Comparative study of the energy efficiency of diesel and hybrid powered light aircraft

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Abstract. In recent times many research efforts are aimed at the development of aircraft hybrid propulsion technology as a means to minimize the aircraft environmental impact. The objective is to provide an additional assessment of the energy efficiency of the hybrid powered light aircraft in comparison with current state of the art Diesel powered aircraft. Typical 4-seat propeller airplane is chosen as a prototype and its flight performance model is created using open source data. The comparison is performed for a realistic flight profile including the power demands for go-around and flight to alternate destination airport. Parallel hybrid architecture is considered. A parametric study with variation of the mass, efficiency and energy capacity parameters of the hybrid power train is performed in order to determine the parameter regions where the hybrid power train has advantage over the Diesel one. The present study is based on the previous experience of the authors in assessing the energy efficiency of alternative power sources for the aviation.

1. Introduction
Presently the electric hybridization of the aircraft power plants is envisioned as a necessary step towards fully electric aircraft propulsion. Profound study of the subject is performed by Brelje and Martins [1]. The main conclusion of their work is that successful implementation of electric propulsion, including hybrid electric, remains challenging and should be an object of multidisciplinary design analysis and optimization. More practical efforts in this direction are undertaken in the horizon 2020 program MAHEPA, targeted at designing and testing real hybrid aircraft [2].

A massive effort in the field of aircraft hybrid-electric propulsion was undertaken by a team of researchers from FH Aachen University of Applied Sciences and RMIT University [3-7]. Booth serial hybrid [3, 6] and parallel hybrid [4, 5] architectures were investigated in regard with the sizing of the hybrid propulsion system (HPS). Some important conclusions, concerning the optimum wing loading and power loading of the hybrid aircraft were drawn. As baseline aircraft for the studies were chosen Cessna-172S and Cirrus SR-22 light aircraft. In [7] the research was outspread also on larger aircraft and unmanned aircraft.

Researchers working on MAHEPA also published the results of their studies on sizing serial HPS for light commuter and for regional passenger aircraft [8]. Additional degree of freedom in this study was the inclusion different energy management strategies during the flight.

The objectives of the present work are somehow different from those of the publications listed above. The main objective is not to size and optimize the HPS, but to compare its energy efficiency with that of the state of the art present days technology. Another objective is to assess the viability of
HPS as a drop-in replacement of the conventional aircraft engines for light aircraft. For this reason the approach chosen in this work is to create a flight performance model of an existing light airplane with an internal combustion engine (ICE), modify the model for a hypothetic HPS, and compare the results of the both modes over typical flight scenario.

2. Baseline aircraft and hybrid system description

2.1. Baseline aircraft
As a baseline aircraft Diamond DA-40 NG light aircraft is chosen. It is diesel-powered, single engine, composite structure aircraft. The combination of the low weight and smooth aerodynamic shape of the composite structure and the diesel engine with very low specific fuel consumption puts the DA-40 NG among the most energy efficient airplanes of its class [9]. The Austroengine AE-300 turbo diesel engine of the aircraft is rated at 123.5 kW of takeoff power, with dry mass of 186 kg. The turbo diesel possesses also good high-altitude performance, which contributes to the good cruise performance of the aircraft. The airplane is capable of taking 84 kg of diesel fuel in its main tank.

Among the disadvantages of the diesel engine of the baseline aircraft are the relatively low specific power of 0.66 kW/kg, together with relatively high nitrogen oxides and particulate matter (PM) emissions and higher prize, compared to the aviation gasoline engines.

2.2. HPS structure.
A general assumption of this study is that the HPS replaces the diesel engine without any ‘external’ changes of the aircraft. This includes:
- The propeller remains the same;
- Aerodynamic performance is unchanged (outer shape of the airplane is generally unchanged);
- Maximum take-off mass of the aircraft is unchanged;
- Useful load is unchanged;
- The power ratings of the power plant (take-off, maximum continuous, idle ratings) remain as of the diesel engine, in accordance with DA-40 NG Aircraft flight manual (AFM).

Another assumption is that a serially produced aviation ICE should be used as part of the HPS to reduce the development costs and time. A logical choice is the Rotax 914 turbocharged spark ignition engine, rated at 84.5 kW at takeoff and installation mass of roughly 75 kg. As alternative engine Rotax 915, rated at 104 kW was considered, but it was rejected, mainly because of its higher specific fuel consumption. It should be stated that the ICE optimization for the HPS is a separate, complex task, which is not an object of the current research.

Given these assumptions the practically feasible HPS may look as the one shown on figure 1. It is arguably the simplest form of parallel HPS. The freewheel unit is added as a safety feature in case of ICE failure. The permanent magnet electric motor is mounted directly on the propeller shaft.

![Figure 1. Simplified scheme of the parallel HPS investigated in the present work.](image)
### Table 1. Specific properties of the electric components of the HPS.

| Electric motor specific power | $P_{sp.EM}$ kW/kg | 6 |
| Motor power electronics specific power | $P_{sp.PE}$ kW/kg | 15 |
| Electric efficiency of the system | $\eta_{el}$ | – | 0.85 – 0.95 |
| Battery specific energy | $E_{sp.Bat}$ Wh/kg | 200 – 500 |

With regard to the relatively low energy density of the batteries the following energy strategy is adopted:

- At take-off and climb the ICE (Rotax 914) provides its take-off or maximum continuous power respectively. The electric motor provides the rest of the power needed to reach the power of the original diesel engine;
- At cruise the ICE provides the required power. If the required power is higher than the maximum continuous power of the ICE at the current ambient conditions, the ICE works at maximum continuous power and the electric motor provides the excess power needed.
- At descent the electric motor is not working.

With regard to this strategy the electric motor should provide around 40 kW at take-off and climb. Also an electric motor power reserve should be considered as a safety feature in case of ICE failure. The electric power reserve needs special study, but in this work a guess of 20% is accepted, resulting in electric motor maximum power $P_{EMmax}$ of 50 kW. With regard to the specific power of the motor and its electronics (table 1) the weight of the electric part of the HPS is roughly 12 kg.

The sizing of the battery is made with the presumption that the electric energy will be used for take-off and climb, but also for eventual go-around and climb to safe altitude. An initial guess, providing enough safety margin is to have enough stored electric energy for 30 minutes of the electric motor full power ($t_{FullPWR}$). In this case the mass of the batteries is determined as follows:

$$m_{Bat} = \frac{P_{EMmax}t_{FullPWR}}{\eta_{el}E_{sp.Bat}} \quad (1)$$

For the values of $E_{sp.Bat}$ and $\eta_{el}$ in table 1 the mass of the batteries will be between 147 kg ($E_{sp.Bat}=200$ Wh/kg, $\eta_{el}=0.85$) and 52.6 kg ($E_{sp.Bat}=500$ Wh/kg, $\eta_{el}=0.95$).

With regard to the assumptions stated in p.2.2. the power plant mass equilibrium between the baseline power plant (diesel) and the HPS is expressed in equation 21:

$$m_D + m_{Dfuel} = m_{ICE} + \frac{P_{EMmax}P_{sp.EM}}{\eta_{el}E_{sp.Bat}} + \frac{P_{EMmax}P_{sp.PE}}{\eta_{el}E_{sp.Bat}} + m_{Bat} + m_{HPSfuel} \quad (2)$$

where $m_D=186$ kg is the mass of the baseline diesel engine, $m_{Dfuel}=84$ kg is the baseline aircraft fuel tank capacity, $m_{ICE}=75$ kg is the mass of the ICE of the HPS and $m_{HPSfuel}$ is the fuel capacity of the HPS retrofitted aircraft. For the stated values of the $m_{Bat}$ the HPS retrofitted aircraft fuel will be between 36 kg and 130 kg (nearly as much as in the avgas powered version of Diamond DA 40).

### 3. Aircraft flight performance model

The aircraft flight performance model is realized in Matlab and is based on the model described in [10]. The model is adapted for propeller aircraft, by dividing the power plant sub-model into two parts for the ICE performance and the propeller performance. The generic features of the flight model are described below:

The system of ordinary differential equations (ODE) that describes all of the stages of the flight are:

$$m \ddot{V} = F_x - c_d \frac{pV^2}{2} S - mgsin(\gamma) \quad (3)$$
\[
\frac{dh}{dt} = V \sin(\gamma) \tag{4}
\]
\[
\frac{dL}{dt} = V \sqrt{1 - \sin^2(\gamma)} \tag{5}
\]
\[
\frac{dm}{dt} = -psfcP_{ICE} \tag{6}
\]

where \(V\) is the air speed, \(t\) is the time, \(F_x\) is the engine thrust force, \(\rho\) is the density of the air, \(S\) is the wing area, \(m\) is the mass of the aircraft, \(\gamma\) is the trajectory angle, \(h\) is the altitude above the mean sea level, \(L\) is the distance and \(psfc\) and \(P_{ICE}\) are respectively the power-specific fuel consumption and the shaft power of the ICE.

To solve the ODE system, at every step of the numerical solution an algebraic system consisting of the following equations is to be solved:

\[
c_l = \frac{2mg\sqrt{1-\sin^2(\gamma)}}{\rho V^2 S} \tag{7}
\]
\[
c_d = c_{d0} + \frac{c_l^2}{\pi A e} \tag{8}
\]

Equation 7 expresses the aerodynamic lift coefficient \(c_l\) in the case of rectilinear flight (true for cruise and acceptable for climb and descend flight phases). Equation 8 is the classical expression of the airplane drag polar, where \(c_d\) is the aerodynamic drag coefficient, \(c_{d0}\) is the drag coefficient at zero lift, \(A\) is the wing aspect ratio and \(e\) is the Oswald factor.

The trajectory angle in the phases of climb and descend with constant instrumental airspeed (IAS) is calculated by solving together equation 1 and the following equation:

\[
\sin(\gamma) = \frac{\frac{dh}{dt} - \frac{1}{V}}{\frac{dV}{dt}} = \frac{\frac{dh}{dt} \frac{1}{V} - \frac{dV}{dt}}{1} \tag{9}
\]

The derivative \(dh/dV\) can be calculated, bearing in mind that constant IAS means \(\rho V^2/2 = \text{const}\) and \(\rho\) is a function of the altitude \(h\).

For all of the sub-parts of the model the ambient conditions are presented in terms of altitude above the sea level and deviation from the international standard atmosphere (ISA) temperature at that altitude.

### 3.1. Propeller model

The input parameters for the propeller performance model are the shaft power and the rotational speed provided by the engine, the true airspeed (TAS) of the airplane and the ambient conditions.

The main output parameter is the effective thrust of the propeller, taking into account the interference with the parts of the airplane. The methodology of estimating the effective propeller parameters based on the isolated propeller performance chart and the airplane body characteristic areas is taken from Torenbeek [11, pp.190-194].

The propeller performance charts are represented in the model as look-up tables. As far as there are no freely available charts for the original DA-40 NG propeller –MTV-6-R/190-69, in this study generic characteristics (Hamilton Standard PDB 6101 [11, p.193]) are used.

### 3.2. ICE models

The models are based on look-up tables and polynomial approximation. The engine performance data is taken from the respective Operation manuals (Austro Engine Operation Manual Doc.No.: E4.01.01, Rev.20 and OPERATORS MANUAL FOR ROTAX® ENGINE TYPE 914 SERIES Rev.0).

#### 3.2.1. Baseline engine

The input parameters for the baseline engine model are the required power as a fraction of the maximum available power and the ambient conditions. The influence of the airspeed ram effect on the engine power is neglected.
The output parameters are the shaft power, the shaft rotational speed, and the fuel flow to the engine.

3.2.2. **ICE of the HPS.** The ICE sub-model is constructed as a supplement to the baseline engine model. In accordance with the energy management strategy of the HPS it modifies the output of the baseline model in the following way:

- At take-off mode it recalculates the fuel flow and calculates the required shaft power of the electric motor $P_{EM}$, based on the difference between available power of the baseline engine and the HPS engine;
- At modes between the maximum continuous (max cont.) of the baseline engine and max cont. of the HPS engine, in addition to the fuel flow and $P_{EM}$ the supplement model modifies the shaft speed to be equal to the HPS engine speed at max cont. These modes of operation generally correspond to high speed cruise flight;
- If the required power is less than the available HPS engine max cont. power at the current ambient conditions, the sub-model modifies only the fuel flow to match that of the Rotax 914 at given power and shaft speed.

It should be noted that the shaft speed modification at high-speed cruise modes affect the performance of the propeller so the overall performance of the aircraft is not exactly the same, but this difference is negligible.

3.3. **Airplane aerodynamic characteristics**

The coefficients $c_{d0}$ and $e$, required in equation 8, are initially estimated, using the methodologies given in [11, section 5.3.2.] for ‘clean’ configuration of the airplane and [11, appendix G] for take-off and landing configurations (flaps extracted at 20° and 42° respectively).

The fine tuning of the coefficients for airplane in cruise configuration is achieved by comparing the DA-40 mathematical model results for the true airspeed (TAS) with the data given in chapter 5 of the Flight Manual (AFM). The tuning is made by varying the values of the coefficients until the difference between the calculated and AFM value of TAS at given altitude, engine power setting and ambient conditions is minimized to a sufficient level (in this case 3%). The results of the tuning and verification are shown in table 2.

| AFM, ISA conditions | Initial estimate | Final estimate |
|---------------------|------------------|----------------|
| $c_{d0}=0.033, e=0.8$ | $c_{d0}=0.031, e=0.825$ |

| Power setting, % | TAS, m/s | Calculated TAS, m/s | Difference, % | Calculated TAS, m/s | Difference, % |
|------------------|----------|---------------------|--------------|---------------------|--------------|
| **Flight altitude 610 m** |
| 92 | 69.96 | 68.95 | -1.45 | 70.60 | 0.91 |
| 45 | 49.39 | 46.40 | -6.05 | 48.10 | -2.60 |
| **Flight altitude 1829 m** |
| 92 | 72.54 | 72.10 | -0.60 | 73.80 | 1.74 |
| 45 | 50.42 | 48.80 | -3.20 | 50.50 | 0.17 |
| **Flight altitude 4267 m** |
| 92 | 77.68 | 74.00 | -4.74 | 76.00 | -2.16 |
| 45 | 51.96 | 49.50 | -4.73 | 52.90 | 1.81 |

The drag coefficients for the airplane in take-off configuration are tuned and verified by comparing the model results for take-off run and take-off distance to 15 m above the runway with the data in section 5.3.7 of the DA-40 NG AFM for two altitudes, at ISA. The values $c_{d0}=0.040, e=0.56$ are adopted.
For the airplane in landing configuration there is no appropriate data in the AFM to check the drag coefficients. On the other hand the final approach and landing phase has minor influence on the overall energy efficiency of the flight, so $c_{d0}=0.050$, $e=0.5$ are adopted without further verification.

It should be noted that the obtained drag coefficients produce good model results only together with the propeller and engine sub-models presented above. They are not the ‘real’ aerodynamic drag coefficients of DA-40 NG.

3.4. Flight scenario

For the purpose of comparing energy efficiency of aircraft with different power plants it is assumed that flight at maximum range with maximum fuel and take-off mass is most characterizing. The flight scenario includes also energy reserve (fuel and electric) for go-around and 30 min navigation reserve.

The flight model is constructed to follow close as practical the AFM procedures in terms of IAS, calibrated for sensor errors (CAS), engine power settings (PL), wing high-lift devices configuration, etc. Airspeeds for take-off rotation ($V_R$), initial climb to 50ft/15m ($V_{50}$) and landing approach ($V_{ref}$) are calculated as functions of the airplane mass, in accordance with the AFM. The phases of the flight scenario are presented in table 3.

| Phase description       | PL, % | Airspeed  | Flaps pos., deg |
|-------------------------|-------|-----------|-----------------|
| 1 Take-off run          | 100   | $0 \rightarrow V_R$ | 20              |
| 2 Climb to 15m above runway | 92    | $V_{50}$ | 20              |
| 3 Climb to safe altitude 300m | 92    | CAS=72 kts | 20              |
| 4 Acceleration (in horizontal flight) | 92    | CAS=88 kts | 0               |
| 5 Climb to cruise       | 92    | CAS=88 kts | 0               |
| 6 Cruise at selected altitude | 45.75 | Depending on PL | 0               |
| 7 Top of descend deceleration | 0 (idle) | 0       |
| 8 Descend               | 0 (idle) | CAS=88 kts |                  |
| 9 Deceleration          | 0 (idle) | $V_{ref}$ | 20              |
| 10 Final aproach        |       |           | 42              |

The phases of the reserve flight are the same except that the take-off run and climb to 15m are replaced by acceleration from $V_{ref}$ to CAS=72 kts, the cruise altitude is always set to 1000 m above the sea level and reserve flight duration is set to 30 minutes.

To calculate the flight at maximum range the following calculation procedure is used:

- Reserve flight is reversely calculated, starting from the landing with empty fuel tanks ($m_{fuel} = 0$ kg). The aircraft mass at beginning of reserve flight is determined;
- Descend from cruise altitude and final approach are reversely calculated, starting with the aircraft mass at beginning of reserve flight and determining the aircraft mass at the top of descend and the descend distance and time;
- Take-off and climb are calculated. Aircraft mass flight distance and time at end of climb are determined;
- Cruise is calculated, starting with aircraft mass equal to that at end of climb and ending when precalculated mass at top of descend is reached;
- Finally the data for all flight phases is brought together.

In the case of HPS the electric energy taken from the battery $E_{el}$, for every flight phase is also calculated.

3.5. Energy efficiency calculations

As in [9] the energy intensity is used as a metric to characterize the transport efficiency of the airplane:
where \( m_{fuel} \) is the mass of the fuel spent for the flight in kg, \( H_u \) – the net heat of combustion of the fuel that is used in MJ/kg, \( m_{pld} \) – the payload in kg and \( L_{flt} \) – the distance of the flight in km. The dimension of \( E_i \) could be either MJ·(ton·km)\(^{-1}\) or kJ·(kg·km)\(^{-1}\). In the case of HPS the energy intensity equation becomes:

\[
E_i = \frac{E_{el} + m_{fuel}H_u}{1000m_{pld}L_{flt}}
\]

(11)

4. Simulation results and discussion

By simulating flights at different altitudes and different engine power setting it is found that minimum energy intensity of the baseline airplane \( E_{i_{min}} = 7.09 \) kJ/(kg·km) is achieved at values of PL=52\%, at cruise altitude of 4267 m (14000 ft). For this reason the simulation of the baseline airplane with these parameters is selected as a reference and the simulations of the HPS airplane are made for the same conditions.

4.1. Take-off and climb

The described HPS at PL=52\% uses only the ICE. This means that the electric energy is used only at take-off and climb phases of the flight. For this reason the take-off and climb are analyzed first. It should be noted that the size of the battery do not affect the flight performance at this stage of the flight and only the electrical efficiency plays role. The simulation results are given in table 4:

| Table 4. Simulation results for take-off and climb to 4267 m. |
|-------------------|-------------------|-------------------|-------------------|
| Flight distance, km | 67.243            | 67.396            | 67.423            |
| Duration, h        | 0.378             | 0.379             | 0.379             |
| Fuel consumed, kg  | 9.735             | 6.876             | 6.881             |
| Electric energy, kWh | -                 | 18.65             | 16.71             |
| Energy intensity, kJ/(kg·km) | 19.19 | 16.55             | 16.24             |
| Relative fuel economy, % | 29.37 | 29.31             | 29.31             |
| Relative energy economy, % | 13.72 | 15.36             | 13.72             |

The results show significant reduction of the fuel consumption and to less extent of the energy consumption during the take-off and climb phases of the flight. The electric efficiency of the HPS do not affect strongly these results. It can be concluded that even HPS with current state of technology will have positive local ecological effect in the airport areas.

4.2. Flight at maximum distance

The results for the flight at maximum range are presented in table 5:

| Table 5. Simulation results for maximal distance flight at 4267 m and PL=52\%. |
|-------------------|-------------------|-------------------|-------------------|-------------------|
| Flight distance, km | 1395.52           | 446.90            | 645.67            | 1316.50           |
| Duration, h        | 6.349             | 1.894             | 2.815             | 5.867             |
| Fuel consumed, kg  | 74.816            | 25.240            | 40.766            | 119.921           |
| Energy intensity, kJ/(kg·km) | 7.089 | 7.468             | 8.635             | 12.185            |
| Relative increase of the fuel consumption per hour, % | - | 13.08             | 22.89             | 73.47             |
Because of the lower fuel consumption of the baseline diesel engine at long distances it is superior to the HPS with gasoline ICE. In case of electrically efficient HPS with high energy density battery, it may allow higher fuel capacity and reach the maximum range of the diesel airplane, but at the cost of much higher fuel consumption. At shorter distances the relative weight of the climb and descend phases of the flight where the HPS surpasses the efficiency of the diesel at cruise, the energy intensity of the HPS airplane is much closer to the diesel one. A general conclusion can be drawn, that the optimization of the ICE will continue to play very important role in the HPS.

5. Conclusions
In this study the energy efficiencies of a light airplane with diesel engine and with HPS are compared. The diesel is superior to HPS at long distances. The HPS has distinct advantage over diesel at take-off and climb, which may contribute to the reduction of the emissions in the airport areas. In terms of energy intensity the simulations show that the perfection of the electric part of the HPS and the energy density of the battery do not play as important role as the efficiency of the ICE of the HPS. It can be concluded that the improvement and optimization of ICS will continue to play important role in the light aviation even with the coming of the airplanes with HPS.

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