Design, Modelling and Measurement of Hybrid Powerplant for Unmanned Aerial Vehicles (UAVs)

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Thesis submitted in fulfilment of the requirements for the award of the Master of Engineering degree.

November 2012
Statement of Original Authorship

The work contained in this thesis has not been previously submitted for a degree or diploma at any other higher education institution. To the best of my knowledge and belief, the thesis contains no material previously written by another person except where due reference is made.

Signature: QUT Verified Signature

Date: 15/04/2013
Key Words:

Aircraft, Hybrid, Propulsion, Hybrid-Electric, Drone, UAV, UAS, Engine, Propeller, Efficiency, Performance, Mission, Take-off, Climb, Motor, Engine, Fuel, Battery, Runway, Launcher, VTOL, Turbo-Electric
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1 ABSTRACT

The success or effectiveness for any aircraft design is a function of many trade-offs. Over the last 100 years of aircraft design these trade-offs have been optimized and dominant aircraft design philosophies have emerged. Pilotless aircraft (or uninhabited airborne systems, UAS) present new challenges in the optimization of their configuration.

Recent developments in battery and motor technology have seen an upsurge in the utility and performance of electric powered aircraft. Thus, the opportunity to explore hybrid-electric aircraft powerplant configurations is compelling. This thesis considers the design of such a configuration from an overall propulsive, and energy efficiency perspective. A prototype system was constructed using a representative small UAS internal combustion engine (10cc methanol two-stroke) and a 600W brushless Direct current (BLDC) motor. These components were chosen to be representative of those that would be found on typical small UAS.

The system was tested on a dynamometer in a wind-tunnel and the results show an improvement in overall propulsive efficiency of 17% when compared to a non-hybrid powerplant. In this case, the improvement results from the utilization of a larger propeller that the hybrid solution allows, which shows that general efficiency improvements are possible using hybrid configurations for aircraft propulsion.

Additionally this approach provides new improvements in operational and mission flexibility (such as the provision of self-starting) which are outlined in the thesis. Specifically, the opportunity to use the windmilling propeller for energy regeneration was explored. It was found (in the prototype configuration) that significant power (60W) is recoverable in a steep dive, and although the efficiency of regeneration is low, the capability can allow several options for improved mission viability.

The thesis concludes with the general statement that a hybrid powerplant improves the overall mission effectiveness and propulsive efficiency of small UAS.
2 Introduction

A UAV must perform airbourne mission tasks, therefore getting into and out of the air to fly a payload is ubiquitous. Since the purpose of UAVs is ordinarily to perform aerial work, the powerplant and airframe design are tailored to suit specific missions, but there are many generic operational characteristics, and most aircraft are adaptable for various payloads, speeds, range, endurance and airfield requirements. The process of launching and recovering a UAV is critical to the overall success of the mission operation and is often constrained by propulsion systems flexibility. Many small UAVs cannot be launched without ground infrastructure and personnel interaction, which limits autonomy. The payload purpose sets the operational requirement. The size and mass of the UAV airframe to carry a certain payload on a certain mission varies, and many different airframe configurations are in design or production and use. The mass or volume of payload combined with distance, time or speed requirements are the defining characteristics of any aeroplane. Hence usual measures of effectiveness are Range, Endurance, Speed, and Cost, for a particular payload mission or a set of missions. Also, the UAS must be managed by people, and a good UAS will be “user friendly”. The powerplant in combination with the airframe, transfer energy from storage into ‘PayloadDistance’, or ‘PayloadTime’ at some rate. The utility of this process for the operator is a significant factor in overall effectiveness.

A large part of the aircraft effectiveness derives from its energy efficiency. The direct cost of energy in dollar terms is not usually a significant problem for Small (under 150 kg) sized UAVs. The real cost of energy in either liquid fuel or battery storage is the limit of payload range or payload endurance which carrying it implies. Any development in powerplant technology is usually focused on energy efficiency constrained to yield high power to weight ratios (P/W) at low mass specific fuel consumption (SFC). While the problem inevitably involves the whole aircraft design, some balanced combination of P/W and SFC drives engine design usually separately from specific aircraft. The range of engines suitable for small UAVs (under 50kg take-off weight) is based largely on radio controlled hobby equipment. These engines have various levels of optimization and range from 0.5cc to 250cc swept displacement piston or rotary two stroke and four stroke types, producing between 50Watts and
25kW. The other main type of propulsion power for propeller driven UAVs is the electric motor. These range from 50 Watt to 5 kW units. A representative size suitable for experimentation was selected, in the form of a 2kW rated engine and 500W rated Brushless DC motor.

Usually these two forms of powerplant are used completely independently, either the aircraft is powered solely by a liquid fueled engine, or battery charge supplied electric motor. Each powerplant has different characteristics when transferring energy through an output shaft to drive the load, which is the aircraft propeller. Some differences are typically within the overall power and energy mass density which often favour liquid fuelled engines, but may encompass the ease of handling and reliability, which often favour electric motors. A fundamental difference is the mass change inherent in fuel burning aircraft, which improves the overall efficiency of the aircraft during the flight. Also, electric motors are inherently self-starting, while internal combustion piston engines require application of external torque to the crankshaft, usually from an electric motor, before becoming self-sustaining.

An Aircraft Hybrid Powerplant (AHP) consists of an internal combustion engine (ICE) and electric motor (EM) system designed to transfer energy between sources and loads. The loads driven by the powerplant are the propeller power and the electrical services power. The load driven by the electric motor is the engine starting power and propeller power. Possible energy sources include fuel, battery, excess potential or kinetic momentum and solar radiation. The mechanical combination of the ICE and EM may be effected in several distinct layouts. The two main hybrid schemes are Parallel and Series configurations. Figure 2-1 shows the schematic layout and energy flow paths of a Parallel Hybrid powerplant suitable for a typical UAV.
Introduction

Figure 2-1 Parallel Hybrid Architecture
The parallel system architecture has advantages due to the inherent redundancy offered by the capability of utilizing either prime mover independently or together. Also there is a requirement for only one electric machine (motor generator) whereas the series architecture as shown in Figure 2-2 requires at least two electric machines, an ICE starter/generator and separate propeller motor/generator.

Figure 2-2 Series Hybrid Architecture
The series hybrid system can be advantageous where distributed prime movers are desired. A single ICE and generator set can be used to drive multiple propeller motors.

In this work, a parallel hybrid architecture has been selected for analysis due to the lower system weight and relative simplicity of installation for candidate UAVs.

Theoretical analysis of a hybrid powerplant considers parameters associated with propeller, engine, motor, fuel, battery, power-conditioning, airframe, and mission. Some properties are generic while some specific data must be measured. Once a representative qualitative analysis is available for the constituent parts, the powerplant design can proceed to combine the parts to fit particular requirements. This research explores a particular method to analyze an appropriate degree of hybridization (proportion of installed power) for the candidate engine and motor.

Simulations of a powerplant based on the characteristics determined above are compared to baseline non-hybrid systems. The simulation is necessarily constrained to certain arbitrary representative aircraft mission scenarios.

A representative airframe analytic simulation model has been developed to enable powerplant performance comparisons with the design payload affecting the size and mass of the airframe. The design speed, combined with the aerodynamic efficiency, required climb rate and angle parameters dictate the thrust and power requirements.

An overriding assumption of the airframe and propulsion analysis is that the aircraft will be fitted with a fixed pitch propeller, and should take-off independently from external auxiliary launch equipment. These conditions are a realistic baseline for improvement and optimization for several reasons. Variable pitch propellers are extremely rare in small UAS. The technical and mechanical complexity, weight, reliability, relative fragility, and expense associated with variable pitch UAV propellers has not been outweighed by the efficiency performance improvements they may yield. This is a similar case to many human carrying light and ultra-light aircraft, where fixed pitch propellers are in extensive use. The capability of independent launch requires an onboard powerplant starting mechanism, which is inherent for electric units but which requires an auxiliary system to be carried for internal combustion engines. Independent launching and propulsion initiation is desirable for
many reasons some of which may be subject to tradeoff considerations while others would be of necessity.

Fixed pitch propellers are optimized for certain combinations of rotational and forward speed, while internal combustion engines are optimal over a certain range of rotational speed and power settings. The ability to produce good slow speed thrust is necessary for takeoff, approach and landing capability. A propeller exhibits best absolute thrust at low forward speed but must be optimized for efficient cruise speeds, involving airframe and engine parameters. Extra thrust can be produced at any speed within blade Mach speed limits by increasing rotational velocity, but installed power is limited. By combining an electric boost capability, significant extra thrust can be made available within battery storage and motor mass and thermal limits. Electric boost may be tuned to enable the engine to operate at its optimal load and rotational speed. A fixed pitch cruise propeller may be selected which gives good efficiency at cruise, but is not well matched to an engine at static and low forward speeds. Additionally the overall fuel efficiency of the aircraft can be improved by the propeller selection available given the presence of the auxiliary boost motor. The power control system for electric motors is advanced particularly with small Brushless Direct Current (BLDC) machines. The electronic speed control (ESC) system can supply current between the motor and battery with high efficiency and can be precisely controlled and programmed. Thus it is easy to manage power flow into or out of the battery, engine and propeller remotely or autonomously.

### 2.1 Research questions

The following research questions are raised in this research project:

1. **Can an AHP yield energy efficiency advantages for small UAS?**
2. **Can an AHP yield overall productivity and effectiveness advantages for small UAS?**
3. **How can they be determined and measured?**
4. **What are the problems in implementing a practical hybrid system?**

The methods of measuring, sizing and installing power inputs and outputs are discussed below.
Payload Range and Payload Endurance are primary indicators of aircraft economic productivity and rely on aircraft efficiency parameters dependent on powerplant matching to particular airframe and mission requirements.

Effectiveness includes productivity as above but also the ease, reliability and safety of operation as well as the adaptability and flexibility of usage modes to enable varying mission requirements.

In terms of UAS effectiveness and productivity, some subjective, probabilistic, and historical considerations are relevant.

Engine failures are not uncommon on small UAS.[1] Research on forced landing systems is specifically directed toward reducing the negative consequences from human injury and property loss to hull loss. It is often the case that small engines (and aircraft engines in general) may suffer a temporary fault which could lead to the loss of the aircraft. The ability of onboard restarting capability could prevent such loss, moreover, the existence of a parallel energy path to the propeller could enable recovery or at least a better guided descent.

Routine UAS handling in operations suffers from the necessity for external starting systems. Some remote starter systems have been developed which remove the human contact danger from the process, however these are not in normal use across the range of small UAS and represent significant extra operating infrastructure. These elements of effectiveness and productivity are primarily in the domain of piston engine powerplants, all-electric systems mitigate these problems but bring several other problems. The outstanding problem of electric powered aircraft is the low energy mass density of electrical energy storage in comparison with liquid hydrocarbon fuel. A power system energy density delta between all-electric and liquid fueled powerplants generally favours liquid fuel by a factor of 10:1. Sections, 6.1, 6.2 and 6.3 detail the energy density and conversion efficiency typical for EM and ICE powerplants used on typical small UAS. The existence of many all-electric UAS and some emerging human carrying aircraft [2-4] is due to the development of battery technology as well as the improvements in motors and current control systems. The combined hybrid powerplant benefits from these same improvements.
2.2 **UAV missions**

The nature of small UAS mission scenarios is inherently diverse. A key advantage of UAS over conventional onboard human piloted aircraft is the adaptability and flexibility of airframes suited for new roles. The lead times for operating new designs in new missions is not necessarily constrained by traditional airworthiness and developmental processes. A new or modified aircraft design can enter service in a new mission scenario literally within hours or minutes from the operational requirement specification. Since aircraft performance capability is strongly dependent on powerplant performance, the flexibility and adaptability of the powerplant can be of extreme importance.

For example, if a given mission was not feasible due to runway length constraints for take-off, and either full fuel or maximum payload were not required, the extra power afforded by the mass of an electric motor and battery capacity would reduce the runway length required. Typical present measures against this problem include the use of launch catapults and recovery apparatus, however these represent significant extra infrastructure requirements and may not be available where and when necessary.

The extra mass of implementing an onboard starting system is often considered to be too detrimental to payload, range or endurance capability. However it will be shown that the overall system efficiency can be improved using onboard starting equipment to the extent that there is no benefit in omitting it.
3 Literature Review

Very much literature and experience exists in design of engines, electric motors, aeroplanes and propellers. Specific literature for aircraft of the size of small UAS is less common, however there has been increasing activity in this area during the past 10 years.

Primary source material for this research falls into several categories;

- Aircraft Mechanics of Flight
- Fluid Propellers
- Liquid Fuel Piston Engine
- Electric Drives
- Powerplant Dynamometry
- UAS
- Hybrid Vehicles
- Hybrid UAS Propulsion

3.1 Aircraft Mechanics of Flight

Classical Newtonian mechanics have been applied to aeroplane design for over a century. The most significant advances were achieved by the 1960s. Newer developments in aerodynamics are occurring in fields such as stealth design and fluid visualization however the basic material applicable to most UAS is available in many standard textbooks. In particular, the relationships determining energy flow are relevant. Most of these texts also contain various methods of propeller performance analysis, and some engine analysis.

3.2 Air Propellers

An important aspect for Small UAS is the size of the propellers used, typically sourced and suited to engines used in hobby model aircraft engines. These propellers are very well tested and developed, however little specific performance data is available. The main difference from larger more extensively analyzed propellers used for human carrying aircraft is the relatively low Reynold’s number.

A multitude of texts describe fluid mechanics principles are applied to propeller motion in air. Two main lines of analysis are momentum and blade section theory. The first approach considers bulk mass flow energy transfer, while the second finite element wing theory. Neither of the two can yield a completely accurate solution, but
numbers within 5% are considered reasonable on careful examination. Parameters such as Thrust, Torque, Size, and Efficiency can be determined theoretically and also measured.

3.3 Liquid Fuel Internal Combustion Piston Engines (ICE)
Marine, automotive and aeronautical power systems have utilized these units in vast numbers. Thermodynamic gas cycles liberate fuel energy as waste heat and useful work. The piston engine under consideration is a form of Carnot Cycle. Ultimately input fuel energy can be related to useful output shaft power.

Specific sources of information about relevant engine performance is relatively limited. The conditions of use of particular engines vary the performance considerably, manufacturers usually provide limited information.

Measurement systems for determining output power characteristics and efficiency are well developed. Engine Dynamometers have always been used in conjunction with engine development, to ascertain and assess performance outcomes.

3.4 Electric Drives
Developed before the internal combustion engine, electric machines are extensively analyzed in literature. A main distinction in types occurs depending on the source current. Both Alternating Current and Direct Current electric motor drives are used in a vast range of technology. The use in aircraft as a primary propulsion power began seriously only 20 years ago. Microchip Technology Corporation Inc [5] publishes comprehensive technical and performance information regarding generic Brushless Direct Current (BLDC) motors.

A multitude of electric motors designed for hobby model aircraft and small UAS exists. These motor designs are subject to identical theory as for any scale electric machine.

3.5 Powerplant Dynamometry
The performance of small UAS engines is not well documented. Few published references are available concerning specialist small-scale dynamometers of the type required to test these engines. The basic principles of the device are well known in
mechanics science, and applications are principally within the domain of engine manufactures or hobbyists.

Conner and Arena [6] developed a system capable of resolving measurements of relevant magnitude to variance within 1.15%, 2.75% and 2.35% for propeller torque, thrust and efficiency respectively with accuracies of ±1.11%, ±2.04% and ±2.50%. These figures were obtained using a purely electric drive, the influence of using an ICE drive with its inherent vibration was not addressed.

Asson and Dunne [7] have reported on measuring propeller performance using a dynamometer, “capable of providing accurate performance measurements to within ±5% error over a wide range of operating conditions.” Also the concept of in-flight testing was considered feasible.

An interesting and relevant undergraduate design thesis was completed by Hewton and Miller [8] in 2007. Considerable effort was directed at preventing vibration problems in the structure of their dynamometer and electronic filtering was implemented on the strain gauge signal. Problems of hysteresis in the load cell were noted. Measurement accuracy within 5% was claimed. Their dynamometer was designed for measuring and tuning road vehicle engines and consequently a reaction brake type unit was developed with no propeller performance measurement capability.

### 3.6 Small UAS

The history of small UAS of the type relevant to this research is comparatively short, however much research and development is currently being devoted to the field. Several commercially successful examples have emerged which can be used as key representative benchmarks. The “Aerosonde” UAV [9] has been well documented and used as a modeling simulation sample in Matlab Aerosim [10]. The Aerosonde engine was subject to significant development to convert it from a Commercial Off The Shelf (COTS) hobby-grade engine into a high efficiency unit suitable for the very long range and endurance application.[11] The Insitu “ScanEagle”[12] is another UAV which has become an industry standard. Notably, both these UAVs feature onboard electrical generators and are launch assisted by catapult or car rooftop mechanisms. These aircraft are designed for maximum range and endurance, but
cannot be landed remotely then re-launched without human assistance. The compromise for maximum range or endurance requiring omission of undercarriage leads to reduced mission flexibility since the extra weight and drag of the undercarriage structure will inevitably reduce fuel capacity and increase fuel consumption. The Aerosonde lands conventionally by belly skidding, while the ScanEagle utilizes a sophisticated autonomous recovery device.

Figure 3-1, Figure 3-2 and Figure 3-3 show popular UAS ScanEagle, Aerosonde and MLB BAT respectively. Each aircraft utilizes a specialized mechanism to enable reliable take-off.

Another commercially successful small UAV is the MLB Bat [13] shown below. Again this UAV uses a catapult launch system, but nevertheless features a rugged undercarriage for conventional ground landing.
Also notable about all these aircraft is the use of fixed pitch propellers. Clearly the fixed pitch propeller is the most simple, inexpensive and reliable type, but it also appears to have satisfactory performance and efficiency for these aircraft.
It is clear that many popular small UAVs today utilize fixed pitch propellers, onboard generators, and auxiliary takeoff and recovery mechanisms. None of these examples yet operate with Hybrid powerplants. Investigation should yield quantitative and qualitative information about the respective compromises. Means to optimize aircraft performance across a broader operational capability may be found.

### 3.7 Hybrid Vehicles

The definition of a hybrid powerplant is the installation of at least two separate source energy conversion paths within a vehicle, leading to a common final output path. Surface based hybrid vehicles have existed in various forms throughout transport history, and continue to be prevalent. Early examples in water vessels include combination sail and steam engine types [14] such as “The Rising Star” from 1822. Series hybrid configurations are the norm for applications as diverse as rail transport to mining equipment, and submarines. There must be good reasons for introducing what may appear to be more complex systems to power transport equipment. The two main types of hybrid architecture were discussed in the introduction, the Parallel, Figure 2-1 and the Series, Figure 2-2.

In the earliest example above, the hybrid steam sailing ship, the reasons include the unreliability and inefficiency of early steam-plant and water propulsion mechanism development. The benefit of utilizing the engine was to enable navigation irrespective of prevailing wind, however the wind sail propulsion methods were highly developed and clearly required no onboard fuel supply, so a combination increased the overall effectiveness. Later in the development of marine transport, ICE engine and water propulsion systems became efficient and reliable enough that the advantages of wind power no longer were dominant over the disadvantages and sails were removed entirely. It is interesting that changes in the overall energy supply and cost structure of international shipping is beginning to see the re-emergence of ICE Sail hybrids [15].

The use of hybrid powerplant in modern rail transport is practically ubiquitous. The diesel electric locomotive has not been superseded. This unit can transform the energy of the liquid fuel into very precisely controlled output power across a very broad torque and speed range with a minimum mechanical transmission complexity. The
other main form of rail locomotive power is via overhead or third-rail electrical delivery. This is still a hybrid system with the ICE engine plant remotely located.

Energy cost and availability constraints have more recently begun to affect smaller scale surface transport such as automobiles.[16] For a certain performance requirement, when the extra cost and complexity of hybrid powerplant can be balanced by fuel use reduction in an appropriate time span, a hybrid vehicle will sell.

So where and why are hybrid systems inherently more efficient and effective than singular systems? Many transport systems utilize singular powerplant systems. Most notably here, the dominant aircraft propulsion systems are singular ICE, with vast amounts of money spent on improving aircraft efficiency. This thesis will show that hybridization can yield significant benefits.

There are general and specific parameters related to hybrid benefits. Clearly a parallel hybrid system can offer redundancy in the case of failure of one energy path, or the option of utilizing only one path for some benefit, as in the case of hybrid sail ICE ships. A series hybrid might reduce the mechanical complexity of the transmission system as in the case of rail locomotives.

3.8 Hybrid UAS

Of the limited literature available in this field, Frederick G. Harmon [17] appears to be most prolific with a doctoral thesis and several papers to his credit. Harmon has concluded “Parallel hybrid-electric propulsion systems would be beneficial for small unmanned aerial vehicles (UAVs) used for military, homeland security, and disaster-monitoring missions.” His research developed system control algorithms and techniques to optimize energy use in thrust generation from parallel hybrid configurations. A sophisticated computer modeling environment was used to characterize the system components resulting in a very comprehensive computerized system model. Propeller output parameters were determined using some basic empirical input coefficients on well-known derived relationships. This technique indicates the usefulness of developing techniques and equipment to supply and validate such input coefficients. The thrust and torque coefficients are inherently approximate, a single value must represent the combined effects a multitude of variations. The error due to these variations can be reduced by discretizing the
coefficients across smaller increments of their relevant functions. Harmon’s computer propeller model includes coefficient look-up tables which would address this issue. Harmon recommends some future research be directed at developing powerplant dynamometry and specifically within wind tunnels. Also he notes the desirability of utilizing variable pitch propellers to maintain best efficiency at different flight conditions, and that particular aircraft model used in his analysis had an advance ratio which was “similar at cruise and endurance speed”. Given the lack of availability of variable pitch propellers for small UAVs a simplified technique for sizing fixed pitch propellers for particular powerplant and mission scenarios would be advantageous.

Harmon was able to show that the use of parallel hybrid powerplant configurations can lead to energy use reductions of between 54% and 22% in realistic UAS operating scenarios. This reduction in energy requirement would directly translate to increased payload range or endurance. If such significant improvements are fundamentally available by implementing hybrid powerplants, a simplified method of selecting or modifying the required system components on existing UAVs would be advantageous.

Bental Industries [18] has recently announced a range of “hybrid” UAV motors and electronic controllers.
Bental advertise the advantages of the integrated starter and generator functions of the unit and “Leveraging the Operational Advantages of Electric Motor and Internal Combusting Engine”, also that “Benefits of this advanced hybrid motor include fuel savings, quiet operation, reduced weight and size, and high survivability due to the low thermal and noise signatures during mission implementation.” A particular UAS operator would need a method to determine the best combination of the available motor units with ICE and propeller units for their airframe and intended mission. A simplified method for sizing appropriate ICE and propeller combinations and estimating overall system performance and capability will be highly advantageous.

### 3.9 Summary

The use of hybrid drive propulsion systems for other forms of transport is common. The use of hybrid powerplant is gaining acceptance in UAS. The usefulness and potential for net increase in effectiveness and productivity of the system should be investigated.
4 Theoretical Evaluation

The items of performance involving Aircraft Mechanics of Flight, Fluid Propellers, ICE, EMs, and UAS operations can be modeled mathematically. Notable variables and relationships associated with the aforementioned individual items can be derived and put together as a system as demonstrated in this chapter.

4.1 Airframe Mechanics

Fundamental characteristics are determined from the interaction of the solid body moving through a fluid and under the influence of gravity and thrust. The aircraft motion through the air results in a relative velocity, i.e., relative airflow. In this work, the relative airflow will always be assumed to be normal to the propeller disk area. There are errors associated with this assumption, but they are most often considered insignificant compared to the resulting aerodynamic properties and are neglected. The forces arising from the fluid (air) interacting with the solids are conventionally analyzed as defined by Lift at 90 degrees with respect to remote free-stream fluid velocity. Drag is defined as the force generated in line with the free-stream. Weight acts at the mass centroid of the airframe and contents, directed toward the earth center. Thrust acts normal to the plane of rotation of the propeller, and therefore as defined above, in line with the flight direction, opposite drag.

![Figure 4-1 Basic Aerodynamic Forces](image-url)
All the forces acting on the airframe are subject to variation, however for the present purposes, the weight will be considered constant. For steady state conditions, all forces must sum to zero therefore different airspeeds, climb and descent angles require different combinations of Lift, Drag and Thrust. The Lift and Drag are interdependent, while the thrust can be set independently. The Lift Drag relationship is defined by the physical shape of the airframe, and is a function of airspeed and climb or descent angle. As discussed later, the Lift/Drag ratio for any given airframe geometrical configuration can usually be considered a function of airspeed only. The drag is composed from two separate origins. Induced Drag directly due to the production of Lift, and Parasite Drag arising from all other sources. For straight level un-accelerated flight, the Lift is equal to the Weight and therefore constant, however speed changes alter the efficiency of the lift production and hence the associated drag. Higher speeds lead to reduced Induced Drag and Increased Parasite Drag, leading to the characteristic Total Drag curve shown below. Given that the Lift is constant, the Total Drag curve also represents the Lift/Drag ratio as a function of airspeed.

Figure 4-2 Drag Characteristics

Assumed values for this Characteristic Total Drag will be used in later analysis.

For un-accelerated level flight the Thrust must equal the Drag, and so the Total Drag curve is equivalent to the Thrust Required Curve as shown below. The Thrust Available curve can be plotted also to show the difference or Reserve Thrust.
The theoretical evaluation of thrust and drag characteristics involves understanding the airspeed at which the maximum Reserve Thrust is achieved. This airspeed is the speed at which the maximum climb angle can be obtained. Where the thrust required equals the thrust available, this is the maximum attainable steady state level flight airspeed.

Power flowing to the airframe from the powerplant is realized as Thrust Power, which is the product of the normal component of force on the propeller disk and the flight speed in that direction. The output thrust power per input engine power is propeller efficiency.

The maximum thrust power for the range of airspeeds is plotted as Propulsion Power Available. Drag occurring on the airframe multiplied by the airspeed equates to Airframe Power Required.

**Figure 4-3 Thrust and Drag Characteristics**

The airspeed at which there is maximum Reserve Thrust is the speed at which the maximum climb angle can be obtained. Where the thrust required equals the thrust available is the maximum attainable steady state level flight airspeed.
Figure 4-4 Airframe and Propulsion Power Characteristics

The difference between the power required and power available is Reserve Power. The airspeed at which maximum reserve power occurs is the airspeed at which maximum climb rate can occur. The minimum and maximum attainable un-accelerated flight speed corresponds to the point at which the curves intersect.

The Power Required curve also indicates the Maximum Endurance speed as well as the Maximum Range speed as shown below. These two speeds are often relatively close.
4.2 Propellers;

Thrust produced is equivalent to the rate of change of momentum of the air mass affected by the propeller along its axis of rotation. The momentum change can be effected either by the magnitude of the acceleration or the magnitude of the mass flow affected. If all other factors are equal, increasing the magnitude of acceleration of the mass-flow results in reduced efficiency while increasing the magnitude of the mass accelerated will increase the efficiency. Inefficiency is manifested by losses from any resultant non-axis accelerations or heating.

The important parameters for propeller performance are subject to dimensional analysis to yield fundamental relationships as follows;

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Symbol</th>
<th>Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>propeller diameter</td>
<td>D</td>
<td>m</td>
</tr>
<tr>
<td>propeller speed</td>
<td>N</td>
<td>rev/s</td>
</tr>
<tr>
<td>torque</td>
<td>Q</td>
<td>Nm</td>
</tr>
<tr>
<td>thrust</td>
<td>T</td>
<td>N</td>
</tr>
<tr>
<td>fluid density</td>
<td>p</td>
<td>kg/m³</td>
</tr>
<tr>
<td>fluid viscosity</td>
<td>ν</td>
<td>m³/s</td>
</tr>
<tr>
<td>fluid bulk elasticity modulus</td>
<td>K</td>
<td>N/m²</td>
</tr>
<tr>
<td>flight velocity</td>
<td>V</td>
<td>m/s</td>
</tr>
</tbody>
</table>
4.3 **Thrust, $T$**;

$T = k_T \rho n^2 D^4$, where $k_T$ is called the thrust coefficient and in general is a function of propeller design, Reynolds Number $R_e$, Mach Number at blade tip, $M_{tip}$ and Advance Ratio, $J$.

4.4 **Torque, $Q$**;

$Q = k_Q \rho n^2 D^5$, where $k_Q$ is called the torque coefficient and in general is a function of propeller design, $R_e$, $M_{tip}$ and $J$.

The factors $k_T$ and $k_Q$ are specific to particular propeller designs, shape, size and operating conditions.

4.5 **Advance Ratio, $J$**;

$J = \frac{V}{nD}$

4.6 **Efficiency,**

The power supplied to the propeller is $P_{in}$ where

$$P_{in} = 2\pi nQ$$

The useful power output is $P_{out}$ where

$$P_{out} = TV$$

Therefore the efficiency is given by;

$$\eta = \frac{TV}{(2\pi)nQ} = \frac{k_T\rho n^2 D^4 u_0}{k_Q \rho n^2 D^5 (2\pi)} = \frac{1}{2\pi} \frac{k_T}{k_Q} J$$

Where $u_0$ = remote free stream velocity, taken as $V$, flight velocity.

Two means of increasing the mass-flow rate of air affected by the propeller are the diameter of the propeller and the airspeed of the aeroplane. The combination of mass-flow effects from varying either the forward speed of the aircraft, the propeller diameter and angular rate is contained in the factor $J$, the advance ratio. For any given propeller geometry, the efficiency is a function of advance ratio and peaks at one particular value. This is a result of the variation of the ratio $\frac{k_T}{k_Q}$ which involves blade
angle of attack and geometric pitch optimum angles. A typical plot of efficiency versus advance ratio for different pitch settings is shown below.

![Figure 4-6 Propeller Efficiency](www.aoe.vt.edu)

This shows that any fixed configuration of propeller can exhibit maximum efficiency of around 85% only at one specific advance ratio. In order to maintain high efficiency at different aircraft flying speeds or engine speeds, the propeller geometry must change, typically this involves adjusting the blade pitch angle. As seen above, high efficiency can then be maintained across a broad range of flying and engine speed combinations, $V$ and $n$. The availability and use of variable pitch propellers in small UAVs is however extremely rare. Thus, as for many light and ultra-light aircraft, a fixed pitch propeller is used and selected on the basis of optimising efficiency at a particular operating point. The airframe and powerplant characteristics define the propeller selection. As discussed elsewhere, a particular airframe will exhibit certain performance efficiency outcomes depending on its speed and weight. The designer may choose to match the speed for maximum airframe range or endurance to a propeller, then take-off and climb performance will be compromised.

Propulsive efficiency is not always the primary requirement. Irrespective of efficiency, thrust requirements must be provided to suit the operationally necessary aircraft performance.

Propeller diameter may be limited by structural or practical operational limits such as ground clearance.
4.6.1 Propeller Performance Analytical Estimation

The key performance outcomes of propellers are the Thrust produced and the Power Required to produce that thrust at various forward speeds. Computational and Analytical modeling techniques for predicting these parameters are available. Computerized Computational Fluid Dynamics processing tools, and Goldstein Vortex theoretical prediction methods are two techniques in common use.

The equations and relationships noted above require input values which depend on the specific dimensions and shape of a given propeller. The accuracy of techniques for theoretically predicting performance is highly dependent on input data and assumptions. For the purposes of this work, it was decided to utilize empirical measurement and interpolation methods rather than analytical estimation.

4.7 Engines:

The engines under consideration are liquid fuelled reciprocating piston types. The engine used in experimental testing and validation is a glow plug assisted compression ignition 2-stroke cycle unit. The general power and efficiency characteristics for such engines are plotted either on a Torque versus Output Shaft Angular Rate (RPM) or Power versus RPM plane. The nature of the fluid flow and combustion process through cycles leads to the following characteristic curves showing torque and power as a function of RPM with a single maximum. Typical representative plots of torque and power characteristics for such engines are shown in figure 4-7 below.
Theoretical Evaluation

The two curves are always related since Power is a function of Torque and RPM (product of Torque and Angular Velocity). The torque and hence power of this type of engine depends on the ambient air density, therefore the data shown is only valid for a specified ambient condition.

Engine torque and power plots are not easily obtainable for the majority of UAV engines, however occasionally manufacturers do provide them as shown in figure 4-8 below.

Figure 4-7 Engine Torque and Power Characteristics
Figure 4-8 Example Manufacturer Torque and Power Curves

Note that the data extends from 6000 RPM through to 10 000 RPM, showing a clear peak, and that the test conditions are specified. Most often a candidate engine will have to be independently tested to yield this data.

The efficiency of the engine is specified in terms of Specific Fuel Consumption (SFC) which is usually specified in terms of kilograms of fuel consumed per kilowatt hour of energy delivered (kg/kWh). SFC can also be plotted on the Torque versus RPM plane to form “Efficiency Islands” [19] pp. 43 as shown in figure 4-9 below.
4.8 Electric Drives

The type of electric motor under consideration is the Brushless Direct Current type. These motors combined with the associated Electronic Speed Controller are functionally equivalent to conventional permanent magnet direct current electric motors. The BLDC motor characteristics can be shown on Torque and Power versus RPM plots similar to those of engine characteristics.
The torque, power and efficiency characteristics for electric motors are more often provided by the manufacturer. Unlike internal combustion engines, the characteristics are not significantly affected by ambient conditions, however the voltage and in some cases the current supply (ESC) needs to be specified.

Figure 4-10 Direct Current Motor Torque Characteristics

BLDC Torque Characteristic
(www.microchip.com)

Figure 4-11 Characteristic Direct Current Motor Torque and Power Characteristic Curves
The efficiency for electric motors is usually quoted as percentage of mechanical output power at the output shaft per input electrical power. A typical plot using manufacturer provided data is shown below.

![Figure 4-12 Characteristic Direct Current Motor Efficiency Characteristic Curve](image)

Again, a clear peak in the efficiency across an RPM range is usual.
5 Experimental Setup and Methodology

Analysis of the propulsion system is directed at answering the research questions, while the ultimate development goal of the hybrid powerplant is installation and use on research and commercial small UAVs, many fundamental parameters need to be isolated, qualified and quantified separately. Measurements of representative specific example engine, motor and propeller properties were made. The interaction of the major components of the powerplant in a particular configuration was measured to validate analysis and prediction techniques. The same process will be used when the powerplant is installed in an aircraft, however at this stage, only theoretical airframe modeling has been conducted.

Can an AHP yield energy efficiency advantages for small UAS?

The primary variables relevant to propulsive efficiency can be measured in laboratory conditions. The power developed by the powerplant was directly measured using a dynamometer. The thrust at various airspeeds was measured using the dynamometer contained within a wind-tunnel. The only significant variable which this system could not simulate is the ambient air density changes which occur with altitude and temperature changes in the aircraft operating environment. Various techniques exist to model the effects of different air densities on the engine and propeller and in combination with the airframe. However, the basic premises for influencing the overall propulsive efficiency can be effectively explored and tested under the ambient laboratory air densities before extending the analysis. This provides an important benchmark, and is quite valid on its own, given that there is no predefined operating requirement apart from the baseline mission parameters.

Can an AHP yield overall effectiveness advantages for small UAS?

This question can be somewhat subjective in nature, since it is partly based on the objectives of human UAS operators and their personal expectations. With respect to laboratory experimental setup, the powerplant frequently had to be installed and operated within the inaccessible environment of the wind-tunnel. The testing process required frequent powerplant starting and shutdown cycles. Several key advantages of operating the hybrid system became quite clear, particularly with reference to the author’s frequent UAS operations in the field using non-hybrid powerplants both ICE
Experimental Setup and Methodology

and EM. Remote hands-free engine starting and the capability of near instant energy replenishment by way of refueling are excellent features for a powerplant.

5.1 Dynamometer

Clearly, data concerning the engine, motor and propeller performance characteristics must be available for analysis of hybridization and airframe matching. The device for measuring these qualities is a Dynamometer. The dynamometer directly measures the forces and torques developed in the powerplant. Measurement and experimental errors are inevitable but a useful set of data across various parameters can be obtained in laboratory conditions. The useful quantities are usually represented by torque or power required or developed versus some rate. The rate concerning engine power is angular speed and fuel flow, while for propellers it is angular speed versus useful thrust power.

The dynamometer is a physical measurement apparatus, in this case capable of measuring Thrust, Torque and Angular rates of motion.

5.1.1 Dynamometer Types

There are two main types of dynamometer, which differ in the way in which the load is presented to the prime mover. As discussed below the torque may be determined using direct reaction measurement, or acceleration. There are advantages and disadvantages for each technique.

5.1.1.1 Reaction Measurement

This technique utilizes the principle that the torque applied to the load by the prime mover will equal the torque applied by the load to the prime mover. Hence either the load torque or the prime mover torque may be measured. In some cases it is convenient to measure the load torque since the load mechanism may be suitably compact and easier to support on low friction bearings than the prime mover. A bearing system which can allow control of the path of the balancing force is always required. The balancing force maintains the orientation of the prime mover with the structure of the dynamometer, there may be a load structure which requires balancing force also. For example, a large engine driving a friction brake load will generate mounting forces on both the engine mount and brake mount in proportion to the
torque developed. Either the engine and or the brake could be supported such that the support forces may be measured. The bearing system must constrain translation and rotation movement in all but the desired measurement axes. This movement is constrained by separate structure through which the force is measurable. Most commonly a spring is used and the force calculated according to the resultant displacement. Alternatively a mass balance system may be used. Increasing use is made of electronic instrumentation techniques to expedite analysis using digital computers. The most convenient tool for generating electronic signals in response to material loading is the Strain Gauge.

5.1.1.2 Acceleration Measurement
This technique is useful for torque measurement in some cases due to its inherent simplicity. No force measurement is required and so the apparatus construction is very simple. A flywheel load is driven by the prime mover shaft. The technique utilizes the principle that the torque applied to the load is not balanced. There is therefore a resulting acceleration of the prime mover output shaft. The only parameter to be measured is the angular velocity of the shaft. This will yield the angular acceleration and in combination with the value of the mass moment of inertia of the flywheel, the torque is determined for each particular speed.

Due the fact that propeller torque and thrust relationships were required in addition to prime mover characteristics, the versatile Reaction Dynamometer type was developed.

5.1.2 Construction
A reaction type dynamometer was designed on the basis of the types of powerplant it was to measure. In this case two internal combustion engines were available to test. The dynamometer must measure applied torque, and simultaneously measure the linear thrust force. Several combination type load cells are available commercially which could be very suitable for this application [20, 21], however cost constraints led to the implementation of a custom built solution. Ultimately the dynamometer has to transfer an applied torque to a measurable force at a radius, as well as the linear thrust force. The technique of measurement was decided to be strain gauged beams, therefore the beams had to be loaded by application of a force system. The beams
could act as a Simple Beam, Cantilevered Beam or other configurations. The torque beam must constrain rotational movement of the spindle with a few degrees such that there is not excessive movement of the engine. The applied torque will vary from clockwise to anticlockwise depending on the engine rotation direction and also on the externally applied starting torque. Construction simplicity indicated the use of a fully cantilevered beam. This type of beam has the advantage of simplified mounting arrangement without affecting the bending characteristic, and forces can be applied from either direction. The same arguments applied for the thrust beam.

The spindle necessarily has to support the overhung load of the engine mass. The bearings are not required to operate in the usual manner of continuous rotation so conventional bearing selection techniques were not applicable. The bearings were selected on the basis of the spindle diameter and cost. The spindle was sized according to the applied moments with a generous safety factor. The entire chassis which supports the spindle was mounted on a low friction linear slide bearing to allow deflection of the thrust strain beam.

The mass and size of the dynamometer are not particularly critical for this application, however the inertial moments of the spindle and the friction developed in the bearings were kept relatively low. Other important design factors were the availability of materials and the convenience of transporting the unit.

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![Figure 5-1 Dynamometer Schematic Diagram](Image)
5.1.2.1 Load Cell Torque beam
Having selected a cantilever beam configuration, the beam dimensions could be
determined on the basis of the forces to be applied at the end of the beams. The
magnitude of the torque beam force depends on the radius of application from the
center of the spindle as well as the torque applied, while the magnitude of the applied
thrust force is simply the maximum predicted propeller thrust. A suitable deflection of
each beam to the applied loads is necessary for best strain gauge response.

Torque characteristics of several different propellers, were sought. In order to drive
the larger type propellers to high speed, the most convenient method was to use a
suitable engine. The larger propellers required an engine significantly larger than the
basic 10cc unit selected to comprise the experimental hybrid powerplant, and as
explained below, this requirement influenced some aspects of the design.

The basic layout and important parameters of the torque strain beam are shown below.
Dimensions L and b were set by practical considerations of the overall dynamometer
construction, the thickness, t, had to be determined.

![Figure 5-2 Load Cell Beam](image)

The well known beam bending equation derived from mechanics theory can be
applied.

$$\frac{M}{I} = \frac{\sigma}{E} = \frac{E}{R}$$

Where M = the applied moment

I = the Second moment of inertia for the section
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\[ \sigma = \text{the stress at position } y \]
\[ y = \text{the distance from axis } x \]
\[ E = \text{the Young’s modulus} \]
\[ R = \text{radius of curvature} \]

The variable of interest is the maximum stress applied to any beam material. It is required that the beam not yield in service. Permanent deformation would cause several problems. The angular position of the engine setup would change and the related excessive strain would damage the strain gauge or at least alter the calibration. Therefore the beam should be designed such that the maximum applied stress due to expected normal engine torque will not exceed the yield stress of the material.

The two requisite conditions, moderate strain and high yield strength indicate the use of “spring” steel. A high-carbon steel which has been hardened and tempered. AISI 4340 grade alloy steel with yield strength approximately 417 MPa was used.

The yield strength of 417 MPa is significantly higher than most mild steels which usually yield at around 200MPa, while the Elastic (Young’s) modulus for all steels is similar at around 200GPa.

Having selected this material, and the dimensions L and b, the thickness, t, had to be decided.

From the bending equation above,

\[ \sigma_{\text{max}} = \frac{M y_{\text{max}}}{I} \]

And,

\[ y_{\text{max}} = \frac{t}{2} \]

Also the second moment of inertia, I for a rectangular section is,
\[ I = \frac{bt^3}{12} \]

And from the diagram below the point force \( P \), can be related to the torque \( T \).

![Figure 5-3 Torque beam Loading Scheme](image)

\[ T = hP \]

Then,

\[ P = \frac{T}{h} \]

The maximum moment applied to the beam will exist where the beam is constrained at the beam mounting block, distance \( L \) from the force \( P \).

Then,

\[ M_{\text{max}} = PL \]

Putting all this together,
Experimental Setup and Methodology

\[
\sigma_{\text{max}} = \frac{12TL}{2hbt^3}
\]

Or

\[
t = \sqrt[3]{\frac{TL6}{\sigma_{\text{max}}hb}}
\]

The expected engine torque was determined by inspection of the manufacturer’s power data. The two engines to be tested had quoted power figures as below.

**Zenoah, 3kW @ 7000 RPM**

**OS, 2kW @ 15000 RPM**

Therefore the Zenoah torque was expected to be around

\[
\frac{60 \times 3000}{2 \times \pi \times 7000} \text{Nm} = 4 \text{Nm}
\]

The expected torque from the OS engine,

\[
\frac{60 \times 2000}{2 \times \pi \times 15000} \text{Nm} = 1.3 \text{Nm}
\]

Given that \(\sigma_{\text{max}}\) must be less than or equal to 400MPa, and for

\[
L = 0.04 \text{m}
\]

\[
h = 0.04 \text{m}
\]

\[
b = 0.015 \text{m}
\]

\[
T = 4.0 \text{Nm}
\]

\[
t_{\min} = \sqrt[3]{\frac{4 \times 0.04 \times 6}{400 \times 10^3 \times 0.04 \times 0.015}} = 0.002 \text{ m}
\]

Thus, the minimum thickness \(t\), had to be approximately 0.002 meters.
The expected strain, is a function of the applied stress and the modulus of elasticity.

\[ \varepsilon = \frac{\sigma}{E} \]

The modulus of elasticity is constant, however the stress varies with the applied moment which clearly varies along the beam. So the strain will vary according to the position with the maximum occurring immediately adjacent to the beam constraint blocks. The strain gauges were mounted allowing a few millimetres clearance, so in order to calculate the average strain on the gauge, the distance from the load point to the middle of the gauge could be used. However the actual strain occurring is not directly relevant for the purposes of torque measurement. Once the dynamometer is constructed a calibration using known applied torque and measuring the resulting strain is possible to yield a constant. Then for any measured strain, the applied torque can be inferred.

An approximate strain near the gauge is worth calculating as a reference, and to ensure the strain gauge will be operating within its recommended range.

Using the above representative value of 4Nm applied torque, imposed on the 2mm thickness beam, a maximum stress of 400 MPa, then Strain;

\[ \varepsilon = \frac{\sigma}{E} \]

\[ \varepsilon = \frac{400 \times 10^6}{200 \times 10^9} \]

\[ \varepsilon = 2000 \, \mu \varepsilon \]

So the expected maximum strain is within the recommended range specified for the gauge.

Also of interest is the expected deflection of the beam under the applied load. For a built-in full cantilever beam the maximum deflection can be given by;

\[ D_{\text{max}} = \frac{PL^3}{3EI} \]

Again the second moment of inertia for this beam is;
Experimental Setup and Methodology

\[ I = \frac{bt^3}{12} \]

Then,

\[ D_{\text{max}} = \frac{4PL^3}{Ebt^5} \]

\[ D_{\text{max}} = \frac{(100)(0.000064)}{(200)(10^3)(0.015)(7)(10^{-5})} \]

\[ D_{\text{max}} = 3\text{mm} \]

This is the maximum deflection allowable before the beam will yield. Knowledge of this limit became very useful when in practice the torque required to start a large Zenoah brand engine was well in excess of 4Nm. The starting torque is composed of the effects of inertia, friction as well as the cylinder compression. Also it was evident that angular oscillation in combination with the spring loading effect of the strain beam gave rise to torsional vibrations which amplified the load on the beam. Consequently the yield stress was exceeded while starting the Zenoah and the load cell suffered permanent deformation and damage to the strain gauges.

Figure 5-4 Dynamometer Torque Load Cell
The solution to this problem was the modification of the dynamometer to include adjustable limiting stops which prevented excess movement of the spindle. These stops were adjusted such that the maximum deflection of the beam was limited to 2.5mm which turned out in practice to be sufficient to allow full and free deflection under the applied engine running load. However, initial conditions when starting the Zenoah could not be measured.
5.1.2.2 Deflection Verification

To check the veracity of the above calculations a measurement of the deflection due to a specified applied torque was carried out. A known mass of 400 grams was suspended on a beam connected to the spindle of the dynamometer at a horizontal distance of 0.25m from the spindle axis. Thus a torque of \((0.4) \times (9.81) \times (0.25) = 0.981\text{Nm}\) was applied.

\[
T = hP
\]

And

\[
P = \frac{T}{h}
\]

So the Force applied to the beam,

\[
P = \begin{pmatrix} 0.981 \\ 0.04 \end{pmatrix}
\]

\[P = 24.525 \text{ Newton}\]
Experimental Setup and Methodology

\[ D_{\text{max}} = \frac{4PL^3}{Ebt^3} \]

\[ D_{\text{max}} = \frac{(4)(24.525)(0.04)^3}{(200 \times 10^3)(0.015)(0.002)^3} \]

\[ D_{\text{max}} = 0.000262 \text{ meters, or 0.262mm, according to theory.} \]

The actual deflection of the beam was measured using a dial indicator.

![Beam Deflection Verification](image)

**Figure 5-7 Beam Deflection Verification**

The measured deflection was 0.295mm, indicating good agreement with the theoretical value.

### 5.1.3 Load Cell Thrust Beam

A similar design process was followed as for the torque beam, however the value of maximum thrust was simply measured from an aircraft in the field using a spring balance. The maximum thrust applicable was determined as 200N. For convenience, the same strain beam as used for torque measurements was desirable, so again;
Experimental Setup and Methodology

\[ t_{\text{min}} = \sqrt[6]{\frac{PL}{\sigma_{\text{max}} b}} \]

Where P=200

\[ t_{\text{min}} = 2.78 \text{mm} \]

So, the same physical size strain beam was suitable for both torque measurement and thrust measurement under the given geometry.

5.1.4 Dynamometer accuracy and precision

The torque measurement was calibrated under static conditions using known mass weights suspended from an arm.

The weights provided a known amount of torque to the spindle. The strain was measured at discrete increments of weight and the gradient of the resulting graph became a calibration constant for converting measured strain into applied torque.

Figure 5-8 Dynamometer Calibration Rig

The first beam trialed using this method was made from an unknown grade of stainless steel. The thickness was 3mm but the yield strength and other properties were unknown. Early in the calibration process it was noted that the strain reading
was not returning to near zero upon removal of the load. Either the beam was suffering permanent deformation at relatively low stress, there was a problem with the bonding of the strain gauge, or the particular material properties were not completely elastic. The Strain Gauge amplifier function of the Yokogawa DL750 enables a rebalance of the gauges at any time to return the output to zero. The effect of such a rebalance part way through the calibration attempt for the stainless steel gauge can be seen in the following plot.

![Figure 5-9 Unsuitable Load Cell Calibration](image)

Clearly this load cell would be entirely unsuitable as gross errors in strain measurement as well as calibration would occur.

The 4340 grade steel beam was trialed next. This load cell exhibited far superior elastic properties and consistently returned to within 20 microstrain of the balanced condition.

<table>
<thead>
<tr>
<th>Table 5-1 Load Cell Torque Calibration Data</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass [grams]</td>
</tr>
<tr>
<td>-----------------</td>
</tr>
<tr>
<td>100</td>
</tr>
<tr>
<td>150</td>
</tr>
</tbody>
</table>
The calibration graph for this load cell is shown below.

<table>
<thead>
<tr>
<th>Load (kg)</th>
<th>Voltage (V)</th>
<th>Error (µV)</th>
</tr>
</thead>
<tbody>
<tr>
<td>200</td>
<td>415</td>
<td>-403</td>
</tr>
<tr>
<td>250</td>
<td>525</td>
<td>-515</td>
</tr>
<tr>
<td>300</td>
<td>624</td>
<td>-614</td>
</tr>
<tr>
<td>350</td>
<td>730</td>
<td>-724</td>
</tr>
<tr>
<td>400</td>
<td>830</td>
<td>-828</td>
</tr>
<tr>
<td>450</td>
<td>934</td>
<td>-919</td>
</tr>
<tr>
<td>500</td>
<td>1051</td>
<td>-1024</td>
</tr>
<tr>
<td>550</td>
<td>1143</td>
<td>-1125</td>
</tr>
<tr>
<td>600</td>
<td>1251</td>
<td>-1220</td>
</tr>
<tr>
<td>650</td>
<td>1345</td>
<td>-1320</td>
</tr>
<tr>
<td>700</td>
<td>1455</td>
<td>-1416</td>
</tr>
<tr>
<td>750</td>
<td>1570</td>
<td>-1510</td>
</tr>
<tr>
<td>800</td>
<td>1662</td>
<td>-1600</td>
</tr>
<tr>
<td><strong>850</strong></td>
<td><strong>1772</strong></td>
<td><strong>-1715</strong></td>
</tr>
</tbody>
</table>
Figure 5-10 Good Torque Load Cell Calibration

The desirable features for the load cell calibration are a straight line passing through the origin, clearly this load cell worked very well. The gradient of this plot (2.21/1820) Nm per microstrain became the strain constant for all subsequent torque calculations. A calibration check after the dynamometer had been extensively operated showed no significant variation in the gradient.

5.1.4.1 Vibration

The influence of vibration on the accuracy of the torque measurements was not ascertained. Many modes of vibration are present during powerplant operation in the dynamometer. The nature of piston driven ICE units ensures that there will be cyclic variation in torque output even at “constant” power settings. For 2-stroke single cylinder engines of the type under test these variations occur at the frequency of the output shaft speed. The nature of the strain gauge signal conditioning as well as the rotational speed measurement system lead to averaging across multiple revolutions, which is the most useful data for the present purposes. The influence of the high frequency torque variation as well as other vibration modes may be beneficial in
modifying the effects of friction in the load paths from propeller shaft toward the load cells.

5.1.4.2 Independent Power Measurement Comparison
An interesting view of the accuracy and precision of the torque and thrust measurements is seen in the wind turbine experimental results. In this case two completely independent methods could be simultaneously applied to measure the power flowing through the system. The mechanical power delivered to the turbine was measured by the torque applied by a generator to the dynamometer, at various rotation speeds. At the same time, the electrical power developed by the generator was determined from measurements of current flow and terminal voltage. These measurements were made across a range of generator loads and airspeeds, for the purposes of determining wind turbine efficiency and viability as noted elsewhere, however they also constituted an independent verification method.

It is well known that in such a system, there must be a difference between the power absorbed by the turbine, and that delivered by the generator. Unfortunately there is no available truth value for this difference, which is the generator efficiency for the given machine. The magnitude of the discrepancy which should be seen is unknown, but we do know that the generator power should always be less than the absorbed turbine power, and if generator efficiency is assumed as a continuous and narrowly limited function, that under varying speeds and loads the independent power measurements should follow a very similar trend. The exact true function of power absorbed by the turbine with RPM is not known, but a third order polynomial fits well, especially the more precise electrical data. By observing the residual error between the fitted curve and measured data points, for each of the respective measurement techniques, an indicative measure of the dynamometer precision can be assessed.
Experimental Setup and Methodology

Figure 5-11 Dynamometer Torque Measurement Precision and Accuracy
These data are shown below individually with the associated residual plots.

Figure 5-12 Torque Precision Low Load
The Norm of the Residuals for the 20 m/s electrical measurement is 4.16 Watts, the Norm of the Residuals for the mechanical measurement is 9.26 Watts.
Figure 5-13 Torque Precision at Medium Load

The Norm of the Residuals for the 25 m/s electrical measurement is 6.01 Watts, the Norm of the Residuals for the mechanical measurement is 32.4 Watts.

Figure 5-14 Torque Precision at Higher Load

The Norm of the Residuals for the 30 m/s electrical measurement is 22.5 Watts, the Norm of the Residuals for the mechanical measurement is 31.2. Clearly the errors for
all loads may be problematic, but particularly toward the lower power measurements where the errors can be a significant percentage of the true value.

Of particular interest though, is that each curve closely follows the shape of the other at widely varying speeds and loads. The difference in the curves should be composed of generator electromagnetic efficiency and mechanical friction. The zero crossings for electrical measurements were determined under open circuit conditions.

Reasons for such dynamic measurement error, despite the good static calibration throughout the measurement range, may be attributed to unaccounted for airflow effects on the dynamometer apparatus, or possibly the effects of electromagnetic noise on the strain gauge amplification and measurement system.

The dynamometer complete with all generator equipment except propeller was subjected to the wind-tunnel airflow, and torque measurements were recorded to correct the effects of the airflow on the rig. However it cannot be assumed that exactly the same effects will be present when the propeller turbine is operating on an aircraft in the open environment airflow.

It was noted that the electronic frequency control for the windtunnel drive motor did have a significant effect on the Yokagawa Scopeorder, however the noise was high frequency and appeared to average to zero.

### 5.2 Completed Aircraft Hybrid Powerplant and Dynamometer

The experimental powerplant was constructed from the selected prime-movers using a basic spur gear and bracket, then mounted on the dynamometer as shown below.
Experimental Setup and Methodology

Auxiliary equipment necessary for powerplant testing included:

Motor Electronic Speed Controller (ESC)
Radio Control for engine throttle and ESC
Multi-meters for ESC current and voltage measurement
Batteries for motor power
Photo-tachometer for propeller RPM

A full view of the hybrid powerplant experimental rig can be seen in figure 5-16 below.
Experimental Setup and Methodology

5.3 Wind Tunnel

The available wind tunnel had a square test section with sides approximately 450mm. This size enabled the use of propellers of the correct size for the type of powerplant under test, although inevitably there would be undesirable influence between the sidewall of the tunnel and the propeller tips. Also, due to practical constraints, the propeller had to be mounted close to the exit of the tunnel. Analysis of the interference effects due to these problems is beyond the scope of the project, but again the specific implementation of the powerplant in a candidate aircraft is undefined and could lead to similar conditions if for example the propeller were mounted behind fuselage structure or within a shroud. Provided the resulting data is in broad accordance with expected outcomes predicted by other methods or experience, the effect of the errors should not adversely affect the veracity of the experiment with respect to determination of the wider system trends sought. The airspeed was variable up to 40m/s which is representative of the real relevant UAV flight speeds.
5.3.1 Propeller Turbine Experimental Apparatus

The experimental setup for propeller turbine analysis allowed measurement of the input power by the mechanical parameters and the output power by electrical parameters. The mechanical power due to torque and rotational rate absorbed by the propeller from the fluid flow is directly transferred to the generator shaft. The generator is attached to the dynamometer which measures the torque, and also the associated drag. Simultaneously, the electrical output of the generator is measured as voltage across and current through a variable resistive load as shown in the schematic diagram below.
The variable resistor (rheostat) allowed control of the turbine RPM, and resulting current and voltage produced by the generator. Hence the turbine power and drag characteristic could be assessed across a broad range of loads. This method of independent measurement provides a basis for determining gross errors, in conjunction with an estimate of generator efficiency.

5.3.2 Powerplant Experimental Apparatus
The prototype hybrid powerplant and propeller characteristics were also measured in the wind tunnel to obtain dynamic performance data.
Experimental Setup and Methodology

Figure 5-19 Hybrid Powerplant Experimental Test Rig

The Dynamometer setup was identical to that used for the propeller turbine experiments however the electrical measurement system was different. The electrical parameters of interest were the voltage and current applied to the ESC.
The DC current flowing to the ESC could often exceed 50 Amps, therefore a “Current Clamp” type ammeter was used. The combined measurements of Voltage and Current entering the ESC were used to estimate the motor power.

The purpose of this chapter is to explain how the important energy and power properties of the propulsion system were determined.

Energy flow from storage through to thrust power can be modeled using knowledge of energy density in storage mediums and approximate efficiency characteristics of each component in the flow.

The two storage mediums for the parallel hybrid system are battery and liquid fuel. The two energy conversion mechanisms are the electric motor/generator and internal combustion engine.

6.1 Battery:

Various chemistry exists in battery technology which leads to particular energy storage density on both mass and volume basis. The battery types seeing the most use in present use UAVs are Sealed Lead Acid (SLA), Nickel Cadmium (NiCd), Nickel Metal Hydride (NiMH) and Lithium Polymer (LiPo). Each type has different characteristics which may suit different usage requirements aboard a UAV. Different energy flow rates, current draw versus time profiles, lead to different outcomes in terms of useable energy available from batteries, similar characteristics exist in recharging. Also a great many other variables, such as battery age, number of charge/discharge cycles, temperature etc will influence the actual storage efficiency in practice. Since the efficiency of discharging and recharging cycles for batteries is dependent on many complex factors, a simplified model using reasonable assumptions is warranted for the purposes of this study.

The primary variable of interest is the Energy Density. The highest energy density battery type of those mentioned above is the LiPo at approximately 165 Watt hour per kilogram [22]. On the basis that 1 Watt equals 1 Joule per second, 165 Watt hours will be approximately 594 kilojoules, therefore the energy density would be approximately 0.6 MJ/kg. To verify this estimate, data from a representative COTS product is analyzed as follows.

A typical candidate battery in use in small UAVs is the “FlightPower EVO25” made by Energon. This battery is rated at 3700 mAh and is nominally 14.8 Volt. The rating
scheme is usually based on a standard current draw to discharge to a predefined voltage in 1 hour. Assuming a 1 hour discharge at the nominal voltage, the rating implies that the battery should deliver \(3.7 \times 3600 \times 14.8 = 0.197\) MJ. This particular battery weighs 387 grams, giving a rated energy density of 0.509 MJ/kg.

The value of 0.5 MJ/kg will be used as indicative of the energy density of batteries installed on small UAVs. Other mass factors in the system are present and will be included in analyses as for engine system masses.

### 6.2 Liquid Fuel:

The two main fuel types in use for small UAVs are methanol based, and petrol. Methanol engines typically require up to 20% lubrication oil while petrol types generally use around 3%.

The energy density of Methanol is 19.7 MJ/kg and for Petrol 46.9MJ/kg. Nitro-methane is often used in concentrations up to 20% in methanol burning engines as a combustion enhancement additive. The nitro-methane has the effect of improving the power output of the engine by increasing the mass of fuel burnable for a given air/fuel mixture, however it has a lower specific energy than the methanol it replaces, at 11.3 MJ/kg.

**Table 6-1 Methanol Fuel Energy Density**

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Methanol</td>
<td>19.7</td>
<td>70%</td>
<td>13.8</td>
</tr>
<tr>
<td>Nitro-methane</td>
<td>11.3</td>
<td>10%</td>
<td>1.13</td>
</tr>
<tr>
<td>Castor Oil</td>
<td>*Unburnt</td>
<td>20%</td>
<td>0</td>
</tr>
<tr>
<td>Totals</td>
<td></td>
<td>100%</td>
<td>14.92</td>
</tr>
</tbody>
</table>
Table 6-2 Petrol Fuel Energy Density

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Petrol</td>
<td>46.9</td>
<td>96%</td>
<td>45.0</td>
</tr>
<tr>
<td>2-Stroke Oil</td>
<td>*Unburnt</td>
<td>5%</td>
<td>0</td>
</tr>
<tr>
<td>Totals</td>
<td></td>
<td>100%</td>
<td>45.0</td>
</tr>
</tbody>
</table>

After allowing for the typical unburnable lubrication oil and likely presence of Nitro-methane, the value of 15 MJ/kg will be used for Methanol based fuel. Petrol based fuel will be considered to contain 45MJ/kg.

6.3 Engines:
Conversion of liquid fuel energy into mechanical energy available at the output shaft of an engine is a complex thermodynamic process. Maximum limits of efficiency may be determined by thermodynamic analysis techniques, such as Temperature versus Entropy diagrams. Engines of the type used in small UAVs may be analyzed as a Carnot Cycle [23]. Effectively, the upper limit of efficiency is bound by the difference in temperatures achieved between the combustion process and the environment. For the type of engines and operating conditions applicable, the maximum possible cycle efficiency is around 40%. This is the thermodynamic limit for a perfect engine with no mechanical friction, pumping or heat losses. In practice, engines suffer significant losses, do not completely burn all the fuel delivered or allow complete expansion. The efficiency will also depend on the operating speed and load as discussed in chapter 4.7, in this analysis, the engine efficiency will be measured or determined from manufacturer’s data, efficiencies between 5% and 25% are normal.

6.4 Motor:
The efficiency of the electric motor is one of its primary advantages. As for ICEs the efficiency will vary according to the speed and load, but it is typical for electric
motors to yield between 50% and 95% efficiency. The efficiency plot for the representative motor used in this work is shown below using manufacturer data [24].

![Plettenberg Motor Efficiency](image)

**Figure 6-1 Plettenberg Motor Efficiency**

For simplicity, a conservative constant value of 70% will be assumed.

The BLDC type motor is dependent on the ESC, which also has a certain efficiency. The fact that significant losses are incurred by ESC is clearly inferred by the necessity for substantial heat sinks to be incorporated in the structure of the unit. The real efficiency characteristic of ESCs suitable for small UAVs is not usually published. It will be assumed that a constant value of 70% is likely.

The combined efficiency of the Motor and ESC will therefore be conservatively considered as 50%.

### 6.4.1 Internal Combustion versus Electrical Energy Density Comparison

On the basis of the above analysis, the average energy flow density from batteries and fuel to the final output shaft of the powerplant can be summarized thus;

**Engine:**

On the basis of relatively poor engine efficiency, $\eta_{\text{engine}} = 10\%$, (see Chapter 8.5) the converted energy density of methanol fuel equates to 1.5 MJ/kg. For a highly efficient engine, $\eta_{\text{engine}} = 25\%$ using petrol fuel, the converted energy density is 11.25 MJ/kg.
Battery:

Using the efficiency estimates above, and LiPo battery data the converted energy density for electric systems is 0.25 MJ/kg.

This illustrates the well known advantage of utilizing even the lower energy grade liquid fuel over the best available electrical energy storage. Techniques and analysis of methods for comparing the aircraft performance outcomes of these storage and conversion systems is presented in Chapter 8.

6.5 Electrical Energy from Engine Power.

Electrical energy can be derived from engine power and used directly, or used to replenish battery state of charge.

6.5.1 Generator Efficiency.

Using approximately 75% as for Motor.

Power Conditioner, again 75%

So around 56% of the energy delivered to the generator should be available for use directly by electrical services. From the above fuel and engine conversion parameters, electrical energy sourced from fuel will yield 0.84 MJ/kg, versus 0.25 MJ/kg for carrying battery.

The decision whether to carry more battery or more fuel plus generator system, has in part, to be based on the weight of the generator system versus the quantity and availability characteristic of electrical and propulsive energy needed for the flight. The mass of the generator system will have an equivalent in battery energy storage. In the present case, the Plettenberg motor and ESC weigh 0.215kg and 0.115kg respectively, giving a combined mass of 0.33 kg. This is equivalent to the 3700mAh LiPo. The fuel required to give the same energy as contained in this LiPo would be 0.197 / (1.5 x 0.56) = 0.235 kg.

Using the fuel energy to replace battery storage energy will be achieved with further losses. Battery charging losses will depend again on many complex factors such as the particular electronics, rates and battery chemistry, an assumed conservative figure of 25% could be used. Then by the time the energy manifests as battery state of charge, around 40% of the energy delivered by the engine to the generator will remain.
Hence, the process of using the engine to directly supply electrical services yields better efficiency than when restoring battery state of charge. The hybrid powerplant discussed in this work relies on some amount of battery depletion and therefore recharging. However the fundamental premise which is developed does not require significant analysis of battery charge states. Continuous or frequent discharge and recharge cycles due to aircraft mission profiles which are enabled by the hybrid powerplant have been explored and analyzed in other work [25].
7 Experimental Results

7.1 Engine Torque Curves

The first important characteristic which must be determined in developing a hybrid powerplant is the engine torque curve. This curve provides the foundation for analysis of what loads the engine can drive. As discussed in the Experimental Setup chapter, the maximum torque from a selected engine was measured across its design speed range. The resulting data is presented below.

![Torque Curve Mutunuc 10cc](image)

**Figure 7-1 Mutunuc Engine Torque Curve**

A few features regarding IC piston engines, and 2-stroke cycle engines in general are notable. At zero speed there is zero torque, the engine must be running at some minimum speed to be capable of providing any torque in excess of what it needs to sustain itself.

The torque rises steadily until a particular speed range when it rises rapidly before levelling to a peak and then declining. The abrupt changes shown after 12500 RPM are likely to be artefacts of the measurement process. The torque measurement process utilised for the Mutunic engine involved applying an external braking load using a handheld friction brake. The engine was equipped with a base load in the form of a significantly undersized propeller. This allowed the engine to attain its highest recommended RPM while in static conditions on the dynamometer. The hand held friction device was then steadily applied to the propeller hub with the engine at full throttle until the rotation ceased.
Experimental Results

The torque can be multiplied by the rotational speed to yield a Power Curve as shown below.

![Power Curve Mutunuc 10cc](image)

**Figure 7-2 Mutunuc Engine Power Curve**

The power curve again shows a distinct characteristic peak, in this case around 11 000 RPM.

The maximum torque and hence power output of an engine will vary according to several conditions. The ambient air density is the most fundamental, but other items including the fuel quality and mixture tuning, ignition system and the general mechanical condition of the engine will influence the magnitudes and resulting shape of the torque curve. Without detailed analysis and or testing across a wide range of these variables it will be assumed that the ambient air density is the only factor which will change the maximum torque available and that torque will vary in an inverse proportional relationship with density altitude [26].

The curves shown above were measured with approximately 10 000 data points using a Mutunuc 10cc 2-stroke methanol glow-plug engine. Subsequent development of the hybrid powerplant utilized a newer OS FX series 10cc 2-stroke methanol glow-plug engine. This change was made because the OS engines were current production units and were being used in the working UAS at the time of study. The technique employed to measure the torque across the full range of engine speed was not practical to determine the actual torque available once the engine was in the
Experimental Results

The presence of the gearing system and EM will reduce the torque available due to friction losses, and restrict the maximum operating speed. Also, the specific nature of the curve below the practical operating range is inconsequential provided there is sufficient power to run the loads up to speed, and this is very obvious in practice. Therefore a torque curve derived from relatively few data points, but using the measured performance of the hybrid engine as installed in the experimental rig is used for subsequent analysis. A curve is constructed by fitting a polynomial through the restricted OS data points, as shown below and compared to the Mutunuc curve.

![10cc Engine Torque Curves](image)

**Figure 7-3 Measured OS and Mutunic Torque Comparison**

The different shape of the curves is expected since the Mutunuc engine was fitted with a tuned exhaust system. Tuned exhaust systems are used to increase the available torque for a particular engine but are well known to reduce the speed range at which high torque is available. The OS engine was fitted with a standard type exhaust system and is in general designed to provide a wide speed range of high torque to suit the target market of aero-model hobbyists. The Mutunuc torque curve is similar to the hypothetical curve used to illustrate some theoretical analysis for sharply peaking type engines in Chapter 10.

### 7.2 Propeller Characteristic

Three different propellers were tested at two different airspeeds. The propellers ranged from the standard recommended size for the 10cc engine, 12” diameter 6”
pitch (12 x 6) through to a large size known to be an overload and never ordinarily used on the 10cc OS fx engine, a 16” diameter 6” pitch (16 x 6) unit. The intermediate size 14.5 x 8 is sometimes used on the given engine, but is considered the limit.

### 7.2.1 Propeller Torque Curves
As discussed earlier, it is of interest to determine the change in torque required by the propeller for given rotational rates at different forward speeds. The propeller torque was measured at various rotational rates at zero forward speed (Static) and at 30 meters per second forward speed. These speeds represent the initial take-off condition and typical cruise speed of a candidate UAV.

The hybrid engine and motor were used to drive the propeller for data collection. Approximately six data points were measured for each combination of propeller and forward speed. The data points corresponded to increments of throttle opening and electrical boost power. In order to clearly show the characteristics of interest, an assumed polynomial fit was applied to generate continuous curves up through the maximum measured data point.

#### Figure 7-4 Propeller Torque Curves
Several key features are notable from these plots.

- The torque required curve is steeper for larger diameter propellers.
- There is a negative torque applied when the propeller is not turning but is in relative airflow, i.e. windmilling configuration.
• Less torque is required for any RPM when in forward motion, up to a limit. As the RPM increases, the inefficiencies such as compressibility effects at the tips begin to dominate.

7.2.2 Propeller Thrust Curves
The useful output from propellers is thrust. Below are shown the thrust curves for the propellers tested. Again the difference in steepness and offset of the curve according to the airspeed is very distinct.

![Propeller Thrust Curves](image)

Figure 7-5 Propeller Thrust Curves

7.2.3 Propeller Efficiency Curves

![Propeller Efficiency Curves](image)

Figure 7-6 Propeller Efficiency Curves
Relatively few data points were recorded, but there is a clear pattern of declining efficiency beyond an optimum value for J as expected from the theory presented in Chapter 4.

### 7.2.4 Thrust versus Shaft Power Curves

The thrust measurements are presented below;

![Figure 7-7 Propeller Thrust versus Power Curves](image1)

### 7.2.5 Fuel Flow Measurement.

Back to back testing of the 12 x 6 propeller equipped powerplant with the 16 x 6 was conducted to measure fuel flow at maximum engine only power output at 30 m/s cruise conditions.

<table>
<thead>
<tr>
<th>Propeller</th>
<th>Airspeed [m/s]</th>
<th>RPM</th>
<th>Thrust [N]</th>
<th>Fuel Flow Rate [liter/hour]</th>
</tr>
</thead>
<tbody>
<tr>
<td>12 x 6</td>
<td>30</td>
<td>11 000</td>
<td>16.5</td>
<td>1.76</td>
</tr>
<tr>
<td>16 x 6</td>
<td>30</td>
<td>7860</td>
<td>15.7</td>
<td>1.28</td>
</tr>
</tbody>
</table>

Table 7-1 Fuel Consumption Data
Experimental Results

The fuel flow rate using the 16 x 6 propeller on the same engine at the same airspeed as the 12 x 6 was reduced by 27.3% while producing 5% less thrust. The slightly reduced thrust at the same airspeed implies a reduction in Thrust Power delivered also of 5%. The reduced thrust would in practice reduce the maximum flight speed of an aircraft, all other factors being equal, by much less that 5%, since the Thrust and Power Required curves depend on the square of the flight speed.

Hence, the larger propeller is more efficient on this powerplant at 30m/s than the smaller propeller.
8 Results Analysis

8.1 Load Curves Analysis

A main outcome of this research is to verify the effectiveness of Load Curve Analysis techniques in sizing aircraft hybrid powerplant components. The engine and propeller torque curves which are illustrated in the Experimental Results, Chapter 7, have to be combined with electric motor torque curves to yield Combined Torque Available. The combined torque available is plotted with the torque required by the propeller for various conditions, to show clearly the state of the system. Points where torque available equals torque required are referred to as “Operating Points” for the system. In order to achieve this state, the torque available must have been in excess of that required throughout the range of running states up to that point. The magnitude of the excess margin of available versus required torque is proportional to the acceleration of the system.

8.2 Electric Motor Torque Curves

The torque available from the motor used in the experiments was not independently measured. In this case motor manufacturer has provided sufficient data [24] to construct the required curves.

The motor characteristics depend on the applied voltage, several sets of data are published referring to different voltages. The Plettenberg 18 Volt and 24 Volt data are used in the following plots, since these were in the range of applied voltage used in the experiments.
The maximum torque is available at zero RPM, but cannot be used for any extended period as there is high current flow, and very limited cooling. In practice the motor very quickly accelerates until the torque equilibrates with the opposing torque due to turning over the engine and propeller. It must be noted that the motor torque curves presented here represent the maximum output, there are limits to the time which the motor can run at maximum output without overheating. Different motor sizes, internal windings, number of magnetic poles, and output shaft gear ratios change the slope and intercept point of the torque curve, however all DC motors including BLDC, exhibit this same characteristic linear shape. For a given motor, the applied voltage and output shaft gearing may be designed such that the motor output can match different loads. The effect of changing the output gear ratio for a given motor construction is shown below in figure 8-2.
Figure 8-2 Effect of Motor Gearing on Torque

The starting torque can be seen to be reduced, however the usable torque extends to higher shaft speeds.

### 8.3 Combined Torque Available

For the given transmission system, a near 1:1 ratio spur gear set, the final shaft output torque can be assumed to be the algebraic sum of both input prime mover torques. In practice the friction in the transmission system used will decrease the output torque by some amount, but for the present purposes this will be considered negligible. The combined torque available plot is shown below in figure 8-3.

Figure 8-3 Combined Hybrid Torque Curves
8.4 System Curves

Having determined the system required and available torque curves for particular conditions, they may be plotted together to give a complete picture of the hybrid powerplant performance at those conditions.

![Propulsion System Curves](image)

**Figure 8-4 Propulsion System Curves**

Note that Operating Points can occur where ever a torque availability can equal a torque demand. Reducing the engine throttle setting or the ESC electrical current will reduce the respective available torque and the propeller RPM will settle at a lower value. The items of interest here are the Maximum Operating Points at Full Power.
8.4.1 12 x 6 Propeller System Curves

A detailed system curve for the 12 x 6 propeller, 10cc OS fx engine and Plettenberg 220 BLDC motor at sea level test conditions is shown above in figure 8-5.

The intersection of the torque available and torque required curves determines the maximum RPM and hence the maximum thrust which can be expected. Also the fuel flow and electric power for specific part-load and full-load conditions can be determined with reference to the engine and motor efficiencies at those conditions.

It can be seen in this case that the limit of the electric motor torque availability occurs very near the full throttle operating point RPM of the engine for this propeller at static sea level conditions. At 30 m/s the operating point is well beyond the limit of motor torque availability. This implies that there is no potential for boost torque application in this system. If it was desired that the 12 x 6 propeller were used in conjunction with the engine and motor with significant boost potential, a different gearing ratio between the motor and engine/propeller shaft could be used. Using the Load Curve Plotting technique, the effect of different gear ratios can be clearly seen, and desired outcomes can be engineered.

A system curve for the same components as above, but with a 2:1 step up gear ratio between the motor output shaft and the engine/propeller shaft is shown below for comparison.
The limit of measured data for the propeller and engine torque was exceeded at 12000 RPM, however the extrapolated curves show the expected new operating points at significantly higher output speed than possible without the step up gear. This increase in propeller speed would allow significant increase in available thrust at the expense of using electrical power. The zero speed starting torque is also reduced in proportion to the gear ratio, so for this system to be viable for engine starting operation, consideration of starting torque requirements would be necessary.

For original 1:1 gear configuration, the excess torque available at lower RPM indicates that a larger propeller could be used.
8.4.2 14 x 8 Propeller System Curves

Figure 8-7 14x8 Propeller System Curves

These results are discussed further in section 8.5.

8.4.3 16 x 6 Propeller System Curves

Figure 8-8 16x6 Propeller System Curves
The hybrid powerplant constructed for experimentation in this research was developed from components which happened to be conveniently available. Therefore the matching of these components was not based on full prior analysis, and may be suboptimal. The method of determining the optimization is illustrated and can be used to compare outcomes of different prospective system components as set out below.

8.5 **Comparison of 12x6 and 16x6 data.**

Having determined the operating points for the system with two significantly different size propellers, the performance outcomes can be assessed. This comparison is based on the powerplant thrust and efficiency performance at static conditions and at 30 m/s cruise at the same density altitude. In each case the engine is operating at full throttle. For these data points the original measured raw data used to create the above plots will be used.

8.5.1 **12 x 6 Thrust**

Referring to Table 8-1 the RPM for engine only and full boost (maximum combined torque) is 10791 RPM and 10 388 RPM respectively, for Static conditions and 11000 RPM for engine only at 30m/s cruise speed.

<table>
<thead>
<tr>
<th>Airspeed</th>
<th>Engine Only Maxima</th>
<th>Full Boost Maxima</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>RPM</td>
<td>Thrust [N]</td>
</tr>
<tr>
<td>Static</td>
<td>10 791</td>
<td>41.6</td>
</tr>
<tr>
<td>30 m/s</td>
<td>11 000</td>
<td>16.5</td>
</tr>
</tbody>
</table>

*The reduction of thrust encountered upon attempted application of full boost EM via the ESC is an interesting consequence of the motor speed and gear ratio matching. The electric motor was being run above the maximum speed for which the applied voltage could exceed the back emf, and hence it was operating as a generator. See Chapter 11. Further, given that the EM gearing in this case did not allow its application at static conditions, it could not be used for boost at the 30m/s cruise condition with the 12 x 6 propeller.
8.5.2 16 x 6 Thrust

The prototype hybrid powerplant was next trialed with the over-size 16 x 6 propeller, and the results are tabulated below.

### Table 8-2; 16x6 Propeller Thrust Summary

<table>
<thead>
<tr>
<th>Airspeed</th>
<th>Engine Only Maxima</th>
<th>Full Boost Maxima</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>RPM</td>
<td>Thrust [N]</td>
</tr>
<tr>
<td>Static</td>
<td>6360</td>
<td>31.2</td>
</tr>
<tr>
<td>30 m/s</td>
<td>7860</td>
<td>15.7</td>
</tr>
</tbody>
</table>

When fitted with the 16 x 6 propeller, the prototype hybrid powerplant can make full use of EM boost power from static conditions through to cruise speed.

8.5.3 Comparative Propeller Thrust Summary

### Table 8-3; Comparative Propeller Thrust Summary (nil EM boost)

<table>
<thead>
<tr>
<th>Airspeed</th>
<th>12 x 6 Maximum Thrust [N]</th>
<th>16 x 6 Maximum Thrust [N]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Static</td>
<td>41.6</td>
<td>31.2</td>
</tr>
<tr>
<td>30 m/s</td>
<td>16.5</td>
<td>15.7</td>
</tr>
</tbody>
</table>

This data shows that the thrust generated at the 30 m/s cruise condition using ICE power only with the 12 x 6 propeller and the 16 x 6 propeller is within 4.8% of each other. If either powerplant were installed in the same aircraft at the same all-up weight and drag, then that aircraft would cruise at practically the same speed with either powerplant since the thrust is practically the same. Such an aircraft would however suffer serious degradation in take-off performance using the 16 x 6 propeller due to the 25% decrease in static thrust unless the boost power available were used. The static thrust available from utilizing only the ICE power, declines from 41.6 N for the 12 x 6 to 31.2 N for the 16 x 6. It should be noted that the static thrust available is a primary determinant of take-off distance and climb angle [23]. Thus a 25% reduction
in static thrust would restrict the aircraft to using much longer or smoother runways, and could render a successful take-off impossible.

If the full boost power were used, the maximum thrust available of 83.6N at static conditions with the 16 x 6 propeller is clearly in excess of the 41.6N available with the un-boosted 12 x 6 propeller. To compare the static thrust available from a 12 x 6 with boost, it is necessary to consider a more appropriate motor gear ratio such as used in Figure 8-6. Then the Thrust may be estimated by extrapolation of the curve in Figure 7-5. Using these approximations, 12200 RPM would be achieved giving 45 N of static thrust. This modest increase in available 12 x 6 thrust (7%) with the application of significant boost power (50%) is indicative of the reduced efficiency of the smaller diameter propeller in producing static thrust.

### 8.5.4 Cruise Fuel Efficiency

The cruise thrust availability has been shown to be equivalent using either the 12 x 6 or 16 x 6 propeller. For equivalent take-off performance, it is necessary to utilize electric boost power when using the 16 x 6, this involves carrying extra weight, therefore to yield equivalent payload range or payload endurance, the fuel consumption of the 16 x 6 hybrid powerplant must be reduced. Fuel flow measurements were conducted as recorded in Chapter 7.

### 8.5.5 Fuel Mass Density

The fuel used in the experiment was a blend of Methanol, Nitro-methane and Castor Oil. The specific mass of these at standard conditions is shown below [27, 28].

<p>| Table 8-4 Methanol Mass Density |
|-------------------------|----------------|----------------|</p>
<table>
<thead>
<tr>
<th>Fuel Constituent</th>
<th>Mass Density</th>
<th>Fuel Fraction</th>
<th>Mass per Liter of Fuel</th>
</tr>
</thead>
<tbody>
<tr>
<td>Methanol</td>
<td>0.786</td>
<td>70%</td>
<td>0.5502</td>
</tr>
<tr>
<td>Nitro-methane</td>
<td>1.137</td>
<td>10%</td>
<td>0.1137</td>
</tr>
<tr>
<td>Castor Oil</td>
<td>0.956</td>
<td>20%</td>
<td>0.1912</td>
</tr>
<tr>
<td>Totals</td>
<td></td>
<td>100%</td>
<td>0.853</td>
</tr>
</tbody>
</table>
Therefore the fuel mass flow rate at cruise can be calculated and is listed in table 8-5 below;

**Table 8-5 Propeller Specific Thrust and Fuel Flow**

<table>
<thead>
<tr>
<th>Propeller</th>
<th>Airspeed [m/s]</th>
<th>RPM</th>
<th>Thrust [N]</th>
<th>Fuel Flow Rate [liter/hour]</th>
<th>30 m/s Cruise Fuel Flow [kg/hour]</th>
</tr>
</thead>
<tbody>
<tr>
<td>12 x 6</td>
<td>30</td>
<td>11 000</td>
<td>16.5</td>
<td>1.76</td>
<td>1.5</td>
</tr>
<tr>
<td>16 x 6</td>
<td>30</td>
<td>7860</td>
<td>15.7</td>
<td>1.28</td>
<td>1.09</td>
</tr>
</tbody>
</table>

Given the condition that electrical boost power is required for takeoff when using the 16 x 6 propeller, the difference in fuel mass flow rate will favour the hybrid powerplant aircraft only beyond a certain minimum flight time. This minimum flight time is the time at which the fuel mass saving equals the extra installed mass of the hybridization components required for boost power. This and further analysis based on the reduced fuel flow for equivalent thrust is discussed in Chapter 9.

The reduced cruise fuel flow may occur due to improvement in propeller efficiency and or engine efficiency. Some engines display significant changes in efficiency according to the torque and speed setting, while others are less sensitive. The Specific Fuel Consumption (SFC) and Engine Thermodynamic Efficiency (ETE) of the OS fx 10cc engine can be determined for cruise conditions with each propeller as follows.

### 8.5.6 Fuel Energy Density

**Table 8-6 Methanol Fuel Mass Density**

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Methanol</td>
<td>19.7</td>
<td>70%</td>
<td>13.8</td>
</tr>
<tr>
<td>Nitro-methane</td>
<td>11.3</td>
<td>10%</td>
<td>1.13</td>
</tr>
<tr>
<td>Results Analysis</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Castor Oil</th>
<th>*Unburnt</th>
<th>20%</th>
<th>0</th>
</tr>
</thead>
<tbody>
<tr>
<td>Totals</td>
<td>100%</td>
<td>14.92</td>
<td></td>
</tr>
</tbody>
</table>

Table 8-7 Propeller Specific Torque and Power

<table>
<thead>
<tr>
<th>30 m/s sea level cruise</th>
</tr>
</thead>
<tbody>
<tr>
<td>Propeller</td>
</tr>
<tr>
<td>------------</td>
</tr>
<tr>
<td>12 x 6</td>
</tr>
<tr>
<td>16 x 6</td>
</tr>
</tbody>
</table>

Engine SFC is usually quoted in terms of fuel mass consumed per engine work delivered, in the units, kilogram per kilowatt hour, \( \frac{kg}{kW \cdot hr} \)

SFC = Fuel Mass Flow Rate [kg/hour] / Power [kW]

8.5.6.1 Engine Thermodynamic Efficiency using 12 x 6 Propeller

When fitted with the 12 x 6 propeller in 30 m/s cruise conditions the engine produces 710 Watts using 1.5 kg of fuel per hour.

\[
SFC = \frac{1.5}{0.710} \left( \frac{kg}{kW \cdot hr} \right) = 2.12 \left( \frac{kg}{kW \cdot hr} \right)
\]

ETE takes into account the mass specific energy of the fuel.

ETE = Fuel Mass Flow Rate [kg/hour] x Fuel Mass Specific Energy [kJ/kg] / Power [kW]

\[
ETE = \left( \frac{1.5 \cdot 14.9 \times 10^6}{(710) \cdot (3600)} \right) \left( \frac{kg \ Joule \ second}{hour \ kg \ Joule \ second} \right)
\]
8.5.6.2 Engine Thermodynamic Efficiency using 16 x 6 Propeller
When fitted with the 16 x 6 propeller in 30 m/s cruise conditions the engine produces 545 Watts using 1.09 kg of fuel per hour.

\[
SFC = \left( \frac{1.09}{0.545} \right) \frac{kg}{kW \cdot hr} \\
= 2.0 \frac{kg}{kW \cdot hr}
\]

\[
ETE = \left( \frac{(1.09)\times(14.9\times10^6)}{(545)\times(3600)} \right) \frac{kg \ Joule \ second \ hour}{hour \ kg \ Joule \ second} \\
= 8.3\%
\]

8.5.6.3 Efficiency Summary
The change in cruise fuel flow in this case stems mainly from propeller efficiency, the engine efficiency has in fact reduced slightly.

| Table 8-8 Propeller Specific Overall Propulsive Efficiency |
|-----------------|----------------|----------------|----------------|----------------|
| 30 m/s sea level cruise | Propeller Efficiency | Engine Efficiency | Overall Propulsive System Efficiency | Baseline Efficiency Improvement |
| 12 x 6 | 70% | 8.7% | 6.09% | 17% |
| 16 x 6 | 86% | 8.3% | 7.14% | |

It is notable that the overall efficiency of the propulsion system has been improved significantly (17%) despite the fact that the engine efficiency has declined. In the present example, the engine is directly coupled to the propeller and so the operating speeds must be matched. There is no fundamental reason for the engine to be directly
coupled to the propeller other than simplicity, therefore it would be possible to configure the hybrid system to allow the engine to operate in a higher efficiency condition. The outcome of varying the gear ratio between the engine and propeller can be modeled using the Load Curve techniques shown above. Different engines and detail engine tuning outcomes, propellers etc could also be modeled to compare efficiency. Chapter 10 provides exploration of some of these theoretical outcomes.
9 Payload Range and Endurance Modeling

This chapter will develop an appropriate model to compare the range and endurance (Flight Time) of an aircraft fitted with either (i) a conventional engine only powerplant, (ii) engine with generator and (iii) hybrid powerplant system. The propulsion system modeling developed in the previous chapters will be used in conjunction with the efficiency outcome based on the experimental procedure documented in Chapter 8. Some further assumptions and modeling will be developed to define a baseline airframe for comparison.

9.1 Airframe model

The purpose of this analysis is to compare the range or endurance outcomes of varying powerplant configuration, therefore the airframe model need only address some very basic parameters. The propulsion system model has been developed according to a particular thrust output at a particular flight speed. An airframe will always have an optimum flight speed to give either maximum range or endurance as discussed in Chapter 4.1.

Other factors remaining equal, this flight speed is a function of the weight. The weight change with the use of liquid fuel as an energy storage carrier over the use of batteries favours the use of fuel, however as a conservative approach, this benefit will not be considered. Also, the speed for maximum range will differ from the maximum endurance speed, usually this difference is relatively small, and the range speed is affected by the further environmental factor of wind. For the purposes of this analysis, the range will be simply assumed to be equal to the maximum endurance multiplied by the given flight speed.

The flight speed is assumed to be 30 m/s at which there will be total drag of 16 Newtons, since for a UAV with a Lift per Drag ratio of 6:1 (at 30 m/s), this drag corresponds to a flying weight of 96 Newtons, or a system mass of approximately 10 kilograms. These assumed values correspond very closely with known performance of UAVs which the author has operated.
9.1.1 Aircraft Mass Fractions

Given that the aircraft mass will be assumed as 10 kg, suitable definitions for the various mass fractions must be determined as follows.

**Takeoff Weight (TOW)**

The mass of the fully loaded aircraft at commencement of takeoff. As noted above, 10kg will be assumed as the mass of the aircraft for the entire flight. This mass includes all airframe components, fuel, batteries, propulsion and payloads.

**Airframe Basic Empty Weight (ABEW)**

The mass of the airframe including all systems necessary for flight except payload, propulsion fuel and payload batteries. This will be assumed as 4kg. The mass due to flight control system energy requirements will be assumed to be entirely in the form of battery mass and included in ABEW.

**Payload Weight (W\text{pay})**

The mass of payload systems, energy storage and transfer systems necessary to for payload function. In order to quantify the energy storage mass due to the energy requirement of the payload, an assumed power consumption of 50 Watts will be used. The payload energy can be considered to be sourced from either a battery, generator only, or a combination of both. For simplicity in summary, the energy will be considered to come either from a battery or generator, but a linear estimation will be presented for combinations. Also, a generator in a non-hybrid configuration will be considered to weigh less than a Motor/Generator as used in the hybrid powerplant.

**Propulsion Weight (W\text{pro})**

The mass of all propulsion system components including engine, fuel, electric motor, propulsion battery fraction, ESC, and any propulsion related system. Masses of propeller, fuel system components and electrical wiring are considered insignificant.

Known measured values of items in $W_{pro}$ include;

- Engine: 0.7 kg
- Motor/Generator: 0.215 kg
- Generator: 0.10 kg
- Transmission and Mounting: 0.10 kg
- ESC/PowerConditioner: 0.115 kg
Unknown values of items of $W_{\text{pro}}$ include:

- Fuel, $W_{\text{fuel}}$
- Battery, $W_{\text{bat}}$

Hence the following relationships can be defined:

- $W_{\text{pro}} + W_{\text{pay}} + \text{ABEW} = \text{TOW}$
  - $W_{\text{pro}} + W_{\text{pay}} + 4 = 10$ [kg]
  - $W_{\text{pro}} + W_{\text{pay}} = 6$ [kg]

- $W_{\text{gensys}} = \text{Generator} + \text{Transmission and Mounting} + \text{ESC/PowerConditioner}$
  - $W_{\text{gensys}} = 0.1 + 0.1 + 0.115$ [kg]
  - $W_{\text{gensys}} = 0.315$ [kg]

- $W_{\text{hybrid}} = \text{Motor/Generator} + \text{Transmission and Mounting}$
  - $W_{\text{hybrid}} = 0.215 + 0.1 + 0.115$ [kg]
  - $W_{\text{hybrid}} = 0.430$ [kg]

- $W_{\text{fuel}} = W_{\text{fuelPropulsion}} + W_{\text{fuelPayloadPower}}$

### 9.2 Propulsion Weight Comparative Case Studies

Three cases are considered:

i. **Singular ICE operation with no onboard generator,**

   $W_{\text{pro1}} = W_{\text{proIC}} = W_{\text{engine}} + W_{\text{fuel}}$ [kg]
   - $W_{\text{proIC}} = 0.7 + W_{\text{fuel}}$ [kg]

ii. **Singular ICE operation with onboard generator,**

   $W_{\text{pro2}} = W_{\text{proICgen}} = W_{\text{engine}} + W_{\text{gensys}} + W_{\text{fuel}}$ [kg]
   - $W_{\text{proICgen}} = 0.7 + 0.315 + W_{\text{fuel}}$ [kg]
   - $W_{\text{proICgen}} = 1.015 + W_{\text{fuel}}$ [kg]

iii. **Hybrid operation,**
\[ W_{\text{pro3}} \equiv W_{\text{proHybrid}} = W_{\text{engine}} + W_{\text{hybrid}} + W_{\text{bat}} + W_{\text{fuel}} \quad [\text{kg}] \]

\[ \rightarrow W_{\text{proHybrid}} = 0.7 + 0.430 + W_{\text{bat}} + W_{\text{fuel}} \quad [\text{kg}] \]

\[ \rightarrow W_{\text{proHybrid}} = 1.13 + W_{\text{bat}} + W_{\text{fuel}} \quad [\text{kg}] \]

### 9.3 Flight Time Equation

The flight time or endurance of the aircraft will be defined according to the magnitude of storage supply and rate of consumption of relevant energy sources. These figures were developed in Chapter 8.

Fundamentally, \( \text{Power} = \frac{\text{Work}}{\text{Time}} \)

Then,

\[
\text{FlightTime} = \left( \frac{\text{Energy Storage Density} \cdot \text{Energy Storage Mass} \cdot \text{Conversion Efficiency}}{\text{Power Consumption} \cdot (60)} \right) \left( \frac{\text{second} \cdot \text{Joule} \cdot \text{minute} \cdot \text{kilogram}}{\text{Joule} \cdot \text{kilogram} \cdot \text{second}} \right)
\]

\[ \rightarrow \text{[minutes]} \]

The flight time is limited either by the payload energy availability or the propulsion energy availability. It is assumed that the total payload operating time should equal the total propulsion operating time.

#### 9.3.1 Case 1: Singular ICE operation with no onboard generator.

As determined in Chapter 6, noting that ICE only operation with ground take off implies utilizing the 12 x 6 propeller, and experimental outcomes for 12x6 propeller, the following quantities will be used;

Battery Energy Density = 0.50 MJ/kg
Battery Conversion Efficiency = 100%

Fuel Energy Density = 14.9 MJ/kg

Engine Efficiency = 8.7%

Engine Power Required for Propulsion = 710 Watts

Electrical Power Required for Payload = 50 Watts

\[
\text{FlightTime} = \left( \frac{\text{Energy Storage Density} \cdot \text{Energy Storage Mass} \cdot \text{Conversion Efficiency}}{\text{Power Consumption} \cdot (60)} \right)
\]

\[
\text{FlightTime}_{\text{Payload}} = \left( \frac{(0.50 \times 10^6) \cdot (W_{\text{bat}}) \cdot (1)}{50 \cdot (60)} \right)
\]

\[
\rightarrow \text{FlightTime}_{\text{Payload}} = W_{\text{bat}} \times (166) \quad [\text{minutes}]
\]

\[
\text{FlightTime}_{\text{Propulsion}} = \left( \frac{(14.9 \times 10^6) \cdot (W_{\text{fuel}}) \cdot (0.087)}{710 \cdot (60)} \right)
\]

\[
\rightarrow \text{FlightTime}_{\text{Propulsion}} = W_{\text{fuel}} \times (30.4) \quad [\text{minutes}]
\]

9.3.2 Case 2: Singular ICE operation with onboard generator.

In this case the quantity of battery mass for payload power will be variable to explore the resulting effects on endurance. At some point, for heavy payloads and relatively short flights, carrying the generator and associated equipment may be less effective than carrying sufficient battery.

The assumption that generator power will be available rests on the assumption that sufficient engine power above that required for generating the thrust at this condition is satisfied. Conceptually it is possible to retain the propeller RPM and thrust while operating the engine at the higher power, using gearing. A further assumption regarding the efficiency or SFC of the engine at the higher output must be made. It has been found already that the SFC remains fairly constant for this engine through the range considered, with a slight increase toward the higher end. For a conservative estimate, the original 12 x 6 efficiency (8.7%) will be assumed.

The following quantities will be used;
Battery Energy Density = 0.50 MJ/kg  
Battery Conversion Efficiency = 100%  
Fuel Energy Density = 14.9 MJ/kg  
Engine Efficiency (ETE) = 8.7%  
Engine Power Required for Propulsion = 710 Watts  
Electrical Power Required for Payload = 50 Watts

\[
\text{FlightTime} = \left( \frac{\text{Energy Storage Density} \cdot \text{Energy Storage Mass} \cdot \text{Conversion Efficiency}}{\text{Power Consumption} \cdot (60)} \right)
\]

Noting that the Payload Power will be provided by a combination of the battery and the generator, the resultant operating time will be the sum of the times provided by each source. Thus;

\[
\text{FlightTime}_{\text{Payload}} = \left( 0.50 \times 10^6 \right) \cdot W_{\text{bat}} \cdot (1) + \left( 14.9 \times 10^6 \right) \cdot W_{\text{fuelPayloadPower}} \cdot (0.087) \cdot (0.50) \]

\[
\Rightarrow \text{FlightTime}_{\text{Payload}} = (W_{\text{bat}} \times (166)) + W_{\text{fuelPayloadPower}} \times (216) \quad \text{[minutes]}
\]

\[
\text{FlightTime}_{\text{Propulsion}} = \left( 14.9 \times 10^6 \right) \cdot W_{\text{fuelPropulsion}} \cdot (0.087) \]

\[
\Rightarrow \text{FlightTime}_{\text{Propulsion}} = W_{\text{fuelPropulsion}} \times (30.4) \quad \text{[minutes]}
\]

\[
W_{\text{fuel}} = W_{\text{fuelPropulsion}} + W_{\text{fuelPayloadPower}}
\]

### 9.3.3 Case 3: Hybrid

As for case two, the requirement for the engine to produce more power to service both propulsion and electrical generation is assumed to be feasible by suitable gearing. The Propulsion power required from the engine is based on the thrust required, airspeed and propeller efficiency. For the baseline airframe, these factors do not change. In full hybrid mode then, the engine will be required to operate at significantly higher RPM
to generate the 100W extra power required for the assumed payload requirement after 50% power-conditioning efficiency. It has been shown that at the higher RPM the engine will be slightly more efficient, however as a conservative estimate, the lower efficiency (8.3% when using the 16 x 6 propeller) will be assumed.

The following quantities will be used. Note that the engine efficiency and propulsion power required are altered to reflect the measured experimental outcomes of changing the propeller;

- Battery Energy Density = 0.50 MJ/kg
- Battery Conversion Efficiency = 100%
- Fuel Energy Density = 14.9 MJ/kg
- Engine Efficiency (ETE) = 8.3%
- Engine Power Required for Propulsion = 510 Watts
- Electrical Power Required for Payload = 50 Watts

The flight time can be calculated as:

\[
\text{FlightTime} = \left( \frac{\text{Energy Storage Density} \times \text{Energy Storage Mass} \times \text{Conversion Efficiency}}{\text{Power Consumption} \times 60} \right)
\]

Again, the Payload Power will be provided by a combination of the battery and the generator:

\[
\text{FlightTime}_{\text{Payload}} = \left( \frac{0.50 \times 10^6 \times W_{\text{bat}} \times (1)}{50 \times 60} \right) + \left( \frac{14.9 \times 10^6 \times W_{\text{fuelPayloadPower}} \times 0.083 \times 0.50}{50 \times 60} \right)
\]

\[
\rightarrow \text{FlightTime}_{\text{Payload}} = (W_{\text{bat}} \times 166) + W_{\text{fuelPayloadPower}} \times 206 \quad \text{[minutes]}
\]

\[
\text{FlightTime}_{\text{Propulsion}} = \left( \frac{14.9 \times 10^6 \times W_{\text{fuelPropulsion}} \times 0.083}{545 \times 60} \right)
\]

\[
\rightarrow \text{FlightTime}_{\text{Propulsion}} = W_{\text{fuelPropulsion}} \times 378 \quad \text{[minutes]}
\]

\[
W_{\text{fuel}} = W_{\text{fuelPropulsion}} + W_{\text{fuelPayload}}
\]
For each case a system of linear equations relating FlightTime for various proportions of battery mass and fuel mass has been developed. The Endurance of the aircraft can be determined by solving these equations. The operating time for both the payload and the propulsion system will be set equal.

9.4 Case 1: Singular ICE operation with no onboard generator.

\[ \text{FlightTime}_{\text{payload}} - W_{\text{bat}} \times (166) = 0 \]

\[ \text{FlightTime}_{\text{propulsion}} - W_{\text{fuel}} \times (30.4) = 0 \]

\[ \text{FlightTime}_{\text{payload}} - \text{FlightTime}_{\text{propulsion}} = 0 \]

\[ W_{\text{fuel}} + W_{\text{bat}} + W_{\text{proIC}} + W_{\text{pay}} = 6 \]

\[ W_{\text{proIC}} - W_{\text{fuel}} = 0.7 \]

Solving for FlightTime as a function of \( W_{\text{pay}} \)

\[ \text{FlightTime} = \frac{5.3 - W_{\text{pay}}}{\frac{1}{30.4} + \frac{1}{166}} \text{ [minutes]} \]

For \( W_{\text{pay}} \) incremented by 1 kg and constant 50W power requirement the resulting FlightTimes (Endurance) are shown below.

<table>
<thead>
<tr>
<th>Payload [kg]</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum Endurance [minutes]</td>
<td>110</td>
<td>85</td>
<td>59</td>
<td>33</td>
</tr>
</tbody>
</table>

9.5 Case 2: Singular ICE operation with onboard generator.

Let \( p = \) proportion of payload energy mass as battery mass

\[ R_{pp} = \frac{W_{\text{bat}}}{W_{\text{fuel/payloadPower}}} = \left( \frac{p}{1-p} \right) \]
Payload Range and Endurance Modeling

\[\text{FlightTime}_{\text{Payload}} = (W_{\text{bat}} \times 166) + W_{\text{fuelPayloadPower}} \times 216\]

\[\text{FlightTime}_{\text{propulsion}} = W_{\text{fuelPropulsion}} \times (30.4)\]

\[\text{FlightTime}_{\text{payload}} - \text{FlightTime}_{\text{propulsion}} = 0\]

\[W_{\text{fuelPayloadPower}} + W_{\text{bat}} + W_{\text{proICgen}} + W_{\text{pay}} = 6\]

\[W_{\text{proICgen}} - W_{\text{fuelPropulsion}} = 1.015\]

\[W_{\text{fuel}} = W_{\text{fuelPropulsion}} + W_{\text{fuelPayloadPower}}\]

Solve for FlightTime as a function of \(W_{\text{pay}}\) and \(R_{pp}\)

\[\text{FlightTime} = \frac{(4.985 - W_{\text{pay}})}{\left(\frac{1}{216} + \frac{1}{30.4}\right) \left(\frac{166}{216} \cdot \frac{R}{216} - 1\right)}\] [minutes]

The following figure shows the flight time plotted for all values of payload and generator utilization factor, \(p\) or Proportion of Payload Powered by Battery.

![Diagram showing flight time plotted for all values of payload and generator utilization factor.](image-url)
Given that a generator is fitted, it is always better for endurance to use it for all values of payload mass. There may be situations where carrying some proportion of payload energy as battery is required, such that the payload remains operational with the engine shutdown. The figure above shows the cost to endurance of this trade-off.

In this case, the change in flight time is slight due to the inefficiency of the engine, the low energy density of the fuel, and the high energy density of the assumed battery type.

<table>
<thead>
<tr>
<th>Table 9-2 Case 2 Endurance</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload [kg]</td>
</tr>
<tr>
<td>Maximum Endurance [minutes]</td>
</tr>
<tr>
<td></td>
</tr>
</tbody>
</table>

In comparison to Case 1, the endurance has decreased. A consequence of the low engine efficiency, fuel energy density and moderate payload power consumption in this example is that carrying the extra generator weight is not as beneficial as carrying that weight in extra fuel and battery. Referring to Table 9-2 above, it is clear that the aircraft could fly slightly longer at all payloads, if no generator were fitted.

For comparison, a plot showing the value of utilizing a generator with better values for engine efficiency (22%), and fuel (40 MJ/kg) and higher payload power consumption (100W) on the same airframe and payload is shown in Figure 9-2 below. These engine and fuel parameters are consistent with commercial operational UAVs such as Aerosonde and ScanEagle, the following plot exemplifies why generators are present on these aircraft.
Figure 9-2 Endurance, Payload and Generator Utilization for Case 2 High Efficiency

A marked improvement in endurance is seen for all values of payload when utilizing the generator to power the payload, rather than batteries, when high efficiency engine and fuel is used.

Table 9-3 Case 2 High Efficiency Fuel and Engine with Generator Endurance

<table>
<thead>
<tr>
<th>Payload [kg]</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum</td>
<td>Generator Only (p=0)</td>
<td>719</td>
<td>539</td>
<td>359</td>
</tr>
<tr>
<td>Endurance</td>
<td>Battery Only (p=1)</td>
<td>236</td>
<td>177</td>
<td>117</td>
</tr>
</tbody>
</table>

9.6 Case 3: Hybrid Powerplant

Let \( R_{pp} = \frac{W_{bat}}{W_{fuelPayloadPower}} = \left( \frac{p}{1-p} \right) \)

\[
\text{FlightTime}_{\text{payload}} = (W_{\text{bat}} \times (166)) + W_{\text{fuelPayloadPower}} \times (206)
\]
Payload Range and Endurance Modeling

\[ \text{FlightTime}_{\text{propulsion}} = W_{\text{fuelpropulsion}} \times (37.8) \]

\[
\text{FlightTime}_{\text{payload}} - \text{FlightTime}_{\text{propulsion}} = 0
\]

\[
W_{\text{fuelpayloadpower}} + W_{\text{bat}} + W_{\text{hybrid}} + W_{\text{pay}} = 6
\]

\[
W_{\text{hybrid}} - W_{\text{fuelpropulsion}} = 1.130
\]

\[
W_{\text{fuel}} = W_{\text{fuelpropulsion}} + W_{\text{fuelpayloadpower}}
\]

Solve for FlightTime as a function of \( W_{\text{pay}} \) and \( R_{\text{pp}} \)

\[
\text{FlightTime} = \frac{(4.87 - W_{\text{pay}})}{\left(\frac{1}{206} + \frac{1}{37.8}\right) - \frac{p}{\left(166 \times p + 206(1 - p)\right)}} \text{ [minutes]}
\]

Figure 9-3 Case 3 Hybrid Endurance, Payload and Generator Utilization

This plot shows that the hybrid powerplant equipped aircraft has increased endurance at all payloads and payload power battery usage compared to the generator only equipped aircraft. It has greater range than either other configuration. The extra mass
of the electrical boost motor and subsequent loss of fuel mass capacity is more than offset by the improved overall propulsive efficiency at the design conditions.

**Table 9-4 Case 3. Hybrid Endurance**

<table>
<thead>
<tr>
<th>Payload [kg]</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Maximum Endurance [minutes]</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Generator Only (p=0)</td>
<td>124</td>
<td>92</td>
<td>60</td>
<td>29</td>
</tr>
<tr>
<td>Battery Only (p=1)</td>
<td>119</td>
<td>88</td>
<td>58</td>
<td>27</td>
</tr>
</tbody>
</table>

**9.7 Summary**

The following table summarizes the theoretical endurance of the given airframe model utilizing the three powerplant options.

**Table 9-5 Powerplant Specific Endurance Comparison**

<table>
<thead>
<tr>
<th>Payload [kg]</th>
<th>1</th>
<th>2</th>
<th>3</th>
<th>4</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Case 1, Maximum Endurance [minutes]</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>No Generator or Hybrid</td>
<td>110</td>
<td>85</td>
<td>59</td>
<td>33</td>
</tr>
<tr>
<td><strong>Case 2 Maximum Endurance [minutes]</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Generator Only</td>
<td>106</td>
<td>80</td>
<td>52</td>
<td>26</td>
</tr>
<tr>
<td><strong>Case 3 Maximum Endurance [minutes]</strong></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Hybrid</td>
<td>124</td>
<td>92</td>
<td>60</td>
<td>29</td>
</tr>
</tbody>
</table>

The critical payload at which generator or hybrid powerplant becomes desirable on the basis purely of maximum endurance can be determined by setting the relevant linear equations equal.

Comparing Case 1 with Case 2, the following equation is solved for $W_{\text{pay}}$ with $p=0$;
\[
\frac{(5.3 - \text{W}_{\text{pay}})}{\frac{1}{30.4} + \frac{1}{166}} = \frac{(4.985 - \text{W}_{\text{pay}})}{\frac{1}{216} + \frac{1}{30.4} - \frac{p(166 - 1)}{(166 \times p + 216(1-p))}}
\]

\[\text{W}_{\text{pay}} = -3.45 \text{ kg}\]

The negative payload value indicates that combined effect of low grade fuel and engine inefficiency prevent the generator from yielding an advantage at all payload weights for this model.

Comparing Case 1 with Case 3, the following equation is solved for \(\text{W}_{\text{pay}}\) with \(p = 1\);

\[
\frac{(5.3 - \text{W}_{\text{pay}})}{\frac{1}{30.4} + \frac{1}{166}} = \frac{(4.87 - \text{W}_{\text{pay}})}{\frac{1}{206} + \frac{1}{37.8} - \frac{p(166 - 1)}{(166 \times p + 206(1-p))}}
\]

\[\text{W}_{\text{pay}} = 3.09 \text{ kg}\]

Despite still using low grade fuel and inefficient engine, the hybrid powerplant shows increased endurance compared to the Case 1 engine only powerplant at all payloads up to of 3.09 kg.

The improvements of Case 3 hybrid endurance over the Case 1 and 2 powerplants are due to the increased propeller efficiency (16 x 6). It is clear that The Case 1 and 2 powerplant configurations could be fitted with a Case 3 propeller to yield the same cruise performance, however it must be emphasized that the main conditional assumptions are that the aircraft must be capable of independent takeoff performance and use a fixed pitch propeller. If fitted with the larger propeller (16 x 6), both Case 1 and 2 powerplants would suffer a 25% decrease in static thrust compared to the original propeller, and would have 62% less takeoff thrust than the boosted Hybrid, see Tables 8-2 and 8-3. Neither Case 1 nor Case 2 powerplants can deliver the static and low-speed thrust which the Case 3 hybrid powerplant is capable of.
Under these conditions the hybrid powerplant appears to be superior for all but the highest payload mass and shortest duration missions. The difference seen in Table 9-5 is modest due to the poor efficiency of the given engine and fuel energy density, but that even under these conditions the aircraft has extremely improved utility and effectiveness in addition to improved endurance and range.
10 Further Development Opportunities

This chapter will present some theoretical features of aircraft hybrid powerplants which were not developed experimentally but which build on the concepts shown in this thesis as successful.

10.1 Peaky Engines

The hybrid performance advantage shown in Chapter 9 is primarily a result of allowing the use of a higher efficiency propeller for some given set of operational requirements. A hybrid powerplant would be much more effective if it would also enable an increase in engine efficiency. Internal combustion engines typically exhibit a peak efficiency at some combination of output torque and speed. Where the range of torque and speed necessary for the application can be constrained, the design may be optimized toward the highest efficiency. Much of the overall efficiency gain from hybridization of automobile powerplants is due to the constraint of the range of engine operating points without compromising the overall vehicle’s range of operating conditions. An aircraft powerplant typically operates through a smaller range of output conditions than an automobile engine. The use of an in-flight variable pitch propeller would also contribute to restricting the necessary output range for the aircraft engine and a hybrid powerplant combined with variable pitch propeller would be ideal.

An automobile engine is often used throughout the range from 1500 RPM to 6000RPM and 10% to 100% rated power, but the cruise condition is usually 30% rated power[19]. The relevant UAV engine will be used most of the time between 7000RPM and 9000RPM with cruise between 50% and 75% rated power. If the requirement for the engine to be capable of smooth or efficient operation below these limits is removed, enhancement of operation within the limits becomes easier.

The OS fx engine used for testing is designed for reliability and ease of operation from 2000RPM through to 14000RPM, the torque curve is smooth and relatively flat. Also the SFC of this engine was found to vary from 2.04 kg/kWhr to 1.94 kg/kWhr through the range from 7860RPM to 11000RPM. An engine optimized for absolute efficiency between narrower constraints could exhibit a much sharper peak. Such an engine could exhibit very poor efficiency if operated too far from the design optimum,
under such conditions the use of electrical motor torque to either supplement or replace the engine torque could yield a more efficient use of the available energy. The effect of hybridizing such an engine for aircraft use is explored below.

10.1.1 **SFC Contour Plots**
Engine SFC data can be gathered across the operating range of an engine and plotted on the Torque or Power Vs RPM plane. The plot below is based on information contained in Matlab Aerosim [42].

![Figure 10-1 Aerosim Aerosonde Fuel Map](image)

**10.1.2 Hypothetical Peaky Engine Characteristics**
The corresponding power curves for the OS and hypothetical engines are shown below. The high efficiency hypothetical engine in this case cannot produce as much power (760W) as the OS engine (830W), but produces 92% of the OS power using 15% of the OS fuel flow.
Further Development Opportunities

Figure 10-2 Hypothetical “Peaky” Engine Power Comparison

Increasing the fuel efficiency of an engine usually compromises the maximum power output, and therefore the power per weight ratio. Hence for the same engine mass installed in an aircraft, the highly fuel efficient engine will usually develop less maximum power.

For the purpose of a comparison of fuel efficiency gained at the expense of operating flexibility some hypothetical fuel consumption figures will be used. The value of 0.3 kg/kWhr is quite reasonable for a well developed small UAV engine. The Aerosonde engine as modeled in Aerosim quotes a best figure of 0.325 kg/kWhr, across a reasonably broad range. Hypothetically assumed SFC contour plots for two engines are shown below.
Further Development Opportunities

Figure 10-3 Hypothetical Fuel Maps
The hypothetical engine SFC is very sensitive to changes in speed and load. In order to gain the advantage of the very high SFC available, it is imperative to load the engine only within the narrow optimum range.

The plot below shows how a hybrid powerplant configuration can enable a fine tuned “peaky” engine to be operated across the region of torque demand where it cannot generate sufficient power.

Figure 10-4 Hybrid Combined Torque Enabling Engine Viability
For all thrust and therefore torque requirements beyond the operating capability of the engine, the electric motor is used to supply and accelerate the load as shown in the region 0RPM to approximately 5200RPM in figure 10-4. Once the engine is operating in the design region (approximately 6500RPM to 7800RPM), it can efficiently supply the necessary torque on its own with no assistance from the motor. If motor torque is combined with engine torque, the propeller can be operated at significantly higher thrust.

The plots below indicate the effect of incorrectly matching a load to an engine with a very sensitive SFC.

Figure 10-5 Efficient (left) and Inefficient (right) Load Matching

The OS engine as shown on the left of figure 10-5 would drive the load at the operating point shown with a SFC of 2Kg/kWhr, while the hypothetical engine would deliver 2.5 kg/kWhr under these circumstances. Usually gearboxes are used to alter the load conditions to permit the engine to operate in an efficient region, such as ubiquitously seen in automobiles. The gear ratio selected does not significantly alter the transmission efficiency. A similar effect is available on aircraft fitted with variable-pitch propellers, the load conditions may be altered, but the change in pitch and RPM may lead to significant changes in propeller efficiency. The trade-off will result in a local optimum operating point, however unless the system happens to match a global design optimum, neither the propeller nor the engine can work at maximum efficiency.
Also notable is the fact that the hypothetical engine, although capable of producing more torque, is unable to drive this load at as high a speed as the OS example, since altering the gearing cannot increase the power output. In the case shown below in figure 10-6, the required load point is beyond the capability of the hypothetical engine.

**Figure 10-6 Viable (left) and Non-viable (right) Load Matching**

The plot below in figure 10-7 shows the result of altering the gear ratio between the hypothetical engine and the propeller. This particular desired propeller speed of 9200 RPM cannot be maintained by this engine at any gear ratio. However if this propeller RPM was required for extended periods the appropriately geared engine torque could be augmented with electric motor torque. Under these conditions the full engine torque could be delivered at the appropriate RPM for maximum SFC, while using only a small amount of electric motor power.
Figure 10-7 Effect of Engine to Propeller Gearing on Overload Condition

An alternative approach to solving this loading mismatch is available with a hybrid powerplant shown in figure 10-8.

Figure 10-8 Application of Electric Motor Torque to Satisfy Temporary Overload

Given a particular propeller speed requirement, in this case around 9200 RPM, the hypothetical engine cannot produce the required torque. Reducing the propeller pitch to reduce the torque required will allow the engine to speed up, but the operating point will be moving along the torque curve further toward lower SFC, and the thrust
produced may be inadequate. At this gear ratio, the application of augmenting electric motor torque allowing the engine to produce maximum torque, would drive the propeller at the desired RPM, but the engine SFC would still be very low. It may be advantageous to temporarily disengage or shut down the engine and provide the required propeller power with the electric motor alone.

The temporary application of motor power will use some battery energy which may be regenerated at a time and conditions when the engine is operating at optimum efficiency.

Given the same propeller load, a hybrid powerplant can alter the load characteristic by implementing generator load. The engine will perform more work, but a large proportion of the generator work energy will be stored, and the efficiency of the engine can be optimized, with resulting decrease in fuel flow. Referring to the figure 10-9 below, a particular propeller operating point is assumed.

**Figure 10-9 Hypothetical Propeller Load Only Operating Point**

The desired thrust output for the aircraft performance is being achieved but the engine is operating inefficiently in the 2 kg/kWhr region. Application of generator torque load is possible without reducing the engine RPM, as under the present conditions the engine is not developing full power.
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Figure 10-10 Hypothetical Operating Point with Generator Load

The new operating point of the system is shown in figure 10-10 above with the engine SFC at 0.3kg/kWhr. Extra power is being produced above that necessary for propulsion and is being converted to electrical power for direct use or storage. The efficiency of the propulsion energy production and the electrical energy production is optimized.

10.2 Density Altitude (air Density) Considerations

A normally aspirated engine will exhibit increased or reduced performance according to the ambient density of the air. The density is a function of temperature and pressure, both of which change with altitude. Pressure always reduces with altitude, while temperature may vary, but usually declines. The net result is that the density declines as a function of altitude and this has the effect of reducing the torque and power of normally aspirated (non induction boosted) engines. This factor is easily modeled by the load curve analysis method if the engine performance for the various density altitudes are known or approximated. The trend plot is shown below in figure 10-11.
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The absolute ceiling (highest attainable steady state altitude) of an aircraft occurs where the power required at that altitude is equivalent to the power available at that altitude. The power required at high altitude is in excess of that required at lower altitudes[26] . A major benefit of electric powerplants is that the power output is insensitive to altitude variation. Hence a hybrid powered aircraft could be operated at higher altitudes than an equivalent internal combustion only equipped aircraft, under certain conditions. The current world altitude record for any non-rocket powered aircraft is held by an electric powered, propeller driven type, Helios [4] .

10.3 Problems using the same motor for starting, charging and boosting.

The speed and torque specification of the electric machine depend on its physical construction. The number and type of coil windings as well as the method of field excitation are the principle variables. For BLDC type motors, there is a linear output speed versus torque function. The applied voltage is the primary control for motor speed, while the current controls the torque. Limits to the available voltage and current capability of the system as well as mechanical strength considerations define the motor performance capability.
For engine starting, the motor must be able to provide sufficient torque across a relatively small speed range to overcome the combustion cylinder compression, friction and inertia loads. The propeller aerodynamic load is insignificant on start up, and its inertia load may be removed depending on the transmission system used.

For effective boost power, the motor must provide torque across the range of output shaft speeds appropriate to the engine and propeller. The electrical supply battery must be specified in such a way that for a given motor, engine and propeller configuration it can supply the maximum required current for maximum desired boost power at maximum output shaft speed. The battery no-load voltage must be sufficient that under conditions of maximum current draw, it remains able to provide the highest motor output speed. However at lower speeds when the motor is being used as a generator, the battery voltage must be lower than the resulting open terminal voltage of the driven motor, or its electronic controller.

There are two clear methods of resolving this problem. The supply battery or electronic controller can be arranged to vary between voltage levels as required, or the transmission ratio between the electric machine shaft and engine or propeller shaft can be varied.

The current provided by the generator is a function of the terminal voltage, the lower the battery voltage, the higher will be the current for a given shaft speed. There are two purposes for the generator in the present context, battery recharging, and providing current direct to loads, however battery charging is considered the only purpose since in practice it will always be part of the circuit. Thus, maintaining battery voltage is the primary requirement of the generator. The specifics of electrical load control and equipment is beyond the scope of this work. It will be assumed that in order to charge the battery the open terminal voltage of the generator must exceed the terminal voltage of the battery. Further, it will be assumed that the open terminal voltage difference between the battery and generator will provide a proportional torque load on the generator.

For any flight condition, excess engine power must be available if there is to be fuel energy directed into electrical generation. For a fixed pitch propeller installation, the existence of excess engine power available implies that either, (a); the throttle is not
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wide open, or if the throttle is fully open, that (b); the engine is operating at a speed beyond its best power output speed.

10.3.1 Case a:
In the first case, opening the throttle while simultaneously loading the generator can maintain the engine speed and hence that flight condition while providing electrical power. The overall energy efficiency of this operation will depend first on the aerodynamic flight conditions, but whatever that baseline, the engine efficiency can be optimized using the generator. See Figure 10-10. Always it is preferable on a carburetor type engine to have the throttle wide open for best pumping efficiency, however the resulting engine speed can only be controlled by the ambient air density and the load. Now for the given flight condition, the ambient air density is not variable, but the charge rate and hence the generator load may be variable. Then if the charge rate can be set to provide a generator load which will maintain the desired engine rpm at full throttle, maximum “local” efficiency will result.

10.3.2 Case b:
This should be an unusual condition, most engines of type under consideration will show best SFC closer to maximum power speeds, if the engine is running beyond its best power speed, given fixed gearing and propeller pitch;

- the propeller is undersized, or
- the aircraft must be descending at high speed

Applying more load to the engine under these circumstances may be appropriate or necessary. Providing the shaft speed reduction did not adversely affect the desired propeller performance, there could be significant power available and it would be gained without extra fuel consumption.

10.3.3 Discussion
Case a, is certainly the usual condition which should be designed for to provide continuous electrical power from the engine. Although there will always be operational variables, different cruise speeds and altitudes, a reasonable average range of generator speeds and loads can be determined for a particular aircraft. The electric unit can be selected to provide an appropriate voltage at this speed, which will be above the battery voltage. Clearly then, this battery will not be able to drive the
generator as a motor at this or any higher speed as required to provide boost power, without some form of step change in the system.

In the experiments performed in this research, the significant problem was encountered in obtaining sufficient range of motor torque across the speed range of the engine for both starting and boosting purposes was battery voltage level. The motor and gear ratio selection were inappropriate for the engine and propeller combination, the resulting effect specific to battery voltage is relevant here.

Difficulty was experienced in driving the motor at the high end of the speed range where the torque was required to allow the engine to operate in its best torque range and hence drive the propeller load. The motor was within the design operating limit, but at the high end where the 700 rpm/Volt no-load speed constant would require at least 15 Volts, under load, and thus around 20 Volts should have been sufficient.

In principle, the required voltage and current supply can be made available, however some practical and experimental limitations were present. The major limiting factor was the ESC, one available unit [29] was restricted to maximum 16V input. The available SLA batteries could be configured to give a nominal 18V, but at full charge would often measure around 20V. This ESC had over-voltage protection and would not operate with applied voltage above 16V. A regulation system was not available, but a set of NiMH type batteries was setup to give 15.6 Volts nominal. Although when fully charged, the NiMHs had slightly more than 16V at the terminals, the ESC worked, however with at best 16V applied to the motor, it could never achieve the desired output speed. Moreover, at the current draw required, the terminal voltage of the battery quickly dropped well below the nominal voltage and exacerbated the problem. A different ESC [30] was available which had a 36V limit. Two SLAs in series were applied to give 24V nominal, and for short periods, the desired system output was achieved. Unfortunately, this ESC was under-rated in terms of current flow and quickly overheated under full load, also the effect of on-load voltage drop continued to be significant, nevertheless some useful data were collected.

These experiments highlighted the necessity for the entire electrical power supply, conditioning and control system to be specifically designed as part of the overall hybrid propulsion system. If the mechanical gearing of the test system had been
different or variable, the desired motor or generator performance could have been achieved with the given battery and ESC constraints. Conversely, if the Battery and ESC were re-configurable, particularly in voltage levels, the desired motor and generator performance could have been achieved without mechanical gearing changes. Within some limits, a standard non re-configurable system, in either gear ratio or voltage will work, if the aircraft performance qualities and operational requirements are suitably matched to the hybrid powerplant. Given that electronic power control and conditioning systems are inherently necessary for motor and generator control, and that mechanical transmission systems are always best avoided or simplified, the extension of hybrid performance will rely on the electronics development.

**10.4 Power Regeneration**
The inclusion of an electric generator and appropriate transmission system on a hybrid powered aircraft enables the possibility of regenerating excess airframe energy to electrical energy. The excess airframe energy will be in the form of continuous relative airflow. The capability to sustain this airflow will occur from mission scenario outcomes or environmental opportunities.

**10.4.1 Mission scenario;**
Excess altitude may be transformed into distance by gliding an aircraft. The efficiency of this transformation is dependent on the glide ratio. If a mission scenario called for a reduction in altitude without glide or at very steep glide angles the surplus energy has to be disposed of by increasing the drag by use of flaps or airbrakes, or by allowing a very high speed. These drag increasing methods waste energy and the high speed dive may result in excess speed at the new desired altitude. The use of the aircraft propeller as a turbine will provide drag and recuperate some power into electrical storage.

**10.4.2 Environmental Opportunities;**
It is well known that atmospheric wind and thermal convection systems can be utilized to keep non-motorized aircraft airbourne. The reliability and availability of these energy sources is variable, however where they exist, the power available can be quite significant. Convection columns and orographic currents rising at rates of 6m/s are normal across many areas. Aircraft usually make use of these vertical air flows to
increase altitude, however if altitude gain is unnecessary or unwanted, the available energy can be harvested using a turbine. The use of a propeller as a turbine in this situation can allow continuous flight while increasing onboard stored energy.

10.4.3 Efficiency
It is of interest to determine the efficiency at which a propeller turbine can convert the available excess energy. In some cases the decision as to the use of a propeller turbine will be governed by the efficiency, and in other cases the excess energy must be disposed of and any regeneration will be a bonus.

Regardless of the reason for the availability of the airflow energy, there is a limit to how efficiently it can be converted to shaft power and supplied to a generator. The same techniques as for land based wind power turbines can be applied, however there is a major difference in the design of a propulsion propeller which inevitably leads to reduced efficiency. The airfoil section of a propeller is never symmetrical, and when used in reverse flow, the camber will effectively be inverted and very poor section lift per drag ratios occur. This translates to low efficiency. If the propeller were capable of complete pitch reversal, the camber relative to the airflow would be appropriate, however the longitudinal camber orientation would be reversed. Effectively the airfoil section will run either upside-down or back to front when a propeller is used as a turbine, and both configurations are relatively inefficient.

10.4.4 Flow Power
The fluid power available from a fixed diameter of fluid flow is determined by the following equation.

\[ P_{flow} = \frac{1}{2} A \rho V^3 \] [W]

Where;

A is the flow cross-section area in square meters.
\( \rho \) is the fluid density in kg per cubic meter
V is the fluid velocity in meters per second.
There is a maximum limit to the quantity of power which may be continuously removed from the flow by the turbine, as described by Betz [31] 59%.

As discussed in Chapter 5, measurements for various propeller sizes at various flow rates and RPM were made. The most relevant data here concern the 16 x 6 propeller.

![Figure 10-12 Propeller Turbine Power versus Turbine RPM at Different Airspeeds](image)

This plot shows the input mechanical power delivered by the turbine to the generator as well as the resulting generator power production for three different airspeeds. The difference between the input power and output power is a result of the generator electromagnetic efficiency and friction effects. The DC generator used in this experiment was a brushed permanent magnet type. The characteristics are similar for BLDC machines. Also shown in the plot is evidence of the dynamometer precision and accuracy as discussed in Chapter 5.

The plot below shows a maximum energy recovery of around 60 W at a flight speed of 30 m/s.
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If 60 Watts is feasibly recoverable on a UAV of this size, a turbine recovery and electrical generator may be quite valuable considering the baseline payload power requirements assumed elsewhere. However this maximum power was recovered with the imposition of a certain amount of additional drag from the propeller. To be viable the aircraft must continue to fly with this extra drag for some reasonable time period. The associated drag is shown in the plot below.

![Figure 10-13 Propeller Turbine Power Available 20, 25 and 30 meters per second Airspeed](image1.png)

![Figure 10-14 Propeller Turbine Drag at 20, 25 and 30 m/s Airspeed](image2.png)

Notably there are two distinct values of RPM for most values of Drag. Also there are two distinct values of RPM for each value of Power below the maximum. Hence the drag associated with power production can be varied, according to the RPM selected. However, different RPM settings imply different terminal voltage on the generator, so
utilization of this variable is only feasible according to the nature of the power conditioning system.

**Figure 10-15 Propeller Turbine Power Available versus Drag**

A simplified aircraft model will be used to determine the performance with the imposition of this extra drag. For each of the given airspeeds, the aircraft will have a particular Lift per Drag ratio excluding the drag associated with the turbine. This L/D can be assumed for a typical small UAV and the addition of the turbine drag will indicate the resultant glide angle and sink rate. This analysis should include density altitude effects, since the premise of energy recuperation from excess altitude implies significant density altitude change. The concept of Equivalent Airspeed [32] also known as Indicated Airspeed (IAS) is proportional to the dynamic air pressure and allows comparison of aircraft performance speeds at different altitudes. True Airspeed (TAS) refers to the velocity of the relative airflow.

\[ V_e = \left( \frac{\rho}{\rho_0} \right)^{\frac{1}{2}} V \]

Where

\( \rho_0 \) = Reference Density, sea level International Standard Atmosphere (ISA)

\( \rho \) = Density at given altitude

Or
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\[ IAS = \left( \frac{\rho}{\rho_0} \right)^{\frac{1}{2}} TAS \]

Now, the following assumed L/D figures for the aircraft model occur at identical IAS for any altitude, but clearly the TAS differs. The change in TAS will affect the Vertical Velocity but not the Glide Angle.

A method of determining the resultant glide angle and descent rate is required to ascertain the effect and feasibility of operating the turbine. When the turbine is operating it produces drag, which may be modeled as negative thrust. The standard steady state aerodynamic model as shown below can be used.

![Figure 10-16 Basic Aeroplane Forces](image)

For steady-state zero acceleration the aerodynamic, gravitational and thrust forces sum to zero. For typical L/D ratios, the required thrust is significantly less than the weight.

For a descent angle \( \gamma \) to be introduced, for a steady state, either the thrust must be reduced or the drag increased or both.

The reduced thrust and no extra drag case is illustrated below.
If a steeper descent is required, additional drag is applied as follows.

\[ f = \frac{D}{W} \sin \gamma = \beta \cos \gamma + \sin \gamma \]

Where \( \beta = \left( \frac{D}{L} \right) \) and \( f = \left( \frac{T}{W} \right) \)

Then, for small \( \gamma \);

\[ \gamma = \sin^{-1} \left( \frac{T - D}{W} \right) \]
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Note that the magnitude of the lift vector decreases as steeper descents are required, the lift vector also decreases as steeper climbs are required. Climb and descent angles depend on the difference in magnitude of thrust and drag.

For the same descent angle and speed, additional drag could be applied as a negative thrust, rather than by flaps or airbrakes and represented as follows.

\[
\gamma = \sin^{-1}\left(\frac{-T - D}{W}\right)
\]

And, Vertical Velocity

\[
V_y = V \sin \gamma
\]

\[
V_y = -V\left(\frac{\text{Turbine Drag} + \text{Airframe Drag}}{\text{Weight}}\right)
\]

Clearly then, for a steady continuous descent to occur, the combined drag of the airframe and the turbine cannot exceed the weight. If the combined drag equals the weight, the aircraft can descend vertically in still air at whatever speed gives this drag. If the aircraft is in a rising column of air, under these conditions it will climb. By controlling the turbine drag, the descent rate and angle can be controlled. If there is atmospheric energy (thermal convection or orographic lift, or horizontal gradient energy [33], then the turbine can supply energy continuously with no net loss of altitude.
The airframe drag can be modeled in terms of assumed airframe zero thrust L/D for different airspeeds. The L/D can be considered identical to the Glide Ratio, for any angle with minimal error. [32]

And,

\[
\tan^{-1}\left(\frac{V_y}{V_x}\right)
\]

Unfortunately the only Turbine Drag data obtainable was for speeds up to 30m/s but not in excess. If the turbine is to be used aboard a “diving” (descending at a speed beyond best L/D) aircraft either a different airframe model from that used in previous analysis Chapter 9 must be used, or the unknown turbine drag may be extrapolated to suit higher speeds on the existing airframe model assumptions.

The following analysis is based on an assumed airframe Lift per Drag model to yield glide angles and drag magnitudes for a range of airspeeds. The known turbine drags as well as extrapolated values are then used to determine resultant total drags and hence the effect of turbine operation on aircraft performance. The combined turbine output and airframe performance then yields total energy recovery.

**Table 10-1 Propeller Turbine Modelling Outcomes for Standard Airframe Model**

<table>
<thead>
<tr>
<th>IAS [m/s]</th>
<th>20</th>
<th>25</th>
<th>30</th>
<th>35</th>
<th>40</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glide Angle L/D Assumed</td>
<td>4:1</td>
<td>5:1</td>
<td>6:1</td>
<td>5:1</td>
<td>4:1</td>
</tr>
<tr>
<td>Drag for 10 kg AUW [N]</td>
<td>24.5</td>
<td>19.6</td>
<td>16.4</td>
<td>19.6</td>
<td>24.5</td>
</tr>
<tr>
<td>Max. Turbine Drag [N] (extrapolated for +30m/s)</td>
<td>8</td>
<td>11</td>
<td>14</td>
<td>18.5*</td>
<td>22.7*</td>
</tr>
<tr>
<td>Total Drag [N]</td>
<td>32.5</td>
<td>30.6</td>
<td>30.4</td>
<td>38.1*</td>
<td>47.2*</td>
</tr>
</tbody>
</table>
Further Development Opportunities

<table>
<thead>
<tr>
<th>Glide Angle with Turbine [deg]</th>
<th>19.3</th>
<th>18.2</th>
<th>18</th>
<th>22.9*</th>
<th>28.8*</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glide Ratio</td>
<td>2.85 :1</td>
<td>3.04 :1</td>
<td>3.07 :1</td>
<td>2.4 :1*</td>
<td>1.8 : 1*</td>
</tr>
<tr>
<td>Vertical Velocity [m/s]</td>
<td>6.6</td>
<td>7.81</td>
<td>9.27</td>
<td>13.6*</td>
<td>19.3*</td>
</tr>
<tr>
<td>Turbine Power (extrapolated for +30m/s)</td>
<td>24</td>
<td>42</td>
<td>62</td>
<td>100*</td>
<td>149*</td>
</tr>
<tr>
<td>Flow Power Available</td>
<td>622</td>
<td>1216</td>
<td>2101</td>
<td>3337</td>
<td>4981</td>
</tr>
<tr>
<td>Turbine Efficiency</td>
<td>4.0%</td>
<td>3.4%</td>
<td>3%</td>
<td>3%*</td>
<td>3%*</td>
</tr>
<tr>
<td>Time for 1000 meter Descent [minutes]</td>
<td>2.5</td>
<td>2.13</td>
<td>1.8</td>
<td>1.22*</td>
<td>0.86*</td>
</tr>
<tr>
<td>Energy Recuperated for 1000m descent. [J]</td>
<td>3600</td>
<td>5370</td>
<td>6700</td>
<td>7320*</td>
<td>7690*</td>
</tr>
<tr>
<td>Energy Recuperated [WattHour]</td>
<td>1.0</td>
<td>1.5</td>
<td>1.8</td>
<td>2.0*</td>
<td>2.1*</td>
</tr>
</tbody>
</table>

* Based on extrapolated data

The analysis above is intended to give some indication of the benefit of operating the propeller turbine as part of the hybrid powerplant installed on an airframe of the type.
used in the propulsive efficiency analysis. The minimum vertical velocity of 6.6 meters per second combined with the modest electrical power recovery of 24 Watts indicates that the regeneration capability would be of marginal usefulness for this airframe. It is possible however that this aircraft could be flown in atmospheric conditions which could sustain this sink rate for significant periods of time without reducing altitude. Vertical atmospheric currents of this magnitude are comparatively rare in distribution, compared to magnitudes of 2 to 3 meters per second which ordinarily occur during normal weather over many continental locations during the day. A regenerative turbine installed on a more aerodynamically efficient airframe could sustain useful turbine output more often and for longer periods. An example using Lift per Drag values of a high efficiency sailplane of the same mass is shown below.

Table 10-2 Propeller Turbine Modelling Outcomes for High Efficiency Airframe Model

<table>
<thead>
<tr>
<th>IAS [m/s]</th>
<th>20</th>
<th>25</th>
<th>30</th>
<th>35</th>
<th>40</th>
</tr>
</thead>
<tbody>
<tr>
<td>Glide Angle</td>
<td>15:1</td>
<td>20:1</td>
<td>25:1</td>
<td>20:1</td>
<td>15:1</td>
</tr>
<tr>
<td>L/D Assumed</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Drag for 10 kg AUW [N]</td>
<td>6.5</td>
<td>4.9</td>
<td>3.9</td>
<td>4.9</td>
<td>6.5</td>
</tr>
<tr>
<td>Max. Turbine Drag [N] (extrapolated for +30m/s)</td>
<td>8</td>
<td>11</td>
<td>14</td>
<td>18.5*</td>
<td>22.7*</td>
</tr>
<tr>
<td>Total Drag [N]</td>
<td>14.5</td>
<td>15.9</td>
<td>17.9</td>
<td>23.4*</td>
<td>29.2*</td>
</tr>
<tr>
<td>Glide Angle with Turbine [deg]</td>
<td>8.5</td>
<td>9.28</td>
<td>10.5</td>
<td>13.8*</td>
<td>17.3*</td>
</tr>
<tr>
<td>Glide Ratio</td>
<td>6.7 :1</td>
<td>6.1 :1</td>
<td>5.4 :1</td>
<td>4.1 :1*</td>
<td>3.2 :1*</td>
</tr>
<tr>
<td>Vertical Velocity [m/s]</td>
<td>3.0</td>
<td>4.0</td>
<td>5.5</td>
<td>8.3*</td>
<td>11.9*</td>
</tr>
</tbody>
</table>
Further Development Opportunities

<table>
<thead>
<tr>
<th>Turbine Power (extrapolated for +30m/s)</th>
<th>24</th>
<th>42</th>
<th>62</th>
<th>100*</th>
<th>149*</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flow Power Available</td>
<td>622</td>
<td>1216</td>
<td>2101</td>
<td>3337</td>
<td>4981</td>
</tr>
<tr>
<td>Turbine Efficiency</td>
<td>4.0%</td>
<td>3.4%</td>
<td>3%</td>
<td>3%*</td>
<td>3%*</td>
</tr>
<tr>
<td>Time for 1000 meter Descent [minutes]</td>
<td>5.5</td>
<td>4.2</td>
<td>3.0</td>
<td>2.0*</td>
<td>1.4*</td>
</tr>
<tr>
<td>Energy Recuperated for 1000m descent. [J]</td>
<td>7920</td>
<td>10580</td>
<td>11160</td>
<td>12000*</td>
<td>12510*</td>
</tr>
<tr>
<td>Energy Recuperated [milliAmpHour@12V]</td>
<td>183</td>
<td>245</td>
<td>258</td>
<td>278*</td>
<td>290*</td>
</tr>
</tbody>
</table>

* Based on extrapolated data

Here the resulting vertical velocity of 3 m/s while generating 24 Watts would be sustainable in average thermal convection and orographic lift conditions. 62 Watts would be sustainable in thermal and orographic lift conditions which are not unusual.

The change in TAS will alter some of the characteristics of the turbine. For moderate altitude differences the variation between IAS and TAS is small. Using ISA data [34],

### Table 10-3 True Airspeed Variation with Standard Altitude

<table>
<thead>
<tr>
<th>Elevation [m]</th>
<th>$\frac{\rho}{\rho_0}$</th>
<th>$\frac{V}{V_e}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>1.0000</td>
<td>1</td>
</tr>
<tr>
<td>500</td>
<td>0.9529</td>
<td>1.025</td>
</tr>
<tr>
<td>1000</td>
<td>0.9075</td>
<td>1.050</td>
</tr>
<tr>
<td>1500</td>
<td>0.8638</td>
<td>1.076</td>
</tr>
</tbody>
</table>
Further Development Opportunities

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td>622</td>
<td>30</td>
<td>367</td>
<td>4.8</td>
</tr>
</tbody>
</table>

The experimental drag data will be assumed constant for constant IAS.

The efficiency measurements below should be considered with reference to the Betz limit which states that the theoretical maximum turbine energy extraction from continuous fluid flow is 59.3%.

The available power at sea level is;

\[ P_{flow} = \frac{1}{2} A \rho V^3 \]

\[ A = \pi \times \left( \frac{16 \times 0.0254}{2} \right)^2 \]

\[ \rho = 1.2 \]

\[ V = 20, 25, \text{ or } 30 \text{ m/s} \]

So the flow power available = 622, 1216, and 2101 Watts respectively.

The efficiency of the turbine is based on the mechanical power absorbed. Figure 10-13 shows the mechanical power absorbed. Experimental error is present and since the true generator efficiency is unknown, a conservative estimation would give mechanical power absorbed as 30, 60 and 100 Watts respectively. Hence the efficiency of regeneration of power and thus energy conversion from the turbine generator system due to drag during a period of descent is estimated to be 4.8%, 4.9% and 4.8% respectively for airspeeds of 20m/s, 25m/s and 30m/s.

Table 10-4 Propeller Turbine Efficiency
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<p>| | | | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>25</td>
<td>1216</td>
<td>60</td>
<td>717</td>
<td>4.9</td>
</tr>
<tr>
<td>30</td>
<td>2101</td>
<td>100</td>
<td>1240</td>
<td>4.8</td>
</tr>
</tbody>
</table>
11 Conclusions and Recommendations for Future Work.

A Hybrid Aircraft Powerplant has been designed, built, tested and analyzed and found to improve aircraft performance, efficiency and effectiveness under certain conditions. Propeller and engine efficiency can be maximized while simultaneously increasing aircraft operating flexibility. Methods of sizing powerplant components on the basis of overall aircraft efficiency have been demonstrated.

This work complements other investigations on the subject while providing a more visible intuitive and practical approach. Integrated computer simulations would enable much more flexibility with the many complex variables which have been largely held constant or linearized here. However the approach presented here clearly indicates the advantages which are available from a hybrid aircraft powerplant, and therefore, a justification for more development of sophisticated methods.

A large part of the overall efficiency analysis and prediction relies on accurate propeller and engine characteristic data. While the inherent thermodynamic and fluid mechanical complexity of obtaining engine data will likely continue to rely on empirical testing methods, useful propeller data has become much easier to obtain from computerized methods. Computerized Fluid Dynamics can enable a large range of alternate propeller types and configurations to be characterized quickly and economically. Such data can then easily be integrated with further simulation or used with the load curve plotting techniques as presented in this thesis.

Given the importance of powerplant dynamometry, and notwithstanding the successful results of the unit developed for this work, a more precise and accurate system is called for. The utilization of high quality COTS electronic load cells would be appropriate. The problem of accurate measurement and seamless recording of air density and temperature variations as well as other important variables such as precise continuous fuel flow, should be included in a fully integrated dynamometer.

It has been shown in this thesis that the benefits of the Aircraft Hybrid Powerplant can be summarized as;
Conclusions and Recommendations for Future Work.

**Self Starting;**

Self Starting is clearly advantageous for remote autonomous and independent UAV operation. Increases mission flexibility, reliability and survivability.

**Boost Power;**

The ability to provide additional power creates opportunities for use of finely tuned fuel efficient engines. The increased power capability markedly improves take-off, climb and dash speed performance.

**Battery Charging;**

Increases payload, range and endurance capability for most mission scenarios. Generating electrical energy from fuel rather than batteries minimizes take-off weight in most cases, and allows the possibility of wind turbine regeneration.

**Regulation of Thrust Characteristic;**

Electronic EM controls allow more precise thrust control characteristics than single mode IC engine power control system.

The capability for precision thrust control was not specifically studied in this work, but is a natural consequence of utilizing the electrical propulsion equipment. Conventional fixed wing aircraft generally do not require particularly high precision thrust control but a clear application where it may be essential is Vertical Take-off and Landing (VTOL) types of aircraft using vectored propeller thrust. Whereas helicopter VTOLs vary thrust utilizing variable pitch rotors, a VTOL using a fixed pitch propeller may rely more on precise control of RPM. The response characteristic of an internal combustion engine is significantly less precise than an equivalent electric motor. Hence a hybrid powerplant would provide the inherent power per weight advantage of the engine with the control of the electric motor.

**Availability of Electronic Control and Conversion Equipment;**

An absolute requirement for hybrid powerplant development is the availability of suitable electronic controllers and power conditioners. COTS equipment available in the hobby market at present shows many promising capabilities but none has been
Conclusions and Recommendations for Future Work.

found to be particularly suitable. Personal communication with several manufacturers has so far elicited limited interest in the hybrid requirements. One UAV manufacturer is known to be producing a suitable ESC but the limited market scale at present is reflected in a price which excludes viability for low cost UAS. It would seem that the benefits shown for high performance commercial UAVs would be just as valuable for mass market hobby model aircraft users. The application for small hybrid aircraft powerplants shares many features with other developing commercial ESC equipment such as high efficiency electric or hybrid scooters, where energy conservation and mass constraints are important factors. Perhaps the confluence of technical and market factors will see the emergence of a range of low cost power electronics suitable for small UAV hybrid powerplants in the near future.

Larger Scale and Recent Developments;

The full automation of an aircraft will ultimately require complete autonomous management of thrust and electrical power supply. This can best be accomplished using an electrical prime mover, however the energy density of liquid fuelled engines will likely be superior for the near and medium term future. Implementation of hybrid powerplant options will yield benefits for both efficiency and effectiveness for UAV operations and conventional manned aircraft.

Since the original manuscript of this work was completed during 2008, further developments in Hybrid Powerplant for both UAS and larger manned aircraft have occurred to date (2012). Notable examples include, Flight Design [35], producing a Rotax parallel hybrid powerplant prototype. The Boeing Company has released a technology concept design for a hybrid powered passenger airliner, the “Sugarvolt” [36]. Politecnico di Torino has developed a record breaking serial hybrid electric fuel cell aircraft the “ENFICA-FC” project [37]. Siemens has demonstrated a series hybrid electric wankel engine powered aircraft, the “DA36 E-Star” [38] demonstrated at the Paris airshow. Scaled Composites has developed a series hybrid technology demonstrator Personal Air Vehicle (PAV), the “BiPod” [39]. A significant advantage of hybrid propulsion techniques is to allow distributed propulsion. A single high performance internal combustion engine can supply multiple electric motor thrust units. These can be located around an airframe to optimise propulsive effectiveness,
Conclusions and Recommendations for Future Work.

efficiency, as well as aerodynamic and mass distribution properties. No doubt many other examples are being developed around the world.

Recent work at the Wright-Patterson Airforce Base by Hiserote [40], Ausserer [41] and by Hung [42] has further explored hybrid propulsion concepts.

The advantages of Hybrid Propulsion for aircraft can synergize with modern automated navigation and control technology to yield UAVs and Personal Air Vehicles (PAVs) and other aircraft types with superior utility and capability for a wide market.
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