

A REVIEW OF IN SITU PROPELLANT PRODUCTION
TECHNIQUES FOR SOLAR SYSTEM EXPLORATION

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by

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FOREWORD

This study was conducted between June 1982 and March 1983 as part of the work performed by Science Applications, Inc. under Task 7 of Contract No. NASW-3622 for the Earth and Planetary Exploration Division, Code EL, NASA Headquarters. The level of effort expended on this study subtask was 480 man-hours. The results are intended to assist NASA planners in assessing the requirements and capabilities of in situ fuel production on remote bodies as a means of improving the performance of future exploration of the solar system.

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1. INTRODUCTION

As the exploration of the solar system evolves through investigative phases of increasing focus and detail, spacecraft of greater size and complexity will be required to accomplish the desired missions. In many cases these vehicles exceed the current ability of the Shuttle and any upper stage to place them on the desired trajectory. This has caused mission planners to resort to low thrust upper stages and planetary swingbys to deliver the spacecraft to its target, often at the expense of increased travel times. Many times even these techniques do not provide the needed mass relief for those missions which are designed to investigate the surfaces of other bodies in the solar system. The use of locally acquired materials to produce all or part of the propellant required for certain legs of these missions could reduce the mass of the vehicle at launch such that present or foreseen launch vehicles could be used. The phrases Extra Terrestrial Chemical Production (ETCP) and In Situ Propellant Production (ISPP) have been coined to describe these processes.

While the idea of producing propellants locally has been mentioned in the past in a general sense, the paper by Ash, Dowler and Varsi in 1978 (Ash, et al. 1978) was the first to identify specific equipment to accomplish the propellant production and apply it to a specific mission. In this case the mission identified was the Mars sample return mission which would use Martian water and carbon dioxide to produce methane and oxygen. This was found to reduce the vehicle's mass by several thousand kilograms when compared to what was then considered the best means of returning a surface sample. This latter method relied on a solid propellant, two stage ascent vehicle to deliver a small collected sample (less than 10 kg) to an orbiting Earth return vehicle (Moore and Scofield 1975).

The ISPP concept was found to provide similar mass relief at bodies other than Mars. In fact any body which had readily available materials in the form of gases, liquids or ices would be a candidate for such a mission. The body need not even be the primary target for the mission since propellants could be produced at one body for use by the vehicle to move to another target or could be transferred to another spacecraft not associated with the propellant-producing

vehicle. Details concerning these concepts as they apply to various missions and planets will be discussed in greater detail in the following sections.

In addition to its primary function of producing propellants for a spacecraft, the ISPP system can provide electrical, thermal and chemical energy for other purposes as well as performing certain other physical tasks. Studies have identified many potential benefits of this system, some of which can be performed using more than one type of energy. The electrical power produced by the system should be used whenever possible because the nominal power source typically has some excess power, is efficient, and can be easily enlarged, if necessary. Thermal power from the power subsystem and from the chemical processing subsystems is essentially free; that is, it is necessary to reject significant amounts of heat which must go to the surrounding environment if it is not used for some worthwhile purpose. Chemical energy in the form of the propellants already being produced can be used for purposes other than rocket-powered flight. This can only be accomplished, however, if there is sufficient throughput to provide an excess of propellants. Fuel and oxidizer to power a small rover or to drive a fuel cell for possible manned applications are but two examples of alternative applications for the propellants.

These promising benefits are not acquired without incurring liens in the form of additional direct costs, design conflicts and operational conflicts. Studies have identified a number of potential liens which are associated with the ISPP system. The new direct costs are associated with the development of new subsystems, namely the propellant production equipment and a propulsion system which uses the fuel and oxidizer that are produced. The design and operational conflicts arise from some unique characteristics of such a chemical processing plant; namely, that the landed system has some large, empty propellant tanks, that the system has significant environmental effects, and that in some forms, it must collect raw materials which require extensive surface operations and restricted landing sites. Another area of concern is that missions now become sensitive to surface stay times while the propellant is being produced. Surface stay time is important because it, and contingencies related to it, are used to determine the production capacity of the propellant producing system. Both benefits and liens will be discussed in greater detail in following sections.

The objective of this document is to present a compendium of some of the representative studies which have been done in the area of extraterrestrial chemical production as it applies to solar system exploration. Little or no previous familiarity with the subject has been assumed, but descriptions and discussions concerning individual topics will be kept to a minimum.

A description of the ISPP system will be presented first. Various propellant combinations and direct applications along with the previously mentioned benefits and liens will be discussed. Following this, a series of mission scenarios will be presented which, of all the suggested missions, have been studied in the greatest detail. A general description of the method(s) of analysis used to study each mission will be provided. Each section will be closed by an assessment of the performance advantage, if any, that can be provided by ISPP. A final section will briefly summarize those missions which, as a result of the studies completed thus far, should see a sizable benefit from the use of ISPP.

2. THE ISPP SYSTEM

The basic idea behind In Situ Propellant Production which makes it attractive is that it can reduce spacecraft mass by taking advantage of extended stay times to convert nuclear energy into chemical energy. Thus the use of two dissimilar energy sources is combined in such a way that the desirable qualities of one replace the undesirable qualities of the other making the system as a whole more useful. Nuclear energy sources have the advantage of a relatively high energy density when compared to chemical propellants. This allows the same amount of energy to be moved from one place to another without moving as much mass. The only limitation on this otherwise promising system is that extended periods of time are required to convert the nuclear energy to chemical energy. This is due to design considerations which allow the mass of the ISPP equipment to be reduced as the processing time is increased. There are, however, certain classes of missions which are either forced to accept, or would find highly desirable, an extended encounter on the surface of another body in the solar system.

The ISPP system can be divided into a number of subsystems which are typical of all missions identified for this concept thus far. These subsystems include the raw material gathering system, the chemical separation system, the power system, and finally the storage system. Variations in the details of each of these will occur depending upon the propellant combination used and the form of the raw materials being gathered.

Fuel and oxidizer combinations which have been investigated in the studies conducted to date have all used liquid oxygen as the oxidizer. Fuels that have been studied include hydrogen, methane, and carbon monoxide. Table 1 lists these combinations along with various properties which can be used to compare the usefulness of each.

TABLE 1. COMPARISON OF POTENTIAL ISPP DERIVED PROPELLANTS

<u>Propellant Type (Fuel/Oxidizer)</u>	<u>Isp (sec)</u>	<u>Typ. Mixture Ratio (Fuel:Oxidizer)</u>	<u>Critical Temperature Fuel</u>	<u>Oxidizer</u>
H ₂ /O ₂	426	1:6	33.3 K	154.8 K
CH ₄ /O ₂	342	1:3.4	191.1 K	154.8 K
CO/O ₂	259	1:0.5	133.2 K	154.8 K

Feedstocks from which either the fuel and/or oxidizer can be manufactured vary greatly depending upon the location within the solar system. Among the inner planets, gaseous carbon dioxide and water in various forms are the most readily available sources. The outer planets are for the most part limited to water ice found on satellites orbiting these bodies. Asteroids offer water as the potential feedstock in the form of solid hydrates or ice, depending upon the orbit of the body.

The remainder of this section will be devoted to brief descriptions of four types of ISPP systems which have been analyzed for specific applications but could be applied to a wide range of missions. The critical steps involved for propellant production in each system will be noted as well as the locations in the solar system where each can be used.

The first system to be discussed, and the one on which the most analysis has been performed, produces only oxygen. The fuel to be used by this system must be brought with the ISPP system from Earth. The basic assumption made for applying this system is that a carbon dioxide atmosphere is present. This limits the system to use at either Venus or Mars.

The critical steps involved in the oxygen production are illustrated in Figure 1. Carbon dioxide is drawn into the system and heated to a temperature at which the gas dissociates into oxygen and carbon monoxide. The oxygen is separated from the carbon monoxide with the use of a solid electrolyte (Richter 1981). The oxygen is then cooled to cryogenic temperatures and stored as a liquid. A refrigeration unit will be required to maintain the oxygen at this temperature. The entire system is powered by an RTG unit which provides both thermal and electrical energy.

Collection of the carbon dioxide will depend upon the planet at which the system is operating. At Venus the ambient pressures are high enough to allow a simple vacuum system to draw in the gas. The atmospheric pressures at Mars are too low to allow a vacuum intake to be used without devoting an excessive amount of mass to this subsystem. An alternative method would use the cryogenic propellants as a working fluid to sublimate the carbon dioxide into "frost" which would then be collected and processed.

The second type of system which would produce both oxygen and methane, is described in detail in the paper by Ash, Dowler and Varsi (Ash, et al. 1978). Such a system requires carbon dioxide in the atmosphere and a source of water. This situation can only be found at Mars.

The first step in the propellant production sequence is to gather water and carbon dioxide from the local surroundings. The water is separated into hydrogen and oxygen by the familiar method of electrolysis. The oxygen is cooled to a liquid and stored. The hydrogen, however, is combined with the carbon dioxide to produce methane and water. This process can be carried out in a straightforward manner using a nickel based catalyst (Lalancetta 1975 and Seglin 1975). The water is returned to the electrolysis unit while the methane is cooled and stored in liquid form. This entire process is shown in Figure 2.

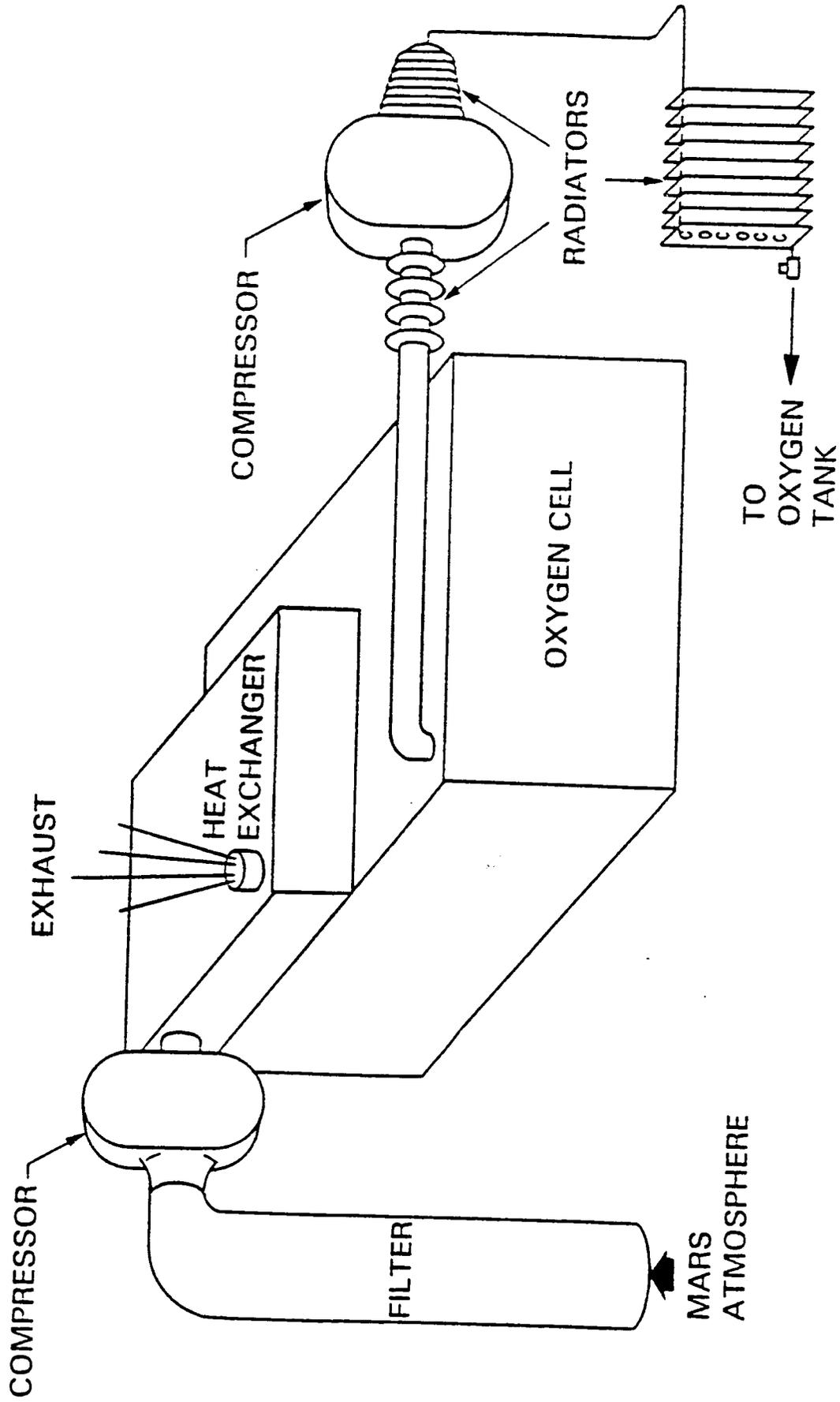


FIGURE 1. POSSIBLE CONFIGURATION ISPP OXYGEN ONLY PRODUCTION SYSTEM (HANSON 1982)

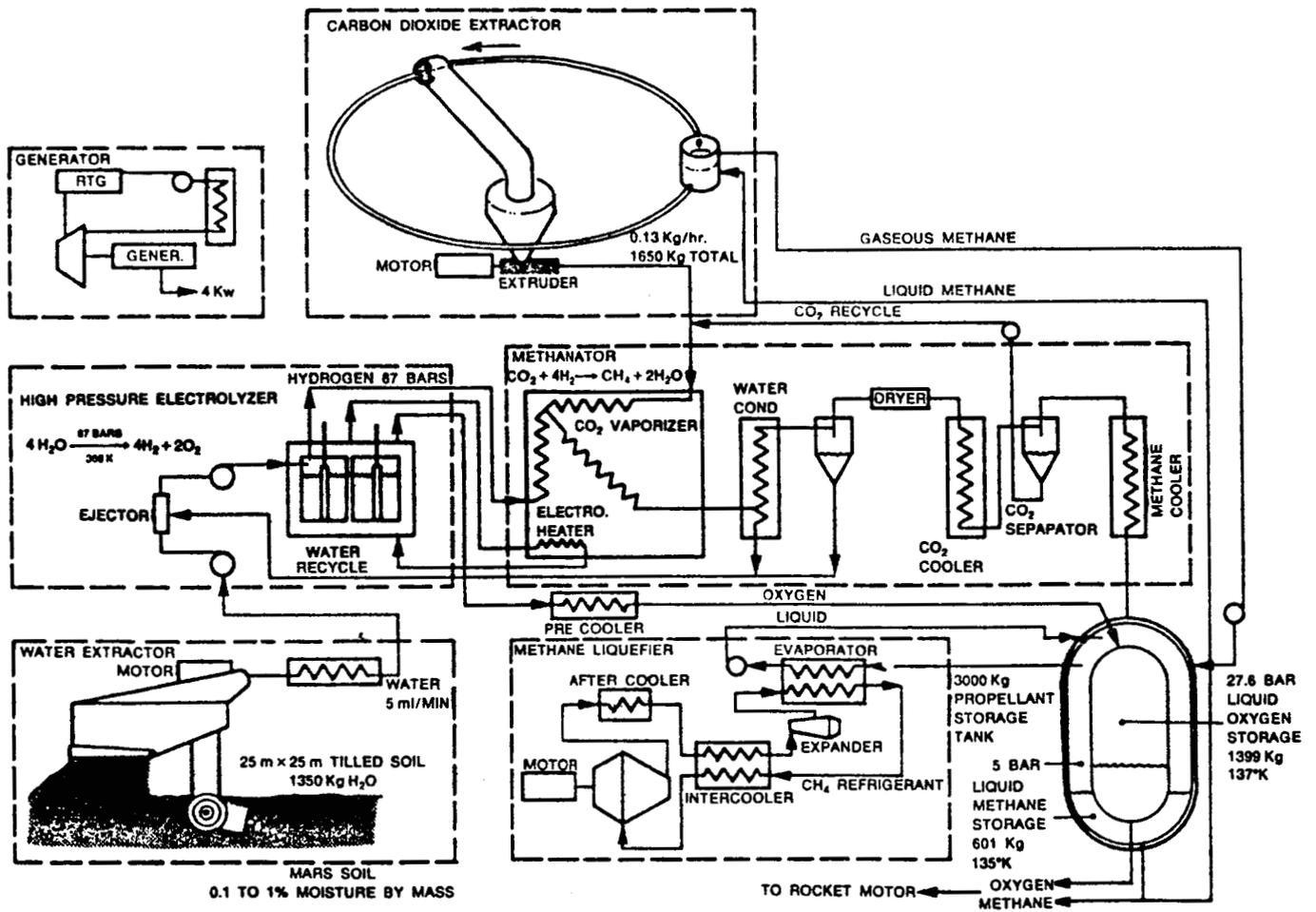


FIGURE 2. SCHEMATIC OF POSSIBLE METHANE/LOX ISPP SYSTEM (ASH, et al 1978)

Since this system is only applicable at Mars, the sublimation method of collecting carbon dioxide as mentioned above would be used. The collection method for the water would depend upon the local conditions. At the poles, water ice should be available directly from the surface and could be obtained by drilling or scraping. At lower latitudes, water is assumed to be trapped as subsurface permafrost. In this case, both soil and water would be gathered for processing. The water could then be separated out by heating the soil until the water evaporates and then condensing the vapor for delivery to the electrolysis unit.

The third system would also produce both fuel and oxidizer from local materials. In this case the fuel is carbon monoxide which would be burned with oxygen. The system would be the same as the oxygen-only system with the exception that the carbon monoxide is now stored rather than vented as waste. Again the system would only find the necessary raw materials at Venus or Mars. This option has not received as much attention as any of the other three due to the fact that the carbon monoxide/oxygen combination has relatively poor rocket performance ($I_{sp} < 300$ sec).

The final system would produce liquid hydrogen and oxygen from water. The raw material for this system could be in the form of liquid water, water ice or solid hydrates. Water in one or more of these forms can be found at Mars, some of the asteroids, and on the satellites of the outer planets. While Mars is mentioned here for completeness, the relatively high ambient temperatures found in the Martian atmosphere would require a rather large refrigeration unit. The mass penalty involved with such a unit has tended to limit the use of this system to the outer planet region where ambient temperatures are much lower (Ash et al. 1980).

The procedure used to manufacture this propellant combination is the same as that used on Earth; namely, splitting water into hydrogen and oxygen through the use of electrolysis. These gases are then dried of excess water vapor, cooled to liquid form, and stored. Depending upon the local temperature conditions, refrigeration may be required to maintain one or both of these liquids in this state. A diagram of a proposed system is shown in Figure 3 with system requirements listed in Table 2.

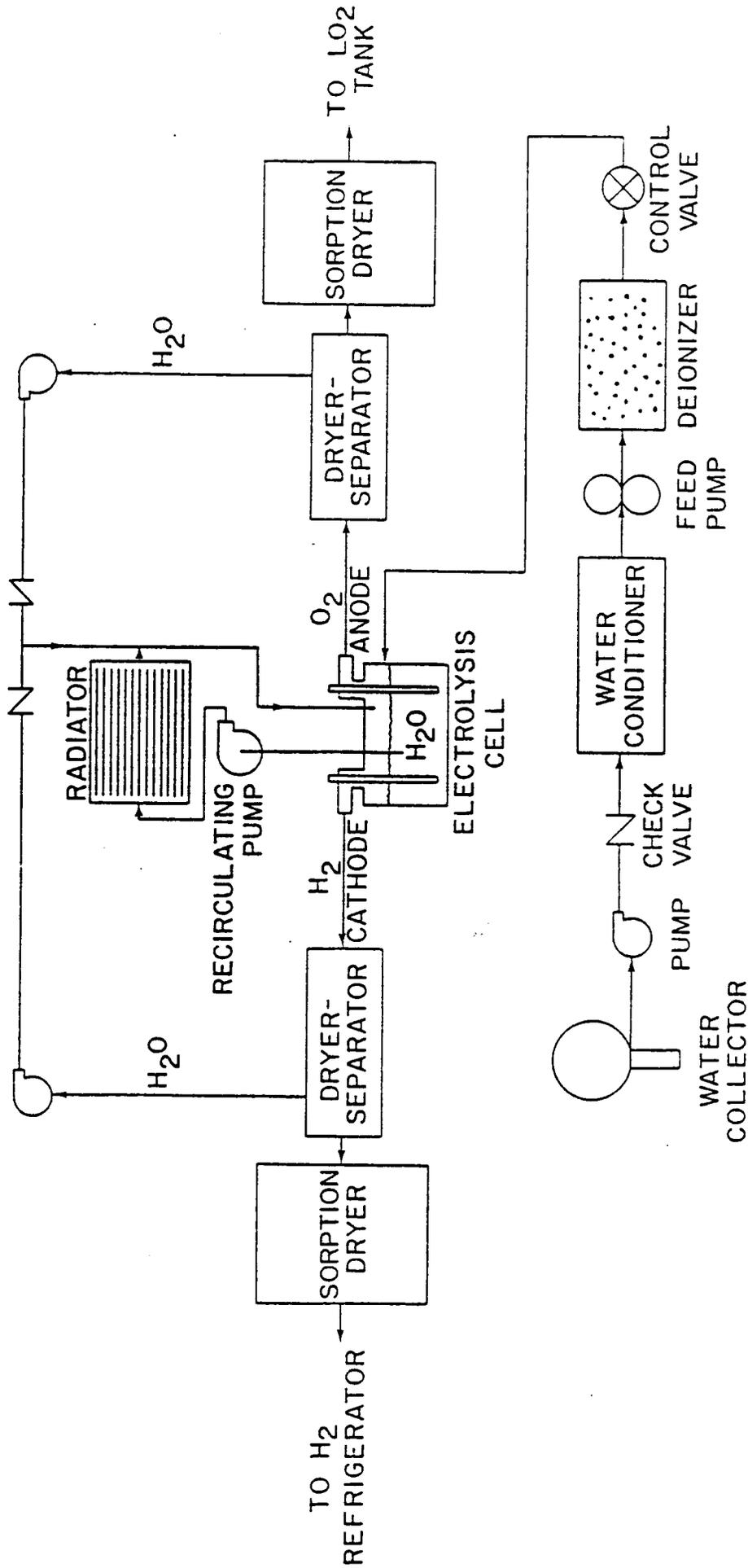


FIGURE 3. POTENTIAL CONFIGURATION FOR HYDROGEN/LOX ISPP SYSTEM

TABLE 2. MASS AND POWER PERFORMANCE OF H₂/O₂ ISPP SYSTEM

<u>PROCESSOR ELEMENTS</u>	(Flow Rates: 2.9 to 4.8 kg of H ₂ O/day)		
	<u>Mass</u>	<u>Power</u>	<u>Volume</u>
Water collection	14-21 kg	---	10 to 15 ℓ
Pumps and valves	5 kg	10-20 W	1 ℓ
Water treatment	5 kg	10 W	5 ℓ
Electrolysis Cell	10-15 kg	440-700 W	7-12 ℓ
Dryer	1-2 kg	10 W	1-2 ℓ
Radiator	1-2 kg	---	(1-2 m ²)
Piping and Structure	4-5 kg	---	1-2 ℓ
	<u>40-55 kg</u>	<u>450-740 W</u>	<u>27-37 ℓ</u>
Hydrogen Refrigerator	35-50 kg	210-400 W	100-250 ℓ
<u>POWER REQUIREMENTS</u>	<u>CH₄ + (O₂)</u>	<u>+(H₂/O₂)</u>	
J _S SR*	450 W	660 W	
J _S ² SR**	740 W	1200 W	
<u>SYSTEM MASS</u>			
J _S SR	110 kg	180 kg	
J _S ² SR	190 kg	300 kg	

* SINGLE GALILEAN SATELLITE SAMPLE RETURN

** DUAL GALILEAN SATELLITE SAMPLE RETURN

Possible collection schemes for ice and permafrost have already been discussed. This leaves the hydrate form which must be either drilled or scraped as was mentioned for ices. These minerals must then be chemically cracked to release the water.

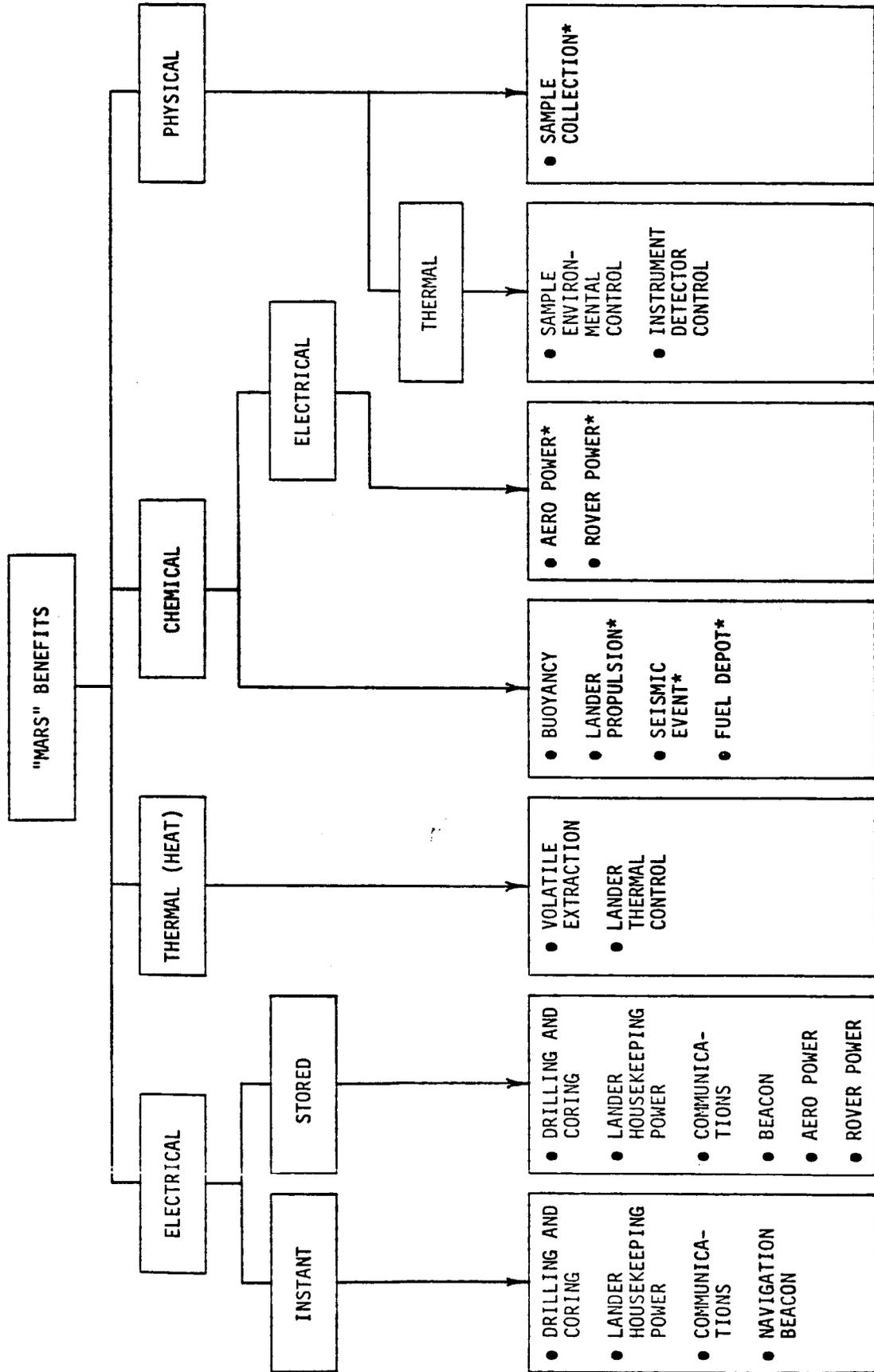
In addition to its primary function of producing propellants, the ISPP system can provide electrical, thermal and chemical energy for other purposes as well as doing certain other tasks. Figure 4 identifies 15 potential benefits of the

ISPP system in its various forms, some of which can be performed using more than one type of energy. Each is described in Table 3. The electrical power produced by the system should be used whenever possible because the nominal power source, an RTG, typically has some excess power, is efficient, and can be enlarged easily, if necessary. Thermal energy from the power system and from the chemical processing systems is essentially free; i.e., it is necessary to reject significant amounts of heat which must go to the surrounding environment if it is not used for some worthwhile purpose.

When an ISPP system is used, there are additional direct costs, design conflicts and operational conflicts which are incurred. Another area of concern is that missions become sensitive to surface stay time when propellants are produced on the surface. Eleven potential liens of this type are identified in Figure 5 and described in Table 4. The design and operational conflicts arise from some unique characteristics of an ISPP system, namely; the landed system has some large, empty propellant tanks, the ISPP system can have significant environmental effects, and, in some of its variants, the system must collect raw materials in a form that requires extensive surface operations and restricted landing sites. Finally, surface stay time is important because it and contingencies related to it are used to determine the production capacity of the ISPP system.

The topics discussed in this section have shown the ISPP concept to be relatively versatile in the types of propellants which can be produced and the sites where suitable raw materials can be found. Applications in both the inner and outer solar system are thus considered feasible. Of the various options discussed, the system which produces just O_2 is currently the only one which is under advanced study and then only for application at Mars. It thus represents the only system which would be available in the near term. The other options, while benefitting from the work done on the O_2 only system, will nonetheless require more development effort.

The following sections will discuss specific missions for which analysis has shown that the use of ISPP can increase the payload performance.



* SECOND GENERATION "MARS" ONLY (i.e. HYDROGEN/LOX PRODUCTION)

"MARS" = MARS AUTOMATED REFUELING STATION

FIGURE 4. BENEFITS DERIVED FROM USING THE ISPP SYSTEM

TABLE 3

SUMMARY OF "MARS" SYSTEM SPINOFF BENEFITS

Benefit	Category(s)	Subcategory(s)	Description
Drilling/Coring	Electrical	Instant, Stored	Collection of subsurface samples is accomplished by drilling/coring. Power demands are high but brief and could be met by an idle MARS or by surplus MARS power accumulated in a battery.
Housekeeping Power	Electrical	Instant, Stored	Housekeeping power demand for the lander is modest and relatively constant. It can be met with surplus MARS power supplemented by batteries when no MARS power is available.
Communications	Electrical	Instant, Stored	Communications power requirements are modest and intermittent. Surplus MARS power could be used directly or via batteries.
Navigation Beacon	Electrical	Instant, Stored	A navigation beacon would be used by a rover or an airplane. The low and steady power demand can be provided by surplus MARS power supplemented by batteries.
Airplane Power	Electrical or Chemical*	Stored Electrical	An airplane could be used for deployment of penetrators to make a network of landed science instruments for a detailed characterization of the geological unit around MARS or for a study of the vertical structure of the atmosphere. A propeller airplane driven by an electric motor can be designed for operation in the thin martian atmosphere. The source of electrical energy could be either a battery charged using surplus MARS power or a fuel cell powered by excess MARS CH ₄ and O ₂ . Internal or external combustion engines are thought to be less effective for this application.

*MARS" = MARS AUTOMATED REFUELING STATION

TABLE 3 (cont'd.)

SUMMARY OF "MARS" SYSTEM SPINOFF BENEFITS

Benefit	Category(s)	Subcategory(s)	Description
Rover Power	Electrical or Chemical*	Stored Electrical	A rover can be used for surface science investigations including deployment of small science stations and utilization of imagery and geophysical methods on geological traverses. The electric motors that are usually assumed to provide the rover's traction force could be powered by a battery charged using surplus MARS electrical energy. Rover power could also be derived from a fuel cell using excess CH ₄ and O ₂ .
Volatile Extraction	Thermal		The extraction of volatiles from a surface sample is a necessary part of some science experiments (e.g., rare gases used for age dating and biochemically important molecules). Heat for the extraction process could be provided from the thermal waste of the radioisotope power source at temperatures up to 500°K. Higher temperatures are possible but would only be available when electrical power demand is below rated capacity.
Lander Thermal Control	Thermal		Thermal control of the lander can make use of the waste heat from either the radioisotope power source or from condensation and other process steps. When CH ₄ /O ₂ are produced, 300 watts of heat from a water condensation step is available at 285°K (12°C or 54°F), a suitable temperature for lander systems.
Buoyancy	Chemical		A tethered balloon could be used by the lander to extend its horizon, thereby obtaining better characterization of the site via imagery. The balloon could be filled with any gas lighter than CO ₂ , especially the H ₂ produced for the CH ₄ /O ₂ process. Each day about 1.0 kg of H ₂ is produced which can lift 20 kg of gross weight (payload + balloon). If compressed to about 70 bar, the H ₂ can be returned to the process.

TABLE 3 (cont'd.)

SUMMARY OF "MARS" SYSTEM SPINOFF BENEFITS

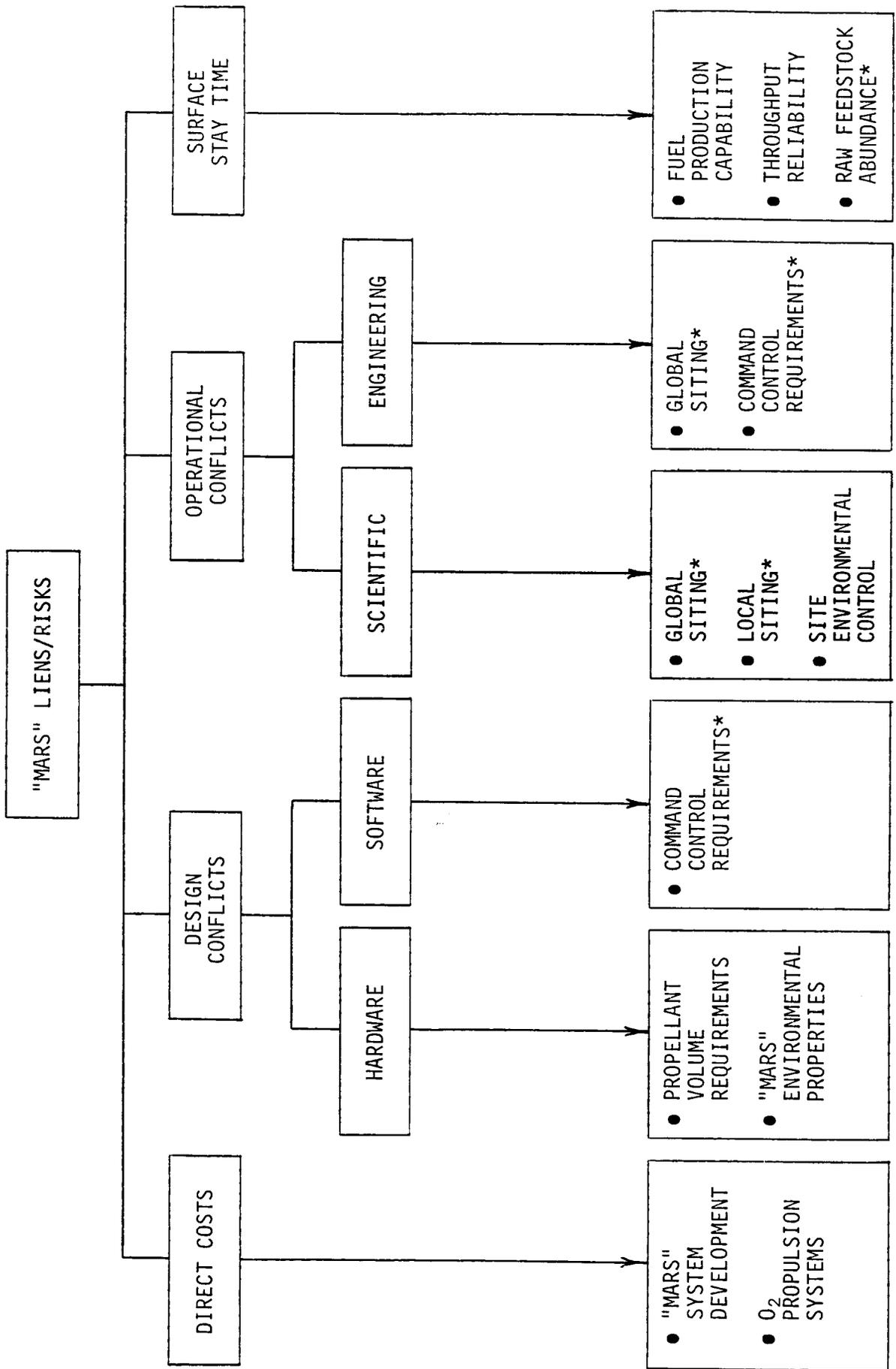
Benefit	Category(s)	Subcategory(s)	Description
Lander Propulsion	Chemical*		Propellants produced by MARS after its primary mission is over could be used to relocate the lander using rocket motors, perhaps the same units used for terminal descent. Range is likely to be measured in tens of kilometers, at most, and a considerable amount of fuel must be produced before each translation maneuver.
Seismic Event	Chemical*		An active seismic experiment needs a source of seismic waves. Small explosions of CH ₄ and O ₂ gases could be used although normally the explosive used for this purpose is a solid which can be placed in a borehole for better coupling of the explosion to the surface.
Fuel Depot	Chemical*		After its primary mission, MARS could continue to produce propellants to be placed in a surface fuel depot. Subsequent missions could use this fuel. This requires that these missions be able to transport either the fuel to a desirable location where it will be used or the system needing the fuel to the site of the MARS. This means that a single MARS could provide the rocket fuel for several returned samples if new samples are brought to it by long range rovers and if empty rockets are also provided.
Sample Environmental Control	Physical	Thermal	There is a desire to keep the chosen return sample at a cold temperature, 243°K (-30°C) has been suggested. The MARS system has liquids stored at 125°K so that using them as a refrigerant, a temperature like 243°K is easily maintained.

TABLE 3 (cont'd.)

SUMMARY OF "MARS" SYSTEM SPINOFF BENEFITS

Benefit	Category(s)	Subcategory(s)	Description
Instrument Detector Control	Physical	Thermal	Detectors of infrared, microwave and nuclear radiation often have improved performance if operated at low temperatures. These detectors could be cooled to about 125°K by using liquid methane as the refrigerant. One application that has special merit is the use of high energy resolution x- and gamma-ray detectors for experiments sensitive to surface composition. The best detectors for this are made of lithium-drifted germanium which must be cooled at all times. These detectors would continue to use MARS cooling if sent to the surface of Mars and could probably use radiative cooling if placed in orbit around Mars.
Sample Collection	Physical*		Small rovers bring surface material containing water for the MARS to process into CH ₄ and O ₂ . These rovers could also be used to bring samples for the return sample. These small rovers have ample payload (about 5 kg) and a nominal range of 25 m. However, they have no instrumentation which could document the sample before it is collected.

*SECOND GENERATION "MARS" ONLY (HYDROGEN/LOX)



*SECOND GENERATION "MARS" ONLY (HYDROGEN/LOX)

"MARS" - MARS AUTOMATED REFUELING STATION

FIGURE 5. LIENS AND RISKS RESULTING FROM ISPP USAGE

TABLE 4

SUMMARY OF "MARS" SYSTEM MISSION LIENS

Liens	Category(s)	Subcategory(s)	Description
"MARS" System Development	Direct Costs		In addition to a performance advantage over conventional methods, MARS may have a cost advantage, if the development cost is not too high. The break even development cost is about \$100 million (this is discussed later).
O ₂ Propulsion System Development	Direct Costs		Along with the "MARS" system, it is necessary to develop new rocket propulsion systems for this mission which use cryogenic liquids, particularly oxygen. A conventional system would use either solids or room temperature liquids as propellants. They must also be sterilizable, but several such propellants are available using already developed technology.
Propellant Volume Requirements	Design Conflict	Hardware	As currently envisioned, MARS has pressurized tanks for storing propellant. In addition there is a multi-stage rocket, also with empty tanks. Both must be delivered to the surface, so while the MARS concept may reduce the mass of landed systems, it may have a larger volume. This may contribute to packaging problems, particularly in the aeroshell.
"MARS" Environmental Properties	Design Conflict	Hardware	The MARS contains a 3.0 kw (electrical) radioisotope power source. Such a power source will produce much more waste heat and more nuclear radiation than the power source needed for a conventional system. It is essential that the thermal and radiation environments of both the lander systems and the collected surface sample be kept within reasonable bounds. Again, packaging in the aeroshell is the major problem because the distance between the radiation and heat and the sensitive electronic systems cannot be as large as it can be during other mission phases.

"MARS" = MARS AUTOMATED REFUELING STATION

TABLE 4 (cont'd.)

SUMMARY OF "MARS" SYSTEM MISSION LIENS

Liens	Category(s)	Subcategory(s)	Description
Command, Control Requirements	Design Conflict* or Operational Conflict*	Software Engineering	As presently conceived, the collection of water bearing soil is both a design conflict and an operational conflict. One possible approach is to provide a high level of artificial intelligence for the soil collecting rovers so that they can carry out their work with little intervention from mission controllers. This requires sophisticated computer software as well as advanced computer hardware and sensors. The other extreme is to allow mission controllers to make most of the necessary decisions. This puts great strains on mission operations and on the communications systems which must bring sensor data to the Earth and many commands up to Mars. The final design will probably be a blend of these two approaches so that both types of conflict will be present, but minimized.
Global Siting	Operational Conflicts*	Scientific, Engineering	The site selected for the second generation MARS must have abundant subsurface water that is expected at higher latitudes including the polar caps. This requirement restricts the choice of landing sites and excludes many areas of high scientific interest. The site may have operational conflicts of an engineering nature such as poor solar illumination, difficult communications with the Earth, or entry conditions which are not optimum.
Local Siting	Operational Conflicts*	Scientific	To assure success of the second generation MARS, the water bearing soil must be easy to collect and process. This may cause selection of a site which is not preferred for best scientific results.

TABLE 4 (cont'd.)

SUMMARY OF "MARS" SYSTEM MISSION LIENS

Liens	Category(s)	Subcategory(s)	Description
Site Environmental Control	Operational Conflict	Scientific	If the science sample is collected first, it must be maintained at an appropriate temperature and in an uncontaminated environment throughout the time while the MARS is producing fuel, waste heat and nuclear radiation. A sample collected after fuel has been produced may be altered by the changes in the local environment due to the MARS.
Fuel Production Capability	Surface Stay Time		The design of the MARS must provide sufficient propellants within the surface stay time allowed for a specific launch opportunity. In some launch years, the stay time is shorter than in others so that is necessary to have a larger MARS with more production capability. The stay times in the 1990, 1992 and 1994 opportunities are among the shortest known. The Mars departure energy (and thus total fuel) requirement in 1990 is above average, so that short stay times are not compensated by low fuel needs.
Throughput Reliability	Surface Stay Time		The MARS must produce propellants at a steady rate to meet its goal. The rate must be maintained despite changes in atmospheric conditions (including dust content), soil temperatures, etc. If any of these variable conditions makes the rate uncertain, then an increase in the size of the MARS is needed to provide a production contingency.

TABLE 4 (cont'd.)

SUMMARY OF "MARS" SYSTEM MISSION LIENS

Liens	Category(s)	Subcategory(s)	Description
Raw Feedstock Abundance	Surface Stay Time*		It is necessary to have a source of subsurface water if methane and oxygen are to be produced by the second generation system. Water is a known atmospheric constituent and is almost certainly very abundant in the ice caps. However, it is not clear that it is present in other areas in the required abundance (2% by weight). It appears that additional mission data, particularly orbital γ-ray spectroscopy, is needed to determine what sites, if any, are suitable.

*SECOND GENERATION "MARS" ONLY (HYDROGEN/LOX)

3. MARS MISSIONS

The mission options thus far considered for Mars which would utilize ISPP technology can be divided into two general categories. A Mars sample return was the first of these to be recognized as benefitting from ISPP and is the option on which the most analysis has been focused. The second category covers those missions which would use the locally produced propellants to conduct extended explorations of the surface of the planet.

The use of ISPP for the Mars Sample Return (MSR) had its origins in the paper by Ash, Dowler and Varsi (Ash, et al. 1978). The paper basically outlined a method by which propellants could be produced at Mars and showed that the technology was available to implement this method. A report by Stancati, Niehoff and Wells (Stancati, et al. 1978) evaluated various combinations of flight options including the then baseline case of Mars Orbit Rendezvous (MOR) as well as Direct Entry (DE) and Direct Return (DR). Using information generated by Ash, et al. and the ongoing Mars Program at JPL, this study was able to show that the DE/DR option provided sizable mass margins when the Shuttle/IUS was considered as the launch vehicle. These study results have since been further refined at JPL (Hanson 1982) resulting in a scenario which uses the DE/DR concept for the landing and return phases at Mars.

The most frequently mentioned use of ISPP after the sample return mission is the concept of conducting extensive surface exploration by refueling an excursion vehicle. Both short (i.e. less than a kilometer) and long (i.e. global) range missions have been proposed for study. To date a preliminary analysis of only the long range missions has been completed (Hoffman et al. 1982). This study considered two types of vehicles: a version of the previously designed Mars airplane modified to allow refueling and a ballistic "hopper" which uses rocket engines to fly a ballistic trajectory from one point on the surface to another.

Mars Sample Return

The arrival and departure scenarios for the MSR mission have to date tended to focus on the MOR and DE/DR options. Depending upon the assumptions made, these

are the most mass efficient options of four arrival and departure combinations which are illustrated in Figure 6.

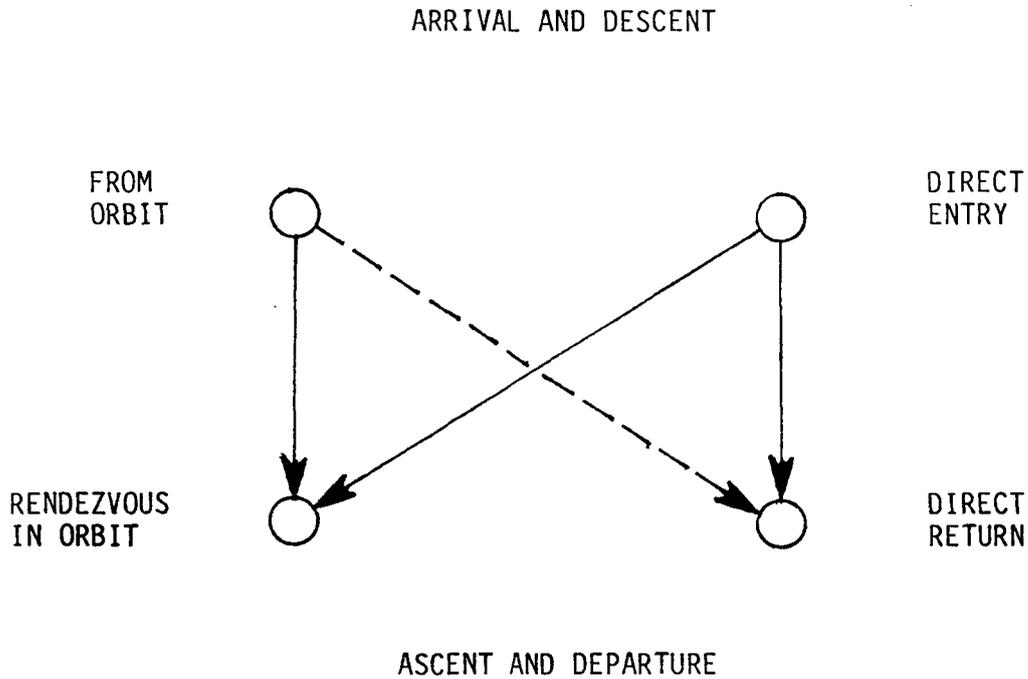


FIGURE 6

The study carried out by Stancati, Niehoff and Wells (Stancati, et al. 1978) sought to determine whether the use of ISPP with either the MOR or DE/DR option could improve the mass performance of the mission. To this end a comparison was made of the injected mass requirement for a vehicle using conventional space-storable propellants with that of a similar vehicle which used ISPP-produced propellants. In this study, the ISPP system was assumed to produce only oxygen to be used with methane or to produce both oxygen and methane. These two system types are identified as either first or second generation systems respectively. Commonality between the two vehicles was obtained by assuming that all subsystems not directly affected by the type of propulsion system would be the same for both. The subsystem masses were taken from the ongoing JPL Mars Program. While this did not allow either vehicle to be optimized to take full advantage of either propulsion system, it did remove all variations in the injected mass except those caused by the type of propulsion used.

This comparison was carried out for two different launch opportunities. These dates were selected to be representative of what could be considered a worst case (1990) and a best case (1994) in the launch opportunity cycle.

The MOR mission option was the previous baseline design for the MSR mission because it resulted in the lowest total injected mass when only conventional (i.e. non-ISPP) propellants were considered. In this scenario two vehicles are required to complete the mission. The first is the Mars Ascent Vehicle (MAV) which lands on the surface to collect the sample and then launch it into a low altitude parking orbit. At this point, an autonomous rendezvous with the second vehicle, the Earth Return Vehicle (ERV), is made and the sample container is transferred from the MAV to the ERV. The ERV then returns the sample to Earth where it is placed in a low Earth orbit to be retrieved by the Shuttle.

Two options exist for placing these vehicles in the vicinity of Mars. The first would combine the MAV and ERV for a single launch. The pair would be placed into a suitable parking orbit at Mars and the MAV would descend from this orbit to conduct its portion of the mission. The sequence of events then proceeds as described above. The second option, and the one used for this study, would use two launches to deliver the MAV (with a suitable carrier vehicle) and the ERV to Mars separately. Once the MAV has landed on the surface, the sequence of events again proceeds as above.

Aside from replacing all or part of the MAV propulsion system with ISPP equipment, the only major difference between the conventional system and the ISPP system is that the ISPP MAV will be the active partner in the rendezvous. This is due to the ISPP's ability to produce enough propellant to accelerate the necessary avionics from the surface into orbit and, once there, to execute the required maneuvers.

Having thus set the bounds for the study, Table 5 lists the fixed mass allowances made for both vehicles.

TABLE 5

FIXED MASS ALLOWANCES FOR MARS SAMPLE RETURN

-	RENDEZVOUS AND RETURN ORBITER	446 kg
-	CARRIER FOR LANDER	485
-	BASIC LANDER ALLOWANCE	400 *
-	MAV -- PASSIVE RENDEZVOUS PARTNER	95
-	MAV -- ACTIVE RENDEZVOUS PARTNER	150 *
-	MAV STRUCTURE FOR MOR 1st AND 2nd STAGES (each)	20 *
-	MAV STRUCTURE FOR DR 1st AND 2nd STAGES (each)	40 *
-	EARTH RETURN VEHICLE	155
-	EARTH ORBIT CAPSULE	
	o NON-PROPULSIVE	30
	o 1990 WET MASS	55
	o 1994 WET MASS	58
-	MARS SYSTEM 1st GENERATION (O ₂)	690 †
	2nd GENERATION	750 †

* ESTIMATE

† REFERENCE VALUE, SUBJECT TO MODIFICATION BELOW

Results from the study determined the following injected mass requirements for each of the vehicles in the dual launch scenario.

TABLE 6. COMPARISON OF INJECTED MASS REQUIREMENTS FOR MARS SAMPLE RETURN

<u>Vehicle</u>	<u>1990</u>	<u>1994</u>
Conventional		
ERV/Carrier	2998 kg	2630 kg
MAV/Carrier	<u>3938</u>	<u>3621</u>
Total	6936 kg	6251 kg
ISPP (LOX ONLY)		
ERV/Carrier	3468 kg	2996 kg
MAV/Carrier	<u>4493</u>	<u>4182</u>
Total	7961 kg	7178 kg
ISPP (METHANE/LOX)		
ERV/Carrier	3468 kg	2996 kg
MAV/Carrier	<u>4162</u>	<u>3874</u>
Total	7630 kg	6870 kg

As can be seen, the conventional system is superior to the ISPP system by several hundred kilograms for both opportunities and thus eliminates the ISPP from consideration for this type of mission.

The DE/DR mission scenario will use only one vehicle for both legs of the interplanetary transfer and thus does not require an autonomous rendezvous at Mars. This fact is attractive from a cost point of view since it does not require the expensive development of the rendezvous capability. However, past investigations of this possibility have yielded injected masses so large that designers were forced to use the MOR option.

The basic sequence of events for DE/DR consists of the vehicle making a direct descent to the surface from its interplanetary transfer orbit. Having collected the sample, the ascent vehicle, typically consisting of three stages, is launched from the surface directly onto the return trajectory. Given this scenario, it is easy to see why conventional methods could not possibly satisfy the mission requirements. The additional propellant needed to inject all of the return propellant onto a Mars transfer orbit and then, once at Mars, decelerate

that mass for a soft landing quickly drives the total injected mass at Earth to over 9000 kg. By contrast, the ISPP system will produce 80 percent of its return propellant at Mars and thus considerably reduce the injected mass requirement. A summary of these mass requirements for the two opportunities is shown in Table 7. In all cases, the ISPP based spacecraft has an injected mass which is considerably below that needed by the conventional system.

To summarize the comparisons which have been made for the MSR mission, the effects of using conventional propellants versus ISPP-produced propellants for two different mission scenarios were investigated. The results for two different launch dates were also analyzed to gauge the effects of a good and a poor launch opportunity. The bar chart shown in Figure 7 illustrates the outcome of choosing one system over the others.

As mentioned previously the conventional system outperforms the ISPP system if MOR is used and the opposite proves to be true for DE/DR missions. However, a comparison of MOR with DE/DR shows that the ISPP system, when used in the DE/DR mode, has a considerable advantage over all of the other possibilities. This result has caused a shift to occur in the Mars program away from MOR and towards DE/DR utilizing ISPP. Two major benefits are derived from this shift. The first is a consequence of the lower injected mass. It now becomes possible to use a single Shuttle launch with an IUS upper stage to place the MSR vehicle on to the necessary transfer orbit. The second benefit is that the development cost should be lower due to the fact that the ISPP option requires fewer hardware systems of comparable technology than the MOR option.

The previous analysis served the purpose of assessing the effects of changing the propulsion system on an otherwise unchanged vehicle. While such a restriction was necessary to allow a meaningful comparison of those effects, it did not allow for a fully integrated and optimized design. Work to this end continues at JPL (Hanson 1982) with the result that the injected mass has continued to be reduced. Some of the refinements which have been incorporated thus far include the use of the ascent engines for descent braking, the integration of the lander and ascent vehicle structure, and the use of more accurate determinations of the gravity and drag losses. The outcome of these improvements has been the reduction of the injected mass from 5260 kg to 3950 kg for the 1994 opportunity. The

TABLE 7

DE/DR: MARGINS WITH VARIOUS MARS SYSTEMS

OPPTY.	THREE-STAGE MAV (O_2/CH_4)			MASS (KG)			SURFACE STAY TIME	
	GEN.	SIZE (KG)	PRODUCTION (KG/DAY)	EARTH LAUNCH	LAUNCH MARGIN	SEPARATED LANDER/MAV		REQUIRED PRODUCTION
1990	2nd	750	10*	3101	+1816	2947	3867	290
	2nd	900	14.5	3299	+1618	3145	3867	290
	1st	690	10*	4043	+ 874	3889	2633	290
1994	2nd	750	10*	3041	+2282	2887	3559	340
	2nd	900	14.5	3239	+2084	3085	3559	340
	1st	690	10*	3902	+1421	3748	2423	340

*REFERENCE MARS SYSTEM PERFORMANCE

"MARS" = MARS AUTOMATED REFUELING SYSTEM

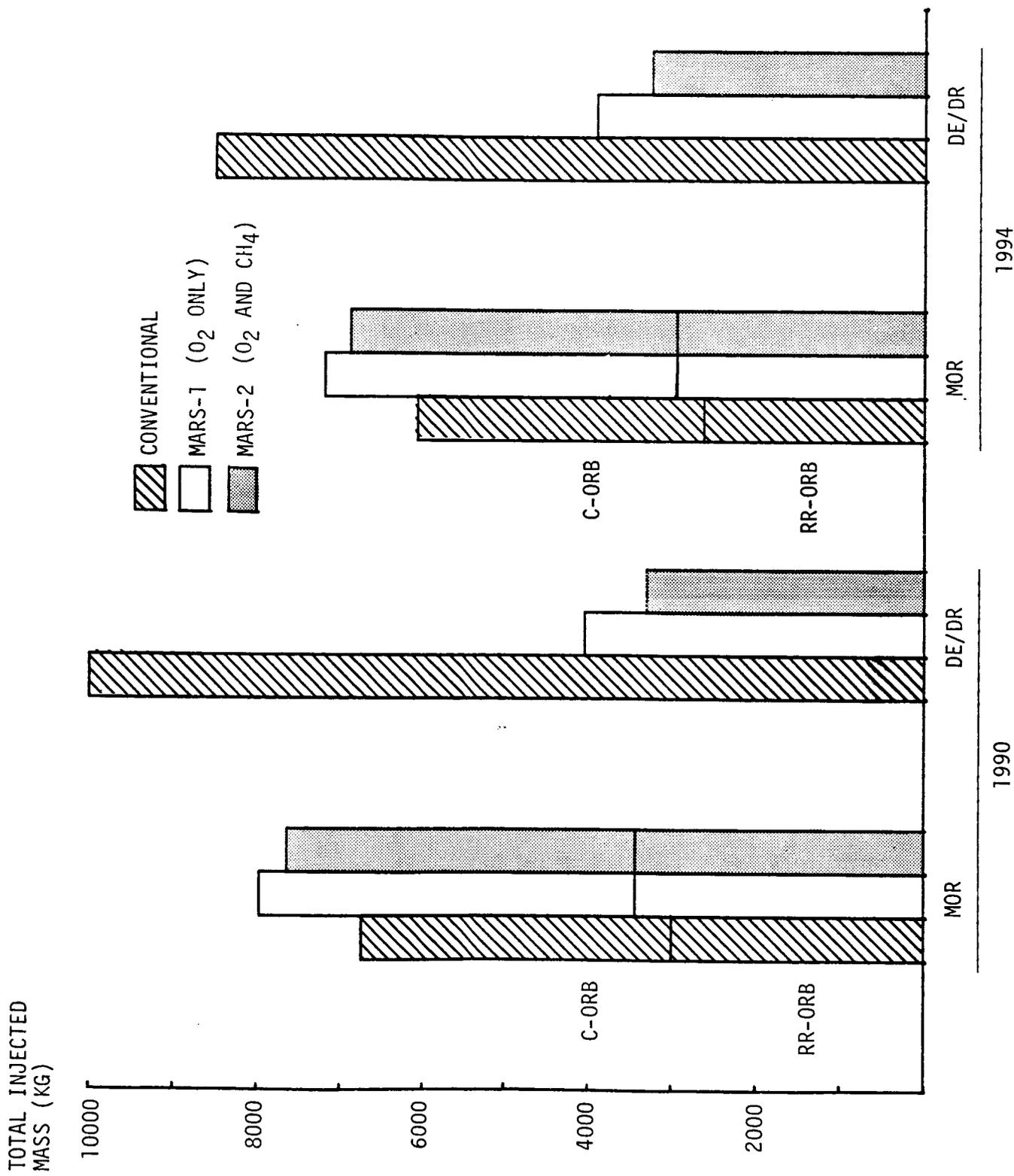


FIGURE 7. ISPP SYSTEM PERFORMANCE SUMMARY

initial mass estimated of 5260 kg is higher than that used in the Stancati et al. report due to the use of a much more detailed and realistic design study. This increased initial mass should not cast doubt on the earlier statements made in association with the Stancati study since all of the results and conclusions were consistent within the scope of the study. It does, however, lend considerable confidence to the conclusion that the ISPP system can be constructed with a mass equal to or less than the autonomous rendezvous vehicle. It would also support the statement that the mission can be accomplished, with a sizable mass margin, using only a single Shuttle/IUS launch. Of course, if the wide-body Centaur is used in place of the IUS, then considerably more mass margin is available for use as a hedge against, or to support, growth in payload/system design mass.

Mars Surface Exploration

An ongoing area of study related to, but independent of, the Mars sample return mission is that of extended exploration of the Martian surface. Previous studies have analyzed both short and long range vehicles to accomplish this mission. The short range vehicles have tended to be either wheeled or tracked rovers and, depending on the type of investigation to be carried out, could be tethered to a fixed lander or be fully autonomous (Paine 1978-1, Paine 1978-2, Minear and Friedman 1978). The maximum range of any of these rovers did not exceed roughly 100 km during its lifetime and would only traverse 1-2 km per day. Two other vehicles with ranges on the order of several thousand kilometers are the Mars ball (Minear and Friedman 1978) and the Mars airplane (Anon. JPL 1978-1, Anon. JPL 1978-2). The Mars ball, whose motion would be only partially controllable, might follow a tumbleweed's path at the mercy of its environment. The airplane would be able to range several thousand kilometers in only a few hours of flight. However, when its supply of hydrazine fuel was exhausted, the mission would be over.

The objective of the study by Hoffman, Niehoff and Stancati (Hoffman, et al. 1982) was to determine whether the feasibility of an "ideal" long range Mars mobility concept would be enabled by the use of an ISPP system. For this investigation two mobility concepts were considered: 1) a rocket-powered

"ballistic hopper" vehicle, and 2) a modification of the previously mentioned Mars airplane. Both systems were analyzed within the context of a Mars landed mass capability similar to that of an MSR mission. Sortie range and total range were regarded as the key measures of performance.

Basic assumptions made which affected both of these systems included use of the ISPP system, described previously, with a mass allotment of 750 kg. This mass was split by assigning 400 kg to the ISPP equipment and 350 kg to the RTG power system. In addition, it was assumed that a Centaur upper stage would be used with the Shuttle. When a worst case transfer orbit during the 1985-95 time period was used, it was determined that a maximum of 7100 kg could be placed on the transfer orbit. Using the Direct Entry option for landing the vehicle on Mars, Table 8 shows the various events which take place between injection onto the transfer orbit and landing on the surface. Once on the surface, two possible propellant combinations were considered for use by either vehicle. As in the MSR study, these combinations consisted of methane/oxygen and carbon monoxide/oxygen.

TABLE 8. LANDED MASS CAPABILITY FOR DIRECT ENTRY AT MARS

INJECTED MASS	7100 KG
LESS:	
TRAJECTORY CORRECTION MANEUVERS $\Delta V(50 \text{ m/sec with } I_{sp} = 215 \text{ sec})$	166
BALLUTE AND PARACHUTE	110
TERMINAL DESCENT $\Delta V(150 \text{ m/sec with } I_{sp} = 365 \text{ sec})$	280
LANDED MASS	6544 kg

At this point the analysis of each system followed an independent path. For the hopper, the equations of motion for a ballistic trajectory were numerically integrated to determine the actual range which could be obtained for a given amount of onboard propellant. Gravity and drag losses were taken into account

in this process. The performance of the airplane was determined by retaining its previously determined flying characteristics but modifying the power plant to use other propellants. A specific fuel consumption for each propellant combination was determined which would allow a range to be calculated based on the amount of propellant which could be carried.

It was decided at the outset of the study that a one-way range of approximately 1000 km and more than one sortie (i.e. ballistic hop or airplane flight) would be desirable to allow diverse regions of the surface to be explored. Failure to meet these criteria constituted grounds for dropping the concept from further consideration.

Four hopper concepts were identified as potentially being capable of meeting these criteria. These concepts included:

- 1) A round-trip ballistic hopper
- 2) A one-way ballistic hopper
- 3) A round-trip hopper with a lifting aerobody
- 4) A one-way hopper with a lifting aerobody

The round-trip hoppers were defined to be ballistic flight vehicles which carried sufficient propellant onboard to move a significant distance away from and then return to a fixed ISPP site. The one-way hoppers would carry the ISPP equipment along to the new site which would then allow the vehicle to be refueled in place for the next hop. Finally, it was assumed for the aerobody cases that minor aerodynamic, structural and/or packaging changes could be made to a Viking class aeroshell which would increase its lift-to-drag ratio. This additional lift could then be used to extend the range of the hopper during the descent phase of the flight.

Analysis of the round-trip hoppers quickly indicated that this option should be eliminated from any further consideration. Two limitations of this particular concept lead to these conclusions. The first is that a fixed ISPP depot limits the area that can be investigated to a circle centered on the depot and having a radius equal to half of the maximum range of the hopper. Carrying the processor along permits repetitive one-way hops which effectively increases the total range. The second limitation is that in order to reach the desired 1000 km

range, analysis indicated that the round-trip hopper would require a total delta-V of about 6 km/s, the propellant for which will exceed the propellant needed to carry the ISPP along to the new site.

The results of the aerobody cases indicated that even when flying a maximum lift entry profile, the range extension obtained by a typical aeroshell cone ($L/D < .5$) was negligible. L/D 's greater than 1.0 were required to produce any noticeable increase in range. This appears to be due to the fact that at the speeds involved, the Martian atmosphere is too tenuous to provide significant amounts of lift without using a device that begins to resemble a winged vehicle. Any such appendages would need to be retracted during ascent and then redeployed for entry. Even with the larger values of L/D , range was not significantly improved. Hence, the high L/D aerobody concept was dropped from further analysis.

The concept option remaining to be investigated was the choice of the propellant combination to be used. As mentioned earlier, the choices consisted of methane and oxygen (CH_4/O_2) or carbon monoxide and oxygen (CO/O_2). The ratio of initial mass to final mass for CO/O_2 was found to be roughly 50 percent higher than that for CH_4/O_2 . This result along with the fact that 80 percent of the CH_4/O_2 propellant mass is produced locally by the ISPP system indicated that CH_4/O_2 was the propellant combination of choice. The CO/O_2 option was thus dropped from further consideration.

At this point, the vehicle configuration has been narrowed to a multiple sortie ballistic hopper using a low L/D aeroshell decelerator. The vehicle carries the ISPP system to each new site to produce oxygen to be used in combination with transported methane for the primary propellant. The final task was to determine the actual range possible when gravity and aerodynamic drag losses are taken into account from liftoff to landing. In order to characterize the performance of the vehicle, two parameters remained to be investigated. These were the number of sorties and the number of engines. Since the amount of methane is fixed, the number of sorties will affect the range per sortie and the cumulative range of all the sorties. The number of engines used will also affect the range in two ways. First, as the number of engines is increased that amount of methane must be decreased to maintain the same total mass. But secondly, as the number of engines is increased the gravity losses will decrease since there is a larger acceleration placed on the vehicle and shorter burn times are required.

To quantify these trade-offs, some assumptions about the mass breakdown of the hopper were required. The assumptions made are listed in Table 9. These assumptions yield a net vehicle dry mass of approximately 4000 kg to which must be added the mass of the engines, tankage and deceleration devices. The total mass for these latter three items is dependent upon both the number of sorties made and the number of engines used.

TABLE 9. DRY MASS FOR BALLISTIC HOPPER

ISPP	750 kg
SCIENCE	50 kg
ROVER	50 kg
STRUCTURE AND SUPPORT SUBSYSTEMS	1963 kg
AEROSHELL	1178 kg
ENGINES (2000 LB _f MAXIMUM THRUST EACH)	47 kg EACH
BALLUTE/PARACHUTE	100 kg PER SET
PROPELLANT TANKAGE	--- 15% OF PROPELLANT MASS

At this point a parametric analysis was carried out to determine the effect on range of the number of sorties and the number of engines used by numerically integrating the equations of motion. The flight profile used for this integration made the following assumptions. The methane fuel would be equally divided between each of the sorties. This allowed a fixed amount of oxygen tankage to be used to its maximum capacity on each sortie. It would, however, cause the initial sorties to be shorter than those that follow since the lift-off mass of each sortie is reduced by the methane fuel burn in the previous hop. At lift-off, the vehicle would be launched in a near vertical direction and use a "gravity turn" to cause it to follow a ballistic arc. On entry, an expendable ballute would be deployed at Mach 5 followed by an expendable parachute deployment at Mach 1. In the terminal descent phase, a delta-V budget of 150 m/sec was allowed. In all the cases examined, this budget was found to be adequate. Finally, it was assumed that the 400 kg ISPP unit would be jettisoned before the final sortie but the 350 kg power system would be retained to provide electrical support for the other subsystems.

The results of the analysis in terms of the minimum and maximum range per sortie as well as the total range of all sorties are summarized in Table 10. A plot of the total range of all hops versus the parameters which were varied is presented in Figure 8.

A 10-engine configuration was necessary to achieve the guideline range of 1000 km in a single hop. It should be noted that the ranges for multiple hops begin to fall off when more than eight engines are used. This is due to the fact that the loss of propellant mass brought about by the increased engine mass more than offsets the reduction in gravity losses due to increased acceleration. These results indicate that the case which provides the best results that are consistent with the multiple sortie and longest range objectives uses eight engines and investigates three sites (i.e. two hops). This combination has a minimum range of about 380 km and a maximum range of roughly 640 km for a total maximum range of 1030 km.

The other mobility concept investigated in this study was a variation of the Mars airplane designed by JPL and Developmental Sciences, Inc. (Anon. DSI 1978). For purposes of comparison, it was assumed here that the airplane would retain the same configuration and aerodynamic characteristics. The maximum weight of the vehicle was kept at 300 kg, but the science instrument complement was resized for the reconnaissance mission and the propellant combination was then varied to determine the effect on range.

Combining the Mars airplane with ISPP capability essentially meant replacing both the cruise and lift-off/landing engines with those using either CH_4/O_2 or CO/O_2 propellants. Since carrying the ISPP onboard the aircraft would exceed the imposed weight limitation, this analysis assumed that the airplane would have to be refueled from a fixed propellant production base. All sorties, therefore, are round-trips. While this limits the area accessible for airborne reconnaissance, it does permit gathered data to be collected and processed through the ISPP base rather than having this task accomplished onboard the aircraft.

TABLE 10. HOPPER RANGE AS A FUNCTION OF ENGINES AND HOPS

NUMBER OF HOPS	NUMBER OF ENGINES*				
	5	6	8	10	
1	RANGE _{TOT}	0.0	9.80	980.14	1294.9
	RANGE _{MIN HOP}	0.0	9.80	980.14	1294.9
	RANGE _{MAX HOP}	0.0	9.80	980.14	1294.9
2		469.36	794.41	1030.14	989.22
		37.27	202.21	386.30	385.17
		432.09	592.20	643.84	604.05
3		463.36	592.33	662.12	612.33
		100.79	149.81	159.09	154.25
		206.71	254.44	302.97	270.48
4		363.14	417.07	415.89	390.94
		44.72	66.73	73.55	72.06
		158.71	164.81	152.01	136.80
5		267.47	289.86	289.99	269.82
		33.86	39.85	43.62	42.87
		88.61	89.63	82.59	72.72

* 47 kg per engine; 8900 N (2000 lb) maximum thrust per engine.

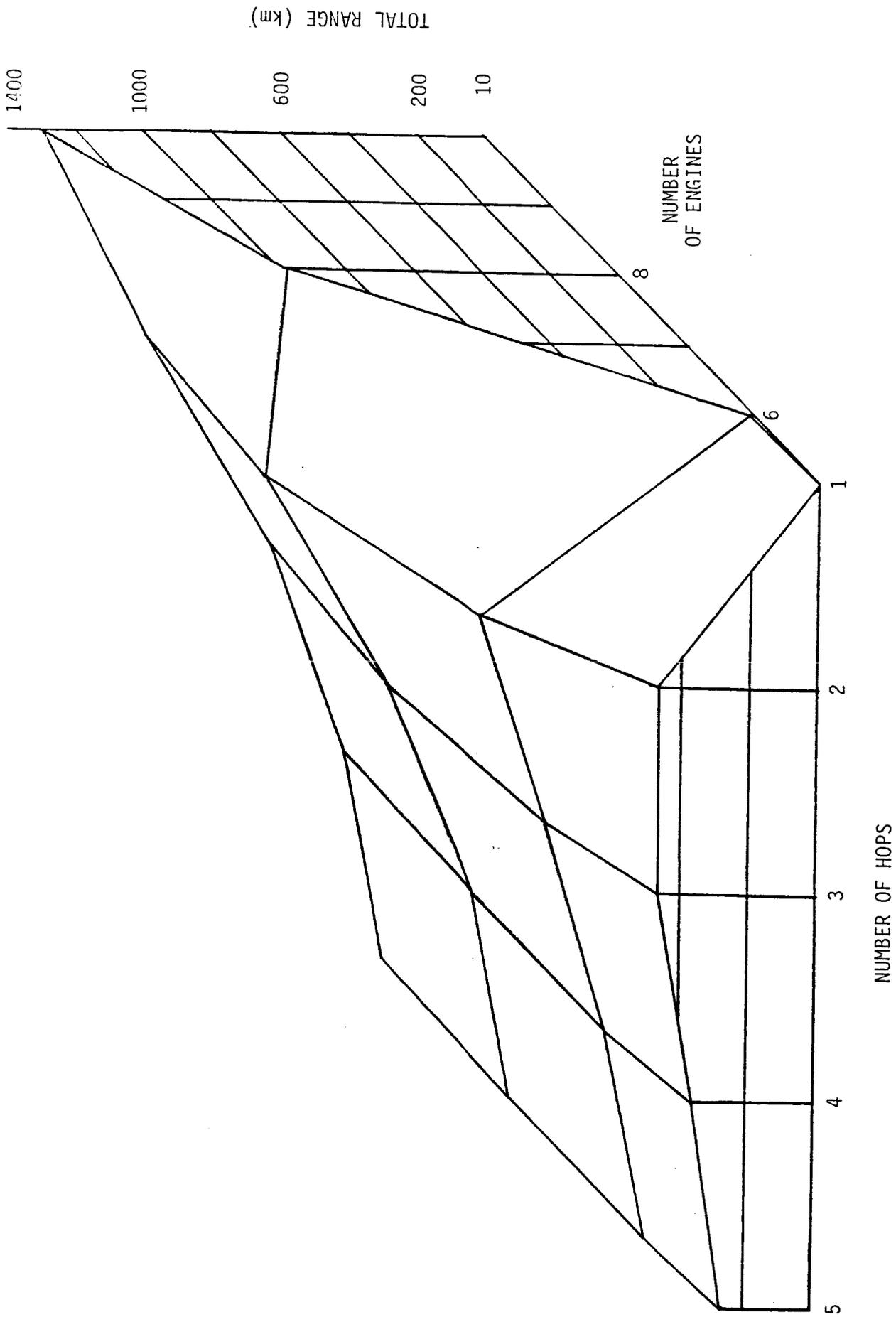


FIGURE 8. TOTAL HOPPER RANGE

The mass breakdown of the entire vehicle (both the processing plant and airplane) is essentially the same as that for the ballistic hopper and is shown in Table 11. This table indicates that 2754 kg are available on the lander for propellants, tankage and the ISPP equipment. The depot storage tankage mass was assumed to be 15 percent of the maximum storable propellant mass at the ISPP base. In addition, the table shows that a propellant mass budget of 90 kg has been assumed for the airplane. This mass will be used for lift-off and landing as well as cruise.

TABLE 11. MASS ALLOCATION ASSUMPTIONS FOR AIRPLANE AND SUPPORT LANDER

TOTAL LANDED MASS CAPABILITY	6544 kg
LESS:	
LANDER SCIENCE	50 kg
REFUELING ROVER	50 kg
TERMINAL DESCENT ENGINES	339 kg
STRUCTURE AND SUPPORT SUBSYSTEMS	1963 kg
AEROSHELL	1178 kg
DRY AIRPLANE	210 kg
	2754 kg
UNALLOCATED MASS	2754 kg

The flight profile assumed for the aircraft was to cruise at one kilometer above the surface at an average velocity of 295 km/hr. After taking into account the efficiencies of the propeller and gearbox, it was determined that the powerplant would be required to produce 3740 watts (5.11 hp) during cruise. This amount of power is independent of the type of powerplant used. The delta-V budget for one take-off and landing was fixed at 316 m/sec and was assumed to apply to all propellant types.

The heart of the analysis consisted of characterizing the performance of a powerplant which uses either methane and oxygen or carbon monoxide and oxygen. Any number of energy conversion devices could be used with these propellants.

Examples include a gas generator-turbine combination, a positive displacement Rankine cycle engine (e.g. the "Mini-Sniffer" engine, Akkerman 1979) or a conventional Otto cycle internal combustion engine. Each of these devices can be characterized by a thermal efficiency which is defined to be the actual amount of heat transformed into work divided by the amount of heat released by the propellants during combustion. Since no data was available to allow an outright selection of any of these engines, three efficiencies: 0.30, 0.50 and 0.70 (the first for the Otto cycle and the last two for the Rankine cycle) were carried through the remainder of the analysis.

With a knowledge of the energy content of the various propellant types, a series of calculations allowed a sortie range for each of the three efficiencies to be determined. These have been listed in Table 12. All ranges shown here are total ranges and thus the distance that the airplane can travel from the base is only half the indicated amount. All ranges are also more than double that which can be obtained by the hydrazine powered aircraft. This additional range cannot be accomplished without a price, however. Recall that at least 750 kg worth of ISPP processing equipment must be brought to the surface before either the CH_4/O_2 or the CO/O_2 version of this aircraft could fly. This mass could just as easily have been used for hydrazine and its associated tankage at a central depot. Figure 9 presents total range versus surface mass for propellant, depot tankage, and ISPP equipment, if used.

TABLE 12. ENDURANCE AND RANGE CAPABILITIES FOR VARIOUS ENGINE EFFICIENCIES

<u>ENGINE EFFICIENCY</u>	<u>ENDURANCE (HOURS)</u>		<u>RANGE (KM)</u>	
	<u>CH_4/O_2</u>	<u>CO/O_2</u>	<u>CH_4/O_2</u>	<u>CO/O_2</u>
0.3	13.5	7.21	3990	2128
0.5	22.5	12.0	6651	3546
0.7	31.5	16.8	9310	4965

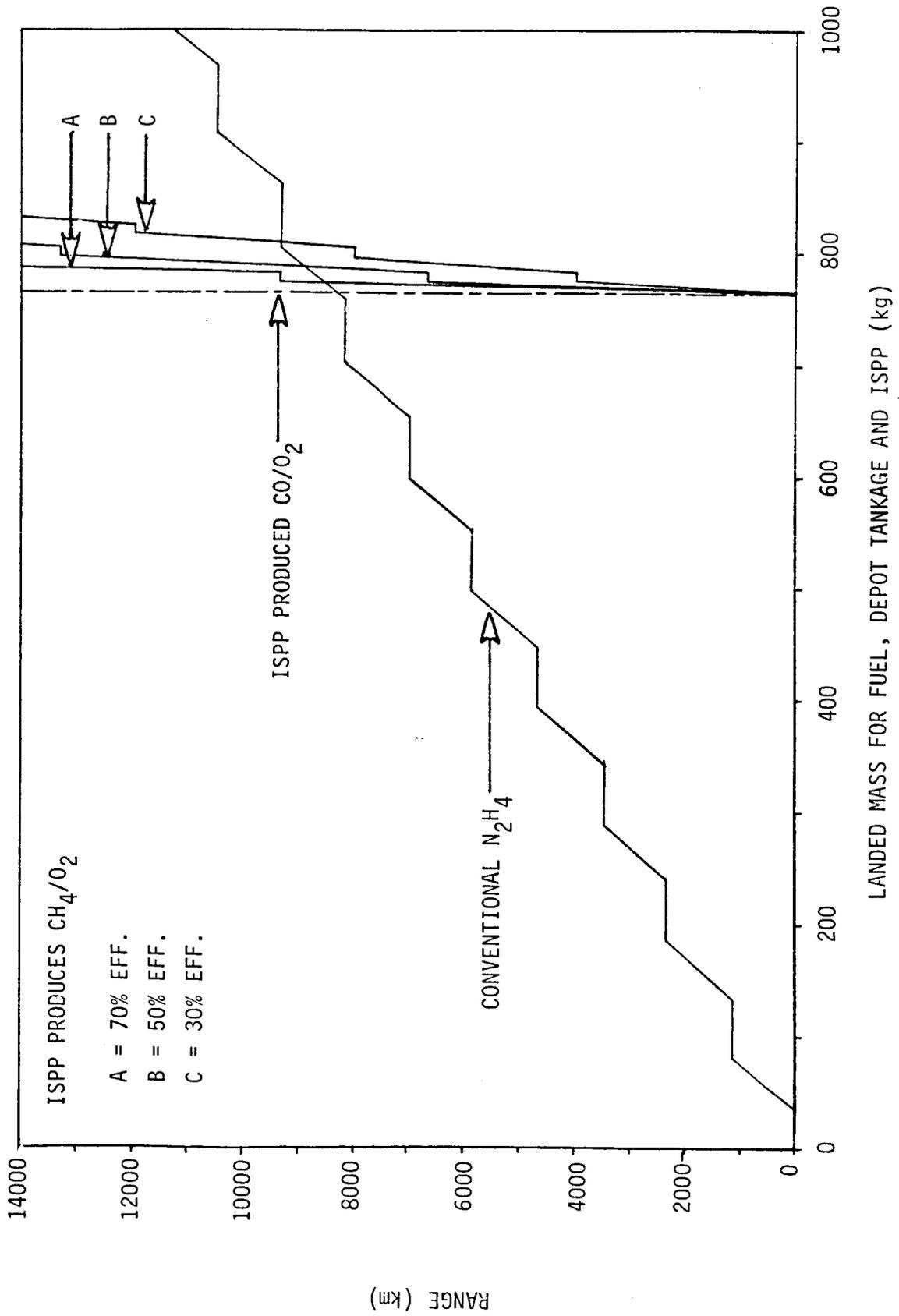


FIGURE 9. AIRPLANE RANGE CAPABILITIES USING VARIOUS PROPELLANTS

As can be seen, both the ISPP-based cases intersect the hydrazine case very shortly after the mass for the ISPP equipment has been reached. For masses below this point, a trade-off must be considered between using the depot for the storage of hydrazine and the storage of CH_4/O_2 and its associated refrigeration equipment. ISPP is clearly superior for masses above the 750 kg point.

In summary, the ISPP system allows for the exploration of diverse sites on the surface of Mars at ranges greater than any of the previously investigated concepts. However, the ranges are still not great enough to allow for global exploration by one or at most a few vehicles. Of the two concepts analyzed in this section, the ballistic hopper provides rather marginal performance at best for the mass invested in the system. The Mars airplane, on the other hand, is more promising in that multiple sorties from a fixed depot site can, in the case of a CH_4/O_2 -propelled vehicle, cover areas approaching one quarter to one half of the surface area of the planet.

4. SMALL BODIES

Among some of the more difficult solar system exploration missions under consideration are those involving a sample return from primitive bodies; that is from comets and asteroids. If suitable raw materials are present on these bodies, then ISPP may provide sufficient mass relief to make the missions more attractive. At present, only one study (Stancati, et al. 1978) has been completed which tries to confirm this possibility.

Two targets, comet Encke and asteroid 19 Fortuna, were selected for investigation since present knowledge indicates that both have the required raw materials necessary to produce oxygen and methane. In addition, comet Encke was chosen for its relatively short period. As indicated by the raw materials, the return leg of the mission would use the methane/oxygen propellant combination with the ISPP producing just oxygen or both oxygen and methane.

To provide a basis for comparison, each mission was analyzed using two possible means of returning the sample to Earth. The first method, which is also the current baseline mission, would use a low thrust Solar Electric Propulsion Stage

(SEPS) for the return trip. The other method would use the previously mentioned methane/oxygen system. Both options would use a SEPS stage for the outbound leg and would return a one kilogram sample by direct entry into the Earth's atmosphere. The Earth entry capsule (EEC) would have a mass of 30 kg and would be limited to entry speeds of less than 50,000 ft/sec.

The all-SEPS mission was assumed to require a 600 kg (wet) lander to land, collect, and return to the SEPS vehicle with the sample. The lander would then be jettisoned and the SEPS would return the EEC to the vicinity of the Earth.

The ISPP mission would require a 200 kg (dry) lander to deliver the ISPP, Earth return vehicle (ERV; basically the same as that used for the Mars sample return mission), and the EEC. A mass budget of 155 kg was assigned to the ERV, but no fixed mass was set for the ISPP. Due to uncertainties in the form of the raw materials and thus in the gathering and processing subsystems, the analysis provided only the mass margin for the ISPP system. A rough estimate of feasibility could then be obtained by comparing this mass margin with the necessary production rate.

Comet Sample Return

The following table shows the results of the analysis for comet Encke.

TABLE 13. PERFORMANCE COMPARISON FOR COMET ENCKE MISSION

	<u>All-SEPS</u>	<u>O₂ ONLY</u>	<u>O₂/CH₄</u>
Launch Date	Feb. 2, 1991	Feb. 2, 1991	Feb. 2, 1991
Stay Time (days)	100	454	454
Trip Time (years)	6.1	7.1	7.1
Mass Margin at Rendezvous (kg)	289		
Mass Available for ISPP (kg)		379	588
Required Propellant			
Production Rate (kg/day)	---	1.7	2.2

Recall that the All-SEPS mass margin is a truly free margin with no liens placed upon it. The ISPP margins indicate the mass available for the ISPP system. In both ISPP cases, the low production rates indicate that packaging the ISPP in the given mass allocation is not out of the question.

Asteroid Sample Return

Results of the analysis for the asteroid 19 Fortuna are as follows.

TABLE 14. PERFORMANCE COMPARISON FOR ASTEROID 19 FORTUNA MISSION

	<u>All-SEPS</u>	<u>O₂ ONLY</u>	<u>O₂/CH₄</u>
Launch Date	Aug. 16, 1990	Aug. 24, 1990	Aug. 24, 1990
Stay Time (days)	292	655	655
Trip Time (years)	4.8	4.7	4.7
Mass Margin at Rendezvous (kg)	364		
Mass Available for ISPP (kg)		607	1021
Required Propellant			
Production Rate (kg/day)	---	2.3	2.9

The comments following the comet mission mass margins also apply here. The ISPP margins in this case are higher but the production rates have also increased. The net effect of this is neither to improve nor diminish the chances of packaging the ISPP in the given mass margin when compared to the comet mission.

The results from this study can be summarized as follows. The ALL-SEPS modes appear to hold a performance advantage over the SEPS/ISPP systems in terms of smaller SEPS propellant mass, shorter trip time, and/or rendezvous mass margin. The SEPS/ISPP modes, on the other hand, remain at the target much longer to process fuel, which may have certain unique science benefits. In addition, their impulsive ballistic return transfers are shorter and simpler to control. Taken as a whole, the All-SEPS modes probably hold an edge in overall performance.

5. GALILEAN SATELLITE EXPLORATION

The four Galilean satellites of Jupiter (Io, Europa, Ganymede, and Callisto) have often been described as making up a miniature solar system. This offers an excellent opportunity for a comparative study of the evolution of the two systems. In addition, the geological age of the surface of each moon varies from extremely old (Callisto) to very young (Io) and thus argues for a comparative study of the moons relative to one another. There has also been speculation that Europa may have all the ingredients to support life below its surface. Such wide open possibilities point to an interactive investigation of this group of moons with the spacecraft segment having been designed for redirection of its mission based on results obtained from data which it has collected. This could significantly reduce the need for follow on missions.

One of the previously mentioned advantages of ISPP is the large amount of electrical, chemical and/or thermal energy which can be made available. This energy could be used to power robotics and/or rovers on the surface, propel hoppers to other points on the surface or to the other moons, or, finally, to return samples to the Earth for analysis. The last two possibilities have been studied in some depth and will be discussed in the following paragraphs.

These two mission types will be referred to as one-way hoppers and sample returns. The former name indicates that the vehicle remains in the vicinity of Jupiter and does not return to Earth. Each of these mission types can be further subdivided depending upon the number of moons the vehicle or vehicles will visit. The one-way landers would then use the ISPP derived propellant to visit multiple sites on one of the moons or use it to travel to another body. In this case, the option exists of either landing or orbiting that satellite. Both options have been investigated. The sample return missions have basically the same option; that is, visit only one satellite and return a sample from it or to visit and collect samples from more than one of the Galilean satellites.

The propellant combinations which have been examined in the course of these studies include LOX/methane and LOX/hydrogen. The latter would produce both components on those moons which possess water. The methane, if used, would be transported from Earth and only oxygen would be produced locally. In either situation, there is a strong case for basing the ISPP system at Europa.

Given that there are equally compelling scientific reasons to investigate each of the four satellites, the needs of the ISPP system provide a reasonable means of choosing one of the moons over the others as the primary landing site. Table 15 lists some of the important physical characteristics of the four targets.

TABLE 15. PHYSICAL PROPERTIES OF GALILEAN SATELLITES

Satellite	Orbit Period (days)	Mean Surface Gravity (m/sec)	Escape Velocity (km/sec)	Maximum Temperature (K)	Bond Albedo
Io	1.77	1.44	2.30	141 + 11	0.56 + 0.12
Europa	3.55	1.31	2.01	139 + 12	0.58 + 0.14
Ganymede	7.15	1.34	2.73	154 + 6	0.38 + 0.11
Callisto	16.69	1.03	2.27	167 + 3	0.13 + 0.06

(From Ash, et al., 1980)

The mean surface gravity and escape velocity are virtually the same for all bodies which means that the propellant production requirements at each will be nearly identical. However, a source of water is required for both of the potential propellant combinations which eliminates both Callisto and Io. Studies by Mandevill, Geake, and Dollfus (Mandeville, et al. 1980) indicate that the spectra from Europa's surface is much more heavily dominated by water than is Ganymede's. In addition, the maximum surface temperature of Europa is below that required to passively store both liquid methane and LOX. Ganymede would require active cooling for LOX, by far the larger component of either the LOX/hydrogen or LOX/methane combinations. For these reasons, Europa is the preferred site for basing the ISPP system.

One-Way Landers

The first option examined for the one-way lander mode was that of a single target, multi-site vehicle. The lack of a substantial atmosphere on any of the moons allowed a quick assessment to be made of the velocity change, and thus the propellant requirement, needed to move from one point on the surface to any other point using a simple ballistic flight path. Figure 10 shows these velocity changes, in terms of the central angle traversed, for each of the four possible targets.

The analysis for the single target, multi-site landers assumed that all tankage and ISPP equipment would be transported to each new site. The only decrease in lift-off mass at each subsequent site would be the result of a loss of hydrazine used during terminal descent maneuvers. Thus the propellant requirement remains almost constant regardless of the number of hops that are made so long as those hops all travel equal distances.

Of the two targets where oxygen and hydrogen (if used) can be produced, only Europa was examined in detail. Results for Ganymede would be similar but somewhat larger due to the higher surface gravity. Figures 11 and 12 show the Earth launch mass and propellant production requirements for one, two, and three hops using LOX/hydrogen.

Two points are worthy of note. First only a ten percent increase in launch mass is required to allow three hops instead of only one. This would allow investigation of four rather than two sites. The ten percent mass growth is a result of the need to carry additional hydrazine and the increased structural mass to support the larger weight. The second point is that for any given number of hops, the launch mass is virtually insensitive to the distance travelled per hop. What increase there is results from the augmentation in tankage mass needed to hold the near tripling in propellant mass.

The second lander option studies involved visiting more than one target and, if a landing is made, visiting no more than one site on that target. Figure 13 shows the Earth launch mass for a number of different target combinations.

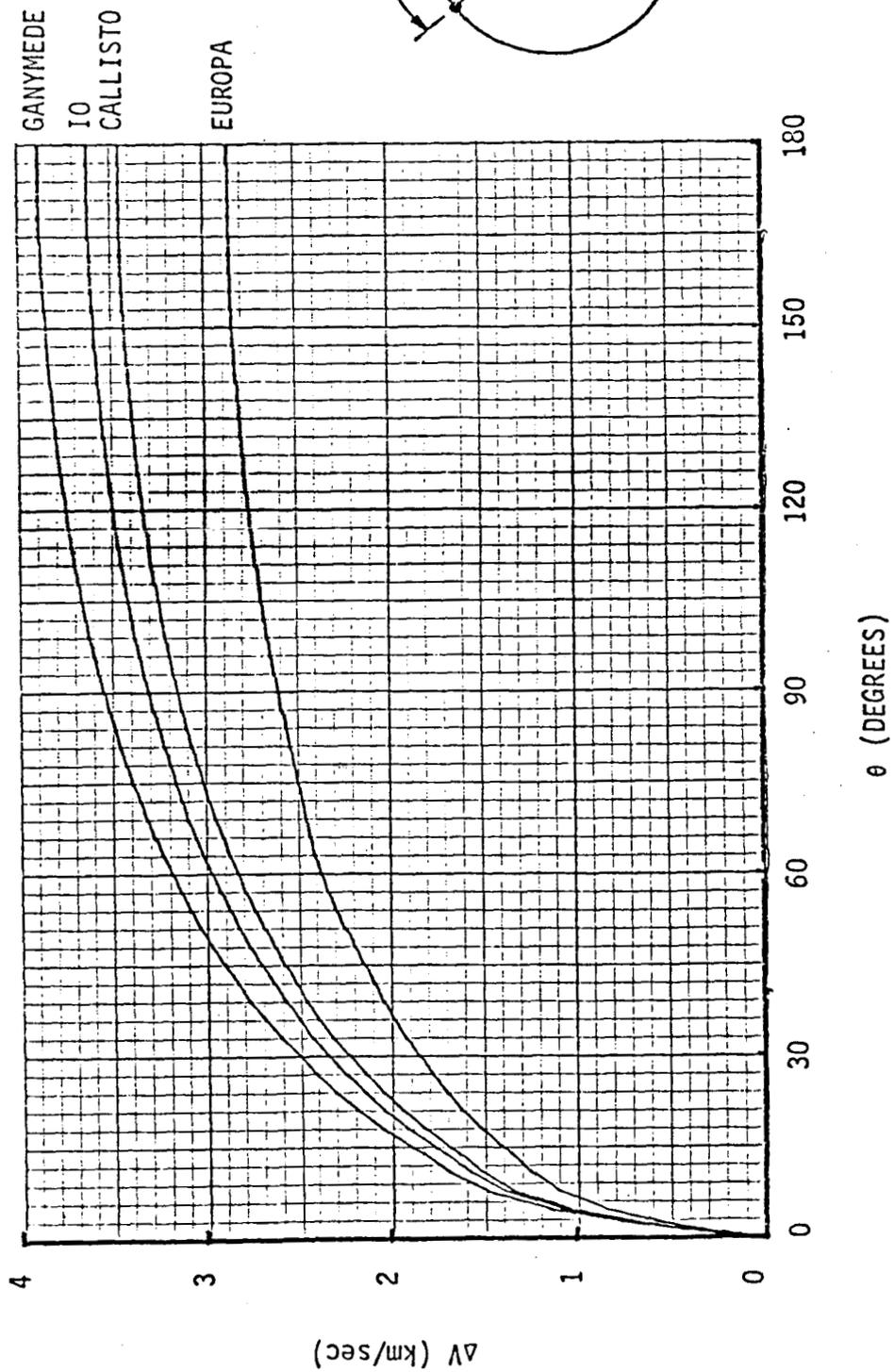


FIGURE 10. TOTAL VELOCITY IMPULSE REQUIRED FOR TRAVEL BETWEEN ANY TWO SURFACE SITES

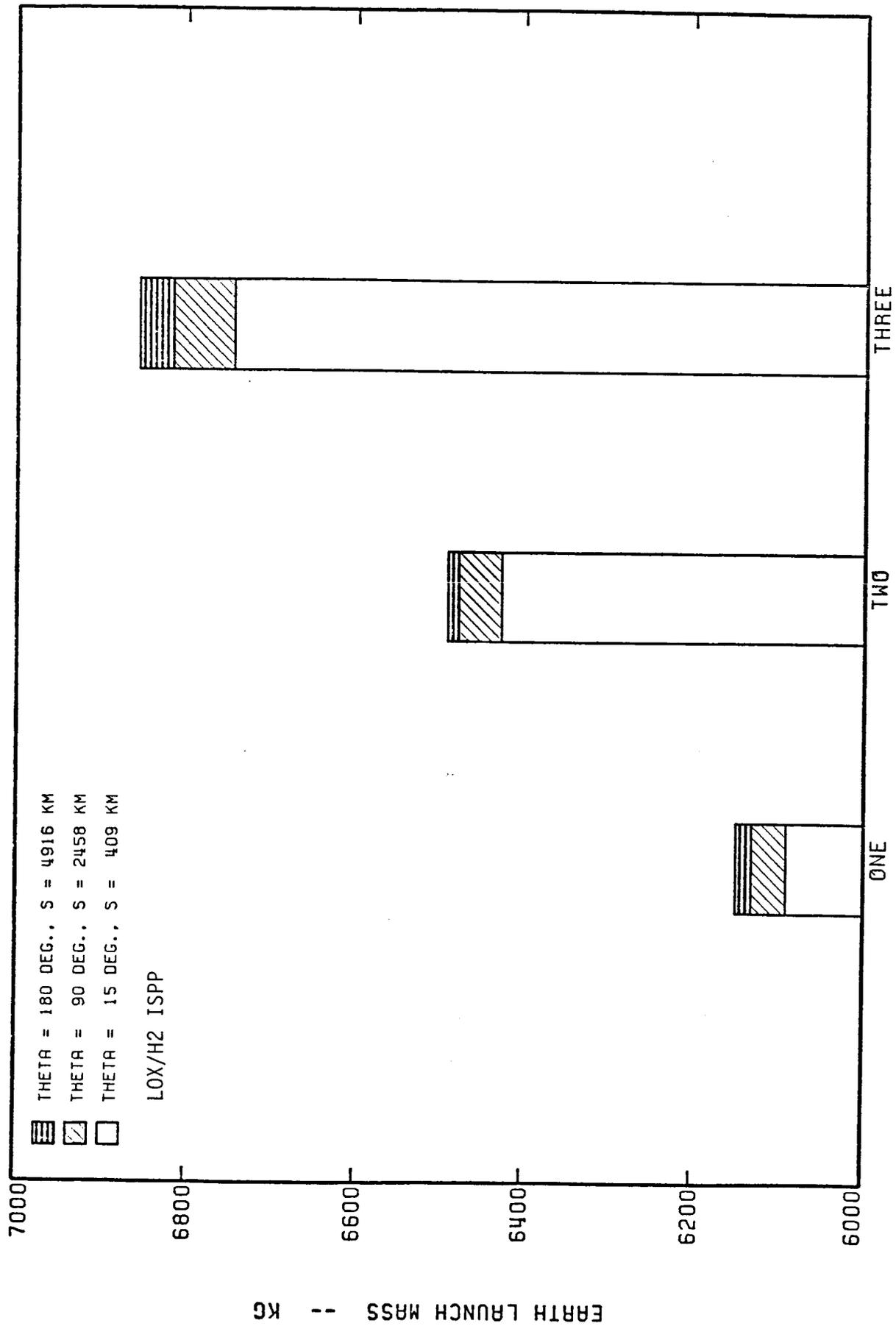


FIGURE 11. INJECTED MASS REQUIREMENTS FOR EUROPA LANDER

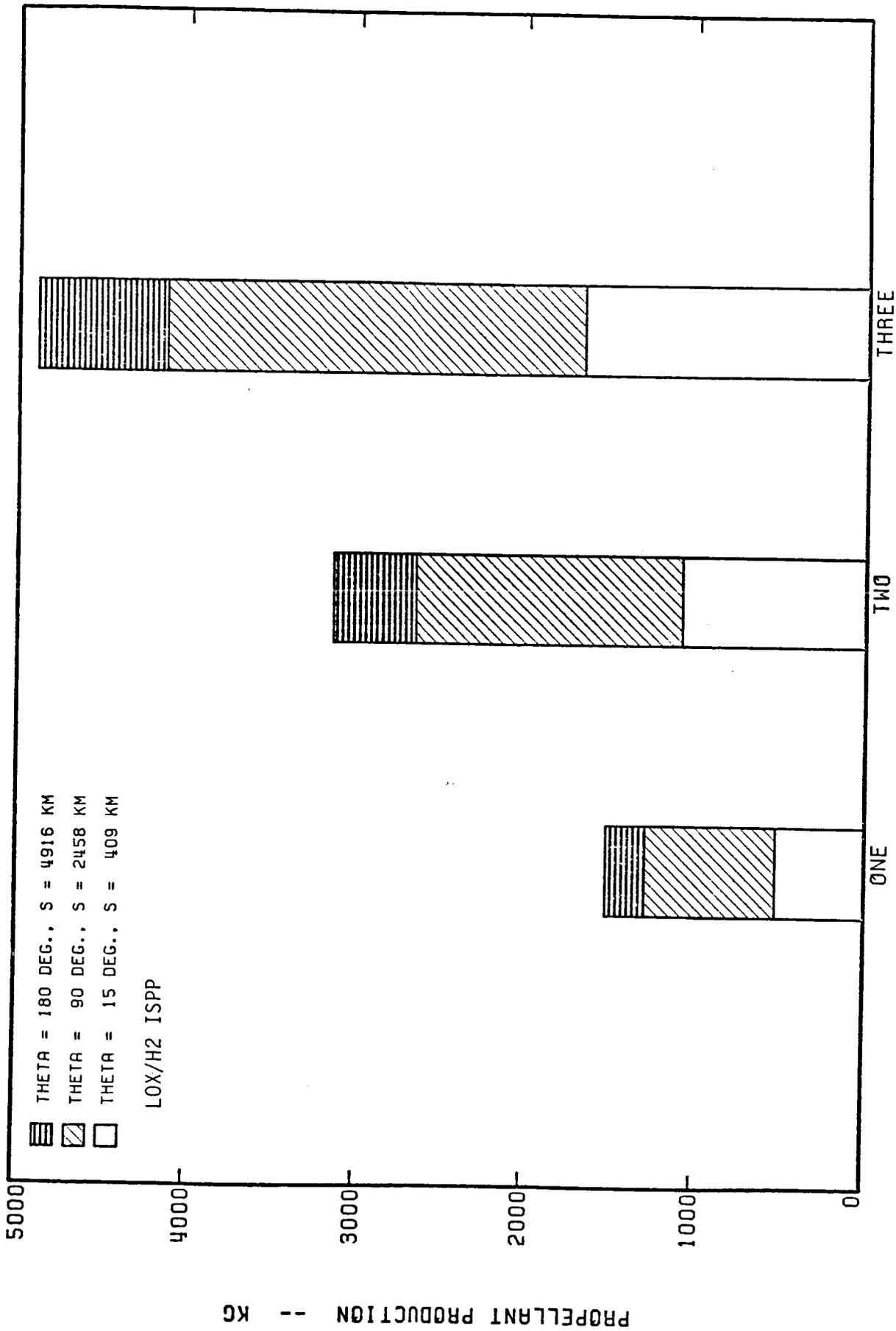
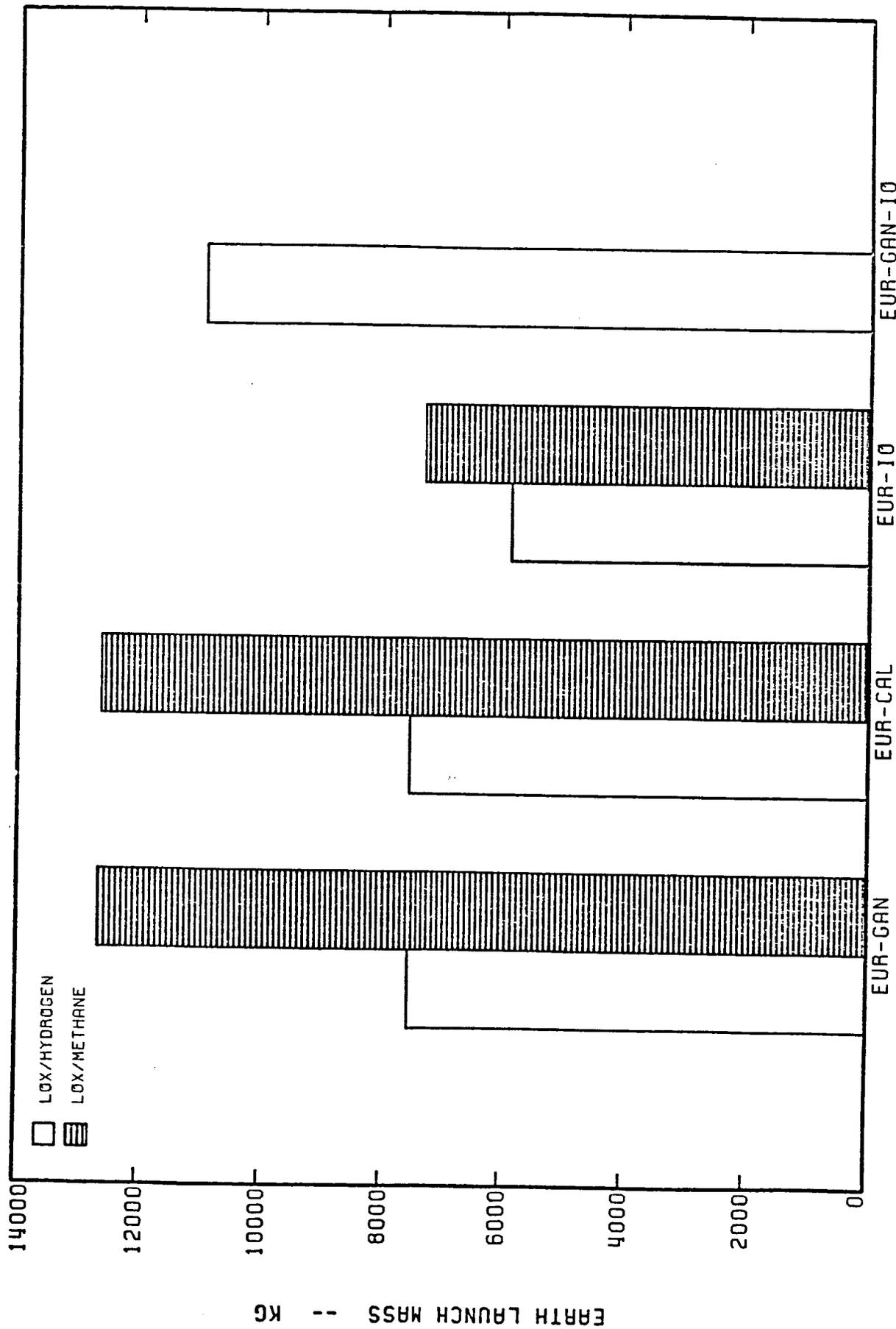


FIGURE 12. PROPELLANT PRODUCTION REQUIREMENTS FOR EUROPA BALLISTIC HOPPER



MULTI-TARGET LANDER MISSION

FIGURE 13. INJECTED MASS REQUIREMENT FOR MULTI-TARGET LANDER MISSION

The results of those cases involving Io are lower than might be expected due to the fact that the vehicle was assumed to only orbit this moon and not land. The launch masses of the LOX/methane cases are uniformly higher than the LOX/hydrogen results because of the lower specific impulse for LOX/methane and the fact that all of the methane must be transported from Earth.

When these results are compared to an equivalent vehicle using space storable propellants (i.e. fluorine and hydrazine), no significant mass savings occur. It appears that the price of carrying the ISPP system into and out of several gravity wells is too great.

Sample Return

Sample return missions from the Galilean satellites using ISPP show a significant improvement in performance when compared to conventional methods. This is true of both the single and multiple target missions. In fact, a dual sample return using ISPP has an Earth launch mass slightly less than a single target mission using fluorine/hydrazine.

The analysis for both the single and multiple target options assumed that a 2+ type delta-VEGA trajectory would be used to place the spacecraft in the vicinity of Jupiter. Satellite touring would be used to lower the velocity of the spacecraft at approach to the target. Flight profiles are illustrated in Figures 14 and 15. A five kilogram sample would then be collected at each moon. Satellite touring would again be used to increase the energy of the sample vehicle which would then return to Earth on a direct trajectory. The resulting total mission times were found to be on the order of ten years. As in the lander analysis, the ISPP missions were compared with those using fluorine and hydrazine as the propellant combination.

This analysis also looked at both LOX/hydrogen and LOX/methane ISPP systems, with all methane transported from Earth. Due to the extended times spent in satellite touring for the return flight, possibly severe hydrogen boiloff caused a modification to be made in the LOX/hydrogen system. Since methane and LOX could be stored relatively easily as liquids during the tour, LOX/hydrogen would

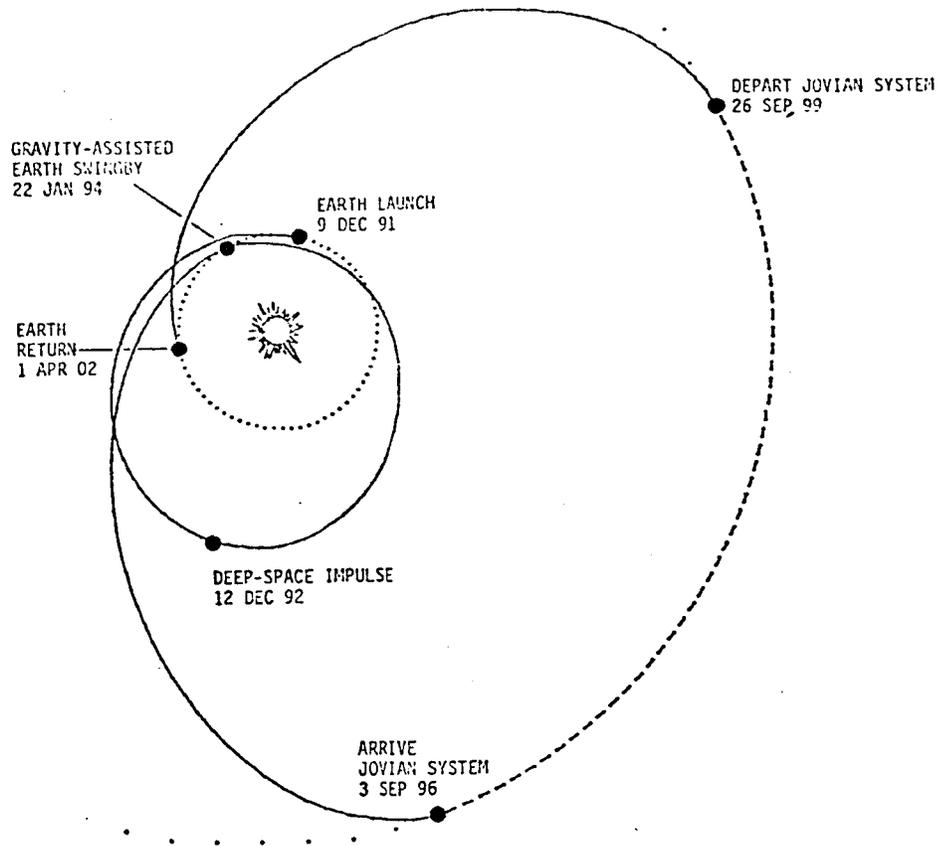


FIGURE 14. OUTBOUND AND RETURN TRAJECTORY FOR GALILEAN SATELLITE SAMPLE RETURN

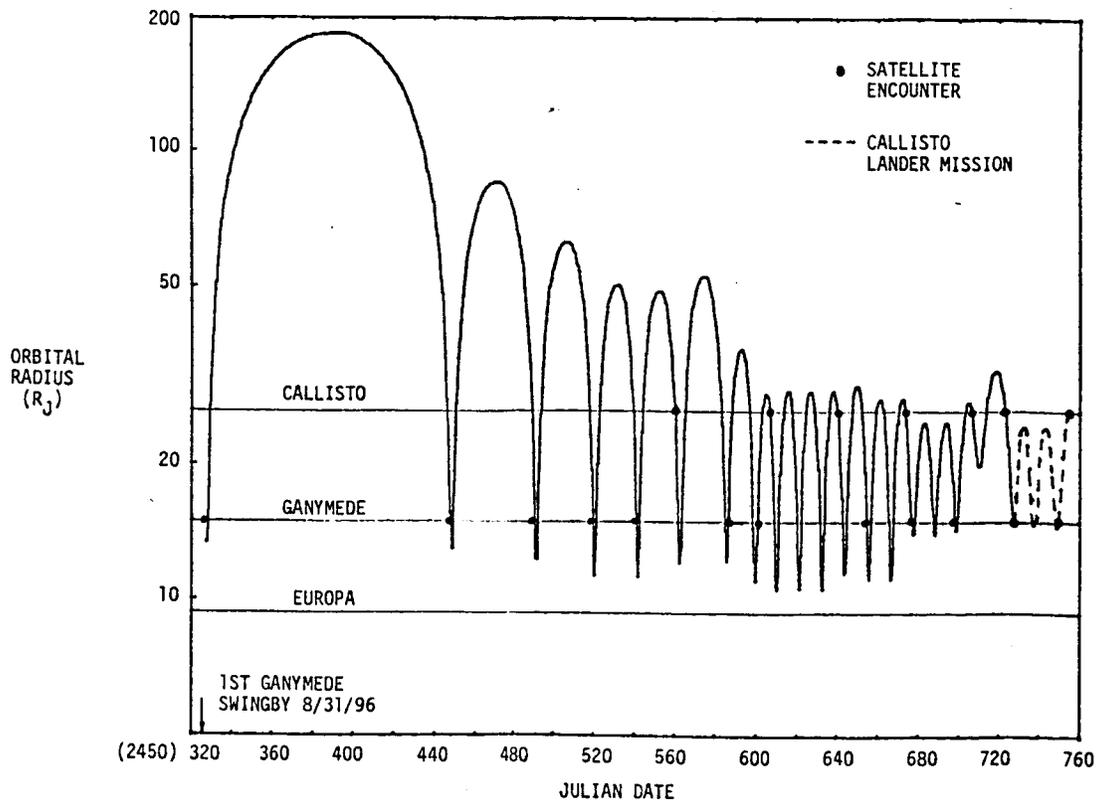


FIGURE 15. SAMPLE SATELLITE TOURING PROFILE

only be used at lift-off while LOX/methane was used for the Earth return maneuver. Given these assumptions, Figure 16 shows the Earth launch mass for a single target sample return. In all cases, the ISPP reduced this mass by at least 4000 kg.

For the multiple target option, the only case studied would visit two of the Galilean satellites, one of which would be Europa and the other being either Ganymede or Callisto. Three possible methods of conducting this mission were considered. The first case would use one vehicle which would visit the targets sequentially as was discussed for the multi-target lander. The second would use two landers which would be targeted for Europa and one of the other targets. The Europa lander would use the LOX/hydrogen system (with the previously mentioned modification) and would be the vehicle which would eventually return to Earth. The second lander would use a LOX/methane ISPP system to generate enough propellant to place the sample in a parking orbit around its target. The Europa ascent vehicle would rendezvous with the second vehicle in its parking orbit and the sample would be transferred. Return to Earth would then be accomplished as discussed above. The final option would be identical to the second with one exception. At departure the rendezvous would take place during a hyperbolic flyby of either Ganymede or Callisto (whichever happened to be the second target) during the orbit pumping phase of the ERV's departure.

Of these three options, the last was found to be the most mass efficient. Assuming that Callisto is the second target, then the estimated Earth launch mass is 10,000 kilograms which is less than that for the single sample return using conventional propellants (Figure 16).

In summary, this section has looked at ways in which the exploration of the Galilean satellites might be enhanced by the use of ISPP. Two types of missions, a one-way lander and a sample return, were identified as potentially benefitting the most from this technology. Each of these mission options was further subdivided into single and multiple target modes. Of the options considered, only the one-way lander which would visit more than one target was found not to gain in performance when compared to conventional systems. Visiting multiple sites on one target by the one-way lander as well as single and multiple target sample returns were found to have significant improvements in

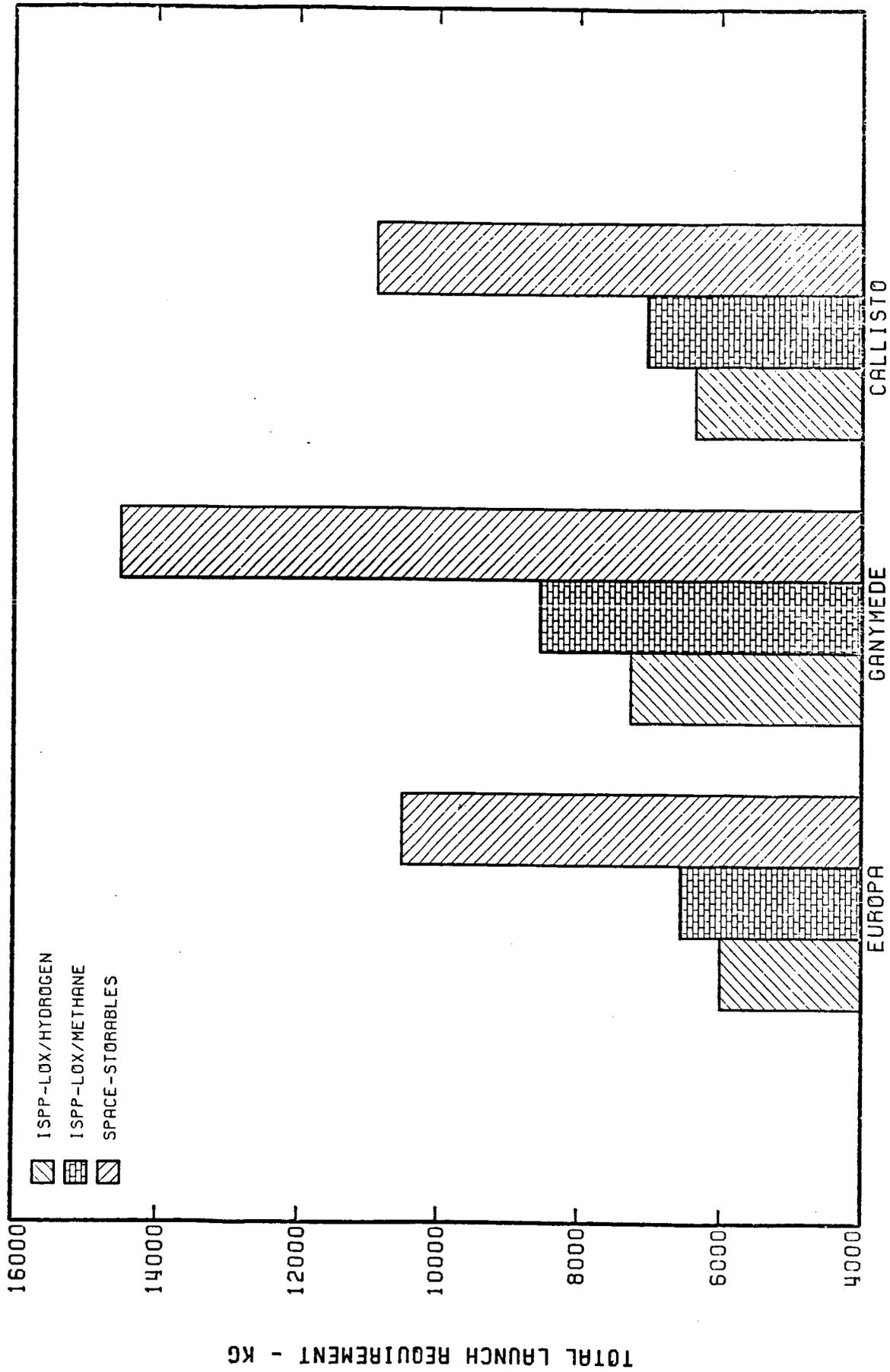


FIGURE 16. INJECTED MASS REQUIREMENTS FOR GALILEAN SATELLITE SAMPLE RETURN

mass performance. It was even found that the Earth launch mass requirement for a dual sample return would be less than that for a single sample return accomplished by conventional means.

6. ADDITIONAL MISSION APPLICATIONS

This section will briefly discuss other applications of the ISPP system for missions ranging throughout the solar system. At this point, no analytical work has been completed for any of these missions, but most follow logically from the uses described in earlier sections.

The atmosphere of Venus can be considered similar to that of Mars in that an ISPP system could be used to generate oxygen from the local carbon dioxide. The problems associated with operating at high temperatures and pressures at the Venusian surface can be avoided by basing the ISPP in a buoyant station. Operations carried out in the 20 to 30 kilometer altitude region would then encounter Earth-like atmospheric conditions. The propellant could be used to launch a number of small sounding rockets to explore the upper atmosphere of that planet. Investigations using a small aircraft as proposed for Mars would also be feasible. And, finally, the power subsystem could be used to give some controllable mobility to the buoyant station itself rather than allowing it to drift with the local atmospheric currents.

The Saturn system is in many respects similar to that of Jupiter. Thus the same types of missions described for the exploration of the Galilean satellites apply at Saturn as well. Three missions in particular have been identified as having great potential when augmented by an ISPP system. These include a Titan surface sample return, a multi-satellite tour, and a Saturn ring rover. The first two of these missions are duplicates of those proposed for Jupiter. The third mission, however, would establish a tight elliptical orbit about Saturn, with apsides bracketing the innermost and outermost ring radii. The orbit would be inclined out of the ring plane to avoid the collision hazard of a sustained fly-over mission to study ring phenomena. ISPP would be used to offset the high energy requirements for establishing such an orbit. Propellant production for

each of these missions would take place on one of the smaller satellites, where apparent feedstocks of suitable raw materials have been indicated by Voyager observations.

Missions to the far outer planets would follow the same pattern as that established for Jupiter and Saturn. Multiple satellite surface investigations and sample returns are considered good prospects for augmentation by ISPP. Upper atmospheric skimmers may also benefit by using ISPP-generated propellants to lower the periapse radius to the proper altitude. This would allow extensive aeronomy experiments to be conducted at these gas giants. Exploration of the surface of Pluto and its moon will become feasible by using an ISPP supported hopper. Data on the raw materials available at these outer planets is incomplete at present but should be improved by observations made by Voyager and the Space Telescope.

Small bodies other than the single comet and asteroid mentioned in a previous section are worth investigating and may prove more suitable for ISPP application. Aside from sample return missions, ISPP may allow multiple targets to be investigated by a single spacecraft if the vehicle is refueled at each target.

The final mission opportunity to be discussed here is that of a manned mission to Mars. In this case, an unmanned ISPP vehicle would be sent to Mars in advance of the manned mission. The ISPP would then begin producing and storing propellants for use on the return leg of the flight. By producing all or part of the return propellant locally, it becomes possible to generate enough propellant to use a non-Hohmann return trajectory and thus shorten the total mission time. Relieving the manned vehicle of the need to carry return propellant to Mars may also allow a non-Hohmann outbound leg, further reducing the total mission time. In addition, the large amounts of power required to run the ISPP system would be available to the surface exploration party since most if not all of the return propellant would have been produced before their arrival. Water collection and O_2 production could be used to augment or restock life support systems.

7. CONCLUSIONS

In reviewing the discussions presented throughout this document, one gains a sense of the large number of applications in which In Situ Propellant Production provides a significant performance advantage over conventional methods. Missions ranging from local exploration to sample returns have been identified for targets in all parts of the solar system. The only requirement is that suitable raw materials be readily available. In fact, most of the major bodies and the more interesting small bodies have the desired feedstocks on their surface, within their atmosphere, or, in the case of the large planets, on their moons.

Of the various applications discussed, the most promising in a general sense appears to be for sample return missions. A considerable reduction in the initial launch mass when compared to conventional alternatives was found to be typical of virtually all missions of this type which were examined. In particular, the results for a Mars sample return have sparked the most interest for a near-term application. The Galilean satellite sample return results are also quite favorable and thus place this mission in a good position as a candidate for early implementation.

While the idea of using locally produced propellants for some of these missions has just recently come under serious consideration, the means for accomplishing it has not. The process of collecting and splitting the raw materials as well as storing and, if necessary, refrigerating the resulting propellants is an existing, well understood technology. There are three areas, however, which will require more extensive research and development before the entire system can be used. The first of these is the engine which will use the propellants produced by the ISPP system. A small rocket motor using either liquid methane/LOX or liquid hydrogen/LOX has never been built although the necessary technology has been applied to larger engines. The second area concerns the construction and use of multi-kilowatt RTG power systems. Methods for assembling that much nuclear material for a space system, as well as the means for conditioning and distributing it, must be found. Finally, a number of methods for separating the propellants from one another and from waste products have been proposed. But most exist only on paper or as laboratory demonstration models. More development work in this area is therefore required. As might be

expected with any new system, there are questions of reliability which arise, especially due to the extended operation times and the large amounts of raw materials which must be processed. For example, studies conducted for the Mars sample return mission have raised questions concerning the lifetime of the electrolysis cell and the compressor unit. The latter must generate a relatively high compression ratio due to the tenuous Martian atmosphere. In addition, possible difficulty exists with the small but continuous presence of contaminants, most notably dust, in the atmosphere which must be filtered out before delivering the carbon dioxide for processing. Work on these developmental issues in support of an endorsed early mission application would assure the implementation of ISPP technology. The enabling capability of ISPP would have a profound impact on the future of solar system exploration.

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