Some Aspects of Space Propulsion with Extraterrestrial Resources

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Extraterrestrial resources for space processing of chemicals, in general, and propellants, in particular, are explored quantitatively. It is seen that, for several candidate space mission scenarios, space processing of both space resources and Earth-carried resources can make decisive differences in the mission success for a given payload. To fix ideas and to demonstrate trends, the specific case of water splitting to extract oxygen, discard (or use without storage) the resulting hydrogen, and burn Earth-carried noncryogenic liquid fuel(s) in a simple rocket motor, designed for periodic thrusting, is treated in some detail. Experimental hardware is assembled and demonstrated to perform adequately, besides showing compactness of the space-packaged "capsule" module that is self-contained. Building upon previous studies, the concept of in situ propellant production (ISPP) is reexamined in light of more recent energy and materials technologies. Missions to comets and Mars Sample Return are mentioned as candidate scenarios. The mission duration, reliability-repairability of hardware, resource availability in low Earth orbit (LEO), and the thrust requirements are considered in turn. It is seen that space storage of hydrogen for extended durations (5–10 years) involves problems that require detailed studies, besides involving many presently unanswered issues. A study of the energy option in LEO and in deep space is developed in simple terms. The different solar, radioisotope, and nuclear power sources are mentioned. Storage and handling of raw and processed chemicals are considered.

Introduction

IT has been well recognized for a long time[1] that utilizing extraterrestrial resources can significantly improve payload ratios and endurance of space missions. In the specific context of propulsion and propellants, the concept of in situ propellant production (ISPP) has been examined in some detail.[2-5] The basic idea is to use available local resources to manufacture specific propellants on site rather than transport them (and the necessarily associated gear) from Earth. This concept has been shown to be particularly applicable to missions (unmanned or manned) to Mars.[6,7] It was also shown that the propellant itself becomes one aspect of a multifaceted scenario, and other considerations such as energy, power, materials, containment, refrigeration, etc., are all inextricably interwoven into the general fabric of space missions. Instead of analyzing specific missions and evaluating the options available, it was deemed desirable at this stage to take a broader view of the entire issue of extraterrestrial resource utilization as it affects propulsion. To render the view a little more specific, three missions are mentioned.

It is the objective of this paper to examine, from a technology point of view, the various options, and combinations thereof, available for near- and intermediate-term space missions of a peaceful nature. It can be readily appreciated that, depending upon the mission, different technologies and combinations may appear superior to others. This is not to say that extraterrestrial processing (ETP) is highly mission specific. There do seem to exist some broad classes of energy and material options that can be treated on a common basis. The area of space exploration, space habitability, space processing, space manufacturing, and space resource utilization is growing rapidly. Current technologies could change dramatically in the near future. The highly nebulous nature of the subject and the paper length limitations preclude a detailed technical analysis of each technology. The basic facets are identified to be: processible materials, energy (and power) sources, structural issues, and material/energy explulsion or rejection. Different options are discussed in some detail.

In the cases of planetary (lander) explorations and sample retrieval, use of local materials seems logical, provided the materials are reasonably well characterized already. The obvious examples include the lunar materials. The martian soil and atmosphere have also been characterized, although to a lesser degree, and can be included in the same class with minor reservations. When other missions are considered, the uncertainties in the local materials make it imperative that we rely on proven (well characterized) materials to avoid mission risks. Here, typical examples are comet rendezvous or sample retrieval and return. While it can be argued that asteroid and comet materials are frequently well characterized from some points of view, utilization of such materials for reliable mission dependency must certainly await further studies. The risks involved do not seem to justify the approach. In such cases, Earth-carried material resources seem to offer the most reliable overall mission performance;

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however, these material resources need not be dedicated resources—there lies the newer approach of ETP. Several materials are available naturally in any mission and can be utilized in "unconventional" ways. An interesting example was discussed by Davis[8] and Taylor.[9] The shuttle external tank, which is jettisoned to Earth after attaining 98% of orbital velocity, has been identified as a valuable resource that could be better utilized (carried into orbit) instead. The empty tank could serve as an additonal cargo bay, experimental laboratory, and similar "vehicle" needs. A different approach could make the structural materials available in LEO for other uses. For example, they could be processed (with energy) to serve as a propellant ingredient. These would, of course, require several fairly major modifications in the way the current STS (space shuttle) is designed and operated. The modifications are not unthinkable, besides probably providing an economical next step in lieu of a major developmental program. An extreme generalization of this material utilization approach may lead to technologically feasible "cannibalization" of the space "junk" in LEO and on some of the planets (Mars, Moon, Venus). Collection of naturally occurring gases and plasmas in LEO is also seen to provide significant quantities of material, despite the extremely rarefied corridor, because of the extremely large volumes involved. All of these represent no fundamentally new technology but only novel ways of processing extraterrestrially.

To fix ideas and show immediate applicability, use of the space station and two other specific missions are mentioned in some detail. One is the much publicized Mars sample return (MSR) and the other is the comet sample retrieval (CSR). In the former, the main propulsion system is considered; and in the latter, only the secondary propulsion need (attitude control) and the retrothrust module are considered. In the latter case, the preliminary design and fabrication of a "breadboard" hardware is also completed and discussed. It is emphasized that various options are available within the general class of extraterrestrial processing. Specific missions could draw upon different combinations of these basic elements. The options are so varied that the tradeoffs must be left to the users.

Missions, Materials, Energy, and Processing

Missions

We are particularly fortunate to find ourselves in the wake of an extensive set of guidelines established for planetary exploration. The Solar System Exploration Committee of the NASA Advisory Council spent a lot of effort and careful thought developing a "core program" through the year 2000.[10] In it, not only do we have the benefit of extensive and intensive experience of a wide range of experts, but also the fundamental details of the recommended missions. The initial core missions are

	Launch	Data Returned
Venus Radar Mapper	1988	1988–89
Mars Geoscience/Climatology Orbiter	1990	1990–92
Comet Rendezvous/Asteroid Flyby	1990–92	1994–2000
Titan Probe/Radar Mapper	1988–92	1995–97

Among the candidate subsequent missions mentioned are the Mars Surface Probe (MSR) and Comet Atomized Sample Return (CASR). It would be prudent to examine the mission details within the framework of these committee recommendations. An important addition to the resources available for these missions will be the orbiting permanent space station currently in the preliminary stages of design.[11] Several of the mission (execution) features will be influenced by the space station's availability. In general, the primary features of the missions include low gravitational forces, low/high temperature at equilibrium, vacuum, and radiation consisting not only of the optical and IR but also charged particles, x rays, and γ rays (commonly grouped under plasma or cosmic radiation). These need careful attention in specific light of the mission duration.

The microgravity necessitates alternative technologies for separation of species that would terrestrially be accomplished by density gradients, for example. The technology of microsieves is fairly well developed and is readily applied in space. The state-of-the-art technology promises energy-efficient membrane separation almost tailor made for specific applications. Electrophoresis is a powerful tool for zero gravity separation.[12] In the specific context of propellants and propulsion needs, it is seen that separation of oxygen from water is a necessary operation. For this gas separation from a liquid, it would appear that a simple membrane separator is admirably suited, as was demonstrated recently.[13]

Another aspect of microgravity needs to be studied further. This is the general dynamic and thermal behavior of large structures. At the current time, the problems are beginning to be addressed.[14] In addition, the nature of damping of mechanical oscillations needs to be studied and understood. Here, a novel technique using piezoelectricity may offer hope.[15]

The natural tendency for spherical symmetry to be attained in microgravity may have strong implications to combustion zones. However, this tendency may have greater relevance to the first safety of the craft than to the propulsion problem at hand.

The temperature in space has some interesting aspects. In the near-Earth region (1–2 AU), the equilibrium temperature reached by an object is such that cryogenic liquid will certainly need refrigeration for storage at moderate pressures. Also, the temperatures attained enable the heat rejection (to the dark side) to be fairly easily achieved. In addition, solar concentrations can be called upon to attain fairly high (>1000 K) temperatures with modest collector areas. By far the most important implication of the temperatures attained with equilibrium with this radiation has to do with storage over extended time periods of liquids such as hydrogen.

Missions to comets and asteroids have to consider the possibility of craft impact and damage by particles. The probability of impact is much higher in such missions compared with a Venus probe, for example. Full autonomy and quick response times are obviously needed to avoid collisions. (Commands from Earth or LEO take too long for receipt and execution.) A low projection area in design may have to be inherently incorporated into all such missions. Such a projected area limit design could influence the entire system. For the SSEC-recommended core missions, there do not appear to be any fundamentally new problems that need a technological breakthrough for mission success.

For deep space (outer planets) missions, the issues change: solar energy is not viable, mission durations increase to decades, and temperatures drop below 100 K for extended durations. These issues are considered in the energy and materials subsections.

Materials

Conventional propulsion requires the expulsion of mass (except in photon propulsive devices. The concept of "recycled" mass[16] for generalized propulsion needs further evaluation for wide acceptance). The higher the exit velocity per mass (specific impulse), the greater the system economy. Thus, the energy of the exhaust stream enters the picture. Deferring the energy source to the next subsection, we can now examine a choice of materials available as propellants. The two generic classes are Earth-carried and space-derived propellants. Oxidizer fuel ratios for maximum specific impulse typically are in the range of 2–6 on a mass basis. Thus, a large fraction of any propellant is the oxidizer and not the fuel. Therefore attempts to make fuel available for space missions would really be addressing a question of lesser importance. In this connection, it is worth noting that practically any reasonable material could be used as a fuel, provided the enthalpy of formation (Δh_f^0) is not too large (negative). Most metals (by definition) have a $\Delta h_f^0 = 0$ and would form good fuels. Many of the modern space-qualified materials generically called plastics (hydrocarbon or others) also qualify as fuels. In addition, "atmospheric" data gathered above the Earth show that at 300 km altitude (LEO), several useful constituents are available (Table 1) in small concentrations. With a reasonable collector of 30 m diam, it is easily seen that a toroidal volume of 3×10^6 m^3 is traversed per orbit around the Earth. The potential quantity of several gases collected is shown in Table 2. Several kilograms of atomic oxygen can thus be collected per day. The dissociation equilibrium constant has a value that will recombine the O into O_2 after collection (because of the higher pressure); this recombination is also a source of energy. Strictly speaking, this is a source of solar energy since the O_2 dissociation is driven by solar energy in the first place. The solar wind flux of protons show 4×10^8 protons/cm^2 s at 1 AU. Using a stationary 30-m-diameter disk, the mass collected is 5×10^{-7} kg/day. Such quantities do not appear significant enough to provide the usual propulsion needs. Details of these LEO class materials are discussed in Refs. 17-20.

The shuttle external tank has an empty (control) weight of 35,000 kg. If means can be found (see later in the energy subsection) for melting, cutting, or processing the aluminum portion of the fuel source, approximately 480,000 Btu of energy can be obtained, assuming all Al can go to Al_2O_3. Used as a propellant, along with some hydrogen, a specific impulse of 250 s can be envisioned. This means that a typical mission such as a comet probe ($\Delta v = 5$ km/s) can be achieved with a significant payload of 18,000 kg. Generalizing this concept to the space junk in LEO, similar numbers seem applicable. There is surely an enormous potential for propellants and propulsion besides cleaning up the LEO corridor; however, it is recognized that careful monitoring and compliance with space law are necessary as nontechnical issues.

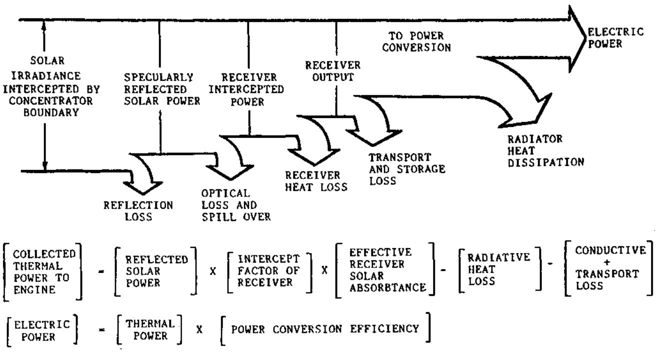

Fig. 1 Solar thermal energy balance relationship.[23]

Planetary materials are available in large amounts. Only those well characterized can be relied upon in the first missions. Some of these are discussed in "ETP for Mars Sample Return" in relation to one specific example MSR. As more data become available, more economy is foreseen for future propulsion. Thus, initial missions not only serve the scientific community, but also render future missions more attractive.

The large quantities of oxidizer needed in propulsion pose the key question of the means of transporting the oxidizer. It is felt that one of the best sources of this oxidizer is water, having 16 parts in 18 on a basis. Transportation and extended storage seem to pose no major problems. It is also foreseen that water will be a key material on board the space station and would thus naturally enter the system dynamics, operations, and economics. In addition, several asteroids and comets are presumably rich in water[21] and thus a water-based propulsion technology can easily find itself in harmony with missions to such bodies. The extraction of oxygen from water needs energy, and this is the topic of the next subsection.

Energy and Processing

Ideas for space processing with solar energy are not new. Fairly detailed quantitative analyses for the production of hydrogen fuel and carbon (for structural materials) starting with Earth-carried methane were discussed 10 years ago.[22] While the fundamental concept is very appealing, some details are not clear in that reference and similar studies. For example, storage of LH_2 in LEO presents problems, and extended storage (several years) has not been demonstated yet. In addition, the solar radiation absorption with carbon black may not be a homogeneous transfer considering the large outer surface absorption. The times for thermal equilibrium under microgravity also need to be studied further. Nevertheless, the availabiliy of "free" solar energy has certainly stimulated a lot of studies to reap this source in LEO and even in outer space missions up to 6 AU. For this reason, it would seem appropriate to address the issue at a fundamental level, leaving the specifics to later studies.

Available solar flux varies from 10^4 w/m^2 at Mercury (0.39 AU), to $1.3\text{-}1.4 \times 10^3$ w/m^2 at Earth (1 AU), down to 1 w/m^2 at Pluto (30 AU). Typical space missions are predicted to require power levels of 500–2000 W_e for the immediate future. (This number is for planetary, asteroid, and comet missions. By comparison, the space station is expected to need 25 kW_e as the base line.) Thus, it would seem logical to consider solar energy up to no more than 6 AU (50 to 60 w/m^2) if excessive collection areas are to be avoided. Solar photovoltaic electricity generation is inherently low in efficiency—13% seems a reasonable number for silicon and no more than 20% even for the advanced GaAs. If some of the ultimate use can be accomplished by nonelectrical means (although some electrical power is surely needed), the solar dynamic systems offer an interesting alternative. The major subsystems include concentrators/collectors, receivers/storage, heat engines, and radiators. An energy balance diagram from Roschke and Wen[23] is reproduced here as Fig. 1. Idealized system efficiency

Table 1 Mass of constitutents per cubic meter; 300 km altitude above Earth

Con- stituents	Atomic weight	Number density Min, m^{-3}	Number density Max, m^{-3}	Mass/volume Min, g/m^3	Mass/volume Max, g/m^3
H	1.00797	7×10^{12}	3×10^{10}	1.2×10^{-11}	5.0×10^{-14}
He	4.003	6×10^{12}	7×10^{12}	4.0×10^{-11}	4.7×10^{-11}
O	15.9994	1×10^{14}	2×10^{15}	2.7×10^{-0}	5.3×10^{-8}
O_2	31.9880	8×10^{10}	7×10^{13}	4.2×10^{-12}	3.7×10^{-9}
N_2	28.014	4×10^{12}	1×10^{15}	1.9×10^{-10}	4.7×10^{-8}
Ar	39.948	–	7×10^{11}	–	3.3×10^{-11}

Table 2 Mass of constituents per orbit; 30 m (collector diameter)

Con- stituents	Mass/m^3 Min, g/m^3	Mass/m^3 Max, g/m^3	Mass/orbit Min, g	Mass/orbit Max, g
H	1.2×10^{-11}	5.0×10^{-14}	0.36	1.5×10^{-3}
He	4.0×10^{-11}	4.7×10^{-11}	1.2	1.4
O	2.7×10^{-9}	5.3×10^{-8}	80	1570
O_2	4.2×10^{-12}	3.7×10^{-9}	0.12	110
N_2	1.9×10^{-10}	4.7×10^{-8}	5.6	1290
Ar	–	3.3×10^{-11}	–	0.98

is shown in Fig. 2. The baseline radiator temperature is assumed to be 300 K. The engine inlet temperature is 30 K lower than the receiver temperature. The collector is in sun 65% of the time and in shadow the rest of the time.

It is seen that the idealized system performance is governed by the matching of the concentrator quality and operating temperature. At 700 K, the system $\eta = 39\%$ and rises to 56% at 1800 K. It is quite sensitive to slope error. For example, a 10 m rad slope error concentrator operating at 1400 K will produce zero net power! The needed concentrator area is also a function of the radiator area, receiver temperature, and the slope error (Fig. 3). It is easily appreciated that the overall electrical power-generation efficiency can exceed that of photovoltaic generators. The area of the concentrator needed to generate power needs to be studied. Detailed numerical studies and tradeoffs[24] have shown that the optical quality of the concentrator and the system temperature levels are major drivers; the electrical power output per unit of concentrator area optimizes at increasingly higher values of receiver temperature as the concentrator surface quality increases. This effect is more pronounced at larger AU. This is quantitatively shown in Fig. 4. Translation of this to mass of hardware needed is at an especially critical juncture now (1985). The classical glass/metal reflectors are facing competition from such reflectors as graphite fiber honeycomb core reflectors[25] with a factor 100 decrease in the mass/unit area. Also, the stretched membrane (metallized polymer) technology promises further reductions,[26] especially with a taut graphite fishnet backing.

The authors believe that a combined photovoltaic/thermal dynamic system may offer the best technology. A high-temperature receiver can also be operated from the "waste heat" of a PV collector. These concepts need further study.

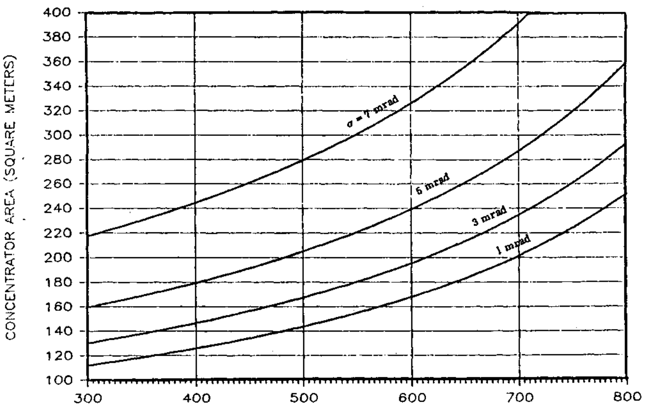

Fig. 2 System efficiency [$e = 1.0$, $C = 1.0$, $Tc = 300$ K (Ref. 23)].

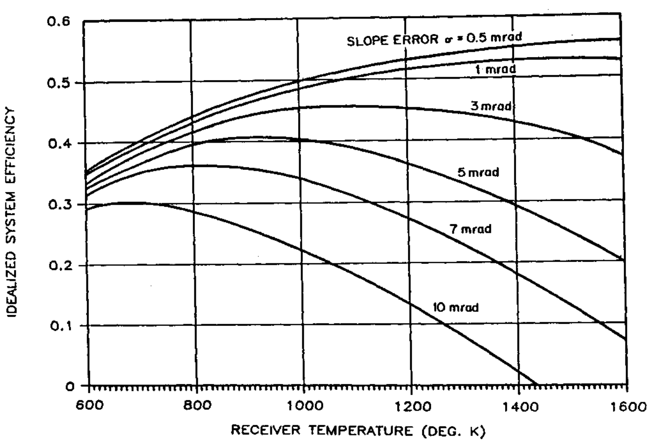

Fig. 3 Concentrator requirement [receiver temperature = 1200 K, $e = 0.5$ (Ref. 23)].

If the utilization of the energy is indeed to manufacture propellants only, some interesting possibilities exist. Space limitations restrict us to including only some basic concepts in this paper.

The option is to use solar thermal energy and eliminate the thermal energy losses in the conversion to electric power; this has the effect of decreasing the size and mass of the solar energy collection system and providing the additional hydrogen required for the fuel-oxidizer ratio needed for conventional oxygen-hydrogen engines. A combination of solar thermal with solar thermal electric provides several interesting propellant production options. There are several reactions which could be used to convert solid or liquid mass lifted from Earth into chemical propellants. In selecting these, one has to consider the temperature of the reaction, the amount of thermal energy needed per kilogram of reactant, the use of cyclic thermal energy as produced in LEO, and the management of liquid phases in near-zero gravity. These reactions have been studied extensively or used in Earth-based industrial chemical processes.

The thermal production of hydrogen from water would require a reactor temperature that is beyond the current capability of solar thermal technology. However, the thermal production of hydrogen can be obtained by reacting water (steam) with solid carbon:

$$C\,(\text{solid}) + 2\,H_2O\,(\text{gas}) \rightarrow CO_2\,(\text{gas}) + 2\,H_2\,(\text{gas})$$

This reaction would proceed with the carbon heated at temperatures below 1000 K and require 19 kcal/g-mole of thermal energy for the heat of reaction. For each kilogram of hydrogen produced, 12 kg of mass would have to be lifted from Earth. Separation of the carbon dioxide from the hydrogen could be accomplished by absorption in a water solution of monoethanolamine or by liquefaction. The carbon dioxide could be used in a resistojet for auxiliary propulsion needs. The carbon dioxide would also provide the carbon

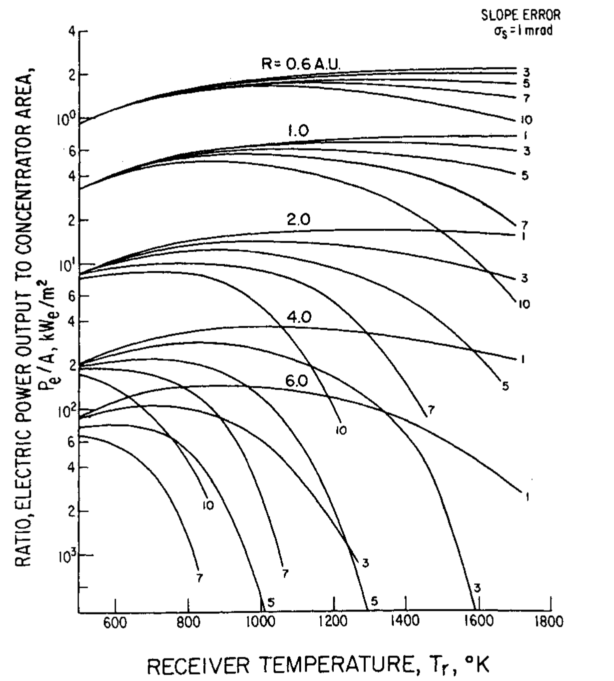

Fig. 4 Electric power generated per unit concentrator area for various distances from the sun.[24]

Table 3 Key technologies in evolution of ISPP

Material concentration
Purification/decontamination
Chemical processing/separation
Distribution and storage
Instrumentation and control
Autonomy

Table 4 Mars atmosphere constituents

Constituent	Percent
CO_2	95
N_2	2.76
Ar	1.6
H_2O	0.03

Table 5 Manufactured propellant and required feedstock

Propellant	Feedstock
O_2	CO_2
O_2/CO	CO_2
O_2/CH_4	CO_2,H_2O
O_2/H_2	H_2O

source needed for space plant photolysis where high concentrations of carbon dioxide in the growing atmosphere would be desirable.

An alternative reaction would be to increase the reaction temperature to 1400 K so that carbon monoxide and hydrogen are produced. Earth-based solar thermal systems have demonstrated received temperatures near this temperature:

$$\mathrm{C\,(solid) + H_2O\,(gas) \rightarrow CO\,(gas) + H_2\,(gas)}$$

This reaction would require 15 kg of mass lifted from Earth for each kilogram of hydrogen produced and 38.9 kcal/g-mole of thermal energy. Carbon monoxide can easily be separated from hydrogen by liquefaction, which would be the process of choice; alternatively, for a gas engine, the mixture of carbon monoxide and hydrogen could be used. The reaction products could also be used as propellants for a carbon monoxide-oxygen engine to deliver a specific impulse of about 250 s. This is significantly below the specific impulse of 400 s for a oxygen-hydrogen engine, but it compares favorably with hydrazine thrusters; however, carobn monoxide-oxygen engines have not been developed.

Because of the energy required for liquefaction of hydrogen and the difficulty of storing liquid hydrogen in Earth orbit, additional space processing could convert the hydrogen to methane or methanol; however, there is little benefit to be gained because these liquids at the normal O/F ratio can be transported rather easily from Earth. One potential process is the production of ammonia from hydrogen and nitrogen from the upper atmosphere that could be collected in low Earth orbit; ammonia synthesis is a well-developed industrial technology.

The total energy required for these processes is significantly below the 136 kcal/mole that is required for electrolytic conversion of water to oxygen and hydrogen.

Three Specific Examples

Immediate Applications: The Space Station Scenario

While it is not the purpose of this section to develop advanced ISPP technologies, it should be noted that the space station will reside in an environment where resources are available. That is, oxygen and hydrogen are present at concentrations where it would be possible to collect and concentrate them.[27] The source for space station aerodynamic drag is also a source for dilute quantities of O, N_2, O_2, and H_2. These substances could be concentrated using a coated conical collector attached to the space station and then removed with a turbo pump.

Much of the technology required to develop a space station-based fuel processing system will evolve naturally from operation of the space shuttle and the first-generation space station. To assist in the discussion of that evolution in more detail, important ISPP technologies are summarized in Table 3. The technology requirements were discussed briefly and, subsequently, an assessment of base line space station operations was made in terms of how they address ISPP. Space limitations here make it imperative to merely refer the reader to a recent paper.[28]

Table 6 Propellant combination performance

Propellant	Specific impulse
O_2/CO	2550 m/s (260 s)
O_2/CH_4	3530 m/s (360 s)
O_2/H_2	4320 m/s (440 s)

ETP for Mars Sample Return

Mission Considerations

ISPP shows promise for a variety of missions, including round-trip missions to the surface of planetary bodies. Studies have been conducted involving its use on Mars and the Galilean satellites of Jupiter.[29-31] The most thoroughly studied missions involve Mars, and indeed, round-trip missions to this body probably are the most likely near-term applications of ISPP. In this section, we will evaluate round-trip missions to the surface of Mars. While the primary focus will be upon manned missions, most of the conclusions apply equally well to unmanned sample return missions. The differences lie primarily in the size of the system and the greater complexity permissible in a man-deployed and man-tended system.

As discussed elsewhere in this paper, three ISPP concepts appear viable: 1) production of oxygen only to use with fuel brought from Earth, 2) production of oxygen and carbon monoxide (CO) fuel, and 3) production of oxygen and methane (CH_4) fuel. While it may be argued that oxygen/hydrogen is also an option, the difficulty in liquifying and storing hydrogen makes this approach seem impractical for early missions. However, oxygen/hydrogen performance will be shown to provide a reference.

In terms of martian resources required, the first two options seem most desirable. The atmosphere of Mars is predominantly carbon dioxide (CO_2), as shown in Table 4. Carbon dioxide can be electrochemically decomposed into oxygen and carbon monoxide, which can then be separated as discussed elsewhere. Thus, the primary resource for these two options can be obtained by simply compressing the atmosphere. The third option, synthesis of oxygen and methane, requires water as well as CO_2 (Table 5). Water is only known to exist in the permanent polar cap. Even if the landing site were in the polar region (not desirable for a variety of mission reasons), the "mining" of low-temperature ice mingled with sand or similar material could be very difficult. The water in the atmosphere is inadequate to provide the quantities needed for propellant manufacture.

The considerations in the previous paragraph may push the choice toward options 1 or 2, at least for early missions. If reasonably available water is discovered by these early missions or by a precursor unmanned mission, the choice might well shift toward option 3, especially if a permanent facility capable of maintaining mines or wells is established.

In order to better understand the impact of choosing a propellant combination, it is instructive to compare their

Table 7 Mass ratio requirements

Mission	Stages	Mass ratio O_2/CO	O_2/CH_4	O_2/H_2
Low Mars orbit	1	6.07	3.68	2.90
$v = 4.6$ km/s	2	2.46	–	–
Earth return	1	13.3	6.49	4.61
$v = 6.6$ km/s	2	3.64	2.55	–

Table 8 Vehicle liftoff mass (payload = 10 metric tons)

Mission	Vehicle mass (metric tons) O_2/CO	O_2/CH_4
Low Mars orbit	87.5 (73 for 2 stages)	44
Earth return	178 (2 stage)	98

performances. In this comparison, the propellant fuel imported from Earth for option 1 is CH_4. The reasons for this choice are several: 1) the oxidizer-to-fuel mass ratio is high (about 4:1), thus maximizing effectiveness of ISPP; 2) performance is good; 3) liquid temperature is compatible with O_2; and 4) it is a good refrigerant. This choice has the serendipitous advantage of reducing the recombinations to be dealt with here.

Table 6 presents the performance of the propellant combinations under discussion, along with comparative data for O_2/H_2. Note that the latter offers a substantial performance advantage but is probably impractical because of the uncertainty in obtaining water and the storage problems mentioned earlier. It will be noted that the O_2/CH_4 combination has almost a 20% penalty in performance. This may be slightly offset by the greater density of the propellants. By comparison with either of these, the O_2/CO combination does not look highly impressive. As observed previously, its major virtue is that it can be generated entirely from the one thoroughly known and readily available martian resource.

To properly evaluate the options, it is necessary to look at applications to specific missions. We will consider two: delivery of a payload from the surface of Mars to low circular orbit around Mars, and delivery to an Earth-bound interplanetary trajectory. Typical incremental velocities (Δv) of these two missions, making allowance for drag and gravitational losses, are 4.6 and 6.6 km/s, respectively. (The latter is based upon a moderate but not unreasonable velocity at infinity of 2.4 km/s.)

Table 7 compares the mass ratio (defined as liftoff mass divided by mass remaining at attainment of the final velocity) for the propulsion optimal for both missions. Single stages with mass ratio greater than about 6 lead to excessively large stages. For these cases, a two-stage vehicle is calculated. The structural factor ϵ of 0.04–0.05 is assumed as reasonable, considering the state of the art.

It will be noted that the O_2/H_2 combination can do either mission with one stage, while the O_2/CH_4 combination can perform the Mars orbit mission handily with one stage but would benefit from staging for the escape mission. O_2/CO demands staging for the escape mission and would benefit from it for the orbital mission.

We will now consider the actual vehicle sizes required to perform the mission. In each case, the goal will be to deliver a payload of 10 metric tons (22,000 lb) into low Mars orbit or an Earth return trajectory for rendezvous with a return vessel. Henceforward, only O_2/CO and O_2/CH_4 will be considered. Table 8 presents the results. Even when staging is used for O_2/CO, the liftoff mass is much higher than that of the single-stage O_2/CH_4 vehicle. This clearly indicates the superiority of this combination and, if water is readily

Table 9 Vehicle empty mass and propellant mass including payload

Mission	Mass (metric tons)	
Vehicle dry		
Propellant	14.5	12
Earth return	73	32
Vehicle dry	49	15
Propellant	129	83

Table 10 ISPP system masses

Mission	ISPP Mass (kg) O_2/CO	O_2
Low Mars orbit	1480	1150
Earth return	2600	2900

Table 11 Useful mass delivered to surface

Mission	Mass (metric tons) O_2/CO_2[a]	O_2/CH_4[b]
Low Mars orbit	16	20
Earth return	52	35

[a]Vehicle dry mass plus ISPP system. [b]Vehicle dry mass, ISSP system, return fuel (CH_4).

available, there is no doubt that it would be used. If methane must be brought from Earth, then the real comparison is mass delivered to Mars.

In order to assess mass delivered to Mars, we must evaluate ISPP system mass. Allowing for a manufacturing period of one Earth year, we can roughly estimate ISPP system masses for the four cases of Tables 8 and 9 to be the values shown in Table 10. Note that both O_2 and CO are produced for that case while only oxygen is being produced for the O_2/CH_4 case. These values are rough extrapolations based upon data in Ash et al.[31]

Using the estimated ISPP mass, the vehicle dry mass, and fuel mass for the O_2/CH_4 case, we can now estimate a useful mass to be delivered to the surface of Mars. This appears in Table 11. It can be seen that the mass penalty for carrying return fuel renders O_2/CH_4 less attractive than O_2/CO for the low Mars orbit case. For the case of escape to Earth return, however, the effect of the superior performance of O_2/CH_4 makes itself felt and the mass for that case is substantially lower. Since mass delivered to Mars has an amplification factor of over 3 when its effect on Earth orbit departure mass is considered, this may be significant.

It should be noted that power requirements for these various cases fall in the 20–50 kw_e range. This mass is not included in the calculations discussed above. Since the entire base would require electrical power, it is probable that a 100-kw_e nuclear reactor system, perhaps based upon a design similar to SP-100, would be used to provide both base and ISPP power. The power vs mass curve for these units is quite flat in the range of interest, and thus the variance in power levels will probably not impact total mass significantly.

Comet Rendezvous/Sample Retrieval

As a third example, the specific technologies involved in ETP for sample retrieval from the tail of a comet is considered. The purpose of this section is to simply show some hardware design work that has been performed—not necessarily the optimal. Consistent with the philosophy of specifics in this section, Comet Kopff was chosen as the candidate. The fact that ground-based viewing will be excellent in 1996 makes this mission an especially interesting test case for

Table 12 Mass comparison of water electrolysis vs O_2

	Electrolysis	O_2
M_S (final mass)	197.34	22.08
M_A (initial mass)	563.83	65.91
M_{O2} tank	56.19	6.72
$M_{JP\text{-}4}$ tank	11.11	1.33
M_{helium}	0.02	0.002
$M_{helium\ tank}$	0.036	0.004
$0.08\,M_A$	45.11	5.27
Payload	8	8
M_{H_2O} tank	28.7	-
M_{H_2}	31.57	-
$M_{electrolysis\ equipment}$	15	-

the propulsion system. The incremental velocity needed from Earth is $\Delta v = 2.3$ km/s, which would not tax the propulsion system too severely. The launch year would be 1991 and the flight time 4.8 years; thus we also have a technology issue one step removed from the previous two examples—namely, the mission duration of several years ($\simeq 10$). However, the closest approach to Earth of 1.57 AU (in 1996) means that the probe spacecraft will always be within the "sunbelt" (in space solar vocabulary). Whether it makes overall mission sense to use the available solar energy to process propellants is an open question. Two separate approaches seem feasible. One considers Earth-carried gaseous oxygen in a modern high-pressure container and claims mass advantages; the other carries the needed oxidizer in the form of water, splits the water during travel, and stores the oxygen gaseously in a low-pressure balloon. Both the designs are mentioned quantitatively.

The scenario is as follows. A scoop/probe (initial payload) weighing no more than 5 kg is launched from LEO. The fully autonomous and self-contained craft encounters Kopff's tail approximately 4.8 years after launch. Any midcourse trajectory correction or maneuvers (to avoid meteors, for example) will be handled by on-board propulsion. After collecting approximately 5 kg of the comet's tail as a sample, a short-duration high thrust will inject the craft into an Earth-capture trajectory. The JP-4/O_2 rocket provides the retrothrust and any midcourse trajectory correction thrusts. The craft (with comet sample) is retrieved at LEO by the shuttle (or the space station). Thus, there exist power needs for switching and related robot-type operations, low-thrust propulsion needs intermittently for nearly 10 years, and at least one high-thrust propulsion need at the end of sample collection. The craft has to survive in hard vacuum and space radiation during this time, also.

In the first design, called Design 1, a photovoltaic power source is used. Based on proven reliability, silicon cells were chosen, despite poor efficiency. A conservative estimate led to 80 w_e/kg. Chemical propulsion was selected. The initial thrust is delivered by a solid rocket such as the Star48, after which the case is jettisoned. The on-board propulsion utilizes a JP-4 oxygen rocket capable of 10–20 lbf thrusts. The Earth-storable JP-4 (melting point approximately 216 K) is carried passively in a lightweight container. The provision exists for intechanging with propane (melting point 89 K) if the design should change in the future, depending on the equilibrium temperature of the craft in space. A specific impulse of 270 s is calculated, with an expansion ratio of 25:1. The oxidizer (oxygen) is carried passively as water. Electrolysis in-route generates oxygen which is stored in a large balloon coated with aluminum for thermal control. The hydrogen is discarded because of storage problems mentioned elsewhere.

The key novelty of Design 1 is electrolysis, which is used to generate oxygen, and is described below. The distinction from an excellent earlier study[32] is that hydrogen use is avoided.

Design I—Electrolysis

The total oxygen requirement for the Kopff mission (20 burns, JP-4 fuel) has been determined to be about 37 kg, almost half the initial mass. Storage of this much gaseous or liquid oxygen for long periods, such as the proposed 5- to 10-year duration, presents considerable problems. Cryogenic storage requires heavy equipment and a large energy drain, at least within the orbit of Mars (past that point, passive refrigeration may be very useful). High-pressure gas storage means a substantial weight penalty for a reliable storage vessel, a high probability of leakage over the life of the mission, and a great risk to members of the shuttle transport crew. Our solution is to carry the oxygen in liquid water, which is very stable and occupies low volume. The water is then "split" by electrolysis during the mission as oxygen is required. The hydrogen left over is simply discarded.

Electrolysis is the process by which water is decomposed into hydrogen and oxygen by passing current through electrodes separated by a conducting aqueous solution, or electrolyte. Several systems have been developed for splitting water in space, that is under zero-gravity, no-atmosphere conditions. Most reseach in this area has been concentrated on manned missions (recovery of oxygen from wastewater aboard space stations), but the technology is still applicable to our smaller scale problem.

Liquid electrolysis[33]: Early electrolysis systems developed for space used an alkaline or acid solution (usually 25–30% KOH or H_2SO_4) as the electrolyte, not unlike the simple tank-type units still in use on Earth. Asbestos matrices or ion-exchange membranes prevented mixing of gas and liquid in zero gravity. Units of this type were fabricated and tested extensively; they showed promise but had considerable problems with electrode corrosion, leakage, and system stability over long periods. Also, since it is difficult to maintain electrolyte balance without maintenance, systems that used pure water were more desirable.

Palladium-silver electrode[33]: This type of cell is characterized by the use of a nonporous hydrogen diffusion cathode made from palladium-silver alloy tubing which has a closed end immersed in water. Hydrogen diffuses into the tube directly, and very high O_2 and H_2 purities result. A model of this type showed many improvements over earlier prototypes, most importantly, low weight (only 20% of previous weights) and power requirements. Still, the system involves a very caustic electrolyte that tends to poison the cathode over time.

Solid polymer electrolyte[34–36]: A solid polymer electrolyte (SPE), developed by General Electric, has been the focus of most recent testing and development where space oxygen systems are concerned. The electrolyte is a solid plastic sheet of perfluorinated sulfonic acid polymer which is similar to Teflon. Since hydrogen ions move through this membrane, there is no need for a conducting alkaline solution. Such a cell can be operated under various conditions of temperature and pressure, although a typical NASA "one-man" (0.907 kg O_2/day) module works best at about 94 C (200 F) and 2758 kPA (400 psig). SPE systems have been shown to be quite stable and reliable and, since very thin membranes and platinum/niobium electrodes are used (e.g., 0.1 mm), the cells are very compact and lightweight.

A related system, high-temperature solid electrolyte electrolysis,[34,37] is also under development for space. This cell uses an yttrium oxide-stabilized zirconium oxide electrolyte and platinum electrodes. Although a test cell of this type was shown to be very low weight (0.9 kg) and have few leakage problems, this system requires extremely high water temperatures (1200°C) and therefore presents feed and storage problems our little comet sample unit cannot overcome.

Testing: Although most testing of electrolysis systems has been concerned with relatively large-scale manned systems, we can make some predictions based on the "scaling down" of such systems. For example, an early prototype developed for NASA using liquid electrolyte produced 17.6 kg O_2/day with 960 W of power.[33] The unit weighed 84 kg, or 185 lb. This is far too large for our application; we need only 0.15 kg O_2/day for a firing every 3 months (20 burns in 5 years). The power

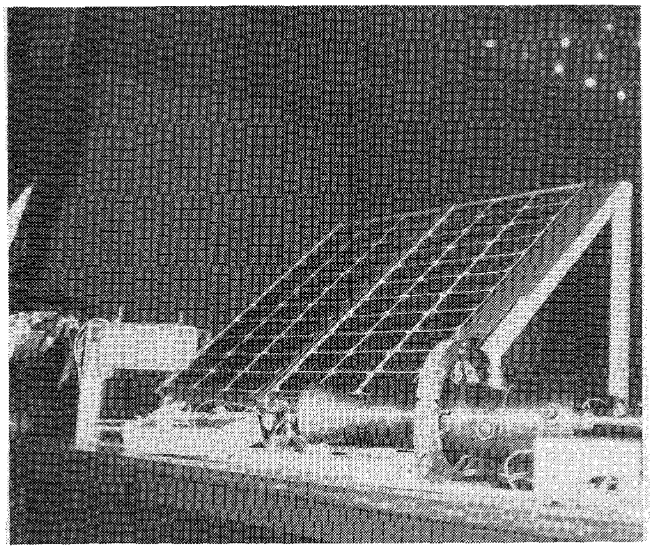

Fig. 5 Assembled hardware of comet proble: Design 1.

would scale down proportionally since O_2 production is directly related to current, so only about 10 W are required. We increase this to 50 W to allow for pumps, etc., whose current requirements may not decrease linearly with size. The weight would also be considerably reduced, especially with the SPE system, since the number of cells and pump size would be deceased. One SPE electrolyzer developed by base line produces about 0.3 kg O_2/day and has a mass of only 12.3 kg.[38] By 1990, then, it can be assumed that a small oxygen generator can be produced for a mass penalty of only 7 or 8 kg. This is an evolutionary future technology.

In the 1970s, attention was focused on the reliability of SPE systems. NASA tested small modules for 9000- and 11,000-h periods without degradation or loss of performance.[36] Later, improvements were made on the pumping systems, pressure seals, and electrode stability, among other things.[39] If a reliable electrolysis system is to accompany our comet catcher, further design and testing is required to optimize space SPE technology and to modify existing cells to fit our low-power, long-life mission requirements.

Design II—Storage

In this design, oxygen is stored gaseously in a polyethyleneterrapthalate (PET) balloon. Its properties include high flexibility over a wide temperature range, low cost, good resistance to uv degradation, and an excellent strength-to-weight ratio. The inside and the outside are coated with a few-angstrom-thick aluminum. Inside, the aluminum would oxidize to present an impermeable barrier of Al_2O_3 to the rest of the oxygen. To increase reliability, the balloon is sectioned into compartments. The design pressure is 3 psia. This Δp results in a very low leak rate of oxygen. To store 36 lbm of O_2 at a (space) equilibrium temperature of 306 K, the volume is 2200 cubic feet. For a sphere, this results in a diameter of 16 ft. The total (fuel + oxidizer) mass of the propellant for the mission is 45 lbm.

Module: The basis framework consists of an aluminum trapezoidal shape. The rocket and the electrolysis hardware are stored in the system. The solar panels form the outer cover. The assembly is shown in Fig. 5. It is interesting that not only is Design I complete, but it also translates into working hardware. Long-term reliability in space needs to be proven, however.

Summary

Based on extensive studies, of which only small a fraction could be mentioned here, it is clear that ETP can make important contributions to space exploration at little extra cost or risk. There are many technologies to pick from. These are based on Earth commercial ("off-the-shelf") or industrial systems and space-derived systems. The tradeoffs needed for any specific optimization are complex. In order for these technologies to be used in the future, work must begin now. We could begin with simple needs and gradually expand. No radically new or big (large developmental scope) technology need be embarked upon at this time. Several state-of-the-art technologies need to be studied to extend their proven benefits beyond their current use. A simple example is the H_2/O_2 rocket for propulsion. With its excellent specific impulse and consistent superior performance, it is an ideal candidate for future missions, also. Long-term space storage of H_2 and operation are issues that need to be addressed. No fundamentally new system may be needed.

It is especially fortunate that the space station will be ready in a few years to test some of these ETP concepts and to use as a platform for future missions. Some of the simpler missions described here may actually aid the space station design through preliminary hard data from long-term space missions. A water-based propulsion technology may find itself in harmony with not only the space station, but also with many planetary, asteroid, and comet resources as well. Since human (manned) missions will surely need water on board, this technology could well be an interesting one to focus attention on and develop to the point of reliable-repairable autonomy. Water also provides almost 90% oxidizer by mass. Since the oxidizer (and not the fuel) is the key in space missions, the advantages appear to be too numerous to be taken lightly. The energy needed for extraction of oxidizer appears to be easily managed, at least until 6 AU. Storage problems of LH_2 can be completely avoided by passively using it in a resistojet, for example, or, in the extreme case, discarding the H_2 and using any (combinations) of the abundant fuel resources available practically everywhere; the fuel constitutes only (typically) 16–33% of the total propellant.

Several recent studies are actively pursuing space resources and processing.[40-42] It is hoped that this paper can help in the logical, gradual, small-step transformation of technologies from the fantasyland to the tomorrowland of space exploration and space resource utilization for the benefit of mankind.

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