3-11-2011

Sizing Analysis for Aircraft Utilizing Hybrid-Electronic Propulsion Systems

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SIZING ANALYSIS FOR AIRCRAFT UTILIZING HYBRID-ELECTRIC PROPULSION SYSTEMS

THESIS

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THESIS

Presented to the Faculty

Department of Aeronautics and Astronautics

Graduate School of Engineering and Management

Air Force Institute of Technology

Air University

Air Education and Training Command

In Partial Fulfillment of the Requirements for the

Degree of Master of Science in Aeronautical Engineering

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March 2011

AFIT/GAE/ENY/11-M26

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AFIT/GAE/ENY/11-M26

Abstract

Current conceptual aircraft design methods use historical data to predict and evaluate the size and weight of new aircraft. These traditional design methods have been ineffective to accurately predict the weight or physical dimensions of aircraft utilizing unique propulsion systems. The mild hybrid-electric propulsion system represents a unique design that has potential to improve fuel efficiency and reduce harmful emissions. Hybrid-electric systems take advantage of both reliable electric power and the long range/endurance capabilities of internal combustion engines. Desirable applications include general aviation single-engine aircraft and remotely-piloted aircraft. To demonstrate the advantages of mild hybrid-electric propulsion, a conceptual design code was created that modified conventional methods. Using several case studies, the mild hybrid conceptual design tool was verified. The results demonstrated potential fuel savings for general aviation aircraft and expanded mission capability for remotely-piloted aircraft.

Acknowledgments

What a piece of work is a man,
how Noble in Reason, how infinite in faculty,
in form and moving, how express and admirable
in Action, how like an Angel,
in apprehension, how like a God?
the beauty of the world, the paragon of animals

-William Shakespeare

I would not have made it this far without the love and support of my parents. Their work-ethic inspired me to be the person I am today. Thanks Mom and Dad!

To my sister, I would like to thank her for always going the extra mile and setting the bar so high. As a role model she unknowingly helped guide me to all of my success.

To my fiancé, the never ending support and sacrifices made by her were much appreciated and will not soon be forgotten. Her encouragement kept me focused on the task at hand and the result was an outstanding academic experience.

To the hybrid design team and machine shop guys, it was a lot of fun building and testing such a revolutionary concept. I learned a lot from every single person and hope to see everyone in the near future.

To the entire faculty and staff at AFIT, no words can describe how much I appreciate what they did every day. Their helpful attitudes and desire to share knowledge was phenomenal.

Finally, I would like to thank my thesis advisor. Without his assistance I never would have made it to AFIT and had such a positive experience.

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List of Abbreviations

BDC	Bottom Dead Center
CI	Compression Ignition
COTS	Commercial Off-the-Shelf
DDD	Dull Dirty Dangerous
EM	Electric Motor
GA	General Aviation
GTOW	Gross Takeoff Weight
ICE	Internal Combustion Engine
IED	Improvised Explosive Device
ISR	Intelligence Surveillance Reconnaissance
FBW	Fly By Wire
MEA	More Electric Aircraft
MEP	Mean Effective Pressure
ODIN	Observe Detect Identify Neutralize
RPA	Remotely-Piloted Aircraft
RPM	Revolutions Per Minute
RPV	Remotely Piloted Vehicle
SI	Spark Ignition
SUAS	Small Unmanned Aircraft System
TDC	Top Dead Center
UAS	Unmanned Aircraft System
UAV	Unmanned Aerial Vehicle

Nomenclature

Symbol	Description (Units)
AR	Aspect Ratio
$Batt_{SpecificEnergy}$	Specific Energy of the Battery
m_{batt}	Battery Mass
C_D	Drag Coefficient
C_L	Lift Coefficient
d	Engine Stroke Displacement
D	Drag
$D_{D,0}$	Drag Polar
e	Oswald Efficiency Factor
g	gravity
HF	Hybrid Factor
hp	Horsepower
L	Lift
ρ	Density
η	Efficiency of a Component
n	Load Factor
nm	Nautical Miles
P	Power
ROC	Rate of Climb
S	Wing Cross Sectional Area
s_{LO}	Takeoff Distance
T	Thrust
v	Specific Power
V	Velocity
W	Weight

Subscript	Description
∞	Free Stream Condition
0	Initial
a	Maneuver Speed
A	Quantity at Altitude Condition
alt	Altitude
Cruise	Quantity at Cruise Condition
EM	Electric Motor
max	Maximum Quantity
PD	Power Device
propulsive	Power Dedicated to Propulsion
R	Required Quantity
ICE	Internal Combustion Engine
Stall	Quantity at Stall Condition
Takeoff	Quantity at Takeoff Condition

SIZING ANALYSIS FOR AIRCRAFT UTILIZING HYBRID-ELECTRIC PROPULSION SYSTEMS

I. Introduction

1. Background

Physically, humans were never meant to fly among the clouds, but because of the imagination of a few, mankind has been able to cheat nature. Man's first recorded attempt to achieve the freedom of flight was the Grecian myth of Daedalus and his son Icarus. Since that time the theory of powered flight eluded the most creative minds. It wasn't until the early 20th century that two men, who built and maintained bicycles, took it upon themselves to overcome the limitations of mundane earth treading. On December 17, 1903, Orville and Wilbur Wright gave mankind flight.

To put the Wrights' engineering feat in perspective, sliced bread was still a mere 25 years away in 1928 (which was first introduced in Chillicothe, Ohio, the same home state as the Wright brothers). Since that faithful day, man has desired higher, faster, and more efficient aircraft. The military applications of such a tool were staggering. One such application removed the pilot, so that dangerous missions could be executed without putting a pilot's life at risk. The origin of Remotely-Piloted Aircraft (RPA) stems from the pioneering work of Elmer Sperry, Charles Kettering, and even Orville Wright himself. Charles Kettering immediately realized the potential use of unmanned aircraft. Once Sperry was able to demonstrate gyro stabilization allowing semi-autonomous flight, Sperry and Kettering worked with Orville Wright to create the first military RPA 'the Bug'[1]. Transitioning to the year 2010 the RPA presents a widely expanded design space, compared to conventional manned aircraft, and offers a moldable framework in terms of new and unique applications [2].

Since 9/11 the conflicts in Afghanistan and Iraq have created the most widespread use of unmanned vehicles. Though Charles Kettering and the US military were the first to develop a RPA for military use, Israel was the first country to use an RPA in combat [1]. The Israeli military reasoned "that for reconnaissance missions a loss of a relatively inexpensive remotely-piloted vehicle (RPV) was better than the loss of a pilot and multi-million dollar plane" [1]. Israel remains one of the leading allied countries for the development of unmanned aircraft systems (UAS) for military application. The Ryan Firebee was the first RPA technology to be used by the US during the Vietnam War. The Firebee was used extensively to perform imagery reconnaissance, communication intelligence, and leaflet dispensing missions [1]. Since the end of the Cold War RPA development was part of a growing trend. Not until the first decade of the 21st century, because of the recent US conflicts, has RPA development seen such acceleration [1].

There has been a continuous Warfighter need for sustained presence and multiple payload capabilities of RPAs performing the 'Dull, Dirty, Dangerous' (DDD) missions to help end the conflict [3]. Some of these missions require innovative platforms that can satisfy multiple mission capabilities. The aircraft that were most commonly recognized by civilians are the Predator and the Global Hawk. By looking at the RPA Worldwide Roundup published by AIAA in 2009, readers can see the hundreds of RPA designs featured from many countries ranging from just a few pounds to the scale of Global Hawk (around 30,000lbs) [4]. The wide ranging capabilities of these systems were still not enough to satisfy the needs of the Warfighter. As with any new development, RPAs were limited by state of the art technologies. The most dominant technology that has limited the development of new RPAs has been the area of propulsion. Electric aircraft lack the endurance and range of combustion driven RPAs. Combustion engines were limited as well because they offer little stealth to the soldier performing surveillance at low altitudes. A propulsion system that utilized both was highly desirable.

To meet the demands of both stealthy operation and long duration Lt. Col. Frederick G. Harmon proposed a novel propulsion system using hybrid-electric technology [5]. The analysis of aerodynamic forces applied to cars has had dramatic effects on the fuel economy of some cars, “thus the parallel development of the airplane and the automobile over the past few years has been mutually beneficial” [1]. By using the proven hybrid technology that has been applied to cars, Harmon developed a conceptual design for a hybrid-electric RPA [5]. This RPA has the potential to bridge the gap between the capabilities of electric and combustion driven RPAs and could have immediate effect in the theatre.

From a general aviation (GA) perspective hybrid-electric technology would help overcome the transition from internal combustion engines to full electric propulsion systems. In 2010, the automobile market released several plug-in hybrid and full electric vehicles. The traditionally modeled cars boast electric power plants that can provide 100 miles/charge for the Nissan Leaf [6], and 35 miles/charge with an additional 340 using an internal combustion generator for the Chevy Volt [7]. The range and performance for hybrid vehicles has finally reached a point to make them practical for everyday driving. Making the same comparison for electrically powered GA aircraft the practical solution has yet to be discovered. Yuneec Aviation’s e 430 aircraft has an estimated flight time of

2 hours with a 83 kg battery [8], similarly the Cessna 172 modification proposed by Cessna and Bye Energy uses a 295 kg battery to achieve the same flight time [9]. Without knowing the specific requirement specifications for payload, flight speed, or maneuvering capability these aircraft can only be considered technology demonstrators.

To make electrification more practical for the general aviation community, hybrid-electric technology may bridge the gap. While battery technology continues to mature internal combustion engines using hydrocarbon fuels offer the most weight conscious propulsive solution. Hybrid-electric technology that can augment the power needs for specific aircraft applications presents a practical electric energy solution that avoids sacrificing excessive aircraft performance. The hybrid-electric propulsion system would be able to transition along with battery technology slowly phasing out the energy needs from the fuel, while increasing electrical energy storage.

2. Motivation

2.1. Remotely-Piloted Aircraft

Urban conditions and ominous highways in Iraq and Afghanistan make it challenging for the Joint Force to safely maneuver through war zones. "The troops rely on timely information from intelligence, surveillance, and reconnaissance (ISR) aircraft and other sources to detect insurgents in the act of emplacing improvised explosive devices (IEDs), The IED threat is expected to worsen" [10]. Accordingly, the Obama administration, prior to sending 30,000 additional troops to Afghanistan, made it a priority to increase the use of remotely-operated aircraft to protect Joint Force soldiers. As a result, the forward commands in both Iraq and Afghanistan have pushed many battle ready RPAs into action. The secretary of the Air Force, Robert Gates, reports that since the beginning of the war “the Air Force has significantly expanded its ISR capability” and adds, “we intend to keep expanding it” [10]. There was a high priority for reliable information to provide situational awareness enabling decision makers to minimize troop losses as well as identify high value targets through ISR [11]. Task Force ODIN (Observe Detect Identify Neutralize) was the product military unit that used the increased ISR capabilities to counter the rising toll of IEDs being used as roadside bombs [10].

Unfortunately, currently fielded UAS only satisfy discrete points for the broad spectrum of mission needs. As outlined in the 2009-2047 UAS Roadmap, the Department of Defense (DoD) requested the design of a platform that will satisfy the gap between the medium and high altitude mission segments [12]. The “Unmanned Aircraft Systems Flight Plan: 2009-2047” also calls for the multi-mission SUAS that can bridge the gap between man-portable RPA and predator aircraft [13]. Companies are investing more time and money into this problem than ever before [14]. Since current propulsive technologies were inadequate to satisfy the entire flight regime, research needed to be taken beyond the traditional methods of aircraft design.

Research conducted at University of California at Davis and AFIT suggest that a full hybrid-electric power plant could be employed on a small RPA. Most recently, a variety of configurations and discharge strategies were optimized using a MATLAB code developed at AFIT that yielded results that encourage continued research [15]. The driving force behind the hybrid design was to increase the endurance of an electric driven RPA and decrease the mission compromising signature created by internal combustion engines (ICE). “The key performance requirements for future UAS, depending on mission requirements, will be, speed, maneuverability, stealth, increased range, payload,

[and] endurance” [12]. The most effective platform has to be one that can satisfy a majority of these requirements. An RPA can have stealth while operating with an electric motor (EM), speed using an ICE, increased range, and multiple payload capabilities if a hybrid design can be exploited. The benefits of hybrid cars have been realized for years. The recent study of hybrid-electric aircraft suggests the same benefits could be realized for aircraft. The study of the limiting design factors must be explored to realize the potential of hybrid RPAs and GA aircraft.

2.2. General Aviation

With the energy crisis still looming, hybrid-electric power helps reduce fuel usage and promotes the More Electric Aircraft (MEA) initiative. The desire for MEA was not a new concept, but has struggled to remain in the forefront of aircraft development. The most notable hurdle was the fly by wire (FBW) system used on some aircraft replacing the hydraulic pneumatic systems on larger aircraft [16]. The installation of an all electric propulsion system posed a seemingly impossible barrier. Today several all electric aircraft exist proving that electric propulsion was possible. However, each one of these aircraft was limited by the energy storage capability. The potential advantages when using all electric propulsion would be; no emissions, increased performance (especially at altitude since air density does not affect motor performance), and lower operating costs. To get to that point more research was needed in the area of energy storage and conceptual aircraft design to continue the push toward all-electric aircraft.

3. Problem Statement

Gaps between the high altitude Global Hawk, medium-altitude predator and the variety of man-portable RPAs need to be analyzed to meet the growing demand of commanders. Mild hybrid-electric designs could be the answer to meeting the multiple mission needs of commanders. USAF requires that the design of "future UAS should be multi-mission, [and] should also be able to carry any standard payload within its performance envelope" [13]. It is unlikely that current propulsive technologies alone will fulfill the unique mission capabilities proposed by the multi-role RPA outlined in the 2009 UAS Roadmap. Man portable systems were limited by low battery power densities that reduce endurance time. Unmanned aircraft that utilize large or small internal combustion engines/turbines were hindered by low efficiency, and large heat and acoustic signatures made them vulnerable to detection. Currently, this has resulted in the Warfighter needing two systems to satisfy two mission requirements. RPAs utilizing mild hybrid propulsion could provide one platform for multiple missions.

The need of the Warfighter has prompted the rapid deployment of RPAs that have trouble meeting transforming needs. This can be attributed to the lack of AF doctrine defining mission requirements that continue to grow more complex [12]. From an operational standpoint the use of multiple platforms makes it difficult to effectively perform missions. Often in a military division multiple capabilities were needed, and require the logistical hardship of carrying two RPA ground control stations. Versatile unmanned systems would greatly reduce the forward footprint of ISR equipment [14]. Another important requirement of the DoD was that new RPAs have the ability to interchange payloads for specific missions. Operational modularity can allow platforms to evolve with improving technologies. The most desired technology would be a high power and high energy density source which would offer long endurance and high speed capability. Until this energy source can be discovered a more reliable intermediate step must be explored.

New design methods need to be explored for unique propulsion systems. Hybrid-electric technology has been one of many alternative power plants considered to replace internal combustion engines. Some other examples include fuel cells, all electric, solar powered, and multiple combinations of each. The traditional aircraft design methods have been useless to accurately predict weight or the physical dimensions of aircraft with unique propulsion systems. The most cost effective method has been to retrofit existing airframes with a new propulsion system and hope to equal the performance. To take advantage of new propulsion systems beyond retrofitting, useful conceptual design methodologies must be created. The product would be a method that optimized aircraft designed around unique energy and power sources. For hybrid-electric propulsion this strategy must account for battery weight, energy usage from battery and fuel, and the effective delivery of power using the engine and motor.

4. Research Objective

This research was another milestone in the development of a novel propulsion system. The hybrid-electric system takes advantage of both reliable electric power, and the long range/endurance capabilities of ICEs to satisfy the growing mission needs of the Warfighter. To evaluate the usefulness of such a propulsion system, limiting factors must be addressed to gauge aircraft performance. The most important constraining factor was the physical size of the mild hybrid-electric components. The goal was to examine state of the art technologies and establish an optimized conceptual design code using simplified aircraft design methods. Constraints were added to the design code to account for structural integrity, specific mission requirements, and power plant optimization. The author predicted that the size limit would depend greatly on the payload requirements and battery weight since the motor's energy supply would be heavy compared to the ICE's fuel. The increased propulsive efficiency, and reliability of using a synthesis between the two, leads the author to believe that the pure hybrid design proposed by Harmon could be taken beyond small class RPAs where mild hybrid designs may be applicable.

The hybrid-electric propulsion system could have similar benefits in the commercial general aviation industry satisfying the More Electric Aircraft initiative. Analysis was conducted to demonstrate the feasibility of a conceptual mild hybrid design that assisted the aircraft's takeoff and climb. The benefit of this research was that fuel consumption of general aviation aircraft could potentially be reduced using the smaller engine associated with the mild hybrid design. As well as providing an effective propulsive redundancy by using the motor to effectively extend glide slopes to find a safe landing location. The mild hybrid system would replace engines that were oversized for the cruise condition with a smaller engine and electric motor running in parallel. The engine would be optimized for a cruise condition suitable for the airplane and the motor would provide the additional power needed for transient conditions such as takeoff and climb.

The research objectives were to:

- Scale mild hybrid-electric systems to various sizes of GA aircraft and RPAs
- Develop conceptual design code to analyze mild hybrid-electric systems for GA aircraft and RPAs.

- Use case studies to validate conceptual design code by retrofitting existing platforms.
- Determine hybrid capabilities for multi-mission RPAs

5. Research Scope

This research was meant to help researchers understand the scaling possibilities for hybrid-electric aircraft to meet different mission needs. Mission capabilities were defined as takeoff distance, range, climb rate, max altitude, and payload. The design code includes traditional sizing methods for a conventional aircraft and any changes that the author deems necessary to account for the unique propulsion system. The propulsion system uses a heavy fuel ICE and electric motor in parallel configuration. The design parameters will be for a simple aircraft consisting of rectangular wings, constant airfoil shape, fuselage, and tail. The product will be an optimized airframe coupled with optimal propulsion component weights and power.

Research included investigating an applicable conceptual design tool for a mild-hybrid configuration. The mild-hybrid could provide power-assisted takeoff and, currently unavailable, redundancy. The study for these aircraft considers a limited structural element, neglects stability and control analysis, and limits the aerodynamic analysis. Code validation will be the successful demonstration of several case studies of GA and RPA aircraft with regards to aircraft performance.

6. Methodology

To demonstrate the advantages of mild hybrid-electric propulsion, a conceptual design code was created based on traditional methods. The methods used to determine the gross takeoff weight and aircraft performance were from Raymer's *Aircraft Design: A Conceptual Approach* and Anderson's *Aircraft Performance and Design* [17] [18]. Cruise power was optimized for the cruise condition to avoid oversized engines. Additional power was supplied for the motor for any transient power needed. Evaluation of the resulting propulsion system and aircraft component weight fractions was similar to the methodology used by Harmon and Hiserote [5] [15]. Comparing the weight fractions and performance of several traditional aircraft to the mild-hybrid design would provide the necessary validation for the optimized results.

7. Overview of Thesis

The following chapters were organized in such a way so that readers can understand the development of the hybrid aircraft conceptual design process. Chapter II provides the literature background needed by the author to develop the knowledge base necessary to continue hybrid-electric propulsion research. Chapter III describes the methodology used by the author to establish the conceptual design code for feasible hybrid concepts. Finally, Chapters IV and V provide results and recommendations for the hybrid conceptual designs, respectively.

II. Literature Review

1. Overview

As of 2010, hybrid-electric propulsion has been applied to every ground based mode of transportation: trains, buses, cars, etc. With high gas prices, efficient hybrids become a highly desirable alternative. The automotive industry has helped alleviate the shortcomings of high fuel consumption by means of hybrid technology. The fuel efficiency of automobiles and other internal combustion applications were comparable to propeller driven aircraft. If hybrid technology can be applied to aircraft, the same fuel saving benefits for automobiles may be achieved and possibly increase the capabilities of today's RPAs and GA aircraft. Considering the state of the art of RPAs, electric and combustion propulsion were used separately for discrete mission capabilities, thus a compromise was made between the advantages of engines and motors. Correspondingly, a push for all-electric GA aircraft has caused a need for improved fuel consumption and reduced fuel emissions. For GA aircraft hybrid propulsion can be a stepping stone to the eventual electrification of larger aircraft.

Motors and batteries provide an efficient electric alternative to the ICE, but the specific energy of hydrocarbon fuels was still far superior to batteries. For unmanned aircraft this translates to a decision between efficiency and endurance. To maintain high efficiency of the on-board energy, all-electric systems must be used. The endurance of all-electric RPAs has been restricted by the weight penalty current batteries possess. For long range missions ICEs were more effective but have been compromised by thermal and acoustic signatures. Each system offers a desired advantage to unmanned aviation however improvements to these systems individually would not provide the immediate solution needed by today's Warfighter. A mesh between these components realized in a hybrid-electric design would be the best propulsive solution to meet the RPA market demands in Iraq and Afghanistan. This chapter was meant to give the reader a general knowledge base for hybrid-electric technology and how it can be used to design ground-breaking aircraft, with an emphasis on unmanned applications and single engine GA aircraft.

2. Hybrid-Electric Technology

Hybrid power systems would effectively transition from internal combustion engines to all electric applications. The automotive industry has worked hard to make hybrid driven cars available to the public. Improvements in battery technology, control algorithms, and traction motors have revolutionized the application of efficient electric power to satisfy transportation needs. Designing these systems has been difficult because of the balancing act needed between energy storage and high power output. Unique combinations of motors and batteries have been used to investigate the strengths and weaknesses of each component. The end goal was the complete electrification of cars for every day travel needs. To make this a reality research has been done to optimize a variety of drive train configurations.

The rationale behind using hybrid technology has been to take advantage of improved energy efficiency and reduce fossil fuel use that impacts the environment. There were multiple configurations available for hybrid vehicles. A series hybrid uses an engine running at an optimal operating condition powering a generator that converts the fuel's energy into electrical energy that powered an electric motor. This configuration was commonly used on trains and larger applications, but contains energy losses due to conversion inefficiencies and has a single energy path [19]. The hybrid configuration used in cars was most often a parallel hybrid. A parallel hybrid system uses the motor and engine concurrently, and has the ability to use the engine as a generator in addition to driving the vehicle. In parallel the IC engine was designed to operate at its most efficient condition for highway driving, while the electric motor was used for transient accelerations at slower speeds [19]. The parallel system also allows two independent energy paths. For an RPA application, a parallel hybrid offers significant advantages such as stealthy operation and power redundancy. For this reason a parallel configuration was the best option for a small hybrid-electric RPA (and was explored by Frederick G. Harmon at the University of California-Davis [5].

Hybrid technology has the potential to encompass both small RPAs and single engine GA platforms. The benefits would be similar but the hybrid application would be much different. For an RPA, the goal would be to use a pure parallel hybrid configuration. A pure hybrid means that the aircraft could be powered solely by the motor or engine at any given time. For general aviation aircraft, passengers provide an added weight penalty and would need an excessive amount of motor power and energy storage to be able to fly on electric power alone. The sensible alternative would be to augment the power of the engine with additional electric power during certain phases of flight. The boost power provided by the motor has the potential to improve takeoff and climb performance. Also, the augmented power has an inherent safety feature and could be designed to provide enough backup power to extend a glide in the case of an engine failure mid flight. In both cases the energy storage and power delivery must be carefully sized to meet the mission demands.

The automotive industry has greatly benefited from the use of hybrid technology and has perfected the balance between efficiency and control. By glancing at the 2011 car sales lots there were more available hybrids than ever before. All the major motor companies have invested in the technology, and continue pushing toward all electric vehicles that can be charged using a home wall socket. Unfortunately, all-electric vehicles have been restricted by their battery energy storage capacity, much like the RPAs using electric propulsion. One design advantage that hybrid cars had over aircraft designs was the overall weight was not as critical. The weight penalty of batteries in aircraft has a greater impact and was a critical design consideration that cannot be ignored. This and other important characteristics of hybrid vehicles must be evaluated before aircraft can incorporate the technology.

3. State of the Art: Hybrid-Electric Aircraft

The direct application of automotive hybrid designs to general aviation aircraft would be difficult because of the weight penalty associated with battery packs. Much like the jet engine, "electric propulsion has the potential to be the next significant leap in aircraft propulsion technology" [20]. The benefits of electric propulsion in terms of efficiency, noise reduction, and capabilities would be endless. Proper steps must be taken to transition from hydrocarbon fuels to all-electric aircraft.

Weight has always been a critical design consideration for aircraft. Simply replacing the engine with a motor and battery combination, modern aircraft would suffer in performance. The reason that aircraft suffer in performance for these retrofits was attributed to the lack of energy storage available in current batteries. Yuneec aviation has designed a light sport aircraft using this method and the aircraft can only maintain flight safely for 1.3 hours demonstrating the inadequacy of the energy storage [8]. The Yuneec E 430 electric aircraft boasted an aspect ratio close to 20 and during flight the propeller was tucked away in the fuselage for gliding flight. Thus the 1.3 hour flight may not be powered for the whole duration [8]. The shortfall for these types of aircraft has been replacing the enormous amount of energy hydrocarbon fuels provide compared to batteries. The limited endurance of aircraft like Yuneec Aviation's E 430 and small RPA's such as Aerovironment's Raven (endurance of 60-90 minutes [21]), indicates that until battery technology improves, practical aircraft and potential multi mission RPAs must take advantage of hybrid drive systems. The safest and most efficient starting point for GA aircraft propulsion would be a mild-hybrid that used a motor to provide a power assist during takeoff, climb, and dynamic performance. This model would follow similar precedent established by the automotive industry and would progress along with available technology. For the RPA design, since no passenger payload was required, a full hybrid drive system may be more beneficial.

3.1. Remotely-Piloted Aircraft

With regards to the small RPA design, Harmon et al provide evidence, through simulation, that a parallel hybrid-electric RPA improved mission capabilities [5]. Using the logic that a mission can be broken into segments that have a variety of power needs, Harmon conceptualized a two-point hybrid design. The first design point, similar to the parallel car model, was an ICE designed to satisfy the cruise condition of the aircraft. Then, taking advantage of quiet operation and high energy efficiency, the EM was then sized for the loitering mission segment [5]. The output of the engine and the motor was derived from the power required curve from an optimized airframe. Using aircraft performance equations Harmon developed a MATLAB code that optimized an aircraft that weighed 13.9 kilograms [5]. The result was a reasonable aircraft design that could be physically constructed and tested. The difficulty was deciding how the components would be integrated.

The initial design was to use an electro-magnetic clutch between the engine and motor that could be disengaged during loiter operation [5][22]. Using a clutch Harmon anticipated that when the engine was shut off during loiter operation it could be restarted by powering the motor with the clutch engaged. The torque necessary to do this was substantial and needed evaluation. Figure 1 depicts a clutch configuration model constructed at Wright State University by this author and a senior design team.

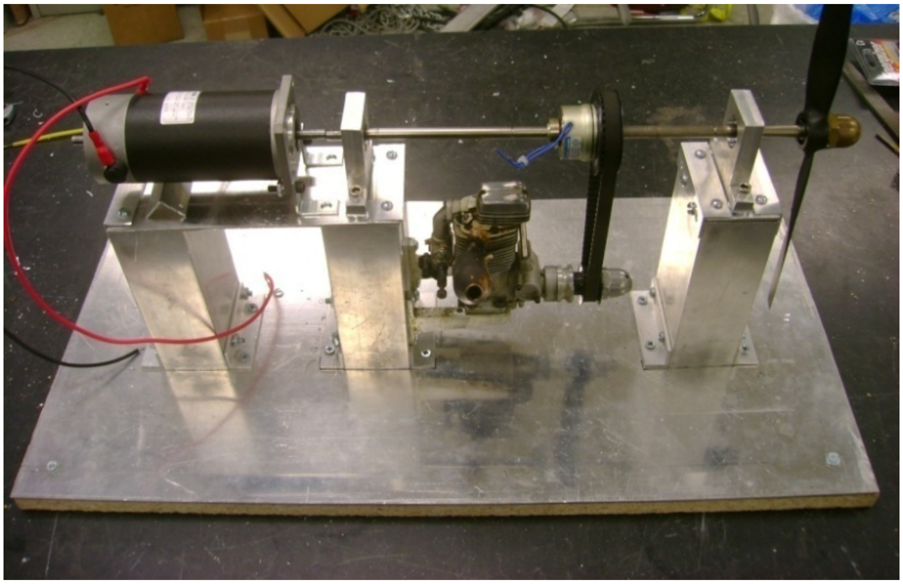

Figure 1: Clutch Configuration Experimental Model [23]

Using a hobby glow engine, electric motor, and clutch consistent with Harmon's two-point design the team tested the clutch configuration. It was discovered for this experiment that the clutch could withstand the torque, but the motor was unable to handle the load to restart the engine [23]. The team concluded that the clutch configuration could work if the motor could be designed for the hybrid design point and satisfied the torque required for starting the engine at low speeds. With the knowledge gained by this experiment the author wanted to investigate alternatives to the clutch design.

The mechanical complications caused by the clutch parallel configuration warranted the need for different methods to integrate the hybrid-electric RPA components. The first alternative used a geared secondary motor to start the engine independently of the primary motor. The engine could then be connected to the primary motor using a one-way bearing so that when operating the engine, the system could use the motor as a generator [22]. This design could only replace the clutch design if the geared motor attached to the engine had comparable weight to the clutch. The second alternative considered was to separate the engine and motor completely and use a centerline thrust configuration with two propellers [22]. This configuration offered the least mechanical complications, but when the engine was decoupled from the motor it can no longer use the motor as a generator to charge the batteries. Using these three configurations, Hiserote evaluated the advantages of each using a similar code that was used by Harmon for the clutch configuration.

By considering different charging strategies, Hiserote was able to characterize the three configurations. Hiserote determined that a charge sustaining, charge depleting, or a segmented charging strategy would have unique benefits for each configuration [15]. Using three configurations and three charging strategies, 9 conceptual designs were evaluated for the parallel hybrid-electric RPA. Hiserote found that for a mission that required charge sustainment that used rechargeable batteries, the clutch start would be the unanimous choice [15]. If the clutch was found to be unreliable the electric start represented the next most viable option. Finally, the center line thrust using two propellers offered the best charge depletion capability and because of the redundancy was the most survivable [15].

After deciding that a clutch start configuration would be the best solution a team of graduate students at the Air Force Institute of Technology decided to build a working prototype. The prototype was meant to verify the RPA full hybrid design, but the determination of whether or not additive torque could be achieved would also verify the mild-hybrid aircraft model. A group of five students have characterized a prototype model of a parallel hybrid propulsion system using a dynamometer. In order to implement the propulsion system into a flying aircraft several questions had to be answered. First, accurate engine maps have been developed to allow an on board controller to optimize operation [24]. Next, a reliable method for matching the integrated hybrid components was created to minimize energy losses in the drive train [25]. Then a controller was developed to use fuel and battery energy in the most efficient manner [26]. Lastly in order to determine the useful application of mild hybrid-electric GA and RPA aircraft several case studies were performed to verify a conceptual design code.

In Australia Richard Glassock has designed and tested a similar parallel hybrid and confirmed that a RPA would benefit from the hybrid configuration. The experimental setup used by Glassock et al can be seen in Figure 2. The motor is mounted underneath the ICE output shaft and can provide additional torque or serve as a generator for the avionics. The experimental results illustrated in Figure 3 demonstrate the ability EM and ICE to provide additive torque to the propeller giving improved performance. The thrust can be shown to increase only if an oversized propeller was used to account for the motor speed and gear ratio matching between the engine and motor [27]. The challenge became matching the engine, motor, and propeller.

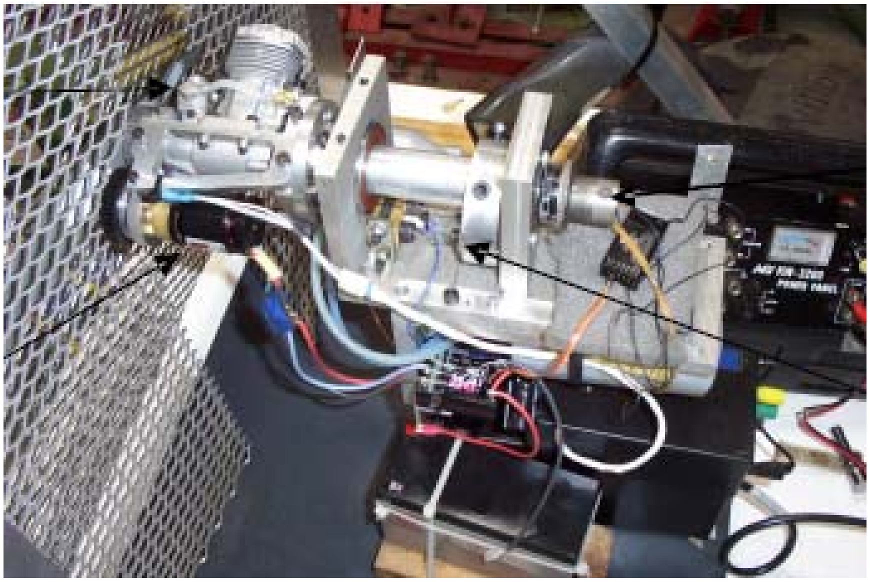

Figure 2: Australian Hybrid Design

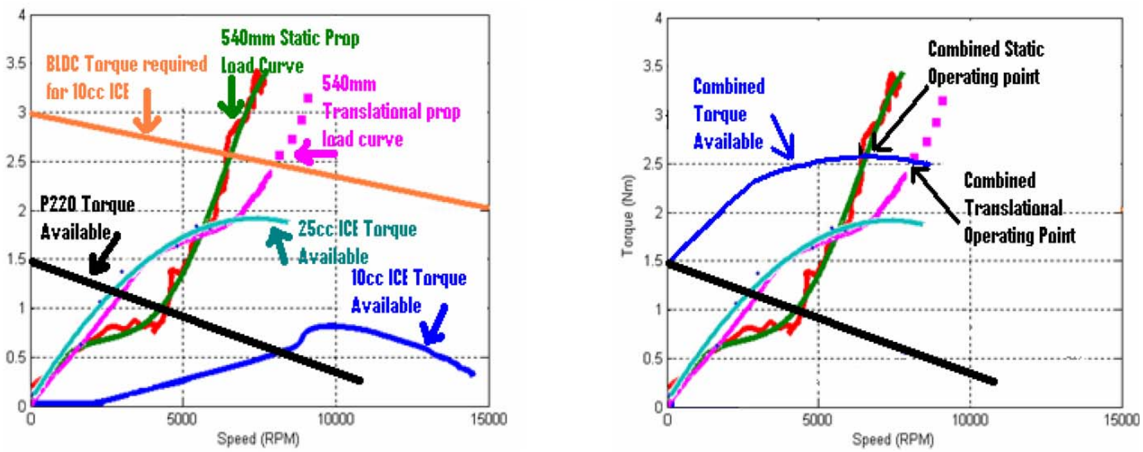

Figure 3: Experimental Torque Measurements [28]

3.2. General Aviation Aircraft

To improve takeoff and climb performance for general aviation aircraft a mild parallel hybrid design was needed. Since battery technology has struggled to keep up with the energy delivery available in carbon based fuels, a full hybrid would not be reasonable for larger aircraft. The most concerning design constraint for the GA platform was the gross takeoff weight (GTOW), takeoff distance, and climb performance. The engines in most single-engine aircraft were oversized in order to takeoff and climb. At the engine cruise condition only 55% of the power was needed [29]. A parallel design can improve the takeoff and climb performance, and provide added redundancy in the case of engine failure. A mild hybrid-electric aircraft could use the motor to supplement the engine power when needed and be used independently if an engine failure occurred.

The physical configuration of the mild parallel hybrid design could use either a side by side belt drive or a single shaft direct drive. A German aviation company, Flight Design,used a side by side belt drive on the 116.25 kW prototype using a 30 kW motor and Rotax 914 (86.25 kW) engine [30]. Real experimental data for the design has yet to be released but the parallel power-plant showcased at the Oshkosh Air Show in 2009 and 2010 can be seen in Figure 4. An alternative design would be to mount the motor on the engine shaft and use a clutch to disengage the motor from the engine in the case of ICE failure. Regardless, the main concern and design constraint will ultimately be the battery weight to provide energy to the motor. Using the direct drive configuration for the RPA and GA aircraft a design process was developed to take advantage of the energy usage of the mild parallel hybrid design.

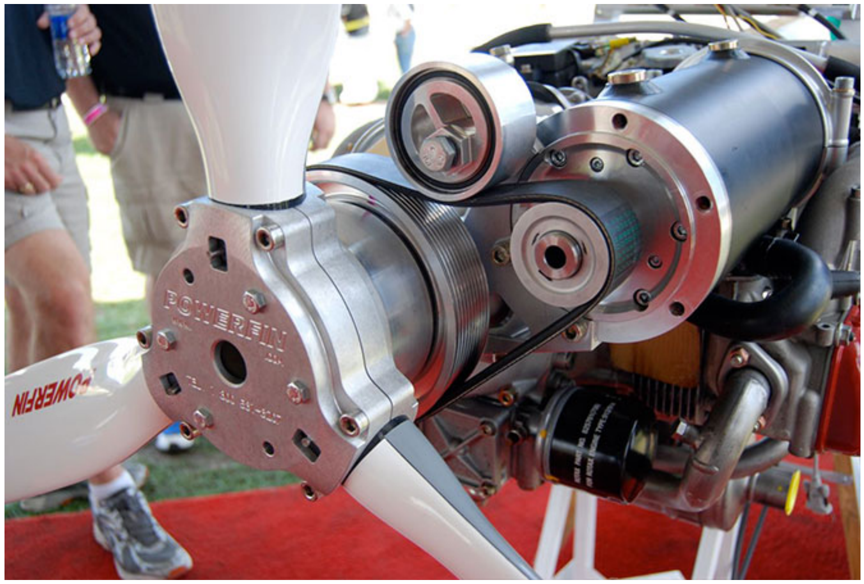

Figure 4: Flight Design's Parallel Hybrid Design [30]

Beyond what Flight Design has done there has been minimal research conducted concerning hybrid-electric GA aircraft. Much of the attention has gone to completely electrifying the aircraft. Minimal progress has been made to produce an all electric aircraft that has adequate performance for the current GA flight profiles. Hybrid-electric aircraft can be the stepping stone.

4. Engines

Hybrid-electric aircraft use two methods of propulsion, an ICE designed for the cruise speed and an electric motor that maximized loiter endurance. For the RPA engine it was highly desirable to use readily available commercial off the shelf (COTS) products for convenience, but the engines available were inefficient and have limited information in terms of performance and reliability[5]. The drive was to build, inexpensive RPAs using compact, thermally efficient, and heavy fuel burning engines that were commercially available[31]. The design of a new RPA propulsion system was dependent on finding the most reliable power plant that utilized a field available fuel. Additionally there were three important aircraft design factors to consider when selecting an engine. Low fuel consumption engines offer increased range for the same amount of fuel which was essential for conducting ISR type missions. To minimize takeoff weight the engine must have a large power to weight ratio, which means the largest output in the smallest package [31]. Finally, the engine must be simple and easy to maintain so that the aircraft can maximize operational use. The same characteristics were assumed for the GA engine.

4.1. Internal Combustion Engine Fundamentals

Propeller driven aircraft use reciprocating engines that operate on a two-stroke or four-stroke cycle. Each engine cycle transmits power through the compression of a fuel air mixture that was ignited, driving a piston up and down turning a drive shaft [32]. The following explains the processes for a four-stroke engine process illustrated in Figure 5.

1. Intake stroke

A fuel air mixture enters the cylinder through an opening or valve.

2. Compression stroke

The residual momentum of the drive shaft pushes the cylinder up compressing the fuel air mixture.

3. Power stroke

Once compressed the fuel air mixture is ignited by the compression or a spark, driving the cylinder down.

4. Exhaust stroke

When the power stroke is completed a valve or opening allows the burned mixture to escape as the piston is pushed up.

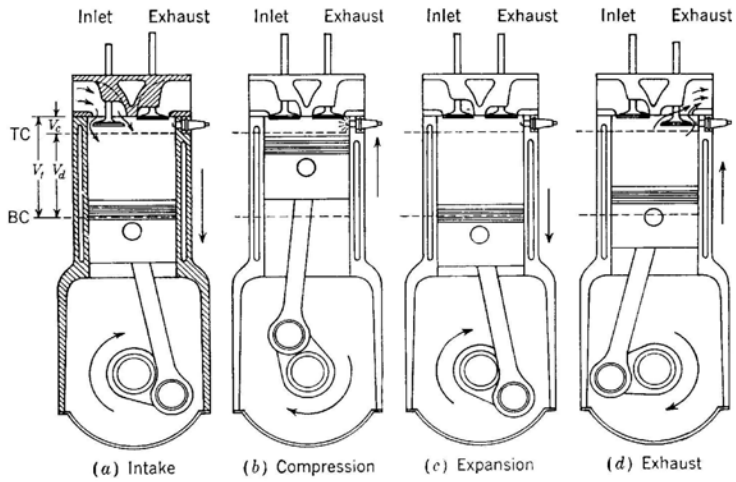

Figure 5: Four-Stroke Operating Cycle [32]

The four-stroke engine completes two revolutions per ignition cycle, but has a low power to weight ratio. To obtain a higher power to weight ratio a two-stroke engine was invented that simplified the process [32]. The two-stroke engine cycle is explained below and illustrated in Figure 6.

1. Compression

The compression allows a fresh fuel mixture to enter the crank case below the piston. Combustion is then initiated using the compression or a spark.

2. Power/Expansion stroke

As the piston moves down an exhaust port in the side wall of the cylinder is uncovered at the same time as an intake port. New fuel is pushed in below the cylinder and the burned mixture escapes through the exhaust.

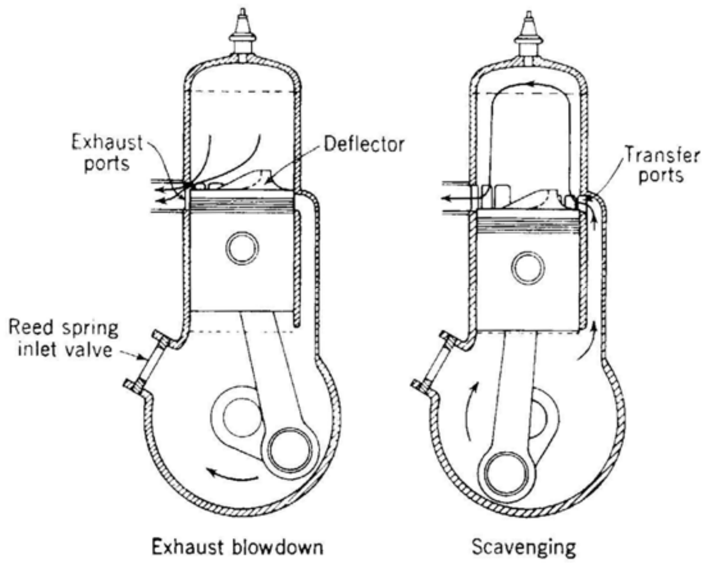

Figure 6: Two-Stroke Operating Cycle [32]

The mechanically simple two-stroke engine only provided one revolution per cycle. Also, during the power stroke, unburned fuel escapes through the exhaust port making the engine inefficient. Choosing the best engine cycle for the hybrid design, will depend on the engine with the highest performance.

The performance of an engine depends on physical dimension, fuel efficiency, and durability. The power of an engine was proportional to three important features. First, the displacement of the engine describes the distance of the piston stroke from top dead center (TDC) to bottom dead center (BDC) [18]. A higher displacement means a longer power stroke. The number of power strokes was represented by the second feature, revolutions per minute (RPM). Power output can be increased for a small displacement engine, to an extent, if the RPMs were increased. Finally, the mean effective pressure (MEP), which can be determined from an average pressure calculation during the power stroke, defines how hard the expansion pushes on the piston. Together all three define the power output of the engine. Equation 1 represents how displacement *d*, RPMs, and MEP are proportional to the power output [18].

$$P \ \alpha \ (diameter) \times (MEP) \times (RPM) \tag{1}$$

Fuel efficiency of engines can be improved by using compression ignition, use of heavy fuels, and electronic fuel injection. The first decision to be made would be to select the method of igniting the air fuel mixture. Spark ignition (SI) engines use a spark plug that introduces an electrical charge that begins the combustion process. Compression ignition (CI) engines use fuels that will combust based on the pressure created by the compression stroke [32]. The compression ignition process was thermally more efficient than the spark ignition [19] and can be used with heavy fuels. The use of heavy fuels was a high priority for newly deployed aircraft because of the availability in the field and the high energy content heavy fuels offered [33]. Lastly, electronic fuel injection will allow engines to perform at higher altitudes by reducing icing problems [34]. Electronic ignition also allows for better fuel flow and mixture control [35]. The engine can use fuel more effectively and operate at the highest efficiency at low and high RPMs. If engines having these qualities cannot be found then it would be necessary to modify existing engines. The trouble was that engines that were needed for RPAs exist in the hobbyist world and little performance data was available.

4.2. Small Internal Combustion Engines Using Heavy Fuel

Over 20 years ago Lawton announced the need for heavy fuel engines to be used by unmanned aircraft of the future [36]. The gas turbine engine suffers poor fuel consumption and was primarily used for large fighter and transport aircraft. Turbine engines were also cumbersome in terms of the weight penalty and the required maintenance. By using ICEs that operate on multiple heavy fuels and have low specific fuel consumption operators can extend the usefulness of an airframe [36]. In the aircraft design process the optimal weight and efficiency of the propulsion system will ultimately result in the maximum payload capacity for the user. Heavy fuel piston engines impart better fuel consumption and have power to weight ratios suitable for small unmanned aircraft [36]. Unfortunately the availability of piston engines in the range necessary for RPAs weighing 5-200kg was limited [35]. This means that the hybrid-electric system for RPAs may be dictated by the accessible power range of current engines.

4.3. Scaling Engines Using Dynamometer Testing

There was a large variety of missions that RPAs were designed for and the market for new missions will demand engine capabilities to match mission requirements. Limited available data of small hobby engines has lead researchers to conduct experiments to characterize these engines using dynamometer testing [31]. This research was conducted at the Air Force Research Labs propulsion directorate located on Wright Patterson Air Force Base and the University of Maryland. The propulsion directorate at AFRL wanted to see how engines designed for glow fuel would handle fuel conversion and improve engines already used in the field [33]. Using an OS 0.91 glow fuel engine, AFRL researchers were able to run regular unleaded fuel gasoline by installing a spark plug and fine tuning the spark advance. A Fuji-Imvac four stroke engine that was used in the Silver Fox RPA was also tested to see if performance can be improved. For both the OS engine and the Fuji engine the output power measured by the dynamometer was nowhere near the manufacturers suggested power rating for the engines. Giving evidence that more research was needed to characterize these engines.

At the University of Maryland several engineers have attempted to predict performance by scaling small IC engines. If the performance of an engine can be scaled the need for engine performance testing can become a secondary requirement when designing an optimized RPA. By performing dynamometer testing for a range of differently sized engines, an understanding of how engine performance scales with size can be estimated [31]. Research has found that the fidelity of the engine data was sufficient to develop scaling laws for small engine performance [31]. It should be mentioned that these power laws presented in the Maryland research were applied to a limited number of engines. More engines need to be tested in order to validate these claims, but the quality of the research was promising for future testing. By plotting available engine data from manufacturer suggested weight vs. power, it can be noted that the trend follows a power law in the form $y = Ax^b$ [37]. Where A and b were constants, x was the mass of the engine in kilograms, and y was the power output in Watts. Similar to the empty weight fraction trend lines, discussed later in this chapter, the constants A and b may change with respect to the class of engines being used. From small scale hobby single-piston engines to large scale multiple piston GA engines.

The efficiency can be related in a similar fashion and was the basis for the performance scalability of small IC engines. The setup and data collection used was suitable for testing and showed repeatability [37]. More tests needed to be run before a scaling trend could be found. Ideally this would encompass at least 30 different engines so that a good sample could be used before generalizing a scaling factor between engines. Experiments were conducted for three engines and as mentioned before the measured performance was found to be nowhere near the manufacturers' specifications. The researchers Menon and Cadou attribute these discrepancies to the standards and fuel used by the manufacturers [37]. The measured performance suggested a similar power law that power increases with weight [38]. Being able to apply this kind of scalability to the engines can be useful in the design routine of the hybrid-electric RPA to determine a maximum size limit.

4.4. Large Heavy Fuel Engines

Over the past few years the military has been working to consolidate the types of fuel used for combat operations. The logistics of transporting multiple volatile fuels has become a burden and a safety hazard for the military. Most of the fuels being considered for this purpose were heavier diesel fuels because of the high flashpoint that makes them less volatile. For jet aircraft that use JP-1, JP-8, or kerosene this would be an easy transition. However, single engine propeller aircraft used for training still used 100LL, avgas, or traditional automotive gas. Research has been done to modify these engines so that they can run on heavy fuels. In order to be considered for new aircraft designs the reliability of these engines must be proven and put through the rigorous inspection of the FAA or military standards. Until that can be accomplished, hybrid alternatives may be useful.

5. Batteries and Motors

For the past century the United States has relied on the automotive industry to satisfy the public's transportation needs. The most reliable and affordable power source has been the internal combustion engine. However, many attempts have been made to produce all electric cars. The common issue was that the electric vehicles had limited range and low speeds. The oil shortage in the 1970's inspired car companies to invest in hybrid-electric car research. By the late 1990's hybrid cars were mass produced but carried an expensive price tag when compared to IC engine powered traditional vehicles. Clean energy and high efficiency motors have been the major advantages of electric propulsive power. The aircraft companies could benefit from the same propulsive efficiency. Only a few, all electric, aircraft have been built and have suffered the same limited range and endurance that motor vehicles have. Designing hybrid systems that use smaller internal combustion engines and high efficiency electric motors would help the transition from hydrocarbon fuels to batteries. In order to better understand the current state of the art technology for batteries and motors, a basic fundamental understanding was required.

5.1. Battery Basics

A battery is a device that stores electrical energy in chemical form. Battery chemistry dictates the performance and application of the batteries based on the energy storage (Wh/kg) and power capacity (W/kg) of the battery cell. Choosing the type of battery can be unique to specific applications. A primary battery cannot be recharged and useful for small low power applications such as a watch battery. Secondary batteries can be recharged, and the number of recharge cycles depended on the chemistry of the battery. The three most commonly used secondary batteries for light weight applications have been nickel-cadmium (NiCad), nickel-metal hydride (NiMH), and lithium (Li-ion, lithium polymer, lithium sulfur). Figure 7 illustrates the energy and power density of some secondary batteries available for hybrid electric vehicles. The blue shaded region in Figure 7 represents the broad application of the Lithium Ion battery.

For specific applications battery selection depends on the energy storage capacity and the power output required. A balance between energy and power can make the battery more

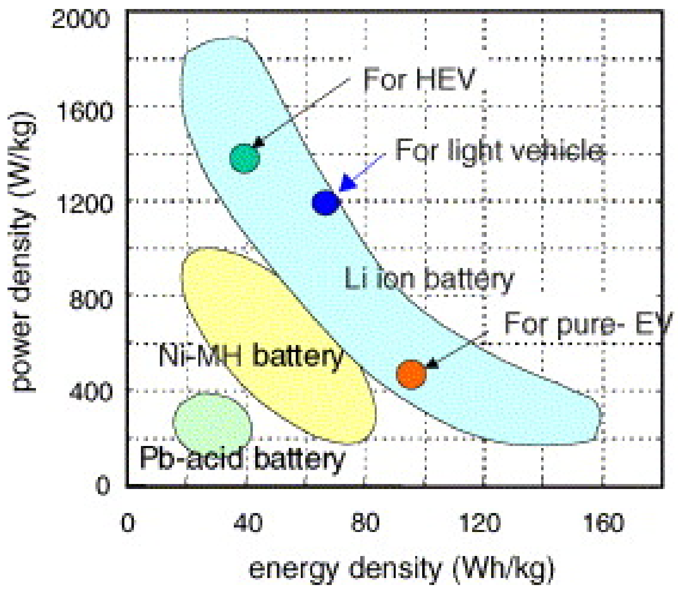

Figure 7: Battery Chemistries [38]

efficient. Lithium ion batteries demonstrated the highest specific power and energy meaning that energy and power can be delivered in the smallest package [39]. A small package was crucial when designing mild hybrid aircraft. The weight of the batteries influences the total weight of the aircraft and impacts the overall performance.

The important question that must be answered for large aircraft applications would be whether or not a battery exists that has the energy storage and power capabilities for a practical mission. Though lithium ion batteries were the superior secondary battery the relative size needed for an all electric aircraft would still be significant. Small li-ion batteries can be found in cell phones, small RC aircraft, and other small electronic devices. Larger scale battery applications were explored. Japan has been one of the leading manufacturers of small li-ion batteries and has worked to expand the application of these batteries. Engineers at GS Yuasa Technology Co. built and tested a

200Ah and 400Ah li-ion battery to serve as a backup power source for high rise buildings. The structural design and complex circuitry were a few of the complications encountered. The eventual completion of the batteries provided a battery that was a 1/3 the weight of the current backup power source, a lead acid battery. One more important aspect of li-ion batteries was that if they short circuit they can explode or catch fire. Tests were conducted on the large batteries and no fire or explosions occurred, meaning that stability of the battery was acceptable for industry use [40].

Another large battery application was presented at the 5th International Advanced Battery and Ultracapacitor Conference by Lithion. The application was for a Navy Seal underwater delivery system that needed a 150V 85.7kWh battery. Since the battery was sealed underwater the thermal management was an important issue. Heat generation was minimal because of the low impedance and high coulombic efficiency. These engineers were also well aware of the dangers due to overcharging and short circuits. Fail safes were implemented that sensed current and temperature to prevent overcharging and short circuits [41]. These two examples, Yuasa Technology and Lithion, demonstrate the possibilities for large lithium ion batteries. In both cases weight was not a driving constraint and these batteries may not be readily applied to aircraft designs. However, future work developing large capacity, light weight batteries, all electric flight may be feasible. Until then current battery technology may only allow hybrid technology to perform at the same level as the traditional engine driven aircraft.

5.2. Motors

There are many types of motors that are used for a variety of applications and each motor type has specific advantages. For hybrid electric volume and power are the driving design factors to consider. When considering aircraft the weight, volume, power, and reliability must be optimized so that electric propulsion can be feasible. The power output of a motor can be determined using the following equations. Equation 2 describes how the output torque of a motor can be determined by evaluating a simple equation using the measured current, no load, current, and torque constant of the motor.

$$Q_m = \frac{i - i_o}{K_Q} \tag{2}$$

The torque constant would be found experimentally and would depend on the design of the motor. The power being delivered to the propeller of an aircraft would be dependent on the rotational speed as well. Equation 3 calculates the rotational speed of the motor shaft based on the voltage and speed constant of the motor. The K_v value of the motor

$$\Omega = v_m K_v = \left(v - iR\right) K_v \tag{3}$$

can also be found experimentally and will become an important motor characteristic when selecting an appropriate motor for the hybrid system. Finally the output power of the motor can be determined using the results of Equations 2 and 3. The electrical power delivery can be determined by simply multiplying the measured current by the measured voltage. The physical power delivery can be found using the measurements for torque and speed of the shaft and propeller recorded here in Equation 4.

$$P_m = \Omega Q_m = \left(i - i_o\right)\left(v - iR\right) \tag{4}$$

The efficiency of the motor can then be determined using the ratio of the power calculations using the characteristic motor values and the physically measured torque and rotational speed in Equation 5.

$$\eta_m = \frac{P_m}{P_e} = \frac{P_m}{iv} = (1 - i_o / i)(1 - iR / v) \tag{5}$$

6. Requirements-Driven Aircraft Design

6.1. RPAs

At the outset of the Global War on Terror coalition forces had a desperate need for unmanned aircraft that could satisfy DDD missions. The environments in Iraq and Afghanistan were vast and unforgiving, manned aircraft can simply not provide the coverage necessary to track a sparsely distributed insurgent force. Unmanned vehicles were the ideal solution to this problem, but the recent rapid deployment of inadequate systems lack desired mission capabilities. The Warfighter was demanding specific capabilities that were unrealistic because of the confusion between the soldier and the aircraft designer [42]. This was attributed to the requirements creep of the Warfighter. Requirements creep by the user was the desire to change the mission capabilities of an existing system without considering the limitations of the aircraft's design [42]. Therefore it was essential that during the design process requirements were clearly defined and the system had the ability to adapt to changing mission needs.

The US military needed to provide realistic, clearly defined design requirements so that new RPA platforms can satisfy the needs of the soldiers on the front line. These requirements need to include information pertaining to the mission profile, payload, desired cruise speed, loiter speed, maintainability, usage, and range [43]. Once these can be clearly defined designers must pay close attention to the specific design requirements and anticipate increasing the capabilities of RPAs once designed and fielded. Field modification has recently been accomplished on the AAI Shadow 200 shown in Figure 8.

Figure 8: AAI's Shadow 200 (www.defenseindustrydaily.com)

The platform was made over by applying expanded wings and implementing a more efficient fuel injection system to the engine [34]. These improvements have increased the endurance by 2 hours and enable the Shadow to carry a weapons payload. Modifications of this type improved performance but were expensive to make post-production. Anticipating the need for multi-mission capability, the hybrid-aircraft will allow mission flexibility without the need for expensive alterations.

The mild hybrid conceptual design tool was meant to optimize flexible component requirements producing aircraft that met the specifications of the user [44]. Since mission design analysis frequently points towards new concepts and technologies the hybrid-electric concept was developed [17]. Making the components moldable to specific mission profiles, a hybrid platform can be made to satisfy the multiple mission capability desired by the United States military. Typical long endurance and extended range mission profiles were illustrated in Figure 3a and 3b. Hybrid-electric RPAs incorporate these missions into one platform, one for quiet reconnaissance and one for increased range. For aircraft on the scale of GA aircraft, a mild hybrid-electric system was the best solution. Furthermore, users could take advantage of the hybrid's

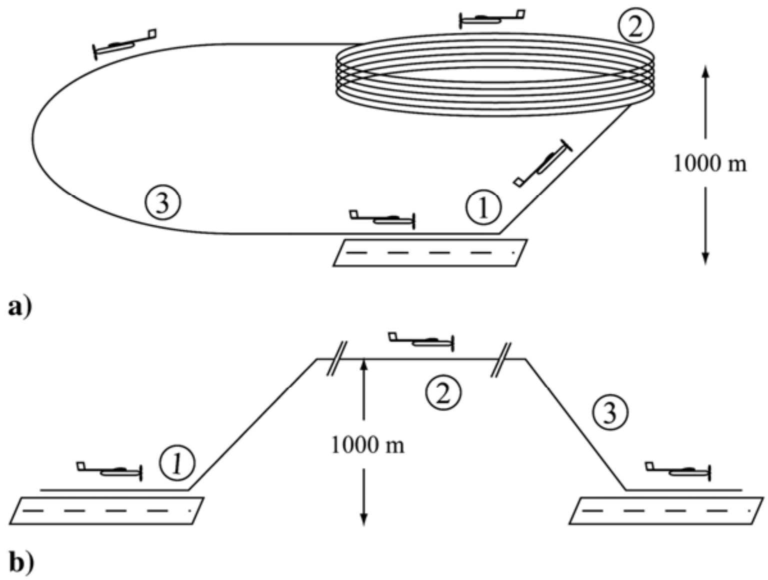

Figure 9: Endurance and Cruise Mission Profiles

propulsion system by the tradeoffs between the battery, fuel, and payload. A greater portion of battery storage could be used on stealthy low altitude ISR. A greater fuel load could extend the reach of a needed mission. Finally, modular sensor payload capability would be available [43]. With the hybrid technology, sensors could be a wide variety of types and weight by the simple adjustment of the battery and fuel weights.

6.2. General Aviation

The requirements for GA aircraft were well established by the existing platforms that currently exist. Unlike the RPAs, most single engine aircraft share a similar mission profile. The typical single engine GA aircraft carried two passengers and some additional baggage payload. The performance criteria for GA aircraft were found in flight manuals. From a design perspective the driving force behind the design were maximum range, minimum weight, and safety. So for the design of a mild hybrid-electric there would be much less concern for modularity or mission capability than the RPA design. Once a GA hybrid aircraft can be flown the design considerations would turn toward improving energy storage and transition from the dominant ICE to a more efficient EM.

7. Aircraft Performance

The most accurate way to determine the true performance of an aircraft has been to perform flight tests. To avoid major design changes of aircraft prototypes, a robust design method must include all aerodynamic quantities. Dimensionless coefficients for the aerodynamic forces and moments applied to aircraft present a more fundamental description of airframe performance than the forces and moments themselves [18]. The use of dimensionless quantities has enabled aircraft engineers to simulate real world condition on scaled models used in wind tunnels. For the hybrid-electric RPA the use of scaled aerodynamic quantities should help identify the aerodynamic qualities needed for the airframe. The accurate simulation for new aircraft makes the final product more affordable in terms of modification and meeting desired performance.

7.1. RPA

The hybrid-electric RPA design has used two unique design points for the engine and the motor. The motor was sized for the highest endurance possible for loitering mission segements. The minimum of the power required curve defined the slowest velocity the aircraft can travel above the stall speed, and would be optimal for loitering [2]. The hybrid-electric RPA must also achieve a higher speed to ingress and egress from target locations. This would be provided by engine power, and was more dependent on the mission requirement than optimal design. The two speeds would be difficult to achieve with traditional ICEs or EMs alone[15]. The combination of the two used in the hybrid design would consume power and energy more efficiently. Each of the two design points were considered when the aircraft was in steady level un-accelerated flight (SLUF). During SLUF, Equations 6 and 7 shows the force balance for the aircraft, where T was the thrust of the aircraft, D represents the drag, L was the lifting force, and W represents the weight of the aircraft.

$$T = D = \tfrac{1}{2}\rho V_\infty{}^2 S C_D \tag{6}$$

$$L = W = \tfrac{1}{2}\rho V_\infty{}^2 S C_L \tag{7}$$

Below, using Equations 8 and 9 the lift and drag coefficients can be found.

$$C_L = \frac{W}{\tfrac{1}{2}\rho V_\infty{}^2 S} \tag{8}$$

$$C_D = C_{D,0} + \frac{C_L{}^2}{\pi e AR} \tag{9}$$

The thrust of the aircraft must equal the drag so Equation 10 was formed

$$T_R = \frac{W}{C_L / C_D} \tag{10}$$

$$Power = \frac{energy}{time} = \frac{force \times distance}{time} = force \times \frac{distance}{time}$$

The power required to maintain SLUF can be found to be Equation 11

$$P_R = T_R V_\infty \tag{11}$$

Knowing the quantities for weight, wing area, and local density calculations can be performed to generate the thrust and power required curves for an airframe.

These calculations were simplified to demonstrate the design method. The use of these equations alone provides an estimate for the conceptual design. Harmon’s code takes into account many more aerodynamic principles and quantities but must be improved so larger sized RPAs can be modeled. Making the hybrid-electric RPA design more robust would require a stronger knowledge of the aerodynamic and structural behavior of larger aircraft. Revolutionary methods to define them were needed because of the complexity of the hybrid-electric system.

7.2. GA Aircraft

Using the same step by step design process as the RPA a conceptual mild hybrid GA aircraft could be designed. The same aerodynamic equations used for the RPA can be applied to the mild hybrid design with a few minor changes. The mild hybrid design needs the electric motor to assist at the takeoff, climb, dash, and possibly landing conditions. The engine requirement would satisfy the cruise speed because the cruise is the longest mission segment in a typical GA platform. After optimizing the engine at the cruise condition all additional power needs would come from the EM. Additional power requirements would be governed by the largest power needed for takeoff distance or climb rate. Simple calculation can be used to evaluate the necessary motor power and will be explained later in Chapter III.

8. Aircraft Design

Since the Wright brothers, there have been many advances in the techniques used to design aircraft. Airplanes needed to fly higher and faster to be effective commercial transports and be more useful to the military. The size of aircraft was limited by materials because the wooden models created by the Wright's and others had little structural integrity. The weight of aircraft was limited by the available engine power. Until the jet engine, piston powered propellers were standard and rapidly became inadequate. Since the first flight, materials and engine technology have significantly improved as well. The design of aircraft has been dictated by these available technologies and the improvements have expanded aircraft applications. The majority of these improvements have been applied to manned aircraft. Subsequently there were well documented design strategies to build new aircraft. Conversely, modest efforts had been given to improving small unmanned aircraft design until the Global War on Terror began. The urgent need for RPAs to provide unique ISR and combat capabilities has revealed the inadequacy of current RPA design methods. Similarly, the recent concern for fuel efficient aircraft has called for unique propulsion concepts for all aircraft including GA platforms.

8.1. Traditional Conceptual Design

For commercial aircraft the complex process of decision making with regards to design has been supplemented by vast historical data that provide empirical relationships of important design variables [45]. The difficulty for RPA design was the wide variety of capabilities that must be met without a database of historical reference to allow decisive action by unmanned aircraft designers. Mission requirements have been a consistent starting point for most aircraft. At the conceptual stage simple optimization methods can be implemented using constraints driven by historical data to minimize cost. This was useful for the mild hybrid GA aviation case but the progression to all electric aircraft may make this method ineffective for both RPA's and GA aircraft. Due to the inexpensive nature of RPAs it has been easy to use conceptual design methods that have been used on larger aircraft using trial and error. However the traditional methods cannot capture the full potential of hybrid RPAs. The development of a robust conceptual design tool for single engine aircraft and RPAs may help make hybrid-electric propulsion a reality.

The most important ISR mission requirements for RPAs relate to the payload capacity, range, and endurance. If a specific mission profile was desired a new aircraft could be developed using data mining. Neufeld and Chung at Ryerson University in Canada created a database that interpolates categorized RPA data to aid configuration decisions [45]. The goal was to develop the empirical relationships that were available for larger aircraft. The algorithm accepts a desired mission, then the data mining extracted useful information from the database, then returned the information to the algorithm, which proceeded to iterate on a design until converged. The method was tested using existing RPA platforms and the associating mission profile. Results indicated that the algorithm improved the efficiency of RQ-7 Shadow and the Gnat 750 airframes. However, Neufeld and Chung concluded that the limited RPA database entries for certain categories made these results invalid [45]. The framework of the algorithm needed to include structural analysis to allow a more detailed design analysis. A more direct approach to RPA design would be to use the traditional methods used on conventional aircraft.

A popular source for aircraft designers has been Daniel P. Raymer's book *Aircraft Design: A Conceptual Approach* [17]. Raymer has presented a simplified method for estimating the initial design of an aircraft based on mission requirements. The first step, identified the desired mission and determined the estimated gross weight of the aircraft [17]. For commercial aircraft this has been where empirical data was useful, but for RPA design more thought was needed to estimate takeoff weight. Using the fuel weight that burns during mission segments, Raymer defined fuel weight fractions for mission segments by calculating the ratio of weight before and after each segment. Segments can be takeoff, cruise, loiter, maneuvers, and landing. To maintain an accurate mission profile each segment should have a fuel fraction of 0.8 or greater. In order to calculate the total weight of the aircraft, an initial guess has to be made. The analysis prescribed by Raymer produced a calculated takeoff weight from the initial guess. If the guess did not match the calculated value for the takeoff weight the designer iterated using a new guess until guessed takeoff weight matches the calculated takeoff weight [17]. This process provided a simple baseline that can give engineers the insight as to whether or not a conceptual design would satisfy given mission requirements.

These back-of-the-envelope type calculations can be important so computer time and research money would be saved on a project that might have failed initial design limitations. With the lack of statistical data for electric or hybrid propulsion systems traditional sizing methods like Raymer's need to be used on a case by case basis to represent the most accurate hybrid design. The most crucial difference between engine only and hybrid propulsion will be the weight fraction considerations. Traditional weight fractions take into account that fuel was being burned during a given segment. However, if battery power was used this assumption becomes false and segments using battery power should be adjusted accordingly. More in depth analysis was needed for battery powered aircraft that can follow the same basic principles that were used in the simplified approach used by Raymer. Especially for the hybrid design, the unique characteristics of using both an IC engine and an electric motor for propulsion should be accounted for in the design.

8.2. Unconventional Aircraft Design

Aircraft that have used electrical or fuel cell based propulsion have not followed the same trends as traditional conceptual design would suggest[46]. Aircraft that use IC engines have low energy efficiency causing needed alternatives to hydrocarbon fuel usage. The expanding market for RPAs has encouraged increased complexity that has allowed the use of revolutionary propulsive systems[47]. However system-level studies have attempted to force revolutionary propulsion systems into conventional architectures without considering the need for revolutionary design methods[46]. In order to design a hybrid-electric aircraft a new design methodology must be used. Hiserote helped identify the mission capabilities but a sizing limit for the hybrid's takeoff weight was still needed to gauge the usefulness it might have based on mission analysis [15]. The application of different methods must be explored and possibly melded together to create an accurate sizing model.

Research at the Georgia Institute of Technology has developed a generalized power based sizing method [48][46]. The method used traditional methods for sizing using power constraints and mission analysis. The traditional method analyzed point performance, such as climb, sustained turn, and acceleration expressed as wing loading and thrust loading. These values were then used to define a geometry and propulsive need. In order to be applied to aircraft consuming unconventional energy a great deal of modification in the formulation was required [48]. The most basic modification was to account for the limited knowledge of SFC and scalability of revolutionary concepts. By analyzing the specific energy, the SFC and power characteristics of revolutionary systems can be described for the purpose of aircraft sizing [46]. The following, Equations 12 and

13, reflect the method used. This was also applied to propulsive systems using more than one energy source.

$$P_{propulsive} = \eta_n \eta_{n-1} ... \eta_1 \eta_0 P_0 = \Pi(\eta) P_0 \tag{12}$$

$$W_{PD} = \sum_{k=1}^{n_{PD}} W_{PD} = \sum_{k=1}^{n_{PD}} \frac{P}{\upsilon_{PD_k} \Pi_{n \to k}(\eta)} \tag{13}$$

The revolutionary idea of using multiple energy sources to propel aircraft complicated the traditional conceptual design approaches. Typically aircraft lose weight during flight because of fuel burn. Using a battery, the weight would not change according to the mission segment weight fraction calculations discussed. Researchers from the Georgia Institute of Technology have published methods to overcome the difficulties for initial sizing of aircraft using multiple energy sources. For each mission segment they identified the individual energy and power paths taken and determined whether consumable energy or non consumable energy was propelling the aircraft. Using these calculations for each mission segment the overall mission analysis would yield initial size and weight estimation [46] [48].

The approach used at the Georgia Institute of Technology optimizes the energy storage and the power needed to complete a specific mission of a new aircraft [48]. Researchers were upset because the performance of recent revolutionary propulsion systems have suffered because the systems were retrofitted into an existing architecture. They advocated that the full potential of a revolutionary concept can only be realized once accurate sizing methods can be established for the new propulsive system [46]. This thesis was meant to demonstrate that a full hybrid system could be applied to RPA designs and a conceptual mild hybrid design could replace the large IC engine in some

GA aircraft. Traditional conceptual design approaches were modified accordingly. Since the selected mission profile used the IC engine was used in all mission segments, the weight fractions were calculated based on fuel lost. The goal was to improve the fuel consumption of existing airframes using hybrid-electric technology by incorporating a smaller engine. Once the conceptual tool can be validated new aircraft designs can be generated based on mission requirements. Careful consideration was taken to evaluate weight fractions and calculate energy need based on multiple energy sources in the hybrid-electric system. The following chapter outlines how the different conceptual design methods were applied to the mild hybrid-electric propulsion system for GA aircraft and RPAs.

III. Methodology

1. Chapter Overview

Traditional aircraft design was not readily applicable to aircraft that use multiple energy sources. Therefore a new design method was needed to account for the differences when using hybrid propulsion. The purpose of this chapter was to explain in detail the design method used to establish a basic aircraft design code that takes advantage of hybrid propulsion. Beginning with the selection of the hybrid configuration, and then walking through how the design process was established, this chapter clarifies how a mild hybrid-electric aircraft would be designed. For the design of a small full hybrid-electric RPA, please refer to Hiserote and Harmon [5] [15]. The middle of the chapter explains the optimization routine that was developed using aerodynamic equations and assumed variable quantities. The chapter finishes with a description of how the case study hybrid aircraft was comparably measured.

2. Hybrid Configurations

In the automotive industry hybrid electric cars have used both series and parallel hybrid configurations. Series hybrids were most often used for heavy vehicle applications such as buses and trains. The popular plug in hybrids can be classified as parallel hybrids because both power sources can be used independently. In order to accurately choose the correct configuration for aircraft a specific mission profile must be developed to identify the power needs at different design points. For RPA applications, the design points considered were; the endurance speed for long on station loiter time and the cruise condition so that the aircraft can get on and off station quickly. For a GA aircraft excessive power was needed at takeoff and climb. Otherwise cruise power was significantly lower at altitude. Therefore, a mild hybrid configuration was the optimal choice for the large RPA and single-engine GA aircraft platforms.

2.1. Design Process

The purpose of designing a hybrid propulsion system was to improve the performance and fuel consumption of selected platforms. Revolutionary propulsion systems require the use of unconventional design strategies [49]. However, the design process for traditional aircraft using internal combustion engines can be used as a frameword but must be modified accordingly. The current sizing methods anticipate each mission segment burning fuel and reducing aircraft weight throughout the mission. For an aircraft using only electrical energy no weight would be loss due to fuel burn. These unique characteristics were not taken into account since the mild hybrid's electrically powered mission segments were short and still used engine power. The anticipated fuel savings would be the result of properly sized components operating at optimal efficiency. The performance requirements and constraints were based on the present configurations of several viable airframes, and were modified with hybrid-electric propulsion systems. The resulting performance was measured. The design process featured in Figure 10 demonstrates how the aircraft design would iterate until a converged solution yields a feasible design.

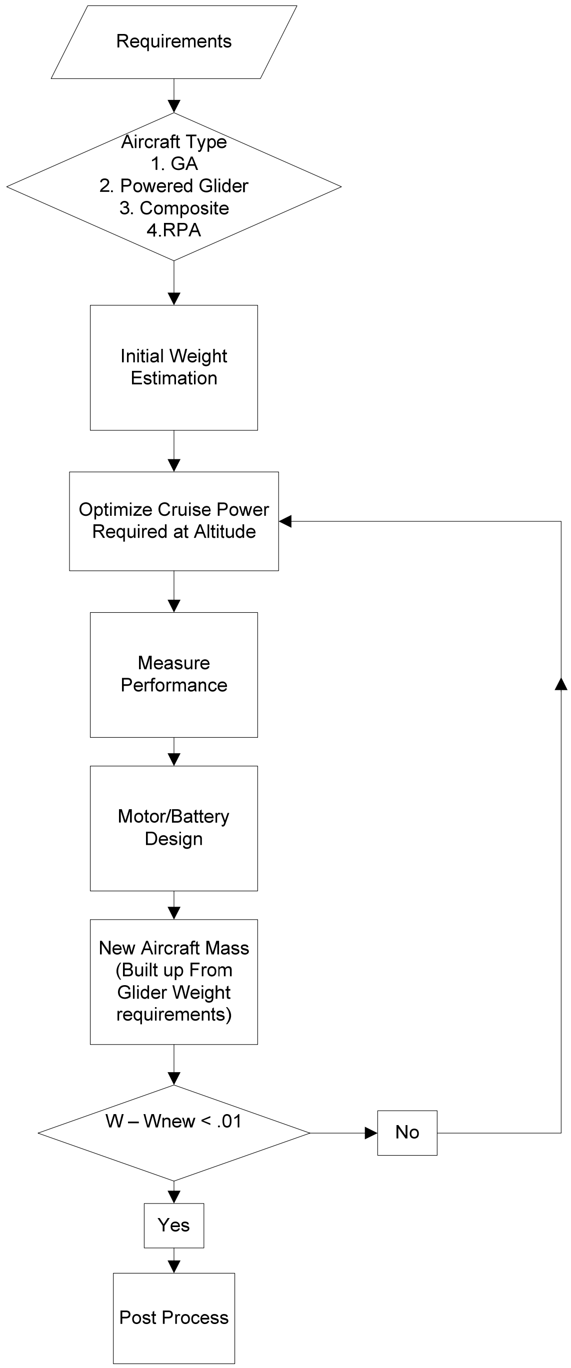

Figure 10: Design Process

The process was then written in MATLAB so that user inputs could produce a rough aircraft design and hybrid propulsion system including engine, motor, and battery.

2.2. Requirements

The requirements defined for the hybrid RPA systems and GA aircraft were from the original design performance for selected aircraft. The most important point performance characteristics considered were takeoff ground roll, altitude performance, rate of climb, power required at cruise condition, and sustained turn g-loading. Each parameter was carefully determined using existing data for each aircraft. The requirements were used to develop constraints for an optimization routine, as well as calculate component sizes for the hybrid power plant. The MATLAB code developed allows the user to input estimated aerodynamic parameters such as Oswald efficiency, $C_{L,Max}$, and prop efficiency. Then the user was able to set desired performance requirements for cruise speed, rate of climb, payload, and takeoff distance. The user can then limit wing area, wingspan, or aspect ratio constraining the rest of the optimization. The last user input was the existing weights of aircraft components so that a comparison can be made between the original and a hybrid substitute system. Once these requirements were input, the code determined a conceptual design of a hybrid aircraft using modified traditional sizing methods.

2.3. Weight Estimation

The most traditional form of determining the conceptual weight of aircraft has been the use of historical data. Many major aircraft companies have resources that allow them to quickly determine a rough estimate of initial aircraft weight based on past designs. These databases use the performance criteria of similar platforms and extrapolate based on regression lines that fit the historical data. One of the most recognizable aircraft designers Daniel Raymer includes some of this data in his book. Raymer's method used the weight fractions derived from a desired mission profile and the empty weight regression lines in Table 3.1 of his book, *Aircraft Design: A Conceptual Approach*, to iterate upon an initial guess until the guess matched the calculated weight [17]. This same method was used for the hybrid propulsion system design to come up with an initial weight estimate. To more accurately represent the weight of a hybrid power system a new iterative method was necessary to determine the final hybrid aircraft weight.

To complete the initial weight estimation the weight fractions associated with each mission segment needed to be estimated. To establish the most basic conceptual design tool the fuel burned, at each hybrid mission segment, was calculated as if the engine were providing all power. This estimate was used because the segments using the electrical motor were short and would be difficult to estimate the fuel savings and was considered negligible. Also since the engine and motor work together at takeoff and climb some fuel was used regardless. So the changes in aircraft design that were a concern for all electric aircraft using a battery as a non-consumable energy source can be avoided since the mild hybrid uses the battery energy for a relatively small portion of the flight profile [20] [48]. This allowed the original range estimation and fuel fraction estimates found in Raymer to be useful for the conceptual mild hybrid design. However, the contribution to the GTOW of the battery and the motor can provide significant insight for the future design tools applied to to all electric aircraft. Until then reliance on traditional methods was necessary.

Since the hybrid system was meant to be retrofitted to an existing aircraft, a weight buildup from the original glider weight can be used to calculate the final conceptual weight. To produce the glider weight the fuel, engine, and payload was subtracted from the max GTOW of the original aircraft. With just the airframe left the hybrid propulsion system could be added to this weight for the hybrid design GTOW. The optimization routine and other subroutines in the MATLAB code were used to calculate the component weights for the engine, battery, motor, fuel and payload. By adding these components to the glider weight the final conceptual weight was found. The weight was important to determine first because the rest of the performance calculations use a weight in order to calculate cruise power required, rate of climb, and takeoff distance. To meet all the performance criteria, the hybrid system's conceptual weight buildup must be iterated through the optimization routine until the weight converges to satisfy each requirement. Once the weight buildup was calculated, the converged solution yielded an estimate for the physical dimensions of the aircraft.

3. Optimization Routine

An optimization routine was developed to calculate the optimal physical dimensions and determine the power needed for the ICE at the cruise condition. The important parameters for the optimization of aircraft were wing loading, wing lift coefficient, wingspan, wing area, and aspect ratio [5]. It would be difficult to anticipate the final weight and aircraft size if a new airframe were being developed. This code was only meant to serve as the first conceptual blueprint for a hybrid propulsion system being retrofitted to an existing airframe.

3.1. Cost Function

Since the optimization routine was meant to calculate the ideal engine for cruise the cost function was derived from the SLUF equations discussed earlier in Section 4 of Chapter II. Using the equations for the lift coefficient and drag coefficient simultaneously a single equation can be derived for the power required at a given flight condition. This relationship was found in Raymer's text in the form of Equation 14.

$$P = \frac{1}{\eta_{prop}\eta_{mech}}\left[\frac{1}{2}\rho V_{Cruise}{}^{3} S C_{D_0} + \frac{W}{S}\frac{KW}{\frac{1}{2}\rho V}\right]$$

$$where\ K = \frac{1}{\pi e AR} \qquad (14)$$

Equation 14 was a strong function of weight, cruise velocity, and wing loading which was consistent with power required using the SLUF equations. The power needed to overcome an increase in speed was parabolic because the drag function has a velocity squared term. Since the drag was equivalent to the thrust required in SLUF the drag was multiplied by velocity again to yield power required which was why the power in Equation 14 has a velocity cubed term.

The easiest way to calculate the engine power required for the hybrid electric system was to calculate the engine size based on the cruise condition. The typical power profile for single engine aircraft consisted of full power for takeoff and climb, and then the power needed to maintain steady flight was dramatically reduced [20]. As a result the cost function was centered on the engine power required at the cruise condition. Minimizing this power allows the motor to pick up any additional power needed through the mission profile. Meaning the engine selected can then operate at the ideal operating line for maximum efficiency of the given mission and would reduce the fuel wasted on engine inefficiency at cruise. The desired cruise power must then be adjusted to account for the altitude effects on the engine's operation. Anderson estimated the power loss at altitude using the density ratio compared to sea level [2]. Equation 15 was used in the conceptual design code to determine the engine's necessary horsepower needed at altitude.

$$hp_{A,alt} = \frac{\rho_{alt}}{\rho_0} hp_{A,0} \tag{15}$$

The propeller efficiency was accounted for in the cost function so no additional power would be needed. Using this method allowed the engine size to be more accurate for varying altitude requirements.

To minimize the power required at cruise the important design variables were wing span, wing area, and the relationship between them, aspect ratio. The weight was found earlier using the weight estimation, air density and cruise velocity were then determined from the design requirements. Finally, the Oswald efficiency (e) and drag polar were estimated from historical data. Once constrained the cost function yielded results for the design variables wingspan and wing area.

3.2. Constraints

The constraints for the cost function in Equation 14 were found considering performance and structural limitations. Without constraints the design variables of wingspan and wing area would attempt to reach unreasonably high aspect ratios and would produce useless aircraft. Simply bounding the aspect ratio would limit the robust nature of the code. Constraints needed to be found that were easily applicable to all GA and RPA aircraft.

The wing loading of aircraft was found to be an important parameter in multiple design strategies [17][46] [48]. At the end of Raymer's text a sample conceptual design was found that illustrated the step by step process needed. To determine a desirable wing loading several wing loading conditions were calculated for stall, takeoff, climb, and cruise. The most conservative of these calculations was the wing loading at stall and was used for the rest of the design problem [17]. Other wing loadings could be used but may cause problems in the end meeting certain performance criteria. Equation 16 below was the constraint produced from the stall wing loading condition.

$$\frac{W_0}{S} - \frac{1}{2} V_{Stall}{}^2 C_{L,Max} \leq 0 \tag{16}$$

The next concern was the structural limits for the aircraft. A sustained turn can generate some of the largest load factors experienced by the aircraft. As a precaution many aircraft manuals restrict large control surface movements above a certain speed that was called the maneuver speed or V_a so that high load factors were not reached. Equation 17 gives insight for the maximum turn rate of aircraft based on their aerodynamic quantities.

$$L = nW_0 = \frac{1}{2} \rho V^2 S \sqrt{C_{D_0} \pi e AR} \tag{17}$$

Judging by Equation 17, high load factors can be achieved by increasing the aspect ratio. The maximum sustained load factor allows for the maximum turn rate for the aircraft. For the hybrid-electric aircraft, high load factors and large turn rates were not desirable therefore a load factor of 2 at the maneuver speed would be sufficient. Substituting the maneuver speed and load factor into Equation 17, a constraint that limits the aspect ratio was produced by using Equation 18.

$$AR - \left[\frac{2nW_0}{\rho V_a{}^2 S} \right]^2 \frac{1}{C_{D_0} \pi e} \leq 0 \tag{18}$$

3.3. Outputs

The output of the design code would be the wing span and wing area that can then be used to calculate the rest of the hybrid-electric propulsion system. By using the fmincon function in MATLAB the cost function and constraints change the design variables until a minimum cruise power can be found. MATLAB then displays the engine power needed at the cruise condition, wing span, and wing area. These values can then be passed to the next portion of the code.

3.4. Initial Physical Dimensions

The aircraft sizing was used to determine the physical scale of the configuration to satisfy the mission requirements [48]. The wing span and wing area for the hybrid design were determined using the optimization routine that was developed to minimize the power required for the engine at the cruise condition. Left unbounded the wing span and wing area became very large since the power can be greatly reduced by increased aspect ratio. To avoid unreasonable dimensions and verify that the propulsion system could be placed in an existing system the wingspan was constrained to the span of the original aircraft. By doing this the design was verified if the wing area calculated matched the wing area of the original aircraft. Now that the weight, wing area, and wingspan have been estimated, performance equations can be used to judge how well the engine alone can satisfy the requirements.

4. Performance

The important performance characteristics of the hybrid aircraft were directly related to the requirements of cruise, takeoff distance, and climb rate. The initial matched hybrid design was meant to prove that GTOW and performance could be near the original using a motor and battery to assist an ICE. In the future, the improvement of battery technology may lead to full-electric aircraft. In order to not sacrifice the endurance and performance of current ICE powered aircraft, electrically assisted hybrids could be the first step toward full electrification. The greatest challenge was establishing an accurate conceptual design strategy that can reasonably estimate the propulsive power needed from multiple sources to satisfy discrete performance criteria. Although the Georgia Institute of Technology researchers have previously defined a robust sizing algorithm based on performance constraints [48], a much simpler conceptual tool was developed for modified general aviation aircraft. Performance constraints were developed based on a specified mission profile and the defined design point for each power source.

Additional power requirements were met by an electrical motor that was sized based on initial performance goals. The defined requirements were met by optimizing the ICE altitude cruise power, for an existing airframe. With the use of Equation 19 the takeoff ground roll was determined based on the optimized wing area, weight, and other aerodynamic quantities found for several different airframes. Since it was hard to

$$s_{LO} = \frac{1.44W^2}{g\rho_\infty SC_{L,Max}T} \tag{19}$$

determine the thrust output of the original aircraft the best estimate for the thrust at takeoff was taken from the thrust produced at the cruise condition annotated here in Equation 20. Once the thrust can be determined from the power output of propulsion

$$Thrust = \frac{Power}{V_\infty} = \frac{P_{R,Cruise}}{V_{Cruise}} = T_{max} \tag{20}$$

system or using the previous method the power required at takeoff for a desired ground roll can be calculated. By rearranging the takeoff distance equation, Equation 21 demonstrated how the power for takeoff can be calculated. Now that takeoff ground roll

$$P_{R,takeoff} = \frac{1.44 V_{Cruise} W^2}{g \rho_\infty S C_{L,\max} s_{LO,desired}} \tag{21}$$

was determined the rate of climb performance requirement needed to be evaluated. The simplest approach for the rate of climb was recorded in Equation 22. The power available

$$ROC = \frac{P_{Available} - P_{\text{Re}quired}}{W_0} \tag{22}$$

would be the total power of the engine and motor, the smallest power required for the airframe was the minimum of the curve developed in Section 4 of Chapter II. So the largest rate of climb would be when the motor and engine were at max power while the aircraft was flying at the speed associated with the lowest power required.

The performance points discussed were implemented in the design code to ensure the mild hybrid-electric propulsion system could meet desired requirements. The requirements had a strong influence on the outcome of the code. The requirements acted like constraints for the GTOW iteration and could be made more stringent or relaxed to yield a desirable result.

5. Motor and Battery Design

The motor and battery design were determined after the engine was optimized for the initial physical dimensions at the cruise requirement. Since the engine was optimized for a single operating condition the motor needed to supply additional power for the other design points. Two such requirements would govern the size of the motor, desired takeoff distance and the required rate of climb. Both performance criteria were taken from the original propulsion system in the aircraft. Using Equation 21 from before the motor size was calculated for takeoff using Equation 23. The additional power needed for climb was

$$P_{EM} = P_{R,Takoff} - P_{R,Cruise} \tag{23}$$

then calculated using Equation 22 rearranged to form Equation 24. The minimum power required was used because this would yield the maximum climb rate for the available power. motor size so that every performance parameter was met.

$$P_{EM} = W_0 (ROC)_{Desired} + P_{R,\min} - P_{R,Cruise} \tag{24}$$

The battery size was calculated by determining how much energy was needed to climb to the operating altitude with an additional 5 minutes (600s) for an emergency procedure. The method for calculating the battery size was derived from the fundamental battery specific energy storage (Wh/kg) and the motor power multiplied by the desired time needed shown in Equation 25. Judging the accuracy of this estimate, there were

$$m_{batt} = \frac{P_{EM} \left({}^{MaxOperatingAltitude}\!/_{ROC_{\max}} + 600 \right)}{Batt_{SpecificEnergy}} \tag{25}$$

multiple situations to consider. First, the time to climb was purposely conservative since GA aircraft rarely reach maximum operating altitudes. Also, aircraft using IC engines to climb generally lose thrust at higher altitudes so the rate of climb would decrease. However, since the motor was used for climbing only a portion of the thrust would decrease since the motor would be unaffected at higher altitudes. Therefore it was concluded that the estimate was reasonable and leans toward the conservative side of the spectrum.

6. Code Validation

Each performance requirement must be met so that a hybrid propulsion system can replace existing IC engines in several case study aircraft. The performance criteria were unique for each aircraft investigated and input into the MATLAB code accordingly. By comparing the physical dimensions of the mild hybrid-electric design code to several original GA aircraft designs the code could be validated. The first measure of validation would be how close the hybrid-electric design's size and weight matched the original configuration.

A second measure of the validation would be the power ratio between the electric motor and engine. According to simulations performed by Lukic and Emadi at the Illinois Institute of Technology, the ratio between the engine and motor can be defined as hybridization factor, and should be between 0.3 and 0.5 [50]. Equation 26 was the equation used to determine the ratio between the electric motor and internal combustion engine. To validate the hybrid drive train, the hybridization factor will be calculated.

$$HF = \frac{P_{EM}}{P_{ICE} + P_{EM}} \tag{26}$$

The hybridization factor defines both full and mild hybrid designs. The lower values near 0.3 would be classified as mild-hybrid propulsion systems. Higher values near 0.5 would be categorized as full-hybrid [50]. Previous work by Flight Design on a GA aircraft shall be used for comparison to confirm the appropriate ratio [30]. The prototype constructed by flight design has a hybrid-factor of 0.26. Once validated, the code can be used for the conceptual design of unique aircraft designs that may be more suitable to the revolutionary hybrid-electric system.

IV. Results and Discussion

1. Overview

The following chapter summarizes the research conducted concerning the conceptual design for hybrid-electric aircraft. The beginning of the chapter outlines how the governing performance requirements were determined for both the general aviation and remotely-piloted aircraft cases. Then several economical aircraft were selected based on reasonable flight profiles and physical size. The following case studies were performed using the code outlined in Chapter III to determine how well the hybrid-electric propulsion system performed compared to the original configuration. Since most of the conceptual design was not easily applicable to the RPA designs, explanation was given for how the same code could be modified to include RPAs. Finally, the usefulness of the code was evaluated based on the case study aircraft.

2. Requirements Analysis

Defining the relevant performance characteristics was a challenge because of the limited information available for some aircraft. The easiest way to find accurate data was the use of flight manuals for the GA aircraft and extensive web-based searches for the RPAs. The essential physical parameters were determined from similar conceptual design strategies used by Raymer, Anderson and others [18][17][46][5][43]. The most important parameters were; wing loading, wingspan, wing area, aspect ratio, payload mass, lift coefficient, and drag coefficient. These parameters dictate the performance of the aircraft and can be manipulated to optimize the hybrid-electric propulsion system. For this conceptual design the takeoff ground roll, rate of climb, power required for SLUF, and maximum sustained turn rate were the performance criteria that needed to be examined.

3. Case Studies

Several case studies were performed to evaluate the effectiveness of the conceptual design code. Three airframes were selected to ensure the code could handle multiple aircraft types including general aviation and remotely-piloted aircraft. The Diamond Aircraft DA 20 and Cessna 172 Skyhawk represent two popular general aviation airframes, and the General Atomics Predator represents a highly capable RPA. The hybrid-electric system has the potential to be applied to any aircraft that uses a single engine as the primary propulsion system. This research determined the optimal components necessary to replace existing propulsion systems. Though many aircraft may be able to use a hybrid system, these studies demonstrate how the weight distribution of mild hybrid electric systems changed. The simple retrofit of existing GA platforms would be the easiest solution for now. These three case studies were meant to validate the performance of the code so that new aircraft could be designed using the same code. The following sections outline how each case study was approached and executed.

3.1. RPA Design Considerations

The historical data available for RPAs was not readily available and was spread across many resources. To apply the same conceptual design code to RPAs several issues needed to be addressed. The design of GA aircraft had the benefit of historical data and consistent trends making the design process simple [45]. The most notable differences between GA and RPA designs were the dramatic differences between the weight fraction distributions as seen in Figure 11. The weight fractions were calculated by taking the component mass and dividing by the GTOW. The glider weight fraction, or structural weight fraction, was taken by subtracting the engine mass from the empty mass and dividing by the GTOW. All other calculations were straight forward and represented the component weight fractions. The Heron RPA was chosen for this illustration instead of the Predator because the Heron had an equivalent GTOW to the Cessna 172, making

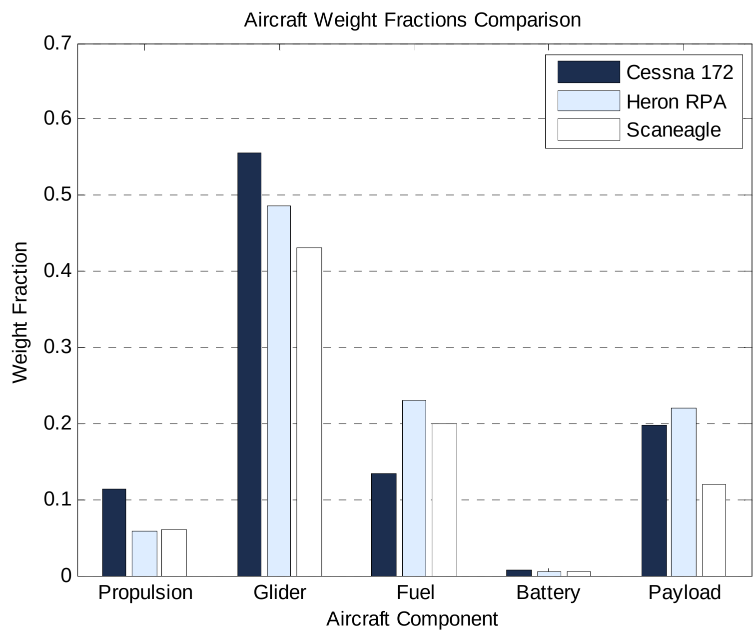

Figure 11: GA vs. RPA Weight Fraction Comparison

comparison easy. Much more weight was distributed to fuel and less to structure for the Heron RPA. This was because RPAs carry no humans and environmental control systems for humans were not needed. A greater difference was observed for the smaller Scaneagle RPA in Figure 11. It was difficult to readily apply the same conceptual design to the RPA design. A more in depth look across the spectrum of RPA sizes suggested that the only plausible application for the mild hybrid-electric design code was on large RPAs. This conclusion was made after several trials using small RPAs were performed. The code was unable to converge on a practical solution for the mild-hybrid design to use the same airframe. The decision was made to perform a case study for the General Atomics Predator because of it size and potential multi-role capabilities.

4. Inputs

For the conceptual design of the case study aircraft, many of the variables were made constant and several were manipulated respective to each platform. The constant variables were parameters that were independent of the aircraft variance. Many of these values must be assumed since many were not well defined for hybrid-electric aircraft. Table 1 lists each of these variables along with their MATLAB variable name followed by the corresponding value and unit. Gravity was assumed constant regardless of the

Table 1: Constant Parameters

Description	MATLAB Variable	Value	units
Gravity	g	9.81	m/s^2
Sea Level Density	p_sl	1.22	kg/m^3
Propeller Efficiency	n_prop	80	%
Mechanical Efficiency	n_mech	97	%
Battery Specific Energy	Batt_SpecEGY	150	Wh/kg
Motor Specific Power	Motor_SpecPWR	2250	W/kg

operating altitude or location of operation and was a safe assumption since the relation between location and altitude on the force of gravity was negligible. Sea level density was selected based on standard day conditions. Propeller efficiency was difficult to estimate since it can be dependent on altitude, propeller speed, and torque. However a reasonable value of 80% was given since the aircraft would be at the cruise condition for a majority of the design mission profile, and the propeller can be optimized for that condition. Mechanical efficiency depends greatly on the mechanical configuration selected for the hybrid system. A conservative value of 97% was selected based on the findings in Raymer's text for single engine aircraft with mechanical efficiency near 99% [17]. The present day specific energy for lithium ion batteries peaked at 150 Wh/kg. For the conceptual design the maximum was selected since many large lithium ion battery applications have been demonstrated exhibiting the highest specific energy [51][39] [40]. Finally, the specific power for motors was estimated based on the advanced motors designed by Yuneec aviation that averaged a specific power of 2250 W/kg [8]. These variables remained constant for each case study and were used to help measure performance and size the propulsion system.

The next group of constant variables helped determine the initial weight estimation of each aircraft. Raymer tabulates these values in his book for the general aviation, powered glider, and homebuilt composite cases using best fit lines of historical data [17]. Raymer did not include any fit line or historical data to evaluate present day RPAs. Using the same method as Raymer historical data was found for several RPAs representing two groups. Group 1 was RPAs with a mass less than 70 kg, Group 2 RPAs had a mass greater than 70 kg. The resulting fit lines and the RPA historical data can be found in Appendix C accompanying this document and all aircraft types were summarized in Table 2 for the empty weight fraction estimation.

Table 2: Variables Needed for Empty Weight Fraction Calculations

Empty Weight Fraction vs. W_0 ($W_e/W_0 = AW_0^C$)			
Aircraft Type	**Drag Polar (Cd_0)**	**A**	**C**
General Aviation Single-Engine	0.03	2.36	-0.18
Powered Glider	0.02	0.91	-0.05
Homebuilt Composite	0.018	1.15	-0.09
RPA Group 1	0.035	0.6209	-0.0161
RPA Group 2	0.035	0.5728	-0.0015

The final variables that were needed for the conceptual design were the independent variables that represented the desired performance characteristics. Remembering that the conceptual design included a simplified aerodynamic model with rectangular wings and traditional wing body tail configuration, the only aerodynamic variables included were the max lift coefficient and Oswald efficiency. Other variables for takeoff and climb performance included stall speed, desired takeoff distance, and desired rate of climb. The desired payload mass and range (fuel mass) helped determine the overall mass of the aircraft. Cruise speed, operational altitude, and maximum wingspan determined how large the mild hybrid's engine must be to maintain steady level flight at the desired altitude. Finally, max wing loading and the maneuver speed determine the load factor during a sustained turn that indirectly limits the aspect ratio that was discussed in Chapter III section 3.1. Table 3 records the MATLAB variables used for each described parameter.

Table 3: Design Requirement Inputs

Description	MATLAB Variable
Oswald Efficiency	e
Max Lift Coefficient	Cl_max
Stall Speed	V_Stall
Density @ Altitude	p_alt
Cruise Speed	V_cruise
Manuever Speed	Va
Desired Rate of Climb	des_ROC
Desired Takeoff Distance	des_TO_dis
Desired Range	RangeDes
Payload	Payload
Maximum Wingspan	max_b
Maximum Wingloading	max_Wingload
Specific Fuel Consumption	C

The variables for the DA 20 and Cessna 172 were found using the relevant flight manual used by pilots [52][53], and the values found for the Predator were found using Jane’s [54]. From Raymer’s text specific fuel consumption (SFC) for general aviation aircraft was estimated to be on average 0.4 lb/hr/bhp [17]. Taking advantage of a revolutionary DeltaHawk turbo-charged diesel engine the DA-20, Skyhawk, and Predator could benefit from significantly improved SFC near 0.35 lb/hr/bhp [55]. These numbers were optimistic and not yet achieved for the DeltaHawk engines. Still it was assumed that the SFC performance was improved since the engine would be optimized for the cruise condition. An enhanced SFC of 0.37 lb/hr/bhp was used for each aircraft. Once the performance values were determined for each aircraft the MATLAB code was utilized to size the hybrid-electric propulsion system.

4.1. Performance Evaluation

The advantage of the hybrid system was measured by how much performance was gained or sacrificed compared to the original platform. To maintain the same performance, an increase in the GTOW was expected because of the battery mass. If the same GTOW could be achieved by manipulating the mission requirements, some aircraft could still benefit from the fuel savings of the hybrid technology. Therefore, two different sets of variable inputs were used, a matched case and adjusted case, to evaluate each platform. For each case study, the first variant maintained the same initial aircraft performance;

4.2.1 Matched Performance

For the mild-hybrid matched performance variant the variable inputs were selected based on the original commercial specifications for each case study aircraft.

These variables were determined from available flight manuals. They include all variables outlined in Table 3. After running the code a few times for each aircraft a few interesting facts appeared. If no performance was sacrificed the engine, optimized for altitude cruise, was small causing a large demand from the motor for takeoff and climb. A large motor meant a larger battery and heavier overall aircraft. This caused increased wing area for the hybrid designs making it impossible to achieve matched performance.

4.2.2 Adjusted Performance

The second variant manipulated the performance characteristics to yield a match to the original aircraft's weight and airframe so that a retrofit was possible. To maintain the same aircraft weight and physical size, performance was altered for the mild-hybrid adjusted design. The easiest way to reduce the overall weight of the aircraft was to reduce the electrical energy storage in the batteries. This could be achieved by making the motor smaller. If the takeoff distance requirement was increased or the rate of climb reduced the additional motor power necessary was decreased. The smaller motor meant that a lighter battery was required. Range could also be sacrificed to reduce the GTOW by using less fuel. The two variants were then compared to the original platform. The relation between the performance results will be given in more detail with each individual case study.

Plugging in the appropriate variables and performance characteristics three designs were compared, the original, a mild hybrid matched performance, and a mild hybrid adjusted performance. The code used two design variables to converge on an optimized cruise power requirement. The first design variable was the wingspan. For each case study the wingspan was set equal to the original airframes span. The second variable, wing area, was allowed to vary from 1m^2 to 50 m^2 so that the cruise power was minimized at the cruise condition. The code attempted to drive the wing loading down and the aspect ratio to the highest possible value. If the wingspan were allowed to vary unreasonably large wingspans resulted from the high aspect ratio and low wing loading. By keeping the wingspan constant the wingspan varied and optimization was constrained using Equation 17 and 18. If the resulting wing area was smaller or matched the original airframes, a retrofit could be possible. Any increase in the wing area meant that new wing would need to be designed. After verifying the code any future designs using this code should allow the wingspan to change. The goal of this research was to demonstrate for each case study that a hybrid propulsion system could be retrofitted into an existing airframe to validate the conceptual design code.

5. DA 20

The DA 20 is a two place aircraft developed by the Diamond aircraft company for general aviation enthusiasts and has become a highly capable training aircraft. The main reason this aircraft was selected was because the United States Air Force used the DA 20 as their primary flight training platform. The use of a hybrid-electric system on this platform has the potential to reduce the cost of training USAF pilots and lower the need for AVGAS and 100LL fuels. AVGAS and 100LL fuels have been subjected to EPA regulations and the usage of such fuels needed to be phased out. The hybrid-electric system's performance was measured to account for how much fuel was saved. The fuel savings must be weighed against the sacrificed performance that was necessary to be able to retro fit the existing DA 20 airframe. The following evaluates the conceptual design and performance of the mild hybrid DA 20.

5.1. Mild Hybrid Applied to Original DA 20 Matched Performance

The current DA 20 propulsion system allows the aircraft to have desirable characteristics for training purposes. The original configuration used a Continental IO 240, 4 cylinder, 4 stroke engine that can produce 93.75 kW at 2800 RPM. The physical wingspan was 10.9 m and wing area was 11.6 m^2 [52]. The maximum gross takeoff weight (GTOW) was 800 kg and carried 65 kg of fuel. Fuel mass and engine power can be reduced by implementing a hybrid propulsion system. To be able to retrofit the existing airframe the GTOW of 800 kg cannot be exceeded. The most desirable product was to have matching performance compared to the original. The appropriate variables were adjusted in the MATLAB code and the matched mild hybrid DA20 results follow.

The performance of the original aircraft and hybrid with matched requirements can be found in Table 4. At first glance the two aircraft seem similar. Many of the same

Table 4: Matched Hybrid DA 20 Performance

	unit	Original DA 20	Hybrid DA 20 Matched
Engine	kW	93.75	87.75
Wingspan	m	10.9	10.9
Wing Area	m^2	11.61	12.35
Max TO Weight	kg	800	873
Payload	kg	220	220
SFC	lb/hr/bhp	.4	.37
Fuel Mass	kg	65	61
TO Distance GR	m	400	390
ROC	m/min	304.8	300
Operating Altitude	m	4000	4000
Range	nm	547	547
Stall Speed	m/s	24	24
Cruise Speed	m/s	71	71
Empty Weight	kg	529	584
Wing Loading	kg/m^2	69	70
Manuever Speed	(m/s)	54	54
Motor Power	kW	NA	21.75
Battery Mass	kg	NA	57

flying performance requirements were met by the hybrid DA 20, but the increase in GTOW from 800 kg to 873 kg caused a proportional increase to the physical dimensions. The wing area for the hybrid design was 12.35 m^2, 6% higher than the original meaning that the airframe would need a new wing designed. This was unacceptable for a retrofit design. However, more results were needed to evaluate the rest of the mild hybrid conceptual design.

Using the weight fractions of the energy storage and power delivery allowed for a relative comparison between the original and hybrid configurations even if the GTOW varied. The variation in the weight fractions give little information about the individual component weights but can give insight for the effectiveness of the conceptual design.

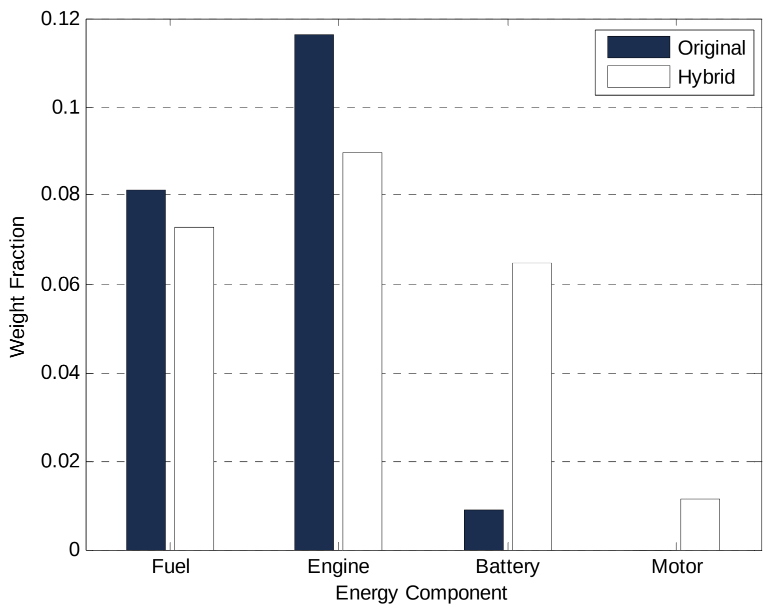

Figure 12: Energy Component Weight Fraction DA 20 Matched Performance

The weight fractions were taken with respect to each configuration's GTOW. For the hybrid design the new GTOW was 873 kg an increase of 10% from the original. Figure

12 plotted the original versus the hybrid to compare relative energy weight fractions for fuel, engine, battery, and motor.

The benefit of the hybrid system was that the weight fractions for the fuel and engine were reduced. However, by only looking at the weight fractions in Figure 12 the reduced mass could not be concluded since the GTOW was different for these two designs. Other outputs of the code needed to be observed. After further investigation, the fuel mass was only reduced 4 kg. The increased GTOW made the engine power required equal the original engine making the matched case an unlikely candidate for the mild hybrid propulsion system. The significant increase in the battery weight fraction in Figure 12 was expected because of the limited specific energy capability of batteries to drive the electric motor. The motor weight fraction was a non factor before since no motor was present on the original but the new component weight fraction can now be monitored.

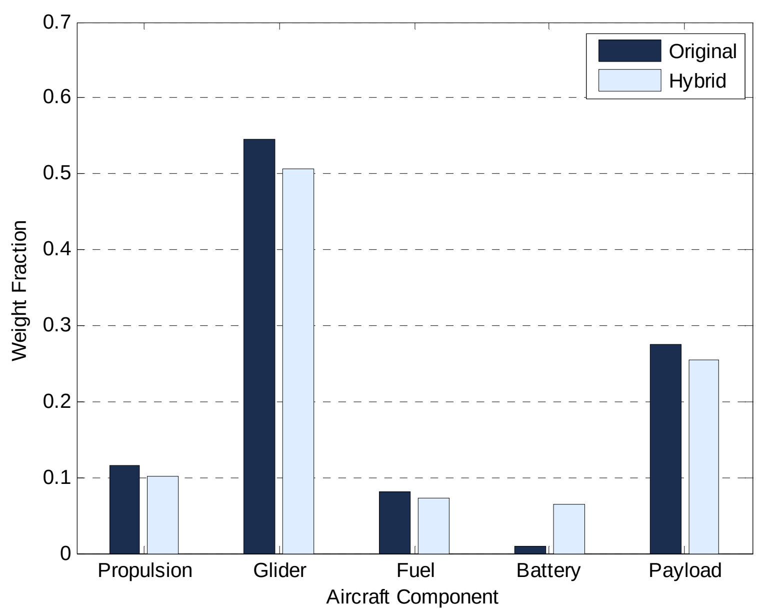

Figure 13: Aircraft Weight Fraction DA 20 Matched Performance

Other aircraft component weight fractions were observed to compare the two aircraft. Remembering that the difference in GTOW makes it difficult to judge each component's mass directly. Figure 13 recorded the calculated aircraft weight fractions for the overall propulsion mass, glider mass, a repeat of the energy component, and the payload. All five components together represent 100% of the GTOW. Comparing the distribution of weight helped evaluate the hybrid conceptual design against the traditional GA aircraft. The propulsion mass included the engine and motor mass for the design. The engine mass reduction was greater than the motor mass required for the hybrid so the weight fraction was reduced. This meant that a smaller percentage of the GTOW was allotted to the propulsion system. The glider weight fraction represented the structural weight of the aircraft stripped of all propulsion and payload components. No structural weight was added to the airframe but the additional GTOW caused the decrease of the glider weight fraction. Only the battery weight fraction increased since large energy storage was required to power the motor. Payload mass remained the same but the weight fraction was reduced because of the GTOW increase. The battery was identified as the greatest driving force for the hybrid system's weight fraction distribution, which was expected.

The flying performance power constraints were used to help size the components that affect the weight fractions seen above. The power required to maintain steady level flight was important in order to determine the smallest possible engine that could be used for propulsion at altitude. Figure 14 depicted the power required curve for the hybrid DA 20 configuration. To satisfy the cruise performance at altitude the power required from the engine can be found be moving along the altitude curve until the cruise speed of

71m/s was lined up. This value was around 60 kW but this calculation only included the inefficiency of the prop and not the altitude affect that was noted in Chapter III. Equation 15 estimated the necessary engine power adjustment required using the density effect. The power needed was 87.75 kW which was not much less than the Continental IO 240. The reduced power lead to the reduced fuel needed.

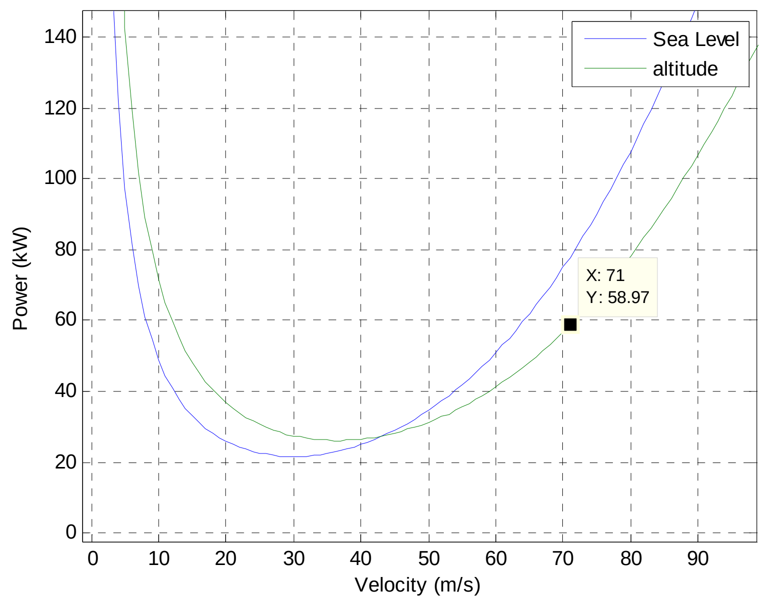

Figure 14: Hybrid Power Required Curve DA 20 Matched Performance

The next performance criteria were the rate of climb and sustained turn rate. Once the engine power was determined the additional power required to meet the climb requirement determined the motor power necessary for the aircraft. The relationship between the rate of climb and altitude impacted the battery energy storage. For the DA 20 the original requirements were ROC of 5 m/s to 4100 m. Figure 15 provided the spectrum of climb rates for the hybrid configuration. The best rate of climb was at the speed associated with the minimum power required. The excess power provided the

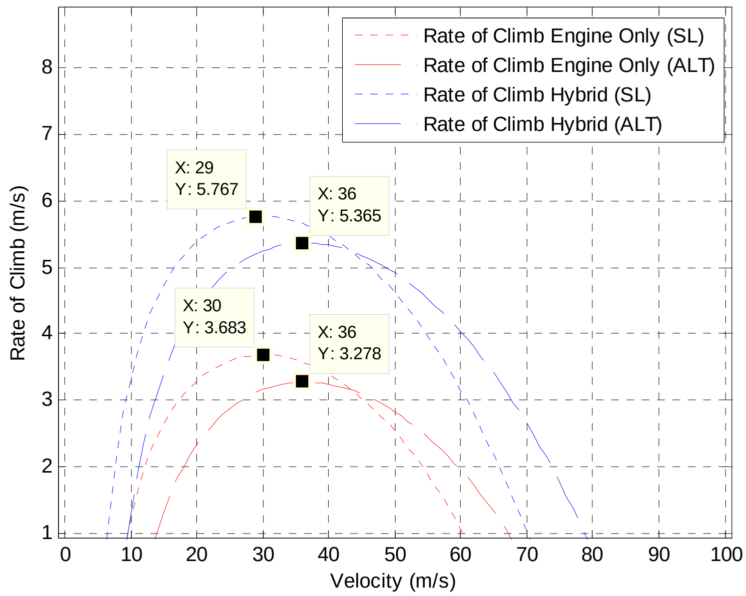

Figure 15: Hybrid Rate of Climb DA 20 Matched Performance

climbing ability. At altitude the climb rate was significantly reduced, however this effect would not be as severe when using a motor since no power would be lost at altitude like the ICE. The sustained turn rate was a secondary performance criterion that was observed more as a constraint rather than a performance requirement. By limiting the g-load factor to a comfortable value of 2 (n=2) at the maneuver speed, the aspect ratio of the airframe was limited. A greater aspect ratio allows larger turn rates and greater g-loads. Figure 16 graphed the maximum sustained turn rate for the hybrid's physical configuration. The intersection of the maximum sustained turn line and the load factor of 2 (n=2) at the maneuver speed would represent the aspect ratio limit. For the DA 20 the intersection was at 56 m/s and V_a was 54 m/s meaning the sustained load factor constraint was not active for the matched performance DA 20 hybrid.

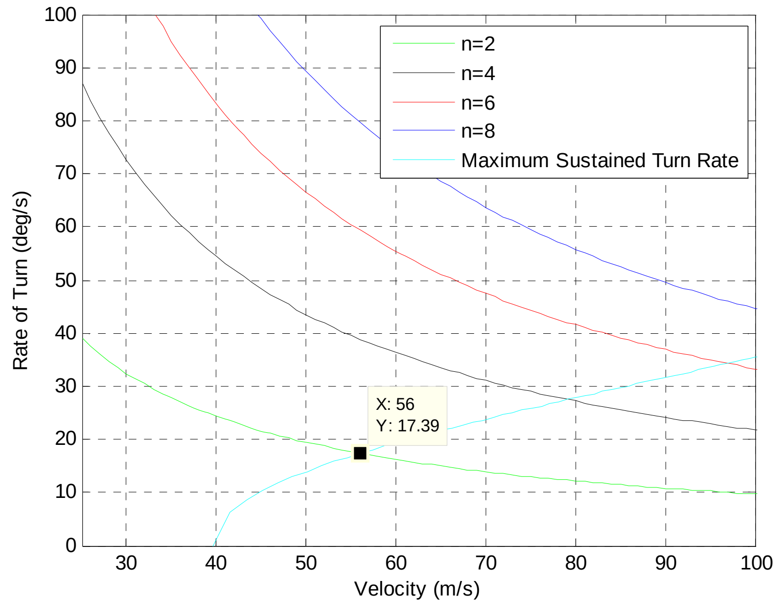

Figure 16: Maximum Sustained Turn Rate DA 20 Matched Performance

The matched performance of the mild hybrid DA 20 caused the undesirable weight increase that can be alleviated by adjusting the performance requirements. The increased weight may cause a redesign of the DA 20 airframe which would cost more money and was not the purpose of this evaluation. Some performance loss was expected when the mild hybrid was proposed to be retrofitted into existing airframes. Where these losses come from were scrutinized based on the importance of the mission and may be different for other aircraft. The variable inputs found in Table 3 that were set to the original configuration can now be manipulated to find the adjusted mild hybrid design to allow a direct replacement in the DA 20 airframe.

5.2. DA 20 Mild Hybrid Adjusted Performance

To avoid a redesign of the DA 20 airframe several performance requirements were reduced. The Diamond Aircraft Company built the DA 20 to be a capable general aviation aircraft for leisure and has become a valuable training aircraft for USAF. The success of the airframe as a training platform made it appealing for this research. To make the hybrid suitable for the DA 20 little performance was surrendered, but payload and range (fuel mass) were reduced. To help adjust the overall takeoff weight of the aircraft range was given up to reduce the fuel mass carried on board. Finally, the baggage allowance was removed from payload since the hybrid DA 20's primary role would be as a trainer and has little need for baggage. Table 5 summarized the adjustments made for

Table 5: Performance Comparison for Diamond DA 20

	unit	Original DA 20	Hybrid DA 20 Matched	Hybrid DA 20 Adjusted
Engine	kW	93.75	87.75	80.25
Wingspan	m	10.9	10.9	10.9
Wing Area	m^2	11.61	12.5	11.35
Max TO Mass	kg	800	882	800
Payload	kg	220	220	181
SFC	lb/hr/bhp	.37	.37	.37
Fuel Mass	kg	65	61	40
TO Distance GR	m	400	390	390
ROC	m/min	300	300	313
Operating Altitude	m	4000	4000	4000
Range	nm	547	547	400
Stall Speed	m/s	224	24	24
Cruise Speed	m/s	71	71	71
Glider Mass	kg	430	430	430
Wing Loading	kg/m^2	69	70	70
Maneuver Speed	(m/s)	54	54	54
Motor Power	kW	NA	21.75	21
Battery Mass	kg	NA	57	53

the DA 20 and compared the outcome against the original design and matched performance hybrid design. The wing area calculation of 11.35 m^2 was better for the adjusted mild hybrid design because it was less than the original DA 20 configuration and would not require a new wing design. Payload mass was reduced from 220 kg to 181 kg, an 18 % reduction and the range was reduced from 547 to 400, a 26% decrease. Both adjustments were made because they had little impact on the flying qualities of the aircraft, except to reduce GTOW.

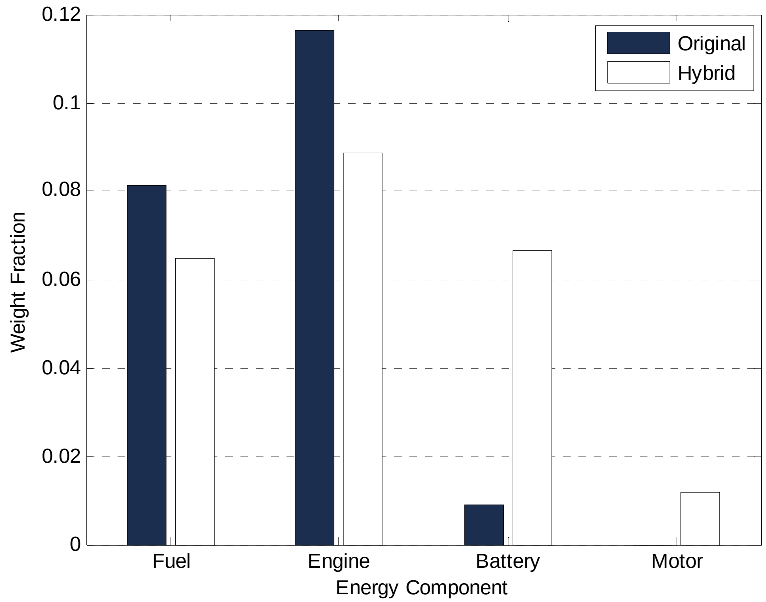

Figure 17: Energy Component Weight Fractions DA 20 Adjusted Performance

The adjusted mild hybrid and original DA 20 had similar GTOW so any weight fractions calculated would represent an equivalent relation to the weight from the original to the hybrid. Figure 17 illustrated the weight savings for the fuel required and the engine mass. A smaller GTOW meant a smaller engine was needed because less power was required at altitude. Fuel mass was greatly reduced because of the smaller range requirement and smaller engine. The battery weight fraction was still a significant increase from the original. Comparing Figures 12 and 17, there were similar battery and motor weight fractions for the matched and adjusted mild hybrid. This was expected since the takeoff and climb requirements were unchanged for both hybrid DA 20 configurations. The augmented power necessary to meet the takeoff and climb performance was proportional to the GTOW and thus the weight fraction for the motor went unchanged.

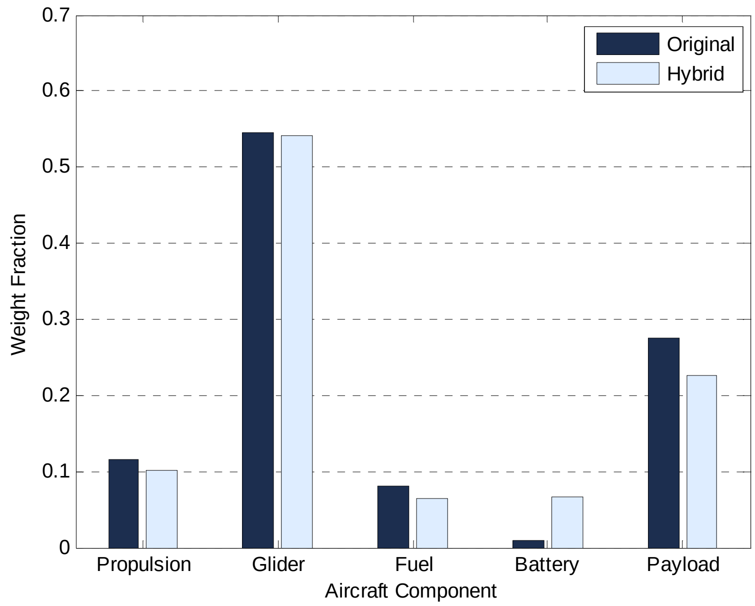

Figure 18: Aircraft Weight Fractions DA 20 Adjusted Performance

The aircraft weight fractions for the adjusted hybrid yielded the same trends that the matched performance hybrid design demonstrated. Figure 18 displayed the aircraft weight fraction for the adjusted case. The overall propulsion mass was slightly reduced for the adjusted hybrid which meant a weight fraction drop. The glider weight was equivalent for the original and the adjusted hybrid design. Referring back to Table 5 the fuel mass for the adjusted hybrid was 40 kg, 35% less than the 61 kg for the matched performance hybrid and 38% less than the 65 kg required for the original. The fuel weight fraction for the adjusted hybrid reflected these relationships. Finally, payload mass was reduced 18%, which translated to the proportional weight fraction reduction from the original seen in Figure 18. The battery weight fraction still sustained a large increase. Weight given to the batteries was taken from other components of the aircraft such as payload negatively impacting the design. Improved specific energy would alleviate this problem and reduce battery weight in exchange for payload.

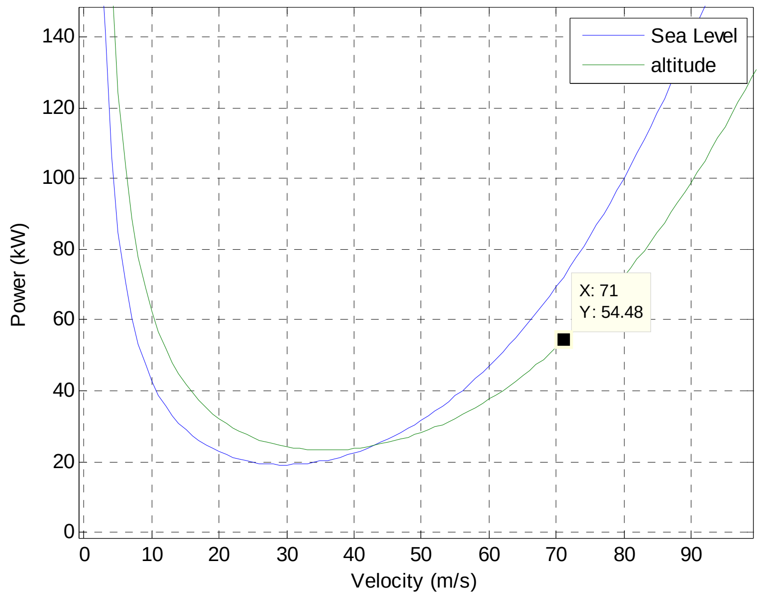

Figure 19: Hybrid Power Required Curve DA 20 Adjusted Performance

There was little change in the overall performance for the adjusted hybrid DA 20. The slight differences from the matched performance hybrid DA 20 design can be attributed to the reduced GTOW. All other parameters such as altitude, cruise speed, rate of climb requirement, and maneuver speed were consistent between the two hybrid designs. Figure 19 depicted the adjusted hybrid's power required. Figure 20 revealed an increased rate of climb compared to the matched hybrid's rate of climb. The relationship between power and weight evident in Equation 22 caused the increase in climb rate. Finally, the sustained turn rate results shown in Figure 21 were altered by the change in wing area for the adjusted hybrid design. The smaller wing area meant a larger aspect ratio pushing the ratio closer to the constraint, shifting the sustained turn rate from Figure 16 to the left in Figure 21.

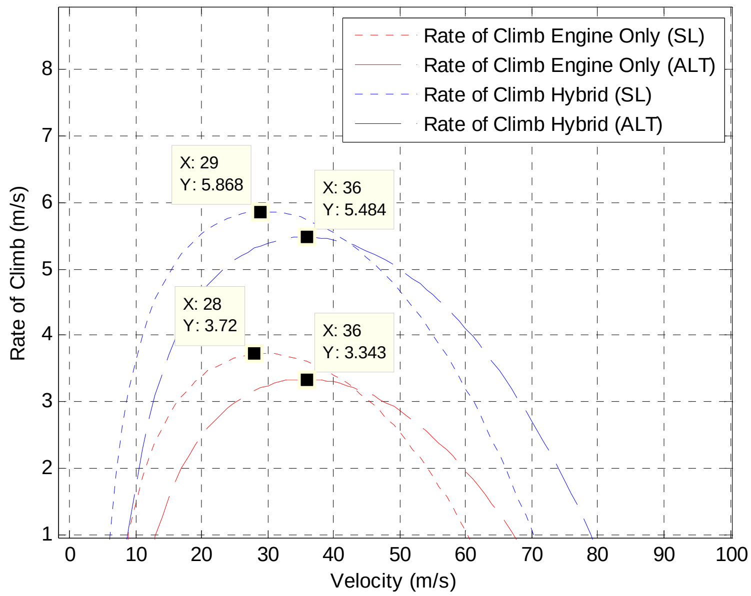

Figure 20: Hybrid Rate of Climb DA 20 Adjusted Performance

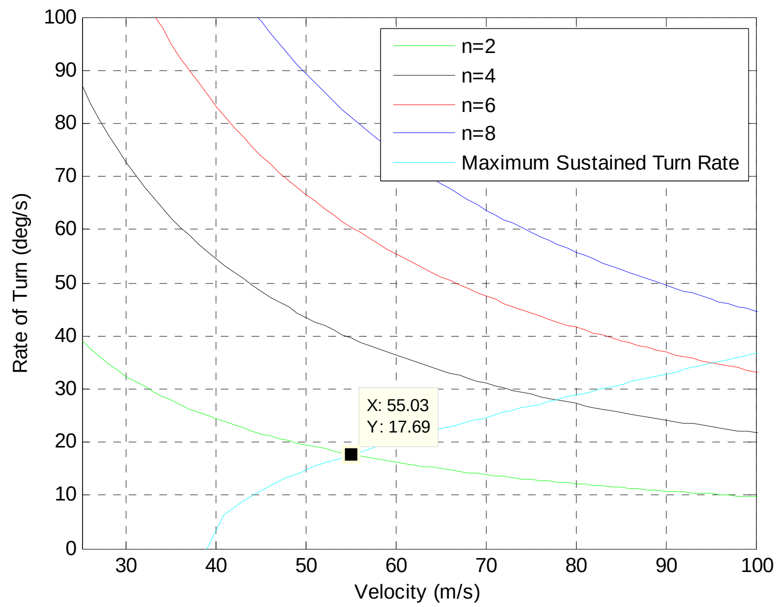

Figure 21: Maximum Sustained Turn Rate DA 20 Adjusted Performance

The DA 20 was determined to be a good candidate for a mild-hybrid electric propulsion system if range and baggage load could be reduced. Fortunately none of the flying qualities needed to be sacrificed to integrate the hybrid system into the DA 20. With similar flying qualities the adjusted hybrid-electric DA 20 saved nearly 25 kg worth of fuel that would be needed from a larger engine oversized for takeoff and climb. Nearly half of the fuel mass savings came from the 100nm of range sacrificed. The other half came from the smaller engine needed for takeoff and climb that possessed improved SFC at altitude. Unfortunately, the conceptual tool could only estimate the fuel savings. Flight testing would need to be done on a retrofitted DA 20 to verify the amount of fuel saved using the mild hybrid-electric technology. Further simulations representing a training mission, with more transient power requirements, should yield greater benefit for the adjusted mild hybrid design.

6. Cessna 172 Skyhawk

For a long time, the Cessna 172 Skyhawk has been one of the superior four place GA aircraft. Introduced in 1956, the Skyhawk has been a nostalgic airframe for GA pilots. To make the modern Skyhawk energy efficient alternative propulsion systems were needed. Recently, Cessna teamed up with Bye energy to create an all electric Skyhawk. They anticipate no significant performance fall off in terms of the flying characteristics of the airplane [9]. Research has suggested that this would be a lofty goal for a first attempt at a large scale all-electric aircraft. A hybrid-electric propulsion system offers a much more practical power plant that exhibits the reliability of internal combustion engines supplemented by efficient electric power. The hybrid system was also meant to be assembled with available commercial materials making it much more affordable than revolutionary battery packs and motors. The following outlines how well the Cessna Skyhawk would perform with the mild hybrid propulsion system.

6.1. Mild Hybrid Applied to Original Cessna 172 Matched Performance

The Cessna 172's propulsion system provided the necessary power required for a payload near 272 kg. The original Skyhawk used a Lycoming IO-360-L2A, 4 cylinder, 4 stroke engine that can produce 120 kW at 2400 RPM. The physical wingspan was 11 m and wing area was 16.2 m^2. The maximum gross takeoff weight (GTOW) was 1114 kg and could carry 144 kg of fuel. Fuel mass and engine size can be reduced by implementing a hybrid propulsion system. To be able to retrofit the existing airframe, the GTOW of 1114 kg cannot be exceeded. The appropriate variables were input into the MATLAB code to make the Cessna 172 a candidate for the mild hybrid propulsion system.

The same evaluation process was given to the Skyhawk as the DA 20. The first evaluation uses the original performance requirements to see how close the physical size of the mild hybrid could come to the original Cessna 172. Table 6 outlined the mild hybrid results for the matched performance. Similar to the DA 20, the GTOW increased from 1114 kg to 1204 kg (8% increase). The wing area was also increased from 16.2 m^2 to 16.28 m^2 a 0.5% increase meaning that the original airframe was close to the physical size needed for the hybrid system. However, increased wing loading (68.8 to 74, a 7.5% increase) made structural integrity a concern. The engine size was significantly reduced to 91.5 kW giving an 18% fuel savings. The fuel savings could be improved more if the performance could be adjusted to reduce the GTOW. The weight fractions were evaluated to see how the weight was distributed.

Table 6: Matched Hybrid 172 Performance

	unit	Original Cessna	Hybrid 172 Matched
Engine	kW	120	91.5
Wingspan	m	11	11
Wing Area	m^2	16.2	16.28
GTOW	kg	1114	1204
Payload	kg	220	220
Fuel Mass	kg	150	123
TO Distance GR	m	514	348
ROC	m/min	220	220
Operating Altitude	m	4115	4115
Range	nm	700	700
Stall Speed	m/s	24	24
Cruise Speed	m/s	60	60
Glider Mass	kg	619	619
Wing Loading	kg/m^2	68.8	74
Manuever Speed	(m/s)	50	50
Motor Power	kW	NA	45
Battery Mass	kg	NA	140

The energy storage for the batteries was again the largest weight fraction increase. The modest fuel and engine mass savings were overshadowed by the battery weight fraction that jumped from just under 1% to over 12% of the GTOW. Figure 22 demonstrated the same trends seen before for the DA 20 energy weight fractions. The total energy storage for the Skyhawk can be found by adding the fuel and battery weight fractions together. The total energy storage for the Skyhawk increased from 10% to over 20% of the GTOW. This leaves little available weight for the payload or structural weight of the aircraft. Eventually adjusting the performance of the aircraft was critical to making the Skyhawk a valid mild-hybrid design.

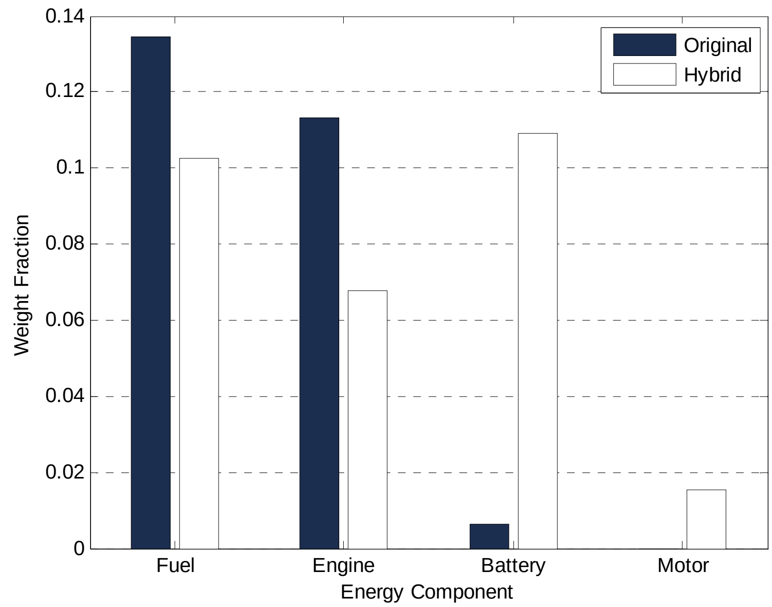

Figure 22: Energy Component Weight Fractions Cessna 172 Matched Performance

Additional weight fractions were observed to evaluate the propulsion, glider, fuel, battery, and payload weight fractions. The same trends from the DA 20 were evident in Figure 23, increase in the overall propulsion system and battery weight fractions, and decrease in the payload weight fraction. The glider fraction was decreased because of the increase of the GTOW. The payload mass remained the same but the increase in GTOW made the weight fraction smaller. Finally, much like the DA 20 matched hybrid, the battery weight fraction significantly impacted the weight distribution for the aircraft. As explained before the energy storage required for the batteries needed to be reduced so that the Skyhawk did not need to be redesigned. This could be accomplished by adjusting the constraints for takeoff and climb.

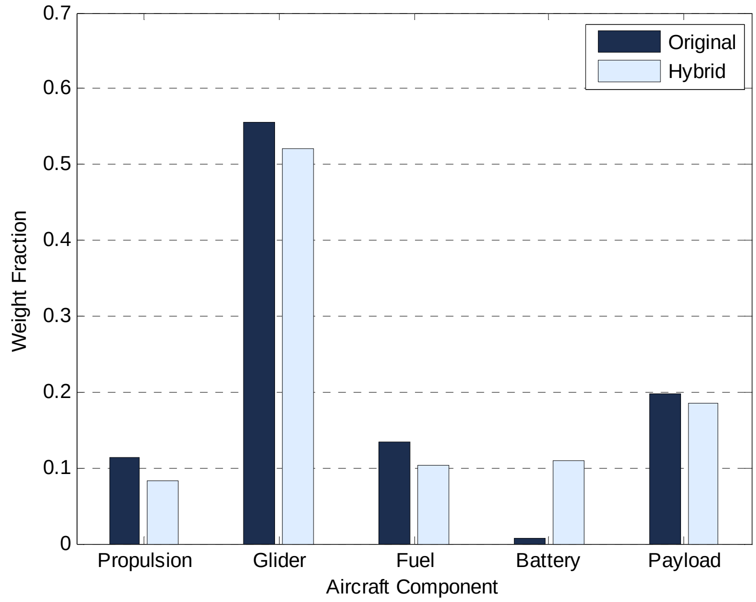

Figure 23: Aircraft Weight Fractions Cessna 172 Matched Performance

The power required at altitude was dramatically changed from 120kW for the original airframe to a reduced 62.2 kW for the hybrid. The altitude effects required that the engine was scaled up to 91.5 kW, but still was smaller than the original 120 kW Skyhawk engine. Figure 24 highlighted the power required curves at sea level and altitude. The design point represented the cruise velocity and the corresponding power required. Once the engine power was established the motor power was calculated to meet the climb performance shown in Figure 25. Observing the climb rate curves, if the motor or engine was to fail little climbing ability was available. The engine alone had climbing ability only near sea level and the smaller motor would only provide enough power for extended glides. This was the intention of the hybrid drive train so that if either power source failed, the other would have adequate power to allow an emergency glide. The redundant power source would make this aircraft a good candidate for certification with the FAA.

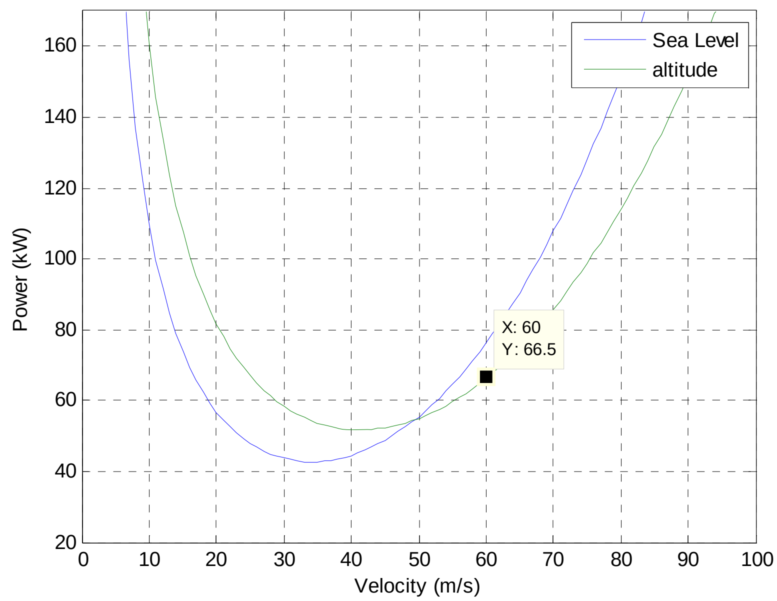

Figure 24: Hybrid Power Required Curve Cessna 172 Matched Performance

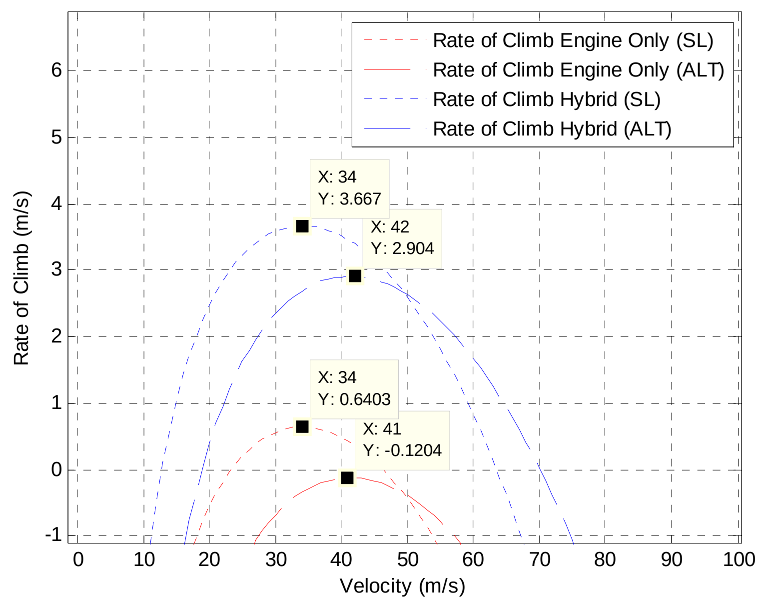

Figure 25: Hybrid Rate of Climb Cessna 172 Matched Performance

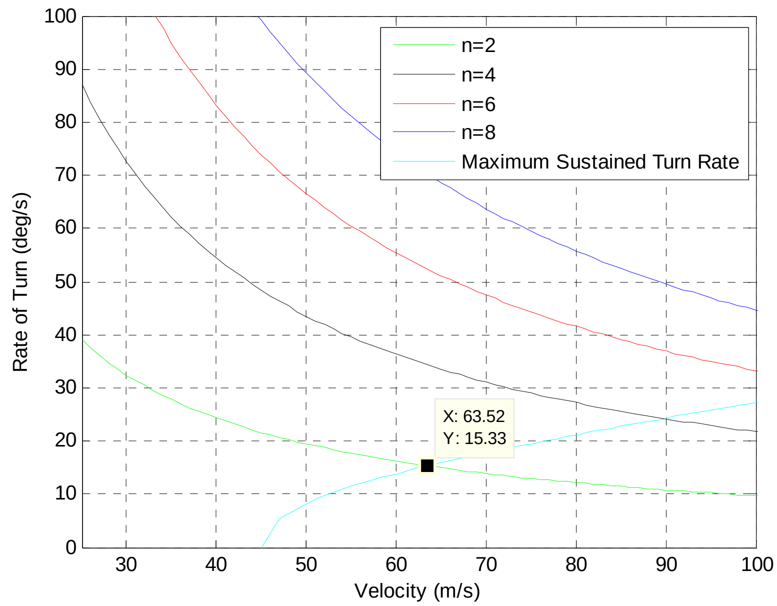

Figure 26: Maximum Sustained Turn Rate Cessna 172 Matched Performance

The physical design of the aircraft was altered because of the increased weight. The wing area was increased to avoid wing stall. The increase in wing area caused a decrease in the aspect ratio, because the wingspan was unchanged. Once the aspect ratio was reduced, the maximum sustained turn rate suffered and the constraint became inactive. For the constraint to be active the maximum sustained turn rate would have to intersect with the g-load curve of 2 (n=2) at the maneuver velocity (V_a = 50 m/s) in Figure 26. Since the wingspan was kept constant for these case studies the aspect only changed as a function of the wing area. Once the code can be validated and the wingspan can vary, higher aspect ratios could be desirable for newly designed hybrid aircraft.

6.2. Cessna 172 Mild Hybrid Adjusted Performance

The adjusted performance for the Cessna 172 sacrificed a small payload amount, range, and rate of climb. Following the same procedure as the DA 20 the adjusted performance for the Skyhawk was recorded in Table 7. The relative improvements inherent with the adjusted hybrid

Table 7: Performance Comparison for Cessna 172

	unit	Original Cessna Skyhawk	Hybrid Skyhawk Matched	Hybrid Skyhawk Adjusted
Engine	kW	120	91.5	82.5
Wingspan	m	11	11	11
Wing Area	m^2	16.2	17.34	15.09
GTOW	kg	1114	1204	1116
Payload	kg	220	220	193
SFC	lb/hr/bhp	.37	.37	.37
Fuel Mass	kg	144	138	95
TO Distance GR	m	514	348	375
ROC	m/min	220	220	200
Operating Altitude	m	4115	4115	4115
Range	nm	700	700	600
Stall Speed	m/s	24	24	24
Cruise Speed	m/s	60	60	60
Glider mass	kg	619	619	619
Wing Loading	kg/m^2	68.8	73	74
Manuever Speed	(m/s)	50	50	50
Motor Power	kW	NA	45	35.25
Battery Mass	kg	NA	140	118

were measured against the original and matched hybrid configurations. The most noticeable improvement was the battery mass. The new GTOW was nearly equivalent to the original aircraft. The wing area for the adjusted requirements was smaller than the original wing area so that no modifications were necessary. Payload was reduced by 27 kg which was the estimated baggage allowance for the aircraft. Only 9% of the rate of climb was sacrificed from 220 m/s to 200 m/s, leading to a decreased motor power requirement, so that a smaller battery could be used. The takeoff ground roll constraint was not adjusted but the value changed because of the new power available and takeoff weight. The energy and aircraft weight fractions proved optimistic for the adjusted hybrid Cessna 172. The weight fractions for the adjusted hybrid were calculated and compared to the original Cessna 172 in Figure 27. The adjusted hybrid weight fractions were also contrasted to the matched hybrid case in Figure 22. Using the data recorded in Table 7

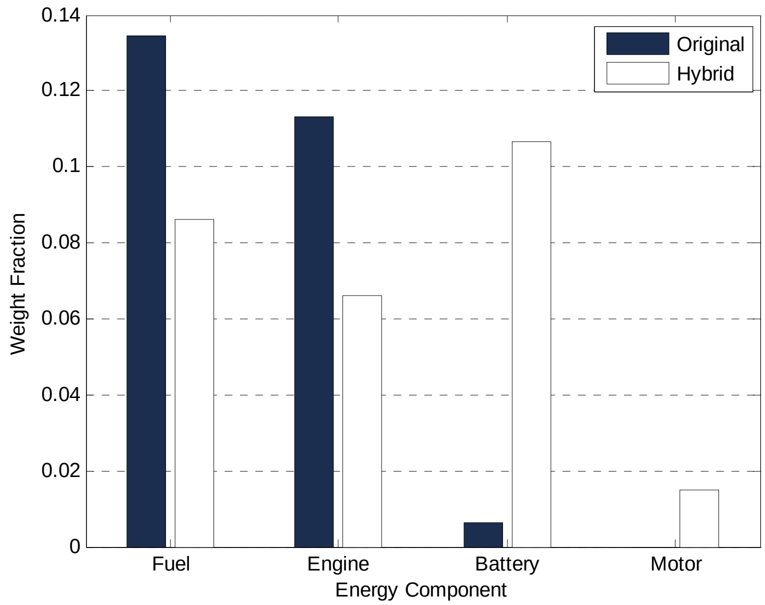

Figure 27: Energy Component Weight Fractions Cessna 172 Adjusted Performance

the comparison between the two mild hybrid designs could be made. The energy storage of the battery was smaller for the adjusted performance so battery mass was reduced 16% from 140 kg to 118 kg. The corresponding GTOW reduction meant that the adjusted mild hybrid design was close to the GTOW of the original platform. So the battery weight fraction did not see a dramatic change between the two hybrid designs, 0.116 for the matched to 0.106 for the adjusted a 9 % change. The fuel weight fraction was also slimmed down from 0.102 to 0.085, a 16% adjustment. This meant that the overall sum for the energy storage was reduced by 12% from 0.219 for the matched case to 0.192 for the adjusted. With less weight fraction allocated to the energy storage and power, proportional weight could be distributed to the payload and structural weight. The more important result was that with similar GTOW the original airframe could be used.

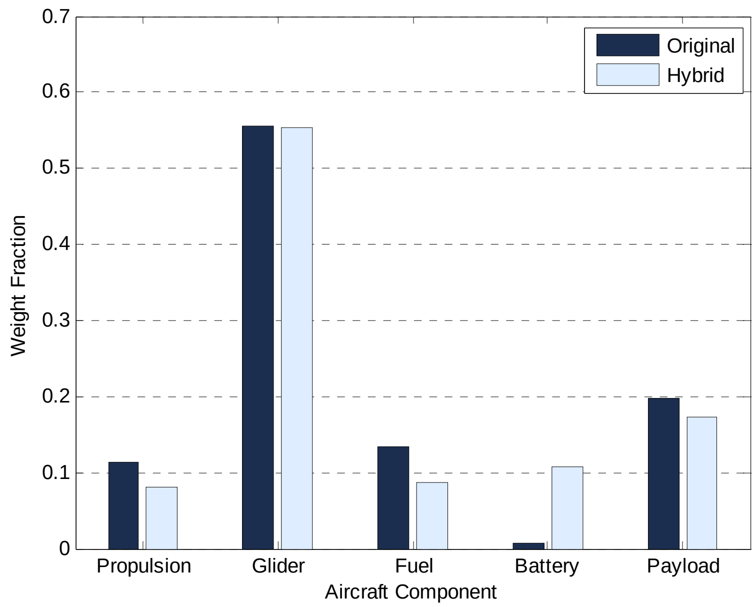

Figure 28: Aircraft Weight Fractions Cessna 172 Adjusted Performnace

The aircraft weight fraction distribution was improved for the adjusted 172 hybrid case. The energy storage improvements discussed previously meant that more weight could be distributed to the structural and payload weight fractions. The similar aircraft GTOW of the original and adjusted hybrid made the comparison easier using Figure 28. The glider weight fraction was maintained so that the original airframe was sufficient. The increased battery weight fraction from the original to the adjusted hybrid was equivalent to the sum of the smaller weight fractions for propulsion, fuel, glider and payload. The negative impact was on payload and performance of the adjusted hybrid 172. The payload impact was clearly indicated in Figure 28, and the adjustment was necessary to allow the improved weight fraction distribution.

The sacrificed performance requirements for the adjusted hybrid shifted the power required, rate of climb, and maximum sustained turn rate curves accordingly. Since the GTOW was cut down, the engine power required to stay in the air dropped to 56.25 kW

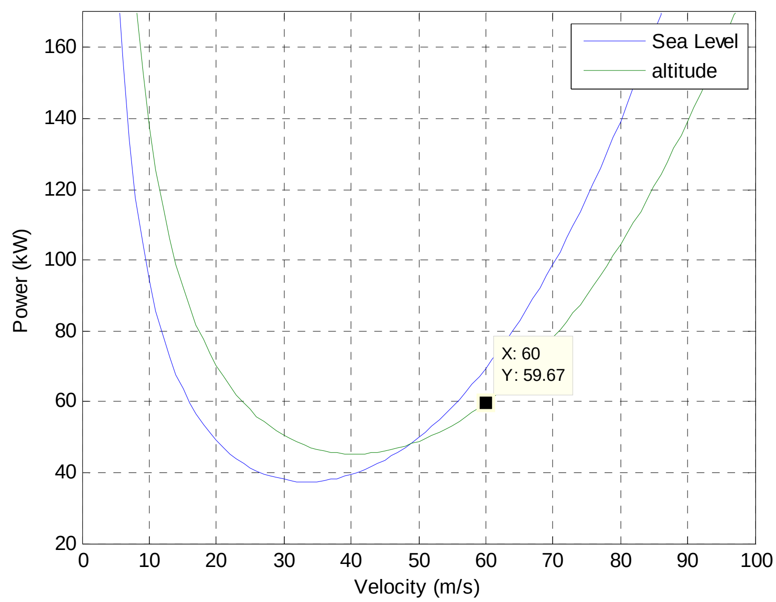

Figure 29: Hybrid Power Required Curve Cessna 172 Adjusted Performance

in Figure 29. The rate of climb was adjusted so that the motor power was lessened so that a smaller battery could be used. The resulted climb rate curves were plotted in Figure 30. Lastly, since the wing area was reduced the aspect ratio was increased. This shifted the maximum sustained turn rate curve in Figure 31 to the left, closer to the constraint boundary.

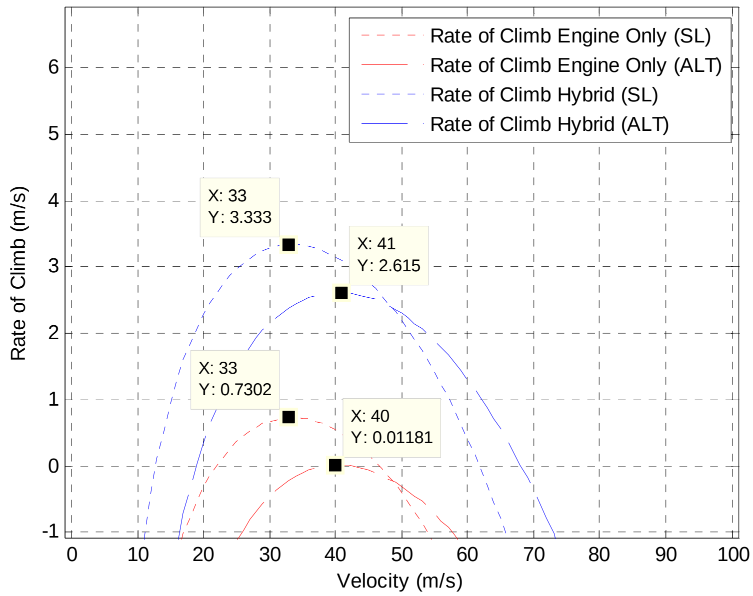

Figure 30: Hybrid Rate of Climb Cessna 172 Adjusted Performance

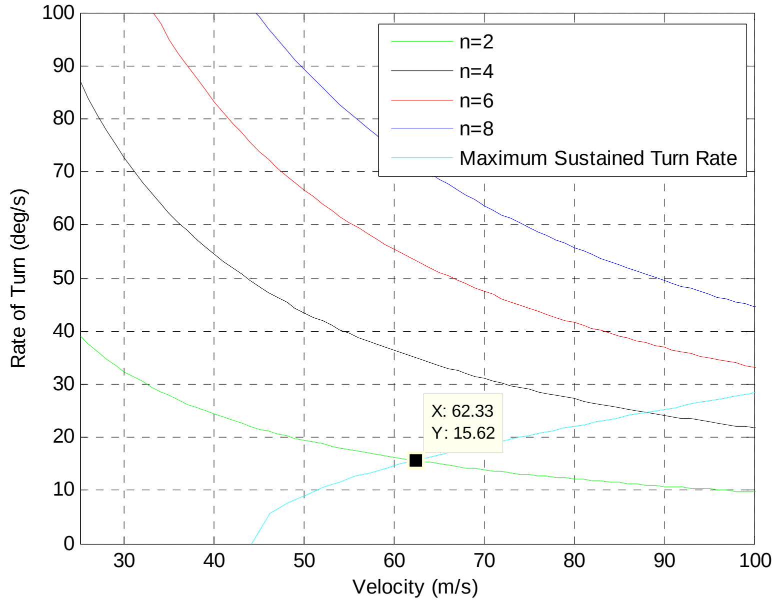

Figure 31: Maximum Sustained Turn Rate Cessna 172 Adjusted Performance

The Cessna 172 Skyhawk proved to be a viable candidate for the mild-hybrid propulsion system. Performance that was sacrificed was minimal to adjust the component weights of the Skyhawk. The potential fuel savings was 54 kg or 75 liters, but sacrificed 100 nm and added 118 kg worth of batteries. Additional simulations could be conducted to determine a more accurate fuel savings and range for a given mild-hybrid configuration. Dynamometer testing of engines for the SFC vs. Power and mission profile analysis would be beneficial to future hybrid research. The mission profile selected for both DA 20 and Cessna 172 case studies was for takeoff, climb, cruise, and land. A typical mission profile for GA pilots in training or flying for pleasure would have more transient conditions including turns and multiple climbs/descents. Simulating these in the conceptual design may provide evidence for enhanced benefits using the mild-hybrid propulsion system for general aviation aircraft.

7. Predator

The General Atomics Predator has been one of the more celebrated RPAs used in the Air Force. Predators have been used during the Wars in Afghanistan and Iraq. The Predator and its sensors provide real time surveillance for commanders on the frontline. Reasoning behind the Predator's selection was the high aspect ratio characteristics to minimize the power required at altitude. The Predator's mission matches the profile used on the two previous hybrid case studies that demonstrated fuel savings. Hopefully, the same fuel savings would benefit the adjusted Predator hybrid. Rrange sacrificed for the Predator would be willingly exchanged for multi-mission capability. If the matched conceptual design could demonstrate some fuel savings, the exchange of fuel, battery, and payload mass would allow multi-mission capabilities for the hybrid Predator RPA. Any mission designed for the hybrid predator should take advantage of the fuel savings during transient flight. An alternative mission would be where the target was far away and a fast ingress and egress was necessary. The loiter time would be much less but depending on the hybridization results the motor could still supply a short term stealth operation on station. For another mission in which a small payload was needed the weight distributed to the payload could be replaced by fuel that would increase the range. The added mission capability would depend on the following.

7.1. Matched Mission Requirements for Predator RPA

Compared to the general aviation case studies the RPA distributed more weight to the fuel and not as much to the glider structural weight. The mission requirements made it difficult for the design code to converge because of the unique mission profile. The same conceptual design method used for the general aviation case studies needed to be adjusted in order to yield a converged solution. The fuel weight fraction was adjusted so that using a realistic SFC, a reasonable fuel mass was calculated. Typically the fuel weight fraction could be used to iterate toward a final GTOW. Since the fuel weight fraction was so large for the Predator the formula was unable to converge. To alleviate this issue the original fuel weight fraction was hard coded into MATLAB and the design code converged on a solution. Another unique characteristic was that the glider weight fraction was much less than the GA cases because no human comforts were needed on the aircraft. The rest of the results for the performance were documented in Table 8.

The hybrid configurations for the Predator RPA demonstrated similar weight fraction trends to the GA case studies. A matched performance hybrid design for the Predator caused a 16% GTOW increase from 1022kg to 1190kg. This made it difficult to evaluate the weight fractions. A decreased weight fraction did not immediately translate to a mass reduction for that component. However, both fuel and engine mass were reduced observing the calculated results in Table 8. To achieve the 2000nm range of the original Predator the matched hybrid only saved 11kg of fuel over the entire mission

Table 8: Matched Predator Performance Comparison

	unit	Original Predator	Hybrid Predator Matched
Engine	kW	86.25	62.25
Wingspan	m	14.84	14.84
Wing Area	m^2	11.5	14.8
Max TO Mass	kg	1022	1190
Payload	kg	207	207
Fuel Mass	kg	300	289
TO Distance GR	m	1524	464
ROC	m/min	220	220
Operating Altitude	m	4500	4500
Range	nm	2000	2000
Stall Speed	m/s	28	28
Cruise Speed	m/s	47	47
Glider Mass	kg	422	422
Wing Loading	kg/m^2	90	96
Manuever Speed	(m/s)	35	35
Motor Power	kW	NA	54.75
Battery Mass	kg	NA	185

but added 185kg worth of battery mass. The fuel savings of typical hybrids were maximized at transient conditions. The Predator RPA's mission was primarily cruise and loitering and does not have many transient mission segments. Since the Predator's mission profile was not as ideal for hybrid propulsion compared to the GA trainers the potential benefits were not as obvious. The 185kg battery pack was the ultimate cause of the increased GTOW since modest fuel and engine mass were saved. The results below in Figure 32 were misleading because of the difference in the GTOW. Observing each energy component weight fraction the total energy storage fraction for the hybrid was significantly increased from the original Predator. The battery specific energy was less than the fuel so a heavier battery was needed for an equivalent amount of fuel. The small benefit was that some fuel was saved for the same mission with the additional GTOW. The large battery mass made the Predator an unlikely candidate for the mild-hybrid design. Still, some advantage could be found in the hybrid design if payload, fuel, and battery masses could be interchangeable for specific missions.

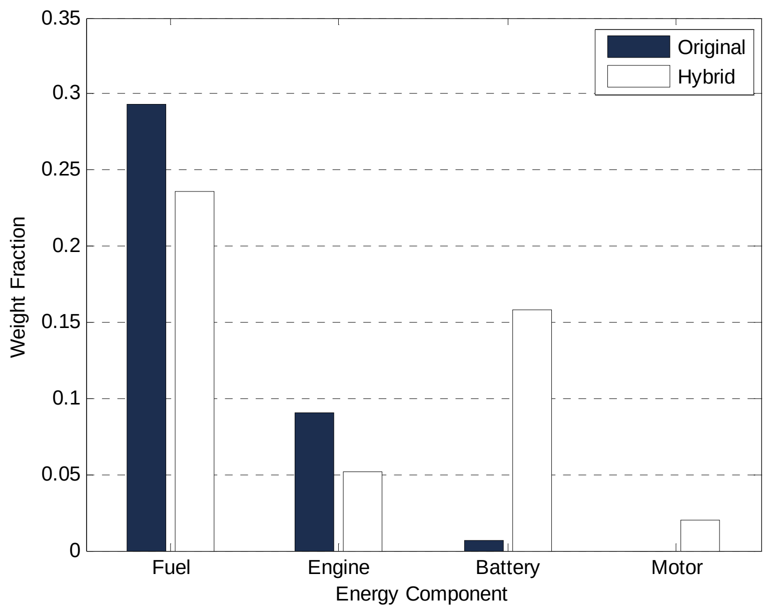

Figure 32: Energy Component Weight Fractions Predator Matched Performance

The weight fraction distribution for the whole aircraft was different than the GA case, but the use of the mild-hybrid design followed similar weight fraction changes. Compared to the Cessna 172 or DA 20, for the initial Predator design, a smaller weight fraction was needed for the glider and larger weight fraction for the fuel. Weight allocated to the propulsion system and payload weight fractions for the Predator were both consistent with the GA aircraft. Again the increased GTOW made it so that a weight fraction comparison between the Predator matched-hybrid and original was not a one to one relationship in Figure 33. If the GTOW for the hybrid was equal to the original and the component masses were unchanged, the weight fraction distribution comparison would not be as dramatic as indicated in Figure 33. For example the glider mass of the original

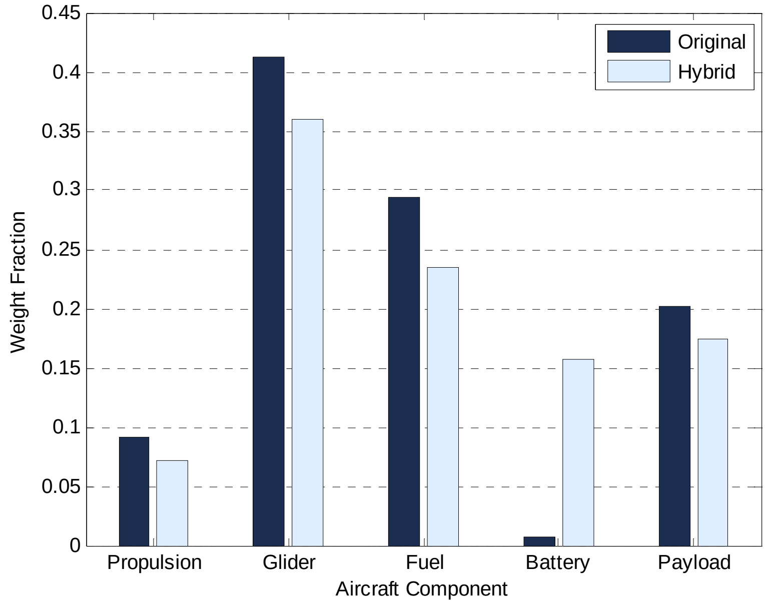

Figure 33: Aircraft Weight Fractions Predator Matched Performance

and the hybrid were identical but because of the large GTOW, the hybrid weight fraction was smaller. The battery mass caused a larger GTOW that exaggerated the effect on other component weight fractions. To make the Predator a viable candidate a similar weight distribution was desired so that airframe modifications could be avoided.

Unexpectedly, the converged solution for the matched hybrid Predator did not achieve the original AR. As discussed in Chapter III most of the equations used to calculate the power required and turn performance were dependent on AR. For the power required the AR was found in the denominator, so if the hybrid Predator's AR could be increased, the power curve in Figure 34 would shift down. Since the power required could be lowered by improved AR the optimized matched hybrid result was not anticipated. Upon further investigation it was realized that the wing loading constraint was active and would not allow a smaller wing area that would result in an improved aspect ratio. To avoid airframe modifications the wing area and AR should have matched the original. Referring back to Table 8, the wing area was increased from 11.5 m^2 to 14.8 m^2, and the calculated AR was 14.9. GTOW and wing loading must be adjusted accordingly to produce the original AR near 20. To increase the wing

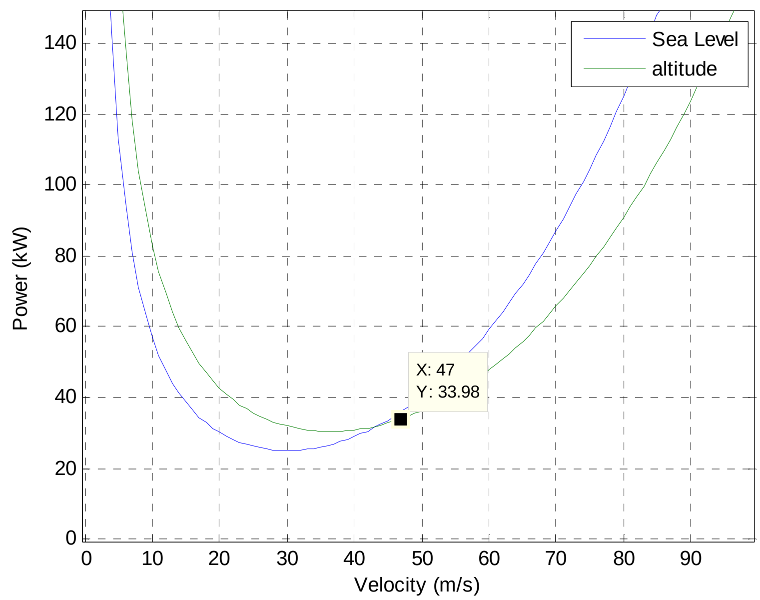

Figure 34: Hybrid Power Required Curve Predator Matched Performance

loading constraint found in Chapter III Equation 16, the designed stall speed could be increased, or a larger $C_{L,max}$ achieved using additional high lift devices. The sustained turn rate AR constraint was not active at the converged solution in Figure 35 but may become active by making the previously described adjustments to GTOW and wing loading. Careful alterations must be made to achieve a more favorable AR.

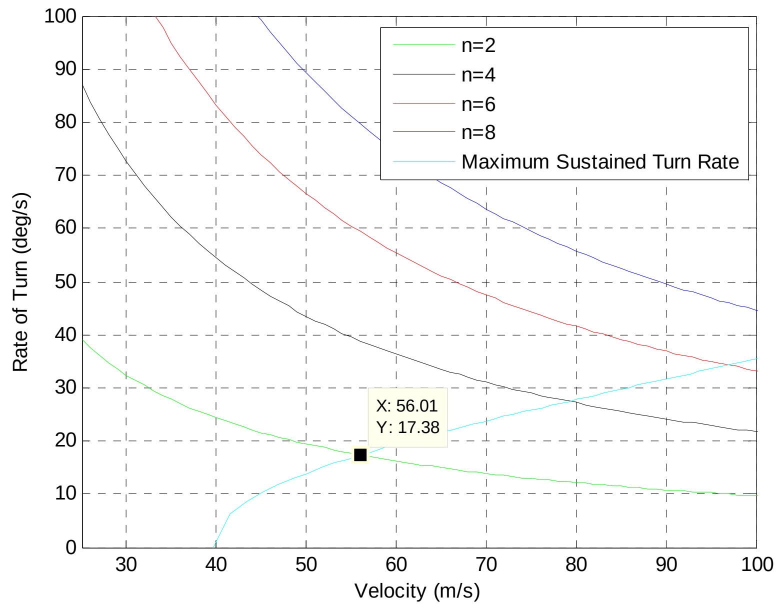

Figure 35: Maximum Sustained Turn Rate Predator Matched Performance

Though the rate of climb was not directly impacted by the AR, the augmented power necessary to provide the climb rates dictated the size of the batteries. If the engine power at cruise could be minimized further, a larger motor would be required to match the climb rate. Having the motor power equal to the engine power approaches the practical hybridization factor limit of 0.5 defined earlier in Chapter III [50]. The rate of climb performance for the engine and the motor augmented power was shown in Figure 36. The ratio between the engine and motor suggest that a full hybrid would be needed for the Predator RPA. The DA 20 and Cessna 172 verified that a mild hybrid design could be used on GA aircraft. The RPA design has a unique weight distribution that was not consistent with the same design strategies used to develop the code. By making some adjustments to the Predator RPA, the outcome may be more favorable.

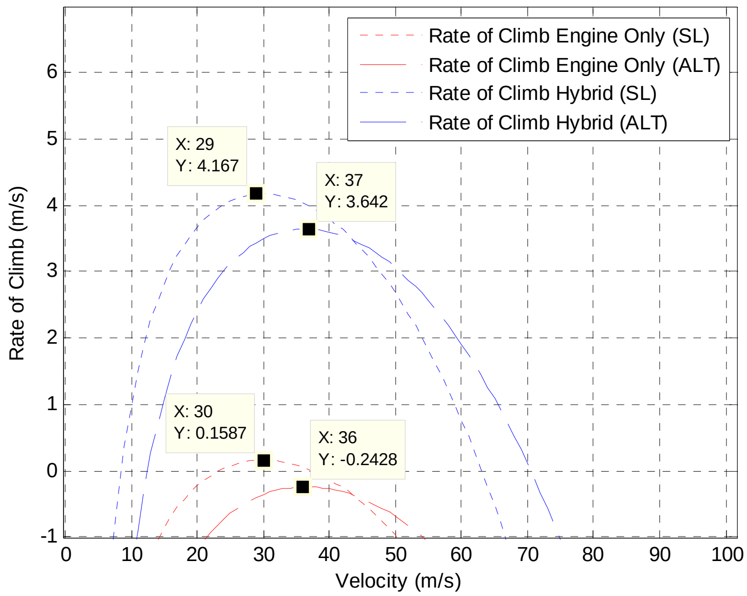

Figure 36: Hybrid Rate of Climb Predator Matched Performance

7.2. Adjusted Mission Requirements for Predator RPA

By manipulating the range, payload, and climb performance the Predator could utilize mild-hybrid propulsion. Unfortunately, the Predator's adjusted performance sacrificed too much to maintain the same mission capabilities. The General Atomics Predator has been used to monitor targets for up to 21 hours [54]. The adjusted mild hybrid would have to sacrifice a large amount of that endurance to allow the batteries to be carried on board instead of fuel. Although the electric energy from the battery was used more efficiently than fuel, more energy could be stored in an equal amount of fuel. The GA aircraft had little trouble because the initial engines were oversized, and baggage allowance could be thrown out of the payload. The Predator had a much larger portion of the weight allocated to fuel. This made it hard to justify trading high specific energy fuel for low specific energy batteries using a hybrid configuration. The only way to match the original Predator's GTOW was to give up 800 nm of range, 14 kg of payload, and 30 m/s of climb rate. The resulting adjusted hybrid configuration for the Predator was given in Table 9.

Table 9: Performance Comparison for General Atomics Predator

	unit	Original Predator	Hybrid Predator Matched	Hybrid Predator Adjusted
Engine	kW	86.25	62.25	49.5
Wingspan	m	14.84	14.84	14.84
Wing Area	m^2	11.5	14.8	11.1
Max TO Mass	kg	1022	1190	1022
Payload	kg	207	207	193
Fuel Mass	kg	300	289	138
TO Distance GR	m	1524	464	604
ROC	m/min	220	220	190
Operating Altitude	m	4500	4500	4500
Range	nm	2000	2000	1200
Stall Speed	m/s	28	28	30
Cruise Speed	m/s	47	47	47
Glider Mass	kg	422	422	422
Wing Loading	kg/m^2	90	96	91
Manuever Speed	(m/s)	40	40	40
Motor Power	kW	NA	54.75	50.25
Battery Mass	kg	NA	185	174

The fuel weight fraction was the only noteworthy change between the matched and adjusted Predator hybrid cases. Having the same GTOW made it easier to compare the adjusted results to the original configuration. The battery mass needed was still too large for enough fuel to be carried on board to support the 2000 nm range. The amount of fuel mass lost due to the 800 nm range sacrificed translated to the similar GTOWs. Observing Figure 37, the battery weight fraction gain was proportional to the fuel and engine weight fraction drops. The energy storage demand for the 50.25 kW motor was overwhelming for the adjusted hybrid design. Adding any arbitrary amount of engine power could reduce the power and energy demand for the takeoff and climb portions of the mission. This would defeat the purpose of the mild-hybrid design because one of the initial problems of aircraft was the oversized engines at cruise. Since there were demonstrated fuel savings for the matched case, the adjusted hybrid should possess the same savings regardless of the range sacrificed. Investigating the Predator further gave insight for the design considerations needed for hybrid RPAs.

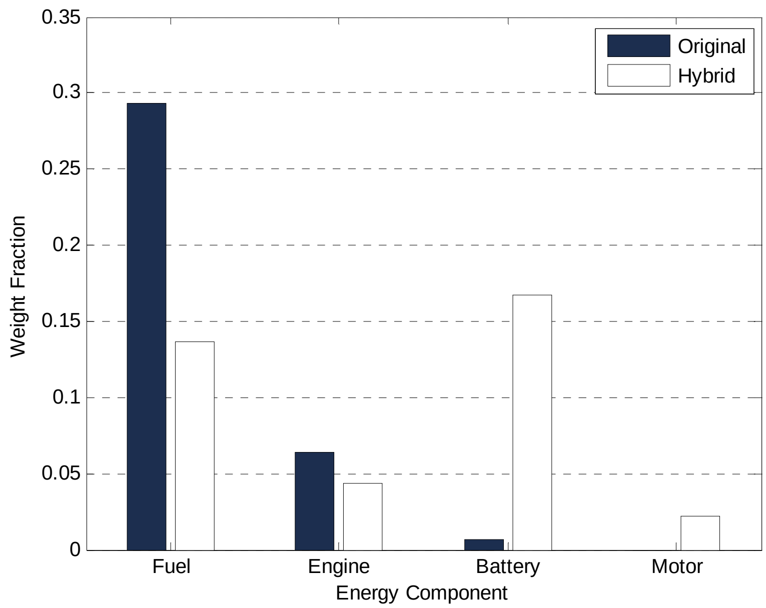

Figure 37: Energy Component Weight Fractions Predator Adjusted Performance

Much like the matched hybrid, the adjusted hybrid predator shifted the weight distribution for the aircraft. Contrary to the GA aircraft case studies, the hybrid weight distribution only changed for the fuel and battery fractions. Both, DA 20 and Cessna 172, studies saw changes for each aircraft weight fraction. The adjusted hybrid comparison in Figure 38 showed a small decrease in payload and traded the form of energy storage from the fuel to the battery. This supported the idea that battery, fuel, and payload could be interchanged for an RPA depending on the desired mission. When interchanging these components, the designer must realize that the ratio between the engine and motor may change. Assuming that the appropriate measures were taken to interchange the components a specific airframe becomes a capable muilti-mission RPA.

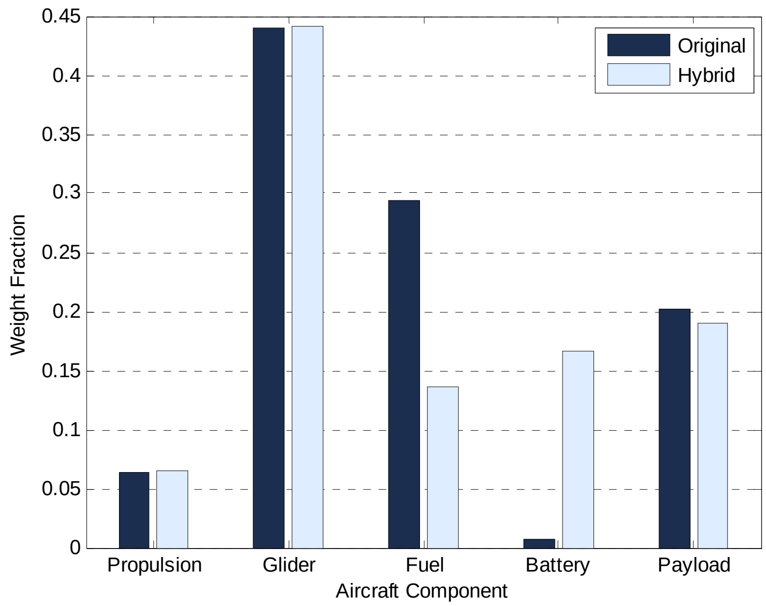

Figure 38: Aircraft Weight Fractions Predator Adjusted Performance

The flying qualities of the adjusted Predator were consistent with the changes made to the performance requirements. Using different mission profiles the following graphs would change based on the desired outcome. Figure 39 illustrated the engine power required at altitude and how close the Predator's cruise speed was to stall. The General Atomics Predator was designed to cruise near stall for the same reason as the U-2 Spy Plane or the Northrop Grumman Global Hawk, long endurance. Having a long endurance makes these aircraft effective ISR platforms, but the slow flying speeds limits the performance of the aircraft. To achieve a reasonable climb rate, the Predator used a 86.25 kW Rotax 914 engine. The hybrid only needed a 49.5 kW to maintain SLUF at altitude. Figure 40 compared the large difference between the 49.5 kW engine's ability to climb, against the augmented power provided by the motor. Having a large AR allows the Predator to fly at low power settings and slow speeds for persistent ISR. The sustained

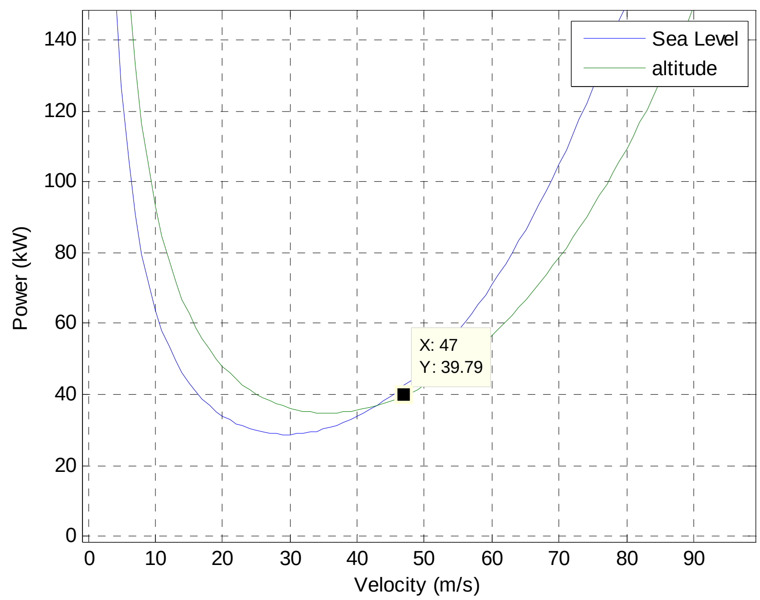

Figure 39: Hybrid Power Required Predator Adjusted Performance

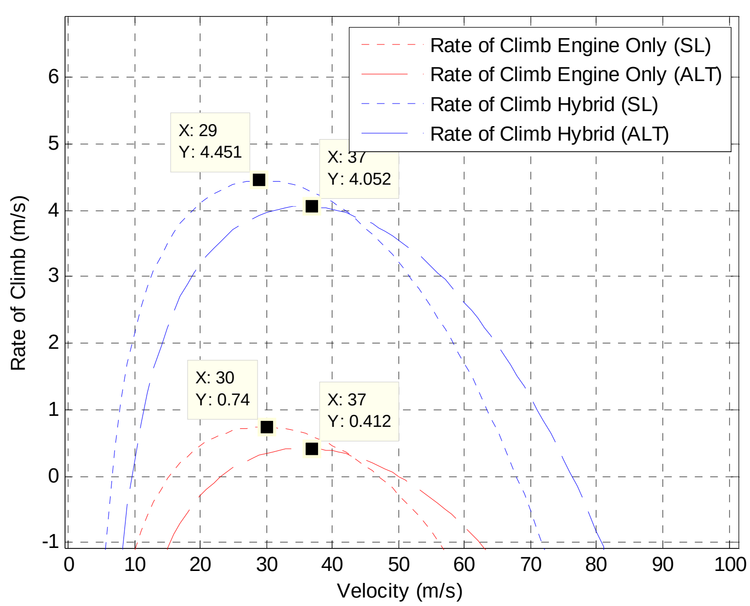

Figure 40: Hybrid Rate of Climb Predator Adjusted Performance

turn rate can be large for high AR aircraft but the high wing loading makes it difficult to support the large g-loads. Figure 41 outlined the sustained turn rate for the adjusted hybrid Predator.

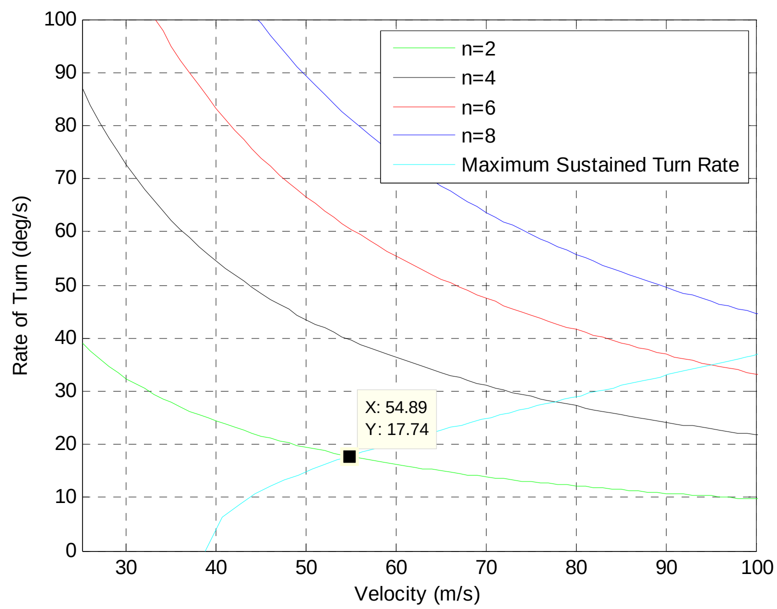

Figure 41: Maximum Sustained Turn Rate Predator Adjusted Performance

Promising results were found for a mild-hybrid Predator, as long as a new mission was defined that was suitable for the adjusted performance case. Original performance requirements could not be met using the hybrid propulsion system. Manipulating the mild-hybrid's propulsion system and performance criteria has the potential to make any hybrid RPA multi-mission capable. Regarding the design code's ability to handle RPAs, small adjustments were necessary. The unique weight distribution for an ISR capable RPA caused a departure from the traditional conceptual design method used for the GA cases. Only minor changes were made to the code so that convergence could be achieved. With a more robust database of RPAs a similar design strategy could yield better results for hybrid RPAs.

8. Code Validation

Promising results suggest that the conceptual design code effectively sized a mild hybrid-electric propulsion system for several aircraft. The DA 20, Cessna 172, and Predator RPA case studies provided evidence that the current propulsion systems could be replaced with an electric motor and smaller internal combustion engine. The large batteries necessary to drive the electric motor meant that some aircraft mass needed to be exchanged for battery mass. To avoid sacrificing payload, the range for each platform was lowered, ultimately trading fuel mass for battery mass. This may be acceptable if the aircraft was primarily used for training mission with numerous transient load conditions. The physical dimensions for each mild hybrid adjusted performance configuration became consistent with the original airframe making a retrofit possible. Though the wing area was allowed to change between $1m^2$ and $50m^2$, the constant wingspan allowed the optimization code to drive the wing area to the original platform's value. With a converged solution that produced equivalent GTOW, wingspan, and wing area the same physical structure could support the aerodynamic performance for the power delivered by the mild hybrid system.

To further validate the results of the case studies, Flight Design's mild hybrid prototype was compared to each aircraft. The 120 kW hybrid power plant designed by was the only comparable mild hybrid aircraft propulsion system. Hybridization factor was compared for each aircraft studied and were recorded in Table 10.

Table 10: Hybridization Factor

	Flight Design	DA 20	Cessna 172	Predator
HF	0.26	0.21	0.3	0.5

Both the DA 20 and Cessna 172 were consistent with the mild hybrid ratio described earlier in Chapter III section 6. The Predator and other RPAs lean toward the full hybrid configuration because of the minimal power requirements needed for the specific missions they conduct. The hope was that using a mild hybrid configuration on RPAs would enable them to be multi mission capable. Further investigation into the appropriate design strategy for RPAs may help support this potential.

Finally, a case was run that allowed the wingspan and wing area to vary. Since the Cessna 172 was the most successful case study, the design requirements for the validation case matched the Cessna's. The results suggested that the conceptual design tool was able to optimize an aircraft for hybrid-electric propulsion with performance that matched the original Cessna 172. Table 11 recorded the outcome. As expected the code attempted to

Table 11: Validation Case Study Results

		Original	**Hybrid**
GTOW	kg	1114	1048
Engine	kW	120	62.25
Motor	kW	NA	35.86
Battery	kg	7.2	84
Wingspan	m	11	16.55
Wing Area	m^2	16.2	14.16
Aspect Ratio		7.5	19.35

maximize the aspect ratio in order to reduce the power required at cruise. Such a large increase to the wingspan would cause stress concentrations near the root of the wing. Structural analysis was outside the scope of this research but in the future constraints could be added to account for the added stress large wings would create.

V. Conclusions and Recommendations for Future Research

The initial goal of this research was to investigate how to scale mild hybrid-electric propulsion systems using conventional conceptual design strategies applied to GA and RPA platforms. Previously, Harmon and Hiserote demonstrated the benefits for small full hybrid-electric RPAs using similar conceptual design methods. The resulting question posed by Hiserote was how large can a RPA or any hybrid aircraft be [15]. The conceptual code developed for mild- hybrid systems was limited to single engine aircraft, but supported the scalability of a mild hybrid-electric system up to a GTOW near 1115kg (Cessna 172). Validity of the code was accepted based on several encouraging case study results. Acceptance of the mild hybrid design code opens the doors to the many benefits of hybrid-electric aircraft propulsion beyond expected fuel savings.

1. Conclusions of Research

The mild hybrid conceptual design tool demonstrates the relationship between the important weight fraction considerations. The make-up of an aircraft's overall weight can be represented by the component weight fractions. For the mild-hybrid designs this distribution was significantly altered by the battery mass needed to power the electric motor during certain phases of flight. This effort monitored and compared the weight fractions before and after hybridization. For the adjusted hybrid GA aircraft the weight distribution was not dramatically changed if the energy storage for battery and fuel was calculated together. The mass of energy stored increased because of the low specific energy of the batteries. Likewise, the RPA case followed the same trend but the adjusted hybrid case sacrificed a greater percentage of performance. Once battery specific energy can be improved less performance would need to be sacrificed.

From the perspective of aircraft design, the mild hybrid was a less efficient means of storing energy on the aircraft by replacing fuel with heavy batteries. The cost outweighs the benefit in terms of range performance and payload ability. The specific energy of fuel was far greater than the specific energy of batteries. Reasoning behind using the electric energy stored in batteries was that it can be delivered via an electric motor with 90% or greater efficiency. Efficiency of gas engines has peaked around 30%. So, the high specific energy of the fuel is wasted on thermal inefficiency. Specific energy of batteries only needs to reach 30% of hydrocarbon fuel's specific energy to be just as effective. If more fuel could be replaced by batteries the efficiency of the energy stored on-board would increase. The mild hybrid-electric design demonstrated fuel saving potential and improved the overall efficiency of the energy delivery.

The improvement of the RPA propulsion design to satisfy multiple missions was supported by this research. With a hybridization factor near 0.5 the mild hybrid RPA realistically becomes a full-hybrid system similar to the Harmon and Hiserote design [5] [15]. Their design was for 1 hr cruise ingress, 1 hr loiter, and 1 hour cruise egress. Most RPAs were designed for this type of ISR missions, a mild-hybrid RPA may not be ideal for this type of mission but could be beneficial for attack or quick response missions. The added performance provided by the motor could get the platform to altitude quickly and provide boost power in combat. Much like the full hybrid, the interchange of fuel, battery storage, and payload would allow multiple missions to be conducted on one platform.

The added power source inherent in the mild-hybrid design not only provided increased efficiency but also added a redundant safety measure. Redundancy has been common for aircraft systems to improve safety. Multiple power supplies, pneumatic devices, and control surface linkages were common on many GA aircraft. Most engines use multiple sets of spark plugs that fire simultaneously to avoid engine failure during flight. The mild hybrid-electric motor would not be adequate to sustain flight, but would be large enough to help maintain a powered glide. Extending the glide slope of the aircraft could allow the pilot to locate a safe landing spot to ditch the airplane. Helping the pilot to avoid trees, water, or possibly residential areas, even a small amount of power could save the pilot and innocent bystanders. A redundant power source would be highly desirable for the FAA and GA pilots.

2. Recommendations for Future Research

If aircraft can benefit from the mild hybrid design for a simple cruising mission, more benefits would be expected for more typical training missions. More simulations need to be run so the mild hybrid-electric system can be more effectively used for different mission profiles. Compared to the automotive industry the hybrids were more effective in the city where the motor was more capable of augmenting the power of the engine. Takeoff and climb were not the only mission segments that could benefit from the added power source. During a typical training mission a student pilot may conduct multiple climbs, sustained turns, slow flight, and missed landing scenarios that may need power beyond the hybrid engine. Keeping the engine at its ideal operating line, increasing the throttle would require power from the motor instead of increasing the fuel use. To simulate these conditions different fuel fractions could be calculated for each scenario.

The necessary motor power would dictate how much energy would be needed from the batteries. Summing each mission segment from takeoff to landing the total fuel mass and battery mass could be determined. Expected results would be that added fuel savings would result.

Eventually the physical integration of the engine, motor, and battery will become a challenge. Multiple suggestions have been proposed to the mechanical configuration of the components. Without knowing the performance of the transmission, additive torque between power sources was not guaranteed. Additive torque between the components was essential for the conceptual design code to be accurate. Other issues arise when placing each component into the aircraft. Each component mass must be placed appropriately so the center of gravity location yields a stable aircraft. The obvious decision would be to place the engine and motor within the forward cowling. The battery mass could be split by using multiple batteries to assist with acceptable weight and balance. Flight testing would be the ultimate confirmation on the stability of the aircraft. Future work will be needed to integrate the power sources using an effective transmission, and determining the proper placement of each component in the aircraft.

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Appendix A: MATLAB Code Equations

Function [Rippl_Mild_Hybrid_Design_Code]

$$W_{Glider} = W_{E,Original} - W_{Engine,Original}$$

$$W_0 = W_{Glider} + W_{Payload} + W_{Motor} + W_{Battery} + W_{Engine} + W_{Fuel}$$

$$W_{Engine} = \frac{PR_{Cruise}}{\rho_{Alt}/\rho_0} \times EngineSpecPower$$

Function [weightestimation]

$$WF_{WUTO} = .99$$

$$WF_{\text{climb}} = .98$$

$$WF_{Cruise} = \exp\left[{}^{-(Range)(6076)(SFC)}\big/_{V_{Cruise}(3.28)(L/D)} \right]$$

$$WF_{Loiter} = .99$$

$$WF_{Land}.995$$

$$WF_{mission} = WF_{WUTO} WF_{\text{climb}} WF_{cruise} WF_{loiter} WF_{land}$$

$$WF_{fuel} = 1.06\left(1 - WF_{mission}\right)$$

$$WF_{Empty} = AW_0^{\,C}$$

$$W_{new} = \frac{W_{Payload}}{1 - WF_{fuel} - WF_{Empty}}$$

Function [optimizemildhybrid]

$$AR = \frac{b^2}{S}$$

$$P = \frac{1}{\eta_{prop}\eta_{mech}}\left[\frac{1}{2}\rho V_{Cruise}{}^{3} S C_{D_0} + \frac{W}{S}\frac{KW}{\frac{1}{2}\rho V}\right]$$

$$where\ K = \frac{1}{\pi e AR}$$

$$\frac{W_0}{S} - \frac{1}{2}V_{Stall}{}^{2} C_{L,Max} \le 0$$

$$L = nW_0 = \frac{1}{2}\rho V^2 S\sqrt{C_{D_0}\pi e AR}$$

$$AR - \left[\frac{2nW_0}{\rho V_a{}^{2} S}\right]^2 \frac{1}{C_{D_0}\pi e} \le 0$$

Function [performance]

$$C_L = \frac{W}{\frac{1}{2}\rho V_\infty{}^2 S}$$

$$C_D = C_{D,0} + \frac{C_L{}^2}{\pi e AR}$$

$$\frac{C_L}{C_D} = \frac{L}{D}$$

$$T_R = \frac{W}{C_L / C_D}$$

$$Power = \frac{energy}{time} = \frac{force \times distance}{time} = force \times \frac{distance}{time}$$

$$P_R = T_R V_\infty$$

$$ROC = \frac{P_{Available} - P_{Required}}{W_0}$$

$$s_{LO} = \frac{1.44 W^2}{g \rho_\infty S C_{L,\max} T}$$

$$W_{fuel} = SFC \times PR_{cruise} \times \frac{1852 \times Range}{V_{Cruise} 3600 \times 2.2}$$

Function [motordesign]

$$P_{R,takeoff} = \frac{1.44 V_{Cruise} W^2}{g \rho_\infty S C_{L,\max} s_{LO,desired}}$$

$$P_{EM} = P_{R,Takoff} - P_{R,Cruise}$$

$$P_{EM} = \left(\frac{W_0 (ROC)_{Desired}}{60} + P_{R,\min} - P_{R,Cruise} \right)$$

$$E_{Battery} = P_{EM} \left(\frac{Alt_{Operating}}{ROC_{best}} + 600 \right)$$

$$W_{Battery} = \frac{E_{Battery}}{BatterySpecificEnergy}$$

$$HF = \frac{P_{EM}}{P_{EM} + P_{Engine}}$$

Appendix B: Mild Hybrid-Electric Conceptual Design Code

```
function []=Rippl_Mild_Hybrid_Design_Code()
%% Mild Hybrid-Electric Propulsion System
%% Matthew Rippl
%% Air Force Institute of Technology
%% Masters Degree Program
%% Grad Date March 2011

clear all; clc; close all;
% Title and Date-Time Stamp
timestamp = clock;
disp('Hybrid-Electric UAS Sizing Program');
disp(['Date: ',date,'      Time: ',num2str(timestamp(4)),':', num2str(timestamp(5))]);
disp(' ');
global Cd_0 p_0 W_0 A C Aircraft_mass Calc_Aircraft_mass PR_Cruise_HP Aircraft_mass
Glider_Wgt_org Payload motor_mass Batt_mass Engine_mass WF_fuel e n_prop n_mech W_0g
p_alt p_sl Batt_SpecEGY Specific_Power Glider_Wgt_org max_Wingload Motor_SpecPWR
Org_eng_mass Approx_Eng_SpecPWR g V_cruise RangeDes Payload min_S max_S min_b max_b
Operating_Altitude des_TO_dis Cl_max V_Stall n_prop max_AR des_ROC Max_TOGW_org
Fuel_Wgt_org Engine_Wgt_org Payload_Wgt_org Empty_Wgt_org Battery_Wgt_org EM_Wgt_org Va
Fuel_mass

e       = .8;                   % Wing Efficiency
Cl_max  = 2.0;                  % Maximum lift coefficient (Raymer pg. 96)
p_sl    = 1.2;                  % Air Density at sea level
g       = 9.81;                 % Acceleration due to gravity
V_Stall = 24;                   % Stall Speed Maximum (m/s)
n_prop  =.8;                    % Propeller Efficiency
n_mech  =.97;                   % Mechanical Efficiency
Batt_SpecEGY    = 150;          % Battery Specific Energy (Lithium Ion Wh/kg)
Motor_SpecPWR   = 3;            % Specific Motor Power (HP/kg)
Approx_Eng_SpecPWR = 1.25;      % (HP/kg)


% Select Altitude for the Calculations
    h_TO=input('Enter takeoff altitude (meters AMSL):  ');
    h_AGL=input('Enter mission altitude (meters AGL):  ');
    disp(' ');
    h = h_TO + h_AGL;
    [T_TO, a_TO, P_TO, rho_TO] = atmosisa(h_TO);
    [T, a, P, rho] = atmosisa(h);
    disp(['Mission Altitude Density (kg/m^3) = ', num2str(rho)]);
    p_alt = rho;
    Operating_Altitude = h_AGL;
    disp(' ');

% Requirements
V_cruise        = input('Desired Cruise Speed (m/s):   ');
Va              = input('Manuevering Speed (m/s):      ');
des_ROC         = input('Aircraft Desired Rate of Climb at SL (m/min):  ');
des_TO_dis      = input('desired takeoff ground roll (m)');
RangeDes        = input('Desired Range (nm)');
Payload         = input('Desired Payload (Pounds)');

% Constraint Input Limits
max_S           = input('Maximum Wing Area Constraint:  ');
min_S           = input('Minimum Wing Area Constraint:  ');
max_b           = input('Maximum Wingspan Constraint (m):  ');
min_b           = input('Minimum Wingspan Constraint (m):  ');


% Original Aircraft Component Mass (kg)

Max_TOGW_org = 800;     input('Max take off weight for original aircraft')
Fuel_Wgt_org = 65;      input('Fuel weight for original aircraft')
Engine_Wgt_org = 93;    input('Engine weight for original aircraft')
Payload_Wgt_org = 212;  input('Payload weight for original aircraft')
Empty_Wgt_org = 523;    input('Empty weight for original aircraft')
Battery_Wgt_org = 7.2;  input('Battery weight for original aircraft')
```

```
EM_Wgt_org = 0;
W_0g = Max_TOGW_org*2.2;
% Glider weight for original aircraft. This value will become the basis for
% the weight buildup of the hybrid design. Both glider weights are meant to
% be identical since the hybrid system is to be installed in the existing
% airframe

Glider_Wgt_org = Empty_Wgt_org - Engine_Wgt_org;

% The Weight estimation calculation is based on historical data using
% Raymer's approximations. Different values are chosen for A and and C for
% the empty Weight Equation found in Table 3.1 in Raymer. A switching case
% is used to determine what type of original aircraft will be used for the
% calculation.

    disp(' ');
    disp('Select approximate aircraft type:');
    disp('  1:  General Aviation Single-engine');
    disp('  2:  Powered Glider');
    disp('  3:  Homebuilt Composite');
    disp('  4:  RPA');
    disp(' ');
    Aircraft_type=input('Enter your selection:  ');
    disp(' ');

   switch Aircraft_type
       case 1
           Aircraft_type = 'General Aviation Single Engine';
           disp('Begin Weight Estimation using General Aviation Single Engine
estimation');
           Cd_0 = .022;
           A = 2.36;
           C = -.18;
           weightestimation()
       case 2
           Aircraft_type = 'Powered Glider';
           disp('Begin Weight Estimation using Powered Glider estimation');
           Cd_0 = .02;
           A = .91;
           C = -.05;
           weightestimation()
       case 3
           Aircraft_type = 'Homebuilt Composite';
           disp('Begin Weight Estimation using Homebuilt Composite estimation');
           Cd_0 = .018;
           A = 1.15;
           C = -.09;
           weightestimation()
       case 4
           Aircraft_type = 'Remotely Piloted Aircraft';
           disp('Begin Weight Estimation using Remotely Piloted Aircraft estimation');
           Cd_0 = .03
           disp('Select Appropriate RPA Grouping');
           disp('  1:  RPA < 120 lbs');
           disp('  2:  RPA > 120 lbs');
           disp('')
           RPA_Group=input('Enter your selection:  ');
           disp('')
           switch RPA_Group
               case 1
                   RPA_Group = 'Group 1: < 120 lbs';
                   A = .6209;
                   C = -.0161;
                   weightestimation()
               case 2
                   RPA_Group = 'Group 2: > 120 lbs';
                   A = .5728;
                   C = -.0015;
                   weightestimation()
           end
```

```
   end

% The while loop is used to iterate the mass of the aircraft. Initailly the
% weight estimation from Raymer is used. However the hybrid configuration
% has unique mass considerations that are accounted for by building up the
% weight of the new component from the glider weight of the original
% aircraft. The new mass is then input into the optimization routine and is
% iterated until the convergence criteria is satisfied.

convergence1 = 1;
while convergence1 > .01
    % Optimizehybrid sends the requirement values to the optimization
    % routine to minimize power required at alititude.
    optimizemildhybrid();
    % performance takes the wingspan, wing area, and weight to calculate
    % the power required and thrust required curves to establish the
    % maximum rate of climb and the maximum lift to drag ratio.
    performance();
    % motordesign uses the engine size calculated from the optimization
    % routine and calculates the neccessary excess power needed to meet the
    % takeoff ground roll and rate of climb requirement.
    motordesign();
    % Current Weight buildup of hybrid aircraft
    Calc_Aircraft_mass = W_0/9.81;
    % Current engine mass after PR_Crusie_HP changed during iteration
    Engine_mass= (PR_Cruise_HP/((p_alt/p_sl)*Approx_Eng_SpecPWR));
    % New aircraft weight buildup for hybrid system
    Aircraft_mass = Glider_Wgt_org + Payload/2.2 + motor_mass + Batt_mass + Engine_mass +
Fuel_mass;
    % new weight passed to optimization routine for next iteration.
    W_0 = Aircraft_mass*9.81;
    convergence1 = Aircraft_mass - Calc_Aircraft_mass;
end
    % postprocess displays all pertinent information about new hybrid
    % design and publishes figures of merit.
    postprocess()
end

function [] = weightestimation(W_0)
global  W_0g RangeDes Payload WF_WUTO WF_climb WF_cruise WF_loiter WF_land WF_empty
WF_fuel W_0 A C initial_W_0 WF_mission V_cruise

   WF_WUTO = .99;           % Fuel Weight Fraction Warm-Up and Takeoff
   WF_climb = .98;          % Fuel Weight Fraction Climb
   WF_cruise = exp(-(RangeDes*6076*.0001)/(V_cruise*3.28*17));  % Fuel Weight Fraction
Cruise
   WF_loiter = .99;         % Fuel Weight Fraction Loiter
   WF_land = .995;          % Fuel Weight Fraction Land
   convergence2 = 2;

    while convergence2 > .01
   % Total Mission Weight Fraction
   WF_mission = WF_WUTO*WF_climb*WF_cruise*WF_loiter*WF_land;
   WF_fuel = 1.06*(1-WF_mission);           % Total Fuel Weight Fraction
   WF_empty = A*W_0g^C;                     % A and C are determined from switching case
   W_new = Payload/(1-WF_fuel-WF_empty);    % Calculated Takeoff Weight
   convergence2 = abs(W_new-W_0g);          % Value of Convergence after each iteration
   W_0g=W_new;                              % Makes the Calculated weight the New Guess
   end
   % The iterative weight estimation is performed to yield pounds so value
   % must be converted to N for the remainder of the calculations.
    W_0 = W_0g/2.2*9.81;
    initial_W_0 = W_new/2.2;
end

function []=optimizemildhybrid()
global  W_0 PR_Cruise_HP min_b max_b AR S b Wingload min_S max_S
% Initial inputs
LB = [min_b min_S];
UB = [max_b max_S];
```

```
options = optimset('Algorithm','interior-
point','MaxFunEvals',5000,'TolCon',.01,'TolX',1e-9);
x_0 = [8;9];
% Variables
    % x(1):  Wingspan, b (m)
    % x(2):  Wing Area, S (m^2)
[x,PR_Cruise_HP,ExitFlag]=fmincon(@optim_aircraft,x_0,[],[],[],[],LB,UB,@optim_aircraft_c
ons,options)
S = x(2);
AR = x(1)^2/x(2);
b = x(1);
Wingload = W_0/x(2);
end

function [f] = optim_aircraft(x)
global Cd_0 e n_prop n_mech W_0 V_cruise p_alt
% Objective Function Taken from Raymer Equation 17.17 at the Cruise
% Requirement



f = (1/n_mech) * (1/n_prop * ( (.5*p_alt*V_cruise^3*Cd_0*x(2))) +
(W_0/x(2))*(2*W_0/(p_alt*pi*V_cruise*e*(x(1)^2/x(2))) ) )/750;
end

function [C,Ceq]=optim_aircraft_cons(x)
% Constraint functions
global  Cl_max V_Stall W_0 p_sl e Va Cd_0
 % Nonlinear Constraints for Cost Function
    % Variables
    % x(1):  Wingspan, b (m)
    % x(2):  Wing Area, S (m^2)
    % Inequality Constraints
    C=[(W_0/x(2)) - V_Stall^2*p_sl*Cl_max/2;
       (x(1)^2/x(2)) - ((2*2*W_0)/(p_sl*Va^2*x(2)))^2/(Cd_0*pi*e);
       ];
   Ceq=[];
end

function []=performance()
global Cd_0 e g W_0 S AR p_sl p_alt CL CD TR PR V PR_Cruise_HP ROC V_cruise g TO_dis
best_ROC_eng n_prop Cl_max LOD_max load Turn Vel rate RangeDes Fuel_mass
V = 1:1:120;        % Velocity matrix (m/s)
x = size(V,2);      % Makes x the size of velocity matrix
p(1) = p_sl;        % Air Density at sea level
p(2) = p_alt;       % Air Density at altitude
for j = 1:2         % for loop to evaluate both sea level and altitude
for n = 1:x         % for loop to evaluate parameters at each velocity
    % Lift Coefficient Anderson Eqn 6.17
    CL(j,n) = W_0/(.5*p(j)*V(1,n)^2*S);
    % Drag Coefficient Anderson Eqn 6.1c
    CD(j,n) = Cd_0 + CL(j,n)^2/(pi*e*AR);
    % Lift to Drag ratio
    LOD(j,n) = CL(j,n)/CD(j,n);
    % Thrust Required Anderson Eqn 6.16
    TR(j,n) = W_0 / (CL(j,n)/CD(j,n));
    % Power Required Anderson Eqn 6.24
    PR(j,n) = (TR(j,n)*V(1,n))/750/n_prop;
    % Rate of Climb Anderson Eqn 6.50
    ROC(j,n) = (PR_Cruise_HP*750*n_prop - PR(j,n)*750)/W_0;
end
end
    % Takeoff Distance Anderson Eqn 6.104
    TO_dis = (1.44*W_0^2*V_cruise)/(g*p_sl*S*Cl_max*PR_Cruise_HP*750)/n_prop;
    % Best Rate of Climb at Sea Level
    best_ROC_eng = max(ROC(1,:));
    % Maximum Lift to Drag Ratio
    LOD_max = max(LOD(1,:));
    % Fuel mass Calculated using specific fuel consumption of typical
    % general aviation aircraft engine
    Fuel_mass = .37*PR_Cruise_HP*(RangeDes*1852/(V_cruise*3600))/2.2;
```

```
% Measured Turn Performance
load = 1:8;                             % Load factor integers
for z = 1:8
for i =1:120
    Turn(z,i) = (g*sqrt(load(z)^2-1))/(V(i))*57.3;
end
end
n = 1:.1:8;                             % Load Factor
limit = size(n);
for y = 1:limit(1,2)
% Velocity as a Function of Load Factor Raymer Eqn. 17.55
Vel(y) = sqrt(n(y)*W_0*2/(p_sl*S*sqrt(Cd_0*pi*e*AR)));
% Turn Rate as a Function of Load Factor and Velocity Raymer Eqn. 17.52
rate(y) = g*sqrt(n(y)^2-1)/Vel(y)*57.3;
end
end

function []= motordesign()
global W_0 g p_sl p_alt des_TO_dis S Motor_SpecPWR motor_mass Batt_mass Cl_max n_prop
PR_Cruise_HP PR best_ROC_eng_em TO_dis_assist V_cruise motor_power des_ROC
Operating_Altitude Batt_SpecEGY Hybrid_factor
% Anderson Eqn 6.104 rearranged with T = P/V where cruise condition at
% altitude is considered (additional prop inefficiency included for takeoff)
Power_needed_TO = ((V_cruise * 1.44 * W_0^2) /
(g*p_sl*S*Cl_max*des_TO_dis))/(750*n_prop);
% Motor Power calculated to satisfy takeoff condition
motor_power(1) = (Power_needed_TO - PR_Cruise_HP);
% Motor Power calculated to satisfy climb condition at sea level
motor_power(2) = (W_0*(des_ROC/60) + min(PR(1,:))*750 -
PR_Cruise_HP*750*n_prop)/(750*n_prop);
% Best rate of climb based on the maximum motor size.
best_ROC_eng_em = ((PR_Cruise_HP + max(motor_power))*n_prop  - min(PR(1,:)))*(750/W_0);
% Best takeoff distance with selected motor and engine combined (additional
% prop inefficiency included for takeoff)
TO_dis_assist = (1.44*W_0^2*V_cruise)/(g*p_sl*S*Cl_max*((PR_Cruise_HP+
max(motor_power))*750))/n_prop;
% Battery energy needed based on time to climb with additional 10 minutes
Battery_Energy = max(motor_power)*750*(Operating_Altitude/best_ROC_eng_em + 600);
% Battery mass calculated based on specific energy estimation for Lithium
% Ion batteries
Batt_mass = Battery_Energy / (Batt_SpecEGY*3600);
% Motor mass estimation based on specific motor power regression
motor_mass = max(motor_power)/Motor_SpecPWR;
% Hybrid factor to measure the degree of Hybridization typical values range
% between .1 to .5
Hybrid_factor = max(motor_power)/(PR_Cruise_HP/(p_alt/p_sl)+max(motor_power));
end

function [] = postprocess()
global e W_0 b p_sl V_cruise RangeDes Payload S V n_prop Aircraft_mass LOD_max ROC TR PR
p_alt Cl_max AR Wingload Glider_Wgt_org Fuel_Wgt_org PR_Cruise_HP WF_WUTO WF_climb
WF_mission WF_cruise WF_loiter WF_land WF_empty WF_fuel TO_dis motor_power TO_dis_assist
best_ROC_eng best_ROC_eng_em motor_mass Batt_mass Approx_Eng_SpecPWR Hybrid_factor
Max_TOGW_org Fuel_Wgt_org Engine_Wgt_org Payload_Wgt_org Empty_Wgt_org Battery_Wgt_org
EM_Wgt_org Turn Vel rate initial_W_0 Fuel_mass
% Current engine mass after PR_Crusie_HP changed during iteration
Engine_mass = PR_Cruise_HP/((p_alt/p_sl)*Approx_Eng_SpecPWR);
% Weight fractions for original aircraft
WF_Empty_org    = Empty_Wgt_org/Max_TOGW_org;
WF_Fuel_org     = Fuel_Wgt_org/Max_TOGW_org;
WF_Engine_org   = Engine_Wgt_org/Max_TOGW_org;
WF_Payload_org  = Payload_Wgt_org/Max_TOGW_org;
WF_Battery_org  = Battery_Wgt_org/Max_TOGW_org;
WF_Motor_org    = EM_Wgt_org/Max_TOGW_org;
WF_propulsion_org = (Engine_Wgt_org + EM_Wgt_org + Battery_Wgt_org)/Max_TOGW_org;

% Empty Weight of aircraft with hybrid system (should be close to original)
Empty_Wgt = Aircraft_mass-Fuel_mass-Payload/2.2;
% New Weight Fractions of Hybrid system
WF_empty = Empty_Wgt/Aircraft_mass;
```

```
WF_fuel = Fuel_mass/Aircraft_mass;
WF_engine = Engine_mass/Aircraft_mass;
WF_Payload = Payload/2.2/Aircraft_mass;
WF_battery = Batt_mass/Aircraft_mass;
WF_motor = motor_mass/Aircraft_mass;
WF_propulsion_Hyb = (Engine_mass + motor_mass + Batt_mass)/Aircraft_mass;
% New Hybrid Propulsion Mass
Hybrid_propulsion_mass =
((PR_Cruise_HP/(n_prop*.65))/Approx_Eng_SpecPWR)+motor_mass+Batt_mass;
% Stall Speed calculated at altitude (should be less than cruise speed)
V_Stall_alt = sqrt((2*W_0)/(p_alt*S*Cl_max*e));

PR(3,:) = PR(1,:);
PR(4,:) = PR(2,:);
ROC(3,:) = (PR_Cruise_HP*n_prop + max(motor_power)*n_prop - PR(3,:))*750/W_0;
ROC(4,:) = (PR_Cruise_HP*n_prop + max(motor_power)*n_prop - PR(4,:))*750/W_0;

disp([' Hybrid Design Results'])
disp([' Initial Aircraft Mass Estimate :', num2str(initial_W_0),    ' kg']);
disp([' Aircraft Mass :                 ', num2str(Aircraft_mass),  ' kg']);
disp([' Range:                          ', num2str(RangeDes),       ' nm']);
disp([' Payload Mass:                   ', num2str(Payload/2.2),    ' kg']);
disp([' Cruise Speed:                   ', num2str(V_cruise),       ' m/s']);
disp([' Stall Speed at Altitude         ', num2str(V_Stall_alt),    ' m/s']);
disp([' Aspect Ratio:                   ', num2str(AR)]);
disp([' Wing Area:                      ', num2str(S),              ' m^2']);
disp([' Wingspan:                       ', num2str(b),              ' m']);
disp([' Wingloading (W/S)               ', num2str(Wingload/9.81),  ' kg/m^2']);
disp([' Maximum Lift to Drag ratio (L/D)', num2str(LOD_max)]);
disp([' Oswald Eff. Factor:             ', num2str(e)]);
disp(' ');
disp([' Power Required for Crusie Speed:                            ',
num2str(PR_Cruise_HP), ' HP']);
disp([' Engine Power for Aircraft:                                  ',
num2str(PR_Cruise_HP/((p_alt/p_sl))), ' HP']);
disp(' ');
disp([' Original Weight Fractions'])
disp([' WF Empty Original:              ', num2str(WF_Empty_org)]);
disp([' WF Fuel Original:               ', num2str(WF_Fuel_org)]);
disp([' WF Engine Original:             ', num2str(WF_Engine_org)]);
disp([' WF Payload Original:            ', num2str(WF_Payload_org)]);
disp([' WF Batteries Original:          ', num2str(WF_Battery_org)]);
disp([' WF Motor Original:              ', num2str(WF_Motor_org)]);
disp([' Glider Weight Original:         ', num2str(Glider_Wgt_org)]);
disp(' ');
disp([' Hybrid Weight Fractions'])
disp([' WF Empty Hybrid:                ', num2str(WF_empty)]);
disp([' WF Fuel Hybrid:                 ', num2str(WF_fuel)]);
disp([' WF Engine Hybrid:               ', num2str(WF_engine)]);
disp([' WF Payload Hybrid:              ', num2str(WF_Payload)]);
disp([' WF Batteries Hybrid:            ', num2str(WF_battery)]);
disp([' WF Motor Hybrid:                ', num2str(WF_motor)]);
disp([' Glider Weight Hybrid:           ', num2str(Glider_Wgt_org)]);
disp(' ');
disp([' Mission Weight Fractions'])
disp([' WF Warm Up Takeoff:     ', num2str(WF_WUTO)]);
disp([' WF Climb:               ', num2str(WF_climb)]);
disp([' WF Cruise:              ', num2str(WF_cruise)]);
disp([' WF Loiter:              ', num2str(WF_loiter)]);
disp([' WF Land:                ', num2str(WF_land)]);
disp([' WF Mission:             ', num2str(WF_mission)]);
disp(' ');
disp([' Takeoff Distance with Engine alone:         ',num2str(TO_dis),             '
m']);
disp([' Motor Power:                                ',num2str(motor_power),        '
HP']);
disp([' Motor Mass:                                 ',num2str(motor_mass),         '
kg']);
disp([' Battery Mass:                               ',num2str(Batt_mass),          '
kg']);
```

```
disp([' Fuel Mass:                                  ',num2str(Fuel_mass),           '
kg']);
disp([' Takeoff Distance with Motor assist:         ',num2str(TO_dis_assist),       '
m']);
disp([' Best ROC with engine:                       ',num2str(best_ROC_eng*60),     '
m/min']);
disp([' Best ROC with engine and motor (SL):        ',num2str(best_ROC_eng_em*60),  '
m/min']);
disp(' ');
disp([' Original Engine Mass:                   ',num2str(Engine_Wgt_org),          '
kg']);
disp([' Fuel Savings:                           ',num2str(Fuel_Wgt_org-Fuel_mass)   '
kg']);
disp([' Hybrid Engine Mass:                     ',num2str(Engine_mass),             '
kg']);
disp([' Hybrid Propulsion Mass:                 ',num2str(Hybrid_propulsion_mass),  '
kg']);
disp([' Hybridization Factor:                   ',num2str(Hybrid_factor)])

figure(1); colormap('bone');
bar1=bar([WF_Fuel_org WF_fuel ; WF_Engine_org WF_engine ; WF_Battery_org WF_battery ;
WF_Motor_org WF_motor], 'group');
set(gca,'YGrid','on','XTickLabel',{'Fuel';'Engine';'Battery';'Motor'},'fontsize',10);
set(bar1(1),'FaceColor',[0.1098 0.1804 0.3098]);
xlabel('Energy Component','fontsize',10); ylabel('Weight Fraction','fontsize',10);
legend2=legend('Original','Hybrid');
title('Energy Weight Fractions Diamond DA 20');

figure(2); colormap('bone');
bar1=bar([WF_propulsion_org WF_propulsion_Hyb; WF_Empty_org WF_empty ; WF_Fuel_org
WF_fuel ; WF_Payload_org WF_Payload ], 'group');
set(gca,'YGrid','on','XTickLabel',{'Propulsion';'Empty';'Fuel';'Payload'},'fontsize',10);
set(bar1(1),'FaceColor',[0.1098 0.1804 0.3098]);
xlabel('Aircraft Component','fontsize',10); ylabel('Weight Fraction','fontsize',10);
legend2=legend('Original','Hybrid');
title('Aircraft Weight Fractions Diamond DA 20');
set(bar1(2),'FaceColor',[0.8706 0.9294 1]);

% Plot Thrust Required vs. Airspeed
figure(3)
plot(V,TR)
title('Thrust Required for Design')
axis([15 80 0 2000])
xlabel('Velocity(m/s)')
ylabel('Thrust(N)')
grid on

% Plot Power Required vs. Airspeed
figure(4)
plot(V,PR(1,:),V,PR(2,:)); hold on;
plot(V_cruise,PR_Cruise_HP,'bo');hold on;
legend('Sea Level' , 'altitude')
axis([0 100 20 170])
title('Power Required for Design Diamond DA 20')
xlabel('Velocity(m/s)')
ylabel('Power(hp)')
grid on

% plot Rate of Climb vs. Airspeed
figure(5)
plot(V,ROC(1,:),':r',V,ROC(2,:),'--r',V,ROC(3,:),':b',V,ROC(4,:),'--b')
legend('Rate of Climb Engine Only (SL)' , 'Rate of Climb Engine Only (ALT)','Rate of
Climb Hybrid (SL)' , 'Rate of Climb Hybrid (ALT)')
axis([0 100 0 8])
title('Rate of Climb Comparison Diamond DA 20')
xlabel('Velocity(m/s)')
ylabel('Rate of Climb (m/s)')
grid on

figure(6)
plot(V,Turn(2,:),'g',V,Turn(4,:),'k',V,Turn(6,:),'r',V,Turn(8,:),'b',Vel,rate,'c')
```

```
legend('n=2' , 'n=4','n=6' , 'n=8' , 'Maximum Sustained Turn Rate')
axis([25 100 0 100])
title('Sustained Turn Rate Diamond DA 20')
xlabel('Velocity(m/s)')
ylabel('Rate of Turn (deg/s)')
grid on
end
```

Appendix C: Empty Weight Fraction Analysis for RPAs

Results for Group 1 UAVs

General model:

f(W_0) = a*W_0^b

Coefficients (with 95% confidence bounds):

a = 0.6209 (0.3648, 0.8771)

b = -0.01609 (-0.1313, 0.09914)

Goodness of fit:

SSE: 0.0345

R-square: 0.01349

Adjusted R-square: -0.1098

RMSE: 0.06567

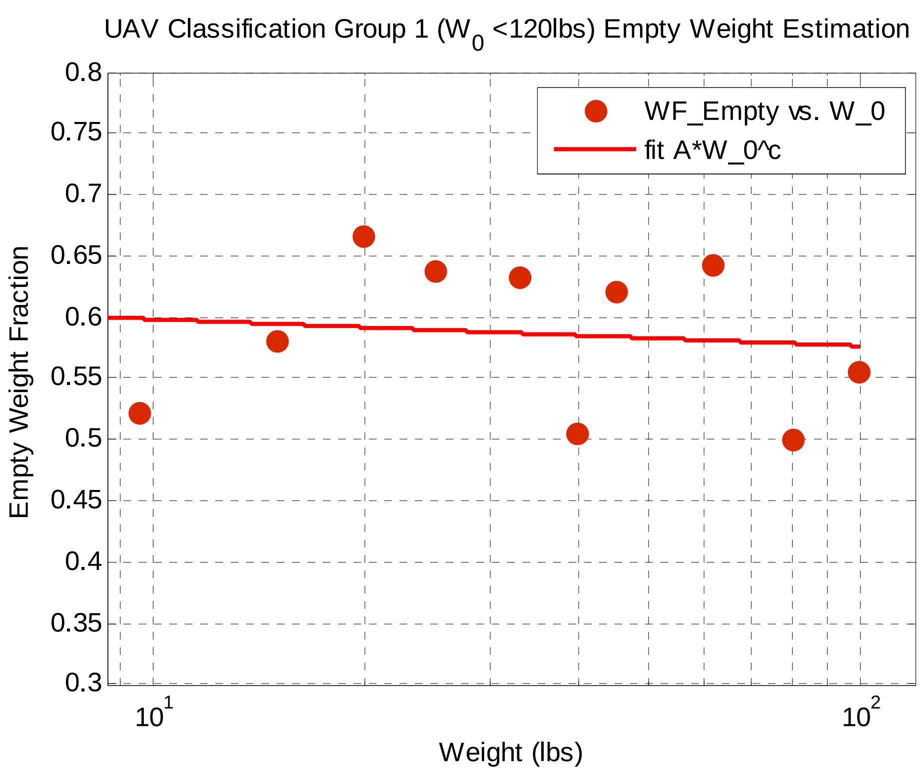

Figure C-1: Group 1 RPA Empty Weight Fraction Best Fit Curve

Results for Group 2 UAVs

General model:

f(W_0) = a*W_0^b

Coefficients (with 95% confidence bounds):

a = 0.5728 (0.2392, 0.9064)

b = -0.001489 (-0.09585, 0.09287)

Goodness of fit:

SSE: 0.1355

R-square: 8.337e-005

Adjusted R-square: -0.07134

RMSE: 0.09839

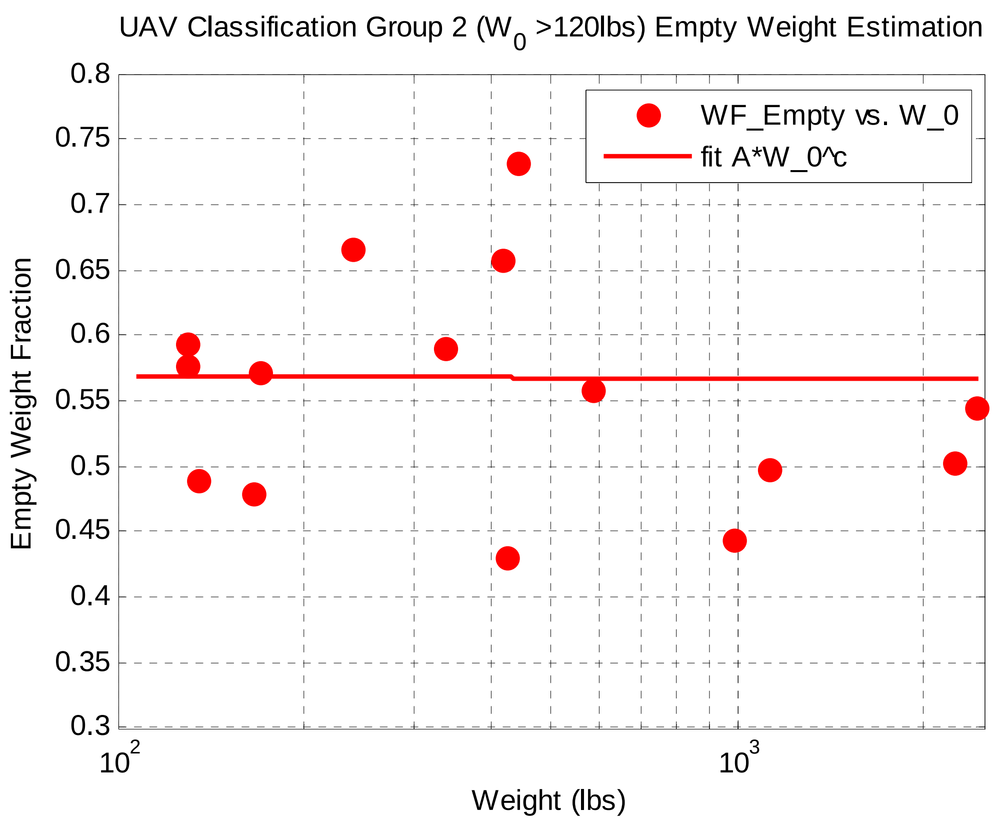

Figure C-2: Group 2 RPA Empty Weight Fraction Best Fit Curve

Table C-1: RPA Review

	TOGW (kg)	*Empty mass (kg)*	*Weight in lbs*	WF_{empty}
Pointer	*4.35*	*2.27*	*9.57*	*0.5218*
Javelin	*6.8*	*3.95*	*14.96*	*0.5809*
Biodrone	*9*	*6*	*19.8*	*0.6667*
Scan Eagle	*18*	*9.1*	*39.6*	*0.5056*
Aerosonde	*15*	*9.5*	*33*	*0.6333*
Silverfox	*11.4*	*7.28*	*25.08*	*0.6386*
T-15	*20.45*	*12.72*	*44.99*	*0.622*
CSV 40	*28*	*18*	*61.6*	*0.6429*
T-16	*36.36*	*18.18*	*79.992*	*0.5*
Brumby Mk3	*45*	*25*	*99*	*0.5556*
XPV Tern	*59*	*35*	*129.8*	*0.5932*
XPV Mako	*59*	*34*	*129.8*	*0.5763*
Integrator	*61.2*	*30*	*134.64*	*0.4902*
Cana Guardian	*77*	*44*	*169.4*	*0.5714*
T-20	*75*	*36*	*165*	*0.48*
Geneva Aerospace Dakota	*109*	*72.6*	*239.8*	*0.6661*
Shadow 200	*154*	*91*	*338.8*	*0.5909*
Pioneer	*190*	*125*	*418*	*0.6579*
Isis	*193*	*83*	*424.6*	*0.4301*
Shadow 400	*201*	*147*	*442.2*	*0.7313*
Shadow 600	*265*	*148*	*583*	*0.5585*
Hermes 450	*450*	*200*	*990*	*0.4444*
Gnat	*511*	*254*	*1124.2*	*0.4971*
MQ5A Hunter	*726*	*540*	*1597.2*	*0.7438*
Predator	*1020*	*513*	*2244*	*0.5029*
Heron	*1100*	*600*	*2420*	*0.5455*

REPORT DOCUMENTATION PAGE	*Form Approved* *OMB No. 074-0188*

1. REPORT DATE *(DD-MM-YYYY)*	**2. REPORT TYPE**	**3. DATES COVERED** *(From – To)*
03/18/2011	Master's Thesis	09/2009-03/2011

4. TITLE AND SUBTITLE **SIZING ANALYSIS FOR AIRCRAFT UTILIZING HYBRID-ELECTRIC PROPULSION SYSTEMS**	**5a. CONTRACT NUMBER** **5b. GRANT NUMBER** **5c. PROGRAM ELEMENT NUMBER**
6. AUTHOR(S) Mr. Matthew D. Rippl	**5d. PROJECT NUMBER** **5e. TASK NUMBER** **5f. WORK UNIT NUMBER**
7. PERFORMING ORGANIZATION NAMES(S) AND ADDRESS(S) Air Force Institute of Technology Graduate School of Engineering and Management (AFIT/ENY) 2950 Hobson Way, Building 640 WPAFB, OH 45433-8865	**8. PERFORMING ORGANIZATION REPORT NUMBER** AFIT/GAE/ENY/11-M26
9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) Dr. Fred Schauer (frederick.schauer@wpafb.af.mil) Air Force Research Laboratory 1950 Fifth Street WPAFB, OH 45433-7251	**10. SPONSOR/MONITOR'S ACRONYM(S)** AFRL/RZTC **11. SPONSOR/MONITOR'S REPORT NUMBER(S)**

12. DISTRIBUTION/AVAILABILITY STATEMENT

APPROVED FOR PUBLIC RELEASE; DISTRIBUTION UNLIMITED

13. SUPPLEMENTARY NOTES

14. ABSTRACT

Current conceptual aircraft design methods use historical data to predict and evaluate the size and weight of new aircraft. These traditional design methods have been ineffective to accurately predict the weight or physical dimensions of aircraft utilizing unique propulsion systems. The mild hybrid-electric propulsion system represents a unique design that has potential to improve fuel efficiency and reduce harmful emissions. Hybrid-electric systems take advantage of both reliable electric power and the long range/endurance capabilities of internal combustion engines. Desirable applications include general aviation single-engine aircraft and remotely-piloted aircraft. To demonstrate the advantages of mild hybrid-electric propulsion, a conceptual design code was created that modified conventional methods. Using several case studies, the mild hybrid conceptual design tool verified potential fuel savings for general aviation aircraft and expanded mission capability for remotely-piloted aircraft.

15. SUBJECT TERMS

Hybrid-electric, Propulsion, mild hybrid, Optimization, Power

16. SECURITY CLASSIFICATION OF:			**17. LIMITATION OF ABSTRACT**	**18. NUMBER OF PAGES**	**19a. NAME OF RESPONSIBLE PERSON**
					Frederick G. Harmon, Lt Col, USAF
a. REPORT	**b. ABSTRACT**	**c. THIS PAGE**			**19b. TELEPHONE NUMBER** *(Include area code)*
U	**U**	**U**	**UU**	148	DSN 786-3636 x7478 (frederick.harmon@afit.edu)

Standard Form 298 (Rev. 8-98)
Prescribed by ANSI Std. Z39-18