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Hybrid-electric propulsion integration in unmanned aircraft

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ABSTRACT

Hybrid-Electric Propulsion Systems (HEPS) have emerged as a promising area of research in aerospace engineering as they combine the complementary advantages of internal combustion and electric propulsion technologies while limiting the environmental emissions. Despite the promising benefits, the insufficient energy densities and specific energies of electrical storage devices are major challenges as they induce severe weight and volume penalties. Significant opportunities are nonetheless emerging thanks to optimised propulsive profiles, energy harvesting techniques and more electric aircraft technologies. To support further research on hybrid electric aircraft, the aim of this study is to develop a HEPS retrofit design methodology for existing Remotely Piloted Aircraft Systems (RPAS). The implemented HEPS models use power state variables, allowing more accurate predictions of energy converter efficiency than with power-based approaches. Data from commercially available products is introduced and a case study is presented assuming a reference RPAS platform and performing parametric studies for traditional, electric and hybrid configurations. Range and endurance performances are investigated in depth and the most significant dependencies on design parameters are analysed. The results suggest that HEPS technology represents a viable trade-off solution in small-to-medium size RPAS, promoting the mitigation of noxious and greenhouse emissions while providing adequate range and endurance performance.

1. Introduction

Modern aircraft propulsion technology revolves around the use of petroleum-based Internal Combustion Engines (ICE), in the form of either Aviation Turbine Fuel (ATF) or Aviation Gasoline (AvGas). Despite their widespread adoption, these fossil fuels have a significant impact on the environment, both in terms of pollutants such as Carbon Monoxide (CO), Nitrogen Oxides (NO_X) and Unburnt Hydrocarbons (UHC) and in terms of greenhouse emissions, the principal of which is Carbon Dioxide (CO_2). Improvements are therefore currently pursued in alternative fuels and hybrid-electric technologies [1]. Some non-mainstream propulsion technologies are expected to mitigate considerably the environmental impacts in aviation [2]. One of the most promising concepts for aircraft integration of hybrid-electric technologies exploits the synergy with distributed propulsion configurations [3–7]. Distributed propulsion is based on subdividing the desired thrust across a number of smaller engines with benefits in terms of noise reduction, higher reliability, shorter take-off and landing distances, better specific fuel consumption and improved stability [3,5,7]. From a research and development perspective, small size Remotely Piloted Aircraft Systems (RPAS) are seen as a stepping-stone, enabling experimental scaling studies for larger aircraft applications. This paper therefore focusses on hybrid RPAS, employing a combination of ICE and electrical propulsion systems. Medium-large Blended-wing-Body (BWB) transport aircraft integrating distributed hybrid-electric propulsion technology are assumed as the long-term target for this research as they can combine most of the benefits offered by all these emerging technologies. Various emerging BWB concepts are proposing hybrid-electric distributed propulsion configurations and the optimisation of size and number of ICE systems and electric motors is an areas of active research.

An electric aircraft concept was considered in conjunction with research projects on solar-powered aircraft undertaken by NASA and AeroVironement Inc. in the early 1970s. Among the various emerging design concepts, solar powered aircraft such as the Helios Prototype typically featured a multi-propulsor arrangement consisting of brushless electric motors [7]. Electric aircraft research highlights potential for future commercial exploitation, however

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encounters some fundamental technological limitations. Weight considerations are among the key challenges, due to components such as batteries, electric motors, generators and converters introducing significant penalties while still not achieving the typical range and endurance performances of ICE propulsion. Hybrid electrical propulsion can nevertheless help increase system reliability and power distribution/quality. One notable approach is the use of renewable energy sources by implementing suitable harvesting technologies. These may include solar radiation and wind speed, for instance exploiting solar Photovoltaic (PV) array, Fuel Cell (FC) stack and battery with a provision for onsite hydrogen (H_2) generation, in use of electrolyser and H_2 tank. Harvesting energy in the most efficient way possible and optimizing the propulsion configuration and mission profile are among the the major difficulties. Energy harvesting from renewable sources as opposed to relying on just one power source (ICE) can enhance endurance and range performances, increase safety and decrease carbon emission. This paper focusses on piston engines and electric motors for light aviation use. Range and endurance can be determined for both piston propeller and jet aircraft via the *Breguet equation*. Estimations of range and endurance for electrical aircraft are less established, but consideration of the electrical energy storage nature suggests equating power delivered by the battery (accounting for losses due to propeller motor) and the motor controller, to the power required to overcome drag as an acceptable method to produce initial performance estimates. Subsequently, the actual behaviour of the battery and its effective capacity/energy density can be analysed in a more comprehensive design trade-off study. This paper will focus on a Hybrid-Electric Propulsion System (HEPS) retrofit for RPAS. Critical parameters for HEPS considered are flight endurance, range and velocity required to perform RPAS missions such as intelligence, surveillance and reconnaissance (ISR).

1.1. Range and endurance performance of ICE-powered aircraft

The *United States Air Force Institute of Technology* developed a number of HEPS concepts [8]. One particular concept considered an ICE engine to be activated in the cruise flight phase and provide primary power. In this case, range can be correlated accurately to fuel burn. Fuel flow, symbolized by $\dot{m}_f$, becomes the reference metric. The amount of fuel available m_f is also required to assess potential energy. If we assume a steady level condition (cruise), range becomes a function of fuel flow, fuel available and velocity, as outlined in equation (1). This equation assumes that the weight of the aircraft remains constant, however this is not accurate. Fuel is burned and thus weight is reduced through the flight and must be taken into consideration. The reduction in weight also has an effect to the amount of lift and drag generated as the overall weight of the aircraft reduces so will the amount of lift required for steady level flight. Therefore, a modified form of the Breguet equation can be adopted for propeller driven aircraft [9]. This modified empirical relationship is shown in equation (2), where W_1 and W_0 is the initial fuel weight and final fuel weight on the aircraft respectively, L/D is represented as the lift-to-drag ratio, and $C_{SFC,}$ is the power specific fuel consumption as defined in equation (3), as the rate of fuel consumption per unit shaft power [8–10]. These equations do not explicitly account for density however this is taken into account when L/D ratio is numerically approximated. Furthermore, equation (2) does not account for wind speeds.

$$R = \frac{m_f}{\dot{m}_f} V_\infty \tag{1}$$

$$R = \frac{\eta_{prop}}{C_{SFC}} \frac{L}{D} \log \frac{W_0}{W_1} \tag{2}$$

$$C_{SFC} = -m_f \big/ P \tag{3}$$

Following equations (1)–(3), to maximize the range for reciprocating engines, design efforts are aimed at:

- Highest propeller efficiency η_{prop}.
- Lowest fuel consumption, C_{SFC}
- Highest W_0/W_1 ratio, equating to largest fuel weight W_f since $W_0 = W_1 + W_f$

Anderson and Payne also present a Breguet endurance equation derived to maximize loiter time, as shown in equation (4), where t_{end} is the endurance time. Similarly to the range equation, the endurance equation accounts for fuel burn during loiter, suggesting to maximize operations at $[c_L^{3/2}/c_D]_{\mathrm{MAX}}$ to enhance endurance performance.

$$t_{end} = \frac{\eta_{prop}}{C_{SFC}} \cdot \sqrt{2\rho_\infty S} \cdot \frac{c_L^{\frac{3}{2}}}{c_D} \cdot \left(\frac{1}{\sqrt{W_1}} - \frac{1}{\sqrt{W_0}} \right) \tag{4}$$

As a result, the maximum endurance for propeller driven airplanes powered by reciprocating engines can be achieved in the following conditions:

- Highest propeller efficiency, η_{prop}
- Lowest fuel consumption, C_{SFC}
- Highest W_0/W_1,
- Aerodynamic condition, $\frac{C_L^{\frac{3}{2}}}{C_D}$
- Flight at sea level, where density is maximum

1.2. Range and endurance performances of battery-powered aircraft

In case of battery-powered aircraft, the mass remains constant throughout the flight, hence simplifying the range equation. The range of an aircraft is defined by flight speed and time:

$$R = V_\infty \cdot t \tag{5}$$

Specifying endurance for battery powered RPAS by simply adapting equation (4) is not possible for battery powered aircraft, due to singularity of the solution in the case of constant weight (i.e., $W_1 = W_0$). Similar considerations apply to the range equation (2). This reinstates the original range strategy suggested by Payne as appropriate based on battery capacity, E_{BAT}, and the average current and voltage associated with the battery drain, respectively $\overline{I}$ and $\overline{V}$ [8,10]. The battery capacity can be further elaborated as the product of the mass of the battery and of the specific energy of the battery. Consequently, the maximum endurance conditions will correspond to the velocity of minimum power required. In particular, the endurance t_{MAX}, measured in units of time, can be calculated as a function of mass ($m_{battery.}$) and mass specific energy, E^* of the battery as:

$$t_{MAX} = E_{BAT} / \overline{IV} = \frac{m_{battery} \cdot E^*}{P_{Drain}} \tag{6}$$

Introducing this flight time into the range equation (5) yields: