Cryogenic Liquid O_2/H_2 Reaction Control Systems for Space Shuttle

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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

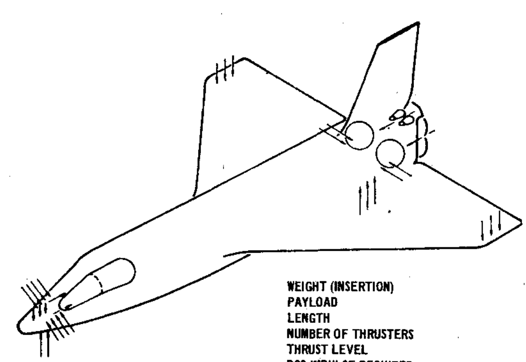
TO provide the technology base required for the Space Shuttle design, NASA has sponsored several technology programs related to Reaction Control Systems (RCS). Among these were a series of studies to provide system design data for selection of preferred system concepts and delineation of requirements 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 Refs. 1–4. 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.

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.

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 liquid O_2/H_2 (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 Fig. 1. Vehicle characteristics are based primarily on the results of MDAC–E studies of fully reusable orbiters and boosters defined in Ref. 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 Sec. 6. These show RCS weight at design requirements corresponding to smaller orbiter configurations of the type designed to use external, main engine tankage.



WEIGHT (INSERTION)	331,780 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

Fig. 1 RCS design study orbiter vehicle.

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Reference 6 provides the detailed analyses and rationale used to develop the RCS requirements tabulated in Fig. 1. The RCS uses 33 engines at 1150 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 fps. 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 Ref. 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 Ref. 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 flow rates 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 Ref. 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; above average demands are met by accumulator supplementation.

3. Liquid System Operation

The liquid hydrogen storage conditions are shown in Fig. 2. 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 Fig. 3 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 lb/ft^3.

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 Fig. 4 that at all pressure levels of interest, significant hydrogen density changes to levels below 3.0 lb/ft^3 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

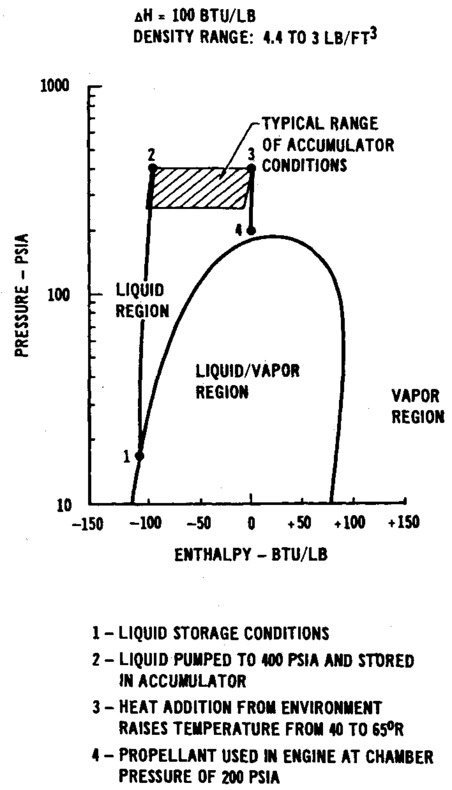


Fig. 2 Liquid hydrogen storage conditions.

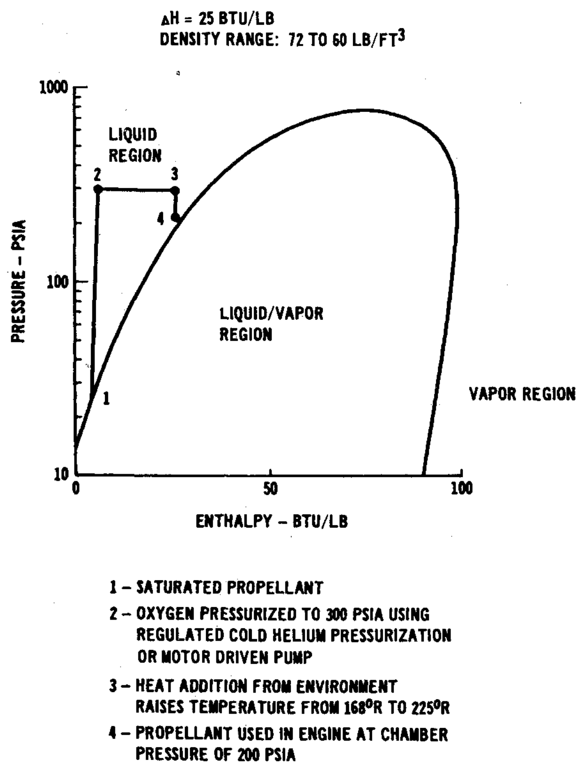


Fig. 3 Liquid oxygen storage conditions.

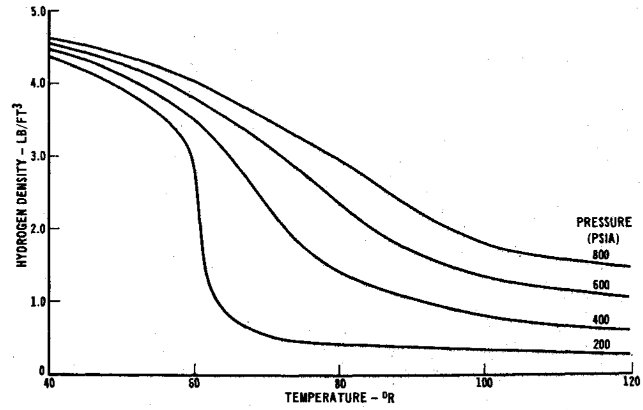


Fig. 4 Liquid hydrogen density variations.

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 towards detailed thermal analysis.

The second question concerns the feasibility of liquid ignition in a pulse mode engine, since the ignition temperature limits are significantly lower than those previously considered. A detailed examination of ignition was beyond the scope of the current study, however, 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. In fact, 167 ignition tests have been successfully conducted by the NASA Lewis Research Center under contract NAS 3-16775, (Ref. 7). No propellant ignition problems were encountered; the only ignition failures were caused by electrical failures.

The storage conditions shown in Figs. 2 and 3 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 Fig. 5. 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 temperature increases, the injection velocity increases, but again, if the hydrogen temperature is limited to 65°R, these changes are minimal.

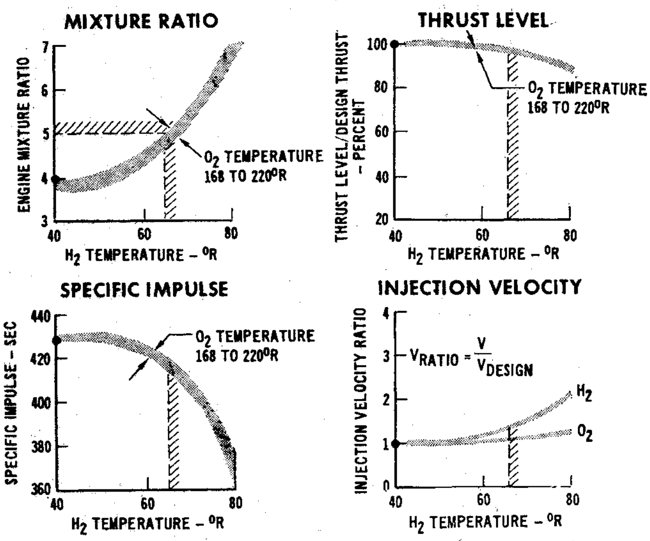


Fig. 5 Effect of propellant temperature on engine performance.

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 Fig. 6. The smallest practical hydrogen usage rate would be associated with a $\pm 20°$ 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 $\pm 6°$) 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 ½ lb/hr. (Recent Shuttle vehicle studies have shown that a vehicle deadband of $\pm 5°$ is more desirable than a $\pm 20°$ deadband. 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 Fig. 6 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 Fig. 7 was designed. The manifold consists of

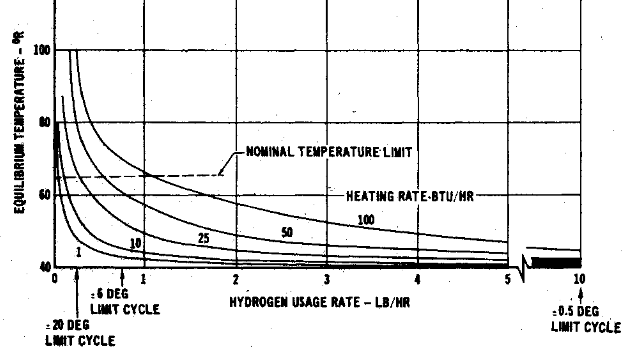


Fig. 6 Effect of propellant consumption on hydrogen temperature.

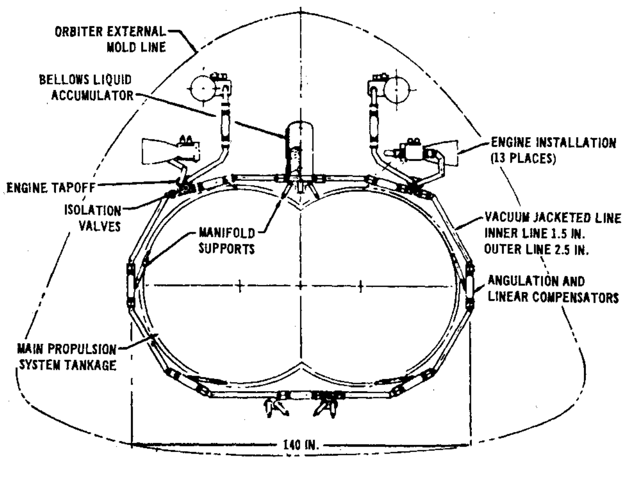


Fig. 7 **Liquid H_2 forward manifold installation.**

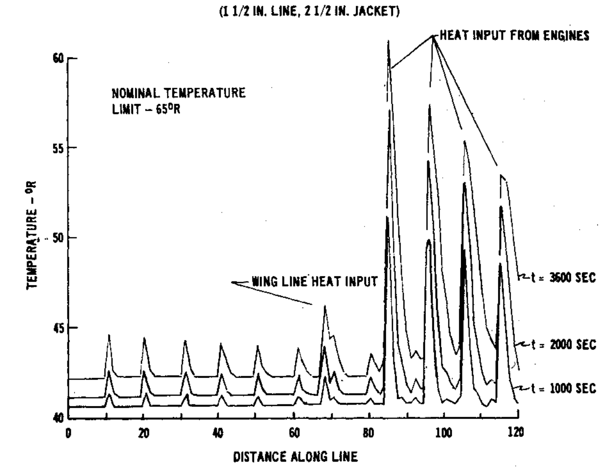


Fig. 9 **Hydrogen line temperature history**

vacuum jacketed propellant lines with multilayer insulation (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°R 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.

A thermal model was prepared to evaluate, in detail, significant heat transfer effects and is summarized in Fig. 8. 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.

Initial propellant distribution 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 Fig. 9. 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/feed line 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 Fig. 10, decreasing the feed line spacing from 10 to 2½ ft 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 temperature profiles are similar to those of hydrogen except that the nominal temperature limit (225°R) is not encountered for times of 15–20 hr. Again, temperature spikes occur 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.

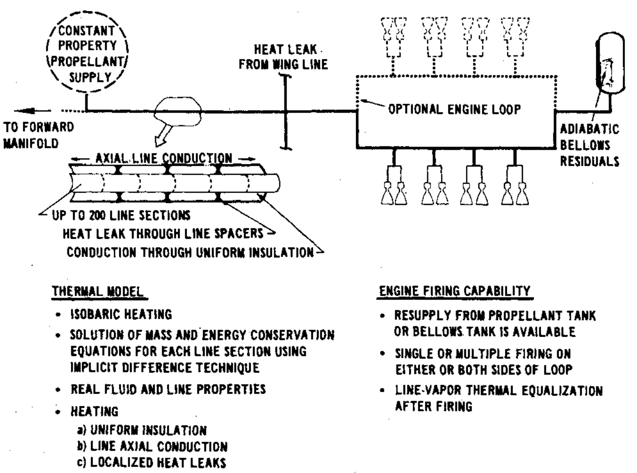


Fig. 8 **Feed line thermal model.**

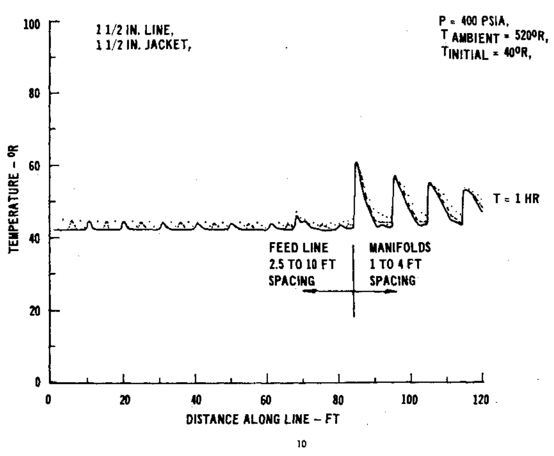


Fig. 10 **Line support spacing effect on hydrogen line temperature profile.**

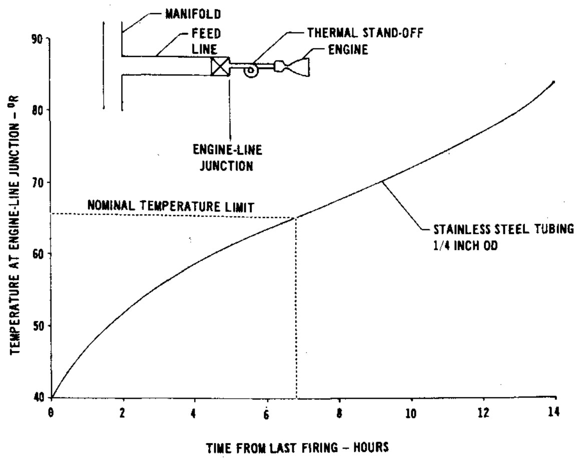


Fig. 11 Hydrogen line heating near engine.

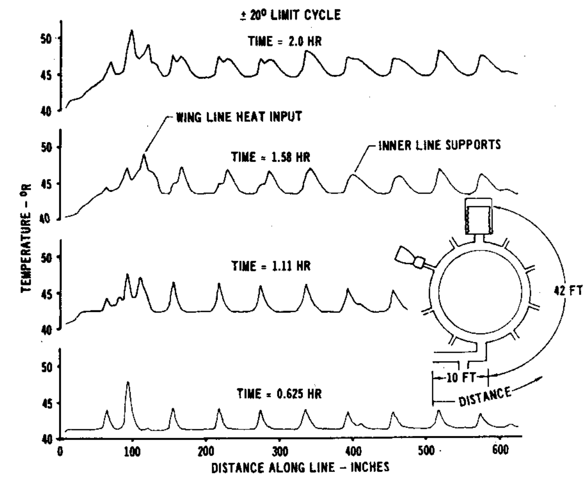


Fig. 12 Hydrogen temperature profile with engine usage.

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 $\frac{1}{4}$ in. in diameter and 6 in. 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 Fig. 11.

With the standoff, the temperature limit at the engine valve is not achieved until approximately 7 hr as opposed to approximately 1 hr without thermal standoff. The hydrogen distribution system temperature profile with thermal standoffs employed and propellant usage simulated are shown in Fig. 12. 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. The hydrogen temperature would stabilize at the limit of 65°R, 12 to 14 hr 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. For larger deadbands the hydrogen temperature would exceed the 65°R limit. Similar calculations for the oxygen line with thermal standoffs and engine usage predict temperatures much below the 225°R limit assigned, indicating there are no significant problems with oxygen thermal control.

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 Fig. 13. The oxygen side of the system operates fully pressurized and a hydraulic motor pump/liquid accumulator combination is used for the hydrogen. Using a 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.

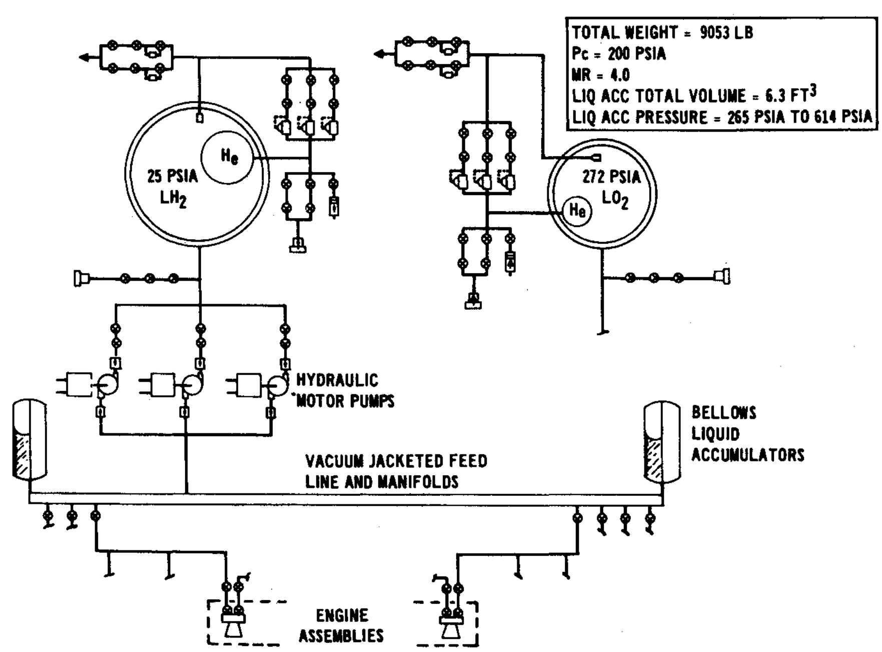


Fig. 13 Liquid system baseline summary.

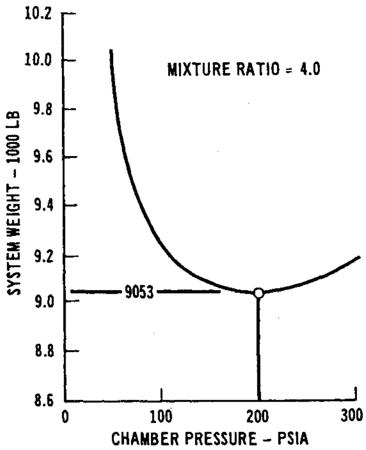


Fig. 14 Weight sensitivity to chamber pressure

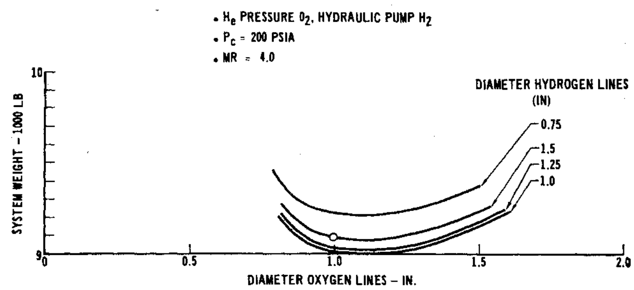


Fig. 16 Weight effect of line sizing hydraulic hybrid-liquid O_2/H_2 RCS

The baseline liquid system, Fig. 13, which includes the component redundancy necessary to satisfy Shuttle failure criteria, weighs 9053 lb. Minimum system weight, shown in Figs. 14 and 15, 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 Fig. 16. 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. diam was selected on the basis of line heat transfer. This size provides additional heat capacity (larger liquid residuals) for a small weight penalty.

A baseline hybrid system weight breakdown is shown in Fig. 17. A large portion of the system weight is associated with the propellant feed lines (738 lb), 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, including the demand imposed on the Auxiliary Power Unit (APU), is given in Fig. 18. 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 MLI 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 335 lb but this weight improvement is not justified when the additional complexity and development cost is considered. Storing the oxygen pressurant in the hydrogen tank will save 127 lb but would require the addition of a small, passive, structural heat exchanger to heat 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 flow rate 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 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.

The turbopump system is the preferred concept. A simplified turbopump system schematic is shown in Fig. 19. 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

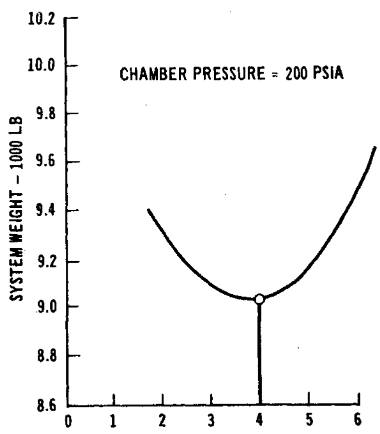


Fig. 15 Weight sensitivity to mixture ratio

- 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
APU 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	

Fig. 17 Hydraulic hybrid—liquid O_2/H_2 RCS.

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

Fig. 18 Comparison of alternate liquid system concepts.

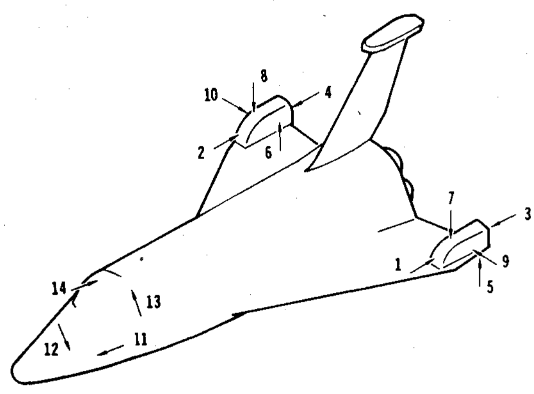
not be required. The second choice would be a hydraulic motor operated, vane pump which is attractive from a weight standpoint, but would interface significantly with the APU.

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 (Ref. 8) 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 Fig. 20. 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.

A weights comparison for modular storable systems and for oxygen/hydrogen systems integrally mounted in the vehicle is shown in Fig. 21 for the Mark II vehicle. Monopropellant hydrazine (N_2H_4) and storable bipropellant [N_2O_4/monomethylhydrazine (MMH)] system weights were taken from the preliminary Task C effort (Ref. 9). 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.

It should be noted that the total impulse values are higher for modular systems due to control cross coupling effects that result with modular system engine installations. Further penalties are assessed against both the modular and integral systems because of the effect of increased structural weight. Vehicle studies show that weight penalty equal to 40% of the total inert system weight is required to account for resizing the aerodynamic surfaces, landing gear, etc. With these effects included, the liquid oxygen/liquid hydrogen system for the Mark II vehicle is shown in Fig. 21 to weigh from 3500 to 4500 lb less than comparable storable propellant systems. Thus, liquid oxygen/liquid hydrogen systems offer large potential weight savings for the Space Shuttle.



	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

Fig. 20 Baseline orbiter for storable RCS studies.

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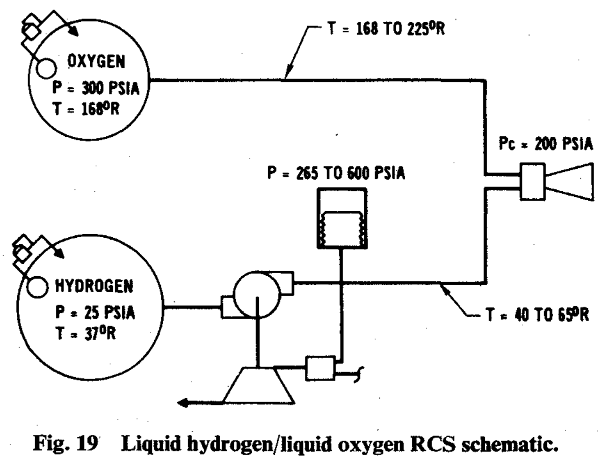

Fig. 19 Liquid hydrogen/liquid oxygen RCS schematic.

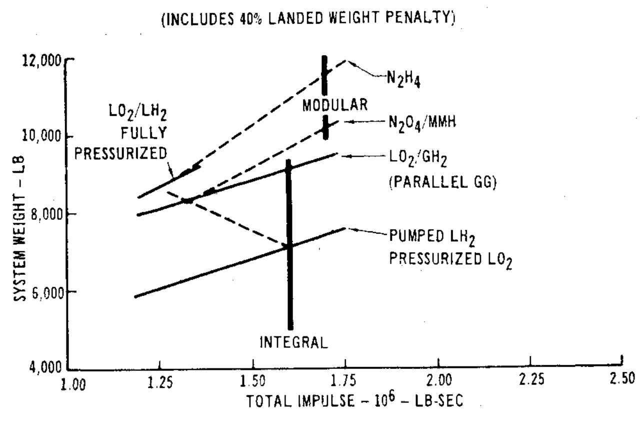

Fig. 21 Liquid oxygen/liquid hydrogen system for Mark II vehicle.

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Development of a 5-cm Flight-Qualified Mercury Ion Thruster

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A 5-cm structurally-integrated ion thruster (SIT-5) has been advanced from a predesign concept to a flight-qualified system for application to attitude control and stationkeeping of synchronous satellites. All elements of the system are structurally and thermally matched with one another and also with respect to a spacecraft with which the system will ultimately be associated. A thorough program of component and system testing has qualified the SIT-5 for operation at a thrust $T = 2.1$ mN with a specific impulse $I_{sp} = 3040$ sec and a total elecrical power consumption $P_T = 72.2$ w. With its vectorable beam-extraction system, thrust can be deflected electrostatically by up to 10° in any azimuthal direction. The system is 31.2 cm long by 12.1 cm in diameter. It weighs 2.2 kg including tankage for 6.8 kg of mercury propellant which is sufficient for over 25,000 hr of full thrust operation. A launch-enviroment test has structurally qualified the system for shock (30 g), sinusoidal (9 g) and random (19.9 g rms) vibrations.

Introduction

MISSION and system analyses have demonstrated the suitability of electron-bombardment ion thrusters for attitude control and stationkeeping of synchronous satellites where low system mass and long life are major requirements.[1–3] In initial research investigations with mercury bombardment thrusters at the NASA Lewis Research Centre (LeRC), a thrust level of 2.2 mN was chosen as being representative of a number of applications for satellites weighing 220 kg to 680 kg. At this thrust level, the duty cycle to provide the North-South stationkeeping function is estimated[2] at about 0.5 for each of two thrusters operated back-to-back, and at 0.01 for the attitude control function. With these duty cycles, a satellite lifetime of 3 yr requires an operating lifetime for the stationkeeping thruster of about 13,000 hr and includes 1000 on-off cycles.

Thruster operation at LeRC reported in 1966 by Kerslake et al.[4] demonstrated the suitability of an experimental thruster of 5-cm anode diameter. With an oxide-coated brush cathode, the thruster system was endurance tested for 1553 hr. At a specific impulse $I_{sp} = 3050$ sec, this system generated a thrust $T = 2.9$ mlb with an over-all power-to-thrust ratio of 50 w/mN.

In subsequent investigations at LeRC by Reader et al.,[5] somewhat more efficient operation was demonstrated at lower specific impulse ($I_{sp} = 1800$ sec) with a 5-cm thruster which utilized technology developed for the NASA Space Electric Rocket Test II (SERT II),[6,7] including the use of hollow cathodes[8] for both the discharge-chamber and neutralizer electron sources. Both the discharge and neutralizer emitters were of the enclosed hollow cathode type.[9] The neutralizer cathode was operated successfully at an equivalent mercury flowrate below 2 ma while coupled to an operating thruster beam. The total discharge-chamber flow was directed through the discharge cathode, and post-cathode propellant deversion was provided by ports passing through the walls of the cathode-cup polepiece. All thruster tests were conducted with a single glass-coated accelerator grid.[10]

Based on the encouraging results of the LeRC research program, a development effort was initiated at Hughes Research Labs. (HRL) to advance the 5-cm mercury-bombardment thruster subsystem to flight-qualified status. Progress on this effort was first reported by Nakanishi et al.[11] in conjunction with the simultaneous investigation at LeRC of a 5-cm prototype thruster of different design. The intent of the present paper is to report on development by HRL of the 5-cm flight qualified unit which has now been completed.

Technical Program

A 5-cm structurally integrated ion thruster (SIT-5) has been advanced from a predesign concept to a flight-qualified system for application to attitude control and stationkeeping of synchronous satellites. In design of the SIT-5 system, special

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