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STUDY OF HYDROGEN AS AN AIRCRAFT FUEL

by

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The conversion to hydrogen as a naval aviation fuel would allow for independence on fuel cost and supply, as hydrogen is globally accessible. The biggest obstacle to using hydrogen is its very low density, a property that even combined with hydrogen's high heat of combustion still results in very large fuel tanks. Liquid hydrogen (LH2) with its higher density would still require a larger volume than kerosene for the aircraft to achieve the same mission. Another problem with using LH2 is its cryogenic nature, a property that requires complicated fuel tanks and more careful fueling. A design study has been conducted for this report to determine the feasibility of using LH2. A Lockheed-Martin P-3 Orion configuration was modified to accommodate LH2 as its fuel, its mission parameters kept unchanged. It is concluded from this design study that using LH2 would significantly limit the amount of usable cabin space, as the fuel tank takes up 65% of the aircraft's internal volume. Despite the large LH2 tank weight of about 14,865lb, due to the low fuel weight the aircraft's takeoff gross weight is only 113,646lb, about 80% of the current petroleum-fueled P-3. The total cost of LH2 as fuel is currently undetermined.

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STUDY OF HYDROGEN AS AN AIRCRAFT FUEL

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ABSTRACT

The conversion to hydrogen as a naval aviation fuel would allow for independence on fuel cost and supply, as hydrogen is globally accessible. The biggest obstacle to using hydrogen is its very low density, a property that even combined with hydrogen's high heat of combustion still results in very large fuel tanks. Liquid hydrogen (LH_2) with its higher density would still require a larger volume than kerosene for the aircraft to achieve the same mission. Another problem with using LH_2 is its cryogenic nature, a property that requires complicated fuel tanks and more careful fueling. A design study has been conducted for this report to determine the feasibility of using LH_2. A Lockheed-Martin P-3 Orion configuration was modified to accommodate LH_2 as its fuel, its mission parameters kept unchanged. It is concluded from this design study that using LH_2 would significantly limit the amount of usable cabin space, as the fuel tank takes up 65% of the aircraft's internal volume. Despite the large LH_2 tank weight of about 14,865lb, due to the low fuel weight the aircraft's takeoff gross weight is only 113,646lb, about 80% of the current petroleum-fueled P-3. The total cost of LH_2 as fuel is currently undetermined.

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EXECUTIVE SUMMARY

This nation and much of the developed world depends heavily on petroleum to fuel its economy and its military. The few nations who control the petroleum supplies, its distribution and its cost, significantly influence the world economy. As our petroleum dependence has become a critical national vulnerability, alternative fuel sources are being considered. Also, proponents for alternate fuels are searching for a "green fuel," one that would produce fewer byproducts harmful to the environment, especially carbon dioxide (CO_2) gases. This work discusses the feasibility of hydrogen as an emerging petroleum-replacement fuel, specifically in Naval aviation applications.

Due mainly to its high heat of combustion, global accessibility, and its clean combustion with air, hydrogen is currently the most promising replacement for petroleum to fuel aircraft. However, hydrogen's very low density poses a problem of storage space on aircraft. Carrying hydrogen in its liquefied form decreases the amount of volume necessary to transport the fuel, although liquid hydrogen (LH_2) still necessitates about four times the volume than that of current petroleum fuels for the aircraft to accomplish a similar mission. Another benefit of LH_2 as an aviation fuel is its low weight, as the amount of LH_2 necessary to achieve the same mission as with petroleum fuels weighs less than one-third that needed from current petroleum fuels. Not only does using LH_2 require a modified engine, but also the larger volume needed for and the cryogenic property of LH_2 requires a much different fuel

tank design and fueling process than is currently used for petroleum fuels. As the most critical design constraint is the LH_2 tank, current uses and studies were consulted to design the most reliable LH_2 tank with minimal volume and weight. A recent attempt at a composite LH_2 tank is that designed for the X-33, NASA’s experimental reusable launch vehicle (RLV) that was abandoned in early 2001. This tank design was finalized after a detailed study, and while it failed, its design is still promising.

To determine the feasibility of converting current Navy aircraft to LH2 fuel, a design study has been conducted specifically for this report on a converted Lockheed Martin P-3 Orion, the Navy’s primary anti-submarine (ASW) aircraft. To make the transition as simple as possible, the outer structure of the aircraft is unchanged. The P-3’s Allison T-56 engine was modeled using GasTurb computer software, and then modified with the software to operate with LH2 as its fuel. Using GasTurb, a volume of 4,316ft^3 is arrived at as the requirement for a P-3 to operate as it does currently. The fuel tank is designed using titanium facesheets and titanium honeycomb core, a design similar to that of the X-33’s tank. To contain the required volume of LH2, the tank would weigh about 14,865lb. Despite the large tank weight, due to the low fuel weight the aircraft’s takeoff gross weight is only 113,646lb, about 80% of the 140,000lb of the current petroleum-fueled P-3. Even by placing LH2 in the wings, the fuel tank takes up 65% of the aircraft’s internal volume, which means that the crew’s operational space would be much less than with the petroleum-fueled aircraft.

Despite the benefits to using hydrogen as an aircraft fuel, it has yet to be widely implemented. Hydrogen production facilities would have to be constructed, along with liquefaction facilities, LH_2 storage facilities, and an entire fuel distribution system would have to be developed. Also, refueling systems would have to be converted to deliver the cryogenic hydrogen. As the X-33 project demonstrates, the critical LH_2 tanks are much more complicated and fragile than current fuel tanks, and are much more difficult to inspect and repair. Perhaps the most inhibiting consequence of using hydrogen is the large storage volume required to achieve the same mission performance as with conventional fuels. The costs of converting to LH2 fuel, and the comparison between operating with LH2 as opposed to JP-5, have not been determined at this time.

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I. INTRODUCTION

A. NEED FOR ALTERNATIVE POWER SOURCES

1. Sociological Considerations

This nation and much of the world depends heavily on petroleum to fuel its economy and its military. Those who control the petroleum supplies, its distribution and its cost, significantly influence the world economy. Throughout the past century, control of petroleum supplies has quickly emerged as one of the United States' critical vulnerabilities. Much of the world's oil is concentrated in Middle Eastern countries, which collaborated to form OPEC in 1973. This vulnerability has become evident, from the 1973 Arab Oil Embargo to the Gulf War of 1991 into today. As our petroleum dependence has become a critical national vulnerability, alternative power sources have been considered to protect our current way of life. This document discusses the feasibility of hydrogen as an emerging petroleum-replacement fuel, specifically in Naval aviation applications.

2. Ecological Considerations

Besides availability and cost, perhaps the largest argument for an alternate fuel source centers on the environment. The combustion of petroleum in motor vehicles produces carbon dioxide, carbon monoxide, and nitrous oxide gases that pollute the atmosphere and lead to unnatural climate changes, especially in heavily populated areas. Also, the need for petroleum has forced issues on drilling for oil in lands presently protected from development, such as those in Alaska. Not only can the presence of modern man disrupt an area’s wildlife, but also the oil itself is harmful to wildlife if it should spill. Most proponents for alternate fuels are searching for a “green fuel,” one that would produce fewer byproducts harmful to the environment.

B. VEHICLE TYPE COMPARISON

1. Automobiles

The availability and cost of oil most visibly affects the individual American through his personal vehicles. Because the nation relies on cars, trucks, and vans for transportation, the cost of fuel is an issue for everyone who stops at a gas station or shops at a store. Important considerations besides cost that go into determining the practicality of an alternative fuel for automobiles are ease of refueling and fuel tank size. Even if the cost were less, few people would accept an alternate fuel if it meant giving up a significant amount of passenger or storage space in their vehicle. Also, refueling would have to be about as simple as it is now with gasoline, so that the average driver would not spend extra time and effort at the refueling station.

Hydrogen internal combustion engines resemble those using more familiar petroleum fuels. When hydrogen is used in a standard gasoline internal combustion engine in stoichiometric proportions, the hydrogen takes up about 30% of the volume in the cylinder, compared to 2% for gasoline vapor, and produces about 20% less output power. But hydrogen is flammable over a wide range of concentrations and can be burned lean, thereby increasing the energy efficiency of the engine and reducing the flame temperature. But with direct injection, hydrogen engines can be run at higher compression ratios than gasoline engines, increasing engine efficiency. Overall, well-designed hydrogen engines are estimated to have 20-25% higher energy efficiency than comparable gasoline engines, while eliminating all pollutant emissions except for low levels of nitrogen oxides (NO_x). Although small in comparison to water vapor and nitrogen products, NO_x emissions result due to the higher temperatures of the combustion of hydrogen than the combustion temperatures using hydrocarbons. Practical hydrogen internal combustion engines can achieve efficiencies of about 45%.

2. Aircraft

Cost and refueling complexity are factors that go into determining the practicality of alternative fuels for aircraft as well. However, because aircraft are not owned and operated by individuals on the scale of the automobile, these factors are perhaps less of an issue with aircraft. The most significant factor that separates aircraft from automobiles in this design study is weight. Because an aircraft must lift its weight, it is not beneficial for an alternative fuel system to weigh much more than current fuel systems, as the range of the aircraft or its mission should remain essentially unchanged. For hypersonic airbreathing engines, such as supersonic ramjet (or scramjet) engines, a lean burning fuel such as hydrogen is optimal.

3. Spacecraft/Rockets

Since spacecraft are operated much less than aircraft, the tankage, cost of fuel, and ease of refueling are even less significant issues with spacecraft than with aircraft. The need to lift its own weight makes fuel weight an issue as it is with aircraft. But because of the intense amount of thrust needed to lift the craft and its payload out of the earth’s atmosphere, hydrogen has been, and will likely remain, the best chemical fuel for rocket engines. Hydrogen combined with fluorine is the most energetic combination of two chemicals theoretically possible for a rocket propellant system. This combination in a rocket engine operating at a chamber pressure of 1,000psi produces a specific impulse of 410 seconds. Fluorine, however, is extremely difficult to handle, very expensive, and very toxic, properties that have thus far prevented, and probably always will prevent, its use in a an operational rocket system. Fortunately, the combination of hydrogen with oxygen can produce a specific impulse of 390 seconds, only 5% less than that of fluorine and hydrogen. Oxygen is relatively inexpensive, easy to handle, and nontoxic, and its combination with hydrogen will continue to be the propellant of choice for rocket systems.[1] Because spacecraft must operate their engines at extremely high altitudes, they cannot rely on oxygen in the atmosphere to react with the fuel as aircraft and automobiles do. Consequently, rockets must always carry a supply of oxygen onboard, which adds to the size and weight consideration of the propellant.

[1] Williams, *Hydrogen Power*, pg 45.

II. ALTERNATIVE FUEL CANDIDATES COMPARED WITH CONVENTIONAL AIRCRAFT FUEL

A. JP-5

Since this study is to look at replacing current aviation fuel, it is necessary to first understand the characteristics of current fuels in order to compare potential alternatives. JP-5 is the aviation grade of kerosene that is currently used in military aircraft such as the Lockheed-Martin P-3 Orion. With a boiling point of about 332^{o}F at a pressure of 1 atm, and a freezing point of -41^{o}F, JP-5 exists as a liquid for almost all handling and storing scenarios. It has a liquid density of 51.1 lb_m/ft^3, a specific gravity of 0.819, and a heat of combustion of 18,456 Btu/lb_m. It combusts with air according to the following equation, not including secondary reactions:

$$C_{10}H_{19.185} + 14.80(O_2 + 3.76N_2) \Rightarrow 10CO_2 + 9.59H_2O + 55.61N_2$$

B. GASEOUS HYDROGEN

The combustion equation for hydrogen with air is as follows, excluding secondary reactions:

$$H_2 + 0.5(O_2 + 3.76N_2) \Rightarrow H_2O + 1.88N_2$$

As is evident by this equation, the combustion of hydrogen produces no carbon oxides, unlike current hydrocarbon fuels. Gaseous hydrogen, stored at a temperature of 60^{o}F and a pressure of 2400 psi has a heat of combustion of 51,590 Btu/lb_m, a value more than twice that of JP-5. This reduces the overall fuel weight by a factor of about 2.8. Also, this higher heat of combustion means that the engines and thus the overall aircraft would operate quieter than with

JP-5. Gaseous hydrogen has a specific heat of 2.32 Btu/lb_m-^oF, which could allow the fuel itself to be used to cool the engine and vehicle hot parts. More importantly, having a higher specific heat than JP-5 allows for a higher turbine inlet temperature and overall pressure ratio, which equates to a further reduction in specific fuel consumption and further weight savings. The energy required for hydrogen compression, while significant, is much less than for liquefaction.[2] And hydrogen, unlike petroleum, can be globally accessible.

However, gaseous hydrogen has a density of only 0.787 lb_m/ft^3, and likewise a specific gravity of 0.013. The fuel load for an aircraft would thus require about 23 times as much volume if hydrogen were carried in gaseous form, even at this high pressure. This volume problem, combined with consideration of the obvious weight of the containers and of the safety problems associated with the high-pressure storage, eliminates gaseous hydrogen as a viable candidate for aircraft applications.[3] Onboard storage systems for compressed hydrogen are bulkier and heavier than those for liquid fuels or compressed natural gas.

C. LIQUID HYDROGEN (LH_2)

Liquid hydrogen, or LH_2, requires 5.6 times less volume than gaseous hydrogen. LH_2 must be heated before entering the engine to gain the same benefits of using gaseous hydrogen while significantly reducing the fuel storage space required.

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[2] Ogden, “Hydrogen: The Fuel of the Future?” pg 69-75.

[3] Brewer, *Hydrogen Aircraft Technology*, pg 151.

Liquid hydrogen still has an extremely low density compared to JP-5, only 4.43 lb_m/ft^3 at its normal boiling point, or alternatively, LH_2 has a specific gravity of 0.071. Despite the benefits from its other properties, liquid hydrogen storage requires about 4.15 times more volume than JP-5. This may lead to a lower lift to drag ratio and a lower wing loading at takeoff in a new design. As it has a boiling point of -423^{o}F at 1 atm, hydrogen in its liquid form is most definitely cryogenic, a property that makes its handling more complicated. As a result, using liquid hydrogen requires a large, heavy tank and fueling system, special tank fill and vent procedures, and a constant tank pressure to minimize boil-off.[4] Hydrogen's extremely low boiling point combined with a heat of vaporization of only 192 Btu/lb_m contribute to a rapid rate of boiling from the smallest heat leakage into the storage vessel. Thus, an airtight insulation system is a major consideration when storing hydrogen as a liquid.[5]

D. HYDRIDES

A newer method of storing hydrogen is to trap it in a metal hydride, which is an inter-metallic compound that soaks up hydrogen like a sponge. Hydrides require moderate pressures but are currently expensive, need to operate at high temperature to store a lot of hydrogen, and are typically very heavy. One other drawback is they must have only very pure hydrogen supplied or they will get contaminated and stop operating properly. From a safety standpoint, hydrides are intrinsically safe, as the hydrogen must be released from the hydride before it can

[4] Brewer, pg 11, 14.

[5] Williams, pg 98.

oxidize or burn rapidly. Metal hydrides offer the possibility of storing hydrogen compactly and safely, but their excessive weight is an overwhelming problem. Typically a metal hydride can currently store hydrogen only between 2% and 4% by weight. Thus, to store 1000lb of hydrogen, at least 25,000lb of hydride would be required. Although carbon nanofibre technology may have the capacity to store up to 70% of hydrogen by weight, this technology is currently still in the laboratory stages.[6] Because of their safety and density advantages, hydrides may eventually be adopted as hydrogen storage systems for nonweight-sensitive vehicles, but they are not viewed as practical candidates for aircraft.[7]

E. LIQUID METHANE (LCH_4)

Having a heat of combustion of 21,500 Btu/lb_m, liquid methane is 15% more energetic than JP-5. However, it is only little more than half as dense, having a liquid density of 26.4 lb_m/ft^3 and a specific gravity of 0.423. Compared to hydrogen, it is only 41% as energetic but is six times as dense. Its boiling point of -258^{o}F classifies liquid methane as a mild cryogen, because this temperature is high enough that oxygen in the air will not liquefy upon contact with an uninsulated portion of the fuel system as it would with liquid hydrogen. Methane combusts with air according to the following equation, excluding secondary reactions:

$$CH_4 + 2(O_2 + 3.76N_2) \Rightarrow CO_2 + 2H_2O + 7.56N_2$$

[6] Hart, “Hydrogen storage and transportation technology.”

[7] Brewer, pg 151.

Since methane could be injected into the combustor as a saturated liquid at elevated temperature, it can be expected to mix and burn more evenly than kerosene, thus producing somewhat less NO_x than kerosene.[8]

F. AMMONIA

Ammonia, or NH_3, is another compound with some potential to replace petroleum as a fuel. It combusts with air according to the following equation, excluding secondary reactions:

$$NH_3 + 0.75(O_2 + 3.76N_2) \Rightarrow 1.5H_2O + 3.32N_2$$

It has a density of 6.96 lb_m/ft^3 and a specific gravity of 0.112. Three gallons of ammonia is equivalent to one gallon of gasoline in energy content, or in other terms, 2.35lb_m of ammonia is equivalent to one pound of gasoline in energy content. Ammonia is essentially nonflammable and is readily obtained and handled in liquid form without the need for expensive and complicated refrigeration technology. In addition, ammonia contains 1.7 times as much hydrogen as liquid hydrogen for a given volume. Therefore liquid anhydrous ammonia is an excellent storage medium for hydrogen, even though the endothermic ammonia cracking results in some efficiency penalty.[9]

Ammonia contains no carbon, and can be easily made from hydrogen or natural gas. Anhydrous ammonia is stored in the same manner as propane, as a liquid under approximately 100-psi vapor pressure at room temperature. If released into the atmosphere, ammonia's density is lighter than that of air and thus dissipates rapidly. In

[8] Brewer, pg 11, 13-14.

[9] Faleschini, "Ammonia for High Density Storage."

addition, because of its characteristic smell the nose easily detects it in concentrations as low as 5 ppm. Finally, ammonia has such a narrow flammability range that it is generally considered non-flammable when transported.

Ammonia is widely available, making it much easier to obtain than petroleum fuels. In the United States alone, 35 trillion pounds of anhydrous ammonia are produced per year. Ammonia is produced and distributed worldwide in millions of tons per year. Approximately one million US farms have access to anhydrous ammonia. The distribution infrastructure already exists to deliver approximately 8 billion pounds of anhydrous ammonia to these farms for direct use as a nitrogen soil supplement. Farmers are already handling anhydrous ammonia and could easily use it as a fuel for their farm equipment if an efficient utilization engine were available.

The use of ammonia also has the potential to reduce unwanted emissions from combustion. After combustion, any generated NO_x emissions can be readily reduced by reaction with ammonia over a zeolite according to one of the following two reactions:

$$4NO + 4NH_3 + O_2 \Rightarrow 4N_2 + 6H_2O$$
$$6NO_2 + 8NH_3 \Rightarrow 7N_2 + 12H_2O$$

Measurement of the ammonia and NO_x emissions from typical operating conditions has shown approximately equal quantities (400ppm) of both ammonia and NO_x. Thus, ammonia addition to the exhaust stream may not be required.[10]

Despite all these benefits to using ammonia as a replacement for petroleum, ammonia’s largest failing is

[10] Blarigan, “Advanced Internal Combustion Electrical Generator.”

that it has a heat of combustion of only 7992 Btu/lb_m. This value is so low that the weight of fuel required to enable an airplane to fly a given mission would be up to 2.5 times as much as with conventional aircraft fuel, leading to vehicles of enormous size and limited capability.[11]

G. FUEL-CANDIDATE SUMMARY

Because of its high heat of combustion, along with its high specific heat, hydrogen is a favorable alternative to conventional aircraft fuels. Also, hydrogen fuel usage would lead to a cleaner environment, improved safety, improved aircraft performance, less energy required from resources to manufacture the fuel, lower direct operating cost, universal availability, and a favorable economic impact.[12] Although liquid hydrogen would require more volume than current fuels, it would use significantly less volume than in its gaseous form, and less weight than hydrides. Thus, as the most feasible alternative aircraft fuel, LH_2's cryogenic properties must be carefully considered in any system design considerations.

[11] Brewer, pg 13.

[12] Brewer, pg 400-402.

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III. PRESENT USES OF HYDROGEN AS A FUEL IN AEROSPACE APPLICATIONS

A. ROCKETS

The United States’ spacecraft initially evolved out of military missiles. The Mercury Redstone was used to launch America’s first satellite into space in 1958. The Mercury Redstone rockets used liquid oxygen, ethyl alcohol, and water as propellants. It wasn’t until the use of the Saturn V rocket used in the Apollo program for lunar flights that LH_2 was used in conjunction with liquid oxygen in American spacecraft.

B. THE SPACE SHUTTLE

The Space Shuttle, currently NASA’s main vessel for space travel, saw its first launch into space with the orbiter Columbia in April 1981. Hydrogen and oxygen are the main engine propellants in the Space Shuttle system. The Space Shuttle has two large, solid rockets that are part of the initial lift-off propulsion, but the main engines are fueled with liquid hydrogen and the oxidizer is liquid oxygen. A schematic of the shuttle is shown in Figure 1. Underneath the orbiter is a very large external tank designed to hold the liquid hydrogen and oxygen in separate compartments. Large feedlines run from these tanks into the orbiter, where they are attached to the engines. The orbiter also has small inside tanks of hydrogen and oxygen for the final insertion into orbit.[13] The external tank is 154ft long and has a diameter of 27.6ft. When the Space Shuttle main engines are done and the fuel depleted, the

[13] Williams, pg 46.

external tank is jettisoned, enters the Earth’s atmosphere, breaks up, and impacts in a remote ocean area. It is not recovered.[14] Currently, alternative ideas for external tanks for use as platforms for commercial space endeavors are being considered.[15]

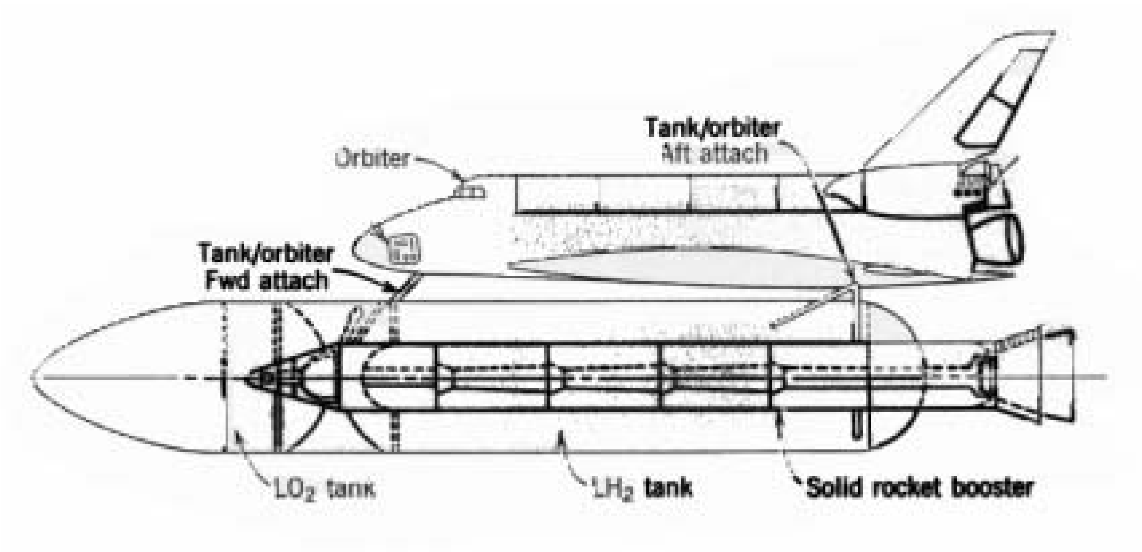

Figure 1. NASA Space Shuttle Vehicle

C. THE X-33 REUSABLE LAUNCH VEHICLE (RLV)

The X-33 was a half-scale prototype of a reusable spacecraft that NASA and its prime contractor Lockheed-Martin teamed up to develop. The unpiloted X-33 technology flight demonstration vehicle was designed to cut the cost of going into space. The 69ft-long, wedge shaped prototype was to be launched upright like a rocket and fly back to Earth like an airplane. Had the prototype been successful, a full-scale version called Venture Star would have been developed to possibly replace the Space Shuttle fleet.[16] The

[14] NASA, “Shuttle Reference and Data.”

[15] David, *Space News*.

[16] Bull, *Aerotech News and Review*.

sub-orbital X-33 was designed to demonstrate advanced technologies that would dramatically increase launch vehicle reliability and lower the cost of launching payloads to low Earth orbit from $10,000 to $1,000/lb$_m$.[17]

This single-stage rocket concept began in the early 1990's. Along with Lockheed Martin, Boeing/McDonnell-Douglas and Rockwell International vied for the lead design role. In 1996 NASA chose Lockheed-Martin to test the feasibility of replacing the space shuttle with a fully, reusable rocket by 2012. Along with NASA's funding, industry officials planned to make a significant investment of their own with hopes that dramatically lower launch costs will open up new markets, including space tourism.[18]

But a composite fuel tank, the first of two LH_2 flight tanks for the X-33, structurally failed after a series of tests Nov. 3, 1999, at NASA's Marshall Space Flight Center in Huntsville, AL. An investigation team found that the unexpected severity of a condition called microcracking was instrumental in the failure of the tank's composite skin, a small portion of which split following the tests. After the tank failure, work on the X-33 continued. NASA and Lockheed-Martin proceeded with design of aluminum LH_2 tanks for the X-33, replacing the experimental composite tanks originally planned. Composite hydrogen fuel tanks would have reduced the weight of the craft, which would be vital in a single-stage craft attempting to reach Earth orbit.

[17] NASA Spacelink, 2000.

[18] Iannotta, *Space News*, 1995.

The project continued through March 2001, but was thereafter dropped due to lack of funding and was never completed.[19]

[19] NASA Spacelink, 2000.

IV. SUPPORT EQUIPMENT FOR LIQUID HYDROGEN ENGINES

A. FUEL TANK

Due to the complex nature of LH_2 storage, the tank is one of the most crucial technical challenges confronting use of LH_2 in operational aircraft. The development and fabrication of reusable cryogenic tanks has been a significant technical barrier to overcome in the development of an operable RLV.[20] The cancellation of the X-33 due to the failure in its LH_2-tank design is evidence of this statement. For a reference, the liquid hydrogen tank in the Space Shuttle is an aluminum semimonocoque structure of fusion-welded barrel sections, with an operating pressure of 32 to 34 psia. The LH_2 tank is 27.6ft in diameter, 97ft long, and has a volume of 53,518ft^3 and a dry weight of 29,000lb.[21] In his text *Hydrogen Aircraft Technology*, Brewer conducts a detailed study of tank concepts for liquid hydrogen tanks for use in aircraft. Materials used for tank construction must be resistant to hydrogen embrittlement, impermeable to gaseous hydrogen, and capable of retaining satisfactory ductility and fracture resistance at cryogenic temperatures. In Brewer's design study, combinations of nonintegral and integral tankage, together with both internal and external insulation, were the candidate design possibilities for fuel containment systems. Brewer chooses external insulation for his preliminary studies because of the difficulty of meeting the requirement for a liner

[20] Johnson, et al, "Thermal Structures Technology Development For Reusable Launch Vehicle Cryogenic Propellant Tanks."

[21] NASA, "Liquid Hydrogen Tank."

impervious to gaseous hydrogen if internal insulation is used. Nonintegral tanks are those that are designed to take only loads associated with containment of the fuel. They are supported within conventional fuselage skin/stringer/frame structure. On the other hand, integral tanks are an integral part of the basic aircraft structure. In addition with the loads taken by nonintegral tanks, integral tanks must also be capable of withstanding all the usual fuselage stresses resulting from the critical aircraft loading conditions. Comparison of nonintegral and integral tank concepts has led Brewer to conclude that the potential of the integral tank concept is superior to that of the nonintegral. This is due to the integral tank concept having a greater structural efficiency and a higher volumetric efficiency, and because the integral design is more readily accessible for inspection and repair.[22]

Three sources have been used to determine the best candidate material for the LH_2-fuel tanks. Brewer has adopted an all-metal, aluminum alloy (2219) design over stainless steel options, noting the possible benefits of using other aluminum alloys, such as 2021 aluminum, as well. He considers four types of wall concepts, namely, blade-stiffened, zee-stiffened, and tee-stiffened, and unstiffened designs. The design found to be preferred for the integral tank concept used zee stiffeners in the tank in the upper and lower quadrants where bending stresses must be resisted, and no stiffeners in the side quadrants. A frame spacing of 50 in. has been found to be optimum.[23] A more contemporary source being used to determine tank

[22] Brewer, pg 26-30.

[23] Brewer, pg 158-161.

materials is “Thermal Structures Technology Development for Reusable Launch Vehicle Cryogenic Propellant Tanks.”[24] This 1998 report describes studies conducted at the NASA Langley Research Center for investigating integrated cryogenic propellant tank systems for an RLV. This report states that the cryogenic tanks of an RLV must not only function as pressure vessels at cryogenic temperatures, but that they also must carry primary structural loads and support the Thermal Protection System (TPS). Although that study was for a vehicle that would be exposed to much higher temperatures and stresses than an aircraft, its use of advanced materials and concepts prove pertinent to the design of any LH_2 fuel tank. This report not only considers external foam concepts similar to Brewer’s design, but also considers honeycomb sandwich cryo-insulation concepts. The results of the analytical studies identify a honeycomb sandwich tank with mechanically attached metallic TPS as a possible approach for a reusable LH_2 tank system for an RLV. The two most attractive honeycomb sandwich concepts have been found to be IM7/5260 Graphite-Bismaleimide (Gr-BMI) facesheets with Hexcel™ glass Reinforced Phenolic (HRP) core, and titanium facesheets and titanium honeycomb core.[25] Table 1 lists combined aerial masses of both materials. TPS concepts will be discussed in more detail in the next section. They are included here when discussing the RLV tank concepts due to the combinational nature of these concepts.

[24] Johnson, et al.

[25] Ibid.

Table 1. Combined Aerial Masses of LH_2 Tank Concepts

Tank/TPS	Cryogenic Insulation & TPS Aerial Mass (kg/m^2)	Tank Structural Mass (kg/m^2)	Total Aerial Mass (kg/m^2)
(Gr-BMI/HRP/Gr-BMI)/(SA/HC)	10	6.0	16.0
(Ti/Ti/Ti)/(SA/HC)	8.9	8.9	17.8

The final report of the X-33 LH_2 tank test investigation team details the design of the tank. Although it failed during testing, its design was still considered for this report. The tank design was highly innovative, pushing the limits of technology and combining many unproven technology elements. The interaction and integration of these elements created a highly complex system, both technically and programmatically.[26] Through the lessons learned from the X-33 LH_2 composite tank, it may be feasible to build a working tank, should the concept be properly desired and funded. Each LH_2 tank for the X-33 was a multi-lobe IM7/977-2 graphite/epoxy (Gr/Ep) tank with integrally bonded, woven composite joints. The bulkheads were a sandwich panel construction of Gr/Ep tape facesheets and woven graphite core (Ultracore).[27]

B. INSULATION

The LH_2 tanks used in the upper stages of the Saturn V launch vehicle and the drop tank used on the Space Shuttle use a 4 to 6in layer of plastic foam insulation. These applications do not require very high quality insulation.

[26] Goetz, "Final Report of the X-33 Liquid Hydrogen Tank Test Investigation Team," pg 7-8.

[27] Goetz, "Final Report of the X-33 Liquid Hydrogen Tank Test Investigation Team," pg 9.

The vehicle is placed on the pad, and the LH_2 tank is filled only an hour or so before lift-off. During the time it sits on the pad before actual lift-off, the LH_2 tank is continuously vented and topped-off to maintain the proper fuel load. During the flight, the hydrogen is used so rapidly (completely used in 4 to 5 minutes) that there is essentially no effect from the slight amount of boil-off that occurs. Plastic foam insulation is also considered to be suitable for use in hydrogen-fueled airplanes, for much the same reason that it is adequate for space launch vehicles. An LH_2-fueled airplane would be fueled just before takeoff, and the boil-off rate could be controlled with insulation so that the fuel use rate exceeds boil-off. Therefore, in addition to withdrawing the boil-off vapor for fueling the engines, liquid will also have to be withdrawn and vaporized in a heat exchanger to provide the fuel required to power the airplane.[28] The heat exchanger must be designed to operate in the coldest environment the aircraft will encounter.

The Space Shuttle external tank TPS consists of sprayed-on foam insulation and premolded ablator materials. The system also includes the use of phenolic thermal insulators to preclude air liquefaction.[29] In Brewer's text, he describes a comprehensive study conducted by Lockheed of insulation systems for LH_2 tanks for aircraft. Fifteen candidate insulation systems incorporating nonintegral and integral tank designs were studied. Out of those suited for integral tanks, two candidates were preferred and subjected to a more detailed analysis, one using rigid, closed-cell

[28] Williams, pg 98-99.

[29] NASA, "Shuttle Reference and Data."

polyurethane foam and the other using microspheres. The selected thickness for each concept was 3.6in and 2.4in, respectively, keeping in mind that Brewer's design was optimized for a 400-passenger, 5500-nmi-range subsonic transport aircraft. On the basis of gross weight, fuel weight, operating empty weight, fuselage length, engine size, aircraft price, direct operating cost, and energy utilization, the latter was considered the best choice.[30] It should be noted that a preliminary assessment was made in Brewer's text of three refrigeration systems to determine the potential of this approach. Refrigeration systems would be the alternative to accepting boil-off of LH_2 as the penalty for the heat allowed by the cryogenic insulation system to leak into the fuel. However, for all of these systems the weights associated with the refrigeration devices, shields, and plumbing lines exceeded the weight of fuel saved by a minimum of 2200lb per tank, relative to a passive insulation system.[31]

In the 1998 NASA report on RLV cryogenic propellant tank development, both external foam concepts and honeycomb cryogenic insulation concepts were considered. External concepts consisted of adhesively bonded Rohacell™ as the cryogenic foam insulation and either Alumina Enhanced Thermal Barrier (AETB) or Tailorable Advanced Blanket Insulation (TABI) as the TPS. Honeycomb concepts consist of a sandwich tank wall with an evacuated core for insulation and Superalloy/Honeycomb (SA/HC) metallic panels as TPS. As

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[30] Brewer, pg 168-207.

[31] Brewer, pg 168.

honeycomb concepts were preferred in the study, SA/HC metallic panels were chosen as the TPS.[32]

C. FUEL SYSTEMS AND COMPONENTS

The flow path of LH_2 begins in the fuel containment system, traveling through the boost pumps, tank shutoff valve, and fuel supply lines to the high-pressure pump mounted on the engine. From there it goes through the three cowl-mounted heat exchangers before passing to the fourth heat exchanger, the exhaust fuel heater mounted in the core exhaust. The fuel then passes through the engine flow-control valve before it is injected into the combustor.[33] A diagram from Brewer’s text of the fuel system of a theoretical subsonic transport aircraft fueled with LH_2 is included as Figure 2. Brewer details the design of a boost pump, a high-pressure pump, and engine fuel delivery lines in his text. Of note for this design study is that an LH_2-fueled airplane should have multiple boost pumps per engine in order that failure of a single pump will not compromise aircraft safety. Aircraft LH_2 tanks should thus be divided into compartments to make a separate compartment to feed fuel to a particular engine. The fuel delivery lines are constructed from Type 321 stainless steel, and are optimized to a 1.0-in inner diameter and a wall thickness of 0.016-in. Of the two types of insulation systems evaluated, namely, vacuum-jacket (VJ) and rigid, closed-cell foam, the foam insulation system was chosen. Despite a small weight and operational advantage for the VJ system, lower susceptibility to incapacitating damage, lower

[32] Johnson, et al.

manufacture and maintenance costs, and easier, quicker repair capability are the overwhelming benefits of the foam system. The foam insulation is 1.5-in thick. The outer tube is made of 6061 aluminum alloy, and has a 4.0-in inner diameter and a wall thickness of 0.016-in.[34]

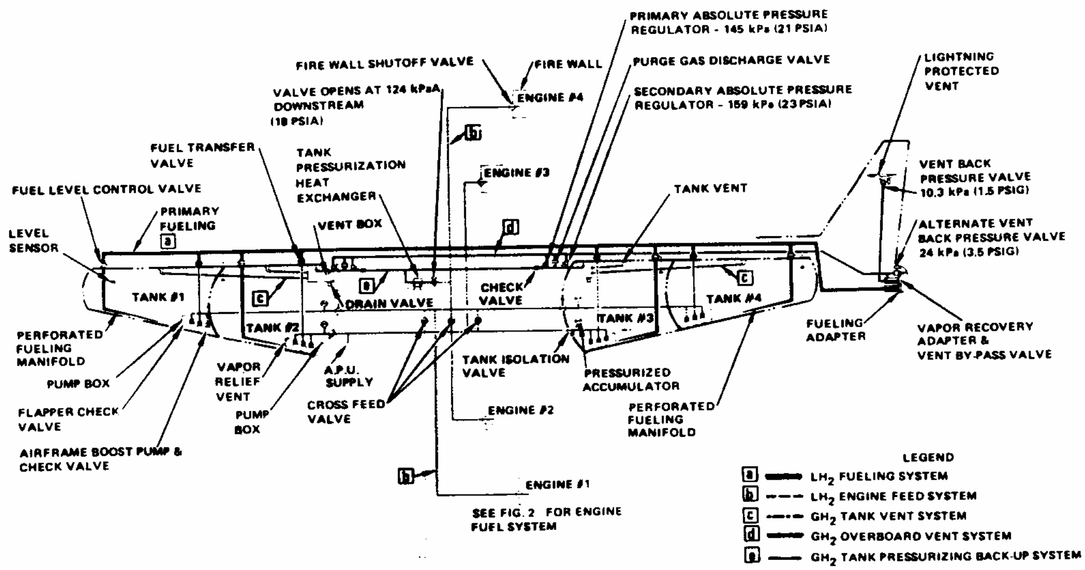

Figure 2. Sample Aircraft LH_2 Fuel System Schematic, Brewer, *Hydrogen Aircraft Technology*, pg 106.

[33] Brewer, pg 97-98.

[34] Brewer, pg 131.

V. DESIGN STUDY: THE LOCKHEED MARTIN P-3 ORION FUELED WITH HYDROGEN

A. CHOICE OF THE P-3 VS. OTHER NAVAL AIRCRAFT

To determine the feasibility of using hydrogen as an aircraft fuel, a design study was conducted on the Lockheed Martin P-3 Orion, the US Navy's antisubmarine aircraft. A table of P-3 specifications used in this design is listed in Table 2. The P-3 was selected for this design study for a number of reasons, although its size was the deciding factor. The P-3 is unique among naval aircraft in that it has a large, cylindrical body similar to a cargo plane, but carries only electronic surveillance equipment. As the LH_2 tanks would be four times larger than current fuel tanks, any large presently unused internal area in the P-3 would be critical to a successful design. Another important factor in choosing the P-3 is that it is land-based, eliminating the need for LH_2 to be stored onboard a ship. Also, the P-3 is based at only a few locations, thereby minimizing the number of bases that would have to be provided with LH_2 manufacturing, storage, and handling facilities. Initially, these aircraft would have to operate entirely from their assigned bases, to eliminate the problem of having to refuel a hydrogen-fueled airplane at an airfield not yet equipped with LH_2 facilities.

Table 2. P-3 Specifications

Take-off weight (lb)	139,760
Fuel weight (lb)	59,530
Payload (lb)	8,279
Cruising altitude (ft)	20,100
Cruising speed (kt)	320
Combat radius (nmi)	1,225
Mission time (hr)	11.8
Weapons load	2 MK-46 / 4 AGM-84A

B. ENGINE

The engine currently used in the P-3 is the Rolls-Royce Allison T-56-A-14 turboprop. The P-3 uses four of these engines, each driving a Hamilton Standard 54H60-77 four-blade constant-speed propeller. To minimize differences between the current JP-fueled P-3 and an LH_2-fueled variant, the same engine is envisioned in this design study but modified to operate with liquid hydrogen. Modeling the engine is necessary to determine its net thrust and its specific fuel consumption, values that would determine the amount of fuel required for the given mission. This engine has been modeled using GasTurb computer software (see Appendix). To ensure that GasTurb would adequately model the modified engine, the engine was first modeled with JP-5 fuel using data on the engine from both *Jane's All The World's Aircraft* and Rolls-Royce's internet fact sheet. The T-56 data used is listed in Table 3.

Table 3. Rolls-Royce Allison T-56-A-14 Turboprop Data

Propeller	
Diameter (ft)	13.48
Overall gear ratio	13.54
Power section rpm	13,820
Compressor	
Pressure ratio	9.5
Mass flow (lb_m/s)	32.35
Turbine	
T-O gas temperature (^oF)	1,970
Performance	
T-O Power (SLS) (shp)	4,591
SFC--max rating (lb/h/shp)	0.468

The goal in modeling the JP-fueled engine is for GasTurb to output the two performance parameters, namely, power and specific fuel consumption (SFC) as given in Table 3. The engine data is input into GasTurb without the propeller to model a turboshaft engine, and was run at standard-day sea-level (SLS) conditions. Since values for the compressor and turbine efficiencies are not found in source material, they were modified in the program so the performance parameters output by GasTurb most closely matched those listed in Table 3. The compressor and turbine efficiencies were set to 0.875 and 0.9, respectively. The output performance parameters using GasTurb are listed in Table 4.

Table 4. GasTurb Performance Values (SLS) Modeling Allison T-56-14 Engine

Power (shp)	4,326
SFC (lb/h/shp)	0.467

Using GasTurb, the Allison engine was successfully modeled with only a 6.1% error in shaft horsepower and a 0.3% error in SFC. To model the engine for use with liquid hydrogen, only the fuel heating value (or heat of combustion) was changed in GasTurb to reflect the change in fuel. The performance data for the engine obtained using GasTurb is listed in Table 5.

Table 5. GasTurb Performance Values (SLS) Modeling LH_2-Fueled T-56 Engine

Power (shp)	4,614
SFC (lb/h/shp)	0.159

Once the engine is modified to operate using LH_2, GasTurb is again modified, this time off-design, to operate with the propeller as a turboprop at cruise speed and altitude during a hot day. The values for net thrust and specific fuel consumption used in this design study are obtained from this run of GasTurb and are listed in Table 6.

Table 6. GasTurb Performance Values (Cruise) Modeling LH_2-Fueled T-56 Engine

Net Thrust (lb_f)	2,380
SFC (lb/h/shp)	0.1702

It should be noted that the LH_2-fueled engine differs in thrust from the JP-fueled engine only by an added 7%. The two values listed in Table 5, along with the density of liquid hydrogen, were used to determine the volume of fuel required for the four combined engines. The total amount of LH_2 required for a P-3 to operate the estimated 13.5 hours is 4,316ft^3, or about four times as much volume as is required by JP-5. However, due to the low density of LH_2, the mission fuel would only weigh 22,000lb, almost one-third that of the amount of JP-5 required for the same mission.

C. TANK PROPERTIES

The total volume for the LH_2 tank is calculated using the weight of the LH_2 required, the density of LH_2 at 21 psia (= 4.326 lb/ft^3), and a 7.2% sizing allowance.[35] For this design, the total tank volume is 5,421ft^3. The Ti/Ti/Ti-sandwich tank with metallic thermal protection system integrated tank system concept listed in Section IV.A was chosen for the tank in this design. A diagram of the tank wall is included as Figure . Using the tank size given in next section and the tank properties from Section IV.A, the tank would weigh approximately 14,865lb.

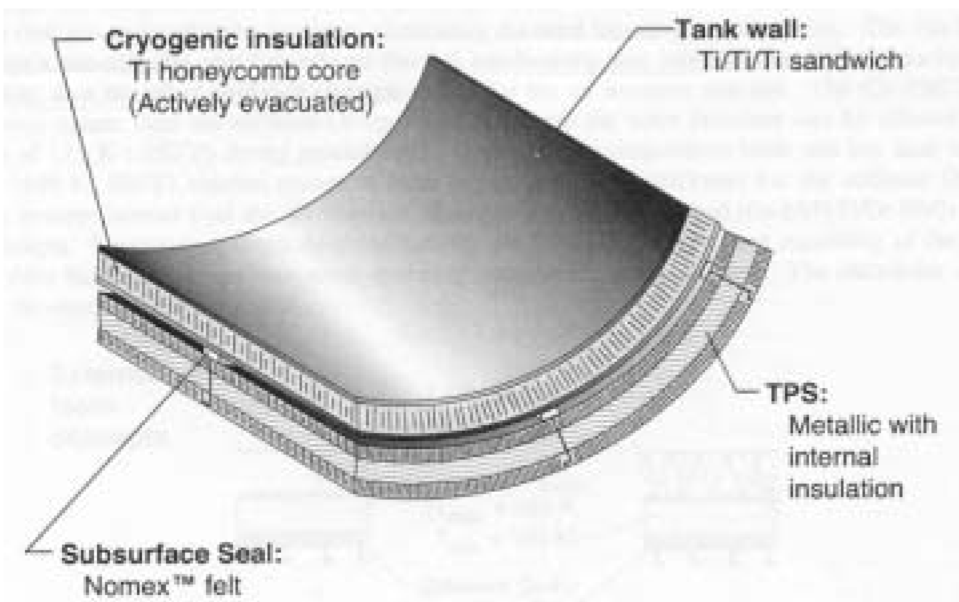

Figure 3. Proposed LH_2 fuel tank wall cutaway

D. FUEL STORAGE

The optimum shape of a propellant tank is spherical, because for a given volume it results in a tank with the least weight. A spherical tank also would provide for the lowest surface to volume ratio, thus minimizing the amount

[35] Brewer, pg 154.

of heat transfer between the tank's internal and external environments. Unfortunately, larger spheres, which are needed for the principal propulsion systems, are not very efficient for using the storage space in a vehicle. Most are cylindrical with half ellipses at the ends, but they can be irregular in shape. From the total tank volume determined in the previous section, it is possible to calculate the approximate length of the tank inside the aircraft assuming an integral tank. The tank is sized to fill the entire diameter of the aircraft, allotting a 0.5-ft difference between aircraft external diameter and tank internal diameter for the aircraft bulkheads, the tank insulation, and the tank walls. The majority of the tank was designed as a cylinder, with the 37-ft of tank in the tail accounting for less volume due to the shape of the tail. The total length of the LH_2 tank is thus approximately 86-ft forward from the tail. Assuming a single tank, this number places the tank in 70% of the aircraft internal volume, as shown in Figure 4. Obviously, this would not allow the aircraft to operate its given mission, having limited space (about 20ft behind the cockpit) for the crew to operate the necessary surveillance and other electronic equipment and no weapons bay. Although 6,554 gal of JP-5 is currently stored in the wings as shown in Figure 5, only about half that amount of LH_2 could be kept in the wings due to insulation volume. This would free up another five feet aft of the cockpit for other equipment. While this is an improvement, it still would be difficult to achieve the desired mission.

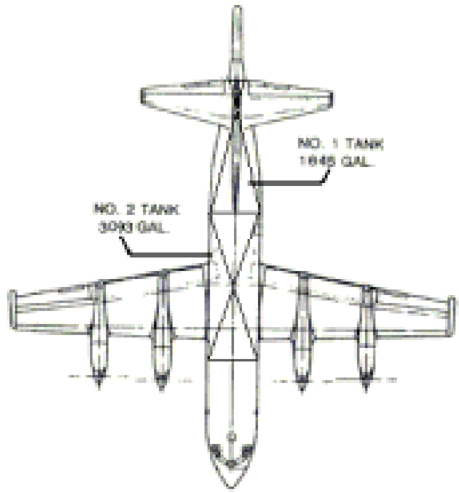

Figure 4. P-3 Orion with liquid hydrogen tanks

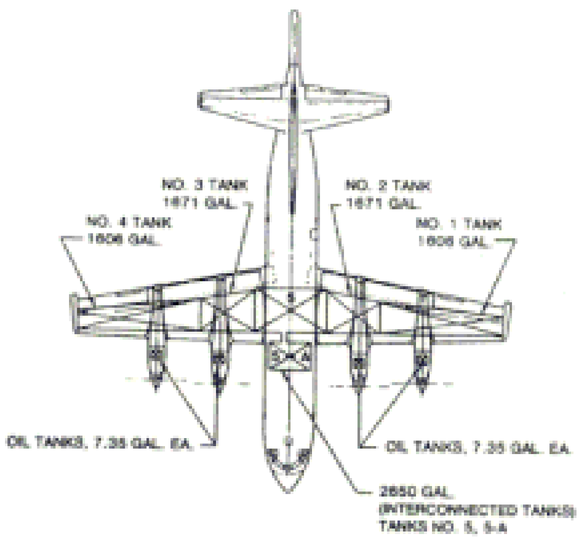

Figure 5. Conventional P-3 Orion

E. FUELING

For simplicity, it can be assumed that gaseous hydrogen is available at the boundary of the airfield. However, this assumption necessitates a liquefaction facility near the airfield. Also, LH_2 storage tanks and transportation methods would be required. Brewer describes three different methods of transporting LH_2 from the liquefier to the storage tanks located at the airport were analyzed, namely, a vacuum-jacketed (VJ) pipeline, truck-trailers, and railroad tank cars. He concludes that transport of LH_2 via the VJ pipeline is the most economical method for distances under 40mi. For distances greater than 40 mi, the railcar transport is the most economical.[36]

Although an airfield's fueling equipment would have to be redesigned to accommodate the LH_2, the fueling process with LH_2 need take no longer than with the equivalent JP-fueling process. However, defueling and refueling the LH_2 aircraft, necessary steps during major aircraft maintenance or fuel tank inspections, are complicated and potentially damaging procedures. It was after draining and purging the X-33 LH_2 tank that the tank structurally failed due to microcracking. If the LH_2 fuel were exposed to the ambient atmosphere, the change in temperature and pressure would cause the fuel to expand and rupture the tank. It is only after draining the LH_2, purging the remaining hydrogen gas with nitrogen, and then flushing the nitrogen from the tank that the tank can be considered completely defueled. Also, fueling an empty tank that has been allowed to warm up must be performed at low rates to avoid overpressurizing the tank.

F. COMPARISON WITH CURRENT P-3

Externally, the LH_2-fueled P-3 looks the same as the current P-3 fueled with JP-5. The main structural design was left unchanged to minimize the difficulty in converting the aircraft to a new fuel system. Internally, LH_2 tanks replace the JP-5 tanks in the wings. This is also done in the main body of the aircraft, which eliminates much of the

[36] Brewer, pg 322.

cabin space. However, removing the JP-5 tanks eliminates the restraint on the level of the cabin floor, as fuel tanks will no longer be in the aircraft's belly. Although due to the weight of the LH_2 tanks the empty weight of the aircraft will be greater than the current P-3, because of the extremely low fuel density the takeoff gross weight of the aircraft will be only 76% of its current counterpart. The net thrust produced by the engines will be about the same as the current aircraft, as will the mission time and range, critical parameters that the LH_2 aircraft was designed to meet.

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VI. CONCLUSIONS

A. DESIGN SUMMARY

Due mainly to its high heat of combustion, global accessibility, and its clean combustion with air, hydrogen is currently the most promising replacement for petroleum to fuel aircraft. However, hydrogen’s low density poses a problem of storage space on aircraft. Carrying hydrogen in its liquefied form decreases the amount of volume necessary to transport the fuel, although liquid hydrogen (LH_2) still necessitates four times the volume than that of current petroleum fuels for the aircraft to accomplish a similar mission. Another benefit of LH_2 as an aviation fuel is its low weight, as the amount of LH_2 necessary to achieve the same mission as with petroleum fuels weighs less than one-third that of current petroleum fuels. Not only does using LH_2 require a modified engine, but also the larger volume needed for and the cryogenic property of LH_2 requires a much different fuel tank design and fueling process than is currently used for petroleum fuels. As the most critical design constraint is the LH_2 tank, current uses and studies were consulted to study the most reliable LH_2 tank design with minimal volume and weight. Currently, the only operable examples of an engine fueled with LH_2 are modern spacecraft, such as NASA’s Space Shuttle. This 30-year-old design uses an aluminum semimonocoque structure for the LH_2 tank. However, current technology may provide alternative, more desirable composite tank designs. A recent attempt at composite LH_2 tank is that designed for the X-33, NASA’s experimental reusable launch vehicle (RLV) that was abandoned in early 2001. The X-33's LH_2 tank bulkheads were a sandwich panel construction of IM7/977-2 graphite/epoxy (Gr/Ep) tape facesheets and a woven graphite core (Ultracore) for insulation. This tank design was finalized after a detailed study, and while it failed, its design is still considered promising.

To determine the feasibility of converting current Navy aircraft to LH2 fuel, a design study was conducted specifically for this report on a converted Lockheed Martin P-3 Orion, the Navy's primary anti-submarine (ASW) aircraft. To make the transition as simple as possible, the outer structure of the aircraft has been left unchanged. The P-3's Allison T-56 engine was modeled using GasTurb computer software, and then modified with the software to operate with LH2 as its fuel. The analysis using GasTurb resulted in an engine with a specific fuel consumption of 0.159 producing 4,614shp. Using GasTurb, an LH_2 tank volume of 4,316ft^3 is arrived at as the requirement for a P-3 to operate as it does currently. The fuel tank is designed using titanium facesheets and titanium honeycomb core, a design similar to that of the X-33's tank. To contain the required volume of LH_2, the tank would weigh about 14,865lb. Despite the large tank weight, due to the low fuel weight the aircraft's takeoff gross weight is only 113,646lb, about 80% of the 140,000lb of the current petroleum-fueled P-3. Even by placing LH_2 in the wings, the fuel tank takes up 65% of the aircraft's internal volume, which means that the crew's operational space would be much less than with the petroleum-fueled aircraft.

B. BENEFITS OF A HYDROGEN-FUELED AIRCRAFT

The most significant benefits to using an aircraft fueled with hydrogen are a favorable economic impact due to its universal availability, and a cleaner environment due to its lack of pollutant byproducts. Because hydrogen can be made from water using any available electrical energy source, as well as being manufactured by conventional processes using the fossil fuels, it can be produced locally almost anywhere in the world. On the other hand, oil is not so universally accessible, and its demand requires the world to be dependant on a few oil-rich countries. Decreasing the amount of oil usage in the United States would decrease dependence on other countries and would support the domestic economy instead of a foreign one. Also, water vapor plus a small quantity of NO_x are the only exhaust products resulting from combustion of hydrogen and air, eliminating the carbon, sulfur oxides, and smoke produced from burning other fuels.

C. OBSTACLES TO A HYDROGEN-FUELED AIRCRAFT

Despite the benefits to using hydrogen as an aircraft fuel, it has yet to be widely implemented due to a number of reasons. Hydrogen production facilities would have to be constructed, along with liquefaction facilities and LH_2 storage facilities. An entire fuel distribution system would have to be created, which would increase in complexity with the distance between liquefaction facilities and refueling stations. Since it is thus optimal to have liquefaction facilities near refueling stations, many such facilities would have to be constructed. Also, refueling systems would have to be converted to deliver the cryogenic hydrogen. As the X-33 project demonstrates, the critical LH_2 tanks are much more complicated and fragile than current fuel tanks, and are much more difficult to inspect and repair. Besides all these factors, perhaps the most inhibiting consequence of using hydrogen is the large tank volume required to achieve the same mission performance as with conventional fuels.

APPENDIX

A. GASTURB

GasTurb is a trademark of Dr. Joachim Kurzke, copyright 2001. It is computer software designed to predict engine performance. It allows the user to select a wide variety of engine types, including turboshaft, turboprop, and turbojet. Once an engine type is selected, the user can select an on- or off-design approach to determine engine performance. The program allows a wide variety of input parameters, both for the engine design and the operating conditions, as shown for a JP-5-fueled engine in Figure 6. The program also has example modules for each engine type, with sample inputs that can be modified according to the user. Once the user is satisfied with the inputs, he can run the program, which will then compute and display the output engine parameters, as shown for a JP-5-fueled engine in Figure 7. The numbered stations indicated in Figure 7 are shown in Figure 8. The program also has other functions, such as performance optimization, which were not utilized for this report.

Figures 9 and 10 are GasTurb printouts of input and output parameters, respectively, for an LH_2-fueled engine. Input parameters were modified to achieve the desired results for both the JP-5-fueled engine and the LH_2-fueled engine. The mass flow rate, pressure ratio, burner exit temperature, and spool speed are properties of the Allison T-56 engine. The fuel heating values reflect the heats of combustion for the specific fuels. The compressor and turbine efficiencies were modified on GasTurb for the JP-5-

fueled engine to obtain similar results to an actual Allison T-56 engine. These efficiencies were unchanged when determining the LH_2-fueled engine's performance. Figure 11 is the GasTurb output of the LH_2-fueled turboprop engine operating at an altitude 20,100ft on a hot day. Figure 12 is a diagram of the numbered stations listed in Figure 11.

Altitude	ft	0
Delta T from ISA	R	0
Relative Humidity [%]		0
Mach Number		0
Basic Data		
Inlet Corr. Flow W2Rstd	lb/s	32.35
Intake Pressure Ratio		0.99
Pressure Ratio		9.5
Burner Exit Temperature	R	2309.67
Burner Design Efficiency		0.9999
Burner Partload Constant		1.6
Fuel Heating Value	BTU/lb	18456
Water-Fuel-Ratio		0
Steam-Fuel-Ratio		0
Overboard Bleed	lb/s	0
Mechanical Efficiency		0.9999
Burner Pressure Ratio		0.97
Turbine Exit Duct Press Ratio		0.98
Design Exhaust Pressure Ratio		1.03
Air System		
Rel. Handling Bleed		0
Rel. Overboard Bleed W_Bld/W2		0.01
Rel. Enthalpy of Overb. Bleed		1
Turbine Cooling Air W_Cl/W2		0.05
NGV Cooling Air W_Cl_NGV/W2		0.05
Comp Efficiency		
Isentr.Compr.Efficiency		0.875
Comp Design		
Nominal Spool Speed		13820
Turb Efficiency		
Isentr.Turbine Efficiency		0.9

Figure 6. GasTurb Inputs For Single Spool Turboshaft SL Static, ISA, using JP-5 Fuel

```
Station    W          T          P          WRstd        PWSD       =    4325.7
 amb                518.67     14.696                    PSFC       =    0.4665
  2      32.026     518.67     14.549     32.350         PWSD/W2    =    135.07
  3      32.026    1044.00    138.215      4.831         Therm Eff=      0.2957
 31      30.105    1044.00    138.215                    WF         =   0.56053
  4      29.064    2309.67    134.069      6.723
 41      30.665    2248.84    134.069      7.000         s NOx      =   0.20138
 49      30.665    1417.76     15.446                    incidence=     0.00000
  5      32.267    1399.95     15.446     50.440         XM8        =    0.2100
  6      32.267    1399.95     15.137                    A8         =    432.57
  8      32.267    1399.95     15.137     51.470         P8/Pamb    =   1.03000
P2/P1 = 0.9900  P6/P5 = 0.9800
Efficiencies:   isentr  polytr    RNI     P/P            W_NGV/W2 =     0.05000
 Compressor     0.8750  0.9070  0.990   9.500            WCL/W2     =   0.05000
 Burner         0.9999                  0.970            Loading %=      100.00
 Turbine        0.9000  0.8721  0.786   8.680            WBld/W2    =   0.01000
 Spool mech     0.9999  Nominal Spd     13820            ZWBld      =   0.00000
                                                         PWX        =   0.00000

  Fuel             FHV     humidity   war2
JP-4            18456.0       0.0    0.0000
```

Figure 7. GasTurb Outputs For Single Spool Turboshaft SL Static, ISA, using JP-5 Fuel

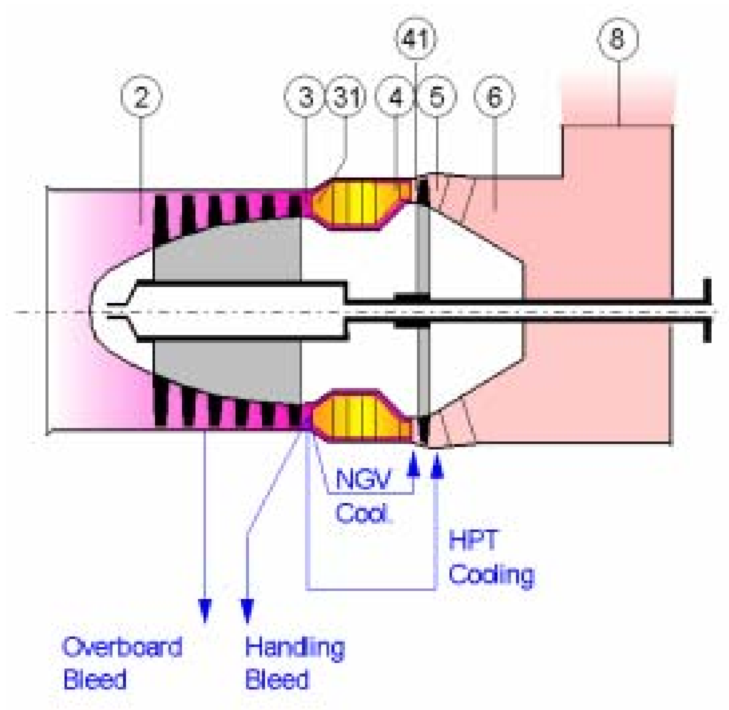

Figure 8. One spool turboshaft engine diagram

Altitude	ft	0
Delta T from ISA	R	0
Relative Humidity [%]		0
Mach Number		0
Basic Data		
Inlet Corr. Flow W2Rstd	lb/s	32.35
Intake Pressure Ratio		0.99
Pressure Ratio		9.5
Burner Exit Temperature	R	2309.67
Burner Design Efficiency		0.9999
Burner Partload Constant		1.6
Fuel Heating Value	BTU/lb	51590
Water-Fuel-Ratio		0
Steam-Fuel-Ratio		0
Overboard Bleed	lb/s	0
Mechanical Efficiency		0.9999
Burner Pressure Ratio		0.97
Turbine Exit Duct Press Ratio		0.98
Design Exhaust Pressure Ratio		1.03
Air System		
Rel. Handling Bleed		0
Rel. Overboard Bleed W_Bld/W2		0.01
Rel. Enthalpy of Overb. Bleed		1
Turbine Cooling Air W_Cl/W2		0.05
NGV Cooling Air W_Cl_NGV/W2		0.05
Comp Efficiency		
Isentr.Compr.Efficiency		0.875
Comp Design		
Nominal Spool Speed		13820
Turb Efficiency		
Isentr.Turbine Efficiency		0.9

Figure 9. GasTurb Inputs For Single Spool Turboshaft SL Static, ISA, using H_2 Fuel

```
Station    W          T          P          WRstd       PWSD      =    4614.2
 amb                518.67     14.696                   PSFC      =    0.1591
  2      32.026     518.67     14.549     32.350        PWSD/W2   =    144.07
  3      32.026    1044.00    138.215      4.831        Therm Eff=     0.3102
 31      30.105    1044.00    138.215                   WF        =   0.20394
  4      28.708    2309.67    134.069      6.773
 41      30.309    2250.15    134.069      7.050        s NOx     =   0.20138
 49      30.309    1412.07     15.446                   incidence=    0.00000
  5      31.910    1394.85     15.446     50.678        XM8       =    0.2101
  6      31.910    1394.85     15.137                   A8        =    433.69
  8      31.910    1394.85     15.137     51.712        P8/Pamb   =   1.03000
P2/P1 = 0.9900    P6/P5 = 0.9800
Efficiencies:     isentr   polytr    RNI     P/P        W_NGV/W2  =   0.05000
 Compressor       0.8750   0.9070   0.990   9.500       WCL/W2    =   0.05000
 Burner           0.9999                    0.970       Loading %=     100.00
 Turbine          0.9000   0.8717   0.786   8.680       WBld/W2   =   0.01000
 Spool mech       0.9999   Nominal  Spd     13820       ZWBld     =   0.00000
                                                        PWX       =   0.00000

  Fuel              FHV       humidity    war2
Hydrogen          51590.0        0.0     0.0000
```

Figure 10. GasTurb Outputs For Single Spool Turboshaft SL Static, ISA, using H_2 Fuel

```
Station    W          T          P          WRstd       FN        =   2179.13
 amb                486.50      6.725                   TSFC      =    0.1832
  2      17.889     510.48      7.877     33.111        FN/W2     =   3919.19
  3      17.889    1063.37     79.366      4.743        Prop Eff  =    1.7414
 31      16.816    1063.37     79.366                   WF        =   0.11088
  4      16.032    2294.72     77.067      6.555
 41      16.927    2236.67     77.067      6.826        s NOx     =   0.17049
 49      16.927    1353.35      7.105                   incidence=    0.00000
  5      17.821    1339.78      7.105     60.273        XM8       =    0.2209
  6      17.821    1339.78      6.949                   A8        =    492.35
  8      17.821    1339.78      6.949     61.626        P8/Pamb   =   1.03328
P2/P1 = 0.9900    P6/P5 = 0.9780
Efficiencies:     isentr   polytr    RNI     P/P        W_NGV/W2  =   0.05000
 Compressor       0.8472   0.8870   0.551  10.076       WCL/W2    =   0.05000
 Burner           0.9997                    0.971       Loading %=     187.81
 Turbine          0.8838   0.8476   0.456  10.847       WBld/W2   =   0.01000
 Spool mech       0.9999   Nominal  Spd     13820       ZWBld     =   0.00000
                                                        PWX       =   0.00000

  Fuel              FHV       humidity    war2
Hydrogen          51590.0        0.0     0.0000
```

Figure 11. GasTurb Outputs For Single Spool Turboprop 20,100ft, hot day, using H_2 Fuel

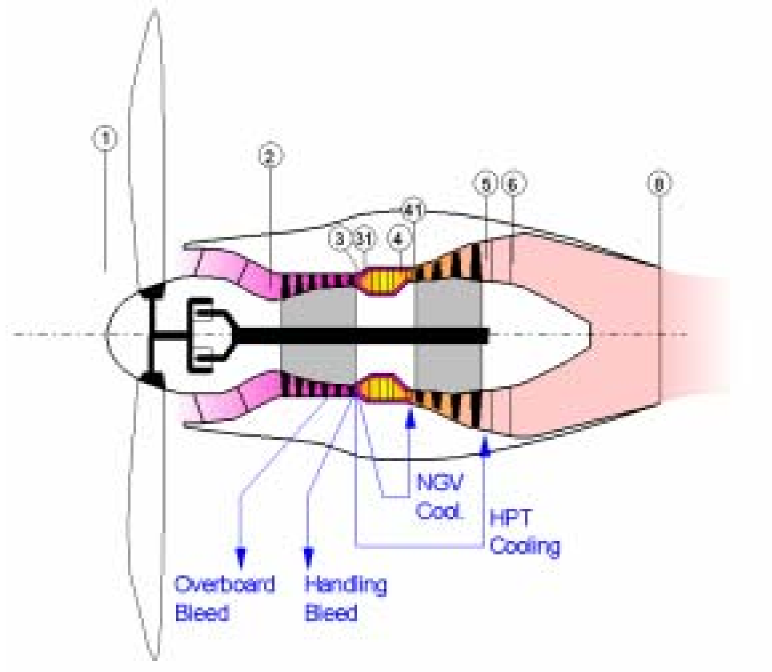

Figure 12.   One spool turboprop engine diagram

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