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PREFACE

The work described in this report was performed by the Propulsion Division of the Jet Propulsion Laboratory.

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ABSTRACT

The feasibility and application of hybrid rocket propulsion to outer-planet orbiter missions is assessed in this study and guidelines regarding future development are provided. A Jupiter Orbiter Mission was selected for evaluation because it is the earliest planetary mission which may require advanced chemical propulsion. Mission and spacecraft characteristics which affect the selection and design of propulsion subsystems are presented. Alternative propulsion subsystems, including space-storable bipropellant liquids, a solid/monopropellant vernier, and a hybrid, are compared on the basis of performance, reliability, and cost. The comparisons which assess performance, reliability, and cost independently do not yield a conclusive evaluation of each alternative propulsion subsystem's competitive position. This handicap was overcome by comparing the alternative propulsion subsystems with a cost-effectiveness model which combines the above three variables into a single parameter. Cost-effectiveness comparisons are made for a range of assumptions including variation in (1) the level of need for spacecraft performance (determined in part by launch vehicle injected mass capability), and (2) achievable reliability at corresponding costs. The results indicated that the hybrid and space-storable bipropellant mechanizations are competitive.

I. SUMMARY

The objectives of the hybrid study were to assess the feasibility and application of hybrid propulsion to planetary missions and to provide guidelines regarding future development.

A hybrid which uses a throttling monopropellant vernier subsystem for thrust vector control (TVC) was compared with two liquid-bipropellant designs (OF_2/B_2H_6, F_2/N_2H_4) and a beryllium solid/monopropellant vernier configuration. The OF_2/B_2H_6 design is a conventional liquid-liquid bipropellant, whereas the F_2/N_2H_4 is capable of both monopropellant and bipropellant operation, i.e., dual mode. The solid/mono alternative is basically the same as the hybrid/monopropellant vernier configuration with the exception that a fixed impulse, burn-to-completion, berylliumized solid-propellant motor is substituted for the hybrid.

The major conclusions of this study are the following. The most obvious area of improvement of the hybrid/mono is to eliminate the throttling monopropellant vernier subsystem. In doing so, the reliability and performance would be improved such that the hybrid could become the preferred subsystem and so could influence the final choice. Elimination of the vernier subsystem will require the hybrid to provide those functions previously implemented with the verniers. Major developments confronting the hybrid would then be:

(1) Demonstration of a nozzle to withstand long burn times and multiple restarts.

(2) Demonstration of a grain capable of multiple starts in space with long periods of space exposure between burns.

(3) Thrust vector control.

(4) High accuracy, low thrust capability for multiple, small trajectory correction maneuvers.

Additional conclusions of this study are based on performance (mass), reliability, cost, and cost-effectiveness comparisons. A comparison of the hybrid/monopropellant vernier system with the competing designs on the basis of performance, reliability, and cost provided the following results:

(1) The hybrid/mono has good performance, equivalent to the dual-mode F_2/N_2H_4, considerably better than the Be solid/mono, but poorer than the OF_2/diborane bipropellant.

(2) The hybrid/mono has the poorest reliability, due primarily to the numerous components in the monpropellant vernier subsystem.

(3) The hybrid/mono has intermediate cost, higher than the Be solid/mono, but lower than the bipropellant liquids.

Comparisons based on performance, cost, and reliability independently do not yield conclusions regarding the hybrid's relative merits. Cost-effectiveness techniques allow combination of these parameters into one parameter giving a more effective comparison. The four advanced propulsion concepts were then compared under five specific conditions. They are:

(1) Propulsion performance when spacecraft mass is important.

(a) Low-propulsion reliability.

(b) High-propulsion reliability.

(2) Propulsion performance when spacecraft mass is of reduced importance.

(a) Low-propulsion reliability.

(b) High-propulsion reliability.

(3) Contemporary propellant technology versus advanced propulsion.

Results of these comparisons are as follows. The hybrid/monopropellant vernier alternative with a requirement for high reliability and when spacecraft mass is important is competitive from a cost-effectiveness viewpoint with the liquid-propulsion options considered. The beryllliumized solid/monopropellant vernier design is generally not competitive with the hybrid/mono except when mass has reduced importance (for example, excess launch vehicle capability).

The hybrid/mono is not competitive with the other three alternatives when mass is not constraining (excess launch vehicle capability) and when each design does not require high reliability. Under the same constraint on performance, all systems considered are equivalent when high reliability is required.

Contemporary propellant technology represents lower cost and less development risk than advanced propulsion and would be more favorable than advanced propellant subsystems when mass has reduced importance and cost and reliability have greater importance.

Since it is not possible to discern at this time how important spacecraft mass will be, future developments must be tailored to the case when mass is important, so that if this case should result, the propulsion subsystems needed will be developed. Also, it is clear that reliability is very important to long-term missions. Under these conditions it can thus be concluded that hybrid propulsion is competitive with other advanced propulsion subsystems for outer-planet orbiter missions.

II. GROUNDRULES AND CONSTRAINTS

This report presents mission and spacecraft characteristics that affect the selection and design of propulsion subsystems for use in outer-planet orbiters. These missions were selected since they are the earliest missions which may require advanced chemical propulsion systems. The specific propulsion numerical values presented here apply to a 1981-1982 Jupiter Orbiter Mission, although a 1984 Saturn Orbiter Mission was also investigated. The same type of maneuvers and mechanizations apply to a Saturn Orbiter, even though it offers its own peculiar complications. Material used for this study was drawn from the Outer Planet Orbiter Study (Ref. 1), carried out at the Jet Propulsion Laboratory and from associated studies (Refs. 2 and 3). Because of the limited duration of the study, only planetary orbiter propulsion subsystems were considered. However, propulsion includes subsystems which combine trajectory correction, retro, and subsequent in-orbit changes.

The spacecraft injected mass was constrained to be within the capability of Titan-based launch vehicles with appropriate upper stages. Since this is a study of a mission 10 yr in the future, there is considerable uncertainty as to launch vehicle performance in that era, availability of stages, and eventual spacecraft hardware weight. Hence, the study has considered the Titan IIID (T-IIID) launch vehicle family, including:

(1) T-IIID (1205)/C/B-II.

(2) T-IIID (1205)/C/F_2-H_2.

(3) T-IIID (1207)/C/B-II.

(4) T-IIID (1207)/C/F_2-H_2.

These vehicle combinations represent uprating in both the 305-cm (120-in.)[1]-diameter solid strap-on rockets (5 versus 7 segments) and upper stage propulsion (Burner-II class versus fluorine/hydrogen).

The propulsion mechanizations studied were:

(1) Hybrid/monopropellant verniers.

(2) Beryllium solid/monopropellant verniers.

(3) OF_2/B_2H_6 bipropellant liquid.

(4) F_2/N_2H_4 dual-mode liquid.

The hydrazine attitude propulsion subsystem included in Ref. 1, because of propellant sharing with the main propulsion subsystem, is not included in this analysis, although subsystems having monopropellant on board could benefit by sharing tankage with the attitude propulsion system.

The study approach consisted of comparing the hybrid with competing propulsion candidates. They were first compared on the basis of performance, reliability, and cost. Then, to gain added insight, cost-effectiveness techniques were used to evaluate the alternative subsystems. The cost-effectiveness technique utilized is outlined in Ref. 4 where trajectory correction propulsion alternatives were compared for the Grand Tour spacecraft. The cost-effectiveness methodology trades off mass (performance), R&D and hardware costs, and reliability. The importance and effect of these parameters were evaluated under varying program constraints.

[1] **Values in customary units are included in parentheses after values in SI (International System) units if the customary units were used in the measurements or calculations.**

III. PROPULSION REQUIREMENTS

A. Mission Characteristics

The propulsion subsystem must provide the impulse for in-transit course corrections, typically post-earth and pre-encounter with the planet, and the impulse to establish the desired planetocentric orbit change from the hyperbolic approach trajectory. After establishing the prescribed orbit, subsequent orbital apsides adjustment will be provided by a discrete number of small impulse expenditures. For the purpose of this report, the propulsive maneuvers are categorized chronologically as (1) post-injection near earth, (2) pre-encounter, near Jupiter or Saturn, (3) orbit insertion, and (4) post-insertion.

Mission characteristics for the selected Jupiter Orbiter Mission are summarized in Table 1 and a mission profile showing when each burn occurs is schematically represented in Fig. 1. The mission is approximately 5 yr in duration: 2 yr in transit, and 3 yr in orbit. Total ΔV required by this mission is 1781 m/s. The total guidance correction ΔV is equal to 104 m/s and is used to correct for flight path errors. Post orbit insertion maneuvers, excluding the orbit trim which occurs 10 days after insertion and which is included in the total guidance correction ΔV value, are allocated for orbit changes for satellite encounters, periapsis correction, and inclination change in order of decreasing priority. The total allotted propellant for post-orbit insertion maneuvers is equivalent to a 200-m/s ΔV. This is considered sufficient ΔV capability to favorably position the spacecraft for observations of Jovian satellites such as Io, Europa, Ganymede, and Callisto, and to provide late mission orbit changes.

Saturn Orbiter Mission characteristics are summarized in Table 2. The mission is approximately 8.5 yr in duration: flight time is 5.5 yr and desired active life in orbit is 3 yr. The total guidance correction ΔV equal to 104 m/s was assumed to be the same as that for the Jupiter Mission pending completion of the Saturn Orbiter flight path error analysis. Orbit insertion ΔV is equal to 1036 m/s. A post-insertion ΔV equal to 129 m/s is required to encounter the rings of Saturn late in the mission.

B. Launch Vehicle Availability

The impact of launch vehicle capability on outer-planet orbiter missions is unresolved at this time. Uncertainties as to launch vehicle performance, availability of upper stages, and final spacecraft mass could impact the degree of importance attributed to spacecraft propulsion performance. Figures 2 and 3 from Ref. 5 are curves of T-IIID(1205)/Centaur and T-IIID(1207)/Centaur performance with appropriate upper stages. Jupiter and Saturn Orbiter spacecraft injected mass ranges as a function of C_3 are superimposed on both curves. The T-III(1205)/C/B-II class of vehicle provides inadequate or marginal capability even for a Jupiter Orbiter spacecraft using energetic onboard propulsion, i.e., OF_2/B_2H_6. Furthermore, this class of performance is inadequate for Saturn Orbiter launch energies. By going to an energetic F_2/H_2 upper stage, more than adequate capability exists at Jupiter Orbiter C_3's; however, the Saturn Mission is still questionable.

The T-IIID(1207)/C/B-II class of vehicle provides adequate capability for Jupiter, but is inadequate for Saturn; whereas the T-IIID(1207)/C/F_2/H_2 combination provides more than adequate capability for both missions.

Therefore, one can see the conditions when spacecraft propulsion performance is critical (whenever launch vehicle capability is just adequate or marginal) and the conditions at which spacecraft performance is not constraining (when more than sufficient launch vehicle capability exists). With the launch vehicle unresolved, the need for a high-performance spacecraft propulsion subsystem is unresolved. However, to allow for all possibilities, advanced development should stress high-performance spacecraft propulsion subsystems so that they will be available if needed.

C. Functional Requirement

The functional objective of the spacecraft propulsion subsystem is to provide the impulse requirement from a mechanically separable propulsion module mated to the flight spacecraft with minimal umbilical connections crossing the mutual interface. In addition, it is desired that the module be capable of being loaded and serviced prior to spacecraft mating. The module will incorporate required safe/arm mechanizations and design features such that the loaded module may be safely handled by personnel.

D. Interface Criteria

Acceleration, acceleration rate, and environmental considerations are summarized in Table 3. In the spacecraft's fully deployed state, long flexible booms and antenna are extended and at high-acceleration levels could deflect appreciably under the inertial loads developed by high retro thrust during orbit insertion. Since the deployed appendages are designed to be tested at 1 g on the ground, this appears to provide a good upper-limit design value for propulsion thrust/weight ratio. The onset of thrust, and its decay, is also important since it fixes the rate of elastic energy development in deployed appendages and contributes to "flapping" of the booms. An ignition thrust profile consisting of a 0.2-g step function followed by a 0.2-g/s ramp up to 0.8 g (Ref. 6), and a 0.2-g/s ramp function during cutoff appears to be adequate to control these difficulties when using solid-propellant designs.

Environmental compatibility requirements for the Jupiter Mission are considered in terms of two phases: transit and orbital. Each phase is of concern to all propulsion candidates. Throughout the entire mission, the propulsion subsystem is exposed to vacuum storage and operation.

During the transit phase, the propulsion subsystem will be exposed to gamma and neutron radiation developed by the radioisotope thermoelectric generator (RTG) power source and to space radiation. The asteroid belt, located between Mars and Jupiter, is also a potentia hazard to the propulsion module. During the orbital phase of the Jupiter Mission, intense Jovian electron and proton radiation, RTG radiation, and space radiation will exist.

Propellant thermal conditioning is required in order to ensure satisfactory propulsion operation. Maximum and minimum propellant temperature ranges for each propulsion mechanization are listed in Table 4. The storage temperature range for each propellant combination can be derived from the information in this table.

E. Development Schedule Requirement

The probable launch dates of the Jupiter and Saturn Orbiters are 1982 and 1984, respectively. The technology cutoff date would precede the launch date by 3 yr. This requires propulsion technology to be developed by 1979 for the Jupiter Orbiter and by 1981 for the Saturn Orbiter.

IV. PROPULSION MECHANIZATIONS STUDIED

A. Solid/Monopropellant Verniers

This design is based on a simplified version of the Surveyor propulsion concept; characteristics of this design are summarized in Table 5 and are schematically represented in Fig. 4. The philosophy of the solid-propellant/monopropellant propulsion subsystem is to provide the bulk of the required energy for the mission, which is needed during orbit insertion, by a single high-mass-fraction, high-performance, low-thrust, burn-to-completion solid-propellant motor.

The solid-propellant motor of this propulsion option would be coupled with four throttleable monopropellant engines which provide thrust vector control (TVC) during solid motor operation. The monopropellant engines also provide the precision control and multiple restart flexibility for the small velocity increment, high accuracy, mid-course, post-insertion trim, and orbit change maneuvers.

During the orbit insertion maneuver, the four monopropellant engines would be started shortly before the solid motor to provide an autopilot-controlled stable spacecraft prior to firing the solid motor. Upon command, the solid motor would be ignited at a low-thrust level (0.2 g). After ignition, the spacecraft acceleration would be increased to a steady-state level, less than 1.0 g, by a slow increase in thrust, 0.2 g/s. The solid motor burn-out would also be characterized by a slow decrease in thrust. After the solid burns to completion, the monopropellant engines can continue operating in order to provide additional incremental velocity flexibility.

The preliminary motor design has assumed the use of predicted technology applicable for a 1981-1982 flight (Ref. 7). Because of its greater performance potential, a beryllium ized propellant served as the basis for design. The propellant grain configuration is a regressive end-burning charge in order to achieve a low and decreasing thrust versus time profile and therefore maintain a low and constant acceleration (~1.0 g) on the spacecraft. Also, during ignition a "g-dot" igniter (Ref. 8) will provide a soft, slow increase in thrust to accelerate the spacecraft to a steady-state level.

The liquid-propellant portion of this subsystem would use a hydrazine-fueled engine. It is a blow-down design with propellant and pressurant gas contained within a common tank. Neat hydrazine, N_2H_4, or the hydrazine-hydrazine mononitrate blend, N_2H_4 HNO_3 H_2O, can be used as the monopropellant. Helium pressurant gas is used as on the other propulsion options.

B. Hybrid/Monopropellant Verniers

This design is functionally similar to the solid/monopropellant vernier concept. A multiple start-stop, high-performance, hybrid motor would provide the bulk of the mission's required energy (orbit insertion and post-insertion). Small, relatively simple, liquid-monopropellant engines would provide precise control and multiple restart capability for the small velocity increment, high-accuracy maneuvers. They would also provide thrust vector control during hybrid motor burn. Characteristics of this design are summarized in Table 5 and Fig. 5.

The monopropellant vernier subsystem of this mechanization is similar to that for the solid/monopropellant design described earlier.

The following description of the hybrid design was abstracted from Ref. 3. An attempt has been made to simplify the propulsion design as much as possible and to utilize only those components, liquid management techniques, and technology that are fairly well established. The thrust section of the hybrid motor, however, is an extrapolation of the current state-of-the-art. The coolant material in the nozzle design and the location of the transpiration cooling gas ports in the nozzle represent new technology which will be investigated in a currently planned NASA contract. The reason for selecting this nozzle design approach is that the orbit insertion burn time at 205 s is substantially longer than the useful life of currently available ablative nozzles.

The selected propellant has been changed for the revised design from FLOX oxidizer and 25 lithium/25 lithium hydride/50 polybutadiene fuel to oxygen difluoride (OF_2) oxidizer and 25 Li/10 LiH/65 PBD fuel. The delivered specific impulse of those two propellant combinations is equivalent and a high degree of similarity exists in all other characteristics such that the performance does not normally change by substituting one combination for the other in a given application. The storability of OF_2 is somewhat better than FLOX due to a higher boiling point. The improved storability may be required to accommodate the final burns assigned to the hybrid thruster. However, propellant cost is much higher for OF_2.

The adjustment in the fuel formulation from 50 to 65% binder is made to provide maximum performance with OF_2 oxidizer. Peak performance in the FLOX - Li/LiH/PBD family occurs when sufficient oxygen is blended into the oxidizer to oxidize the carbon in the binder to carbon monoxide. When OF_2 is used, the oxygen is about 30% of the oxidizer by weight and additional carbon is necessary to react with the oxygen; hence, the maximum performance occurs at 65% binder. As in the previous fuel formulation (25 Li/25 LiH/50 PBD), the ratio of Li to LiH is selected to provide a fuel that quenches rapidly when oxidizer flow is terminated. Both propellants are supported by extensive test data and the confidence in both is very high.

C. OF_2/B_2H_6 Bipropellant Liquid

This subsystem would employ a single bipropellant rocket engine fed by fuel and oxidizer contained in two separate tanks. The propellants are mild cryogens; the fuel is diborane (B_2H_6), and the oxidizer is oxygen difluoride (OF_2). Propellants would be forced into the engine by means of helium gas pressurant which would be stored in high-pressure vessels and subsequently regulated to the desired feed pressure through a gas pressure regulator. This bipropellant system would perform all required propulsive maneuvers. This propulsion candidate represents advanced liquid-rocket development which promises a significantly higher specific impulse [3923 N-s/kg (400 lbf-s/lbm)] than the other candidates. A summary of this design is presented in Table 5 and is schematically represented in Fig. 6.

D. Dual-Mode Bipropellant Design

This mechanization employs a single-rocket engine fed by fuel and oxidizer contained in two separate tanks (Fig. 7). The fuel is neat hydrazine (N_2H_4), and the oxidizer is fluorine (F_2), a cryogenic. Helium pressurant gas is regulated through a pressure regulator to each propellant tank. The thrust chamber is a combination monopropellant/bipropellant engine. The engine contains a quantity of catalyst to initiate and maintain the decomposition of the hydrazine. Oxidizer is injected downstream of the decomposed hydrazine, thereby achieving bipropellant operation. Small guidance correction maneuvers (midcourse, pre-encounter) and propellant settling are performed in the monopropellant mode. Large maneuvers, such as orbit insertion and in-orbit changes, are performed using the dual-mode or bipropellant combination. This design is summarized in Table 5.

V. PROPULSION COMPARISON METHOD

The hybrid/monopropellant vernier propulsion subsystem was first compared with the other propulsion subsystems on the basis of performance, cost, and reliability. These comparisons, as will be shown, did not lead to definite conclusions regarding the competitive position of the hybrid/mono subsystem. Cost-effectiveness techniques were then employed to combine the three basic parameters – performance (mass), cost, and reliability – into a single cost-effectiveness parameter. This enabled a more effective comparison. The following section presents the performance, cost, reliability, and cost-effectiveness comparisons.

A. Performance Comparisons

Propulsion subsystem mass for each mechanization are given in Table 6. The nonpropulsive payload mass is 685 kg (1450 lbm) in each case and the injected mass is allowed to vary as a result of the propulsion subsystem mass. The calculations were done this way because Ref. 1 had defined the payload mass, but uncertainties in the launch vehicle prevent definition of the injected mass. The propulsion system masses are based on the data of Table 5 and the ΔV requirements given in Table 6. The mode of operation for providing each velocity increment is also summarized in Table 6. The last column in the table shows the difference between the hybrid/mono and the other subsystems. Note that the hybrid/mono has a considerable mass advantage over the solid/mono; it is equivalent to the dual mode, but has a mass disadvantage when compared with the OF_2/B_2H_6 bipropellant design.

B. Cost Comparisons

Propulsion costs are divided into two categories: (1) nonrecurring or development cost, and (2) recurring or hardware costs. Cost estimates for the four competitive subsystems are given in Table 7. The hybrid propulsion subsystem costs are based upon previous work performed for NASA by United Technology Center and McDonnell-Douglas, modified by JPL analysis. The costs for the other options are based upon previous JPL work. Note that the hybrid is neither the highest nor the lowest cost subsystem.

C. Reliability Comparisons

The general subject of predicting mechanical component and system reliabilities is extremely difficult and controversial. Problems associated with reliability analysis and alternative approaches are documented in Ref. 4. These difficulties are summarized in the following paragraphs in order to place the ensuing reliability study in proper context.

Mechanical component failure causes can be attributed to three general areas: engineering design, fabrication and quality control, and test and handling. The first problem to be identified by Ref. 4 concerns the lack of agreement on sources of unreliability. Test failure report (TFR) data from the Surveyor Program indicates that "propulsion components generally show a lower percentage of failures attributable to engineering and design, and higher percentages attributable to fabrication and quality control, relative to the other spacecraft subsystems" (Ref. 4). However, as stated in Ref. 4, Planning Research Corporation Report 948 (Study of Reliability Data from In-Flight Spacecraft, March 1967) "presents a distribution of failure causes quite different from those resulting from ground tests of the Surveyor spacecraft. Sixty percent of the reported in-flight failures from 225 launched spacecraft (35 programs) were attributed to 'various aspects of the spacecraft design,' twenty percent to manufacture, and twenty percent to spacecraft operation. Eighty-five percent of the reported in-flight anomalies were reported to have little or no effect on the accomplishment of the mission, while only 71 percent of the TFR's during the Surveyor testing were judged to be non-mission critical..."

Mechanical designers have typically not used "standard derating policies or worst-case analysis techniques, both of which are standard disciplines of the electronic designer. Design margins of mechanical systems are difficult to verify by testing; whereas electronic systems can be life and load tested quickly, easily, and economically" (Ref. 4).

Reliability prediction is further complicated by incomplete design knowledge and variations resulting from manufacturing processes, material, environment, and functional considerations. The use of "safety factors" raises the question of reliability equivalence since there is no one-to-one relationship between factors of safety and reliability.

Very poor agreement between mechanical component failure rate data has stemmed from the lack of agreement between manufacturers and between manufacturers and users. This, for example, results in solenoid valve failure rates distributed over three orders of magnitude.

To further complicate matters, spacecraft mechanisms have exhibited the disturbing characteristic "of sometimes degrading with test experience and having lower reliabilities associated with the later tested spacecrafts" (Ref. 4). Failure mechanics has almost always used a constant failure rate approach in reliability prediction. This technique is inadequate in that time and cycle dependency are not included. Moreover, meaningful time and cycle dependent data are unavailable.

Consequently, mechanical component and system reliability prediction is difficult at best. One is faced with the dilemma of either using available low confidence component failure data or attempting to include meaningful time and cycle dependent parameters for which no data exist.

This study will therefore consider a range of component reliabilities in order to attempt to establish some reliability "ranking" of the alternative propulsion mechanizations. High- and low-component reliabilities shown in Table 8 will be used to make relative comparisons of the propulsion alternatives.

Each subsystem component was assumed to be in series with the remaining components such that propulsion unreliabilities were assumed to lead to catastrophic failures (negation of spacecraft ability to obtain further data). Resulting subsystem reliabilities are summarized in Table 9. Note that the hybrid/mono subsystem has the lowest predicted reliability. This is due primarily to the large number of components comprising the monopropellant vernier subsystem. Elimination of this subsystem, then, using simpler methods for TVC and midcourse correction, could improve the reliability. Also note in Table 9 that the differences in reliability between the subsystems diminishes as greater reliability is achieved. Hence, for long-term missions where efforts are made to achieve high reliability, the differences in reliability between the alternative subsystems diminish and reliability becomes a less important factor in subsystem selection. This will be shown quantitatively in the cost-effectiveness analysis section.

D. Cost Effectiveness

The comparisons of the alternative subsystems on the basis of performance (mass), cost, and reliability do not provide sufficient information to determine the competitive position of the hybrid/monopropellant vernier system. Cost-effectiveness techniques provide a means for combining these three parameters into a single parameter, and allows the determination of the design's relative merit based on a single parameter. Also, the technique provides a means for determining the relative importance of performance, cost, and reliability.

The four propulsion alternatives were compared using the cost-effectiveness method presented in Ref. 4. This technique is a systems tradeoff analysis involving propulsion subsystem mass, R&D and hardware costs, and reliability. The first step in the analysis is to determine the nominal cost-effectiveness number for the mission. Cost effectiveness (CE) is defined as expected mission return per dollar spent:

$$CE = \frac{\text{Expected mission return}}{\text{Total mission cost per flight}} \qquad (1)$$

The expected mission return (EMR) in "units of value" is defined by the following summation:

$$EMR = \sum_{i=1}^{n} P_i W_i = P_1 W_1 + P_2 W_2 + \cdots + P_n W_n \qquad (2)$$

where W_i is the worth in "units of value" for the ith phase of the mission, and P_i is the corresponding spacecraft probability-of-success of accomplishing that mission phase.

1. Mission Cost Effectiveness. Although Ref. 4 analyzed a four-phase mission corresponding to a four-planet Grand Tour Mission, this study, in keeping with the limited time available, considered a "lumped" or single-phase Jupiter Orbiter Mission. Cost effectiveness can then be written as

$$CE = \frac{PW}{C_T} \qquad (3)$$

Mission worth (W) for this single-phase mission then becomes 100 units of value. Total mission cost per flight (C_T) is assumed at 200 million dollars for purposes of this study only. The baseline Jupiter Orbiter Mission (Ref. 1) is approximately 5 yr in duration. The last propulsion burn occurs approximately 2 yr after orbit insertion or 4 yr after launch. Thus, the required operative lifetime of a propulsion subsystem is 4 yr. Consequently, the probability-of-success has a value corresponding to that for a 4 yr period. The Jupiter Orbiter spacecraft was assumed, as in Ref. 1, to be a derivative of the Grand Tour spacecraft, thereby enabling curves of spacecraft probability-of-success as a function of mission time to be generated from TOPS data (Refs. and 11) and to be used in this study. The probability-of-success is 64.4% for a Grand Tour-type spacecraft, with 68 kg (150 lbm) of redundancy, 4 yr after launch.

The resultant nominal mission cost effectiveness is

$$CE = \frac{(0.644)(100) \text{ units of mission value}}{200 \text{ M\$}}$$

$$CE = 0.322 \frac{\text{units of mission value}}{\text{M\$}} \qquad (4)$$

2. Cost-Effectiveness Method for Propulsion Comparisons. Cost-effectiveness comparisons for spacecraft having different propulsion subsystems can be made with Eq. (3) by modifying the equation as follows:

$$CE = \frac{W P_e P_p}{C_e + C_p} \qquad (5)$$

where

W = mission worth

P_p = propulsion subsystem probability-of-success

P_e = probability-of-success of the rest of the spacecraft

C_p = propulsion cost

C_e = cost of everything else

Using the above expression one could compute the relative total cost-effectiveness advantages and disadvantages of the hybrid/mono subsystem with respect to competing options. However, such a comparison would not reveal the relative importance of reliability, cost, and performance. Instead, Eq. (5) can be differentiated to give

$$\Delta CE = \frac{\partial CE}{\partial M}\Delta M + \frac{\partial CE}{\partial C_p}\Delta C_p + \frac{\partial CE}{\partial P_p}\Delta P_p \qquad (6)$$

where M is mass and P is equivalent to reliability. The expression of Eq. (6) defines the difference in cost effectiveness from a baseline, which in this analysis will be taken as the hybrid/mono. The ΔCE is defined in terms of mass, cost, and reliability. These terms can be calculated separately and their relative influence noted. In order to calculate the ΔCE's, the data to compute the Δ's for mass, cost, and reliability have already been presented in Tables 6, 7, and 9. However, the partial derivatives, or influence coefficients, must be developed.

$\partial CE/\partial C_T$ and $\partial CE/\partial R_p$ can be obtained by differentiating Eq. (3) and substituting the appropriate numerical values. $\partial CE/\partial M$ is more difficult since it is a function of mission worth. The development of these influence coefficients is described in the following section.

3. Relative Importance of Mass or Performance. The hybrid motor/monopropellant vernier alternative is established as the reference subsystem because the objective is to compare the hybrid with alternatives in order to determine the applicability of hybrids to planetary missions. Propulsion masses above and below the reference mass and resultant mass differences from Table 6 can be equated to (1) added mission capability (increased experiments), (2) increased spacecraft probability-of-success through the use of redundancy, and (3) reduced cost.

The influence of mass on cost effectiveness can be determined by taking the total derivative of CE. This can be done by, first, taking the total differential of Eq. (3) which is a function of P, W, and C_T:

$$CE = f(P, W, C_T)$$

$$dCE = \frac{\partial CE}{\partial P} dP + \frac{\partial CE}{\partial W} dW + \frac{\partial CE}{\partial C_T} dC_T \qquad (7)$$

The total derivative of CE with respect to M is

$$\frac{dCE}{dM} = \frac{\partial CE}{\partial P}\frac{dP}{dM} + \frac{\partial CE}{\partial W}\frac{dW}{dM} + \frac{\partial CE}{\partial C_T}\frac{dC_T}{dM} \qquad (8)$$

This equation shows that a change in mass (dM) within the total spacecraft system could affect changes in (1) spacecraft probability-of-success, dP; (2) mission worth dW; and (3) total mission cost per flight dC_T. Each effect is considered below in detail.

a. Influence of mass on increased spacecraft probability-of-success. This case considers the allocation of potential savings in propulsion mass to spacecraft reliability improvement and is analyzed in the following manner.

The first term on the right side of Eq. (8) can be further reduced by differentiating Eq. (3):

$$\frac{\partial CE}{\partial P} = \frac{W}{C_T} \qquad (9)$$

Then,

$$\frac{\partial CE}{\partial P}\frac{dP}{dM} = \frac{W}{C_T}\frac{dP}{dM} \qquad (10)$$

Equation (10) can be written in incremental form as

$$\frac{\Delta CE}{\Delta M_p} = \frac{W}{C_T}\frac{\Delta P}{\Delta M} \qquad (11)$$

The probability-of-success with 68 kg (150 lbm) of redundancy distributed throughout the spacecraft is 0.644; without redundancy, it is 0.418 (Refs. 4 and 11). The influence coefficient relating cost effectiveness to redundancy is

$$\frac{\Delta CE}{\Delta M_p} = \frac{(100 \text{ units})(0.226)}{(200 \text{ M\$})\ (68 \text{ kg})}$$

$$= 1.66 \times 10^{-3} \frac{\text{units/M\$}}{\text{kg}}$$

$$\left(0.753 \times 10^{-3} \frac{\text{units/M\$}}{\text{lbm}}\right) \qquad (12)$$

Addition of redundancy to the spacecraft and the resultant effect on probability-of-success over a mass range is not well defined at this time. The resultant impact on $\Delta P/\Delta Mass$ is well summarized by Ref. 4:

> "A straight line... between the two known points would infer that a constant increment of reliability can be gained per pound of mass as mass is added to the spacecraft — since this cannot be true... a decreasing $dP_s/dMass$ must be assumed."

The contribution of mass redundancy to cost effectiveness can be bounded in order to overcome the unknown $\Delta P/\Delta Mass$ slope problem. Consequently, the cost effectiveness/redundancy influence coefficient can be considered, as in Ref. 4, to lie within a range given by 1/6 to 5/6 of the value stated in Eq. (12), or

$$\frac{\Delta CE}{\Delta M_p} = \begin{cases} 0.277 \times 10^{-3} \dfrac{\text{units/M\$}}{\text{kg}} \\ \left(0.126 \times 10^{-3} \dfrac{\text{units/M\$}}{\text{lbm}}\right) \\ \text{to} \\ 1.383 \times 10^{-3} \dfrac{\text{units/M\$}}{\text{kg}} \\ \left(0.628 \times 10^{-3} \dfrac{\text{units/M\$}}{\text{lbm}}\right) \end{cases} \qquad (13)$$

Further interpretation of this coefficient will be discussed in a later section.

b. Influence of mass on increased mission capability. This case considers "excess" mass to be allocated to increased experiments instead of spacecraft reliability improvement.

Science payload for the 1982 Jupiter Orbiter is 87 kg (192 lbm) (Ref. 1). Differences in propulsion mass (Table 6) between the hybrid/mono and the lightest and heaviest alternative designs range from approximately +52 kg (+114 lbm) (OF_2/B_2H_6 < hybrid/mono) to -156 kg (-344 lbm) (hybrid/mono < Be solid/mono), respectively. These differences can be examined in terms of additional science payload mass as a result of a reduction in propulsion mass due to increased performance or vice versa. In no case would science mass be lower than the 87 kg (192 lbm) required by the mission.

A change in cost effectiveness with respect to a change in mission worth can be shown by differentiating Eq. (3):

$$\frac{\partial CE}{\partial W} = \frac{P}{C_T} \tag{14}$$

and

$$\frac{\partial CE}{\partial W}\frac{dW}{dM} = \frac{P}{C_T}\frac{dW}{dM} \tag{15}$$

Equation (15) can be written in incremental form as

$$\frac{\Delta CE}{\Delta M_W} = \frac{P}{C_T}\frac{\Delta W}{\Delta M} \tag{16}$$

$\Delta W/\Delta M$ is the relation between changes in mission worth and science payload mass variations and is the slope of the mission worth versus science payload curve shown in Fig. 8 from Ref. 4. The orbiter which is a derivative of the Grand Tour spacecraft (Ref. 1) is assumed to have a mission worth versus science payload curve of comparable shape. The science payload range of interest is the region above 87 kg (192 lbm) because propulsion mass is to be traded against additional science capability. This range is in the neighborhood of approximately 91 kg (200 lbm) plus 45 to 136 kg (100 to 300 lbm) additional mass. The slope of the mission worth curve in the 136- to 227-kg (300- to 500-lbm) region is conservatively assumed to be one order of magnitude lower than the final slopes shown in Fig. 8. This more than compensates for the decrease in cost effectiveness due to instrument costs.

It is of interest and importance to note that the mission worth curve could conceivably rise as more experimental payload is added, resulting in an increasing $\Delta W/\Delta M$ slope. This could be the case of a Grand Tour-type spacecraft carrying an atmospheric probe, or additional engineering experiments could highly enhance the worth of the mission. In either case, if $\Delta W/\Delta M$ were to rise instead of falling off as additional capability is added, then high propulsion performance would be of even greater importance.

The influence coefficient relating cost effectiveness to mission worth through additional capability is given by

$$\frac{\Delta CE}{\Delta M_W} = \left(\frac{0.644}{200\ M\$}\right)\left(\frac{0.3\ \text{units}}{0.454\ \text{kg}}\right)$$

$$= 2.128 \times 10^{-3}\ \frac{\text{units/M\$}}{\text{kg}}$$

$$\left(0.966 \times 10^{-3}\ \frac{\text{units/M\$}}{\text{lbm}}\right) \tag{17}$$

However, approximately 0.27 kg (0.6 lbm) of structure, telecommunications, and power are required to support 0.454 kg (1 lbm) of science payload on the Thermoelectric Outer-Planet Spacecraft (TOPS) (Ref. 4). This effectively reduces the magnitude of the above influence coefficient to

$$\frac{\Delta CE}{\Delta M_W} = \frac{2.128 \times 10^{-3}}{1.6}\ \frac{\text{units/M\$}}{\text{kg}}$$

$$= 1.33 \times 10^{-3}\ \frac{\text{units/M\$}}{\text{kg}}$$

$$\left(0.604 \times 10^{-3}\ \frac{\text{units/M\$}}{\text{lbm}}\right) \tag{18}$$

Note that the magnitude of this coefficient is equivalent to the magnitude of the cost-effectiveness/redundancy coefficient given by Eq. (13). Thus, the effect of using excess mass for reliability improvement or increased scientific benefit results in approximately equal cost-effectiveness benefit to the spacecraft.

c. Influence of mass on total mission cost per flight. The effect of mass differences on total mission cost per flight (C_T) and cost effectiveness will now be examined. Differentiating Eq. (3) with respect to C_T gives

$$\frac{\partial CE}{\partial C_T} = \frac{-WP}{C_T^2} \tag{19}$$

and

$$\frac{\partial CE}{\partial C_T}\frac{dC_T}{dM} = \frac{-WP}{C_T^2}\frac{dC_T}{dM} \tag{20}$$

or, in incremental form,

$$\frac{\Delta CE}{\Delta M_{C_T}} = \frac{-WP}{C_T^2}\frac{\Delta C_T}{\Delta M} \tag{21}$$

The effect of this term on Eq. (8) can be determined by examining its magnitude. Consider the effect to be due to 68 kg (150 lbm) of mass difference (ΔM). Assume that the cost differential due to ΔM is 1% of C_T or 2 M$. This estimate is based on Viking Orbiter design data (Ref. 12). There are, however, differences between Viking's usage of the data and ours. Viking in effect is paying a certain dollars per pound to reduce the mass of support subsystems in order that this mass can be used for payload. Our dollar per pound usage for this case, however, is aimed at reduced cost through reduced development and/or heavier hardware and so is different from Viking. The two processes are not equivalent nor are they unrelated. Therefore, as a first estimate the Viking data is used to calculate the influence of mass on the mission cost.

$$\left|\frac{\Delta CE}{\Delta M_{C_T}}\right| = \frac{(100\ \text{units})\ (0.644)\ (2\ M\$)}{(200\ M\$)^2\ (68\ \text{kg})}$$

$$= 0.473 \times 10^{-4}\ \frac{\text{units/M\$}}{\text{kg}}$$

$$\left(0.214 \times 10^{-4}\ \frac{\text{units/M\$}}{\text{lbm}}\right) \tag{22}$$

This influence coefficient is seen to be one order of magnitude lower than those derived from reliability and scientific benefit considerations, and thus will be neglected in the ensuing analysis.

d. Advanced study emphasis on performance. It has been found that influence coefficients for increased spacecraft probability-of-success (Eq. 13) and increased mission worth (Eq. 18) are of equivalent magnitude. This implies that both parameters result in approximately the same mission cost-effectiveness benefit to the spacecraft. Also, recall that both influence coefficients are derived from the cost-effectiveness model stated in Eq. (3) and are dependent on "conversion" or reallocation of mass to redundancy and increased experiments, respectively. These coefficients, in effect, attributed a certain "importance" to mass within the mission cost-effectiveness model used in this study. Therefore, a single "mass" influence-coefficient representative of redundancy and mission worth, equal to the low value given in Eq. (13), will be used to characterize the case where mass is important.

e. Flight project emphasis on performance. Flight project emphasis on performance is constrained by cost. Flight programs emphasize low cost more than high-propulsion performance and resultant low-propulsion mass. However, this does not say that propulsion performance and mass have zero importance; they just have less importance during projects than during advanced studies since project costs are established without knowledge of all development work required. For example, Viking Orbiter "design changes... are implemented in a stepwise manner, using what appears to be the most cost-effective [low cost] design changes first, etc." (Ref. 12). Typical improvements in Viking Orbiter propulsion, which can increase payload (lander/capsule mass), cost from \$2900 to \$17200 per kilogram (\$1300 to \$7800 per pound mass). These design changes are summarized in Appendix A.

Jupiter Orbiter propulsion hardware cost per kilogram (or pound mass) can be determined from Eq. (13) and Eq. (33) discussed in Section V-D-4-b:

$$\frac{\Delta C_{PH}}{\Delta M} = \left(\frac{\Delta CE}{\Delta M}\right)\left(\frac{\Delta C_{PH}}{\Delta CE}\right) = -172{,}000\,\frac{\$}{kg} \quad \left(-78{,}300\,\frac{\$}{lbm}\right) \tag{23}$$

This ratio is approximately one order of magnitude larger than current Viking program expenditure. Therefore, a single "mass" influence coefficient equal to one-tenth of the low value of Eq. (13) will be used to represent project emphasis on performance, or a reduced importance of mass:

$$\frac{\Delta CE}{\Delta M} = 0.0277 \times 10^{-3}\ \frac{units/M\$}{kg} \quad \left(0.0126 \times 10^{-3}\ \frac{units/M\$}{lbm}\right) \tag{24}$$

4. Relative Importance of Propulsion Costs. A previous section determined the cost effectiveness influence coefficient due to mass. One of the factors investigated was the influence of mass variation on total mission cost per flight which was found to be relatively insignificant. In this section the influence of propulsion costs on cost effectiveness will be computed. Mission cost effectiveness has been defined as

$$CE = \frac{PW}{C_T} \tag{3'}$$

Total mission cost per flight (C_T) can be divided into spacecraft propulsion cost (C_P) and the cost of everything else (C_E), including launch vehicle, spacecraft, operations, etc. Propulsion cost can be further divided into hardware (C_{PH}) and development (C_{PD}) costs. C_T can then be written as

$$C_T = C_E + C_P = C_E + C_{PH} + C_{PD} \tag{25}$$

Substituting Eq. (25) into Eq. (3') gives

$$CE = \frac{PW}{C_E + C_{PH} + C_{PD}} \tag{26}$$

a. Development cost. The effect of a change in development cost on CE can be determined by differentiating Eq. (26) with respect to C_{PD}:

$$\frac{\partial CE}{\partial C_{PD}} = \frac{(C_E + C_{PH} + C_{PD})\dfrac{\partial(PW)}{\partial C_{PD}} - PW\dfrac{\partial}{\partial C_{PD}}(C_E + C_{PH} + C_{PD})}{(C_E + C_{PH} + C_{PD})^2} \tag{27}$$

Assume that changes in spacecraft propulsion development cost have negligible effect on (1) expected mission return PW, (2) cost C_E, and (3) propulsion hardware cost C_{PH}. Then,

$$\frac{\partial(PW)}{\partial C_{PD}} = 0, \quad \frac{\partial C_E}{\partial C_{PD}} = 0, \text{ and } \frac{\partial C_{PH}}{\partial C_{PD}} = 0$$

Equation (27) reduces to

$$\frac{\partial CE}{\partial C_{PD}} = -\frac{PW}{(C_E + C_{PH} + C_{PD})^2} = -\frac{PW}{C_T^2} \tag{28}$$

The Jupiter Orbiter Mission described in Ref. 1 was based on a single launch. By considering a flight unit plus a spare, nonrecurring propulsion cost can be spread over two units. The resultant development cost influence coefficient is

$$\frac{\partial CE}{\partial C_{PD}} = -\frac{PW}{2C_T^2} \tag{29}$$

Propulsion development cost effects on CE can be examined in terms of a unit million dollar change:

$$\frac{\Delta CE}{\Delta C_{PD}} = -\frac{(0.644)(100 \text{ units})}{2(200 \text{ M\$})^2}$$

$$= -0.805 \times 10^{-3} \frac{\text{units/M\$}}{\text{M\$}} \quad (30)$$

b. Hardware cost. The effect of a change in hardware cost on CE can be determined by differentiating Eq. (26) with respect to C_{PH}:

$$\frac{\partial CE}{\partial C_{PH}} =$$

$$\frac{(C_E + C_{PH} + C_{PD})\frac{\partial(PW)}{\partial C_{PH}} - PW\frac{\partial}{\partial C_{PH}}(C_E + C_{PH} + C_{PD})}{(C_E + C_{PH} + C_{PD})^2} \quad (31)$$

Assume that changes in spacecraft propulsion hardware cost have negligible effect on (1) expected mission return PW, (2) cost C_E, and (3) propulsion development cost C_{PD}, such that

$$\frac{\partial(PW)}{\partial C_{PH}} = 0, \quad \frac{\partial C_E}{\partial C_{PH}} = 0, \text{ and } \frac{\partial C_{PD}}{\partial C_{PH}} = 0$$

Equation (31) is then reduced to

$$\frac{\partial CE}{\partial C_{PH}} = -\frac{PW}{(C_E + C_{PH} + C_{PD})^2} = -\frac{PW}{C_T^2} \quad (32)$$

Hardware cost effects on CE are analyzed in a fashion similar to development cost. Thus, in terms of a unit million dollar change in propulsion hardware cost:

$$\frac{\Delta CE}{\Delta C_{PH}} = \frac{-(0.644)(100 \text{ units})}{(200 \text{ M\$})^2}$$

$$= -1.61 \times 10^{-3} \frac{\text{units/M\$}}{\text{M\$}} \quad (33)$$

5. Relative Importance of Reliability. Changes in propulsion affect CE as follows. Spacecraft probability-of-success (P) can be considered as a product of propulsion reliability (R_P) and the reliability of everything else (R_E):

$$P = R_E R_P \quad (34)$$

Equation (3) can then be written as

$$CE = \frac{WR_E R_P}{C_T} \quad (35)$$

Differentiating Eq. (35) with respect to R_P gives

$$\frac{\partial CE}{\partial R_P} = \frac{WR_E}{C_T} = \frac{\frac{WP}{C_T}}{R_P} \quad (36)$$

The reference hybrid/mono propulsion subsystem reliability (R_P) is chosen as the baseline for computation. Recalling that a range of reliabilities were computed, the influence coefficient corresponding to low-component reliabilities is

$$\frac{\Delta CE}{\Delta R_{PL}} = \frac{0.322 \text{ units/M\$}}{0.8491} = 0.379 \text{ units/M\$} \quad (37)$$

and the influence coefficient corresponding to high-component reliabilities is

$$\frac{\Delta CE}{\Delta R_{PH}} = \frac{0.322 \text{ units/M\$}}{0.9885} = 0.326 \text{ units/M\$} \quad (38)$$

6. Cost-Effectiveness Comparisons. Cost-effectiveness comparisons were made on the four competing subsystems for the Jupiter Orbiter Mission. Two component reliability extremes were considered. Also, two extremes representing varying importance of mass were considered. The comparisons for the resulting four cases are discussed below and a sample calculation is shown in Appendix B.

a. Low-component reliabilities. Table 10 shows the cost-effectiveness comparisons for low component reliabilities when mass is important. The table shows percent cost-effectiveness difference from the reference hybrid/mono system. Note that the differences are of the order of 8 to 12% of the total mission cost effectiveness, which is enough of a variation to provide meaningful conclusions. Also note that the mass and reliability cost-effectiveness differences are generally much more significant than cost. This is true for most of the comparisons. For the comparison of Table 10, the hybrid is seen to be midway between the liquid and solid alternatives, with the liquid alternatives having a definite superiority. Note that the superiority is primarily due to the better reliability of the liquid subsystems. For this case, where the component reliabilities are low, the spread between the reliability values is large resulting in the large differences in the cost-effectiveness component due to reliability.

Table 11 shows the case where low-component reliabilities have again been assumed, but now mass is considered less important. Mass could be considered less important if, for example, excess launch vehicle capability were available. This case is the only one in which the hybrid is not competitive with at least one of the alternate subsystems. All other systems are superior. Note that the only difference between this table and the previous one is in the mass-related cost-effectiveness numbers which have been reduced by an order of magnitude. Since the hybrid/mono option is a high performing subsystem with low relative reliability, it fares poorly in comparison with the other alternatives for this case.

b. High-component reliabilities. The assumption of low reliability, as discussed in the previous section, is open to question for the missions of interest. These missions are of very long duration that will require high reliability for their accomplishment. Thus, efforts on these programs will be to achieve high reliability. The following two assessments, based on the high reliability assumption, are therefore more realistic.

Table 12 shows the case where high reliability has been assumed in conjunction with the assumption that mass is of reduced importance. Note that the reliability numbers are now relatively low. This is due to the small differences between the high reliability numbers. That is, as high reliability is strived for, the cost-effectiveness differences due to reliability between alternative designs becomes less. These low numbers combined with the low mass-related cost-effectiveness numbers result in total cost-effectiveness differences which are quite small. Thus, for this case all subsystems are competitive.

The final comparison, shown in Table 13, is probably the most realistic. The assumptions for it are that the subsystems must have high reliability and that mass is important. The high reliability assumption has already been shown to be a more reasonable assumption than low reliability. Also, the case of mass being important is more realistic than the case where mass is of reduced importance. The latter assumes more than adequate launch vehicle capability, which with the uncertainties in launch vehicles could prove to be a bad assumption. And if technology were developed which did not emphasize the importance of mass, or the need for improved propulsion subsystem performance, then the advanced technology program may not provide the propulsion necessary for the mission. Thus, the advanced technology program must be directed with the assumption that spacecraft mass is important. For this case, as shown by Table 13, the hybrid/mono is seen to be superior to the beryllium solid/mono, equivalent to the dual-mode liquid, and is slightly inferior to the OF_2/diborane liquid. It can therefore be concluded that the hybrid/mono is competitive with the alternative subsystems for the mission.

VI. HYBRID PROPULSION DEVELOPMENT REQUIREMENTS

The previous analysis and comparisons were based upon the hypothesis that each of the propulsion options can be made to function within the constraints of the missions. There are two basic technology requirements which must be satisfied by the hybrid/mono design in order to establish feasibility:

(1) A nozzle must be developed to operate for the long burn times under conditions of multiple restart.

(2) The fuel grain must withstand long-term space storage conditions.

In order for the hybrid option to develop into a superior subsystem, it is clear that the monopropellant vernier subsystem must be eliminated. This would increase both the reliability and performance of this design. To accomplish this will require a thrust vector control subassembly and a throttling or dual-thrust capability for the hybrid. If the hybrid could be so developed, its schematic diagram would be as shown in Fig. 9. Comparisons were made of this hybrid design with the other propulsion alternatives. Table 14 shows that only the OF_2/diborane liquid design is competitive with the hybrid, but the hybrid is only approximately 3% superior to the dual-mode alternative. If the performance of the hybrid can also be improved to 3923 N-s/kg (400 lbf-s/lbm), then the hybrid increases 1.9% in ΔCE as shown by Table 15; however, some of this advantage may be lost due to increases in cost and reduced reliability. In summary, the development requirements for the hybrid are:

(1) Nozzle for long burn times and multiple restarts.

(2) Space storable fuel grain.

(3) Thrust vector control.

(4) High-accuracy, low-thrust capability for trajectory correction maneuvers.

(5) Improved I_{sp}.

VII. CONCLUSIONS

The basic conclusions of the study are:

(1) The hybrid/monopropellant vernier alternative is competitive with competing bipropellant liquid-propulsion subsystems and superior to the Be solid/mono option.

(2) If the monopropellant vernier subsystem is replaced with a TVC subsystem and accurate dual-thrust capability, the hybrid subsystem could be superior to the solid/mono and the dual-mode liquid and directly competitive with the OF_2/diborane liquid design.

These conclusions are based on the need for high reliability and when the launch vehicle does not provide excess capability.

VIII. TOPICS NOT COVERED BY STUDY

Topics not detailed in this study are development risk, materials compatibility, and Grand Tour technology spin-off. Development risk for each propulsion alternative has not been fully evaluated beyond estimated R&D cost required for a successful technology demonstration. The probability-of-success of developing a long burn time berylliumized solid-propellant motor, a long burn time hybrid nozzle, an OF_2/diborane engine assembly, a dual-mode bipropellant engine, temperature-actuated propellant valves, etc., is difficult to assess at this time. Even though certain propellants (for example, composite solid propellants, neat hydrazine, etc.) have less severe material compatibility problems than their more energetic counterparts, no factor was included in the cost-effectiveness comparison to quantify this aspect.

The Jupiter Orbiter spacecraft was assumed as in Ref. 1 to be a derivative of the Grand Tour flyby spacecraft. In spite of this, no attempt has been made during this study to account for foreseeable propulsion developments which could benefit orbiter propulsion. The long life in space monopropellant hydrazine technology that will result from the outer-planet flyby missions has not been applied to appropriate propulsion alternatives considered in this study, nor has the hydrazine attitude propulsion subsystem with its propellant sharing features been included in the study.

Only one type of hybrid propulsion was considered in this study. It was assumed to be most representative of high-performance designs. However, there are other candidate hybrid systems which should be evaluated in the future

Table 1. Jupiter Orbiter Mission characteristics

Nominal launch	Dec. 25, 1981 to Jan. 8, 1982
Arrival date	Jan. 25, 1984
Flight time, days	753 ±8
Launch energy (C_3), km^2/s^2	85
Orbit dimensions	4 × 98.8 Jupiter radii
Orbital period, days	45.4
Active life in orbit, yr	3

Parameter	ΔV requirement, m/s	Time	Propulsion accuracy, m/s
Total guidance correction allocation	104		
Midcourse	(18)	Launch +7 to 10 days	0.1
Pre-encounter	(21)	Encounter - 40 days	0.1
Orbit trim	(65)	Insertion + 10 days	0.1
Orbit insertion, nominal	1477	Jupiter encounter	7.5
Minimum	(1461)		
Maximum	(1488)		
Post insertion maneuvers allocation	200		
Satellite encounters	---	{Insertion + 43 to 46 days} {Periapsis + 12 h}	0.1
Major orbit change	---	Insertion + 2 yr	0.1
Total requirement	1781		

Table 2. Saturn Orbiter Mission characteristics

Nominal launch	Jan. 9 to 2[illegible], 1984
Arrival date	Aug. 1, 1989
Flight time, days	2025 ±8
Launch energy (C_3), km^2/s^2	123
Orbit dimensions	3.5 × 120 Saturn radii
Orbital period, days	85.1
Active life in orbit, yr	3 (desired) 1 (minimum)
ΔV requirement, m/s	
Total guidance correction allocation	104
Orbit insertion	1036
Ring encounter	129
Total ΔV requirement	1269

Table 3. Interface criteria

Interface parameter	Jupiter orbiter	Saturn orbiter
Acceleration, g		
Initial step	0.2	0.2
Maximum	1.0	1.0
Acceleration rate, g/s		
Ignition	0.2	0.2
Cutoff	0.2	0.2
Radiation environment		
RTG	Gamma, neutron	Gamma, neutron
Space	Protons	Protons
Planetary	Electron, proton	?
Space storage, yr	5	8
Geometrical constraint	3.35-m (11-ft) diam Titan shroud	3.35-m (11-ft) diam Titan shroud
Temperature environment	See Table 4	See Table 4

Table 4. Propellant temperatures

Propulsion alternative	Temperature, K (°F)	
	Minimum	Maximum
Hybrid/monopropellant verniers		
N_2H_4 monopropellant	278 (40)	306 (90)
FLOX	56 (-360)	100 (-280)
or		
OF_2	56 (-360)	156 (-180)
Fuel grain	233 (-40)	317 (110)
Solid/monopropellant verniers		
Be solid motor	233 (-40)	317 (110)
Hydrazine nitrate	261 (10)	306 (90)
OF_2/B_2H_6 bipropellant liquid		
OF_2	56 (-360)	156 (-180)
B_2H_6	117 (-250)	222 (-60)
F_2/N_2H_4 dual mode		
F_2	56 (-360)	100 (-280)
N_2H_4	278 (40)	306 (90)

Table 5. Propulsion characteristics

Characteristics	Solid propellant-liquid monopropellant verniers	Hybrid-liquid monopropellant verniers	OF_2/B_2H_6 bipropellant (mild cryogens)	Dual-mode bipropellant
Propellants	Berylliumized solid Nitrated hyrazine monopropellant	Li/LiH/PBD fuel FLOX or OF_2 oxidizer Neat hydrazine monopropellant	Oxygen difluoride Diborane	Fluorine Neat hydrazine
Specific impulse, N-s/kg (lbf-s/lbm)	3089 (315) (solid) 2452 (250) (monopropellant)	3805 (388) (hybrid) 2256 (230) (monopropellant)	3923 (400)	3727 (380) (bipropellant) 2256 (230) (monopropellant)
Mass fraction	0.90/0.80	0.858/0.75	0.84	0.84
Thrust, N (lbf)	9341 (2100) solid motor (max)-throttleable liquid	8006 (1800) hybrid motor (max)-throttleable liquid	4448 (1000)	2669 (600) (bipropellant) 667 (150) (monopropellant)
Pressurization mode	Blowdown liquid	Pressure regulated	Pressure regulated	Pressure regulated
System joints	Welded	Welded	Welded	Welded
Tank material				
Fuel	Titanium	---	Aluminum-lined boron composite	Titanium
Oxidizer	---	Aluminum		Aluminum
Propellant acquisition device	Surface tension	Surface tension	Surface tension	Surface tension (N_2H_4 side only)
Pressurant gas	Helium	Helium	Helium	Helium
Thrust vector control	Throttling	Throttling	Gimbaled engine	Gimbaled engine
Motor case	Titanium	Filament wound composite	---	---

Table 6. Jupiter Orbiter propulsion subsystem masses

Propulsion alternative	Injected mass, kg (lbm)	Propulsion mass, kg (lbm)	Mass difference from hybrid/mono, kg (lbm)
Hybrid/mono			
Mode 1	1215 (2677)	557 (1227)	0
Solid/mono			
Mode 2	1372 (3021)	713 (1571)	-156 (-344)
OF_2/B_2H_6 bipropellant	1164 (2563)	505 (1113)	+52 (+114)
F_2/N_2H_4 bipropellant			
Mode 1	1213 (2671)	554 (1221)	+3 (+6)

(-) Indicates a heavier subsystem than the hybrid/mono.

(+) Indicates a lighter subsystem than the hybrid/mono.

Payload mass = 685 kg (1450 lbm).

Jupiter Orbiter ΔV requirement, m/s:

Pre-insertion	=	39
Orbit insertion	=	1477
Orbit trim	=	65
Orbit change	=	200
Total	=	1781

Mode 1 - Pre-insertion ΔV done with monopropellant. Orbit insertion, orbit trim, and orbit change ΔV done with combination.

Mode 2 - Pre-insertion, orbit trim, and orbit change ΔV done with monopropellant. Orbit insertion ΔV done with combination.

Table 7. Propulsion cost estimates

Propulsion alternative	R&D cost, M$		Hardware cost, M$	Total cost two flights high reliability, M$
	Low reliability	High reliability		
Hybrid/monopropellant vernier	16	18.5	4.5	27.5
Hybrid	11		3	
Monopropellant vernier	5		1.5	
Be solid/monopropellant vernier	11.5	14	3.5	21
Be solid	6.5		2	
Monopropellant vernier	5		1.5	
OF_2/B_2H_6 bipropellant	22	23.75	4	31.75
F_2/N_2H_4 bipropellant	19.5	21.25	4.5	30.25

Table 8. Summary of component reliabilities[a]

Component	Low	High
Solenoid valve:		
Monopropellant	0.990	0.9996
Bipropellant valve assembly	0.985	0.9994
Pressure regulator	0.994	0.9996
Bipropellant TCA	0.996	0.9997
Helium lines, fittings, connections	0.9980	0.99985
Relief valve	0.9984	0.99985
Nitrogen lines, fittings, connections	0.9985	0.99990
Monopropellant lines, fittings, connections	0.9992	0.99994
Bipropellant lines, fittings, connections	0.9985	0.99990
NO/NC valve	0.9990	0.99990
Fill valve	0.9994	0.99992
Pressurant tank	0.9996	0.99995
Fuel tank	0.9997	0.99996
Monopropellant TCA	0.99985	0.99997
Pressure transducer	0.99985	0.99997
Filter	0.99988	0.999972
Temperature transducer	0.99988	0.999972
Be solid motor[b]	0.990	0.995
Hybrid motor[c]	0.9930	0.9980
Gimbal subassembly	0.9950	0.9999

[a] Reference 4.

[b] Reference 9.

[c] Reference 10.

Table 9. Propulsion reliability estimates

Propulsion alternative	Low	High
Hybrid/monopropellant	0.8491	0.9885
Hybrid	0.9529	0.9947
Monopropellant verniers	0.8911	0.9938
Solid/monopropellant	0.8821	0.9888
Solid motor	0.990	0.995
Monopropellant verniers	0.8911	0.9938
OF_2/B_2H_6	0.9240	0.9943
F_2/N_2H_4	0.9213	0.9937

Table 10. Cost-effectiveness comparisons: mass is important, low-component reliabilities

Comparison parameter	ΔCE, %			
	Be solid/mono	OF_2/B_2H_6 liquid	Dual-mode liquid	Hybrid/mono
Mass	-13.42	+4.47	+0.26	0
R&D cost	+ 1.12	-1.50	-0.88	0
Hardware cost	+ 0.50	+0.25	0	0
Reliability	+ 3.89	+8.82	+8.50	0
Total ΔCE	-7.91	+12.04	+7.88	0

(+) ΔCE signifies advantage with respect to hybrid/mono.

(-) ΔCE signifies disadvantage with respect to hybrid/mono.

Table 11. Cost-effectiveness comparisons: reduced mass importance, low-component reliabilities

Comparison parameter	ΔCE, %			
	Be solid/mono	OF_2/B_2H_6 liquid	Dual-mode liquid	Hybrid/mono
Mass	-1.34	+0.45	+0.02	0
R&D cost	+1.12	-1.50	-0.86	0
Hardware cost	+0.50	+0.25	0	0
Reliability	+3.89	+8.82	+8.50	0
Total ΔCE	+4.17	+8.02	+7.64	0

(+) ΔCE signifies advantage with respect to hybrid/mono.

(-) ΔCE signifies disadvantage with respect to hybrid/mono.

Table 12. Cost-effectiveness comparisons: reduced mass importance, high-component reliabilities

Comparison parameter	ΔCE, %			
	Be solid/mono	OF_2/B_2H_6 liquid	Dual-mode liquid	Hybrid/mono
Mass	-1.34	+0.45	+0.02	0
R&D cost	+1.12	-1.31	-0.69	0
Hardware cost	+0.50	+0.25	0	0
Reliability	+0.03	+0.59	+0.53	0
Total ΔCE	+0.31	-0.02	-0.14	0

(+) ΔCE signifies advantage with respect to hybrid/mono.

(-) ΔCE signifies disadvantage with respect to hybrid/mono.

Table 13. Cost-effectiveness comparisons: mass is important, high-component reliabilities

Comparison parameter	ΔCE, %			
	Be solid/mono	OF_2/B_2H_6 liquid	Dual-mode liquid	Hybrid/mono
Mass	-13.42	+4.47	+0.26	0
R&D cost	+ 1.12	-1.31	-0.69	0
Hardware cost	+ 0.50	+0.25	0	0
Reliability	+ 0.03	+0.59	+0.53	0
Total ΔCE	-11.77	+4.00	+0.10	0

(+) ΔCE signifies advantage with respect to hybrid/mono.

(-) ΔCE signifies disadvantage with respect to hybrid/mono.

Table 14. Hybrid comparison: mass is important, high-component reliabilities

Comparison parameter	ΔCE, %			
	Be solid/mono	OF_2/B_2H_6 liquid	Dual-mode liquid	Hybrid
Mass	-15.92	+1.98	-2.24	0
R&D cost	+ 1.12	-1.31	-0.69	0
Hardware cost	+ 0.25	0	-0.25	0
Reliability	- 0.54	+0.01	-0.05	0
Total ΔCE	-15.09	+0.68	-3.23	0

(+) ΔCE signifies advantage with respect to hybrid.

(-) ΔCE signifies disadvantage with respect to hybrid.

Hybrid $I_{sp} = 3805 \frac{N\text{-}s}{kg}\left(388 \frac{lbf\text{-}s}{lbm}\right)$.

Hybrid mass fraction = 0.84.

Table 15. Hybrid comparison (improved hybrid I_{sp}): mass is important, high-component reliabilities

Comparison parameter	ΔCE, %			
	Be solid/mono	OF_2/B_2H_6 liquid	Dual-mode liquid	Hybrid
Mass	-17.89	0	-4.21	0
R&D cost	+ 1.12	-1.31	-0.69	0
Hardware cost	+ 0.25	0	-0.25	0
Reliability	-0.54	+0.01	-0.05	0
Total ΔCE	-17.06	-1.30	-5.20	0

(+) ΔCE signifies advantage with respect to hybrid.

(-) ΔCE signifies disadvantage with respect to hybrid.

Hybrid $I_{sp} = 3923 \frac{N\text{-}s}{kg} \left(400 \frac{lbf\text{-}s}{lbm}\right)$.

Hybrid mass fraction = 0.84.

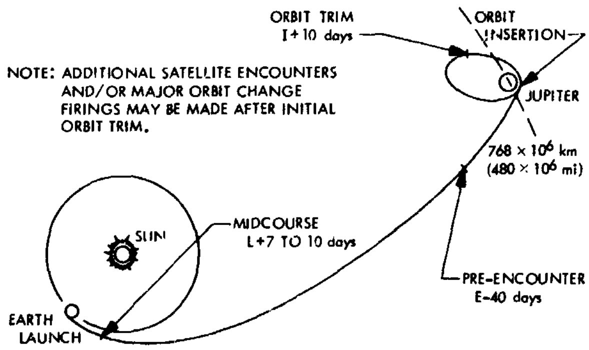

Fig. 1. Mission profile schematic

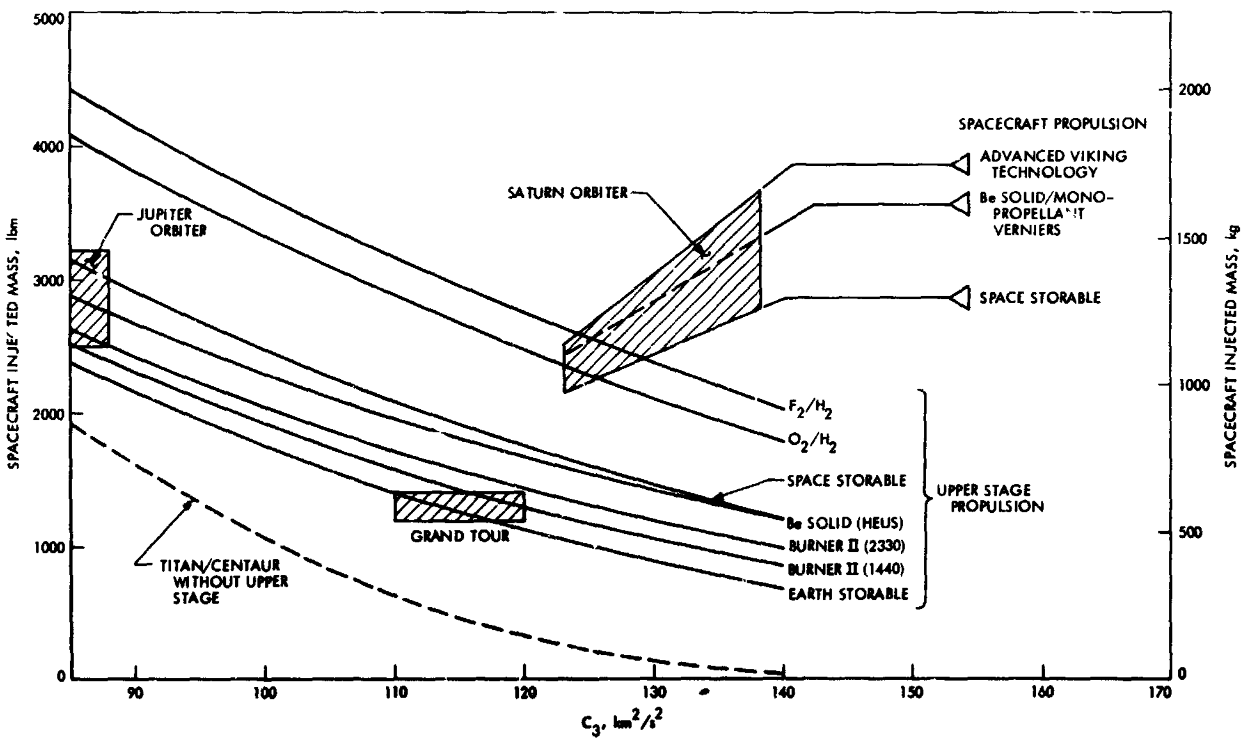

Fig. 2. Optimum upper stage study for Titan III (1205)/Centaur

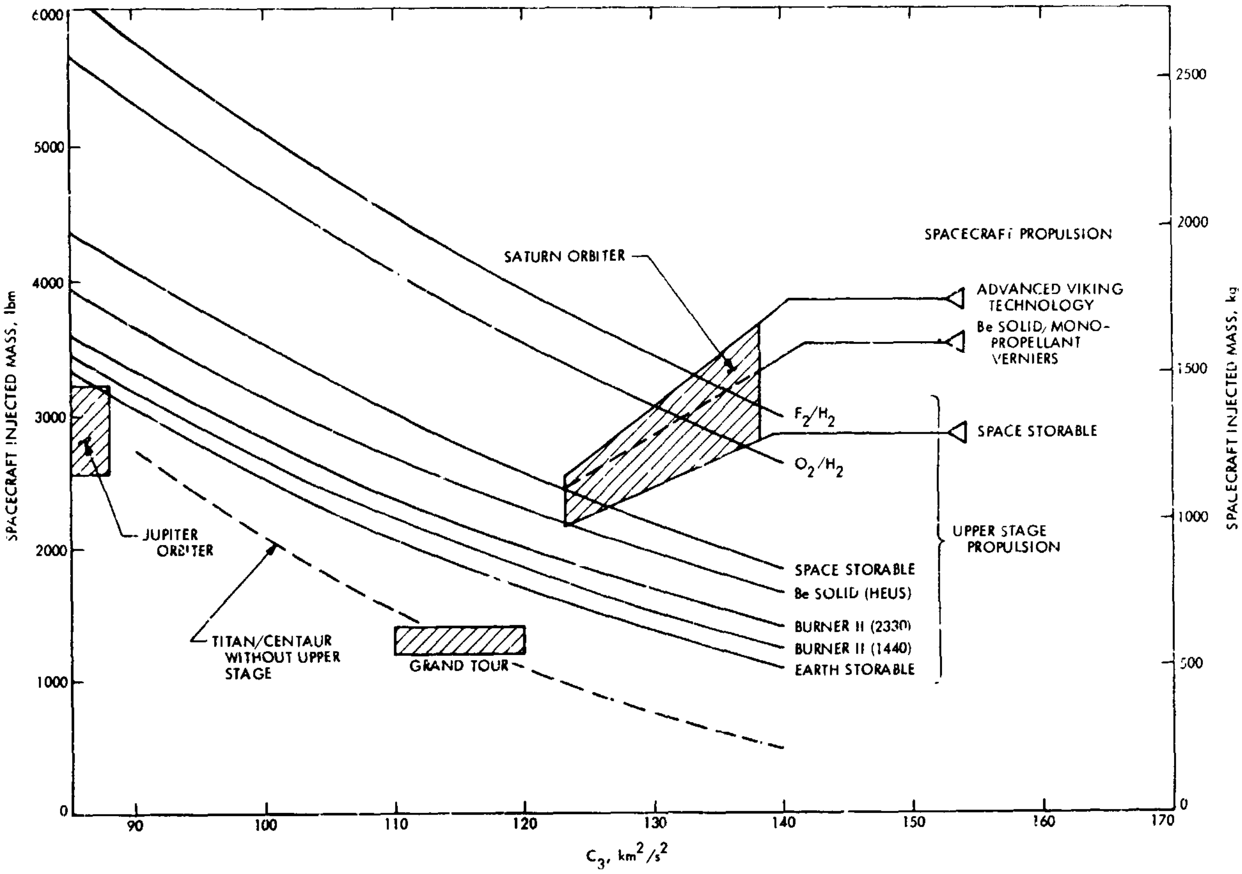

Fig. 3. Optimum upper stage study for Titan III (1207)/Centaur

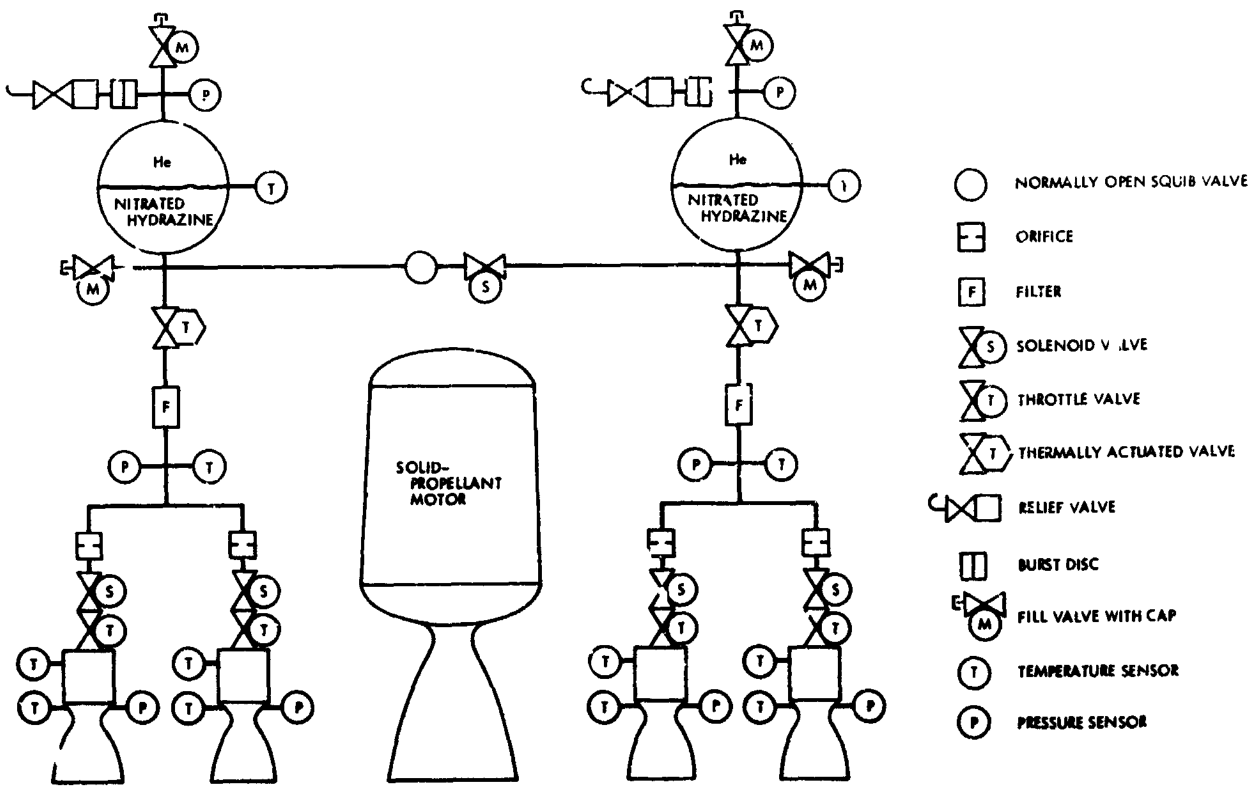

Fig. 4. Solid-propellant/monopropellant vernier alternative

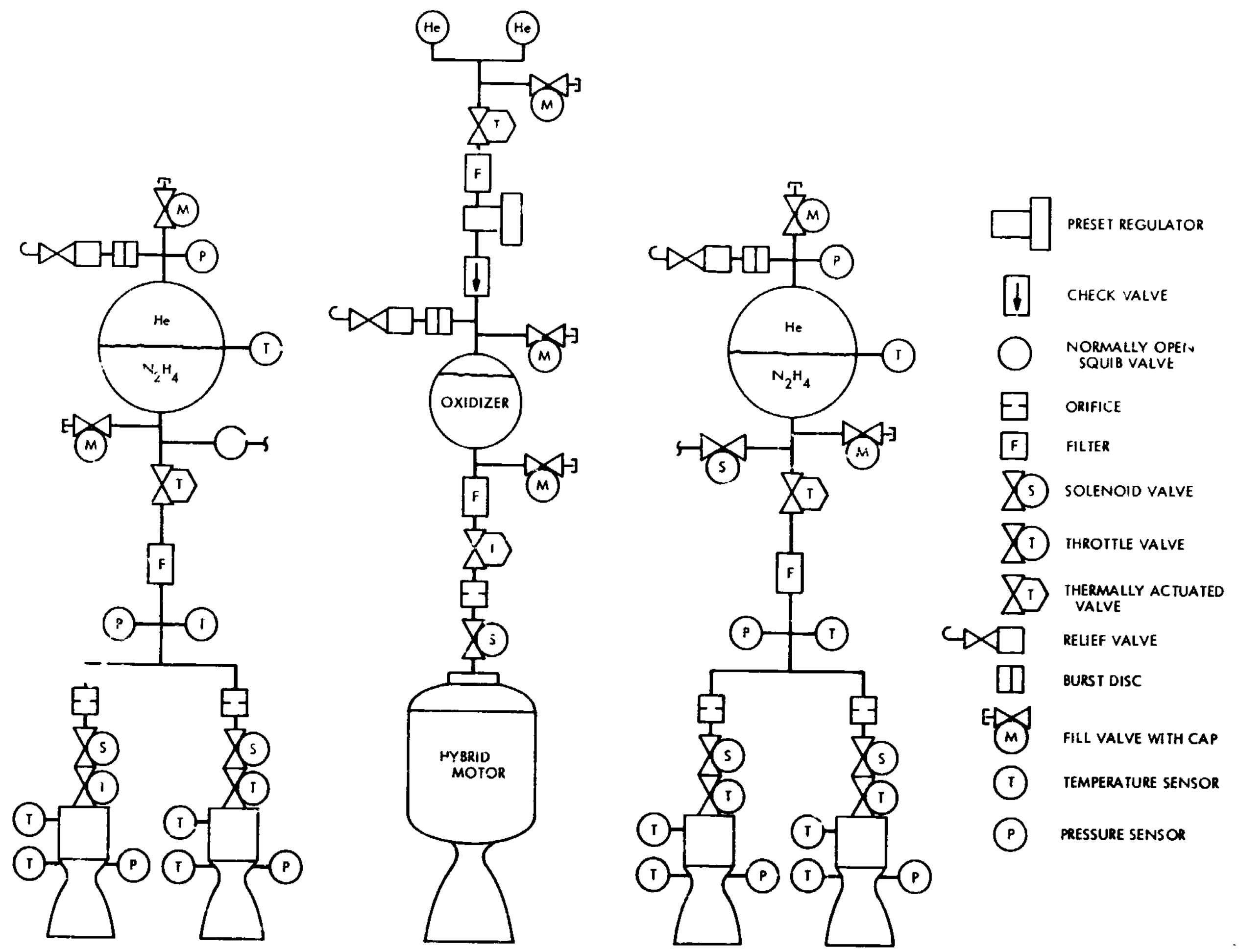

Fig. 5. Hybrid/monopropellant vernier alter.ıative

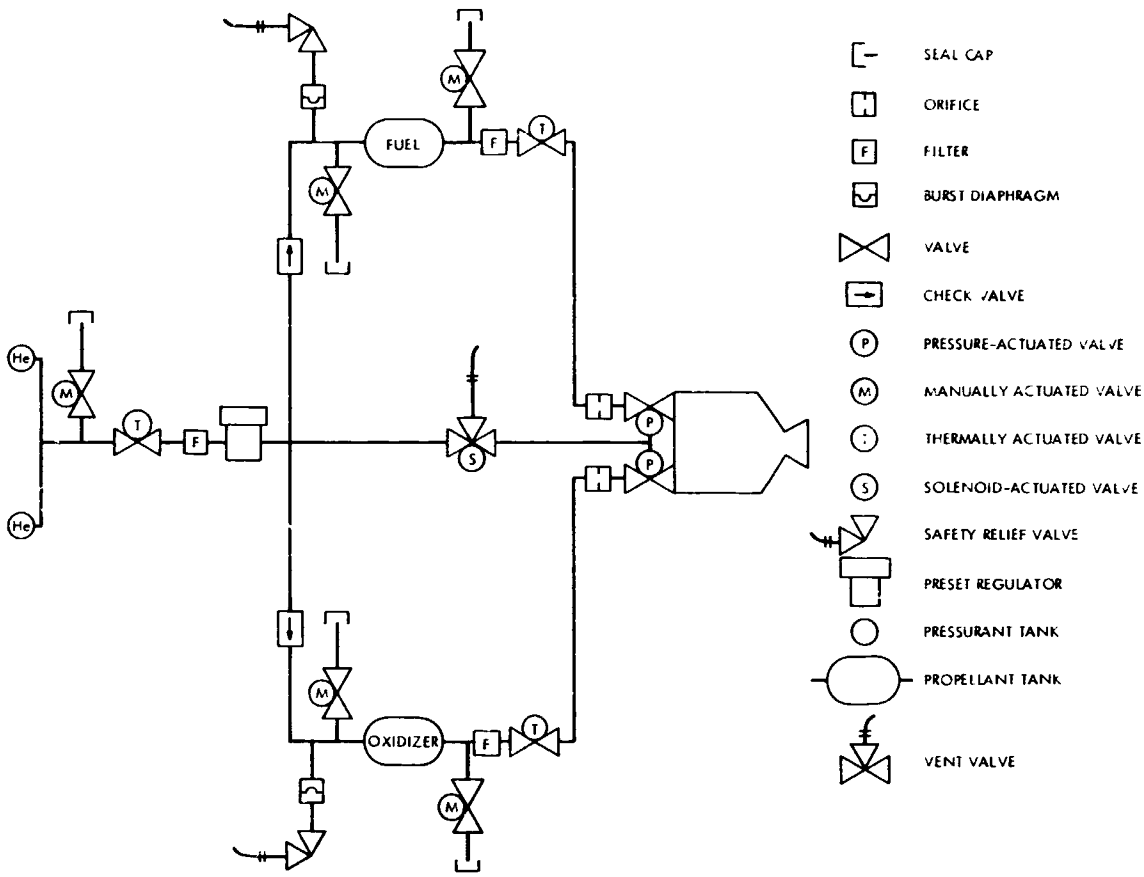

Fig. 6. Two-tank schematic for OF_2/B_2H_6 module

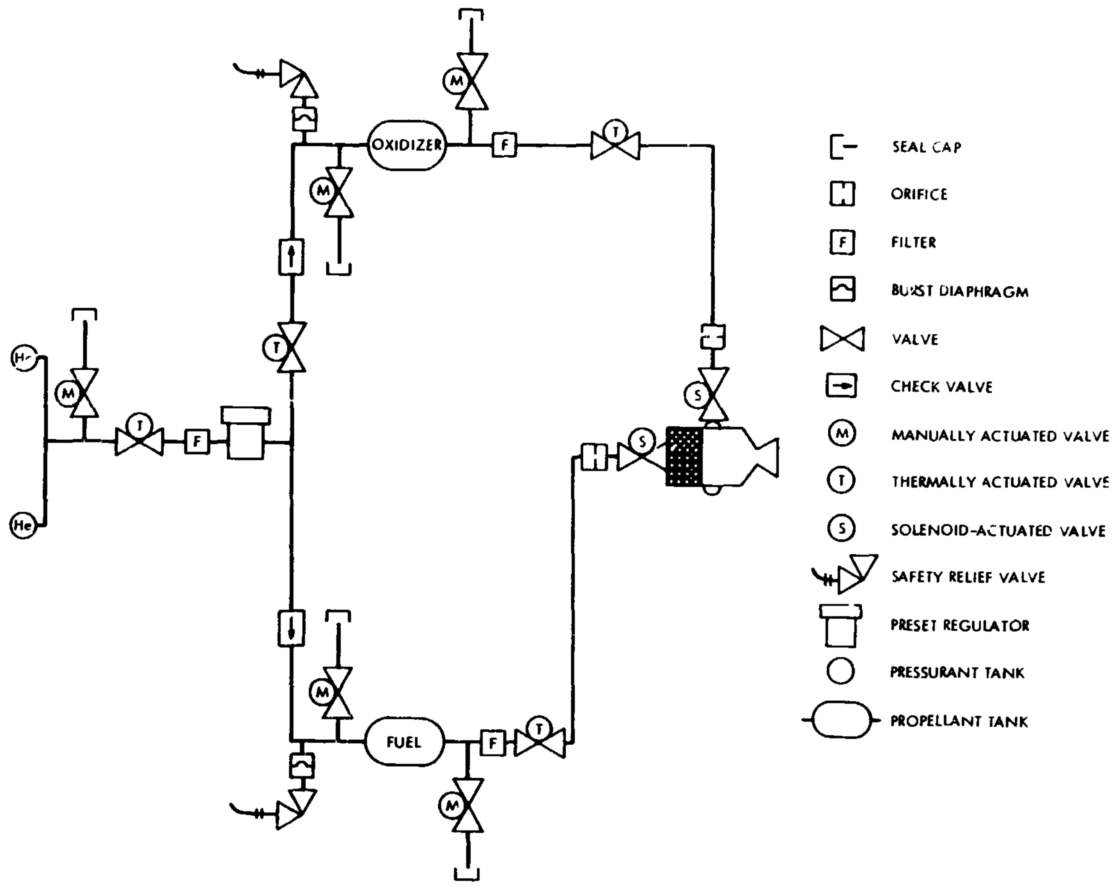

Fig. 7. Two-tank schematic for F_2/N_2H_4 dual-mode module

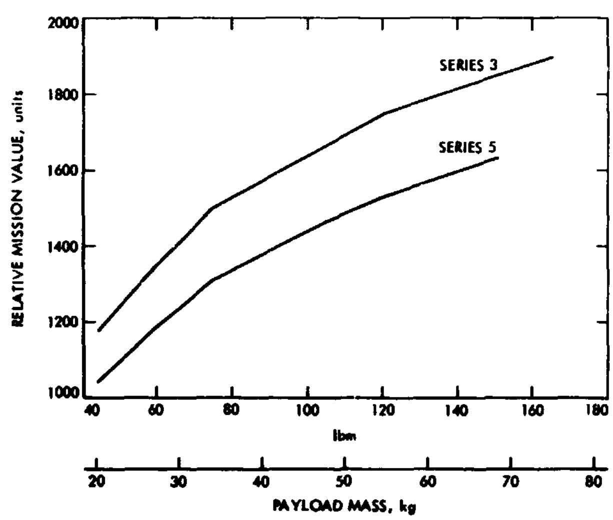

Fig. 8. TOPS mission value versus payload mass

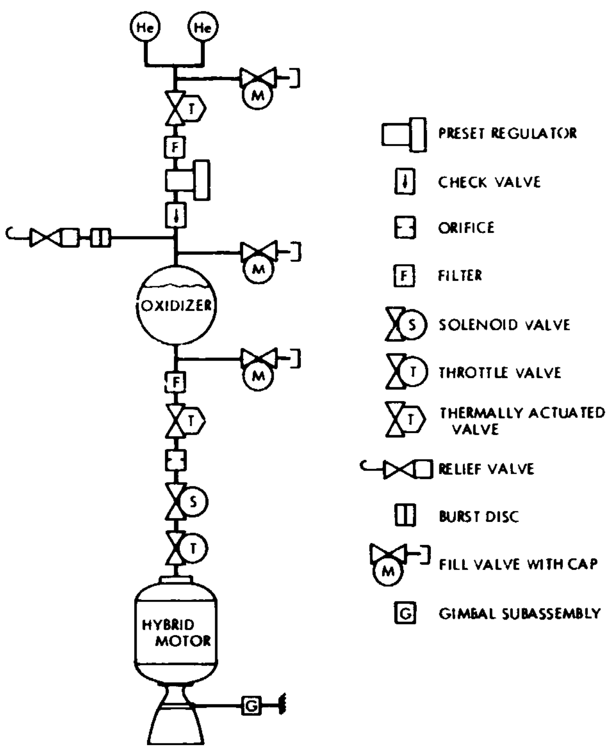

Fig. 9. Hybrid propulsion subsystem

APPENDIX A

TYPICAL VIKING ORBITER IMPROVEMENTS TO INCREASE PAYLOAD

Table A-1. Typical orbiter improvements that can increase payload (lander/capsule mass) (Ref. 12)

Orbiter design option[a]	Cost of increased capsule mass	
	$/kg	$/lbm
Substitute helium for nitrogen for propulsion pressurization system	2900	1300
Increase nozzle area ratio from 40:1 to 60:1	4400 to 6600	2000 to 3000
Utilize lightweight solar cell materials	6200	2800
Improve propulsion pressurant tank strength by heat-treating the titanium spheres	7300	3300
Utilize programmed pitch maneuver vs fixed pitch during orbit insertion	4400 to 6400	2000 to 2900
Increase nozzle area ratio from 60:1 to 80:1	5100 to 15200	2300 to 6900
Select only higher performance injectors	17200	7800
Fabricate propulsion pressurant tanks of boron filament	11000	5000
Use lightweight solar cell/structure array	14500	6600
Reduce cold gas system redundancy	27500 to 275000	12500 to 125000

[a] Ranked in the order of preferred implementation including cost, system impact, schedule risk, and mission risk.

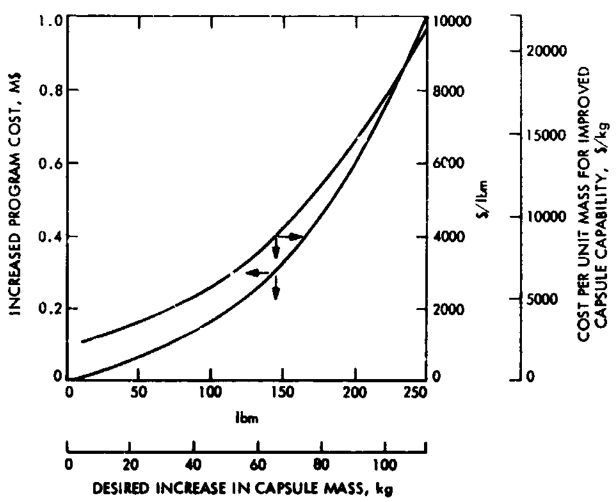

Fig. A-1. Costs of orbiter improvements to increase lander/capsule mass

APPENDIX B

SAMPLE COST-EFFECTIVENESS CALCULATION

APPENDIX B. SAMPLE COST-EFFECTIVENESS CALCULATION

Examine the dual-mode bipropellant design and consider the case for which mass is important and the subsystem has components with high reliability.

Mass

$$\Delta M = 557 - 554 = 3 \text{ kg}$$

$$\Delta CE = \left(0.277 \times 10^{-3} \frac{\text{units/M\$}}{\text{kg}}\right)(\Delta M)$$

$$= \left(0.277 \times 10^{-3} \frac{\text{units/M\$}}{\text{kg}}\right)(3 \text{ kg})$$

$$= 0.831 \times 10^{-3} \frac{\text{units}}{\text{M\$}}$$

$$\% \Delta CE = \frac{(0.831 \times 10^{-3} \text{ units/M\$})(100)}{322 \times 10^{-3} \text{ units/M\$}}$$

$$= +026\%$$

R&D Cost

$$\Delta C_{PD} = 21.25 - 18.5 = 2.75 \text{ M\$}$$

$$\Delta CE = \left(-0.805 \times 10^{-3} \frac{\text{units/M\$}}{\text{M\$}}\right)(\Delta C_{PD})$$

$$= \left(-0.805 \times 10^{-3} \frac{\text{units/M\$}}{\text{M\$}}\right)(2.75 \text{ M\$})$$

$$= -2.21 \times 10^{-3} \frac{\text{units}}{\text{M\$}}$$

$$\% \Delta CE = \frac{(-2.21 \times 10^{-3} \text{ units/M\$})(100)}{322 \times 10^{-3} \text{ units/M\$}} = -0.69\%$$

Hardware Cost

$$\Delta C_{PH} = 4.5 - 4.5 = 0 \text{ M\$}$$

$$\Delta CE = \left(-1.61 \times 10^{-3} \frac{\text{units/M\$}}{\text{M\$}}\right)(\Delta C_{PH}) = 0$$

$$\% \Delta CE = 0\%$$

Reliability

$$\Delta R_{PL} = 0.9937 - 0.9885 = 5.2 \times 10^{-3}$$

$$\Delta CE = \left(0.326 \frac{\text{units}}{\text{M\$}}\right)(\Delta R_{PL})$$

$$= \left(0.326 \frac{\text{units}}{\text{M\$}}\right)(5.2 \times 10^{-3})$$

$$= +1.70 \times 10^{-3} \frac{\text{units}}{\text{M\$}}$$

$$\% \Delta CE = \frac{(1.70 \times 10^{-3} \text{ units/M\$})(100)}{322 \times 10^{-3} \text{ units/M\$}} = +0.53\%$$

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