

Exploration of a Full-scale Hydrogen Fuel Cell Powered Helicopter through Flight Tests and Analysis

Yuval Enciu¹, Anubhav Datta¹, Diallo Ibrahima², Marc-Antoine Pilon³, Asher Leach³, and Mikaël Cardinal³

¹Alfred Gessow Rotorcraft Center, University of Maryland

²United Therapeutics Corporation

³Unither Bioelectronics, Inc.

ABSTRACT

A single pilot, full-scale, proton exchange membrane fuel cell powered helicopter is flight tested with 700 bar compressed gaseous hydrogen as fuel. Models are developed for the fuel cell, hydrogen flow and the helicopter, and validated with flight test data. The data covers powerplant architecture, stack electrical characteristics, hydrogen flow, component weights, radiator drag, and aircraft power measured in hover and forward flight. The validated models are then used to conceptually explore the conversion of a larger, more capable, turbine engine R66-like airframe with liquid hydrogen as fuel. Predictions indicate that payloads of 300 – 600 lb can be carried over a range of 200 nautical miles, with current fuel cell technology, if hydrogen storage weight fractions of 0.2 – 0.3 can be achieved, and if the tank and baggage compartments are both used for fuel. The key conclusion is that hydrogen fuel cell helicopters are feasible and the test data and validations presented here open the door for advancing its performance in the future.

NOTATION

		s_{h2}, s_{tank}	H ₂ , tank volume, m ³
A_c	Cell active area, cm ²	V	Voltage or velocity, V or m/s
C_p	Specific heat of air, J/kg-K	v	Cell voltage, V
c_d	Airfoil drag coefficient	W	Weight, lb, or mass, kg
c_l	Airfoil lift coefficient	$\dot{w}$	Mass flow rate, kg/s
D	Drag force, N	w_f	Weight fraction
E_h, E_r	Ideal, practical reversible cell voltages, V	w_{h2}, w_{tank}	H ₂ , tank weight, kg
f	Leakage or utilization factor	x_O	Mole fraction of oxygen in air
I	Stack current, A	α, α_S	Angle of attack or aircraft pitch, deg
i	Cell current density, A/cm ²	γ	Lock number or specific heat ratio
L	Lift force, N	ΔT	Temperature difference, K
m	Molecular weight, kg/mole	$\theta_{75}, \theta_{1C}, \theta_{1S}$	collec, lat and lon cyclic, deg
N	Moles of electrons	κ_0, κ_1	Proportionality constants
n_c	Number of cells	η_i, η	Ideal, practical thermodynamic efficiency
P	Power, W	v_β	Flap frequency, /rev
p	Cell power density or pressure, W/cm ² or N/m ²	ϕ_s	Aircraft roll, deg
Q	Heat or torque, W or N-m		
q	Cell heat density, W/cm ²		
S_H	Moles of hydrogen		
s_f	Volume fraction		

INTRODUCTION

The overall objective of this research is to develop a certifiable, zero-emissions, liquid hydrogen powered fuel cell helicopter, with a range of 200 nautical miles (nm) or more and a payload of critical life-saving organs, medical equipment, and personnel, equivalent to two passengers or more including a pilot. To this end, a Robinson R44 was re-engineered to a gaseous hydrogen (700 bar) powered, experimental, fuel cell helicopter followed by systematic flight testing and modeling. The objective of this paper is to document the insights gained



Figure 1. Forward flight tests of the Unither Bioelectronics hydrogen-electric R44 helicopter. The first crewed hydrogen powered fuel cell helicopter.

from this endeavor and validate fuel cell tools in preparation for the future. One of the major impediments to hydrogen aviation is the lack of flight test data. This keeps the true state-of-the-art unclear. This research seeks to fill that gap.

To combat the irreversible consequences of climate change and global warming, traditional fuels such as kerosene and gasoline must be phased out with cleaner alternatives. The promise of hydrogen is well known—it is a clean fuel and has three times the specific energy of kerosene—notwithstanding its challenges of production and storage. Compared to conventional combustion engines, modern proton exchange membrane (PEM) fuel cell technology has no CO_2 or NO_x emissions, no engine noise, and can be more efficient.

There is a history of hydrogen in aeronautics—from the first liquid hydrogen-fueled turbojet flown in 1956 on a U.S. Air Force B-57 at NACA-Lewis (Ref. 1) to the more recent liquid hydrogen-fueled piston engine flown in 2012 on the Boeing Phantom Eye unmanned air vehicle at NASA-Dryden Edwards Air Force base (Ref. 2). Fuel cells, first adopted in aerospace during the Gemini V program in 1965, and dramatically improved since then by the automobile industry, have also been adapted over the past two decades to experimental, piloted fixed-wing aircraft (Refs. 3,4). But there has not been any documented program on piloted fuel cell helicopters. The emergence of the brand-new electric-Vertical Take Off and Landing (eVTOL) aircraft industry, and its long term search for greater specific energy—with zero-emissions and no engine noise—warrants a renewed attention to hydrogen and hydrogen fuel cells.

On March 27, 2025 at Roland-Désourdy Airport in Bromont, Québec, Canada, pilot Ric Webb lifted a gaseous hydrogen powered fuel cell helicopter into a short but stable, three-minute and 16-second test flight (Ref. 5). This was the first, crewed, hydrogen fuel cell powered helicopter flight. Since then, the helicopter has been tested in piloted hover, hover

taxi, level flight up to 70 kts, climb/descent and maneuvers. Figure 1 shows the helicopter in a recent forward flight test to collect data reported in this paper. This proof-of-concept helicopter was developed by Unither Bioelectronics (a subsidiary of the United Therapeutic Corporation) by converting a piston engine Robinson R44. The project is named *Project Proticity*, inspired by biology, with the term proticity referring to the flow of protons across the inner mitochondrial membrane, driving the synthesis of energy. This phenomena is akin to the flow of protons across the membrane electrolyte of a fuel cell.

The broader aspiration of the project is to develop a zero carbon footprint vertical lift aircraft to deliver manufactured organs to hospitals for lifesaving human transplant surgeries. The converted hydrogen-electric R44 and its gaseous hydrogen storage is a baseline stepping stone to a larger, more capable hydrogen-electric R66-like airframe in the future with liquid hydrogen supplying the fuel cell. Refining tools with flight test data from the baseline and parametric exploration of the future are the two objectives of this paper.

The concept of a hydrogen fuel cell helicopter is as old as eVTOL aircraft (Ref. 6). Researchers had explored on paper conversions of existing helicopters (Refs. 7,8) as well as new designs (Refs. 9,10), but there was no flight test prototype to validate any conclusion and benchmark the true potential of a hydrogen helicopter. A few notable vertical lift industry efforts have been reported in the recent past, see for example the unmanned liquid hydrogen flight by Joby (Ref. 11) or development by Piasecki (Ref. 12). These pursuits are being fueled by the growth in power density of fuel cells over the last decade and advanced light-weight liquid hydrogen storage tanks achieving high weight fractions and low volumetric footprints. However, none of the efforts have published any technical details or flight test data.

Design tools needed to support such developments require high quality research data to validate models, identify barriers,

ers, and benchmark progress. The goal of this paper is to carry out these tasks using *Project Proticity* as the foundational, full-scale, flight test prototype.

Scope of Paper

Following the introduction, the paper begins by describing the full architecture of the fuel cell powerplant. Models of the fuel cell and the helicopter are then briefly summarized. The next two sections deal with the hydrogen-electric R44 and R66. The last section concludes the paper. Throughout, the term fuel cell is used interchangeably to mean a cell, the stack, or the entire stack and ancillary systems, depending on context, and abbreviated as FC or PEMFC. The converted helicopter is termed fuel cell helicopter, hydrogen helicopter, or hydrogen-electric helicopter interchangeably. The word hydrogen is used interchangeably with H₂ or H₂.

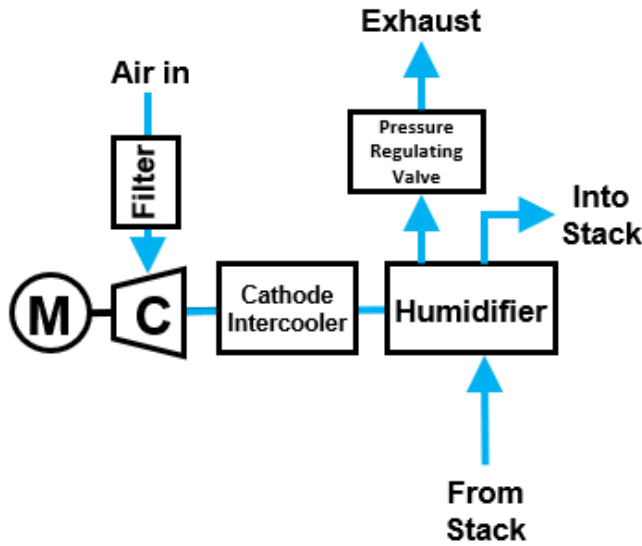


Figure 2. Air system.

FUEL CELL ARCHITECTURE

Electrical power from a Proton Exchange Membrane (PEM) fuel cell (PEMFC) is generated through a chemical reaction that requires hydrogen and air as reactants. Regardless of gaseous or liquid storage, the cell needs gaseous hydrogen, so the architecture remains the same. A fuel cell stack is formed by arranging cells in parallel. Operation of the stack requires an air intake system, a low temperature cooling system (LTC), an H₂ system, a high temperature cooling system (HTC), and supporting water management.

Air System

The air system supplies oxygen to the cathode side of the fuel cell. Figure 2 shows a schematic of the air system. The air is sucked in through a filter (to avoid contaminating the flow with dirt, dust, and other unwanted particles) by a compressor, turned by an electric motor. The compressor increases

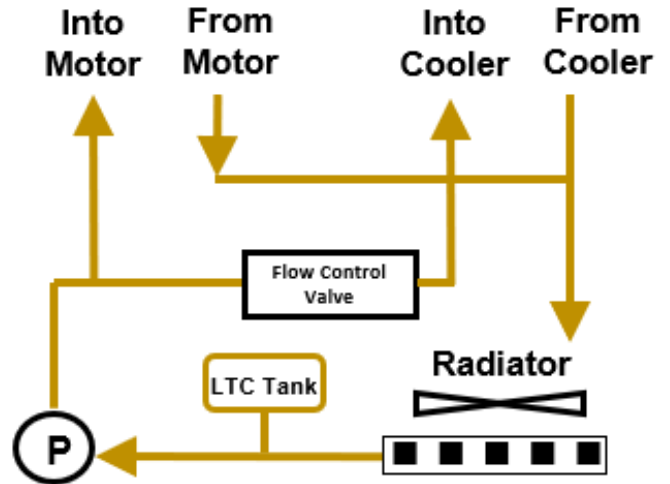


Figure 3. Low temperature cooling (LTC) system.

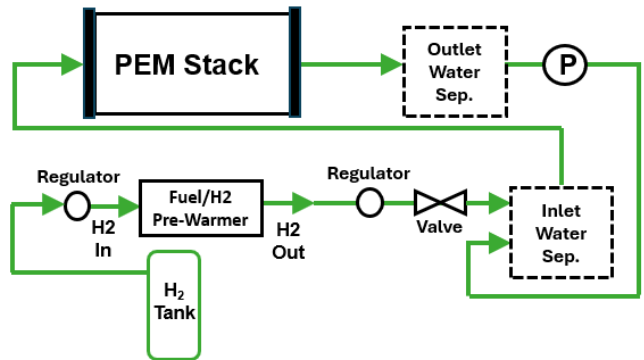


Figure 4. Hydrogen system.

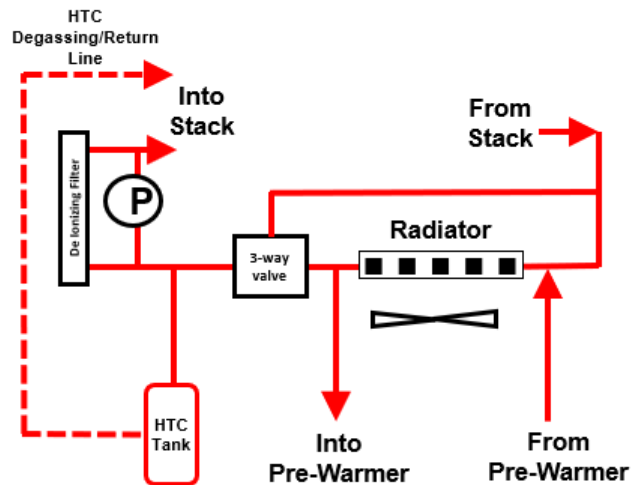


Figure 5. High temperature cooling (HTC) system.

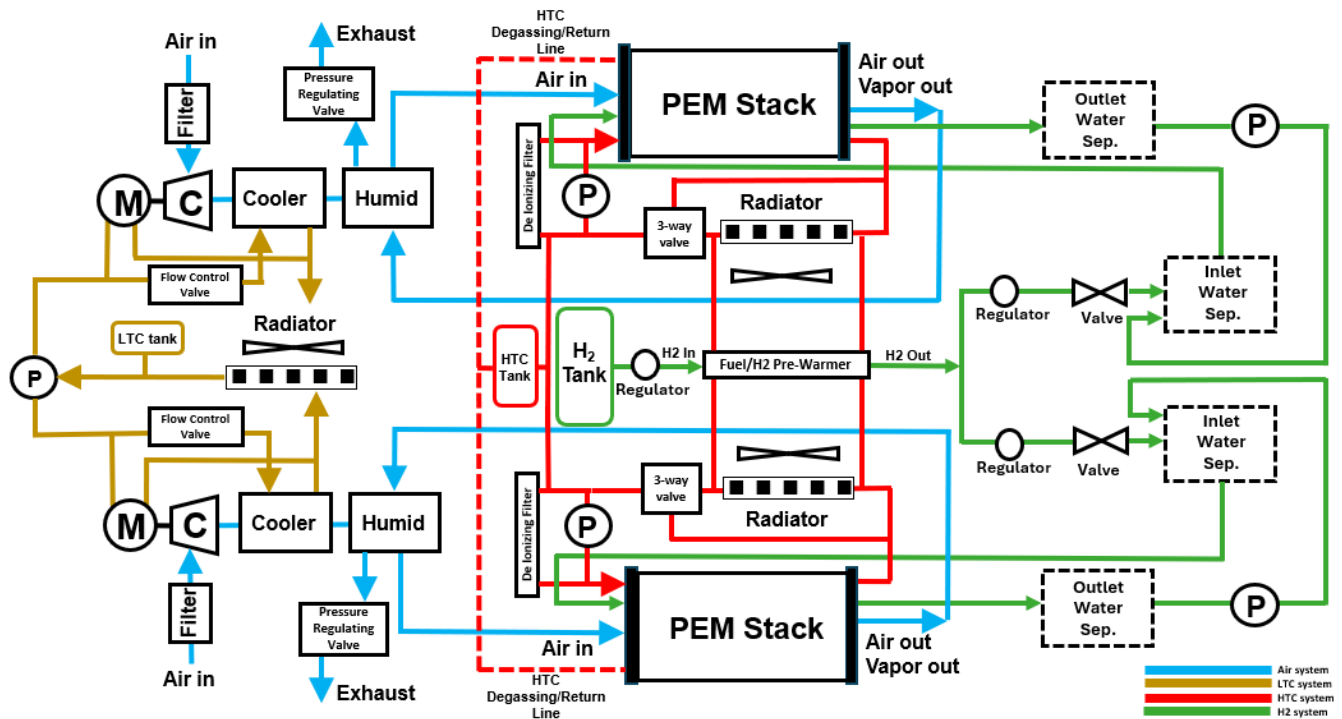


Figure 6. The full fuel cell powerplant architecture.

the pressure of the intake air from ambient to the stack pressure of 2.2 atmospheric pressure (atm). The compressor also heats the air to a temperature higher than the stack temperature (80°C). Therefore, a cathode intercooler is needed to cool the air.

High humidity is essential to prevent membrane dehydration, so air is passed through a humidifier. Recycled air flowing out of the stack transfers heat and moisture to the intake air. This prevents the need of an external humidification system. The air is now ready to enter the stack. A pressure regulating valve is used to discharge the remaining moisture-laden air into the atmosphere.

Low Temperature Cooling (LTC) System

The cathode intercooler and motor need cooling. Because the heat load is low (compared to the stack system described later), a low temperature coolant suffices. The name is borrowed from automobile stacks, although the coolant used in this application is the same throughout. Figure 3 shows a schematic of the LTC system.

Starting from the tank, the coolant is pumped into the intercooler and motor, and its temperature increases. The hot coolant flows into a radiator, which acts as a heat exchanger. Air absorbs the coolant heat, and the cooled coolant is ready to circulate through the LTC loop again.

Hydrogen System

The hydrogen system supplies hydrogen to the anode side of the fuel cell. Figure 4 shows a schematic of the gaseous hydrogen system.

Compressed hydrogen gas at 700 bar exists the tank and passes through a high-pressure regulator, which reduces its

pressure to 10 bar. This hydrogen flows through a pre-warmer to increase its temperature to the stack temperature. It then passes through a mid-pressure regulator, to reach the required stack pressure of 2.2 atm. To enter the stack at a controlled flow rate based on load (power) demand, hydrogen passes through a proportional control valve.

Recirculated hydrogen is mixed with fresh hydrogen at the water separation inlet, and flows into the stack. The stack outlet stream contains unused hydrogen and water (liquid or vapor). It passes through the outlet of the water separation system, where water is stripped from the unused hydrogen and exhausts out. Hydrogen is sent to the separator inlet, where it mixes with fresh hydrogen, and flows back into the stack.

High Temperature Cooling (HTC) System

Compared to a combustion engine, a fuel cell operates at a very low temperature, so the exhaust gases carry very little heat out. Thus, heat must be removed entirely by other means. The high temperature cooling (HTC) system cools the stack. It is the heaviest element of the entire powerplant after the stack system. Figure 5 shows a schematic of the HTC system.

The system is divided into three parts. The first part handles coolant flow into the stack. The second part cools the coolant. The third part degasses the line.

Liquid coolant is pumped out of the tank and passed through a de-ionizing filter. Deionizing prevents high coolant conductivity from ion contamination in the bipolar plates. A highly conductive coolant leads to a high leakage current, which reduces cell efficiency greatly. Coolant is then pumped through the cooling channels built into the cathode bipolar plates of the fuel cell, to absorb stack heat.

Once it exits the stack, it is sent to a radiator. Some of the hot coolant is used to increase the fuel temperature in the hydrogen pre-warmer. This takes advantage of coolant heat that would otherwise be wasted. A three-way valve regulates stack temperature by mixing or diverting coolant between a cooling path through the radiator, and a bypass path.

A coolant degassing line removes trapped gases such as nitrogen or leaked hydrogen from the coolant loop.

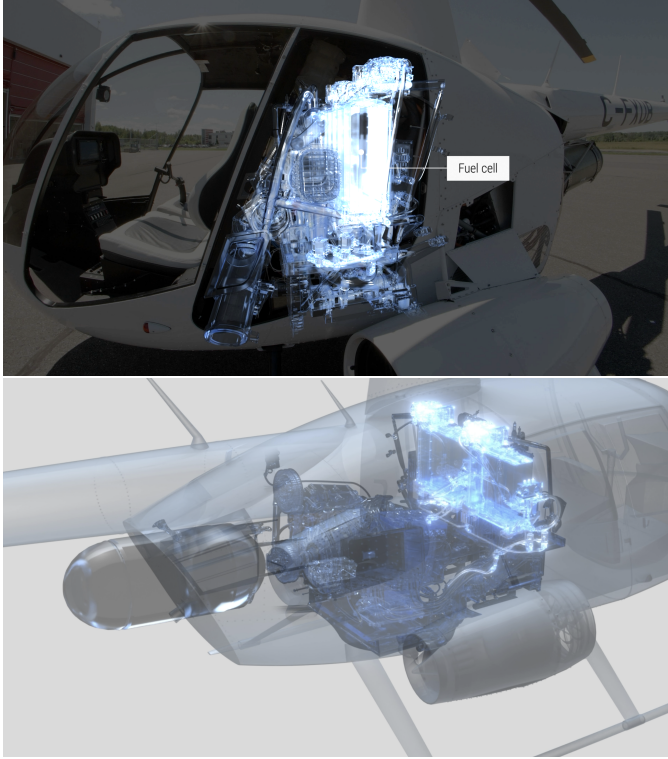


Figure 7. PEM Fuel stack integration on Unither Bioelectronics hydrogen-electric R44; top-isometric view, bottom-side view.

Full PEMFC System

The full system integrates the building blocks above innovatively with proper interconnections. Figure 6 shows a schematic of the full PEMFC system.

The full architecture contains two parallel stacks, each providing exactly half the power. This provides some redundancy, hence improves reliability. Each stack can provide a little more than the cruise power for best endurance (92 kW) which can assist in safe landing if one stack fails. Each stack has its own air, high temperature cooling, and hydrogen supply system. The low temperature cooling system — mainly radiators and pumps that cool the coolant passing into the cathode inter-cooler and motor, are common. Hydrogen and coolant tanks are also shared.

The actual hardware on the fuel cell helicopter is shown in Figure 7, from two different angles. The two stacks in front and the single tank in the back are clearly visible. The tank is a composite type IV tank containing 2.1 kg of hydrogen —

just enough for flight testing. The two nacelles hanging on either side of the fuselage carry the radiators for stack cooling (HTC system).

All of the mentioned components were modeled, their performance and weights predicted, and validated with measurements. The modeling is summarized in the following section.

FUEL CELL MODEL

The fuel cell model sizes a stack to deliver a specified design power, P_D , at a specified design voltage, V_D . The model starts from Ref. 15 and then refines it to include the details of the actual system and the dual-stack architecture. The key elements of the model are summarized in this subsection.

The fuel cell used is a Proton Exchange Membrane (PEM) Fuel Cell (PEMFC). At the specified cell operating temperature of 80°C , the ideal reversible cell voltage is $E_h = 1.4722$ V. The efficiency of a cell is defined relative to this voltage. Due to entropy losses, the true reversible voltage is lower and it is a function of pressure and reactants. At the specified cell operating pressure of 2.2 atm, and with air as reactant (instead of pure oxygen), the true (practical) reversible cell voltage is $E_r = 1.1722$ V. The ideal thermodynamic efficiency is $\eta_i = E_r/E_h = 1.1722/1.4722 = 0.796$. This value is merely of academic interest, because as soon as a current i is drawn, the cell voltage drops again from E_r to its true value v .

Often, the total current produced at the electrodes is $i + i_{\text{leak}}$, of which i_{leak} is lost in side reactions and i is the useful current that flows through the cell. The practical thermodynamic efficiency of the cell is then

$$\eta = \frac{v}{E_h} \frac{i}{i + i_{\text{leak}}} = \frac{v}{E_h} f \quad (1)$$

where f is a leakage factor. The drop in cell voltage from E_r to its true value v is due to three losses: activation loss (due to reaction kinetics), Ohmic loss (due to ionic and electronic resistance), and concentration loss (due to mass transport). Each of these losses is influenced by the cell leakage current, i_{leak} . A simple model of the variation of the cell voltage with current is

$$v = E_r - \eta_{\text{act}} - \eta_{\text{Ohmic}} - \eta_{\text{conc}} \quad (2)$$

which is called the characteristic $i - v$ curve. The three losses can be modeled using seven thermodynamic constants plus the leakage current.

In the $i - v$ curve, current is normalized by the area of the membrane, resulting in the cell current density (typical unit A/cm^2). The corresponding characteristic $i - p$ curve shows variation of cell power with current. Here, $p = iv$, which is the power density (Watt/cm^2).

The $i - v$ curve of the present cell is not published, but it follows the state-of-the-art. Figure 8 shows a range of such curves found in open literature. The UMD data is acquired for this paper from an in-house test on an off-the-shelf unpressurized fuel cell stack (1 atm). The U.S. Department of Energy data, published in Ref. 13, shows the effect of higher pressures

1 – 2.5 atm. The Toyota Mirai data, published in Ref. 14, is also at high pressure, with pressure rising from 1.1 – 2.2 atm as current density rises from 0.05 – 2.2 A/cm². The Air Liquide/Axane data, available in public domain in their brochure (not peer reviewed), is of similar pressure. Figure 9 shows the corresponding $i - p$ curves. The high power density of modern cells are evident. The Toyota and Axane data represent the state-of-the-art, and since altitude operations call for pressurization, these are representative of the discussed stack design.

The basis for the design involves a selection of a cell design point, v_c, i_c , along the $i - v$ curve. Any point can be selected and the stack sized to deliver the specified design power P_D at the specified design voltage V_D . Each selection will produce a different stack, of different weight and fuel consumption characteristics. The selection is made judiciously to minimize the weight of the entire stack system.

If the selected point corresponds to cell power density p_c , then the total membrane area, must be P_D/p_c . The volume, and hence the weight, is ultimately proportional to this area.

$$W_{\text{stack}} = \kappa_0 + \kappa_1 \frac{P_D}{p_c} \quad (3)$$

The constant of proportionality κ_1 and the overhead κ_0 are design factors that depend only on structure and are calculated with a model (detailed later). The specified design voltage can be supplied by splitting the membrane area into n_c cells and arranging them in series (to form a stack) so they add up to $V_D = n_c v_c$ (v_c adjusted for integer n_c). The membrane area of each cell, called the cell active area, is $A_c = P_D/(n_c p_c)$. As design voltage rises, the number of cells increases, while active area decreases. Thus, voltage drives the aspect ratio of the stack, not its weight.

For a selected cell design point, the number of cells, cell active area, and cell efficiency for a specified design power P_D at voltage V_D (or current $I_D = P_D/V_D$) are the following:

$$\begin{aligned} n_c &= \frac{V_D}{v_c} \quad (\text{rounded to integer}) \\ A_c &= \frac{P_D}{n_c p_c} = \frac{I_D}{i_c} \\ \eta_D &= \frac{v_c}{E_h} \frac{i_c}{i_c + i_{\text{leak}}} = \frac{v_c}{E_h} f_c \end{aligned} \quad (4)$$

where f_c is the leakage factor at i_c . If the selected design point is at maximum power density, $p_{c,\text{max}}$, then from Equation 3, the stack has minimum weight. If the selected design point is at maximum voltage, $v_{c,\text{max}}$, then from the third of Equation 4, the stack has maximum efficiency, which later leads to minimum fuel consumption. Conflicts arise as maximum power density occurs at the minimum voltage, and vice versa. Thus, one or the other must be selected, or perhaps a compromise in between. Later, stacks will be shown for such conditions. Once n_c and A_c are fixed, the rest of the design falls out.

The stack structure is modeled based on the template shown in Figures 10 and 11. The membrane is encapsulated within catalyst layers, gas diffusion layers (GDL), and gaskets to form a Membrane Electrode Assembly (MEA), surrounded by the bipolar plates, as in Figure 10. Hydrogen (cathode side) and air (anode side) flow through channels built into these plates. This forms one cell unit. The cell units are placed in series and end with current collector plates, insulating plates, and end plates as in Figure 11, secured together by a fastener.

The measured weight of the stack structure will be used to validate the weight model. In addition to the stack, the electronic stack controller and other structural overhead (covering support structure) are accounted for under stack weight.

Once the stack is sized, the current, voltage, and power at any current density i are the following,

$$\begin{aligned} I &= i A_c \\ V &= n_c v \end{aligned} \quad (5)$$

$$P = V I = A_c n_c$$

with efficiency varying with current density (governed by fuel flow) as in Equation 1. The stack $I - V$ and $I - P$ characteristics for the full current range will be measured and used to validate the model. Since E_h and i_{leak} are hard to measure, the efficiency is difficult to predict. However, i_{leak} can be inferred from fuel flow data and will be used to validate the model.

The fuel (hydrogen) flow is obtained from the anode half reaction stoichiometry: $H_2 \rightarrow 2H^+ + 2e^-$, where a single mole of hydrogen ($S_H = 1$) is oxidized to produce two moles of electrons ($N = 2$). A utilization factor f_H is added if supply is greater than stoichiometric consumption. The fuel flow is

$$\dot{w}_H \text{ in} = f_H S_H \frac{m_H}{N F} \frac{P}{v} \frac{1}{f} \quad (6)$$

where $m_H = 2.01588 \times 10^{-3}$ kg/mole is the molecular weight of hydrogen, $F = 96,485$ Coulomb/mole is the charge carried by a mole of electrons (Faraday's constant), P is the stack power, v is the cell voltage, and f the leakage factor. In the present system $f_H = 1$. Because fuel consumption is inversely proportional to cell voltage, maximum cell voltage gives minimum fuel consumption—in direct contradiction to stack weight.

The weight of the hydrogen carried aboard is specified and the time of operation is calculated. The top-level parameters for conceptual design are the weight fraction, w_f , and volume fraction, s_f .

$$\begin{aligned} w_f &= \frac{w_{h2}}{w_{h2} + w_{\text{tank}}} \\ s_f &= \frac{s_{h2}}{s_{\text{tank}}} \end{aligned} \quad (7)$$

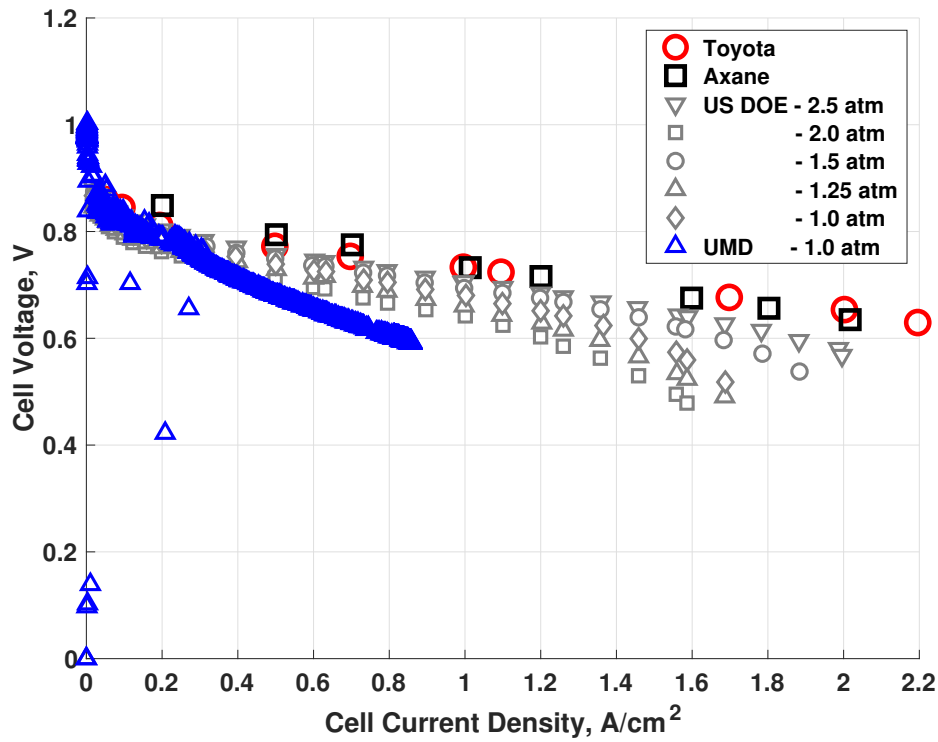


Figure 8. Cell voltage versus current density.

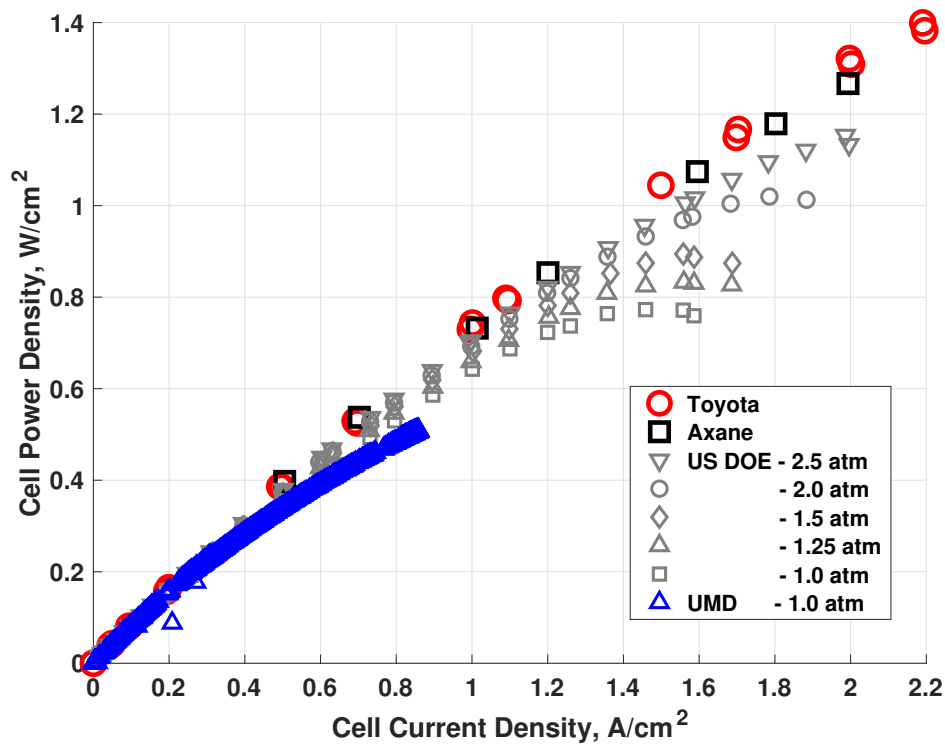


Figure 9. Cell power density versus current density.

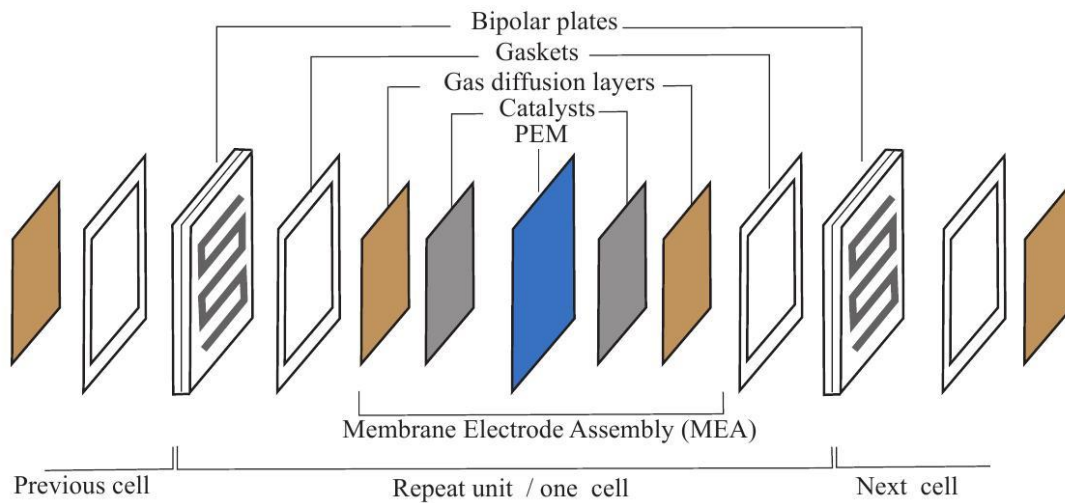


Figure 10. Construction model from membrane to a cell.

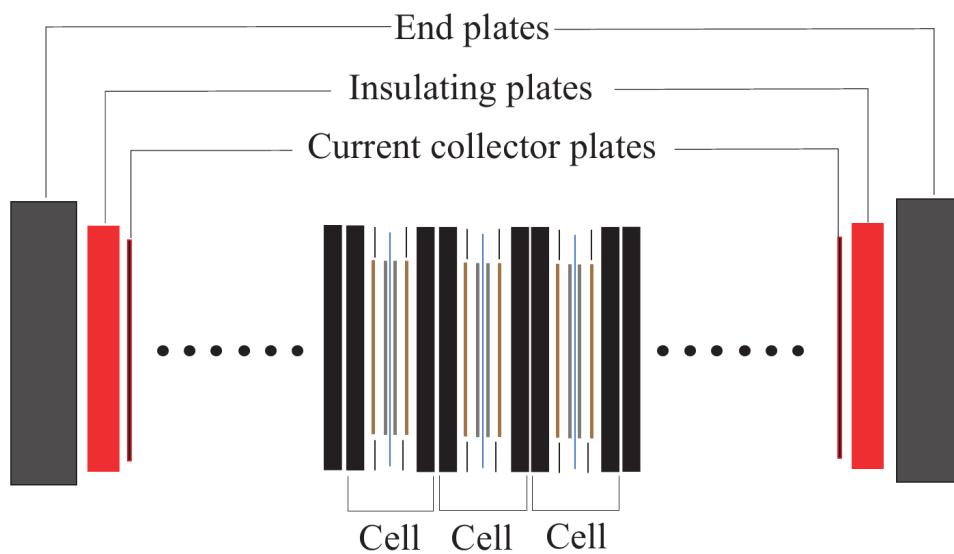


Figure 11. Construction model from a cell to a stack.

where w_{h2} is the mass of hydrogen, w_{tank} is the mass of tank system, which includes regulators, sensors, a high pressure manifold and a hydrogen pre-warmer to raise it to the stack temperature. s_{h2} is the volume of hydrogen and s_{tank} is the volume of tank system. The tank weight is calculated from a specified tank weight fraction. The total weight is measured and used to validate the model.

The air flow is obtained from oxygen flow and mole fraction of oxygen in air. The oxygen flow is obtained from the cathode half reaction stoichiometry: $(1/2)O_2 + 2e^- + 2H^+ \rightarrow H_2O$, where half a mole of oxygen ($S_O = 1/2$) captures two moles of electrons ($N = 2$). A utilization factor f_A is needed as the supply of air is always greater than stoichiometric consumption. The air flow is

$$\dot{w}_{\text{Air in}} = f_A S_O \frac{m_A}{x_O N F} \frac{P}{v} \frac{1}{f} \quad (8)$$

where $m_A = 28.9655 \times 10^{-3}$ kg/mole is the molecular weight of dry air and $x_O = 0.2095$. f_A is hard to measure, but typically $\approx 1.5 - 2.5$ for PEM cells; a value of 2.5 is used here to be conservative. A high cell voltage is beneficial, as it results in lower air flow for the same power, hence requiring a lighter air compression system.

A compressor takes in ambient air at pressure p_a and pressurizes it to the stack pressure p_s . This requires power that cuts into the net power output. The power is calculated from the rise in temperature of the air ΔT as it is compressed. ΔT is found by assuming isentropic compression with an efficiency $\eta = 0.7$.

$$P_{\text{comp}} = C_p \dot{w}_{\text{Air in}} \Delta T$$

$$\Delta T = \left[\left(\frac{p_s}{p_a} \right)^{(\gamma-1)/\gamma} - 1 \right] T_a \frac{1}{\eta} \quad (9)$$

where T_a is the ambient temperature (Kelvin), C_p is the specific heat of air in J/kg-K and γ is the ratio of specific heat. C_p and γ are calculated at the ambient temperature. The compressor weight is calibrated to the maximum power, which is required to deliver maximum stack power. For a pressurized stack, the air at the outlet is at a higher pressure than ambient, and some power can be recovered with a turbine. This is not used in the present system.

The heat generated in each cell is the cell heat density (Watt/cm²), $q = E_h(i + i_{\text{leak}}) - vi = (E_h/f - v)i = (1/\eta - 1)p$, which is the gap between power delivered to the cell and power extracted by the electrons. The heat generated by the stack is simply the heat density multiplied by the active area and number of cells.

$$Q = (E_h/f - v) i n_c A_c = (1/\eta - 1) P \quad (10)$$

The power to operate the subsystems—air, high temperature cooling, and low temperature cooling is called the balance of

plant, P_{BOP} . Thus, if the gross power is P_D , the net useful power is

$$P_{De} = P_D - P_{BOP} \quad (11)$$

reported as kW-electrical or kWe. The final system efficiency at the design condition is then

$$\text{System } \eta = \frac{v_c}{E_h} f_c \frac{P_{De}}{P_D} \quad (12)$$

To predict component weights, the model is refined to include all components of the air intake, high and low temperature cooling, and hydrogen supply. Careful calibration of weights is carried out to match measured weights of the fuel cell R44. The model generates engine charts of current, voltage, fuel flow, and balance of plant loss which will all be compared to test data in a later section.

HELICOPTER MODEL

The helicopter model predicts power versus speed for various gross weights and altitudes. A simplified model is used, adequate for a design level analysis of the fuel cell powerplant.

The main rotor blades have rigid flapping motion with frequency $v_\beta = 1/\text{rev}$ and Lock number $\gamma = 6$ (at 4300 ft). The blades are rigid in lag and torsion, and the details of the teeter/dual flap hinges and blade elasticity are neglected. Uniform inflow assumption is used. The aerodynamic model is linear with sectional lift and drag coefficients of $c_l = 5.73\alpha$ and $c_d = 0.01$, respectively, where α is the sectional angle of attack.

The tail rotor model is similar, except it has no cyclics, and the flapping motion is neglected. The fuselage is a rigid body with a center of gravity (CG) located a distance h below the shaft. Specified CG offsets in the lateral and longitudinal directions are included. The horizontal and vertical tails are not modeled separately, but as part of the fuselage. The flat plate area is refined for a good fit of power at high speed. This compensates for inadequate profile power rise due to the simplified sectional drag model. The flat plate area is increased by the nacelle drag of the fuel cell cooling system, as will be explained in a later section of the paper.

Only steady level flight is considered at this stage. The aircraft is trimmed for equilibrium in six-axes with the six trim variables: collective, longitudinal and lateral cyclics, aircraft pitch and roll, and tail rotor collective.

The aircraft properties are compiled from the R44 and R66 pilot's operating handbook. Model validation for both is performed using published R44 and R66 performance data, taken from Ref. 16 and Ref. 17, respectively.

HYDROGEN R44

The baseline four-seat light utility R44 has a (maximum) gross takeoff weight of 2500 lb. The powerplant is a single Textron-Lycoming IO-540-AE1A5 six-cylinder piston engine with manufacturer's rating of 260 HP (194 kW), derated to a

5-minute takeoff rating of 245 HP (184 kW), and continuous rating of 205 HP (153 kW).

The hydrogen R44 retained the same maximum gross takeoff weight. The fuel cell powerplant provided the same takeoff rating of 245 HP (184 kW). Gaseous hydrogen (700 bar) was used as fuel. The belt drive was retained, and the engine was replaced with an electric motor. The following subsections describe the performance and weights of the powerplant, hydrogen storage, radiator nacelles, and the converted aircraft. Validation of prediction tools is also included.

Fuel Cell Powerplant

The dual-stack configuration is designed so that each stack is required to generate a gross electrical power of 92 kW at a specified stack voltage of 210 V. The environment is SL/ISA.

The variation of stack voltage and power with stack current are shown in Figure 12. The stack current is between 0 – 420 A, and the corresponding voltage and power are between 0 – 210 V and 0 – 92 kW, respectively. The model predictions follow the test data satisfactorily, except for jumps in voltage at low power rating. These are transient phenomena not included in the model and likely artifacts of the test procedure. The power variation with current, however, is smooth. Because there are two stacks in parallel, the stack currents add, whereas the voltage remains the same as that of a single stack.

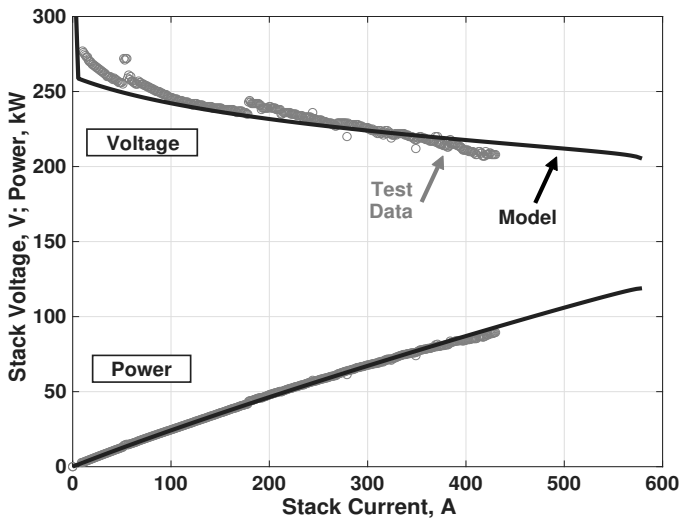


Figure 12. Measured stack voltage and power versus current compared with predictions.

The next important performance parameter is the fuel flow. The total fuel (hydrogen) flow (to both stacks) is plotted versus total current in Figure 13. The test data allows for comparisons with predictions and refinement of the model. The predicted fuel flow is higher than measurements at lower currents but it is conservative.

Figure 13 shows how two leakage currents change the predictions. They suggest a value around 0.2 A/cm², lower than the 0.3 A/cm² value that was obtained from the cell $i - v$ curve. The value of 0.3 A/cm² was retained as it gave a good fit to

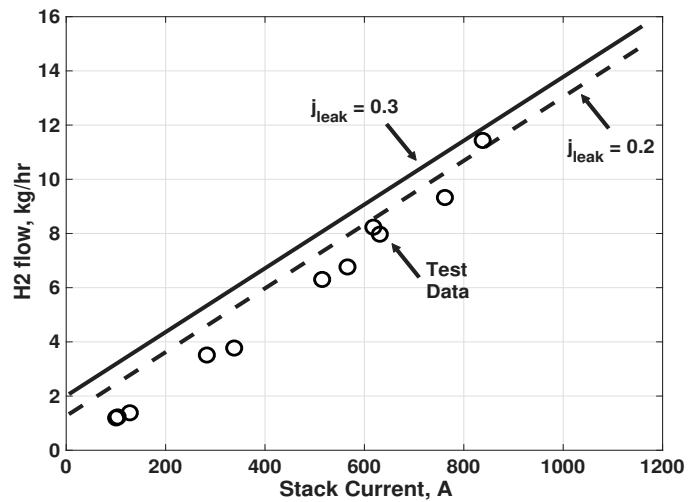


Figure 13. Measured H₂ flow rate versus stack current compared with predictions.

the curve, and would provide a conservative margin to the payload-range assessment for the R66 later in the next section.

The test program allowed for the validation and calibration of the most crucial of all aspects of the design — weights. All parts and pieces of the powerplant were weighed individually. The subsystem weights are presented in Table 1. Recall, the architecture has a dual-stack system, so the breakdown includes both stacks, air, hydrogen, and HTC systems, with the single LTC system.

The stack system consists of the stack, controller, and other weights which include support structure (for dual-stack architecture), fasteners and fittings. The stack weight is well predicted, the controller is over-predicted, and the other weights were not in the model at all, making the value of this exercise clear. With the controller compensating for this other weight, the overall the prediction was only about 5% lower. With test data available, the model was calibrated with the necessary technology factor ($= \text{measured}/\text{predicted}$).

The air system consists mainly of the compressor and humidifier, along with other ancillary parts listed in the table. The compressor weight was over-predicted but the ancillaries were under-predicted. Overall, the prediction was about 13% lower. With test data available, the models were corrected.

The high temperature cooling system (stack cooling) is as heavy as the stack system. It consists mainly of the radiators and coolant along with other ancillary parts listed in the table. The prediction is quite close, and only about 5% lower. The cooling system is a major design challenge and was intentionally over-sized. This was a judicious choice for safety of the first piloted hydrogen helicopter. Hence, predictions used very liberal margins to match the actual system. The present exercise provided the insight and models needed for optimization, which remains an area of focus for future design.

The low temperature cooling system (air cooling) consists mainly of the cathode intercooler, a lower shroud structure, its own radiator and coolant, along with ancillary parts listed

Table 1. R44 fuel cell powerplant masses; measured versus predicted; power $P = 184$ kW.

Sub-system	Components	Measured kg	Predicted kg	Technology factor
2 Stacks	Stacks,	88	91	0.97
	controller, and	6	17	0.35
	Other.	20	0	
	Total	114	108	1.05
2 Air Systems	Compressor,	16	34	0.47
	humidifier			
	filter			
	regulator			
	plumbing & power.	28	5	5.6
	Total	44	39	1.13
2 High Temperature Cooling Systems	Radiator,			
	coolant			
	filter			
	regulator			
	plumbing & power			
	Total	112	107	1.05
1 Low Temperature Cooling Systems	Cathode cooler,			
	lower Shroud,			
	radiator,			
	coolant,			
	filter,			
	regulator,			
	plumbing & power			
	Total	22	8.7	2.53
	Grand total	292	263	1.11
H ₂ System (2.1 kg H ₂)	Tank.	36	35	1.03
	Regulator,			
	sensors,			
	plumbing & prewarmer	19	3	6.3
	Total	55	38	1.4

in the table. This system is much lighter than the stack cooling but significant. The prediction underestimated the weight by a factor of 2.5. With test data available, the models were corrected.

The hydrogen system consists mainly of the tank and the anode hydrogen circuit consisting of the ancillary parts listed in the table. There are also the water separators. The tank weight is well predicted, but the ancillaries are not, hence, the prediction under-estimated the total system weight by a factor of 1.4. With test data available, the models were corrected. Note that the ancillary parts are decoupled from hydrogen weight, and it is the tank that scales up with the amount of hydrogen.

With the models calibrated, a more detailed assessment of the system is given in Table 2. Here, the present design (column 4) is compared with two important hypothetical designs — one for minimum weight (column 5) and another for maxi-

mum efficiency (column 6).

The only change is in the selected design point on the cell $i - v$ curve. For the minimum weight stack, the design point corresponds to maximum power density, p_c . This implies a lower cell voltage, v_c . For the maximum efficiency stack, the design point corresponds to a high cell voltage. This implies a lower power density.

The minimum weight design minimizes the weight of the stack and air system but increases the heat load, hence the weight of the HTC, as it is less efficient. The maximum efficiency design does the opposite. The LTC weight, small as it is, remains unaffected. In the end, the difference in total weight is not substantial (289 kg versus 296 kg). This is due to the relatively flat nature of the state-of-the-art cell $i - v$ curve, which is of significant benefit, as it brings the maximum efficiency solution close to the minimum weight solution.

Table 2. Various designs for net $P = 160$ kWe. SL/ISA. Stack: 210V, 2.2 atm, 80°C .

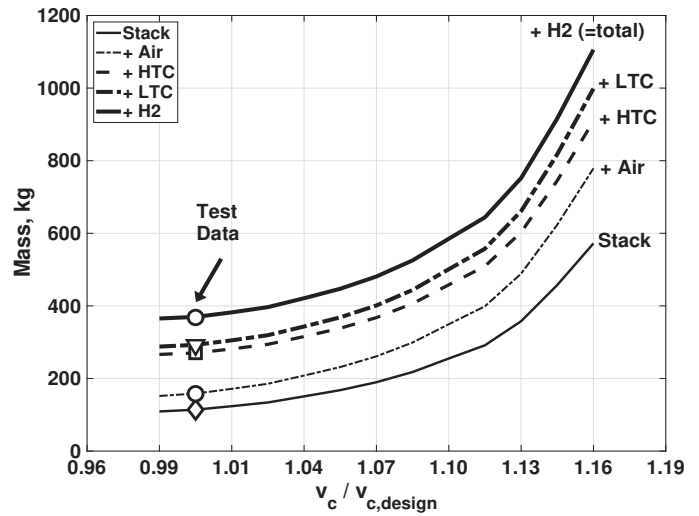
Components	Variables	Units	Present design	min weight	max η
Stack	Weight	kg	114	103	117
	Specific power	kW/kg	1.57	1.8	1.61
Air	Weight	kg	44	43	46
	Weight	kg	112	116	111
HTC	Q generated	kW	224	232	222
	Q load	kW	199	208	197
	$Q/\Delta T$	kW/ $^\circ\text{C}$	3	3.2	3
LTC	Weight	kg	22	22	23
	Q generated	kW	12	11	12
	Power	kW	33	34	33
Stack system					
	P, P_{net}	kW, kWe	184,160	185,160	184,160
	Weight	kg	292	289	296
	Specific power	kW/kg	0.63	0.64	0.62
	Efficiency η		0.3226	0.3225	0.3271

The present design falls in between. Overall, the specific power of the stack alone is $184/88 = 2.1$ kW/kg, stack subsystem is $184/114 = 1.57$ kW/kg, the total system is $184/292 = 0.63$ kW/kg and total system with net electrical is around $160/292 = 0.55$ kWe/kg. The stack efficiencies, after balance of plant losses, are all around 0.32.

Finally, the cell voltage is varied widely to cover the full range, and a series of hypothetical designs is studied. The mass of these designs is plotted in Figure 14. The plot uses a build-up approach, where the curve above adds to the weight of the curve below. The first curve at the bottom corresponds strictly to the weight of the stack, the curves above add the weights of the air system, HTC, LTC, and the hydrogen system, progressively. The cell voltage is plotted as a fraction of the design voltage. The sensitivity of weights to the selection of design voltage is evident. Even a 10% change in design voltage can almost double the weights. Clearly, the stack and the HTC are the heaviest subsystems. With the stack already being at a high specific power, and the HTC oversized for the experimental aircraft, the effect of reducing this weight will be studied later for the R66.

Hydrogen Storage

An external tank stores hydrogen for easy access for experimental purposes. The proof-of-concept R44 has a 36 kg tank, that stores 2.1 kg of gaseous hydrogen at 700 bar. This corresponds to a weight fraction of $w_f = 0.055$ (or 5.5%). The total tank system, including regulators, sensors, plumbing, fuel prewarmer and other pieces weigh about 55 kg, with a weight fraction of 0.036 (3.6 %). This fraction will be significantly increased with liquid hydrogen for the R66.

**Figure 14. Powerplant weight variation with cell voltage.**

Radiator Drag

The majority of the balance of plant is dominated by the high temperature cooling system. Note that it is significantly oversized in keeping with the experimental nature of the aircraft. The radiators are large and built into special nacelles which hang from either side of the fuselage, thus imposing a high penalty on the airframe drag. An experimental test campaign was carried out to measure their flat-plate drag.

For this purpose, a special test rig was constructed. A single nacelle was mounted on a platform on top of a pickup truck, which was driven at speeds up to 55 kts on a runway. The experimental setup is shown in Figure 15.

The drag was measured using a load cell and converted to equivalent flat plate area by dividing the drag with the dynamic pressure. The data is presented in Figure 16, with measurement scatter shown as minimum, mean, and maximum



Figure 15. A single radiator nacelle being truck-tested on the runway.

values. There is significant scatter at low speeds, which is not unusual due to low signal-to-noise ratio, but the values converge to around 2.15 ft² beyond 40 kts. So, to first order, the two nacelles contribute an additional 4.3 ft² of flat plate area to the helicopter, neglecting any interference.

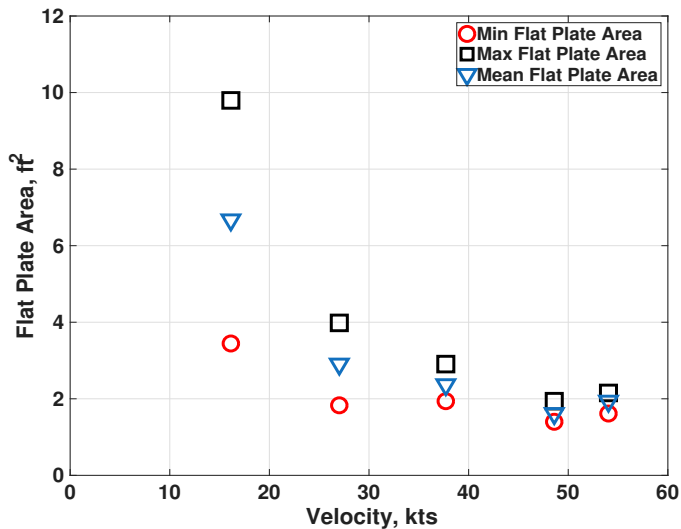


Figure 16. Measured radiator nacelle drag; Min and Max values show scatter in data.

Aircraft Performance

With the powerplant validated and radiator drag measured, attention is turned to flight tests. The fuel cell power output was measured during the tests, accounting for aircraft power plus all the balance of plant losses. Comparing predictions with this data would therefore validate both.

The aircraft power versus speed of the baseline piston engine R44 is shown in Figure 17. Note that this is not the exact aircraft that was converted, but one for which test data is publicly available (Ref. 16). The gross weight of the aircraft was 2200 lb, 300 lb lower than the fuel cell aircraft. The model

assumes main rotor and tail rotor power to yield total aircraft power. To obtain satisfactory predictions at high speeds, the flat plate area in the model was increased by 20% from the assumed baseline of 4.75 ft² to 5.7 ft². Thus, this is not a true prediction, but a calibration, justified by the conceptual nature of the analysis.

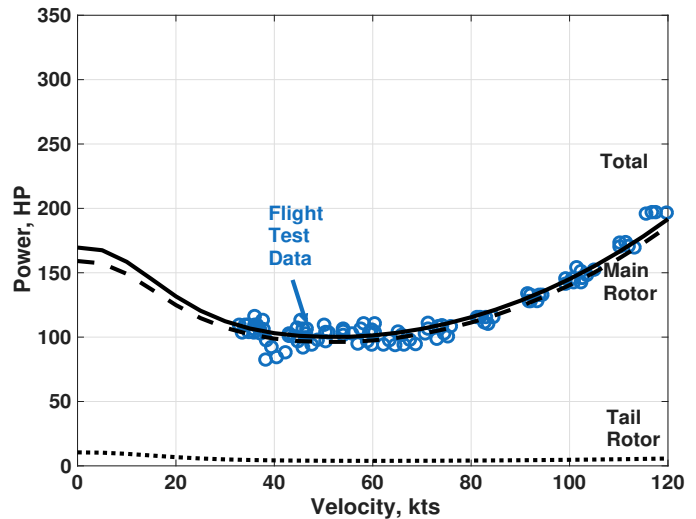


Figure 17. Power versus speed of a normal / generic R44; 2200 lb; Predictions in lines and test data in symbols.

Next, we turn to the fuel cell aircraft. The gross weight is increased to 2500 lb. The drag penalty from the nacelles is added, increasing the total drag area to 10 ft². The predicted aircraft power is shown in Figure 18. From 2200 lb to 2500 lb there is an increase in induced power affecting hover and low speed. From without nacelles to nacelles there is an increase in parasite power affecting cruise beyond 40 kts.

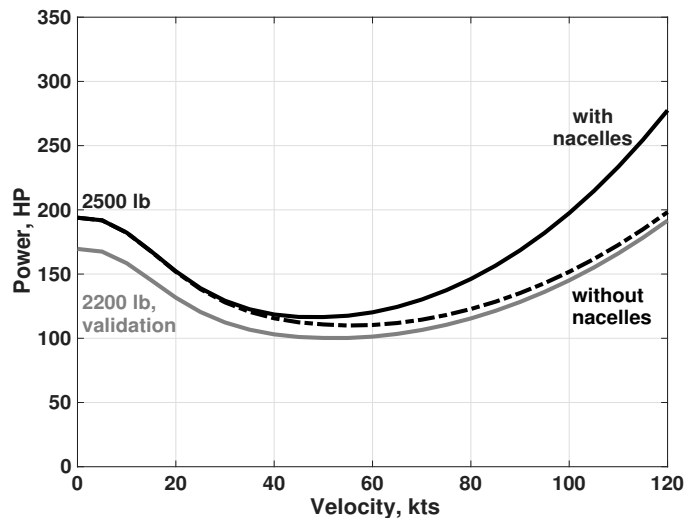


Figure 18. Predicted power of fuel cell R44; 2200 and 2500 lb, with and without radiator nacelles.

The predicted aircraft power, shown again as a dashed line in Figure 19, is assumed to be the net electrical power output of the fuel cell. This power is input to the fuel cell model to

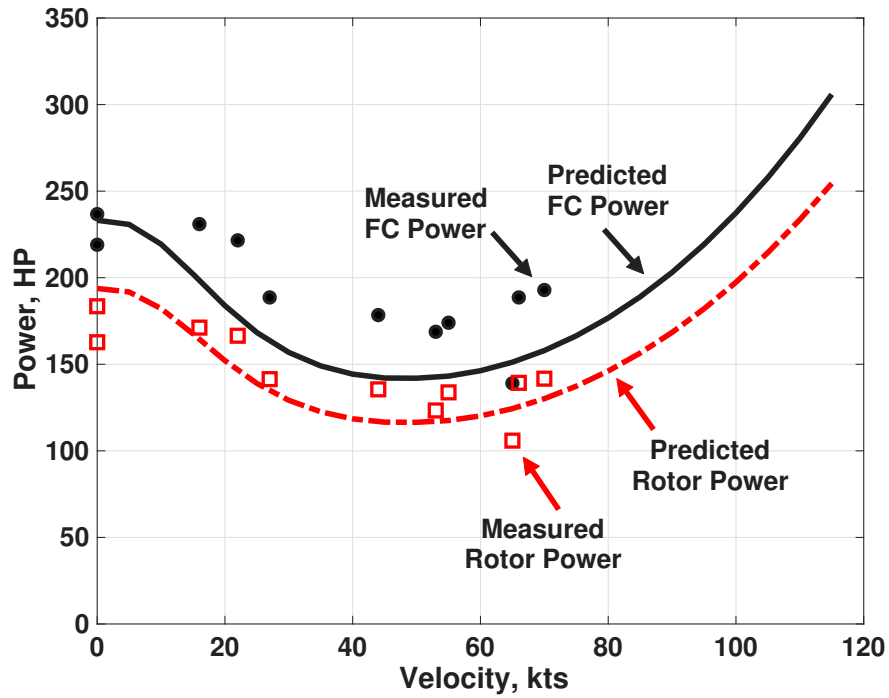


Figure 19. Measured and predicted power of hydrogen-electric R44.

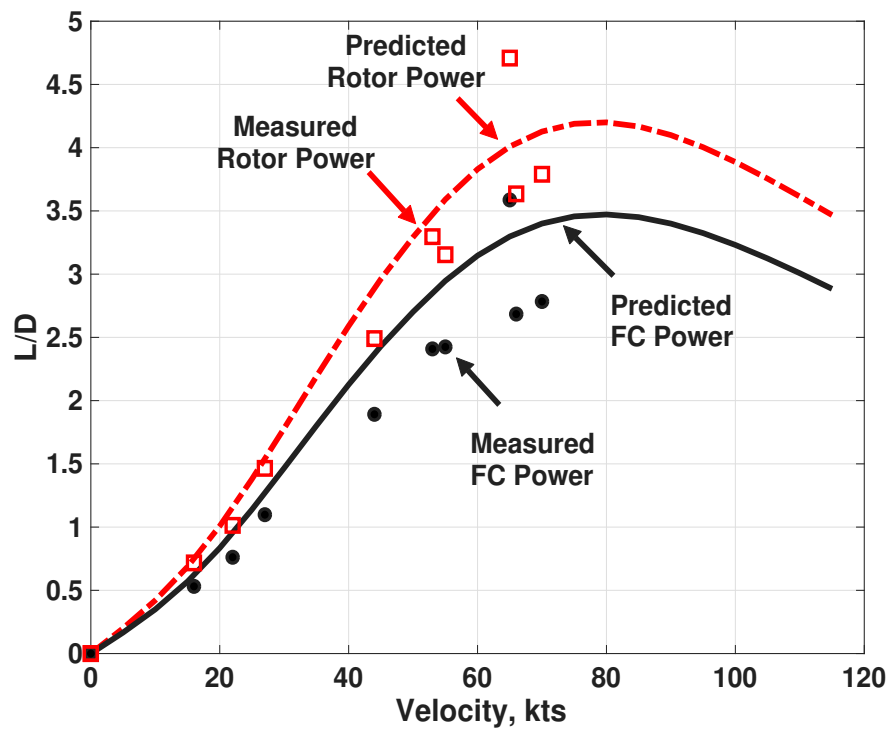


Figure 20. Measured and predicted aircraft L/D of hydrogen-electric R44.

calculate the gross power output, including balance of plant losses, which is shown as the solid line.

The power measured during flight are both the shaft power and the fuel cell power, current $\times$ voltage. These measurements are also shown in Figure 19 as symbols. The shaft power is the aircraft power (labeled as rotor power). The fuel cell power is the shaft power plus all the balance of plant losses (labeled as FC power). They are close in hover but deviate in cruise. The shaft power is generally well predicted, but the fuel cell power shows a difference. The reason is not very clear, especially as the hover data is well predicted, where fuel cell and its balance of power is near maximum. Nevertheless, the general trend is similar.

The power is then converted to lift-to-drag ratio, $L/D = WV/P$, and is shown in Figure 20. The solid line uses predicted fuel cell power (labeled as FC power), and the dashed line uses predicted aircraft power (labeled as rotor power). These are compared with flight test that use measured engine power, shown as symbols. The aircraft L/D peaks to around 4 near 80 kts.

With validations completed for both the powerplant and aircraft tools, attention is turned to the conceptual conversion of a larger, more capable aircraft—the R66, now using liquid hydrogen.

HYDROGEN R66

The Robinson R66 is now converted to a hydrogen powered fuel cell helicopter. This conversion is conceptual, but in preparation for future development. The objective is to maximize its payload-range envelope.

First, the helicopter model of a turbine engine R66 aircraft is validated. Next, the fuel cell is sized, building on knowledge gained from the R44. Finally, its payload-range envelope is explored.

The R66 is similar in gross weight to the R44, so all calibrations can be carried over. Liquid hydrogen is considered instead of compressed gas, so hydrogen storage is the principle difference between the two. The R66 offers more volume for fuel storage, which is a significant benefit.

Baseline R66

The (maximum) gross takeoff weight W_{TO} is 2700 lb. It consists of an empty weight (structural W_S and propulsion W_P) of 1280 lb, fuel W_F of 493 lb, and payload W_{pay} of 927 lb.

$$W_{TO} = W_S + W_P + W_F + W_{pay} \quad (13)$$

The powerplant weight is not available in public, but the isolated engine dry weight is reported as 201 lb. A 25% overhead results in a weight of 250 lb. The structure is therefore assumed to be $1280 - 250 = 1030$ lb.

The 493 lb of fuel is stored in a tank of volume 74.6 US gallons, of which 73.6 US gallons are usable. The fuel is Jet A or A1 with density 6.7 lb/gal (784 kg/m³). An additional volume of 134 US gallons is available in a baggage compartment.

Thus, for purposes of conversion, the extra space of 134 US gallons is available.

The powerplant is a single Rolls-Royce 250-C300/A1 engine (commercial designation RR300) with manufacturer rating of 300 HP (224 kW), derated to a 5-minute takeoff rating of 270 HP (200 kW), and continuous rating of 224 HP (167 kW).

The fuel cell powerplant will be designed for 200 kWe net electrical. This is an over-design, as the 200 kW takeoff rating is in fact the gross power, but it serves to keep ample margin.

Validation of Performance

The performance of the baseline turbine helicopter is validated extensively with level flight test data for a range of gross weights (2200–2700 lb) and altitudes (3000–12000 ft). The conditions presented in this paper however are only those for which data was already published by UMD and Robinson (Ref. 17). They correspond to a gross weight of 2200 lb, altitudes of 3696 ft and 4300 ft, and two aircraft CG locations.

The variation of power with speed is shown in Figure 21. The symbols are test data and the lines are predictions. The tail rotor torque was also published, although the conditions were not stated; nevertheless, that torque was converted to power using the published tail rotor speed, and is also shown in Figure 21. In general, the tail rotor power is within 5% of the main rotor power, which is well captured by the analysis. The last data point, at 145 kts, is known to be a maneuver, not a level flight.

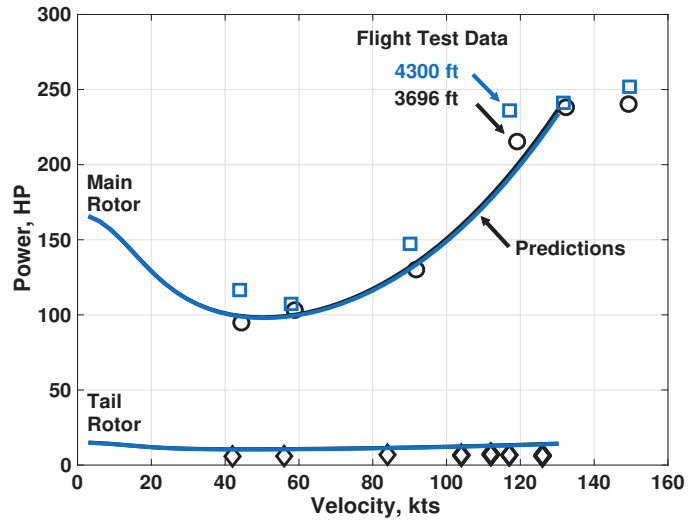


Figure 21. Main and tail rotor power; 2200 lb.

The aircraft $L/D = WV/P$ requires the total aircraft power. These values are not published; however, the addition of main and tail rotor powers provides a reasonable approximation. These approximate values are shown in Figure 22. The predictions follow the trends satisfactorily, within the limitations of a simple analysis.

The predictions of collective θ_{75} , lateral cyclic θ_{1C} , longitudinal cyclic θ_{1S} , aircraft pitch α_S , and roll ϕ_S are indicators of the fidelity of the analysis.

These are shown in Figures 23 to 27. They generally follow the flight test data with a level of accuracy expected from a simplified analysis.

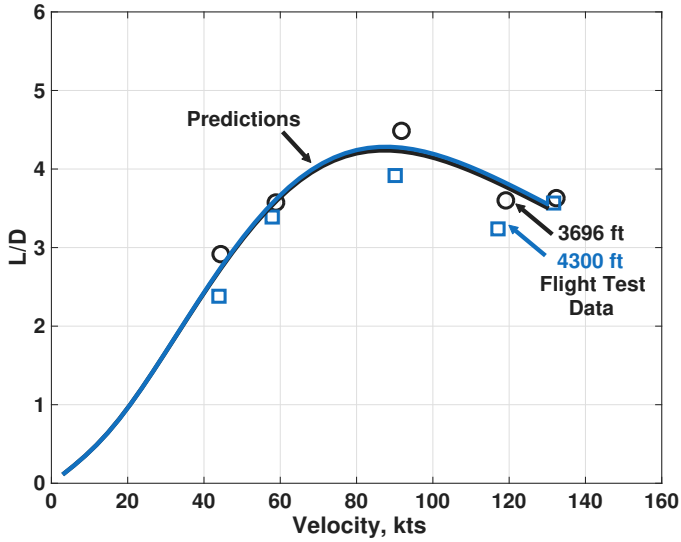


Figure 22. Aircraft L/D; 2200 lb.

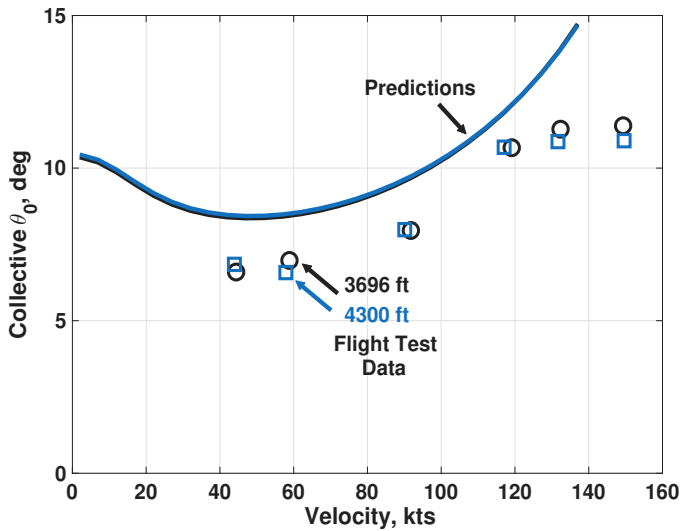


Figure 23. Collective angle; 2200 lb.

Once validated, the analysis is used to predict performance at the target gross weight of 2700 lb, with the same radiator nacelles attached on both sides, just like the converted R44. Because the power levels are similar (200 kW versus 184 kW), and because the R44 was oversized, this is justified.

Figure 28 compares the predicted performance with and without nacelles. Predictions for the 2200 lb without nacelles are also shown. The expected rise in power at hover and low speeds is apparent with the rise in gross weight. The consequence of the nacelle drag penalty is also observed beyond 40 kts, and it is very significant.

Figure 29 shows a loss in maximum L/D of nearly 0.8. It is clear therefore that the HTC designs would significantly

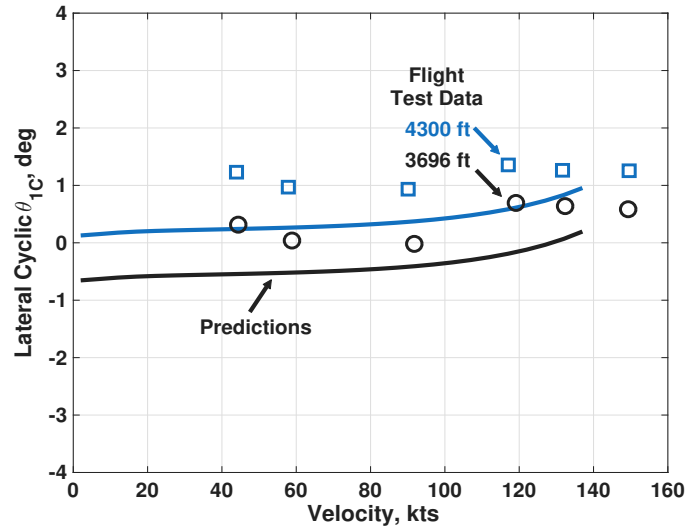


Figure 24. Lateral cyclic angle; 2200 lb.

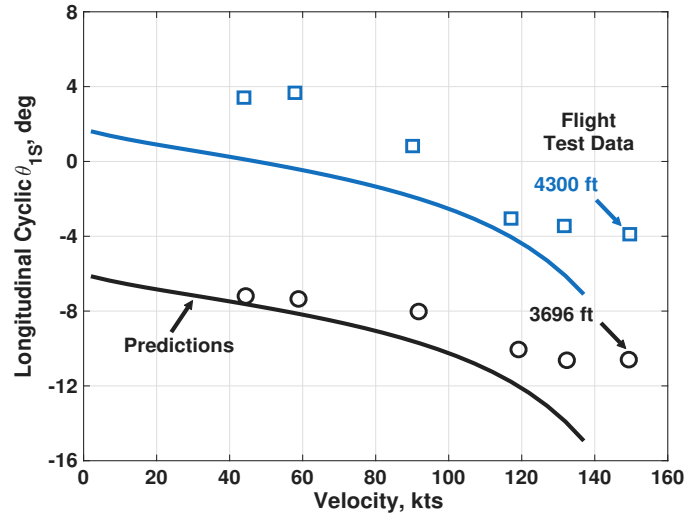


Figure 25. Longitudinal cyclic; 2200 lb.

benefit from designs more streamlined than two large nacelles hanging externally.

Figure 30 shows the specific range for R44 and R66 with and without nacelles. Specific range is the ratio of speed to fuel flow. The speed for best range (location of maximum specific range) drops from 100 kts to 85 kts for both aircraft as soon as the nacelles are included.

Conversion to Electric Drive

With baseline performance established, and nacelle drag included, we transition to the conversion of the powerplant from turbine to fuel cell.

The original drive is retained, only the engine is replaced with electric motor and fuel cell. From the engine to the main rotor there is a 14.75 : 1 reduction in RPM (13 : 37 from engine to drive line and 11 : 57 from drive line to main rotor, per R66 operator manual).

The main rotor power of 200 kW and rotational speed of 408 RPM (from tip speed of 705 ft/s and rotor radius 16.5

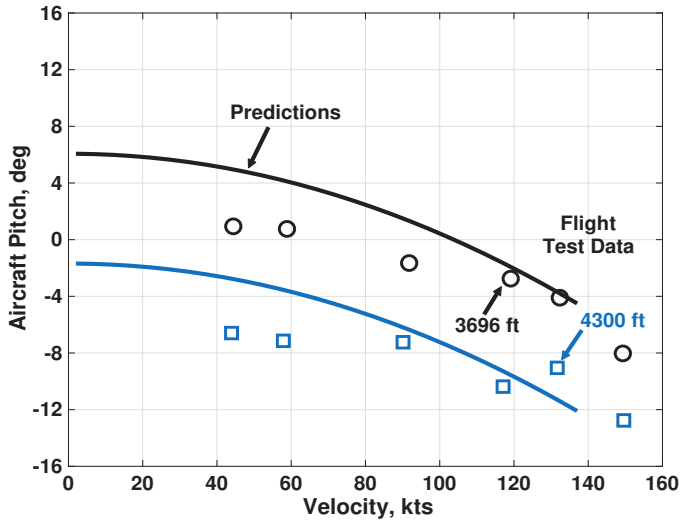


Figure 26. Aircraft pitch; + up; 2200 lb.

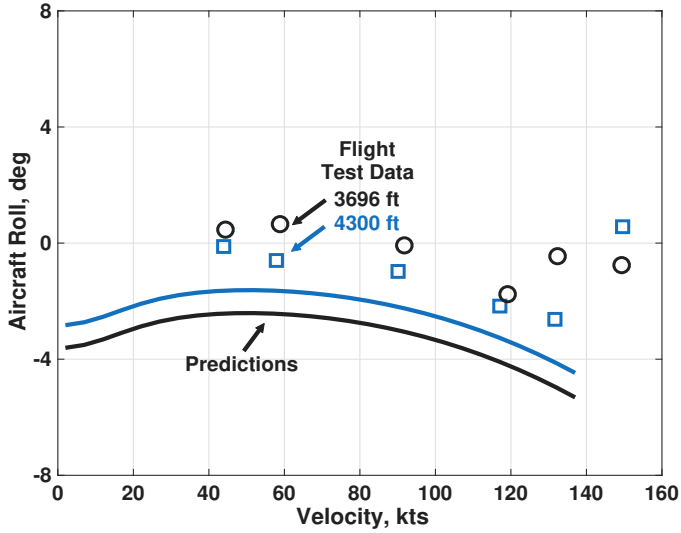


Figure 27. Aircraft roll; + right; 2200 lb.

ft) gives a torque of 4700 N-m. The engine torque is then $Q = 4700/14.75 = 317$ N-m and $RPM = 408 \times 14.75 = 6018$. The motor is sized to this torque.

The regression model by Mills (Ref. 18) is used, as given in Equation 14. The motor model is based on 49 commercial permanent magnet motors up to 400 N-m in torque with $\pm 20\%$ error. W is in kg and Q is in N-m. With torque levels greatly surpassing 13 N-m, the liquid cooled motors are needed.

$$\begin{aligned} W &= 0.2519 Q^{0.5857} & Q \leq 13 \text{ N-m (air-cooled)} \\ &= 0.1062 Q^{1.0422} & Q > 13 \text{ N-m (liq-cooled)} \end{aligned} \quad (14)$$

The motor controller model is based on 45 commercial electronic speed controllers up to 360 kW with $\pm 23\%$ error. Here W is in kg and P is in kW. The equation for liquid cooled is used.

$$\begin{aligned} W &= 0.0497 P^{0.7897} & \text{(air-cooled)} \\ &= 0.4655 P^{0.6188} & \text{(liq-cooled)} \end{aligned} \quad (15)$$

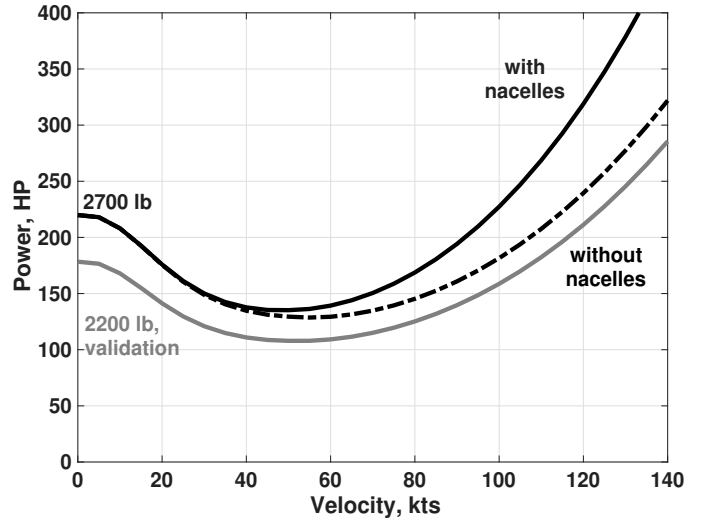


Figure 28. Aircraft power; 2700 lb.

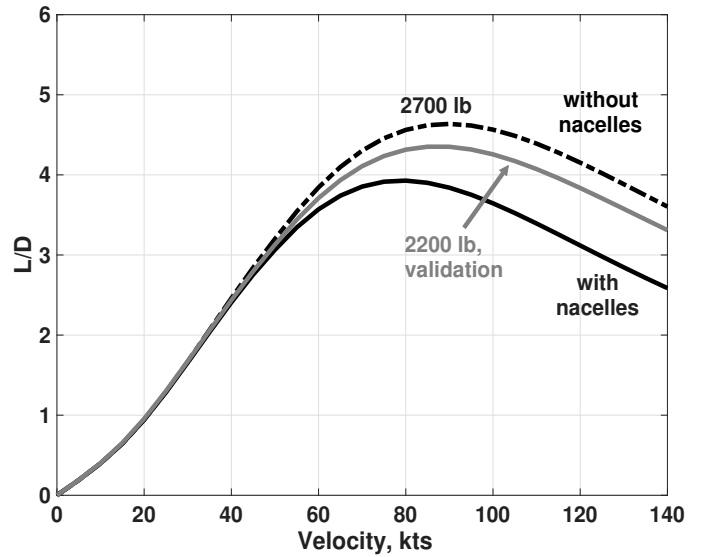


Figure 29. Aircraft L/D; 2700 lb.

Substituting torque and power values in the above equations imply a motor mass of 42 kg and controller mass of 13 kg. For reliability, two motors (each providing half the torque) are desired. Thus, if one motor fails, the other can provide some power to land immediately. There is no weight benefit as the model is almost linear with torque: $2 \times 0.1062 (Q/2)^{1.0422} \approx 42$. An additional 30 kg is allotted for wiring, support structure, and margins, bringing the total estimate to 85 kg.

Fuel Cell Powerplant

From Figure 28, the hydrogen R66 requires 230 HP (170 kW) to hover, and 160 HP (120 kW) for best range cruise. Nonetheless, the stack is sized to the maximum derated power of 200 kW.

Table 3 shows three fuel cell systems, all sized to the same design condition of the earlier hydrogen R44, only varying in net power. The reference powerplant supplies 100 kW per stack, for a combined total of 200 kW. The other two cases

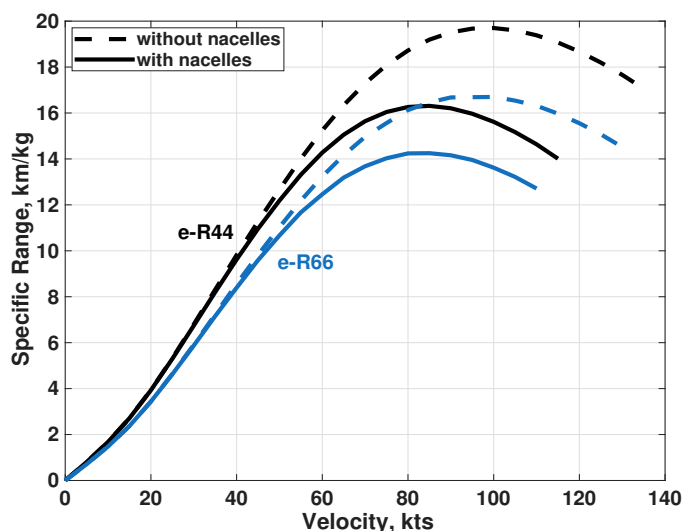


Figure 30. Specific range versus velocity, with and without nacelles.

supply 90 and 80 kW per stack, for a combined total of 180 and 160 kW, respectively.

Table 3. eR66 Powerplant weights (kg) for three design powers.

Subsystem	Unit	80 kWe	90 kWe	100 kWe
Stack	kg	106	116	126
Air	kg	40	45	49
HTC	kg	98	108	117
LTC	kg	19	21	23
Total	kg	263	290	315

Liquid Hydrogen

Liquid hydrogen storage is complex. A notional diagram of a tank is shown in Figure 31. Liquid H_2 occupies an inner tank insulated by vacuum from an outer tank. A pump maintains the vacuum. There are supports, a fill port (and a return channel for pressure relief) and a delivery line, all built of cryogenic materials. Inside the tank, hydrogen vapor accumulates (ullage). A heater controls boil-offs to supply, and level sensors for the liquid ensure reliable operation.

The mass of the tank system should include sensors, regulators, and relief valves, along with cryogenic fuel lines and the controller. The volume of the tank system should account for the outer tank and the space needed for vacuum pump, regulators, controller, and support.

There is a dearth of data on weights and volumes of these components for aeronautical applications. The few works that report useful data are described next.

The US Navy Ion Tiger project in the early 2010s remains the most well-documented work in peer-reviewed literature (Refs. 19,20), albeit in small scale. It flew a small 16 kg airplane using 1.34 kg of LH_2 . The mass of the aluminum storage tank was 3.68 kg, and tank system mass including

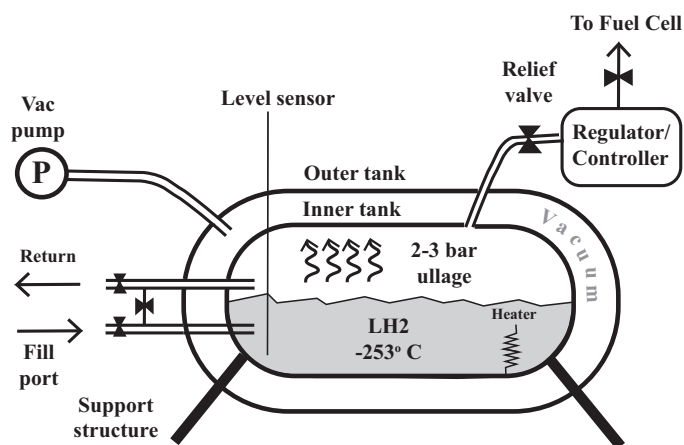


Figure 31. Notional LH_2 storage system.

tank, fuel lines, sensors and regulators was 5.87 kg. Thus, the weight fraction of hydrogen was approximately 0.185. The tank external and internal volumes were 31.63 L and 20.46 L, respectively. The resulting volume fraction of hydrogen storage was reported as 0.65. The mass of hydrogen and the internal volume indicate a density of 65 kg/m^3 , slightly lower than the theoretical 70.99 kg/m^3 , to account for ullage.

Hylum Industries of South Korea have reported the development of a LH_2 tank for an unmanned airplane over the last decade (Refs. 21–23). However, beyond a LH_2 mass of 0.85 kg, aluminum tank mass of 2.8 kg (Ref. 22) or 2.6 kg (Ref. 23) quoted variously, and tank internal volume of 13.2 L, no regulator or controller weights are reported, nor outer vessel volume. The mass of hydrogen and internal volume indicate a density of 64 kg/m^3 .

A European Union-supported project led by Josef Kallo of H2Fly/DLR in the late 2010s and early 2020s, demonstrated LH_2 flight of a piloted aircraft, with two pilots on board. The project, called high power density fuel cell system for aerial passenger vehicle fueled by liquid hydrogen (HEAVEN), is of a scale relevant to the present work. A presentation by Jeunesse at the ICEC29/ICMC Conference in 2024 reports a LH_2 mass of 16 kg, tank mass of 54 kg, and total tank system mass of 125 kg (Ref. 24), resulting in a weight fraction of 0.11. No volumes were reported. The tank material was aluminum, with some titanium and steel parts. The same group at H2Fly later partnered with Joby to demonstrate LH_2 flight of Joby's converted, rotary-wing aircraft. The flight was unmanned. The LH_2 mass was 40 kg; no other details were reported.

Gloyer, of Gloyer-Taylor Industries, have reported the development and testing of LH_2 tanks with advanced composites (Ref. 25). When completed, these tanks are envisioned to contain 19 and 50 kg of hydrogen and weigh 15 and 30 kg, respectively, implying weight fractions of 0.55 and 0.62 for the tank alone. This number will reduce with regulators and other ancillary equipment but still very high compared to aluminum state-of-the-art.

Based on these data, and the volume available in the R66 helicopter, the amounts of LH_2 that can potentially be car-

Table 4. Mass and volume of LH2 in R66; volume fraction 50% ($s_f = 0.5$) and density 65 kg/m³.

	Unit	Small	Large
US units			
Vol useful	US gal	74	207
Vol LH ₂	US gal	37	104
Mass LH ₂	lb	30	83
Mass tank sys	lb	300	830
SI units			
Vol useful	m ³	0.28	0.78
Vol LH ₂	m ³	0.14	0.39
Mass LH ₂	kg	9	25
Mass Tank Sys	kg	90	250

ried are calculated in Table 4. The small volume corresponds to the currently available fuel tank volume. The large volume corresponds to the fuel tank plus baggage compartment volume. The mass of the tank is calculated using the EU project weight fraction of 0.11 as this level of storage has already been demonstrated in (fixed-wing) flight. Later, in the R66 Performance sub-section, the weight fraction will be increased up to 0.3, but the volume will always be restricted to the available volume shown in Table 4.

Aircraft Performance

The powerplant weight is replaced with the new electric drive and the fuel cell system. The structural weight is increased by the hydrogen tank weight. What remains must be payload and fuel.

The fuel is restricted by the volume available. Consider the two cases of small and large volumes available, as presented in Table 4. From the volume available, assume a 50% volume fraction to calculate the volume of liquid hydrogen. This accounts for external tank and other equipment. The weight of hydrogen is obtained by multiplying with density. A reduced value of 65 kg/m³ is considered to account for some ullage.

The weight of the hydrogen tank is calculated assuming a weight fraction of 0.11. This is considered to be the reference case—the worst case scenario, with the most conservative tank weight fraction.

The payload-range plot has two segments. The segment on the top is where more and more fuel is carried. A trade-off occurs between the addition of fuel and payload weight. The take-off weight is fixed. The starting point is where no fuel is carried, so the entire fuel weight can be carried as additional payload. Without fuel, the range is zero. The ending point is where the fuel tank is full. The payload is the design value. The corresponding range is the design range.

The segment on the right is where the tank is full but the payload is reduced gradually to zero. Thus, the take-off weight is reduced. The starting point is the end point of the first segment. The end point is where the payload is deleted. The range is maximum, as the tank is full and the aircraft has minimum weight.

The payload-range calculation is carried out as follows. At every time step, depending on the aircraft weight and condition (hover, cruise, climb/descent), the required engine power is calculated. For cruise, the best range speed is calculated, and the power is taken at that speed (normally 99% of the best range speed, to allow the aircraft to fly faster at the expense of reducing the range by 1%). Power is used to calculate the fuel flow rate. The fuel consumed updates the aircraft weight. The range is calculated by multiplying the cruise speed and the time step. The process is repeated until the tank is empty. Note that the power, best range speed, and L/D changes continuously with weight.

Figure 32 shows the payload-range for the reference case, with a liquid hydrogen weight fraction of 0.11. The solid and dashed lines represent the small and large volumes, respectively. The best range speed is 87 kts, where the L/D is around 3.7 or less.

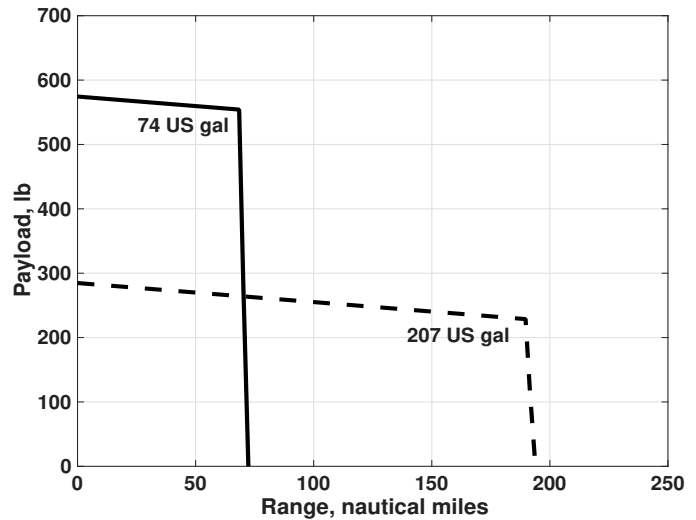


Figure 32. Payload-Range reference configuration.

Six parametric variations are performed next for:

- (1) aircraft lift-to-drag ratio,
- (2) hydrogen volume fraction,
- (3) hydrogen weight fraction,
- (4) HTC system weight,
- (5) stack system weight, and
- (6) design power.

The first two are important for range, while the next four are important for payload. The aircraft cruises at the best range speed (highest L/D) until fuel runs out.

Figure 33 shows the impact of L/D . Because L/D varies with speed (normally increasing with fuel burn), these are average values. As L/D increases, less power is required by the aircraft, resulting in lower fuel burn. From the analysis shown earlier, it is clear that to improve the L/D it is necessary to optimize the design of the high temperature cooling system, and avoid having external nacelles to store the radiators. This

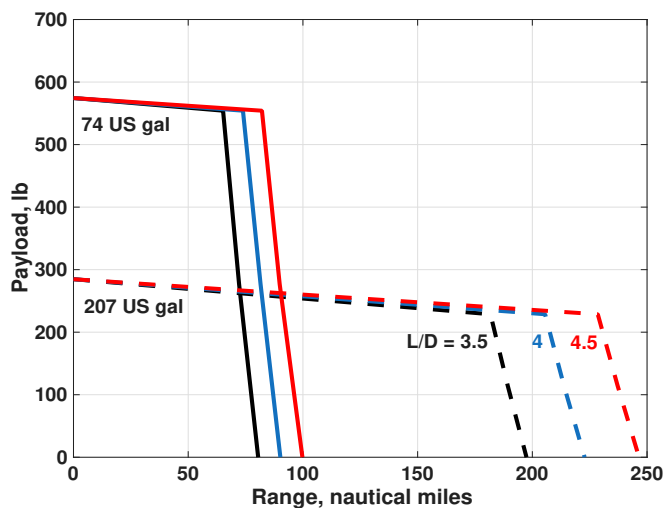


Figure 33. Payload-range with varying lift-to-drag ratio.

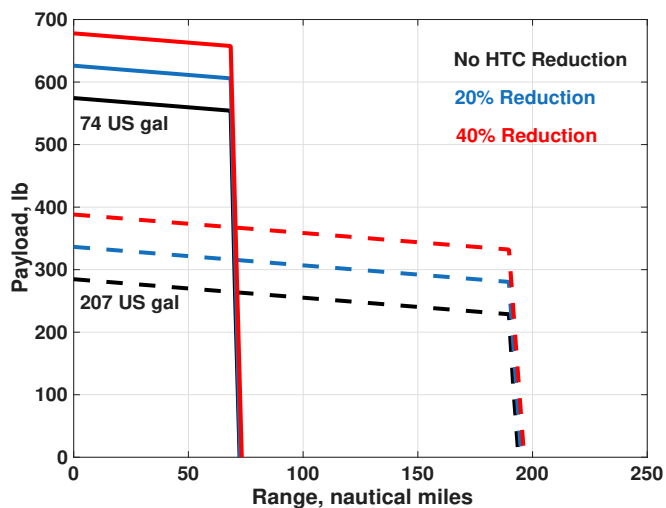


Figure 36. Payload-range with reduction in HTC weight.

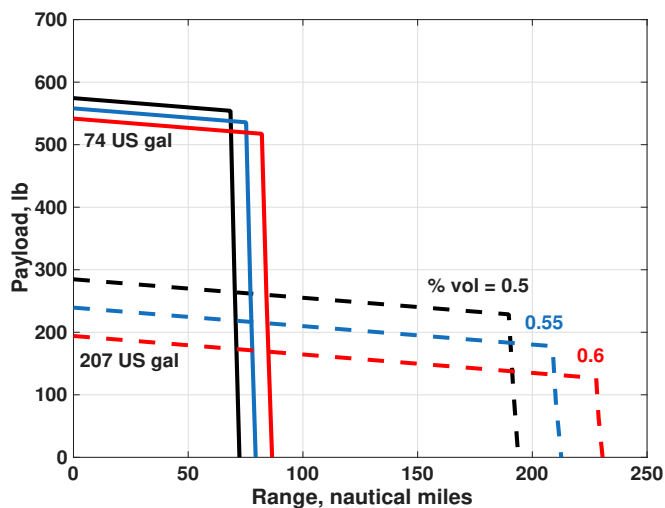


Figure 34. Payload-range with varying volume fraction.

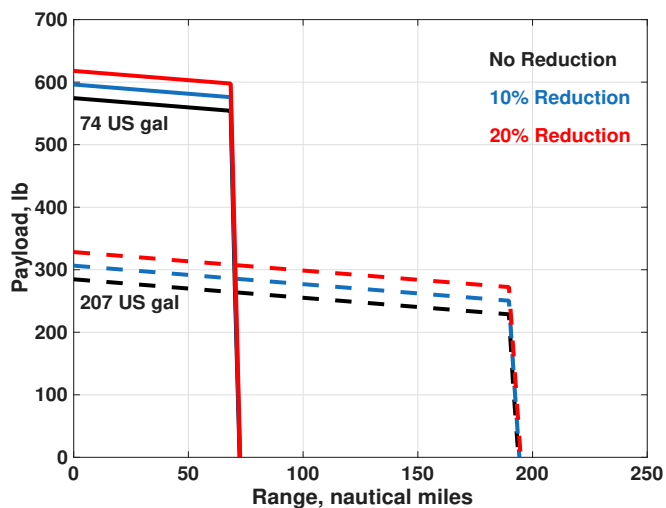


Figure 37. Payload-range with reduction in stack weight.

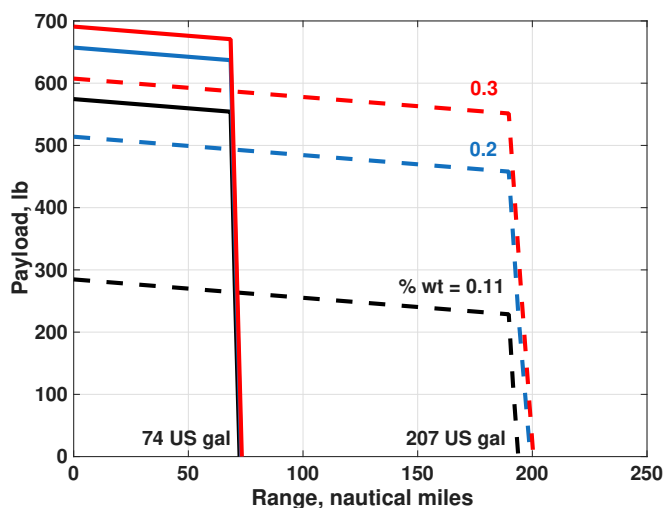


Figure 35. Payload-range with varying weight fraction.

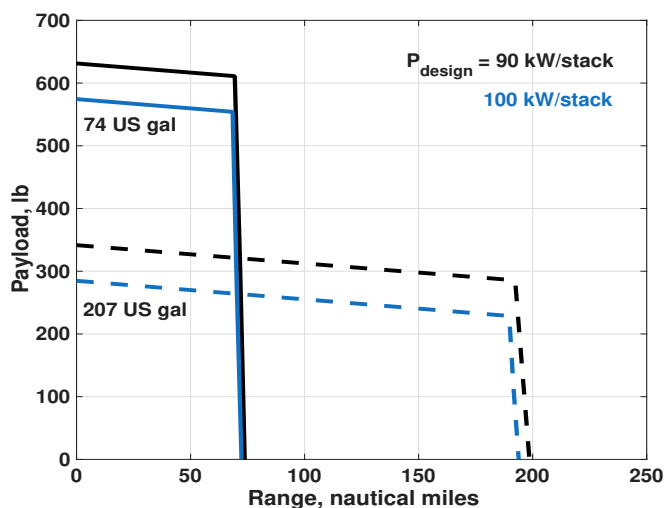


Figure 38. Payload-range with varying design power.

will reduce the drag and allow the aircraft to operate at L/D up to 4.5. This gives a range increase of 20 – 50 nm with no loss in payload.

Figure 34 shows the impact of LH_2 storage volume fraction. An increase in volume fraction allows more liquid hydrogen in the same volume, hence allowing the aircraft to cover a larger range. A range of 15 – 35 nm is gained with some loss in payload.

Figure 35 shows the impact of weight fraction. Increased weight fraction decreases tank weight and allows more payload. A lighter tank also implies less support structure. Weight fraction up to 0.3 is considered. This gives a payload increase of 100 – 300 lb with no loss in range.

Figure 36 shows the impact of high temperature cooling (HTC) system weight. Reduced HTC system weight decreases overall powerplant weight and allows more payload. Reduction of up to 40% is considered. This gives a payload increase of 50 – 100 lb with no loss in range.

Figure 37 shows the impact of stack weight. Reduced stack weight decreases overall powerplant weight and allows more payload. Reduction of up to 20% is considered. This gives a payload increase of about 20 – 45 lb with no loss in range.

The last parameter varied is the design power. Even though the derated engine power of the current aircraft is 200 kW, the hover power of the fuel cell aircraft is only 170 kW. So it is of interest to evaluate the gain, if any, of a less powerful engine. Figure 38 shows the impact of a 180 kW engine. The gain is about 55 lb of payload.

A battery will be integrated to provide burst power when demanded. One of the benefits of a battery is that the powerplant can be resized to an even lower design power, taking advantage of its higher specific power than a fuel cell.

Figure 39 compares the reference fuel cell range with that of a battery powered R66. A pack level specific energy of 200 Wh/kg available and installed is assumed. The original turbine R66 design payload-range is also shown. The battery powered R66 would be able to carry the turbine payload but its range would be limited to about 35 nm. The reference fuel cell R66 carries only 60% of the original payload (554 lb) but reaches around 70 nm. Thus, a hybridization might be of some benefit.

Consider two hybrid fuel-cell/battery systems. In one, the fuel-cell is sized to 180 kW (90 kW/stack), and the battery provides the remaining 20 kW. In the other, the fuel-cell is sized to an even lower 160 kW (80 kW/stack), and battery provides the remaining 40 kW. The battery has the same specific energy of 200 Wh/kg, available and installed as before, and can operate for five minutes (12-C). The payload-range of these two are compared with the baseline fuel-cell sized to 200 kW in Figure 40. Sizing the powerplant to a lower power provides a payload increase of about 80 lb, with no loss in range. Thus, there is some, although little, benefit in hybridizing with a battery.

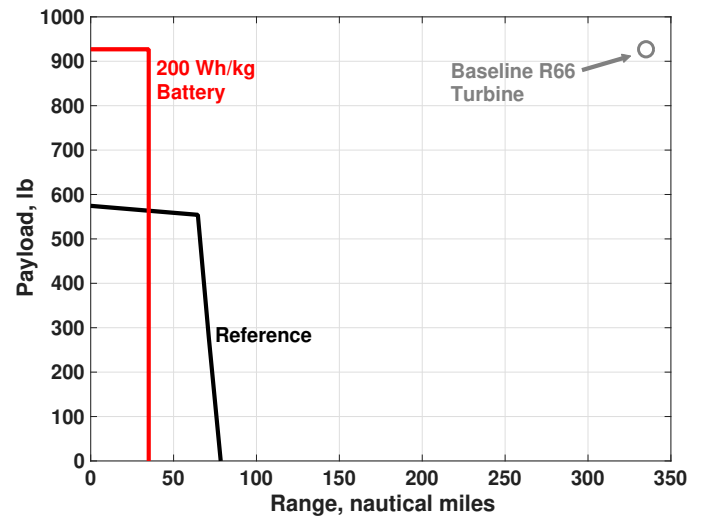


Figure 39. Payload-range of turbine and 200 Wh/kg battery R66.

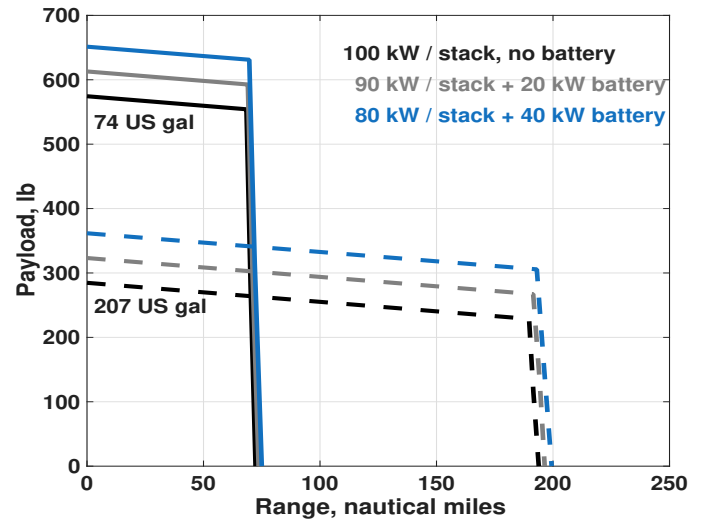


Figure 40. Payload-range for cruise at best range speed with battery.

Three Summary Cases

To summarize the improvements of the most influential parameters on the payload-range performance, three configurations are presented. These are listed in Table 5. While prior results explored the improvements one by one, here they are considered together. Three cases are envisioned. The reference case is the worst of all, where no improvements are achieved. The maximum case is where almost all improvements are achieved but still with sensible margins. The moderate case is in between.

Figure 41 compares the different payload-range capabilities of each configuration. The design payload-range point of a turbine R66 is marked as well. The maximum improved case results in a design payload of 713 lb and achieves a range of 205 nm. This is an improvement of about 30% in payload but more than 3-times the range when compared to what the reference case offers.

Table 5. Powerplant variations to increase fuel cell R66 performance.

	Units	Reference	Moderate	Maximum
	lb	2700	2700	2700
L/D		3.5	3.5	4
LH ₂ volume fraction		0.5	0.5	0.5
LH ₂ weight Fraction		0.11	0.3	0.3
Volume available	US gal	74	178	207
Design Power	kW	100	100	90
HTC Weight		Reference	-20%	-40%
Stack Weight		Reference	Reference	-10%
Payload	lb	554	629	713
Range	nm	64	155	205

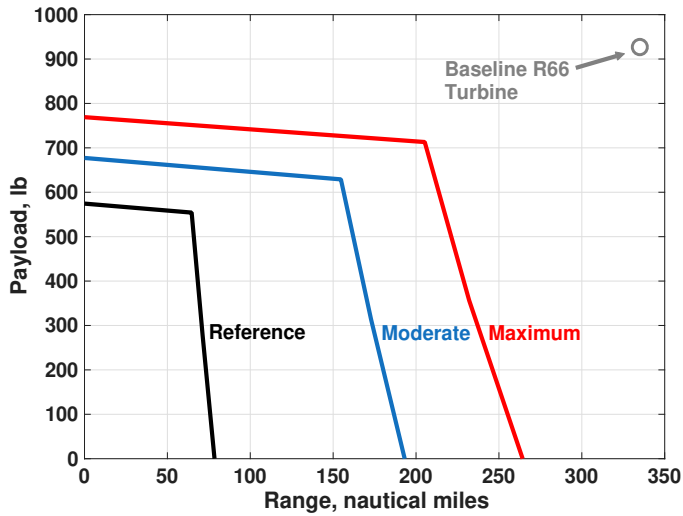


Figure 41. Fuel cell R66 performance for reference, moderate, and maximum improvement configurations.

Finally, Figure 42 shows the impact of hovering time on the range of each of the three cases. During hover, fuel is being consumed although the aircraft remains stationary. To maximize payload-range, hover time should be minimized to save LH₂ for cruise. With a 10-minute hover time, remaining LH₂ suffices to cover a range of 58 nm in reference, 160 nm in moderate, and 193 nm in the maximum improvement cases.

SUMMARY AND CONCLUSIONS

A piloted, full-scale, proton exchange membrane fuel cell powered helicopter, converted from a piston engine R44, was flight tested with 700 bar compressed gaseous hydrogen. Models were developed for the fuel cell, hydrogen flow, and the helicopter, and were validated with flight test data. The data covered powerplant architecture, stack electrical characteristics, hydrogen flow, detailed component weights, radiator drag, and full aircraft power measured in hover and forward flight. The validated models were then used to conceptually explore the conversion of a more capable, liquid hydrogen powered helicopter, using the turbine engine R66 as the target of conversion. Based on this work, the following key conclusions are drawn.

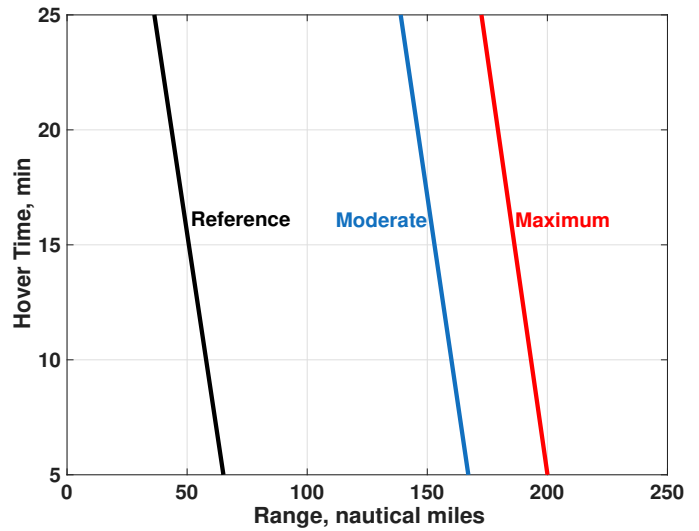


Figure 42. Hover time versus range for reference, intermediate, and maximum improvement configurations.

1. Piloted, full-scale, hydrogen fuel cell helicopters are feasible. The successful flight tests of an experimental, converted R44 helicopter, powered by gaseous hydrogen support this claim. Predictions anchored to test data indicated more capable missions are possible with liquid hydrogen, if larger helicopters like R66 are converted.
2. The test program established the fuel cell technology levels in performance and weight. The specific power was measured as 2.1 kW/kg with the stack alone, 1.6 kW/kg including the controller and support structure, and 0.63 kW/kg with the total system including stack, air, high and low temperature cooling. The overall efficiency of the fuel cell system, after balance of plant has been subtracted, is around 0.32 or so. The hydrogen storage was counted separately. The test program used gaseous hydrogen. A tank weight fraction of 0.055 was measured which is the state-of-the-art for a 700 bar compressed hydrogen storage.
3. Stack heating was a significant challenge, as expected, with the high temperature cooling system (stack cooling) dominating the balance of plant weight, accounting for

38% of the total fuel cell system weight. Note however, the baseline design was significantly over-sized, in view of the experimental nature of the program.

4. The test program established the cooling technology levels in weight and aerodynamic drag. Radiators were specially designed, instrumented, truck-tested on a full-scale rig, then integrated on the aircraft, and finally flight tested up to 70 kts. Measurements showed a 75% increase in airframe drag due to an additional 4.3 ft² or so of drag area. The resulting aircraft L/D was a modest 3.7 or so at around 70 kts.
5. Predictions for a converted R66, with liquid hydrogen replacing the gaseous, while other technologies remain similar, indicated payloads of 300 – 600 lb may be feasible over a range of 200 nautical miles with the current gross take off weight of 2700 lb. The principal requirement to achieve this range is that the baggage storage volume (134 US gallons) should be dedicated to hydrogen storage in addition to the current fuel tank volume (74 US gallons). Substantial benefit can also be derived by optimizing the cooling system, and not over-sizing it, to achieve an aircraft L/D of 4.0 or higher. The principal requirement to achieve the higher end of the payload is that the hydrogen storage weight fraction should be increased to 0.3. This increase is expected from gaseous to liquid storage.

The next step is the development and flight testing of the larger more capable liquid hydrogen helicopter, with the objective of meeting and beating the maximum performance predicted in this paper.

ACKNOWLEDGMENTS

This work is being carried out at the Alfred Gessow Rotorcraft Center at the University of Maryland under the United Therapeutics Corporation sponsored research agreement (no. GR10075) with Diallo Ibrahima and Mikaël Cardinal as technical monitors. We deeply acknowledge the support of Dr. Martine Rothblatt, Chairperson and Chief Executive Officer of United Therapeutics Corporation, for creating this exciting program.

REFERENCES

1. Brewer, G. D., *Hydrogen Aircraft Technology*, CRC Press, 1991.
2. Mills, G. L., Buchholtz, B., and Olsen, A., "Design, Fabrication and Testing of a Liquid Hydrogen Fuel Tank for a Long Duration Aircraft," AIP Conference Proceedings, Vol. 1434, No. 1, June 2012, pp. 773–780.
3. Lapena-Rey, N., Mosquera, J., Bataller, E., and Ort, F., "First Fuel-Cell Manned Aircraft," *Journal of Aircraft*, Vol. 47, No. 6, 2010, pp. 1825–1835.
4. Kallo, J., Flade, S., Stephan, T., and Schirmer, J., "Antares DLR H2 Test Bed for Electric Propulsion," Paper AIAA 2015-1305, 53rd AIAA Aerospace Sciences Meeting, January 2015.
5. Swartz, K. I., "World's First Piloted Hydrogen VTOL Lifts Off in Quebec," *Vertiflite*, Vol. 71, No. 3, May/June 2025, pp. 30–32.
6. Datta, A. and Johnson, W., "Requirements for a Hydrogen Powered All-Electric Manned Helicopter," Paper AIAA 2012-5405, 12th AIAA Aviation Technology, Integration, and Operations (ATIO) Conference and 14th AIAA/ISSMO Multidisciplinary Analysis and Optimization Conference, Indianapolis, IN, 17–19 September 2012.
7. Datta, A. and Johnson, W., "Power Plant Design and Performance Analysis of a Manned All-Electric Helicopter," *Journal of Propulsion and Power*, Vol. 30, (2), March 2014, pp. 490–505.
8. Pichitkul, A., Tiandam, J., Sankar, L. N., and Tantrairatn, S., "Engineering Study of Retrofitting a Piston Engine Powered Helicopter with Hydrogen Fuel Cells," 10th Annual Electric VTOL Symposium, Mesa, Arizona, USA, Jan. 24–26, 2023.
9. Ng, W. and Datta, A., "Hydrogen Fuel Cells and Batteries for Electric-Vertical Takeoff and Landing Aircraft," *Journal of Aircraft*, Vol. 56, No. 5, Sep-Oct 2019, pp. 1765–1782.
10. Ng, W., Datta, A., and Patil, M., "Hydrogen Fuel Cell and Battery Hybrid Architecture for Range Extension of Electric VTOL (eVTOL) Aircraft," *Journal of the American Helicopter Society*, Vol. 66, No. 1, Jan 2021, pp. 1–13.
11. Warwick, G., "Joby Beats Range Target for Hydrogen-Electric Air Taxi Demonstrator," *Aviation Week*, 11 July, 2024.
12. Swartz, K. I., "Piasecki Relaunches the Heliplex," *Vertiflite*, Vol. 69, No. 4, July/August 2023, pp. 30–37.
13. Ahluwalia, R. K., Wang, X. and Steinbach, A. J., "Performance of Advanced Automotive Fuel Cell Systems with Heat Rejection Constraint," *Journal of Power Sources*, Vol. 309, 31 March 2016, pp 178–191.
14. Takahashi, T., Ikeda, T., Murata, K., Hotaka, O., Hasegawa, S., Tachikawa, Y., Nishihara, M., Matsuda, J., Kitahara, T., Lyth, S. M., Hayashi, A., and Sasaki, K., "Accelerated Durability Testing of Fuel Cell Stacks for Commercial Automotive Applications: A Case Study," *Journal of the Electrochemical Society*, Vol. 169, No. 4, April 2022.
15. Datta, A., "PEM Fuel Cell Model for Conceptual Design of Hydrogen Fuel Cell Aircraft," NASA CR-20210000284, January 2021.

16. Holzman, J. C., Kish, B. A., Kimberlin, R. D., and Wilde, M., "Robinson R44 Performance Flight Test," IEEE Aerospace Conference, 6—13 March, 2021.
17. Sridharan, A., Chopra, I., and Turnour Stephen, "Mast Bumping Simulation and Mitigation Analysis for Teetering Rotor Systems," AHS Specialists Conference on Aeromechanics Design for Transformative Vertical Flight, Jan. 16-18, San Francisco, CA, USA, 2018.
18. Mills, B., *Fabrication and Fundamental Studies of a 4-kW, Variable-Voltage, Distributed Hybrid-Electric Powertrain for eVTOL Aircraft*, Doctoral dissertation, Digital Repository at the University of Maryland, 2022. <https://doi.org/10.13016/mxqi-gifl>
19. Swider-Lyons, K., Stroman, R., Rodgers, J., Edwards, D., Mackrell, J., Schuette, M., and Page, G., "Liquid Hydrogen Fuel System for Small Unmanned Air Vehicles," Paper AIAA 2013-0467, 51st AIAA Aerospace Sciences Meeting, 7-10 January 2013, Grapevine, TX.
20. Stroman, R. O., Schuette, M. W., Swider-Lyons, K., Rodgers, J. A., and Edwards, D. J., "Liquid hydrogen fuel system design and demonstration in a small long endurance air vehicle," *International Journal of Hydrogen Energy*, vol. 39, no. 21, July 2014, pp. 11279-11290.
21. Garceau, N. M., Kim, S. Y., Lim, C. M., Cho, M. J., Kim, K. Y., and Baik, J. H., "Performance Test of a 6 L Liquid Hydrogen Fuel Tank for Unmanned Air Vehicles," Institute of Physics Conference Series: Materials Science and Engineering, Vol. 101, No. 1., on Advances in Cryogenic Engineering: Proceedings of the Cryogenic Engineering Conference, 28 June—2 July, 2015, Tucson, AZ.
22. Gavrilovic, N., Mertika, S., Moschetta, J., Schimpf, J., Park, G., and Kim, S. Y., "Experimental Study on a Liquid Hydrogen Tank for Unmanned Aerial Vehicle Applications," *Journal of Aircraft*, Vol. 61, No. 6, November 2024, pp.1805-1816.
23. Mertika, S., Schimpf, J., Kim, S. Y., Moschetta, J., and Turpin, C., "Modeling and Experimental Testing of a UAV Liquid Hydrogen Propulsion Set," *International Journal of Hydrogen Energy*, Vol. 88, October 2024, pp. 1398—1409.
24. Jeunesse, L., "Cryogenic Tests of an Airborne Liquid Hydrogen Tank for a Manned Aircraft in the HEAVEN Project," Presentation 163, ICEC29/ICMC2024, Geneva, Switzerland, July 22-26, 2024.
25. Gloyer, P., "Ultra-lightweight LH2 Storage for LH2 Powered Aircraft and Other Vehicles," Presentation at the 3rd Annual H2-Aero Symposium and SAE AeroTech Conference, March 13–14, 2024, Charlotte, NC.