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## **RECOVERY OF LUNAR SURFACE ACCESS MODULE RESIDUAL AND RESERVE PROPELLANTS**

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### **ABSTRACT**

The Vision for Space Exploration calls for human exploration of the lunar surface in the 2020 timeframe. Sustained human exploration of the lunar surface will require supply, storage, and distribution of consumables for a variety of mission elements. These elements include propulsion systems for ascent and descent stages, life support for habitats and extra-vehicular activity, and reactants for power systems. NASA KSC has been tasked to develop technologies and strategies for consumables transfer for lunar exploration as part of the Exploration Technology Development Program. This paper will investigate details of operational concepts to scavenge residual propellants from the lunar descent propulsion system. Predictions on the mass of residuals and reserves are made. Estimates of heat transfer and boiloff rates are calculated and transient tank thermodynamic issues post-engine cutoff are modeled. Recovery and storage options including cryogenic liquid, vapor and water are discussed, and possible reuse of LSAM assets is presented.

## INTRODUCTION

The Vision for Space Exploration calls for sustained human presence on other celestial bodies, initially a lunar outpost near a polar location [1,2]. This presence will require delivery, storage, and distribution of consumables essential for human life support, power generation, thermal control, and propulsion for spacecraft and ground rovers. Due to the propulsive energy required to transport payload from Earth to the lunar surface, each kilogram of consumables delivered to the outpost will require between 112 and 203 kg of launch mass [3]. Efficient delivery and storage of these consumables and their systems is essential to minimizing launch mass. Based on many years of ground operations experience, NASA KSC has been tasked to lead the investigation of technologies required for a sustainable consumables transfer station for use on the lunar surface. One FY07 deliverable of this task is a detailed study of operations to recover residual propellants from the Lunar Surface Access Module descent stage, presented to the Lunar Architecture Team in their phase II study.

Figure 1 illustrates the many potential interfaces between a consumable station and mission elements. Consumable systems can be high pressure gas as well as a cryogenic liquid. Liquid storage is advantageous from a density point, but requires greater energy if liquefaction is needed and often leads to venting from heat leak. Loss of product is undesirable for two reasons; first because the mass of cryogen lost is magnified due production or transportation costs, and second, because too much venting in the lunar atmosphere has the potential to contaminate the lunar vacuum environment. This work is a follow on to previous lunar operations work done by KSC under the Supportability program [4]. The system will stress energy efficient storage and transfer, which includes thermodynamic efficiency of liquefaction systems as well as storage and transfer efficiency to minimize boil-off and chill down losses. The operations concepts will maximize use of local resources, including the local thermal environment, which in many ways is an ideal cryogenic storage site. The concept of local resources will be ex-

panded to include assets delivered to the surface as part of the descent module which may have no other essential mission functions. For performance and maintenance reasons, ground systems are preferred over flight systems when ground systems can perform the same function. This work is intended as a first order approximation of system power requirements, mass and volume constraints, and thermal performance, future work will refine these estimates through experimentation or detailed analysis.

First, some specific assumptions on mission architecture will need to be made before an analysis of the propellant recovery operations can occur. The ESAS lander configuration is assumed to be spacecraft used. It features 4 liquid hydrogen tanks and two liquid oxygen tanks, feeding four RL-10 engines. The tanks are cylindrical with elliptical end caps, and are insulated with 60 layers of MLI, with a density of 20 layers per cm. The initial missions are for 7 day surface stays, increasing to 30 days within three years. The LSAM has a 4.2 kW fuel cell to power the spacecraft, and there will be long term solar power available for hookup within hours after landing if needed. There is access to connect to the propellant systems via the same umbilical plates used for servicing on Earth, and propellant transfer between tanks is allowable. There will be a large demand for water during periods of prolonged EVA for thermal control of the suits. Lunar surface temperatures range from 120 to 160 K in the polar region. Radiation to deep space will have an effective sink temperature of 4 K, whereas radiators that see the lunar surface will have varying sink temperatures depending on orientation, location and the angle of the sun [5]. The temperature in the permanently shadowed craters near the poles are assumed to be as low as 40 K, but it is not anticipated there will be much activity in these craters. The lunar environment is an ultrahigh vacuum of  $10^{-12}$  torr. Dust contamination is a concern due to abrasions, seal wear, degradation of thermal control surfaces, mechanical breakdowns, and human health concern, all experienced in the much shorter duration Apollo missions.[6]

## LSAM RESIDUAL AND RESERVE MASS

Once the descent stage of the LSAM has completed its primary mission, there will be an unused portion of the propellants that serve as a safety margin. This unused propellant is classified in two categories, referred to as residuals and reserves. The primary type of available propellant is reserve propellant, usable by the engines if needed. There may be some uncertainty regarding the total amount of  $\Delta v$  required during the mission, especially early missions where precise landing areas are not known and some amount of maneuvering may be required to find an acceptable landing site. It is assumed over time that this level of reserve propellants can be decreased if desired, as previous flight data is available and better lunar navigation aides are deployed and regular landing sites are cleared. Current LAT studies show a planned delta-V of 3078 m/s compared to a required delta-V of 2978 m/s, or a propulsive reserve of 3.2%. The total propellant mass is derived from lunar architecture team estimates to be 24,400 kg. Assuming an engine mixture ratio for the RL-10 of 6/1, the total required quantity of propellants can be found. The propellant is assumed to be in a configuration similar to the ESAS LSAM, with 4 LH2 and 2 LOX cylindrical tanks.

Unusable propellant necessary for proper engine operation is classified as residual propellant. Liquid residuals are necessary to ensure the state of the cryogenic propellant is sufficiently subcooled when it reaches the engine turbopump. The net positive suction head of the engine will decrease as the tank empties and the fluid warms up. Values below the required NPSH will cause cavitation in the pump, which must be avoided. Space Shuttle engine cut-off sensors are located at 0.6% tank volume for the hydrogen system and inside the feedline for the oxygen system. Saturn data records LOX cutoff at 1.1%, 0.7%, and 0.7 % for the S-I, S-II, and S-IV stages, and a 3.2% residual for the LH2 S-IV stage. Once the exact LSAM feedline geometry and thermodynamics are known, better estimates on required liquid residuals can be made. Historically, liquid hydrogen requires a greater percentage of residual than oxygen, as

engines don't want to run the risk of shutting down LOX rich and the low density LH2 doesn't provide much elevation head. Landing acceleration forces, minimal opposed to launch forces, increase that problem. For purposes of this work, liquid residuals are assumed to be 1.5% of the initial load for hydrogen and 1% for liquid oxygen.

Gaseous residuals are vapors in the top of the tank needed to pressurize the liquid. Current planning assumes the LOX tank is pressurized with cold gaseous helium to approximately 50 psia. The helium is loaded in thermal equilibrium with the LOX at KSC, and little oxygen is vaporized in the pressurization process. The LOX ullage at shutdown will be greater than 90% helium which may interfere with the recovery operations. The hydrogen is autogeneously pressurized by the RL-10 engines. Calculations on hydrogen vapor residuals requires knowledge of the state of the ullage vapor. Flight data of the RL-10 is not available for this paper, but it is estimated the pressurization flow enters at 200K. The vapor residual in the hydrogen tanks may vary between 22 kg and 218 kg, depending on the heat transfer rate between the liquid and the ullage pressurant. The relative amount of vapor is large, as shown in Figure 2. The low estimate assumes no heat transfer between the ullage and the liquid, and the ullage temperature is approximately 200K. The high mass estimate assumes thermal equilibrium throughout the tank and the ullage temperature is approximately 25K. Centaur flight data shows the actual mass of the hydrogen pressurant required is 2.18 times less than the equilibrium model predicts, or 101 kg [xxx]. If the ullage volume was isothermal, this would correspond to a bulk ullage temperature of 45K. In practice, there will be a temperature gradient in the ullage between 25 K at the liquid interface and 200 K at the pressurization inlet. For greater resolution in the details of this temperature profile, a transient, non-equilibrium CFD model will need to be created that takes into account heat and mass transfer at the liquid to vapor interface, pressurization momentum terms, heat transfer at the solid walls, and temperature dependent heat capacity of the

liqui, vapor, and tank walls. This is beyond the scope of this work, and the ullage will be treated as an isothermal single node in the rest of this work.

Some propellant is considered unable to be recovered. There may be too much residual heat to recover liquid in the feedlines immediately after landing. Precise placement of engine pre-valves may minimize this loss. For this study, a 12m x 10 cm supply line for LOX and 15m x 15 cm line for LH2 are considered unrecoverable. In addition, low pressure vapor in the tanks may not be practical to recover. A final state of 155 kPa at 200 K is assumed for the tank. Table 1 lists the estimated propellant masses, broken down by residuals, reserves, and unrecoverables, for LOX and LH2..

## **TANK HEAT TRANSFER AND THERMODYNAMIC ISSUES**

After lunar landing, the liquid oxygen tanks will be a two species mix of helium and oxygen at an equilibrium temperature around 104 K and a pressure around 344 kPa. In practice, the tanks will not be isothermal, especially with a liquid level around 5%. The warm boundary temperature around the tanks will be highly variable. The relative amounts of radiation, from the spacecraft at 240K, the lunar surface between 120-160K, deep space at 4 K, and possible direct solar incidence, must be analyzed by radiation modeling. An average temperature of 300 K is assumed for initial estimates, but moving the liquid to the shaded side of the LSAM reduces the temperature to 200K. Shielding the tanks will offer the potential to further reduce the heat leak, but this isn't necessary for the LOX system as will be shown. The estimated LOX heat leak rate ( $T_2=300\text{K}$ ) is 21.6 W, which corresponds to a steady state boil off rate of 388 g/hr. If all the liquid was transferred to the tank on the shaded side, the lower temperature and tank area would reduce the heat leak to 3.7 W and boil off rate to 67 g/hr. However, the required fuel cell supply rate is 1500 g/hr, so as long as the fuel cell is operating, extra heat will need to be input into the LOX tank. A minimum of 100W electrical resistance heaters are re-

quired for each tank. Based on the predicted amounts of residuals and reserves, this allows for fuel cell operation for between 88 and 530 hours.

The state of the LH2 tank is less certain. At engine cutoff, the tank will have superheated vapor in the ullage, and if the system is closed the tank will seek thermal equilibrium. Depending on the ratio of the vapor enthalpy to the liquid enthalpy, the equilibrium state of the tank will change. A large mass of subcooled liquid can use its sensible heat to condense a smaller mass of ullage volume, causing an ullage pressure collapse. Likewise, A large mass of warm ullage gas can cause evaporation of liquid, resulting in pressurization and venting. Since the exact conditions in the hydrogen tank after landing are not known, a model has been created to predict the equilibrium state of the tank for a range of initial conditions. This model has two nodes, separate liquid and vapor regions with uniform properties. The system is a constant volume, closed system, with an initial pressure of 344 psia. Figure 3 shows the results of this model.

Once the system reaches thermal equilibrium, there will be a predictable heat leak into the system that will create a pressure rise. This pressurization may or may not be accompanied by evaporation since there may be only saturated vapors in the tank. Using the same warm boundary conditions as the oxygen tanks, the predicted heat leak into the hydrogen system is 57.2 W. When liquid is in the tank, this corresponds to an evaporation rate of 503 g/hr. This is significantly higher than the required 200 g/hr needed for the fuel cells. If all the hydrogen can be transferred into a single shaded tank, the estimate heat leak into the system reduces to 5.3 W and the boiloff reduces to 47 g/hr. However, this is not possible if a large portion of the hydrogen is cold vapor. There are several possible solutions to the problem of long term storage of hydrogen. Shielding all the tanks so they only view deep space can reduce the effective source temperature to below 120K. Transferring all the liquid into a single tank, then using the remaining tanks as a progressively warm ullage space in series may reduce the overall heat



leak to acceptable levels without having to vent the drained tanks. If venting is required, between 5 kg and 60 kg of valuable hydrogen may be lost.

Another issue with the hydrogen storage and delivery is the fact the required delivery rate of 200 g/hr will require variable heat input to maintain at a constant rate. This is an issue with the oxygen as well. The heat of vaporization will decrease as the liquid warms up, changing the boil off rate. When the tank transitions from liquid to vapor, the pressurization rate is decided by vapor thermal expansion as opposed to liquid evaporation. Then there will be a temperature dependence on the volumetric expansion coefficient and specific heat. A model was created to analyze the required heating rate for liquid hydrogen as the tank drains from 5% to the warm vapor state. It describes a transient, open system with a constant mass flow rate of 200 g/hr, and a constant pressure of 344 kPa. The required enthalpy of the final state is found by calculating the enthalpy initially and subtracting the fuel cell supply stream enthalpy, knowing the required mass flow rate and the timestep. Figure 4 shows a required heating rate of 21 W, slightly increasing to 23.7 W at the point the residual liquid is vaporized. The required heating drops to 12.9 W for the remaining vapor, increasing to 24.9 W when the hydrogen tank is depleted at 530 hours. This assumes an initial mass of 252 kg, with a quality of 88%. Tanks with different initial mass will have different slopes, but the overall behavior of the system is similar. More work needs to be done to understand the thermal and fluid behavior of this system as the tank drains, to make sure an on demand fuel cell supply rate can be achieved.

## **SURFACE OPERATIONS**

Once access is obtained, operations to transfer the residual propellants can be initiated. At this point, there are several options for how to recover the consumables. The propellants can be drained to a lunar based mobile tanker in their liquid state, a compressor can be used to transfer the propellant as a gas to a high pressure storage tank, or the propellants can be combined in a catalyst bed or fuel cell and then stored as water. An alternative to transfer opera-

tions is to leave the propellants in their LSAM storage tanks and transfer these tanks to the required usage point. Operations for each of these options, and the hardware required to perform them, are discussed next. Engine shutdown purges will need to be initiated and the feedlines downstream of the MPS supply valves will probably be purged as well. Helium purge valves for the engines and MPS lines are shown in the schematic. Assuming LOX and LH2 line diameters and lengths shown in Figure XX and a density corresponding to saturated liquid at 444 kPa, there will be 16.7 kg of LOX and 3.3 kg of LH2 that will need to be vented and purged from the engines and feedlines immediately after landing. In addition, the remaining liquid in the tank will have more time to absorb the heat from the pressurization gasses, leading to vaporization inside the tank. The easy solution is to vent the storage tanks soon after landing, before the liquid and ullage space have a chance to achieve thermal equilibrium. This will lead to additional losses, from both initial ullage venting as well as vaporization of some liquid during the process of cooling down the remaining liquid. For the liquid oxygen case, the losses are minimal, since the majority of the ullage volume is taken up by helium gas. Oxygen losses are predicted to be approximately 11.4 kg. However, there remain significant quantities of hydrogen vapor in the ullage space, due to the larger storage volume and the relatively low ratio of hydrogen density between the liquid and vapor state. Venting the hydrogen from the ullage space will create losses between 160-165 kg. In fact, depending on the % of residuals remaining, there may be more hydrogen to be recovered from the ullage vapor than there is in liquid form. A mobile LOX and/or LH2 storage tanker is required to support liquid recovery options. Each tank can sit on a mobility platform delivered by an unpressurized rover. The minimum tank sizes are approximately 1.1 m<sup>3</sup> for liquid oxygen and 3.1 m<sup>3</sup> for liquid hydrogen, if recovery of all the residuals is planned. If dedicated tanks are flown for this purpose, this corresponds to a one time up mass of xxx kg. If LSAM descent stage tanks from a previous mission can be scavenged and recycled for use on a mobility platform, there would be minimal addi-

tional mass required for dedicated storage tanks. This option would also allow for greater operational flexibility, since planned tank volumes would be larger than the minimum required. Larger tanks allow for greater storage volume, so tanks will probably always be partially full (hence cold) and have greater ullage volume to fill with evaporated product during transfer operations. Based on data from no vent fill operations conducted at NASA GRC, a partially filled, cold tank should be able to increase its liquid level from 20% to 30% with no venting. This may not be the case for a dedicated empty storage tank designed to be filled to 90%. Other hardware will be required for transfer operations. A cryogenic transfer tube with a flex section to connect the tanks estimated at 6 meters in length and weighing 2.9 kg is needed. The line should be uninsulated, since short duration transfers are dominated more by chilldown losses than heat leak from the ambient environment. Line sizes of 1" diameter are sufficient for this operation but interface connections will be dictated by LSAM servicing QD's needed for pre-launch operations. The umbilical interface should be designed to allow for easy access on the lunar surface, preferably below the storage tanks to allow for gravity drain. Otherwise, pressurized transfer is possible and no pump transfer operations are envisioned. Figure x.x shows an artists conception of what this liquid recovery may look like. Notice two separate umbilical locations are required, one each for hydrogen and oxygen.

For vapor recovery, a mobile storage tank is again required. The total storage volume will vary depending on the desired pressure, and a trade study to determine the optimum pressure can be conducted. For most users of gaseous hydrogen and oxygen, pressure requirements should be pretty low. Except for the 6000 psia oxygen in the current emergency O2 supply for the EVA PLSS, the highest required pressure appears to be 750 psia for the EVA primary O2 supply. There are no defined requirements for high pressure gaseous hydrogen at this time. However, gaseous storage required much greater volume than liquid storage, and pressure vessels can get heavy for large volumes. Figure XX shows the relationship between required stor-

age volume and vessel mass as a function of storage pressure for gaseous oxygen for a range of storage temperatures. If dedicated storage vessels were required to be transported from Earth, the optimum pressure to store the oxygen would be xxx psia. Similarly, hydrogen storage mass has a minimum at a pressure of xxx psia. These mass impacts can be mitigated by using excess helium storage bottles from previous LSAM descent stages. From the ESAS report, the assumed pressurization system for the descent stage oxygen tanks is high-pressure gaseous helium stored in two Inconel 718 lined, graphite epoxy composite overwrapped 6,000 psia tanks. The diameter of the tanks is 0.61 m, giving a total storage volume of 0.24 m<sup>3</sup>. After 3 missions, there should be enough storage volume from the excess helium tanks to store 1 mission worth of oxygen residuals, but it would take 20 missions to obtain enough tanks to store hydrogen as a high pressure gas. It appears the primary storage of residual hydrogen as a high pressure gas is not feasible, given the high mass and volume required for the pressure vessels. Storage of gaseous oxygen may be possible, especially if the primary storage is a liquid but secondary tanks are filled with gas and distributed to smaller scale users that may have infrequent usage rates.

An additional consideration is the hardware and power required for compression. If oxygen vapor is to be compressed for storage, there must also be a purification step to eliminate any helium pressurization gas in the ullage volume. A permeable membrane that allows for the smaller helium molecule to pass while trapping the oxygen is a solution to this issue. Oxygen compression is particularly difficult due to combustion hazards, and mechanical compression would probably require a minimum of four stages to compress the gaseous oxygen from 30 psi to 6000 psi. Each stage would require an intercooler to remove the heat of compression and reduce the compression power required. If the process compresses the oxygen as it is being boiled off, the minimum flowrate is 32.7 g/hr and the power required is estimated at 1.9 W with a total interstage heat rejection of 2.5 W. These flowrates correspond to a timeline of

greater than 3 years, which is impractical. On the other hand if the recovery rate is increased so the oxygen tank is emptied in 7 days, the estimated power for that compression rises to 400 W and the heat rejection becomes 515 W. For oxygen vapor recovery and compression, the connection to the vehicle should probably be done at the outlet of the thermodynamic vent system for the low flowrates, but at flowrates above what the TVS can handle then recovery will require connection to the much larger tank vent QD that is used during cryogenic loading on the launch pad.

Other compression options include a thermal process which takes a set quantity of liquid from the bottom of the tank, and puts it into a sealed volume where ambient heat will increase the temperature and create a pressure rise, or a cold pump/compressor which uses less energy to compress the liquid. Both of these options require less power and system mass, but are require some technology development. For hydrogen storage and/or compression, metal hydride beds may be used to absorb the gas. While the energy storage density of these systems is high, resulting in small volumes, the storage mass efficiency is poor. Assuming at 10% mass storage density, over 3000 kg of hydrides would be required to store the residual hydrogen for each mission. While these beds are recycleable and the mass penalty is a one time charge, these systems are too massive and will not be considered further in this work. There are also power constraints with these systems as the desorbing process is endothermic, and many hydride systems require high thermal energy input at high temperatures. Another advanced technology worth considering for hydrogen compression is an electrochemical compressor. These systems are analogous to a single component high pressure electrolysis system, where power input into an anode dissociates electrons from the hydrogen. The  $H^+$  particles flow across a permeable membrane, against a pressure gradient, where it combines with the free electrons in the cathode to reform elemental hydrogen. The compression power required can be estimated by the Nernst equation and theoretically equals isothermal compression. There are technology hurdles to

overcome from any of these advanced compression techniques, if gas recovery and compression was the chosen method of propellant scavenging. . However, due to storage volume concerns and the need for additional compression equipment and processes, gaseous storage is not recommended as the primary recovery method for LSAM residuals.

The final recovery option is to combine the residual hydrogen and oxygen in a catalyst bed or fuel cell, then recover the product water. Assuming the fuel cells used to produce power during the mission phase are supplied from the propulsion system cryogenic tanks, there will be no additional hardware required for that interface. The water product from the fuel cells will probably go to a temporary storage tank, and during the mission will be supplied to an evaporator system for thermal control. This water may or may not be recoverable on the lunar surface, depending on the needs of the evaporator and thermal control system.

## **LSAM SYSTEM ASSETS**

Keeping with the idea the LSAM has assets that should be recovered, there will be several fluid system components that can be reused during lunar surface operations. Each mission will bring 4 liquid hydrogen tanks, roughly 13 m<sup>3</sup> each, and 2 LOX tanks of 10 m<sup>3</sup> each. There are low pressure tanks that could store cold vapors for the habitat life support system, or as a temporary storage for regenerative fuel cells or ISRU product. Taking off the insulation and placing radiation shields may allow the tank temperatures to get close to 100K. At these temperatures over 100 kg of oxygen and 10 kg of hydrogen can be stored in each tank. There are also smaller oxygen tanks that will be used to supply the LOX/LCH<sub>4</sub> reaction control system. These tanks can be used for multiple purposes as well, including smaller EVA oxygen caches, or as propellant tanks for hoppers of fuel cell powered rovers. In addition, high pressure composite overwrapped helium tanks are available for high pressure oxygen storage. . Special considerations should be given during LSAM final design stages to ensure the maximum reusability of consumables storage assets.

## **CONCLUSIONS AND RECOMMENDATIONS**

In conclusion, residual and reserve propellants can be recovered from the LSAM descent stage and stored for other uses. There is more hydrogen by % than stoichiometric, due to conservative ECO and pressurization gas. Testing for low volume tank thermal performance, transient effects pressurization, should be done. Consider using autogeneous LOX pressurization. The optimum recovery method will depend on the required use and other architecture concerns. In addition, if the fuel cell water product is needed for thermal control of the LSAM systems, recovery of that water will not be possible.

## **ACKNOWLEDGEMENTS**

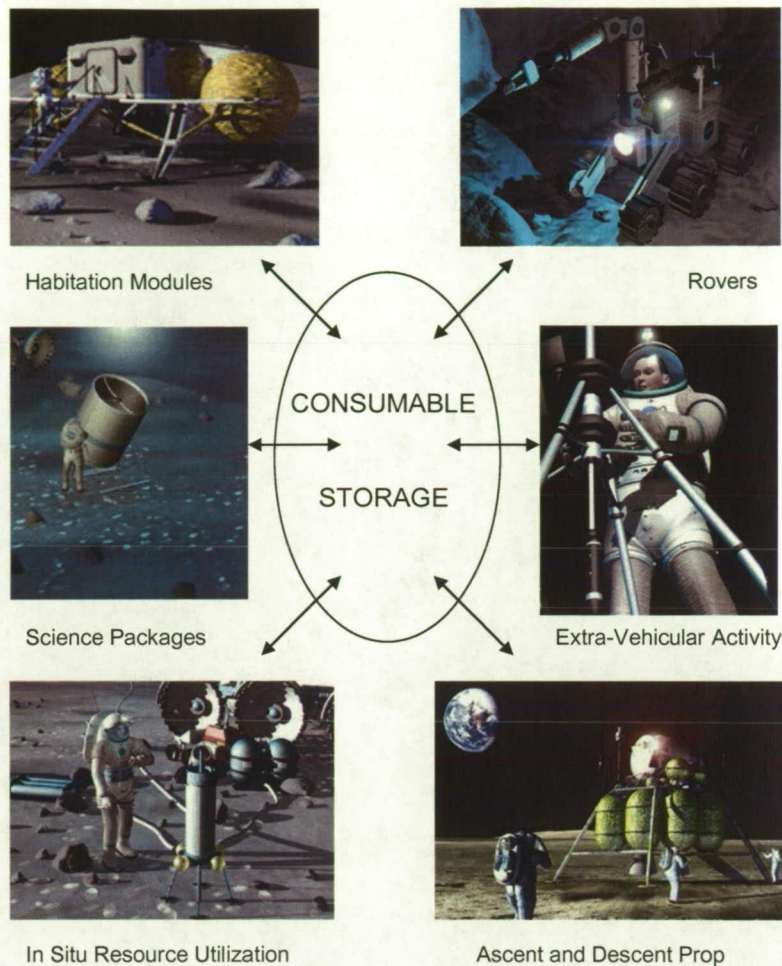
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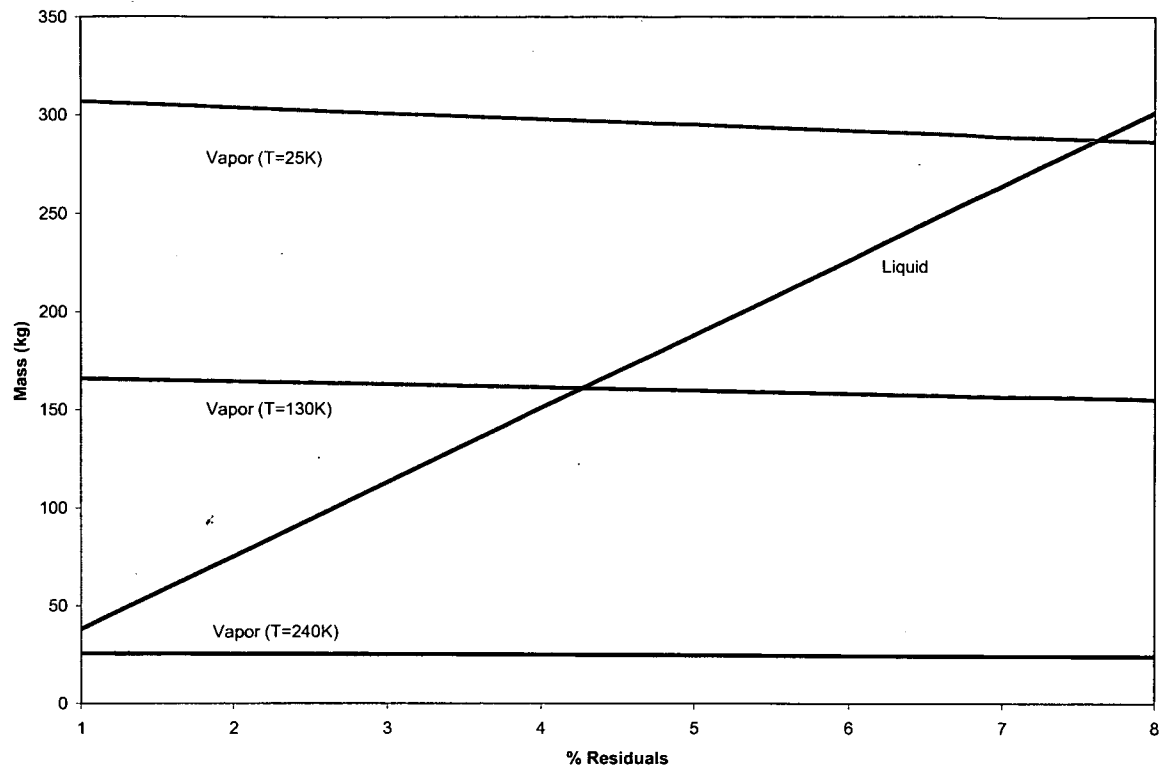
**FIGURE 1.** Various applications of a lunar consumables storage and distribution system are illustrated.



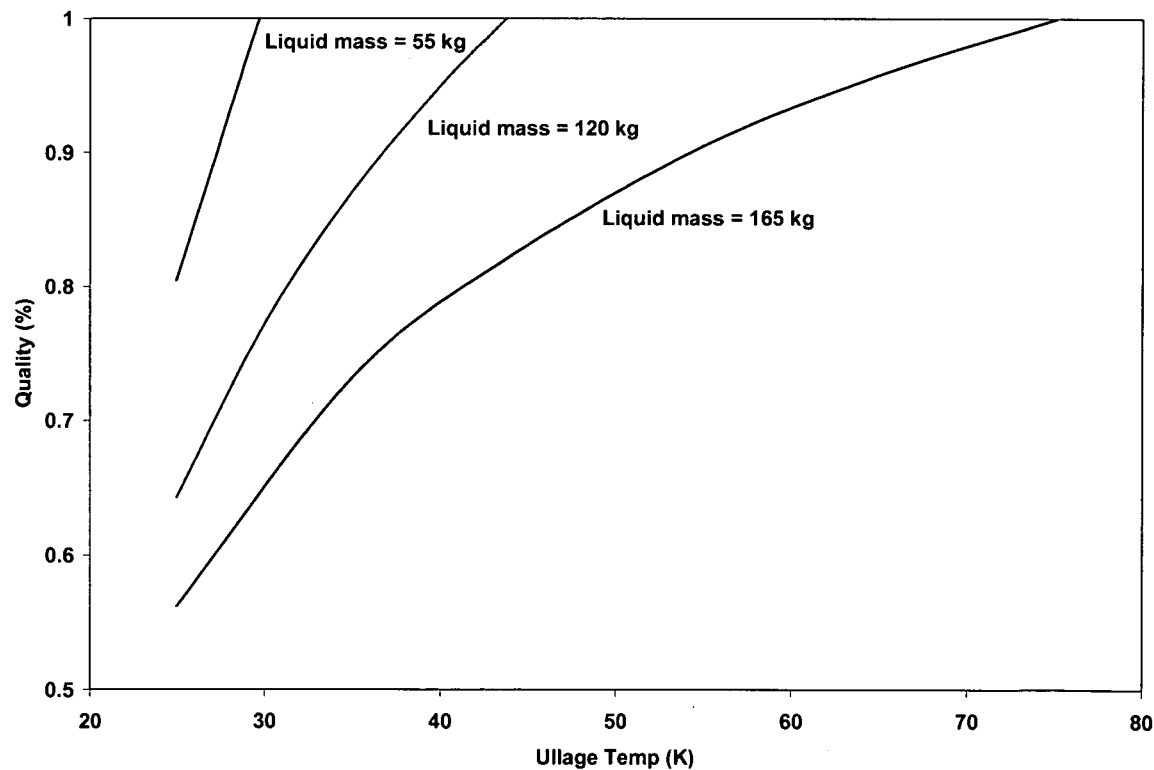
**TABLE 1.** Predicted propellant residuals and reserves based on initial usable propellant.

		LOX		LH2	
<b>Usable Propellant</b>		20891	kg	3482	
	Nominal	20222	kg	3371	
	Reserve	669	kg	111	
<b>Unusable Propellant</b>					
	Residual Vapor	2.23	kg	100.2	
	Residual Liquid	211.0	kg	54.5	
<b>Unrecoverable Mass</b>					
	Feedline Liquid	33.69	kg	5.67	
	Low Pressure Vapor	47	kg	8.27	
<b>Total Recoverable Mass</b>					
	Minimum	132.1	kg	140.8	
	Maximum	800.6	kg	252.2	

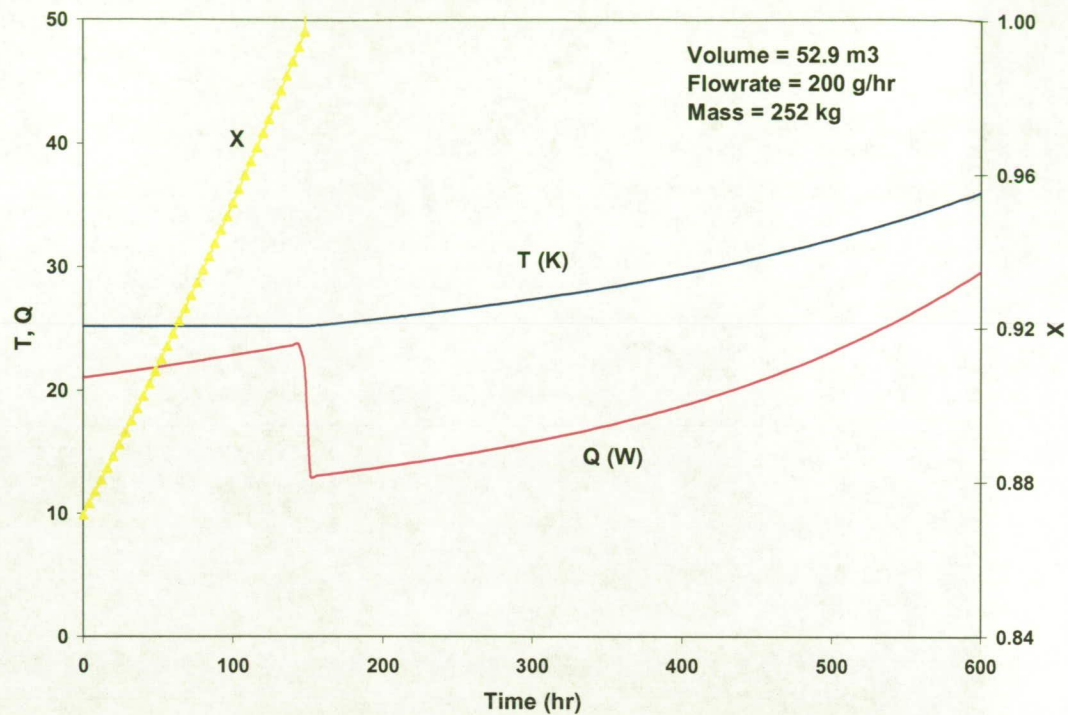
**FIGURE 2.** Mass of liquid and vapor residuals as a function of residual tank volume for a variety of bulk ullage temperatures



**FIGURE 3.** The equilibrium quality as a function of bulk ullage temperature for hydrogen predicts when tank venting will occur during thermal stabilization.



**FIGURE 4.** Transient heating requirements to maintain a constant 200 g/hr vent in a cryogenic hydrogen tank.



**FIGURE 5.** Artist conception of transfer operation of consumable liquid or vapor between LSAM and ground storage tanks.

