



# Hydrogen Fuel System Design Trades for High-Altitude Long-Endurance Remotely-Operated Aircraft

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## Abstract

Preliminary design trades are presented for liquid hydrogen fuel systems for remotely-operated, high-altitude aircraft that accommodate three different propulsion options: internal combustion engines, and electric motors powered by either polymer electrolyte membrane fuel cells or solid oxide fuel cells. Mission goal is sustained cruise at 60,000 ft altitude, with duration aloft a key parameter. The subject aircraft specifies an engine power of 143 to 148 hp, gross liftoff weight of 9270 to 9450 lb, payload of 440 lb, and a hydrogen fuel capacity of 2650 to 2755 lb stored in two spherical tanks (8.5 ft inside diameter), each with a dry mass goal of 316 lb. Hydrogen schematics for all three propulsion options are provided. Each employs vacuum-jacketed tanks with multilayer insulation, augmented with a helium pressurant system, and using electric motor driven hydrogen pumps. The most significant schematic differences involve the heat exchangers and hydrogen reclamation equipment. Heat balances indicate that mission durations of 10 to 16 days appear achievable. The dry mass for the hydrogen system is estimated to be 1900 lb, including 645 lb for each tank. This tank mass is roughly twice that of the advanced tanks assumed in the initial conceptual vehicle. Control strategies are not addressed, nor are procedures for filling and draining the tanks.

## Introduction

The first flight of an aircraft fueled solely by liquid hydrogen occurred in 1988 (Ref. 1) and AeroVironment flight-demonstrated a remotely-operated drone in 2005 powered by a liquid-hydrogen/fuel-cell/electric-motor system (Ref. 2). While the ability to operate aircraft with liquid hydrogen is certain, the tools to provide *optimum* system designs for such hydrogen systems remain incomplete.

To identify the technology issues for optimizing aircraft hydrogen fuel systems and to further advance the design tools in the process, conceptual designs and preliminary design trades were conducted for a representative mission and vehicle. The chosen application, the High Altitude Long Endurance (HALE) Remotely Operated Aircraft (ROA) (Refs. 3 and 4) presents the challenge of long-term storage of liquid

hydrogen (14 days aloft) and presents an example of the type of mission that could benefit from the higher energy-per-mass density of hydrogen (2.80 times higher than JP aviation fuel (Ref. 1). Also, the wider design margins of remotely-operated vehicles, in contrast to human-rated vehicles, provides a more accommodating starting point for these exploratory assessments. The anticipated applications for HALE ROAs include communication relay platforms and environmental monitoring, such as tracking the evolution of hurricanes (Refs. 4 and 5).

To broaden applicability, three different propulsion options are considered: (1) internal combustion engines, (2) polymer electrolyte membrane (PEM) fuel cells powering electric motors, and (3) solid oxide fuel cells (SOFC) powering electric motors. Thermal balance estimates are provided for all three propulsion options over all five mission stages to evaluate the viability of matching the fuel system performance to the operating environment. Hydrogen systems are specified for all three options, including component mass estimates. A parallel task examined the technologies for lightweight cryogenic storage tanks (Ref. 6).

## Representative Mission and Vehicle

It is important to stress that the provisional specifications for the vehicle, its mission profile, and propulsion options are only initial specifications from which to begin more detailed designs and technology development, rather than representing a finalized vehicle concept.

### *Provisional Vehicle*

The HALE ROA studies were jointly investigated by NASA Langley Research Center, Glenn Research Center, and Dryden Flight Research Center (Refs. 3 and 4). The focus is on extending operational duration at high altitude from hours to weeks.

Specifically, the capability set includes highly automated operation with 14-day endurance above 60,000 ft (18.3 km) while carrying a 440 lb (200 kg) payload. The capability set only presents the main features of a future class of vehicle that is envisioned to be feasible after a development span of 5 years; it does not provide the complete set of requirements necessary to bound design solutions. In order to complement the information provided by the capability set, requirements and constraints used in past studies of similar vehicles are applied (Refs. 1, 5, 7, and 8). A maximum velocity capability of greater than 230 ft/s (70 m/s) is specified to ensure station-keeping capability in high winds. Wingspan is limited to less than 250 ft (76 m) in consideration of existing ground infrastructure and ground handling issues. A payload power of 1 kW is assumed. A service ceiling of 65,000 ft (19.8 km) is used in the design to provide adequate altitude performance at 60,000 ft (18.3 km) in off-nominal conditions.

A key performance parameter affecting design optimizations is to increase duration at high altitude toward the 14-day goal. Since duration aloft is strongly affected by the choice of propulsion, three different propulsion options are examined. From a propulsion standpoint, the primary factor limiting endurance is the specific fuel consumption (SFC), which is a measure of the fuel flow rate required to provide a specific power output. Specific fuel consumption is a function of the energy content of the fuel (energy/mass) and the thermal efficiency of the propulsion system (how much of fuel energy is converted into useful thrust energy as opposed to heat). Improvements in SFC can be realized through increased thermal efficiency or increased energy content. The span of propulsion options addresses the impact of efficiency, while the choice of hydrogen fuel addresses the increase in energy content.

The use of hydrogen increases the amount of energy that can be stored on-board an aircraft for a given fuel mass (2.80 times greater than JP aviation fuel (Ref. 1), but presents different design challenges compared to traditional fuels. For example, hydrogen's energy-per-*volume* density (in contrast to *mass* density) is less than aviation fuel, even when stored as a cryogenic liquid (LH<sub>2</sub>) having a mass density of 4.4 lb/ft<sup>3</sup> (342 kg/m<sup>3</sup>). Additionally, the LH<sub>2</sub> tanks must be thermally insulated. The optimum LH<sub>2</sub> tank shape, from a structural and insulation standpoint, is a sphere. This spherical geometry and the relatively large tank volume present different integration challenges when designing the whole aircraft. The use of



Figure 1.—Artist Concept for HALE ROA. The wingspan is 250 ft (76 m) and each engine pod is approximately 9 ft (3 m) wide. To put this into perspective, this wingspan is 83 percent as long as a football field.

more aerodynamically efficient tank shapes (e.g., cylindrical) might provide better aerodynamic performance, but will incur mass penalties.

To fully realize the benefits of hydrogen fuel, design trades are needed to find the optimum combination of tank geometry, insulation mass, operating pressure, and matching the fuel delivery methods to the specific propulsion system needs.

The basic configuration for the reference vehicle is a wing-fuselage-tail layout with two propulsion pods on the wings. Each propulsion pod is sized to fit around a spherical liquid hydrogen fuel tank, and contains a propulsion system to drive a propeller. The primary payload bay is in the centerline fuselage pod. An artist concept of the vehicle is shown in figure 1.

Characteristics of this conceptual vehicle are summarized in Table 1. Ranges in some values reflect variations during the iterations of the vehicle-level studies. In all cases, the lower values are the ones applied through the remainder of this hydrogen system study.

TABLE 1.—PROVISIONAL VEHICLE SPECIFICATIONS

|                      |                   |   |
|----------------------|-------------------|---|
| 65,000 ft            | (20 km)           | Service ceiling   |
| 60,000 ft            | (18 km)           | Operational altitude  |
| 136 kts              | (70 m/s)          | Maximum velocity required at operational altitude   |
| 440 lb               | (200 kg)          | Payload mass  |
| 1.34 hp              | (1 kW)            | Payload power   |
| 143 to 148 hp        | (107 to 110 kW)   | Total combined engine power, at 100 percent throttle (takeoff and climb), provided by 2 engines |
| 75 hp                | (56 kW)           | Total combined engine power at 52 percent throttle (cruise), provided by 2 engines              |
| 2646 to 2755 lb      | (1200 to 1250 kg) | Total hydrogen capacity (Stored in 2 tanks, each in a separate engine pod)                      |
| 632 lb               | (287 kg)          | Assumed advanced technology tank mass (Total, 2 tanks)  |
| 8.5 ft               | (2.6 m)           | Spherical Hydrogen tank inside diameter   |
| 9266 to 9450 lb      | (4203 to 4285 kg) | Gross takeoff weight (GTOW)   |
| 0.015 to 0.016 hp/lb | (25 to 26 W/kg)   | Equivalent power-to-weight ratio  |

As a reflection of the preliminary nature of this study, details such as landing gear and specific component layouts are not considered.

## Representative Mission Profile

The various stages of the mission profile present different operational challenges for the hydrogen fuel system. Each of the five mission stages are identified in Figure 2 and discussed in the subsequent paragraphs. Figure 2 also shows representative values of altitude, pressure, temperature, fuel demands, and tank fill level.

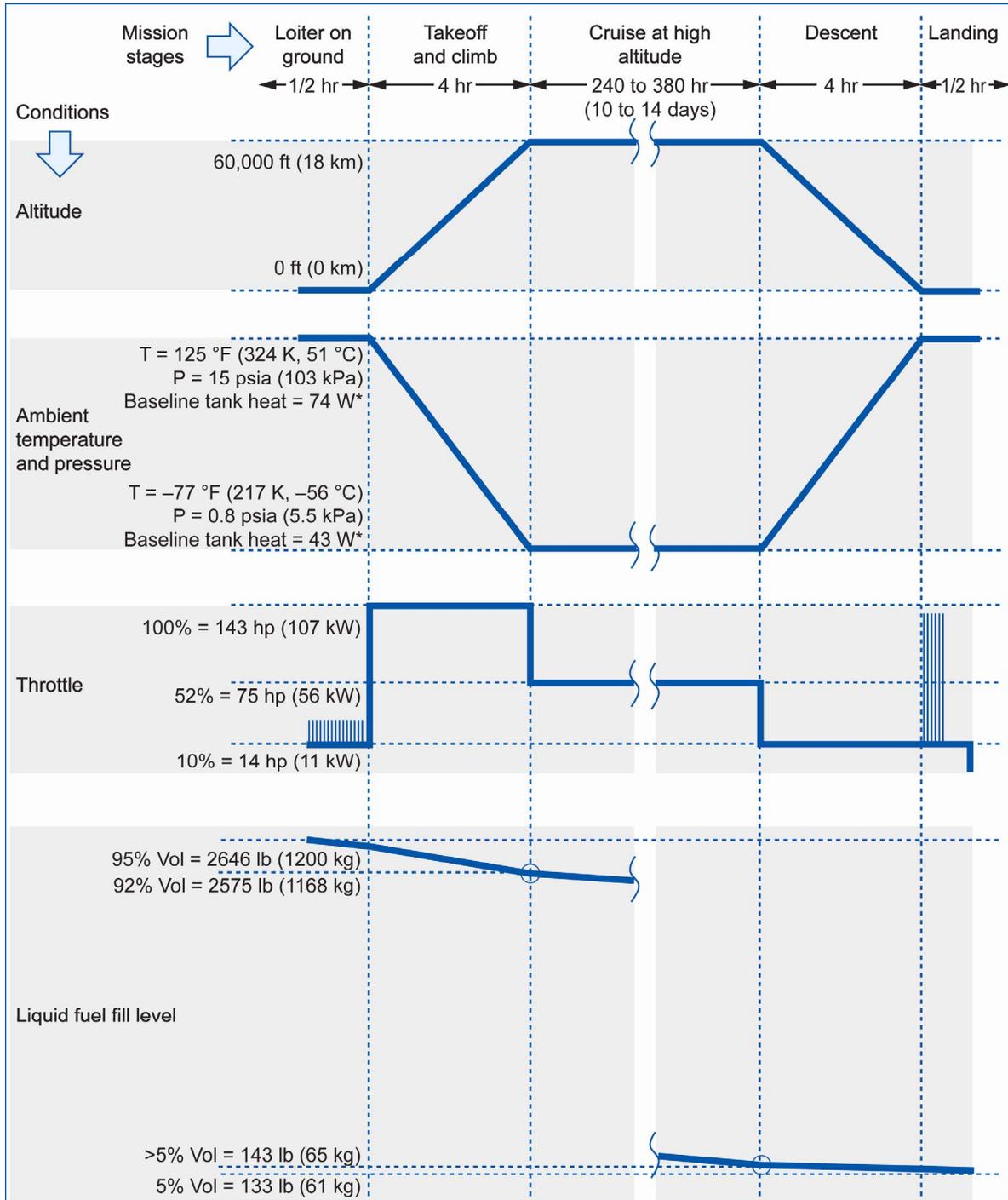


Figure 2.—Mission stages and key variables.

To understand the specific design challenges presented by each of these stages, it is helpful to first review an ideal system. In principle, the intent is to balance the ambient (passive) heat into the tank – which boils off some hydrogen and raises the tank pressure (called “autogenous” pressurization) – with the hydrogen flowing out that supplies the propulsion system with fuel. If the ambient heat and fuel consumption rates are fixed, it would be possible to exactly balance these two such that the normal warming of the hydrogen was just enough to continuously supply the hydrogen to the engines with a completely passive system. Or, in other words, the natural heating of the tank creates the pressure to keep pumping the fuel. Realistically, however, the effects of the ambient heating of the tank will not exactly balance the boil-off flow into the propulsion system, and other design features must be incorporated. For introductory purposes, Equation (1) represents a balanced condition (Ref. 9)

$$mh_{fg}\left(x + \frac{\rho_v}{\rho_l - \rho_v}\right) = Q\left(1 + \frac{t_{Off}}{t_{On}}\right) \quad (1)$$

Where

|           |  |
|-----------|--|
| $m$       | mass flow rate of hydrogen out of the tank   |
| $h_{fg}$  | hydrogen latent heat of vaporization   |
| $x$       | state of hydrogen leaving the tank, where<br>$x = 1$ means saturated vapor<br>$x = 0$ means saturated liquid |
| $\rho_v$  | density of saturated vapor (a function of tank pressure)   |
| $\rho_l$  | density of saturated liquid (a function of tank pressure)  |
| $Q$       | heat into the tank   |
| $t_{Off}$ | duration when the hydrogen flow out of the tank is off   |
| $t_{On}$  | duration when the hydrogen is flowing out of the tank  |

The situation is more complicated than reflected by this introductory equation. For example, the densities,  $\rho_v$  and  $\rho_l$ , are functions of tank pressure. Tank pressure variations are a function of tank fill level. For example, for a given fuel flow rate, tank pressure drops more rapidly when the tank is nearly-full than when near-empty. For a more complete reference on cryogenic propellant storage and delivery, refer to Reference 10.

In addition to just these considerations in balancing the system, the impact of other design options needs to be considered. For example, the fuel can be extracted from the tank as either vapor (tapping the top of the tank, or as liquid (from the bottom of tank), depending on the needs of the propulsion system. The option to tap the vapor is only advantageous in very low-flow situations, since extracting vapor drops the pressure more rapidly than extracting the same mass of liquid, especially when the tank is nearly full. It is a more common practice to extract liquid.

The next paragraphs describe how each of the missions stages present different challenges to this heat/flow-rate balancing act. When designing a system, enough margin and features must be included to encompass this full span of variations.

### ***Mission Stage 1, Loitering Before Takeoff***

For the hydrogen fuel system, loitering before takeoff is the most demanding part of the mission in terms of calculating the required tank insulation. The vehicle is sitting on the ground with full tanks while the ambient pressure and temperature are at their maximum and fuel flow is at its minimum. The worst-case condition includes operation in the desert, where the ambient temperatures can reach 125 °F (324 K, 51 °C). This stage presents the maximum heat rate into the tank, the least pressure mitigation (virtually no

fuel is flowing out of the tank), and offers the least margin for liquid expansion due to the filled-to-capacity tanks (using the standard practice of 5 percent ullage). Without any venting, the tank pressure will rise rapidly. Boil-off losses are expected at this stage, which could be mitigated by ground support equipment designed to top-off the tanks prior to takeoff.

The liquid hydrogen tank pre-fill and chill-down procedures that would precede this stage were not examined in this study.

### ***Mission Stage 2, Takeoff and Climb***

With a 100 percent throttle setting for takeoff and climb, the fuel consumption is at its maximum, which works to mitigate the effects of the high passive heat into the hydrogen tank. As the vehicle climbs, the heating gradually drops as the ambient temperature and pressure diminish. If hydrogen flows out of the tank faster than the passive heating can make up for the pressure difference, other means to recover the tank pressure would be needed. In other words, the full-throttle setting could drop the tank pressure too much when the aircraft reaches the cooler, higher altitudes. Mitigation options include active heaters and an auxiliary pressure system.

Although not detailed in this study, provisions for dumping excess hydrogen in the event of an aborted mission would need to be specified as part of an operational system.

### ***Mission Stage 3, Long Duration Cruise***

At cruise altitude the passive heat into the tank is at its lowest (approximately 43 W), while the throttle setting is around 52 percent. These particular values present a more easily sustainable balance between the ambient heating and fuel consumption. Under these circumstances, it is easier to maintain the hydrogen in liquid form for long periods (days), especially in contrast to the circumstances encountered while on the ground. Minor boil-off rates are still expected at this stage, however.

There are other challenges that arise at these high altitudes, however. At the very low ambient pressure at altitude (0.8 psia), the propulsion system requires significant air compression to meet the oxygen supply demands. A similar Unmanned Aerial Vehicle (UAV) using an internal combustion engine required 2 or 3 stages of compressors (Ref. 5). From another experience, it was found that venting cryogenic propellants at high altitude faced the risk of clogging the vents (Personal communication, Tom Tomsik 2005, May 10). Since pressure at the cruise altitude is 0.8 psia and the triple-point pressure of LH<sub>2</sub> is 1.02 psia, LH<sub>2</sub> could potentially change-phase to its triple-point if direct liquid venting occurred, thereby causing solid slush to form at the vent. Other than noting these issues, accommodating such details in the design was beyond the scope of this preliminary study.

### ***Mission Stage 4, Descent***

During descent the aircraft transitions back to the higher ambient heating, while at minimum fuel consumption. The tanks are nearly drained of liquid, so the pressure rise sensitivity from the ambient heating and fuel flow rate is lower than during the climb. This is due to the larger ullage volume to accommodate the ambient heat effects. In other words, the lower tank fill level makes it easier to maintain safe tank pressure, even though little fuel is being taken out of the tank.

### ***Mission Stage 5, Landing and Post-flight***

In addition to the different tank pressure sensitivities when at low liquid fill levels; this stage of the mission presents the challenge of rapid changes in throttle to accommodate approach and landing maneuvers. This raises the issue of accommodating transients in the pressure and flow thermodynamic balances.

Once the aircraft lands, procedures for safely venting the remaining hydrogen will need to be specified. This study did not go into such procedures, but did include the thermal balance of the condition of loitering on the ground with near-empty tanks.

### ***Propulsion Options***

Three different propulsion options are included as part of the design space to explore the fuel system impacts. Each option has different input pressure and temperature requirements that affect the delivery system specifications. Higher input temperatures require more significant heat exchangers, while higher delivery pressures require either higher tank pressures and/or more powerful pumps. Table 2 outlines the key features distinguishing the various propulsion options. Note that the delivery pressures for all power plants are modest, but the input temperature requirements vary considerably.

TABLE 2.—PROPULSION OPTION SPECIFICATIONS

|   | Internal combustion engine        | Solid Oxide Fuel Cell (SOFC) and electric motors | Polymer Electrolyte Membrane (PEM) fuel cells and electric motors |
|---|-----------------------------------|--|---|
| Estimated efficiency                            | 34 percent                        | 45 percent                                       | 55 percent  |
| *Corresponding maximum fuel flow rate (143-hp)  | 21/lb/hr<br>(10 kg/hr)            | 16/lb/hr<br>(7 kg/hr)                            | 13/lb/hr<br>(6 kg/hr)   |
| *Corresponding cruise fuel flow rate (75-hp)    | 10/lb/hr<br>(5 kg/hr)             | 8/lb/hr<br>(4 kg/hr)                             | 7/lb/hr<br>(3 kg/hr)  |
| Fuel input pressure                             | 60 to 75 psia<br>(410 to 520 kPa) | 15 to 60 psia<br>(100 to 410 kPa)                | 15 to 50 psia<br>(100 to 340 kPa)                                 |
| Fuel input temperature                          | -4 to 212 °F<br>(-20 to 100 °C)   | 1300 to 1700 °F<br>(700 to 900 °C)               | 180 °F<br>(82 °C)   |
| References consulted for values shown in tables | (Refs. 1, 5, 11, and 12)          | (Refs. 13 and 14)                                | (Refs. 14 to 16)  |

\*Flow rates calculated based on efficiency relative to 100 percent utilization of hydrogen energy.

Fuel flow rate is based on both the required power and on the efficiency of the propulsion system. The 100 percent efficiency limit for converting hydrogen flow into engine power is based on hydrogen’s Heat of Combustion of 51,620 Btu/lb ( $120 \times 10^6$  J/kg) (Ref. 17) followed by converting units into terms of power per flow-rate, to yield a value of 20.3 hp-per-lb/hr (33.3 kW-per-kg/hr). Applying this value to the vehicle’s maximum throttle setting of 143-hp (107-kW), yields a hydrogen flow of about 7-lb/hr (3 kg/hr) for a 100 percent efficient engine. For a 50 percent efficient engine, the flow would be twice that, or 14-lb/hr (6 kg/hr). The fuel flow rates shown in Table 2 were calculated in this manner, using the estimated efficiency values shown in the first row of Table 2.

## **Design Trades and Findings**

Using the mission and vehicle constraints as the basis of the hydrogen system design, various design trades were conducted. The trades involve specifying tank geometry, insulation, pressurization control, and options for delivering the hydrogen to the various propulsion systems. Initial trades determine the best configurations for a given application, followed by further refinements to specify component values.

A tank system spreadsheet was generated to examine the effects of mission duration, ambient temperature and pressure, tank pressure, tank geometry, tank material and insulation type. For this specific application, the variables affecting these design trades span considerably different values depending on the stage of the mission, as specified in Figure 2. In some cases this means conducting the trade for the worst-case mission stage and then carrying these values through the other mission stages and subsequent trades.

Hydrogen properties for this investigation were obtained from Reference 17 and in appendix C of Reference 10. Polynomial curve fits of relevant hydrogen properties were generated and used to

automatically calculate the required properties for the analysis. The important liquid hydrogen properties used were density, specific heat, and heat of vaporization.

It should be emphasized that these analyses only considered steady state operation and did not address transitions from one mission stage to another or how to accommodate rapid throttle changes or fluctuations in other operating variables.

## **Tank Geometry**

Tank geometry is a trade between aerodynamic streamlining and mass. Although a spherical tank provides the least tank mass per fuel mass, shifting to a more streamlined tank reduces drag, which in turn, reduces the required aircraft power and potentially reduces aircraft mass. As stated previously, the provisional vehicle for this study dictates that two tanks, in two separate pods, is the baseline configuration, even though this configuration has more drag than a single tank (26 percent increase in frontal area of tankage). For this study, the benefits of streamlining these two tanks are assessed.

It was found that spherical tanks are best for this application. When shifting from a spherical to cylindrical tank, the increased surface area increases both the tank wall mass and the amount of insulation needed. Furthermore, stresses are higher for cylindrical sections than spherical, which increases the minimal allowable tank wall thickness, again increasing weight. A parametric analysis was performed to compare the spherical versus cylindrical tanks with a maximum length/diameter ratio up to 6. It was determined that tank aspect ratios above 4 to 5 (length/diameter) have diminishing drag benefits, and then only offer a 2 percent reduction in drag.

Another complication when considering the use of long cylindrical tanks is how to minimize sloshing and maintain weight balance across the full length of the tank. Since the analysis indicated that a spherical tank is the preferred choice, the issues associated with sloshing were not further explored. To proceed further with an aircraft hydrogen system design, the effects of sloshing would have to be considered even for spherical tanks, because of the potential for rapid pressure changes in the tank that can be brought on by sloshing.

## **Tank Operating Pressure and Wall Thickness**

Aluminum alloy is a common material used for cryogenic storage tanks. It has superior strength-to-weight characteristics when compared to other cryogenic-compatible materials (Ref. 1), and can be easily built using state of the art pressure vessel construction techniques. Reducing the tank pressure reduces the structural requirement of the tank, but two limitations can be encountered. First, if there is a significant pressure difference between the tank pressure and the required delivery pressure, the mass savings might be cancelled by the added mass of pumps or compressors required to produce the required delivery pressure. Second, there is a lower limit to pressure vessel wall thickness so pressures lower than some minimal value do not necessarily result in lower tank mass.

For this application, the inlet pressures for all the propulsion systems are modest, 15 to 75 psia (100 to 520 kPa), so pumping requirements are minimal. The analysis considered tank pressures spanning from 20 to 50 psia, and found that a tank operating pressure of approximately 30 psia was optimum. Pressures lower than this provided no further benefit since this operating pressure coincides with the minimum tank wall thickness of 0.020 in., including a safety factor of 1.5 for the tank strength properties at cryogenic operating temperatures. Tank wall thickness is based on standard strength of material calculations. The issue of whether to use a vacuum jacket tank to surround the hydrogen tank is addressed under *Insulation Trades*.

## Autogenous or Helium Pressurization System

In general, when considering hydrogen delivery from a tank, the options include extracting liquid from the bottom of the tank or the vapor from the top of the tank. While using vapor reduces the requirements for downstream heat exchangers – since the fuel has already been vaporized – it risks dropping the tank pressure excessively. Also, downstream compressors would likely be required to raise the pressure to match the propulsion system input requirements. In contrast, extracting liquid is the more common approach, which will likely require liquid pumps followed by heat exchangers to vaporize the fuel. The specifics of each application affect which option is best.

Given the modest tank pressure (30 psia) and the wide variation of mission regimes, it was concluded that this application is better served by extracting the liquid instead of vapor and having an auxiliary helium pressurization system. Even though autogenous pressurization seems like it might be feasible over the span of *steady-state* operating conditions, the margins are too close to ensure sustained pressure during possible transients. Maintaining net-positive-suction-head across the pumps is crucial to the pumps' operation. Given the modest pressure differences between the tank pressure and delivery requirements (Delta Pressure 0 to 45 psi), a pump with a power of 0.03 hp (20 W) is sufficient for this application. Heaters within the tank are also included as a contingency to raise pressure, although these have slower response times compared to a helium pressure system.

## Passive Versus Active Thermal Control

Another operation choice is whether to use active cryogenic cooling technologies such as thermodynamic vent systems (Refs. 9 and 18) or cryocoolers similar to those evaluated for zero boil-off (ZBO) space missions (Refs. 19 to 22). While these add weight, they are designed to reduce pressure rises and boil-off. Most of this technology was originally developed for spaceflight without consideration for aircraft operations. Compared to aircraft, in-space environments have lower background temperatures and the vibrations are considerably lower than with aircraft operations. In the microgravity of space and in the absence of any atmosphere, convective heat transfer into the tank is eliminated and the only source of vibration is from the spacecraft equipment itself. Aircraft motion is considerably more dynamic than that of a spacecraft, and the resulting hydrogen sloshing in an aircraft could induce rapid pressure change in the tanks. All of these differences affect the applicability of this technology to aircraft operations. These trades are application-specific, being affected by mission duration, operating environment, and propellant delivery schedules.

To compare passive to active thermal control, the masses of insulation and auxiliary equipment is compared. For the passive case, thermal control of the propellant is provided only by the insulation system, such as spray on foam insulation (SOFI) (Ref. 23) or vacuum jacketed multilayer insulation (MLI) (Ref. 24). For the active case, SOFI tank insulation was augmented with cryocooler to offset heat leak by actively cooling the liquid hydrogen. This was compared to a tank insulated with vacuum jacketed MLI with no cryocooler to determine if the additional cryocooler mass would result in a net lower total system thermal control mass. Heat leak into the cryogen was calculated by considering the total heat leak through the insulation and supports. Heat leak for both of these contributors is calculated using the same relationship

$$Q = \frac{kA\Delta T}{x} \quad (2)$$

Where

- $Q$  Heat leak
- $k$  Thermal conductivity of insulation or support
- $A$  Surface area through which heat passes (tank surface area, or cross sectional area of support)

$\Delta T$  Temperature difference between ambient conditions and the cryogen  
 $x$  Insulation thickness or length of support

Results are shown in Table 3. A nominal SOFI thickness of 6 in. (0.152 m) is assumed. For this thickness, an additional 700 W (40 Btu/min) of heat need to be removed from the propellant tank when compared to the MLI insulated case. To remove this amount of heat, cryocooler with an input power of 26.7 kW and a mass of 3,090 lb (1,440 kg) is required. Cryocooler input power and mass is estimated based on a parametric study of historical cryocooler data in Reference 22. Equation (3) shows the relationship used to determine cryocooler input power.

TABLE 3.—COMPARISON OF SOFI WITH CRYOCOOLER TO MLI THERMAL CONTROL

|   | SOFI with cryocooler | MLI     |
|---|----------------------|---------|
| Insulation thermal conductivity (w/m-K) | 0.02421              | 0.00035 |
| Insulation density (kg/m <sup>3</sup> ) | 62.47                | 24.03   |
| Heat leak through insulation (W)        | 776.5                | 70.3    |
| Tank Surface Area (m <sup>2</sup> )     | 18.12                | 18.12   |
| Delta T (K)                             | 270                  | 270     |
| Insulation thickness (m)                | 0.152                | 0.024   |
| Cryocooler size (W)                     | 26720                |         |
| insulation and jacket mass (kg)         | 380.47               | 119.55  |
| Cryocooler mass (kg)                    | 1440                 |         |
| Tank Thermal Control Mass (each) (kg)   | 1820.26              | 119.55  |

$$P_{\text{input}} = \frac{Q_{\text{out}} * (T_{\text{hot}} - T_{\text{ch}})}{T_{\text{ch}} * \eta} \quad (3)$$

Where

- $P_{\text{input}}$  Required cryocooler input power
- $Q_{\text{out}}$  Cryocooler output power required (set to 40 Btu/min)
- $T_{\text{hot}}$  Cryocooler rejection temperature
- $T_{\text{ch}}$  Cryocooler cold head temperature
- $\eta$  Cryocooler Carnot efficiency (estimated at 20 percent)

For the case analyzed, the addition of the cryocooler results in a significantly higher over all system mass for thermal control. Therefore, passive cooling (insulation) is the better option for this application. Mass of the cryocooler is based on the following relationship (Ref. 22)

$$m = 0.1422(P_{\text{input}})^{0.905} \quad (4)$$

### Insulation Trades

With the option of active cooling dismissed for this application, the next trade determines the best insulation option. This trade evaluates the minimum *total* system mass, where the system consists of the tank, insulation and hydrogen boil-off masses. Additional insulation reduces heat leak and consequently reduces hydrogen boil-off, but at the expense of additional insulation mass. Less insulation results in lower insulation mass, but is offset by additional hydrogen propellant and tank mass required to accommodate boil-off losses. The weight penalty of too much insulation has the additional negative impact of adversely affecting autogenous tank pressurization.

Part of this trade determines if a vacuum jacket is needed over the fuel tank, or if foam insulation will suffice. A number of variations were assessed, with the recurring conclusion that this application requires a vacuum-jacketed tank with a modest (approximately 1.5 in.) amount of multilayer insulation (MLI) inside a vacuum gap of 1.7 in. between the hydrogen tank and its vacuum jacket. Figure 3 is an example of one of these assessments which shows that the relative difference in total tank mass for various insulation options. The heat leak for the various mission segments was calculated based on Equation (2), and the ambient temperatures for the mission segment. This heat leak was translated into equivalent boil off by dividing heat leak by the heat of vaporization and assuming that all the heat entering the system converted liquid to vapor.

In a parallel study, the performance projections for advanced materials is assessed, comparing nanoclay-enhanced graphite/epoxy composites and polymer electrolyte cross-linked silica aerogels to the standard vacuum-jacketed, MLI insulated, aluminum tank (Ref. 6). This assessment also found that the vacuum-jacketed aluminum tank is better for this particular application, with Figure 3 also representative of these parallel findings. More specifically, at its current stage of development, the graphite/epoxy construction does not yet provide a significant weight savings compared to aluminum, and the nanoclay-enhanced graphite/epoxy still has hydrogen permeation issues for limiting vacuum pressure loss to less than 0.10 psi per day. The nanoclay-based materials are presently better suited for the vacuum jacket construction. The aerogel insulation was not found to be advantageous for this long-duration application, being better suited for shorter-duration applications. The parallel study also identified technology goals and approaches to improve and test the materials' performance. The details underlying these parallel conclusions can be found in Reference 6.

When calculating the mass of the vacuum jacket, wall thickness must be sufficient to prevent buckling, since the vacuum jacket is subjected to external pressure. The required wall thickness for this outer jacket is 0.12 in. Additional stiffeners would most likely be required between the vacuum jacketed shell and the inner tank. Again, aluminum alloy is the conventional material used for constructing vacuum jackets for cryogenic storage tanks.

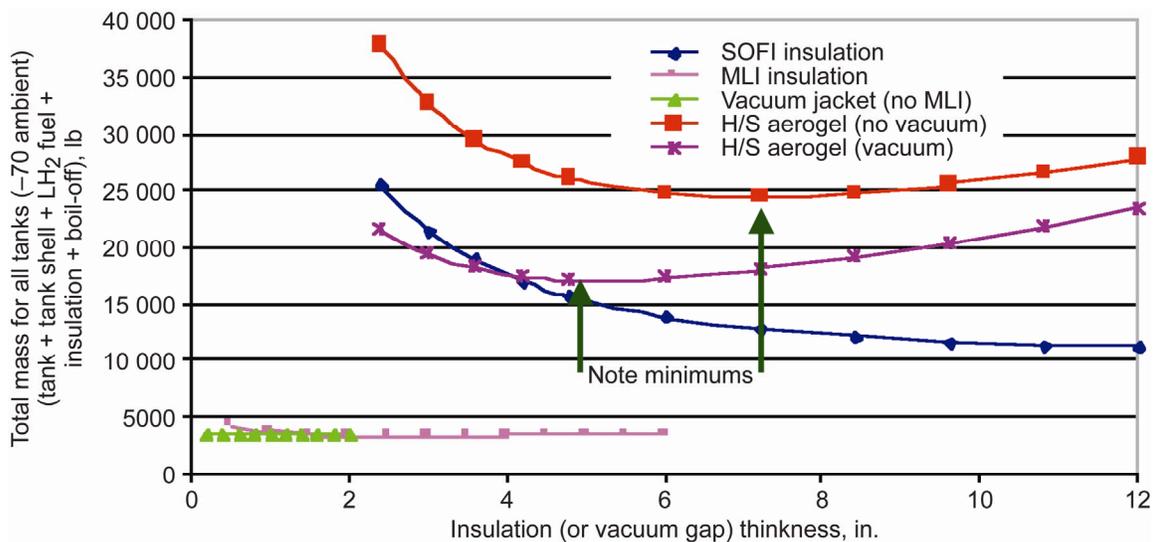


Figure 3.—Example of insulation trade analyses.

## Resulting Baseline Cryogenic LH<sub>2</sub> Storage Tank

Combining the findings of the prior trades and applying existing cryogenic tank technology (Ref. 1), the design presented in Figure 4 represents the spherical, insulated aluminum tank recommended for this application. The tank is not integrated into the vehicle structure, but rather is a separate structure mounted inside the aircraft. Given the spherical geometry, instead of a streamlined shape that could be integrally designed as part of the aircraft structure, the weight of the tank, its vacuum jacket, and hydrogen load is carried through to the airframe via a center column structural support. Multilayer insulation, 1.5 in. thick, is located in the 1.7 in. gap between the inner hydrogen tank and its outer vacuum jacket tank. Hydrogen mass is measured using a capacitance fill-level probe. Liquid and vapor piping penetrations are provided for filling, draining, pressurization and venting.

## Compatibility With Propulsion Systems

Since the delivery pressures required for each of the propulsion systems are relatively similar, as noted previously in Table 2, the system schematics for each configuration have few variations. The most prevalent distinctions are with the heat exchangers and fuel reclamation equipment, which are considered part of the propulsion system rather than part of the delivery system. The propellant delivery schematics for each of the propulsion options, described below, are shown in Figures 5 to 7.

For all three propulsion options, both hydrogen tanks (one in each wing pod) and tank pressurization scheme are the same. The tanks are pressurized using a common helium pressurant system. Each tank is equipped with liquid level sensing and redundant pressure relief devices. Isolation valves are installed upstream and downstream of the liquid hydrogen pump and in other portions of the system to meet anticipated servicing operations. Fill and drain valves are provided for each tank and the two liquid hydrogen tanks are linked together through the use of a common branch with an isolation valve that can be automatically toggled to allow for liquid hydrogen balancing between the two tanks. The liquid hydrogen pump performance requirements vary slightly based on the pressure and flow delivery requirements unique to each propulsion system, but overall a pump power of 0.03 hp (20 W) is sufficient.

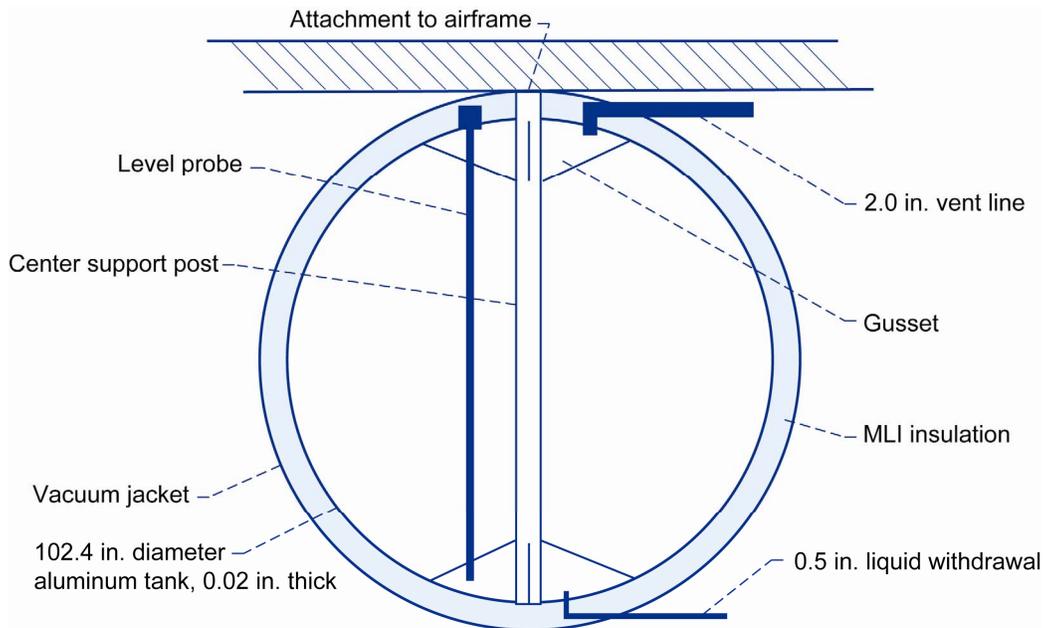


Figure 4.—Baseline aluminum vacuum-jacketed cryogenic tank.

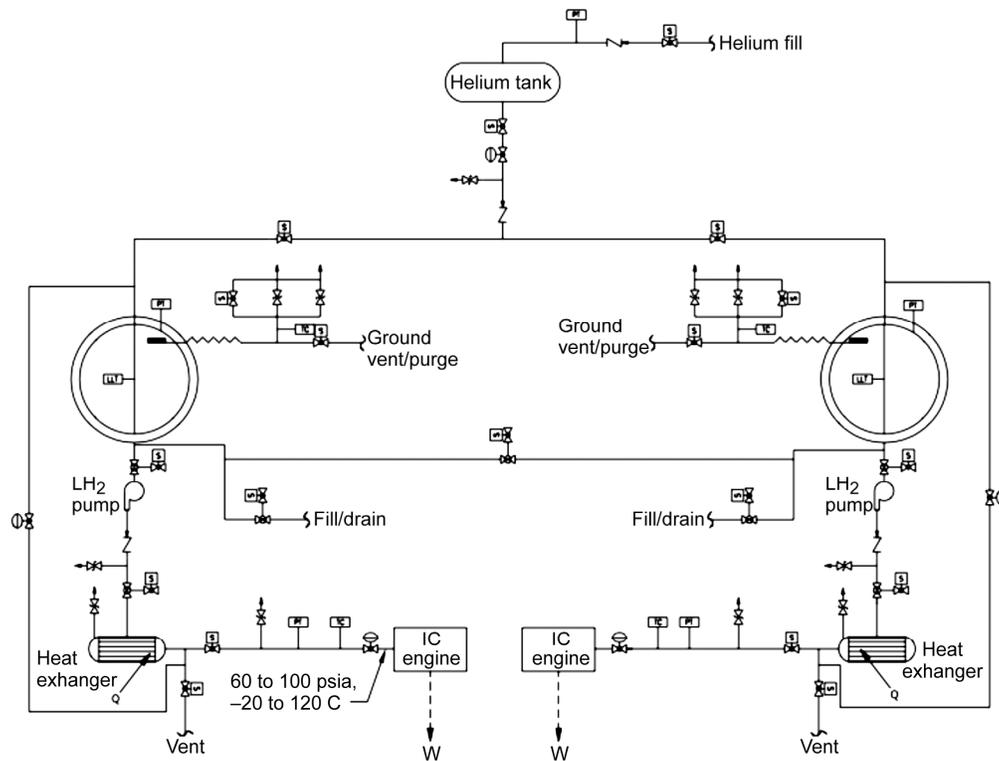


Figure 5.—Internal combustion engine hydrogen supply schematic.

### ***Internal Combustion Engine System (Fig. 5)***

For the internal combustion engine system, the liquid is vaporized through a heat exchanger downstream of the liquid hydrogen pump to deliver hydrogen gas to the engine in a temperature range of  $-20$  to  $120$  °C. A pressure regulator is used to provide hydrogen inlet pressures in the range of 60 to 100 psia.

### ***Solid Oxide Fuel Cell System (Fig. 6)***

For the SOFC system, the liquid hydrogen is vaporized through a heat exchanger using the cathode side exhaust that is downstream of the liquid hydrogen pump. During initial start-up, prior to the fuel cell acquiring its operating temperature, auxiliary heaters may be required to provide hydrogen vaporization. The hydrogen passes through another heat exchanger to increase the temperature to the required  $800$  °C at the solid oxide fuel cell inlet. This additional heat is provided from the anode exhaust flow. The anode exhaust flow passes through a catalytic combustor for additional energy extraction. After heating the hydrogen, the anode exhaust is used again to provide heating to the incoming air from the compressor.

### ***PEM Fuel Cell System (Fig. 7)***

For the PEM fuel cell system, humidification is required for the fuel cell membranes. Downstream of the liquid hydrogen pump, the liquid hydrogen is vaporized through a heat exchanger, humidified, and regulated to 30 to 50 psia before entering the anode side of the fuel cell. After passing through the fuel cell, the exhaust gas is condensed to liquefy and recover the water for the closed-loop water system. The air is also heated and humidified prior to entering the cathode side of the fuel cell, and is then exhausted.

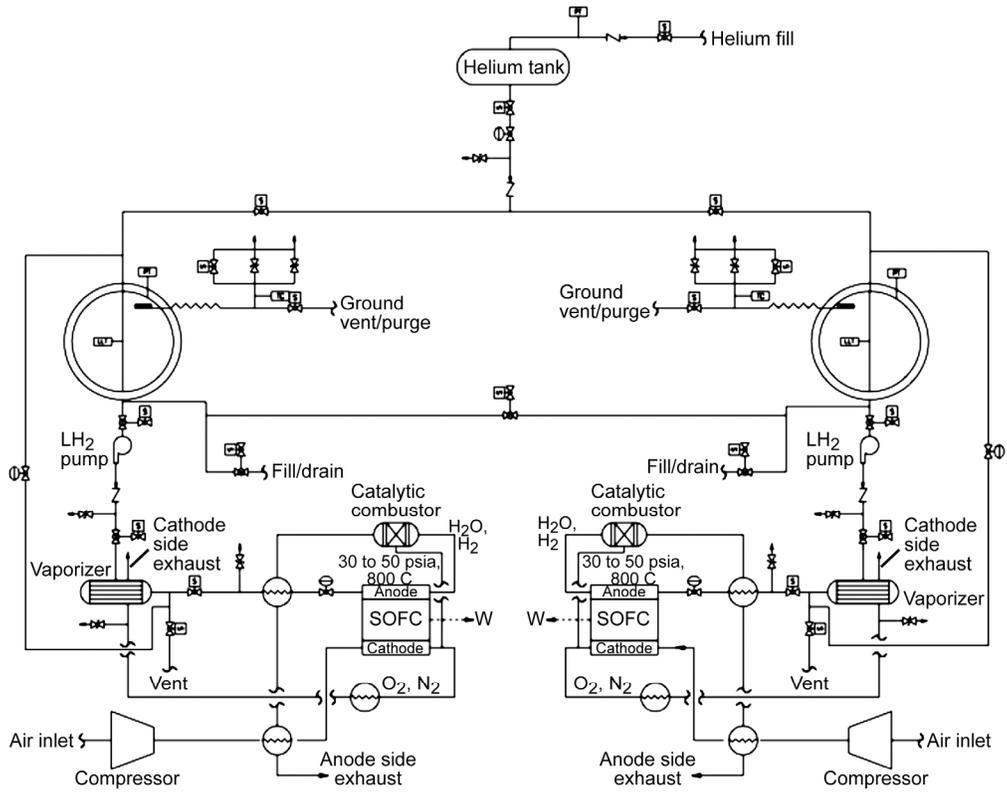


Figure 6.—Solid oxide fuel cell system (SOFC) hydrogen supply schematic.

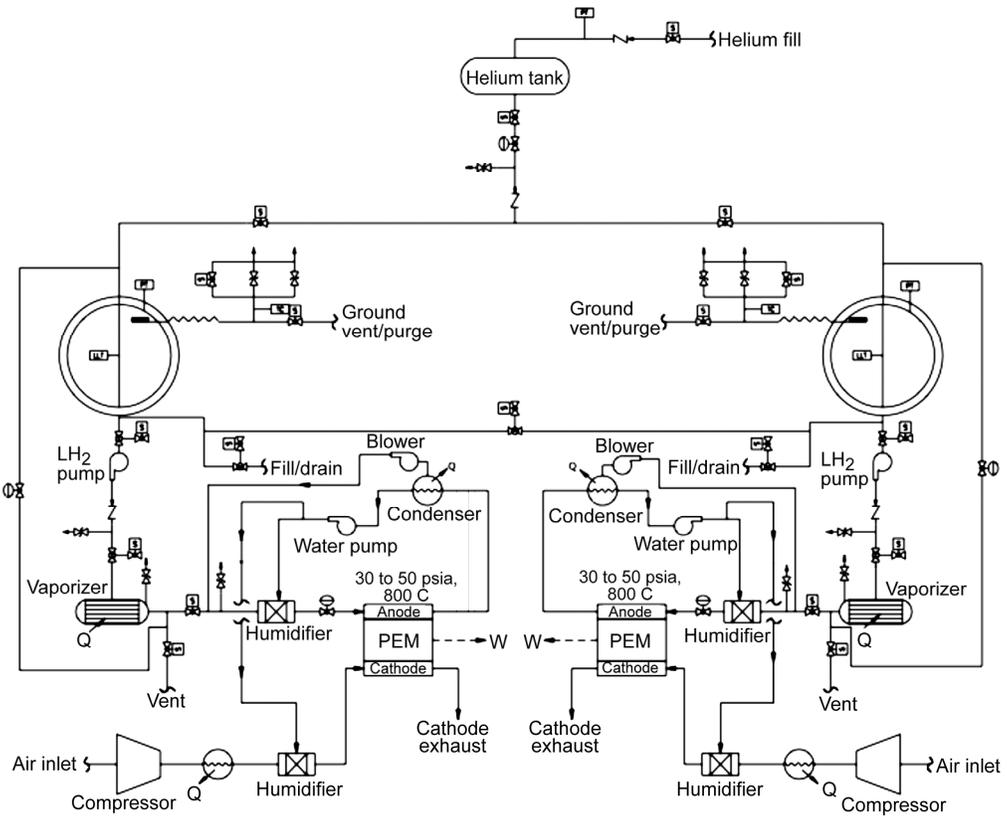


Figure 7.—Propellant feed system (PEM) fuel cell hydrogen supply schematic.

## Thermodynamic Analysis

Based on the system schematics shown in Figures 5 to 7, the mission stages presented in Figure 2, and the propulsion characteristics presented in Table 2, basic thermodynamic balances were conducted to determine if the heat levels, pressures, and fuel flow rates were reasonable, and to determine the anticipated duration aloft. Fuel consumption was determined from using the set durations at each mission stage's power settings, combined with the fuel consumption rate specific to each propulsion system. The boil off rates were calculated from the ambient heat rates, mission stage duration, and tank fill level. The results are presented in Table 4.

The duration at altitude is calculated by determining the amount of hydrogen remaining after taking into account the hydrogen required for loitering, takeoff, climb, descent, and landing. The final amount of hydrogen left in the tank is set by ullage considerations. From this analysis the following major conclusions are drawn:

1. The resulting cruise duration at altitude spans from 10 days to 16 days.
2. Even though boil-off does occur, hydrogen losses at altitude are reasonable. Eliminating boil-off would require increasing the MLI insulation and vacuum gap between the inner and outer tank from 1.7 in. to between 2 to 2.7 in., resulting in an increase in the outer tank mass between 35 to 110 lb, far less than the mass of fuel lost through boil-off.
3. A high portion of the waste power from the Solid Oxide Fuel Cell is required for vaporizing and heating the hydrogen. It is not known how much of the waste power is actually available for this function because detailed thermal analysis of the SOFC were beyond the scope of this study.

In Table 4, the term, “waste power” refers to the power not converted into propulsion (equal to 1 minus the conversion efficiency). Not all of this waste power is accessible for vaporizing and heating the hydrogen to the required operating temperature, but this value does reflect an upper limit useful for preliminary assessments. Actual system designs will require a more thorough thermodynamic assessment of the applicability of using the propulsion system’s waste heat for conditioning the hydrogen. From this preliminary analysis it appears that there is ample waste heat from the internal combustion engines, but the higher temperature fuel cell systems need further investigation.

## Initial System Mass Estimates

Based on similar equipment used for space missions and the tank mass trades previously discussed, Table 5 presents the estimated masses of the storage and delivery system. These mass estimates do not include equipment specific to each propulsion system, such as the hydrogen reclamation equipment, engines or fuel cells, humidifiers, and additional heat exchangers. To reflect the relative proportions of the masses listed in Table 5, a pie chart is presented in Figure 8.

It should be noted that the tank mass assumed in the provisional vehicle, 316 lb (143 kg) for each tank, is considerably less than the mass estimate calculated here for the vacuum jacketed tanks of 645 lb (293 kg) each. This difference was not cascaded back through the vehicle system studies to assess the impact on the whole aircraft system. Recall that even the advanced material study (Ref. 6) came to the conclusion that the vacuum-jacketed aluminum tank presents the minimum mass for this application.

TABLE 4.—THERMODYNAMICS OF THE 5 MISSIONS STAGES AND 3 PROPULSION OPTIONS

|  |         | Internal Combustion Engines | Solid Oxide Fuel Cells (SOFC) | PEM Fuel Cells |       |
|--|---------|-----------------------------|-------------------------------|----------------|-------|
| <b>Power Plant Characteristics</b>                           |         | Units                       |                               |                |       |
| Fuel consumption @ 100%=143 hp                               | lbs/hr  | 20.8                        | 15.7                          | 12.8           |       |
| Power plant operating temperature                            | Celsius | 90                          | 900                           | 82             |       |
| Waste power @ 100% throttle                                  | kW      | 70.6                        | 58.8                          | 48.1           |       |
| Hydrogen delivery pressure required                          | psia    | 75                          | 60                            | 50             |       |
| Hydrogen delivery temperature req'd                          | Celsius | 50                          | 800                           | 82             |       |
| <b>1. Loitering on Ground, 10% throttle, 1/2 hr duration</b> |         |                             |                               |                |       |
| Starting fuel quantity (Set Value)                           | lbs     | <b>2646.0</b>               |                               |                | 1st   |
| Fuel consumed  | lbs     | 1.0                         | 0.8                           | 0.6            |       |
| Boil-off   | lbs     | 0.4                         | 0.4                           | 0.4            |       |
| Ending fuel quantity   | lbs     | 2644.7                      | 2644.9                        | 2645.1         |       |
| Percent waste power for H2 conditioning                      |         | 11%                         | 34%                           | 10%            | 2nd   |
| <b>2. Climb, 100% throttle, 4 hr duration</b>                |         |                             |                               |                |       |
| Starting fuel quantity                                       | lbs     | 2644.7                      | 2644.9                        | 2645.1         |       |
| Fuel consumed  | lbs     | 83.0                        | 62.7                          | 51.3           |       |
| Boil-off   | lbs     | 2.36                        | 2.36                          | 2.36           |       |
| Ending fuel quantity   | lbs     | 2559.3                      | 2579.8                        | 2591.4         |       |
| Percent waste power for H2 conditioning                      |         | 11%                         | 35%                           | 11%            | 3rd   |
| <b>3. Cruise at Operational Altitude, 49% throttle</b>       |         |                             |                               |                |       |
| (1) Duration achievable                                      | days    | <b>10</b>                   | <b>13</b>                     | <b>16</b>      |       |
| Starting fuel quantity                                       | lbs     | 2559.3                      | 2579.8                        | 2591.4         |       |
| Balance of Fuel Available                                    | lbs     | <b>2414.3</b>               | <b>2437.0</b>                 | <b>2450.0</b>  | final |
| (2) Boil-off   | lbs     | 125.61                      | 168.01                        | 207.23         |       |
| Ending fuel quantity   | lbs     | 145.0                       | 142.8                         | 141.4          |       |
| Percent waste power for H2 conditioning                      |         | 11%                         | 35%                           | 11%            | 6th   |
| <b>4. Descent, 10% throttle, 4 hr duration</b>               |         |                             |                               |                |       |
| Starting fuel quantity                                       | lbs     | 145.0                       | 142.8                         | 141.4          |       |
| Fuel consumed  | lbs     | 8.3                         | 6.3                           | 5.1            |       |
| Boil-off   | lbs     | 2.36                        | 2.36                          | 2.36           |       |
| Ending fuel quantity   | lbs     | 134.4                       | 134.2                         | 134.0          |       |
| Percent waste power for H2 conditioning                      |         | 11%                         | 34%                           | 10%            | 5th   |
| <b>5. Landing, 10% throttle, 1.2 hr duration</b>             |         |                             |                               |                |       |
| Starting fuel quantity                                       | lbs     | 134.4                       | 134.2                         | 134.0          |       |
| Fuel consumed  | lbs     | 1.0                         | 0.8                           | 0.6            |       |
| Boil-off   | lbs     | 0.4                         | 0.4                           | 0.4            |       |
| Ending fuel quantity (Set Value)                             | lbs     | <b>133.0</b>                |                               |                | 4th   |
| Percent waste power for H2 conditioning                      |         | 11%                         | 34%                           | 10%            |       |

TABLE 5.—HYDROGEN SYSTEM MASS ESTIMATES

| Description                                      | Quantity | Each mass, lbm | Total mass, lbm |
|--|----------|----------------|-----------------|
| Propellant mass                                  | 2        | 1323           | 2646            |
| <b>Tank (HALE tank is 8.5 ft diameter)</b>       | 2        |                |                 |
| Inner shell                                      | 2        | 80             | 160             |
| Stiffeners                                       | 2        | 10             | 20              |
| Ports/attachments                                | 2        | 10             | 20              |
| Outer shell                                      | 2        | 495            | 990             |
| MLI insulation (1.5 in. thick)                   | 2        | 45             | 90              |
| Tank heaters                                     | 2        | 5              | 10              |
| <b>Total tank mass</b>                           |          | <b>645</b>     | <b>1290</b>     |
| <b>Tank pressurization system</b>                |          |                |                 |
| Helium pressurant                                | 2        | 37             | 74              |
| Helium tank (4500 psi, 23 ft <sup>3</sup> )      | 1        | 300            | 300             |
| Solenoid valve                                   | 2        | 3              | 6               |
| Pressure regulator                               | 1        | 5              | 5               |
| Check valve                                      | 2        | 3              | 6               |
| Relief valve                                     | 1        | 3              | 3               |
| <b>Total pressurization system mass</b>          |          |                | <b>394</b>      |
| <b>Propellant feed system</b>                    |          |                |                 |
| Tubing (1/2 in. w/0.035 in. Wall)*               | 120 in.  | 0.175          | 21              |
| Relief valve                                     | 4        | 4              | 16              |
| Pressure regulator                               | 2        | 4              | 8               |
| Solenoid valve                                   | 9        | 4              | 36              |
| Pump   | 2        | 10             | 20              |
| Check valve                                      | 2        | 4              | 8               |
| Instrumentation                                  | 2        | 10             | 20              |
| Heat exchanger                                   | 2        | 25             | 50              |
| <b>Propellant vent/Purge system</b>              |          |                |                 |
| Solenoid valve                                   | 4        | 3              | 12              |
| Relief valve                                     | 4        | 3              | 12              |
| <b>Propellant fill/Drain system</b>              |          |                |                 |
| Solenoid valve                                   | 2        | 4              | 8               |
| <b>Total propellant system mass</b>              |          |                | <b>211</b>      |
| <b>Total system mass (Excludes propellant) →</b> |          |                | <b>1895</b>     |

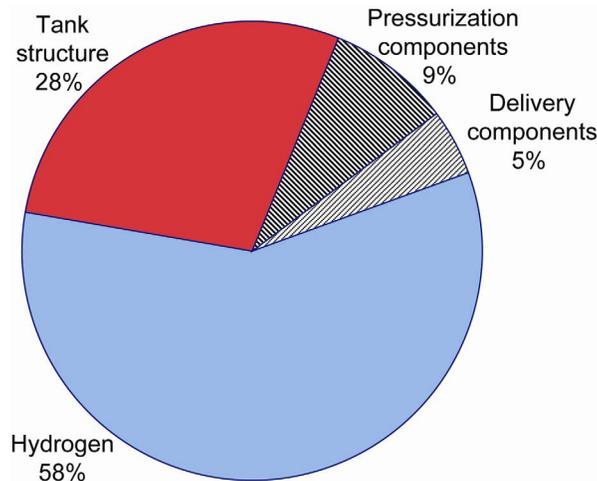


Figure 8.—Hydrogen system mass fractions.

## Technology Options and Issues Not Assessed

### *Control of Transients*

Although steady-state thermodynamic analysis were conducted for each of the 5 mission stages, no transient analyses were conducted to ensure smooth transitions across the stages or to assess the impact of throttle changes or environmental changes on the system. To advance this analysis to the next level, such assessments would be required as part of the control system design.

### *Hydrogen Densification*

Hydrogen densification options were not addressed in the preliminary study, other than to note that the cooling densification strategy could be applied later as a form of ground support equipment. Hydrogen densification means increasing the LH<sub>2</sub>'s mass per volume, so that it is possible to carry more hydrogen in a given tank. Two technologies have been advanced to address this issue, densification through further cooling and densification through gel augmentation.

Further cooling the hydrogen while the tanks are topped-off before the mission can make the hydrogen more dense. Experimental tests have demonstrated that a 7.5 percent increase in storage density is possible (Refs. 25 to 27), but this work was done in the context of space launches rather than aircraft operations. Ground support hardware for this function have been built and tested under operationally relevant conditions and rocket engines have successfully run on densified propellant (Ref. 28). The pressure rise sensitivities for the extra-cooled hydrogen would need to be examined in the context of the aircraft operation, which involves much slower processes than the rocket launches that the densification technology was originally designed for.

Another technique to increase the density of liquid hydrogen is through gel augmentations. Significant fuel density increases approaching 19 percent are possible with densified gelled hydrogen via the addition of hydrocarbons like ethane or methane, which are introduced into subcooled hydrogen as frozen particles that form a gel structure with the hydrogen (Refs. 29 and 30). Component and breadboard systems for this technology have been tested in the laboratory. The gel augmentation densification option has not been explored for this application. Any future investigations for this option would need to address the compatibility of the gelled hydrogen for the various propulsion systems.

### *Operation Details*

Operational issues, such as pre-chilling the tank and pumps prior to filling, and the entire fill and drain sequences were not investigated. It is anticipated that these might be similar to existing spacecraft operations. The operational detail of how to dump excess hydrogen in the event of a mission abort was also raised, but not addressed. These issues would need to be covered in actual designs for hydrogen aircraft systems.

## Summary of Key Issues

In the course of these analyses, the following issues were encountered that would have to be resolved in any further, more detailed, designs.

1. The impact of the higher tank system mass (approximately a factor of 2) has not been iterated with the vehicle-level studies.
2. High altitude operation requires significant compressor requirements for the air consumed by the propulsion system. This affects the power balance when considering how much waste power is available from the propulsion system for conditioning the hydrogen. Since this study focused on just the hydrogen storage and delivery system, such details remain unexplored.

3. High altitude venting of hydrogen presents the risk of hydrogen slush/solid blocking the vents.
4. The longer duration flights and more-varied motion experienced in aircraft operations, compared to spaceflight, presents the risk of inducing rapid pressure changes in the hydrogen tanks from the sloshing of liquid hydrogen. Assessing the likelihood and magnitude of these effects requires further study.
5. Control strategies to maintain tank pressure and fuel delivery conditions over all the mission stages, throttle conditions, and operational transients require further study.
6. Pre and post mission fill and drain procedures would need to be specified and may require special ground support equipment.

## Conclusions

From this initial analysis, hydrogen-fueled, long-duration aircraft is feasible, with durations aloft of 10 to 16 days within reason for the mission considered. The best storage and delivery system is a vacuum-jacketed, spherical tank of relatively low pressure (30 psia), with modest amounts of MLI insulation (1.5 in.) in the vacuum gap (1.7 in.). For fuel delivery, a helium pressurant system is included for operating margin, and low power liquid pumps are sufficient (20 W). Heat exchangers tailored for the power plants will be required, with potential issues for the higher-temperature devices.

Regarding the analysis methods themselves, preliminary assessments can be accomplished with spreadsheet parametric analysis using polynomial curve fits of hydrogen properties. The basic sequence consists of first determining tank geometry (spherical instead of streamlined), followed by determining the tank operating pressure and hydrogen delivery method. For this application, there is little variation in the input requirements of the propulsion options, so the choice of tank pressure and deliver methods are identical. This step is followed by calculations to determine the optimum insulation method. Next, thermodynamic analyses are conducted using the chosen parameters to verify that the delivery system can accommodate all the mission stages. From there, the mass estimates for the system are calculated.

The tank mass estimates derived in this assessment were roughly twice that of the initially projected tank mass assumption used in the vehicle system studies. This change has not been iterated with the vehicle system designs.

To advance to the next level of detail, specifically to assess transients and control strategies, more sophisticated modeling is required. These studies only addressed the steady-state conditions over the 5 mission stages. More detailed studies would have to additionally address the control strategies for transitioning between the different mission stages as well as accommodating transients. Preflight and postflight fill and drain options would also need to be detailed.

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| <b>13. SUPPLEMENTARY NOTES</b>  |                    |   |                                   |   |  |
| <b>14. ABSTRACT</b><br>Preliminary design trades are presented for liquid hydrogen fuel systems for remotely-operated, high-altitude aircraft that accommodate three different propulsion options: internal combustion engines, and electric motors powered by either polymer electrolyte membrane fuel cells or solid oxide fuel cells. Mission goal is sustained cruise at 60,000 ft altitude, with duration-aloft a key parameter. The subject aircraft specifies an engine power of 143 to 148 hp, gross liftoff weight of 9270 to 9450 lb, payload of 440 lb, and a hydrogen fuel capacity of 2650 to 2755 lb stored in two spherical tanks (8.5 ft inside diameter), each with a dry mass goal of 316 lb. Hydrogen schematics for all three propulsion options are provided. Each employs vacuum-jacketed tanks with multilayer insulation, augmented with a helium pressurant system, and using electric motor driven hydrogen pumps. The most significant schematic differences involve the heat exchangers and hydrogen reclamation equipment. Heat balances indicate that mission durations of 10 to 16 days appear achievable. The dry mass for the hydrogen system is estimated to be 1900 lb, including 645 lb for each tank. This tank mass is roughly twice that of the advanced tanks assumed in the initial conceptual vehicle. Control strategies are not addressed, nor are procedures for filling and draining the tanks. |                    |   |                                   |   |  |
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