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CRYOGENIC LIQUID O_2/H_2 REACTION CONTROL SYSTEMS FOR SPACE SHUTTLE

by
P. J. KELLY, H. B. McKEE,
and
T. C. PATTEN
McDonnell Douglas Astronautics Company - East
St. Louis, Missouri

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P.J. Kelly
H.B. McKee*
T.C. Patten

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY–EAST
ST. LOUIS, MISSOURI

ABSTRACT

A Space Shuttle liquid oxygen/hydrogen Reaction Control System (RCS) design analysis has been performed. The system concept considered eliminates propellant conditioning equipment and delivers the propellants to the engines in a liquid rather than a gaseous state. This paper provides system design analyses results and compares various means of implementing the concept on the basis of weight, technology requirements and operational considerations. Additionally, weight comparisons are made between cryogenic oxygen/hydrogen system requirements. These comparisons show that the liquid oxygen/hydrogen system concept could effect marked weight reductions in the Space Shuttle orbiter total impulse range.

1. INTRODUCTION

To provide the technology base required for the Space Shuttle design, the National Aeronautics and Space Administration (NASA) has sponsored several technology programs related to Reaction Control Systems (RCS). Among these were a series of studies to provide system design data for selection of preferred system concepts and delineation of requirements for complementing component design and test programs. The initial system study programs considered a broad spectrum of system concepts, but because of high vehicle impulse requirements, coupled with safety, reuse and logistics considerations, only cryogenic oxygen and hydrogen propellants were considered. Also, engine pulse mode ignition unknowns and concern regarding distribution of cryogenic liquids eliminated liquid-liquid feed systems as candidate concepts. Therefore, only systems which delivered propellants to the engines in a gaseous state were considered for the Reaction Control System. The results of those initial studies are reported in References (1) through (4). These studies indicated that a design approach using heat exchangers to thermally condition the propellants and turbopumps to provide system operating pressure would best satisfy requirements for a fully reusable Space Shuttle. A system concept of this general type is illustrated by Figure 1.

TYPICAL TURBOPUMP RCS

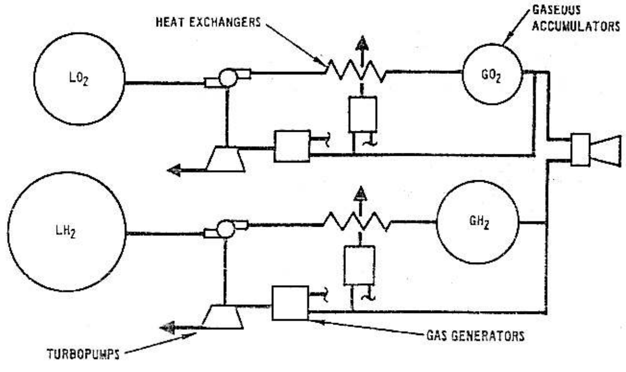

Figure 1

The NASA contracted** with McDonnell Douglas Astronautics Company–East (MDAC–E) in July 1971 for additional study of the Shuttle Reaction Control Systems. One task in this study was to investigate alternate concepts which could both improve performance and reduce the technology concerns associated with fast start up turbopumps in a reusable system application requiring many restarts during each mission. Both gaseous and liquid propellant distribution were considered in the exploratory effort but the most striking study results were obtained with a system concept based on delivery of liquid propellants to the thrusters.

Figure 2 illustrates the liquid O_2/H_2 (LOX/LH_2) RCS concept. The propellants are stored as subcooled liquids in low pressure tankage. Small motor operated pumps supply system operating pressure and are used in conjunction with bellows type liquid accumulators using blowdown helium pressurization. Under pulse mode operating conditions, the thruster assemblies are normally supplied by the pressurized liquid accumulators. When, due to thruster usage, the helium pressure in the accumulators has decayed to a prescribed level, the pumps are restarted and the accumulators are recharged with propellant. For steady state firing the pumps can be sized to provide full system thrust, or the pumps and accumulators can operate in unison to satisfy large single impulse demands. This system design approach was found to provide significant weight savings and a marked reduction in system complexity. This paper describes the analyses and results obtained for the LOX/LH_2 system concept.

2. SYSTEM DESIGN REQUIREMENTS

The orbiter stage for which the LOX/LH_2 RCS studies were conducted is illustrated in Figure 3. Vehicle characteristics are based primarily on the results of MDAC–E studies of fully

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* Member AIAA

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TYPICAL LIQUID-LIQUID RCS

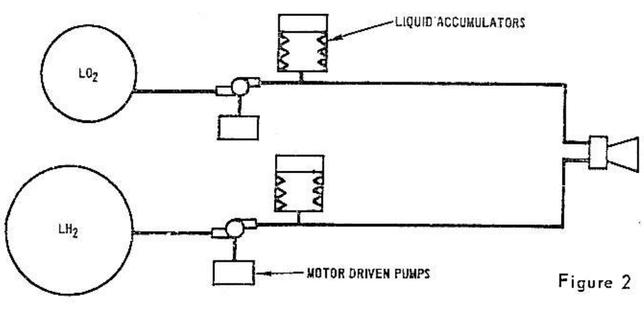
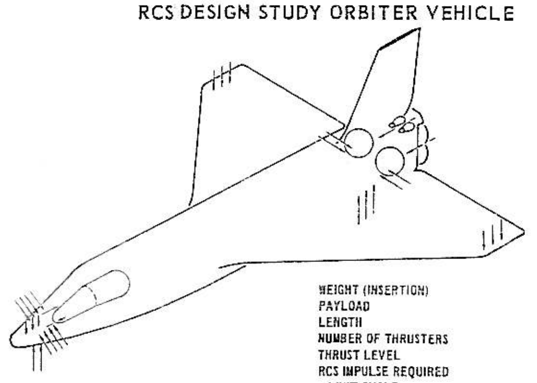


Figure 2

RCS DESIGN STUDY ORBITER VEHICLE



WEIGHT (INSERTION)	331,700 LB
PAYLOAD	65,000 LB
LENGTH	174.7 FT
NUMBER OF THRUSTERS	33
THRUST LEVEL	1,150 LB
RCS IMPULSE REQUIRED	
LIMIT CYCLE	150,000 LB-SEC
ATTITUDE MANEUVER AND DAMPING	830,000 LB-SEC
MANEUVERS (– 20 FT/SEC)	1,270,000 LB-SEC
TOTAL	2,250,000 LB-SEC

Figure 3

reusable orbiters and boosters defined in Reference (5). A distinguishing feature of the orbiter configuration is that the main engine propellant tanks are internal, resulting in a relatively large orbiter stage. Most of the design studies described herein use this orbiter as a reference configuration. The exceptions are in the system weight comparisons of Section 6. These show RCS weight at design requirements corresponding to smaller orbiter configurations of the type designed to use external, main engine tankage.

Reference (6) provides the detailed analyses and rationale used to develop the RCS requirements tabulated in Figure 3. The RCS uses 33 engines at 1,150 lb of thrust each to provide three-axis attitude control. Thrust level and engine arrangement are designed such that, with the failure of any two control engines, the system will still provide torque levels sufficient for safe vehicle entry. The total impulse of the system is 2.25 million lb-sec. This includes total impulse for both attitude control and vernier translation maneuvers of ± 20 ft/sec. These requirements serve to define the basic system design parameters, viz., engine size and storage tank capacity. However, for the systems considered here, two additional, interrelated requirements affect the supply system design. These are: (1) system thrust level, in terms of the maximum number of engines firing simultaneously; and (2) the maximum system total impulse expended during any single maneuver. These are important because they affect pump, pressurization, and liquid accumulator design within the RCS.

As shown in Reference (6), the system must sustain a maximum thrust of 5,570 lb, based on five engines firing simultaneously. This corresponds to the use of four control engines for a translation maneuver and the equivalent of one additional engine for vehicle attitude control during maneuver. The maximum total impulse during any single maneuver is shown in Reference (6) to be 166,000 lb-sec, based on coelliptic ΔV requirements for a resupply mission. These added constraints set the design criteria for the propellant supply system and establish an envelope for tradeoffs between pump flowrates and storage capacities of high pressure liquid accumulators. For example, in a system design without liquid accumulators, the pumps must be capable of satisfying flow demands for full system thrust. Conversely, if the high pressure liquid accumulators are sized to provide 166,000 lb-sec, the pump flow rates required during the maneuver could, theoretically, be zero. However, average impulse expenditure rate during entry determines the minimum pump size. The average system thrust level during entry and, hence, the minimum equivalent pump flow is shown by Reference (6) to be 250 lb thrust. The total entry impulse (500,000 lb-sec) is much greater than the maximum single impulse burn but the time for its expenditure is relatively long. The pump is designed for continuous operation during entry; avobe average demands are met by accumulator supplementation.

3. LIQUID SYSTEM OPERATION

The liquid hydrogen storage conditions are shown in Figure 4. Hydrogen is stored at near saturation conditions and pumped into liquid accumulators at supercritical pressures. The accumulators operate in a blowdown mode and their pressure levels are selected to provide minimum system weight, considering the pressure margins required to insure liquid propellants when the pressure is throttled as low as 200 psia (minimum chamber pressure). For these conditions, the hydrogen density changes from 4.4 to 3 lb/ft^3. Liquid oxygen storage, shown in Figure 5 uses subcooled liquid propellant provided by either pumping or helium pressurization. As the oxygen is stored subcritically, as much as 25 BTU/lb could be absorbed without two phase operation. For these conditions, the oxygen density could vary from 72 to 60 lbf/ft^3.

LIQUID HYDROGEN STORAGE CONDITIONS

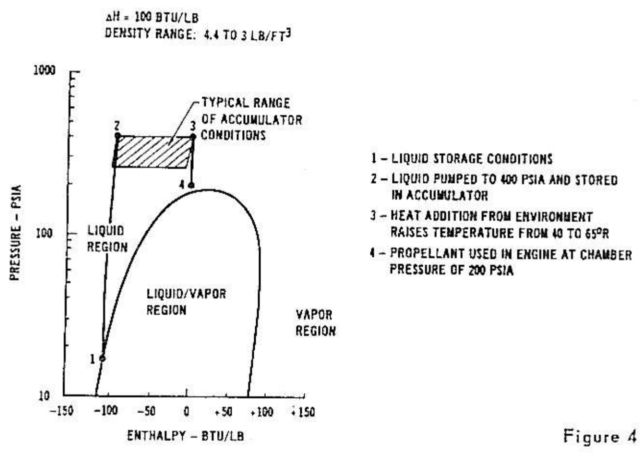


Figure 4

Two primary questions arose with liquid propellant use. First, can a liquid distribution system be designed to provide sufficiently low propellant heating such that density changes are held to levels low enough for satisfactory engine operation? It can be seen from Figure 6 that at all pressure levels of interest, significant hydrogen density changes to levels below 3.0

LIQUID OXYGEN STORAGE CONDITIONS

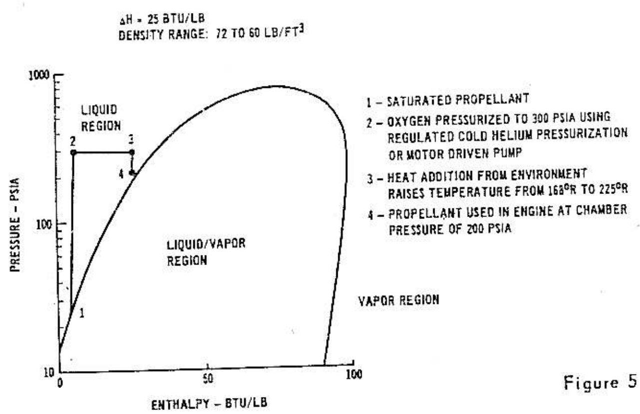


Figure 5

LIQUID HYDROGEN DENSITY VARIATIONS

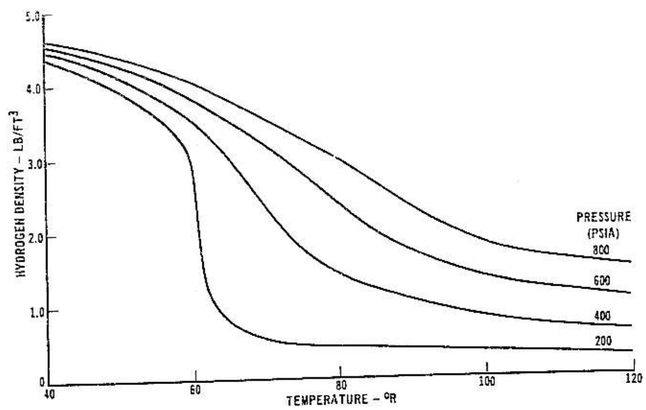


Figure 6

lb/ft³ will occur if hydrogen temperature changes are not controlled.

Cryogenic propulsion systems developed to date have used propellants stored and distributed near saturation conditions. Thus, any heating resulted in propellant vaporization and large propellant density changes. These latter problems are avoided in the liquid system by operating with highly subcooled liquid propellants which can absorb some heat input without propellant vaporization. In fact, the hydrogen could be delivered supercritically. To fully define the system thermal characteristics required that much of the liquid system design effort be directed toward detailed thermal analysis.

The second question concerns the feasibility of liquid ignition in a pulse mode engine. Although the ignition temperature limits are significantly lower than those previously considered, a review of ignition phenomena with engine manufacturers revealed no fundamental reasons that would make liquid ignition doubtful. Additionally, NASA has initiated technology programs by engine manufacturers to fully define engine ignition aspects related to system design. Detailed examination of ignition was beyond the scope of the current study.

The storage conditions shown in Figures 4 and 5 were selected on the basis of system thermal and design analyses. The study initially defined system thermal characteristics, to establish that heating rates were low enough to control hydrogen density changes. Alternate system designs were then evaluated to determine the most desirable system configuration in terms of weight and technology requirements. System thermal and design analyses results are summarized in the sections following.

4. THERMAL ANALYSIS

The first step in the system analysis was to establish propellant temperature rise limits and then to examine the practicality of the limits with respect to the allowable line heat leak with varying rates of propellant usage. Propellant temperature change effects on engine operating characteristics are shown in Figure 7. The major effect is an increase in engine mixture ratio as hydrogen temperature increases. Oxygen temperature effects are shown to be minor for the temperature range of interest and oxygen temperature can be allowed to increase to near saturation (225°R). To maintain a reasonable design range on engine combustion temperature ($4.0 < M_R < 5.0$) the hydrogen temperature must be limited to approximately 65°R. At this temperature both thrust level and specific impulse are reduced 3% below the design values, primarily due to mixture ratio changes. The only other parameter found to be affected by temperature changes was the injection velocity. As propellant temperatures increase, the injection velocity increases, but again, if the hydrogen temperature is limited to 65°R, these changes are minimal.

EFFECT OF PROPELLANT TEMPERATURE
ON ENGINE PERFORMANCE

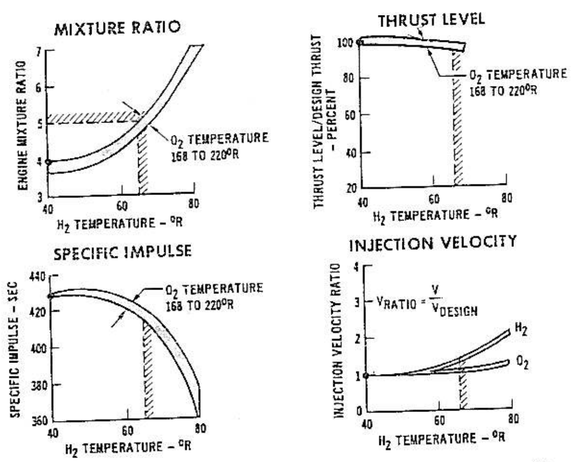


Figure 7

The practicality of a 65°R hydrogen temperature limit was examined by determining equilibrium mean propellant temperatures in the feed lines as a function of heating and hydrogen usage rates. From this it was found that high propellant usage could effectively remove all incoming heat and maintain chilled lines as shown in Figure 8. The smallest practical hydrogen usage rate would be associated with a ± 20° vehicle deadband. This would result in temperatures above 65°R for anticipated heating rates in excess of 25 BTU/hr. To limit the hydrogen temperature to 65°R, the usage rate could be increased. For example, increasing the usage rate by a factor of 3 (by decreasing the deadband to ± 6°) would result in an equilibrium hydrogen temperature less than 60°R for a 25 BTU/hr heating rate. This higher propellant utilization introduces a weight penalty on the order of 1/2 lb per hour (more recent Shuttle vehicle studies have shown that a vehicle deadband of ± 5° is more desirable).

EFFECT OF PROPELLANT CONSUMPTION ON HYDROGEN TEMPERATURE

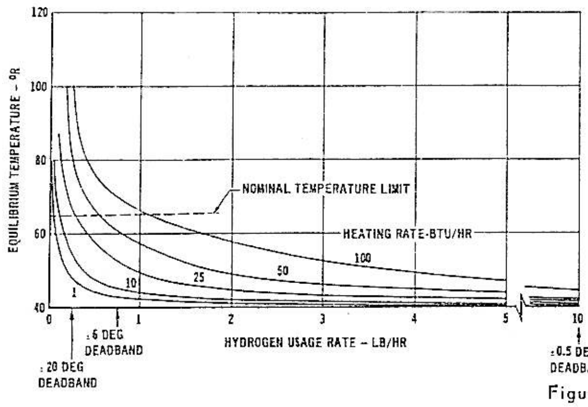


Figure 8

At this design point no system weight penalty would be incurred due to propellant heating. All other operating conditions, such as fine attitude control maneuvers, require much larger usage and would remove sufficient propellant to carry away incoming heat, maintaining chilled propellant and lines. Similar results were obtained for oxygen but were much less restricting than for hydrogen.

The data shown in Figure 8 are for equilibrium conditions where the entire heating rate is uniformly distributed through the total feed system propellant mass. To determine actual temperature gradients and peak heating rates, the manifold shown in Figure 9 was designed. The manifold consists of vacuum jacketed propellant lines with multilayer isolation (MLI) between the inner and outer lines. One manifold is provided forward and one aft in the vehicle. A complete ring manifold was used to eliminate trapped or stagnant propellant regions. Usage of any engine will result in propellant flow through both sides of the manifold, circulating all propellant in the manifold with each engine firing. The outer line is maintained at approximately 520° by the vehicle surroundings. The inner line, however, is chilled to 40°R during filling. Both angular and linear compensators are provided to accommodate inner line thermal contraction.

LIQUID H2 FORWARD MANIFOLD INSTALLATION

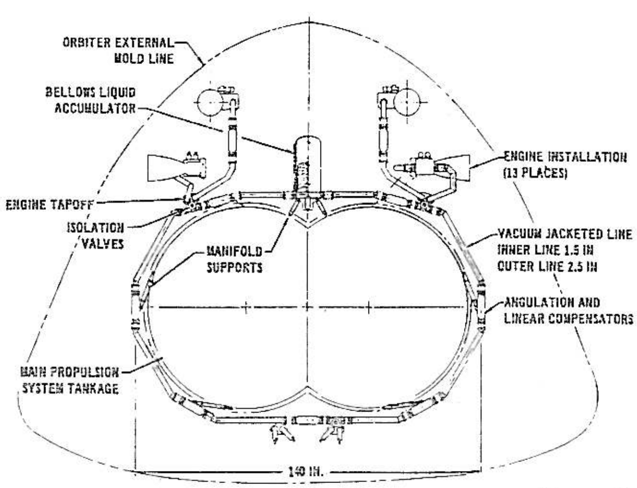


Figure 9

A thermal model was prepared to evaluate, in detail, significant heat transfer effects and is summarized in Figure 10. Heat input considerations included heat leak through inner line spacers (required to prevent MLI crushing) and axial conduction down the lines. Aluminum feed lines were used to conduct the incoming heat away as rapidly as possible from heat short locations and minimize local peak temperatures. Conduction through the liquid was neglected as it contributed little to the thermal characteristics. Real fluid and line properties were utilized in the analysis. The distribution system model considered the main supply line from the propellant tank to the manifold and included heat shorts at junctions with branch feed lines to each wing tip and engine group. A bellows tank was included to provide for fluid thermal expansion that resulted from heating.

FEED LINE THERMAL MODEL

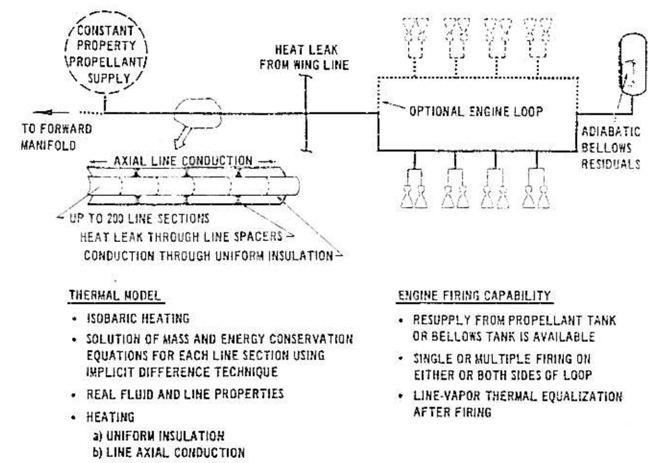


Figure 10

Initial propellant distribution system analyses were exploratory in nature to determine what, if any, modifications should be made in the feed system design to achieve low heating rates. The hydrogen temperatures obtained from these initial studies are shown in Figure 11. It was assumed that no engines were firing and no propellant was used. This is a conservative assumption which resulted in calculation of high localized heating rates. As shown, significant thermal spikes were obtained at the engine/feedline junctions. However, the maximum temperature at the end of one hour was 60°R. This temperature was much lower than originally anticipated for a condition in which the propellant was stagnant. One reason for the low temperature is that any heat input, near the tank or upstream in the line, locally expands the propellant, moving it away from the heat source, allowing migration of fresh, cooler propellant into heat short areas downstream. Significant heating is also evident at the wing lines tied directly to the main line, and from the inner line supports spaced, in this case, at 10 ft intervals along the line. Closer line support spacing was investigated to determine if there would be any significant changes in the temperature profile. As shown in Figure 12, decreasing the feed line spacing from 10 to 2-1/2 feet does not significantly change the maximum temperature encountered, although it does slightly increase the bulk propellant temperature. Thus, closer line supports could be utilized with little temperature effect. Oxygen temperatures, shown in Figure 13, are similar to those of hydrogen except that the nominal temperature limit (225°R) is not encountered until much later. Again, temperature spikes are seen at each engine line junction and smaller spikes are evident where the wing line joins the main feed line and at each inner line spacer.

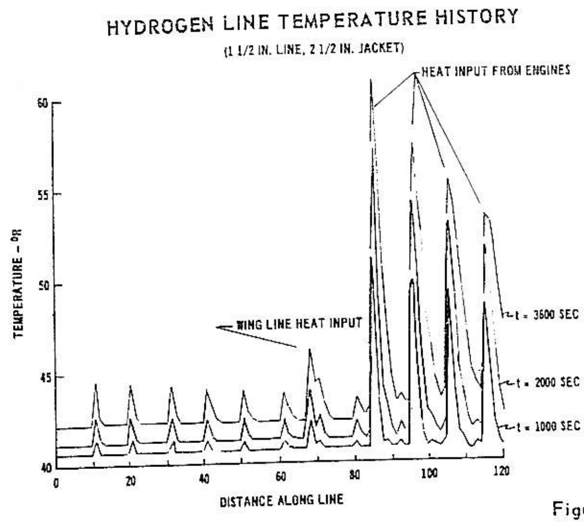


Figure 11

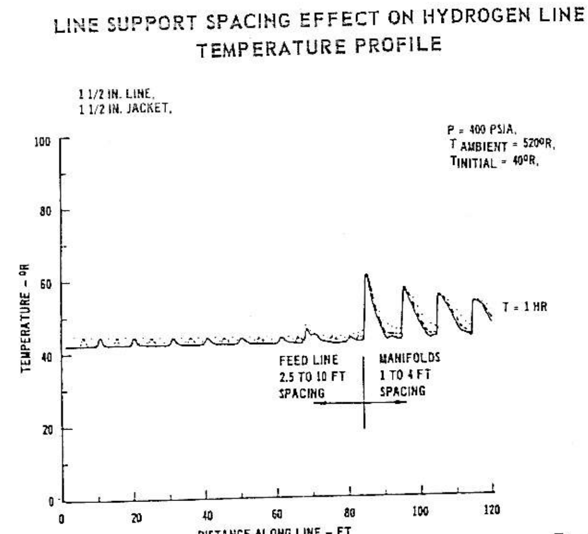


Figure 12

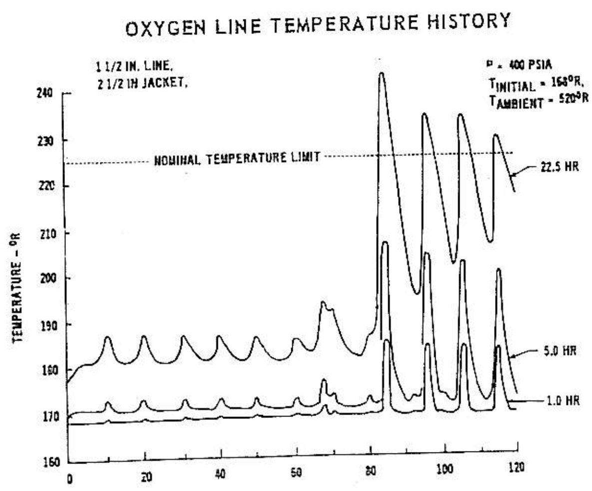


Figure 13

The preceding data indicated that some engine heat input control was mandatory if satisfactory hydrogen temperature limits were to be achieved. A simple tubular thermal standoff, similar to those commonly employed for hydrazine and hydrogen peroxide engines, was evaluated to determine its effectiveness. The standoff selected was a stainless steel tube 1/4 inch in diameter and 6 inches long. Pressure drop through this tube is on the order of 20 psi. Propellant heating at the engine valve junction with this type thermal standoff is shown in Figure 14.

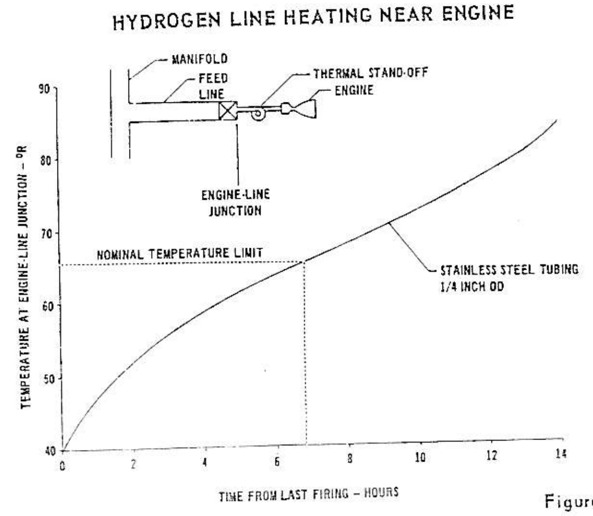


Figure 14

With the standoff, the temperature limit at the engine valve is not achieved until approximately seven hours as opposed to approximately one hour without the thermal standoff. The hydrogen distribution system temperature profile with thermal standoffs employed and propellant usage simulated are shown in Figure 15. The large temperature spikes associated with the engine line junctions have been removed and the only significant heating is from the wing line input and inner line supports. Maximum propellant temperatures from these and similar data are shown as a function of time in Figure 16 for two deadbands. The hydrogen temperature would stabilize at the limit of 65°R, 12 to 14 hours after the start of the mission for a vehicle attitude deadband of approximately 8°. Beyond this time, no further temperature increase is encountered as the propellant heat input is balanced by the heat removed through propellant usage. The oxygen temperature with thermal standoffs and engine usage are shown in Figure 17. These temperatures are much below the 225°R limit assigned, indicating there are no significant problems with oxygen thermal control.

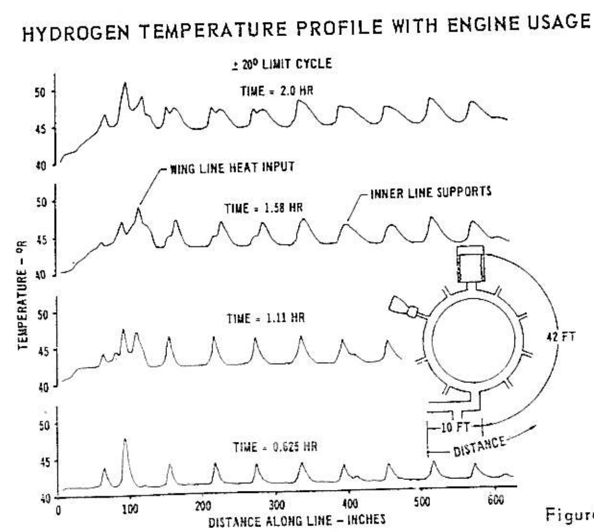


Figure 15

MAXIMUM HYDROGEN TEMPERATURE VS TIME

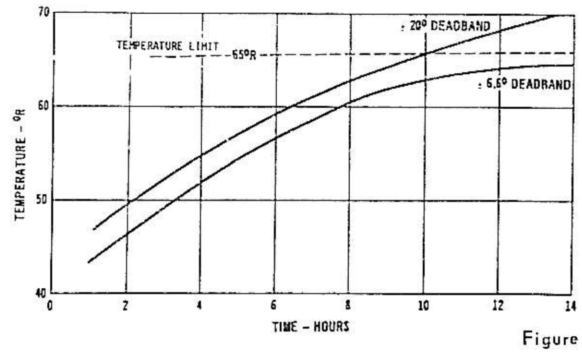


Figure 16

MAXIMUM OXYGEN TEMPERATURES VS TIME

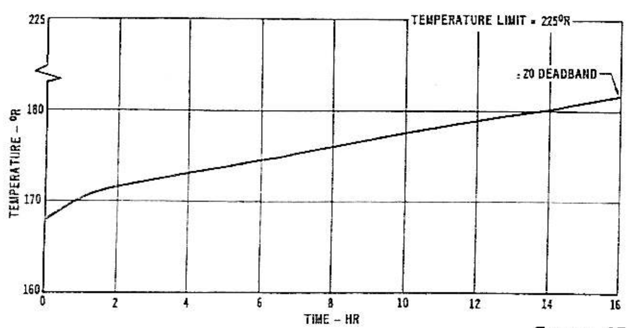


Figure 17

5. SYSTEM DESIGN

The preceding analyses show that thermal management of liquid propellants in the APS distribution system is feasible if proper attention is given to thermal insulation and isolation of major heat inputs such as thruster heat soakback. The remainder of the study effort was directed toward system design and sizing considerations. A hybrid system, using fully pressurized oxygen and pumped hydrogen, was selected as a baseline from which alternate implementation option studies were to proceed. This system approach is illustrated by Figure 18. The oxygen side of the system operates fully pressurized and a hydraulic motor pump/liquid accumulator combination is used for the hydrogen. Using fully pressurized system introduces a weight penalty, but since the oxygen is less than 20% of the total propellant volume, this penalty is small and allows simplification of system design and reduces development risk.

LIQUID SYSTEM BASELINE SUMMARY

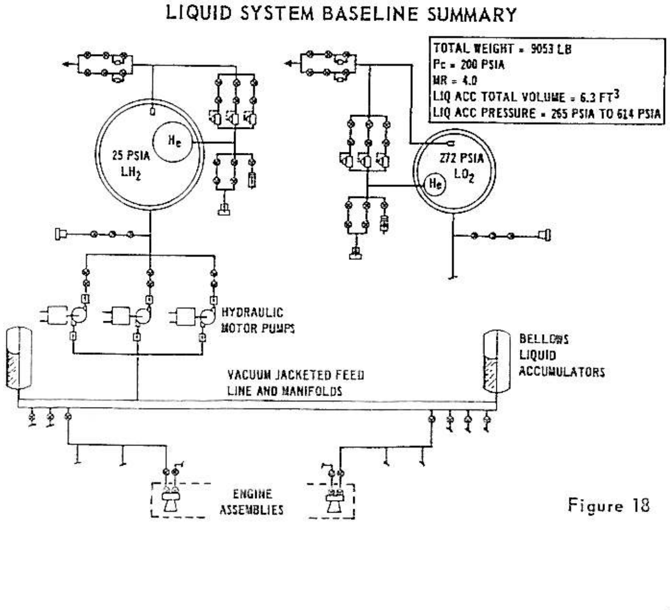


Figure 18

The baseline liquid system, Figure 18, which includes the component redundancy necessary to satisfy Shuttle failure criteria, weighs 9053 lb. Minimum system weight, shown in Figures 19 and 20, is provided at a 200–psia chamber pressure and a 4.0 mixture ratio. The effect of line sizing on system weight is shown in Figure 21. System weights are affected more by oxygen line diameter due to the relatively large residual oxygen weight. An oxygen feed line diameter of 1.0 in. was selected to minimize system weight. Since the weight sensitivity to line diameter was less with hydrogen, a 1.5 in diameter was selected on the basis of line heat transfer. This size provides additional heat capacity (larger liquid residuals) for a small weight penalty.

WEIGHT SENSITIVITY
TO CHAMBER PRESSURE

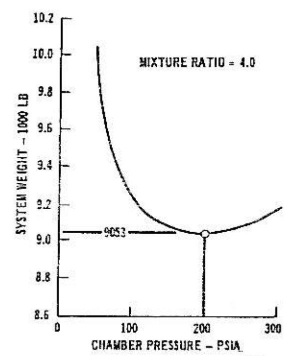


Figure 19

WEIGHT SENSITIVITY
TO MIXTURE RATIO

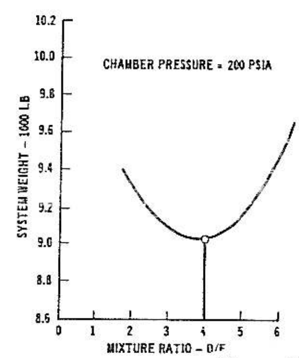


Figure 20

WEIGHT EFFECT OF LINE SIZING
HYDRAULIC HYBRID – LIQUID O_2/H_2 RCS

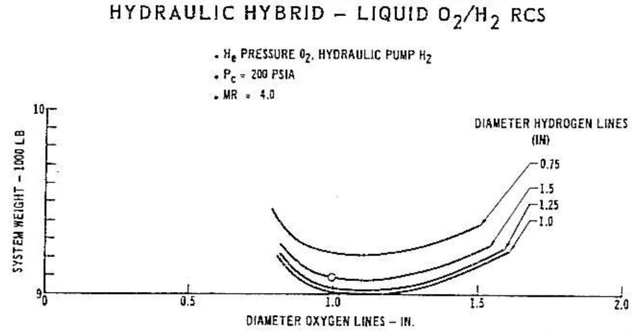


Figure 21

A baseline hybrid system weight breakdown is shown in Figure 22. A large portion of the system weight is associated with the propellant feed lines (728 pounds), the oxygen pressurization system, motors and pumps and liquid accumulators.

Several alternate design approaches to the baseline were investigated to reduce system weight and/or simplify system interactions. These included alternate pressurizations, feed lines, and pumping options. A comparison of feed line and pressurization options is given in Figure 23. The vacuum jacketed feed lines represent a large (23%) portion of the system inert weight. The use of nonjacketed lines could reduce system weight by 365 lb. However, without jacketed lines, the HPI would be exposed to potential handling and atmospheric damage and would, at least, require a soft purge bag for protection. Further investigation of the technology risks involved with nonjacketed lines is warranted before selection of this option.

The use of a motor driven oxygen pump would save 355 lb but this weight improvement is not justified when the additional

HYDRAULIC HYBRID – LIQUID O_2/H_2 RCS

- ENGINE MR = 4.0
- CHAMBER PRESSURE = 200 PSIA
- STORAGE TANK PRESSURE, O_2 = 272 PSIA
  H_2 = 25 PSIA
- H_2 LIQUID ACCUMULATORS, TEMPERATURE = 40°R
  PRESSURE = 265 TO 614 PSIA

COMPONENT	WEIGHT – LB	
	HYDROGEN	OXYGEN
PROPELLANT WEIGHT		
USABLE	1046	4185
RESIDUALS, LINES	22	351
TANKS	21	25
VENTED	194	13
TOTAL	1283	4574
PROPELLANT TANKAGE	381	315
PRESSURIZATION	97	262
MOTORS AND PUMPS	115	0
RCS PROPELLANT	35	
FEED LINES AND INSULATION	230	230
COMPENSATORS	139	139
LIQUID ACCUMULATORS		
TANK	130	0
PRESSURIZATION	22	0
ISOLATION VALVES (28)	60	60
ENGINES (36)	981	
TOTAL SYSTEM WEIGHT	9053	

Figure 22

COMPARISON OF ALTERNATE LIQUID SYSTEM CONCEPTS

SYSTEM	RELATIVE WT (LB)	APU INTERACTIONS	REMARKS
BASIC HYBRID	0	REQUIRES 40–50 APU CYCLES, POWER SATISFIED BY ONE APU UNIT	HIGH PERFORMANCE SYSTEM, USING HYDRAULIC POWER. REQUIRES POSITIVE DISPLACEMENT PUMP DEVELOPMENT
HYBRID-NO VACUUM JACKET	–365	REQUIRES 40–50 APU CYCLES	VACUUM JACKET REMOVAL REDUCES WEIGHT BUT WOULD EXPOSE HPI TO HANDLING AND ATMOSPHERIC DAMAGE. FURTHER STUDY WARRANTED.
ALL HYDRAULIC PUMPED H_2 AND O_2	–355	REQUIRES 40–50 APU CYCLES BUT 2 APU UNITS REQUIRED TO FURNISH HORSEPOWER	COMPLEXITY ASSOCIATED WITH PUMPING OXYGEN IS NOT WARRANTED BY THE WEIGHT POTENTIAL INVOLVED.
HYBRID–O_2 PRESS IN H_2 TANK	–127	REQUIRES 40–50 APU CYCLES AND ONE APU UNIT	USE OF SINGLE H_e SYSTEM IN H_2 TANK WITH PASSIVE THERMAL CONDITIONER FOR O_2 APPEARS ATTRACTIVE.
FULLY PRESSURIZED (100 PSIA CHAMBER PRESSURE)	+1545	NONE	SIMPLE SYSTEM, BUT HEAVY FOR LARGE IMPULSE LEVELS. COULD BE ATTRACTIVE FOR INTERIM SYSTEM

Figure 23

complexity and development cost is considered. Storing the oxygen pressurant in the hydrogen tank will save 127 lb and would require the addition of a passive, structural heat exchanger to heat a small quantity of helium to liquid oxygen temperatures.

The remaining option, a fully pressurized system, would result in an excessive weight penalty of 1545 lb. Although this weight penalty is prohibitive for operational use, the concept could be used on an interim basis for the first few flights and updated later to a high performance configuration.

In addition to the options described above, alternate pump designs were considered. The choice of pump type and power source was not readily apparent from system considerations alone. Various design approaches were available ranging from high-speed, high flowrate designs to small pumps operating for relatively long durations. Also, alternate design points were available to reduce pump power requirements, thereby simplifying pump designs and/or reducing the Auxiliary Power Unit (APU) interface complexity. This could be accomplished either by increasing the maximum accumulator capacity, to lower pump flow rate requirements, or by lowering accumulator pressure.

System weight as a function of pump horsepower and maximum accumulator pressure is shown in Figures 24 and 25. The optimizations shown are for the design chamber pressure of 200 psia, and also for a fixed accumulator capacity which in turn fixes pump flow rate. With pump flow rate fixed, pump horsepower is a function of accumulator pressure only. The data presented in Figures 24 and 25 reflect the effect of variations in accumulator pressure on system weight in terms of both the actual pressure (Figure 24) and the pump horsepower required to produce that pressure (Figure 25). As shown, minimum weight is achieved at 600 psia accumulator pressure and the corresponding pump horsepower is 127.

EFFECT OF ACCUMULATOR PRESSURE ON SYSTEM WEIGHT

EFFECT OF PUMP HORSEPOWER ON SYSTEM WEIGHT

- He PRESSURE O_2, HYDRAULIC PUMP H_2
- BLOWDOWN LIQUID ACCUMULATORS
- MR = 4.0
- Pc = 200 PSIA

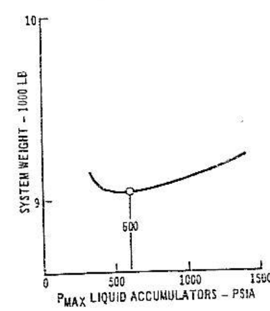


Figure 24

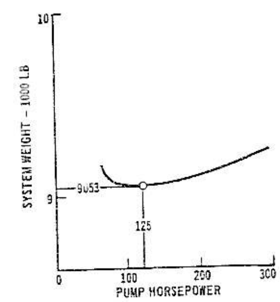


Figure 25

As described in Section 2, the liquid RCS concept allows some flexibility in pump and accumulator design, e.g., system single burn total impulse requirements can be satisfied by a high flow rate pump or by the combined pump and accumulator flow capability must be the equivalent of 250 lb thrust and, at this pump flow rate, the accumulator must be sized to provide 157,000 lb-sec to meet the largest maneuver requirements. After the maneuver, the pumps would continue to operate, recharging the accumulators to full capacity in approximately 17 minutes. Increases in pump flow capacity result in proportional reductions in accumulator capacity. Figure 26 provides the relationships between pump flow and accumulator capacity necessary to satisfy both system thrust and maximum single burn impulse requirements. An increase in the number of pump starts is inherently associated with reductions in accumulator capacity. In the extreme, with no liquid accumulator, pump operation would be required for each engine firing, imposing severe demands on pump cycle life. However, even small liquid accumulators effect marked reductions in the number of pump cycles required. Figure 27 relates the number of pump-accumulator operating cycles in each mission to the liquid accumulator storage capacity.

The effects of pump-accumulator resizing per the above criteria are shown in Figure 28 for other chamber pressures (the 200 psia curve is a duplicate of that shown in Figure 24). Again, as in Figure 24, accumulator capacity and pump flow rate are fixed and the pump horsepower is related to accumulator pressure. Hence, along each constant chamber pressure line, accumulator pressure is variable. The dotted line of Figure 28 describes the minimum system weight locus as a function of horsepower for the fixed accumulator capacity used. Similar weight trends,

PUMP-ACCUMULATOR SIZING REQUIREMENTS

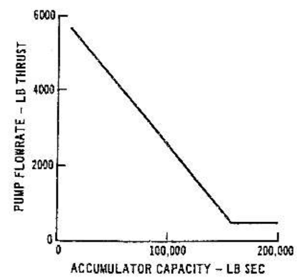


Figure 26

PUMP OPERATING CYCLE REQUIREMENTS

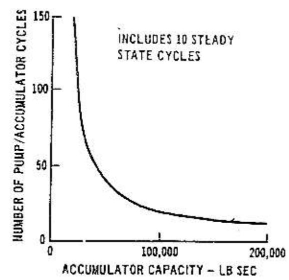


Figure 27

EFFECT OF CHAMBER PRESSURE ON POWER REQUIREMENTS

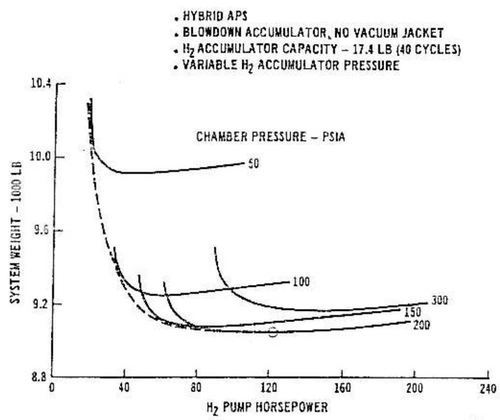


Figure 28

EFFECT OF ACC CAPACITY AND PUMP POWER REQUIREMENTS ON SYSTEM WEIGHT

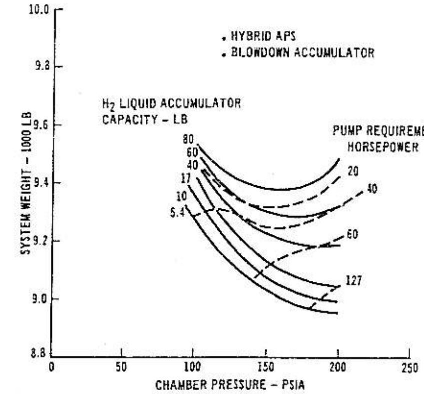


Figure 29

TYPICAL HYBRID SYSTEM OPTIONS

P_c (PSIA)	P_{MAX} (PSIA)	PUMP HP	ACCUMULATOR CAPACITY (LB)	CYCLES	SYSTEM WEIGHT (LB)
200	600	127	17	40	9053
200	400	70	17	40	9153
200	400	20	60	8	9450
200	600	20	64	8	9320

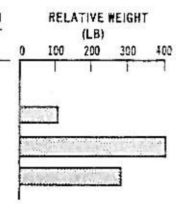


Figure 30

obtained for other accumulator sizes, are shown in Figure 29. These data indicate that the liquid system can accommodate a large range of pump design points for a small weight penalty. For example, the design pump power can be reduced from 127 to 20 hp, for a 100 lb weight penalty, by increasing accumulator capacity from 17 to 70 lb. This could be accomplished at minimum weight by also lowering chamber pressure from 200 to 150 psia but this would reduce the allowable heat transfer (lower heat capacity margin) and would not save sufficient weight to be recommended. Instead, the alternate design points shown in Figure 30 are most attractive. These data show that reductions in accumulator pressure, from 600 to 400 psia, and increases in accumulator capacity, from 17 to 60 lb can greatly reduce pump power requirements for a small weight penalty. Increases in accumulator capacity are particularly attractive in that the number of cycles required to provide the attitude control impulse requirements can be reduced from 40 to 8 cycles. This would reduce the total cycle life requirements (including 10 steady-state cycles) from 50 to 18 per mission, thereby simplifying the APU interface.

Considerable design flexibility is available since, for a small weight penalty, a large range of pump requirements can be accommodated. This could greatly relieve the pump design and development program by allowing relaxation of technology risk through an overall system design compromise. Since the weight difference between design point options was small, a separate subcontract was issued to Pesco Products to evaluate alternate pump designs. Both centrifugal and positive displacement pump designs were considered for outlet pressures from 400 to 600 psia and pump flowrates equivalent to thrust levels from 250 lb to 5750 lb. Also, both electric and hydraulic power sources were investigated.

A comparison of pump types is shown in Figure 31. Centrifugal pumps were shown to be heavy and inefficient at the lower speeds associated with either hydraulic (10,000 rpm) or electric motors (22,000 rpm). To be acceptable, pump speeds near 100,000 rpm would be required. For this reason, centrifugal pumps for this application necessitate a hot gas turbine drive. However, a liquid system pump would present considerably less technology risk than the previous gaseous system turbopump. With liquids, the accumulator capacity can be increased for a small weight penalty, thereby allowing slower pump acceleration and increased bearing life. As a contrast, in a turbopump gaseous system, the 5-sec start time associated with state-of-the-art bearing acceleration limits in long life machines (25,000 to 30,000 rpm/sec) would result in a 1000 lb weight penalty. In addition, the liquid system can accommodate a wider pump performance range, and it would be much easier to integrate the pump into the system than in gaseous systems where pump performance can also affect operation of the propellant heat exchangers. In fact, no significant technology problems are anticipated with liquid APS hydrogen turbopumps.

In addition to the centrifugal pumps described above, piston, gear and vane type, positive displacement pumps were also evaluated. These pumps require slower operating speeds, on the order of 10,000 rpm, and are more adaptable to hydraulic and electric drives. The piston pumps were found to be heavy, due to the large flow capacities associated with low hydrogen density. This required large pistons and heavy rotors which limited pump operating speed. Both gear and vane pumps appear to be better suited for hydrogen pumping than piston pumps. These pumps are lighter and simpler than the piston type. The preferred pump type is a vane pump, similar to a hydrogen vane

CANDIDATE LH_2 PUMP CONCEPTS
(PESCO PRODUCTS)

TYPE	WEIGHT LB	EFFICIENCY PERCENT	COMMENTS
CENTRIFUGAL	307	33	HEAVY AND INEFFICIENT AT LOW SPEEDS (20,000 TO 40,000 RPM). USE OF CENTRIFUGAL PUMPS WILL BE LIMITED TO HOT GAS TURBINE DRIVES
PISTON	460	66	HEAVY DUE TO LARGE PISTONS AND ROTOR REQUIRED TO ACCOMMODATE HYDROGEN FLOW RATE. REQUIRES CASE DRAIN TO REMOVE PROPELLANT LEAKAGE PAST PISTON SEALS
GEAR	194	50	LOW COST, SIMPLE DESIGN. HEAVIER THAN VANE PUMPS
VANE	117	60	LIGHTWEIGHT, EFFICIENT DESIGN FEASIBILITY OF DESIGN FOR LH_2 HAS BEEN DEMONSTRATED

BASED ON 600 PSI HEAD RISE, 2.65 LB/SEC H_2 FLOW, HYDRAULIC DRIVE

Figure 31

pump designed and demonstrated by Pesco Products. This work was part of a General Electric Company contract, "Final Pumping System Liquid Hydrogen/Liquid Methane, J85 Control System," performed for NASA Lewis Research Center. Considerable material development was accomplished, and no critical, new technology effort is anticipated.

The effect of power source on system weight is summarized in Figure 32. Except for the centrifugal turbopump concept, which is shown to be the lightest, the weights are for vane type positive displacement pumps. A hydraulic powered pump is weight competitive and could be provided for 150 to 300 lb, depending on pump outlet pressure. Electric motor concepts, limited to the APU electrical output of 15 KW (20 hp), would weigh 300 to 500 lb more than the turbopump system. Finally, a small DC motor operating from fuel cell power would result in a penalty of approximately 800 lb.

EFFECT OF PUMP POWER SOURCE ON SYSTEM WEIGHT

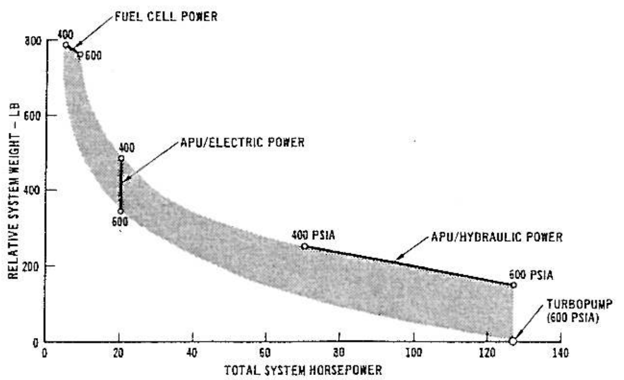


Figure 32

Based on these data, the turbopump system is the preferred concept. A simplified turbopump system schematic is shown in Figure 33. This is the lightest and simplest system and, most importantly, would operate independent of the APU. Thus, the pumps could be operated at any time, the APU could remain inactive throughout the orbital phase of the mission, and resizing of APU electric or hydraulic systems would not be required. The second choice would be a hydraulic motor operated, vane pump which is attractive from a weight standpoint, but would interface significantly with the APU.

LIQUID HYDROGEN/LIQUID OXYGEN RCS
SCHEMATIC

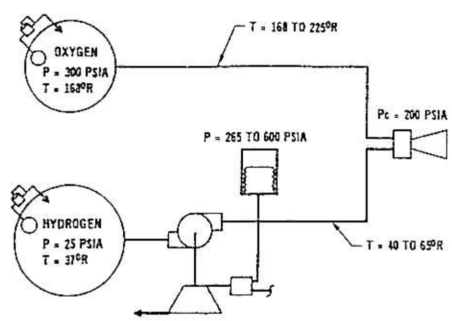
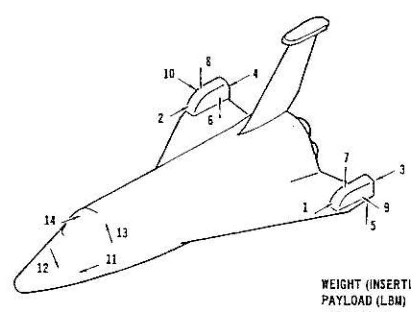


Figure 33

6. SYSTEM COMPARISON AND CONCLUSIONS

The effort described in the preceding sections was based on a fully reusable orbiter with internal tankage. Other phases of the contract effort (Reference 7) considered storable monopropellants for external tank orbiter vehicles. A comparison of the LOX/LH_2 concept with both monopropellant and bipropellant systems was made to determine their relative weights. Two external tank orbiter vehicles, corresponding to a simple, low risk Mark I vehicle to be improved later to a higher performance Mark II configuration, are defined in Figure 34. The impulse requirements for these vehicles range from 1.3 (10^6) lb-sec for the Mark I vehicle to 1.7 (10^6) lb-sec for the Mark II vehicle.

BASELINE ORBITER FOR STORABLE RCS STUDIES



	MARK I	MARK II
WEIGHT (INSERTION) (LBM)	207,200	228,700
PAYLOAD (LBM)	45,000	65,000
LENGTH (FT)	120.7	120.7
NUMBER OF THRUSTERS	34	42
THRUSTER THRUST (LB)	600	600
TOTAL IMPULSE (LB-SEC)	1.343×10^6	1.693×10^6

Figure 34

The study results suggest two approaches to the evolution of a high performance system for a Mark II vehicle. Figure 35 shows these approaches. One approach starts with a gaseous hydrogen-liquid oxygen system. The system uses liquid oxygen because the weight penalties associated with avoidance of oxygen pumps are small and liquid oxygen distribution presents no essential problem. Gaseous hydrogen is used because the engine ignition requirements are state-of-the-art, similar to the Pratt and Whitney RL–10 engine. This system could be updated for a Mark II concept by either increasing the gas generator operating temperature (Reference (8)), thereby increasing system efficiency, or by utilizing a liquid turbopump system which eliminates the gas generator and gaseous accumulator. The decision as to which means of improvement is most attractive could be made on the basis of the relative status of technology demonstration programs in the areas of liquid ignition and high temperature heat exchangers.

HYDROGEN/OXYGEN RCS CONCEPT SUMMARY

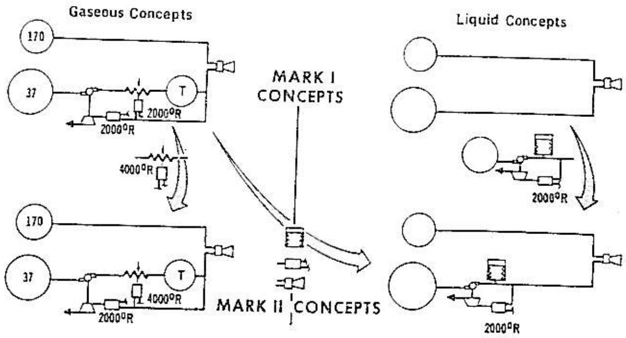


Figure 35

The second approach shown in Figure 35 starts with a simple, fully pressurized liquid-liquid system for the Mark I vehicle. Later a turbopump and liquid accumulator could be added to the hydrogen side providing a high performance Mark II design.

A weights comparison for systems integrally mounted in the vehicle is shown in Figure 36. Monopropellant hydrazine and storable bipropellant system weights were taken from the preliminary Task C effort (Reference (9)). The Mark I and Mark II total impulse values are noted. For the Mark I vehicle, use of fully pressurized liquid oxygen/liquid hydrogen would be slightly heavier than a hydrazine RCS system, but this system could be updated to a pumped liquid hydrogen/pressurized liquid oxygen system for the Mark II vehicle, and would be approximately 3,000 pounds lighter than the hydrazine system. The comparison shown in Figure 36, is for integrally mounted systems. However, the toxic hydrazine and storable bipropellant systems are best designed for installation in removable modules to allow rapid removal of the propellants and transportation to a remote decontamination site. Since the oxygen/hydrogen propellants are not toxic, the modular approach is unnecessary. A more valid comparison then would be between integrally installed cryogenic concepts and modular storable concepts. Weight comparisons on this basis are shown in Figure 37.

It should be noted that the total impulse values for both Mark I and Mark II are higher for modular systems due to control cross coupling effects that result with modular system engine installations. Further penalties are assessed against the modular

COMPARISON OF INTEGRAL RCS CONCEPTS

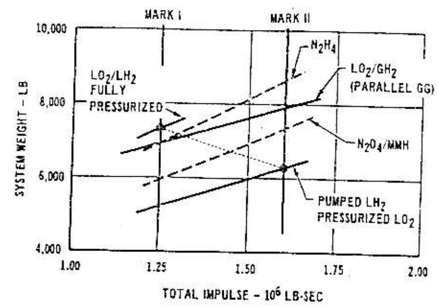


Figure 36

COMPARISON OF MODULAR STORABLE RCS CONCEPTS WITH INTEGRAL CRYOGENIC CONCEPTS

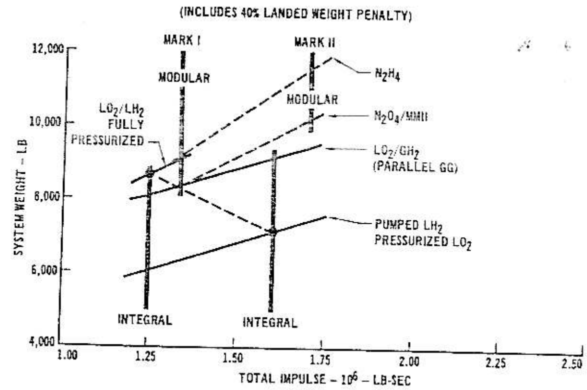


Figure 37

and integral systems because of the effect of increased structural weight. Vehicle studies show that weight penalty equal to 40 percent of the total inert system weight is required to account for resizing the aerodynamic surfaces, landing gear, etc. With these effects included, the fully pressurized liquid oxygen/liquid hydrogen system for the Mark I vehicle is shown in Figure 37 to weigh less than a hydrazine system and only 800 lbs more than a storable bipropellant system. Subsequent development of a pumped liquid hydrogen system for the Mark II vehicle would increase the weight savings to nearly 4,000 pounds. Thus, liquid oxygen/liquid hydrogen systems offer large potential weight savings for the Space Shuttle.

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