

Electric Flight – Potential and Limitations

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ABSTRACT

During the last years, the development of electric propulsion systems is pushed strongly, most notably for automotive applications. For ground based transportation, the system mass is relevant, but of relatively low importance. This paper addresses the application of electric propulsion systems in aeronautics, where all systems are much more constrained by their mass. After comparing different propulsion system architectures the focus is cast on the energy storage problem and here especially on battery storage systems. Using first order principles the limitations and required technology developments are demonstrated by introducing battery powered electric propulsion systems to small and medium sized aircraft.

Abbreviations and Symbols

b = wing span [m]

C = capacity [Wh]

D = drag force [N]

E^* = mass specific energy content [Wh/kg]

f_e = empty mass fraction m_e / m

g = gravity acceleration [m/s^2]

L = lift force [N]

m = mass [kg]

$m_{initial}$ = initial mass at takeoff [kg]

m_e = empty mass without fuel and payload [kg]

m_{final} = final mass at landing [kg]

$m_{battery}$ = battery mass [kg]

$m_{payload}$ = payload mass [kg]

m_{pax} = mass per passenger [kg]

PAX = number of passengers

P = power [W]

$P_{battery}$ = electric power provided by battery [W]

R = range [m]

t = time [s]

T = thrust [N]

V^* = volume specific energy content [Wh/ m^3]

v_∞ = flight speed [m/s]

η = efficiency

ρ_∞ = density of air [kg/m^3]

1.0 INTRODUCTION

Stimulated by the progress in technology developments in the entertainment, toy and car industries, the interest in electric propulsion systems for aircraft has grown considerably during the last years. According to many scientific and general publications the storage of electric energy in battery systems promises a clean new world with fantastic new opportunities. This paper highlights the differences between conventional aircraft propulsion systems, burning fossil fuel and aircraft propelled by electric motors. It tries to show the potential but also the limitations of electric propulsion technology for aircraft. This paper contains assumptions about specific weights, energy densities and efficiencies which are sometimes subject to large uncertainties. Therefore all results must be considered with care but the author believes that the main trends and relations are well represented. The numerical data should not be used to derive final conclusions about the performance of any specific product which is mentioned in this paper.

2.0 MOTIVATION

There are several reasons to study alternative propulsion systems and especially electric systems. The observed growth of air transport is clearly visible despite any temporary drawbacks (Figure 1). This growth leads to increased fuel burn and a stronger impact of aviation on the environment. The aviation industry (manufacturers as well as operators) are currently introducing new technologies and operating schemes to achieving a strong reduction in fuel consumption per flight in the order of 15-20%. However it will take many years until all aircraft are replaced with new generation vehicles and on the other hand the growth of air traffic will lead to increasing overall fuel consumption and hence emissions from air traffic.

Besides the problem of emissions and pollution the amount of oil is limited (Figure 2). Even if it is not clear when the oil resources will be depleted, the price of oil has been steadily climbing for many years following a rather unsteady but clearly visible trend.

Attractive features of electric propulsion systems are that they benefit from better efficiency in the energy conversion chain and that they can be considered as locally zero-emission – albeit depending on how the electric energy is produced.

However, introducing electric propulsion systems into ground transportation seems to be relatively simple compared to the implementation in aviation. While ground based vehicles can easier cope with additional mass due to not fully developed electric storage and propulsion systems, aircraft are much more sensible to mass. Furthermore typical trip distances as flown by aircraft are much larger than trips in ground vehicles (cars, buses, trains) and safety standards in aviation are extremely high. Finally one must acknowledge the fact that current aircraft are already very efficient in terms of fuel burn per payload and distance.

All these facts make it no easy task to develop electric propulsion system concepts for aviation, but it is of strong interest and necessary to determine the physical limitations of using electric energy for aircraft propulsion and to work out the requirements and guidelines for future development.

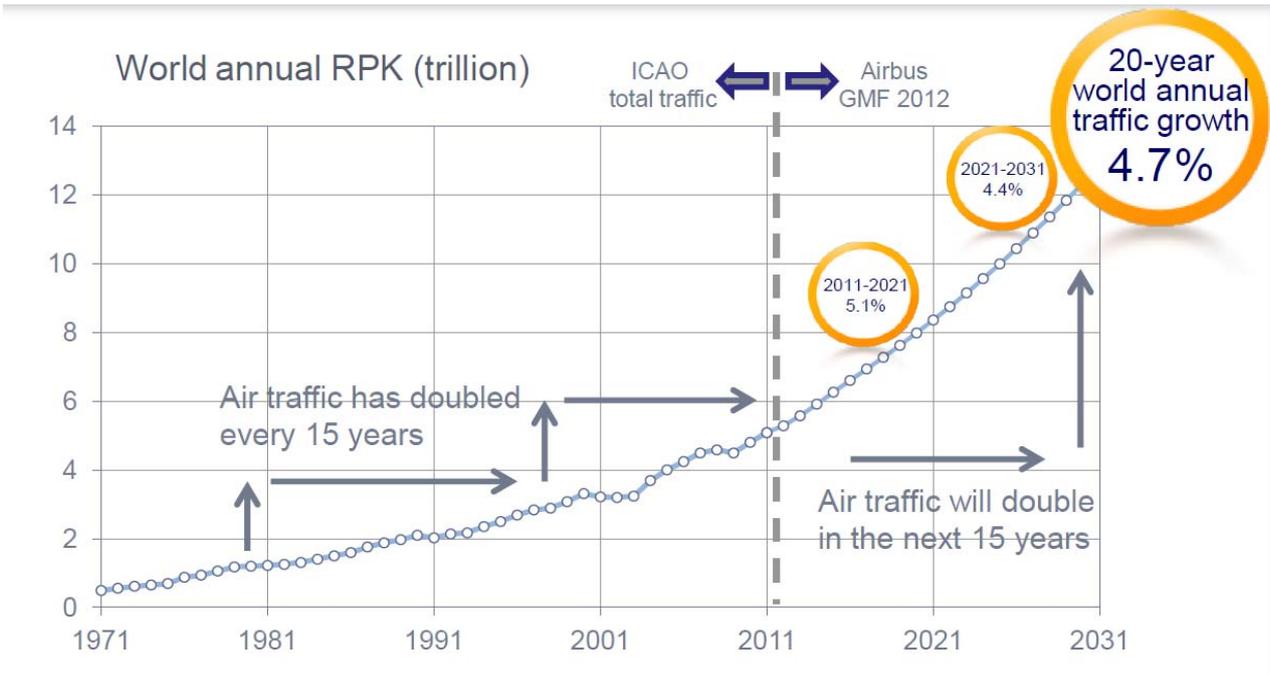


Figure 1 Expected growth of air traffic according to Airbus [1].

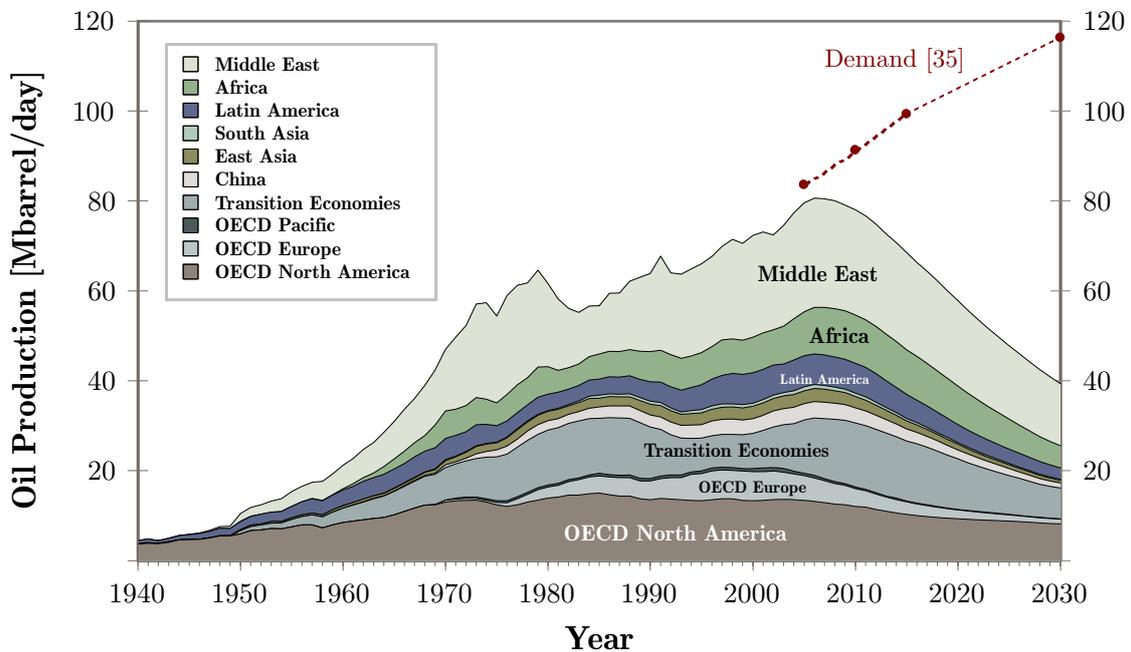


Figure 2 Daily oil production versus time. Extrapolation of oil required by the world [34].

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3.0 THERE IS NOTHING NEW UNDER THE SUN

Electric flight is no new technology. Due to the sensitivity of aircraft to their mass, it was not really feasible for many years. Here only one of the many ancestral lines of electric flight shall be mentioned because it shows a remarkably long development path. Already in 1940 Fred Militky toyed with electric motors to propel model aircraft, but due to heavy brushed motors and lead batteries he was unsuccessful to obtain good results. Nevertheless he followed his idea until he finally was able to introduce a small model airplane into the hobby market in 1960. This free flight model was using a highly efficient electric motor and still small lead batteries. Later in 1972, when Ni-Cd batteries with higher power densities became available he developed the first radio-controlled model which was commercially produced. Finally using the same Ni-Cd batteries he helped to prepare the first electric flight of a manned aircraft. This aircraft was the modified motor-glider MB-E1 which was finally flown in October 1973 by pilot H. Brditschka in Austria. The technology available in these years allowed only for short flights of up to 15 minutes but nevertheless showed the feasibility of electric propulsion for man carrying aircraft. Lack of better battery systems prevented further development. It should be noted that in contrast to some pioneering ultra-light experimental demonstrator aircraft the MB-E1 was a variant of the fully certified airframe HB-3 with minor modifications to carry batteries and electric motor.

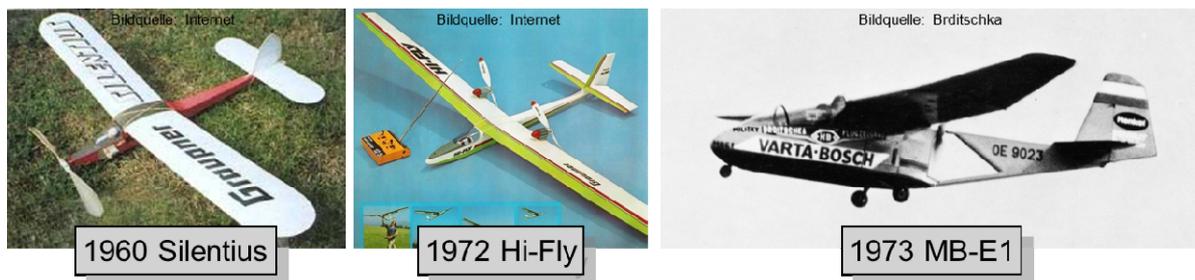


Figure 3 Development steps as performed by Fred Militky leading to man-carrying electric aircraft.

3.1. CONVENTIONAL PROPULSION SYSTEMS

Today's propulsion systems are storing energy in liquid form. This fuel is usually based on oil and is thermally decomposed, i.e. burnt. Technically, with small modifications, these systems are also capable of also using gaseous fuels like hydrogen or natural gas. The fuel is burnt in either piston engines or in gas turbines to produce mechanical power. These machines are connected to a shaft which is linked to an aerodynamic propulsion device, either a propeller (piston-propeller or turbo-propeller) or a fan (turbo-fan). All systems accelerate an incoming mass flow of air to produce thrust. An additional part of the thrust may be created by recovering part of the thermodynamic energy in the exhaust, i.e. by passing the hot gasses through suitable nozzles. In case of propulsion by propellers it is often necessary to use a gearbox to adapt the performance characteristics of the core engine to the characteristics of the propulsion device so that the overall efficiency is maximized. Gearboxes are now also introduced to decouple the fan from the turbine of turbofan engines, thus increasing their total efficiency considerably.

A notable feature of these propulsion systems is that they burn the fuel and the exhaust is emitted into the atmosphere. Besides affecting the environment, this also reduces the mass of the airplane during the flight and therefore affects the performance.

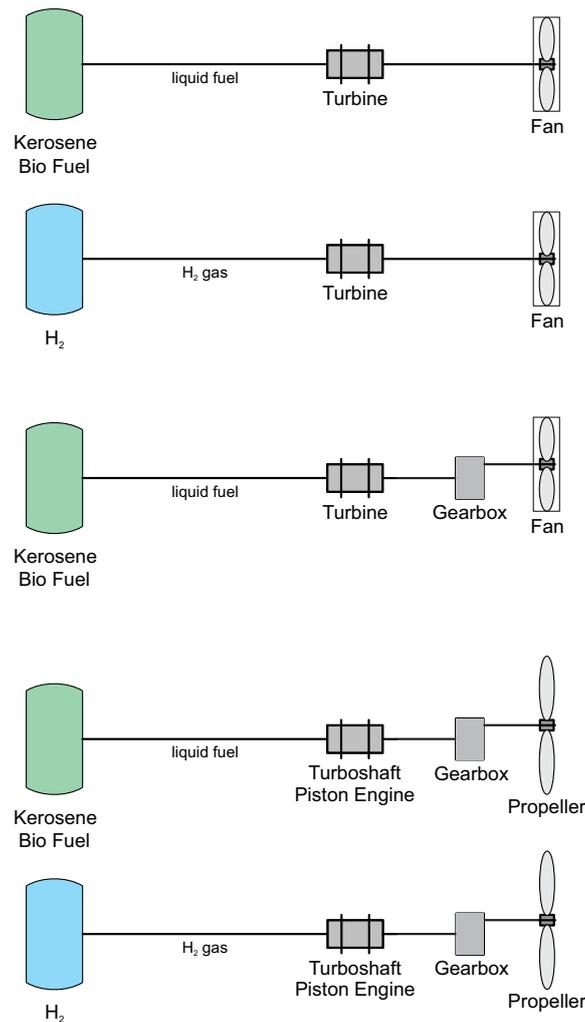


Figure 4 Different propulsion systems exploiting the energy in the fuel by burning it.

3.2.ELECTRIC PROPULSION SYSTEMS

In case of electric propulsion systems there are a large number of possible concepts. They differ in energy storage and conversion. Today mainly two systems are of interest: battery based systems and fuel cell based systems. Both systems are under constant development, driven by automotive and other mobile applications. Most of these developments are currently aiming at power consumptions below 100 kW.

3.2.1 Fuel Cell Systems

These systems store energy in liquid or gaseous form and convert it to electric power by fuel cells. These perform a more or less cold decomposition of the fuel with relatively high efficiency and low emissions. Fuel cell systems have been built for many years for specific applications and a range of power requirements. Still these systems are quite complex and heavy. When the by-products are exhausted to the atmosphere, they lead to a mass reduction during flight like on classical combustion engines. When hydrogen is used for fuel, the emissions contain more water vapor than the emissions of kerosene based fuels and may create more severe contrails in the atmosphere.

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3.2.2 Turbo-Electric Systems

In [3] one of the main motivations to use turbo-electric drive systems is given by the fact that such a system allows to decouple turbine speed and fan speed and thus may be the enabler for distributed multi-fan propulsion systems. The first argument is weakened by recent developments of highly efficient geared turbofan engines (decoupling turbine and fan allows to increase the overall efficiency by 10-20%). However the second argument still holds some opportunities for the development of distributed propulsion systems to improve the propulsive efficiency. These systems can either be driven by conventional fuel or by hydrogen, which still leads to high emission levels and contrail formation.

3.2.3 Battery Systems

Here the energy is stored in batteries which allows for a direct extraction of electric power. The efficiency is limited by the chemical processes occurring during charging and discharging. In most cases the mass of the system is usually not changing, except for some specific air breathing cells like Li-O₂.

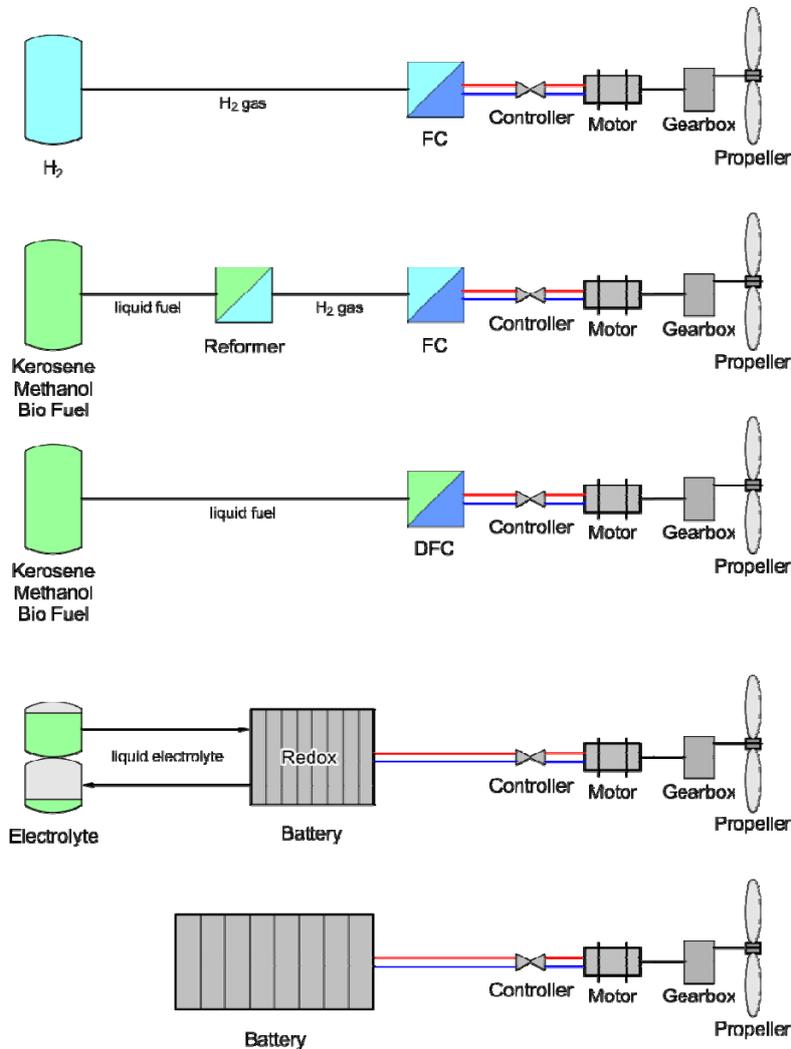


Figure 5 Components of different electric propulsion systems.

3.2.4 Energy Storage

The energy required for a flight must be stored on-board. For application in aircraft the most important parameters are the energy per mass E^* and to a lesser extent the energy per volume V^* . These specific values are shown in Figure 6 for various energy storage systems. It can be seen that even the most advanced current battery storage systems fall short of the parameters of Kerosene. While the factor in specific volume is only about 18, the factor in mass specific energy density is in the order of 60. This is a core problem for application in aviation and imposes strict limits for battery powered systems. The lower volume specific energy content is less critical as long as the aircraft design is not limited by internal volume. Otherwise the vehicle would require larger wings or fuselages or additional external “energy pods” which would lead to losses in overall aircraft efficiency due to larger wetted surface. Of course there are ideas to incorporate batteries and other storage systems into the primary structure, but these are still rather far from practical application. In case of fuel cells hydrogen can e.g. be stored in porous metal, but the weight of these foams is very high and appears unsuitable for aeronautical application. However, such systems have been used in marine applications, where the additional mass is less critical. Table 1 lists some characteristics, advantages and disadvantages of energy storage systems.

As a side-note it is interesting to note that some “bio-fuels”, like milk or cream have rather high energy densities – unfortunately efficient high power conversion technology is still lacking.

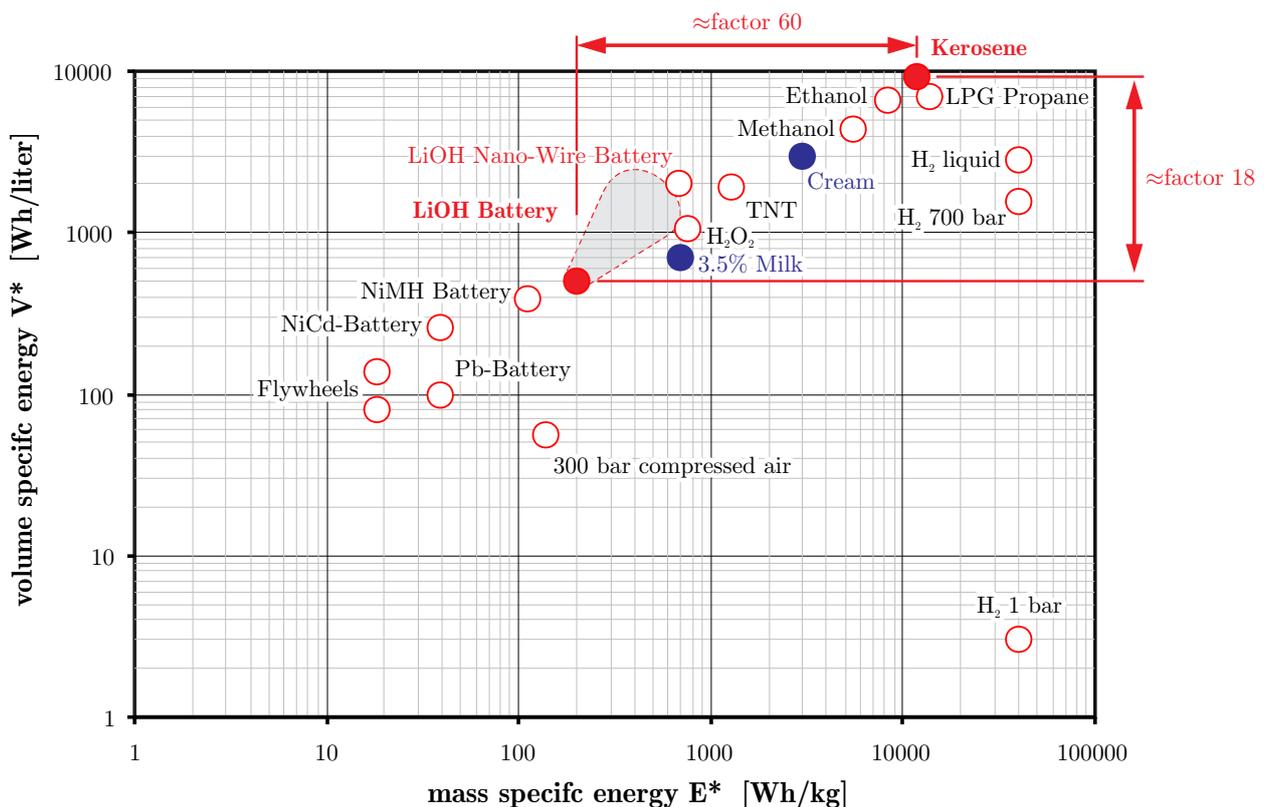


Figure 6 Volume and mass specific energy characteristics of different energy storage systems.

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Kerosene, AVGAS	Can be stored in wing and fuselage tanks. These are often part of the structure and therefore of low mass.
Hydrogen (gas)	Requires high pressure tanks (typically 350 to 700 bars). Tanks are much heavier than the actual fuel and are also a safety risk. If the pressure is reduced, tanks of large volume are required, which affect weight and drag and therefore aircraft performance. Could partially be stored in metallic structures, but practical application in aeronautics is rather limited so far.
Hydrogen (liquid)	Requires cryogenic tanks with insulation to keep the fuel at about -250°C. Tanks are heavy and also require considerable volume with the drawbacks similar to gaseous hydrogen.
Battery	Requires casings with temperature control systems.
Fuel Cell	Requires liquid or gaseous fuel with all the aspects mentioned above. Additional infrastructure like air pumps, water supply or reformer devices is necessary.

Table 1 Characteristics of energy storage systems.

3.2.5 Efficiency Chains

The conversion of energy stored on-board to propulsive power involves several conversion steps. Each of them is affected by losses, which are expressed by individual efficiencies. For the comparison of different systems, the on-board conversion chain must be considered. For a complete picture one could extend this view to the complete energy chain e.g. from “well-to-wheel” in case of fossil fuel, but such an analysis is beyond the scope of this paper and very difficult. Figure 7 shows four typical conversion chains:

- a conventional turboprop
- a conventional turbofan,
- a battery powered system, and
- a fuel cell powered system.

Typical values for the efficiency of the individual components have been assumed¹. It is clearly visible that all electric systems exhibit high overall system efficiency. Especially battery systems which avoid the conversion of fuel to electricity in the airplane offer the highest efficiency of more than 70%, compared with the efficiency of the classical combustion systems reaching efficiencies of up to 40%. Of course these on-board chains do not contain the generation of electricity on the ground nor do they include the generation and distribution of fuel or electricity. For example the charging and discharging cycle of chemical batteries causes energy losses in the order of 15-20%. Nevertheless in terms of efficient usage of on-board energy the battery systems are very attractive. The important aspect of the system mass is examined in the next section.

¹ When looking up these efficiency values one has to be careful, as in some cases (e.g. fuel cells) efficiencies may include the exploitation of thermal power which may not be of interest. For a more refined picture one must consider how to exploit these additional capabilities e.g. for air conditioning purposes or thermal thrust generation. Also note that the latest turbofan engines claim to achieve thermal efficiencies of almost 60%, which would raise the chain efficiencies for turboprop and turbofan to 47% resp. 39%, bringing them closer to the value of the fuel cell chain.

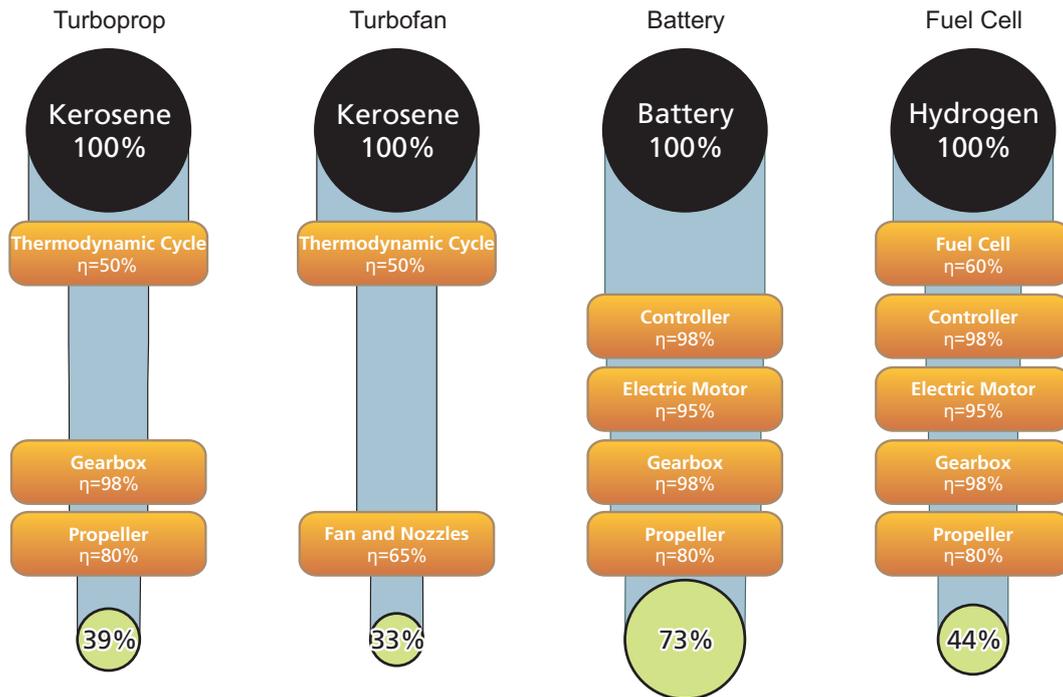


Figure 7 Typical on-board conversion chains with typical component efficiencies and total chain efficiency.

3.2.6 Equivalent Energy Density of Electric Power Sources

In order to compare different electric power sources with a battery system it is necessary to transform their parameters into an equivalent mass specific energy E^* of the complete system. First, we compare the systems as electric power sources, i.e. the chain up to the electric connector where the motor controller would be plugged in. Figure 8 relates the stored energy to the mass of the electric power source.

The example assumes identical flight performance of a general aviation airplane requiring a propulsive power of 50 kW for fast cruise. The required energy is determined from the respective efficiency chains so that 100 kWh are finally available for the mission. As the airplane has not been adapted to the mass of the propulsion system, this yields an optimistic view for the heavier systems.

The mass given below the fuel tanks are for the fuel and the additional mass of the tank with its fuel systems. It can be seen that a high pressure tank for hydrogen outweighs the fuel mass almost by a factor of 20.

In Figure 9 we can add the motor controller and the electric motor so that the chain extends up to the propeller shaft. This allows comparing the chain with an internal combustion engine.

The results show that the classical internal combustion engine offers the lowest mass and hence an effective specific energy of almost 1600 Wh/kg, which is about 14% of the specific energy content of the raw kerosene fuel. This propulsion system is followed by the hybrid electric system which adds a generator and an electric motor to the drive train.

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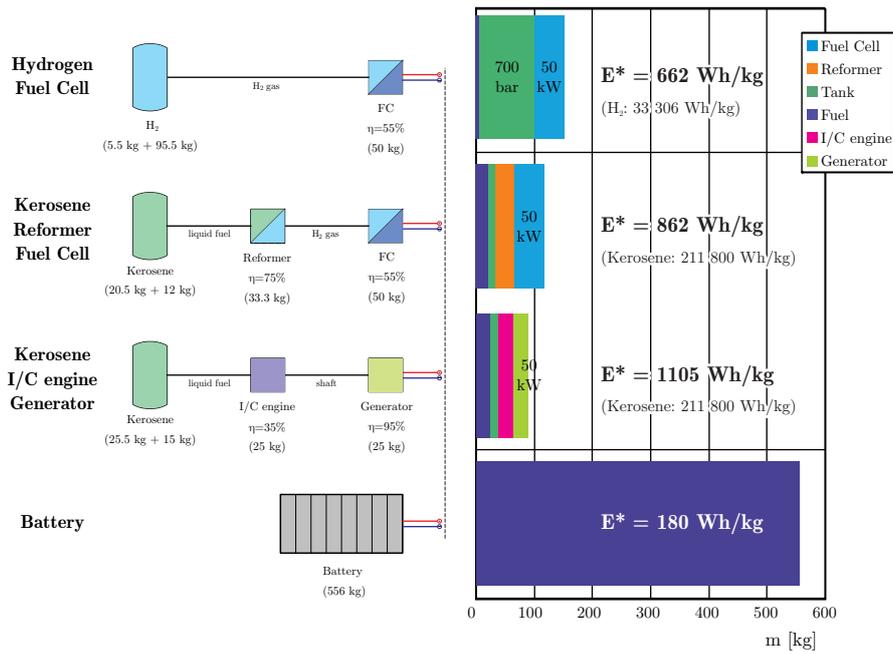


Figure 8 Mass and equivalent energy density of electric power generation systems delivering an electric power of 50 kW for 2 hours.

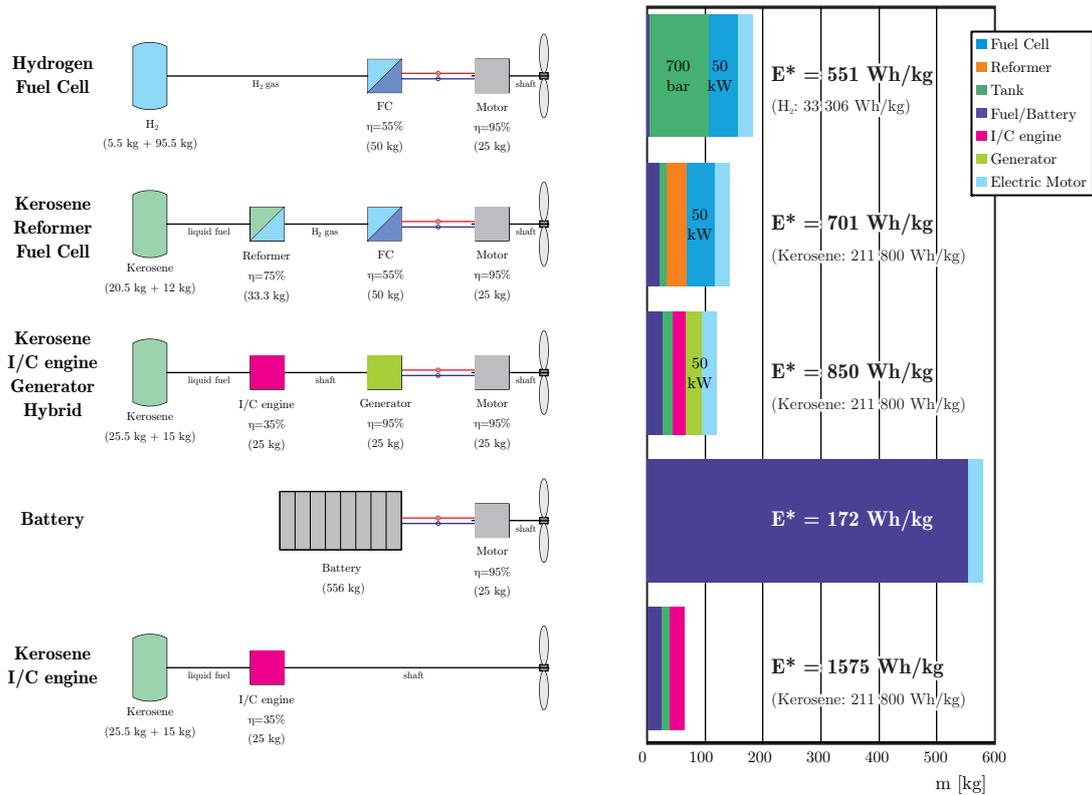


Figure 9 Mass and equivalent energy density of propulsion systems providing a shaft power of 50 kW for 2 hours.

3.2.7 Battery Technology

Today, most electric powered aircraft are using battery systems of the Lithium-Ion type. These are mass produced mainly as power sources for mobile devices and are readily available. Such batteries can be produced relatively cheap and can be scaled to build larger systems of several hundred kWh energy capacity. Currently available systems provide a mass specific energy content of up to 200 Wh/kg (see Figure 10). Further developments may lead to values in the order of 250 Wh/kg, which is insufficient for automotive and larger aerial applications. Today, the development of other Lithium based battery systems is pushed forward. Lithium-Sulfur and Lithium-Oxygen are two systems on which research and development are focused. While sulfur-based systems have reached practical prototype status (after more than 50 years of development), oxygen based systems are still in their infancy. All these systems offer energy densities which are larger by a factor of about 3 to 10. A very complete survey of Lithium based battery technology is given in [3]. Table 2 contains values taken from this publication and shows the expected development of these systems. Concerning the possible application of the air breathing, oxygen based battery systems in aviation many questions are still open and cannot be expected to be answered within the next 10 years. Therefore it seems to be more realistic to expect the application of sulfur based systems within the next 20 years in mobile applications as well as in unmanned and possibly manned air vehicles.

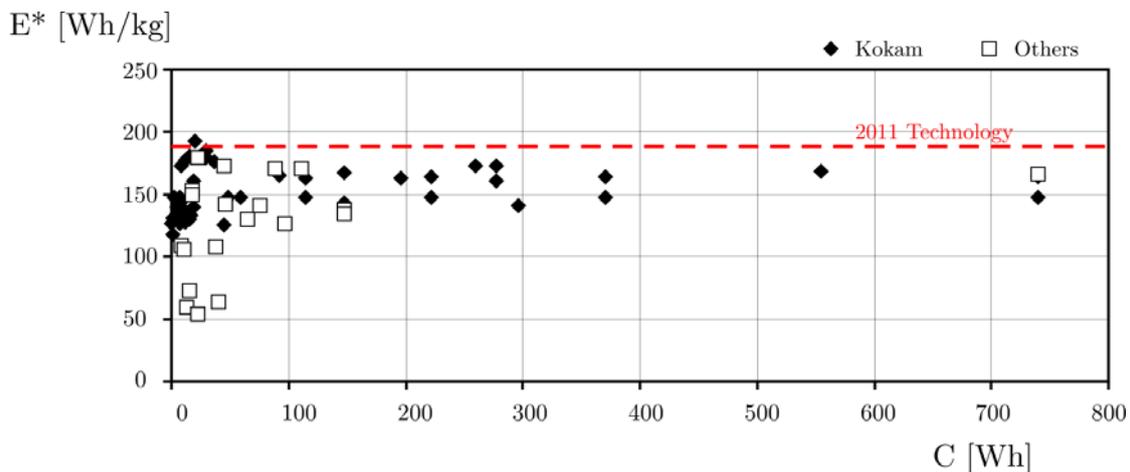


Figure 10 Specific energy density of current Lithium Ion battery systems.

System	theoretical specific energy	expected in 2025
Li-Ion (2012)	390 Wh/kg	250 Wh/kg
Zn-air	1090 Wh/kg	400-500 Wh/kg
Li-S	2570 Wh/kg	500-1250 Wh/kg
Li-O ₂	3500 Wh/kg	800-1750 Wh/kg

Table 2 Specific energy density of current and future chemical battery systems.

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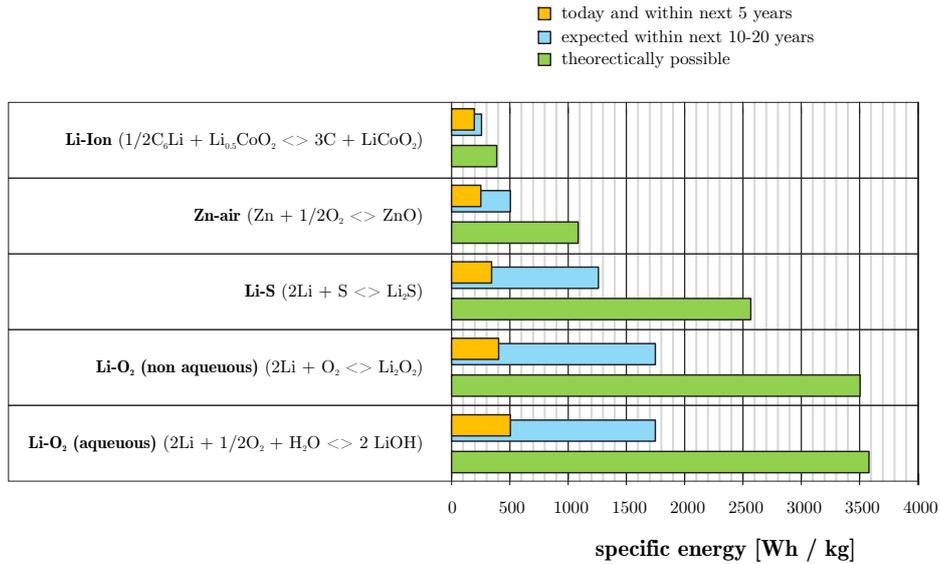


Figure 11 Current battery technology and expected development.

3.2.8 Electric Motor Technology

Electric motors for application in aviation must be designed for continuous operation at cruise power. Motors for hybrid automotive applications or for model aircraft are often designed for short power boost applications so that one must be careful when comparing data. Figure 12 plots the power specific mass of various piston engines and electric motors. The few electric motors available today for aircraft propulsion have a power output of less than 100 kW. Large electric motors are also used in trains, ships and submarines, but here the mass is less important. Today it seems to be possible to build electric motors having a specific mass of about 2 to 4 kW/kg. This compares favorably with the specific mass of larger turboshaft and turbofan engines at cruise power (see Figure 13 and Figure 14). Future developments may extend the range of electric motors to values of up to 8 kW/kg, but there is a strong need for the development of lightweight electric motors, specifically designed for application in aircraft. In the literature assumptions of up to 14 (e.g. in [5]) can be found, but assume superconducting motors with cryogenic cooling systems.

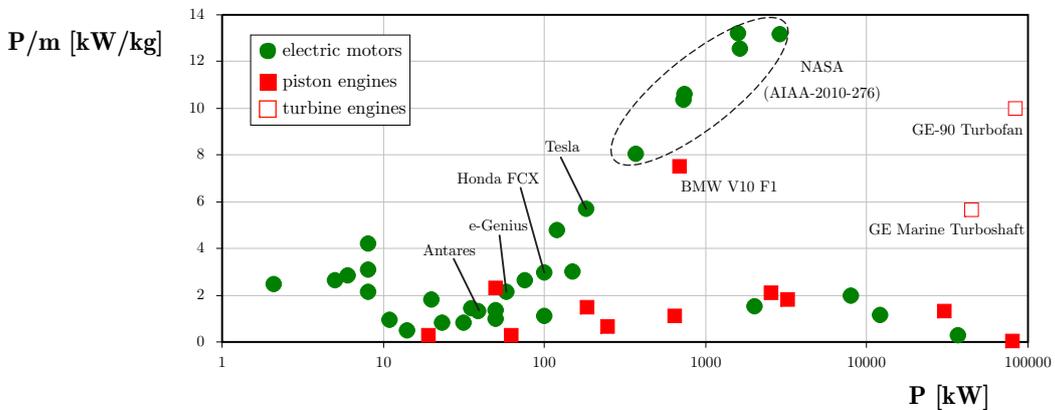


Figure 12 Specific power density of current piston engines (squares) and electric motors (circles).

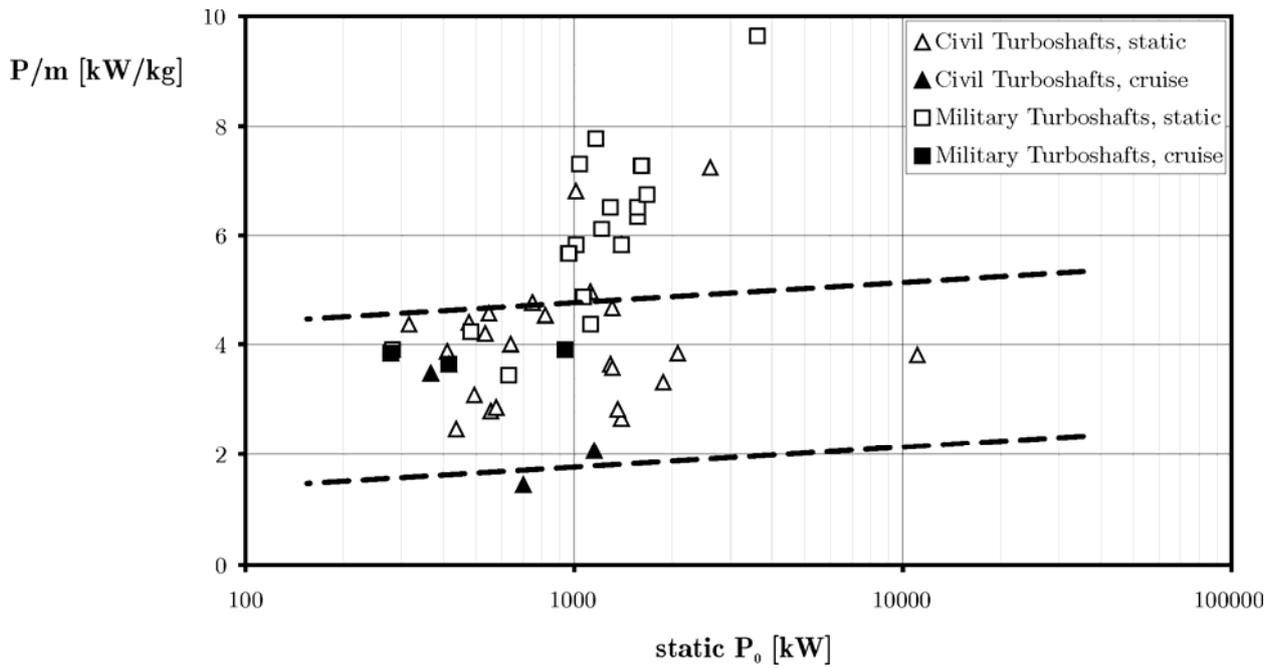


Figure 13 Specific power density of turboshaft engines.

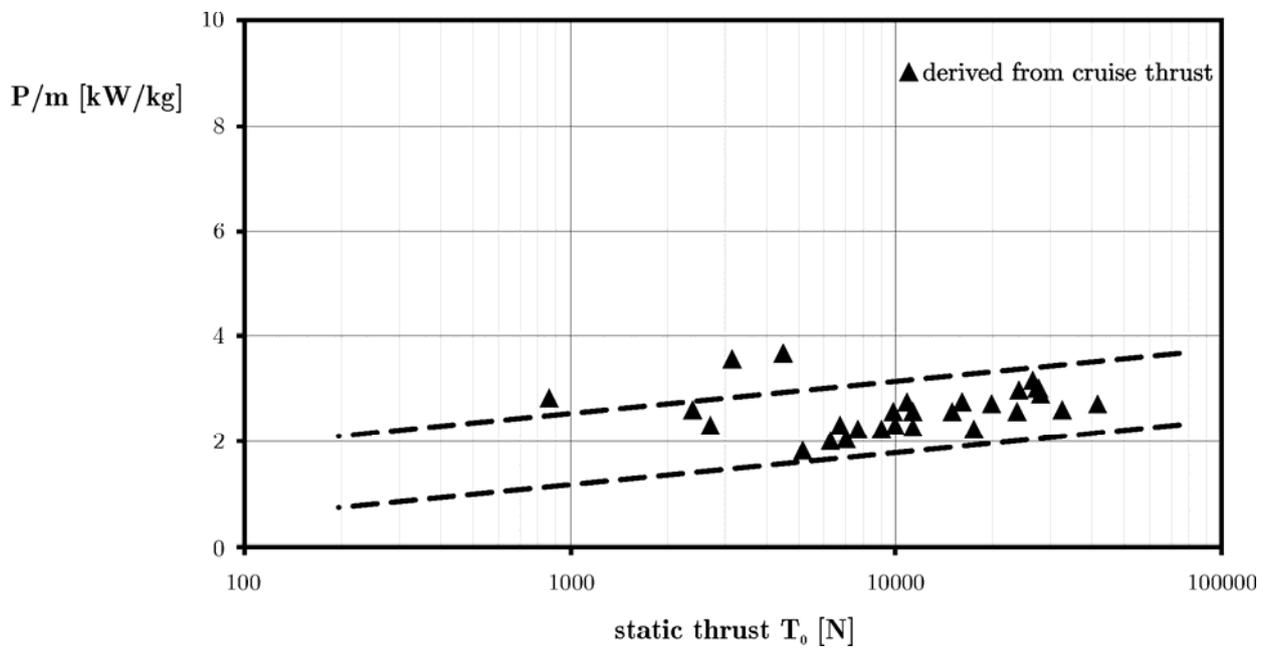


Figure 14 Specific power density of turbofan engines, derived from cruise thrust.

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4.0 RANGE EQUATIONS

The range of an aircraft depends on the available energy, the propulsion system, the mass of the aircraft and the aerodynamic properties of the aircraft. The range equations for classical, fuel burning aircraft with variable mass are well known and have been derived for various cruise flight conditions. In case of battery powered electric aircraft the mass of the aircraft stays constant and hence the range equation is simplified.

The range of an aircraft is defined by flight speed and flight time

$$R = v_{\infty} \cdot t. \quad (1)$$

In case of a battery powered aircraft, the flight time is equal to the time to drain the battery, which, under ideal conditions, is

$$t = \frac{m_{\text{battery}} \cdot E^*}{P_{\text{battery}}}. \quad (2)$$

Inserting this flight time into the definition of the range (1) yields

$$R = v_{\infty} \cdot \frac{m_{\text{battery}} \cdot E^*}{P_{\text{battery}}}. \quad (3)$$

The power drawn from the battery is related to the propulsive power required by the aircraft

$$P_{\text{battery}} = \frac{P_{\text{aircraft}}}{\eta_{\text{total}}} \quad (4)$$

where the power required by the aircraft is linked to its weight, lift over drag ratio and flight speed:

$$P_{\text{aircraft}} = D_{\text{aircraft}} \cdot v_{\infty} = \frac{m \cdot g}{L/D} \cdot v_{\infty}. \quad (5)$$

With (5), the battery power (4) finally becomes

$$P_{\text{battery}} = \frac{m \cdot g}{L/D \cdot \eta_{\text{total}}} \cdot v_{\infty} \quad (6)$$

and can be inserted into equation (3)

$$R = v_{\infty} \cdot \frac{m_{\text{battery}} \cdot E^*}{\frac{m \cdot g}{L/D \cdot \eta_{\text{total}}} \cdot v_{\infty}}. \quad (7)$$

This equation can be simplified to the range equation for battery electric flight

$$R = E^* \cdot \eta_{\text{total}} \cdot \frac{1}{g} \cdot \frac{L}{D} \cdot \frac{m_{\text{battery}}}{m}. \quad (8)$$

Interestingly the range is independent of the flight speed v_{∞} which is only indirectly affecting the result via lift to drag ratio and the total efficiency η_{total} .

This equation clearly shows that for maximum range the following parameters should be maximized:

- the ratio of battery mass to total mass m_{battery} / m ,
- the specific energy capacity E^* of the battery,
- the lift over drag ratio L / D , and
- the total system efficiency η_{total} (from battery to propulsive power).

On the other hand

- the aircraft mass m

should be minimized.

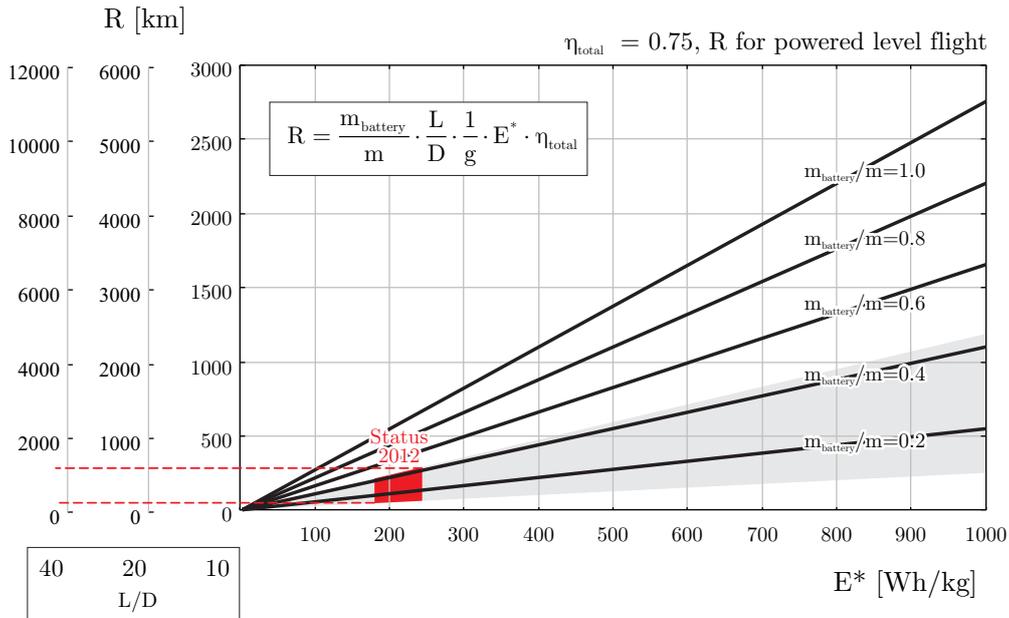


Figure 15 Range versus specific energy for different battery mass fractions. The vertical axes are for three different ratios of lift over drag, ranging from small General Aviation aircraft to sailplanes.

The range obtained from this equation is plotted in Figure 15 versus the specific energy of the battery system E^* . The shaded triangle indicates the range of technically feasible battery mass fractions. It can be seen that for larger ranges the battery technology must be improved to at least five-fold values compared to the state in 2012.

Comparing the range equation for battery powered aircraft with one form of the classical range equation for kerosene burning jet engines

$$R = E^* \cdot \eta_{\text{total}} \cdot \frac{1}{g} \cdot \frac{L}{D} \cdot \ln \left(\frac{1}{1 - m_{\text{fuel}} / m} \right) \tag{9}$$

shows that the two equations only differ in the last term. Figure 16 compares this term for aircraft having different fuel mass fractions respectively ranges. It can be seen that the drawback of maintaining constant mass increases with range and may grow to more than 20% for long range flights. This drawback is inherent in battery powered aircraft and must be compensated by additional energy respectively efficiency.

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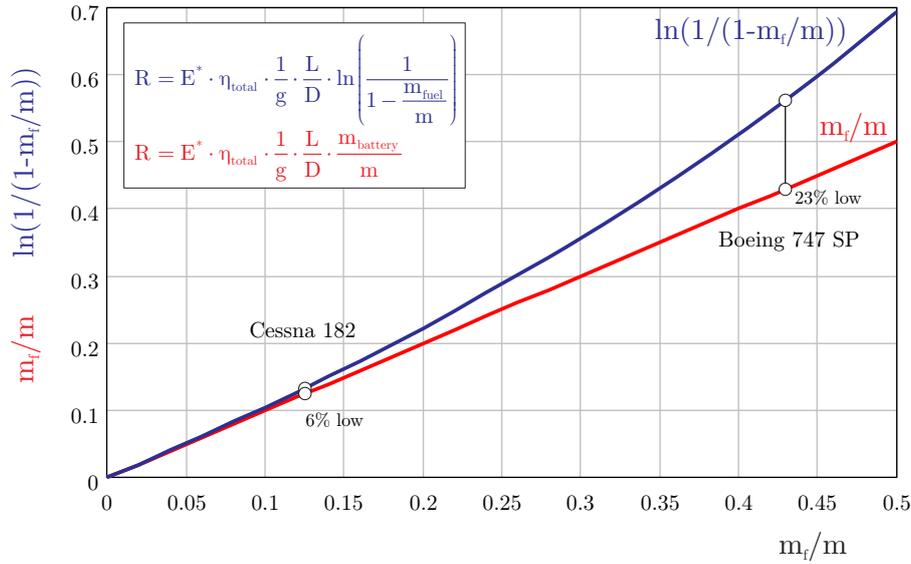


Figure 16 Range terms versus fuel respectively battery mass fraction.

4.1. MAXIMUM RANGE

To gain more insight into typical aircraft design parameters we replace the battery mass

$$m_{\text{battery}} = m - m_{\text{empty}} - m_{\text{payload}} \tag{10}$$

to obtain

$$R = E^* \cdot \eta_{\text{total}} \cdot \frac{1}{g} \cdot \frac{L}{D} \cdot \left(1 - \frac{m_{\text{empty}}}{m} - \frac{m_{\text{payload}}}{m} \right) \tag{11}$$

Reducing the payload mass to zero allows finding the ultimate range

$$R_{\text{ultimate}} = \frac{1}{g} \cdot E^* \cdot \eta_{\text{total}} \cdot \frac{L}{D} \cdot \left(1 - \frac{m_{\text{empty}}}{m} \right) \tag{12}$$

For the given technology parameters, this range can never be exceeded. Also note that for this hypothetical range the aircraft mass would grow to infinity. Therefore the practical range must be lower than this hard limit.

Equation (11) can be solved for the airplane mass which is required for a desired range and payload using a given technology level in structures, aerodynamics and propulsion system:

$$m = \frac{\text{PAX} \cdot m_{\text{payload}}}{1 - f_c - \frac{g}{E^* \cdot \eta_{\text{total}} \cdot L/D} \cdot R} \tag{13}$$

Equation (13) can be used for a very quick first order assessment of the required size of an airplane, using only a few basic technology parameters. The structural design of the aircraft is represented by the empty mass fraction $f_e = m_{\text{empty}} / m$; for a given structural technology and typical design parameters, like aspect ratio and wing thickness, one can assume a certain value for f_e (which is typically between 0.4 and 0.7, see Figure 17). Of course, this ratio should be as low as possible to obtain the lowest aircraft mass. The aerodynamics is represented by the lift over drag ratio, which should be as large as possible to minimize the mass.

In order to obtain a valid solution of (13) with a finite and positive aircraft mass the following conditions must be fulfilled:

The lift over drag ratio must be at least

$$\frac{L}{D} > \frac{R \cdot g}{(1 - f_e) \cdot E^* \cdot \eta_{\text{total}}}, \tag{14}$$

the specific energy of the battery must be larger than

$$E^* > \frac{R \cdot g}{(1 - f_e) \cdot \eta_{\text{total}} \cdot L / D}, \tag{15}$$

and the empty mass fraction of the aircraft must be lower than

$$f_e < 1 - \frac{R \cdot g}{E^* \cdot \eta_{\text{total}} \cdot L / D}. \tag{16}$$

These limits are independent of the payload and can be used to define the required technology level in each discipline or to judge the feasibility of a proposed concept. If a design does not fulfill these constraints, it is unfeasible with the given the technology.

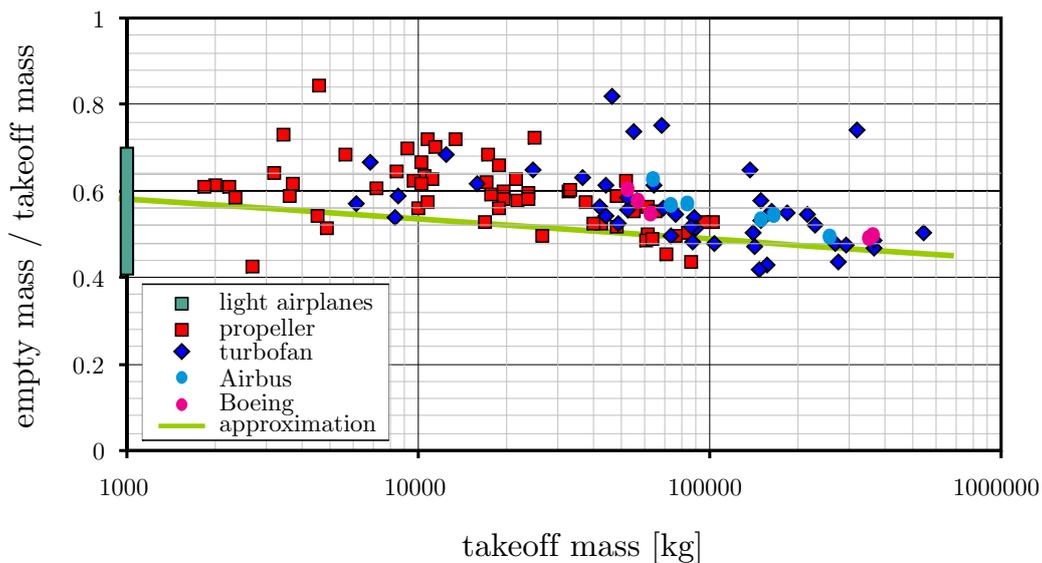


Figure 17 Empty mass fraction of aircraft versus take-off mass.

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A more realistic range limit can be found by monitoring the mass growth per range increase. Differentiating (13) with respect to the range yields the mass growth per distance for a design for a given range

$$\frac{\partial m}{\partial R} = \frac{PAX \cdot m_{pax} \cdot g}{\left(1 - f_e - \frac{g \cdot R}{E^* \cdot \eta_{total} \cdot L / D}\right)^2 \cdot E^* \cdot \eta_{total} \cdot L / D}, \quad (17)$$

where the payload mass has been replaced by $m_{payload} = PAX \cdot m_{pax}$.

If we limit the mass growth per distance to a certain limit value $\left(\frac{\partial m}{\partial R}\right)^*$, we can solve (17) for the limit range R_{max}

$$R_{max} = \underbrace{E^* \cdot \eta_{total} \cdot \frac{L}{D} \cdot \frac{1}{g} \cdot (1 - f_e)}_{R_{ultimate}} - \sqrt{\frac{PAX \cdot m_{pax}}{\left(\frac{\partial m}{\partial R}\right)^*} \cdot \frac{1}{g} \cdot E^* \cdot \eta_{total} \cdot \frac{L}{D}}. \quad (18)$$

The first term in this equation equals equation (12), the second term corresponds to the range reduction imposed by the mass growth limit.

A more complete analysis of existing and prospective electric aircraft showed that the acceptable mass growth depends on aircraft mass: for heavier aircraft a higher limit can be accepted. Based on the numerical results, the following empirical relation has been developed:

$$\left(\frac{\partial m}{\partial R}\right)^* = \frac{1}{4200} \cdot m^{1.27} \left[\frac{\text{kg}}{\text{km}}\right]. \quad (19)$$

For lightweight single seat aircraft ($m = 1000$ kg) this equation yields limit values in the order of 1 kg/km, while the limit for heavy regional aircraft ($m = 15$ tons) reaches values of 50 kg/km.

Numerical values for a hypothetical 30 seat regional aircraft are shown in Figure 18. The realistic range limit is indicated by symbols. Two ways of exploiting an increase of the specific energy E^* from 200 to 400 Wh/kg are shown. First one can maintain the aircraft mass and move horizontally to double the range from 325 to 650 km. At this point (hollow symbol) the mass growth limit is not yet reached so that the practical range can be extended further to about 820 km. Alternatively one could move vertically down, maintaining the range but reducing the aircraft mass from about 14 tons to 8.5 tons.

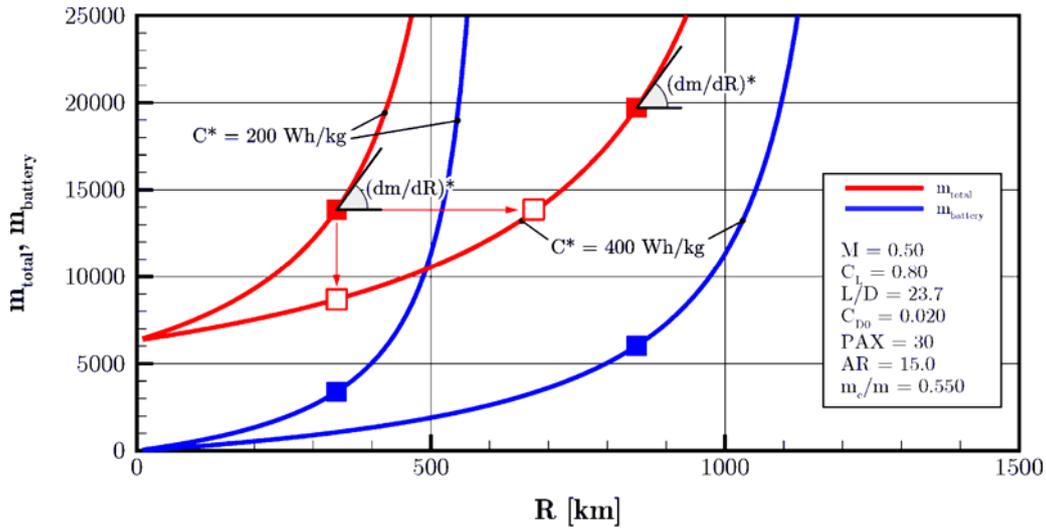


Figure 18 Total mass and battery mass for two values of the specific energy E^* versus range. The solid symbols indicate the range limit imposed by a prescribed mass growth gradient dm/dR .

4.2. RANGE SENSITIVITIES

Equation 11 shows that in order to achieve maximum range for a given aircraft mass one should of course minimize the empty mass fraction $f_e = m_{empty} / m$ as well as the payload mass fraction $f_p = m_{payload} / m$ so that the battery mass is maximized. At the same time one should also maximize the lift over drag ratio L / D . The sensitivities of equation 11 with respect to the mass parameters f_e and f_p

$$\frac{\partial R}{\partial f_e} = \frac{\partial R}{\partial f_p} = -\frac{1}{g} \cdot E^* \cdot \eta_{total} \cdot \frac{L}{D} \quad (20)$$

show that a change in empty weight or payload weight fraction is more critical when the ratio L / D and the system efficiency η_{total} are high. The sensitivity with respect to the total mass

$$\frac{\partial R}{\partial m} = -\frac{1}{g} \cdot E^* \cdot \eta_{total} \cdot \frac{L}{D} \cdot m_{battery} \cdot \frac{1}{m^2} \quad (21)$$

shows a strong dependency on the inverse of the mass. A lightweight aircraft is more sensible to a change in mass than a heavy aircraft.

The impact of the lift to drag ratio can be assessed from

$$\frac{\partial R}{\partial L / D} = (1 - f_e - f_p) \cdot \frac{1}{g} \cdot E^* \cdot \eta_{total} \quad (22)$$

which demonstrates that a lightweight aircraft is more sensitive to changes in L / D . Concerning the effect of the battery technology, the derivative is similar:

$$\frac{\partial R}{\partial E^*} = (1 - f_e - f_p) \cdot \frac{1}{g} \cdot \eta_{total} \cdot L / D \quad (23)$$

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These sensitivities can be used to assess the robustness of a design with respect to changes in operating parameters. A numerical example is shown in Table 3 to demonstrate these sensitivities. The right hand side of the table shows the effect on range of increasing a parameter by 10% respectively of increasing the number of passengers by one. Additionally, derived sensitivities can be used to “trade” one discipline for another discipline; for example the derivative $\partial E^* / \partial f_e$ shows that an increase of 10% of the empty mass fraction ($\Delta f_e = 0.053$) has the same impact on range as a decrease of the specific energy of the battery system by $\Delta E^* = -31.9$ Wh/kg.

assumptions			results		
E^*	200	Wh/kg	R_{\max}	173.1	km
L / D	16.35	-	$\partial R / \partial E^* \cdot 10\% \cdot E^*$	28.2	km
f_e	0.53	-	$\partial R / \partial f_e \cdot 10\% \cdot f_e$	-44.9	km
η_{total}	0.7	-	$\partial R / \partial L / D \cdot 10\% \cdot L / D$	28.2	km
PAX	32	-	$\partial R / \partial \text{PAX}$	-3.4	km
m_{pax}	90	kg	derived sensitivity		
$(\partial m / \partial R)^*$	52	kg/km	$\partial E^* / \partial f_e \cdot 10\% \cdot f_e$	-31.9	Wh/kg

Table 3 Set of parameters (left) for an electric powered aircraft and its range sensitivities (right). Note that the assumed L/D ratio is lower than in Figure 18, therefore the range is lower.

5.0 ELECTRIC PROPULSION FOR REGIONAL AIRCRAFT

In order to assess the potential of an electric propulsion system in larger aircraft a more complete mission simulation tool was developed. Here the mission consists of a climb, a cruise and a descent segment. For each segment the aircraft is flown at minimum energy consumption. The simulation performs the time integration over the three segments for a given aircraft and propulsion system model.

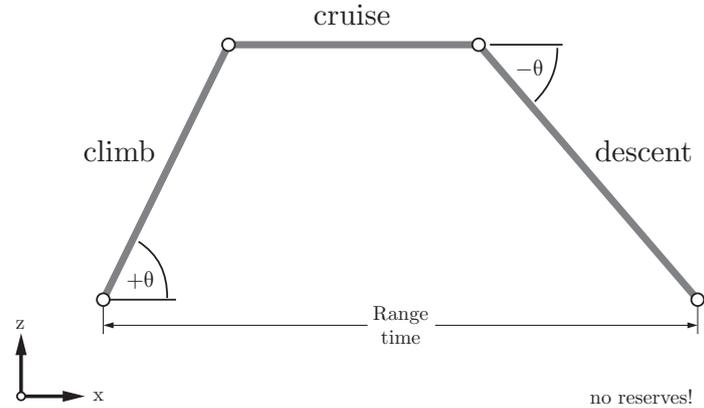


Figure 19 Simplified mission profile.

5.1. CLIMB SEGMENT

This segment is defined by either climb angle or climb rate. The simulation then determines the optimum flight speed for each altitude so that the energy drawn from the battery per altitude dE_{battery} / dz is minimized. It is important to include the efficiency of the propulsion system to avoid the trivial solution of a vertical climb which provides minimum energy consumption of the airframe alone (Figure 20).

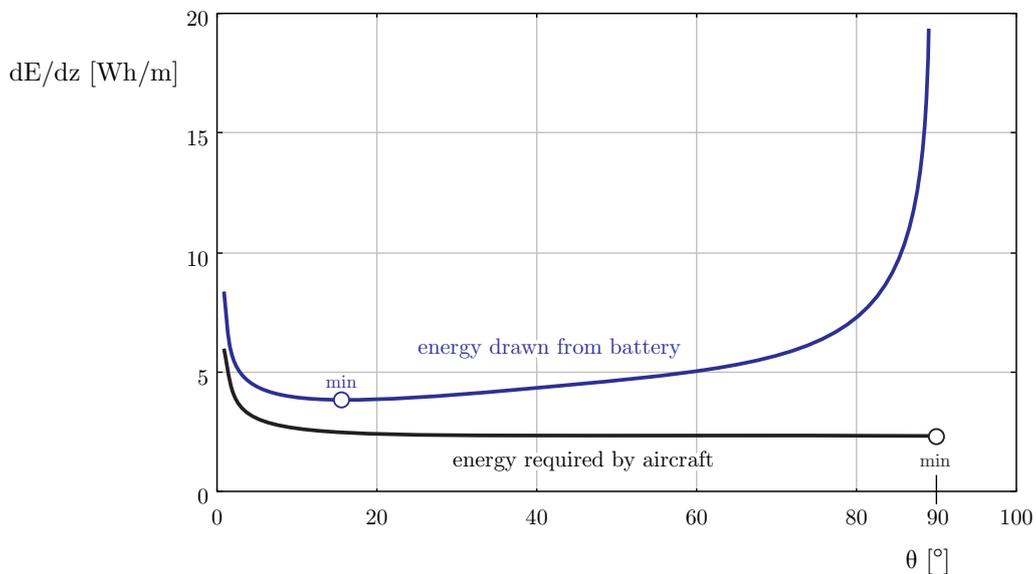


Figure 20 Energy consumed per altitude for the aircraft and for the aircraft with propulsion system chain. The trivial optimum for the airframe would be to climb vertically and as slow as possible to minimize total drag. Including the efficiency of the propulsion system yields a realistic minimum.

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5.2. CRUISE SEGMENT

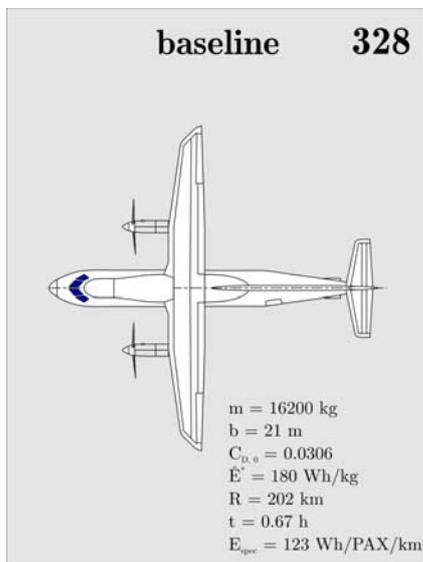
During cruise, the optimum flight speed is determined so that dE/dx is minimized. Again, the efficiency of the propulsion system chain including the propeller is taken into account, so that the optimum speed is not identical to the optimum of the aircraft alone.

5.3. DESCENT SEGMENT

The descent segment is flown with a given descent angle, which is steeper than the maximum of lift over drag ratio of the airframe to minimize the energy consumption. During this flight phase only auxiliary power for the aircraft systems has to be provided by the battery. No recuperation is considered here due to the high drag and low efficiency of the windmilling propellers, which would allow to recover about 5% of the total energy (in case of a 500 km flight of a regional aircraft at $H = 3000$ m the potential energy $m \cdot g \cdot H$ is about 20% of the total energy consumed during the flight).

5.4. APPLICATION

Applying this simulation model to a specific aircraft allows determining the possible performance and the technology improvements required to achieve the desired results. As a test case a regional aircraft similar to the Dornier Do 328 turboprop aircraft was modeled. Step by step several modifications were performed to optimize the aircraft for the electric propulsion system.

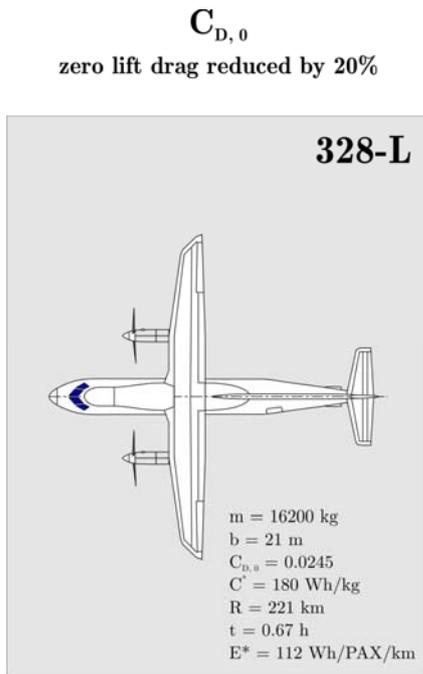


The range of the original aircraft with 32 passengers is about 1200 km. With a reduced payload of 28 passengers the maximum range is approximately 2200 km, while the lift-to-drag ratio is around 16.

The first modification consists of simply replacing the conventional fuel system and the turboprop engines with a battery electric system having the same mass. Using current (2012) technology this aircraft would reach a range of 202 km. The flight time would be about 40 minutes. Cruise speed would be about 300 km/h.

If an additional reserve of 30 minutes for holding at the destination airport would have to be considered, the practical range would drop to 50 km.

Figure 21 Modified baseline aircraft with drop-in electric propulsion system.



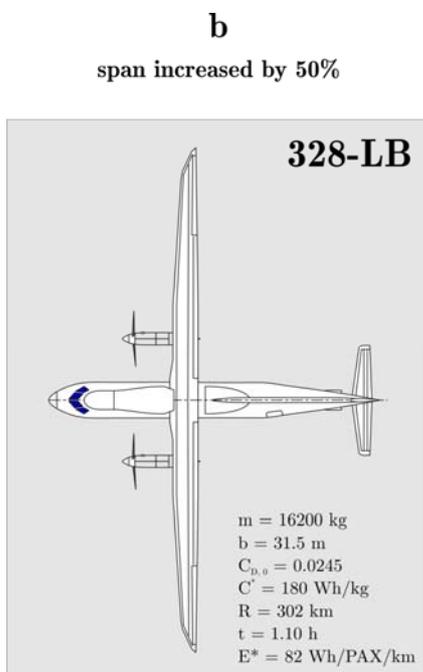
The next step tries to improve the aerodynamic performance by reducing the zero lift drag coefficient by 20%.

This could be achieved by optimizing the installation of the propulsion system (smaller nacelles or fully integrated motors, less cooling air flow) as well as many other detail improvements.

Part of these improvements could be achieved by introducing laminar flow on the wing and on the tailplanes. This is feasible as the wing is unswept and may be produced in composite material with the required smoothness.

Due to the lower drag, the optimum cruise speed would increase to 330 km/h. However, as the zero lift drag is only about half the total drag, this modification would increase the range by only 10% to 221 km.

Figure 22 Modified aircraft with reduced zero lift drag.



Next the induced drag is reduced through an increase in wing span of 50% without increasing the wing mass and the wing area. The new wing span is now 31.5 m compared to the initial 21.0 m.

This change would require substantial changes to the structure, possibly the introduction of a strut braced wing concept (while maintaining zero lift drag) and an optimized composite structure.

This modification would increase the range by about 40% to 302 km.

The higher aspect ratio would move the energy minimum to a lower cruise speed of 275 km/h.

Figure 23 Modified aircraft with increased wing span.

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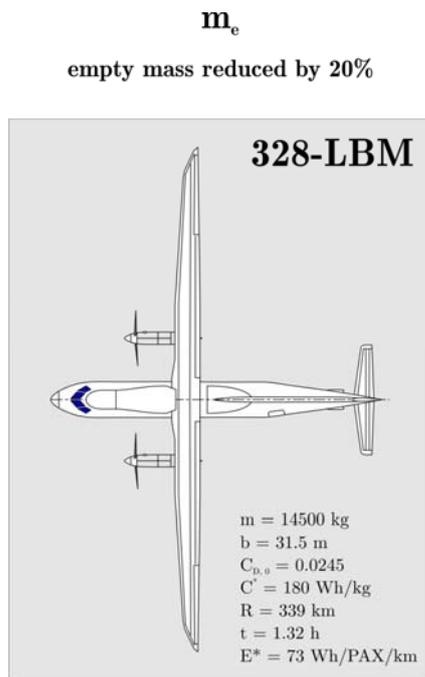


Figure 24 Modified aircraft with reduced mass.

The next modification would reduce the empty mass of the aircraft by 20%. This would require introducing extreme lightweight design features. As many components are unlikely to be reduced in mass the main structural components would have to be reduced by probably 30%.

This modification would increase the range by another 10% to 339 km. Due to the lower mass, the optimum flight speed would be about 255 km/h.

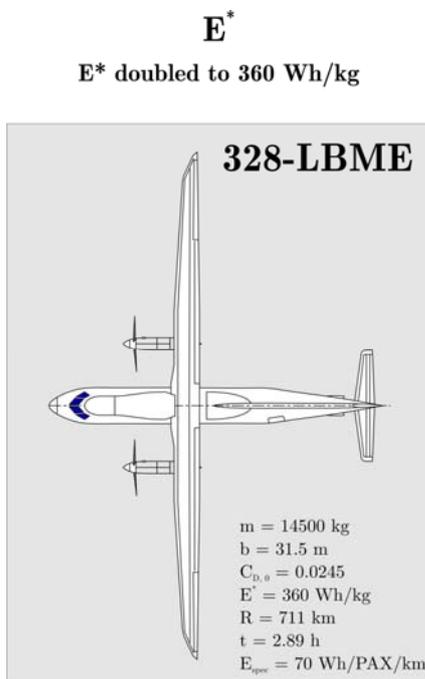
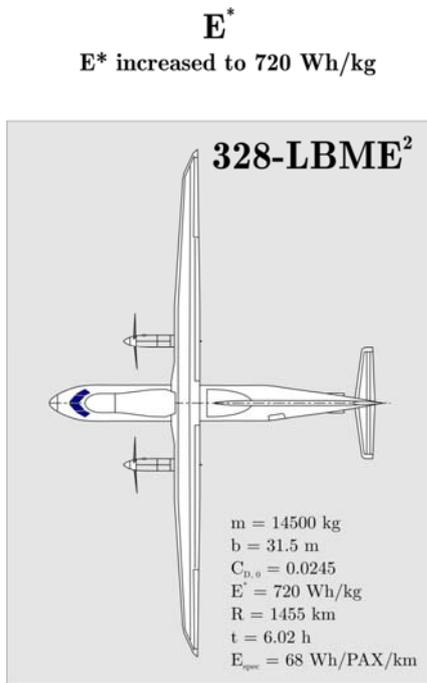


Figure 25 Modified aircraft with improved battery technology.

This step improves the battery technology by doubling the mass specific energy E^* . Such an improvement is quite well possible with future development of Li-S battery systems within the next 15 years.

This modification would double the range to 711 km so that it at least comes into the order of the kerosene based aircraft.

Nevertheless, there is still a factor of 3 in range missing. In order to achieve the range of the original aircraft, the battery technology would have to be improved by this factor, i.e. a factor of 6 compared to today's (2012) technology.



The last modification doubles the mass specific energy density of the battery system again. Such an improvement may be possible with future development of Li-S battery systems in the next 30 years.

This modification would double the range again to 1455 km so that it is almost identical to the kerosene based aircraft. Such an aircraft could be operated profitable if the technology would be available at reasonable costs. Furthermore new infrastructure to replace and recharge the batteries on each airport would be required to make such an aircraft feasible.

However one must consider that this improvement also includes many very optimistic assumptions and improvements in structures and aerodynamics. Just using the improved batteries in the electrified baseline aircraft 328 E would lead to a range of 800 km at the same specific energy consumption of 124 Wh/PAX/km.

Figure 26 Modified aircraft with further improved battery technology.

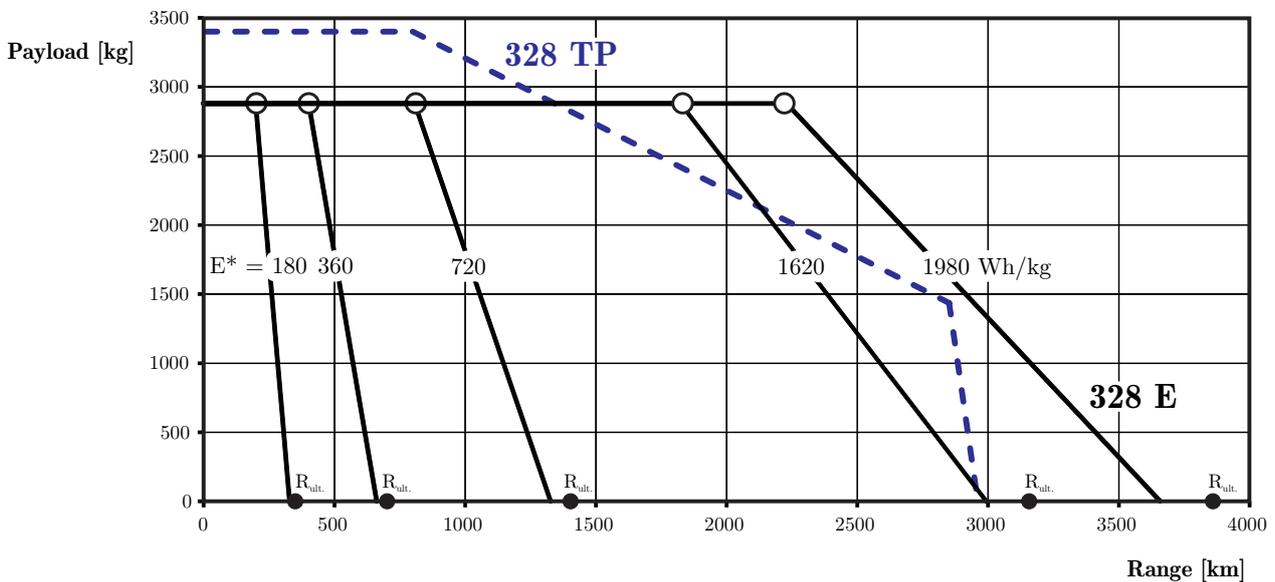


Figure 27 Payload-range chart of the baseline and the electrified aircraft.

Comparing the payload-range characteristics of the baseline turboprop aircraft and the battery powered electric aircraft shows that trading payload for fuel respectively battery has a very beneficial effect in case of kerosene because of its high specific energy. The gain in case of the electric aircraft is smaller, even if the design range and payload are similar. As can be seen, the gradient depends on the specific energy

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content E^* of the battery system, which is always lower than that of kerosene. This means that the range flexibility of the battery powered aircraft is more limited than that of the kerosene powered turboprop variant. Similar payload range capability is obtained when the specific energy of the battery system exceeds about 1500 Wh/kg.

If, on the other hand, the final extremely modified aircraft 328-LBME² would be equipped with a current turbo-prop engine, its fuel consumption would be as low as 1.5 liters per passenger per 100 km, which is about half that of the baseline aircraft.

6.0 CONCLUSIONS

Practical battery electric powered airplanes are today limited to small vehicles up to 2 passengers and by rather short ranges and endurance. Neglecting costs, the current technology is suitable for small Ultra-Light aircraft, but not for commercial aviation. In order to power larger aircraft a dramatic improvement in battery technology would be required. Comparing with today's technology with specific energy values of 150 to 200 Wh/kg, the mass specific energy density would have to be increased at least by a factor of 5 to become useful. More realistic this factor would have to be in the order of 10 to attract commercial interest for larger (regional) aircraft. In this context we must note that all numerical studies presented in this paper did not consider reserves as required for commercial aircraft. These would add 30 to 45 minutes of flight time for holding at the destination airport and deviations.

Currently there is a lot of development in battery technology, mostly driven by mobile devices and by automotive applications. Development history hints that the required improvements may take about 20-40 years until they may become available.

Besides these basic considerations, there are still many questions which have to be and which are currently addressed:

- how does the total energy balance and the environmental footprint including manufacturing look like?
- are there enough raw materials for battery production (automotive plus aviation) available?
- how to achieve the required level of safety and how to adapt certification rules?
- which infrastructure and investment is required at the airport for battery replacement and recharging?
- how can the low noise features of electric propulsion systems be exploited better?
- how does the battery technology compare to future fuel cell technology developments?
- how do fuel cell powered aircraft compare to aircraft burning synthetic fuels or hydrogen?

Returning to the historic flight of the MB-E1 we can apply the same numeric analysis to study the option of repeating the experiment with the technology readily available in 2012. We can replace the Ni-Cd batteries ($E^* = 20$ Wh/kg) with Li-OH cells ($E^* = 180$ Wh/kg) of the same mass and the brushed electric forklift motor ($\eta = 66\%$) with a specifically designed brushless motor ($\eta = 90\%$). This simple modification would extend the powered flight time of the MB-E1 from 7 minutes to 2 hours and 33 minutes. The range of the aircraft would grow from 17 km to 261 km. Mainly due to the improved efficiency of the electric motor the energy consumption of the single passenger (the pilot) per distance would be almost halved from $E^* = 135$ Wh/PAX/km to $E^* = 80.8$ Wh/PAX/km.

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8.0 APPENDIX – RESULTS FOR BASELINE AIRCRAFT “328 E”

Aircraft:

Name	Do 328 E
Wing span	b = 20.98 m
Wing area	S = 40.00 m ²
Aspect ratio	AR = 11.00
k-Factor	k = 1.060
L/D max	L/D = 16.16
v_EAS(L/D max)	v = 79.8 m/s = 287.1 km/h
v_stall	v stall = 53.7 m/s = 193.5 km/h
climb	CD0 = 0.0321
cruise	CD0 = 0.0312
descent	CD0 = 0.0306
passenger+crew load	PAX = 32 @ 90.0 kg
Mass total	m = 15880.0 kg
Mass empty	me = 8500.0 kg
Mass payload	mp = 2880.0 kg (32 PAX)
wing box volume	Vw = 3260.1 liters
empty weight fraction	me/m = 0.535
payload weight fraction	mp/m = 0.181
battery weight fraction	mb/m = 0.283
wing loading	m/S = 397.00 kg/m ²
span loading	m/b ² = 36.08 kg/m ²
range factor	(m/b ²)/(m/S) = 0.091
ultimate range	R ult = 347.4 km

Range sensitivities for 51.5 kg/km, eta_total = 0.700:

max. powered range	R max = 143.0 km
specific energy	dR/dE* = 24.5 km/10%
empty mass fraction	dR/dfe = -40.0 km/10%
lift over drag	dR/d(L/D) = 24.5 km/10%
passenger count	dR/dPAX = -3.2 km/PAX
payload distance	E/(R*mp) = 119.5 - 138.0 Wh/km/PAX (range - endurance)
specific energy	dE*/dfe = -29.4 Wh/kg/10%

Battery:

Energy	E = 793.80 kWh
Mass	mb = 4500.0 kg
Volume	V = 793.8 liters
Voltage	U = 1800.0 V
Capacity	C = 441.0 Ah
C spec.	E* = 180.0 Wh/kg
Limit I	I max = 2205.0 A (5°C/h)
Limit P	P max = 3969.0 kW (5°C/h)

Climb Segment:

prescribed climb angle theta	= 7.5 degrees.
Altitude	H = 0.0 ... 3000.0 m
Time	t = 0.1 h
Distance	R = 22.8 km
Speed avg	v = 333.7 km/h (mean)
Climb rate	vz = 12.1 m/s (mean)
Energy	E = 296.26 kWh (from battery)
Power, a/c	P = 2798.55 kW (mean, for aircraft)
Thrust, a/c	T = 30189.1 N (mean, for aircraft)
CL avg	CL = 0.8 (mean)
CD avg	CD = 0.054 (mean)
L/D avg	L/D = 15.655 (mean)
eta, prop avg	eta = 74.88 % (mean)
eta total avg	eta t = 65.44 % (mean)
Power, batt	P = 4296.77 kW (mean, from battery)
Voltage, batt	U = 1812.3 V (mean)
Current, batt	I = 2389.9 A (mean)

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Cruise Segment:

Altitude	H = 3000.0 ... 3000.0 m
Time	t = 0.4 h
Distance	R = 125.0 km
Speed avg	v = 290.3 km/h (mean)
Speed avg	M = 0.2 (mean)
Energy	E = 493.87 kWh (from battery)
Power, a/c	P = 775.53 kW (mean, for aircraft)
Thrust, a/c	T = 9638.7 N (mean, for aircraft)
CL avg	CL = 1.0 (mean)
CD avg	CD = 0.061 (mean)
L/D avg	L/D = 16.2 (mean)
eta prop avg	eta p = 79.26 % (mean)
eta total avg	eta t = 69.26 % (mean)
Power, batt	P = 1143.94 kW (mean, from battery)
Voltage, batt	U = 1796.7 V (mean)
Current, batt	I = 654.9 A (mean)

Descent Segment:

Altitude	H = 3000.0 ... 0.0 m
Time	t = 0.1 h
Speed avg	v = 332.0 km/h (mean)
Descent rate	vz = -5.7 m/s (mean)
Distance	R = 48.5 km
Energy	E = 3.66 kWh
CL avg	CL = 0.9 (mean)
CD avg	CD = 0.054 (mean)
L/D avg	L/D = 16.2 (mean)
Power, batt	P = 25.00 kW (mean, from battery)
Voltage, batt	U = 1800.1 V (mean)
Current, batt	I = 13.9 A (mean)

Complete Mission:

Time	t = 0.65 h
Distance	R = 196.3 km
Energy a/c	Eac = 532.78 kWh
Energy battery	Eb = 793.80 kWh
spec. Energy	Eb* = 126.4 Wh/PAX/km = 12.64 kWh/PAX/100 km
- Equivalent turbofan propulsion:	
Fuel mass	mf = 173.81 kg Kerosene
Fuel volume	Vf = 231.75 liters Kerosene
Fuel volume	V* = 3.69 l/PAX/100 km
efficiency	eta total = 0.269
- Equivalent turboprop propulsion:	
Fuel mass	mf = 129.95 kg Kerosene
Fuel volume	Vf = 173.26 liters Kerosene
Fuel volume	V* = 2.76 l/PAX/100 km
efficiency	eta total = 0.360
- Equivalent piston-prop propulsion (BMW radial engines):	
Fuel mass	mf = 133.19 kg Kerosene
Fuel volume	Vf = 177.59 liters Kerosene
Fuel volume	V* = 2.83 l/PAX/100 km
efficiency	eta total = 0.351

Powered segments (climb and cruise):

Time	t = 0.50 h
Distance	R = 147.8 km
Energy battery	E = 790.14 kWh
Energy a/c	E = 532.78 kWh
spec. Energy	E* = 125.8 Wh/PAX/km

