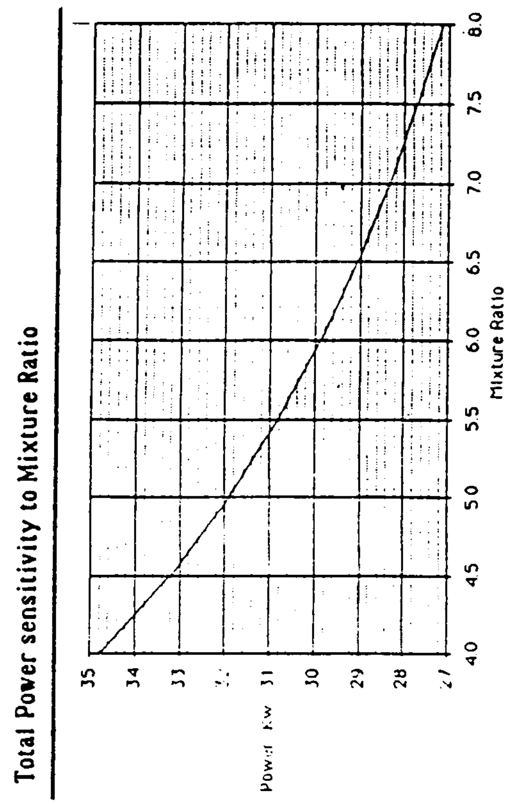
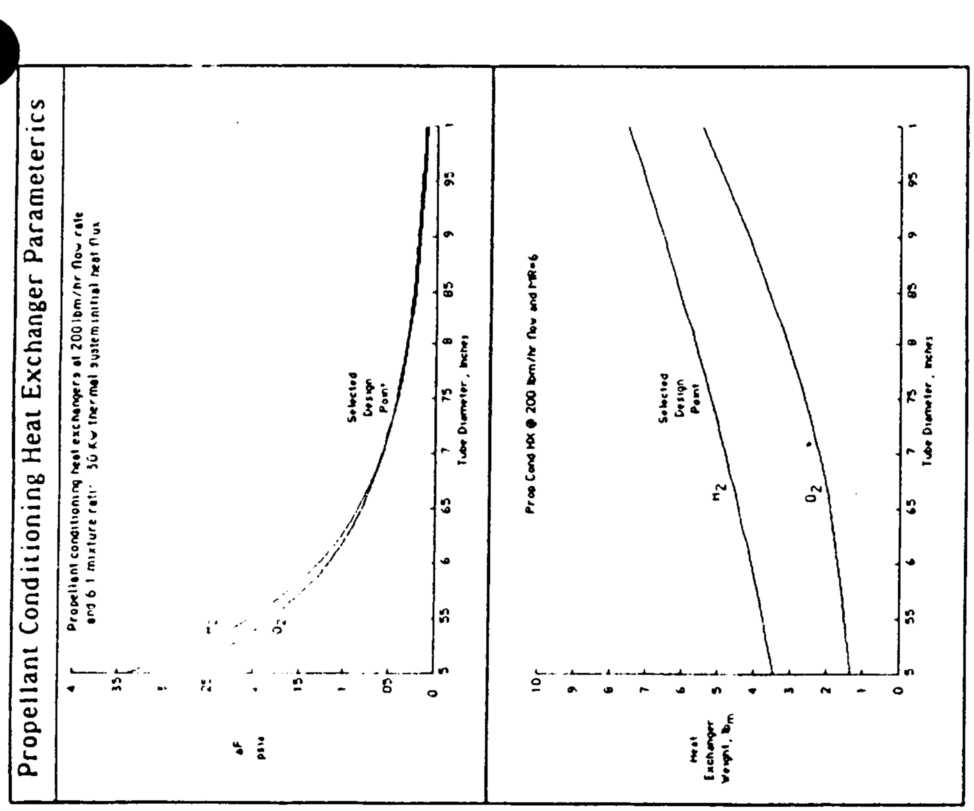


Figure II-5 - Thruster Curve Fit







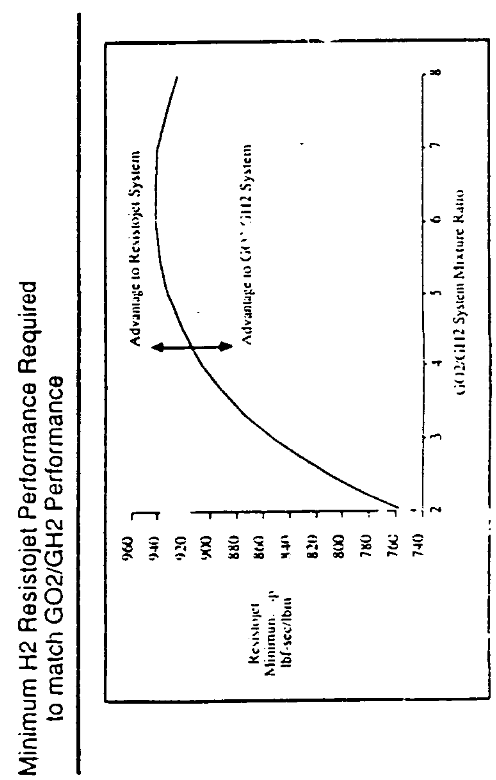


Figure II-6 - Typical System Sensitivities tubing with an assumed 70°F internal wall temperature. The ammonia side heat transfer coefficient is assumed to be 250 BTU/ft 2-hr°R. These heat exchangers will scale with the mass flow rate of the propellant, assuming fixed space station design conditions. The sizing relationship developed returns an "equivalent" heat exchanger tube length for the given design parameters (SS fluid properties and heat exchanger geometry) and the necessary heat flux, propellant mass flow rates and propellant fluid properties. Tube length is then used to estimate heat exchanger mass and cost from empirical relationships.

The fluid simulation model required development of a more detailed physical description of the heat exchangers. This was necessitated by the varying fluid properties of the cryogenic supercritical fluids encountered in the heat exchangers. The standard heat exchanger assumptions for ideal gases were not applicable. Two approaches were examined; a simple linear fluid property fit of c_p vs the bulk fluid and wall temperature difference, and a higher order iterative approach. With an accuracy difference of less than two percent, the simpler approach was used, mainly to speed execution of the program. The equation and its derivation is presented in Appendix A. Essentially, the simulation routine uses the fixed heat exchanger geometry from the sizing program. Using a user supplied mass flow rate for the blower/heat exchanger loop, the routine predicts the heat exchanger exit temperature from which the thermal energy transferred can be determined. For the purposes of fluid simulation, both the thermal storage and condensing heat exchangers are the same. The simulation model does not attempt to model the physics of the thermal storage heat exchanger, primarily due to lack of data, but also because the gross impacts of this type of heat exchanger can be adequately accounted for by the sizing routine.

The phase change or thermal storage type heat exchanger will scale with the maximum quantity of thermal energy to be transferred. Thus they are sensitive to the worst combination of propellant mass and temperature difference required by a single burn. Sizing then depends on the reboost mode. These heat exchangers will therefore become smaller as the burn frequency increases and the individual burn times decrease. The sizing routine uses the mass of the burn and the storage tank end temperatures to determine the maximum amount of thermal energy required. From this, the mass of the wax and required heat transfer area are estimated. Currently, octadecane wax is being used to estimate the mass of phase change material needed. The mass and size of the heat exchanger can then be computed with an assumed efficiency and suitable geometry assumptions. The modeling diagram used is shown in Figure II-7.

The size of the accumulators depends upon the reboost mode and the pressure blowdown range chosen. Accumulator tank diameters are shown as a function of total impulse in Figure II-8. This figure assumes four accumulators each for oxygen and hydrogen. The bottom pressure is determined by allowable thruster inlet conditions, the top pressure by the pump pressure ratio (and solar heat soaking of the accumulator, if allowed). The reboost mode parameters and maximum heat flux available to the propulsion system determine the minimum amount of mass to be stored in the accumulators. However, the absolute minimum amount stored must be enough to handle the shuttle docking disturbances (50,000 lb_f-sec). The sizing model developed uses composite accumulator material properties to estimate spherical tank geometry and weight, with allowances for "nonoptimum " effects. The exact design for the accumulators will be determined during Task III.

The sizing routine developed for supercritical tanks returns estimated volumes and masses of the tanks based upon the mass of propellant required by the reboost strategy and fluid properties at the initial and final conditions. Constant pressure expulsion is assumed so the determining fluid property is the final storage temperature, assuming initial temperatures of 36°R and 110°R for the hydrogen and oxygen, respectively. For the hydrogen supercritical storage tanks, a trade off exists between higher final temperatures reducing the size of the tanks and increasing the size of the storage tank heat exchanger. This results from the increased heat flux required at the end of tank expulsion. Furthermore, the use of a pump in the hydrogen line reduces the temperature required out of the propellant conditioning heat exchanger. For an accumulator temperature of 540°R at 1000 psia, the pump inlet temperature is 370°R. At the end of tank expulsion, no heat flux is required from the hydrogen propellant conditioning heat exchanger, however, the storage tank heat exchanger requires 25 Kw at a 100 lbf flow rate. Conversely, lower tank end temperatures increase the required tank size resulting in increased launch mass. The choice of a final depletion temperature will require a careful balancing of the supercritical tank storage (and logistics) efficiency, with the system operation efficiency.

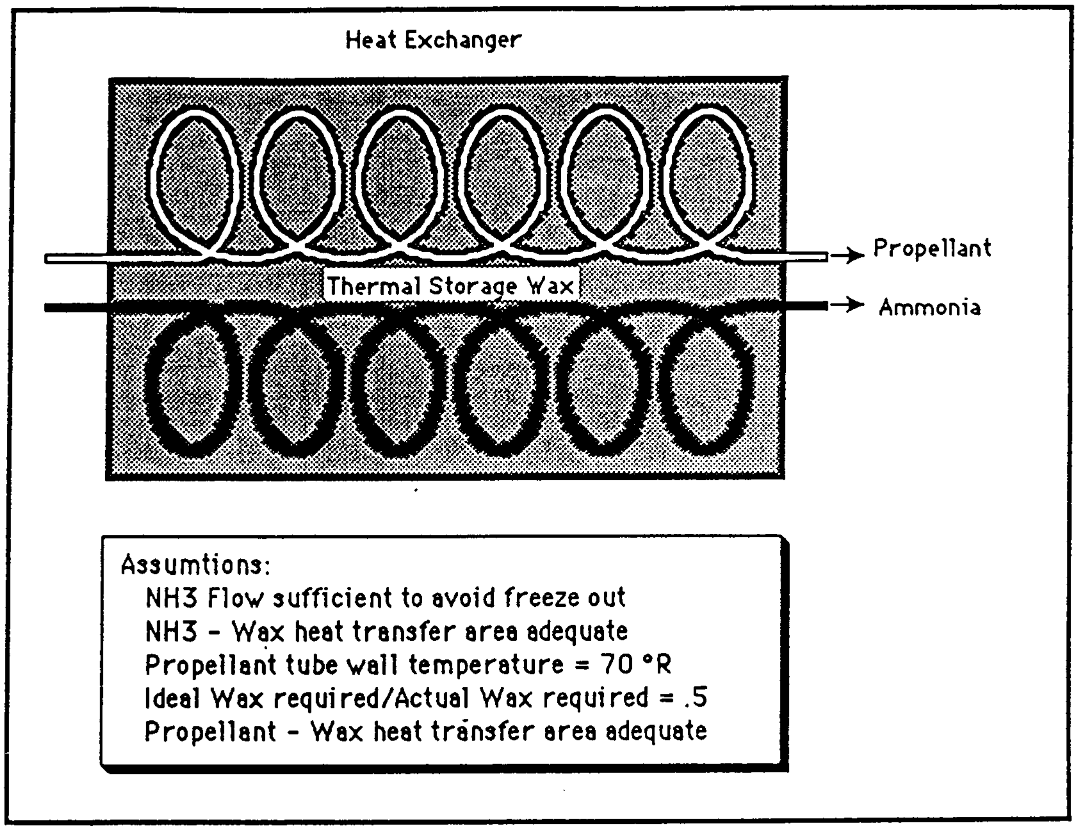


Figure II-7 - Thermal Storage Heat Exchanger Modelling Diagram

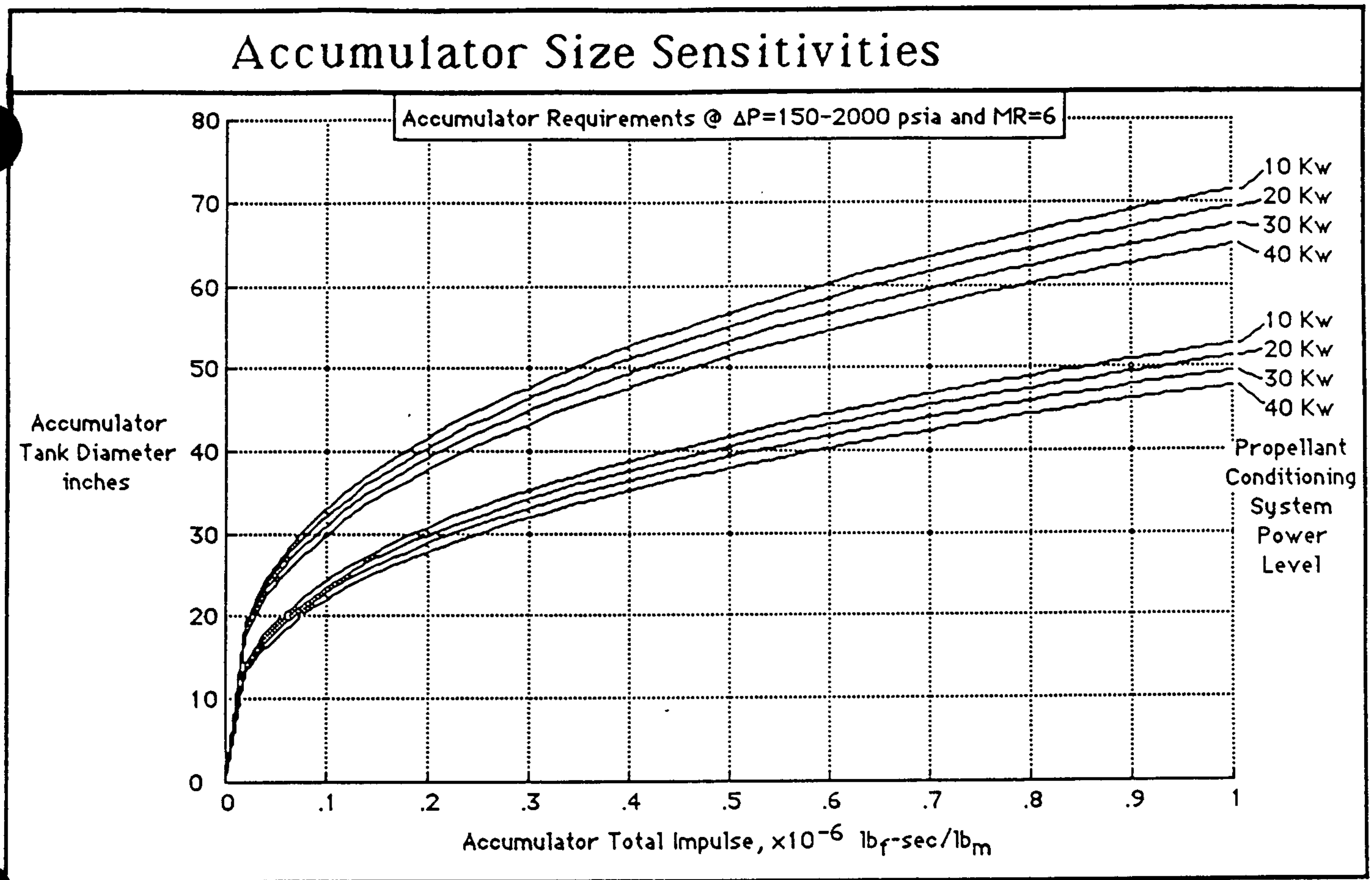


Figure II-8 - Accumulator Volume vs Total Impulse

The fluid simulation program necessitated that a more complete model of the supercritical tank be developed. This model was developed to primarily model the system operation during tank outflow. Hence, the heat transfer between the tank walls and the fluids was neglected relative to the heat being intentionally added to maintain tank pressure during outflow. The model assumes that the fluids are completely mixed so a single node is all that is necessary to represent the tank. During tank outflow, fluid is removed from the tank and sent through the storage tank heat exchanger where heat is transferred to the fluid prior to reintroducing the fluid into the tank. This process is modeled by the simulation programs supercritical tank subroutine. The equations and derivation are shown in Appendix B. Currently, the process modeled is a constant mass flow from the tank with variable heat transfer. A typical outflow plot is shown for the hydrogen tank in Figure II-9. The routine will be modified to model the constant heat flux/variable mass flow case during Task III. The current model can be used to approximate the periods between burns by using a very small external loop mass flow to transfer the heat equivalent to the heat leak which would occur during coast periods. This would be an approximation because the well mixed tank assumption would probably not be valid. However, the allowable heat leak could be determined this way.

E. Model Integration

Two system models were prepared during the performance of Task I, a sizing program used to investigate system level sensitivities to design parameters, and a fluid simulation routine which estimates quasi-steadystate fluid properties as the system operates. The later also serves as a check on the sizing routine as well as the program to be generated in Task IV. The component routines used in the individual programs are described in the previous section. The two routines were developed independently so the check function integrity could be maintained.

The top level organization for the sizing routine is shown in Figure II-10. The program was designed to support interactive analyses and to run on a personal computer. For this reason the input/output was broken into three sections progressing from top level to lowest level. The top level system parameters are concerned with the amount of propellant required and the

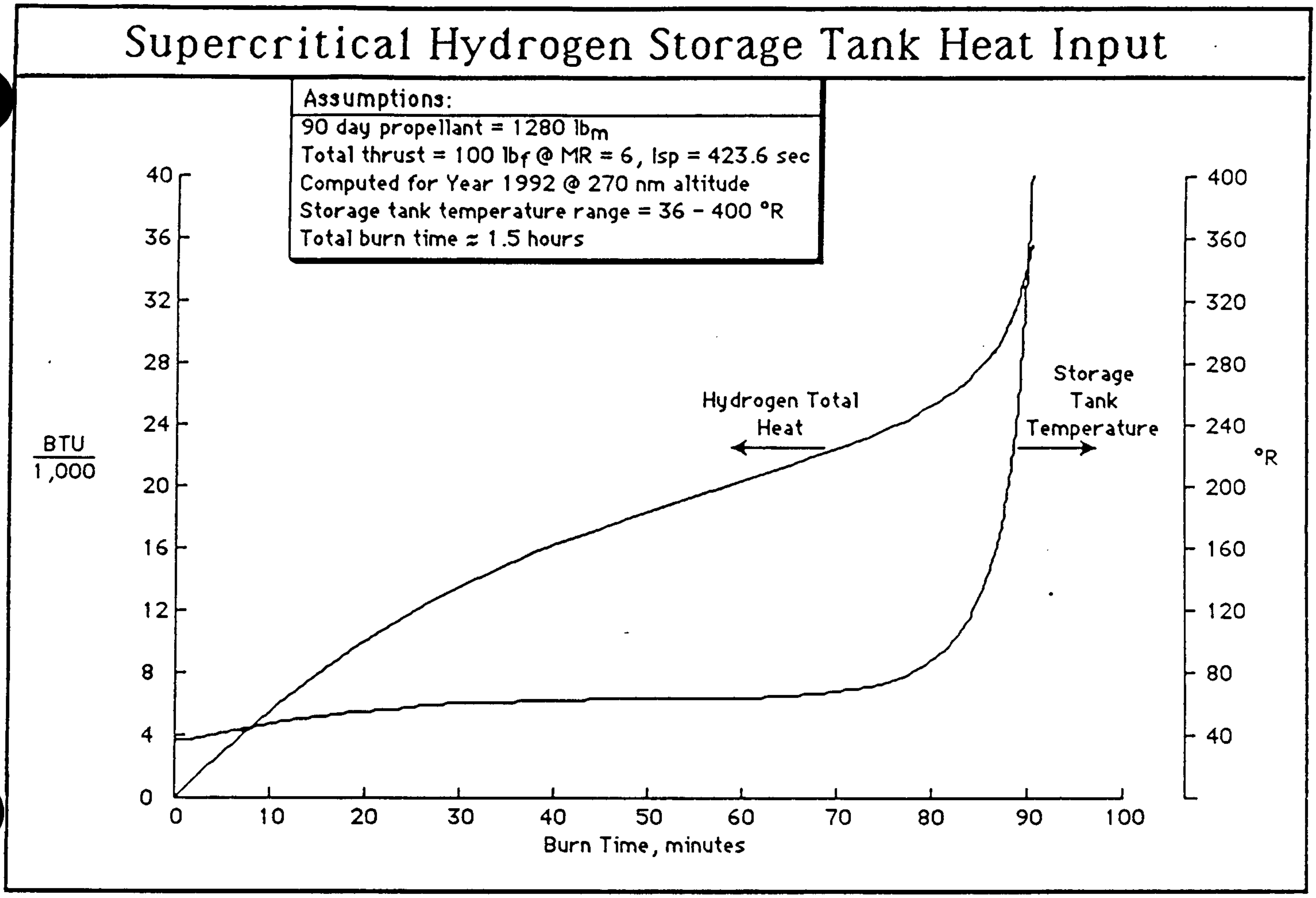


Figure II-9 - H2 Tank Cumulative Heat Flux

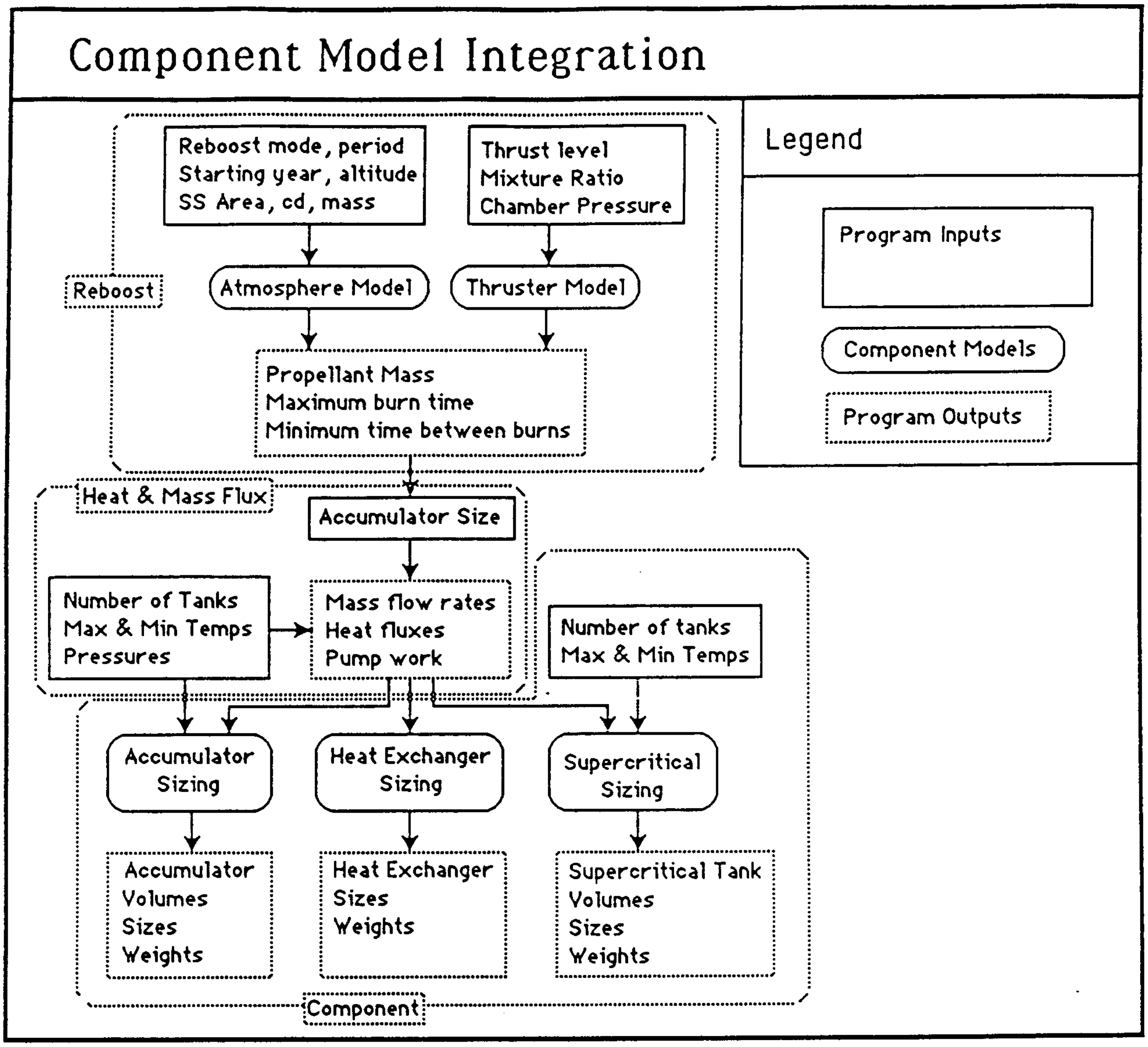


Figure II-10 - Sizing Program Flow Diagram duty cycle by which this propellant is consumed. Space station drag and thruster performance parameters are used by the atmosphere and thruster models to determine the amount of propellant required and the thruster burn time parameters. This section, termed "Reboost", is used to explore Space Station reboost sensitivities, like the mixture ratio trade discussed previously.

Parameters affecting the mass flow and heat flux rates comprise the next section, termed "Heat & Mass Flux". At present, the program uses propellant stored in the accumulators plus "real time" conditioned propellant to make up the propellant mass required by a burn. Therefore, larger accumulators will reduce the mass flow rate required out of the conditioning system for a fixed burn mass. The heat fluxes and pump power scale with the propellant mass flow rate. Hence, once the sizes and operating conditions for the accumulators are specified, the mass flow rates, the pumping power and the heat fluxes required can be determined. As with the Reboost segment, this segment can be used to explore system sensitivities related to mass flow rate. For instance, the accumulator size can be used to adjust the peak power required by the storage and propellant conditioning heat exchangers.

The final segment, termed "Component", is used to size the components after the previous two levels have been defined. The input consists mainly of heat exchanger type selection, and the number of accumulator and supercritical storage tanks. Supercritical tank and accumulator material and geometries are presently fixed, although they will be enabled when the system definition progresses to that level of detail. At that point the program will be modified so that each component is itself a program segment. This segment of the program is used to adjust the packaging of the accumulator and resupply tanks. The program currently includes a 20% contingency for sizing the supercritical tanks. The two types of heat exchangers described above are included as options for both the storage and propellant conditioning locations. The program must treat each unit separately because the design point for each propellant changes by location and type. The thermal storage units scale with the quantity of mass to be conditioned while the condensers scale with mass flow rate. The storage tank heat exchangers must vary heat flux over a 6:1 range if a constant mass flow from the tanks is to be maintained. The design condition for the hydrogen storage tank heat exchanger occurs at the end of tank expulsion unless the final temperature is less than approximately 100°R. The initial storage conditions present the design point for the oxygen storage heat exchangers. The final conditions will size the propellant conditioning heat exchangers for both fluids because the temperature difference is least at that point.

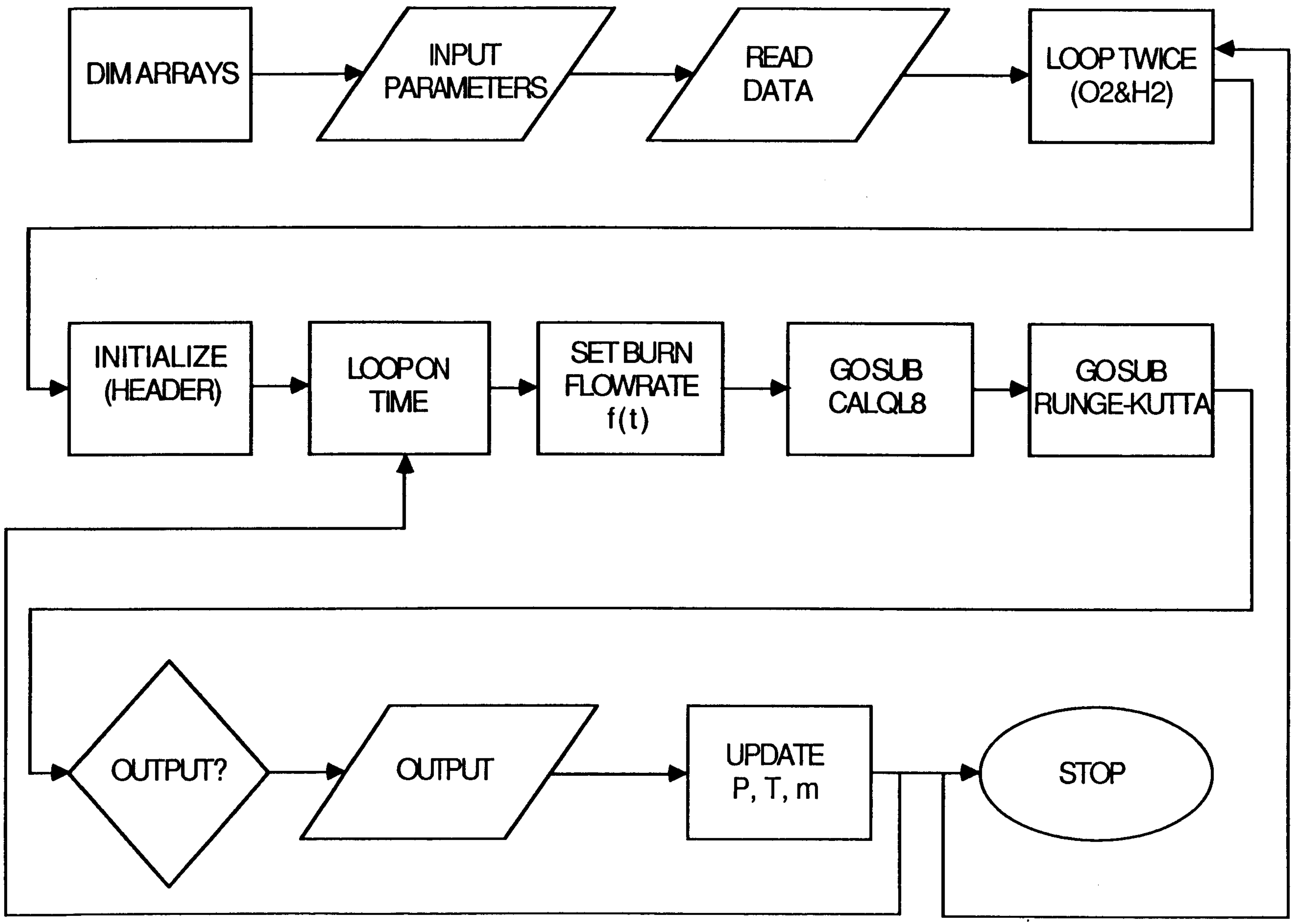
An integrated fluid simulation model was written to model the fluid outflow from the supercritical tanks through to storage in the accumulators. Fluid properties are computed as a function of time. Two basic fluid loops are modeled, one the external heat exchanger loop used to add the heat to maintain tank pressure and the second used to condition the propellants from the storage conditions up to the accumulator conditions. The accumulator blowdown itself was not modeled. The routine sums the energy added to the fluid mass in small time steps so that quasi-steady fluid flow and constant fluid property assumptions can be used. Ideal gas assumptions were used for the accumulator fluids. A flow diagram for the program is presented in Figure II-11 and a schematic is shown in Figure II-12. The two fluids are treated separately for ease of programming. The use of accumulators to separate the fluid conditioning system from the thrusters enables this simplification. The simulation program uses data from the sizing routine as input. A sample run for an earlier design point is shown in Figure II-13. This run used 1280 lbm of propellant stored in one tank per fluid with the initial conditions as slated on the chart. The total energy required will scale with the total propellant mass as long as the fluid end conditions are similar. The newer drag calculations indicate about a 60% reduction in required propellant resulting in a corresponding reduction total energy input.

F. OMV Propulsion Requirements

Propulsion requirements were assembled for OMV. These include total impulse, thrust level(s), duty cycles, and envelope restrictions (where applicable). These propulsion requirements were primarily taken from previously completed studies on OMV. The pertinent OMV propulsion requirements are shown in Table II-3. The missions these apply to include the following list of OMV operations:

Payload placement and/or retrieval
Payload reboost
Payload deboost to re-entry
Payload viewing or in-situ servicing
Provide base support
OTV/Payload transfer

Figure II-11 - Simulation Model Flow Diagram



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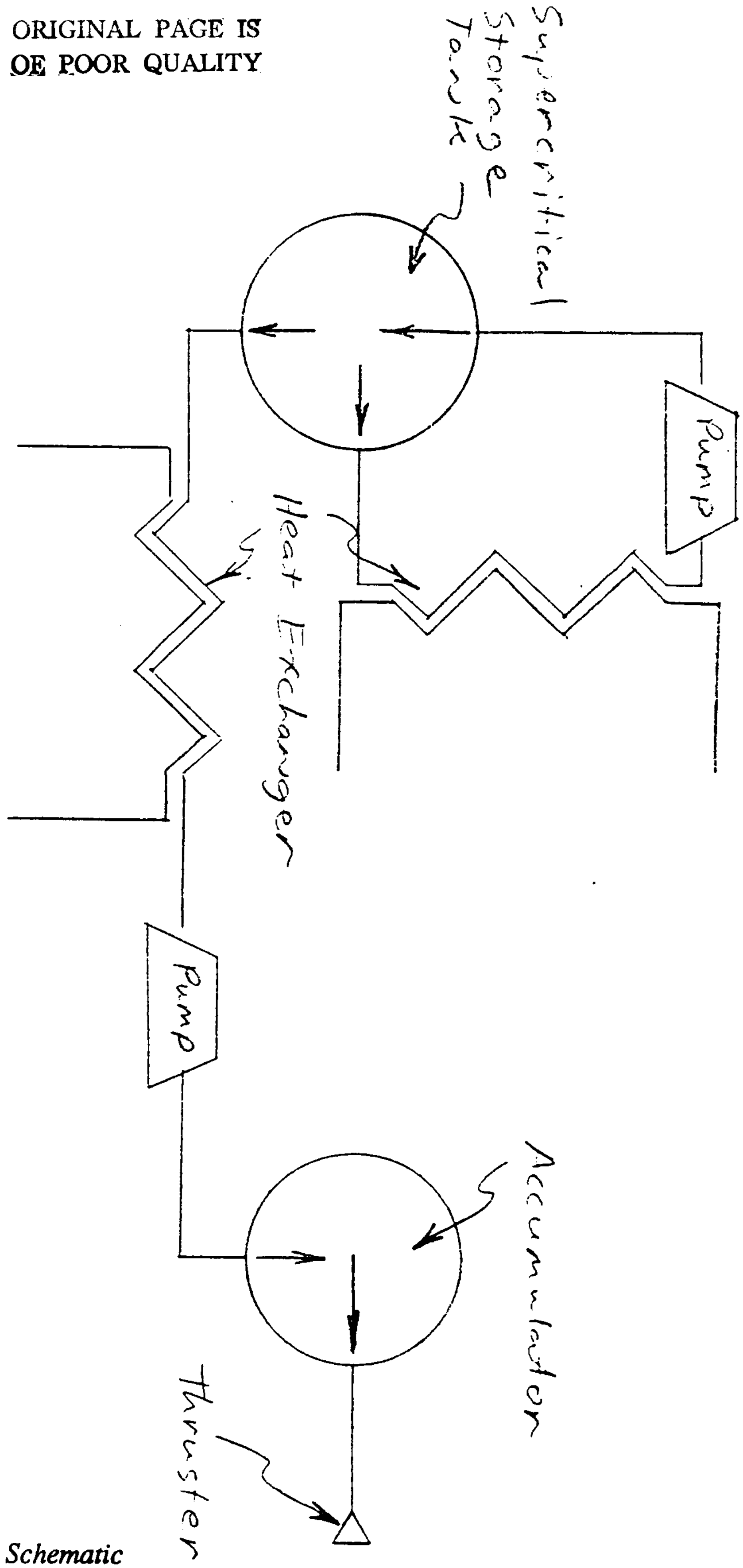


Figure II-12 - Simulation Model Schematic

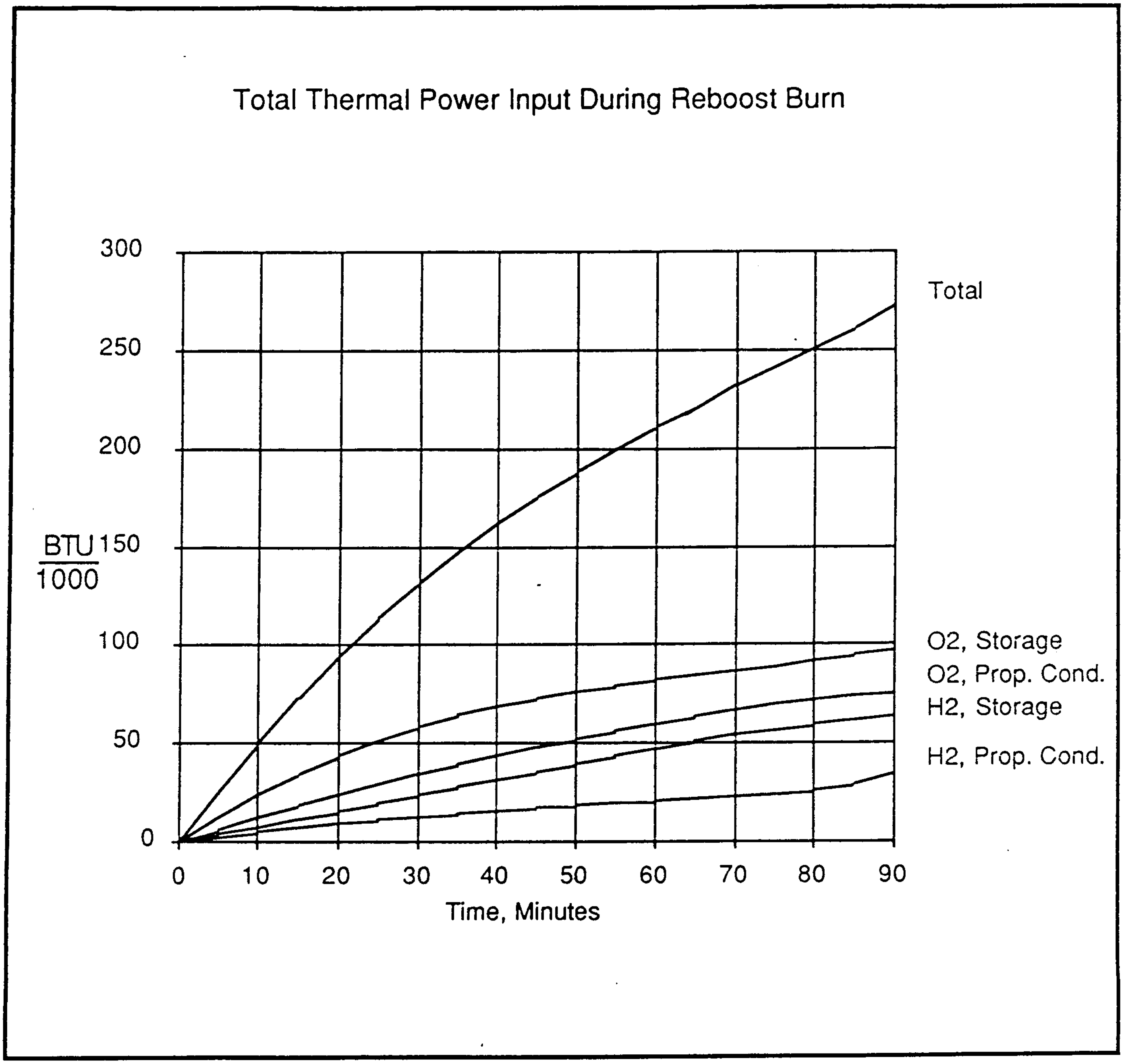


Figure II-13 - Cumulative Heat Flux - 540,000 lbf-sec System

TABLE II-3 OMV PROPULSION REQUIREMENTS

REQUIREMENT	VALUE	REMARKS
Total Impulse	1.9 M lbf-sec	DRMs
Thrust Levels		
- delta V	800 lbf total	Orbit adjust
- ACS	15 lbf; 24	
- prox. ops	5 lbf; 24	Cold gas
Propellant acquisition	Total tank expulsion in zero-g	

Propulsion system dry weight estimates were used in calculating the propellant quantities required to complete the OMV Design Reference Missions with each of the OMV propulsion system candidates. An additional 1436 lbm (electrical power system, video, etc.) was added to the propulsion system dry weight in order to obtain a "full up" OMV dry weight.

The following specific impulse values were used for the candidate OMV propulsion systems:

Storable Bipropellant - 305 sec, (285 sec for low thrust)

Hydrazine - 230 sec

GO_2/GH_2 - 415 sec (405 sec for low thrust)

Supercritical O_2/H_2 - same as GO_2/GH_2

Table II-4 shows the Design Reference Mission (DRM) designation numbers, the quantity of those missions, and the mission propellant quantities for each of the propulsion system (propellant) candidates. Missions 2 and 9 were altered slightly from the Design Reference Mission descriptions to reflect OMV operation out of Space Station rather than the STS. For instance, it was assumed that the payload placement of Mission 2 would be 500 nmi (340 nmi above an STS altitude of 160 nmi or 250 nmi above a Space Station altitude of 250 nmi) rather than 340 nmi above an OMV base (SS or STS) altitude of 250 nmi. Likewise, in Mission 9 it was assumed that the servicing mission was to take place at 1,000 Km (290 nmi above the Space Station orbit at 250 nmi) at a co-orbiting platform rather than 400 nmi above a 160 nmi STS base (which amounts to 1,000 Km) or a 250 nmi Space Station base as specified in the DRM.

Modifying DRM's 2 and 9 allows a GO_2/GH_2 OMV propulsion system sized for 2,600 lbm usable propellant to capture the majority of the OMV missions. A GO_2/GH_2 system sized to capture the ultimate OMV missions would result in either an extremely massive OMV (nearly 20,000 lbm in dry weight and 5,600 lbm propellant) in order to complete DRM 10, or an infinite size OMV (no solution in sizing) to complete DRM 6.

)

Table II-4 - OMV Propellant Requirements (1992-2000)

		Mono N2H4		Bi Prop		Gaseous 02/H2		S/C 02/H2	
DRM	#	Mission	Total	Mission	Total	Mission	Total	Mission	Total
1	4	5824	23296	4594	18376	4295(2577)	17180	4083	16332
*2	9	2339	21051	1676	15084	2459	22131	2216	19944
3	4	2767	11068	1980	7920	2557	10228	2336	9344
4	2	2300	4600	1680	3360	2175	4350	1939	3878
5	1	3377	3377	2492	2492	2511	2511	2376	2376
6	2	4083	8166	2734	5468	No Solution		4708	9416
7	6	1320	7920	966	5796	1313	7878	1195	7170
8	4	1131	4524	832	3328	1041	4164	958	3832
**9	6	2639	15834	1900	11400	2553	15318	2323	13938
10	33	8032	265056	5939	195987	5700***	188100	4812	158796

	Mono N2H4	Bi Prop	Gaseous 02/H2	S/C 02/H2
Totals	364892	269211	271860 (w/o DRM 6)	245026
Totals without Log Module Ferry (DRM 10)	99836	73224	83760 (w/o DRM 6)	86230

* altered to 250 nmi above base (500 nmi)
** altered to 290 nmi above base (1000 Km)
*** vehicle uniquely sized for this mission

0129Y/0019H

The OMV shall also shall also be capable of growth in order to accommodate "Future Capability Missions." These include:

Extended on-orbit operation
Secondary upper stage
Operations in GEO (including servicing and refueling)

G. OTV Propulsion Requirements

The primary driving mission for OTV design as defined in the NASA MSFC Revision 8 OTV Nominal Mission Model consists of delivering a 20000 lbm payload to GEO. The other driving missions include 12000 lbm delivery to GEO with 2000 lbm returned and a manned servicing mission servicing mission with 7500 lbm delivered to and from GEO. Table II-5 shows the resulting propulsion requirements for OTV.

TABLE II-5 OTV PROPULSION REQUIREMENTS

REQUIREMENT	VALUE	REMARKS
delta V	14000 fps 6600 fps	LEO to GEO deorbit & rendezvous
Payload	20000 lbm to GEO 7500 lbm LEO-GEO-LEO	Rev. 8, MSFC
Mission time	3 days 20 days	20,000 lbm delivery 7500 lbm round trip
# Burns	6	LEO-GEO-LEO
Thrust		
- delta V	15000 lbf total	Minimize delta V
- ACS	100 lbf (14 places)	Aerocapture maneuver
Safety		
- Manned	Fail safe	Return to LEO
- Unmanned	Fail operational	Complete mission

H. Platform Propulsion Requirements

Propulsion requirements for polar and co-orbiting platforms derived from Space Station WP-03 studies are shown in Table II-6.

Emphasis on platform propulsion for the polar missions is largely on performance due to the high energy requirements of deorbit, rendezvous with STS for servicing, and reboost to operational altitude. In addition, the long on orbit time (30 mo), imposes some restrictions on propellant choice such as the elimination of cryogenic fluids, both subcritical or supercritical. Therefore, with similar driving requirements of performance and long mission times for OMV, the platform candidate propellants are similar to those chosen for OMV. These include N_2O_4/MMH, N_2O_4/N_2H_4, gaseous oxygen/ hydrogen, and hydrazine.

The operational altitudes for the platform missions range from 400 km (216 nm) to 900 Km (486 nm) for the polar platform and 500 km (270 nm) to 1000 km (540 nm) for the non-coplanar co-orbiter. Servicing for the polar platform is intended to take place every 30 months. The service interval for the non-coplanar co-orbiter is from one to two years depending upon the altitude difference from the Space Station. Otherwise, these platforms must be serviced by a shuttle launched into their plane.

The polar orbit IOC platform requires two shuttle launches to place the service core, payload carrier, and payload into orbit. Using the full carrying capability of the shuttle, an initial mass of 27045 lbm was assumed for the platform. Figure II-14 shows the platform round trip velocity requirements for variations in operational altitude starting at a servicing altitude of 350 km (189 nm) and using the 27045 lbm initial mass. The performance parametrics for five of the platform propulsion options (Bipropellant (N_2O_4/MMH), Dual Mode Bipropellant (N_2O_4/N_2H_4), Hydrazine, Cold Gas O_2/H_2-100% Delivery and Storage Tanks, GO_2/GH_2-Electrolysis-10% Accumulators were determined for several delta velocities in the range. The OMV delivered electrolysis system was not considered for the polar platform due to the high penalty to the payload mass caused by delivering the OMV along with the platform to the servicing orbit. Figure II-15 summarizes the payload capability for each system at the various velocities.

TABLE II-6 PLATFORM PROPULSION REQUIREMENTS

REQUIREMENT	VALUE	REMARKS
Payload		
- Polar	4000-5000 Kg	
- Co-orbiting	6000-10000 Kg	
Altitude		
- Polar	400-900 Km	350Km service alt.
- Co-orbiting	463-1000 Km	SS servicing
Power Available		
- Continuous	0.5 Kw	during on-orbit
- Orbit adjust	5.0 Kw	experiments shut down
Mission Time	30 mo. on-orbit	return to STS for servicing
Acceleration	0.03g MAX	Solar array

I. OMV, PLATFORM & OTV PROPULSION SYSTEM CANDIDATES

The candidates considered for OMV and platforms propulsion are listed below:

OMV	PLATFORM
$N_2O_4/MMH/GN_2$	$N_2O_4/MMH/GN_2$
N_2O_4/N_2H_4	N_2O_4/N_2H_4
N_2H_4	N_2H_4
O_2/H_2 (Gaseous)	O_2/H_2 (Electrolysis)
	O_2/H_2 (Gaseous)

These propellants and resulting propulsion systems configured to meet the requirements discussed earlier were the candidate concepts examined in depth.

Due to the results of the OTV Concept Definition and Study Analysis, Contract NAS8-36108, and the Advanced OTV Propulsion System Study, Contract NAS3-23858 (both performed at Martin Marietta), a cryogenic LO_2/LH_2 concept has been recommended for space based OTV. This recommendation was

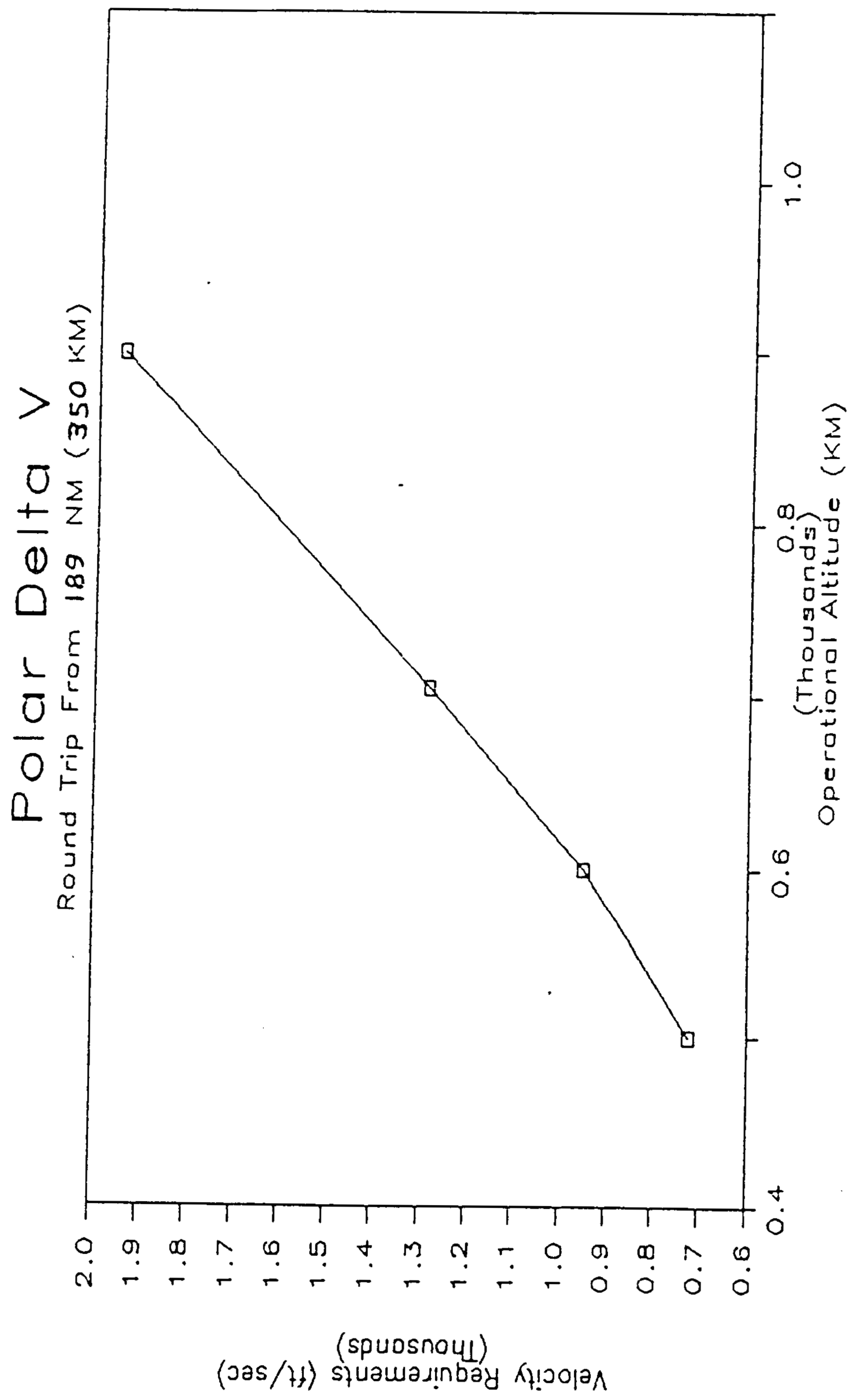

Polar Delta V
Round Trip From 189 NM (350 KM)
Velocity Requirements (ft/sec)
(Thousands)
2.0
1.9
1.8
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
0.9
0.8
0.7
0.6
0.4
0.6
0.8
1.0
(Thousands)
Operational Altitude (KM)

Figure II-14

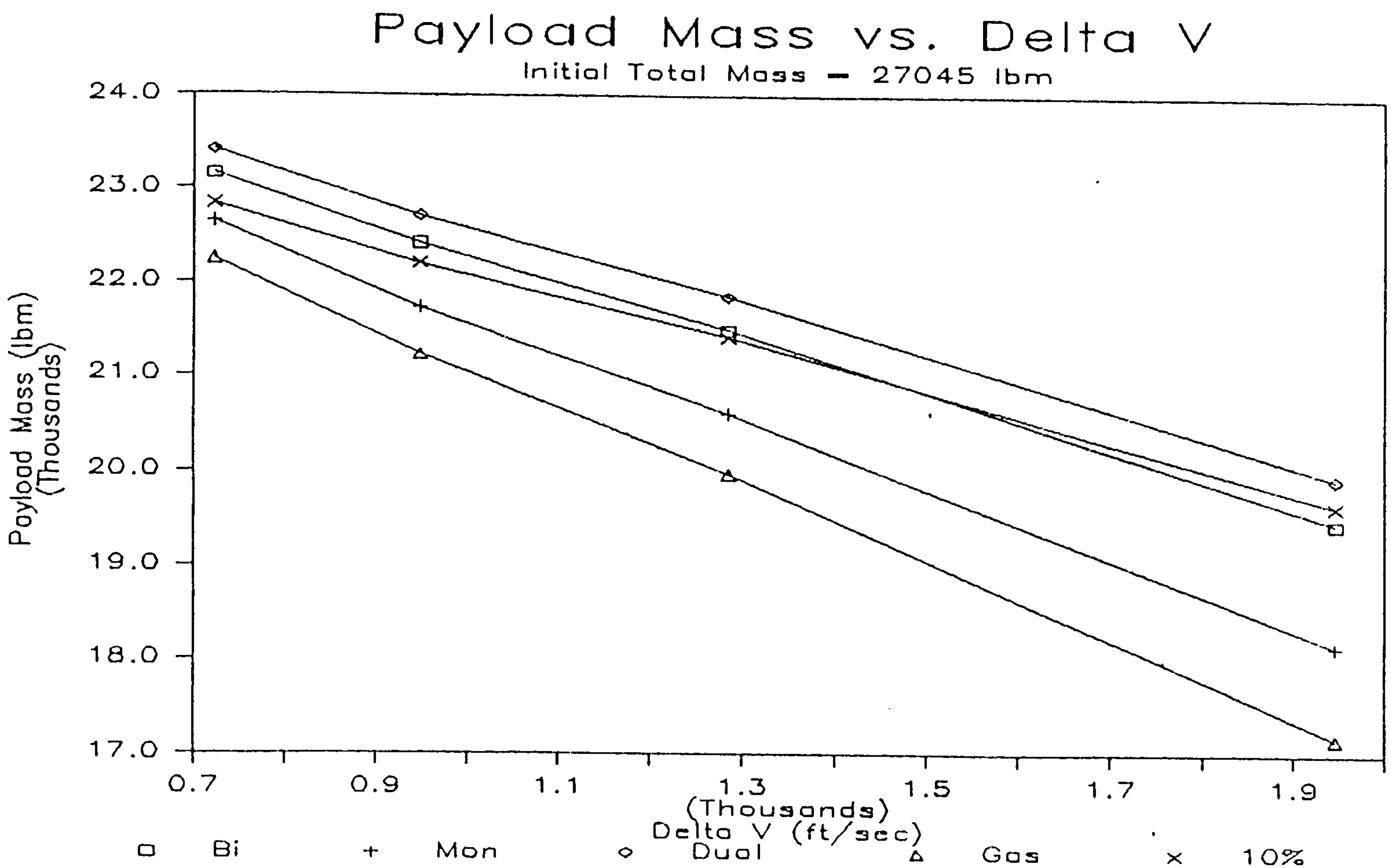

Payload Mass vs. Delta V
Initial Total Mass — 27045 lbm
Payload Mass (lbm)
(Thousands)
24.0
23.0
22.0
21.0
20.0
19.0
18.0
17.0
0.7
0.9
1.1
1.3
1.5
1.7
1.9
(Thousands)
Delta V (ft/sec)
Bi
Mon
Dual
Gas
10%

Figure II-15

based upon the result that large quantities of earth storable propellant would be required, hence lower recurring cost would be encountered for the higher performing cryogenic concept.

The following discussion illustrates why the choice of a high performance propellant combination is necessary for reasonable accommodation of high energy OTV missions. The dry weight of a propulsion system can be expressed by a constant value of hardware weight (independent of total system size) plus a weight of hardware that depends on total propellant loaded. This is expressed as:

$$M_{dry} = A + B\ (M_{prop})$$

where:

M_{dry} = vehicle dry mass

A = constant hardware mass

B = variable hardware factor

M_{prop} = loaded propellant mass

Figure II-16 shows the characteristics of single-stage liquid propulsion vehicles and their limitations. The x-axis is the nondimensional parameter of delta V divided by specific impulse (I_{sp}). The y-axis is the ratio of vehicle and payload initial mass (including propellant) divided by payload mass and vehicle constant hardware mass (A).

Several conclusions may be drawn from the figure. The first is that for a large enough delta V, there is an asymptotic value for a given propellant combination and engine technology (I_{sp}) for which no amount of propellant can deliver any size payload. Therefore, to complete a mission that imposes a delta V above the asymptotic value, either a higher energy propellant must be chosen or the required delta V must be reduced (as with an aerocapture maneuver of some sort). For a low-earth orbit (LEO) to geosynchronous equatorial orbit (GEO) payload delivery with an aerocapture return, the required delta V is about 20,000 fps for a vehicle with a lift to drag ratio (L/D) of zero. Also, a reasonable estimate for B for large propulsion systems is about 0.1 for storable propellants (N_2O_4/MMH) and about 0.125 for LO_2/LH_2 (ref Magnetoplasmadynamic Thruster Definition Study performed for AFRPL, Contract F04611-82-C-0049). So, for a storable propellant combination, the delta V of 20,000 fps places the value of x very

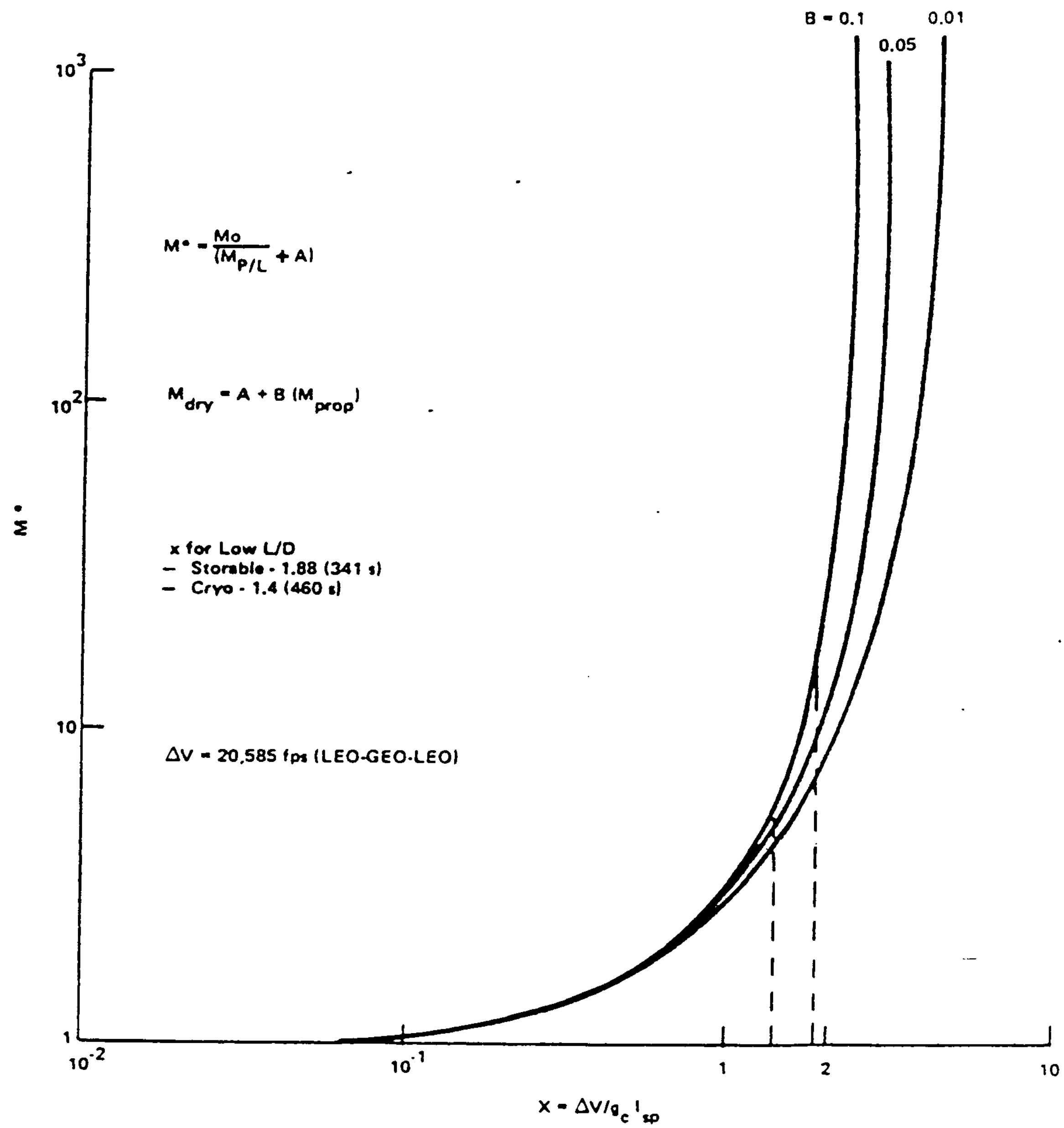


Figure II-16 - OTV Parameters

near the asymptotic value for total vehicle weight. In other words,small savings in delta V and/or increases in I_{sp} result in very great savings in total vehicle mass for a given payload mass. For example, providing a vehicle with a L/D of 1.0 will result in a possible inclination change capability on GEO to LEO return of 14 deg and a delta V savings of only about 800 fps. This, however, is valuable in terms of reducing storable propellant requirements and total size of the resulting vehicle even with a subsequent increase in dry weight because of the higher L/D ratio. Therefore, the higher L/D approach appears necessary for a vehicle with a storable propellant combination in order to achieve a sensible design point for vehicle sizing.

A LO_2/LH_2 propulsion system with an aerobrake and a L/D of zero provides a delta V savings of about 7500 fps when returning from GEO over an all-propulsive system and results in an x value that is further away from the asymptotic value for total vehicle mass. Therefore, the reduction in total vehicle mass is not as dramatic for reductions in delta V by going to L/D ratios greater than zero as for the storable propellant system. This is a result of being on a "flatter" part of the curve relating x and total system mass. For instance, in providing a 800 fps reduction in delta V during return from GEO (via L/D of 1.0), the resulting LO_2/LH_2 total propulsion system mass shows only a reduction of about 5%. This assumes no increase in dry weight as a result of providing the vehicle with a higher L/D capability. Therefore, a mid to high L/D configuration is not recommended for a LO_2/LH_2 vehicle because an inclination change capability does not appear to benefit the system to a great degree. The concept of a low L/D reusable aerobrake, however, is consistent with materials technology evolution for the time frame of interest and the resulting vehicle configuration may be more amenable to component accessibility and servicing.

Present estimates of propellant quantities required for each spacecraft considered in this study are shown in Table II-7 for each of the applicable propellants. The total impulse used for the Space Station, OMV, and platforms are as follows:

Space Station (90 day) - 1,224,000 lb_f-sec
OMV (loaded) - 1,900,000 lb_f-sec
Platforms (loaded) - 1,500,000 lb_f-sec

Table II-7 Candidate Concept Propellant Estimates

PROPELLANT(S)	SPACE STATION	OMV	OTV	PLATFORMS
N_2H_4	5560 lbm	8300 lbm	N/A	6750 lbm includes ACS
N_2O_4/MMH	3950 lbm	6700 lbm and 200 lbm cold gas (N_2) ACS	97500 lbm 1 stage 90800 lbm 2 stages at 2:1 MR	5000 lbm and 900 lbm cold gas (N_2) ACS
N_2H_4/N_2O_4	N/A	6760 lbm includes N_2H_4 ACS	N/A	5260 lbm includes N_2H_4 ACS
O_2/H_2	2900 lbm at MR = 4:1 3200 lbm at MR = 8:1 (electrolysis)	4520 at MR = 4:1 4930 lbm at MR = 8:1 (gaseous)	55000 lbm at MR = 6:1 subcritical liquid	4576 lbm at MR = 8:1 (electrolysis) 3676 at 6.2:1 and 408 sec. 900 lbm O_2 at 67 sec.
Heated H_2 (R-Jet)	2040 lbm	N/A	N/A	N/A
O_2/H_2 S/C	3000 lbm at MR = 6:1 (S/C)	N/A	N/A	4421 lbm at MR = 6:1 (S/C) 3521 at 4.6 and 426 sec. 900 lbm at 67 sec.

Using these total impulse values, the orbit adjust (delta A and/or maneuvering) propellant quantities for the various Space Station element propulsion system candidates were computed. The Space Station on-board propulsion system 90 day impulse used includes that required for ACS also. The propellant amounts shown for OTV are exclusivelv for delta V. For platform propulsion and OMV the delta V propellant amounts are sometimes separated from the ACS (and/or proximity operations) propellant depending upon the choice of delta V propellant(s). For instance, with N_2O_4/MMH use, a cold gas system is also required due to contamination considerations for OMV proximity operations around Space Station and for platform ACS while on-orbit. Therefore, a suitable amount of GH_2 is shown for the OMV and platform cold gas svstems for N_2O_4/N_2H_4 candidates for OMV and platforms is to eliminate the cold gas system and use monopropellant N_2H_4 for the ACS or proximity operations. This extra amount of N_2H_4 is included in the total propellant amount shown.

The O_2/H_2 candidates for Space Station include operations at mixture ratios of 4:1, 6:1, and 8:1. The 4:1 mixture ratio corresponds to a water elecrolysis propulsion system (normally operating at a stoichiomeric mixture ratio of 8:1) integrated with the Space Station ECLSS in order to take advantage of the leftover H_2. A GO_2/GH_2 OMV could feasibly be refueled at Space Station and utilize reactants at the same mixture ratio as the on-board propulsion system. The 6:1 mixture ratio corresponds to the optimum operating point for maximum performance and minimum delivery weight for supercritical O_2/H_2 delivery and storage.

III. TASK II - PROPULSION SYSTEMS CONCEPT SELECTION

The objective of Task II was to select the most attractive propulsion concepts for the space station from the concepts identified in Task I. Criteria used in the selection included: (1) consideration of technology readiness, (2) design, development, test and evaluation costs, (3) life cycle costs, (4) maintainability, (5) reliability, and (6) evolvability. In addition to making the concept selection, a selection of the parameters to complete Task III, Preliminary Design was also to be made. With the revision of the Statement of Work, an objective of selecting the most attractive propulsion systems for each Space Station element/propellant combination and developing descriptions of each of these systems was added to this task.

A. Space Station Propulsion System Final Evaluation/Selection

The Concept Selection Plan was submitted in May, 1985 for approval by NASA LeRC and MSFC. The final approved plan was resubmitted in June, 1985. The system to be carried into the next two tasks was selected using the Concept Selection Plan. System 5-D from the Preliminary Concept Selection task was chosen. This system is shown in Figure III-1 and detailed in Table III-1. The final system ranks are shown in Table III-2. While the recommended system placed third on the absolute ranking used, the gain in actual operational flexibility was felt to justify the choice. As can be seen in the table, the system selection ranks were sensitive to the weighting factors used. Three separate ranking results are shown in the table, the results using the ranking from the Concept Selection Plan are shown first. Upon examination, it was found that complexity, maintenance and reliability were all measuring the same factor and were therefore overly biasing the ranking process. The second two rankings shown were attempts to reduce this overinfluence.

The major features of the system are; large capacity accumulators, low power condensing heat exchangers, and pumps for both fluids. The latest Space Station drag parameters yield a very modest reboost requirement of approximately 350,000 lb_f-sec for the worst case 90 day period. Sizing the supercritical tanks and accumulators each for this capacity will effectively leave the Space Station with a 180 day propellant supply. The accumulators are sized below this maximum amount to first take advantage of the "real time" conditioning capacity available and secondly because the minimum years will not require even this amount of propellant. Besides, the determination of the exact capacity of the tanks will need to await a further stage of Space Station development. Approximate sizes are sufficient to continue with the program.

The operating scenario for the reference reboost scheme begins with the accumulators fully charged when the shuttle visits for Space Station resupply. The empty resupply tanks at the station are then swapped for full ones and the Space Station reboosts. The propellant in the resupply tanks then is moved to the accumulators during the succeeding 90 days. If the Space Station reboost scheme has more frequent reboost periods, the resupply tanks may need a higher degree of thermal protection so the "boiloff" rate from the

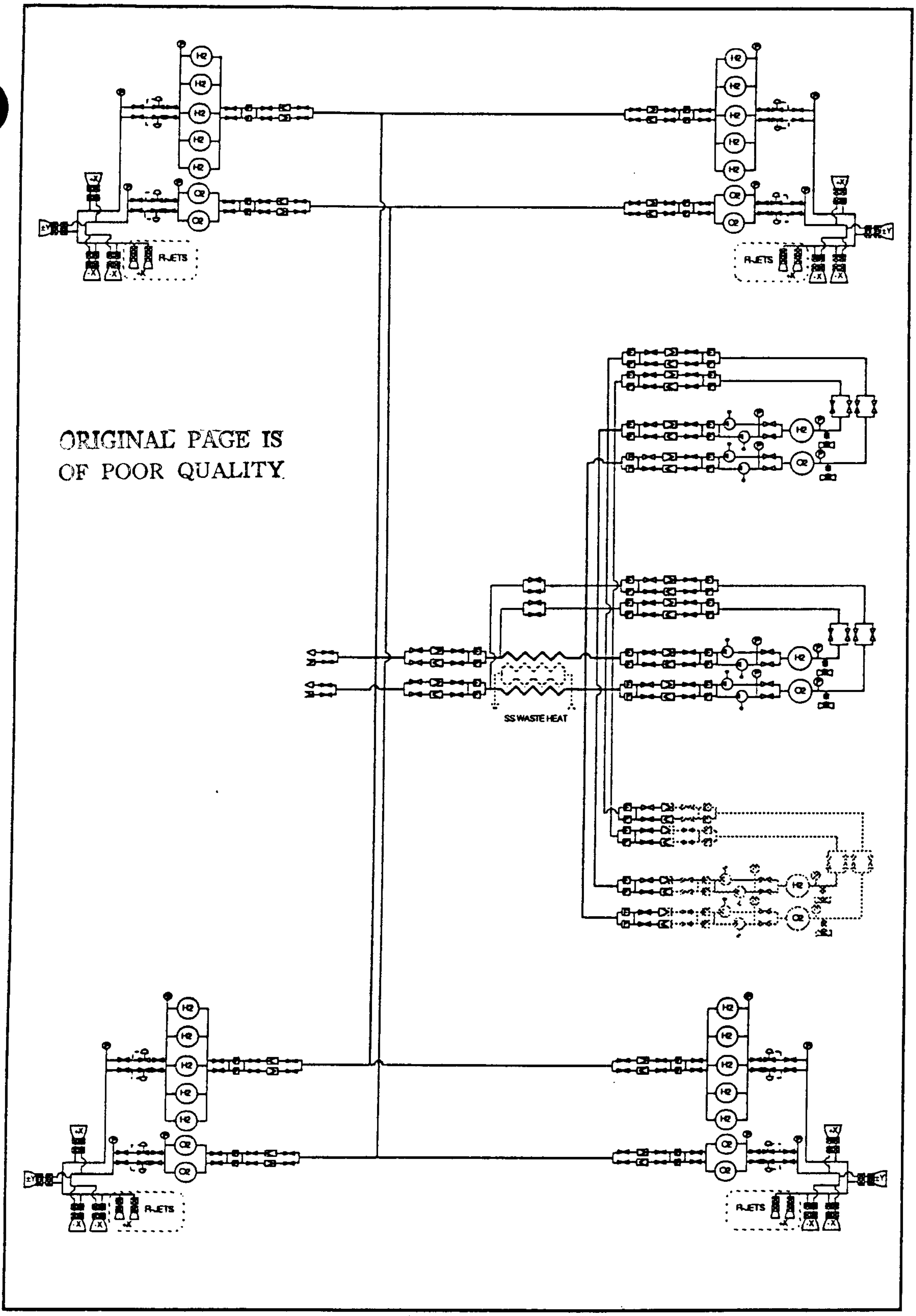


Figure III - 1 - Baseline System Schematic

Table III-1 - Baseline System Design Parameters

INPUT PARAMETERS

Initial altitude = 270 nautical miles starting in year 1993
Reboost mode =Time delta = 90 days
GO_2/GH_2 thruster MR = 6 O/F Chamber Pressure = 150 psia
Thrust level 25 lb_f Specific Impulse 423.5846 lb_f-sec/lb_m
Accumulator TI = 286,000 lb_f-sec/lb_m

	H_2	O_2
Accumulator pressure, psia	1000	1000
Accumulator temperature, °R	450	450
Initial storage temperatures, °R	36	110
Final storage temperatures, °R	150	400
Prop Cond heat exchanger tube dia, inches	.25	.25
Storage Tank heat exchanger tube dia, inches	.5	.5
Number of Accumulators/PM	5	2
Number of Supercritical tanks	3	3

OUTPUT PARAMETERS

Average atmospheric density: 6.57055 e-14 lb_m/ft^3
Days between reboost: 90 Altitude Change: 6.312 nMile
Mass of propellant required for 90 days: 713.35 lb_m
Mass of propellant required for reboost burn: 702.74 lb_m Burn Time = 49.61 minutes

	H_2	O_2	Total
Mass flow rate, lb_m/hr	4.759665	28.55799	33.31765
Pump work,kw	1.195554	1.793286E-02	1.213487
Prop Cond outlet temp, °R	312.8517	441.2144	
Prop Cond Max Q, kw	1.483638	1.35622	2.839858
Storage Init Q, kw	.3957597	1.773241	2.169001
Total Initial Q, kw	1.879397	3.129461	5.008859
Storage Min Q, kw	.1428875	.3011342	.4440217
Storage Fin Q, kw	.5820867	.6022876	1.184374
Storage mass flow rate, lb_m/hr	9.519329	57.11598	66.63531
Prop Cond heat exchanger weight, lb_m	.3984989	.9536706	1.35217
Prop Cond heat exchanger length, ft	1.618198	3.872602	
Prop Cond heat exchanger ΔP, psia	.4506884	.5415571	
Storage heat exchanger weight, lb_m	.2607278	1.869397	2.130125
Storage heat exchanger length, ft	.5073157	3.637413	
Storage heat exchanger ΔP, psia	3.074398E-03	.0144928	
Accumulator volume, ft^3	15.55817	16.02571	439.3691
Accumulator tank dia, in	37.16788	37.53653	
Accumulator weight, lb_m	73.94915	76.1714	2088.354
Supercritical volume, ft^3	7.997017	2.833525	32.49163
Supercritical weight, lb_m	84.00867	52.9396	410.8448
TOTAL WEIGHT, lb_m	1731.668	771.0131	2502.681

Table III-2 - System Selection Rankings

NO.	DESCRIPTION	RANKING: HIGH MAINTAINABILITY & TECHNICAL READINESS	RANKING: HIGH FLEXIBILITY & TECHNICAL READINESS	RANKING: HIGH EVOLVABILITY & FLEXIBILITY
1	BLOWDOWN, 3 KW, SMALL ACCUMULATOR	7	8	9
2	PUMPED, 3 KW, SMALL ACCUMULATOR	4	6 BP	6 BP
3	BLOWDOWN, 2 Ø HX, SMALL ACCUMULATOR	8	7	7
4	BLOWDOWN, 3 KW, LARGE ACCUMULATOR	9 BP	9	8
5	PUMPED, 3 KW, LARGE ACCUMULATOR	6	4	3
6	PUMPED H2, 4 KW, SMALL ACCUMULATOR	5	5	5
7	PUMPED H2, 7 KW, SMALL ACCUMULATOR	2	2	2
8	BLOWDOWN, 13 KW, SMALL ACCUMULATOR	1	3	4
9	PUMPED, 13 KW, SMALL ACCUMULATOR	3	1	1

tanks can be controlled to fit the thruster use rate. The thermal protection for the resupply tanks will need to be traded off against larger accumulators since the concern here is to avoid overpressurizing the supercritical storage tanks. The thermal protection on the resupply tanks needs to be minimized to reduce their launch cost. This trade will be examined during the execution of Task III.

The mass flow rates, heat fluxes and other operating conditions were chosen to allow complete recharging of the accumulators within a 24 hour period. A modest pressure level for the accumulators was used to reduce the pumping power required while yielding a reasonably sized set of accumulators. Five hydrogen and two oxygen accumulators were chosen for the packaging and cost advantages equal size tanks will yield. The individual tanks are plumbed together without isolation to save valve costs. Individual tank isolation is an option that is not strictly required to meet the fail operational/fail safe/fail restorable requirements. Other modules can serve as functional backup units and a tank failure is not a credible failure. The exception to this is a meteoroid penetration which would fail a single tank. Check valves could be installed to guard against the loss of more propellant than that stored in the stricken tank. The number of supercritical tanks was chosen as three each for hydrogen and oxygen so that three resupply modules could be used to capture the variation in resupply quantity as the atmosphere varies. These choices will also receive further attention during the execution of Task III.

The technical effort on Tasks I & II was completed prior to the midterm review at NASA LeRC and MSFC, June 5th &6th, 1985. The recommended system that was to be carried into Tasks III & IV (shown in Fig. III-1) consisted of a low power propellant conditioning system, minimally insulated supercritical storage tanks, and moderate pressure accumulators sized for a "nominal" worst case 90 day period. This system was selected as the best compromise between system complexity and flexibility. The cost and weight differences between systems were not sufficient enough to totally eliminate systems based on those two criteria. Therefore, cost and weight proved unusable as discriminators between competitive systems. An additional consideration in the selection of the recommended system was the desire to retain a representative example of all the components which may be used in a flight design. This would allow the simulation program of Task IV to retain its usefulness as the propulsion system design matures.

B. OMV Final Evaluation & System Design

Six candidate systems were defined for the Orbital Maneuvering Vehicle. These included: a bipropellant system (N_2O_4/MMH) with a cold gas nitrogen system for proximity operations, a dual mode bipropellant system (N_2O_4/N_2H_4) with hydrazine RCS thrusters for proximity operations, a hydrazine system containing sufficient propellant for proximity operations, a gaseous O_2H_4 system with all the required propellant in accumulators, a GO_2/GH_2 system with 10% accumulators filled by an on-board electrolysis system, and a supercritical O_2/H_2 system with the propellant conditioning system on-board. It was assumed that the oxygen/hydrogen would produce no contamination and could be used for proximity operations. All candidates were initially sized using a total impulse of 1.9×10^6 lbf-sec for the first iteration of propulsion system sizing. The valving for all options was placed to provide a fail operational/fail safe system. Each candidate was equipped with four 100 lbf orbit adjust thrusters and sixteen 5 lbf reaction control system thrusters. A rough structure weight was developed using weights from previous programs and proportioning it to the number and size of the tanks. The same controller was used for all systems. Schematics and weight statements for all the options have been included in this report. Table III-3 provides a symbol key for the schematics. Each weight statement is for a dry propulsion unit. An additional 1436 lbm. (electrical power system, video, etc.) plus the weight of the propellant and pressurant must be added to this weight to obtain a "full up" OMV.

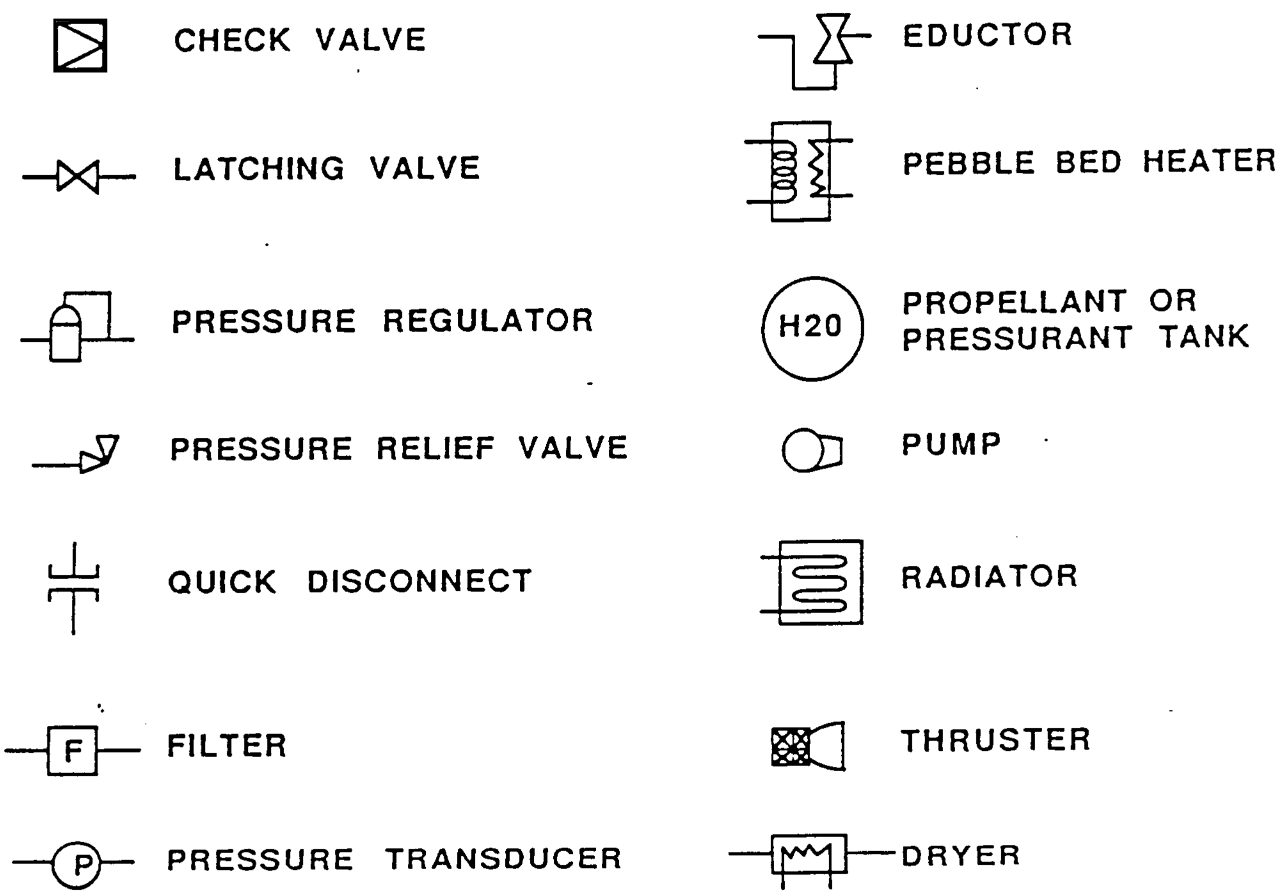
The description of each system follows:

Bipropellant (N_2O_4/MMH) - N_2 Cold Gas For Proximity Operations

Ref: Figure III-2 (Schematic), Table III-4 (Weight Statement)

This system uses the tanks described in the OMV Preliminary Definition Study. The tanks were sized for 6700 lbm of propellant and include a propellant management system. There are four equal volume oblate spheroid tanks: 23 in semi-major axis with 1.414 semi-major to semi-minor axis ratio and a 3.6 in cylindrical section. A 5% ullage fraction and a 2% residual is

Table III-3 - Schematic Symbol Key



OMV N2O4/MMH SYSTEM

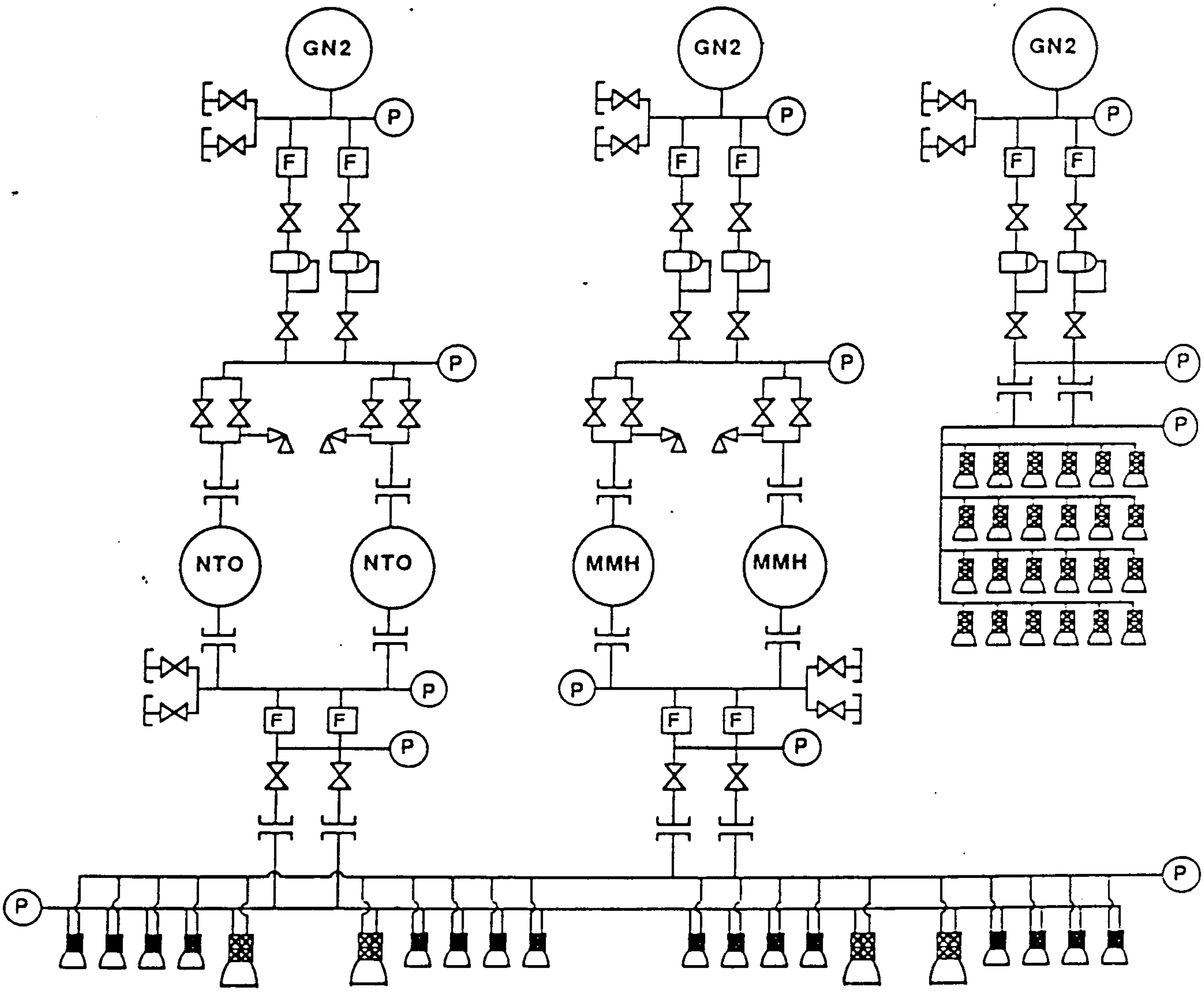


Figure III-2

Table III-4 - OMV N2O4/MMH System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
MMH Tank	2	142.5	285.0
N2O4 Tank	2	142.5	285.0
GN2 Tank (Pressurant)	2	160.0	320.0
GN2 Tank (Cold Gas RCS)	1	120.0	120.0
Latching Valve	8	3.0	24.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	11	0.5	5.5
Quick Disconnects (Halves)	24	3.0	72.0
Propellant Filter	4	1.0	4.0
Temperature Sensors	28	0.5	14.0
Propellant Distribution System			
Lines(length in feet)	52	0.25	13.0
Joints	-	-	-
Thermal Conditioning System			
Heat Trace(length in feet)	52	0.08	4.16
Thermostats/Wiring	16	0.22	3.52
Insulation(length in feet)	52	0.04	2.08
Secondary Structure	1	700.0	700.0
Controller	1	435.0	435.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Cold Gas RCS (5 lbf)	24	1.5	36.0
Pressurization System			
Quick Disconnects (Halves)	8	3.0	24.0
Lines(length in feet)	16	0.25	4.0
Latching Valves	20	3.0	60.0
Pressure Regulators	4	5.0	20.0
Pressurant Filter	4	0.5	2.0
Cold Gas RCS System			
Quick Disconnects (Halves)	6	3.0	18.0
Lines(length in feet)	12	0.25	3.0
Latching Valves	6	3.0	18.0
Regulators	2	5.0	10.0
Filters	2	0.5	1.0

		Propulsion Unit Dry Weight	2560.5

included. The total mass of usable propellant is equal to 6566 lbm. The mixture ratio was assumed to be 1.65 to provide equal propellant volumes. The propellant tank operating pressure is 400 psia and it was sized using a factor of safety of two. The nitrogen pressurization system includes two tanks operating at 4000 psia and 520°R. The system provides propellant to the thrusters at 250 psia. An additional nitrogen tank provides the nitrogen for the cold gas system. This tank also operates at 4000 psia and 520°R. This option includes 24 - 5 lbf cold gas thrusters for proximity operations.

Dual Mode Bipropellant (N_2O_4/N_2H_4) - Hydrazine Proximity Operations
Ref: Figure III-3, Table III-5.

This OMV option uses the same tanks as the bipropellant system for commonality and since the required propellant is approximately the same. A mixture ratio of 1.43 was used to produce equal tank volumes with minimal performance degradation from the maximum Isp mixture ratio of about 1.0. Additional hydrazine (60 lbm) is required for proximity operations. There is sufficient volume in the reference tanks to provide for this requirement. The 24-5 lbf hydrazine thrusters provide the RCS for proximity operations. Elimination of the cold gas system reduces the system dry weight by 155.5 lbm. The size of the nitrogen pressurization system has been slightly increased to provide for the additional hydrazine requirement.

Hydrazine
Ref: Figure III-4, Table III-6

The hydrazine system is very similar to the dual mode system with the exception of incorporating a common pressurization system rather than a system for each propellant. The tanks have been resized for the mass of hydrazine required for the mission. Spherical tanks were sized using a factor of safety of two and additional mass was included for the propellant management system. The 5% ullage and 2% residual figures were used to determine the total volume of the tank. The 16 - 5 lbf RCS thrusters were assumed to perform all proximity operations.

OMV N2O4/N2H4 SYSTEM

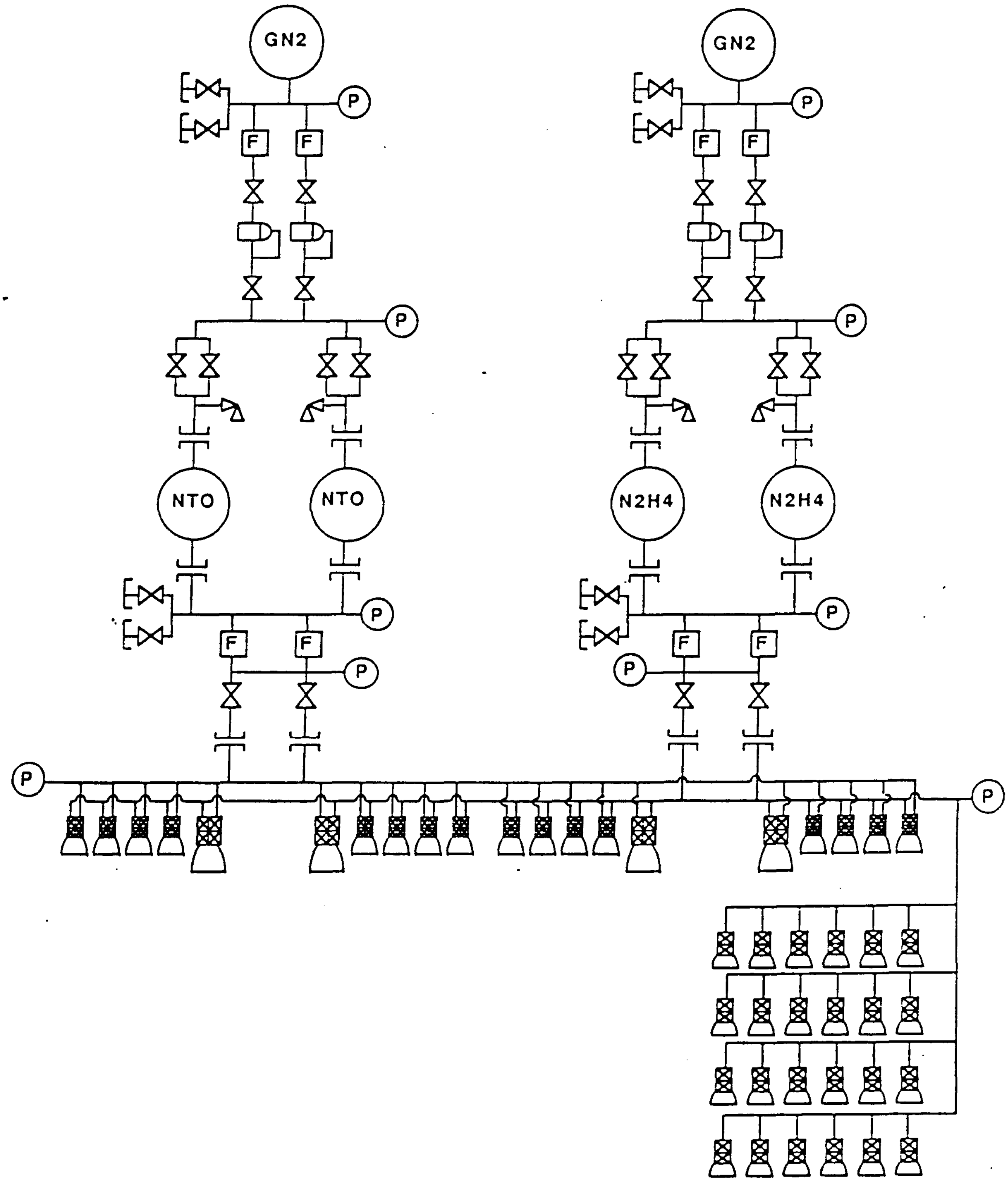


Figure III-3

Table III-5 - OMV N2O4/N2H4 System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
N2H4 Tank	2	142.5	285.0
N2O4 Tank	2	142.5	285.0
GN2 Tank	2	180.0	360.0
Latching Valve	8	3.0	24.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	10	0.5	5.0
Quick Disconnects (Halves)	24	3.0	72.0
Propellant Filter	4	1.0	4.0
Temperature Sensors	24	0.5	12.0
Propellant Distribution System			
Lines(length in feet)	52	0.25	13.0
Joints	-	-	-
Thermal Conditioning System			
Heat Trace(length in feet)	52	0.08	4.16
Thermostats/Wiring	16	0.22	3.52
Insulation(length in feet)	52	0.04	2.08
Secondary Structure	1	675.0	675.0
Controller	1	435.0	435.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Hydrazine RCS (5 lbf)	24	1.5	36.0
Pressurization System			
Quick Disconnects (Halves)	8	3.0	24.0
Lines(length in feet)	16	0.25	4.0
Latching Valves	20	3.0	60.0
Pressure Regulators	4	5.0	20.0
Pressurant Filter	4	0.5	2.0
		Propulsion Unit Dry Weight	2403.0

OMV N2H4 SYSTEM

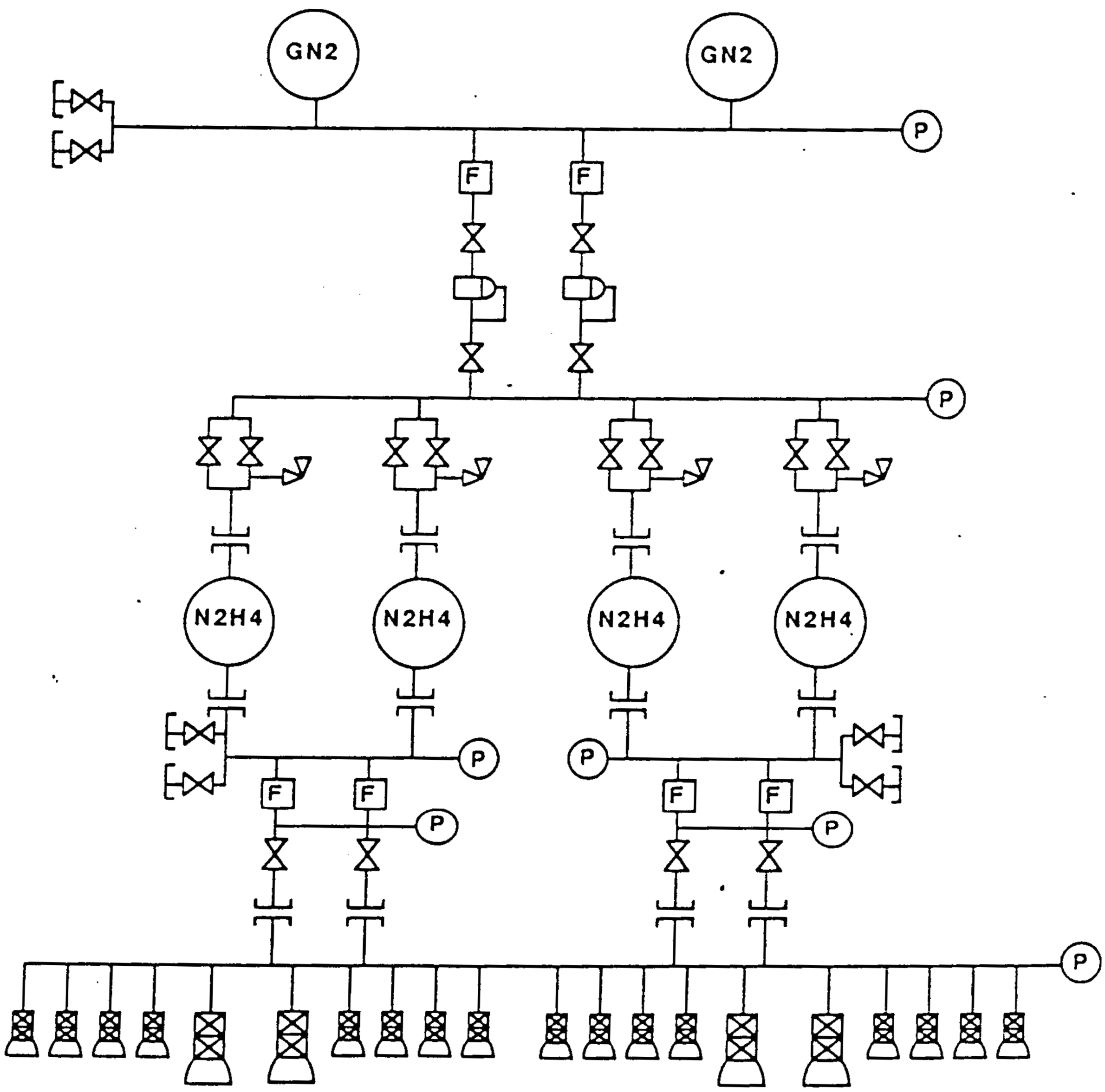


Figure III-4

Table III-6 - OMV Hydrazine System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
N2H4 Tank	4	185.0	740.0
GN2 Tank	2	275.0	550.0
Latching Valve	8	3.0	24.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	7	0.5	3.5
Quick Disconnects (Halves)	24	3.0	72.0
Propellant Filter	4	1.0	4.0
Temperature Sensors	24	0.5	12.0
Propellant Distribution System			
Lines(length in feet)	48	0.25	12.0
Joints	-	-	-
Thermal Conditioning System			
Heat Trace(length in feet)	48	0.08	3.84
Thermostats/Wiring	16	0.22	3.52
Insulation(length in feet)	48	0.04	1.92
Secondary Structure	1	675.0	675.0
Controller	1	435.0	435.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Pressurization System			
Quick Disconnects (Halves)	6	3.0	18.0
Lines(length in feet)	10	0.25	2.5
Latching Valves	14	3.0	42.0
Pressure Regulators	2	5.0	10.0
Pressurant Filter	2	0.5	1.0

		Propulsion Unit Dry Weight	2687.5

GO_2/GH_2 - 100% Accumulators
Ref: Figure III-5, Table III-7

This system was sized such that the amount of oxygen/hydrogen provided by the accumulators is sufficient for all aspects of the OMV mission. The tanks were sized using a 6:1 mixture ratio in order to achieve maximum vehicle performance by balancing high engine performance at lower mixture ratios against lighter weight tankage and structure at higher mixture ratios. The propellant is stored at 2000 psia and 450°R. Composite overwrapped inconel tanks were sized for this option. This was done by sizing an inconel tank and then reducing the weight by 20% to account for the overwrap. The 20% number was obtained from several tanks sized by tank manufacturers. The large tanks (2 - GH_2:9.8 ft. diameter, 2 - GO_2:6.5 ft. diameter) present a packaging problem but additional tanks would drive the weight of the system even higher.

GO_2/GH_2 - Electrolysis - 10% Accumulators
Ref: Figure III-6, Table III-8

In this option, the thrust for the OMV is provided by O_2/H_2 which is stored in accumulators containing 10% of the total impulse required for the highest energy OMV mission. After the accumulators are emptied the spacecraft coasts until the accumulators are recharged by the on-board electrolysis unit. The overwrapped inconel accumulators were sized using a safety factor of 1.5. The smaller accumulators are lighter than those used by 100% accumulator option. However, the weight of the electrolysis unit and the solar panels required to power it more than make up for the weight savings. Table III-9 shows the amount of added solar panel weight required for various duty cycles for the electrolysis unit. The system was sized for an 8:1 mixture ratio to make complete use of the generated propellant. The electrolysis unit provides GO_2/GH_2 to the accumulator at 2000 psia and 450°R. A radiator was sized to eliminate heat in the propellant generated by the electrolysis unit. Vapor cells and dryers are used to remove any remaining water vapor from the propellant. A minimum gage titanium tank was sized for carrying the water.

OMV GH2/GO2 SYSTEM

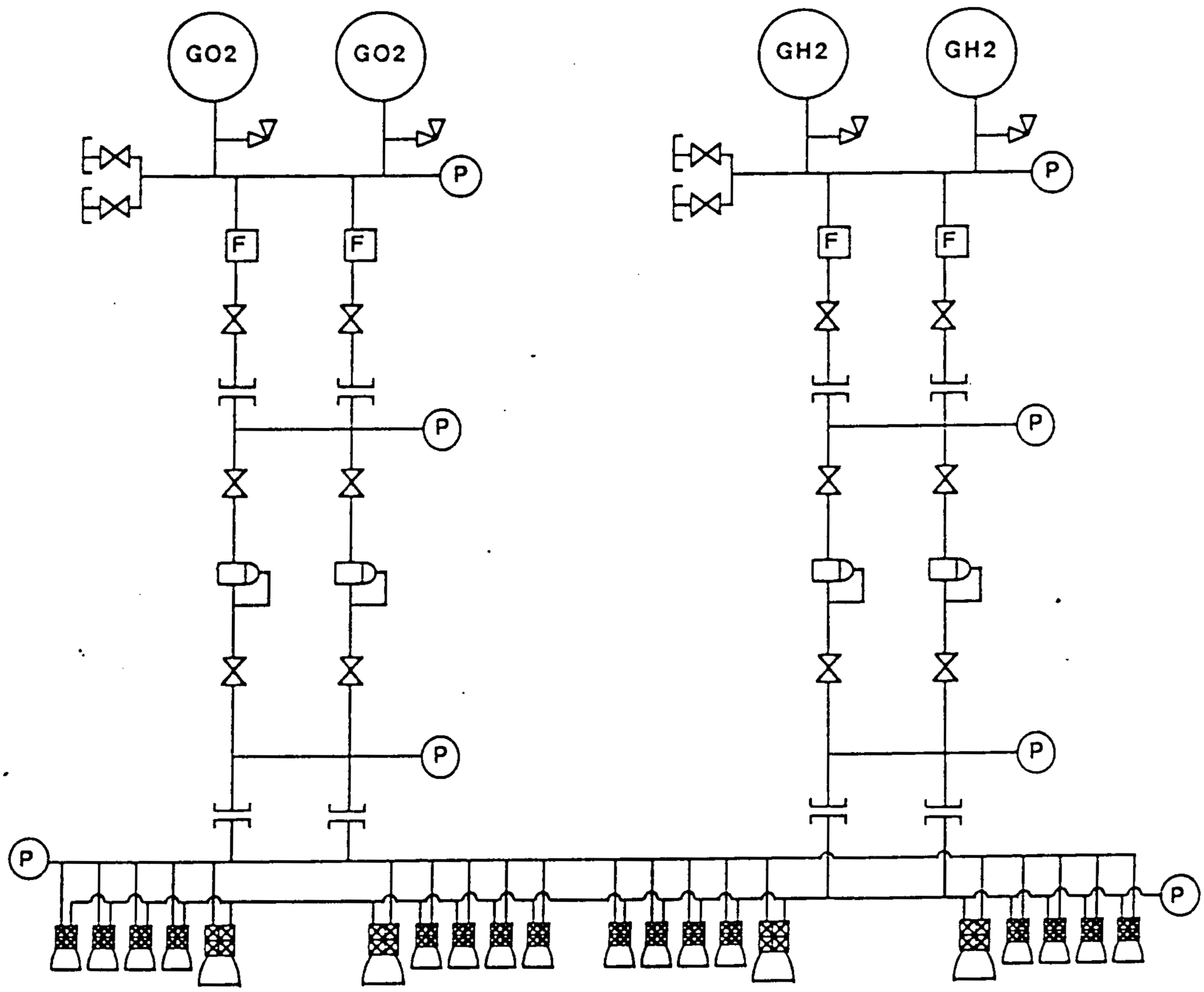


Figure III-5

Table III-7 - OMV GO2/GH2 System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL_WT
GH2 Tank	2	5700.0	11400.0
GO2 Tank	2	1850.5	3700.0
Latching Valve	16	3.0	48.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	8	0.5	4.0
Quick Disconnects (Halves)	20	3.0	60.0
Propellant Filter	4	0.5	2.0
Temperature Sensors	32	0.5	16.0
Pressure Regulator	4	5.0	20.0
Propellant Distribution System			
Lines(length in feet)	100	0.25	25.0
Joints	-	-	-
Secondary Structure	1	700.0	700.0
Controller	1	435.0	435.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0

		Propulsion Unit Dry Weight	16487.2

OMV ELECTROLYSIS SYSTEM
10% ACCUMULATORS

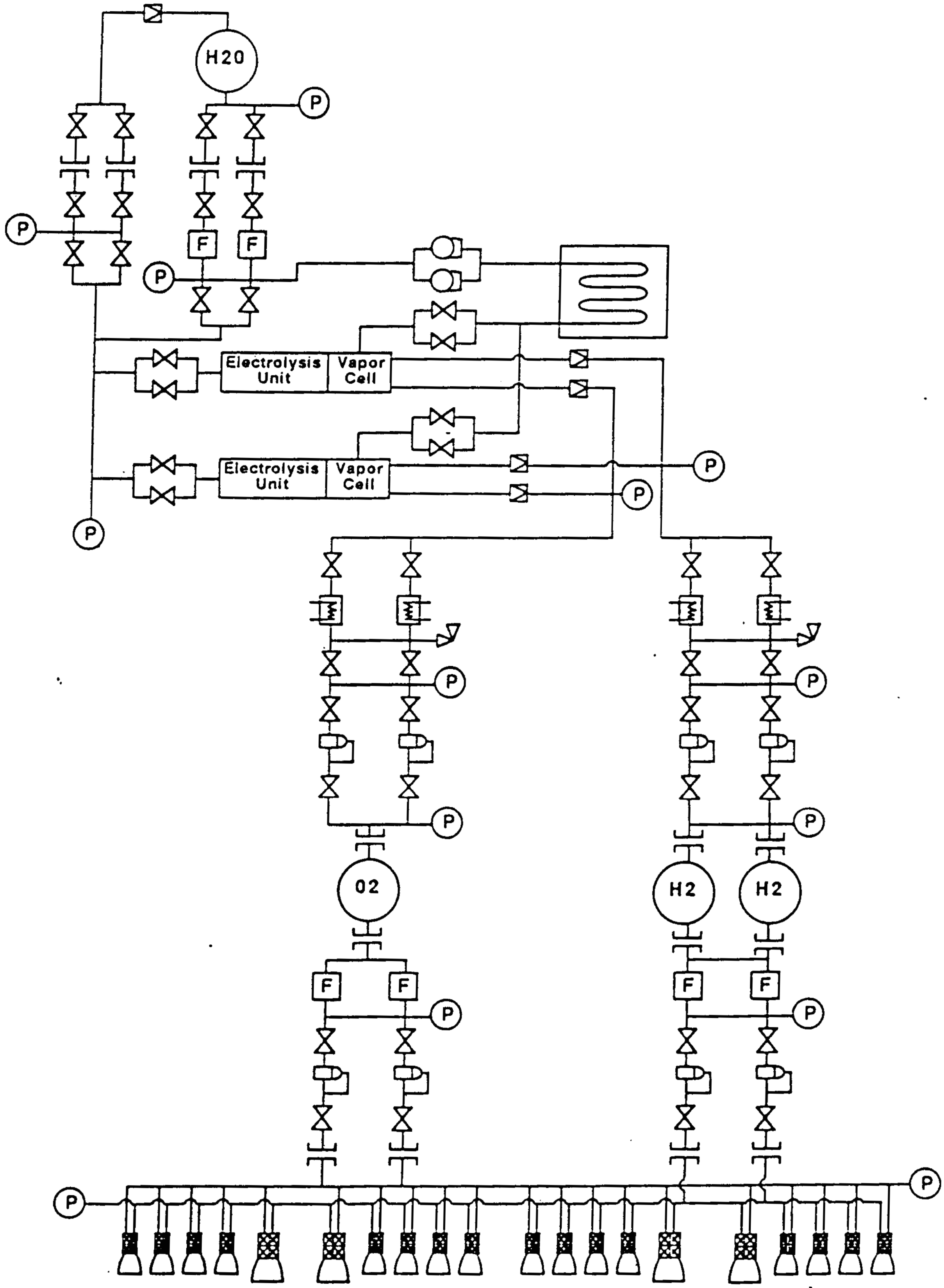


Figure III-6

Table III-8 - OMV Electrolysis System - 10% Accumulators
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
GH2 Tank	2	587.6	1175.2
GO2 Tank	1	463.1	463.1
H2O Tank	1	62.0	62.0
Latching Valve	44	3.0	132.0
Pressure Relief Valve	2	5.0	10.0
Pressure Transducer	16	0.5	8.0
Quick Disconnects (Halves)	28	3.0	84.0
Propellant Filter	6	0.5	3.0
Temperature Sensors	32	0.5	16.0
Pressure Regulator	8	5.0	40.0
Check Valves	5	1.0	5.0
Electrolysis Unit	2	182.0	364.0
Radiator	2	43.0	86.0
Pump	2	10.0	20.0
Desicators	4	24.0	96.0
Propellant Distribution System			
Lines(length in feet)	100	0.25	25.0
Joints	-	-	-
Secondary Structure	1	750.0	750.0
Controller	1	435.0	435.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Thermal Conditioning System			
Heaters	4	5.0	20.0
		Propulsion Unit Dry Weight	3851.5

Table III-9 - Required Solar Panel Mass For Various Electrolyzer Duty Cycles
OMV Electrolysis System - 10% Accumulators

POWER_REQUIRED	DUTY_CYCLE	SOLAR_PANEL_MASS
0.5 kw	2814.4 hr/117.3 days	69.4 lbm
5 kw	281.4 hr/11.7 days	694.5 lbm
25 kw	56.3 hr/2.3 days	3472.4 lbm
50 kw	28.1 hr/1.2 days	6944.7 lbm
100 kw	14.1 hr	13889.4 lbm
200 kw	7.0 hr	27778.8 lbm
500 kw	2.8 hr	69447.0 lbm

GO_2/GH_2 - Supercritical

Ref: Figure III-7, Table III-10

The Aerojet supercritical thermal conditioning concept was applied to the OMV for this candidate. This option circulates propellant through a pebble bed recirculation heater for initial conditioning and then passes it through a pebble bed conditioning heater for delivery to the thrusters. The pebble beds were sized, assuming the amount of material varied linearly with the amount of propellant conditioned. A rough estimate was made for the mass of the container required to hold the bed material. The propellant tanks were sized using the PRSA tanks as a guide. The inner tank consisted of inconel 718 and the outer shell consisted of aluminum 2219. The pebble beds are to be charged prior to the OMV mission. The length of the mission would be limited by the amount of time the pebble beds could hold their charge.

C. PLATFORMS FINAL EVALUATION AND SYSTEM DESIGN

Six propulsion systems were also studied for the platforms. These included: a bipropellant system (N_2O_4/MMH) with a cold gas nitrogen system for on orbit ACS and roll control during orbit adjust, a dual mode bipropellant system (N_2O_4/N_2H_4) with hydrazine RCS thrusters for proximity work, a hydrazine system, a cold gas O_2/H_2 system with 100% accumulators, a GO_2/GH_2 system in which the platform would be delivered by the OMV and returned using an electrolysis unit and 10% accumulators, and a GO_2/GH_2 system using an electrolysis unit and 10% accumulators for the entire mission. The total impulse required for the platforms was assumed to be 1.5 x 10^6 lbf - sec. The valving for all options was placed to provide a fail operational/fail safe system. Each candidate is equipped with four - 100 lbf thrusters (OA) and sixteen - 5 lbf thrusters (RCS). The structure weight was again calculated from information from previous programs. The platform requires a smaller controller since it does not perform as many functions as an OMV controller.

The performance of each candidate platform propulsion system was analyzed with the aid of a microcomputer spreadsheet program. The spreadsheet model was developed to iteratively size the propulsion stages to meet specific delta velocity requirements while keeping one of several system masses constant. These system masses included the following:

OMV - SUPERCRITICAL SYSTEM

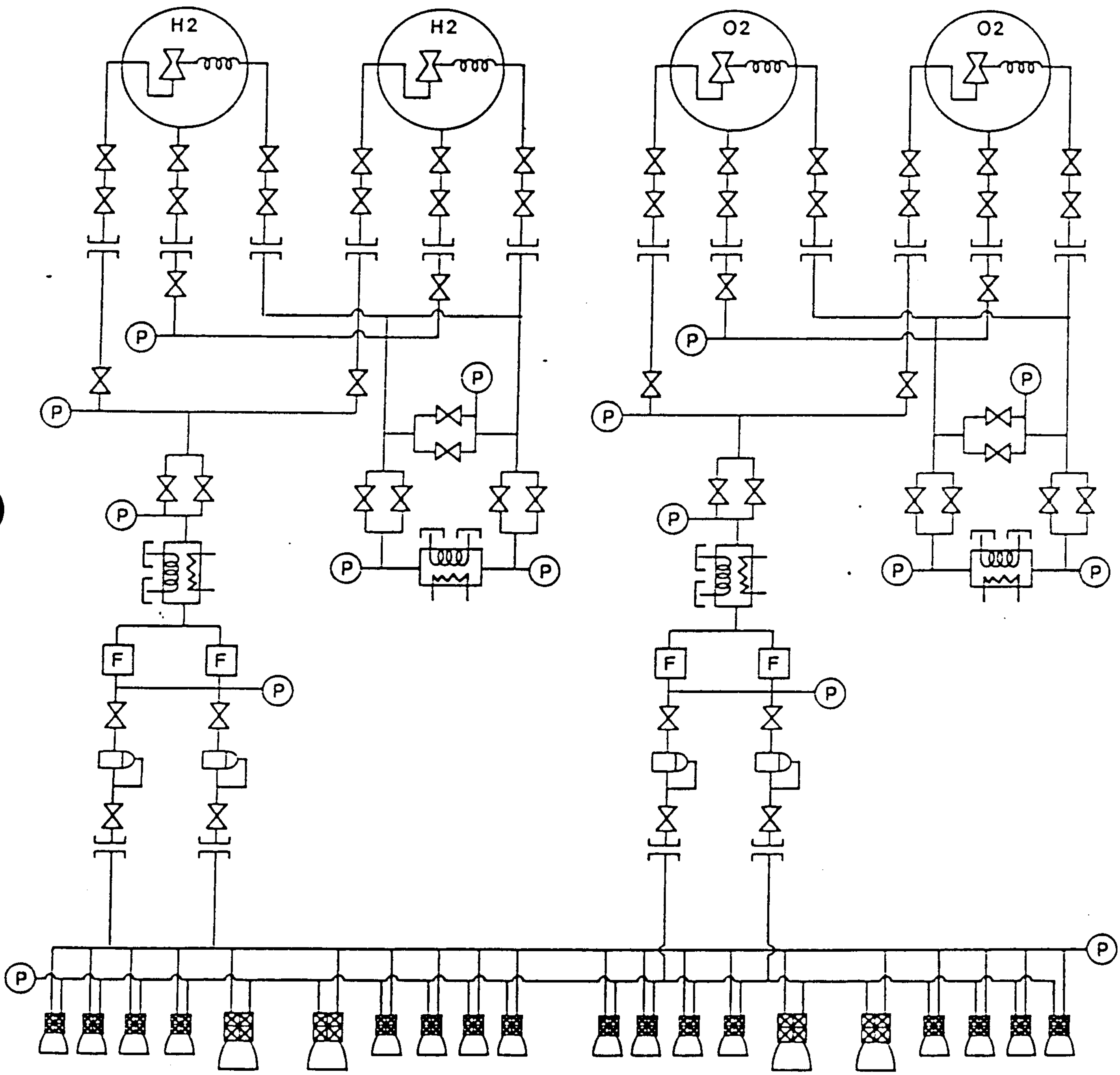


Figure III-7

Table III-10 - OMV Supercritical System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL_WT
Supercritical H2 Tank	2	1185.0	2370.0
Supercritical O2 Tank	2	470.0	940.0
Latching Valve	56	3.0	168.0
Pressure Transducer	16	0.5	8.0
Quick Disconnects (Halves)	40	3.0	120.0
Propellant Filter	4	0.5	2.0
Temperature Sensors	40	0.5	20.0
Pressure Regulator	4	5.0	20.0
Propellant Distribution System			
Lines(length in feet)	100	0.25	25.0
Joints	-	-	-
Secondary Structure	1	800.0	800.0
Controller	1	435.0	435.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Thermal Conditioning System			
Pebble Bed Recir Heater O2	1	1120.0	1120.0
Pebble Bed Recir Heater H2	1	340.0	340.0
Pebble Bed Cond Heater O2	1	1200.0	1200.0
Pebble Bed Cond Heater H2	1	1300.0	1300.0
Insulation(length in feet)	50	0.1	5.0

		Propulsion Unit Dry Weight	8930.2

1. Total initial system mass including payload, propulsion system, and propellant,
2. Payload mass,
3. Resupply fluid mass,
4. Total resupply mass, and
5. Propulsion system dry mass.

It was assumed that the dry mass of each propulsion stage consisted of a fixed mass and a variable mass. Thrusters, avionics, fluid distribution, and structure comprised the fixed mass whereas propellant, pressurization, and ACS tanks made up the variable mass. Since a wide variety of propulsion systems were being considered, no single algorithm could be used to perform all of the parametric analyses, so each system was treated individually. The specific assumptions regarding each propulsion system and its resupply options are listed below.

1. Bipropellant with cold GN_2 ACS

   The specific impulses of the N_2O_4/MMH and cold GN_2 were 310 and 70 lbf-sec/lbm, respectively. The low Isp of the GN_2 would require a large amount of ACS propellant which would, in turn, require large ACS tankage. This seemed to be an unreasonable penalty for this concept, so it was assumed that the GN_2 system would be sized for 20% of the total ACS requirement with a bipropellant system accounting for the remaining 80%. Resupply of this system was accomplished by changeout of the pressurization and GN_2 ACS tanks, whereas the bipropellant tanks were refilled to minimize the resupply dry mass.

2. Monopropellant

   The specific impulse of this system was 230 lbf-sec/lbm. Propellant and pressurant resupply were the same as described for the bipropellant system, except that there is no separate ACS resupply for the monopropellant system.

3. Dual mode N_2O_4/N_2H_4

The specific impulses for the $N_2O_4/N_2/H_4$ and the N_2H_4 systems were 312 and 230 lbf-sec/lbm respectively. Propellant and pressurant resupply were the same as described for the bipropellant and monopropellant systems.

4. Cold GO_2/GH_2

The specific impulse for this system was 412 lbf-sec/lbm, which was obtained by using a 6:1 mixture ratio. This system was resupplied from PRSA type tanks, so the resupply dry mass for this system was larger than for the other systems.

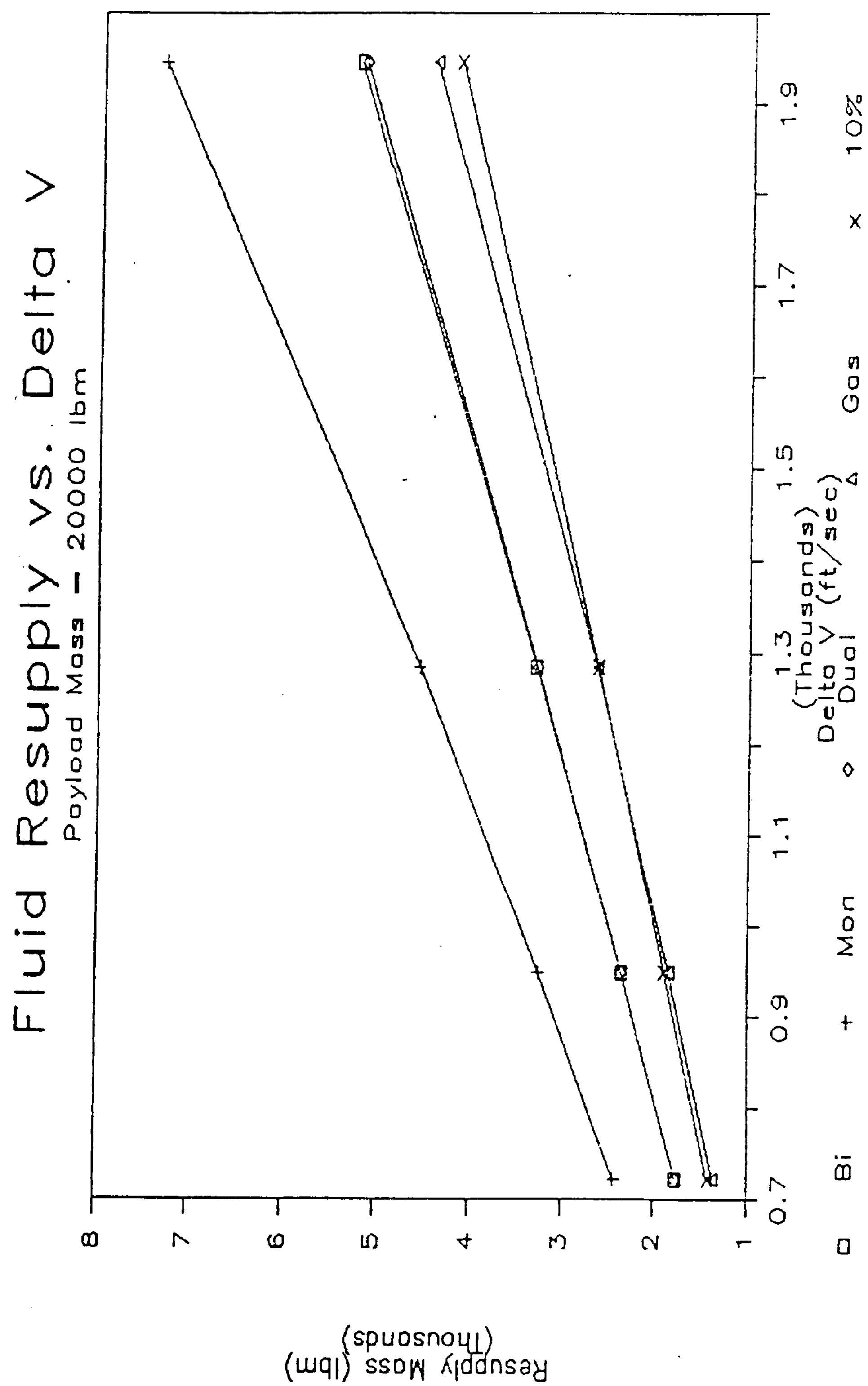
5. Electrolysis GO_2/GH_2

The specific impulse for this system was 385 lbf-sec/lbm, which was a consequence of the 8:1 mixture resulting from electrolysis. This system was resupplied with water delivered from minimum gage resupply tanks.

6. OMV Delivery of Electrolysis System

This system requires that the electrolysis system provide propellant for half of the total delta velocity, with the OMV providing the other half. Consequently, the dry mass of the electrolysis system was reduced while the combined dry mass of the OMV/elecrolysis system was increased. The OMV was resupplied similarly to the bipropellant system.

To determine the resupply requirements for each system the payload mass was held constant at 20000 lbm. Propellant resupply quantities required for each system for the different delta velocities are shown in Figure III-8. Another graph was prepared showing the complete resupply picture (Figure III-9). The mass shown in this figure includes the propellant mass, the tanks used for transporting the propellant, and the associated hardware used to fill



Fluid Resupply vs. Delta V
Payload Mass = 20000 lbm
Resupply Mass (lbm)
(Thousands)
Delta V (ft/sec)
(Thousands)
0.7
0.9
1.1
1.3
1.5
1.7
1.9
1
2
3
4
5
6
7
8
□ Bi
+ Mon
◇ Dual
△ Gas
× 10%

Figure III-8

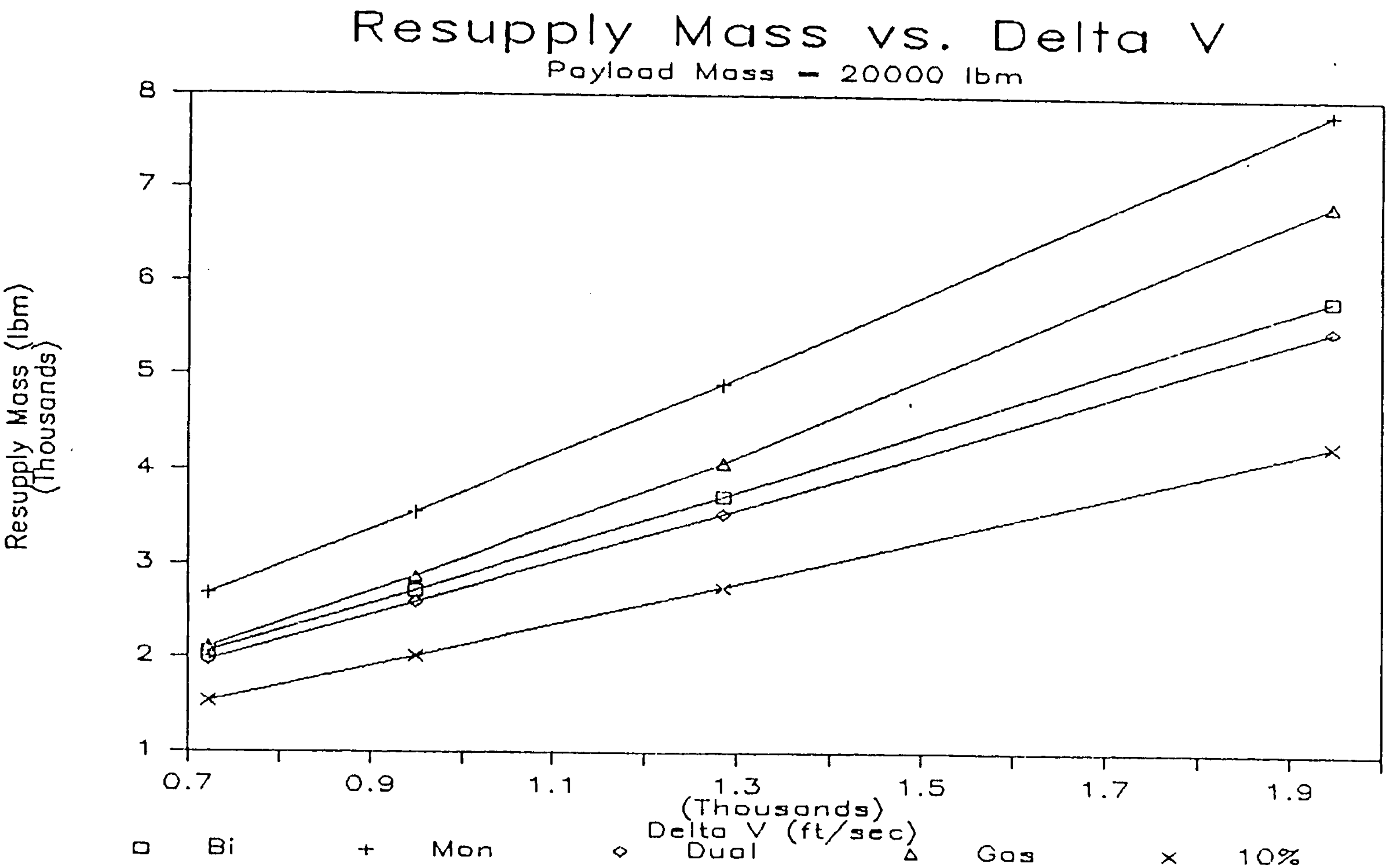

Resupply Mass vs. Delta V
Payload Mass – 20000 lbm
Resupply Mass (lbm)
(Thousands)
8
7
6
5
4
3
2
1
0.7
0.9
1.1
1.3
1.5
1.7
1.9
(Thousands)
Delta V (ft/sec)
□ Bi
+ Mon
◇ Dual
△ Gas
× 10%

Figure III-9

the on-board platform tanks. The resupply module for all the fluid propellants (hydrazine, bipropellant, water, etc.) consisted of minimum gage tanks with a pumping system for filling the on-board tanks and providing the proper pressure. The cold gas system was resupplied using supercritical oxygen and hydrogen brought up in PRSA type tanks. Making these assumptions resulted in resupply modules which were lighter than changing out an entire propulsion module.

Schematics and weight statements have been included for all options. A description of each system follows.

Bipropellant (N_2O_4/MMH) - Cold Gas ACS
Ref: Figure III-10, Table III-11

This system uses the same hardware incorporated in the OMV bipropellant option for commonality. The only difference in the systems would be that the platform would carry less propellant due to its lower required total impulse.

Dual Mode Bipropellant (N_2O_4/MMH) - Hydrazine ACS
Ref: Figure III-11, Table III-12

Again for commonality purposes, this system uses the same hardware used by the OMV dual mode option discussed earlier. Once again the platform would carry less propellant than the comparable OMV option.

Hydrazine
Ref: Figure III-12, Table III-13

The hydrazine system was sized exactly the same as the OMV hydrazine option. It also would carry less propellant than the hydrazine OMV propulsion system concept.

PLATFORM N2O4/MMH SYSTEM

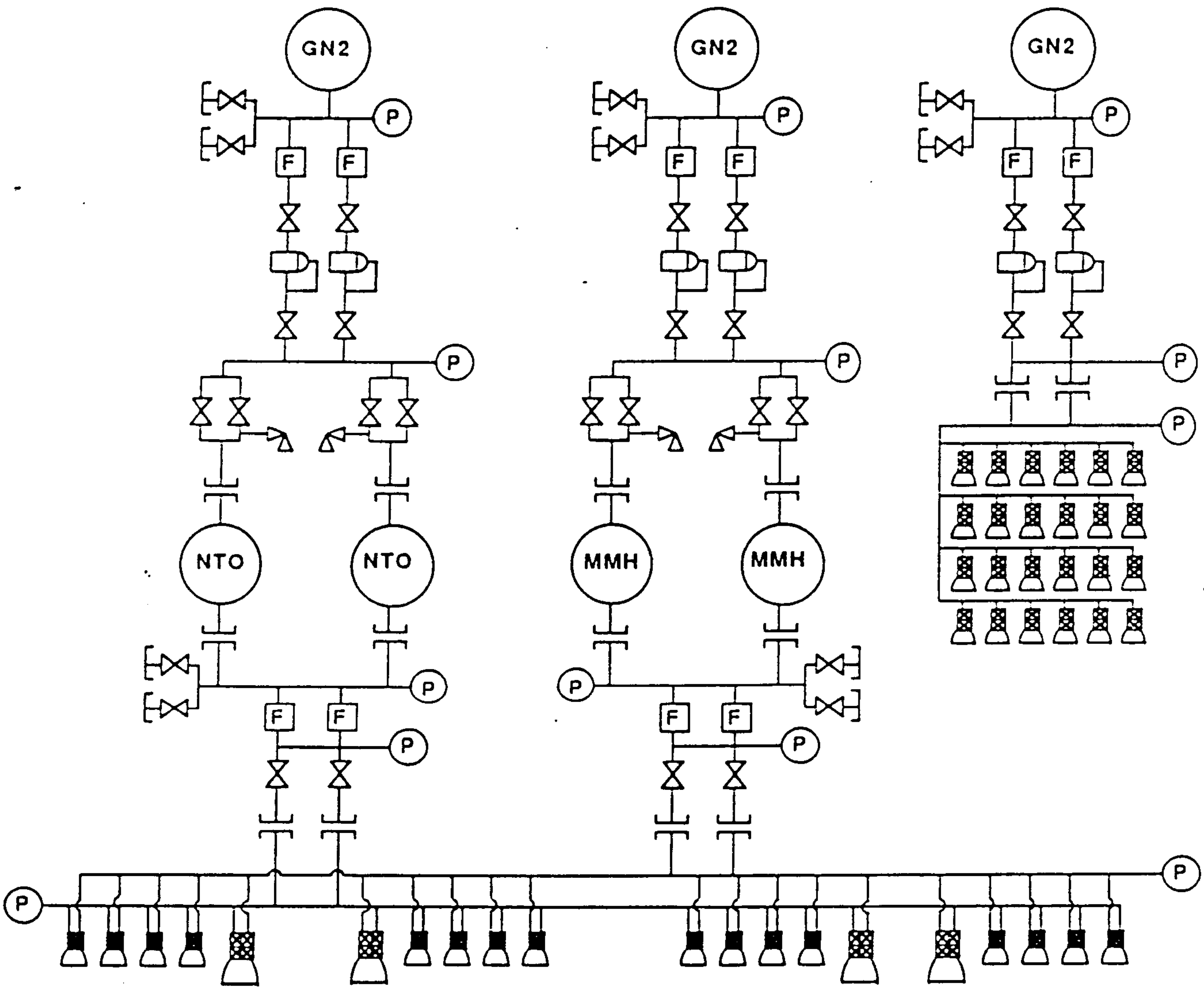


Figure III-10

Table III-11 - Platform N2O4/MMH System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL_WT
MMH Tank	2	142.5	285.0
N2O4 Tank	2	142.5	285.0
GN2 Tank (Pressurant)	2	160.0	320.0
GN2 Tank (Cold Gas RCS)	1	120.0	120.0
Latching Valve	8	3.0	24.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	11	0.5	5.5
Quick Disconnects (Halves)	24	3.0	72.0
Propellant Filter	4	1.0	4.0
Temperature Sensors	28	0.5	14.0
Propellant Distribution System			
Lines(length in feet)	52	0.25	13.0
Joints	-	-	-
Thermal Conditioning System			
Heat Trace(length in feet)	52	0.08	4.16
Thermostats/Wiring	16	0.22	3.52
Insulation(length in feet)	52	0.04	2.08
Secondary Structure	1	700.0	700.0
Controller	1	300.0	300.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Cold Gas RCS (5 lbf)	24	1.5	36.0
Pressurization System			
Quick Disconnects (Halves)	8	3.0	24.0
Lines(length in feet)	16	0.25	4.0
Latching Valves	20	3.0	60.0
Pressure Regulators	4	5.0	20.0
Pressurant Filter	4	0.5	2.0
Cold Gas RCS System			
Quick Disconnects (Halves)	6	3.0	18.0
Lines(length in feet)	12	0.25	3.0
Latching Valves	6	3.0	18.0
Regulators	2	5.0	10.0
Filters	2	0.5	1.0
		Propulsion Unit Dry Weight	2425.5

PLATFORM N2O4/N2H4 SYSTEM

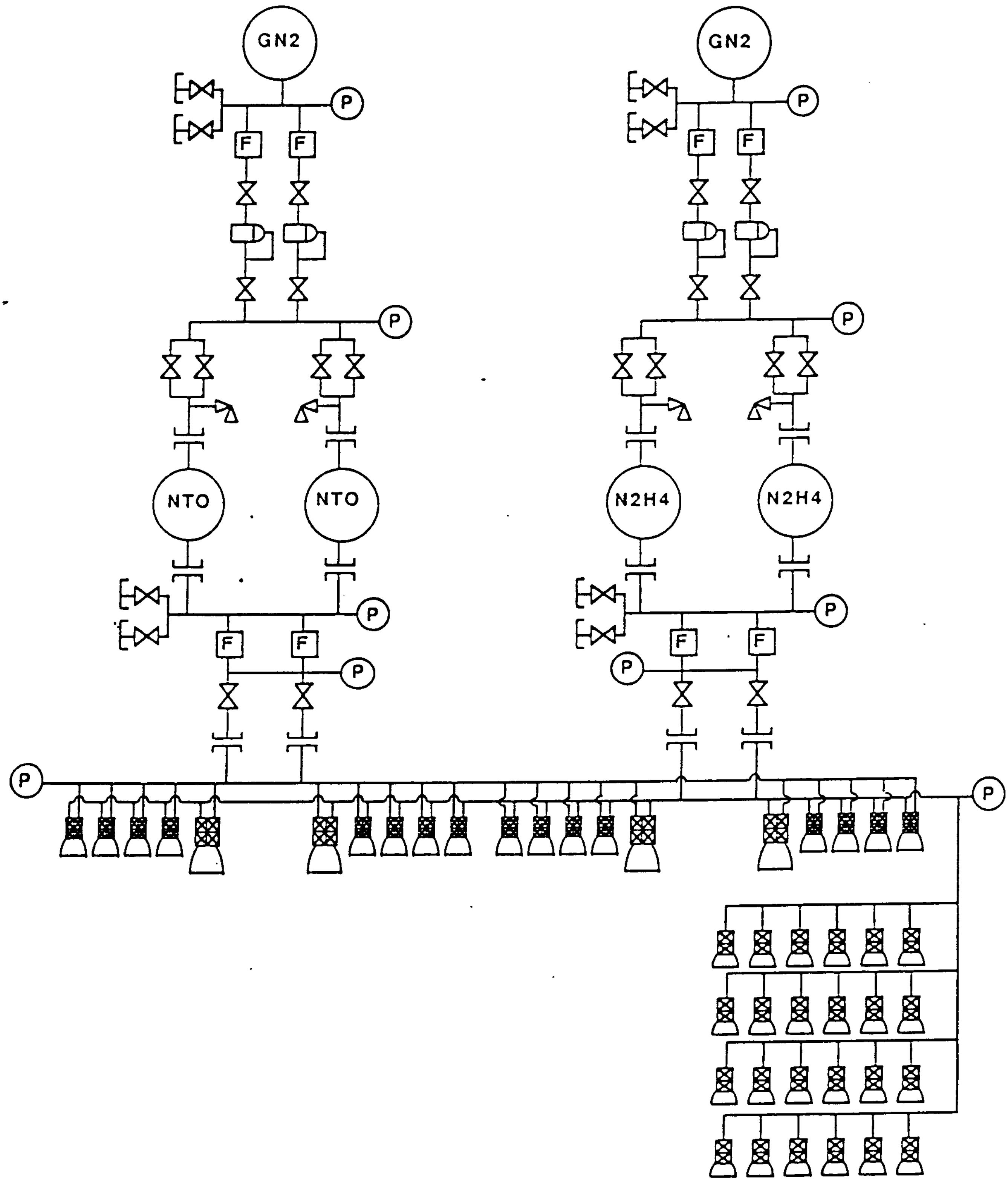


Figure III-11

Table III-12 - Platform N2O4/N2H4 System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL_WT
N2H4 Tank	2	142.5	285.0
N2O4 Tank	2	142.5	285.0
GN2 Tank	2	180.0	360.0
Latching Valve	8	3.0	24.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	10	0.5	5.0
Quick Disconnects (Halves)	24	3.0	72.0
Propellant Filter	4	1.0	4.0
Temperature Sensors	24	0.5	12.0
Propellant Distribution System			
Lines(length in feet)	52	0.25	13.0
Joints	-	-	-
Thermal Conditioning System			
Heat Trace(length in feet)	52	0.08	4.16
Thermostats/Wiring	16	0.22	3.52
Insulation(length in feet)	52	0.04	2.08
Secondary Structure	1	675.0	675.0
Controller	1	300.0	300.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Hydrazine RCS (5 lbf)	24	1.5	36.0
Pressurization System			
Quick Disconnects (Halves)	8	3.0	24.0
Lines(length in feet)	16	0.25	4.0
Latching Valves	20	3.0	60.0
Pressure Regulators	4	5.0	20.0
Pressurant Filter	4	0.5	2.0
		Propulsion Unit Dry Weight	2268.0

PLATFORM N2H4 SYSTEM

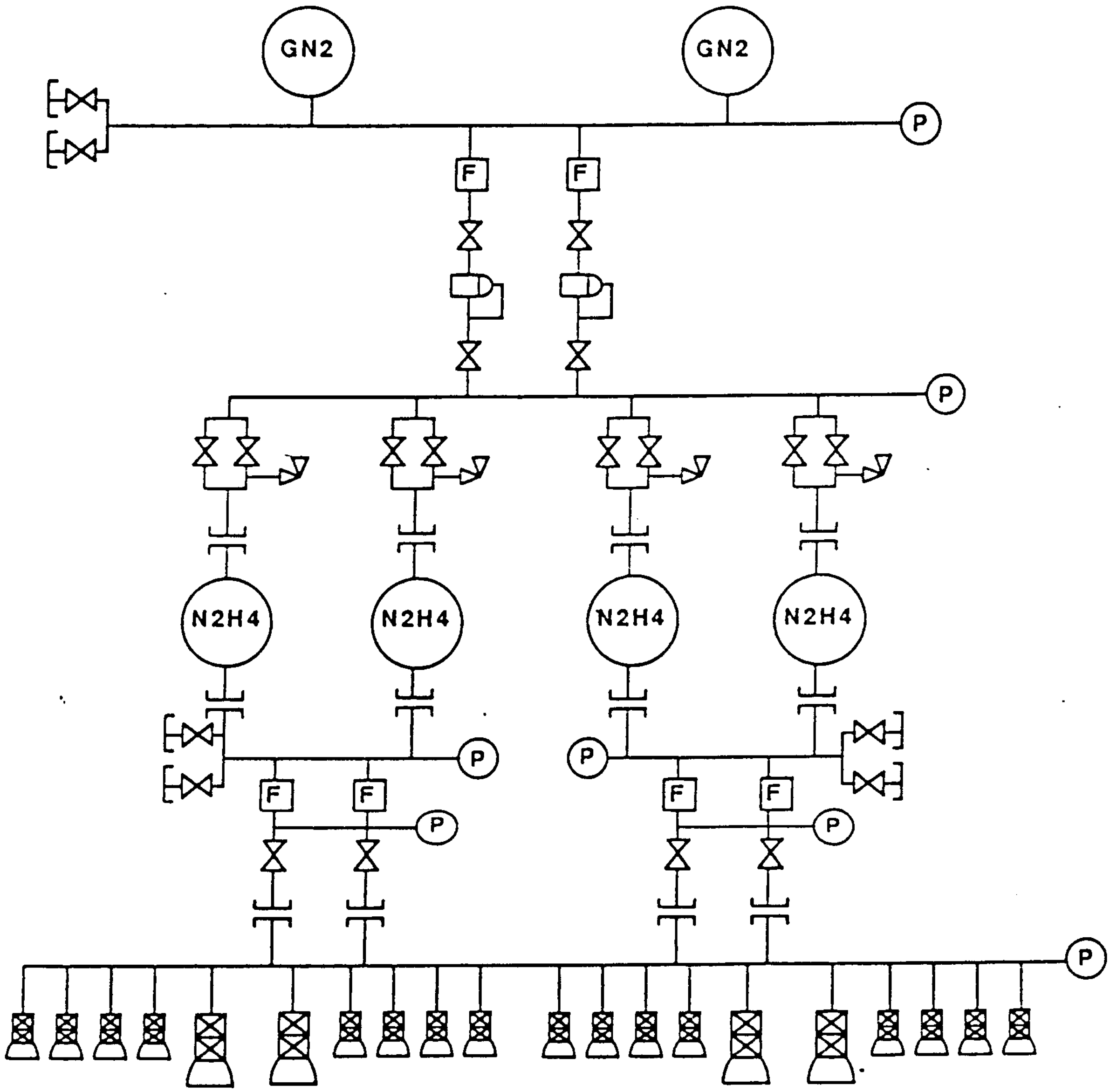


Figure III-12

Table III-13 - Platform Hydrazine System
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
N2H4 Tank	4	185.0	740.0
GN2 Tank	2	275.0	550.0
Latching Valve	8	3.0	24.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	7	0.5	3.5
Quick Disconnects (Halves)	24	3.0	72.0
Propellant Filter	4	1.0	4.0
Temperature Sensors	24	0.5	12.0
Propellant Distribution System			
Lines(length in feet)	48	0.25	12.0
Joints	-	-	-
Thermal Conditioning System			
Heat Trace(length in feet)	48	0.08	3.84
Thermostats/Wiring	16	0.22	3.52
Insulation(length in feet)	48	0.04	1.92
Secondary Structure	1	675.0	675.0
Controller	1	300.0	300.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Pressurization System			
Quick Disconnects (Halves)	6	3.0	18.0
Lines(length in feet)	10	0.25	2.5
Latching Valves	14	3.0	42.0
Pressure Regulators	2	5.0	10.0
Pressurant Filter	2	0.5	1.0

		Propulsion Unit Dry Weight	2552.5

Cold Gas O_2/H_2 - 100% Delivery and Storage Tanks
Ref: Figure III-13, Table III-14

This system uses cold gas launched and stored at 2000 psia to carry more gas in smaller accumulators than a system at 450°R. The tanks were sized for GO_2 at 300°R and GH_2 at 210°R. The oxygen and hydrogen tanks were sized for composite overwrapped inconel. The composite overwrapped tank was again sized using 80% of the mass of a tank made completely of inconel. Heaters are added to the tanks on this system to provide the required thermal conditioning for the propellant. The system was sized using a 6:1 mixture ratio for maximum performance of the O_2/H_2 thrusters.

GO_2/GH_2 - OMV Delivery - Electrolysis Return - 10% Accumulators (Polar Platform Only)
Ref: Figure III-14, Table III-15

In this option the platform would be delivered to its orbit using the OMV. The system then returns to base using propellant supplied by an on-board electrolysis unit. A total impulse of 770,000 lbf - sec is required to return from orbit. The propellant for this impulse is stored in 10% accumulators, returning the platform in stages. The oxygen and hydrogen accumulators were sized as a composite overwrapped inconel tanks. A minimum gage titanium tank was used for water storage. All tanks were sized with a 1.5 safety factor. Table III-16 indicates the amount of additional solar panel weight required for various duty cycles for the electrolysis unit. This mass can vary depending on the required duty cycle. The hardware for the system is essentially the same as the OMV electrolysis system with the exception of the tanks and solar panels. The system was again sized for an 8:1 mixture ratio to make complete use of the generated propellant. The gas is stored at 2000 psia and 450°R.

PLATFORM COLD GAS
H2/O2 SYSTEM

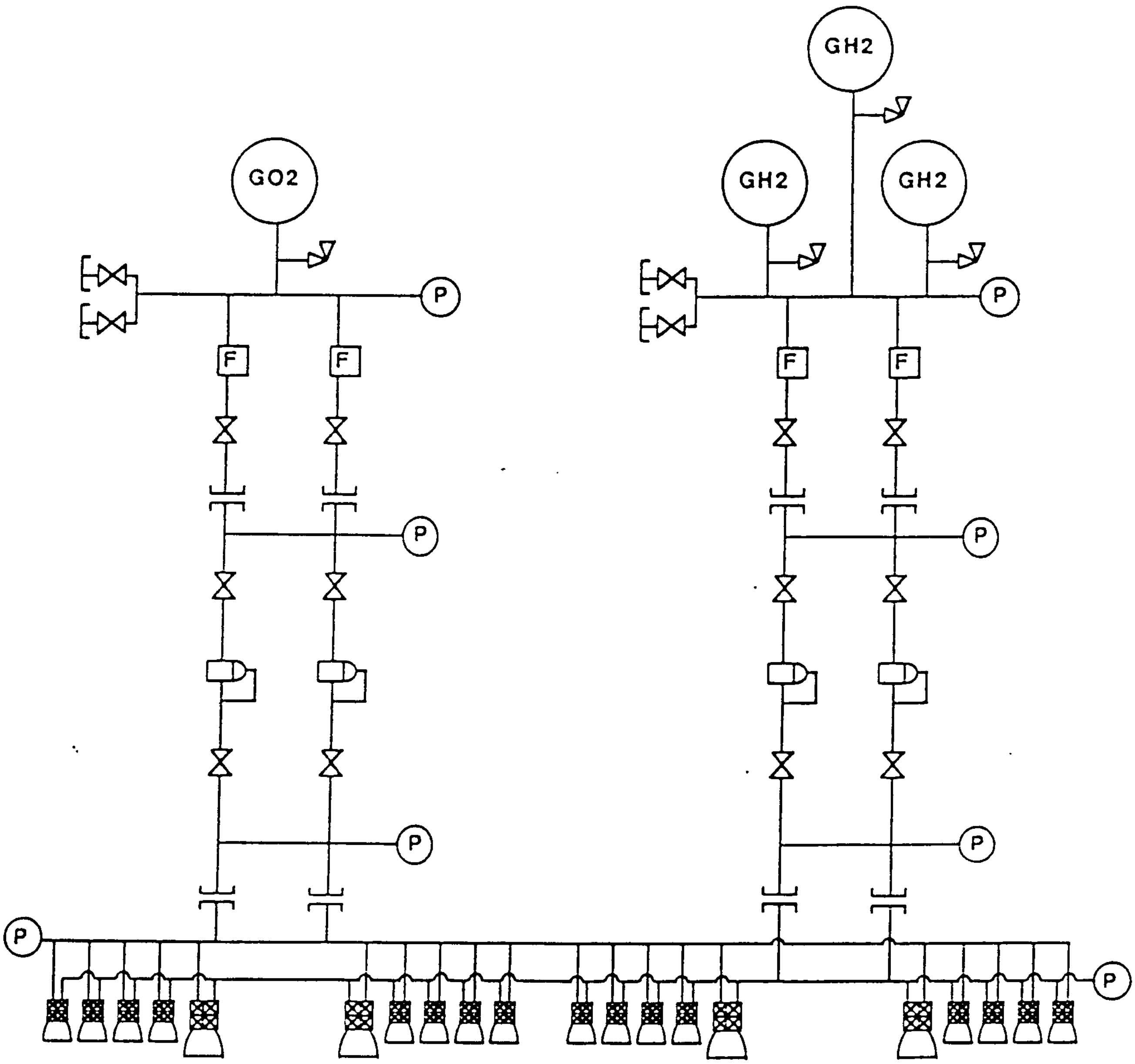


Figure III-13

Table III-14 - Platform Cold Gas O2/H2 System Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
GH2 Tank	3	1838.0	5514.0
GO2 Tank	1	1863.0	1863.0
Latching Valve	16	3.0	48.0
Pressure Relief Valve	4	5.0	20.0
Pressure Transducer	8	0.5	4.0
Quick Disconnects (Halves)	20	3.0	60.0
Propellant Filter	4	0.5	2.0
Temperature Sensors	32	0.5	16.0
Pressure Regulator	4	5.0	20.0
Propellant Distribution System			
Lines(length in feet)	100	0.25	25.0
Joints	-	-	-
Secondary Structure	1	675.0	675.0
Controller	1	300.0	300.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Thermal Conditioning System			
Heaters	1	5.0	5.0
Insulation(length in feet)	450	0.1	45.0
		Propulsion Unit Dry Weight	8654.2

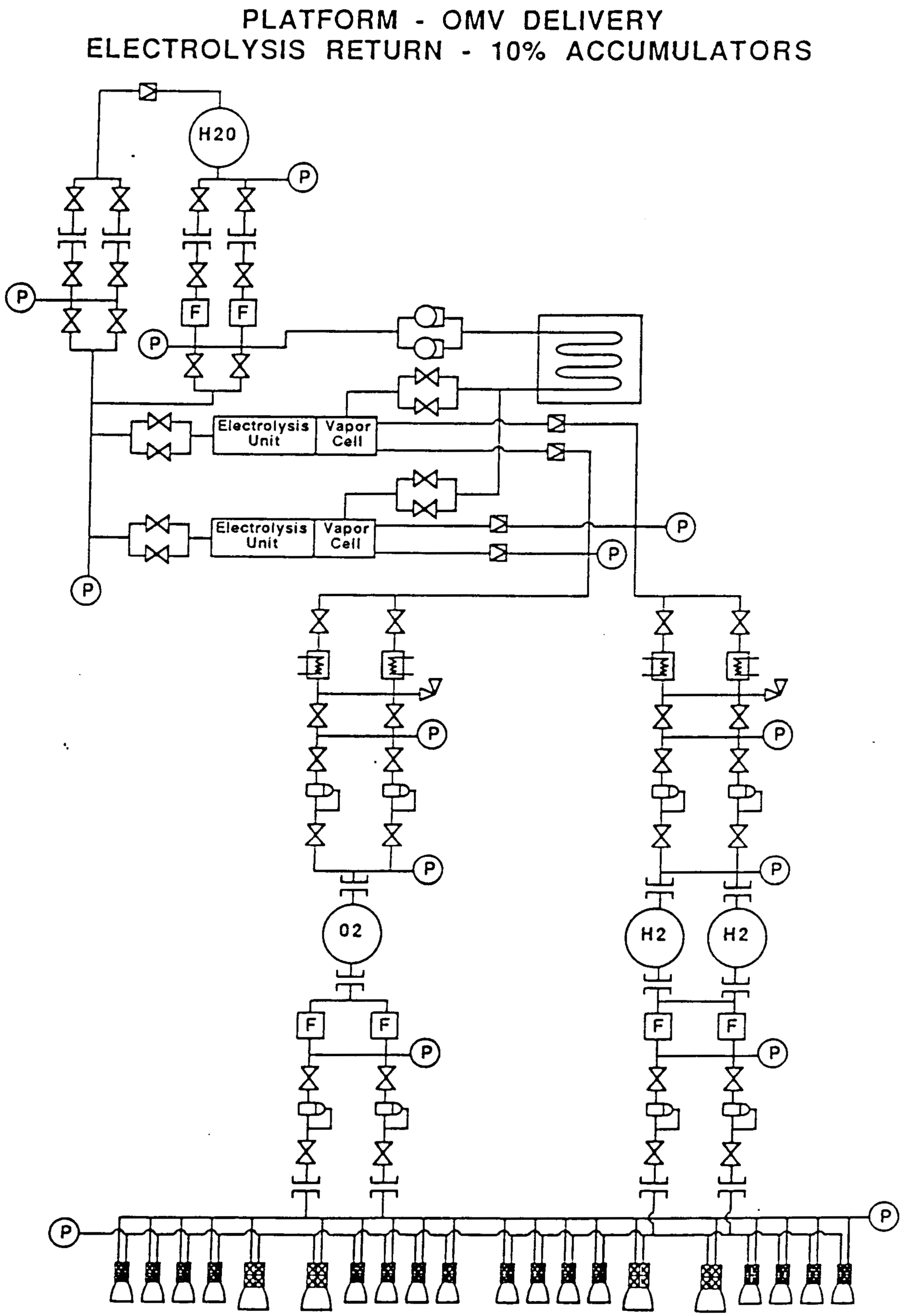


Figure III-14

Table III-15 - Platform - OMV Delivery
Electrolysis Return - 10% Accumulators
Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
GH2 Tank	2	239.0	478.0
GO2 Tank	1	190.0	190.0
H2O Tank	1	40.0	40.0
Latching Valve	44	3.0	132.0
Pressure Relief Valve	2	5.0	10.0
Pressure Transducer	16	0.5	8.0
Quick Disconnects (Halves)	28	3.0	84.0
Propellant Filter	6	0.5	3.0
Temperature Sensors	32	0.5	16.0
Pressure Regulator	8	5.0	40.0
Check Valves	5	1.0	5.0
Electrolysis Unit	2	182.0	364.0
Radiator	2	43.0	86.0
Pump	2	10.0	20.0
Desicators	4	24.0	96.0
Propellant Distribution System			
Lines(length in feet)	200	0.25	50.0
Joints	-	-	-
Secondary Structure	1	800.0	800.0
Controller	1	300.0	300.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Thermal Conditioning System			
Heaters	4	5.0	20.0
		Propulsion Unit Dry Weight	2799.2

Table III-16 - Required Solar Panel Mass for Various Electrolyzer Duty Cycles
Platform - OMV Delivery - Electrolysis Return
10% Accumulators

POWER_REQUIRED	DUTY_CYCLE	SOLAR_PANEL_MASS
0.5 kw	1132.4 hr/47.2 days	69.4 lbm
5 kw	113.2 hr/4.7 days	694.5 lbm
25 kw	22.6 hr	3472.4 lbm
50 kw	11.3 hr	6944.7 lbm
100 kw	5.7 hr	13889.4 lbm
200 kw	2.8 hr	27778.8 lbm
500 kw	1.1 hr	69447.0 lbm

GO_2/GH_2 - Electrolysis - 10% Accumulators
Ref: Figure III-15, Table III-17

The final platform option uses the on-board electrolysis unit to charge 10% accumulators which provide propellant for the total mission. The system is essentially the same as the OMV electrolysis option with the exception of the tanks and solar panels. The oxygen and hydrogen tanks are composite overwrapped inconel. All tanks were sized with a 1.5 factor of safety. Table III-18 indicates the solar panel weights for various duty cycles. An 8:1 mixture ratio was used and the gas was stored at 2000 psia and 450°R.

D. OMV Propulsion System Recommendations

According to Table II-4, the supercritical O_2/H_2 propulsion system candidate and the storable bipropellant candidates require the least propellant of the concepts considered. The gaseous O_2/H_2 concept appears attractive (even though it does require more propellant) since some water may be available on-board Space Station for electrolysis and subsequent "pumping up" of the gaseous accumulators for each OMV mission. This would be a simple resupply operation with none of the safety hazards associated with toxic liquids such as N_2H_4 or N_2O_4. The gaseous O_2/H_2 concept also eliminates the operational problems of dealing with cryogenic fluids.

The power requirements of the gaseous O_2/H_2 concept for resupply at Space Station may be prohibitively large. For instance, for 271,860 lbm required over a 10 year period and an electrolysis specific power value of 4 Kw-hr/lbm, the average continuous power requirement is 12.4 Kw. Without DRM 10 (logistics module ferrying) and DRM 6 this average power requirement is 3.8 Kw. In addition, there is no solution in sizing this concept for DRM 6 (payload viewing mission) due to the large delta V requirement and the high dry weight characteristics of this concept. However, the remaining missions can be accommodated with 2600 lbm propellant capacity and added accumulators for DRM 10. The ECLSS/on-board propulsion fluid economy for excess water availability would probably not nearly support OMV missions anyway. Therefore, the gaseous O_2/H_2 concept was sized at a mixture ratio of 6:1 in order to minimize vehicle weight although this implies that excess H_2 must be supplied via STS launch, ECLSS excess, etc.

PLATFORM ELECTROLYSIS SYSTEM - 10% ACCUMULATORS

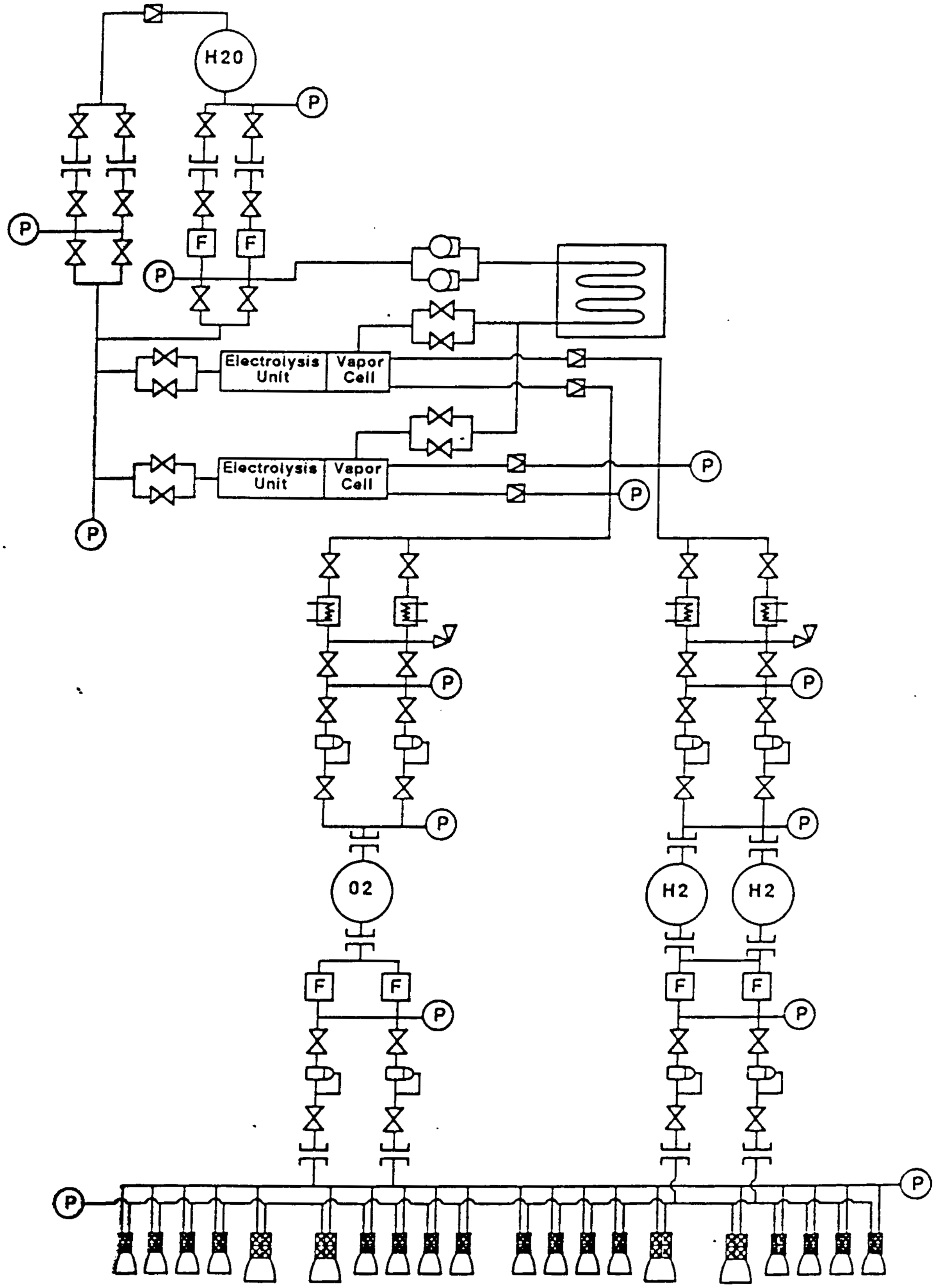


Figure III-15

Table III-17 - Platform Electrolysis System - 10% Accumulators Weight Statement

ITEM	QTY	WT/ITEM	TOTAL WT
GH2 Tank	2	435.0	870.0
GO2 Tank	1	344.0	344.0
H2O Tank	1	61.0	61.0
Latching Valve	44	3.0	132.0
Pressure Relief Valve	2	5.0	10.0
Pressure Transducer	16	0.5	8.0
Quick Disconnects (Halves)	28	3.0	84.0
Propellant Filter	6	0.5	3.0
Temperature Sensors	32	0.5	16.0
Pressure Regulator	8	5.0	40.0
Check Valves	5	1.0	5.0
Electrolysis Unit	2	182.0	364.0
Radiator	2	43.0	86.0
Pump	2	10.0	20.0
Desicators	4	24.0	96.0
Propellant Distribution System			
Lines(length in feet)	200	0.25	50.0
Joints	-	-	-
Secondary Structure	1	800.0	800.0
Controller	1	300.0	300.0
Thrusters			
Orbit Adjust (100 lbf)	4	8.3	33.2
RCS (5 lbf)	16	1.5	24.0
Thermal Conditioning System			
Heaters	4	5.0	20.0
		Propulsion Unit Dry Weight	3366.2

Table III-18 - Required Solar Panel Mass for Various Electrolyzer Duty Cycles Platform Electrolysis System - 10% Accumulators

POWER REQUIRED	DUTY CYCLE	SOLAR PANEL MASS
0.5 kw	2205.9 hr/91.9 days	69.4 lbm
5 kw	220.6 hr/9.2 days	694.5 lbm
25 kw	44.1 hr/1.8 days	3472.4 lbm
50 kw	22.1 hr	6944.7 lbm
100 kw	11.0 hr	13889.4 lbm
200 kw	5.5 hr	27778.8 lbm
500 kw	2.2 hr	69447.0 lbm

The likelihood of a significant portion of OMV propellant being available in the form of "free" water at Space Station is almost non-existant, however. Therefore, since the mass fraction of delivering the gaseous O_2/H_2 is so poor, the concept is not competitive with the other propellant types with higher mass fraction capabilities. Delivery of water would be competitive on a launch mass basis, although not a winner compared to the storable bipropellant candidate. However, the electrolysis power requirements would be a major impact and negative consideration.

"Free" propellant for a bipropellant OMV is a possibility when considering STS OMS scavenging. However, present estimates of the commodities required on logistics missions to and from the Space Station usually exceed the weight limitations of the Shuttle. This means that enough OMS propellant (N_2O_4/MMH) will be loaded to fly the mission and allow maximum STS payload weight capability. It also means that no extra OMS propellant may be added for scavenging once at the Space Station since this would actually degrade the STS launch capability. Therefore, the N_2O_4/MMH candidate may have no advantage of "free" propellant.

Since it appears that "free" OMV propellant may not be a reality, then perhaps OMV propulsion should be chosen on the merit of mission accommodation, minimum propellant launch cost, and minimum resupply operations and maintenance costs. In considering the DRM's, the storable bipropellant and supercritical O_2/H_2 systems have the lowest propellant mass requirements and are nearly equal. The mass fractions of delivering these propellants, however, differ widely. The total launch mass of the supercritical propellants may be approximately 55% higher than the storable bipropellants (mass fractions of 0.55 and 0.85 respectively). Therefore, even though propellant consumption is similar over a 10 year period, the total launch costs favor the storable bipropellant concept.

Dual mode ($N_2O_4/N_2/H_4$) propulsion for OMV may provide some benefits over the N_2O_4/MMH/GN_2 system such as elimination of the GN_2 cold gas system and GN_2 resupply if N_2H_4 is acceptable for use in proximity operations. Over the 10 year period the dual mode system saves between 600 and 1160 lbm of main propellant (bipropellant) over the N_2O_4/MMH system because of less dry weight, and saves 70% of the resupply launch costs for proximity operations propellant. These savings, however, total less than a 1% savings in resupply launch mass over the N_2O_4/MMH concept. So, in addition to considering resupply launch costs, the remaining tradeoffs include the increased development costs and/or tankage design for the dual mode system versus the addition of a GN_2 system and associated resupply operations for the N_2O_4/MMH system. These tradeoffs require detailed assessments and are beyond the scope of this study. Consideration should be given to commonality with platform propulsion in these trades, since the propulsion requirements of polar and non-coplanar co-orbiting platforms are very similar to the OMV requirements. A discussion of appropriate candidates for these platforms follows.

E. Platform Propulsion System Recommendation

The dual mode bipropellant system delivers the largest amount of payload assuming an initial total mass of 27045 lbm. When the resupply picture is examined, however, the electrolysis system with 10% accumulators requires less resupply propellant. When the total mass of resupply module is examined it is clearly the best performer. It is also the second best performer as far as payload delivery is concerned. The system, however, is penalized by the amount of time required to fill the accumulators and the power requirements for the electrolyzer. The dual mode and bipropellant systems are very similar in performance and the amount of resupply propellant required. The hydrazine system is the second to last performer as far as payload is concerned but is the worst system to resupply. The cold gas system is the worst performing system due to the mass of the accumulators but is better than the hydrazine system for resupply through the use of the PRSA type resupply tanks.

The non-coplanar co-orbiting platform was assumed to be identical to the polar platform for commonality since their sizing requirements are nearly the same. Therefore, the 27045 lbm system was once again used for this case. The non-coplanar system was examined for the five platform propulsion options. The OMV delivered electrolysis unit was also included. This was driven by assuming that the OMV would be available for servicing from the Space Station. Since the final propellant selection for the Space Station has not been made, it was assumed that all propellant required for the platform would be brought up by the orbiter. The platform round trip velocity requirements for variations in operational altitude starting at a servicing altitude of 463 km (250 nm) and using the 27045 lbm are shown in Figure III-16 Performance parametrics for the various options were determined for several delta velocities in the range. Figure III-17 summarizes the payload capability for each system at the various delta velocities.

The total resupply mass required for the various delta velocities are shown in Figure III-18. The general trends for the performance and resupply of the non-coplanar co-orbiter are essentially the same as for the polar platform. One difference between the graphs is the addition of the OMV delivered platform option. As far as resupply mass is concerned, the system is comparable to the bipropellant and dual mode bipropellant systems.

The electrolysis propulsion systems require a certain length of time to recharge their accumulators. This amount of time is determined by the power available for the electrolyzer. It was assumed that 5kw would be available for the electrolyzer while traveling to and from the operational altitude and 1/2 kw would be available while on-orbit. The electrolyzer provides 136 lbf sec/kw-hr (per Hamilton Standard). Table III-19 shows the total amount of time required to generate all the propellant for the two electrolysis options and the extra time required for the mission for an electrolysis system compared to a "ready to go" monopropellant or bipropellant system. Two 50% accumulator electrolysis systems have been included for comparison. The first system has 50% accumulators filled from a water tank containing 100% of the mission propellant. The system would be resupplied by changing out the water tank and then waiting for the accumulator to be charged (1105.0 hr or 46.0 days). The second system has previously filled 50% accumulators and a water tank containing the remaining 50% of the propellant. Resupply would involve changing out both the water tank and accumulators. This provides a "ready to go" system but pays a resupply penalty by carrying the heavy accumulators.

By definition the coplanar co-orbiting platform will fly at the same altitude as the Space Station (since an altitude difference would cause the platform to drift out of the Space Station plane). Therefore, some sort of continuous drag makeup or reboost scenario similar to that of the Space Station would seem appropriate. In fact, since no large delta V maneuvers or orbit adjustments are required for the coplanar platform, the reboost system chosen for the Space Station may prove to be the best solution for this platform. Of course, one could attach a "dumb" OMV type propulsion module (storable bipropellant) to the platform and satisfy the mission requirements. However, the low minimum thrust and low impulse requirements as well as the

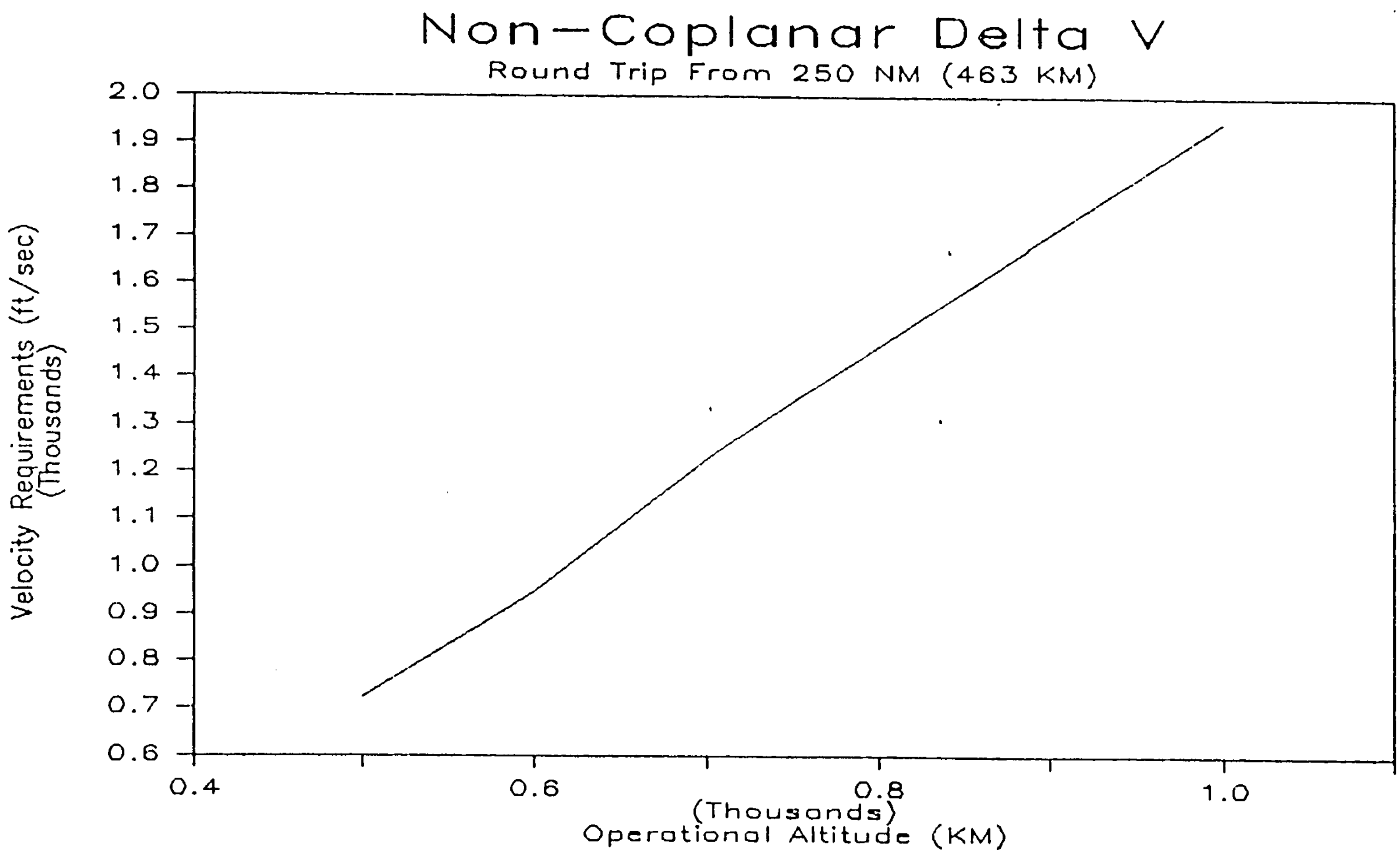


Figure III-16

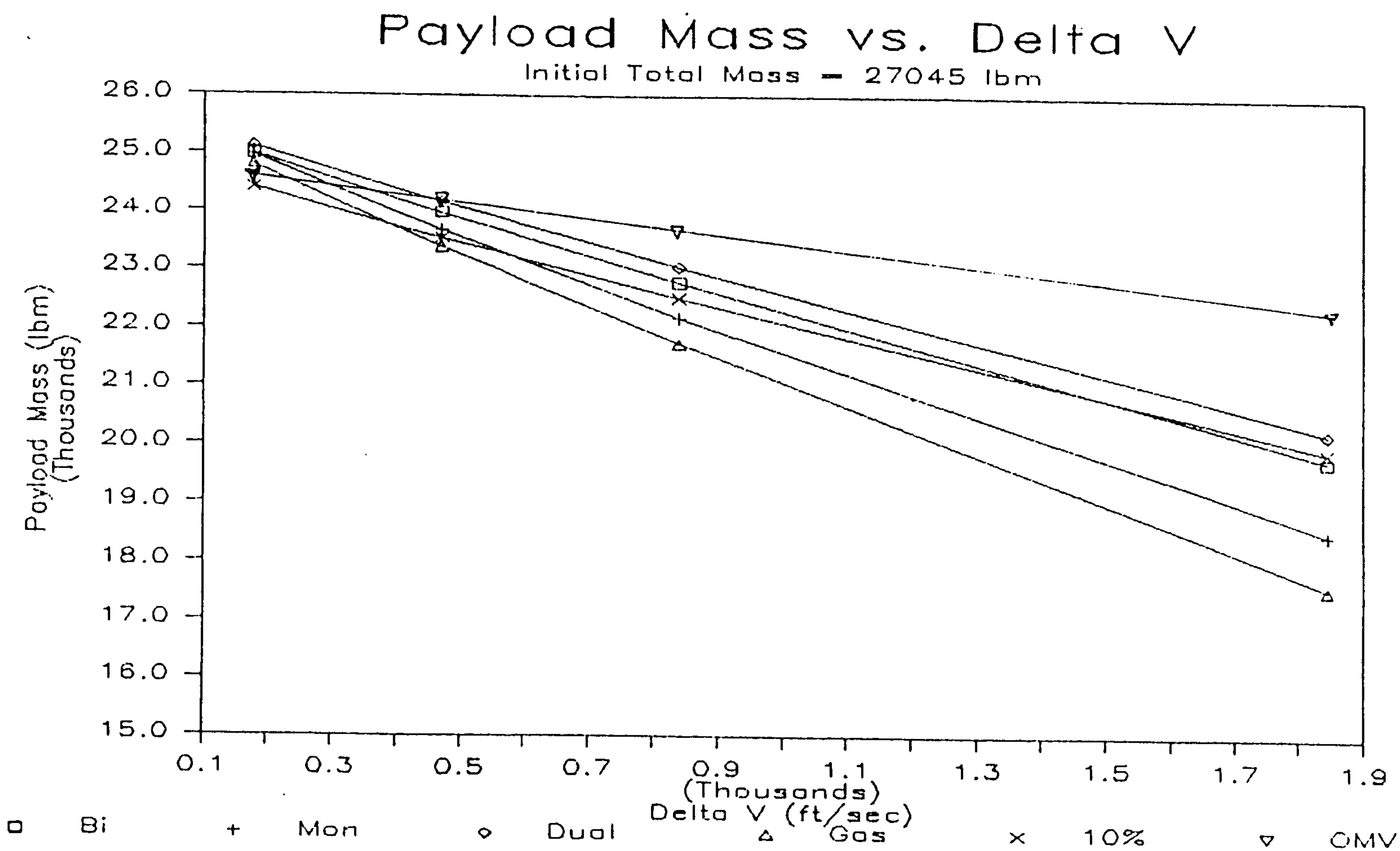


Figure III-17

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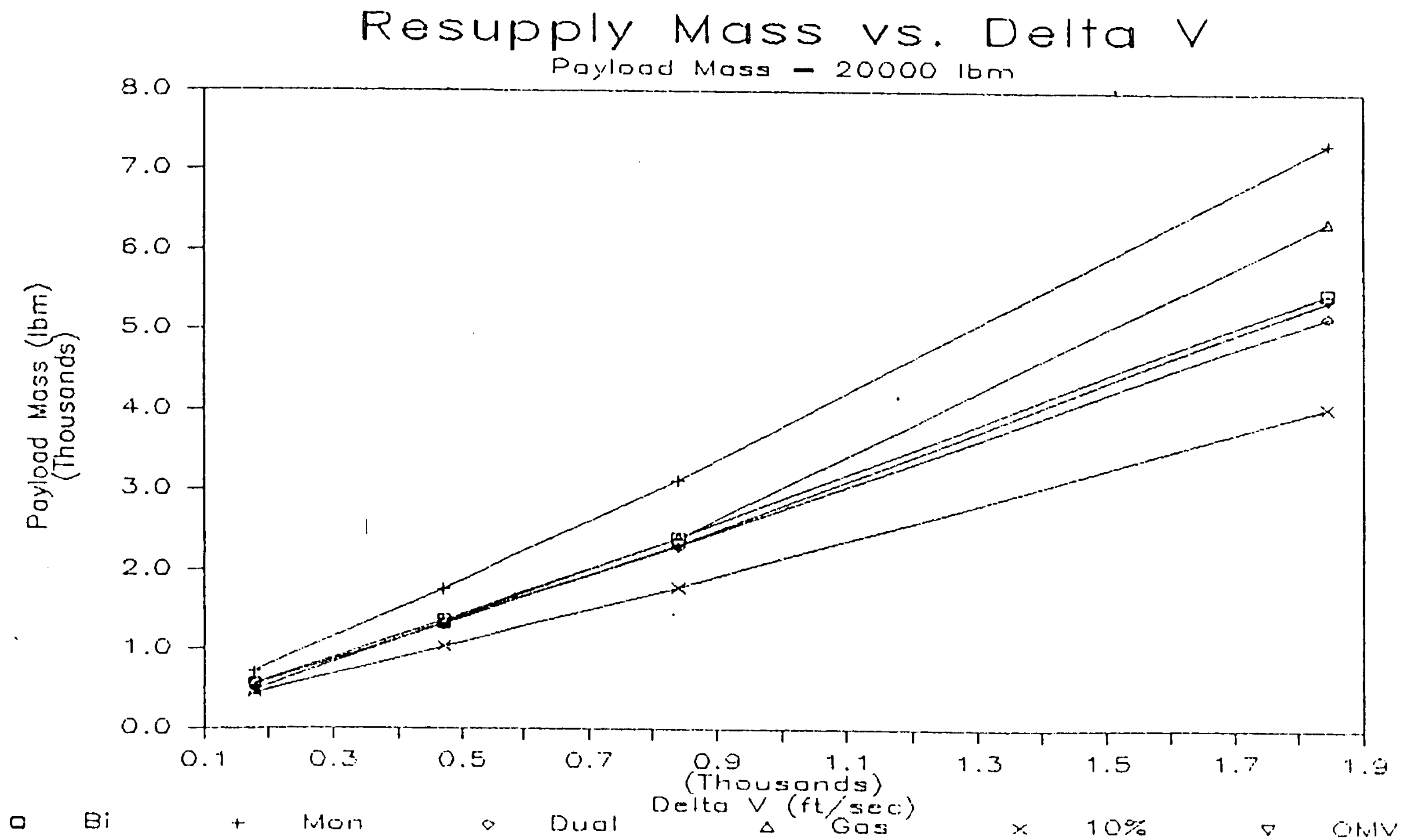

Figure III-18

C-2

Table III-19 - Electrolysis Time Requirements

	TANK REFILL TIME AT SHUTTLE RENDEZVOUS ALTITUDE @ 5KW	ON ROUTE TO OPERATIONAL ALTITUDE @ 5KW	ON ORBIT @ 1/2 KW	RETURN TRIP @ 5KW	TOTAL ELECTROLYSIS TIME	Δ TIME FROM FLIGHT WITH IMMEDIATELY AVAILABLE PROPELLANT
OMV DELIVERY 10% ACCUM	-0- (OMV DELIVERY)	-0- (OMV DELIVERY)	1152.4 HR (48 DAYS)	1036.8 HR (43.2 DAYS)	2189.2 HR (91.2 DAYS)	1036.8 HR (43.2 DAYS)
10% ACCUM	221.0 HR (9.2 DAYS)	884.0 HR (36.8 DAYS)	2210.1 HR (92.1 DAYS)	884.0 HR (36.8 DAYS)	4199.1 HR (175.0 DAYS)	1989.0 HR (82.9 DAYS)
50% ACCUM 100% H2O	1105.0 HR (46.0 DAYS)	-0-	11050.3 HR (460.4 DAYS)	-0-	12155.3 HR (506.5 DAYS)	1105.0 HR (46.0 DAYS)
50% ACCUM 50% H2O	-0-	-0-	11050.3 HR (460.4 DAYS)	-0-	11050.3 HR (460.4 DAYS)	-0-

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insensitivity of propellant consumption to dry weight for this mission (near constant altitude) may point to another system concept.

A water electrolysis propulsion system is well suited to the application to a coplanar co-orbiting platform. A total impulse of 79 K lb_f-sec (worst case nominal) is required for drag make-up and ACS for 90 days for a 25 K lbm platform. This results in a 0.38 Kw continuous power requirement for water electrolysis at a specific power usage of 4.0 Kw-hr/lbm. This power level is less than the 0.5 Kw that will be available to propulsion on a continuous basis on-board a platform. Water for resupply may also be available for the relatively small supply requirements from Space Station ECLSS or MTL leftover. In addition, commonality with Space Station propulsion could be additional rationale.

IV. TASK III - PRELIMINARY DESIGN

Task III was deleted by the revised SOW.

V. TASK IV - COMPUTER SIMULATION

The objective of this task was to provide a computer simulation of the water electrolysis propulsion system for the Space Station. The computer program is capable of simulating design point, planned emergency operation and off-design operation of the propulsion system. It also was written in a "breadboard" fashion to allow each component algorithm to be easily modified. This task also provided full documentation of the computer program in the form of a user's manual that includes a description of the program, description of all subroutines, sample cases and a complete program listing.

The work effort on the computer simulation program began with the development of a software plan. This plan was used to guide the development and testing of the computer program to insure it met all of the requirements of the program. The program was originally intended to model the water electrolysis propulsion system for space station. However, as the program was being developed it was found that it could easily be expanded into a program that would model several other types of propulsion systems. This capability was enhanced by developing a user friendly interface. The interface allows the user to enter all input parameters and define the feed system components through a sequence of questions on the screen. This facilitates trade studies of various systems.

The program consists of a main driving program and subroutines that correspond to each component modeled. The component algorithms can easily be modified by the user because of the modular form of the program. The computer program models all the components that are in the Space Station water electrolysis propulsion system, and several others that are important to propulsion systems in general. These components include: the electrolysis units, dryers, liquid tanks, gas accumulators, thrusters, back-pressure regulators, regulator pumps, and compressors. It also models simple fluid components such as valves, lines, filters, etc.

Documentation of the program was developed concurrently with the programming of the model. This documentation evolved into a user's manual that has been submitted as a separate document for this program*. The user's manual is a complete detailed description of the program. It includes a description of the program; the theoretical background of the mathematical models; the analytical techniques used in deriving solutions; a complete description of all subroutines and sample cases run using the program. For a complete description of the program the user's manual should be consulted. However, some of the highlights of the program are summarized in the following paragraphs.

The electrolysis subroutine has two options, one for the Solid Polymer Electrolyte Electrolysis unit and the other for the Potassium Hydroxide Electrolysis unit. The subroutine calculations are based on the information gathered from two manufacturer's. The information for the SPE unit was provided by Hamilton Standard, and information for the KOH unit was supplied by Life System, Inc. The operating equations provide the flowrates into and out of the unit and are based on the power input and the efficiency of the unit.

The dryer algorithm determines the amount of moisture absorbed from the gas stream. The amount absorbed is based on the operational temperature. Performance curves for the drying materials were obtained from the dryer manufacturers. The algorithm also evaluates the total amount of water absorbed, and determines when the dryers need to be switched for regeneration. During regeneration, the program calculates the power required to liberate the moisture from the dryer material.

* MCR---87, Users Manual for Space Station Electrolysis Propulsion System Simulator

The water storage tank subroutine assumes that the tank is a constant pressure bellows tank. The pressure is input by the user. The routine calculates the pressure and temperature in the tank based on the water flowrate which is determined by the electrolysis unit. Basic mass and energy balances along with heat transfer equations are used to determine conditions in both the gas and liquid regions.

The routine for the gas accumulators has three different methods of calculation depending on what type of component is being modeled. The program models three types of accumulators:

1) an accumulator representing the volume of a component with a regulator downstream,
2) an accumulator representing the volume of a component without a regulator downstream,
3) an actual storage accumulator followed by a backpressure regulator. The calculations for the backpressure regulator are done in a separate routine.

Mass and energy balances and heat transfer equations are used to determine the pressure and temperature in the accumulator.

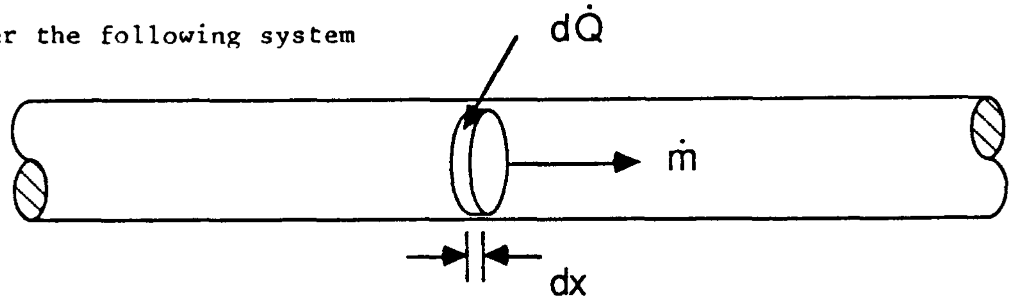
The simple feed system components such as lines, valves, etc., are modeled in the subroutine PDRP. PDRP is a steady-state pressure drop model. The model is designed for a single incompressible fluid. The piping losses are frictional losses, while the line components use a loss coefficient (K) with losses proportional to the square of the fluid velocity.

A plotting routine for the IBM-PC was also delivered with the computer model as requested by MSFC. The routine was limited due to available budget.

APPENDIX A

Constant Hot-Side Temperature Heat Exchanger

Consider the following system



the following relations apply:

$$d\dot{Q} = U\,(T_n - T_f)\,dA \qquad (1)$$

$$d\dot{Q} = \dot{m}c_p dT_f \qquad (2)$$

where: u is the overall heat transfer coefficient, $\frac{BTU}{ft^2 \text{ sec } °R}$

Tn is the constant hot side temperature, Deg R
Tf is the fluid temperature, Deg R
A is the heat transfer area, ft^2
cp is the fluid specific heat, BTU/lbm °R

equating (1) and (2)

$$\dot{m}c_p dT_f = U\,(T_n - T_f)\,dA$$

let $\theta = T_n - T_f$
then $d\theta = -dT_f$

$$-\dot{m}c_p d\theta = U\theta dA$$

assume u varies linearly with A so that:

$$-\dot{m}c_p \frac{d\theta}{\theta} = (a+bA)\,dA \qquad (3)$$

assuming cp to be effectively constant and integrating (3)

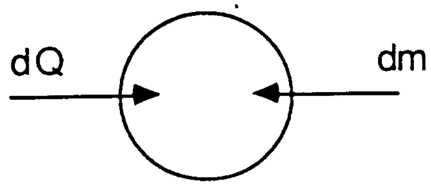
$$-\dot{m}c_p \ln\left(\frac{\theta_2}{\theta_1}\right) = aA + \frac{bA^2}{2}$$

$$\ln\left(\frac{\theta_2}{\theta_1}\right) = \frac{-A}{\dot{m}c_p}\left(a + \frac{bA}{2}\right) \qquad \theta_2 = \theta_1 \exp\left(\frac{-A}{\dot{m}c_p}\left(\frac{u_1 + u_2}{2}\right)\right)$$

equation (4) is used to determine exit fluid temperature.

Storage Tank Thermal Models

1) Energy Equation



$$dU = dQ + hdm$$

where: U is internal energy, BTU
Q is added energy, BTU
h is specific enthalpy, BTU/lbm
m is mass, lbm

(when outflowing, dm is negative and h is based on tank conditions)

taking the differential of

$$dU = mdu + udm \qquad (2)$$

where: u is specific internal energy, BTU/lbm

combining eqs. (1) & (2) solving for mdu

$$mdu = dQ + dm(h - u)$$

$$mdu = dQ + dm\,(Pv)\,J \qquad (3)$$

where: J is unit conversion factor

$$\frac{144}{778.16} \quad \frac{in^2}{ft^2} \quad \frac{BTU}{ft\text{-}lb_f}$$

the general expression for du is

$$du = c_v dT + J\left(T\left(\frac{\delta P}{\delta T}\right)_v - P\right)dv$$

the term $\left(\frac{\delta P}{\delta T}\right)_v$ can be expanded by using the partial derivative relationship:

$$\left(\frac{\delta X}{\delta Y}\right)_Z = -\left(\frac{\delta Z}{\delta Y}\right)_X \Big/ \left(\frac{\delta Z}{\delta X}\right)_Y$$

here $P \rightarrow X$, $T \rightarrow Y$, $v \rightarrow Z$, so that

$$\left(\frac{\delta P}{\delta T}\right)_v = -\left(\frac{\delta v}{\delta T}\right)_P \Big/ \left(\frac{\delta v}{\delta P}\right)_T$$

$$\left(\frac{\delta P}{\delta T}\right)_v = \frac{\beta}{K}$$

where: β is the coefficient of thermal expansion, 1/Deg R K is the isothermal compressibility, - 1/psia

substituting (4) & (5) into (3)

$$mc_v dT + mJ\left(\frac{T\beta}{K} - P\right)dv = dQ + J\,dm\,(Pv) \qquad (6)$$

solving for dT & using $dv = -\frac{v}{m}dm$

$$dT = \frac{dQ + J\,dm\,(Pv) + J\,v dm\left(\frac{T\beta}{K} - P\right)}{mc_v}$$

$$dT = \frac{dQ + J\,dm\left(\frac{T\beta V}{mK}\right)}{mc_v} \qquad (7)$$

Equation (7) is the energy balance used in the supercritical storage model.

2) Pressure Equation

let $v = v(P,T)$

then

$$\frac{dv}{v} = \frac{1}{v}\left(\frac{\delta v}{\delta P}\right)_T dP + \frac{1}{v}\left(\frac{\delta v}{\delta T}\right)_P dT$$

$$\frac{\delta v}{v} = -KdP + \beta dT \qquad (8)$$

solving (8) for dP

$$dP = \left(\beta dT + \frac{dm}{m}\right) \frac{1}{K} \qquad (9)$$

$$dP = \left(\beta dT + \frac{dv}{v}\right) \frac{1}{K} \qquad (9')$$

if β & K do not vary too much, (9') can be integrated to produce

$$\Delta P = \frac{\left(\beta \Delta T = \ln\left(\frac{m_2}{m_1}\right)\right)}{K} \qquad (10)$$

Equation (10) is the pressure relationship used in the supercritical model.

As an aside, if $dP = 0$ (9) becomes

$$dT = \frac{dv}{\beta v}$$

equating (11) with (7)

$$\frac{c_v dv}{\beta v} = \frac{dQ}{m} + \frac{J\,dm}{m}\left(\frac{T\beta v}{K}\right)$$

$$mdv\left(\frac{c_v}{v\beta} + \frac{T\beta J}{K}\right) = dQ$$

since $c_v = \frac{c_p - TV\beta^2 J}{mK}$

$$dQ = dv\left(\frac{m}{v\beta}\left(c_p - \frac{TV\beta^2 J}{mK}\right) + \frac{T\beta J}{K}\right)$$

$$dQ = dv\left(\frac{mc_p}{v\beta} - \cancel{\frac{TV\beta^2 J}{Kv\beta}} \quad + \cancel{\frac{T\beta J}{K}}\right)$$

$$c_p = \frac{v\left(\frac{\delta h}{\delta T}\right)_P}{\left(\frac{\delta v}{\delta T}\right)_P} = v\left(\frac{\delta h}{\delta v}\right)_P$$

$$\therefore dQ = -dm\left(v\left(\frac{\delta h}{\delta v}\right)_P\right)$$