Space Solar Power and Platform Technologies
for
In-Space Propellant Depots

Final Report

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The Boeing Company
in cooperation with
the National Aeronautics and Space Administration
Marshall Space Flight Center

Contract NAS8-99140, Mod.2, Task 3

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ABSTRACT

The cost of access to space beyond low Earth orbit may be reduced if vehicles can refuel in orbit. The cost of access to low Earth orbit may also be reduced by launching oxygen and hydrogen propellants in the form of water. To achieve this reduction in costs of access to low Earth orbit and beyond, a propellant depot is considered that electrolyzes water in orbit, then condenses and stores cryogenic oxygen and hydrogen. Power requirements for such a depot can be met using technology developed through NASA's Space Solar Power (SSP) Exploratory Research and Technology (SERT) program.

A propellant depot is described that will be deployed in a 400 km circular equatorial orbit. It receives tanks of water launched into a lower orbit from Earth (by gun launch or reusable launch vehicle), converts the water to liquid hydrogen and oxygen, and stores up to 500 metric tons of cryogenic propellants. Orbital maneuvering vehicles transfer the Earth-launched propellant tanks from the lower orbit to the depot orbit. The propellant stored in the depot can support transportation from low Earth orbit to geostationary Earth orbit, the Moon, LaGrange points, Mars, etc. The propellant tanks on the depot are modified versions of those used in the Delta IV-Heavy launch vehicle. The tanks are configured in an in-line, gravity-gradient configuration to minimize drag and settle the propellant. Thermal control is maintained by body-mounted radiators; these also provide some shielding against orbital debris. Power is supplied by a pair of solar arrays mounted perpendicular to the orbital plane, which rotate once per orbit to track the Sun. The majority of the power is used to run the electrolysis system.

For comparison, a more conventional propellant depot is also described. This conventional depot does not require the high power levels for propellant production, but it does require delivery of propellants to orbit in the form of cryogenic fluids, which are highly combustible and take up three times as much volume of the same mass of water.

Technology needed for an orbiting propellant depot can be tested and demonstrated in the near-term on the ground, on a Shuttle-deployed free-flyer, and on the International Space Station. In the intermediate future, an orbital depot may be deployed that stores liquid hydrogen and oxygen launched from Earth, to be followed by a full conversion and storage depot.

INTRODUCTION

The following report summarizes the results of a study performed by the NASA Marshall Space Flight Center (MSFC) and The Boeing Company as part of the Space Solar Power Exploratory Research and Technology Program (SERT) Program. Boeing work described herein is part of contract NAS8-99140, "Space Solar Power Systems Studies and Analysis." Other tasks performed under this contract are reported in a separate volume.

Our objective in this study was to develop conceptual designs for two types of Propellant Depot in low Earth orbit:

(1) A Propellant Depot that receives water and uses solar power to convert water into liquid hydrogen and oxygen (LH2 and LOX), then stores and transfers these cryogenic propellants. The basic concept for production of LH2 and LOX is through an <u>electrolysis process</u> commonly used in fuel cells. The process "cracks" the water to form hydrogen and oxygen gas, which is then refrigerated at cryogenic temperatures to convert it into liquid propellants.

(2) A Propellant Depot that only receives, stores, and transfers cryogenic propellants.

The first of these types of Propellant Depot is more complex, requiring significant advances in technology, but it avoids the large volume and safety issues related to containment of cryogenic propellants during launch. Water, in the form of liquid or ice, takes up one third of the volume that would be needed to contain the same mass of liquid hydrogen and oxygen. Cryogenic propellants are hazardous; hydrogen is extremely volatile and flammable, and liquid oxygen is a very powerful chemical oxidizer. Water, in contrast, is chemically inert. As an incompressible liquid, or as solid ice, water can also sustain high payload accelerations during launch. Future, high velocity projectile launch systems could potentially accelerate capsules of water, at several hundred g, to reach orbital velocity. Repeated launches of such a system could potentially transport large masses of water into orbit at a much lower cost than conventional space transportation systems.

In addition, the first of these types of Propellant Depot, the one that converts water into propellants, could serve other future NASA needs:

- The production concept follows Science exploration goals for <u>"following the water".</u> Finding water in the solar system means there is a chance at finding life and sustaining human life. Development of such depot technology will enable sustainable human missions at any location where water can be found, (i.e., Moon, Mars, Europa, etc.).
- This baseline concept is for a cryogen production facility in low-Earth-orbit designed to supply human, robotic, and commercial missions with liquid hydrogen (LH2), and liquid oxygen (LOX) for high thrust <u>chemical engines</u>, LH2 for <u>solar thermal propulsion</u>, and excess <u>LOX for human habitation</u> at other stations.
- Production capabilities would <u>enable new commercial markets</u> for reusable high-energy upper stages, satellite services, and water and oxygen for ongoing human operations.

NASA MSFC, Boeing, and Sverdrup personnel conferred via weekly teleconferences to accomplish this work. Key MSFC personnel included the NASA programmatic leader, David Smitherman, the Contracting Officer's Technical Representative, Dr. Connie Carrington, and the technical Task Leader, John Fikes. Key Boeing personnel included the Project Manager, Mark Henley, and the Principal Investigator, Dr. Seth Potter, as well as Sonia Gutierrez, a student intern who supported this activity on a full-time basis. Sverdrup also supported this work with thermal analysis. NASA Langley Research Center participated in a concurrent, parallel activity. Many other individuals from MSFC, Boeing, and Sverdrup supported this work; the full list of team members was as follows:

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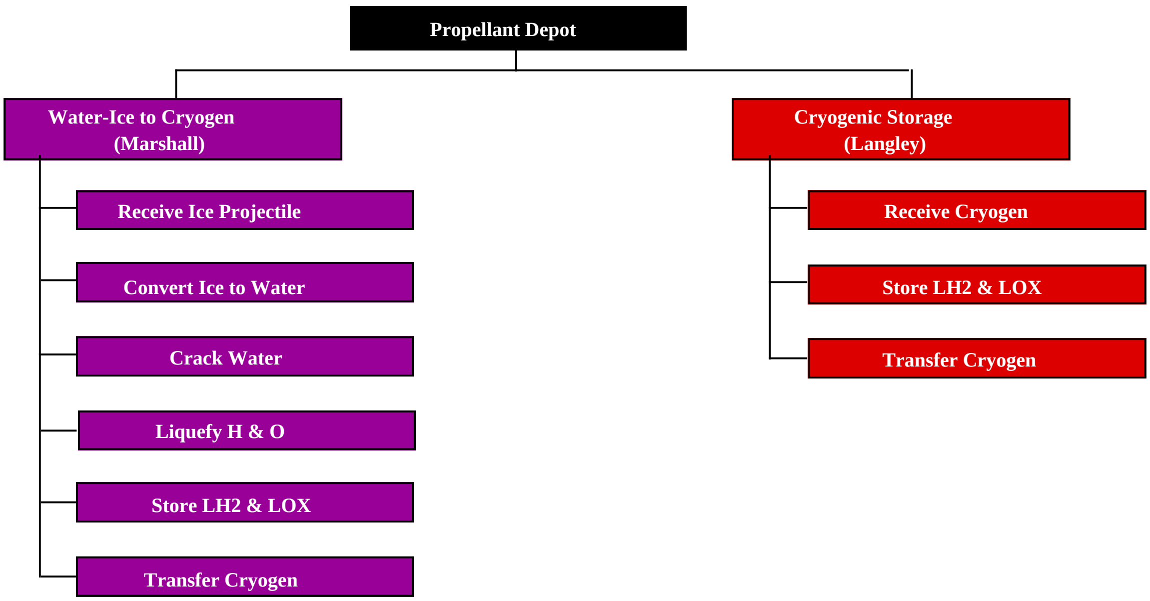
1. SYSTEM REQUIREMENTS

System requirements for the two types of Propellant Depot were derived in parallel with the conceptual design process. The basic functional requirements for the two types of Propellant Depot differ substantially, and the resulting system requirements are also substantially different. Functional requirements are summarized in Table 1.1 for the two types of Propellant Depot.

Table 1.1 Propellant Depot Functional Requirements



Initial emphasis in this study was placed on the Water-Ice to Cryogen propellant production facility. A very high power system was required for "cracking" (electrolyzing) the water and condensing and refrigerating the resulting oxygen and hydrogen. For a propellant production rate of 500 metric tons (1,100,000 pounds) per year, an average electrical power supply of ____ was required. To make the most efficient use of space solar power, electrolysis was performed only during the portion of the orbit that the Depot was in sunlight, so roughly twice this power level was needed for operations in sunlight (slightly over half of the time). This power level mandated large solar arrays, using advanced Space Solar Power technology. A significant amount of this power had to be dissipated as heat, through large radiators.

In contrast to the inherent complexity of the Water-Ice to Cryogen propellant production facility, the Cryogenic Storage-Only facility was designed to be as simple as possible. Without electrolysis and propellant condensation, this facility had relatively modest power and heat rejection requirements, which could be met with small areas of body–mounted solar arrays and radiators. Instead of refrigeration, passive thermal control was possible, using hydrogen boil-off gas to chill the remaining liquid hydrogen and liquid oxygen, and heating the hydrogen gas for use in low thrust station-keeping propulsion.

Table 1.2 summarizes Systems Requirements derived for these two types of Propellant Depot.

Table 1.2. System requirements differ significantly for the two types of Propellant Depot.

System Requirement	1)H20-Cryo Production	2)Storage only	Comments/Rationale
0. Propellant State at Launch	Water (liquid or ice)	LH2 and LOX	Ground-rule of this study
1. Depot O:F Mass Ratio	8:1 (actually 7.8:1)	6:1	Stochiometric vs Optimal
2. Propellant Quantity	500,000 kg (@ 8:1)	389,000 kg	= H2 mass; OK for HEDS
3. Depot Assembly	Automated; 8 tanksets	None (1 launch)	Avoid requiring human help
4. Propellant Tank Size	5 m diam., 12 m long	5 m; 30 m long	Available cryo tank tooling
5. Depot Launch System	Tank = Delta IV stage	Delta IV core	Use ELV Stage tanks in orbit
6. Orbit Altitude	400 x 400 km	400 x 400 km	Balance performance w/ drag
7. Orbit Inclination	Equatorial	Equatorial	Performance & other benefits
8. Depot Propulsion	SEP (krypton) or H2/O2	Thermal H2	Power for SEP; Excess O2/H2
9. Depot Electrical Power	708 kWe (in sunlight)	(minimal)	Derived for Production Depot
10. Solar Array Area	1,660 m2	(minimal)	SERT solar array; 40% efficient
11. Solar Array Orientation	Perpend. to Orbit Plane	E & W surfaces	Entech method for beta angles
12. Depot Orientation	Gravity Gradient	g Gradient	Sufficient for settling propellant
13. Radiator Area	1,560 m2 (x 2 = 3,120)	(minimal)	Body-mounted; doubles area
14. Radiator Orientation	Body Mounted N & S	N & S surfaces	Large area for body mounting
15. Propellant Launch System	OMV + RLV or + Gun	RLV off-load	OMV increases PL for H20
16. OMV Propulsion	H2/O2	N/A	OMV increases H20 PL
17. OMV Docking Port	Near CM	N/A	CM changes only slightly
18. Water tank volume & place	35.5 m3 /near CM	N/A	RLV PL mass; CM changes little
19. Electrolysis volume & place	11 m3 /near CM	N/A	Published data; place available
20. Liquifaction volume & place	3 m3 + 5 m3 /near CM	N/A	Published data; place available

The propellant state at launch was ground-ruled to be either water (in the form of liquid or ice) or cryogenic propellants. Considering the unknown ascent heating loads and unknown time in orbit before reaching the depot, it was assumed that water was received at the depot as a liquid (with a temperature at the melting point of 273 degrees Kelvin). This appears to be a conservative assumption, for if water was received in the form of solid ice, a heat exchanger could make use of the "heat of fusion" (energy absorbed as ice melts) to reduce electrical power and radiator surface area requirements of the Propellant Production Depot. Cryogenic propellants were assumed to be near their normal boiling point at sea level atmospheric pressure, rather than being sub-cooled (which would make them less subject to boil-off during launch and transfer).

1.1. Depot oxidizer to fuel (O:F) mass ratios were a consequence of the propellant state (water or cryogen) at launch. The O:F ratio inherent in water is approximately eight to one (8:1), as each molecule of water contains one atom of Oxygen, with an atomic weight of 16 and two atoms of Hydrogen, each with an atomic weight of 1. [Note that the actual atomic numbers are not integers, due to the presence of various isotopes for each element. The normal stoichiometric ratio for water (O:H2) is 15.9994: 2.01588, close to 7.8:1, but in the interest of simplicity we have used an 8:1 ratio in our analyses for this conceptual study].

An O:F ratio of roughly 6:1 is normally used for chemical H2:O2 propulsion, as this ratio results in a lower combustion temperature and higher exhaust velocity, with corresponding engine life and system performance benefits. The Propellant Production Depot is thus expected to produce surplus Oxygen gas, which could be vented or could potentially be used for industrial space processes, cold gas station-keeping propellant, or for human life support (breathing). For this study, the Propellant Production Depot was assumed to liquify all of the oxygen, and store LOX and LH2 at the 8:1 O:F ratio, whereas the Cryogen Storage Only Depot was assumed to store LOX and LH2 at the 6:1 O:F ratio.

1.2 Propellant quantity requirements are determined by Propellant Depot Mission Requirements. Prospective Depot-supported missions are illustrated below in figure 1-1. The Depot refuels Orbital Maneuvering Vehicles (OMVs) for maneuvers in LEO, such as satellite and payload transfers, satellite servicing and orbital debris removal. The Depot also refuels Orbital Transfer Vehicles (OTVs) for transfer of payloads between LEO and more distant orbits, such as commercial and Government missions to geosynchronous Earth orbit (GEO), Science and Exploration missions to the moon, and large telescope delivery to the Earth/Sun L2 LaGrange point.

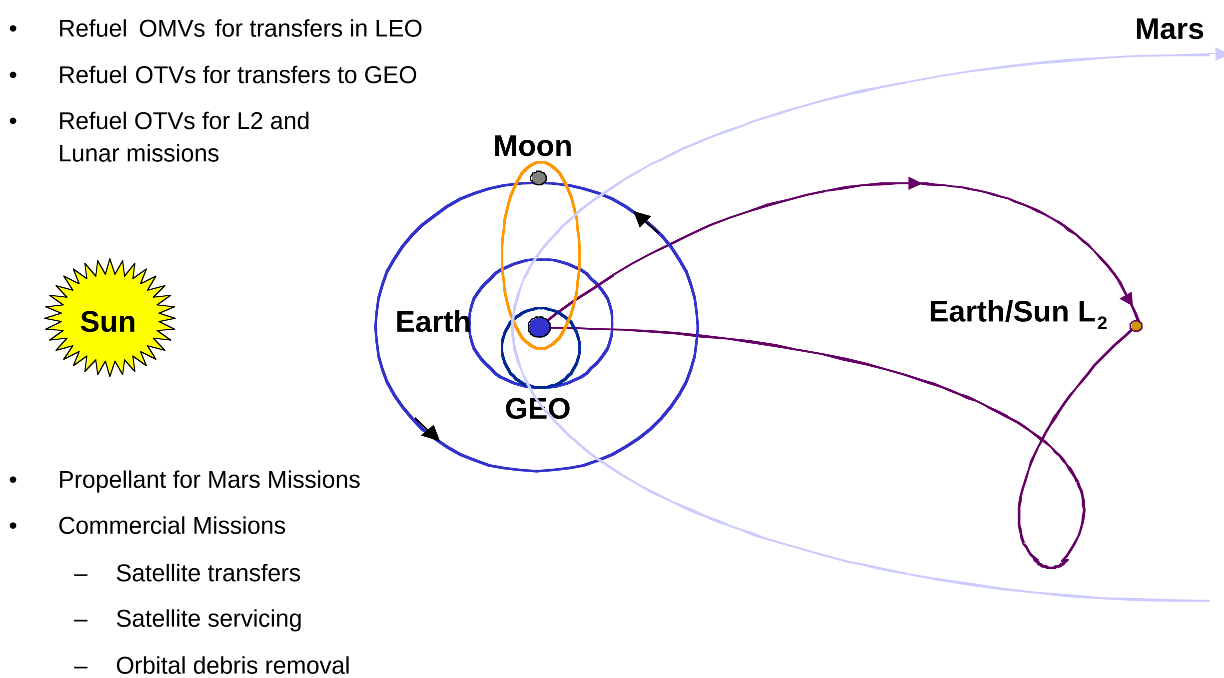


Figure 1-1. Propellant quantity requirements are determined by Depot Mission Requirements.

Depot propellant will also be required to support Mars missions, the most demanding of which is an all-propulsive (Abundant Chemical Propulsion Stage) mission, expected to require roughly 1,000,000 kg of propellant. While this enormous quantity may be reduced in alternate Mars mission scenarios, this requirement was considered in Propellant Production Depot sizing. As the Mars Hohmann Transfer departure window occurs every 2.2 years, approximately 450,000 kg of cryogenic propellant would need to be produced each year to support this mission. For our purposes, the round number of 500,000 kg of was established as a requirement for propellant production per year. As the Mars Transfer Vehicle would be assembled on orbit in advance of the mission, and it could store some of it's cryogenic propellant, prior to launch our Propellant Production Depot storage requirement was assumed to be the 500,000 kg of cryogens produced in one full year of operations.

Considering that a 6:1 O:F ratio is generally used for Oxygen: Hydrogen propulsion; in contrast to the 8:1 O:F ratio for the Propellant Production Depot (PPD), the Cryogen Storage-Only Depot (CSOD) requires a smaller quantity of propellant (the same amount of hydrogen, but less oxygen) to support the same mission model. The corresponding mass of propellant for a 6:1 O:F ratio depot is calculated using the formula:

$$M_{CSOD} = (6 \text{ parts } O2 + 1 \text{ part } H2)/(8 \text{ parts } O2 + 1 \text{ part } H2) \times M_{PPD} = 7/9 \times 500{,}000 \text{ kg} = 389{,}000 \text{ kg}$$

1.3. In-orbit assembly requirements were considered concurrently with analyses of propellant tank size and launch options. The Propellant Production Depot was baselined to use eight (8) tank-sets, each holding 50,000 kg of propellant, whereas the Cryogen Storage Only Depot was baselined to use two (2) tank-sets, each holding 195,000 kg of propellant. Automated on-orbit assembly is assumed to be required for the Propellant Production Depot, as it's large systems cannot be launched together and requirements for manned (EVA) assembly or telerobotic assembly would tend to make the systems heavy and expensive. The assembly approach is one of automated docking of system elements, with prescribed interfaces for power and fluid transfer between the various elements. Cryogenic fluid interfaces do have a tendency to leak, however this technology area must be matured in any case for the transfer of propellants from the Depot to Orbital Transfer Vehicles. The cryogenic storage-only depot is compact enough that two tank-sets can be launched together as a unit, thus it does not require assembly in orbit.

1.4. Propellant tank size was derived based upon the propellant quantity and diameter of available cryogenic propellant tank tooling. In general, because the quantities of propellant are large, large diameter tanks are desired to minimize surface area (heat influx) and mass. The Space Shuttle External Tank (ET) diameter (8 meters) was considered, but the full ET volume was too large for this application, and no practical way was evident to carry shorter ET-derived tanks into orbit. The 5 meter (200 inch) diameter cryogenic oxygen and hydrogen tanks being developed for the Delta IV expendable launch vehicle (ELV) was next in size, and could be launched directly into orbit (see section 1.5). Corresponding propellant tank-set lengths of 12 meters for the Propellant Production Depot, and 30 meters for the Cryogenic Storage-Only Depot were calculated based upon the required propellant mass (see section 1.2) and volume requirements were calculated reserving some volume (5%) for ullage gas and assuming reasonable separations between tanks.

1.5. Depot launches, carrying 5 meter diameter propellant tanks to orbit, were made feasible by using the depot tanks to hold propellants for launch to the final orbit (where they arrive nearly empty). It is assumed that, in the time frame of depot operations, a launch site will be available at the latitude necessary for launching directly into the chosen depot orbit inclination. Figure 1-2 illustrates launch vehicle configurations for the Propellant Production Depot. Depot tank-sets, with an engine, launch in place of a standard Delta IV Heavy cryogenic upper stage. The payload volume, above the full (upper stage) tank-set, is occupied by an additional (empty) tank-set. Other components of the Propellant Production Depot launch using standard Delta IV Heavy ELVs, and the total number of launches required is six. Such a launch system would substantially increase the effective payload capability for Delta IV to LEO (including the Depot tank-set); Payload capabilities would also increase substantially for other destinations of interest to NASA (with an even more dramatic increase for these destinations if the tank-set mass is considered to be part of the payload capability).

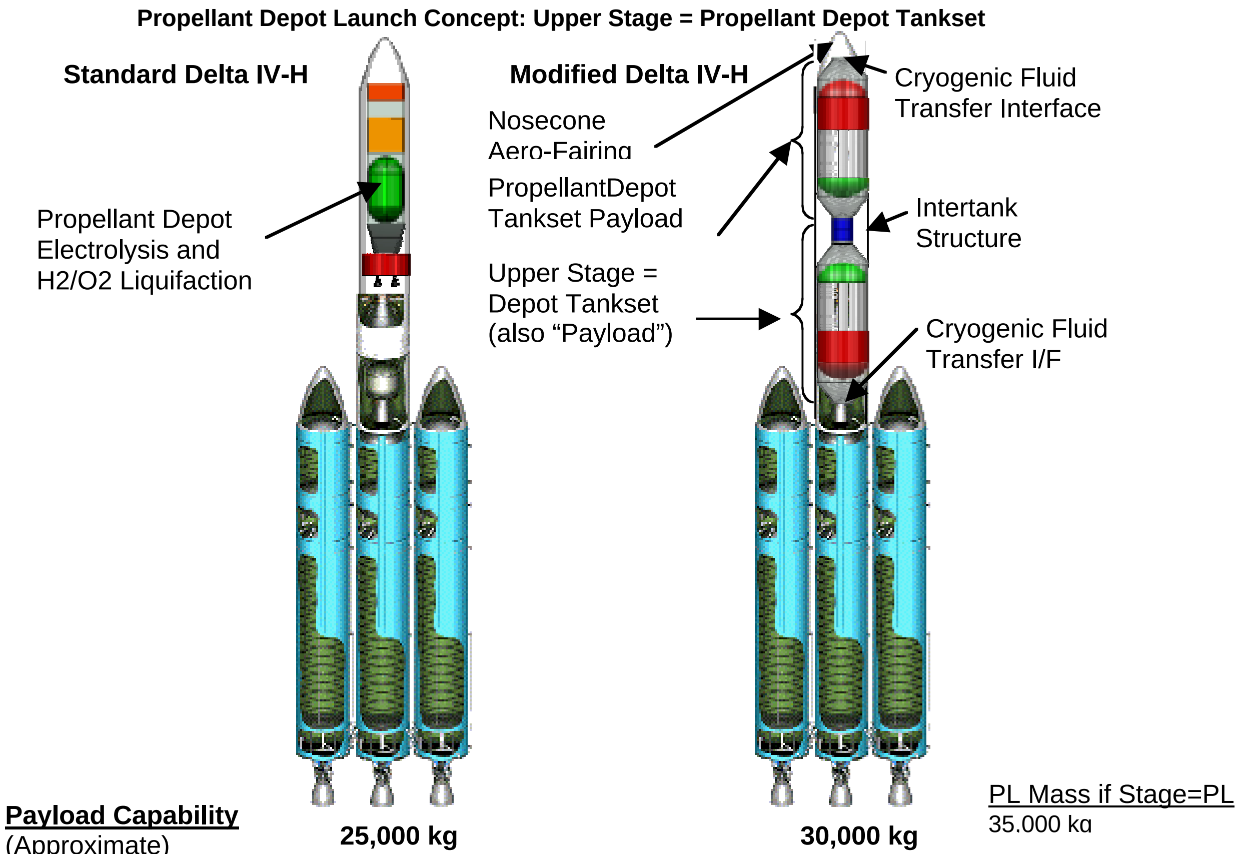


Figure 1-2. Propellant Production Depot tank-sets launch in place of a Delta IV upper stage.

Figure 1-2 illustrates a launch vehicle configuration to carry the entire Cryogenic Storage-Only Depot in one launch, provided that one (or both) of the two large tank-sets carries propellant for launch. When this depot replaces the core stage of the Delta IV-Heavy Expendable Launch Vehicle, it can be launched using either of two strategies:

A) An SSME replaces the RS-68 main engine: Here the main engine might potentially be recovered and returned to Earth for re-use (e.g., on the Space Shuttle).

B) Both the upper and lower tank-sets contain propellant for launch: Here the core propellant load is increased for a longer duration burn.

In either case, the system can carry roughly 60,000 kg into low Earth orbit, more than the weight of the entire (empty) Cryogenic Storage-Only Depot. To circularize at apogee, in low Earth orbit, the system burns approximately 1,000 kg of propellant, rather than requiring a direct injection launch. One method of achieving circularization would be to burn residual ullage gas in the depot tanks through small, low-thrust, pressure-fed gaseous O2-H2 engines, such as the ones Rocketdyne and NASA MSFC developed for Space Station.

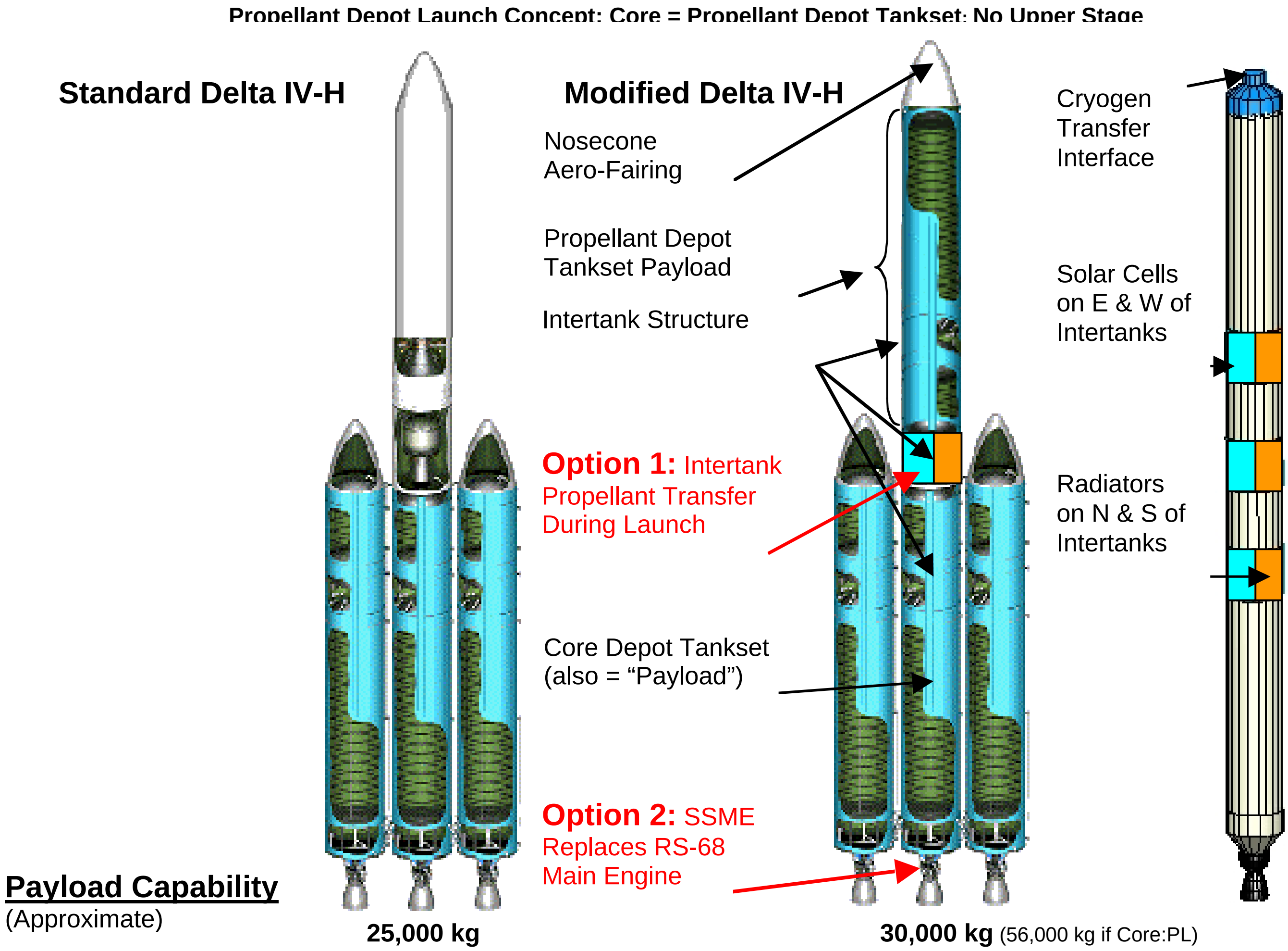


Figure 1-3. Cryogen Storage-Only Depot tank-sets launch in place of the Delta IV core tanks.

1.6. Orbit Altitude was selected to be 400 x 400 km, based on a preliminary analysis that balanced propellant requirements with atmospheric drag. The altitude is high enough that the Depot in the absence of on-board propulsion, would not re-enter the atmosphere in less than two months, even in the worst (2 sigma) case of solar activity (which increases the thickness of upper atmosphere). Related analysis by Dr. Larry Mullins and Craig Cruzen of MSFC is attached as an appendix to this report. A relatively low orbit altitude is desired to minimize the propellant used in OMV retrieval of water payloads or in RLV transportation of cryogens directly from Earth to the Depot (see section 1.15).

1.7. An Equatorial orbit inclination was baselined for both types of Depot because of inherent launch and orbit transfer performance benefits, and other advantages. Equatorial orbit offers benefits for conventional launch, from a launch facility on the Equator, including a slight decrease in delta V required to reach orbit, large expanses of ocean downrange (for range safety) and a launch opportunity every orbit (about every 1.5 hours). For launch of water payloads using a rail-gun or gas-gun, along a fixed track, an equatorial inclination is essential, as it allows transportation of sixteen payloads into orbit every day (16/day), as opposed to one per day

(1/day) for any other orbit inclination. For OTV transportation between the Depot, in an equatorial inclination, and geostationary orbit (GEO, also in an equatorial inclination) the delta V for GEO circularization is reduced substantially. (The required GEO circularization velocity for an OTV departing from a Depot at 0 degrees inclination is about 1 km/s less than that for an OTV departing from the ISS inclination of 51.6 degrees, or about 300 m/s less than that for an OTV departing from 28.5 degrees).

1.8. Depot propulsion propellant requirements are relatively small, compared to the large masses of propellant stored in the Depot for other uses. Related analysis by Dr. Larry Mullins and Craig Cruzen of MSFC, attached as an appendix to this report, assumed that the main propellants LH2 and LOX, were used for chemical propulsion in a periodic re-boost strategy. For the Propellant Production Depot, with plenty of excess electrical power, continuous, low thrust solar electric propulsion (SEP) for drag make-up would require even less propellant, in the form of Xenon/Krypton. Another alternative for this type of depot would be to use the excess O2 as a cold-gas (or potentially, heated gas) propellant. In the case of the Cryogen Storage-Only Depot, H2 is allowed to boil off, without active refrigeration, and this excess gas can be used for cold-gas (or hot-gas) propulsion. Notably, the heat flux and H2 boiloff rate is higher on the sunlit side of the orbit, when atmospheric drag is also higher, which is when propulsion is needed.

1.9. Electrical power requirements were derived for the Propellant Production Depot considering that roughly 500 metric tons of propellant would have to be processed per year. If the conversion process were continuous, this rate would equate to 15.87 grams per second, however the system is simpler and more efficient when electrical power is used for conversion only during the sunlit portion of the orbit. The 400 km equatorial orbit is in sunlight 61.5% of the time, thus the system must convert water at a rate of 25.8 grams per second while it is in sunlight (15.87 g/s / 0.615 = 25.8 g/s).

Power requirements for this process depend upon design details, but, for our design concept, they are separable into the power requirements for electrolysis (482 kWe), for the hydrogen refrigerator's compressor (55 kWe), and for the oxygen refrigerator's compressor (78 kWe). These major contributors combine to form the basic electrical power requirement for propellant production (617 kWe). To account for power management and distribution losses, potential growth, and other systems which will use relatively small power levels (communications and data handling, electric propulsion, etc.) a margin of 15% was added, to reach a total power requirement of 708 kWe while the Propellant Production Depot is in sunlight.

For the Cryogen Storage-Only Depot, the electrical power requirements are much lower. These requirements depend upon design details, and were not assessed in detail.

1.10. Solar array area requirements for the Propellant Production Depot are estimated assuming that NASA's SERT (Space Solar Power Exploratory Research and Technology) program matures related technology sufficiently that an efficiency of 40% will be possible by the year 2015, when the Depot is launched. Considering a slight offset of array pointing (see section 1.11), the area required to supply 708 kWe is approximately 1,660 square meters. For the Cryogen Storage-Only Depot, we surmise that solar array area requirements are small enough that body-mounted solar arrays will be sufficient to supply electrical power for this type of depot.

1.11. Solar array orientation for the Propellant Production Depot is perpendicular to the orbital plane, facing the sun (+/- 22.5 degrees). As the array rotates about an "alpha" axis perpendicular to the orbital plane, the "beta" angle (in equatorial orbit) is not adjusted. The related cosine loss is not significant (for not compensating for this moderate beta angle). When a concentrating trough solar array is used, the trough axes are oriented parallel to the axis of rotation, and the focal length may be adjusted slightly to optimize concentration of sunlight on photovoltaic cells.

For the Cryogen Storage-Only Depot, body-mounted solar arrays are integrated on the East- and West-facing surfaces of selected rigid structures, and solar array orientation is determined by Depot orientation. In equatorial orbit, the East-facing surface generates power for roughly 20 minutes at the beginning of the orbit's passage into sunlight. Similarly, the West-facing surface generates power for about 20 minutes towards the end of the orbit's passage through sunlight.

1.12. Depot orientation was selected through design trades to be fixed with respect to local vertical and the local cardinal points (North South, East and West), using gravity gradient stabilization. The primary reason for is that gravity gradient induced accelerations are sufficient to settle the propellants in large diameter tanks at moderate distances from the center of mass.

1.13. Radiator area requirements were derived for the Propellant Production Depot based upon heat rejection from the propellant production process and other heat generating sub-systems. To reject a total of 290 kWt in thermal energy, a radiator area of 1,560 square meters was needed, with both sides free to radiate. Body-mounted arrays were selected to avoid the complexity of passing working fluids through rotating joints, so only one side of each radiator was free to radiate, thus the area requirement doubled (3,120 m2). Related analysis by Paul Shallhorn, Steve Sutherlin, and Phil Beason of Sverdrup is appended to this report (see Chapter 5).

1.14. Radiator orientation for the Propellant Production Depot, with body-mounted radiators, is on the North- and South-facing surfaces, which never are exposed to direct sunlight at high angles of incidence. In the selected Propellant Production Depot design concept, these surfaces offer a large area for body-mounting. The radiators in this orientation also provide some added protection against orbital debris penetration of the tankage.

1.15. Propellant launch system analyses included options to send a payload from Earth into a low altitude orbit, and an Orbital Maneuvering Vehicle (OMV) to carry the launch system's payload from an initial orbit to the Depot. A variety of water delivery methods are possible depending on the time frame and technology development level for the various systems. For the Propellant Production Depot, water delivery systems could include reusable launch vehicles (RLVs) and gun launch methods (see figures 1-4 and 1-5).

The RLV architecture consists of eight steps from beginning to end. First the RLV delivers a water- or ice-filled tank to an orbit lower than the Depot (nominally 200 km). At 200 km, an OMV captures the RLV's payload and then maneuvers it to the Propellant Depot. Water is then transferred to holding tanks on the depot. The transferring tank is returned to the RLV or is de-orbited and the RLV returns to its launch site. Water is then converted into LH2 and LOX propellants, which are then transferred to OMVs and OTVs and. Finally, OMVs and OTVs transfer to other destinations using this propellant.

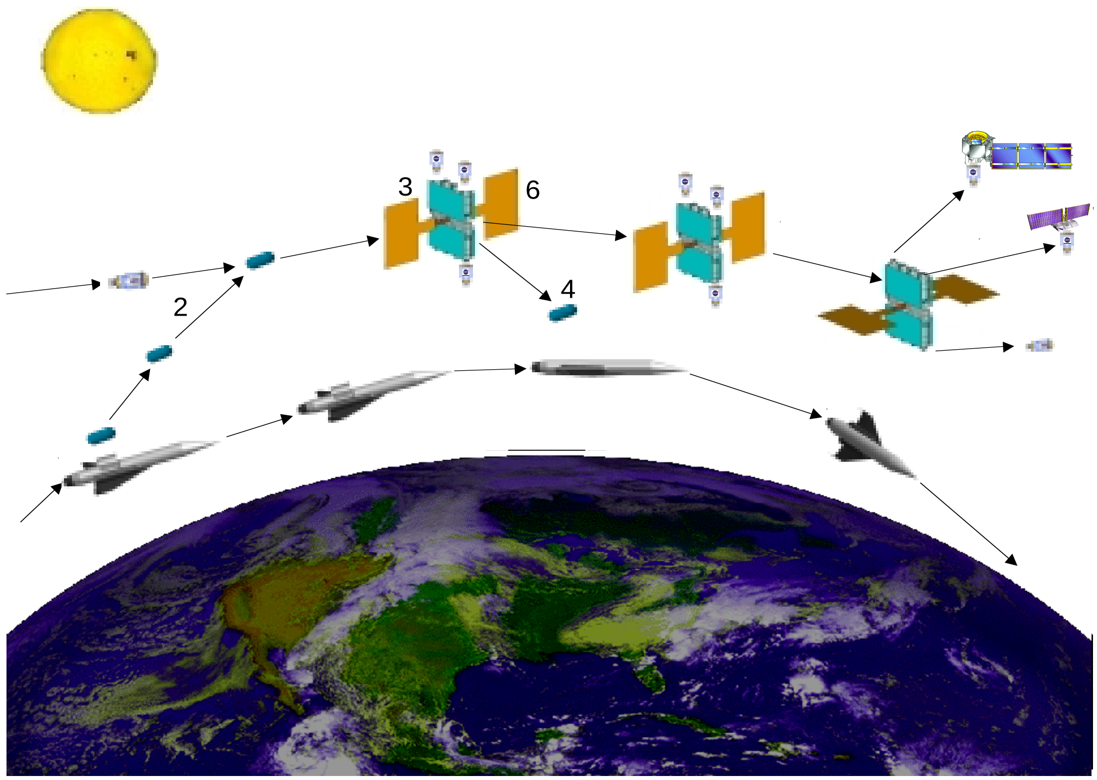


Figure 1-4. RLV launch architecture for Propellant Production Depot

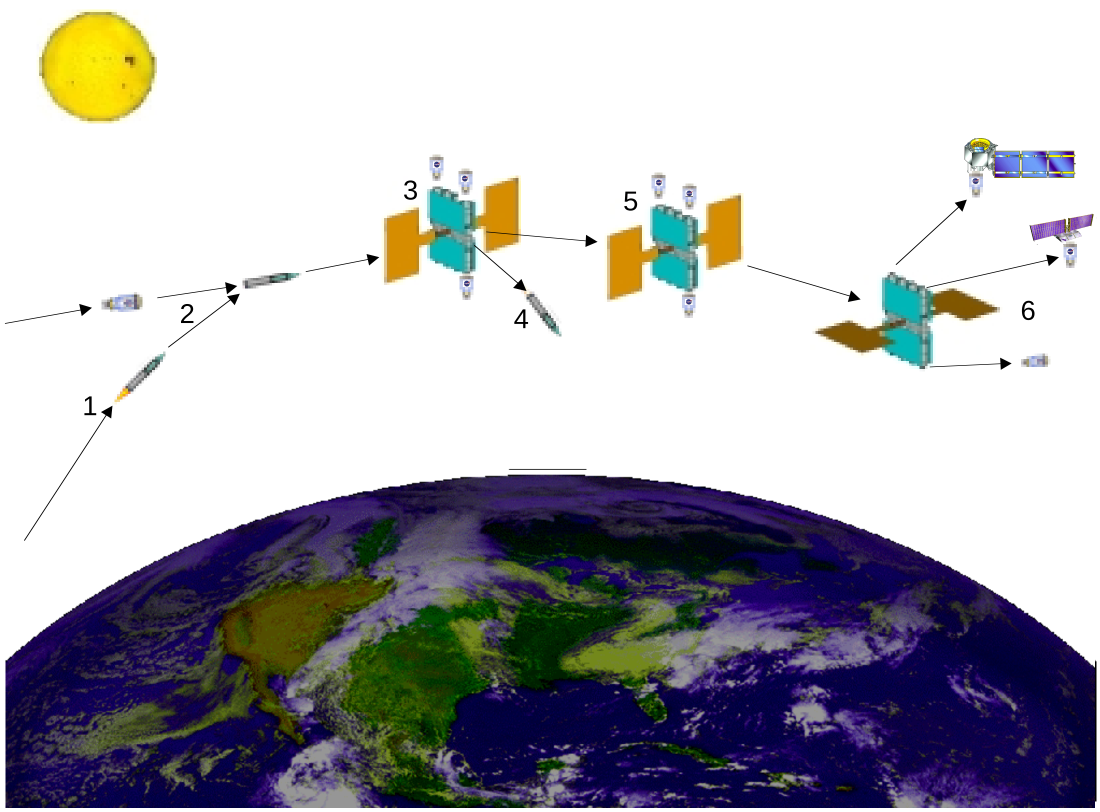


Figure 1-5. Water/Ice projectile Gun-Launch architecture for Propellant Production Depot

Using an RLV, it is anticipated that over 35,000 kg of water would be delivered per launch, requiring roughly 15 launches per year to maintain a baseline propellant production rate of approximately 500,000 kg per year. The OMV propellant requirement is estimated to be 1,266 kg per round trip. OMV propellant analysis by Craig Cruzen of NASA MSFC is included as an appendix to this report. The RLV's large mass is not required to be transported to the Depot orbit and back, and there is no requirement for RLV docking at the Depot.

The Gun-Launch Architecture will complete its mission in six steps. The gun delivers a water projectile to orbit where and OMV captures the projectile and delivers it to the depot. Water is then pulled from the projectile and cracked into LH2 and LOX propellants. Water projectile is then de-orbited and propellants are transferred to OMVs and OTVs. The OMVs and OTVs then transfer to other destinations.

Several gun launch concepts have been considered, including blast wave, "slingatron", and electromagnetic methods. Each delivery method uses an ice filled projectile with a circularization stage. The projectiles measure approximately 1m in diameter by 10m in length and contain <u>250 kg to 500 kg of water-ice</u>, which would require 1000-2000 launches per year (i.e., roughly 3-6 launches per day), to meet the Depot's 500,000 kg per year requirement, excluding any propellant required for OMV operations.

Each projectile is launched to a target orbit at <u>25 km to 75 km below the depot</u>. It is required to have on-board propulsion for an apogee burn to circularize the orbit altitude. A reusable OMV performs rendezvous maneuvers to collect projectiles and deliver them to the depot. The water-ice in the projectile is heated and pumped into storage tanks on the depot.

<u>6 to 9 OMVs are required</u> to deliver 4 to 8 projectiles per day to the depot to maintain a 2000 kg per day accumulation rate. About 25% of the water will be required for use as cryogenic propellants to fuel the OMVs. The remaining 75% will be accumulated at the depot for cryogen production, storage, and delivery to other vehicles.

For the Cryogen Storage-Only Depot, a different type of RLV is considered. In this case, the RLV does not have a separate payload bay, but carries it's payload of cryogenic propellants within it's own cryogenic propellant tanks. An OMV docks with the RLV, in 200 km orbit and gravity-gradient orientation, using an "R-bar" approach. Once attached, it off-loads all propellants that are not needed for the RLVs return to Earth (including flight performance reserves that would otherwise be wasted). Then the OMV returns to the Depot and tranfers propellants into the Depots cryogenic tanks.

1.16. OMV propulsion has been assumed to use oxygen and hydrogen propellants, as these propellants are readily available and offer moderate to high thrust, for rapid maneuvers.

1.17. OMV docking ports for the Propellant Production Depot are at the altitude of the center of mass of the Depot, near the power supply and electrolysis/liquifaction unit(s). In this position, a "V-bar" docking approach is used. Changing the amount of mass (OMV and its payload) at this location will not change the altitude of the center of mass of the (balanced) Depot, and will not effect cryogenic propellant settling by gravity gradient forces. With only slight gravity gradient settling forces at this location, the water is transferred using bladder expulsion tanks.

1.18. Water tank volume for the Propellant Production Depot, to receive the contents of a full tank of water (35,500 kg) delivered by the OMV, is required to be 35.5 cubic meters. Each of two units is required to be near the altitude of the center of mass of the Depot, near the power supply and electrolysis/liquifaction unit(s). Changing the amount of mass (water) at this location will not change the altitude of the center of mass of the (balanced) Depot, and will not effect cryogenic propellant settling by gravity gradient forces. Water is transferred using bladder expulsion tanks.

1.19. Electrolysis unit volume, for each of two units on the Propellant Production Depot, is estimated to be occupy a volume of eleven cubic meters (3.1 m x 2.2 m x 1.6 m), based upon linear scaling of a commercially available unit (IMET-60; www.hydrogensystems.com). It is required to be near the power and water supplies as well as liquification units and radiators.

1.20. Liquifaction units for the Propellant Production Depot have different volume requirements for each of the two hydrogen and oxygen units. Based on published data (Kahout), cryocooler mass is estimated to be 655 kg for oxygen and 1,039 kg for hydrogen. Based on a density of 213 kg per cubic meter, the corresponding volumes are roughly 3 cubic meters for each oxygen unit and 5 cubic meters for each hydrogen unit (cubes of 1.5 m /side and 1.7 m/side, respectively). They are required to be near the power supply as well as the electrolysis units and radiators.

SYSTEM DESIGNS

Two Propellant Depot system designs are evaluated in the following chapter. The first of these, a Water-to-Cryogen Conversion and Storage platform, uses advanced Space Solar Power technology. The second system design, a Cryogen Storage-Only platform, does not require significant advances in SSP technology, but cannot take advantage of inexpensive, safe transport of water to earth orbit (or extra-terrestrial water resources). In addition, potential related future commercial infrastructures are assessed in this chapter.

2.1 Water-to-Cryogen Conversion and Storage Depot

The system design for a water-to-cryogen conversion and storage depot (also known as the Propellant Production Depot) was defined through a process of configuration trade studies which considered spinning, rotating, and gravity gradient propellant settling approaches and culminated in a selected Propellant Production Depot design concept. Key subsystems for this Propellant Production Depot were assessed and analyzed, including the following:

1. Liquid oxygen and liquid hydrogen storage tanks,
2. Liquid oxygen and liquid hydrogen transfer interface,
3. Thermal control subsystem (radiators),
4. Solar arrays,
5. Water transfer interface,
6. Water storage tanks, and
7. Electrolysis subsystem

An integrated system approach was considered for micro-meteoroid and orbital debris shielding, and a preliminary Propellant Production Depot mass estimate was developed, along with a corresponding launch and assembly scenario.

<u>Platform Trade Studies</u>

Several alternative concepts were considered for propellant settling, including rotating, solar inertial, and gravity-gradient facilities. These general approaches are illustrated in figure 2.1-1.

The rotating depot consists of two, three, or more tanks arranged in a spoke-like fashion that rotate around a central hub to generate g-force through a centrifugal effect. It therefore does not need the presence of a planetary gravitational field for propellant settling. However, it requires a de-spun docking port.

The gravity gradient depot utilizes the change in a planet's gravity with altitude for stabilization and propellant settling. As a gravity gradient oriented depot orbit rotates (once per orbit, while keeping the same orientation with respect to the Earth) roughly one third of the “gravity gradient force” is actually due to system rotation. Docking can take place at the ends or center.

A solar inertial depot maintains a fixed orientation with respect to the sun. Its advantage is that it does not require alpha/beta joints for its solar arrays to track the sun. Its disadvantage is that it requires zero-g cryogenic propellant acquisition, a technology that has been studied extensively, but has not yet been demonstrated in space.

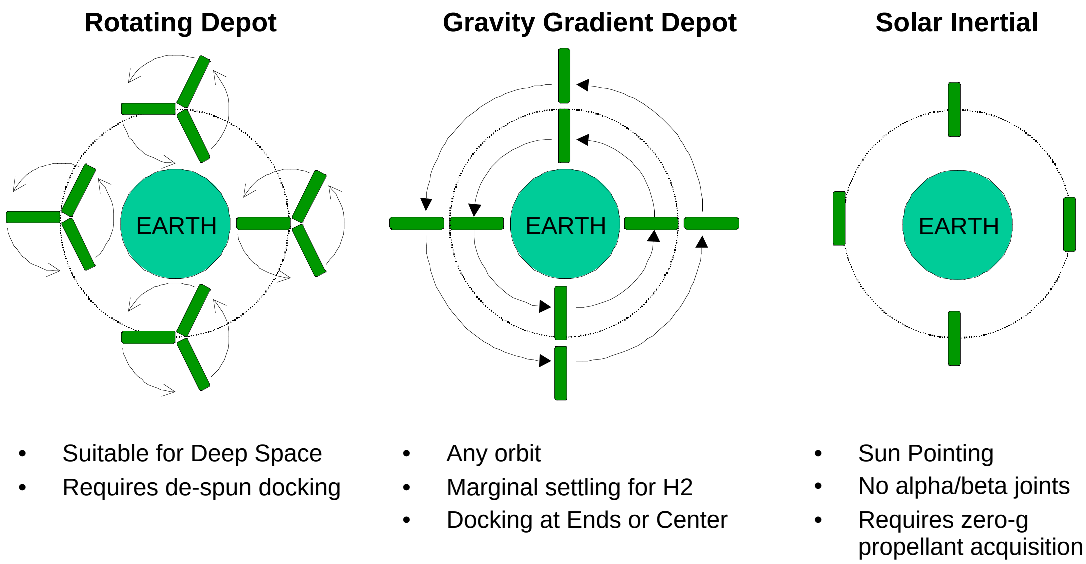


Figure 2.1-1. Several approaches were considered for cryogenic propellant settling.

Preliminary concepts were defined for Propellant Production Depots using these three propellant acquisition approaches. Figure 2.1-2 illustrates these preliminary concepts, as well as a more refined system design for the selected approach, which uses gravity gradient forces for propellant acquisition.

The upper left illustration shows a preliminary spinning system design concept. It's long tanks plus solar arrays form a permanent LEO facility, while short "bee-like" tanks transfer propellant to other customers, and could be building blocks for planetary transfer vehicles. The long tanks, short "bee" tanks, and docking port are spun-up during propellant transfer, and de-spun for docking and undocking maneuvers. The sun-oriented "abacus" solar collector remains de-spun, and therefore requires power transferred across a slip ring between the collectors and the spinning hub.

The upper right illustration shows a quasi-inertial configuration that remains in a fixed orientation with respect to the sun. This concept places the tanks in a semi-shaded position behind the arrays. It has the potential for being quasi-inertially stable, it maximizes shading of the tanks with minimum solar tracking of the arrays, and has a minimum number of power transfer joints.

The lower left illustration shows a preliminary gravity gradient concept, consisting of four large cryogenic propellant tank-sets and a pair of solar arrays. The large diameter of it's propellant tanks reduces the effects of surface tension, which would cause propellant to cling to the walls of small tanks in a low gravity environment. Such large tanks, based on the Delta IV-H expendable launch vehicle core, were also considered for a Cryogen Storage-Only Depot (see section 2.2 of this chapter).

The lower right illustration shows the reference gravity gradient Propellant Production Depot. It will be discussed in more detail in the following pages.

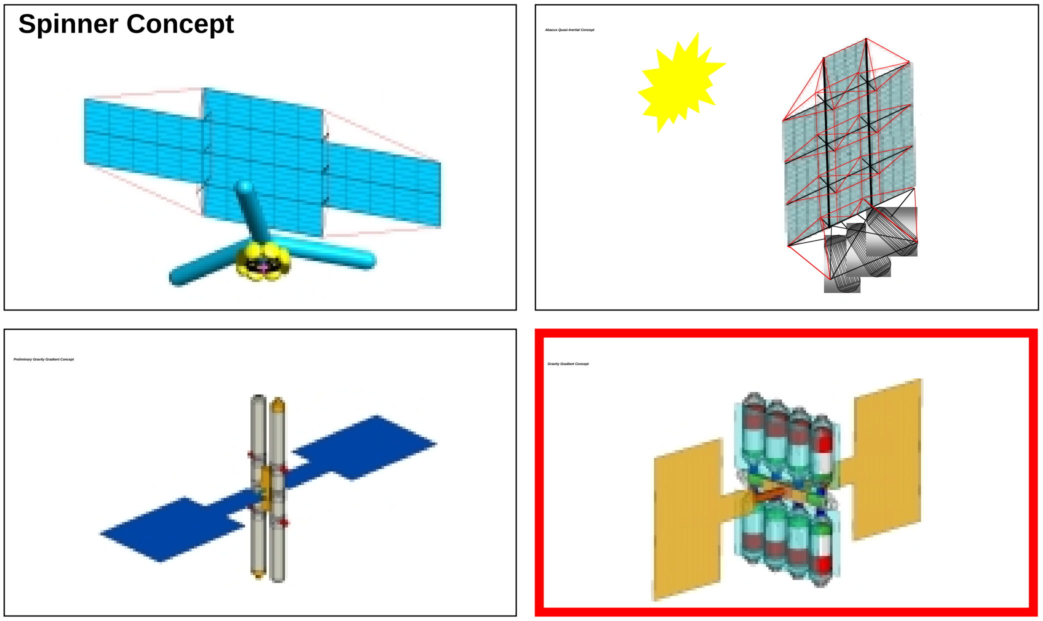


Figure 2.1-2. System concepts were considered for three propellant settling approaches. The selected gravity gradient conceptual approach was then refined into a system design.

Propellant Settling Analysis; Selection of Gravity Gradient Approach

At the center of gravity of an object (i.e., the Propellant Depot) in orbit, gravitational force downward balances the centrifugal "force" upward. Above this, centrifugal "force" dominates, settling liquid propellants upward. Below this, gravitational force dominates, settling liquid propellants downward. The net settling force is proportional to the distance above or below the center of gravity. Figure 2.1-3 illustrates these forces on a cryogenic propellant tank in orbit.

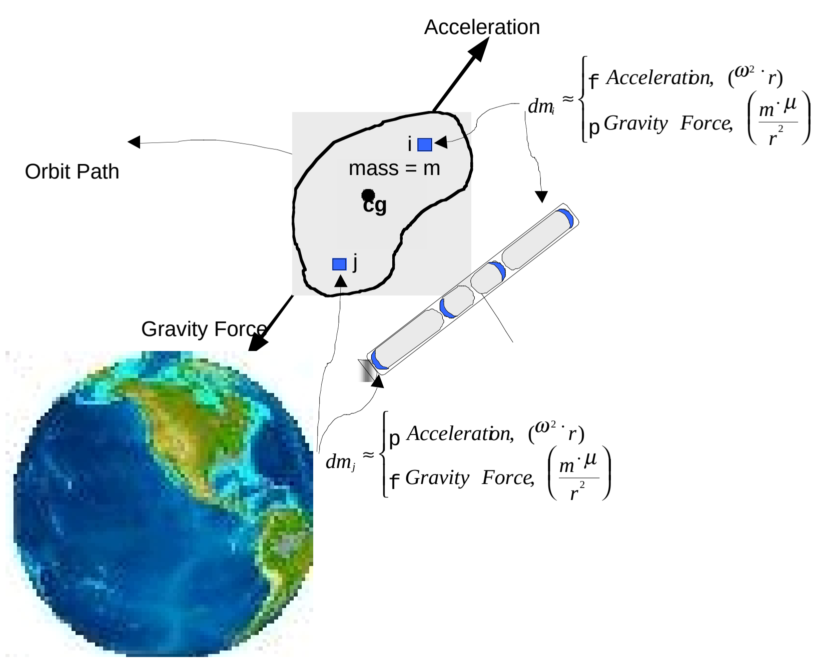


Figure 2.1-3. Gravity-gradient forces can settle cryogenic propellants in large tanks.

This relatively simple settling technique is adequate for large diameter tanks, which the Depot requires to contain large quantities of cryogenic propellants. For a 5-meter diameter tank in a vertical, gravity-gradient orientation, approximately 4 microgees are required to settle LH2; corresponding to a liquid hydrogen/ullage gas interface that is 10 meters from the center of mass of the Depot. A 5-meter diameter LOX tank requires approximately 2 microgees for settling, so its liquid/ullage gas interface need only be ~5 meters from the center of mass of the Depot. To ensure liquid propellant acquisition for propellant transfer, it is only critical that the far end (outflow) of the tank to have this distance from the center of mass. This analysis is based upon an assessment of the "Bond number" or ratio of settling force to surface tension in the propellant. Related analysis is appended in Chapter 5 of this report.

Propellant Production Depot System Design Overview

The Propellant Production Depot system design configuration is illustrated in figure 2.1-4. Its gravity gradient configuration allows for propellant settling, while the in-line configuration minimizes drag. Seven key subsystems are called out in this illustration, and each of these is addressed in the following pages. LOX/LH2 storage tank sets are mounted in eight locations, and each contains docking port for Orbital Transfer Vehicle propellant transfer. Radiators are mounted flush with the tanks, which also minimizes drag. The solar arrays rotate once per orbit to track the sun. Two docking ports are provided for orbital maneuvering vehicles to transfer water on the central structure, which also contains water storage tanks and an electrolysis system.

General system design features of the Propellant Production Depot are illustrated in figure 2.1-5. This Depot is assumed to become an operational in the year 2015, and thus it is designed to use many of the Space Solar Power technology advancements that are planned over the next decade. The system is expected to utilize SSP-related advancements in the areas of solar power generation, power management and distribution (PMAD), advanced structures, robotics, and propulsion. The power system is sized with two large "abacus"-type arrays that produce over 700 kilowatts of power at 150 volts. As the arrays rotate with respect to the gravity-gradient-oriented part of the system, this power is transferred across slip rings. Advanced, inflatable structures are used on the solar array and advanced composites are used in other supporting structure. Robotic operations are anticipated for assembly, vehicle docking, propellant transfer, and maintenance functions. Solar-electric propulsion and control moment gyros are shown as the choices for controlling the altitude and attitude of the Propellant Production Depot.

Many of these subsystem technology choices could be revisited in further study. For example, recent analyses of the Propellant Production Depot propulsion system (see the appendix) have considered the alternative of a higher thrust oxygen-hydrogen system that utilizes the propellant produced on the Depot. Prior Space Station advanced technology development produced small H2/O2 thrusters, designed with this approach in mind, performed very well in burning H2 and O2 gas at an 8:1 oxidizer to fuel mixture ratio. While this alternative, with a lower Isp, would more propellant, the difference may be insignificant in a system of this size, where the propellant is delivered to orbit at low cost. This is just one example of a subsystem detail that is outside the scope of the current activity, but could be refined through further study.

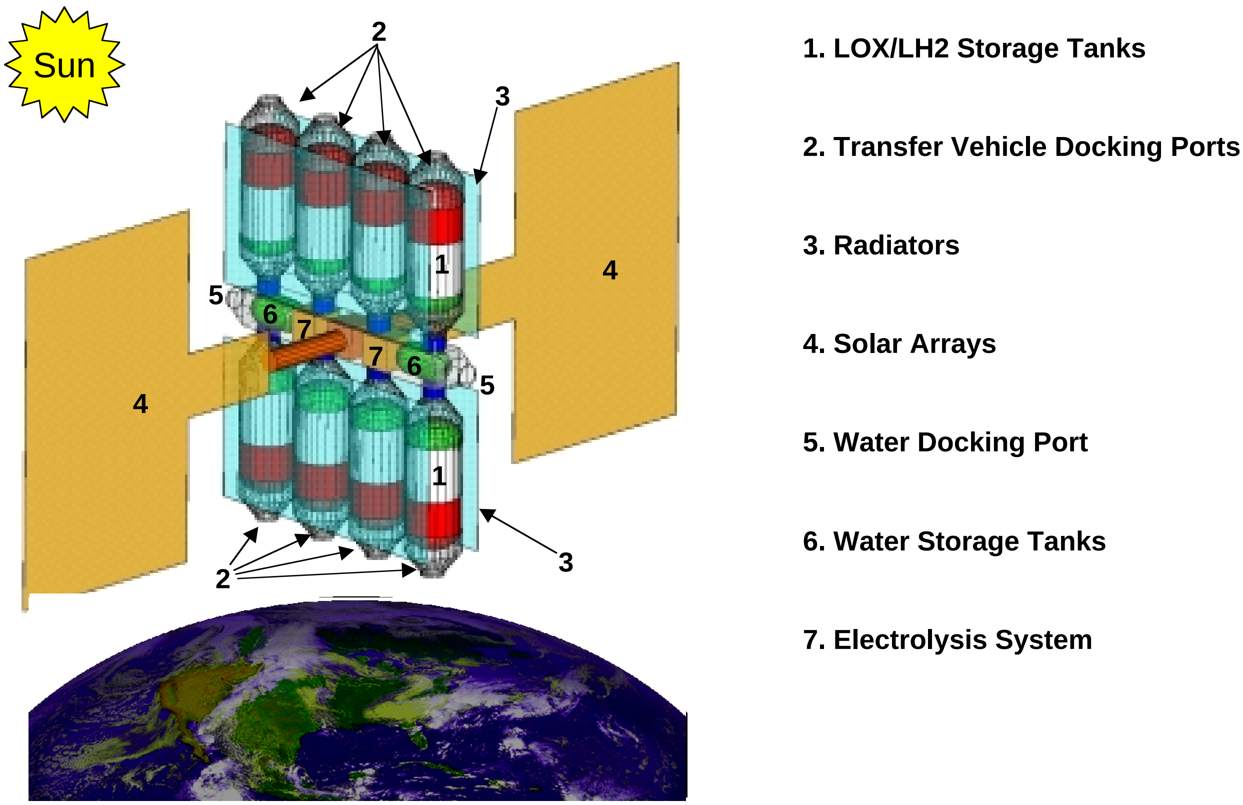


Figure 2.1-4. The Propellant Production Depot design includes seven key subsystems.

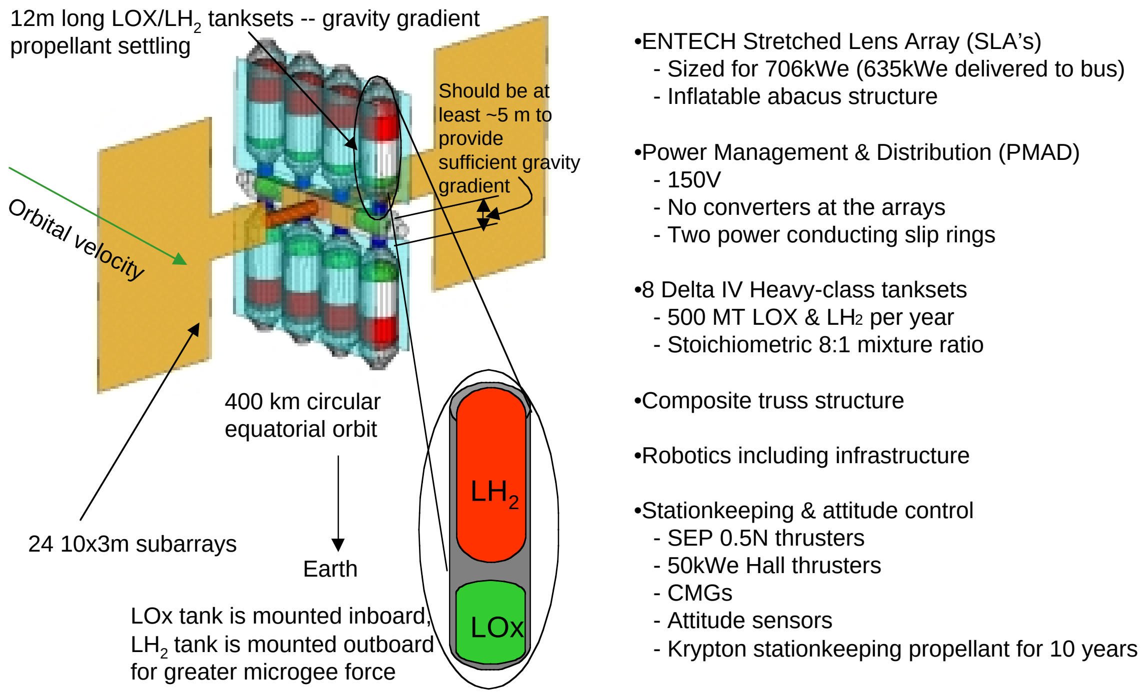


Figure 2.2-5. System Design Features of the Propellant Production Depot

Key Subsystem Assessments and Analyses

2.1.1 Liquid Oxygen and Hydrogen Tanks

Liquid oxygen and liquid hydrogen (LOX and LH2) are stored in large tanks, designed based on the large diameter (5.1 meters or 200 inches) LOX and LH2 tanks developed for the Delta IV expendable launch vehicle. The size of each tank-set allows it to be launched on the Delta IV-H as an upper stage, containing the propellant it requires to reach orbit (along with a substantial additional payload mass). Eight such tank-sets are required to store 500 MT of LOX and LH2 propellants. The 12-m length of the tank-sets, combined with a 5-m spacing provided by the central structure provides sufficient gravity gradient for propellant settling. LH2 requires a greater amount of settling force than LOX, so the LH2 tanks are mounted outboard from the central structure (further away from the center of mass).

The Delta IV Heavy Cryogenic Upper Stage LH2 tank illustrated in figure 2.1-6 is being produced by Boeing in Decatur, Alabama. This tank is composed of an isogrid (load-carrying) cylindrical section, structural interface flanges, forward and aft domes, a cover, sump, tunnel and system support tray. The tank also has a level sensor mast, which provides propellant measurement capability. For Delta IV, it uses a foam insulation, but an aero-thermal shield system has also been designed for it, which is more similar to the shielding system envisioned for Depot tanks.

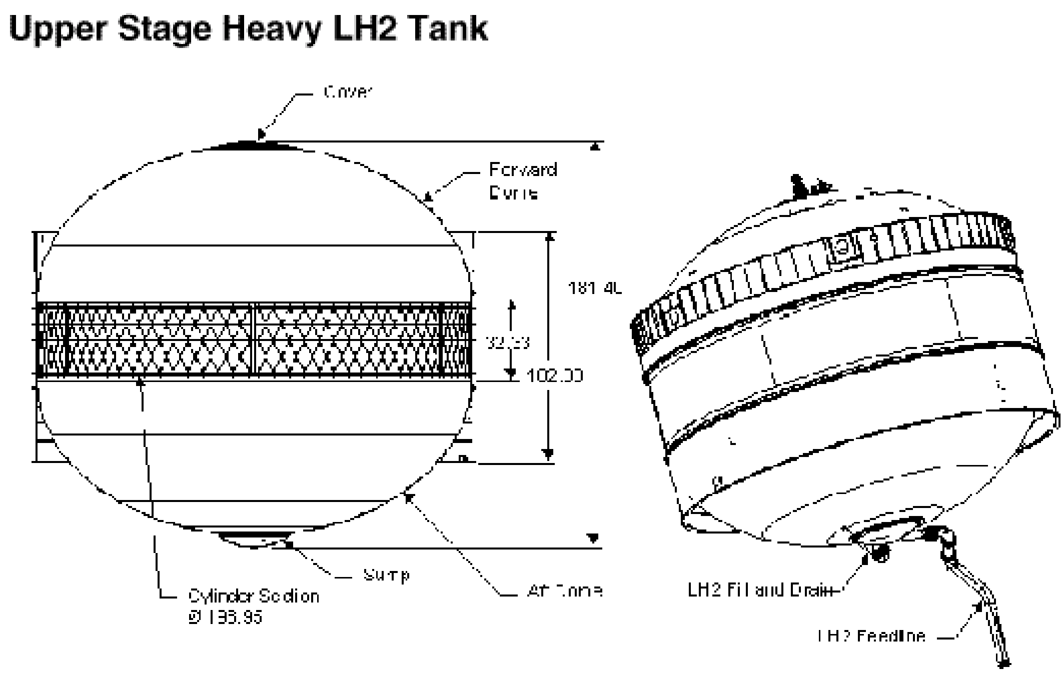


Figure 2.1-6. The Delta IV Heavy Cryogenic Upper Stage Hydrogen Tank

The tank-set illustrated in figure 2.1-7 is based on the Delta IV tank design, using the same 5.1 meter diameter, but changing the length of the cylindrical sections, and adding a propulsion system for autonomous transport into orbit (launched as an upper stage for an ELV). The LOX tank is shown on top, which is a change from the current Delta III and Delta IV upper stages. (Note: more detailed analysis of propellant settling requirements might allow the large Hydrogen tank to be located in its conventional position, on top). Eight such tanksets can store 500,000 kg of propellant at the stoichiometric 8:1 LOX/LH2 mass ratio characteristic of a Propellant Production Depot. Alternatively, the tank-sets could be sized for a 6:1 mixture ratio common to most LOX/LH2 engines, with the excess LOX vented, used for propulsion, or stored in a separate tank, and used later for other functions (e.g. human life support).

- **Total length per tankset** = 3.80 m + 6.8 m + 1.6 m = **12.2 m**.
- **Total propellant mass stored per tankset** = 60,914 kg + 7614 kg **= 68,528 kg**
- **Number of tanksets needed** to meet 500 MT depot requirement ≥ 500/68.528 = 7.3 ~ **8**.

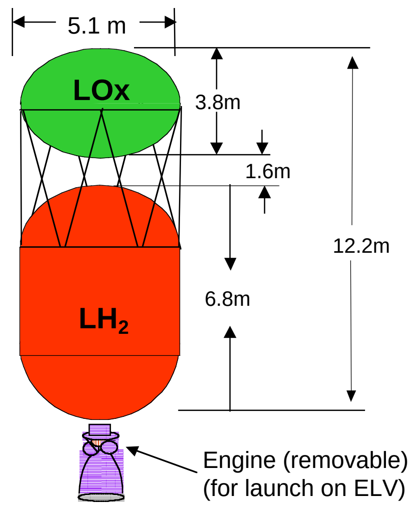


Figure 2.1-7. Propellant Production Depot tanks are launched as ELV upper stages.

2.1.2 LOX and LH2 Transfer Interfaces

The interface for transferring cryogenic propellants from the Depot into Orbital Transfer Vehicles and Orbital Maneuvering Vehicles (OTVs and OMVs) is assumed to be the same interface used for propellant transfer to the engine. OTV studies managed by NASA MSFC in the 1980s included conceptual designs for robotic engine removal and replacement, and this general type of design is assumed to have dual utility for propellant transfer. After the tank reaches orbit and is integrated with the rest of the Depot, the upper stage engine is removed and stored as a replacement engine for OTV and OMV applications. When an OTV or OMV needs to be refueled, it docks (or berths) with this propellant transfer interface and uses the “engine propellant feed” lines for liquid transfer from the Depot tanks, and the “engine autogenous pressurization” lines for ullage gas transfer back into the Depot tanks.

2.1.3 Thermal Control Subsystem

For the Propellant Production Depot, the driving requirement for thermal control is rejection of the heat generated in water electrolysis and propellant liquification. Table 2.1-1 summarizes the temperature of key subsystems for alternative radiator configurations to meet estimated heat rejection requirements. The body mounted option is preferred for several reasons.

Table 2.1-1. Heat rejection requirements can be met by several alternative radiator locations

Radiator Location	Tankset Housing Temp. (K)	Process Equipment Housing Temp. (K)	Equipment Plate Temp. (K)	Solar Array Temp. (K)	Radiator Temp. (K)
Body-Mounted Option	233	274	294	261	268/260
On Solar Arrays	224	273	293	256	279
Axial	218	270	290	251	275
Axial, Articulated	218	270	290	253	272
Lateral	216	269	289	240	275
Lateral, Articulated	216	269	289	253	273

HEAT REJECTION REQUIREMENTS

Source	*Rate (kW)*	*Rejection Temp. (K)*
Electrolyzer	72.3	339
Dryer	2.0	339
O_2 Cryocooler	68.2	339
H_2 Precooler	9.2	339
H_2 Cryocooler	49.7	339
Other	188.3	339

MANAGEMENT CONCEPT

Radiators		
Body-Mounted Option	1560 m^2 effective area	α/ε - 0.1/0.9
Performance (β = 23.44°)	390 kW/0 kW (day/night)	220 K - 295 K (extremes)
Cooling Loop		
Capillary-Pumped Loop	Possible active assist.	
Ammonia (NH_3)	435. psia maximum	

Several of the thermal radiator configurations considered are illustrated in figure 2.1-8. The configuration in the lower left has one large radiator at the rear of the depot, which eliminates one position for an OMV docking port. The configuration in the lower right divides the radiator into fore and aft segments, which eliminates both OMV docking ports (i.e., water would have to be delivered at the cryogenic propellant transfer interfaces at the end(s) of the tanksets). The upper configuration uses a body-mounted radiator, which allows for OMV docking at the both ends of the center structure, and also provides some added debris shielding for the tanks. Radiator temperatures are only slightly higher for this configuration, and it is the option selected for the system design. Figure 2.1-9 provides a thermal map of the body-mounted configuration.

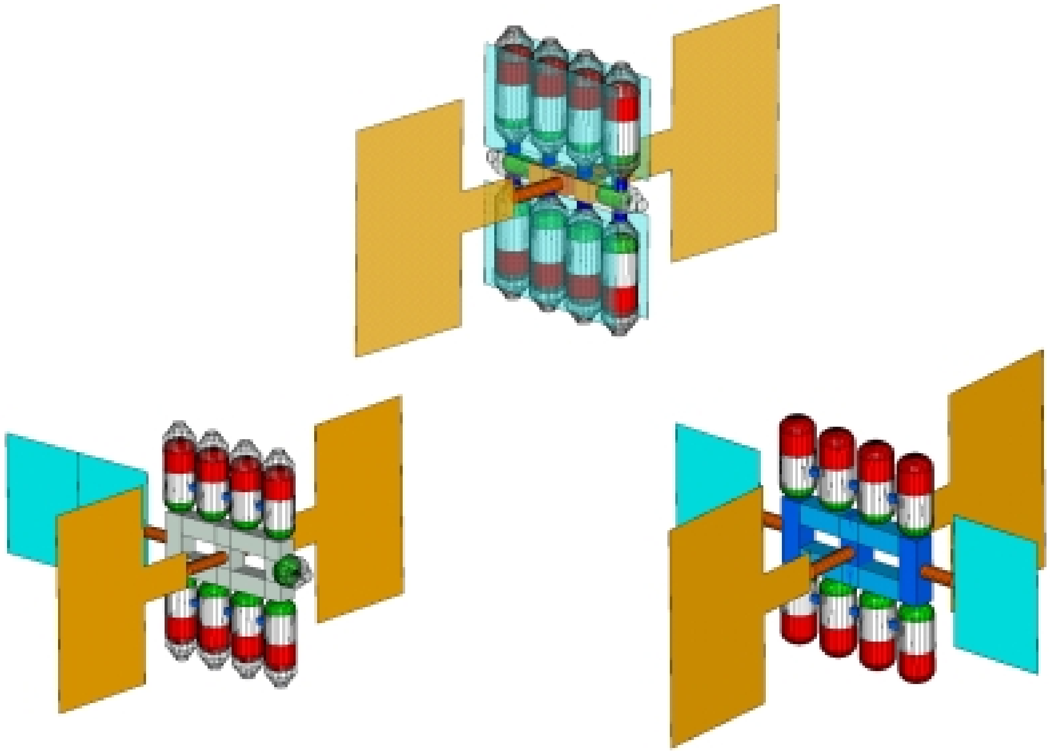


Figure 2.1-8. Three alternative thermal radiator configurations were studied in detail

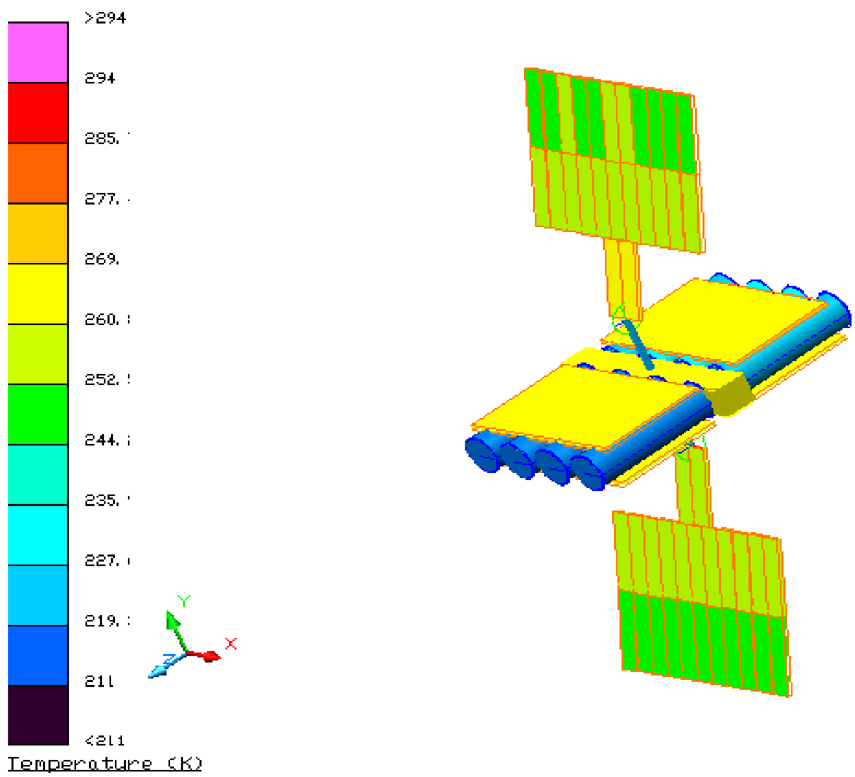


Figure 2.1-9. Body-mounted radiators were selected for the Propellant Production Depot.

Body-mounted radiators have a sufficient view factor to space to maintain the required heat rejection rates, while ensuring that temperatures of the solar arrays and equipment are held at levels that allow for efficient operation. Figure 2.1-11 summarizes thermal conditions versus time for body-mounted radiators. Placing the radiators on the solar arrays could save structural mass and provide a more compact structure, but articulating an axial or lateral radiator would introduce a high degree of system complexity, and provide no significant temperature advantage over non-articulated configurations. Body-mounted radiators also save structural mass, while providing some debris shielding for the tanks and equipment. Temperatures are slightly higher than for axial or lateral arrays, but are still within acceptable limits. Tank-set housing temperature is somewhat higher than for the other options, but is still within acceptable limits, and this option is therefore selected for the Propellant Production Depot.

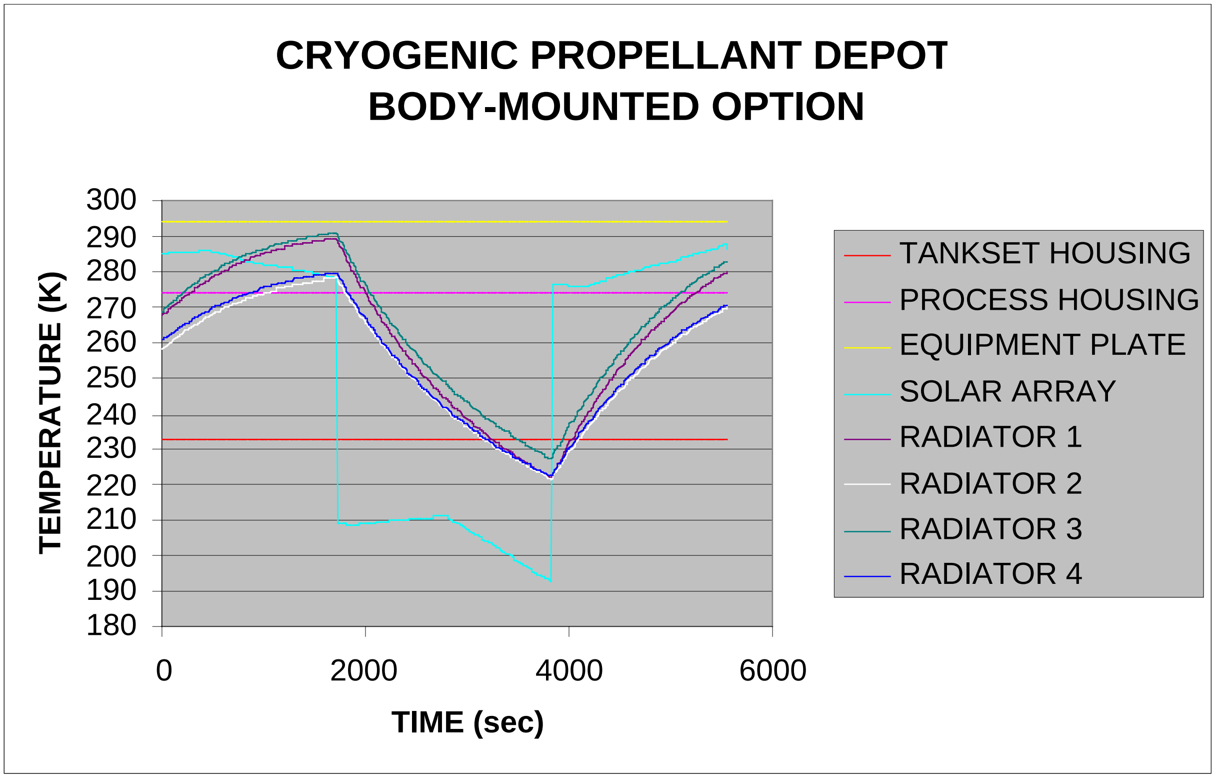


Figure 2.1-10 Thermal Analysis of body-mounted radiators indicates they are sufficient.

2.1.4 Solar Arrays

The size and shape of propellant Production Depot solar arrays allows room for docking of the OMVs and OTVs, while providing sufficient power and sun tracking to support conversion and storage of propellant. A power level of 635 kWe delivered to the bus is sufficient for pumping and electrolyzing water, and cooling the H2 and O2 (see section 2.1.7). Other functions and losses in the system bring the total power requirement to 708 kWe while the system is in full sunlight.

The depot receives power from solar cells on an ENTECH Stretched Lens Array (SLA), as illustrated in figure 2.1-11. Near-term performance for this type of array is expected to reach 300 W/m^2 and 170 W/kg. In order to minimize system complexity, beta angle tracking is not used on the propellant production depot. Instead the focal length of the lens can be adjusted slightly by rocking the lens back and forth over to optimize the concentration of light on a central strip of solar cells beneath each stretched lens in the array.

Figure 2.1-11. ENTECH stretched lens arrays can supply power for propellant production.

2.1-5. Water Transfer Interface

Water transfer is expected to be relatively simple in comparison to transfer of cryogenic propellants. A bladder positive-expulsion system allows transfer of water from OMVs to the Depot, so the system can operate in the absence of any settling force. Water transfer will use automated interfaces and operations similar in nature to proven for storable propellant resupply from Progress to the Mir space station.

2.1.6 Water Storage Tanks

Water storage tanks also use a bladder positive-expulsion system to send water through the electrolysis unit. Each tank is designed to contain 20,000 kg of water, the full load carried by an OMV. The tank is designed for an operating pressure of 34,500 Pascals (50 psia). While system thermal control should be sufficient to prevent water freezing (or boiling) the ullage volume is designed to allow enough gas behind the bladder to absorb contingency pressure changes.

2.1.7 Conversion into Cryogenic Propellant via Electrolysis

Steps in the electrolysis process are illustrated in figure 2.1-12. Water is received in the liquid state and pre-heated to 10 K above it's freezing point. An electrically driven, positive displacement pump circulates the water from the pre-heater and receiver to the electrolyzer, which converts water into oxygen and hydrogen gas streams. A small fraction of water (< .01 percent) is assumed to leak past the proton exchange membrane in the electrolyzer, but a water separator, which is an integral part of the electrolyzer, removes this water from each of the streams and circulates it back to the inlet of the electrolyzer. Heat is removed from the hydrogen stream and radiated to space, then the chilled hydrogen stream is liquefied by the hydrogen liquefier, and liquid hydrogen is routed to storage. An electrically driven, liquid hydrogen re-circulation pump (not shown) may be provided to circulate the liquid hydrogen between the storage tank and liquefier as required to counteract boil-off. The oxygen stream is liquefied directly by the oxygen liquefier, and the liquid oxygen is routed to oxygen storage. Liquid oxygen re-circulation may also be provided to circulate liquid oxygen between the storage tank and liquefier to counteract boil-off, but LOX is produced in excess (at an 8:1 ratio vs the 6:1 ratio commonly used for propulsion) so this is not as important as it is for LH2 storage. Additional heat pipe radiators reject waste heat from the electrolyzer and liquefiers.

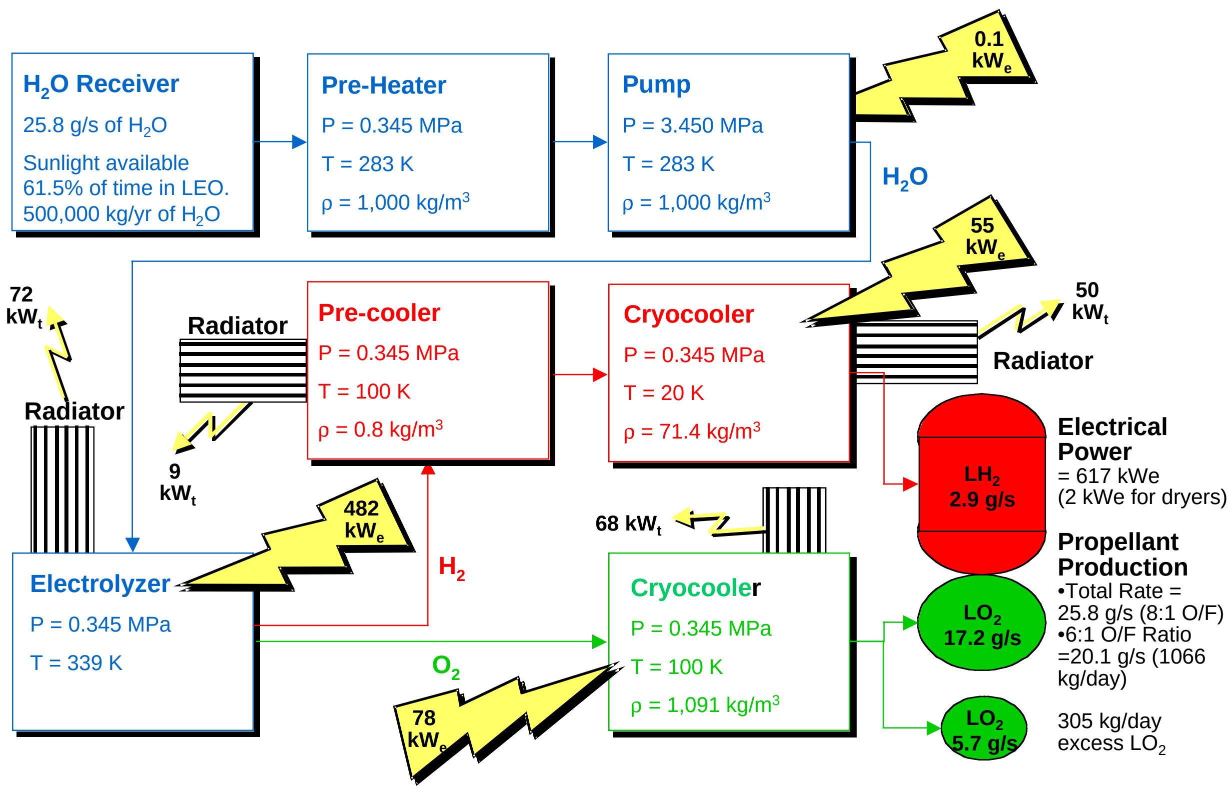


Figure 2.1-12. Electrolysis uses 617 kWe to produce more than a ton of propellant per day.

State points for the various steps in the electrolysis process are summarized in Table 2.1-2. Water is received in the liquid state at near vacuum conditions and a temperature of 273 K. The water is pre-heated to 283 K, using direct solar thermal energy or an electric heater. A nominal pressure of 0.345 MPa (50 psia) is assumed for the closed system. An electrically driven, positive displacement pump circulates the water from the pre-heater and receiver to the electrolyzer. The positive displacement pump is sized to provide a peak pressure of 3.450 MPa (500 psia), to account for the delta-pressures of each of the system components. A heat rejection temperature of 339 K is assumed for the major components with the exception of the water pump. The heat rejection temperature for the water pump was assumed to be 283 K.

Table 2.1-2. Propellant Production System State Points

Node (discharge)	Component	Fluid	Pressure MPa	Temperature K	Density kg/m^3	Enthalpy kJ/kg	Flow Rate g/s
1	storage tank	water (ice)	6.0E-04	273	999.8	0.0	25.800
2	heat exchanger	water	0.345	283	999.9	41.7	25.800
3	pump	water	3.450	283	999.9	41.7	25.800
4	heat exchanger	water	0.345	339	980.2	275.9	25.800
5	electrolyzer	oxygen	0.345	339	3.9	308.4	22.933
6	electrolyzer	hydrogen	0.345	339	0.2	4,785.6	2.867
7	dryer	water	0.345	339	980.2	275.9	0.013
8	dryer	water	0.345	339	980.2	275.9	0.013
9	heat exchanger	hydrogen	0.345	100	0.8	1,564.5	2.867
10	liquifier	hydrogen (para)	0.345	20	71.4	-256.5	2.867
11	heat exchanger	oxygen	0.345	100	1,091.0	-116.4	22.933
11	storage tank	oxygen	0.345	100	1,091.0	-116.4	22.933
12	storage tank	hydrogen (para)	0.345	20	71.4	-256.5	2.867

Input power requirements for each of the major components and radiator mass for heat rejection are summarized in table 2.1-3. Electrolyzer power of 482 kW was based upon the mass flow rate of the water (25.8 g/s), Gibbs free energy (15,890.8 kJ/kg) and an assumed electrolyzer efficiency (80%). Water pump power of 0.1 kW was based upon the peak pressure difference (3.45 MPa), the density of water (1000 kg/m3) and an assumed efficiency (70%). Hydrogen cryocooler power of 54.9 kW was determined based upon hydrogen flow rate (2.87 g/s), enthalpy change (1,564.5 to -256.5 kJ/kg) and an assumed cryocooler efficiency (9.5 %). Oxygen cryocooler power of 77.9 kW was determined from the oxygen flow rate (22.9 g/s), enthalpy change (308.4 to -116.4 kJ/kg) and an assumed cryocooler efficiency (12.5%). The dryer power of 2 KW was assumed. Thermal power rejected from each of the major components was determined based upon the input power and the assumed efficiency. The radiator area assigned to each major component was based upon required rejected power, assumed rejection surface emissivity (0.88), heat rejection temperature (339 or 283 K) and sink temperature (80 K). Total heat rejected by the system is 201.5 kW using a radiator area of 306.9 m2.

The current study has not considered potential heat exchange between the propellants to be liquified and the water to be electrolyzed. For example, if water were delivered in the form of cold ice, perhaps in exchangeable tanks with embedded heat exchange tubing, a system could be designed which chills propellant gas prior to liquification at the same time as it pre-heats the water. While such a system could be beneficial, the energy required for electrolysis and latent heat of liquification are still the drivers for the system.

Table 2.1-3. Propellant Production System Balance Summary

Electrolyzer			Radiators		
Mass flow	25.800	g/s	Epsilon	0.88	
Gibbs free energy	15,890.8	kJ/kg	Radiator sink temperature	80	K
Electrolyzer efficiency	0.85		Pump power rejected	0.0	kWt
Electrolyzer input power	482.3	kWe	Pump rejection temperature	283	K
Specific mass	9.5	kg/kW	Pump radiator area	0.1	m2
Total electrolyzer mass	4,593.2	kg	Electrolyzer power rejected	72.3	kWt
			Electrolyzer rejection temperature	339	K
Cryocooler			Electrolyzer radiator area	110.1	m2
			Dryer power rejected	2.0	kWt
Oxygen mass flow rate	22.933	g/s	Dryer rejection temperature	339	K
Hydrogen mass flow rate	2.867	g/s	Dryer radiator area	3.0	m2
Heat removed (oxygen and hydrogen streams)	117.9	kWt	Oxygen cryocooler power rejected	68.2	kWt
Input power (oxygen and hydrogen streams)	132.9	kWe	Oxygen cryocooler rejection temperature	339	K
Oxygen cryocooler specific mass	8.4	kg/kW	Oxygen cryocooler radiator area	103.8	m2
Hydrogen cryocooler specific mass	18.9	kg/kW	Hydrogen precooler power rejected	9.2	kWt
Total cryocooler mass	1,693.2	kg	Hydrogen precooler rejection temperature	339	K
			Hydrogen precooler radiator area	14.1	m2
Input Power Summary			Hydrogen cryocooler power rejected	49.7	kWt
			Hydrogen cryocooler rejection temperature	339	K
Water pump (70 % efficiency)	0.1	kWe	Hydrogen cryocooler radiator area	75.7	m2
Electrolyzer (85 % efficiency)	482.3	kWe			
Hydrogen cryocooler (9.5 % efficiency)	54.9	kWe	Total power rejected via radiators	201.5	kWt
Oxygen cryocooler (12.5 % efficiency)	77.9	kWe	Total radiator area	306.9	m2
Dryer	2.0	kWe	Specific Mass	5.0	kg/m2
Total input power	617.3	kWe	Total radiator mass	1,534.3	kg

2.1.8 Micrometeroid and Orbital Debris Shielding Integration

Based on analysis performed by Dr. Jeff Anderson of NASA MSFC, the hazard of depot penetration is significant. The probability of <u>no</u> penetration of a single-sheet aluminum wall is used as a standard for comparison. This probability was estimated per year, as a function of surface area, for a depot in a 400 km, circular, zero inclination orbit. As the orbital debris population has been increasing, and is expected to increase further, the projected environment was considered for the year 2020, using the ORDEM96 default solar & debris growth model. Aluminum tank probability of no penetration as a function of thickness and area is as follows:

Thickness		**Exposed Surface Area** (m^2)			
(in)	(cm)	1	10	30	100
0.08	0.2	.89	.30	.03	0.0
0.12	0.3	.967	.71	.36	.03
0.24	0.6	.996	.96	.88	.66
0.50	1.3	.999	.994	.98	.94

Because the depot surface area is on the order of a hundred square meters, the probability of penetration is relatively high (P = ~1-0.94 = ~ 34% per year), assuming that the tank wall thickness is a single wall roughly _ inch (~0.6 cm). Multi-walled impact shielding can improve the Depot's impact resistance dramatically. A typical two-walled shield (a "bumper" separated from the tank wall) will reduce the total thickness required by a factor of four for a given level of protection. A cryogenic liquid storage tank with a vapor-cooled shield (for thermal reasons) may also provide a very significant improvement in impact resistance. Tank insulation will further improve the impact resistance, as will the placement of body-mounted radiators on the North-and South-facing sides of the Depot, which face typical directions of debris impact. Integrated thermal / impact shielding is an important area for future research and technology development.

2.1.9 Propellant Production Depot Dimensions and Mass Estimate

Propellant Production Depot dimensions and mass are summarized in Figure 2.1-13 and Table 2.1-4. Overall dimensions are fairly large at 80 meters in width. Masses are estimated for subsystem for electrolysis (water conversion), solar power generation, telecommunications and commands, liquid oxygen and hydrogen tanks (including insulation), integrating structure, power management and distribution (PMAD), attitude control, and docking ports. Total assembled Propellant Production Depot mass is estimated to be 69,000 metric tons (152,000 lb).

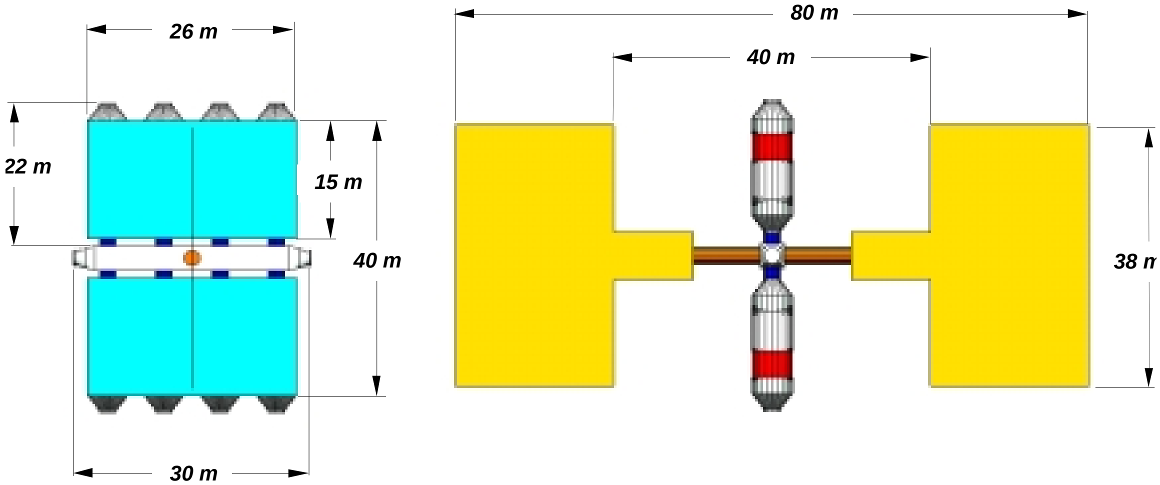


Figure 2.1-13. Propellant Production Depot Configuration Dimensions

Table 2.1-4. Propellant Production Depot Mass Estimate

System Element	Mass (MT)	Comments
Electrolysis Conversion		Devices, Structure; Input Power = 706 kWe
Storage (H20 tanks)	9.001	
Conversion	4.593	
Radiators	1.534	
Cryocooler	1.693	
Structure	0.508	
Add'l Structure Allowance	0.520	Allowance = 3%
Solar Conversion		SLA
Solar Concentrators/Arrays	0.562	Unit Height = 3 m, Width = 10 m, Mass = 0.012 MT, Power = 0.016 MW
Add'l Structure Allowance	0.017	Allowance = 3%
Telecomm & Command	0.338	One set per solar array node (8 sets)
Add'l Structure Allowance	0.010	Allowance = 3%
LOX/LH2 Tanks	18.64	8 LOX/LH2 tanks
Insulation, structure, etc.	8.000	Tank skirts, intertank, vapor-cooled shield, insulation
Integrating Structure	4.790	Abacus, Prop & H2O Tank Structures
PMAD		Cabling & Power Conversion, SPG Power = 706 kWe; Advanced PMAD
Cabling	2.203	Total Length = 3 km @ 0.881 kg/m, Voltage = 0.15 kV
Array Converter Mass	1.011	Mass based on 48 Switches (150 V to 0.15 kV), 0.016 MW Power Out
Electrolysis PMAD Mass	5.503	Mass Includes Voltage Converters, Switches, Harness & PMAD Thermal
Rotary Joints, Switches, Etc.	0.357	Scaled from 79 SPS Study
Attitude Control/Pointing		Sensors, Computers, Control Effectors
Dry Mass	0.701	Thrusters, CMG's, Sensors etc.
Propellant	0.516	10 years, Krypton
Robotics	1.900	7 units @ 200 each, 500 kg infrastructure
Add'l Structure Allowance	0.021	Allowance = 3%
Docking Ports & Structure	6.405	
Add'l Structure Allowance	0.192	
Satellite Mass (MT)	69.015	Without H20 or LOX/LH2

2.1.10 Propellant Production Depot Launch Concept

A preliminary launch vehicle payload manifesting concept for the Propellant Production Depot is illustrated in Figure 2.1-14. For depot tank set launches, the tanks of the Delta IV ELV crygenic upper stage tank are replaced by one Depot tank-set. In addition to this tankset, this “upper stage” carries an additional, empty tank-set as its payload (Element 1 of figure 1.2-14). Other components are launched with cryogenic OMVs as the “upper stages”, each carrying a portion of the depot (i.e., solar arrays or electrolysis apparatus and water holding tanks, etc.). Seven launches in total are required for deployment of the Depot (along with three OMVs).

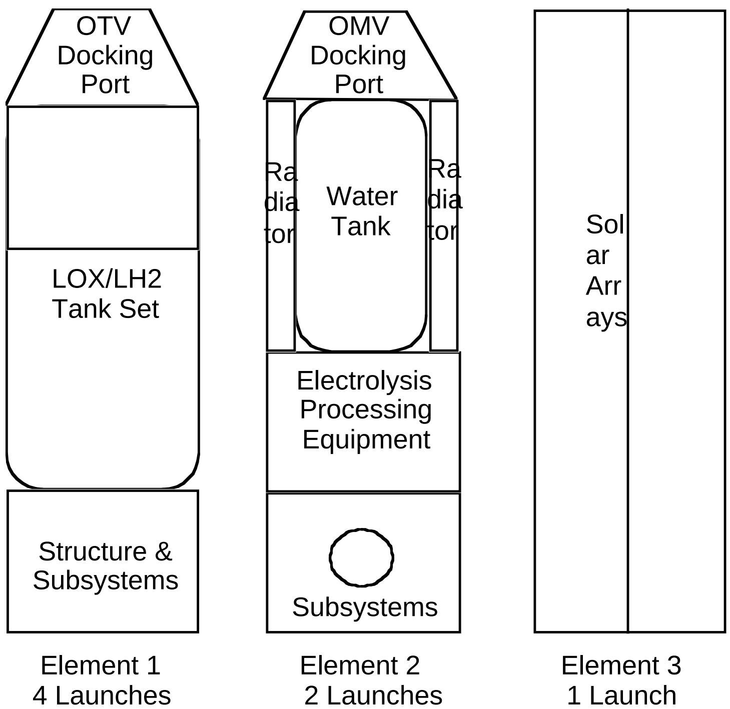


Figure 2.1-14. Seven launches can carry the Propellant Production Depot into orbit.

Based on the proposed configuration and launch packaging, seven launches will be needed to transport all depot elements to orbit.

The following assumptions were made for packaging and assembly purposes:

1. ELV upper stages are replaced or modified for transport of Elements 1, 2, and 3 into the desired orbit. These upper stages have expanded capabilities on orbit: to retrieve, maneuver, and maintain the position of the depot elements as needed.
2. Launch elements are integrated to the greatest extent possible prior to launch.
3. On-orbit assembly is autonomous via capture, deployment, or (tele-) robotic systems.
4. The upper stage maintains attitude control of Depot elements until it is sufficiently assembled to perform this function for itself.

2.2 Cryogen Storage-Only Depot

For comparison purposes, a rough system design was defined, at a conceptual level, for a comparable Cryogen Storage-Only Depot. Figure 2-15 illustrates a simple gravity-gradient oriented system to perform this function. This alternative does not require significant electrical power for operations, as there is no need for cryogen production or refrigeration. Hydrogen gas is allowed to boil off from the liquid, though boil-off is minimized by technologies such as advanced insulation, para- to ortho-hydrogen conversion, vapor cooled shielding, etc. (see Chapter 3). Hydrogen boil-off gas is used for passive cooling of liquid oxygen and then (heated further) and released through a thruster for low thrust thermal Hydrogen propulsion to cancel drag. This Depot also uses a moderate thrust gaseous O2/H2 propulsion system for autonomous circularization and acquisition of the gravity-gradient orientation; this subsystem uses roughly 1,000 kg of ullage gas and residual fluids remaining after main engine cutoff.

Because the power requirement for the Cryogen Storage-Only Depot is small, the area needed for solar cells & radiators is minimal. With a Depot in equatorial orbit, the North and South facing sides receive very little sunlight, and we would mount thermal radiators here, if required. (This was also the case for the body-mounted radiators of the Propellant Production Depot, but a large area was required). In equatorial orbit, the East and West facing sides receive more sunlight, with the sun effectively "rising" in the East and "Setting" in the West on every orbit. Body-mounted solar cells are therefore mounted on the East and West sides of the Depot. Solar cells and radiators are mounted to the relatively small area of three intertank structures on this Depot, where they can interface with rigid aluminum alloy or composite structure, rather than interfacing with cryogenic insulation.

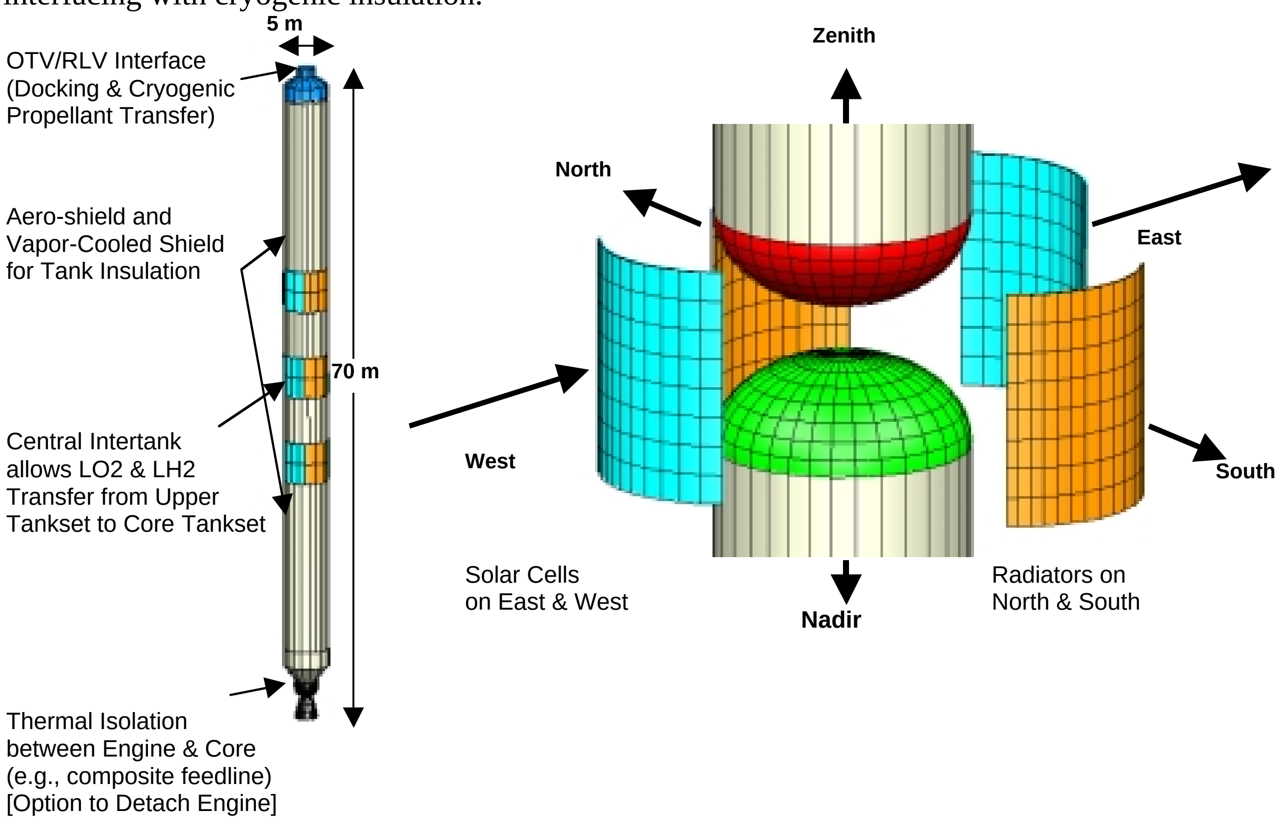


Figure 2.2-1. A simple Cryogen-Storage-Only Depot has been defined for comparison.

This Cryogen Storage-Only Depot can be launched by a single modified Delta IV Heavy Launch Vehicle (figure 2.2-2). In the place of an upper stage and payload, the Delta IV–H carries an inverted core tank set, similar to the central Common Booster Core (CBC) of the launch vehicle.

Two options have been considered which could achieve the performance necessary to place this entire, seventy meter (233 feet) tall structure into 400 km orbit. One option is to carry propellants in the upper tank-set, and feed them down to the lower tank set, so that those propellants may also be used in the main engine during launch. This option does require a somewhat longer engine burn than the basic vehicle. A second option is to replace the RS-68 main engine with a Space Shuttle Main Engine (SSME). SSME Isp is higher and thrust is slightly lower, so more propellant is left in the core after liquid booster staging, and it is burned more effectively. SSME cost may not be prohibitive if an engine is near the end of it's useful life, or if it is to be returned to Earth via RLV. While neither of these concepts has been studied in enough detail to optimize the system, we are confident that either of them launch a combined depot weighing as much as 56,000 kg. [While a detailed weight estimate has not been prepared for this Depot concept, roughly 50,000 kg is needed for it's combined subsystems].

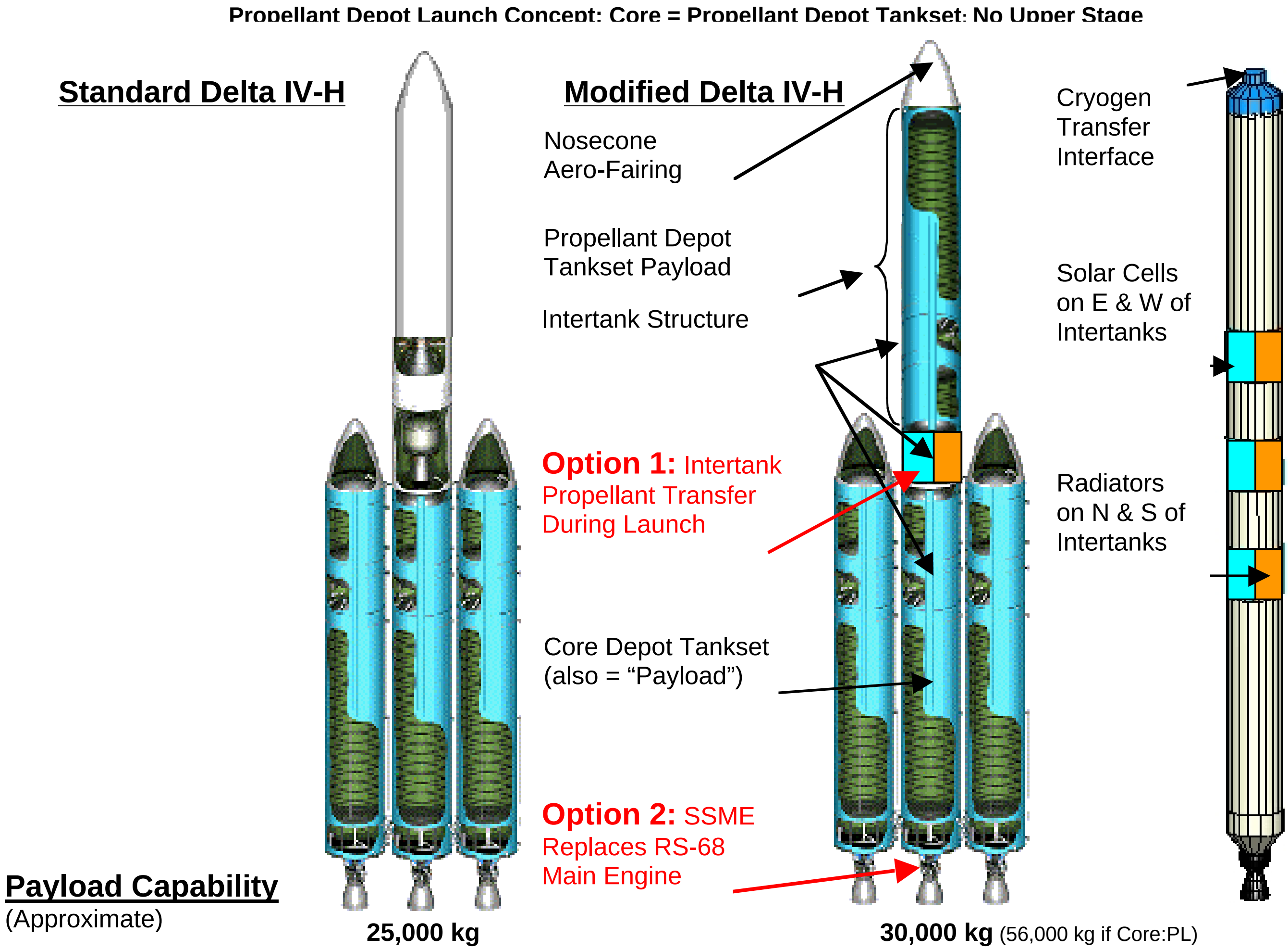


Figure 2.2-2. Cryogen-Storage-Only Depot launches to orbit as a single monolithic structure.

2.3 Support of Commercial Infrastructures

Propellant Depot Missions will refuel OMV's for transfers in LEO and the OTV's will be refueled for commercial satellite transfers to GEO or geosynchronous transfer orbit. The propellant can also be used for future commercial missions of types which are not yet in active use, such as satellite servicing and orbital debris removal. In addition, a Propellant Production Depot may potentially provide excess oxygen, hydrogen, water, power, and expended water tanks for use by future industries in orbit. Laser-based wireless power transmission from this Propellant Production Depot might also be used to power co-orbiting platforms with photovoltaic receivers (i.e., “solar” arrays) and thermal hydrogen engines on co-orbiting transfer vehicles (i.e., “Solar” OTV).

The depot can support commercial infrastructures by providing propellant and/or power to OMVs and OTVs that deliver communications satellites to GEO and other destinations. Water converted to cryogens can be used for propulsion, both as fuel for an OTV, and as an energy storage medium to be recovered by reversible fuel cells onboard the depot. For some commercial applications, the water need not be converted to cryogens, but can be transferred to thermal OTVs for use as reaction mass. Power can be beamed to an OTV from the depot. Lasers of wavelength ~0.8 to 1.1 microns, and focusing optics of ~1-5 meters in diameter may have a range of several thousand km. Power beaming logistics are simplified by the circular equatorial orbit, which does not undergo nodal regression. Power transmission duty cycle can be maximized if more than one depot is used. Thus, the depot can provide the following resources:

(1) Cryogenic propellants to chemically-propelled OMVs and OTVs;
(2) Water for solar thermal OTVs;
(3) Beamed power for Hall effect or ion-propelled OTVs (e.g., using fuel-cell energy);
(4) Water, Hydrogen, and beamed power for laser (solar)-thermal OTVs.

In addition, the depot may become a hub for future commercial use of space, supplying power, communications, and a platform for orientation. Inexpensive launch of water/ice infers that other raw materials for space manufacturing could be launched cheaply as well. Expended water tankage could be a feed-stock of material or a dual-use container/structure for in-orbit commercial applications.

Commercial Vehicle Interfaces with Depot

Vehicles that interface with the Depot (e.g., OMV and OTV) have been based on commercially developed Medium and Heavy upper stages of expendable launch vehicles (figure 2.3-1). The “OMV” configuration is based on the Delta III Cryogenic Upper Stage configuration, with tank volumes that hold up to 22.5 tons of cryogenic propellant. The “OTV” configuration is based on the Delta IV Heavy Cryogenic Upper Stage, which holds 30 tons of usable propellant. This larger upper stage increases the diameter of the LH2 tank to 5.1 meters and the length of the LOX tank by approximately 0.5 meters (20 inches). Delta IV intertank is a composite structure which is longer to accommodate the increased length of 5.1 m domes.

One basic commercial scenario would use a Reusable Launch Vehicle (RLV) to deliver a large GEO communications satellite to the Depot, where it would be mated to a fueled OTVs for transfer to GEO (or GTO). Alternatively, the fueled OTV could descend to a low RLV orbit for payload transfer, which would significantly increase the RLV payload capability.

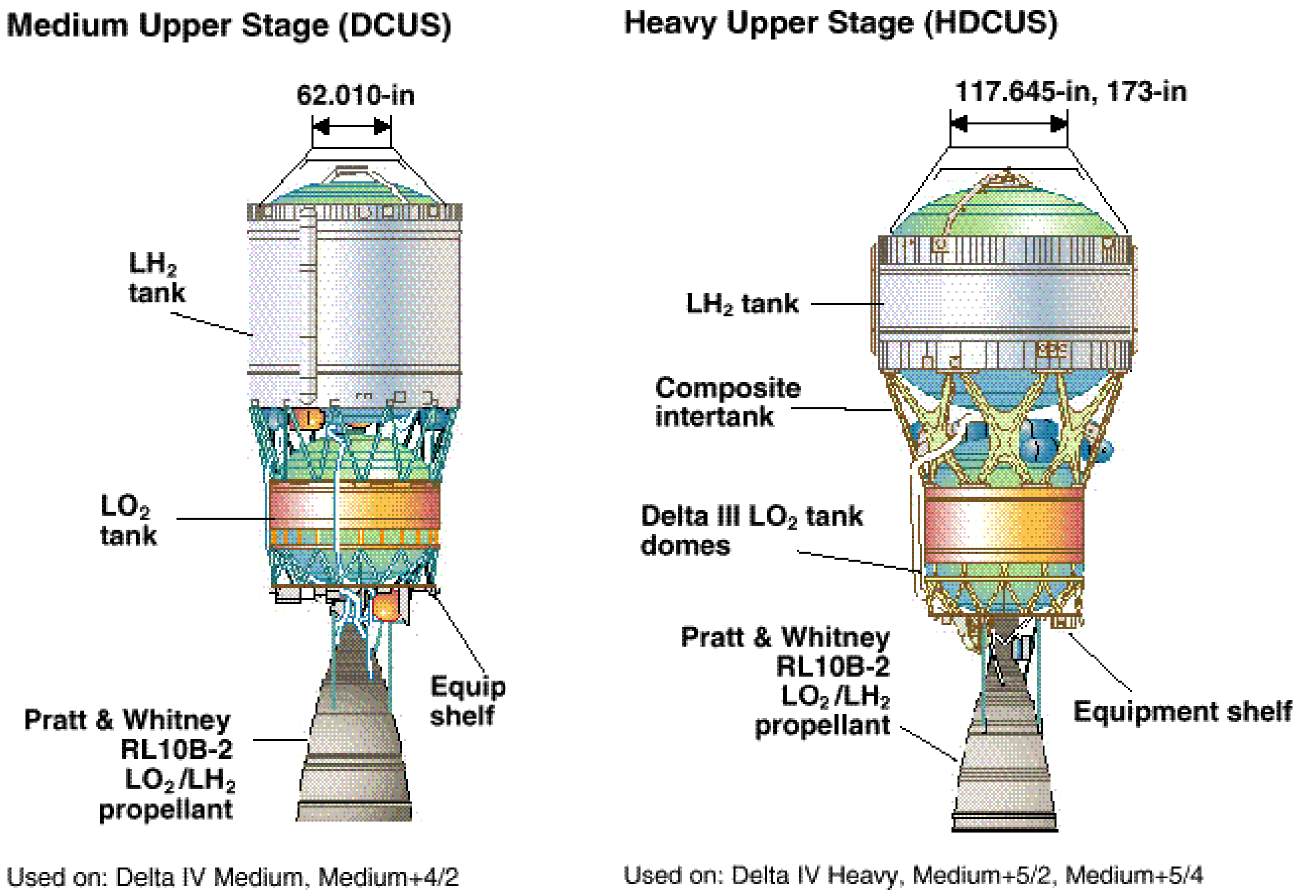


Figure 2.3-1. OMV and OTV configurations are derived from commercial upper stages.

Another alternative commercial scenario would be launch of a large GEO communications satellite on an "expendable" upper stage that reaches LEO almost depleted of fuel. It refuels at the Depot (where basic satellite functions might be checked out following exposure to launch loads. Following refueling, the upper stage carries its payload on to GEO, and the upper stage is disposed of. With this scenario, the mass that could be launched to GEO increases dramatically.

Several other vehicles that may use the Depot for commercial applications, in addition to Government applications, are illustrated in figure 2.3-2. These include a Mars Transfer Vehicle, Hybrid Propellant Module, Projectile Launch System, and Reusable Launch Vehicle.

The reference Mars Transfer Vehicle is an "Abundant Chemical Propulsion Stage", meaning that, because cryogenic propellants are plentiful, they are utilized extensively rather than requiring other advanced technologies (e.g., aero-braking or electric propulsion) to reduce propellant requirements. Its large liquid hydrogen tank and liquid oxygen tank are similar in size to those of the upper stage/depot tank-set used in the Propellant Production Depot Design Concept [and it is possible that one basic design could serve both purposes]. Its stage dry mass and propellant mass are approximately 10,500 kg and 69,500 kg, respectively. The configuration shown uses 5.4 m tank diameters with four off-the-shelf RL 10-B2 engines burning LOX and

LH2 at a mixture ratio of 6:1. This reference vehicle has zero boil-off and has a power requirement of approximately 6 kWe in LEO.

The Reusable Launch Vehicle Hybrid Propellant Module (HPM) concept includes zero boil-off cryogenic fluid management (ZBO CFM), solar arrays, liquid hydrogen, liquid oxygen, and xenon tanks a fuel transfer interface (FTI), and other HPM subsystems. This system could be re-fueled with cryogens at the Depot. Xenon propellant accommodations would require modifications to the system design, but would be similar to the launch and transfer of water.

If projectiles containing water can be gun-launched from Earth to reach the Depot, then other fluids and materials could also be launched similarly, and the projectile casings and water tanks might be used for other purposes after they reach orbit. A wide variety of industrial feed stocks could be sent to orbit with this system, and the Depot would be a natural place to receive and store them. Expended tanks could be rearranged together to form larger structures, or to store the product of future space industries. Projectile casings designed for the aerothermodynamic environment of a gun-launch might be inherently well-suited for re-entry and recovery on Earth, so the projectile itself could potentially become a capsule to deliver the products of future space industries down to Earth.

Reusable Launch Vehicle accommodations at the Depot could include power for extended stays, contingency return to Earth propellant, and propellant for high energy missions.

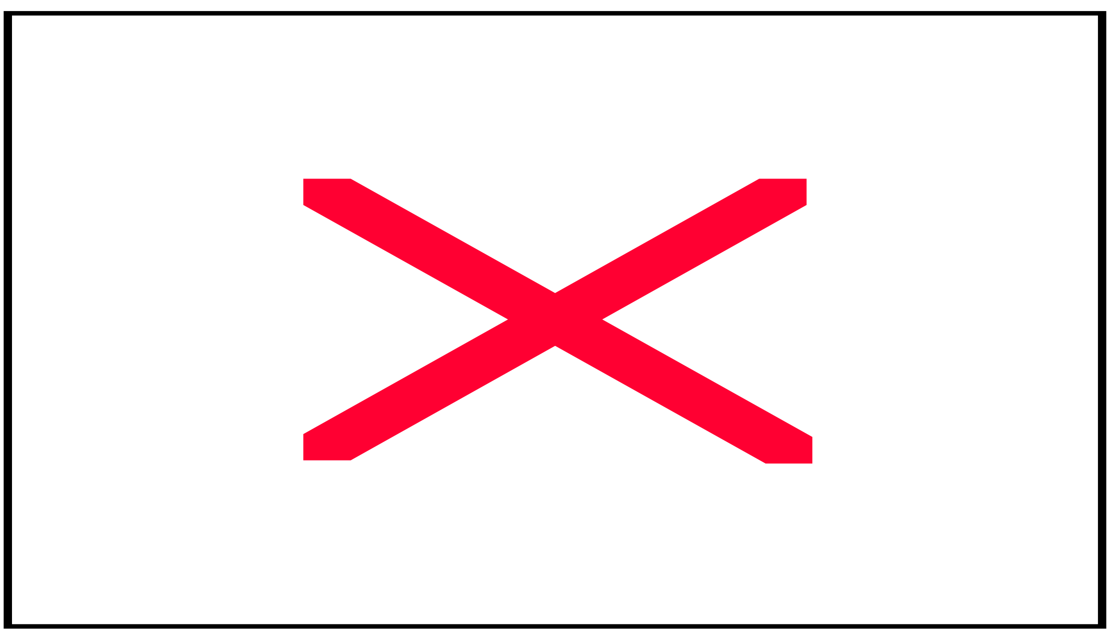


Figure 2.3-2. Advanced vehicles may take part in future commercial infrastructures.

The Solar OTV, illustrated in figure 2.3-3, could also make use of this infrastructure. With Hydrogen as propellant, this system has roughly twice the Isp of an O2/H2 chemical rocket engine, but normally the thrust of this vehicle is fairly low (roughly 0.5 lbf), and thrust is only available on the sunlit side of the orbit. If a laser wireless power transmission (WPT) system were used to send excess power from the Propellant Production Depot to this vehicle (see figure

2.3-4), it could, potentially receive a ten-fold or 100-fold increase in its power level and corresponding thrust level, on both sides of the orbit. Because the Depot and SOTV are both in equatorial orbit, long, repeated intervals would be available for WPT as the vehicle moves out from LEO toward GEO. This scenario would significantly decrease the amount of time needed to transport SOTV payloads.

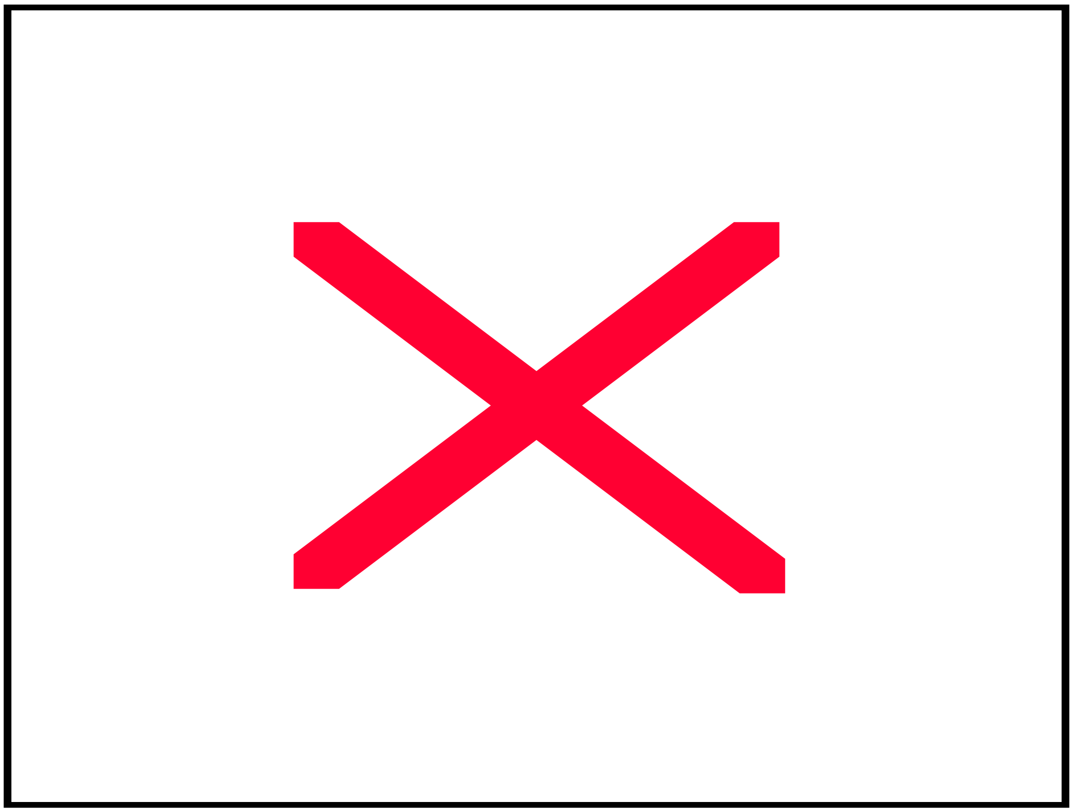


Figure 2.3-3. A Solar OTV could use H2 or H20 propellant and, potentially, laser WPT

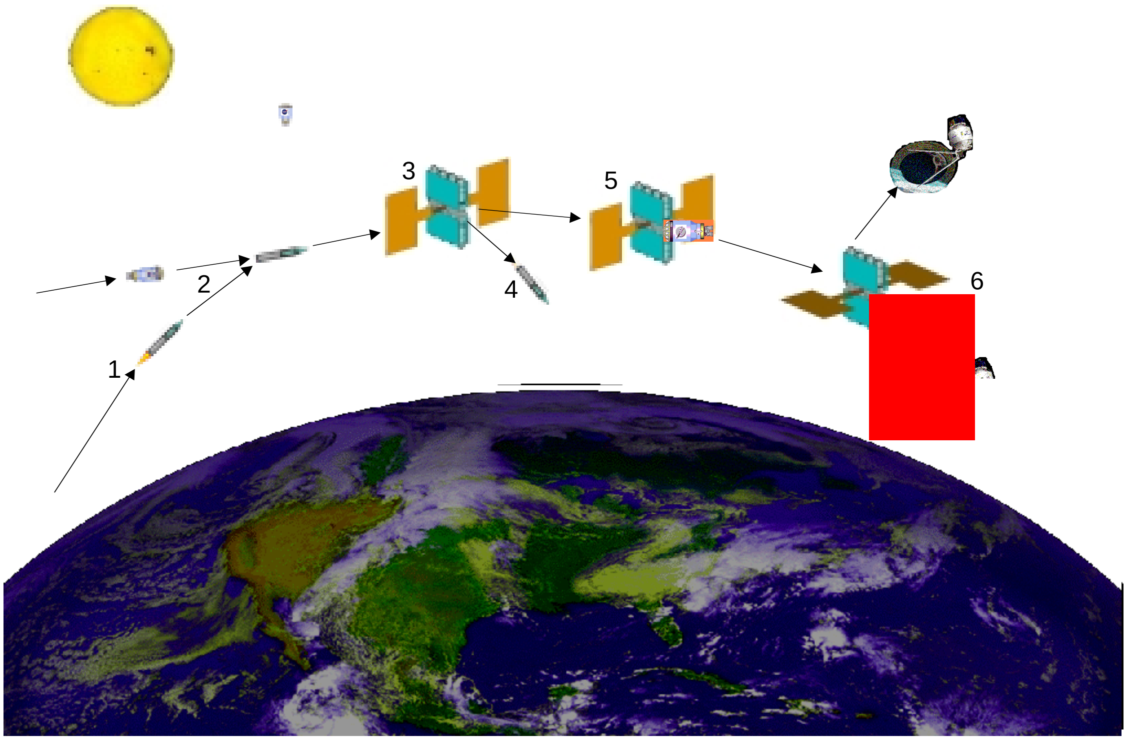


Figure 2.3-4. Laser-thermal propulsion using hydrogen and power from the Propellant Production Depot could significantly reduce “Solar” OTV trip times.

3. TECHNOLOGY

Cryogenic propellant storage depot technology is a unique area, in that it has been studied in detail and tested extensively on the ground, but little research has been accomplished in space, where the unique effects of low gravity come into play. Cryogenic propellant technology needs go beyond the more general Space Solar Power technology, which is a pre-requisite for the Propellant Production Depot (but which is not the focus of this chapter). Technology development road-mapping considers opportunities to mature cryogenic propellant technologies through on orbit testing using platforms such as the International Space Station.

3.1 Technology Development Needs

Cryogenic Propellant Storage Depot Technology needs were assessed based upon a related technical paper, "Evaluation of Cryogenic System Test Options for the OTV On-Orbit Propellant Depot" (Schuster et al). Table 3-1 summarizes key cryogenic storage facility components and their critical technology issues. Technical risks are due to uncertainty that the components will perform their functions as intended in a micro-gravity environment, after being subjected to launch loads. Risks at the component level have been identified as low, medium, or high, depending on the degree of uncertainty that the component would perform as intended. The risk assessment forms part of the basis for defining a technology test program.

NASA Technology Readiness Level (TRL) ratings indicate relative maturity of the various technologies which are needed. Figure 3-1 illustrates this rating scale. An assessment of TRL has been added, along with Risk, to the data in Table 3-1.

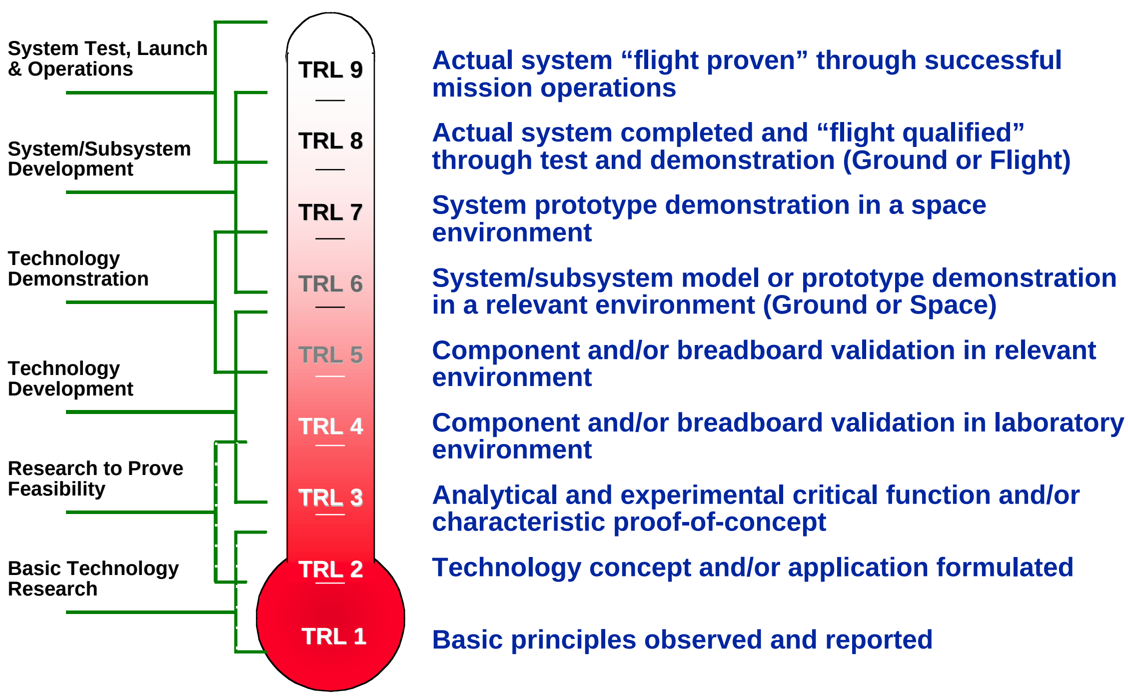


Figure 3-1. Technology Readiness Levels indicate the relative maturity of needed technology.

Table 3-1. Cryogenic Propellant Depot Technologies are identified based on prior studies.

Storage Facility Component	Critical Technology Issues	TRL / Risk
Propellant Tank Set	• Fluid slosh & orientation in μ-g	3 / Medium
Tank Support Structure	• Dynamic response to launch • Dynamic interaction in orbit	6 / Medium 3 / Low
Tank Support Struts	• Support of launch loads • Thermal Performance	6 / Medium 5 / Medium
Integrated Insulation (MLI)	• Insulation thermal performance • Insulation degradation in launch • Atomic oxygen & contamination	5 / Medium 4 / Medium 5 / Medium
Tank Set Solar Selective Cover	• Coating degradation on orbit • Shield thickness & material	6 / Medium 4 / Low
Radiator	• Support during launch • Coating degradation on orbit	5 / Medium 6 / Medium
Micrometeoroid & Debris Shields	• Material & thickness • Performance	3 / Medium **3 / High**
Vapor-Cooled Shield (VCS)	• Performance • Thermal Performance	5 / Medium 5 / Medium
Para-to-Ortho Converter	• Performance • Operating Life • Filtering Requirement	4 / Medium 2 / Low 3 / Low
Penetrations: Inst. & Plumbing	• Thermal Performance	4 / Medium
Warm Tank Chilldown	• Spray Nozzle configuration • Liquid Flow-rate & duration • Number of gas venting steps • Micro-g Performance	3 / Medium 3 / Medium 3 / Medium 3 / Medium
Thermodynamic Vent System (TVS)	• Thermal Performance • Micro-g Heat transfer from fluid	5 / Medium 3 / Medium
Stratification/Hot Spot Management	• Mixing Needs/mixing strategy • Micro-g performance	2 / Medium 2 / Medium
Liquid Acquisition Device (LAD)	• Residual fraction; flow vs. % liquid • P drop; Long term use (corrosion)	3 / Medium 4 / Medium
Pressurization System	• System requirements & performance • Micro-g performance (diffuser flow & T)	3 / Medium 4 / Medium
Liquid pumps	• Operating Life • Micro-g Performance	2 / Medium 3 / Low
Refrigerator	• Thermodynamic efficiency & life • Micro-g performance	**4 / High** 3 / Low
Boil-off Condenser	• Thermal Performance • Micro-g performance	3 / Medium 3 / Low
Boil-off Compressor	• Operating Life • Micro-g performance	**4 / High** 3 / Low
Low Heat Leak Valves	• Operating Life • Thermal Performance	2 / Medium 3 / Medium
Disconnects	• Fluid Leakage, pressure drop • Force & alignment requirements • Thermal performance (heat leak)	3 / Medium 3 / Low 3 / Low
Mass Gauging	• Performance • Micro-g performance	**4 / High** 3 / Medium
Control System	• Performance; life; failure response	2 / Medium
No-Vent Fill	• Procedure • Micro-g condensation & fluid mixing	3 / Medium 2 / Medium
Transfer Line Chilldown	• Procedure & Micro-g Performance	2 / Low

Technology Test Options

Several options are available to test the technology needed for propellant depots. Technology can be tested in the laboratory, or on Expendable Launch Vehicles, the Space Shuttle, the ISS, a Small Scale Depot, and a Full Scale Depot. Table 3-2 summarizes the various types of testing that can be accomplished using these options.

Table 3-2. Six basic options exist for a methodical approach to test needed technology.

Technology Test Option	Lab	ELV	Shuttle	ISS	Small System	Full System
Sub-Scale Tests	√	√	√	√	√	
Full-Scale Tests	√					√
Component Tests	√	√	√	√	√	
Active Component Life Tests	√	?		√	√	√
(Simulated) Launch Environmental Testing	√	√	√	√	√	√
Long Term Orbital Material Tests		?		√	√	√
Integrated System Ground testing	√					
Integrated System Short Term Orbital Testing		√	√	√	√	
Integrated System Long Term Orbital Testing		?		√	√	√

Laboratory testing can use sub- or full-scale tank sets for tests carried out on components, subsystems, and integrated systems on the ground. Identified improvements can be incorporated into subsequent tank sets, which may be used on the ground or in orbital tests. In some cases, a “proto-flight” approach may be used, where the original ground-test tank set can potentially be modified for subsequent testing on-orbit. For example, test requirements may be addressed by building a subscale experiment, which simulates the hydrogen fluid systems of the storage facility, evaluating their performance in a vacuum chamber, and then demonstrating micro-g fluid transfer by performing an orbital experiment.

Expendable launch vehicle based testing consists of subscale component testing with minimal integration. It is intended to limit costs by making use of components in whatever scale they are available, on flights of opportunity. The third Titan-Centaur mission was such a flight, where excess propellant was used in a series of vehicle maneuvers and experiments carried out after the primary mission was accomplished. Testing on this mission helped guide the evolution of cryogenic upper stages. A similar flight of opportunity could potentially become available on the first Delta IV-H mission, which is currently planned for launch without a payload. In this case, as in all flight experiments, the technology demonstration needs to be designed with minimal impact (and no additional risk) to primary mission objectives. Question marks are included in this column of table 3-2 because it possible, in concept, to create a secondary payload that keeps the upper stage alive in orbit (with added on-orbit power) for long-term operations.

Space Shuttle testing is limited in duration, and is subject to numerous manned flight safety considerations, but the Shuttle does use cryogenic propellants in the electrical power system (for fuel cells), as well as the main engines. Figure 3-2 illustratesseveral of the technology experiments which might be suitable for Shuttle demonstrations. An Extended Duration Orbiter pallet could be used for containment of propellants for component-level testing. Free flyers deployed from the Space Shuttle might also serve as a platforms to perform experiments which develop cryogenic propellant depot technology. The scale of such experiments will be smaller than those that can be accommodated by ISS, but they might able to launch in the nearer term.

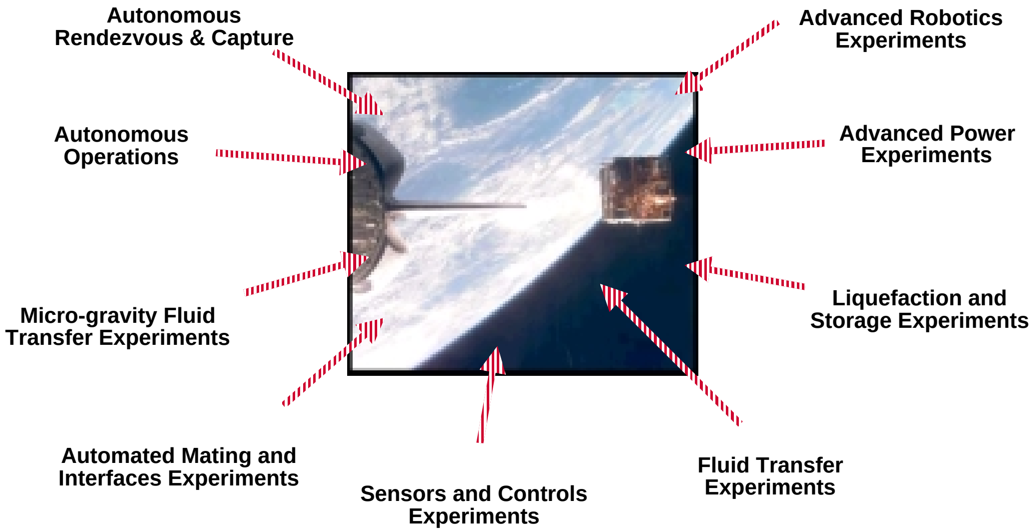


Figure 3-2. Shuttle/Free-flyer Technology Flight Experiment Opportunities Summary

International Space Station testing is also subject to numerous manned flight safety considerations, but allows long term testing, and may potentially aid ISS in performing its mission (e.g., by supplying a second source of contingency power). This option reduces risk by demonstrating post-launch performance of storage facility components in orbit and returning the results in the early stages of the full-scale facility design. The orbital experiment is performed on the ISS rather than the Space Shuttle or Delta IV Cryogenic Upper Stage, which allows much more time for performing transfer experiments, analyzing the results, and repeating the experiments. It also evaluates the longer-term performance of the solar selective cover, micrometeoroid shield, and insulated systems in orbit. Because the altitude of ISS and the dimensions of its components are similar to those of the Depot, it can serve as a platform for technology flight experiments (see figure 3-3). Power available on ISS is expected to be sufficient for such experiments.

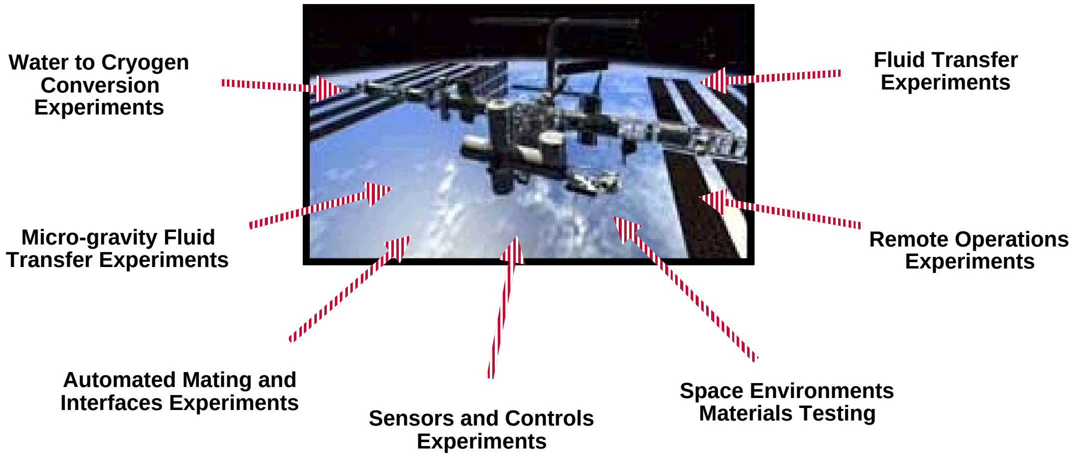


Figure 3-3. ISS Technology Flight Experiment Opportunities Summary

Small Scale Depot testing, as a precursor or pathfinder for deployment of the full-scale facility, provides an intermediate useful capability, which may make this option a most prudent approach. Certainly many lessons may be learned in initial operations of a small-scale propellant depot, which goes a step beyond mere technology demonstration and into actual system operations, with the lessons learned to be applied in full scale systems.

On a full-scale depot, fluids must be transferred on a microgravity environment. The residual gravity present due to gravity-gradient effects must allow for the settling and feeding of propellants. Materials used on the depot must last for perhaps 10 or 20 years. Assembly and operation of the depot will also require that automated rendezvous and docking take place near (but not necessarily at) the center of mass of the structure.

3.2 Technology Roadmap

A plan for technology development is summarized in figure 3-4. This process leads to full conversion and storage depot can be deployed by the middle of the next decade.

Ground demonstrations for cryogenic depot technology can begin in the next two years, with flight demonstrations on ISS and Shuttle following shortly afterward. Among the demonstrations that can take place in the near future are fluid transfer and electrolysis. The full depot will receive tanks containing 250 to 500 kg of water. A demo can test the ability to transfer water and settle it in a microgravity environment at half this scale (i.e., 125 kg), with an option to demonstrate docking of a water delivery vehicle in the longer term. It has a mass of 60 kg, not including stored water or an optional docking mechanism. It has no contamination issues for ISS and may be able to integrate into more advanced demos, i.e. electrolysis of water.

A demonstration of electrolysis of water can take place on ISS, at 1/20 scale of the full depot; i.e., 1.3 g/sec. It requires 25 kW; near maximum of ISS total average payload power. Optionally, separate dedicated solar arrays can be used (e.g., to test general SSP technology), and such advanced arrays could also be useful for ISS upgrades, such as a secondary, contingency solar power supply. The similarity of the ISS orbit to the depot orbit provides an opportunity to demonstrate the duty cycle of the operational depot; i.e., powering down when the Sun is not visible.

Power management and distribution issues and autonomous operations can also be demonstrated in the near term, as a separate demonstration, or combined with a cryogen storage demonstration. On-orbit autonomous fluid connection and transfer may use a Shuttle-deployed free-flying platform similar to Spartan to test the feasibility of autonomously mating two connectors and transferring fluids across the connection. Fluid transfer and autonomous connections can be tested in a vacuum chamber prior to flight.

A small scale storage depot can be deployed by the end of this decade. This depot could provide propellant refueling service for LEO-to-GEO reusable transportation, and LEO, GEO, commercial or military satellites. It would use critical technologies for HEDS, including propellant management, solar power generation, automated rendezvous and control, and on-orbit assembly. This small scale depot is envisioned with a mass of 8,700 kg

A more advanced "small-scale" depot would have the capability for electrolytic conversion of H2O to LOX and LH2 as well as on-orbit cryogenic propellant storage of LOX and LH2. This initial Propellant Production Depot is envisioned with a mass of 25,000 kg and increased mission lifetime, financial returns and reliability. It provides indefinite storage of cryogenic propellants.

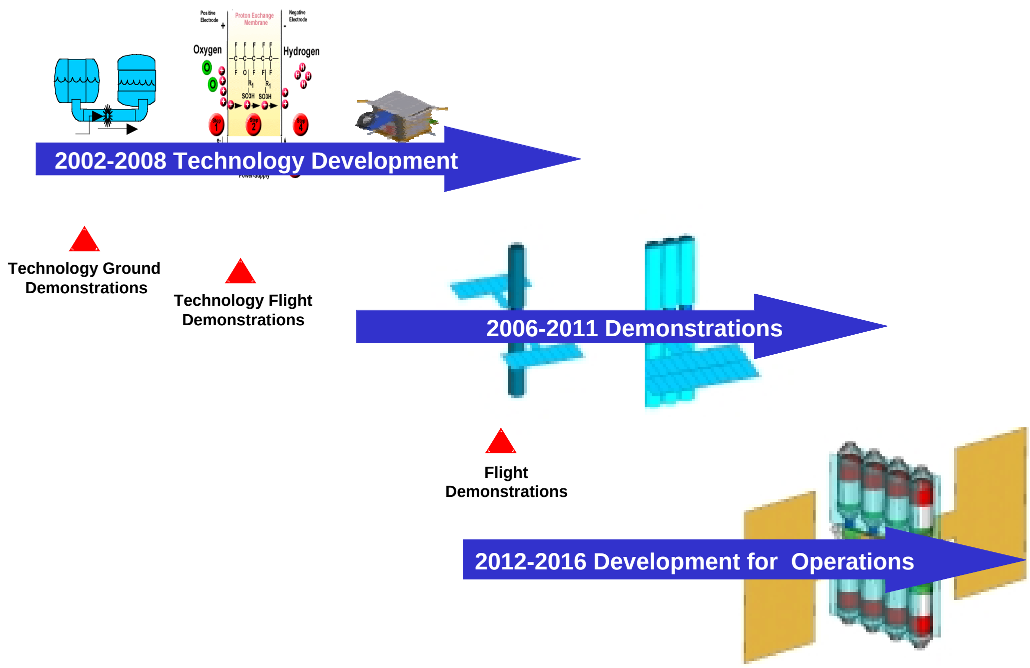


Figure 3-4. Technology matures from ground development and ELV tests, through Shuttle and ISS demonstrations, to small-scale systems operations, before use in the full-scale Depot.

4. CONCLUSIONS

SSP Exploratory Research and Technology enables efficient cryogenic propellant production from H_2O, and can pave the way toward further robotic and human exploration and development of space. Large-scale production of cryogenic propellant from H_2O in Earth orbit is technically challenging, but can help meet future strategic goals (see Figure 4-1). It avoids volume and safety issues related to containment of cryogens during launch and it allows high acceleration of payload (H_2O/ice) during launch. Major components of a cryogenic propellant Depot, such as tanks, can be based on existing technology and launched using vehicles that will become available within the next few years.

In the nearer term, flight experiments can be performed on "expended" cryogenic upper stages, on the International Space Station, and/or on the Space Shuttle. This can be followed by deployment of a sub-scale storage only Depot in LEO.

In the longer term, cryogenic propellant production technology can be applied to a larger LEO Depot, and potentially, to the use of lunar water resources at a similar Depot elsewhere (on the Moon, in lunar orbit, or at an Earth-Moon LaGrange point).

Recommendations

Requirements

Variations in both the size and the function of a cryogenic conversion and storage Depot should be considered. The lower limit on size should be investigated as a means of achieving technological maturation with early beneficial use. Expansion of the functions of the Depot can be considered as well; e.g., use of the cryogenic propellant as stored power for recovery in a reversible fuel cell. Wireless Power Transmission via laser to a "solar" electric or "solar" thermal vehicle is another Depot function that might be added to requirements.

Design

It has been shown that a storage-only propellant Depot can be deployed in one launch. The possibility of deploying a smaller conversion and storage Depot in one launch should be investigated. The next phase of design of a full-scale conversion and storage Depot can be done concurrently. These activities can be supported by an examination of the details of Delta cryogenic storage tank reuse in orbit.

Technology

Near-term flight test opportunities on a Delta cryogenic upper stage should be investigated. An SSP/Depot technology test platform for the International Space Station should be considered in more detail.

SSP and Platform Technologies for Propellant Depots: Conclusions

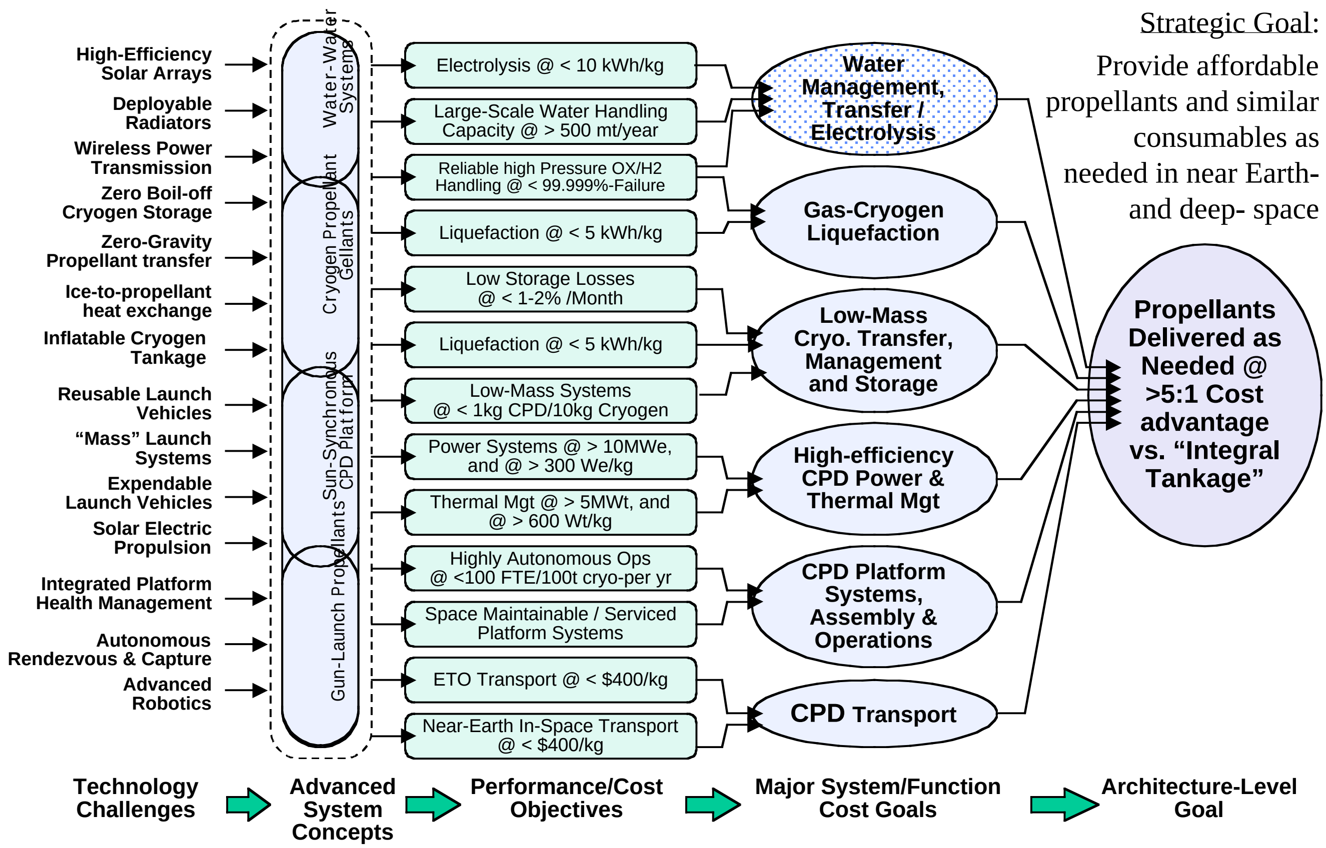


Figure 4-1. Technology for Propellant Depots helps meet top-level strategic goals.

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5. APPENDIX

Comments on "Cryogenic Propellant Depot: Draft Report"

Dr. Han Nguyen, The Boeing Company

1. The required g was calculated from the Bond number (Bo), defined as the ratio of gravitational force to surface tension. A Bond number of 10 was assumed to provide sufficient margin for propellant settling. This led to required g > 3.93e-6 for LH2 and 1.78e-6 for LO2.

2. As the tank radius decreases, the required g increases and the distance from the spacecraft center of mass to the liquid surface, z, also increases. For a 8.33-ft tank, z = 10.1 m (LH2) and 4.6 m (LO2).

3. Another important dimensionless parameter that is useful in the field of capillarity is the Weber number (We), defined as the ratio of inertial force to surface tension. As gravity is reduced to zero, the Bond number decreases to zero, and the fluid motion depends mainly on the Weber number.

   At 400 km, the orbital velocity is V = 7.67 km/s (25165 ft/s). The Weber number are obtained for the various tank radii as follows

   For LH2: density = 4.4 lbm/ft3
   surface tension = 1.20e-4 lbf/f

   For LO2: density = 71 lbm/ft3
   surface tension = 8.76e-4 lbf/ft

r ft	We LH2	We LO2
8.33	6.01E+12	1.33E+13
6	4.33E+12	9.56E+12
5	3.61E+12	7.97E+12
4	2.88E+12	6.38E+12
3	2.16E+12	4.78E+12
2	1.44E+12	3.19E+12
1	7.21E+11	1.59E+12

   The Weber numbers obtained for LH2 and LO2 are very large, indicating that inertial force dominate surface tension. As a result, the fluid is settled, and not shaped by capillary forces.

4. We can also calculate the transition time for zero-gravity phenomena, such as interface formation or wave period, from the Weber number. The time to form the interface is obtained for the various radii as

r ft	T LH2 ms	T LO2 ms
8.33	0.331	0.331
6	0.390	0.390
5	0.427	0.427
4	0.478	0.478
3	0.552	0.552
2	0.676	0.676
1	0.955	0.955

The results show that it takes less than one millisecond to form the interface at the specified orbit. Consequently, the fluid is settled almost instantaneously once the spacecraft is in orbit.

Fuel Depot Orbital Maneuvering Vehicle Sizing Study

Craig Cruzen
NASA MSFC TD54
September 19, 2000

Introduction:

This sizing study was done for a mission using an unmanned Orbital Maneuvering Vehicle (OMV) to deliver tanks of water to a fuel processing facility in Low Earth Orbit (LEO). The water tanks are to be launched by a Reusable Launch Vehicle (RLV) and placed in a low (200 km) circular orbit. The OMV then departs the fuel depot at a higher (400 km) circular orbit, rendezvous and docks with the water tank and returns to the depot. After the water in the tank has been transferred to the depot, the OMV departs with the empty tank and places it on a reentry trajectory. Finally, the OMV returns to the depot. The fuel depot uses electrolysis to convert the water to Hydrogen and Oxygen and these are stored and used to refuel the OMV as well as other space vehicles.

Assumptions:

The following assumptions were made for the study:

Mass of water and tank	35,500 kg	(78,200 lb)
Altitude of the fuel depot in LEO	400 km	(216 nmi)
Altitude of water tank in LEO	200 km	(108 nmi)
Specific impulse of the OMV engine	440 sec	(Based upon a RL10A class engine)

Propulsion inefficiency due to gravity losses	1% of total delta-velocity (Delta-V) required
Propellant required for proximity operations and docking	1% of total Delta-V required

Dry weight of OMV	4,535 kg	(10,000 lb, Based upon detailed NASA OMV design studies)
Frequency of water tank launch	1 per month	

OMV Sizing Analysis:

This analysis involved calculating the Delta-V required to perform the orbit transfers necessary to retrieve the water and return it to the fuel depot. Hohmann transfers were assumed since they are the most fuel efficient method of orbit transfer. Once the Delta-V totals are known, then by using the rocket equation and the assumptions shown above, the propellant masses required to perform the transfers can be calculated.

Delta-V Summary
(with G-loss, proximity operations and docking included)

Delta-V required to transfer from 400 km depot orbit to 200 km water tank orbit	118 m/s
Delta-V required to transfer from 200 km water tank orbit to 400 km depot orbit	118 m/s

Propellant Summary

Propellant required to transfer from 400 km depot orbit to 200 km water tank orbit	156 kg
Propellant required to transfer from 200 km water tank orbit to 400 km depot orbit	1,110 kg
Total propellant required	1,266 kg

Mass Summary

OMV mass departing depot	5,800 kg
OMV mass arriving at water orbit	5,645 kg
OMV and water mass departing water orbit	41,145 kg
MV and water mass arriving at fuel depot	40,035 kg

Fuel Depot Orbit Decay and Orbit Maintenance

Dr. Larry Mullins
NASA MSFC TD54
September 19, 2000

The fuel depot is assumed to fly with the tanks in a gravity gradient attitude along the radius vector. The RLV docking ports are aligned with the velocity vector and the solar arrays have their axes of articulation perpendicular to the orbit plane (i.e., along the orbital angular momentum vector) and will be gimbaled to keep the sun as nearly normal to their surface as possible. This means that at sunrise and sunset the maximum array surface will be directly into the velocity vector and at orbital noon the array surfaces will be "knife-edge" into the velocity vector. During the orbital night, they are assumed to be "knife-edge" into the velocity vector to reduce atmospheric drag.

The maximum area of the solar arrays is 40 m x 33 m = 1320 m^2. This maximum area is not always into the velocity vector but is assumed to be averaged by a cosine function over the day side of the orbit and is zero over the night side of the orbit, giving an orbital average of 420 m^2. The tanks are assumed to have a constant area of 33 m x 5 m = 165 m^2. The sum of these two numbers is 585 m^2.

The mass of the assembly when empty is assumed to be 44 metric tons (44,000 kg) and when full is 554 metric tons. The initial orbit was assumed to be 400 km circular, equatorial. Assuming a coefficient of drag of 2.2, some Lifetime runs were made with $+2\sigma$ solar activity and with nominal solar activity. The results for $+2\sigma$ solar activity with minimum Depot mass was re-entry within about 85 days. The results for nominal solar activity and minimum Depot mass was reentry within about 115 days. The decay curves are shown in figure 1.

If the orbit is allowed to decay to 350 km and then re-boosted to 400 km on a periodic basis, this would occur about every 50 days for $+2\sigma$ maximum solar activity and about every 75 days for nominal solar activity. A Hohmann transfer from 350 km to 400 km would require about 28.5 m/s delta velocity. This works out to a total of about 140 m/s per year for nominal solar activity and about 208 m/s per year for $+2\sigma$ solar activity.

The orbital decay starting at 400 km altitude is very rapid as can be seen from these results. If the starting altitude is raised to 500 km, the resulting orbital decay is shown in Figure 2 and if it is raised to 600 km, the resulting decay is shown in Figure 3. If the starting altitude was at 500 km, one could let the Depot decay to 400 km and then reboost to 500 km periodically. If it started at 600 km, one could let it decay to 500 km and then reboost to 600 km periodically.

The 500 km case would require a reboost about once a year for nominal solar activity and about once every 220 days for the $+2\sigma$ solar activity. Each reboost here would require 56 m/s delta velocity for the Hohmann transfer. The annual delta velocity requirement here would be 56 m/s per year for nominal solar activity and 93 m/s per year for $+2\sigma$ solar activity. The 600 km case would require reboost about once every ten years for nominal solar activity and about once every three years for $+2\sigma$ solar activity. Each reboost here would require 54.8 m/s delta velocity for the Hohmann transfer. The annual delta velocity requirement here would be 5.48 m/s per year for nominal solar activity and 18.3 m/s per year for $+2\sigma$ solar activity.

In each of these cases the RLV could still deliver the water to a 200 km altitude parking orbit. The OMV could come down from the Depot, pick it up and take it back to the Depot at 500 km or 600 km (or at any altitude in between). The delta v requirement for the OMV would be increased in these cases. For the 500 km case, it would be 343.2 m/s per trip and for the 600 km case it would be 452.8 m/s per trip. This compares to 233.7 m/s per trip for the 400 km altitude case. There would be the same number of trips required in each case since the same amount of water would have to be delivered to the Depot regardless of its altitude. This would be a 47% increase in delta v for the 500 km case and a 94% increase for the 600 km case. Thus, for a decrease in orbital maintenance for the Depot, one would pay a penalty in increased delta v for the OMV.

The minimum mass condition will occur when the Depot is first assembled in orbit and will require this orbit maintenance. Later on as the mass is built up, the decay rate will decline. This case has not yet been analyzed.

Figure 1:

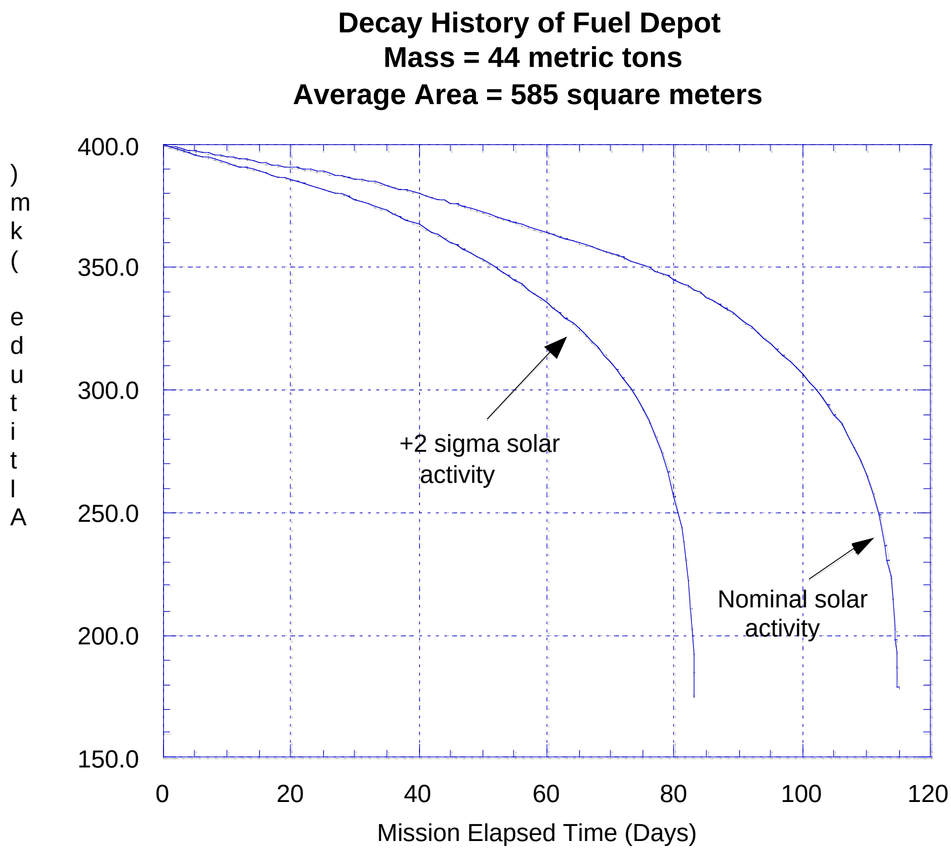


Figure 2:

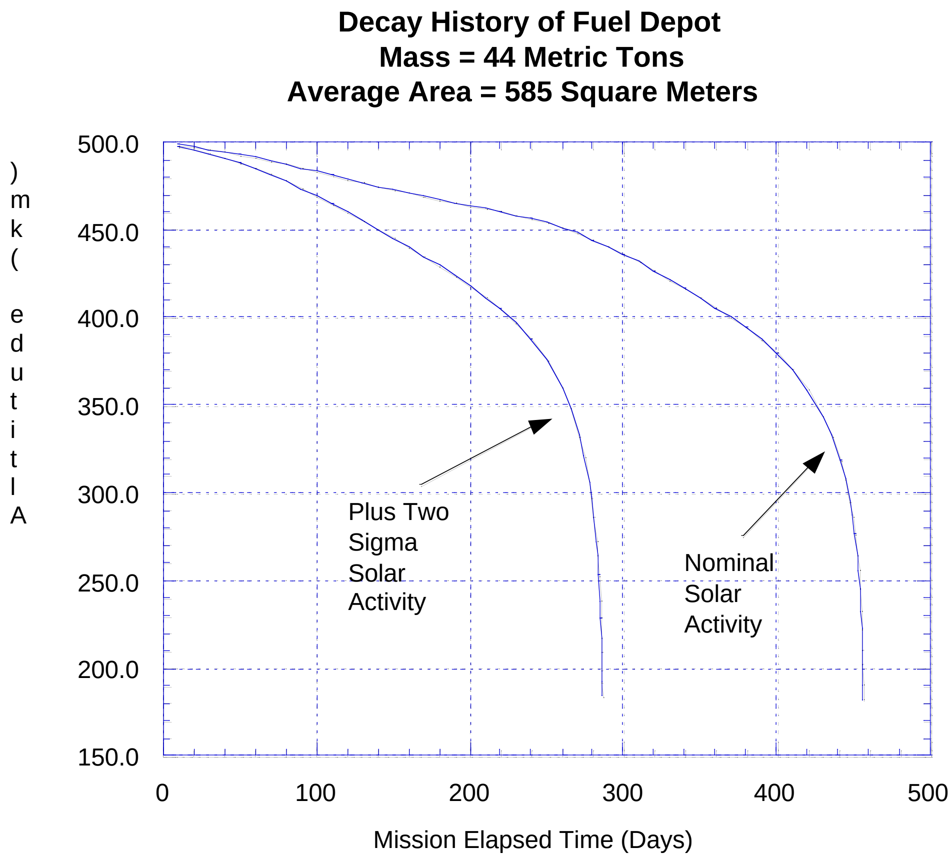
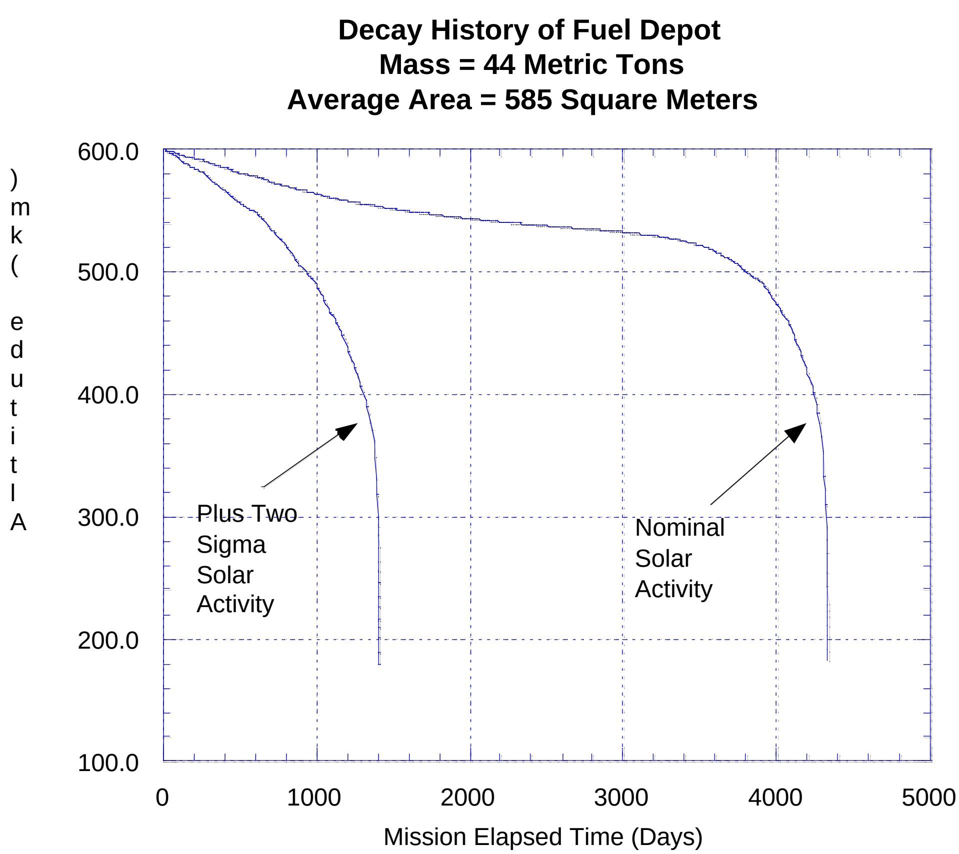

Decay History of Fuel Depot
Mass = 44 Metric Tons
Average Area = 585 Square Meters
Altitude (km)
500.0
450.0
400.0
350.0
300.0
250.0
200.0
150.0
0
100
200
300
400
500
Mission Elapsed Time (Days)
Plus Two Sigma Solar Activity
Nominal Solar Activity

Figure 3:


Decay History of Fuel Depot
Mass = 44 Metric Tons
Average Area = 585 Square Meters
Altitude (km)
600.0
500.0
400.0
300.0
200.0
100.0
0
1000
2000
3000
4000
5000
Mission Elapsed Time (Days)
Plus Two Sigma Solar Activity
Nominal Solar Activity

Fuel Depot Orbit Reboost Propellant Analysis

Craig Cruzen, Dr. Larry Mullins
NASA MSFC TD54
September 19, 2000

Introduction

This study was done for a mission concept using an unmanned fuel depot at a given circular orbit. Once the orbit of the depot has decayed to a given lower altitude, the depot uses a Hohmann transfer, to raise its altitude to a given higher orbit and then recircularizes.

Assumptions

The following assumptions were made for the study:

Specific impulse of the reboost engine	440 sec (Based upon a RL10A class engine)
Propulsion inefficiency due to gravity losses	1% of total delta-velocity (Delta-V) required
Final mass of the depot at the higher altitude	44,000 kg (Empty fuel depot)

Reboost Propellant Analysis

The reboost periods listed below were based upon the orbital decay analysis previously cited. The propellant required for each reboost was calculated with the rocket equation using the required Delta-V for Hohmann type transfers and the assumptions listed above. As seen in the table below, the greater the total altitude of reboost, the more propellant required. However as the upper altitude increases, the time between reboost events increases. Thus the amount of propellant required per year decreases. For example, to reboost the depot from 350 km to 400 km takes 293.8 kg of propellant and this will have to be done every 115 days with nominal solar activity or every 85 days with +2 sigma solar activity. To reboost from 500 to 600 km takes more propellant, 567.2 kg, however the maneuver only need be done every 3800 days given nominal solar activity or 900 days for +2 sigma solar activity. Thus for the 350 to 400 km reboost, the depot will use 932.5 kg per year in nominal solar activity while only 54.5 kg per year is required for the 500 to 600 reboost case.

Lower Altitude (km)	**Higher Altitude (km)**	**Hohmann Delta-V w/ 1% G-Loss (m/s)**	**Propellant Required per Reboost (kg)**	**Reboost Period, Nominal Solar Activity (days)**	**Reboost Period, +2 Sigma Solar Activity (days)**	**Propellant Required per Year, Nom Solar Activity (kg)**	**Propellant Required per Year +2 Sigma Solar Activity (kg)**
350	400	28.7	293.8	115	50	932.5	2144.7
350	500	85.2	877.5	430	260	744.9	1231.9
350	600	140.5	1456.0	4300	1400	123.6	379.6
400	450	28.4	290.6	140	90	757.6	1178.5
400	500	56.5	579.8	360	230	587.9	920.1
400	600	111.8	1154.5	4300	1400	98.0	301.0
500	550	27.8	284.2	2400	400	43.2	259.3
500	600	55.3	567.2	3800	900	54.5	230.0

The following plots show the relationships between the reboost altitudes, periods and propellant amounts required to maintain those altitude ranges. The first figure illustrates that as the distance between the upper and lower reboost altitudes increases, so does the propellant required to perform that reboost maneuver.

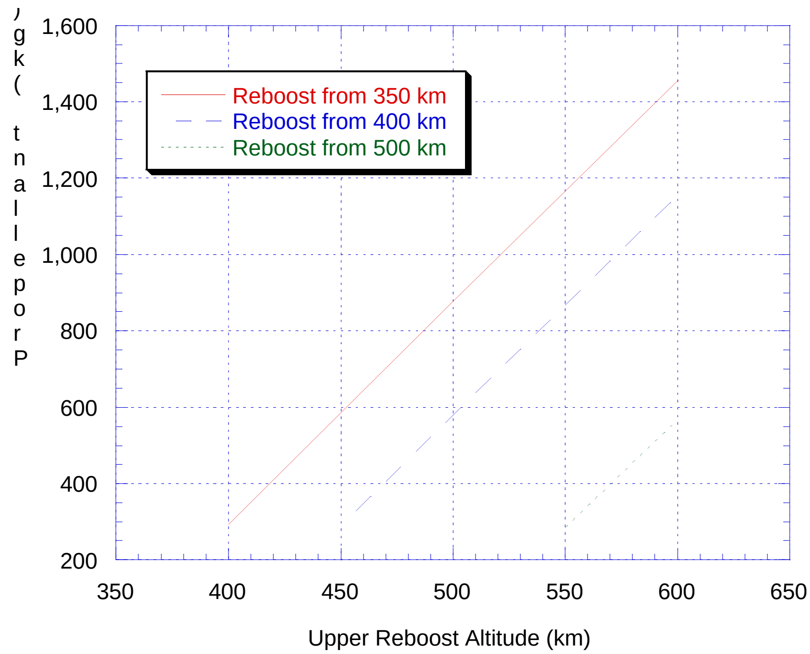


Figure 1: Propellant Required per Reboost Maneuver

Figure 2 shows the relationship between the altitude reboost ranges and the frequency of the reboost maneuvers in a nominal solar cycle. As the range between the upper and lower altitudes increases, the time between reboost maneuvers also increases. This is because as the upper altitude increases, the atmospheric drag on the depot is less. This increases the time it takes for the depot's orbit to decay down to the lower altitude. Note that the curves are not smooth because they include only three points per line. A complete description of the curves would be exponential. Figure 3 shows the same relationship for a +2-sigma solar cycle. In this case, increased drag from the Earth's atmosphere caused by higher than normal solar activity results in an increased orbital decay rate.

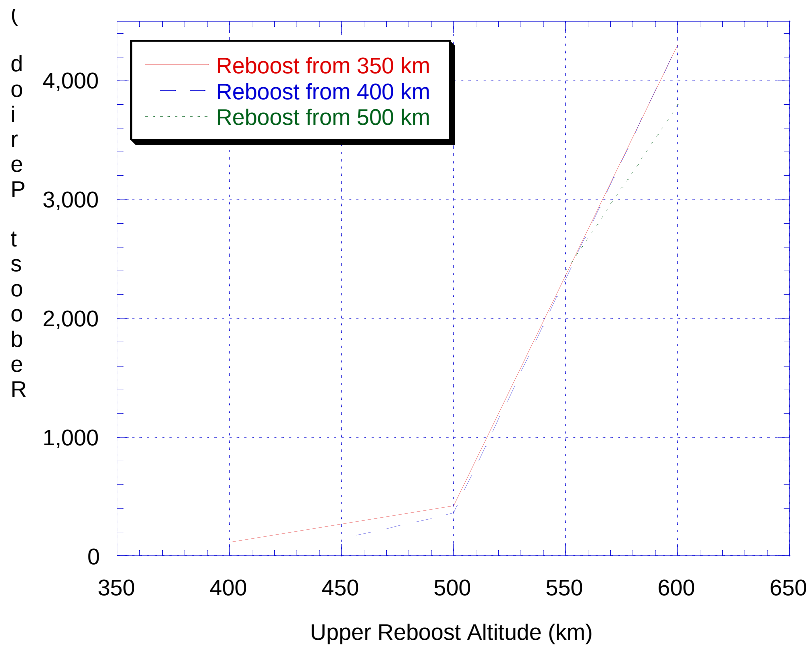


Figure 2: Orbit Reboost Period During Nominal Solar Activity

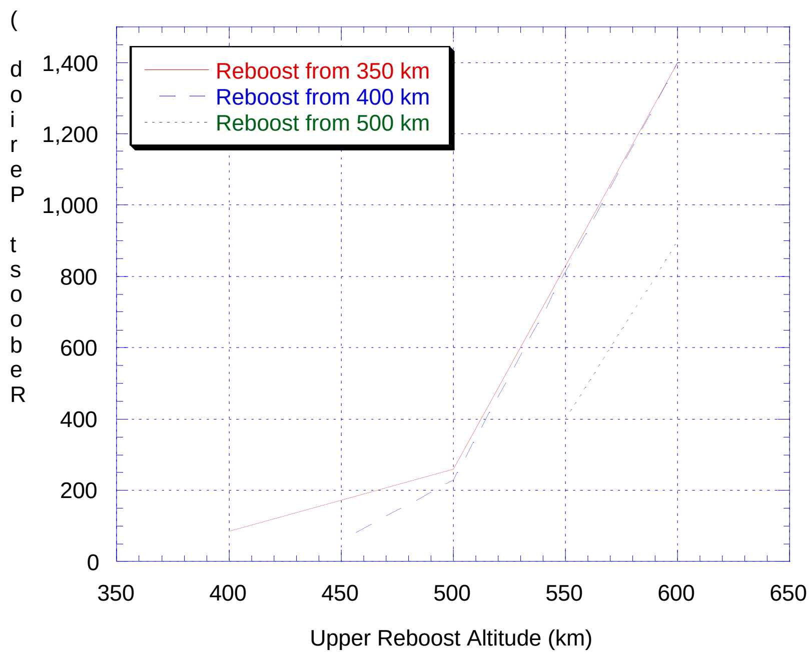


Figure 3: Orbit Reboost Period During +2-Sigma Solar Activity

Figure 4 shows the relationship between the altitude reboost ranges and the amount of propellant required per year for a nominal solar cycle. As the range between the upper and lower altitude increases and the decay rate decreases, the propellant used per year for reboosting naturally decreases. Figure 5 shows the same relationship for a +2-sigma solar cycle.

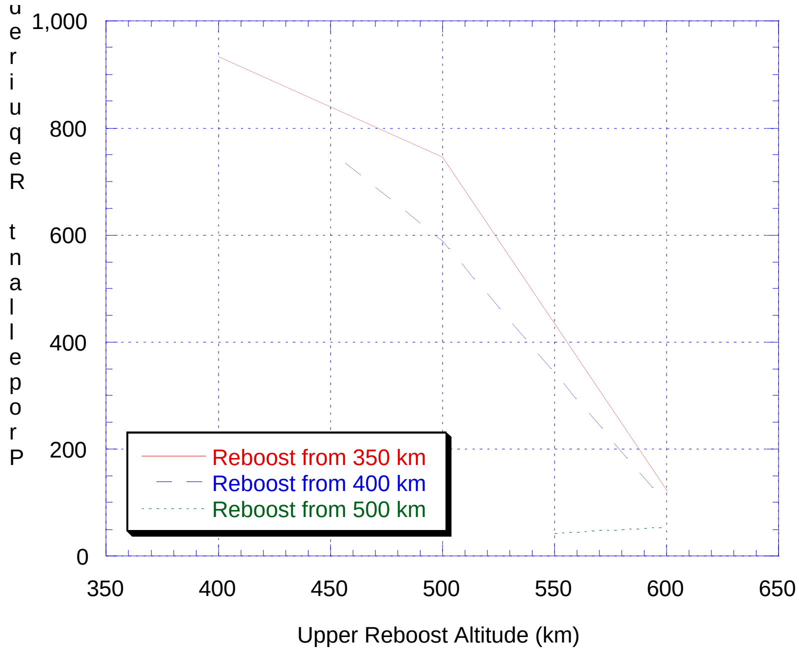


Figure 4: Orbit Reboost Propellant Required per Year During Nominal Solar Activity

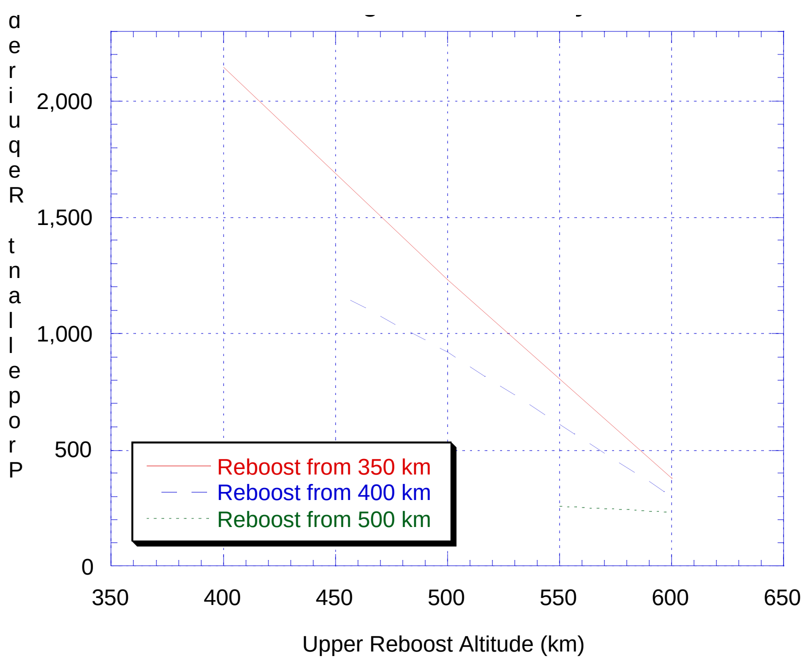


Figure 5: Orbit Reboost Propellant Required per Year During +2-Sigma Solar Activity