Due to their high energy density, proton exchange membrane (PEM) fuel cell systems are becoming increasingly attractive as the primary powerplant for low-power, long-endurance aircraft applications. Although PEM fuel cell technology has been applied for automotive and stationary use, limited design and experimental work has been performed and documented for actual aircraft applications. In order to better understand the design and performance tradeoffs for PEM fuel cell powered aircraft, a high-level conceptual design study of small-scale long-endurance aircraft is performed. This study builds upon design lessons learned through the development and flight testing of a PEM-powered demonstrator aircraft designed and built by the Georgia Institute of Technology. The study focuses on identifying and exploring the concept design space appropriate for small unmanned air vehicles with ranges of up to 5000 km flying at low altitudes with endurances of up to 64 hours. A Quality Function Deployment is used in conjunction with a Matrix of Alternatives to define multiple competing aircraft configurations based on current advanced technologies in PEM fuel cells, hydrogen storage, electric propulsion, aircraft design, and structural materials. A baseline propulsion system consisting of a liquid cooled PEM fuel cell with compressed hydrogen storage powering multiple electric tractor propeller motors was chosen. The corresponding baseline aerodynamic configuration consisted of a high-aspect ratio tapered wing with multiple tractor propellers. Eleven design variables governing the powerplant, propulsion, and aircraft design were chosen and used as inputs to a combination of surrogate and physics based models that were solved using fixed point iteration. Using range, endurance, climb rate, and aircraft mass as metrics, the problem was optimized using a sequential unconstrained minimization technique (SUMT) with an extended interior penalty function using a simplex optimization search algorithm. Several design constraints were active at the optimal solutions for both range and endurance. Results showed that the design was primarily driven by design variables governing hydrogen storage. The analysis also showed that optimizing a design for energy density did not produce the best aircraft design for either long range or long endurance. With the same payload, aircraft optimized for range and endurance were much smaller and had better range, endurance, and climb performance than aircraft optimized for energy density.
I. Introduction

LONG-ENDURANCE unmanned aerial vehicles (UAVs) are the subject of considerable interest among the aerospace community for their potential to perform reconnaissance and sensing missions that cannot justify the cost of space-based satellites because of lower priority, mission duration or value. In addition, long-endurance aircraft can exhibit lower capitol cost, better mission adaptability, faster mission cycle time and they require less of the radiation and temperature hardening of spacecraft. At present, a majority of the long-endurance aircraft available are powered by conventional gas turbine powerplants$^3$.

Polymer electrolyte membrane (PEM) fuel cell powerplants powered by compressed hydrogen have particular advantages over other technologies available for long-endurance aircraft because PEM fuel cell systems have high specific energy, high efficiency, improved environmental performance and can be incorporated into rechargeable energy storage systems$^2$. Where advanced batteries can reach electrical output energy densities of 200Wh/kg at the module level$^3$, fuel cells can achieve $>$800Wh/kg at the system level$^{45}$. Where advanced batteries have charge and discharge efficiency of nearly 100%$^6$, a fuel cell/electrolyzer rechargeable fuel cell system can have a charge efficiency of 80% and a discharge efficiency of 50%$^7$. The 2.5x efficiency advantage of batteries does not make up for the >4x energy density advantage of the fuel cell system. For comparison, a conventional gasoline-fueled internal combustion powerplant (SFC=600g/kWh, 11kg engine$^8$, 10hrs endurance, 3kW cruise, zero fuel tank weight) has a specific mechanical energy of approximately 1030Wh/kg, but is endurance-limited because it cannot easily be recharged in flight.

Although the aviation community’s interest in fuel cell aircraft is now well established, there exists a need for a comprehensive, documented analysis of the performance of practical long-endurance fuel cell aircraft. The Georgia Tech Research Institute, the Aerospace Systems Design Laboratory, the Daniel Guggenheim School of Aerospace Engineering, and the George Woodruff School of Mechanical Engineering at the Georgia Institute of Technology have designed and built a fuel cell aircraft as a technology demonstrator. The aircraft itself is novel as it is the largest fuel cell aircraft yet developed that is fueled by compressed hydrogen, and the largest fuel cell aircraft whose design is in the public domain. Despite the demonstrator aircraft’s achievements, its performance is limited by a relatively low power fuel cell and the performance of components not optimized for aircraft applications. However, the demonstrator aircraft did provide an ideal platform to gather performance data which were used for verifying and calibrating the analysis tools needed for fuel cell aircraft design. By building upon the calibrated design tools and implementing lessons learned during the demonstrator development, it is possible to make a detailed study of the performance enabled by using the state of the art, available fuel cell and hydrogen storage technology.

This paper focuses primarily on examining the feasible design space for medium-sized UAV’s (around 25-100 kg gross weight) based in part on the demonstrator aircraft, but implementing state of the art but commercially available fuel cell and hydrogen storage components. The paper identifies several aircraft with endurance over 2 days and ranges in excess of 3000 km. A methodology for design and analysis of fuel cell airplanes is proposed that includes aircraft platform selection via a matrix of alternatives analysis. The system performance of a baseline aircraft is then simulated by decomposing the aircraft into subsystems modeled using contributing analyses. These contributing analyses are ordered into a design structure matrix that can be solved iteratively. A high-level performance optimization scheme using an extended interior penalty function to handle design constraints is then implemented to maximize the range and endurance of the aircraft while maintaining specified performance and hardware constraints.

II. Technical Approach

A methodology for design and optimization of long-endurance, medium-scale fuel cell powered aircraft based in part on the Integrated Product and Process Development (IPPD) approach was used. First, the problem and requirements were defined. Based on the requirements, a Quality Function Deployment (QFD) was performed to identify conceptual configuration alternatives. These alternatives were organized into a Matrix of Alternatives and using a mainly qualitative analysis, a baseline configuration was selected. Based on the baseline configuration, a set of design metrics and design variables were identified. To calculate the design metrics as a function of the design variables, the analysis of the entire aircraft was decomposed into several contributing analyses that modeled the performance of the aircraft subsystems. Next, mathematical models for modeling the performance of the aircraft subsystems are proposed. The models present in these contributing analyses are validated, parameterized, surrogate mathematical models that are based on data-sets created from experimental data or physics-based modeling tools. The contributing analyses are then assembled intelligently into a design structure matrix that can be solved iteratively for viable aircraft configurations. The result is a computationally efficient mathematical model of the fuel cell aircraft performance that relates the subsystem-level design variables to the aircraft-level performance attributes.
and constraints. In the following sections, the details of the proposed technical approach are presented and discussed.

A. Problem Definition

Since fuel cells are an enabling technology for long endurance flight, the primary focus of this paper is to examine and determine what practical endurance and range values exist for a medium scale fuel cell powered UAV that can be fully developed within the next few years. To demonstrate fuel cell-powered long endurance flight, it was proposed that an aircraft be designed with a range sufficient for an Atlantic Ocean crossing. The shortest distance across the Atlantic Ocean is approximately 2,575 km. To allow for flexibility in starting and stopping locations for a transatlantic flight, a still air range of 2,900 km was set as a minimum. The altitude corresponding to this range was set at 300 m. This altitude is sufficient to allow a small aircraft to fly over shipping traffic and is still well below commercial airspace. The altitude is also the approximate altitude of Atlanta, GA which is ideal for local test flying.

In addition to meeting the still air range criterion, the aircraft must have enough excess power for low power maneuvers during the flight. Since maneuvering requires an excess of available power, a minimum climb rate was used as a metric governing aircraft performance. The minimum climb rate was set at 75 m/min.

The final criterion is based on payload. In order for the aircraft to be useful for more than just demonstration purposes, some payload capacity must exist. It was decided that a payload with mass of 2.27 kg at a continuous power requirement of 25 W would be adequate for many small but useful payloads.

B. Design Space Definition

The problem of designing a fuel cell powered UAV is inherently multidisciplinary in nature. To begin defining a design space, the problem was decomposed into system attributes and requirements that were related to the problem definition. These attributes and requirements were related using a Quality Function Deployment to develop conceptual configuration alternatives that were arranged into a Matrix of Alternatives. The Matrix of Alternatives is simply a collection of several alternative configurations relating to each defined attribute as shown in Table 1.

Table 1. Matrix of alternatives for long-endurance, medium-scale fuel cell aircraft analysis.

<table>
<thead>
<tr>
<th>Attributes</th>
<th>Alt 1</th>
<th>Alt 2</th>
<th>Alt 3</th>
<th>Alt 4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Configuration</td>
<td>Vehicle</td>
<td>Low Wing</td>
<td>High Wing</td>
<td>Biplane</td>
</tr>
<tr>
<td></td>
<td>Planform</td>
<td>Straight</td>
<td>Tapered</td>
<td>Elliptical</td>
</tr>
<tr>
<td></td>
<td>Fuselage</td>
<td>Cylindrical</td>
<td>Oval</td>
<td></td>
</tr>
<tr>
<td>Propulsion</td>
<td>Hydrogen Storage</td>
<td>Chemical Hydrides</td>
<td>Compressed Gaseous</td>
<td>Liquid</td>
</tr>
<tr>
<td></td>
<td>Type</td>
<td>Fuel Cell</td>
<td>Battery Hybrid</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Prop Position</td>
<td>Tractor</td>
<td>Pusher</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Cooling</td>
<td>Air</td>
<td>Liquid</td>
<td></td>
</tr>
<tr>
<td>Structures</td>
<td>Materials</td>
<td>Metal</td>
<td>Composites</td>
<td>Wood</td>
</tr>
<tr>
<td></td>
<td>Process</td>
<td>Monocoque</td>
<td>Space Frame</td>
<td>Combination</td>
</tr>
<tr>
<td></td>
<td>Landing Gear</td>
<td>Fixed</td>
<td>Retractable</td>
<td>Removable Dolly</td>
</tr>
</tbody>
</table>

As shown, the Matrix of Alternatives contains thousands of alternative combinations. Downselecting a design from the Matrix was based on both quantitative and qualitative calculations as well as lessons learned from the construction of the previous demonstrator aircraft.

To achieve a very high aerodynamic efficiency needed for long range flight, it was expected that a high aspect ratio wing would be needed. Based on experience with the demonstrator aircraft and the vast amount of historical data on sailplane planforms, a tapered wing configuration was chosen.

Based on the previous theoretical design studies of similar aircraft, it was known beforehand that hydrogen storage would be of primary design importance\(^5\). Therefore, it was determined that a cylindrical fuselage would be best suited for use with either spherical or cylindrical hydrogen tanks.

The demonstrator aircraft design effort included analysis and testing of both compressed and metal-hydride hydrogen storage. Based on the results, it was determined that for larger scale aircraft with long endurances, compressed hydrogen would offer more benefits and be easier to accurately analyze. Although using liquid
hydrogen has definite advantages, development challenges expected during detailed design would likely extend the timeframe of developing a working aircraft beyond the few years quoted in the problem statement.

For the primary powerplant, a fuel cell only configuration was chosen. A battery-hybrid system was considered to allow the fuel cell to be sized mainly for cruise with the battery sized for small time periods of increased power, but analyzing the hybrid system was not pursued to try and reduce the overall number of design variables and performance assumptions. The temporal modeling necessary to capture the benefits of hybridization were deemed to be beyond the scope of this conceptual design.

For the propulsion configuration, a tractor design was selected. It was expected that the hydrogen storage tank would result in a large diameter cylindrical fuselage that would be less detrimental to a tractor propulsion design if a motor were mounted on the fuselage.

Cooling of the powerplant system was assumed to be accomplished using liquid rather than air. This is based on the balance of plant experience gained during development of the demonstrator aircraft.

For the aircraft structures, the same structural design used in the demonstrator aircraft was chosen. This design was chosen since the authors had more historical information needed to accurately predict the installed weight and strength.

Finally, since the overall goal is achieving long range flight, landing gear are of secondary importance. Rather than adding the extra weight and drag associated with landing gear, it was determined that using a removable dolly for takeoff and landing on a skid would be better suited for the design goals.

In order to design an aircraft with the configuration shown in Table 1, 11 design variables and 2 response metrics were chosen. These variables and metrics fully describe the configuration of the aircraft and are defined in Figure 2.

C. Contributing Analyses

To simplify and make the analysis of the aircraft tractable, the multidisciplinary analysis was decomposed into several contributing analyses (CAs).

Aerodynamic CA

The aerodynamics of an efficient fuel cell powered aircraft are characterized by low power to weight ratio, high aspect ratio and high aerodynamic efficiency. This is the same design space as is occupied by the fuel cell demonstrator aircraft. The demonstrator aircraft has been exhaustively optimized for aerodynamic efficiency subject to Reynolds number, stall, climb, and turning constraints. Since flight tests proved that the design was a good compromise between aerodynamic efficiency, turning performance, and handling qualities, it was decided to maintain the same wing planform. Scaling of the wing area is allowed to accommodate aircraft larger than the demonstrator.

For conceptual design calculations, the wing airfoil used is a Selig-Donovan 7032. This airfoil is a highly efficient, low-Reynolds number airfoil and is used for all of the aircraft configurations considered. The aerodynamic contributing analysis was conducted using both offline and online calculations. Wings2004, a potential flow analysis code developed by Utah State University, was used offline to calculate induced drag, lift, and interaction effects between the wing and tail. The parasite drag of the wing was also calculated offline using profile drag numbers tabulated vs. Reynolds’s number based on wind tunnel tests of the Selig-Donovan 7032 airfoil. Online, the Aerodynamic CA estimated the fuselage lift and drag characteristics and used this information with the offline values to estimate the parasite drag of the aircraft and develop a drag polar of the aircraft. Most of the online calculations are based on the methods and equations provided by Roskam.

Empennage sections were analyzed assuming a NACA 0009 airfoil. Sizing of the empennage is based on maintaining a static margin (scaled by the wing chord) of 20 and an aircraft yawing moment coefficient of 0.15. Sizing of the tail is accomplished using an offline iterative method involving Wings2004 and was scaled online using the resulting tail volume coefficients.

Propeller CA

The propeller analysis is based on Goldstein’s vortex theory of screw propellers using the Betz condition. The propeller geometries used in this analysis are derived from measurements of several commercially available small-scale propellers. To account for propellers of varying diameter and pitch, the baseline propeller aerodynamic pitch distributions and the planform blade shapes are appropriately scaled while assuming that the baseline airfoil distribution along the blade span remains constant.

For any given propeller diameter, pitch, and number of propeller blades, the propeller thrust and power coefficients are calculated as a function of the propeller advance ratio. This allows the torque and the thrust of a given propeller to be calculated as a function of propeller RPM and airspeed. For this design, only two-bladed propellers are considered because the tip Mach numbers are expected to be in the incompressible regime.
Propeller Interference CA

The interference of the fuselage with the propeller is treated using another contributing analysis. This contributing analysis is based on the modification of a method originally suggested in Lowry\textsuperscript{14} which is capable of handling both tractor and pusher type configurations. For multiple motor combinations, interference effects are only calculated for the one motor/propeller is to be mounted to the fuselage for odd numbers of motors.

Fuel Cell Performance and Weight CA

The fuel cell system is modeled as a static polarization curve. The current output of the fuel cell is limited at the maximum power output power of the fuel cell to prevent low-efficiency solutions. The fuel cell stack mass, length, open circuit voltage, and internal resistance are scalable by the fuel cell active area and number of cells. The fuel cell size and mass scaling factors are based on the characteristics of a prototype stack with 0.48cm (3/16in) graphite bipolar plates, aluminum endplates and aluminum through-bolts. The performance of the individual fuel cells is equivalent to the published stack performance of the Gore 58 series membrane electrode assembly\textsuperscript{15}. This membrane electrode assembly is chosen as representative of the state of the art for self-humidified, low-pressure PEM fuel cells. The maximum current density achievable from the fuel cell stack is 1100mA/cm$^2$/cell and the maximum power density is 0.6W/cm$^2$/cell.

Fuel Cell Balance of Plant CA

The fuel cell balance of plant represents the air delivery, hydrogen delivery and regulation, water cooling and power management and distribution subsystems of the fuel cell. The electrical power consumption and mass of the fuel cell balance of plant are based on measurements of previously developed self-humidified, low-pressure fuel cell systems. The compressor power consumption and mass are scaled at 1.76W/L and 37.75g/L of standard air at maximum aircraft power, values representative of a diaphragm compressor. The water pump consumes 13W continuously and the radiator weighs 2.1g/W of fuel cell heat rejected at peak fuel cell power.

Compressed Hydrogen Storage CA

For this study, hydrogen is stored on board the aircraft as compressed hydrogen gas within a composite-overwrapped, aluminum-lined cylinder. The hydrogen storage system model provides a low-fidelity, conceptual design of the hydrogen tank using empirical data and simple mechanics of materials methods. Considerations such as fatigue life, shape optimization, winding schemes, etc., are left for a later, lower-level design task as they are not appropriate analyses for this conceptual design model. A summary of the design variables and relevant constants is provided in Table 2 and Table 3.

Table 2. Design variables used for hydrogen tank conceptual design.

<table>
<thead>
<tr>
<th>Design Variable</th>
<th>Value</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Composite Overwrap Maximum Stress ($\tau_{\text{maxcomp}}$)</td>
<td>1.9GPa</td>
<td>Kevlar-49/epoxy at 55% translation (Refs. 16,17)</td>
</tr>
<tr>
<td>Liner Density ($\rho_{\text{liner}}$)</td>
<td>2700kg/m$^3$</td>
<td>(Ref. 18), Aluminum 6061</td>
</tr>
<tr>
<td>Regulator Mass ($m_{\text{reg}}$)</td>
<td>0.35kg</td>
<td>(Ref. 19)</td>
</tr>
<tr>
<td>Composite Overwrap Density ($\rho_{\text{comp}}$)</td>
<td>1530kg/m$^3$</td>
<td>(Ref. 20)</td>
</tr>
<tr>
<td>Liner Thickness ($t_{\text{liner}}$)</td>
<td>0.762mm</td>
<td>(Ref. 22), Aluminum 6061</td>
</tr>
<tr>
<td>Liner Load Sharing</td>
<td>0%</td>
<td></td>
</tr>
<tr>
<td>Factor of Safety to Yield (FOS)</td>
<td>2.5</td>
<td></td>
</tr>
<tr>
<td>Tank Mounting/Bosses/Tubing Mass Fraction ($f_{\text{mount}}$)</td>
<td>10%</td>
<td>Based on (Ref. 20)</td>
</tr>
</tbody>
</table>

Table 3. Constants used for hydrogen tank conceptual design.

<table>
<thead>
<tr>
<th>Constants</th>
<th>Value</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>Faraday's Number ($F$)</td>
<td>96485C/mol</td>
<td>Ref. 21</td>
</tr>
<tr>
<td>Hydrogen Utilization ($\eta_{\text{util}}$)</td>
<td>90%</td>
<td>Ref. 22</td>
</tr>
<tr>
<td>Standard Hydrogen Density</td>
<td>0.0838kg/m$^3$</td>
<td></td>
</tr>
<tr>
<td>Hydrogen Storage Temperature ($T$)</td>
<td>293K</td>
<td></td>
</tr>
</tbody>
</table>

Based on the fuel cell characteristics and the required output power the mols of hydrogen required to complete the flight can be calculated based on the equation:
\[ n_{H_2} = \frac{n_{\text{cells}} I}{2F\eta_{\text{util}}} \]  

where \( I \) is the cell current, \( E \) is the aircraft endurance, and \( n_{\text{cells}} \) is the number of fuel cells. This equation assumes that 100\% of the hydrogen stored is extractable. The volume \( (V) \) required by this amount of hydrogen stored at a pressure \( (P) \) is calculated using the Redlich-Kwong equation:

\[ P = \frac{RT}{V - b} = \frac{a}{T^{\frac{1}{2}} V (V + b)}; a = 0.1425 K^{\frac{3}{2}} \cdot m^6 \cdot Pa \cdot mol^{-2}; b = 1.817 \cdot 10^{-5} m^3 \cdot mol^{-1} \]

The hydrogen tank is of cylindrical geometry with hemispherical end caps. The tank is subjected only to loading due to the uniform pressure difference between the internal hydrogen pressure and the external atmospheric pressure. The aluminum tank liner is assumed to be of constant thickness and does not contribute to the strength of the tank, but does contribute to its weight. In general, composite hydrogen tanks require metallic or polymeric liners to reduce the hydrogen leak rate. The thickness of the composite overwrap is specified to resist the hoop stress and the axial stress due to the pressure loads. The total composite thickness is equal to:

\[ t_{\text{total}} = FOS \left[ \frac{r(P - P_{\text{atm}})}{\tau_{\text{maxcomp}}} + \frac{r(P - P_{\text{atm}})}{2\tau_{\text{maxcomp}}} \right], \]

and the total tank mass is calculated using the formula:

\[ m_{\text{tank}} = (1 + f_{\text{mount}}) \cdot (m_{\text{liner}} + m_{\text{composite}}) + m_{\text{reg}} + m_{H_2} \]

**Electric Motor CA**

The electric motor analysis uses a conventional lumped parameter equivalent circuit model of the motor, as shown in Figure 1. This model is parameterized using a no-load current \( (I_0) \), motor speed constant \( (K_v) \) and motor internal resistance \( (R_m) \). The current draw from the fuel cell is \( I_m \) and the speed of the motor is proportional to the motor voltage \( V_m \). The motor controller is modeled as a linear resistance \( (R_{\text{cont}}) \). The range of motor constants within the design space is shown in Figure 1. The gear ratio between the motor and the propeller is allowed to vary within the range \( \{0.1:1 \text{ to } 10:1\} \), to avoid multi-stage geartrains. When multiple motors are simulated, the motors are assumed to be connected in electrical series, sharing current and splitting voltage. The effect of this assumption on the results of this study is negligible. To a 1st order approximation, the performance and weight of a fuel cell with two motors connected in parallel is equivalent to a fuel cell with double the cells and half the active area with two motors in series.

**Fuselage Geometry and Weight CA**

The weight and geometry of the fuel cell, balance of plant, hydrogen tank, electric motor and controller are the inputs to the fuselage analysis. The fuselage analysis wraps a minimum-volume cylindrical fuselage around these systems and uses a volume of revolution formula for streamlining the nose and tail sections of the fuselage as suggested by Roskam. The fuselage mass consists of a skin mass and a structural mass. The mass of the skin is scaled by the fuselage surface area. The mass of the internal fuselage structure is scaled by the sum of the mass of the fuel cell system hydrogen storage system. The spar structure weight is scaled based on the weight of the entire aircraft under a 3g loading. Both the skin mass and the structure mass are scaled based on coefficients determined from construction methods used for the demonstrator aircraft construction.
**Wing Weight CA**

The weight of the wings is calculated assuming foam core and composite construction and is inclusive of flap, aileron and servo weights. The scaling of the wing weights does not include the spar weight and is based on the constructed weight of the wings for the demonstrator aircraft. Spar weight is included in the fuselage geometry and weight CA.

**Performance Analysis CA**

The Performance Analysis CA calculates the flight performance of the aircraft using the drag polar, aircraft weight, and the calculated propulsive performance of that configuration. To calculate the cruise performance, the Performance Analysis CA throttles the propulsion system to match the available thrust to the required thrust at the minimum drag airspeed. To calculate climb rates, the CA estimates the maximum climb rate based on available power at the cruise airspeed.

**D. Design Structure Matrix**

In order to analyze a particular aircraft configuration, the CAs are connected into a Design Structure Matrix (DSM). The DSM passes variables between the CAs. The DSM can then be solved iteratively until all variables calculated and passed between the CAs are in agreement. The resulting outputs are then used to calculate aircraft response attributes. The DSM for this problem is presented in Figure 2 and 3. Each CA is shown as a box within the DSM and the CAs are connected by lines which represent information flowing between the CAs. The CAs are processed by the algorithm from top-left to bottom-right. The connections on the upper right side of the CAs represent forward information transferred from one CA to another within the same iteration. The connections on the lower left side of the CAs represent backward information transferred between separate iterations.

For each design configuration, the Propeller/Motor/Fuel Cell CA is run twice. The first run calculates the ultimate performance of the aircraft at 100% electric motor command. The second run calculates the performance of the aircraft under cruise conditions.

The structure of the DSM, and the order in which the CAs are processed is important because it determines the performance required of the DSM solution algorithm. When the DSM contains only feedforward information flow, the DSM can be solved directly without iteration. In general, the higher the number of backward-fed variables, the more computationally expensive the iterative solution of the DSM will be. Intelligent structuring of the DSM improves the numerical stability and efficiency of the solution algorithm. The DSM used for this design has 7 backward fed internal variables and 79 forward fed internal and design variables, and is solved by fixed-point iteration.

Each solution of the DSM represents a convergent solution for a particular aircraft configuration defined by the design variables. Using a Pentium D 3GHz PC each DSM solution requires 15 seconds of CPU time.

**Figure 2. Information flow within the optimization problem and DSM structure.** Details of the Propeller/Motor/Fuel Cell CAs is shown in Figure 3.
Figure 3. Detailed structure of Propeller/Motor/Fuel Cell CAs.

E. DSM Optimization Methods

Varying the design variables changes the performance of the aircraft as modeled using the DSM. To use the DSM to accomplish the design goals of the project, the design of long-range fuel cell aircraft, the DSM was incorporated into an optimization routine, as shown in Figure 2. The optimization routine varies the design variables so as to maximize an evaluation criterion, subject to absolute and side constraints.

Evaluation Criteria

In order to evaluate the importance of each converged solution of the DSM, a scalar Overall Evaluation Criterion (OEC) was defined as a function of the design variable array $X$:

$$OEC(X) = -\frac{R}{\kappa_R} + \frac{m_{to}}{\kappa_m},$$

In Eq. (5), $R$ is range measured in km, $m_{to}$ is the takeoff mass measured in kg and $\kappa_R$ and $\kappa_m$ are constants that weight the importance of the evaluation criteria components. The OEC effectively measures both the range and the takeoff mass of the aircraft. For this analysis, $\kappa_R = 62.137$ and $\kappa_m = 66.14$ were used which allowed range to roughly be an order of magnitude more important than the aircraft takeoff mass.

The design space for this problem is governed by two main constraints. These constraints are listed in Equations 6 and 7.

Climb Rate, $V_c \geq 75$ (m/min) (6)

Propeller Mach Number, $M_{tip} \leq 0.85$ (7)

In addition to these two constraints, there are also several side constraints that limit the values of the design variables. The side constraints are given as Equations 8-17.

$$r_{H2} > 0 \text{ (cm)}$$ (8)

$$1 \leq l_{H2} / d_{H2} \leq 4$$ (9)

$$0 \leq p_{H2} \leq 70 \text{ (MPa)}$$ (10)

$$n_{FC} > 0$$ (11)

$$A_{FC} > 0 \text{ (cm}^2\text{)}$$ (12)

$$773 \leq K_v \leq 2707 \text{ (rpm/V)}$$ (13)
\[ 0 < GR \leq 10 \quad (14) \]
\[ d_p > 0 \; (\text{cm}) \quad (15) \]
\[ p_p > 0 \; (\text{cm}) \quad (16) \]
\[ S_w > 0 \; (\text{dm}^2) \quad (17) \]

Of all the side constraints, only the H\textsubscript{2} tank length/diameter ratio, the H\textsubscript{2} tank pressure, the motor voltage constant, and the gear ratio have upper bounds. These upper bounds are based mostly on commercial availability except for the gear ratio which is based on a performance degradation limit. The remaining side constraints are used to guarantee that all variables are positive to avoid impossible aircraft designs.

Since a negative number for any of the side constraints is impossible, most of the CAs will produce an error if a design variable is negative. Unfortunately, many constrained optimization schemes cannot guarantee that side constraints will not be violated during the solution process. To avoid side constraint violations, a sequential unconstrained minimization technique (SUMT) was used. The SUMT required the objective function to be reformulated as:

\[ \Phi(X, \lambda) = OEC(X) + \lambda \zeta(X) \quad (18) \]

where \( \lambda \) is a scalar multiplier and \( \zeta(X) \) is an imposed penalty function. In order to force the optimization procedure to favor feasible designs and to avoid possible discontinuities caused by the introduction of the penalty function, the following definition for \( \zeta(X) \) was used\textsuperscript{15}.

\[ \zeta(X) = \sum_{j=1}^{n} \tilde{g}_j(X) \quad (19) \]

\[ \tilde{g}_j(X) = -\frac{1}{g_j(X)} \quad \text{if} \quad g_j(X) \leq \epsilon \quad (20) \]

\[ \tilde{g}_j(X) = -\frac{2\epsilon - g_j(X)}{\epsilon^2} \quad \text{if} \quad g_j(X) > \epsilon \quad (21) \]

\[ \epsilon = -C \cdot (\lambda)^a \quad (22) \]

The variable array \( g_j(X) \) represents the absolute and side constraints, where \( n \) is the total number of absolute and side constraints. The scalar values of \( C = 0.246 \) and \( a = 0.417 \) were used in all calculations based on preference weighting of the design criteria. For the first stage of the optimization, \( \lambda = 0.1 \) was used. The converged solution of the first stage optimization provides a better starting point for the next optimization stage. For the next stage \( \lambda \) is decreased by 30\% of its previous value and the optimization routine is repeated using the previous solution as a starting point. This is continued until an acceptable convergence criterion has been met. Graphically, the convergence process for an aircraft with 1 motor is shown in Figure 4. In Figure 4, the design variables and the range and climb rate metric drastically change from the initial guess to the first stage optimized solution. As \( \lambda \) is decreased, the solution tends to move from the feasible region closer to the active constraint boundaries. Note that for 1 motor, the solution slowly moves toward the climb rate, H\textsubscript{2} length to diameter, and gear ratio constraints. As a result, slight increases in range are evident.
**Optimization Algorithm Performance**

The optimization problem specified by minimizing \( \Phi(X, \lambda) \) subject to the constraints specified in Equations 6-17 is not an easy problem to solve. Due to the nature of the problem, continuity of the design space could not be guaranteed, especially if constraints were violated, therefore a non-gradient based simplex search method was used \(^{26}\). The change in the design variables from the beginning of the optimization to the end, as well as the aircraft mass and range are shown in Figure 5-7. Note that the initial and final values in these figures correspond to the initial guess and \( \lambda=0.1 \) curves in Figure 4.

Each of the design variables varies during the optimization scheme until finally reaching their optimum values. Note in Figure 5 that the hydrogen tank length to diameter ratio very quickly goes to its maximum constraint value and stays near the constraint for the duration of the optimization. Also note in Figure 5 that the climb rate immediately goes to its constraint value and this constraint remains active throughout the optimization. The range shown in Figure 5 initially increases, then only slowly increases until making another significant increase before settling at a maximum value near 4000 km. Although the range is only slowly increasing up to around function call 1300, most of the design variables continue to change with no clear indication of what combination will yield the best increase in range. Also note that the hydrogen pressure settles at around 40 MPa except for a small region after function call 1300 where a decrease in hydrogen pressure corresponds to an increase in hydrogen tank radius. Figure 6 shows the motor voltage constant decreasing and the gear ratio increasing as the optimization proceeds. In addition, the fuel cell active area generally increases as the optimization progressed. Decreasing the motor voltage constant and increasing the gear ratio typically is done to facilitate spinning a larger propeller to gain better propulsive efficiency. Note that Figure 7 shows that the propeller diameter does increase which directly correlates with the significant increase in range shown near function count 1300. The increase in range is also due to an increase in wing area (see Figure 7). This increase in wing area also results in an increase in takeoff mass. Finally, note that the propeller pitch over diameter ratio in Figure 7 directly follows the behavior of the cruise airspeed. Thus, maintaining the proper propeller advance ratio is critical for optimizing range.

---

**Figure 4. SUMT iterations for aircraft with 1 motor.**

Normalization factors for plotting are in parenthesis.

**Figure 5. Single motor aircraft hydrogen storage values for simplex optimization with \( \lambda=0.1 \).**
III. Results and Discussion

A. Design Space Exploration – Number of Motors/Propellers

For the purposes of optimization, all of the design variables are treated as continuous inputs except the number of motors and propellers. To compare the optimal performance of the aircraft with numbers of motors and propellers varying between 1 and 4, the optimization is repeated for each case. The characteristics of the optimal aircraft with varying motor and propeller numbers is shown in Figure 8.

An interesting observation for this data set is that the number of fuel cells and therefore the fuel cell operating voltage increases with increasing number of motors. This is expected since adding the motors in series decreases the
voltage applied to each motor. Increasing the number of fuel cells, along with fine tuning of the motor voltage constant and gear ratio, allow the algorithm to optimize cruise efficiency.

The fuel cell active area generally decreases with increasing number of motors. As the fuel cell voltage increases, the current decreases since the cruise power is relatively constant. The fuel cell active area is chosen to optimize the efficiency of the fuel cell system. For each configuration the fuel cell operates at the same optimal cruise condition of 0.785V/cell and 200mA/cm².

The hydrogen tank radius varies between the four configurations, but is lower for the configurations that include a center-mounted propeller and motor (1 motor and 3 motors). For the 1 motor case, the hydrogen tank diameter is 40% of the propeller diameter. Interference between the propeller and the fuselage causes the propeller to lose effectiveness as the fuselage diameter approaches the propeller diameter. Because the hydrogen tank length to diameter ratio is very close to the constraint at 4.0, the hydrogen pressure increases to increase tank capacity for the 1 and 3 motor cases. The hydrogen tank radius increases for the 2 motor and 4 motor cases.

B. Design Space Exploration – Powerplant Energy Density

Energy density is measured as the electrical output power of the powerplant at cruise divided by the sum of the weights of the powerplant, balance of plant, hydrogen storage and hydrogen. Many researchers have proposed that the design of long-endurance fuel cell aircraft may be simplified by designing the fuel cell system so as to maximize energy density without consideration of the aircraft propulsion sub-system performance or aircraft geometry 2,9,27. Other studies using fuel cell propulsion system models integrated with aircraft performance analyses have shown that there exists a tradeoff between power density and energy density for fuel cell-powered aircraft with endurances on the order of hours 28.

The results of this study show that maximizing energy density for fuel cell-powered aircraft with endurances on the order of days achieves neither maximum range nor endurance. For the two-motor configuration, a fuel cell powered aircraft optimized for energy density that meets the design constraints has an electrical energy density of 1045.5Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km. The two-motor configuration with the maximum range exhibits an energy density of 978Wh/kg, a range of 4994km and an endurance of 4621km.
highest endurance exhibits an energy density of 990Wh/kg, range of 4620km and an endurance of 77.3hrs. The high energy density configuration does not represent a performance optimum and is greater than 2.3 times the weight and 130% of the span of the range-optimal configuration. The higher energy density is achieved through increasing the hydrogen tank size which increases aircraft mass. The small increase in energy density achievable is mostly counteracted by the effect of the increased weight on the aircraft performance. Instead, optimization of the fuel cell powerplant within the aircraft system results in higher performing aircraft design, and opens up the design space to more attractive, smaller designs.

C. Optimal Configuration

Based on the results of the optimization study, the two-motor optimal configuration is chosen as the longest range aircraft within the design space. A two-view drawing of the simulated aircraft and its weight breakdown is shown in Figure 9 and the characteristics of the design are shown in Table 4.

The subsystem performance of the two-motor configuration is consistent with the characteristics of real-world aircraft of its class. The propeller tip mach numbers are 0.23. The lift to drag ratio of 27.33 is conservative when compared to gliders which can have a lift to drag ratio of >40 for a similar aspect ratio and planform geometry\textsuperscript{29}. With a hydrogen tank performance factor (PF) defined as:

\[ PF = \frac{P_{\text{burst}} \cdot V}{m_{\text{tank}}}, \]  

the performance factor for the designed tank is 28km. This is quite conservative compared to commercially available composite-overwrapped tanks such as the ATK-PSI P/N 80475-1 which has a performance factor of 33km\textsuperscript{24}.

![Figure 9. Two-view drawing of the two-motor optimal range configuration.](image)

It is also notable that there are a number of improvements that could be made to the aircraft in later stages of the design. For instance, implementation of a lower drag fuselage shape may improve aerodynamic performance. Integration of the hydrogen tank into the aircraft structure may decrease the aircraft structural and fuselage weight. The optimal configuration defined in this study is a conceptual optimum and informed refinement of the aircraft will further improve performance.
D. Sensitivity Analysis

A sensitivity analysis of the range-optimal configuration was performed by using a full factorial design of experiments (DOE) with only two levels for each design variable. The range of variation of the design variable was ±5% around the optimal design point for range shown in Table 4.

An analysis of variance (ANOVA) was performed on the results of the DOE to generate the Pareto plot in Figure 10. Figure 10 tabulates the percent contribution of each of the design variables with respect to the overall range variability calculated from the DOE. Figure 10 shows that the aircraft range of the optimal design is most sensitive to the design variables associated with the hydrogen tank. Propeller pitch, fuel cell active area and aircraft wing area are the least influential design variables. Exclusion of these three design variables results in a model that still explains 95% of the variation of aircraft range over the ±5% variation of all of the design variables.

Figure 10. Pareto plot of estimated design variable main effects on aircraft range.

To understand what the effect of variations in the design variables are near the optimal design point, a central composite DOE was used to generate data for regressing 2nd order response surface equations. The seven most influential design variables from the Pareto plot shown in Figure 10 were included in this analysis. The results are presented in the form of a multi-dimensional profile in Figure 11. The design variables listed along the bottom of the plot are inputs to the surrogate model and the system responses are along the vertical axis. The magnitude of the

Figure 11. Prediction profiler for central composite design
slope of the responses show the amount of influence that each of the design variables has on that response. Figure 11 shows that the optimal solution has settled against the rate of climb constraint. The gradient of the system rate of climb is negative for all of the design variables that could increase the range of the aircraft. This suggests that the numerical optimization technique used for long-range fuel cell aircraft design must be adapted to work in a highly-constrained design space. Also, the definition of the constraint boundaries must be carefully considered since they strongly influence the optimal solution.

In order to gauge the sensitivity of the optimal solution to small changes in intermediate performance variables that are typically difficult to accurately predict, a full factorial DOE was conducted around the optimal 2-motor configuration. The results, presented in Figure 12, show the sensitivity of the aircraft performance to improvements or degradation against key metrics. This plot shows the results of ±5% variation in the aircraft takeoff mass, aircraft drag, fuel cell polarization curve, and thrust. For example a 5% increase in mass over the predicted mass leads to a 258km (~5%) decrease in range. This is an expected result and is in agreement with the Breguet range equation. A 5% degradation in the fuel cell operating voltage leads to a 228km decrease in range and a 13.5% decrease in climb rate. This is a more drastic change and shows the high sensitivity of the aircraft performance to powerplant optimization.

\[\text{Figure 12. Prediction profiler for full factorial DOE around optimal configuration}\]

IV. Concluding Remarks

In this study, a model of a fuel cell-powered UAV has been constructed to examine tradeoffs among aircraft and powerplant design variables and to define optimal configurations. The design space is based on experience with a previously constructed prototype fuel cell UAV. The model incorporates sub-system-level contributing analyses of the aircraft structures, powerplant, hydrogen storage, propulsion system and aerodynamics. The contributing analyses are combined into a design structure matrix that can be used for design of experiments analysis and constrained optimization. Sequential unconstrained minimization was used to define optimal configurations with varying numbers of motor/propeller combinations. The optimization scheme was well suited to the problem because it does not require gradient calculations and does not significantly violate the constraints during the iteration process.

For this design space, the number of motors and their configuration was found to have a large effect on the performance of the aircraft. This effect is particular to fuel-cell aircraft because of the large frontal area required for hydrogen storage. Optimal configurations were found to use two wing-mounted motors so as to avoid propeller/fuselage interference. The optimal configurations were used to show that energy density optimization of the propulsion system alone is an inadequate technique for design of fuel cell aircraft in this class.

The sensitivity of the optimal design was analyzed and it showed that the interaction between the hydrogen storage system and the aircraft has the largest effect on the performance of the system. The optimal aircraft configuration was also shown to lie along the aircraft performance constraint boundaries. The sensitivity analysis shows that small percentage improvements in aircraft weight or fuel cell power can result in an equivalent percentage increase in range.

A verified optimal configuration was defined that achieves nearly 5000km range in a medium-scale PEM fuel cell-powered aircraft. This performance is far beyond the 2900km range required by the problem statement. The performance of the aircraft subsystems are within range of what is commercially available. Full development of this conceptual aircraft requires no technological advances and is feasible within a short developmental time frame.

Acknowledgments

This research was funded in part by the NASA University Research Engineering Technology Institute (URETI) grant to the Georgia Institute of Technology. The authors would also like to thank the many research engineers at the Georgia Tech Research Institute and the Aerospace Systems Design Laboratory that provided valuable expertise and guidance over the course of the project.
References

17 Lark, R. F. “Recent advances in lightweight, filament-wound composite pressure vessel technology,” in Energy Technology Conference, Houston, TX, September 18-23, 1977.