ANALYSIS OF BATTERY WEIGHT REQUIREMENTS IN THE DESIGN OF HYBRID ELECTRIC POWERED AIRCRAFT

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Abstract

Hybrid electric propulsion aircraft uses traditional engine to drive generator to provide electricity, and is equipped with rechargeable energy storage batteries. Both the generator and the battery can provide power for multiple motors / propellers distributed on the wings or fuselage. The weight of the battery is constrained by limited onboard space and load, and the minimum weight that meets the power and energy requirements of the flight mission is usually used as the design objective. Based on a new hybrid power system scheme, this paper studies the calculation method of the battery weight requirement of the aircraft during the takeoff and climb phases. The analysis results show that the battery power requirement during the climb phase can be converted into a battery weight requirement, which is higher than the battery weight requirement calculated according to the power requirement during the takeoff phase; in addition, the total energy requirement of the battery is the accumulation of the requirements of the takeoff and climb phases, which needs to be considered in the conceptual design stage of the aircraft.

Keywords: Hybrid electric propulsion, Distributed propulsion, Battery weight requirements, Aircraft design

1. Introduction

The hybrid distributed electric powered aircraft is an air vehicle with a novel propulsion system that utilizes traditional engines to drive generators to get electricity, providing power to multiple motors distributed on the wings or fuselage, and driving propellers to provide thrust. It can improve the overall aerodynamic performance and propulsion efficiency of the aircraft, expand its range, load capacity, and flight conditions, while also having noise reduction and emission reduction capabilities, and advantages of increasing flight control fault tolerance and safety [1, 2]. Especially, the use of distributed propulsion with "blown lift wings" [3] and high-energy and efficient hybrid systems can greatly improve takeoff weight and available onboard power, making hybrid distributed electric propulsion systems an ideal solution for the power plant of vertical/short takeoff and landing (VTOL/STOL) fixed wing aircraft [4, 5]. In design of such aircraft, the three major balance constraints of lift-weight balance, thrust-drag balance, and energy supply-demand balance under different flight conditions need to be comprehensively considered, especially in the mode transition stage, as well as discipline constraints such as aerodynamics, structure, load, and control [6].

The STOL aircraft with hybrid distributed electric propulsion expands the flight envelope of traditional fixed wing aircraft, avoiding the complex structural design and mode conversion control of vertical takeoff and landing aircraft. It has significant advantages of agile takeoff and landing, fast arrival, persistent hovering, high available energy and high loading capacity, and can be used as a new type of military and civilian air support platform [7].

This paper introduces a new hybrid distributed propulsion aircraft energy system scheme at first. Based on the scheme, the calculation method of the battery weight requirement of the aircraft are studied. Then, the battery weight requirements to meet the roll distance objective during takeoff phase, and a certain climb rate objective during climb phase are analyzed, as well as the differences and connections between the two, providing a reference for the conceptual design of such aircraft.

2. Hybrid Power System Scheme

A new scheme of hybrid power system is proposed, as shown in Figure 1. The main power plant is a piston engine that directly drives the cruise propeller, mainly providing power for the cruise phase. The auxiliary power plant consists of multiple electric motors driven by onboard storage batteries, which drive the symmetrically distributed propellers at the front of the wing leading edge. It mainly provides electric propulsion power for takeoff and climb phases, meeting the high-power requirements for short takeoff and rapid ascend. During the cruise phase, the propellers are folded to reduce drag. During takeoff and climb phases, the piston engine also operates at maximum power to drive the cruise propeller. During the cruise phase, if the aircraft needs to perform maneuvering flight and the piston engine cannot provide adequate power, the power system can also drive distributed propellers to supplement the required propulsion power.

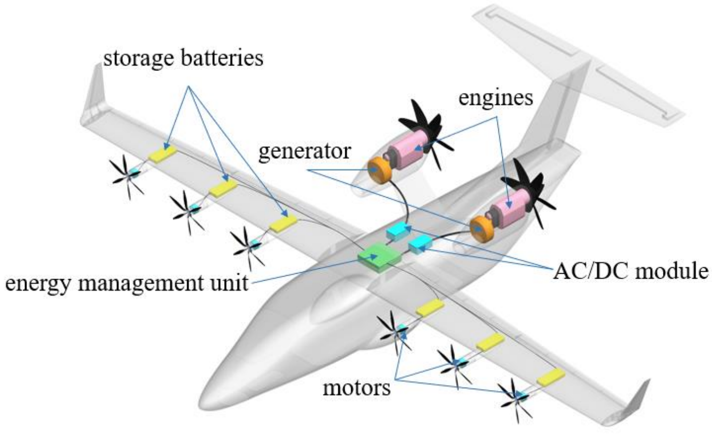

Figure 1 – Hybrid Electric Propulsion System Scheme

3. Method for Calculating Battery Requirements

It can be expected that during the takeoff and climb phases, the battery needs to be discharged at high power for a short period of time, and the weight requirement of the battery can be calculated based on the maximum value required among all phases. During the cruise phase, the aircraft can maintain flight using only the lower power provided by the piston engine, and the remaining shaft power of the engine can drive the generator to operate and charge the storage battery. It is generally believed that the battery energy consumed during takeoff and climb phases is not significant, while the cruising phase takes longer to charge and the generator can provide more energy to fully charge the battery.

3.1 Takeoff Phase

During takeoff, the forces acting on the aircraft include thrust, drag, and rolling friction of the wheels. The friction can be expressed as the rolling friction coefficient μ multiply the weight acting on the wheel (W - L). On a conventional hard runway, the typical value of rolling friction is 0.03 [9].

The acceleration during the rolling process can be derived from Newton's second law. The aircraft acceleration expressed in the equation (1) can be expanded according to the aerodynamic coefficient .

$$\begin{aligned} a &= \frac{g}{W}\left[T - D - \mu(W-L)\right] \\ &= g\left[\left(\frac{T}{W} - \mu\right) + \frac{\rho}{2W/S}\ \ -C_{D0} - kC_L^2 + \mu C_L\ \ V^2\right] \end{aligned} \tag{1}$$

where g is the gravity acceleration, W is the total weight of the aircraft, T is the thrust of the power system , D is the aerodynamic drag, L *is* the aerodynamic lift , ρ is the air density at the altitude, S is the reference area of the wing, C_{D0} is the zero-lift drag coefficient, C_L is the lift coefficient, k is the lift-induced drag coefficient, and V is the flight speed .

The rolling distance is obtained by dividing the speed by the acceleration and integrating it, as shown in the equation (2):

$$s_{to} = \int_{V_i}^{V_f} \frac{V}{a} dV = \frac{1}{2}\int_{V_i}^{V_f} \frac{1}{a} dV^2 \tag{2}$$

The rolling time is obtained by dividing the rolling distance by the speed and integrating:

$$t_{to} = \int_{V_i}^{V_f} \frac{1}{a} dV \tag{3}$$

The takeoff speed V_i must be no less than 1.2 times the stall speed [8]. The stall speed is the speed at which the maximum lift is equal to the weight. The maximum lift coefficient depends on the flap deflection at takeoff, which is taken as 2.25 in this paper.

The rolling distance can be obtained by integrating the equation (2) from V_i to V_f form the equation (4). In the equations (6) and (7), the terms K_A and K_T are defined. K_T contains the thrust term, and K_A contains the aerodynamic term. Through transformation, the above integrals can be converted into integrals of the square of velocity:

$$s_{to} = \frac{1}{2g}\int_{V_i}^{V_f} \frac{d(V^2)}{K_T + K_A V^2} = \left(\frac{1}{2gK_A}\right)\ln\left(\frac{K_T + K_A V_f^2}{K_T + K_A V_i^2}\right) \tag{4}$$

$$\begin{aligned} t_{to} &= \frac{1}{g}\int_{V_i}^{V_f} \frac{dV}{K_T + K_A V^2} = \frac{1}{g\sqrt{K_A K_T}}\left\{\tan^{-1}\left(\sqrt{\frac{K_A}{K_T}}V_f\right) - \tan^{-1}\left(\sqrt{\frac{K_A}{K_T}}V_i\right)\right\}(K_T > 0,\ K_A > 0) \\ &= \frac{1}{2g\sqrt{|K_A|K_T}}\left\{\ln\left(\frac{\sqrt{K_T} + \sqrt{|K_A|}V_f}{\sqrt{K_T} - \sqrt{|K_A|}V_f}\right) - \ln\left(\frac{\sqrt{K_T} + \sqrt{|K_A|}V_i}{\sqrt{K_T} - \sqrt{|K_A|}V_i}\right)\right\}(K_T > 0,\ K_A < 0) \\ &= -\frac{1}{2g\sqrt{K_A|K_T|}}\left\{\ln\left(\frac{\sqrt{|K_T|} + \sqrt{K_A}V_f}{\sqrt{|K_T|} - \sqrt{K_A}V_f}\right) - \ln\left(\frac{\sqrt{|K_T|} + \sqrt{K_A}V_i}{\sqrt{|K_T|} - \sqrt{K_A}V_i}\right)\right\}(K_T < 0, K_A > 0) \\ &= -\frac{1}{g\sqrt{K_A K_T}}\left\{\tan^{-1}\left(\sqrt{\frac{K_A}{K_T}}V_f\right) - \tan^{-1}\left(\sqrt{\frac{K_A}{K_T}}V_i\right)\right\}(K_T < 0,\ K_A < 0) \end{aligned} \tag{5}$$

$$K_T = \left(\frac{T}{W}\right) - \mu \tag{6}$$

$$K_A = \frac{\rho}{2\ W/S}\ \ \mu C_L - C_{D0} - KC_L^2 \tag{7}$$

Integrating equation (4) from any initial speed to any final speed the formula can obtain the roll distance. For takeoff, the initial speed is zero and the final speed is V_f. During the takeoff roll, the thrust varies, so the average thrust value is used in the integration. Because it is integrated over the square of the speed, the average thrust used should be the thrust at about 70% of V_f [9], as shown in the equation (8).

$$T = \frac{P_{\max}}{1/\sqrt{2}\ \ V_f} \tag{8}$$

where $P_{\max}$ is the maximum output power of the entire power system for ground roll, including the maximum power output of the engine and motor:

$$P_{\max} = P_{e,\max}\eta_{p1} + P_{b,\max}\eta_m\eta_{p2} \tag{9}$$

where, P_e and P_b are the output power of the engine and the energy storage battery respectively, the max subscript represents the peak power, η_1 and η_2 are the efficiency of the rear and front propellers respectively, and η_m is the motor efficiency. The mass fraction of the battery required for takeoff is calculated according to the maximum power demand:

$$MF_{b,to,P} = \frac{P_{b,\max}}{W}\frac{g}{\rho_{b,P}} \tag{10}$$

On the other hand, according to the maximum energy demand:

$$MF_{b,to,E} = \int_0^{t_{to}} \frac{P_{b,\max}}{W} \frac{g}{\rho_{b,E}} dt = \frac{P_{b,\max}}{W} \frac{g}{\rho_{b,E}} t_{to} \tag{11}$$

where $\rho_{b,P}$ and $\rho_{b,E}$ are the power density and energy density of the battery (relative to the battery mass).

3.2 Climb Phase

Let R/C be the climb rate [8]:

$$R / C = \dot{h} = V \sin \gamma = \frac{V(T - D)}{W} = \frac{P_a}{W} - \frac{P_R}{W} = \frac{P_s}{W} \tag{12}$$

where $\dot{h}$ is the speed of the aircraft in the flight altitude direction, that is, the component of the flight total speed V in the altitude direction, γ is the track angle, P_a is the available power, P_R is the power required for level flight, and P_S is the residual power.

From the above equation, we can see that when the residual power-to-weight ratio P_s / W is the largest, the climb rate R/C is the largest. Assuming that the total weight W and available power P_a of the aircraft change little during the climb, to maximize the residual power P_s, it means that the power required for level flight P_R is minimized. Then the speed corresponding to the maximum climb rate is consistent with the minimum speed V_{mp} corresponding to the power required for level flight[8]:

$$V_{R/C,\max} = \frac{1}{\sqrt[4]{3}} \sqrt{\frac{2W}{\rho S}} \sqrt[4]{\frac{k}{C_{D0}}} = V_{mp} \tag{13}$$

The minimum required power is:

$$\left(\frac{P_R}{W}\right)_{\min} = \frac{V_{mp}}{0.866 E_m} \tag{14}$$

where E_m is the maximum lift-to-drag ratio, and $0.866E_m$ is the lift-to-drag ratio corresponding to the minimum required power.

If the climb rate requirement during the climb phase is $(R/C)_R$, the available power-to-weight ratio requirement is

$$\left(\frac{P_a}{W}\right)_R = \left(\frac{P_R}{W}\right)_{\min} + \left(R/C\right)_R \tag{15}$$

The power distribution strategy is to give priority to the power generated by the engine, and the insufficient power is supplemented by the battery:

$$\begin{cases} \dfrac{P_e}{W} = \left(\dfrac{P_e}{W}\right)_{\max} \\ \dfrac{P_b}{W} = \dfrac{1}{\eta_{p2}\eta_m}\left[\left(\dfrac{P_a}{W}\right)_R - \left(\dfrac{P_e}{W}\right)_{\max} \eta_{p1}\right] \end{cases} \tag{16}$$

where the maximum output power of the engine decreases slightly with the increase of flight altitude.

The mass fraction of the battery required during the climb phase is calculated based on the maximum power demand during the entire climb process:

$$MF_{b,cl,P} = \left(\frac{P_b}{W}\right)_{\max} \frac{g}{\rho_{b,P}} \tag{17}$$

On the other hand, according to the maximum energy demand:

$$MF_{b,cl,E} = \int_0^{t_{cl}} \frac{P_b}{W} \frac{g}{\rho_{b,E}} dt = \int_0^{h_{cr}} \frac{P_b}{W} \frac{g}{\rho_{b,E}} \frac{1}{R/C} dh \tag{18}$$

The battery mass fraction required in the climb phase is the maximum value of the above two

$$MF_{b,cl} = \max\left(MF_{b,cl,E}, MF_{b,cl,P}\right) \tag{19}$$

Finally, the battery mass requirement needs to comprehensively consider the maximum battery power demand at each flight stage and the maximum total power demand of the entire flight mission profile, and take the maximum value of the two. For the problem studied in this paper, it is necessary to calculate the battery discharge power demand at different times during the takeoff and climb stages and the total energy demand of the entire flight stage. It should be noted that since the battery is in a continuous discharge state during takeoff and climb, the total energy demand of the battery is the accumulation of the demand in the two stages, so the calculation of the battery weight fraction should be corrected to:

$$MF_b = \max \ MF_{b,to,E} + MF_{b,cl,E}, MF_{b,to,P}, MF_{b,cl,P} \tag{20}$$

The battery weight requirement calculation uses the parameters shown in the following table:

Table 1 Aircraft parameters used in battery weight requirement calculation

parameter	describe	value
W	total weight of aircraft (kg)	3000
S	wing reference area (m^2)	20
R/C	specified climb rate (m/s)	5
C_{D0}	zero lift drag coefficient	0.0305
k	lift-induced drag coefficient	0.036222
$C_{L\max}$	maximum lift coefficient	2.25
$P_{e,\max}$	Engine peak power (kW)	207.41
η_1	Rear propeller efficiency	0.8
η_2	Front propeller efficiency	0.8
η_m	Motor efficiency	0.912
$\rho_{b,P}$	Battery power density (kW/kg)	2.16
$\rho_{b,E}$	Battery energy density (Wh/kg)	176

4. Analysis of Battery Requirement During Short Takeoff Phase

By setting different maximum power supply of batteries, the corresponding take off ground roll distance can be calculated, as shown in Figure 2. On the contrary, in aircraft design, once the performance requirement for roll distance (500 meters) is proposed, the takeoff power demand (399.8kW) can be obtained from Figure 2. Subtracting the power provided by the engine and considering the efficiency of propulsion, the battery power supply can be obtained as 211.07kW. Based on the battery power density of 2.16kW/kg (considering grouping efficiency), the battery weight demand can be calculated as 97.77kg. By calculation, the takeoff and ground roll time is 24.83 seconds. Using a battery energy density of 176Wh/kg, the battery weight requirement can be calculated to be 10.35kg (considering a discharge depth of 0.8). The latter is significantly smaller than the former, therefore 97.77kg is taken as the battery weight requirement for the takeoff phase.

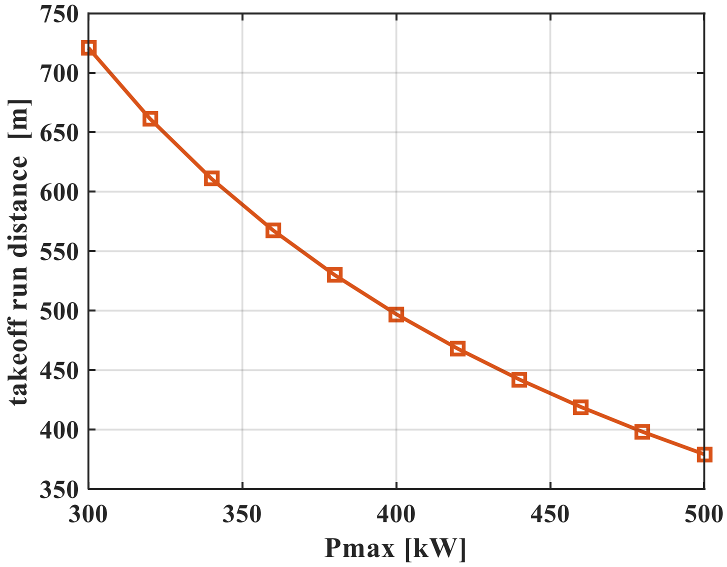

Figure 2 – Takeoff Ground Roll Distance Corresponding to Different Maximum Power Supplies

5. Analysis of Battery Requirement During Short Takeoff Phase

Assuming the aircraft takes off from sea level and climbs to a cruising altitude of 6 km, different average *R/C*s are set, and the total energy output in battery discharge process while climbing is calculated. Combined with the battery energy density, the corresponding battery demand is obtained as shown in Figure 3. From the figure, it can be seen that using the required battery weight during the takeoff phase to provide supplementary power for climbing can only support a low *R/C* of 2.39m/s, with a climbing time of 41.8 minutes. Usually speaking, the RC as well as the climb maneuverability is the significant performance parameter in aircraft design. Assuming a mediocre *R/C* of 5m/s as a design objective, the required battery weight is 296.22kg according to the battery calculating method, which is much greater than the battery demand calculated based on power density during takeoff. In addition, as the two phases are continuous, the battery weight requirement should be calculated (306.57kg) taking into account the battery energy consumption during the takeoff phase, which in turn affects the aircraft conceptual design.

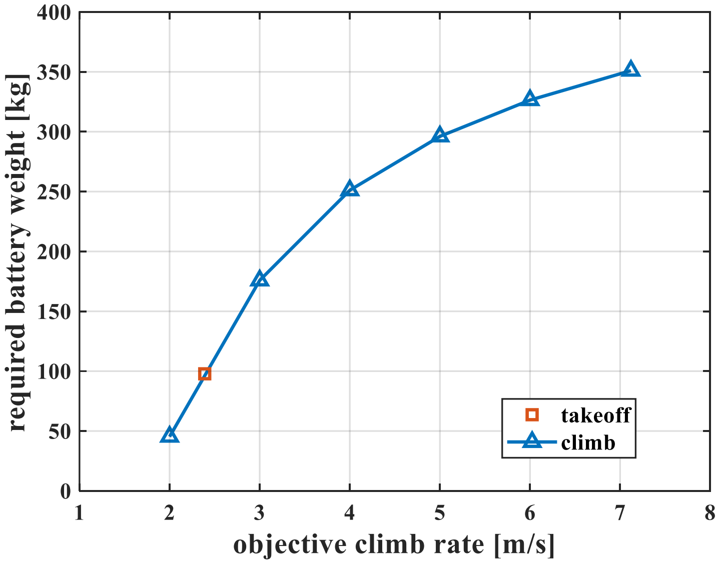

Figure 3 – Battery Demand Corresponding to Different Climb Rates

6. Conclusion and Future Work

In the design of hybrid electric powered aircraft, the current battery weight is configured according to the high-power demand during short takeoff, but during the climb phase, the battery needs to be continuously discharged at high power for a relatively long time, resulting in a significant shortage of battery capacity. The total requirement for battery capacity during the takeoff and climb phases can be converted into battery weight demand, which is higher than the battery weight demand calculated based on power demand during the takeoff phase. Thus, the aircraft should carry heavier batteries than the original design scheme.

After the battery weight is corrected and further significantly increases in the aircraft design scheme, and theoretically, the available battery power in the power system also increases. According to the method of analyzing takeoff performance in this paper, the ground roll distance can be further shortened, and the takeoff performance of the aircraft design scheme can be correspondingly corrected. The selection of battery discharge power and the configuration of battery released energy for each phase need to be investigated. This is also the research work remains to be completed.

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