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Conceptual Study of on Orbit Production of Cryogenic Propellants by Water Electrolysis

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CONCEPTUAL STUDY OF ON ORBIT PRODUCTION OF CRYOGENIC PROPELLANTS BY WATER ELECTROLYSIS

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ABSTRACT

This study was conducted to assess the feasibility of producing cryogenic propellants on orbit by water electrolysis in support of NASA's proposed Space Exploration Initiative (SEI) missions. Using this method, water launched into low earth orbit (LEO) would be split into gaseous hydrogen and oxygen by electrolysis in an orbiting propellant processor spacecraft. The resulting gases would then be liquified and stored in cryogenic tanks. Supplying liquid hydrogen and oxygen fuel to space vehicles by this technique has some possible advantages over conventional methods. These potential benefits are derived from the characteristics of water as a payload, and include reduced ground handling and launch risk, denser packaging, and reduced tankage and piping requirements.

In order to assess the feasibility of this approach, a conceptual design of a water processor was generated based on related previous studies, and contemporary or near term technologies required. The baseline spacecraft processor was sized to support the propellant requirements of one manned lunar mission per year. The resulting spacecraft requires nearly 400 kW of electrical power, and has a dry payload mass of 14,000 kg (30,900 pounds), excluding cryogenic tankage and tank internals. A scaled up version of the processor to support the proposed Mars missions yields a power requirement of 2790 kW, and a mass of 93,700 kg (206,500 pounds).

Extensive development efforts would be required to adapt the various subsystems/components needed for the propellant processor for in space use. In addition, the processor would have an estimated 550 hours of down time annually, and would require astronaut extravehicular activity (EVA) for the associated repair and maintenance operations. Relative to an orbital depot of equivalent propellant capacity (where the liquid hydrogen and oxygen are delivered directly to LEO and stored on orbit), the water processor spacecraft is heavier; requires more power; is costlier to develop, deploy, and maintain; and is less reliable. Based on the cumulative results of this study, propellant production by on orbit water electrolysis for support of SEI missions is not recommended.

INTRODUCTION

Future missions envisioned by the NASA Space Exploration Initiative (SEI) require sizable quantities of liquid hydrogen and oxygen propellant to fuel the proposed space vehicles. Transportation of cryogenic propellants from the ground to low earth orbit (LEO) is a key element of the fuel architecture system needed to support the SEI missions. Contemporary fuel delivery techniques require hydrogen and oxygen to be transported in liquid form to LEO.

An alternative to this delivery method involves launching water at near ambient conditions, and then splitting the water into hydrogen and oxygen on orbit by electrolysis. Electrolysis is an electrochemical process whereby electrical energy is used to produce anode and cathode reactions in a water solution. The process consumes water while generating gaseous hydrogen and oxygen. For on orbit propellant production, the resulting gases must be dried to remove moisture, liquified, and subsequently stored as cryogenic liquids. The complete system is both a propellant processing facility, and an orbital depot where space vehicles can dock for fueling. Figure 1 illustrates the primary operations involved.

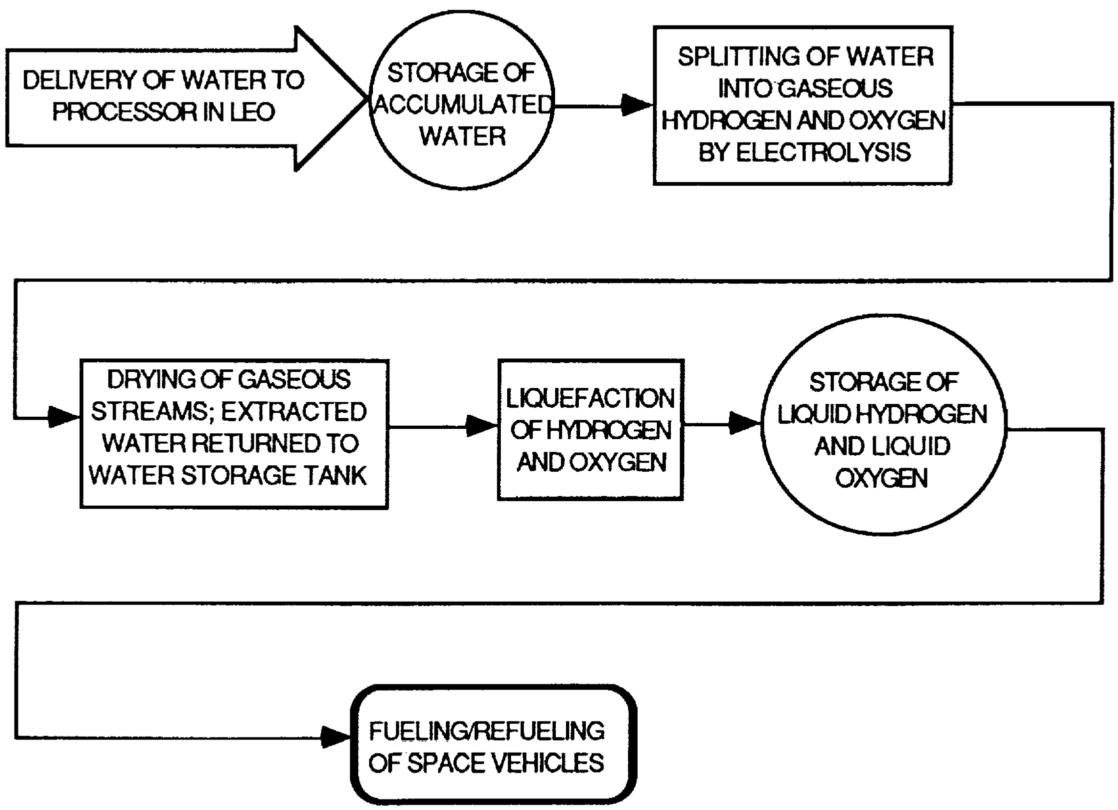

Figure 1: Diagram of primary operations for propellant processor.

The potential advantages of such a system lie in the inherent properties of water as a launch vehicle/ shuttle payload. Primary among these advantages is the reduced safety risk associated with the ground handling and launch of water as compared to liquid hydrogen and oxygen. The overall impact of this advantage is debatable since handling of liquid propellants is required for the launch vehicle propulsion system regardless of the payload. However, some reduction of risk would surely be realized for a water payload, particularly for manned vehicles.

A second advantage involves the near ambient storage conditions attainable with water. Insulation requirements for tankage and piping are effectively eliminated due to near ambient storage temperatures for water, and boiloff is insignificant. Furthermore, tank and piping structural requirements are reduced for water applications. Lastly, water is a dense payload material, providing the opportunity to better utilize volume constrained earth-to-orbit launch systems. Relative to an equivalent mass ratio of liquid oxygen and liquid hydrogen (8:1), water is more than twice as dense.

In contrast, there are several distinct drawbacks to an orbital propellant processor. Chief among these is the development, launch, and maintenance costs associated with the spacecraft. Secondly, many of the required spacecraft subsystems are known to have considerable power and heat rejection requirements (e.g. electrolyzer, liquifiers, and dryers). Finally, although the ground and launch safety risks would be reduced with a water payload, on orbit risks would be increased with an electrolyzer spacecraft. The most obvious areas of risk are the increased sources of potential propellant leakage and the possible electrical hazards posed by the power generation system. In the same vein, while a water payload reduces the tankage and piping requirements for launch, cryogenic fluid storage and handling would still be required on orbit.

The conceptual study described in this paper is undertaken to appraise the technical feasibility and tradeoffs associated with an orbital propellant processor using water electrolysis. The processor spacecraft is initially sized to support the propellant needs of one manned lunar mission annually. An extrapolation of the spacecraft weight and power requirements is then made to accommodate the proposed Mars expeditions consisting of three manned missions spaced at two year intervals.

A review of the literature is undertaken to assess the state-of-the-art performance of needed subsystems/components, and to survey any past work done in the area of propellant production via water electrolysis. Utilizing the gathered data, a spacecraft concept is generated based on current or near term technology (i.e. technology conceivably available within the next five years).

PREVIOUS WORK

The primary reference for this report is a study by Bock and Fisher of General Dynamics Convair Division (ref. 1). The study, conducted in the late 1970's, defines an orbital propellant processor which produces liquid hydrogen and oxygen by water electrolysis. Water is delivered to the processor as a shuttle contingency payload, and the generated propellant supports proposed Orbital Transfer Vehicle (OTV) activity. Spacecraft subsystems design is based on predicted mid-1980's technology.

One of the chief benefits of the water processor concept in Bock and Fisher's study is the utilization of the earth to orbit contingency payload capability of the shuttle, estimated to average more than 12,000 kg (27,000 pounds) per mission at that time. Current operations, however, do not support the contingency payload concept due to greatly reduced shuttle lift capability. Another benefit of the proposed processor is extended earth to orbit capability without the development of new launch vehicles. Once again, the contemporary significance of this advantage is diminished. Nevertheless, the system design and component specifications contained in reference 1 provide a solid point of reference for this study.

A related report released in 1978 by Heald and colleagues (ref. 2) studies propellant architecture systems needed to support expanded space activities. One of the propellant supply concepts, authored by Bock, is the orbital water processor. This report contains more detailed system information, and includes scaling data for the total equipment mass as a function of processor capacity. An economic analysis is performed to compare the water electrolysis concept to other fuel supply methods based on several propellant usage scenarios.

Propellant production by water electrolysis to support future space activities is proposed in a 1987 presentation to NASA by Rocketdyne (ref. 3). An assessment of near term technologies is outlined, and used to estimate the weight and power requirements for an orbital water processor. The resulting system is controlled by a hybrid mix of automation, teleoperation, and man-tended operation. Requisite technology development efforts needed to construct such a system are summarized.

Another major reference in the area of liquid hydrogen and oxygen production by water electrolysis is a recent study by Kohout (ref. 4) of the NASA Lewis Research Center. This report advocates the use of a lunar based regenerative fuel cell system for supplementing the generating capability of a solar power system during the lunar night. The hydrogen and oxygen reactants, produced by electrolysis of water in a closed cycle, are liquified during the sunlit period, and stored for later vaporization and use in the fuel cell. Liquefaction of the reactants results in a substantial savings in the storage tank masses when compared to pressurized gaseous storage. Many of the components and subsystems described in this study are identical to those required by an orbital propellant processor utilizing water electrolysis.

Additional background data is available from reports by Briley and Evans (ref. 5), and Ash, et.al. (ref. 6). Reference 5 reports the results of demonstration tests of a prototype propulsion module for Space Station Freedom. The propulsion system uses water electrolysis to generate gaseous hydrogen and oxygen for the thruster. Reference 6 details an extraterrestrially based propellant production facility for fueling outer planet sample and return missions utilizing the electrolysis of water.

SYSTEM DESIGN

The first step in sizing the spacecraft subsystems is determining the propellant processing rate required. Subsequently, an assessment is made of contemporary or near term technologies necessary for the system. Using the performance criteria gleaned from this assessment, the overall processor is conceptualized based on a consistent set of design assumptions and operating requirements.

Propellant Processing Rate

The baseline scenario for calculating the propellant processing rate is the support of one manned lunar mission annually. A scaled up rate for supplying propellant for the Mars missions is also computed.

Lunar Mission. There are various estimates of the propellant needs for a manned lunar mission (refs. 7-9). Values cited depend primarily on the type of propulsion system assumed, utilization of aerobraking, and the mission scenario. Taking these factors into account, a reasonably conservative estimate of 200,000 kg of total hydrogen and oxygen propellant annually is assumed. This value most closely approximates McDonnell Douglas' preliminary LEO propellant estimate for a chemical injection(40%)/aerobrake configuration without utilization of lunar derived oxygen (ref. 7).

Assuming a 6:1 fuel ratio of oxygen to hydrogen by weight for contemporary space propulsion systems, approximately 171,400 kg of oxygen and 28,600 kg of hydrogen are required. Since electrolysis produces oxygen and hydrogen at an 8:1 ratio, an excess quantity of oxygen must be generated in order to meet the hydrogen requirement. Setting the processing rate, $\dot{W}$, to meet the oxygen and hydrogen requirements described yields:

$$\frac{(1)\ \text{part}\ H_2}{(9)\ \text{parts}\ H_2{+}O} = \frac{(28{,}600)\ \text{kg/yr}\ H_2}{(\dot{W})\ \text{kg/yr}\ H_2O} \tag{1}$$

$$\therefore\ \dot{W} = 257{,}400\ \text{kg/yr} \tag{2}$$

A portion of the gaseous propellants produced are bled off for spacecraft attitude control. Reference 1 estimates an amount equal to 0.65% of the water processing capacity. An additional 2% of the water processing rate is assumed to be lost via leakage and transfer operations by the same study. Finally, Bock and Fisher (ref. 1) predict a 10% system down time based in part on the operating history of existing liquefaction plants. Boosting the processing rate to account for attitude control, fluid loss, and down time results in a processing capacity of:

$$\dot{W} = \frac{(257{,}400)\ \text{kg/yr}\ H_2O}{(0.9935)(0.98)(0.90)} = 293{,}700\ \text{kg/yr}\ H_2O \tag{3}$$

If photovoltaics are used for primary spacecraft power, another adjustment to the processing rate is needed to account for spacecraft shadow time during orbit. An orbit of 250 Nmi with an inclination of 28.5 degrees is desirable for delivery of the water to LEO, attitude control, and space vehicle fueling operations. At this altitude and inclination, the spacecraft is sunlit for approximately 62% of the orbit. Assuming the system operates only during the sunlit portion of the orbit[1], the processing capacity required is:

$$\dot{W} = \frac{(293{,}700)\ \text{lbm/yr}\ H_2O}{0.62} = 473{,}700\ \text{kg/yr}\ H_2O = 54.0\ \text{kg/hr}\ H_2O \tag{4}$$

Water Delivery Payload. Assuming 45 day launch centers, and accounting for the 62% operating cycle and 10% down time, the water payload required to support the calculated processing rate for one manned lunar mission per year is:

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[1] Bock and Fisher (ref. 1), Heald, et.al. (ref. 2), and Rocketdyne (ref. 3), all specify transient system operation in their studies. Heald and coworkers address the effect of cyclic operation on the spacecraft subsystems and conclude that there are no significant difficulties associated with this type of configuration.

$$m_w = (54.0 \text{ kg/hr } H_2O)(0.62)(0.90)(45 \text{ days/launch})(24 \text{ hr/day}) = 32{,}540 \text{ kg } H_2O\text{/launch} \quad (5)$$

This payload mass is beyond the capability of the Shuttle. Therefore, based on 45 day launch centers, a Shuttle C or other HLV would be required to supply the orbital processor for support of one manned lunar mission annually.

Mars Missions. Propellant needs for proposed Mars missions are more difficult to resolve. Estimates vary widely according to the technology assumed. Based on the values cited in references 7 through 9, a total annual propellant requirement of 1,400,000 kg is chosen to support the manned Mars missions. This value approximates the peak annual propellant requirement for a scenario involving three manned Mars excursions at two year intervals (refs. 8 and 9).

The yearly propellant requirement for the manned Mars missions is seven times that chosen for the baseline system supporting one lunar mission annually. Therefore, the processing capacity required to support the Mars missions is seven times the previously calculated value, or 378.2 kg/hr H_2O.

Power Generation

Specification of a power source for the water electrolysis spacecraft is a key part of the overall design. Many of the system components are power intensive. For the purposes of this study, competing power sources are compared by the criteria of power supplied per unit mass (specific power) for a complete power generating subsystem and associated equipment. Figure 2 presents a comparison of current and projected specific power for three power generating technologies; photovoltaics, nuclear, and solar dynamic.

Photovoltaics. Kurland and Stella (ref. 10) cite a power to mass ratio of 25-45 W/kg for existing rigid panel flight arrays. Under the Advanced Photovoltaic Solar Array Program (APSA) funded through JPL, a near term performance goal of 130 W/kg is proposed for a 10 kW system. The ultimate objective of the program is development of a solar array with a specific power of 300 W/kg at power levels of 25 kW by the turn of the century.

Large area planar silicon cells are capable of efficiencies as high as 15% according to Lillington and colleagues (ref. 11). Single junction GaAs cells have demonstrated an efficiency of 18.5%, with efficiencies as high as 24% to 25% expected for two junction cells.

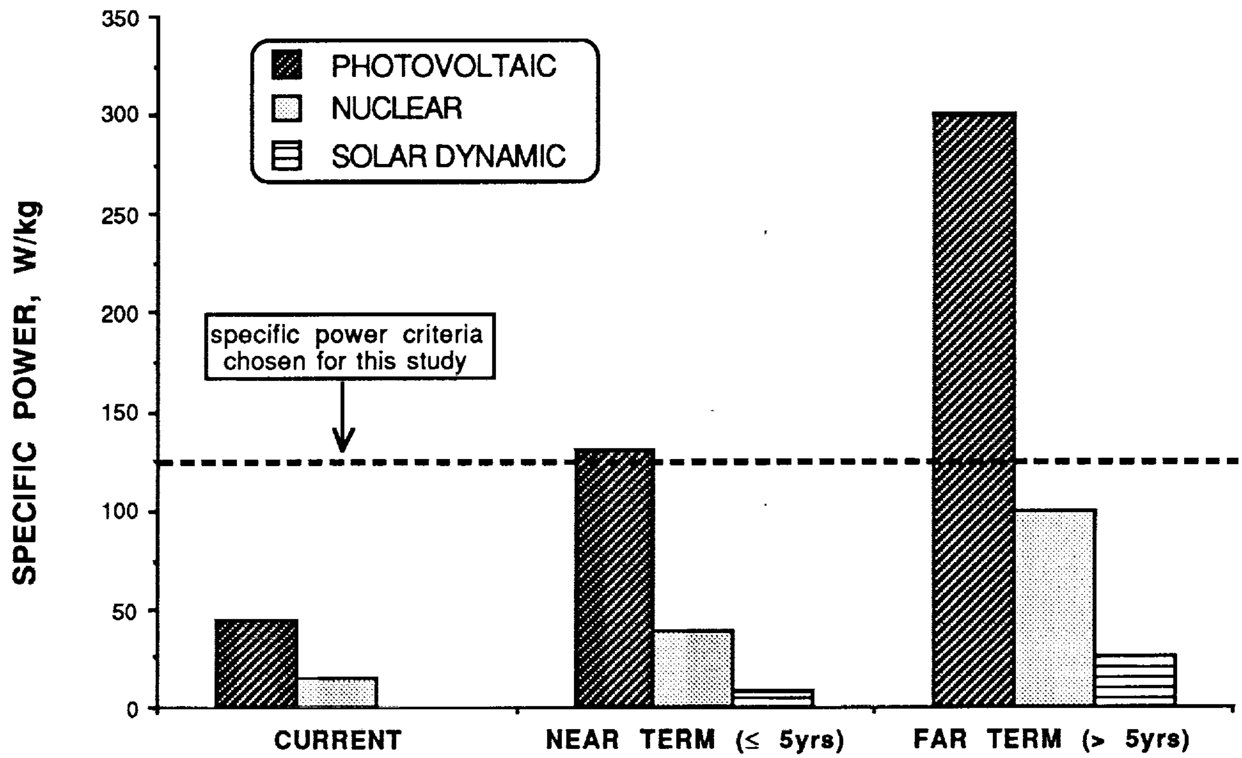

Figure 2: Comparison of current and projected specific power for photovoltaic, nuclear, and solar dynamic power sources based on cited references.

Kohout's study (ref. 4) specifies a 123 W/kg GaAs power source operating on the lunar surface. By comparison, Bock and Fisher (ref. 1) project a highly optimistic performance of 161 W/Kg for mid-1980's technology.

Based on the projections found in the literature, and discussions with personnel from the Power Technology Division of the Lewis Research Center, a reasonable near term performance estimate of 125 W/kg is chosen for this study. This value corresponds to a GaAs photovoltaic solar array with an efficiency of 22.5%.

Nuclear and Solar Dynamic. Current performance for nuclear power sources vary from 5 to 14 W/kg. The SP-100 program proposes systems in the 40 W/kg range with capacities in the hundreds of kilowatts by the early to mid-1990's (see Winter, ref. 12). Far term estimates approaching 100 W/kg are anticipated for power sources in the multi-megawatt range. An inherent disadvantage of using a nuclear power source for a water processing spacecraft is the shielding required to protect astronauts during maintenance and refueling operations.

Solar dynamic power systems are a less competitive option for this application, with future performance estimates of 7 to 25 W/kg (see Warshay and Mroz, ref. 13, and Friefeld and Wallin, ref. 14). These values are representative of systems incorporating thermal energy storage equipment for continuous operation, which constitutes approximately two thirds of the receiver mass. Even without the added thermal energy storage mass, however, solar dynamic systems yield lower specific power than photovoltaic sources.

Power System Design. Based on the performance criteria gathered for current and near term power sources for space applications, photovoltaics is chosen as the baseline power system for the electrolyzer spacecraft. All previous studies of propellant production by electrolysis referenced in this report also specify solar arrays. A conversion and distribution efficiency of 93% is assumed for the power system.

Electrolyzer

Electrolysis of water is an electrochemical reaction whereby water is split into its gaseous constituents, hydrogen and oxygen. The process consumes electrical energy, as the resulting gases collect at the anode and cathode of the electrolyzer. The specific chemical reactions that take place depend on the whether the medium is acidic or alkaline.

Commercial electrolyzers for terrestrially based specialty hydrogen markets are available from a variety of international sources. Research is being conducted in the areas of catalyst and membrane materials, as well as alternative methods of splitting water (see refs. 15-17). Likewise, studies have been performed to assess the feasibility of large scale hydrogen production by electrolysis using both photovoltaic and nuclear power sources (refs. 18-20).

Performance data on electrolyzer units extracted from a variety of references is reasonably consistent. A summary chart of various electrolyzers from two different suppliers was generated by the Space Station Freedom project. Performance of the summarized units ranges generally from 4.4 to 5.1 kilowatt-hours of energy required per kilogram of water consumed (KWh/kg). This criteria is an indication of the electrolyzer's energy efficiency, with a lower value denoting reduced energy requirement per unit mass of electrolyzed water. By comparison, Rocketdyne projects an electrolyzer operating at 4.98 kWh/kg (ref. 3), and Bock and Fisher estimate a comparable performance of 4.85 kWh/kg (ref. 1). Ash and colleagues (ref. 6) use a more energy efficient value of 4.48 kWh/kg for their analysis.

The most recent electrolysis study cited (Kohout, ref. 4) utilizes a computer code developed by Rieker and Hoberecht which simulates an alkaline regenerative fuel cell (refs. 21 and 22). The code generates system and subsystem data, including information on the alkaline electrolyzer modeled, based on the inputted operating parameters. This computer program was employed for the present study to generate data on an electrolyzer unit operating at 4.41 kWh/kg.

To enhance system reliability, two electrolyzers, each with half the total capacity required (27.0 kg/hr H_2O), are specified. This processing rate is used as a convergence criteria for the alkaline RFC code (see ref. 21) in order to size the electrolyzers required. Input parameters to the computer code included an operating temperature of 355 K, operating pressure of 2.17 MPa, current density of 1615 A/m^2, and an active electrode area of 0.093 m^2 per cell. The RFC code is run iteratively until the desired water consumption rate is achieved. Computer program results include electrical power required, equipment weight, exiting gaseous hydrogen and oxygen mass flowrate, and moisture content in the hydrogen and oxygen streams.

Dryers

The drying subsystem, taken directly from Bock and Fisher's study (ref.1), removes the moisture from the gaseous hydrogen and oxygen streams in a two step process. In the first step, some 99.9% of the water is condensed in a cold trap separator. The second step removes the remaining moisture by absorption and adsorption via a corrugated rotor in the flow path which is impregnated with a hygroscopic salt. The salt is regenerated by heated exit gas from the cold trap during a portion of the rotor revolution.

The cold trap separator mass is scaled from reference 1 based on the total mass flow of the respective gas streams leaving the electrolyzer, including the moisture content. This scaling method is also used to estimate the structural support mass, and the electrical power required for the radiator pump. Conversely, the rotor assembly is scaled by the water mass flow rate in the gases exiting the separator.

Heat dissipation requirements, and the associated radiator size needed, are determined by calculating the total energy removed from the gas streams during the cooling process. The total energy removed in the drying system radiator under study is ratioed to the energy removed in the radiator from reference 1, and used to scale the heat rejection needed.

Liquifiers

Hydrogen and oxygen gas produced by electrolysis must be liquified prior to storage in the cryogenic tanks. In addition, boiloff gas from the cryogenic storage tanks is reliquefied. Performance criteria for the liquifiers required to accomplish these tasks must be estimated. Data in the referenced literature on oxygen liquefaction systems is reasonably consistent, whereas estimates of hydrogen liquifier performance are somewhat divergent.

Hydrogen Liquifiers. There is a general lack of data on hydrogen liquifiers with the operating capacity necessary for this study (i.e. liquefaction rate of 6 kg/hr, or equivalently, 2.1 kW of cooling capacity). Much larger liquifiers are routinely used in hydrogen liquefaction plants, while cryocoolers with smaller refrigeration capability are commonly utilized for applications such as sensor cooling. However, in the capacity range of interest for this study, little data is available. Furthermore, severe linear extrapolation of one or more orders of magnitude from existing liquifier data is an undesirable method of generating reliable performance estimates. Table I summarizes the hydrogen liquifier performance data cited in this report.

An NBS report by Strobridge (ref. 23) provides the most in depth generic data on liquifiers and refrigerators of various capacities and operating temperatures. The report is a survey of refrigerators either existing or under development at the time of the study. Parameter trends are plotted for efficiency, volume, and mass, all as a function of refrigeration capacity. Based on these trends, an efficiency of 19 percent Carnot, and an equipment mass per unit of cooling capacity of 5.1 kg/W, is calculated for the hydrogen liquifier in the capacity range required for this study. The equipment mass estimate from this source is higher than would be anticipated for space applications, since a good deal of the data used for the correlation is from ground based liquifiers, which are not weight optimized.

Bock and Fisher (ref. 1) and Kohout (ref. 4) specify hydrogen liquifiers operating at 25 percent Carnot, and with a mass to cooling capacity ratio of 0.7 kg/W. The relatively high efficiency estimates used in these studies are based on a proposed reversed Brayton cycle liquifier. More conservative values of 21 percent Carnot and 1.1 kg/W are presented by Rocketdyne (ref. 3) for their hydrogen liquefaction system.

Waynert and coworkers (ref. 24) report a current state-of-the-art performance of 20-25 percent Carnot for hydrogen liquifiers with a liquefaction rate of 190-1130 kg/hr (5-30 tons/day). Efficiency is expected to drop to 15-20 percent Carnot for scaled down systems in the 40 kg/hr (1 ton/day) range with a conventional cycle. The magnetic liquifier proposed by Waynert and group, however, is projected to have an efficiency of 24 percent Carnot in the 40 kg/hr range. No estimate of the equipment mass is given.

Table I: Hydrogen liquifier performance data

Source	Efficiency[2] (%Carnot)	Size[3] (kg/W)	Comments
Strobridge (1974-NBS) (ref. 23)	19	5.1	correlations based on existing equipment, and equipment under development
Rocketdyne (1987)(ref. 3)	21	1.1	2.8 kW cooling capacity
Waynert, et.al. (1989-Astronautics) (ref. 24)	20-25 15-20		190-1130 kg/hr liquefaction cap. 40 kg/hr liquefaction capacity
	24		projected for magnetic hydrogen liquifier; 40 kg/hr capacity
Bock & Fisher (1978-GDC) (ref. 1)	25	0.7	3.6 kW cooling capacity

Based on the collected data, an efficiency estimate of 24 percent Carnot is selected for this study corresponding to the performance proposed for the hydrogen magnetic liquifier featured in reference 24. Equipment mass estimates are based on the 0.7 kg/W criteria used by Bock and Fisher, and Kohout. Thus, the resulting hydrogen liquefaction system is an optimistic near term prediction in terms of performance and overall equipment mass.

A catalyst is used for ortho to para hydrogen conversion. The cooling load and radiator equipment mass is scaled from reference 1 based on liquified flow rate. Power requirements for the radiator pumps are also scaled by this method. Using the performance criteria for efficiency and equipment mass chosen, the hydrogen liquefaction system is sized to meet the estimated cooling load.

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[2] Percent Carnot indicates the performance deviation of the actual liquifier from an ideal Carnot cycle operating between the liquefaction temperature and the temperature of the surroundings (nominally 300 K).

[3] Sizing criteria expressed as equipment mass in kilograms per watt of cooling capacity.

Oxygen Liquifiers. Reasonable agreement exists in the collected data on oxygen liquifier performance. The correlations by Strobridge (ref. 23) result in a estimate of 20 percent Carnot efficiency and 0.2 kg/W for an oxygen liquifier in the capacity range of interest. Bock and Fisher (ref. 1) and Kohout (ref. 4) use 20 percent Carnot efficiency and 0.1 kg/W, while Rocketdyne (ref. 3) specifies a liquifier operating at 19 percent Carnot and 0.4 kg/W.

An oxygen liquifier with an efficiency of 20 percent Carnot and an equipment mass to cooling capacity ratio of 0.1 kg/W is chosen for this study. Cooling load, equipment mass, and power requirements are calculated as described for the hydrogen liquifier system.

All of the oxygen produced by electrolysis is assumed to be liquified and stored by the processing system. However, since an excess of oxygen is produced by the system due to the difference in generated oxygen to hydrogen mass ratio compared to the ratio required for propulsion (8:1 versus 6:1), some of the oxygen could be dumped overboard. If this excess oxygen were extracted from the system at the outlet of the electrolyzer, the overall spacecraft power requirement would be reduced by 3%, and the overall mass reduced by less than 3%, due to the diminished load on the dryer and liquifier subsystems. The tradeoffs involved with various methods of handling the surplus oxygen were not investigated in this study.[4]

Propellant Storage

The cryogenic storage requirements are calculated from one year's production of propellant. In a year's time, the processing system will generate 260,500 kg (574,000 lbm) of liquid oxygen, and 32,500 kg (72,000 lbm) of liquid hydrogen. Assuming 5% residuals and 90% maximum tank fill level, the minimum tank storage volumes needed are approximately 269 m^3 for the oxygen, and 540 m^3 for the hydrogen.

Bock and Fisher (ref. 1) use a modified shuttle external tank (ET) for storage of the propellants. Modifications include additional insulation, and various fluid management and handling components. The capacity of an ET is slightly more than double the volume required to accommodate an annual yield of propellants for the system under study. Since the ET is part of the shuttle system, its mass is not included in the total payload weight. However, the long term thermal performance achievable with an ET, not to mention the on orbit operations required to modify it, render this option questionable.

[4] For example, retained excess oxygen could be used for life support systems aboard the space vehicles being refueled by the processor spacecraft. Also, a measured quantity of liquid hydrogen could be launched along with the water suppling the processor to offset the surplus oxygen.

For this reason, an ET is not explicitly specified in this study. Instead, the additional insulation mass for the ET option is itemized in the spacecraft weight summary, and the total spacecraft weight does not include the mass of the propellant tankage and internals. Summarizing the spacecraft mass in this way is consistent with the approach of reference 1.

Boiloff estimates of 3.3%/month for the hydrogen tank, and 0.8 %/month for the oxygen tank, are used by Bock and Fisher (ref. 1). These values are reasonably consistent with contemporary predictions of on orbit performance for cryogenic tanks with passive thermal control (e.g. see refs. 7 and 25). All boiloff is reliquefied, and is therefore a part of the cooling load for the hydrogen and oxygen liquifiers[5].

Other Subsystems

Radiators are needed to dissipate the waste heat generated by the dryers and liquifiers in the water processing system. Heat rejection for this system is sizable, particularly for the liquifiers. Consequently, radiator design and heat dissipation requirements are primary drivers in terms of overall spacecraft mass. Contemporary radiator design data predicts a performance parameter of 10 kilograms of radiator mass per kilowatt of heat rejection capability (kg/kW) at a rejection temperature of 340 K. This criteria is comparable to Bock's radiator subsystem designs (refs. 1 and 2), and is therefore chosen for this study.

Power consumption, equipment mass, and heat dissipation requirements for the remainder of the spacecraft subsystems and components are scaled directly from information contained in Bock and Fisher's report (ref. 1).

SYSTEM OPERATION

The primary flow block diagram for the propellant processor spacecraft is illustrated in figure 3. The overall system consumes 398 kW of power during sunlit operation, resulting in an annual accumulation of 260,500 kg (574,000 lbm) of oxygen, and 32,500 kg (72,000 lbm) of hydrogen. Solar arrays supply primary power to the spacecraft during propellant production, while a fuel cell provides housekeeping power during the shadow portion of each orbit. A power and mass summary for the processor spacecraft is shown in table II.

[5] Reliquefaction of the boiloff gases represents a small fraction of the overall cooling load (less than 5% for both the hydrogen and oxygen liquifiers). Therefore, incorporating more advanced insulation systems (e.g. vapor cooled shields, p-o converters, etc.) has little effect on the liquifier power needs.

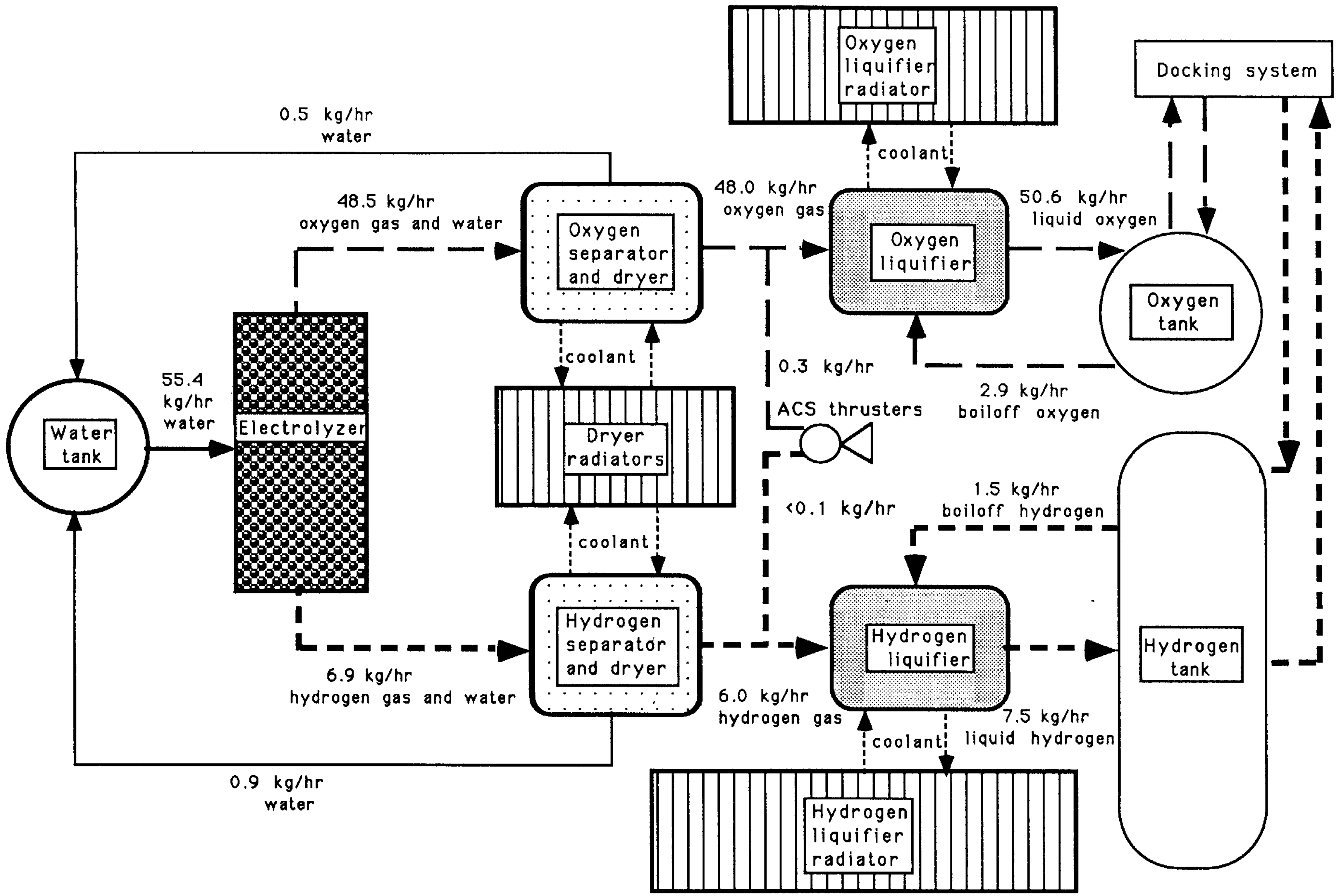

Figure 3: Propellant processor primary flow block diagram (baseline lunar scenario).

Referring to figure 3, water delivered to LEO is stored in the water tank, where it is subsequently pumped to the electrolyzer at a rate of 55.4 kg/hr. The water is electrochemically split into moisture laden streams of hydrogen and oxygen gas in the alkaline electrolyzer. Water in the gaseous streams is then removed in a two step drying process as described earlier. Radiators dissipate the heat generated by the drying process. The 1.4 kg/hr of extracted water is returned to the water tank, resulting in a net water consumption rate of 54.0 kg/hr during the system's 62% operating cycle.

A small portion of the dried hydrogen and oxygen gases are bled off for the attitude control system, with the remaining primary gas flow entering the respective liquifiers. The hydrogen and oxygen liquifiers condense the incoming gas from the drying system, and also reliquefy boiloff gases from the propellant storage tanks. The resulting liquified propellants are transferred to cryogenic storage tanks. The tanks are fitted with fluid handling components and high performance insulation for long term, on orbit cryogen storage and handling. Radiators reject heat generated from the liquefaction process.

A docking system is integrated with the propellant tanks to accommodate space vehicle fueling operations. The overall spacecraft is roughly estimated to have a ten year lifetime based on data for the primary components.

Table II: Spacecraft subsystem summary

Subsystem	Description	Weight* (kg)	Power (kW)	Volume/Area
Solar Array	GaAs, 22.5% efficiency, 125 W/Kg, 2.48 kg/m^2	3420	(428)	1380 m^2
Electrolyzer	Alkaline, 4.41 kWh/kg-H_2O, 355 K, 2.17 MPa	2920	237	2.3 m^3
Hydrogen Liquifier	24 %Carnot efficiency; 16.4 kWh/kg-H_2, 0.7 kg/W	1420	103	1.7 m^3
radiator	10 kg/kW (rejection temp.: 340 K), 7.3 kg/m^2	1980	4	270 m^2
Oxygen Liquifier	20 %Carnot efficiency; 0.9 kWh/kg-O_2, 0.1 kg/W	360	46	1.1 m^3
radiator	10 kg/kW (rejection temp.: 340 K), 7.3 kg/m^2	600	4	83 m^2
Separators/Dryers (incl. radiators)	Two step process: cold trap condensation and absorption/adsorption	100	1	17 m^2
Tank Insulation (ET option)	MLI, 3.3%/mo. H_2 boiloff, 0.8%/mo. O_2 boiloff	1470	-	
Miscellaneous	Structure, piping, pumps, water storage, avionics, fuel cell, etc.	1740	3	
Totals		14,010 *	398	

* Does not include weight of cryogenic tankage and tank internals

DISCUSSION OF RESULTS

The baseline processor spacecraft conceptualized to support the propellant requirements for one manned lunar mission annually has an earth to orbit dry payload weight of 14,000 kg (30,900 lbm), excluding cryogenic tankage and internals. The electrical power needed for the processor is nearly 400 kW, or more than five times the power capability planned for Space Station Freedom at permanent manned capability. A scaled up spacecraft designed to support the planned Mars missions would have a dry payload weight of 93,700 kg (206,500 lbm), and a power requirement of 2790 kW.

Most of the power required, approximately 60%, is utilized by the electrolyzer subsystem. The hydrogen and oxygen liquifier subsystems consume almost all of the remaining power capacity, representing 27% and 12% of the total power requirement, respectively. In terms of equipment mass, the solar array is the single most massive subsystem, followed closely by the hydrogen liquifier and associated radiator, both of which make up approximately 24% of the total mass each. The electrolyzer accounts for an additional 21% of the total mass. The remaining 31% is distributed among the other subsystems.

Maintenance operations on the processor would be substantial, and would require astronaut EVA. A total of 550 hours of down time annually is estimated for repair and maintenance. If the processor is a free flyer, then shuttle flights would be needed to support the maintenance activities. Also, water delivery aboard a shuttle C or other heavy launch vehicle would be required in order to supply the 32,540 kg (71,700 lbm) of water needed per payload, assuming 45 day launch centers.

Another key feature of the system under study is the development effort required for many of the subsystems and components. Table III gives a brief synopsis of the development issues associated with several of the technologies needed[6]. In general, operation of many of the components has not been verified in a microgravity, space environment. In addition, although solar arrays and radiators for space applications are an existing technology, the sizes required for this system are unprecedented. Finally, operation of the processor in an automated mode poses a considerable system control challenge. The spacecraft would essentially be a hydrogen and oxygen generation and liquefaction plant in space, with all the inherent process and operational complexities.

The beneficial tradeoffs associated with orbital production of propellants by water electrolysis lie in three areas, as described earlier in this report. First, there is a presumed reduction in the ground

[6] The development items listed in table III are in addition to the technologies needed for in-space cryogenic fluid storage and handling. Fluid management issues in a microgravity environment must be addressed for any orbital fueling concept, regardless of the specific system used to supply the propellant.

handling and launch risks for transporting water payloads to LEO, although the on orbit safety risks are increased. Second, the ambient storage conditions of water result in reduced structural and insulation requirements for tankage and piping. And third, the greater density of water as compared to an equivalent configuration of liquid hydrogen and oxygen, provides a potential opportunity for increased earth to orbit payload mass. Optimization of payload manifesting to exploit this advantage was not undertaken in this study.

Table III: Propellant processor development issues

Subsystem/Component	Issues
electrolyzer	ground operation established; microgravity fluid dynamics considerations
dryers	in space operation unproven
liquifiers	ground operation in the capacity range required is not established; on orbit operation must be validated
solar arrays	needed technology is currently under development; unprecedented size for space application
radiators	same as for solar arrays
process control	complex multiple processes; automated operations

CONCLUDING REMARKS

The impetus for this study was the assessment of an orbital electrolysis/liquefaction system for supporting the propellant needs of the planned SEI missions. With that objective in mind, it seems appropriate to compare this system, at least qualitatively, with its most likely competitor, namely an orbital propellant depot. Relative to an orbital depot of equivalent propellant capacity, the water processor conceptualized in this study is heavier; requires more power; is costlier to develop, deploy, and maintain; and is less reliable. The water processor system (see figure 3) contains all of the components necessary for an orbital depot, *plus* liquifier, dryer, electrolyzer, and water subsystems. Each of these additional subsystems increases mass, power, development effort, maintenance, system risk, and cost.

In light of these drawbacks, propellant production by on orbit water electrolysis for support of SEI missions is not recommended. It is conceivable, however, that other applications of this system, such as extraterrestrial propellant processing, could prove advantageous.

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16. Abstract

This study was conducted to assess the feasibility of producing cryogenic propellants on orbit by water electrolysis in support of NASA's proposed Space Exploration Initiative (SEI) missions. Using this method, water launched into low earth orbit (LEO) would be split into gaseous hydrogen and oxygen by electrolysis in an orbiting propellant processor spacecraft. The resulting gases would then be liquified and stored in cryogenic tanks. Supplying liquid hydrogen and oxygen fuel to space vehicles by this technique has some possible advantages over conventional methods. These potential benefits are derived from the characteristics of water as a payload, and include reduced ground handling and launch risk, denser packaging, and reduced tankage and piping requirements. In order to assess the feasibility of this approach, a conceptual design of a water processor was generated based on related previous studies, and contemporary or near term technologies required. The baseline spacecraft processor was sized to support the propellant requirements of one manned lunar mission per year. The resulting spacecraft requires nearly 400 kW of electrical power, and has a dry payload mass of 14,000 kg (30,900 pounds), excluding cryogenic tankage and tank internals. A scaled up version of the processor to support the proposed Mars missions yields a power requirement of 2790 kW, and a mass of 93,700 kg (206,500 pounds). Extensive development efforts would be required to adapt the various subsystems/components needed for the propellant processor for in space use. In addition, the processor would have an estimated 550 hours of down time annually, and would require astronaut extravehicular activity (EVA) for the associated repair and maintenance operations. Relative to an orbital depot of equivalent propellant capacity (where the liquid hydrogen and oxygen are delivered directly to LEO and stored on orbit), the water processor spacecraft is heavier; requires more power; is costlier to develop, deploy, and maintain; and is less reliable. Based on the cumulative results of this study, propellant production by on orbit water electrolysis for support of SEI missions is not recommended.

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