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SUMMARY

A study was performed to quantify the benefits of using slush hydrogen instead of normal boiling point liquid hydrogen as a fuel for several space missions. Vehicles considered in the study included the Space Shuttle/Shuttle-C, LEO-to-GEO transfer vehicles, Lunar and Mars transfer vehicles, and cryogenic depots in low Earth orbit. The advantages of using slush hydrogen were expressed in terms of initial mass differences at a constant payload, payload differences at a constant tank volume, and increases in fuel storage time for cryogenic depots. Both chemical oxygen/hydrogen and hydrogen nuclear thermal rocket propulsion were considered in the study. The results indicated that slush hydrogen offers the potential for significant decreases in initial mass and increases in payload for most missions studied. These advantages increase as the mission difficulty, or energy, increases.

INTRODUCTION

Solid-liquid mixtures of hydrogen, known as slush hydrogen, are currently being considered to fuel the National Aero-Space Plane (NASP) (refs. 1 and 2). Slush hydrogen offers the advantages of increased density and heat capacity when compared with normal boiling point liquid hydrogen. The use of dense slush hydrogen rather than liquid as the cryogenic working fluid will yield a reduced NASP vehicle size and correspondingly lighter weight vehicle. The benefits of slush hydrogen use are not limited to NASP applications, however. Previous studies have shown the advantages of using slush hydrogen for space vehicles (refs. 3 to 5), but these studies were usually limited to specific missions or vehicle designs not currently of interest. Therefore, a study to determine the benefits of slush hydrogen use for several existing and planned space vehicles appeared appropriate.

This report summarizes a study performed by Science Applications International Corporation (SAIC) and Martin Marietta Astronautics Group (MMAG), under a NASA task order contract, to quantify the benefits of using slush hydrogen for several space missions. Four applications were considered in the study:

(1) Earth-to-orbit transportation vehicles (Space Shuttle and Shuttle-C)

(2) Low earth orbit (LEO)-to-geosynchronous orbit (GEO) transfer vehicles (expendable and reusable)

(3) Exploration mission transfer vehicles for the Moon and Mars (Lunar and Mars Outpost Missions)

(4) Cryogenic depots in LEO

The exploration missions were based on recent investigations of options for the Lunar Mars Initiative (refs. 6 and 7).

The benefits of slush hydrogen usage have been determined on the basis of the following criteria:

(1) Total system or initial mass differences at a constant payload

(2) Payload differences at a constant tank volume

(3) Increases in fuel storage time for space-based cryogenic depots

These benefits were based on comparisons with similar systems that used normal boiling point and triple point liquid hydrogen. The propulsion systems considered were cryogenic oxygen/hydrogen (O/H) and, for the lunar and Mars missions, nuclear thermal rocket (NTR). See the appendix for a complete list of acronyms used herein.

SYMBOLS

A_t	tank surface area, m^2
D	tank diameter, m
d	insulation thickness, m
I_{sp}	specific impulse, sec
k_{ab}	aerobrake mass fraction
k_c	tank wall thermal conductivity, W/m-K
k_s	tankage fraction
L	tank length, m
q	tank wall heat transfer rate, W/m^2
ΔT	temperature drop across tank wall, K
ΔV	mission velocity increment, m/sec

ASSUMPTIONS

Fluid Properties

The properties of normal boiling point hydrogen ($NBPH_2$), triple point hydrogen (TPH_2), slush hydrogen (SLH_2), and liquid oxygen (LOX) are presented in table I. Note that slush hydrogen, at a solid fraction of 50 percent, provides a 15-percent advantage of increased density over that of normal boiling point hydrogen. Triple point hydrogen offers an 8-percent increase in density over normal boiling point hydrogen. In addition to density increases, slush hydrogen also provides a heat capacity improvement compared with that of normal boiling point liquid hydrogen. The heat sink for the normal boiling point liquid shown in the table is the heat of vaporization. The increase in the heat sink for triple point hydrogen results from the addition of sensible heat (temperature difference), which leads to an increase of approximately 12 percent in heat capacity. The heat sink for slush hydrogen increases from the heat of fusion and sensible heat, which results in an 18-percent improvement. These properties were used throughout the study to determine the benefits of slush hydrogen use.

Earth-to-Orbit Transportation Vehicles

For Earth-to-orbit vehicles, the Space Shuttle (STS) and Shuttle-C (STS-C) were considered. In the case of the existing Shuttle design, it was not deemed appropriate to consider tank redesign; therefore, only payload differences at a constant volume were studied. The baseline case was normal boiling point liquid hydrogen, the fuel used presently by the Shuttle. A 6:1 mixture ratio (by mass) was assumed, with an effective specific impulse (I_{sp}) of 452.4 sec, and an external tank (ET) hydrogen volume of 51 574 ft^3. Although the general ground rule for this study was to ignore specific impulse changes for the denser triple point and slush hydrogen states at constant tank volume, in the case of the Shuttle design, the specific impulse effect was included to show the maximum possible payload gain. Therefore, the hydrogen-rich mixtures can have an increased specific impulse of 452.9 sec for triple point hydrogen and 453.4 sec for the 50-percent slush hydrogen. The amounts of slush hydrogen loaded in the ET for the three states were 227 857, 248 126, and 263 547 lbm, respectively. Performance data were generated by using the ascent trajectory simulation program FLY-IT (ref. 8).

GEO Transfer Vehicles, Lunar and Mars Outpost Missions

For the comparisons made herein, the boiloff of hydrogen was included in the propellant requirements. The boiloff rates were obtained by using a simple calculation for heat transfer through the wall:

$$q = \frac{k_c \Delta T}{d}$$

where q is the heat transfer rate (W/m^2), ΔT is the temperature drop across the wall (K), and d is the insulation thickness (m). The resulting heat transfer was then used with the heat sink values from table I to obtain the boiloff rates shown in table II. In the study it was assumed that double-aluminized mylar, double-silk net multilayer insulation (MLI) at 20 layers/cm would be used; the thermal conductivity of this material is 4.5×10^{-5} W/m-K. Information about various cryogenic insulations is found in reference 9. An outside environmental temperature of 270 K was used for the calculations. It should be noted that these calculations did not include factors such as one tank shadowing another, which would reduce the actual heat transfer.

Depending on the mission application, various values of the insulation thickness were assumed. For GEO missions, because of the relatively short duration of the transfer, no boiloff was assumed and no insulation was required. For the lunar missions, a thickness of 0.02 m was assumed; for the Mars missions and the cryogenic propellant depot in LEO, a thickness of 0.08 m was used. Note, however, that in the case of the Mars transfer, the first stage used for trans-Mars injection (TMIS) had only 0.02 m of MLI because of its relatively short operating time.

Tankage mass plays an important role in determining the vehicle parameters for each mission. Assumptions used for this study are shown in table III. The tankage mass consisted of the insulation, the metal structure, and some integration/holding structure for the oxygen and hydrogen tank sets. The insulation mass is proportional to both the MLI thickness and the tank surface area. For the calculation, the assumed reference geometry was based on a hemispherically domed, cylindrical tank with a length-to-diameter ratio (L/D) of 1 for the cylindrical section. Five percent of the tank volume was assumed to be ullage.

The tanks were constructed of an aluminum-lithium alloy. The structure for these tanks was calculated as a fraction of the fluid contained in the tank. For the case of normal boiling point hydrogen, the relatively high vapor pressure requires that the structure fraction be high enough to account for the pressure hoop stress. For triple point or slush hydrogen, however, the vapor pressure of only about 1 psi allows for a lower tankage fraction, with the launch loads determining the structural mass. For the cases in this study, the tankage fraction was assumed to be 0.09 for $NBPH_2$ and 0.03 for either TPH_2 or SLH_2. For liquid oxygen, the launch loads also determine the tankage fraction, so a value of 0.02 was used for the LOX tankage. It should be noted that, because the tankage fraction plays an important role in the determination of total masses, future studies may be required to determine the actual tankage fractions.

Depending on the particular application and staging assumptions, the number of tanks used in the study was an input for the analysis. Generally, the number of tanks varied from between one and four, with an attempt to obtain reasonable dimensionality. In any given vehicle stage (i.e., ΔV maneuver), equal sized tanks were assumed for the LOX, and similarly equal but different sized tanks were assumed for the hydrogen. An additional 3 percent of the tank set dry mass was included to account for the holding structure.

Table IV shows the assumptions made for the propulsion system and the transfer vehicles. For most cases in which cryogenic O/H propulsion systems were used, a constant mixture ratio of 6:1 and a specific impulse of 481 sec (corresponding to an advanced space engine) were assumed. The only exception to this value of I_{sp} was the trans-Mars injection stage (TMIS), which operated at 475-sec specific impulse. In performance comparisons at constant tank volume, an increase in the amount of hydrogen occurred for triple point or slush hydrogen cases because of the density increase, which decreased the oxygen-to-hydrogen mixture ratio and led to a small increase in specific impulse. For the purposes of this study, however, with operation near the optimal mixture ratio, no change in specific impulse was assumed for the TPH_2 and SLH_2 cases as the mixture ratio changed. For the nuclear thermal rocket propulsion applications, a constant specific impulse of 900 sec was used (ref. 10).

In all cases, 5 percent of the propellant loading was assumed to be unusable for primary thrusting (performance margin and vehicle attitude control). The total mass of a stage included the core stage, the dry tanks, the payload structural adaptor, and the aerobrake. The core stage masses were input constants for each application and are given in table V. The dry tank sets were scaled as previously discussed, and the payload adaptor was assumed to be 3 percent of the payload.

In general, aerobraking was employed for the return to Earth on each of the GEO, lunar, and Mars mission examples. Aerobraking (aerocapture) return was to low Earth orbit for the GEO and lunar missions. Two return modes were examined for Mars: direct entry of a small crew capsule or the aerocapture of the large crew module to LEO. The aerobrake mass for the aerocapture was assumed to be proportional to the mass at entry. An aerobrake fraction of 20 percent was used for GEO and Lunar Outpost missions; however, an aerobrake of 15 percent was used for the Mars Outpost Mission because of the reduced fraction resulting from the probable nonlinearity of the aerobrake fraction for large entry masses.

Performance calculations were made for 1, 2, or 3 stages, depending on the mission. Staging herein refers to the jettisoning of expended tanks after a major ΔV maneuver. The exception here was for the Mars missions in which the first stage was totally jettisoned after the injection burn. Also, the single case in which 3 stages were employed was for the nuclear thermal rocket (NTR) sprint mission to Mars, which required a large midcourse ΔV maneuver. The actual mission velocity increments used in the study are shown in table VI.

Table VII gives the payload masses used in the study. For the GEO mission, both expendable and reusable space transfer vehicles (STV) were considered. The payload masses for these cases were parameterized over the range of 2 to 20 metric tons, and only cryogenic O/H propulsion was used. For the lunar missions, both O/H chemical propulsion and NTR were used. The outbound payload to low lunar orbit consisted of the fully loaded lunar excursion vehicle (LEV), which was 46 metric tons. The inbound, or return, payload was 6.6 metric tons. For the Mars missions, the excursion lander/ascent vehicle (MEV) was treated as payload. The outbound payload mass was 87.36 metric tons; it included Earth-to-Mars consumables, a communications relay orbiter, and the very massive MEV. The return payload was 44.64 metric tons, which included consumables, the large crew module carried throughout the round-trip mission, and a smaller direct-entry capsule carried on all missions for safety, even if the crew module was nominally returned to LEO. Note that the total payload carried on the outbound legs of these missions is the sum of the masses identified in table VII.

The Mars mission propulsion and payload return options considered are shown in table VIII. The options that were chosen for study are indicated by "x". Four missions were considered in the analysis for launch years 2015, 2016, 2017, and 2018. The round-trip times for these missions were 565, 400, 654, and 942 days, respectively. In the case of the crew capsule return, the large crew module was separated from the vehicle on approach to Earth.

Cryogenic Depots in LEO

In order to quantify the benefits of using slush hydrogen for a cryogenic depot in low Earth orbit, two different types of depots were examined. The first, based on a space transportation system utilizing chemical (O/H) propulsion with a mixture ratio of 6:1, assumed a depot sized to 100 metric tons of hydrogen and 600 metric tons of oxygen. The second, based on the use of nuclear thermal propulsion with only hydrogen propellant, assumed a depot size of 300 tons of hydrogen. All individual tanks were sized to contain 100 tons of either hydrogen or oxygen. An additional structural lien of 5 percent of the propellant mass was included in the depot's dry mass. As discussed previously (table II), all tanks were insulated with 0.08 m of double-aluminized mylar MLI. The tank mass fractions used were those discussed previously (table III). Five percent of the tank volume was taken to be ullage.

RESULTS

Earth-to-Orbit Transportation Vehicles

Figure 1 shows a comparison of the Space Shuttle payload with $NBPH_2$, TPH_2, and SLH_2 for delivery of payload to a 220-n.mi. circular reference orbit due east of Cape Canaveral, Florida. As discussed in ASSUMPTIONS, these comparisons were made by considering only the payload gain at a constant volume; no Space Shuttle redesign was considered. The baseline payload using $NBPH_2$ was 39 000 lbm. When triple point hydrogen was used, approximately 1300 lbm in payload was gained if the density increase was considered, and a 2026-lbm increase in payload was evident if both the increased I_{sp} and the density increase were taken into account. The payload increase for triple point hydrogen was, therefore, up to 5 percent. As discussed previously, the specific impulse benefit is attributed to a decrease in the oxygen/hydrogen mixture ratio, which results from additional hydrogen at the higher fuel density. The density increase alone allows for additional fuel to be placed in the external tank, thereby increasing the Shuttle payload capability. If slush hydrogen is used as the fuel, the payload increase will be approximately 2000 lbm for only density increases and 3670 lbm for the combined effects of I_{sp} and density. This payload mass represented a 9-percent payload increase. As a reference point, one flight-ready, Space Shuttle main engine (SSME) weighs about 6885 lbm.

It should be noted that no additional equipment was included in the analysis. For the case of 50-percent slush hydrogen, an additional heat exchanger may be required to assure that liquid reaches the inlet of the turbopump compressor. The added equipment must necessarily be lower than the gains shown here for the slush hydrogen to provide a benefit for Shuttle missions. Also, practical operational considerations, such as the Shuttle hydrogen residuals and structural limits on the external tank, should be considered in future studies.

Figure 2 compares the payload gain when triple point and slush hydrogen are used for Shuttle-C applications. The simulation assumed three SSME's, each operating at a 104-percent thrust level. At a constant tank volume, the payload was 150 000 lbm when normal boiling point hydrogen was used. When triple point hydrogen was used, the payload increased by 2000 lbm (combined I_{sp} and density effects), corresponding to a 1.3-percent increase in payload. When 50-percent slush hydrogen was used, the payload increase, combining both density and I_{sp} effects was 3700 lbm (2.5 percent). The lower percentage gain for Shuttle-C when compared with that of the Shuttle can be explained thus: the Shuttle-C mass-to-orbit is mostly payload whereas in the Shuttle application this mass included the heavy orbiter. However, for both the Shuttle and Shuttle-C, the results indicated payload gains of 2000 to 3700 lbm when slush hydrogen was used.

GEO Transfer Vehicles

Figure 3 shows the results of comparisons of slush and triple point hydrogen with normal boiling point hydrogen for LEO-to-GEO space transfer vehicles (STV). Both one-way expendable (STV(EX)), and round-trip reusable (STV(R)), missions were considered, with a range of payloads from 2 to 20 metric tons. Figure 3(a) shows the initial masses in LEO used for the comparisons to follow. The reusable vehicle requires more initial mass in LEO because a larger propellant mass is needed in comparison with that needed for the expendable vehicle.

Figure 3(b) shows the initial mass reduction at a constant payload for the GEO missions. The results were essentially the same for triple point and slush hydrogen. For the reusable flight mode, a mass savings of 1.6 to 2 percent was calculated, corresponding to an absolute mass savings of approximately 0.30 to 1 metric ton. For the expendable mode, the initial mass savings was approximately 1.5 percent for the range of payloads, leading to decreases of 0.15 to 0.90 metric ton. The mass savings was attributed to reduced hydrogen tankage and propellant loading.

Figure 3(c) shows the payload gain at a constant volume when triple point and slush hydrogen were used. The payload gains ranged from 2.9 to 8.9 percent, depending on the payload level, mission type, and fuel. The ranges of absolute payload gain were approximately 0.1 to 0.6 metric ton for TPH_2 and 0.12 to 0.84 metric ton for SLH_2. This increased payload resulted from the increased fuel load with the denser hydrogen fuels and came partly at the expense of an increased initial mass. Figure 3(d) shows that the increase in initial mass was approximately 1 percent for TPH_2 and approximately 2 percent for SLH_2. The largest initial mass increase was 1.4 metric tons, corresponding to the use of SLH_2 in a reusable STV for a nominal 20-metric ton payload. Therefore, for the LEO-to-GEO transfer vehicle sizes studied, SLH_2 appeared to provide modest benefits, as shown in figures 3(a) and (b).

Lunar Outpost Mission

Figure 4 shows the comparison for the Lunar Outpost Mission considered in this study. The reference initial masses in LEO with $NBPH_2$ (fig. 4(a)) for the flight modes studied are 171.8 metric tons for the cryogenic (O/H) lunar transfer vehicle with aerobrake (CRYO LTV-AB), 110.2 metric tons for NTR propulsion with aerobrake return (NTR LTV-AB), and 126.3 metric tons for NTR with propulsive return to LEO (NTR LTV). The total tankage volume of the $NBPH_2$ (all tanks) is 207 m^3 for the CRYO LTV-AB, 600 m^3 for the NTR LTV-AB, and 818 m^3 for the NTR LTV.

Figure 4(b) shows a comparison of initial mass savings at a constant payload. The mass reductions were similar for both TPH_2 and SLH_2 and were approximately 1.5 percent (2.6 metric tons) for the CRYO LTV-AB, 4 percent (4.4 metric tons) for the NTR LTV-AB, and 6 percent (7.6 metric tons) for the NTR LTV. As expected, the improvement with the NTR was larger because all the propellant was hydrogen. Most of the initial mass savings in the chemical LTV mission was attributed to the reduced LOX load and the reduced mass of hydrogen tanks in the first stage. In the case of the NTR, the savings was mostly in lighter tanks and the reduced hydrogen load of the first stage.

The percentage increases in the payload at a constant tank volume are shown in figure 4(c). The payload gains were compared with the 46-metric ton outbound payload only; the inbound payload was held constant. From the figure it can be seen that the TPH_2 runs showed an increase of 4 percent (1.8 metric tons) for the CRYO LTV-AB case, 17 percent (7.8 metric tons) for the NTR LTV-AB, and 19 percent (8.7 metric tons) for the NTR LTV. The use of SLH_2 further improved the payload gain: for the chemical LTV, the gain was 5 percent (2.3 metric tons); for the nuclear thermal propulsion cases, the gain ranged from 27 to 29 percent (12.4 to 13.3 metric tons). The corresponding increases in initial mass associated with these payload gains ranged from 2 to 19 metric tons (fig. 4(d)). Therefore, the use of SLH_2 for lunar missions appears to provide significant benefits, especially if NTR propulsion is used.

Mars Outpost Mission

Transportation support for the Mars Outpost Mission requires total system mass levels at least 4 times greater than those of the Lunar Outpost Mission. Figure 5 gives data generated for the mission to Mars using cryogenic O/H propulsion for three launch year-flight mode opportunities. The study assumed that only the crew capsule was returned to Earth (the large crew module was separated on approach to Earth). The reference mission initial masses in LEO with $NBPH_2$ (fig. 5(a)) were 814.2 metric tons for the nominal 2015 launch, 657.7 for the 2017 launch, and 557 for the 2018 minimum-energy, conjunction-class launch. Another assumption was that there were four tank sets in the first trans-Mars injection stage (TMIS) and two tank sets in the second stage; for these cases, the total tankage volumes of NBH_2 were 1170, 883, and 689 m^3, respectively.

Compared with the $NBPH_2$ performance at a constant payload, TPH_2 showed initial mass savings ranging from 1.7 to 2.5 percent (fig. 5(b)), with corresponding absolute decreases of 8.4 to 20.4 metric tons. For SLH_2, savings ranged from 1.8 to 2.7 percent, corresponding to mass decreases of 10 to 22 metric tons. Most of the savings resulted from reductions in the propellant load and the hydrogen tankage mass. The propellant boiloff differences were not significant.

Payload gains at a constant tank volume are shown in figure 5(c). These percentages were in reference to the 87.36-ton outbound payload. These gains in payload ranged from 5 to 7 percent for TPH_2 (4.4 to 6.1 metric tons) and 7 to 9 percent for SLH_2 (6.1 to 7.9 metric tons). The corresponding increase in initial mass (fig. 5(d)), ranged from 7 to 10 metric tons for TPH_2 and from 12 to 17.5 tons for SLH_2.

Figure 6 shows the results calculated for the cases in which the entire crew module is returned to LEO via aerobraking. For these, the initial mass requirements increased, especially for the 2015 mission because of its higher ΔV. The initial masses (fig. 6(a)), were 1054 metric tons for the 2015 launch date, 733 for the 2017 launch, and 611 for the 2018 launch. The payloads for these missions were the same as those with the crew capsule return. From figure 6(b), TPH_2 offered an initial mass savings of 1.9 to 3.3 percent, with corresponding mass decreases of approximately 11 to 34 metric tons. Slush hydrogen gave an initial mass savings of 2 to 3.5 percent, or savings of 12 to 37 metric tons. The savings was higher for the case of the crew module return in comparison with that of the crew capsule return because the crew module return requires more energy.

The payload gain at a constant volume is shown in figure 6(c). The payload gains ranged from 5.6 to 9.2 percent (5 to 8 metric tons) for TPH_2 and from 7.5 to 11.1 percent (6.5 to 9.7 metric tons) for SLH_2. The total mass increase corresponding to this increase in payload (fig. 6(d)), ranged from 7.2 to 23 metric tons.

Figure 7 compares the TPH_2 and SLH_2 performance with that of $NBPH_2$ for the Mars Outpost Mission utilizing NTR propulsion. For purposes of comparison, the analysis was limited to the 2015 reference mission with the crew capsule only and with the crew module return. In addition, a "sprint" mission of 400 days was included, with a crew capsule return and a 2016 launch date. Initial mass requirements in LEO for these missions with $NBPH_2$ were 413 metric tons for the 2015 launch with crew capsule return, 468 metric tons for the 2015 launch with crew module return, and 697 tons for the 2016 sprint mission (fig. 7(a)). For the two 2015 missions, four tanks were assumed in the first stage and one tank in the second stage; the total tanked volumes of hydrogen were 2825 and 3445 m^3, respectively. For the 2016 mission, the assumed number of tanks carried in the first, second, and third stages were four, two, and one; the total tanked volume was 6304 m^3.

Figure 7(b) shows the initial mass decreases at a constant payload for the NTR Mars Outpost Missions. Compared to $NBPH_2$ performance, TPH_2 or SLH_2 provided nearly the same decrease in initial mass: 6.6 to 8.2 percent for the 2015 flights and just over 10 percent for the sprint flight. On an absolute scale, the initial mass differences for the three cases examined were 27.2, 36.6, and 70.4 metric tons for the TPH_2 cases and 28.5, 38.5, and 73.8 metric tons for the SLH_2 cases. The savings was attributed mostly to the reduced hydrogen load and the lighter tanks in the first stage.

Figure 7(c) shows payload gains at a constant tank volume. Triple point hydrogen provided gains of 25 to 31 percent, corresponding to mass increases of 22 to 37 metric tons. Slush hydrogen provided gains of 38 to 44 percent, which resulted in an increase of 33 to 39 tons above the initial outbound payload of 87.36 metric tons. Figure 7(d) shows that the corresponding increases in initial mass ranged from 24 to 39 metric tons for the TPH_2 cases and 62 to 71 tons for the SLH_2 cases.

Slush hydrogen has been shown to provide large initial mass decreases as well as significant payload increases for the Mars missions considered herein. These benefits are larger when NTR propulsion is used because the propellant is primarily hydrogen. These initial mass decreases can be translated into a reduced number of launches to LEO, providing the potential for a faster vehicle assembly rate in LEO and a reduced mission launch cost.

Cryogenic Depots in LEO

In addition to benefits for space transfer vehicles, SLH_2 and TPH_2 can provide benefits for an orbiting cryogenic depot. These include lower rates of boiloff and reduced tankage mass (resulting from higher fuel densities and lower vapor pressure in the hydrogen tank). Table IX lists the tank mass values and the derived boiloff rates and includes the refrigeration power needed to prevent any boiloff. In consideration of a depot such as those described herein, a tradeoff can be made between passive cooling (boiloff) and active cooling (refrigeration). For passive cooling, it is assumed that venting through a vapor-cooled shield would keep the vapor pressure in the tank near 1 psi, representing the triple point of hydrogen. For active cooling, turbo-Brayton refrigerators operating at 10 percent of the ideal Carnot efficiency at liquid oxygen temperatures were assumed. This system was chosen because of the demonstrated high reliability (4-yr ground tests) and operation at high power levels (ref. 11). Therefore, if table IX is examined, to obtain no boiloff of the $NBPH_2$, refrigeration power of 11.9 kWe is required. If, on the other hand, no refrigeration power is available or desired, then 0.53 percent/month of $NBPH_2$ boils off. It should be noted that the reason for the higher power requirement of TPH_2 or SLH_2 compared with that of $NBPH_2$ was the reduced Carnot efficiency at the lower temperatures.

Table X lists the total dry mass of the orbiting depot for the different depot types and for passive and active cooling methods. The power system mass was based on an assumed specific mass of 20 kg/kWe, which included solar panels, power conversion, and refrigeration equipment. The use of TPH_2 or SLH_2 offered a fairly significant decrease in depot dry mass, especially in the case of a hydrogen-only depot, such as an NTR depot. These mass savings were 10 percent for the O/H storage system and 37 percent for the hydrogen-only system. It should also be noted that the added mass of an actively cooled depot was calculated to be equivalent to less than 1 month of a boiloff. Therefore, the tradeoff between active and passive cooling may be worth further investigation.

Figure 8 shows the boiloff as a function of storage time. By using slush hydrogen, 5 months of additional storage time is provided to reach 90 percent of the initial tank loading (fig. 8). The increased storage times are attributed to SLH_2 increased heat capacity.

CONCLUDING REMARKS

A study was conducted to determine the benefits of using slush hydrogen for several space missions, which included Earth-to-orbit transportation applications, low Earth orbit (LEO)-to-geosynchronous orbit (GEO) transfer vehicle missions, Lunar and Mars Outpost Missions, and cryogenic depots in LEO. Slush hydrogen (SLH_2) and triple point hydrogen (TPH_2) were compared with baseline normal boiling point hydrogen ($NBPH_2$) results, and benefits were expressed in terms of initial mass decreases at a constant payload, payload gains at a constant tank volume, and increases in storage time for cryogenic depots. Both cryogenic oxygen-hydrogen (O/H) and nuclear thermal rocket (NTR) propulsion systems were used for the benefits comparison.

The results indicated that the use of slush hydrogen offers potential advantages of lighter weight tanks, lower boiloff rates, and higher density propellants, which can translate into significant mass benefits for space transportation. As the difficulty of the missions increased (increase in the mission velocity increment), the benefit of using slush hydrogen also increased. The advantages of using slush hydrogen are summarized as follows:

(1) Space Shuttle/Shuttle-C: 2000 to 3700-lbm payload gain using existing external tank configuration

(2) GEO transfer vehicle: 0.2- to 1.0-metric-ton initial mass decrease (2-percent decrease), 0.1- to 0.8-metric-ton payload gain (4- to 9-percent increase)

(3) Lunar transfer vehicle:

- Cryogenic O/H propulsion: 2.6-metric-ton initial mass decrease (1.5-percent decrease), 2.3-metric-ton payload gain (5-percent increase)

- NTR propulsion: 4.4- to 7.6-metric-ton initial mass decrease (4- to 6-percent decrease), 12.4- to 13.3-metric-ton payload gain (27- to 29-percent increase)

(4) Mars transfer vehicle:

- Cryogenic O/H propulsion: 10- to 37-metric-ton initial mass decrease (2- to 3.5-percent decrease), 6- to 10-metric-ton payload gain (7.5- to 11-percent increase)

- NTR propulsion: 29- to 74-metric-ton initial mass decrease (6.6- to 10-percent decrease), 33- to 39-metric-ton payload gain (38- to 44-percent increase)

(5) Cryogenic depot in LEO: 5-month increase in storage time (to reach 90-percent level of initial tank loading), 10-percent decrease in system dry mass for the O/H storage system or 37 percent for a hydrogen-only system

Several issues still must be examined before slush hydrogen is used for these missions: the ability for long-term storage, the transfer capability in zero gravity, verification and testing of insulation systems, and the definition and testing of any additional components for these space vehicles. In addition, future studies should consider the effects of tankage fraction, aerobrake mass, and specific impulse on the benefits of using slush hydrogen. However, the present study shows that slush hydrogen has the potential to provide significant performance benefits for space transportation and storage vehicles.

APPENDIX–NOMENCLATURE

AB	aerobrake
CRYO	cryogenic depot
ET	Space Shuttle external tank
EX	expendable
GEO	geosynchronous orbit
LEO	low Earth orbit
LEV	lunar excursion vehicle
LMI	Lunar Mars Initiative
LOX	liquid oxygen
LTV	lunar transfer vehicle
MEV	Mars excursion vehicle
MLI	multilayer insulation
MTV	Mars transfer vehicle
NASP	National Aero-Space Plane
$NBPH_2$	normal boiling point hydrogen
NTR	nuclear thermal rocket
O/H	oxygen/hydrogen
R	reusable
SLH_2	slush hydrogen
STV	space transfer vehicle
SSME	Space Shuttle main engine
STS	space transportation system (Space Shuttle)
STS-C	space transportation system-cargo (Shuttle-C)
TMIS	trans-Mars injection stage
TPH_2	triple point hydrogen

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TABLE I.—PROPERTIES OF CRYOGENIC FLUIDS

Fluid	Temperature		Pressure	Density			Heat sink		
	K	R		lb/ft^3	kg/m^3	Normalized	Btu/lb	kJ/kg	Normalized
$NBPH_2$	20.3	36.5	14.7	4.45	71.2	1.0	191.6	445.7	1.0
TPH_2	13.8	24.8	1.1	4.81	77.0	1.081	213.7	497.0	1.115
SLH_2	13.8	24.8	1.1	5.11	81.8	1.149	226.3	526.3	1.181
LOX	90.2	162.4	14.7	71.2	1140.0	-----	90.2	209.7	0.470

TABLE II.—DERIVED BOILOFF RATES

Application	MLI thickness, m	Boiloff, kg/month/m^2			
		$NBPH_2$	TPH_2	SLH_2	LOX
GEO	0.00	------	------	------	------
Lunar	.02	3.27	2.93	2.77	------
Mars	.08	0.818	0.733	0.692	------
Depot	.08	.818	.733	.692	1.25

TABLE III.—TANKAGE MASS ASSUMPTIONS

[Tank material, aluminum-lithium alloy; A_t = 5.251 x (vol.)$^{2/3}$; tank mass, k_t (fluid mass) + 45.2 x d x A_t; holding structure, 3 percent of tank set dry mass.]

Fluid	Tankage fraction, k_t
$NBPH_2$	0.09
TPH_2	.03
SLH_2	.03
LOX	.02

TABLE IV.—TRANSFER VEHICLE PROPULSION ASSUMPTIONS

[Total stage mass, core stage + dry tank sets + payload adaptor + aerobrake; core stage (table V); payload adaptor, 3 percent of payload; aerobrake mass, k_{ab} × (total mass at entry).]

Mixture ratio, O/H	6:1
Specific impulse, O/H, sec	481
Specific impulse, NTR, sec	900
Unusable propellant, percent	5
Aerobrake mass constant, k_{ab} (GEO and Lunar)	0.20
Aerobrake mass constant, k_{ab} (Mars)	0.15

TABLE V.—TRANSFER VEHICLE MISSION CORE STORAGE MASSES

[STV = Space Transfer Vehicle; LTV = Lunar Transfer Vehicle; MTV = Mars Transfer Vehicle; EX = expendable; R = reusable; AB = Aerobrake.]

Mission	Propulsion system	Mission stage		
		1	2	3
		Mass, kg		
GEO	CRYO STV (EX)	1 535	------	------
GEO	CRYO STV (R)	1 535	------	------
Lunar	CRYO LTV-AB	2 310	4500	------
Lunar	NTR LTV-AB	---------	5000	------
Lunar	NTR LTV	---------	7000	------
Mars (2015)	CRYO MTV	28 000	5000	------
Mars (2017)	CRYO MTV	28 000	5000	------
Mars (2018)	CRYO MTV	28 000	5000	------
Mars (2015)	NTR MTV	15 000	5000	------
Mars (2016)	NTR MTV	15 000	------	5000

TABLE VI.—TRANSFER VEHICLE MISSION VELOCITY INCREMENTS

Mission	Propulsion system	Mission stage		
		1	2	3
		Mission velocity increment, ΔV, m/sec		
GEO	CRYO STV (EX)	4212	------	------
GEO	CRYO STV (R)	4212	2195	------
Lunar	CRYO LTV-AB	3310	2576	------
Lunar	NTR LTV-AB	3310	2576	------
Lunar	NTR LTV	3310	5566	------
Mars (2015)	CRYO MTV	4280	3400	------
Mars (2017)	CRYO MTV	4160	1940	------
Mars (2018)	CRYO MTV	3610	1690	------
Mars (2015)	NTR MTV	4280	3400	------
Mars (2016)	NTR MTV	4414	3720	2926

TABLE VII.—MISSION PAYLOADS

[LEV and MEV treated as payload.]

Mission	Propulsion system	Payload mass, metric ton
GEO (EX and R)	CRYO O/H	2 to 20
Lunar Outpost	CRYO O/H and NTR	Outbound, 46 Inbound, 6.6
Mars Outpost	CRYO O/H and NTR	Outbound, 87.36 Inbound, 44.64

TABLE VIII.—MARS MISSION PROPULSION AND PAYLOAD RETURN OPTIONS

["X" indicates case considered in this study.]

Launch year	Trip time, days	Propulsion system			
		CRYO		NTR	
		Crew capsule	Crew module	Crew capsule	Crew module
2015	565	X	X	X	X
2016	400			X	
2017	654	X	X		
2018	942	X	X		

TABLE IX.—CHARACTERISTICS OF CRYOGENIC DEPOT TANKS

[Hydrogen tank capacity, 100 metric tons; insulation, 0.08-m-thick, double-aluminized mylar MLI.]

Tank fluid	Tank mass, metric ton	Cooling method	
		Boiloff (passive), percent/month	Zero-boiloff refrigeration active power, kWe
$NBPH_2$	11.46	0.53	11.9
TPH_2	5.34	.44	15.6
SLH_2	5.24	.40	15.0
LOX	2.39	.13	0.06

TABLE X.—TOTAL DRY MASS OF DEPOT

Depot type	Cooling method	Tank fluid		
		$NBPH_2$	TPH_2	SLH_2
		Tank mass, metric ton		
CRYO O/H (600 tons O_2/100 tons H_2)	Passive	60.8	54.7	54.6
	Active	61.0	55.0	54.9
NTR (300 tons H_2)	Passive	49.4	31.0	30.7
	Active	50.1	31.9	31.6

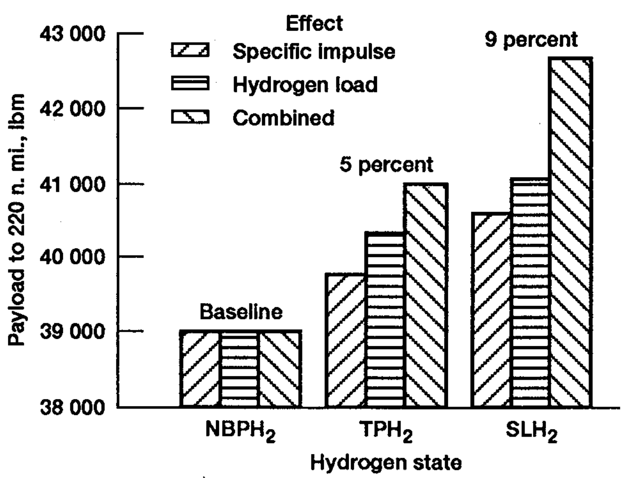


Figure 1.—Shuttle payload gain with triple point and slush hydrogen.

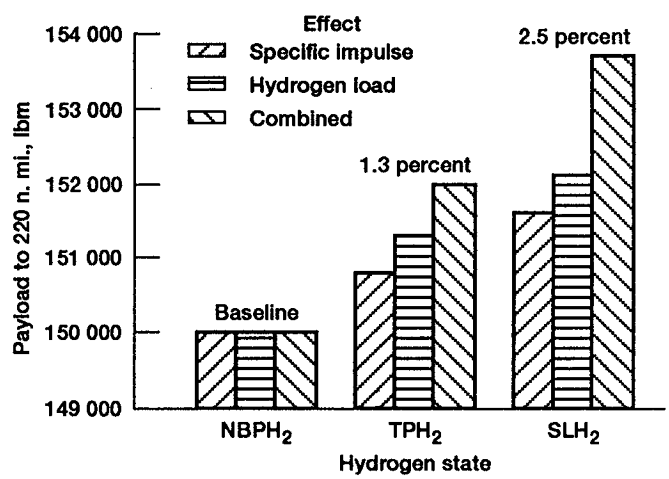
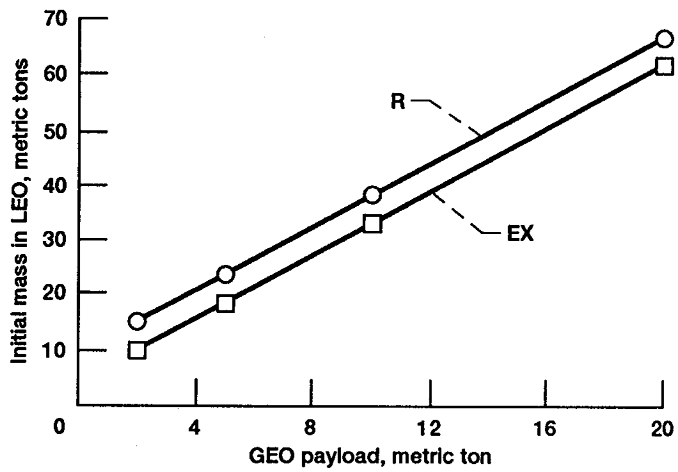
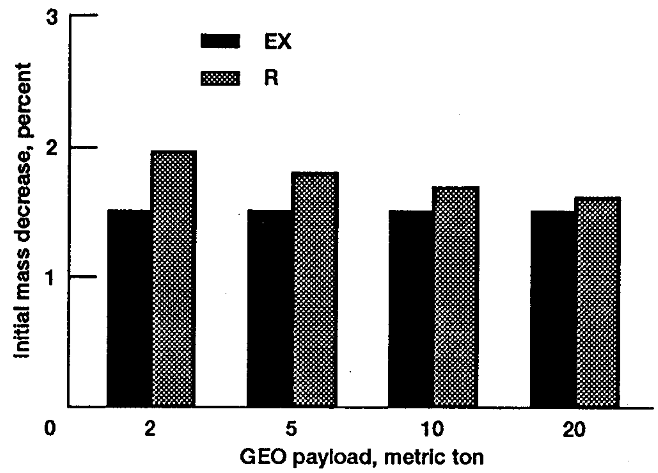


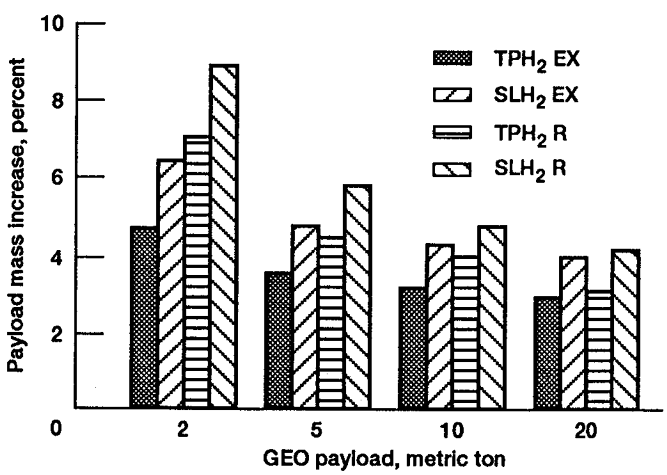
Figure 2.—Shuttle-C payload gain with triple point and slush hydrogen.



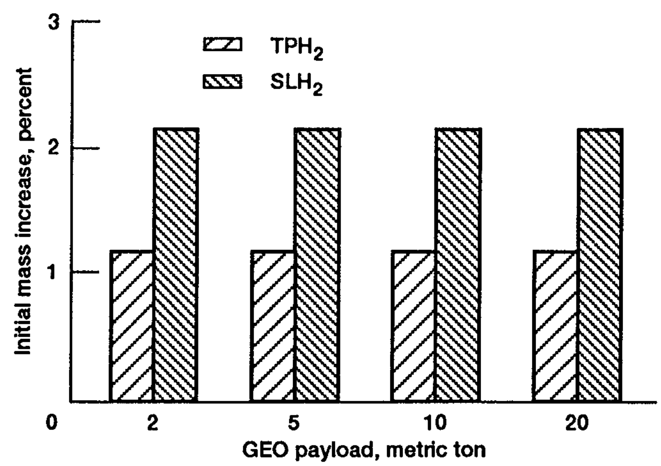
(a) Performance with normal boiling point hydrogen.



(b) Mass savings at a constant payload for triple point and slush hydrogen.

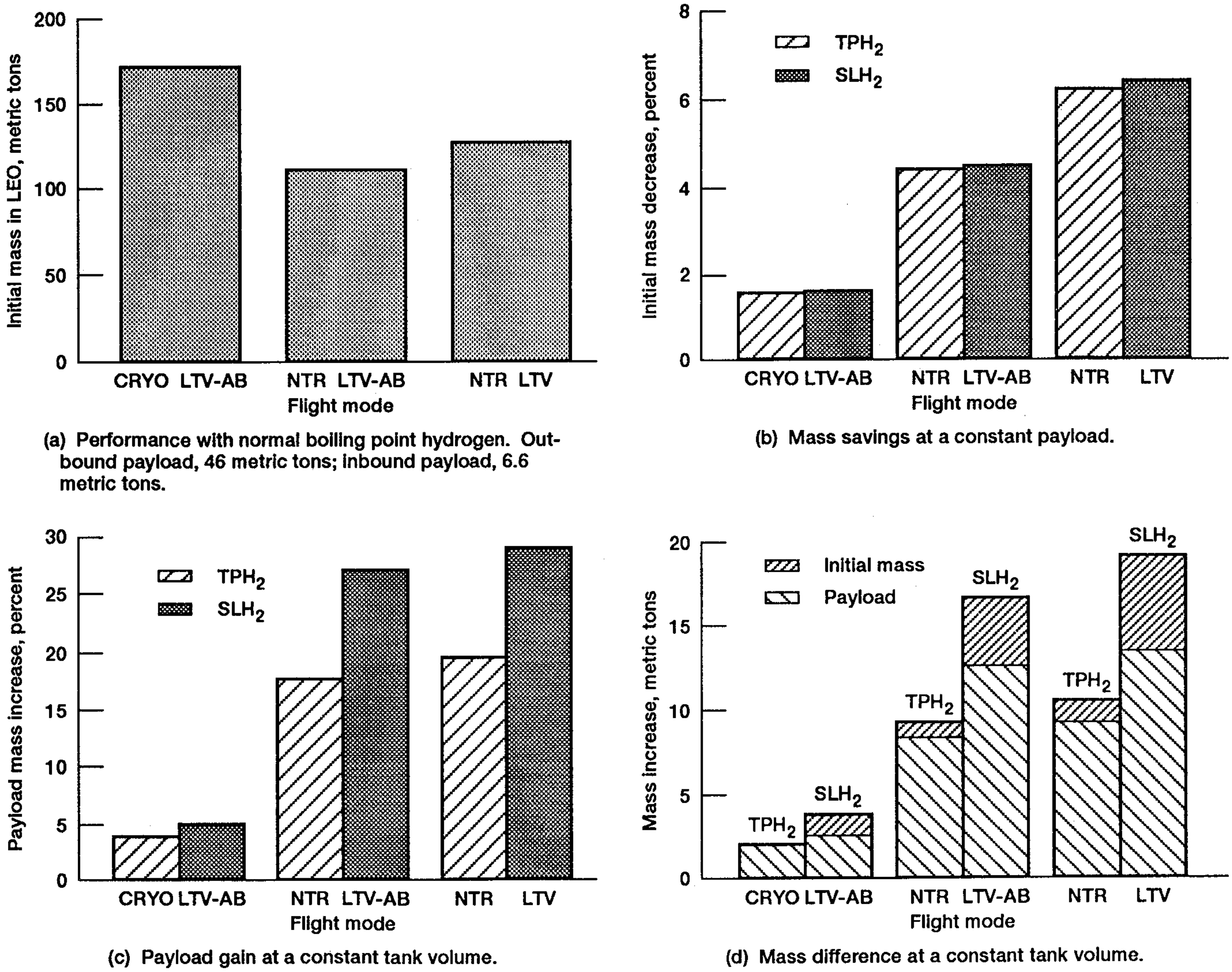


(c) Payload gain at a constant tank volume.



(d) Mass difference at a constant tank volume for EX and R modes.

Figure 3.—Hydrogen-state performance comparison for GEO mission. Space transfer vehicle (STV) flight modes considered were expendable (EX) and reusable (R).



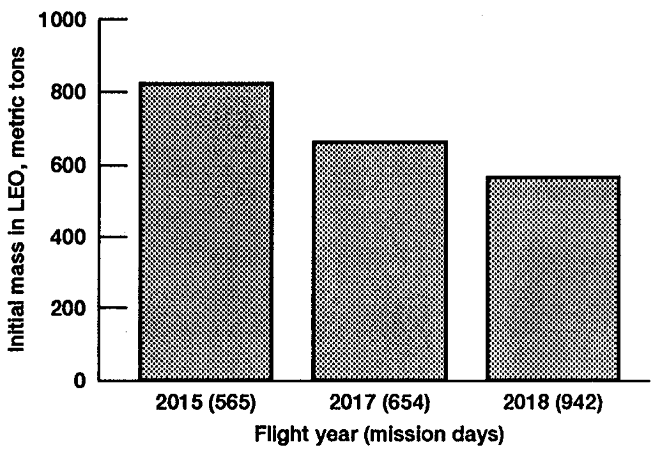
(a) Performance with normal boiling point hydrogen. Outbound payload, 46 metric tons; inbound payload, 6.6 metric tons.

(b) Mass savings at a constant payload.

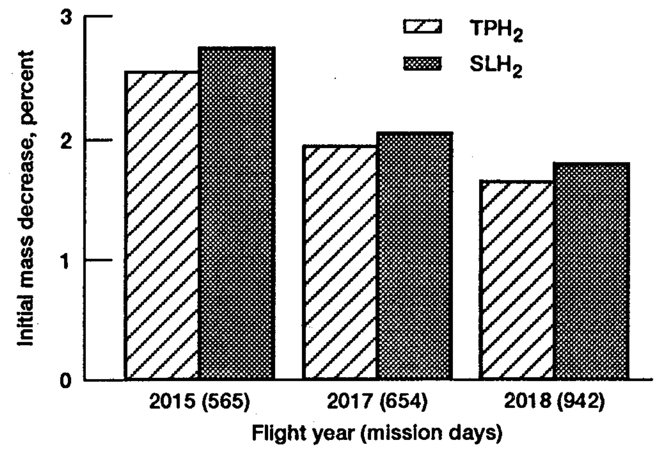
(c) Payload gain at a constant tank volume.

(d) Mass difference at a constant tank volume.

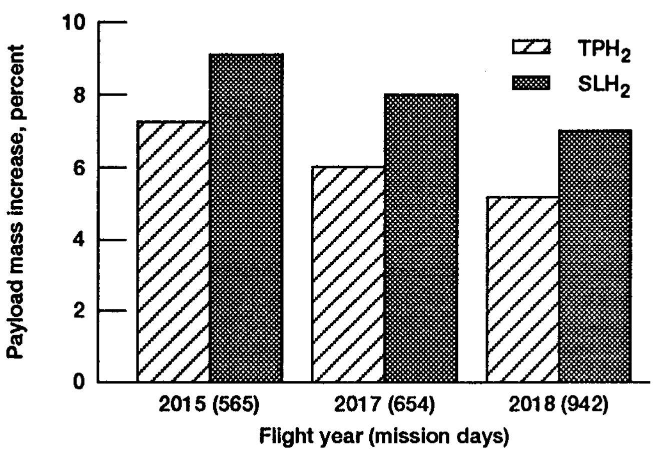
Figure 4.—Hydrogen-state performance comparison for Lunar Outpost Mission.



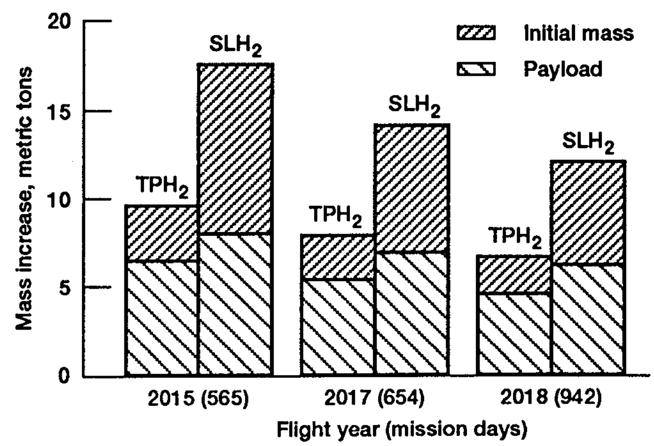
(a) Performance with normal boiling point hydrogen. Outbound payload, 87.36 metric tons; inbound payload, 44.64 metric tons.



(b) Mass savings at a constant payload.

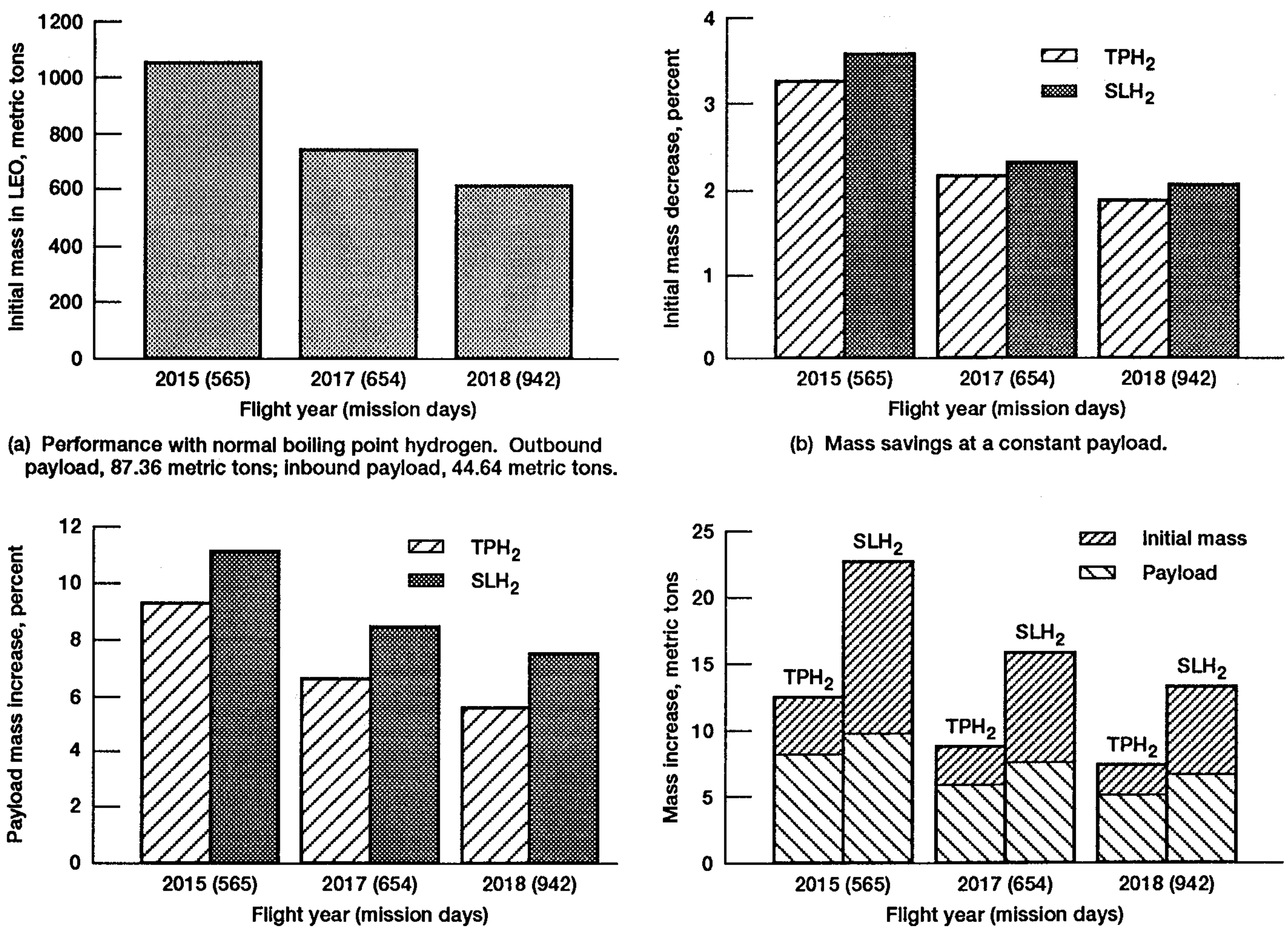


(c) Payload gain at a constant tank volume. Outbound payload 87.36 metric tons.



(d) Mass difference at a constant tank volume.

Figure 5.—Hydrogen-state performance comparison for Mars Outpost Mission (propulsion, cryogenic O/H; payload option, crew capsule return).



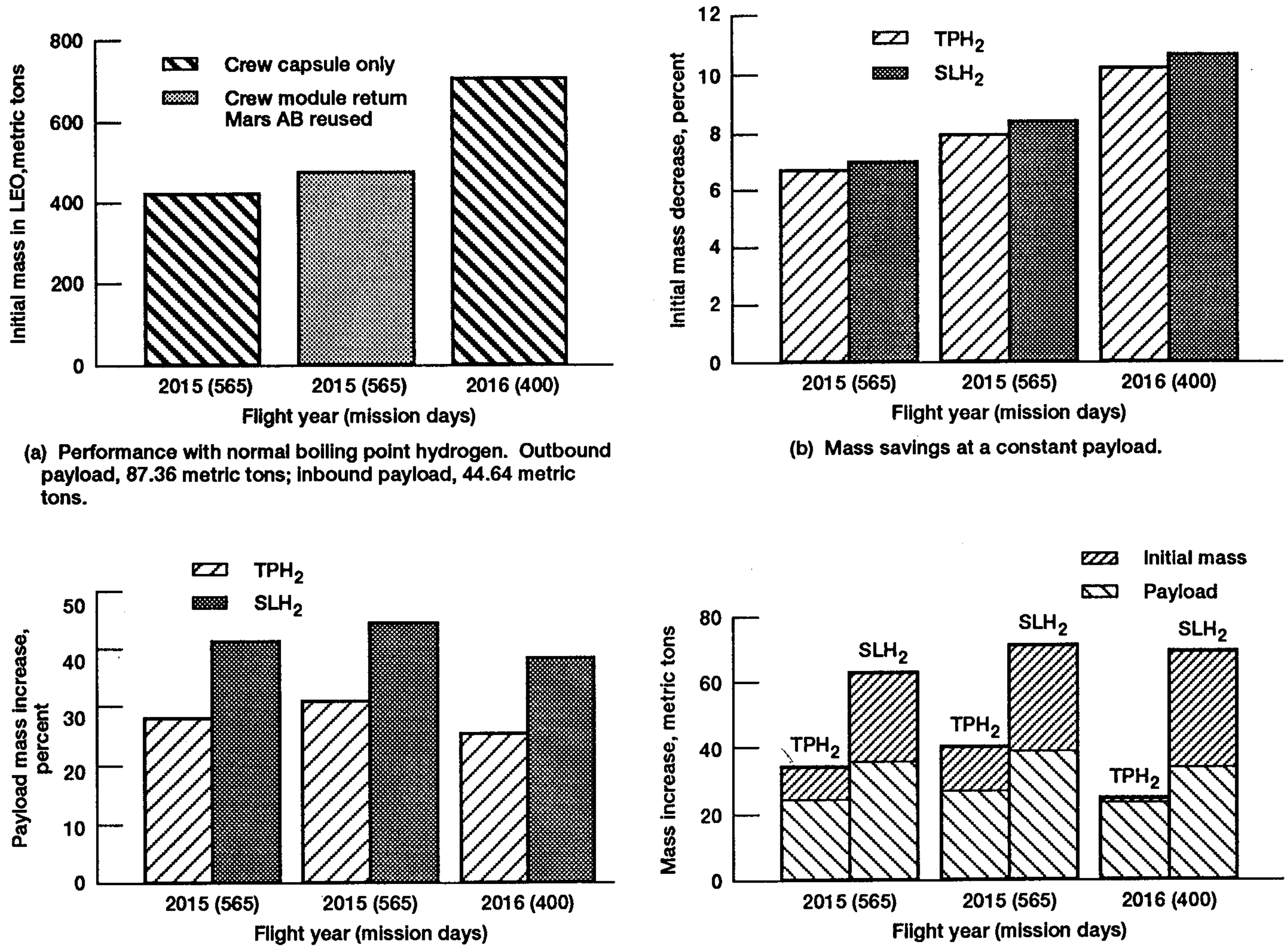
(a) Performance with normal boiling point hydrogen. Outbound payload, 87.36 metric tons; inbound payload, 44.64 metric tons.

(b) Mass savings at a constant payload.

(c) Payload gain at a constant tank volume.

(d) Mass difference at a constant tank volume.

Figure 6.—Hydrogen-state performance comparison for Mars Outpost Mission (propulsion, cryogenic O/H; payload option, crew module return).



(a) Performance with normal boiling point hydrogen. Outbound payload, 87.36 metric tons; inbound payload, 44.64 metric tons.

(b) Mass savings at a constant payload.

(c) Payload gain at a constant tank volume.

(d) Mass difference at a constant tank volume.

Figure 7.—Hydrogen-state performance comparison for Mars Outpost Mission (propulsion, nuclear thermal rocket).

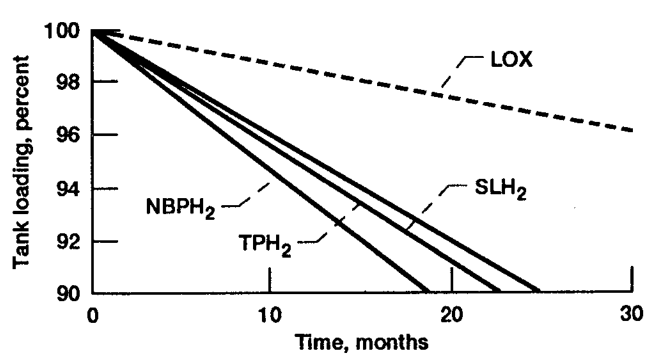


Figure. 8—Boiloff as a function of storage time (cryogenic O/H depot; passive cooling method).

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15. Supplementary Notes

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16. Abstract

A study was performed to quantify the benefits of using slush hydrogen instead of normal boiling point liquid hydrogen as a fuel for several space missions. Vehicles considered in the study included the Space Shuttle/Shuttle C, LEO-to-GEO transfer vehicles, Lunar and Mars transfer vehicles, and cryogenic depots in low Earth orbit. The advantages of using slush hydrogen were expressed in terms of initial mass differences at a constant payload, payload differences at a constant tank volume, and increases in fuel storage time for cryogenic depots. Both chemical oxygen/hydrogen and hydrogen nuclear thermal rocket propulsion were considered in the study. The results indicate that slush hydrogen offers the potential for significant decreases in initial mass and increases in payload for most missions studied. These advantages were found to increase as the mission difficulty or energy increased.

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