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Technology Status of a Fluorine-Hydrazine Propulsion System for Planetary Spacecraft

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The basic technology exists and a systems integration program is well underway to allow incorporation of a fluorine-hydrazine propulsion system into future spacecraft required for unmanned planetary missions. These spacecraft would be inserted in Earth orbit using the Space Shuttle and given its initial sendoff by the Inertial Upper Stage. The design of a typical propulsion system, assessment of thermal and structural impacts on a selected spacecraft, and comparative studies with conventional propulsion systems have been completed. A major part of the current program involves assembly of a 3650 N thrust demonstration system using titanium tanks (0.72 m^3) and flight-weight components and structures. This system will be used to demonstrate the state-of-the-art throughout a representative flight system's qualification. The culmination of the effort will be a full firing verification of the system in an altitude chamber. Completion of this effort will increase the confidence in fluorine-hydrazine propulsion systems technology, such that missions with start dates in 1982 or later will have this higher performance option.

Introduction

THE Space Storable Propulsion Systems Technology Program was initiated in the fall of 1976. The objective is to demonstrate technology readiness of a space-storable liquid propulsion system so that this technology could be applied to planetary missions with project start dates in fiscal 1982 and beyond. Space-storable propellants are defined as a class of high-energy liquid propellant combinations in which at least one of the propellants is a cryogen. Further, it is necessary that each propellant be capable of being maintained in the liquid state while onboard a spacecraft in a space environment with only passive thermal control.

For the current program, the propellant combination of fluorine and hydrazine was selected as the most attractive combination for several reasons: 1) comparative performance, 2) density considerations, and 3) hydrazine's availability for hot-gas attitude control systems (ACS).

The approach taken for this program was to build on the vast fluorinated oxidizer experience[1-3] and to design, assemble, and test a system using procedures and tests that would be required for a flight projects' type approval system.

Concerns for safety when dealing with fluorine are considered paramount throughout the program. A series of reviews with the Space Transportation System (STS) safety office have been initiated to confirm the acceptability of this fluorine system design for Kennedy operations and the Space Shuttle. Evaluation of new safety equipment is part of the support activity presently underway. A report entitled, "Experience with Fluorine and Its Safe Use as a Propellant," which documents the past history of fluorine and its applications, has been published.[4]

A significant part of the current technology effort is associated with the thermal-configuration problems involved in the integration of a cryogen into a spacecraft. All identified applications are being considered; however, only one detailed study has been done at this time. The mission selected for analysis was the Mars Rover class missions where data from the Viking mission was used for comparative information.[5] Thermal-vacuum system tests of a flightlike configuration are planned for 1981 to confirm the more important concepts and assumptions.

Potential Mission Applications

The obvious advantages of increased chemical propulsion-stage performance have been discussed in depth by Meissinger.[6] His conclusion is that the improved performance available in an F_2-N_2H_4 system will benefit all missions studied. The advantage of a fluorine-hydrazine system (specific impulse of 3625 N-s/kg) compared to the much-used nitrogen tetroxide (NTO) monomethylhydrazine (MMH) typical Earth-storable system (specific impulse of 2890 N-s/kg), has attracted the attention of the design study teams for most of the unmanned planetary missions envisioned for potential implementation in the next two decades. Missions such as Mars exploration, Saturn Orbiter Dual Probe, and Solar Probe, have selected fluorine-hydrazine systems for their studies. Missions with early start dates such as Venus Orbital Imaging Radar (VOIR) also could benefit from this improved performance if the development risk can be reduced to an acceptable level by the project start date. The application of an F_2-N_2H_4 system for Earth-orbital transfer stages would be attractive due to the significant size reduction possible when compared to the hydrogen class propulsion systems.

Most performance comparisons are based on the ideal impulse burns disregarding the gravity-burn losses. Figure 1 is a comparative plot for space-storable and Earth-storable propulsion systems for operation at Venus where there are large gravity-burn losses. This figure shows the relative capabilities of space storables and Earth storables for placing payloads into planetary orbits. The curves plotted therein relate the payload-in-orbit (circular) and the orbit altitude. Curves are included for both orbit insertions accomplished by means of theoretically ideal impulse burns and by finite-time burns which entail gravity-burn losses, as would be the actual case.

The 220-350 kg payload advantage for the space-storable system would be welcomed by the VOIR mission planners if the technology demonstration meets their timetable. The figure illustrates that the advantage of space storables over Earth storables is significant whether gravity-burn losses are considered or the simpler ideal burn calculations are utilized.

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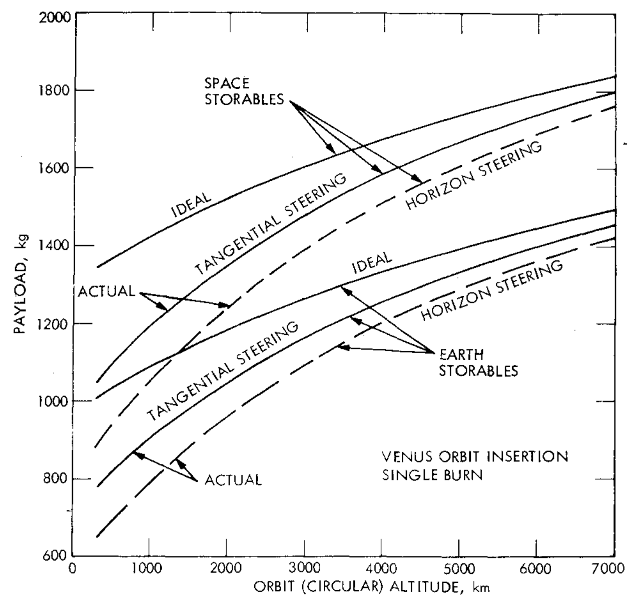

Fig. 1 Comparative payloads for Venus orbit insertion (single burn) for a 5000 kg spacecraft with a hyperbolic approach velocity of 3 km/s.

Even the gravity-burn cases plotted in the figure do not necessarily represent optimum maneuvers. A more thorough comparison of the payload-in-orbit capability of space storables and Earth storables would require a trial-and-error computer optimization which would vary the burn steering law and the intermediate orbits. Curves for two steering laws are included for references: 1) the tangential law wherein the thrust vector is maintained in alignment with the spacecraft trajectory; and 2) the horizon steering law wherein the thrust vector is maintained parallel to the horizon.

Safety

Fluorine is produced in quantity for many applications in industry and its handling has been the subject of many studies.[7] The use of fluorine for space application has also been studied extensively[2,3] for Titan-Centaur and Shuttle-launched missions. The safety concerns identified in these studies have all been addressed in this current program. Reference 4 discusses the history of fluorine and includes current information on safety and equipment together with a full set of references. The conclusion in all the above references is that with proper controls and processes, fluorine is only slightly more hazardous than an equal quantity of the universally used propellant, N_2O_4 or MMH, and that fluorine can be safely used in Shuttle-launched systems.

Cleanliness of a fluorine part, component, or full system is probably the most important factor in the safe handling of fluorine. This requirement is reflected throughout the design, assembly, and test of each item in the system as well as for the integrated system. Passivation of the component and system continues to be the final necessity as well as a required part of the surface protection for many of the materials. Another tool being used to confirm component or system cleanliness is a partial pressure analyzer which will confirm system dryness both in terms of solvents and moisture prior to F_2 exposure.

Design reviews are part of this system design as well as for the major components. A series of reviews with the STS as required for any Shuttle payload has been initiated to add to the confidence in the fluorine system and to identify open areas of concern. A new fluorine safety suit made of Beta cloth, and a new fluorine detector will be tested and evaluated at the hazardous test facility as part of the fluorine operations.

To date no technical reason has been identified which would preclude use of a fluorinated oxidizer in a Shuttle payload.

Table 1 Basic system characteristics

Pressure regulated	2240 kN/m^2 (325 psi)
Independent helium pressurant circuits stored at corresponding propellant temperatures	
Passive thermal control in space	
LN_2 and heaters for ground operations	
Thrust	3560 N (800 lb$_f$)
Duration (total accumulated)	4000 s
Mixture ratio	1.5-1.7
Specific impulse	3630 N-s/kg (370 lb$_f$-s/lb$_m$)
Cold starts (no pulsing)	50
Temperatures	
F_2	−190°C (−307°F)
N_2H_4	20°C (68°F)

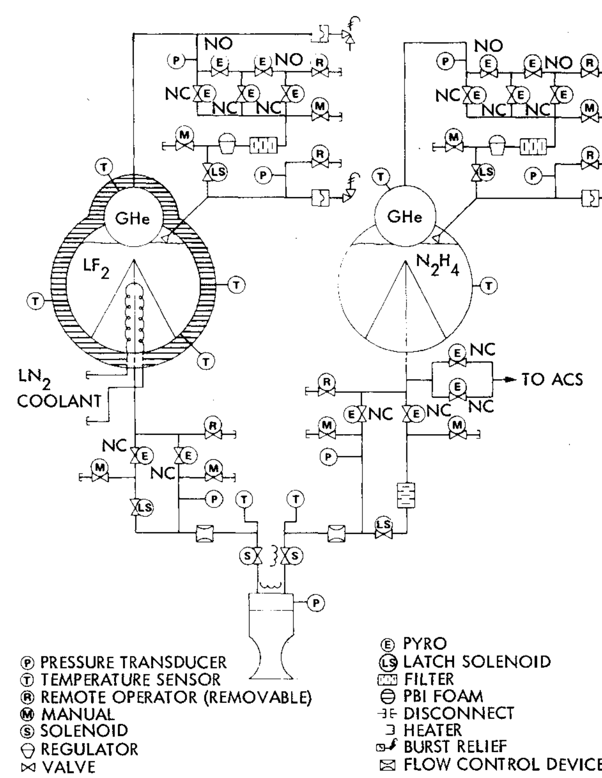

Fig. 2 F_2-N_2H_4 flight system schematic.

Propulsion System Design

The basic propulsion system concepts have been proven on prior missions such as the Mariner 71 and Viking Orbiter propulsion systems.

Starting with these basic systems, the schematic for a typical flight system was evolved taking into consideration the new requirements of a cryogenic propellant and the latest STS requirements.

A summary of the system characteristics is shown in Table 1, with the system schematic depicted in Fig. 2. A pressure regulated system has been selected as the most desirable first design based on rocket engine simplicity (single chamber pressure instead of variable) and smallest tank size for packaging purposes.

The pressurant (helium) for fluorine will be stored cryogenically, while the pressurant for the fuel will be at ambient conditions. This produces the smallest "same size" propellant tanks. The final mixture ratio has not been

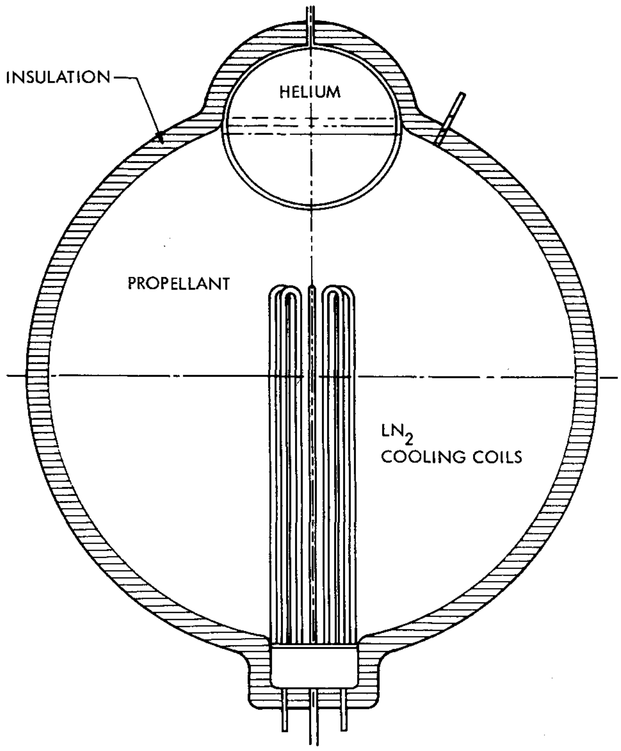

Fig. 3 Integral tank for flight system.

established (rocket engine dependent) but will be between 1.5, which is the equal volume ratio, and 1.7. At mixture ratios of 1.5 and higher, there would be an excess volume of hydrazine available for attitude control applications. At this time the baseline system is assumed to use identical tanks.

The propellant tanks will be made of 6Al-4V titanium. L. Toth[8] has detailed the testing completed to qualify titanium for F_2 service. These tanks will utilize an integral tank concept, as depicted in Fig. 3. This integral tank provides a smaller overall tank envelope and will be no heavier than two separate tanks when the structural supports are considered. The primary reason for the integral tank is a thermal consideration. The low helium storage temperature combined with the small thermal mass of the helium and its container, limits the thermal input allowable to the helium before excessive pressure rise would be encountered. The integral concept thermally connects the helium to the high thermal mass of the fluorine. This produces a significant safety margin for both normal Shuttle operations and abort situations, as well as any inadvertent short-term ground cooling losses. The tank design has only parent metal in the common wall between the tanks, with welds all external.

The cryogenic tanks will be cooled while on the ground with liquid nitrogen. Several concepts are being evaluated, but the most desirable techniques are the use of an internal cooling coil in the fluorine tank or a partial outer shell which would permit LN_2 flow in the annulus on the ground and would become a vacuum dewar in space. All concepts assume use of a foam insulation on the outside to preclude moisture during ground handling and operations. An external umbilical connection on the Shuttle for LN_2 would be utilized and would also be available for easy cooling access in the event a Shuttle return-to-Earth abort situation is encountered. The insulation is a fluorine compatible foam polybenzimidazole (PBI).[8]

The primary plumbing arrangement is similar to previous missions. New items are the completely independent gas circuits (oxidizer and fuel) and the use of a latching solenoid valve in lieu of a check valve in the gas circuit. This valve will complicate the regulator requirements somewhat and necessitate incorporation of a high-pressure activated opening

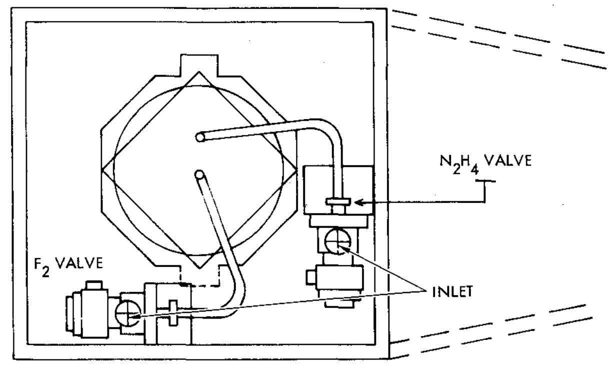

Fig. 4 Thrust box with propellant valves mounted and lines to injector.

of the valve. However, it will allow significant system operational versatility in that multiple burns can be made without multiple pyrovalve actuation during short time periods. Further, the option for blowdown during the final few seconds of each burn by early valve closure will significantly increase the tolerable temperature rise before relief valve actuation and would extend the capability of the system into marginal thermal regions, such as limited adverse orbital maneuvers. For example, at 30% ullage, the temperature increase necessary to raise the pressure 410 kN/m^2 (from regulator lockup to design maximum or relief valve setting) is 6°C. If the system is locked up at 35 kN/m^2 below regulated pressure then the allowable temperature rise is almost double (10°C).

The primary containment devices will remain the pyroactuated valves. These valves, plus the gas circuit latch valve, will reduce the fluorine-exposed lines during ground and Shuttle operations to a safe minimum. The present concept assumes that the valve panels will be maintained warm while on the ground and would be allowed to cool on launch of the Shuttle. This eliminates the difficult component insulation problem and allows service checkouts to be performed similar to previous missions. A small number of low wattage heaters on the structure and lines to the component panel will be required using ground power only.

The location of the propellant valves for the rocket engine assembly has required several system-rocket engine tradeoffs. The rocket engine may be warm or cold depending on spacecraft-sun orientation, so that it is desirable to have valves mounted separate from the thrust chamber assembly. This also allows better thermal isolation between the fuel valve which must be kept above 5°C to prevent freezing and the oxidizer valve which can be as cold as −188°C. Further, the large temperature changes during rocket engine start produce a large thermal stress potential. The solution adopted is to mount the valves on the sides of the thrust box, Fig. 4, and use relatively long lines (~20 cm) from the valves to the back of the injector. This length with the turns involved will reduce thermal stress to acceptable levels. It is possible that for a gimbaled engine the movement can be tolerated in these lines with only minor line changes. Testing on the sea-level hardware with similar line lengths from valves to injector has indicated satisfactory start and shutdown characteristics.

Demonstration System

The demonstration system will use titanium flight weight propellant tanks and helium tanks. The thermal control will utilize internal LN_2 coils for the fluorine and an external LN_2 shell for the helium. The insulation will be PBI on the fluorine system. The helium and propellant tanks will be separate due to use of Viking Orbiter residuals. The propellant tank will hold 975 kg (2150 lb$_m$) of fluorine, assuming a 10% ullage. The helium tanks are oversized, and therefore will be adequate for tests with initial pressurant pressures from as low as 17,930 up to 24,820 kN/m² (2600-3600 psi). The structure uses modified Viking struts equipped with instrumentation to determine thermal characteristics of this standard-type structural configuration.

All component processing, system assembly, cleaning, system proof, leak and functional tests, and handling will use procedures and techniques patterned after prior successful flight programs. After completion of this phase, the system will be shipped to the hazardous test facility at Edwards Air Force Base, California. The system will be installed in the vacuum test facility, and a full set of tests similar to those required at the Kennedy Space Center launch site will be completed. Facility equipment will be substituted for mobile support equipment in these tests. The system will then be passivated and serviced with propellants. A mission sequence will be performed, limited only in its maximum single-burn duration by the Viking-sized tanks. Two propellant loads are planned to attain a total system duration of over 2800 s and to provide the capability of evaluating optional system routines.

The initial start and first firing procedure will be:

1) Open latching solenoid isolation valves (all four).
2) Propellant valve cycle (to evacuate lines).
3) Actuate liquid pyro isolation valves.
4) Actuate gas pyro isolation valves.
5) Perform firing (simultaneous prop valve operations).
6) Close gas isolation latch valves.
7) Close liquid isolation latch valves (at maximum heat soak-back).

A repeat sequence would be:

1) Open all latch valves.
2) Perform firing.
3) Repeat steps 6 and 7 above.
4) Actuate gas isolation pyrovalve if long coast is programmed.

The necessity for waiting to peak heat soak-back prior to closing the liquid isolation valves is necessary for two reasons: 1) to avoid trapping a full line of liquid, which at cryogenic temperatures would be unacceptable; and 2) to intentionally have as large a gas (fluorine vapor) bubble in the line as possible, since conduction heat losses are significantly higher through a liquid filled line than through a corresponding line containing gas. The latching solenoid isolation valve will have a back-relieving characteristic which would also help produce the gas bubble and prevent overpressurization between the valves.

Following the test firings in early 1982, the system will be deserviced and a series of functional and leak tests performed to complete the evaluation.

Rocket Engine

The primary element of the fluorine-hydrazine propulsion is the rocket engine assembly (REA). The detailed results of the first phase of the REA test program was presented by Appel.[9]

Table 2 Rocket engine requirements

Thrust	3650 N (800 lb_f)
Nominal chamber pressure	6895 kN/m² (100 psi)
Total firing duration	4000 s, min
Specific impulse (I_{sp})	3625 N-s/kg (370 lb_f s/lb_m)
Mixture ratio	1.5:1.7
Expansion ratio	60:1 or 80:1
Combustion stability	
Insensitivity to oxidizer lead or lag	
No hydrazine freezing or excessive heat soak-back	

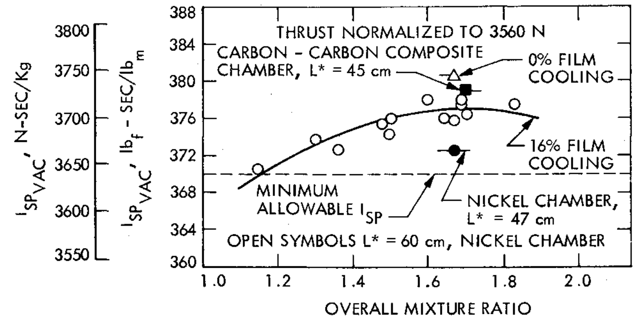

Fig. 5 Like-doublet injector, extrapolation of measured specific impulse to vacuum at expansion ratio of 80:1.

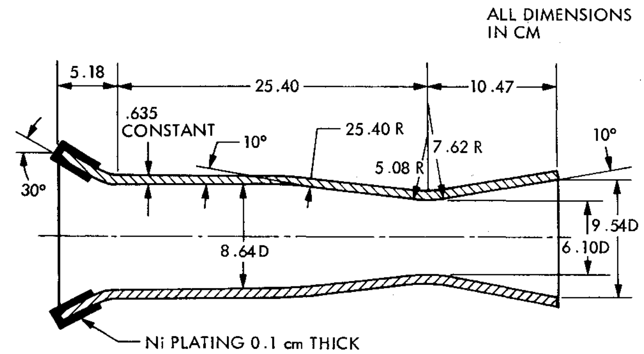

Fig. 6 Carbon-carbon composite thrust chamber (L^* = 47 cm).

For this program the fabrication and testing of the thrust chamber assembly has been separated from the propellant shutoff valve effort during the first phase. The thrust chamber assembly consists of the propellant injector, the thrust chamber, and the necessary flanges and seals required to mate these two elements. The significant design criteria for the assembly is shown in Table 2.

Most of these reported objectives[9] have been accomplished using an 88 element like-doublet injector design with 16% of the fuel as film cooling for the lightweight carbon-carbon composite chamber. Figure 5 is a summary of performance extrapolated from the sea-level expansion ratio test hardware to an 80:1 expansion ratio engine size. The carbon-carbon composite thrust chamber used for the 100-s tests demonstrated the required resistance to the critical thermal shock encountered during the initial 30 s of firing and exhibited no measurable erosion in the 1925°C (3500°F) throat region. Two significant problems were identified: 1) localized erosion of the carbon chamber was observed in the region starting at the plane of the injector face and for approximately 5 cm down the chamber; and 2) the seal between the carbon chamber and the injector was not satisfactory.

The localized chamber erosion was caused by a combination of higher than expected injector end-chamber wall temperatures 1650°C (3000°F), combustion gas recirculation (sometimes referred to as radial winds), and the reaction of hydrazine with the carbon chamber at these elevated temperatures. Two modified injector configurations are presently in test which should eliminate these concerns. The results of this phase of the testing will be reported in a future paper. A cross section of the carbon-carbon composite is shown in Fig. 6 for reference.

The primary chamber-to-injector seal has been accomplished on several test specimens and will also be confirmed during test firings now in progress. The approach used[9] consists of a nickel plating 0.10 cm thick on the injector end of the carbon chamber, shown in Fig. 6. This plating is ground to match the injector contour and gives a large area metal-to-metal seal; a secondary elastomer seal is also incorporated in this design. Alternate techniques are being evaluated which would involve welding the injector to a reinforced portion of the chamber plating.

Following the test phase now in progress, a 60:1 expansion ratio carbon-carbon composite chamber will be fabricated and tested with at least one of the candidate injectors.

Components

The fuel side components are considered to be available and qualified by prior flight programs. A secondary requirement for all oxidizer components being developed is that they be usable in the fuel system so that identical components could be used in both systems for many applications.

The progress on fluorine component development was reported by Weiner.[10] The pyro-actuated valves have completed the component fluorine tests and will next be evaluated at the subassembly level with other fluorine components. The propellant valve and latch valve are both well into the test phase and the regulator is in the design phase. Service valves will probably be facility-type for the demonstration system, but some design effort for flight application is underway. The demonstration system will use facility burst devices and this item will require further development for flight application. Pressure transducers also will be facility-type; an evaluation of available cryogenic transducers is underway and limited testing is planned. The helium filter will be the same unit used by Viking but with added cleanliness and passivation requirements.

The fluorine system plumbing will be stainless steel tubing and all joints will be tungsten inert gas (TIG) welded. The only mechanical joints will be between the propellant valves and the thrust chamber, and the propellant tank's 22.5-cm flange. Flight applications would use a welded tank flange.

Thermal Structures and Configuration

The portion of the effort which deals with the structure and spacecraft configuration for space applications is complicated by the strong influence of the requirement for passive thermal control of the cryogenic fluorine. The scope of our study has been limited to the specific requirement of our demonstration system and a limited amount of design for a specific mission. The feasibility for a Mars Rover class orbiter application was established and the impact of a cryogenic propellant system on spacecraft design has been discussed.[5] The requirement for low thermal structural conductivity is obvious, two structural approaches are shown in Figs. 7 and 8. The first uses a tension wire approach, and the second would require a low conductivity, high-strength material such as the boron fibre composites. This material has been used at ambient conditions, but would require qualification at cryogenic temperatures. The strut approach is preferred from a mass point of view but may require the most development effort.

Configuration studies indicate that compromise will be required between the spacecraft designer and the propulsion subsystem to provide a clear view of deep space. The status of the thermal and configuration studies was discussed by Stultz.[11] Each type of spacecraft and mission will have its own unique considerations. Generally, for a 3-axis stabilized system, space-storable integration for an orbit insertion maneuver will be easy. For single- or dual-axis spin-stabilized systems there will be more complications, with twin boom spacecraft being most easily adapted. Application for extended orbital maneuvers is most difficult, as pointing or science requirements may not allow a configuration with adequate view of space for thermal rejection.

The conclusion reached by Stultz[11] indicates that depending upon specific spacecraft and mission requirements, space storables are feasible. Unlike Earth storables, a space-storable propulsion system does impact the spacecraft configuration and mission design. A thermal design that is compatible with the science objectives must be arrived at for each spacecraft. The following configuration guidelines should be kept in mind by the spacecraft designers.

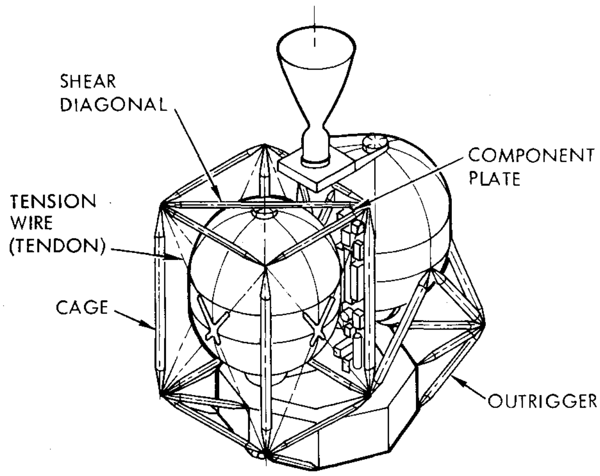

Fig. 7 Space-storable propulsion tension wire configuration.

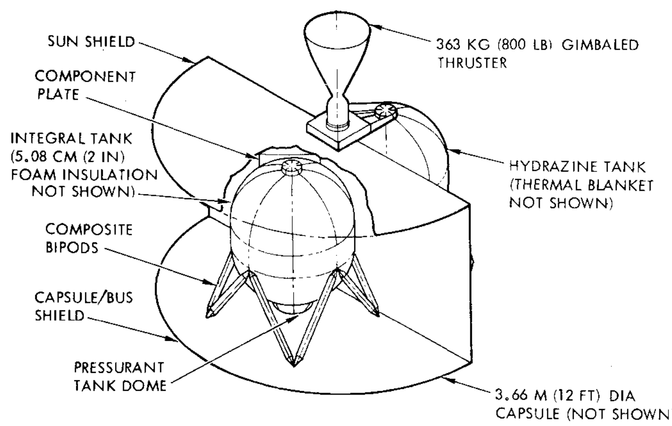

Fig. 8 Space-storable propulsion bipod configuration.

1) The fluorine system must have a maximum possible view of space.

2) Hot spacecraft assemblies, such as the solar panels/radioisotope thermoelectric generators (RTG's), lander, bus, and high-gain antennas, must be placed so that they can be shielded from the fluorine system without substantially reducing the fluorine system view to space. An asymmetric spacecraft configuration is probable and thermally desirable.

3) Shields are required to block the direct and indirect solar and infrared energy that would otherwise be absorbed by the fluorine system. The shields are large and numerous and a necessary part of the early configuration studies.

4) Except for the engine, the various fluorine assemblies (pressurant, pressurant control, and propellant isolation) should be located in proximity of each other to afford making the fluorine system isothermal and to help maximize its view of space.

Conclusion

The current technology demonstration program is at the halfway mark of its five year plan. The component and rocket engine effort should be completed by mid 1980. The system assembly and test will provide the needed confidence in the system designs and operational procedures. This work is scheduled for completion in April, 1982. Configuration and thermal design studies have confirmed the feasibility of a fluorine-hydrazine system for most orbit insertion liquid propulsion applications, but have identified several areas where the propulsion system would impact spacecraft configuration.

The impacts on configuration require early interaction between spacecraft and the propulsion designers.

Most planetary missions of the next few decades can benefit from improved performance in the chemical propulsion system. No significant operational, safety, or technical reasons have been identified which would preclude incorporation of such an improved system using F_2 and N_2H_4 for Shuttle-launched payloads with project start dates in 1982 or later.

Acknowledgments

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