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Research paper

Effect of energy balance evaluation on conceptual design of electric and hybrid aircraft

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Abstract: Nowadays, all the stakeholders, policy makers, regulators, aircraft designers, producers, operators, etc.) are intensively working on development of the aircraft with full electric and hybrid propulsion systems. However, the technical, technological constrains (like limit on accumulator energy density) require introducing a new approach to conceptual design of such aircraft. The new methods is based on energy and mass balance evaluation. This paper analyses the identified constrains; integrates the energy and mass balance equations into the preliminary definition and calculations of the aircraft performance. By this way, the technological constrains might be transferred into the limitation on the aircraft energy and mass breakdown, that initiates a new approach to aircraft conceptual design uses the knowledge based multidisciplinary optimization. The paper describes the developed methodology for conceptual design of aircraft. It show results of implementing this new development philosophy to conceptual design of a four-seat small electric/hybrid aircraft and a special hybrid cargo UAV. The discussion of the results including got by using the emerging and enabling new technologies and new methods and solutions (including for example distributed propulsion system, unconventional forms, morphing, biomimics, etc.), demonstrates the possible implementation of the new development philosophy, new approach to aircraft conceptual design.

Keywords: full electric aircraft, hybrid aircraft, energy and mass balance equations, conceptual design, constraints

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1. Introduction

Nowadays the aeronautics faces a non-classical market pull – technology push innovation [1]. The market pull appears as society needs in greener, more effective, on demand transport that is supported by visions and actions of the policy makers, regulators (see for instant the [2 – 5]). The same documents try to define the possible directions and areas of the technology developments, technology push. The system is non-classical, because the aeronautics including air transport is in transition from the second "S" curve of its development to the third one. This new "S" curve might be defined by several emerging technologies and radically new solutions [3 – 5] (like morphing, active flow control, biomimicry, etc.). The basic visions published for millennium or could not predict the introducing the electric aircraft concept (as Vision 2020 [2]) or they put only the electric propulsion, advanced fuel cells, high efficiency electric motors on the list of revolutionary vehicles – technologies as NASA BluerPrint [5]). Later, the IATA Technology Roadmap [6] published at 2013, predicted the entry into the service of the more electric aircraft, namely aircraft with electric systems, before 2020; while the mid-size full electric aircraft might be introduced into the operation after 2035 (if the accumulator power density would be increased considerable about 3 times comparing to the recent technologies [7]).

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The policy makers, regulators, stakeholders' teams define too ambitious goals as reduction of 90% in NO_x-emissions, and 75% in CO_2-emissions [3, 4]. Other international, national organizations are on the same positions [8-10]. By using the simple innovative solutions, e.g. improving the existing technologies, solutions (applying more light materials, advanced aerodynamics, increasing engine and propulsion system efficiency, engine – airframe integration, new ATM, rethinking of the operational concepts, ground and flight operations, etc.) that goals cannot be achieved. Therefore, the radically new technologies, original and unconventional solutions are required (developing by using the thinking out of the box principle [11 - 13]).

Probably, the most powerful solution enabling to achieve the targeted emission reduction is the development of the new aircraft with electric or hybrid propulsion systems.

It seems the electric aircraft concept is not so new. According to the list of electric aircraft published by Wikipedia [14], the first electrically driven flying machine was a dirigible "La France", constructed by Charles Renard, Arthur Constantin Krebs and Hervé Mangon and flown in 1883 – 84. The second electric aircraft was the PKZ-1 rotorcraft developed by Petróczy, Kármán and Žurovec for replacing the dangerous hydrogen filled observation balloons. The rotorcraft built in Hungary in 1917 under control of the young von Kármán had 3.9 m four-blades rotor driven by an Austrian – Daimler electric motor generating 140 KW (190 hp) power and supported by the ground based generator fed direct current through tethering cables.

Because the low power/weight ratio, the electric aircraft could not find their real implementation. At 1943, the Lockheed developing the bomber L-249 (got XB-30 code in army) studied the use of electric power assisted control, but it was found the hydraulic systems are considerable efficient [15]. The next large step was done by developing and implementing a 63 kW electric fuel pump and controller for F-16 [15]. Finally after revolutionized power electronic weight since 1992 the electronically based hardware became enabling to use in aircraft.

Nowadays, all the stakeholders policy makers, regulators, aircraft designers, producers, operators, etc.) are intensively working on development of the aircraft with full electric and hybrid propulsion systems. Figure 1. shows a series of electric aircraft according to their first flight. This figure demonstrates that, the first airplanes were developed as modification of the existing airplanes or developing aircraft of conventional forms.

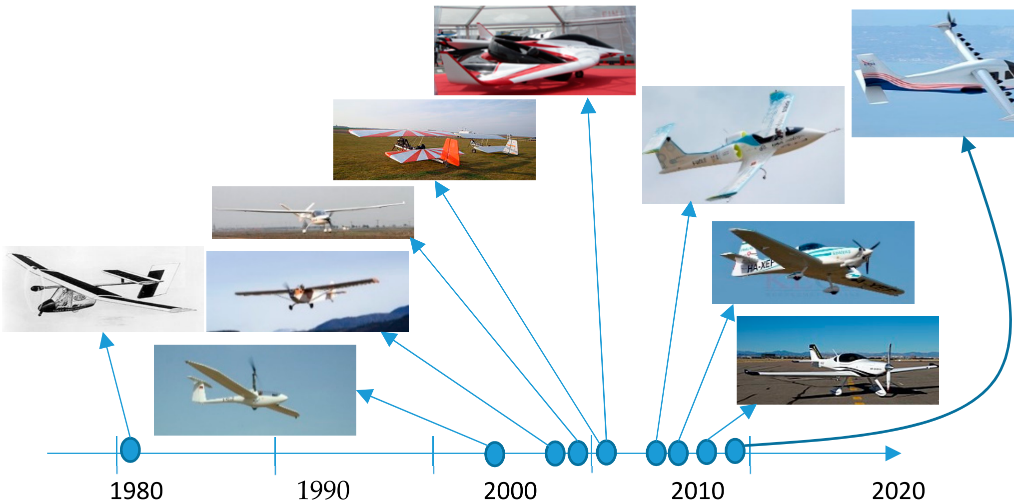

Figure 1. Some electric aircraft [16]: 1.- Solar Challenger, 1981 (first flight); 2.- Air Energy AE-1 Silent (ultralight sailplane with retractable electric motor), 2005; 3.- French BL1E Electra F-PMDJ (the first registered electric aircraft in the world), 2007; 4.- Yuneec International E430, China, 2009; 5.- APEV Pouchelec, 2011; 6.- AugustaWestland (Leonardo) Project Zero, 2011; 7.- Airbus E-Fan, 2014; 8.- Magnus –Siemens eFusion, 2016; 9.- By Aerospace Sun Flyer 2, 2018; 10.- NASA X-57 Maxwell, (18 distributed engines)

The recent and further developments of electric / hybrid aircraft face with series serious problems, technological barriers like low specific energy, thermal instability of batteries. The specific energy of accumulator packs, energy per mass (kWh/kg), equals to 340 – 360 Wh/kg as best. In real applications, the specific energy (SE) of accumulators is about 250 Wh/kg. (So, using 300 Wh/kg is a good approximation.) That means accumulators weighting about 12.4 kg (see 2.1.) may store energy equals to 1 kg kerosene in aircraft usage. Sophisticated cooling system and thermal management must be developed for managing the thermal instability of the battery banks. These are serious barriers in developing the full electric (and hybrid) aircraft. Therefore, there are three possible ways for developing the full electric aircraft. (i) The developers may wait for appearing radically new energy storage technologies, which is not a good option. (ii) Using the traditional, conventional developing philosophies and design methods that result to aircraft with considerable reduced performance, especially 5 – 8 times less range (in case of the analogical other performance). (iii) Creating a new approach to the preliminary and conceptual design of the aircraft.

This paper follows the third option. The new methods may base on energy and mass balance analysis. This paper analyses the identified constrains and recommends to use the aircraft mass and energy equations for understanding the problems and barriers balking and delaying the developing the full electric aircraft and their quicker introducing into the operation. It describes a series of new, even radically new ideas how to improve the performances of the electric and hybrid aircraft. It integrates the energy and mass balance equations into the preliminary definition and calculations of the aircraft performance. By this way, the technological constrains might be transferred into the limitation on the aircraft energy and mass breakdown, that initiates a new approach to aircraft conceptual design. The paper shows results of implementing this new development philosophy to conceptual design of a four-seat small aircraft and a special cargo UAV. The discussion of the results got by using the emerging and enabling new technologies and new methods and solutions (including for example distributed propulsion system, unconventional forms, morphing, biomimics, etc.) demonstrates the possible implementation of the new development philosophy, new approach to aircraft conceptual design.

2. Preliminary considerations

2.1. Required energy and engine power

In first and very draft order, the aircraft required energy support can be defined as choosing the engine type, its sizing and determining the required energy, fuel, for mission.

The demand using as input in new aircraft developments defines the engine types. For example, the Pplane and Esposa projects [17 - 19] demonstrated, the total-life cycle cost related to the one flying hours, safety and security risks, demand in availability and door-to-door performances determine the major performance and engine types (Figure 2.).

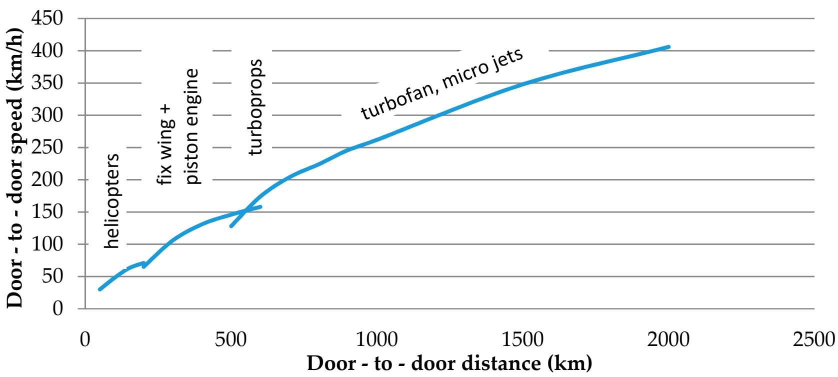

Figure 2. Relationship between the door – to – door speed and aircraft categories of the small conventional aircraft

The time, τ_a, of availability of air taxi service (minutes from requires until service availability) might be approximated as $\tau_a = 15 + 0.02R$, where R is the travelling distance in km. The approximation formulas $V_{DD} = (2.1 + 0.0004R)^{0.67}$, $\tau_{DD} = R/V_{DD}$ define the door-to-door speed (V_{DD} in km/h) and time (T_{DD} in h) for small aircraft.

The operational concept describes using the developing product or service from the user point of view. The flight mission origins from the market needs and it is an important part of the operational concept. It contains realization of the full flight including the flight modes: taxiing, take-off, take-off flight path, climb, cruise / en route flight, descent / gliding, loiter, landing approach, landing (and taxiing). The energy must be determined for such mission, in which the altitude and the speed of cruise flight (as level flight with constant velocity) are the most important variables (for commercial aircraft). The total energy (fuel) required includes the energy needed for solving the flight mission and additional a half hour flight as reserve.

The aircraft design process is a multidisciplinary nonlinear optimization process with large series of constraints [20, 21 (Part I., and II.), 22]. There are several types of constraints as legal, economic, technological, mechanical, aerodynamic, flight performance, flight mission, flight dynamics, stability, control, etc. The legal constraints are defined by the airworthiness requirements as mechanical (stress, weight) constraints or as flight performance, stability and control criteria.). For example, as it identified [23, 24], the very light airplanes (with take-off mass less than 750 kg and stalling speed less than 83 km/h)) should have take-off distance (horizontal part of flight path from start up to reaching the 11 m above the take-off surface) less than 500 m [25]. According to the Certification Specification, CS 23 [26] the normal, utility and aerobatic category reciprocating engine-powered aeroplane of 2 722 kg or less maximum weight must have a steady gradient of climb at sea level of at least 8.3% for landplanes and 6.7% for seaplanes and amphibians with retracted landing gears and flaps in take-off position. The analogical aeroplanes with gas turbines and extended landing gear should have 4 % steady gradient of climb after take-off.

In any case, the regulations as usually follow only the technological changes. Now, there is a time for creating the certification specifications for full electric, hybrid and highly automated aircraft including the unmanned and autonomous versions, too. The regulation must cover the testing and certification of the new technologies like batteries thermal instability or load test of operating gas turbine – electric energy generation units rotating with more than 100 000 rpm.

By the way, the two seats plane Pipistrel Alpha Electro is the first all-electric trainer aircraft getting the FAA airworthiness certification [27] that validates, the new technologies might be certified within the existing system of airworthiness and requirements.

Future greener aviation should use the electric or (because the technology barriers – see next sub-chapter) hybrid propulsion systems. The governing characteristics is the hybridization factor defined by Appendix I. that gives information about the charging, discharging characteristics of batteries, too.

The conceptual design intents to develop the aircraft in conceptual form. It has three major steps [20]. (i) The first guess sizing based on analysis of future needs, requirements, available and emerging technologies and on the "sketches". (ii) The second step is an iteration process of sizing and performance optimisation developing the initial layout. The process is based on analysis and optimisation of aerodynamics, weighting (weight balances), propulsion and flight performance. (iii) Finally, the third step of the conceptual design deals with second iteration with goal of refined sizing and optimisation including the structure, cost, etc.

A citation from know basic book on conceptual design [20] published by Daniel Raymer: "Sometimes a design will begin as an innovative idea rather than as a response to a given requirements." may underline the role of conceptual design in developing the future, unconventional aircraft.

The well-known and applied methods of the aircraft conventional conceptual design [20, 21 (Part I., and II.), 22] have been adapted to electric and hybrid aircraft design by several researchers, institutions, universities [7, 28 – 33]. The applied methodologies include three groups of improvements: (i) reformulation of the operation concept and mission, (ii) correcting the weight/mass formulas and (iii) improving the aircraft performance (range) determining methods.

Figure 3. depicts the improved conceptual design methodology. The operational concept derived from market needs, available and emerging technologies and qualified by the airworthiness requirements. All these provide inputs for the conceptual design.

The preliminary calculation uses the available data on the existing solutions, constructions, aerodynamic, etc. characteristics of analogical aircraft and theoretical and semi practical methods and provides the aerodynamics coefficient, preliminary defined mass balance and relative masses (weights), etc. supporting the future studies. Principally, this is the initial guess sizing.

There are two major cycles embedded. First deals with calculation of the required (maximum) power. The second determines the total required energy for supporting the full flight mission. The objectives of these actions are the synthesis the full aircraft in optimal form depending on the constraints. This is the second step of the conventional conceptual design process.

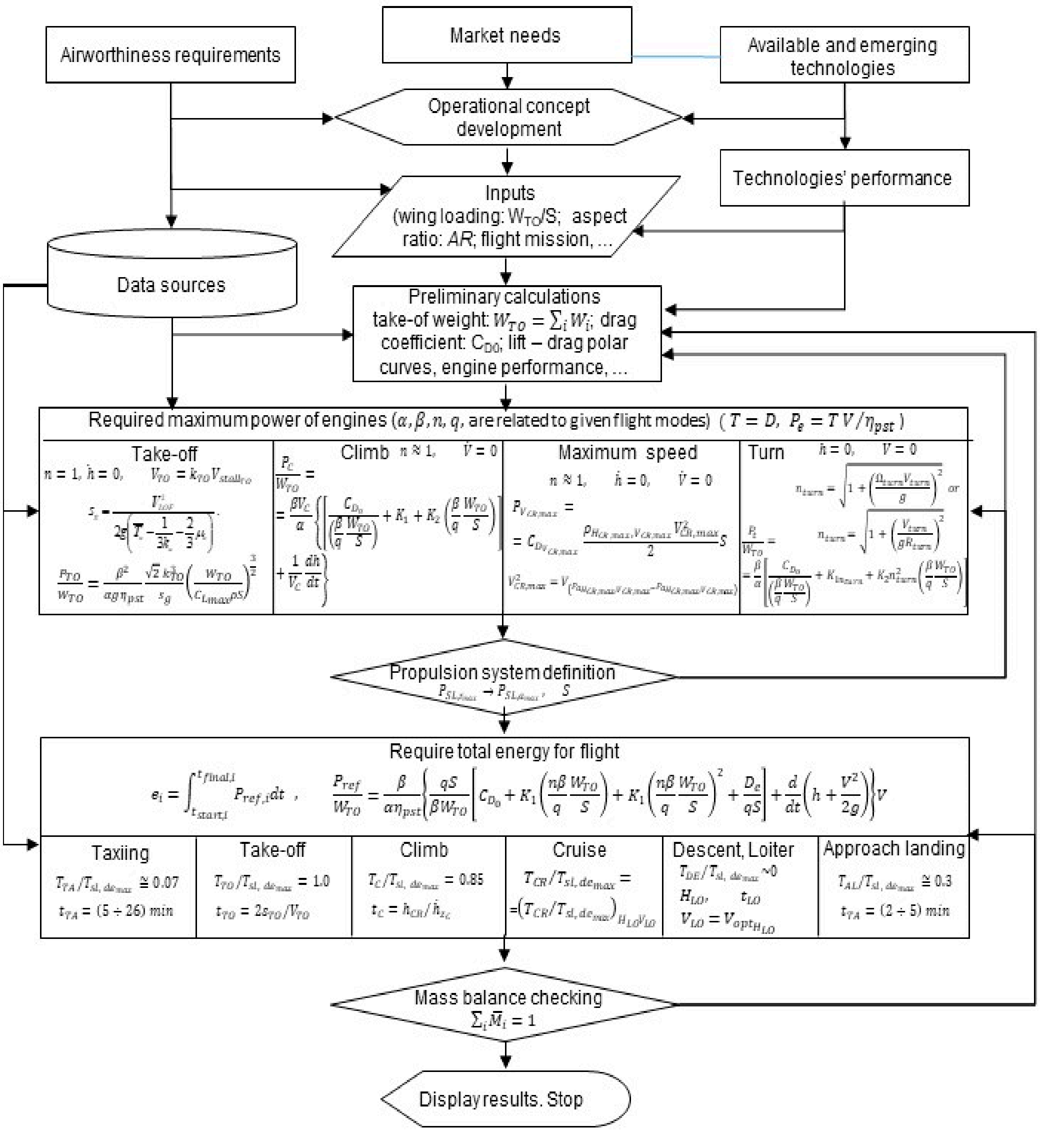

Figure 3. The conventional conceptual design methodology adapted to developing the electric and hybrid aircraft

The design process might be stop, if the mass balance checking demonstrates the acceptable and realizable results. Of course the aerodynamics, flight performance, stability and control, etc. requirements should be complied, too. However, the revised layout, the refined sizing and performance optimisation including for example the structural solutions, life-cycle cost and environmental impact minimisation, etc. may really finish the conceptual design process.

Figure 3. summarizes the methods of calculation the required maximum power and mission total energy for different flight phases. The Appendix II. and III. define the applicable formulas based on the theoretical and semi practical methods.

The Appendix IV. shortly describes the weight and energy balances equations that are very important parts of methodology, too. Chapter 3 describes their role in developing methodology and the ne methodology.

2.2. Possible energy support

There are several methods may improve and greening the future aviation starting from considerable drag reduction (like laminar wings, laminar aircraft, active flow control by use of MEMS (Micro-Electro-Mechanical Systems) technologies) up to introducing new energy generation methods (as fuel cells) and alternative "fuels" including electric energy as fuel, too.

This paper focuses its activity on the electric / hybrid propulsion systems, and their implementation only. This means, the paper deals with full electric propulsion system and with hybrid propulsion aircraft using the full electric modes in all flight and movements under 500 m of flight height (for small aircraft) and under 1000 m (for larger aircraft). Therefore, the applied methods of electric energy generation (as fuel cell, electric generators driven by internal combustion engines / gas turbines, or integrated photovoltaic elements into aircraft surface / structure) are less important in this study.

Figure 4. comparing the characteristics of the electric batteries to gasoline (jet fuel) [34 – 36], calls up attention on a very serious problem, small specific energy of the available accumulator technologies. It is a very important barrier delaying the electric / hybrid aircraft developments.

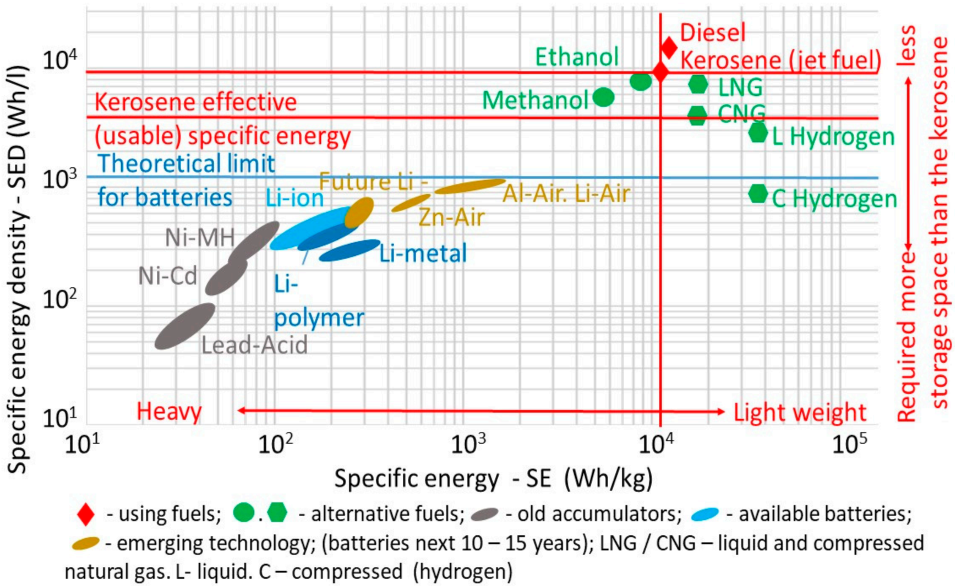

Figure 4. Energetic comparison of the applicable fuels

This figure may explain several interesting problems. At first, the specific energy of kerosene about 30 – 40 times greater than the specific energy of the available batteries. By taking into account the total efficiency of the propulsion systems (equal to about 24 – 28 % in case of conventional and around 76 – 82 % in case of full electric systems including the propellers' efficiency, too), a kg of kerosene contains usable (useful) energy that might be stored by 12 – 13 kg batteries. That means the useful energy of 1 litre kerosene can be stored in electric form in 9 – 10 kg batteries.

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The law specific energy of the battery technologies does not allow to make a full electric aircraft without considerable reduction in their performances. The mass of full electric aircraft increases for 40 - 400 % depending on the accepted radical reduction in range. Figure 5. demonstrates, the changes in relative mass balance of the he possible small size (around 50-seat) regional hybrid aircraft [28] equipped by electric butteries with 500 Wh/kg specific energy (that unit is about 60 – 70 % greater than specific energy of the today available batteries). Commercial load relative mass of such aircraft will be less for 48 %, (cutting down from value 25,8 % to 13,1 %) in case electrification factor equals to 75 %. At the same time, the take-off gross mass increases for 96 %, the required wing area for 134 %. So, such large electrification today is not realizable; the cost of transporting the goods, people may double and more.

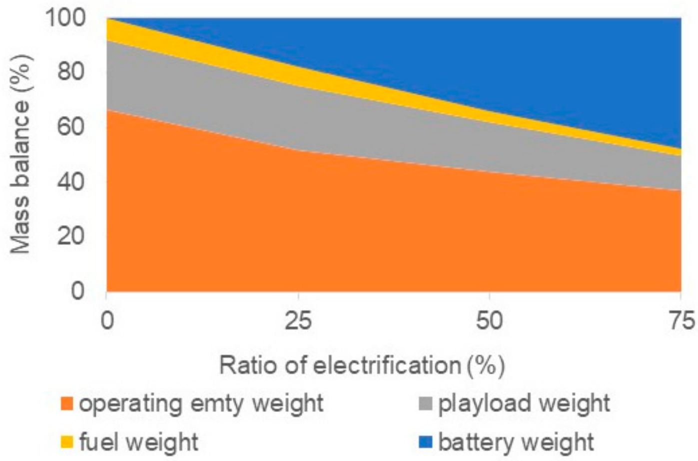

Figure 5. Weight balance of the middle size regional hybrid aircraft depending on electrification (drawn by use of results published by Antcliff et al. [28]

People may staying in position, this technological barrier (low specific energy) is not so hard problem, because the sciences and technologies are developing very actively in accelerated form and the diffusion time of new products into the market considerable and continuously reduces. At first, as it can be seen in Figure 4., the specific energies of the available lithium-ion, lithium-polymer accumulators are close to their technical limits. The future, emerging technologies allow to develop the aluminium-air, lithium-air accumulators, which may have already the acceptable specific energy density. At second, the accumulator technology developments are not follow the Moore's law [37], according to which the transistors on a chip doubles every year while the costs are halved. So, the availability of the required emerging accumulator technologies at 2030 – 2035 might be too optimistic expectation. Plus to it, the emerging accumulator technology may reach the theoretical limits of the known battery technologies. For success absolutely new energy storage technology should be developed.

Another important problem is generating by a false hope: the electric aircraft have zero emission. The media, green organisations, foundations and generally the societies accept, diffuse and propagandize this incorrect information as a "governing" principle. They are forgetting about the emissions of electric energy generation, accumulator and generally the aircraft production, building the required infrastructure, etc.

By taking into account all these effects and determining the total environmental impacts, the picture is not so nice [38, 39]. At first the electric energy generation causes large impact on the environment (Figure 6.) depending on the mix of energy generation [40]. At second, the total life cycle emission of existing and electric /hybrid aircraft are not so different. According to the Figure 7. the full electric aircraft may have less total environmental impact in case when the performance, especially the range of aircraft really cut down. The hybrid aircraft with light hybridization factor (equals to 10 – 25 %) may reduce the environmental impact in airport regions and generally the total emission for 4 – 7 %.

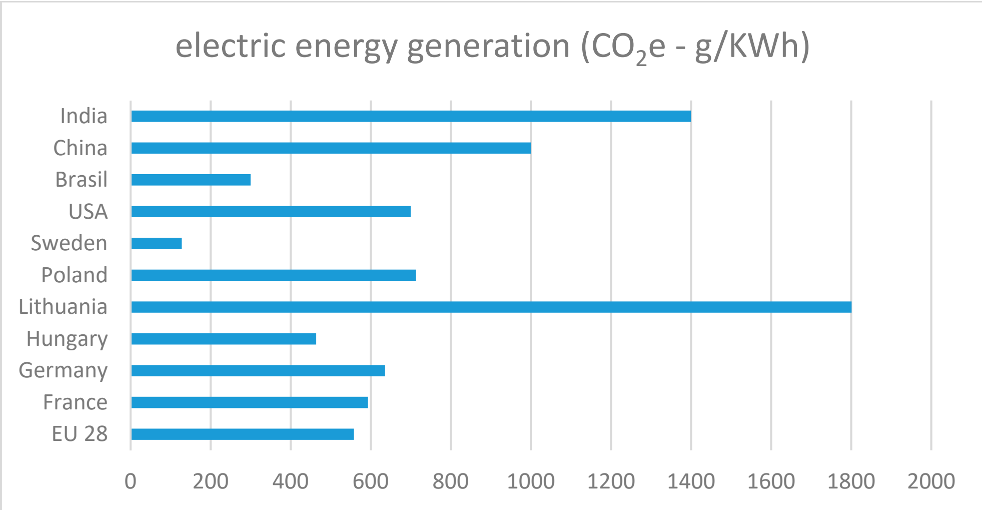

Figure 6. Greenhouse gas effect of electric energy generation in several selected countries (drawn by use of data published by [40] (and given in CO_2 equivalent form)

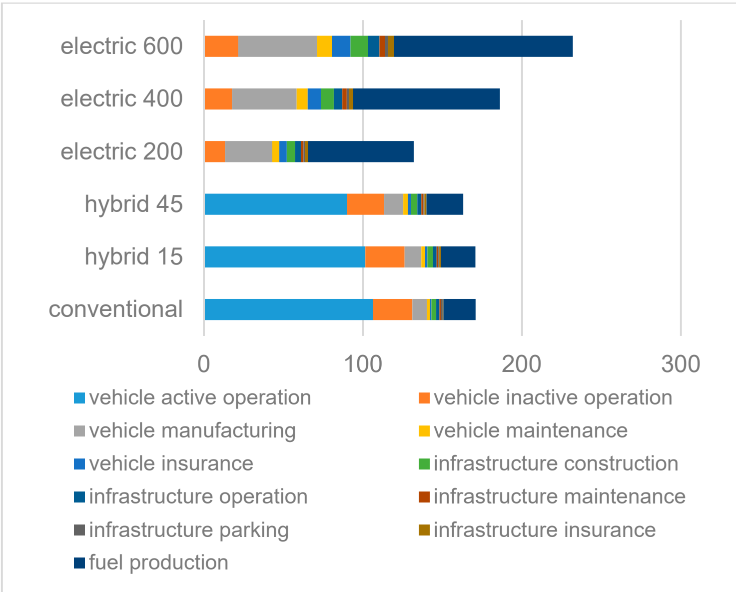

Figure 7. The total life cycle CO_{2e} emission of the investigated aircraft given in g/pkm form (hypothetical conventional aircraft analogical to Cessna 172, electric aircraft with buttery banks 600, 400, 200 KWh, hybrid aircraft applying only electric power for 45 or 15 minutes during its flight)

3. New approach to aircraft conceptual design

3.1. Aircraft mass and energy balance equations

The conventional conceptual design methodology adapted to developing the electric / hybrid aircraft (Figure 3.) applies the mass and energy balances for evaluating the concept and possible realization of the new aircraft. Here weight balance is used not as simple balance defining the aircraft centre of gravity. This is an equation describing the sum mass (weight) fractions.

Raymer [20] defined the aircraft take-off mass in very simplified form:

$$M_{TO} = M_e + M_{pl} + M_f + M_b \, . \tag{1}$$

By introducing the mass fractions (relative masses) for fuel and empty mass

$$M_{TO} = M_e + M_{pl} + \frac{M_f}{M_{TO}} M_{TO} + M \frac{M_b}{M_{TO}} M_{TO} \, , \tag{2}$$

the formula

$$M_{TO} = \frac{M_e + M_{pl}}{1 - \frac{M_f}{M_{TO}} - \frac{M_b}{M_{TO}}} \tag{3}$$

can be used for determining the take-off mass with use of estimated fuel and empty mass. This is often called as weight budget formula that uses the weight breakdown defined by (1).

The equation (1) can be given in more generalized form:

$$M_{TO} = M_a + M_{sy} + M_{ps} + M_f + M_b + M_{pl} + \cdots = M_{TO} \sum_i \bar{M}_i \ . \tag{4}$$

Because the mass fractions (relative masses) might be defined as function of the available technological and production levels (see Appendix IV.) this equation might be called as mass balance equation. In more general form, this equation represents the possible realization existence of aircraft. So, it is the aircraft existence (or life) equation. If the $\sum_i \bar{M}_i > 1$, than the aircraft with predefined performance can not be built in the given production environment (in given country) with planned performances. The $\sum_i \bar{M}_i < 1$ condition shows that even better aircraft with better performance might be built. Finally, the $\sum_i \bar{M}_i = 1$ means the best aircraft is built.

There are many general [20, 21 (Part I., and II.), 22, 41 - 43] and specialized [44, 45] books deal with defining, description and estimated semi practical formulas of the aircraft parts including for example mass of wings or fuselages depending on their real structural solutions and load conditions, load management.

These practical formulas are widely used in aircraft conceptual and preliminary design.

The energy balance is an analogical equation estimating the total energy required for realization the aircraft flight mission (sum of energy consumptions for the mission legs - see Appendix IV.):

$$e_{efm} = e_{TA} + e_{TO} + e_C + e_{CR} + e_{DE} + e_{LO} + e_{AL} = e_{efm} \sum_i \bar{e}_i \ . \tag{5}$$

This equation can be applied for optimization of the energy consumption, energy management.

3.2. New conceptual design methodology for developing the electric / hybrid aircraft

The conventional conceptual design methodology (Figure 3) had been applied to developing five types of 4-seat small aircraft with different propulsion systems [19, 23, 24]. The aircraft performance were compared with a hypothetical conventional aircraft analogical to Cessna 172N with range 1300 km and cruise speed 226 km/h. The draft names (aircraft 200, aircraft 400, …, hybrid 15, …) of developing aircraft "explain" the major characteristics of the propulsion systems. The batteries of developing aircraft may storage the 200, 400 and 600 KWh. In cases of using the hybrid propulsion system, the aircraft are able to fly 15 or 45 minutes in full electric modes.

Figure 8. shows the radical increases in total masses and required mass of the battery banks. The conventional and hybrid 15, hybrid 45 aircraft have the same range 1300 km. The electric aircraft with 200, 400 and 600 kWh battery capacity have ranges 360, 520 and 640 km, only.

Figure 9. underlines another important problems associated by mass balance (and energy) balance. As it can be seen in figure, the relative mass of propulsion system (including the engines or electric motors, propellers, system elements like axis, wires, cooling system and batteries masses) increases up to 65 %; while the relative mass of commercial load (payload) reduces from 36 to 10 %. These results rather similar with result got in conceptual design of larger (50-seaters) aircraft (see Figure 5.).

This changes in mass breakdown generated new idea: the future electric / hybrid aircraft should be developed by introducing a series of preliminary defined extra constraints on the mass and energy balances. Of course another constraints (for example on total life cycle cost) might be defined, too.

Generally, by developing the technology for electric / hybrid aircraft the design space increases dramatically [46] that requires radically new solutions like synergetic, tightly-coupled morphological and systems integration emphasizing propulsion – as exemplified by the potential afforded by distributed propulsion solutions [46 – 48]. Investigation of the potential and limitations [49]

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introduced by electric flight concept [19, 23, 24, 28, 33, 50] leads to rethinking the conventional conceptual design methodologies of aircraft.

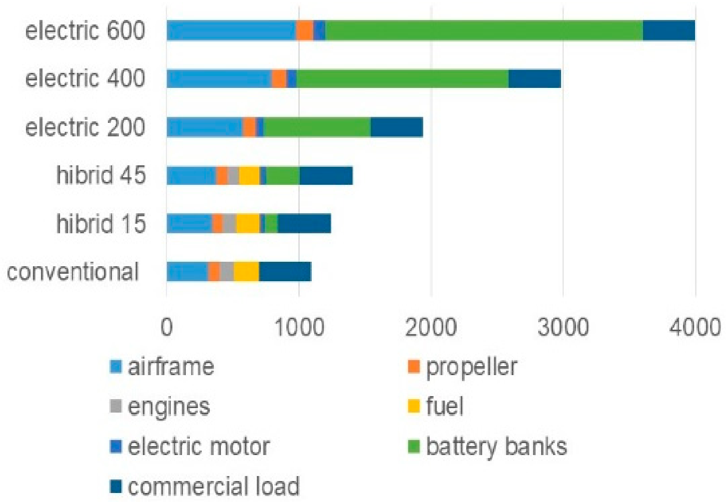

Figure 8. The mass breakdown (kg) of the a series of conceptually designed aircraft analysed aircraft

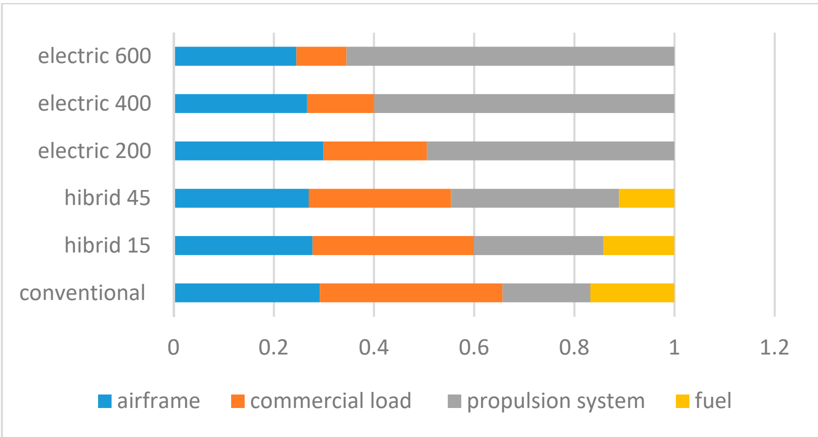

Figure 9. The mass balance (mass fractions) of the conceptually designed aircraft.

The developed and proposed new approach to aircraft conceptual design methodology has the following important novelties:

- adaptation of the conventional conceptual design methodology to electric / hybrid aircraft developments based on iteration on required maximum power and energy required for realization the predefined flight mission,
- introducing new, predefined constraints for mass and energy balances (mass and energy fractions),
- generating original new solutions by applying the thinking out of the box concepts.

The adapted methodology uses four types of inputs (see Appendix I. and II.):

- performance derived from developing operational concept (as number of seats, n_{se}, range, R, cruise speed, V_{CR} maximized take-off distance, s_{TO}, etc.) and airworthiness requirements (as take-off distance, s_{TO}, or rate of climb, $\dot{h}_z$, load factor, n);
- predefined constrains (legal, economic, etc.) requirements including the extra constraints on mass and energy balances;
- technical, technological characteristics of the propulsion system (like hybridization factor, HF, battery discharge and recharge rates, C-rates, efficiency of elements of the propulsion systems,

battery specific energy, SE_b, engine specific energy (or / and power-to-weight ratio) of the conventional engines (ICE, GT, TP), SE_{ce};

- aerodynamic data and flight performance (as polar curve diagrams, $C_L = f(C_D)$, wing aspect ratio, AR, wing loading, $w_L = W_{TO} / S$).

Using extra constraints on mass and energy balances calls for developing very special aircraft, operating by applying radically new solutions. For example, the mass of airframe can be reduced by deployment of the new materials, lightweight flexible airframe, like wing (of course with gust effect elimination and life management), morphing, structural solutions based on biological analogies (biomimicry), etc.

Some constraints introduce accepted changes in performances. For example accepting reduction in flight performance, less cruise speed, longer take-off distance, etc. may lessening the mass of required batteries.

The radically new, original solutions may implement the ideas introduced by projects developing the "out of the box thinking" solutions [11 – 13, 51 - 52], including take-off by ground energy support (Figure. 10.), aerial refuelling or cruiser – feeder concept. Recently, most of revolutionary new ideas as ground support by energy the aircraft during take-off or replacing the battery packs during cruise flight would seem too radical, unrealizable solutions, while delays in their implementation caused by problems of society, stakeholders acceptances rather than technical unsolved problems.

Figure 10. Take-off and landing assisted by magnetic levitation system supported from ground

The developers may define new constraints partly subjectively.

4. Applicability of the new conceptual design methodology

4.1. Conceptual design of a 4-seats aircraft

The professional task of IDEA-E project [50] was defined as to analyse and develop the electric and hybrid aircraft. According to the market analysis, there were identified needs in 4 –seaters small aircraft and cargo UAV.

The latest results of sciences and technologies allow emerging a safety, environmental friendly and economic new Personal Air Transportation System (PATS) [54 – 55].

There are many projects developing the future small aircraft and small air transportation systems. EPATS - European Personal Air Transportation Systems [56, 57], PPlane - Personal Plane [58, 59], SAT-Rdmp Small Aircraft Transport – Roadmap [60], or Esposa – Efficient Systems and Propulsion for Small Aircraft [61] are the EU supported projects. Others, as NASA SATS - Small Aircraft Transportation System [62, 63] or IDEA-E [50] are the national projects.

All these projects have developed new small aircraft, small engines, new technologies and solutions that open new business sector in air transport system. This new business supports the personal air transport including even daily use of aircraft. This new system required new small aircraft, new set of smart airport, new way of integration of this transport into the general transport.

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Finally, the projects resulted to definition the personal aircraft that may have up to 9 seats and limited range (about 800 km) and limited flight time (less than 3 hours). This performance may allow to use the electric or hybrid propulsion system.

The objectives of a national project IDEA-E [50] includes development of a 4 seats hybrid aircraft. The project team has made large work and has published the first results already according to investigation of elements of the propulsion systems (as batteries [36], hydrogen cells [64] and developing aircraft [19, 23, 24] (see Figures 8. and 9., too).

Figure 11. proves an example of mass balance of an electric aircraft developed by using the predefined additional constraints (as limitation on the relative mass of airframe) comparing with conventional and electric 600 aircraft. As it can be seen, the electric predefined aircraft has relative mass of power storage system (batteries) considerable smaller than the electric aircraft 600 (designed by using the conventional conceptual methodology) but – of course – much more greater than the mass of fuel used by conventional aircraft.

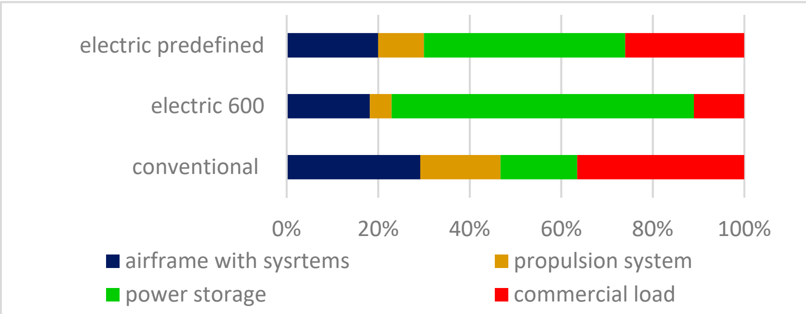

Figure 11. Effect of limitations on the mass balance (conventional, electric 600 and future electric aircraft)

The available technologies (as new composite materials, new lighter engines, etc.) are ready to support the small electric / hybrid aircraft developments. Two factors, reducing the empty mass fraction for 20 – 25 % and range for about 30 – 60 % comparing to the existing aircraft (most of them were developed 20 – 60 years ago, and give rooms for considerable improving the structure and efficiency) may results to required performance. The range of small 4-seaters aircraft might be defined by the modified Breguet range equation (see Appendix V.):

$$R_{hybrid} = \frac{\eta_{pst_{ce}}\eta_{pst_e}}{\left(PSFC\frac{H_{fuel}}{g}(1-PS)\eta_{pst_e}+PS\eta_{pst_{ce}}\right)} k \frac{H_{bat}H_{fuel}}{g(\psi H_{fuel}+(1-\psi)H_{bat})} \ln\left(\frac{\frac{\psi H_{fuel}+(1-\psi)H_{bat}}{H_{bat}H_{fuel}} g E_{total}+W_{empty}+W_{payload}}{W_{empty}+W_{payload}}\right). \quad (6)$$

Figure 12. defines the change distribution in range of 4 seaters small aircraft (analogical to the Cessna 172) depending the hybridization (energy split) factor, ψ, and the cruise speed, M [65].

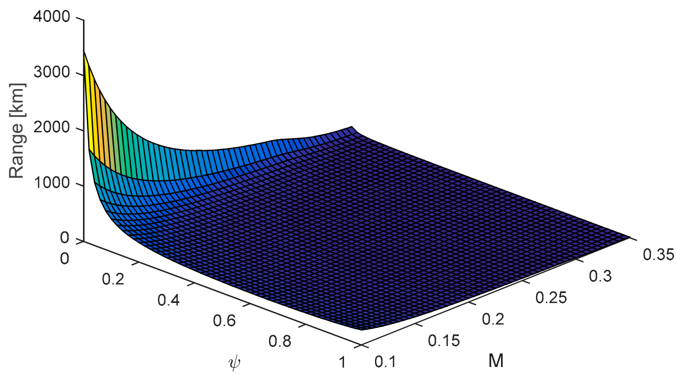

Figure 12. Range depending on hybridization factor and cruise speed

As it can be seen, the best solutions are the hybrid aircraft with light hybridization.

The introduced conceptual design method applying to 4 seaters hybrid aircraft design (in case of using the weight estimation functions provided by Torenbeek [22] and limiting the range as 400 km) results to stable take-off mass after 8 – 10 iteration already [66].

4.2. Conceptual design of a cargo UAV aircraft

This sub-chapter describes and shortly discusses the first results of the conceptual design of family of cargo UAVs with electric / hybrid propulsion systems [24, 64, 67, 68].

The large cargo UAV developments, as PUKA [69] and small drones' application for goods delivery [70] in cities are well known. However, there are a lack in developing the moderate cargo UAVs. For instant, Boeing develops a special cargo air vehicle as electric vertical take-off and landing (eVTOL) aircraft for transporting goods of weight 339 kg (Fig. 9.). As it might be recognized this cargo UAV has rather unconventional form. This is a way, how the effective UAV might be built.

Figure 13. The Boeing HorizonX is an unconventional eVTOL cargo air vehicle [71]

The preliminary market analysis identified needs in cargo UAV carrying 1 – 10 hundreds of kg payload for distance 100 – 300 km. This wide range in required performance calls for developing a series of aircraft. There are three different sizes, but analogical aircraft under development. Each aircraft may adopt to transfer 1 – 3 containers. The smallest may hold containers of 50 kg mass, that is to say, the aircraft commercial load might be 50, 100 or 150 kg. It has a range 100 km. The larger airplane may carry containers of 200, 300 or 400 kg mass for distance 200 km. While the larges can transport goods of 600, 800 or 1000 kg for up to 400 km [24].

The flight mission is defined very simple: taxiing, take-off, climb, cruise, descent, approach and landing. The cruise modes, (velocity and altitude) are optimized for block time, and actual commercial load. The block time (time from starting the take-off until stopping the aircraft after landing) is defined as 1 hour 30 minutes, 2 hours and 2 hours 30 minutes for the 3 different size of aircraft with full loads. Because the highly adaptive wings the users may operate, the aircraft flying on smaller cruise speed (that increases the block time). The defined take-off distances are 60, 100 and 200 m. Rate of climb is defined as minimum 3 m/sec

It seems the largest aircraft carrying 600 – 1000 kg loads rather similar with small 6 – 9 seats aircraft. On the other hand, developing the smallest version faces critical problems. At least the known smallest aircraft like Cri-Cri (Fig. 10.) or SD-1 have relative empty mass (empty mass per take-off mass) equals to 0.45 or higher. (By the way, on 9 July 2015 an electric version of Cri-Cri, built by Electravia flew across the English Channel hours before the Airbus E-fun.)

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Figure 14. Cri-Cri, the World smallest twin engine manned aircraft designed by French Colomban (https://en.wikipedia.org/wiki/Colomban_Cri-cri)

The IDEA-E [50] team defined to start the developing process with the smallest cargo UAV in its maximized form carrying 150 kg payload for distance 100 km during 90 minutes. For reaching the objectives a series of extra constraints were defined [24]:

- the structural mass (as relative empty mass without the elements of the propulsion system) must not go over 0.2;
- the relative mass of batteries should be less than 10 %;
- the commercial load should be greater than 40 %;
- under 500 m height of flight, the aircraft must use full electric modes;
- the electrification factor must be below 0.2; and
- the power of the aircraft must be enough for recharging, the batteries fully during the cruise flight.

The developed aircraft has the following geometrical characteristics (for case of carrying the 150 kg cargo): length: 4.7 m; wingspan: 7.6 m; wing area: 9.4 m^2; take-off mass: 350 kg, take-off distance: 60 m.

The developing aircraft must have really unconventional form for guarantee the required efficiency and performance.

It seems the airfoil selection / design and determination of the wing / aircraft aerodynamic characteristics are the central problems of developing this new, unconventional airplane. The modified (in thickness, camber, positions of maximum thickness and camber) and combination of two different airfoils and specially developing airfoils were studied. The applied solution requires further aerodynamic investigation and development.

The applied unconventional form results to considerable reduced empty weigh of airplanes, but the parasite drag (drag at zero lift) is about 30 – 40 % greater, than the aircraft with analogical weight / size. Therefore, the cruise speed should be reduced and kept moderate.

The hybrid propulsion system are selected because the technological barriers delaying the full electric aircraft developments. The system is based on a small size gas turbine driving an electric generator. This engine – generator is mounted in central part of the fuselage. Air inlet is at the upper side of the noise (Fig. 15.). The inlet is at upper side too, or at the both lower side of fuselage after gas turbine and commercial load packages. 4 – 8 electrically driven propellers having own small electric motors equip the aircraft. The propellers and their motors are integrated into the wing and they “work” on full span.

There is not conventional fuselage and cockpit. The airframe is built from the minimum elements: fuselage as beam truss box with gas turbine – electric energy generator and batteries inside, wings of variable geometry and tail (Fig. 15.)

The control unit is mounted into the “head” of aircraft that is streamlined small body made as independent composite unit.

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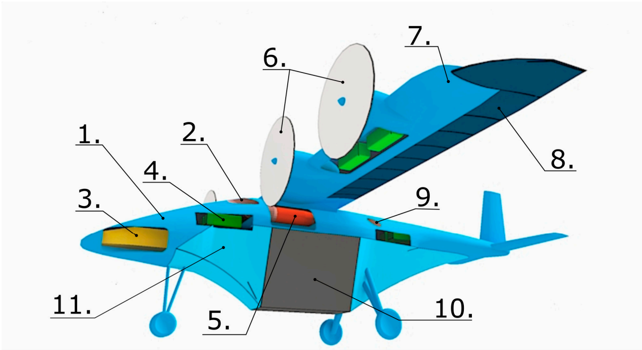

Figure 15. Layout of developing small cargo UAV with hybrid propulsion system (1.- fuselage, 2.- engine inlet, 3.- control box, 4.- batteries, 5.- gas turbine with electric generator, 6.- propellers, 7.- fix wing, 8.- flexible part of wing, 9.- exhaust gas inlet, 10.- containers, 11.- covering linen)

The carrying containers are fixed to the fuselage beam and they are cover by linen strained by shape-retaining frames.

The wing has two different sections. The central "rigid" body section holding the propellers driven by individual electric motors (Fig. 16.). The mean beam is a tube. The fixed wing holds batteries, too.

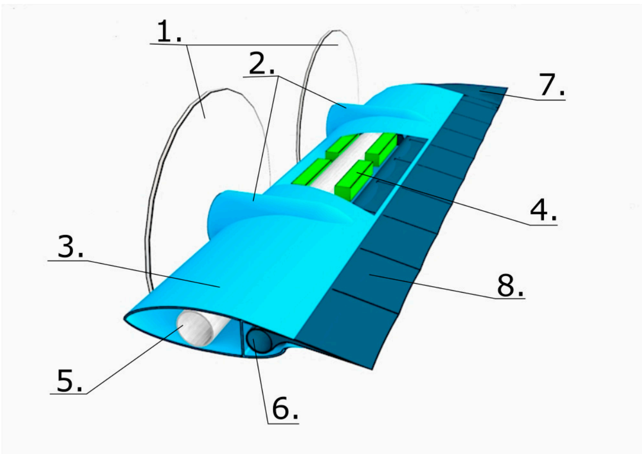

Figure 16. Wing principal structure (1.- propellers, 2.- electric motors, 3.- hard (composite) wing section, 4.- batteries, 5.- beam – tube, 6.- rod rolling the linen, 7.- flexible (composite), deflectable tip rod, 8.- flexible part of wing (linen))

The variable part of wing is a simple, curved linen. It is partly reeled up on a tube at middle section of wing. The trailing edge is a cable in the linen. This cable is stretched between flexible (and deflectable) tip rod and an electric motor built into the fuselage. This motor controls the stretching and position of cable. The motor may move back and down for increasing the wing area and the generating extra lift. This is the flap function. The tip rod can be turned, too, down and up for using the wing tip areas as ailerons.

The conceptual design methodology (Fig. 5.) results to the following major mass and energy balances (Tables 1. and 2.) [24].

The results demonstrate the special, unconventional cargo UAV might be configured and realized.

The project has not finished yet. At first, further investigation is planned for developing and CFD study the airfoil and airplane aerodynamics, improving the structural realisation of the created original solutions and their testing by using the FEM analyses, further investigation and analyies of the stability and new control solutions, and developing the gas-turbine- electric energy generator unit. Finally, the up-sizing effects require further studies and evaluations.

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Table 1. Mass balance of developed UAV

elements	mass (kg)	mass / MTOW
empty mass (without the propulsion system)	69.8	0.199
gas turbine	21.5	0.061
energy generator	8.2	0.023
fuel system	3.6	0.010
fuel	22.8	0.065
electric system including controllers for batteries	12.2	0.035
batteries	38.7	0.111
4 electric motors	12.8	0.037
4 propellers	10.4	0.030
commercial load	150	0.429
total	350	1.000

Table 2. Energy balance of developed UAV

flight phases	energy (kWh)	electric energy (kWh)	fossil used energy (kWh)	battery mass (kg)	total fuel (l)
taxiing	0.07	0.07		0.29	0.05
take-off (finished at 500 m - that is 4 minutes on P_{max})	3.67	3.67		14.67	2.29
climb to 3000 m (climb rate 3 m/s and 0.9 P_{max})	11.46		11.46		7.16
cruise (altitude 3 km. speed 25 m/s (90 km/h.))	20.32		20.32		12.70
descent	0.00				
approach/landing (from 500 m)	1.80	1.80		7.20	1.13
reserve	12.40	4.13	8.27	16.53	5.17
total	49.72	9.67	40.05	38.69	28.49

5. Discussions

The future cleaner air transport needs accelerated implementation of the new available and emerging technologies. The most promising new solutions might be realized by developing the aircraft with full electric and hybrid propulsion systems. The total life cycle cost and environmental impact might be reduced for 12 – 27 % depending on the size and flight performance of the aircraft. Using the full electric mode at under 1000 m of flight height may eliminate the chemical emissions in airport regions. Very intensive and active research, development and innovation processes have started for developing the full electric and hybrid aircraft. It is expected to introduce the first full electric and hybrid passenger aircraft (with 50 – 100 seats) into operation around 2035.

There is an interesting occurrence has appeared: the scientist, aircraft developer makes their calculation with battery performance 2 – 3 times better than the existing, available best accumulators have. Unfortunately, the available and emerging batteries' technology (because their small specific energy) may not support developing such new aircraft without radical improvements in battery technologies, introducing new energy storing technologies or considerable reductions in aircraft flight performance.

This paper identified the most serious problems and barriers generate delays in full electric and hybrid aircraft development.

It seems all the problems are induced by low specific energy (kWh/kg) of the batteries packs (Fig. 4.). It means, one kg kerosene support the conventional aircraft by effective used energy that can be stored in best available today battery pack having weight 12 – 13 kg.

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The Figure 8. shows, the full electric aircraft (even the small 4-seaters) aircraft can not be realized by using the existing battery technologies. The aircraft mass increases for 3.6 times while the range of aircraft reduces for 50 – 52 %. In case of developing a middle size (50 seats) regional aircraft and using the electric energy in 75 % of required total flight mission energy (75 % of hybridization factor), the take-off gross mass increases for 96 %, the required wing area for 134 % (Fig. 5). At the same time, the payload factor (commercial load mass fraction) will be less for 48 %, (cutting down from value 25,8 % to 13,1 %).

These large changes in performance are caused not only by large mass of required battery packs, but with fact, this mass must be carried during all the flight. Therefore, the conventionally applied Breguet range equation is improved, too.

Implementation of batteries for energy storage introduces series other problems as charging, discharging (see Appendix I.) of batteries, thermal instability, effects of integration of different propulsion units, distributed propulsion systems into one system on safety, security, cost breakdown, operational conditions and methods, certification aspects, etc.

The society false expectation: having absolutely green electric aircraft (Fig. 7.), has not direct influence on the aircraft performance, but it may affect on developing the hybrid aircraft with increased hybridization factor reducing the flight performance.

This paper recommends using a new approach to conceptual design of the future full electric and hybrid aircraft by introducing additional constraints and increasing the role of mass and energy balances. Here mass balance is a term used for equation summarizing the mass fractions of elements, which might be called as aircraft existing or life equations (see equation (4) that is well described and used by conventional conceptual design methods, too). This mass balance may show, the possible realization of an aircraft with predefined performance in the given production environment (see Appendix IV.). The paper introduced another balance equation, the energy balance as defining the total energy required for solving the flight mission. Together these mass and energy balances may show how the aircraft with predefined performance might be realized.

The major problem is how to select and define the inputs (including the extra constraints) for conceptual design, how to sketch the first layout of aircraft, and how analysing, selecting and applying the formulas for preliminary calculations. This is a real dilemma, is an electric / hybrid aircraft designers' dilemma (Appendix II.).

The designers' dilemma might be dissolved by

- introduction and definition of extra constraints,
- developing original and radically new, disruptive (out of the box) technologies and solutions and
- utilizing the art and creativity of designers,

The identification, analysis, selection and definition of the proposed extra constraints require further studies. About 40 – 60 years ago, a large step in progress was made by developing the methods of aircraft conceptual design and creating the semi theoretical functions for calculating the mass fractions. Still the same methods are applying with some adaptation to the new developments. Now, it is a time for considerable improving the aircraft conceptual design methods, methodology. For example, definition of the wing mass fraction models for the very flexible wing made from new lightweight materials, using the new structural solution, morphing technologies, MEMS based active flow control distributed propulsion integrated into the structure, etc. lead not a simple, considerable improvements in methodology. This is a new approach to conceptual design of aircraft, because the extra constraints give flexibility in application and allow realizing the multi-disciplinary optimization subject to implementation of the new technologies lie electric / hybrid technologies. (Because the large variety in technologies and solutions having influence on more accurate determination of the weight and energy fractions, a special international project is recommended to initiate for developing the new semi empirical, semi-theoretical formulas mass and energy fractions.)

By using this new approach and creating original (revolutionary new) solutions the full electric and hybrid aircraft might be developed by applying the existing battery technologies. Of course, original (radically new) solutions must be developed (by using the thinking out of the box). For instant, supporting the aircraft take-off by energy from the ground sources, using in-flight energy transfer from distances, batteries re-charging or changes in-flight, etc.

The paper demonstrates the applicability of the introduced ideas in developing new methodology for aircraft conceptual design (calculating with some extra constraints defined for aircraft mass and energy balances) by developing 4-seaters small electric / hybrid aircraft and by conceptual design of a family of original cargo UAV for carrying 1 - 10 hundred kg payload. The developed cargo UAV airplane families are recommended for use to delivering the fast international shipping and same day parcel delivery systems. These UAVs may apply in remote control modes or in autonomous transport systems.

Appendixes

Appendix I. Hybridization factor and the battery discharge and charge rates

Comparing to the conventional aircraft conceptual and preliminary design, some new inputs must be defined that characterize the electric batteries, and electric / hybrid propulsion systems. One of the important new inputs is the hybridization factor [46, 72]. Initially, the hybridization factor, (energy degree of hybridisation) , ψ, as the ratio of the energy stored in the batteries and total energy stored in fuel and batteries.

$$HF = \psi = \frac{E_{bat}}{E_{total}} = \frac{E_{bat}}{E_{bat}+E_{ce}} \ . \tag{7}$$

It was found [33, 46, 72], the hybridization factor only is not enough for defining the real "hybridization" of the systems. The other factors are the power split, (power degree of hybridisation),

$$PS = \frac{P_{em}}{P_{total}} = \frac{P_{em}}{P_{em}+P_{ce}} \tag{8}$$

characterizing the continuous shaft power providing by electric motor related to the total power (generated by the electric motor and applied conventional engines) and the supply power ratio,

$$SPR = \frac{\int P_{em}}{\int P_{total}} = \frac{\int P_{em}}{\int P_{em}+P_{ce}} \tag{9}$$

depicting the total electric motor power integrated over the flight mission in relation to the total shaft power integrated over the mission.

All these characteristics equal to zero in cases of using the conventional engines, only, and 1 for full electric aircraft.

According to the hybridization factor the hybrid propulsion system might be classified as [73]

- full hybrid system is the full electric system, when due to battery support the system may advance the aircraft on its full (standardized) mission,
- mild hybrid, when the purely electric mode of operation is not able to support the full flight,
- minimal hybrid system, when the aircraft can not fly on pure electric mode, the electric power works together with the conventional engines, in case of required.

The hybrid propulsion system (as well as the hybridization factor) depends on the battery discharge and charge rates This C-rate (Coulomb's rate) measures the rate at which a battery is discharged or recharged relative to its maximum capacity. The C-rate of most promising Li-Po accumulators equals to 10 – 30 depending on the temperature [74]. For general case the C = 12 is recommended to use [73] for discharging. That means the current battery pack with energy density 0.3 kWh/ kg, may support the electric motor by 0.3 kW for one hour or by 3.6 kW for 5 minutes. For three hours flight the continues usage of the power of such battery pack will have C = 1/3 discharge rate. At the same time, the recharging rate 2 C means the battery pack (of 0.3 kWh/kg) might be recharged fully during a half an hours. So, if during 5 minutes, the aircraft was using the battery full energy for take-off (in full electric regime or for maximizing the take-off power of the conventional engine(s) working together with electric motor(s)) then minimum a half an hour is required for recharging the batteries.

Appendix II. Delimemma of the electric /hybrid aircraft developers – inputs and preliminary calculations

This Appendix deals with appearing dilemma of the electric / hybrid aircraft developers in selection and definition the required inputs defined by sub-chapter 3.2. and using the inputs in preliminary calculations.

All the methods developed for aircraft preliminary and conceptual design [20, 21 (Part I. and II.), 22, 42 – 45, 75], based on the available data of existing aircraft and their statistical evaluation.

Unfortunately, all the aircraft (input) data describes the existing aircraft applying the existing (not future) technologies and all the preliminary calculation, namely in aerodynamics design and calculations [21 (Part VI.), 41, 76 – 78], aircraft flight performance determination [79 -81] are based (at least partly) on the semi-theoretical / semi-practical approximation formulas. Best example is giving the mass / weight fractions in statistical forms [21 (Part VI.), 41 – 45]. Because of it and opinion of Raymer [20] (cited already in main text) "Sometimes a design will begin as an innovative idea rather than as a response to a given requirement." – an interesting dilemma of electric /hybrid aircraft developers is appearing. The major problem is how to select and define the inputs for conceptual design, how to sketch the first layout of aircraft, and how analysing, selecting and applying the formulas for preliminary calculations. This dilemma causes the required definition of extra constrains for introduced new approach to conceptual design. These constraints must take into account the variety of emerging technologies and future developing solutions and technologies.

For example Gal (et al.) [19] analysed characteristics of 9 well-know and used 4-seater metallic and composite aircraft. As it found, the structural mass / total mass ratio equals to 0.29 – 0.32, while the empty mass / total mass fraction is between 0.55 – 0.65. As it was demonstrated and used, the structural mass of electric / hybrid aircraft should be less than 25 % . Sub-chapter developing a cargo UAV used structural mass fraction defined as less than 20 % defined it.

The identification, analysis, selection and definition of the recommended extra constraints and the semi-theoretical mass, energy fraction models require further large investigations. For example, this paper and the IDEA-E team have defined the empty structure fraction mass less than 20 % as input for a cargo UAV design. The project results have demonstrated, this input can be accepted and realized. However, this is not an ab ovo understandable input. It is very depend on future technology developments and applying the radically new structural solutions.

There are many books, methodologies (as it was cited two paragraphs earlier) and large number of publications [7, 19, 23, 24, 28, 30 – 34, 50, 65, 67, 82 – 91] describing the applicable equations and formulas, including integrated and optimization methods [92 - 94], design supporting software and manuals [95, 96] and a large series of theses [31, 65, 66, 68, 97 – 105].

In simplified case, the preliminary calculations deals with determination of the aerodynamics characteristics of the aircraft drawn in first layout and first estimation of the take-off weight.

According to the aerodynamics the most important characteristics should determine is the model of the drag coefficient. The general form of drag coefficient is

$$C_D = C_{D_0} + \frac{C_L^2}{\pi e AR} \ . \qquad (10)$$

Here, the Oswald efficiency factor, e, defines the span efficiency, namely change in drag with lift as ratio of changes of real three-dimensional wing to a wing with same aspect ratio and ideal elliptical lift distribution. Its typical value is between 0.7 – 0.85 for conventional wing with retracted flaps.

The drag coefficient at zero lift, C_{D_0}, as parasite drag might be determined by using the semi-theoretical methods developed by aerodynamics, or by CFD methods. Introducing the CFD methods to preliminary calculation and determining the zero lift drag of the aircraft with unconventional forms and revolutionary new flow control, etc. might be rather costly. Roskam [21 (Part I.)] provide an well applicable statistical regression model, f, depending on the take-off mass and related to the wetted area,

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$$C_{D_0} = \frac{f}{S_w} = \frac{f_0}{S_w} 10^{a+b\left(c+d\log_{10}(2.2M_{TO})\right)} \ , \tag{11}$$

where f_0 = 0,3048, and the a, b, c, d coefficients can be identified from the table given by Roskam [21 (Part I.)] depending on the skin-friction coefficient

$$C_{D_f} = \frac{0.148}{Re_{CR}^{0.2}} \ .$$

Here the Reynolds number is determined for the preliminary defined cruise speed and reference chord, C_{Ref}.

This method seems very good, while the coefficients must be recalculated depending on the real unconventional forms, implemented flow, boundary layer control, effects of distributed propulsion system, and so on. Authors of this paper recommend to initiate a large international project for revising the mass and energy fraction formulas, developing new constraints with taking into account the rapid technology developments).

The lift coefficient, C_L, can be calculated easy for level flight (altitude and speed are constant) from weigh – lift equality:

$$C_L = \frac{2W}{\rho V^2 S} \ . \tag{12}$$

For preliminary design of aircraft, the take-off situation is very important. The take-off speed (according to the airworthiness – safety – requirements)

$$V_{TO} = 1.2\, V_{Stall} \ , \tag{13}$$

is associated with the stalling (minimum) speed (at level flight of aircraft in take-off condition – flaps open for take-off defining the $C_{L_{max}}$)

$$V_{Stall} = \sqrt{\frac{2W_{TO}}{C_{L_{max}}\rho S}} = \sqrt{\frac{2\frac{W_{TO}}{S}}{C_{L_{max}}\rho}} \ . \tag{14}$$

The ratio W_{TO}/S is called as wing load and it is one of the most important characteristics of the aircraft in preliminary design.

Flaps (or generally the wing mechanizing devices, methods) generates extra lift and unfortunately extra drag, too. Therefore, the stalling speed is calculated for maximum lift generation, when the flaps are deflected for maximum degree for the given flight mode.

About 92 – 95 % of lift origins from wings. Therefore, the aerodynamic design process focuses on the wing. Expect the large effects of wing geometry; the major action is selecting the wing profile, namely airfoil for wing. There are many airfoils are developed for different flight condition, for law, medium and high speed. Special airfoil books, websites [106, 107] provide the aerodynamic characteristics for airfoils. The normal airfoils have maximum lift coefficients equal to 1.2 – 1.9, while the different and extensively used flaps may generate 0.2 – 0.4 during take-off and 0.4 – 1.8 extra (adding) lift coefficient in landing modes [21 (part. I.)]. The larger values are generated on business and transport jets. From the conceptual design point, the maximum lift might be determined from the stalling or take-off velocities (14), if the wing load is given as input.

The investigation of the take-off flight mode may serve useful data, too. it interconnects the take-off distance, applicable take-off velocity, required take-off power (see Appendix III.)-.

During take-off and landing the drag and the lift coefficients are increased by opening the landing gears and the flaps. For example, the drag coefficient for take-off (in simplified case) equals to

$$C_{D_{TO}} = C_{D_0} + \Delta C_{Gear} + C_{D_{Flaps}} + \frac{C_{L_{TO}}^2}{\pi e_{TO} AR} \ . \tag{15}$$

The extra drags can be evaluated by using practical (ΔC_{Gear}) and semi-practical ($C_{D_{Flaps}}$) formulas, too, that are associated with extra lift calculations.

Another important element of the preliminary calculations is the first approximation of the take-off mass. The equation (3) or even better to say, the equation (4) supports the application of the new conceptual design approach, because the mass fractions might be preliminary defined as inputs depending on the ideas on the original solutions unconventional forms, expected changes in layout.

Appendix III. Required maximum power and mission energy (aircraft sizing)

This Appendix shortly describes the applicable "conventional" methodology for determining the required maximum power and mission energy. So, this is a short explanation to Figure 3. that was created to show the process of the conceptual design of the electric / hybrid aircraft. Creation of new design philosophy and methodology shown in this figure was initiated by the ideas and figures appeared in references [29, 89, 92, 101].

Appendix II. had introduced the references on methods of conceptual design, aerodynamic calculation and determining the flight performance, already. The required inputs are listed at the end of sub-chapter 2.1.

Because the limited space, here are only two formulas are proved for calculation the take-off required power and the cruise phase energy. The other formulas are defined, only (here or by the Figure 3.).

Required maximum power

There are five categories of aircraft depending on the size of the installed power:

- general aviation small aircraft, UAVs : P < 1 MW;
- commuters, helicopters large UAVs: P < 2 - 3 MW;
- business jets; turboprop regional airliners (50-100 PAX): P < 5 MW;
- turbofan regional airliners (100-150 PAX): P < 5 MW;
- airliners (200-300 PAX): P > 10 MW.

The power split, PS, and hybridisation factor, HF (see Appendix I.), identify the conventional engines – electric engines „cooperation". There are 6 classes:

- simple conventional propulsion system $PS = 0$;
- multi-source non propulsive energy; $PS < 0.05$; it is used for reducing the fuel consumption or increasing the overall efficiency (like on taxiing, or take-off);
- electrically assisted conventional engines, $PS < 0.10$; possible increasing the efficiency on whole mission, supporting the conventional engines by electric engines;
- parallel hybrid or partially electric propulsion; $PS = 0.20 - 0.50$; conventional, thermal engines and the electric motors are working parallel, that as usually reduces the overall emissions, but the weight and cost might be increased;
- series hybrid (or turboelectric) systems; $PS = 1$, $HF < 1$; conventional (thermal) engines provide energy, only for electric motors and energy storage;
- full electric aircraft; $PS = 1$ and $HF = 1$; when full electric mode fed by batteries.

The required power might be determined easy as $P = TV$. The velocity, V, as usually is well-definable, while estimation of the thrust, T, equals to drag, D, is not so simple.

The required power is a net or using power. The required engine power is

$$P_r = PT/\eta_{pst}, \tag{16}$$

where η_{pst} is the propulsion system total efficiency. One of the important and serious problem of sizing the electric / hybrid aircraft caused by unknown specific fuel consumption (SFC) or calculating the total efficiency of the system. Especially, the efficiency of the different devices as propellers, or electric motors, are investigated well and their efficiency are well studied and described. However, integration of these elements (conventional engines and electro motors, accumulators, propellers, etc.) caused problems of calculation their total efficiency. The total efficiency depends on the hybridization factor, power split and supply power ration, too.

$$\eta_{pst} = \prod_{i=1}^{n} \eta_i \, (HF, PS, SPR) \tag{17}$$

This formula follows the ideas improved by references [102, 108].

There are several models providing the solutions from the simplified to rather complex. For example, take-off power of the turboprop engines can be simple identified by using the approximation formula

$$P_{CR} = P_{TO} a M_{CR}^{m} \left(\frac{\rho(H_{CR})}{\rho_0} \right)^{n} \tag{18}$$

introduced by Nita [102] and well tested by Lenssen [104]. The values for approximation constants *A*, *m* and *n* are respectively 1.089, 0.091 and 0.92. (See P_{CR} estimation later.)

By using a more complex approach, the take-off is investigated by using the take-off distance estimation. It is completed from two, ground and airborne parts (up to fly over the 10.7 altitude) The trust, drag and friction drag define the ground acceleration

$$M\dot{V} = T - D - \mu(W - L), \tag{19}$$

where all the parameters (M, V, T, D, L) are related to the take-off flight modes (flaps are in take-off positions and landing gears are opened).

After some manipulation [81]

$$\dot{V} = \frac{dV}{dt} = \frac{dV}{dt}\frac{ds}{ds} = \frac{ds}{dt}\frac{dV}{ds} = V\frac{dV}{ds},$$

$$mV\frac{dV}{ds} = T - D - \mu(W - L),$$

$$ds = \frac{mVdV}{T - D - \mu(W - L)};$$

and integration

$$\int_0^{s_g} ds = \int_0^{V_{LOF}} \frac{mVdV}{T - D - \mu(W - L)},$$

$$(s_g - 0) = \frac{m_a}{T_a - D_a - \mu_a(W_a - L_a)} \frac{V_{LOF}^2}{2},$$

the take-off ground distance can be determined as

$$s_g = \frac{V_{LOF}^2}{2g\left(\frac{T_a}{W_a} - \frac{D_a}{W_a} - \mu_a(1 - \frac{L_a}{W_a}) \right)}. \tag{20}$$

Here, index *a* depicts the average for ground running.

In simple case (when the weight of aircraft may change only for a small value), the

$$L_a \cong \frac{1}{3} W_a$$

and the aircraft take-off ground distance equals to

$$s_g = \frac{V_{LOF}^2}{2g\left(\frac{T_a}{W_a} - \frac{D_a}{3L_a} - \mu_a(1 - \frac{L_a}{3L_a}) \right)} = \frac{V_{LOF}^2}{2g\left(\frac{T_a}{W_a} - \frac{D_a}{3L_a} - \frac{2}{3}\mu_a \right)},$$

$$s_g = \frac{V_{LOF}^2}{2g\left(\overline{T}_a - \frac{1}{3k_a} - \frac{2}{3}\mu_a\right)}. \tag{21}$$

This formula provides possible estimation of the required thrust by assuming that, the lift-off speed equals to the take-off speed (13) (actually it is smaller, but should be minimum 10 % greater than minimum unstick speed), taking aerodynamic goodness factor from aerodynamic calculation and choosing the friction coefficient from the operational concept (type of runways for which the aircraft will be designed).

The changes in total (sum of kinetic and potential) energy can be approximated by $\Delta e \cong T_{aa} s_a$

$$T_{aa} s_a = m\frac{V_{TOF}^2}{2} + mgh_{TOF} - m\frac{V_{LOF}^2}{2},$$

because the average thrust direction (in airborne flight) has a small deviation only from direction of runway:

$$s_a = \frac{m\frac{\left(V_{TOF}^2 - V_{LOF}^2\right)}{2} + mgh_{TOF}}{T_{aa}}. \tag{22}$$

The aircraft take-off distance is a sum of ground (21) and airborne (22) distances

$$s = s_{TOF} = \frac{V_{LOF}^2}{2g\left(\overline{T}_a - \frac{1}{3k} - \frac{2}{3}\mu_a\right)} + \frac{m\frac{\left(V_{TOF}^2 - V_{LOF}^2\right)}{2} + mgh_{TOF}}{T_{aa}}. \tag{23}$$

Because, still using the (23) seems rather complicated, several assumptions can be recommended. The required thrust and power might be determined for the ground-troll and airborne parts of the take-off separately and the greater must be accepted. Another assumption is $s = 1{,}66 s_g$, therefore the predefined (input) value of take-off distance, s, might use in formula (21) instead of s_g.

Another approach provides numerical integration of motion of aircraft along the runway by using Taylor series [79]:

$$V(t + \Delta t) = V(t) + a(t)\Delta t\ , \tag{24}$$

$$s(t + \Delta t) = s(t) + V(t)\Delta t + a(t)\frac{(\Delta t)^2}{2}\ . \tag{25}$$

In this approach, the engine thrust can be approximated by formula:

$$T = a + bV + cV^2\ , \tag{26}$$

The take-off ground-roll distance approximation formula may contain the thrust weight ratio and wing loading [109],

$$s_g \propto \frac{(W_{TO}/S)}{\sigma^{1.8}(T/W_{TO})C_{L_{\max TO}}}\ , \tag{27}$$

that can be approximated by simple formulas (for the same category aircraft having nearly the same take-off maximum lift coefficient) :

$$s_g = a + b\frac{(W_{TO}/S)}{\sigma^{1.8}(T/W_{TO})}\ . \tag{28}$$

This type of constraint equation might define in more general form proposed by [101, 110]:

$$\left(T - (D + D_e)\right)V = W\frac{d}{dt}\left(h + \frac{V^2}{2g}\right)\ , \tag{29}$$

defining the equality of the specific access power (left side) with changes in total (potential and motion) energy.

$$\frac{T}{W_{TO}} = \frac{\beta}{\alpha}\left\{\frac{qS}{\beta W_{TO}}\left[C_{D_0} + K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right) + K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right)^2 + \frac{D_e}{qS}\right] + \frac{d}{dt}\left(h + \frac{V^2}{2g}\right)\right\} . \tag{30}$$

Here the power (or thrust) lapse ratio, α, is a ratio of power (or thrust) in the given flight mode, flight condition and the reference power (or thrust), namely power in the designed point, design flight mode; β is the weight fraction (identifies the ratio of weight in given flight mode comparing to the take-off weight) and n is the load factor. Of course here all the parameters as, α, β, n, q, are related to the take-off regime. The drag coefficient is approximated by Taylor series:

$$C_D = C_{D_0} + K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right) + K_2\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right)^2 .$$

This equation as master equation allows to generate a series of the constraint equations [29, 101, 110]. For example, equation of the ground roll motion (19) can be rewritten by applying the assumptions $(n = 1, = 0,\ V_{TO} = k_{TO}V_{stall},\ T = const >> D + R)$ into simple form:

$$\frac{T_{TO}}{W_{TO}} = \frac{\beta}{\alpha g}\frac{dV}{dt} . \tag{31}$$

From here

$$ds = \frac{\beta}{\alpha g}\frac{W_{TO}}{T_{TO}} V dV,$$

$$s_g = \frac{\beta}{\alpha}\frac{W_{TO}}{T_{TO}}\frac{V_{TO}^2}{2g} \tag{32}$$

by taking into account the

$$V_{TO} = k_{TO}V_{stall} = k_{TO}\sqrt{\frac{2W_{TO}}{C_{L_{max}}\rho S}}$$

the trust / weigh ratio equals to

$$\frac{T_{TO}}{W_{TO}} = \frac{\beta^2}{\alpha g}\frac{k_{TO}^2}{s_g C_{L_{max}}\rho}\frac{W_{TO}}{S} , \tag{33}$$

therefore the power / weight ratio is

$$\frac{P_{TO}}{W_{TO}} = \frac{\beta^2}{\alpha g \eta_{pst}}\frac{\sqrt{2}\,k_{TO}^3}{s_g}\left(\frac{W_{TO}}{C_{L_{max}}\rho S}\right)^{\frac{3}{2}} . \tag{34}$$

The other power / weight formulas might be proved by applying the analogical approaches. the basic formulas appear in figure 3.

Traditionally the energy consumptions as fuel consumption for the mission legs (5) are determined from the changes in weigh during the flight (flight mission legs):

$$\frac{\Delta W_{ffm}}{W_{TO}} = \sum_{i=1}^{n}\frac{\Delta W_{ffml_i}}{W_{TO}} . \tag{35}$$

Each kg of aviation fuel (diesel or kerosene) as consumable energy source support the aircraft by about 3 kWh energy. This is the fuel specific energy used by aircraft, SE_{fu}, therefor for

$$e_{efm} = \Delta W_{f_{fm}} SE_{fu} = \frac{\Delta W_{ffm}}{W_{TO}} = W_{TO}\sum_{i=1}^{n}\frac{\Delta W_{ffml_i}}{W_{TO}} SE_{fu} . \tag{36}$$

There are many references provide functions for determining the weight reduction depending on the flight performance. For example, in case of cruise flight (constant level, constant speed, i.e. steady level flight),

$$T = D, \quad L = W, \quad T = D\frac{L}{L} = \frac{W}{\frac{L}{D}} = \frac{W}{k} \tag{37}$$

the change in weight (during the cruise flight) equals to

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$$\frac{dW}{\mathrm{dt}} = -TSFC\ T = -TSFC\ \frac{W}{k}\ . \tag{38}$$

By separating and integrating the equation

$$\mathrm{dt} = -\frac{k}{TSFC}\frac{dW}{W}\ . \tag{39}$$

the cruise flight might be determined as

$$t_{CR} = -\frac{k}{TSFC}\ln\frac{W_{CR_{initial}}}{W_{CR_{final}}}\ . \tag{40}$$

The required fuel (mass) for the level flight equals to

$$W_{f_{CR}} = TSFC\ T_{CR}\ t_{CR} = TSFC\ T_{CR}\frac{R_{CR}}{V_{CR}}\quad \left(or\ W_{f_{CR}} = kT_{CR}\ln\frac{W_{CR_{initial}}}{W_{CR_{final}}}\right). \tag{41}$$

The required power for level flight, of course, can be determined by using a simple relationship between the energy and power:

$$e_{CR} = P_{CR}t_{CR}\ . \tag{42}$$

The required energy can be defined by relatively simple way,

$$e_{CR} = P_{CR}t_{CR} = T_{CR}V_{CR}\frac{R_{CR}}{V_{CR}} = T_{CR}R_{CR}\ . \tag{43}$$

or the required power can be determined from modified equation (30):

$$\frac{P_{ref}}{W_{TO}} = \frac{\beta}{\alpha\eta_{pst}}\left\{\frac{qS}{\beta W_{TO}}\left[C_{D_0} + K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right) + K_1\left(\frac{n\beta}{q}\frac{W_{TO}}{S}\right)^2 + \frac{De}{qS}\right] + \frac{d}{dt}\left(h + \frac{V^2}{2g}\right)\right\}V\ . \tag{44}$$

All the flight mission legs might be studied and described by using the analogical methods.

In case of developing the electric aircraft, using these methods for calculation of the power and energy fractions is rather simple. Major different is that the mass of aircraft will not change during the flight.

In case of designing the aircraft with hybrid (or multiply power sourced) propulsion systems, the power and energy fractions must be distributed between the power and energy sourced by using the power split (PS) and hybridization factor (HF). (Appendix V. is an example, how to use the hybridization factor)

Appendix IV. Mass and energy balance equations

This appendix explains the goals of using the mass and energy balance equations.

The mass balance introduced by equation (4) define the sum of mass fractions:

$$M_{TO} = M_{TO}\sum_i \bar{M}_i\ . \tag{45}$$

This equation does deal with the traditional mass and balance or determination of the aircraft centre of gravity. (This last better to call as weight and balance or weigh balance.)

The mass fractions depends on the design and production performances. For example, in case of cruise flight, (level flight with constant altitude and speed), the relative mass might be defined from the flight condition (see (37)):

$$T_{cr} = \frac{W_{CR}}{k_{CR}}\ , \tag{46}$$

while the mass of engines might be defined as

$$M_{en} = \rho_{en}T_{ref} = \rho_{en}\frac{T_{CR}}{k_{en_{CR}}}\ , \tag{47}$$

where ρ_{en} is the engine (mass /thrust) density, and $k_{en_{CR}}$ is a coefficient defining the reduction in thrust depending on the altitude and engine rpm. Therefore the engine mass fraction is

$$\bar{M}_{en} = \frac{M_{en}}{M_{TO}} = \frac{1}{M_{TO}}\frac{k_{W_{CR}}}{k_{en_{CR}}}\rho_{en}\frac{M_{TO}}{k_{CR}} = \frac{k_{W_{CR}}}{k_{en_{CR}}}\rho_{en}\frac{1}{k_{CR}} \;, \tag{48}$$

where $k_{W_{CR}}$ is a coefficient defining the reducing the take-off mass until beginning the given flight mode, here cruise flight.

Equation (48) shows that the engine mass fraction depend on the engine thrust density characterizing the engine and goodness factor depending on aircraft development, technological and production levels. Development and technological levels mean levels of applied latest results of sciences and technologies. For example, better engine efficiency and better aircraft goodness factor (L/D ratio) can be reached by using best CFD methods in designs in shaping, better materials allowing to increase the operational temperatures in engines, reducing the weight, implementing the direct flow control, etc. The production levels depend on the production technologies, production management, production culture (including theoretical and practical competences of the employees, and production companies) and even on the social habits economic conditions (as wage rates or taxation systems). So, in general form, mass fractions are

$$\bar{M}_i = f\begin{pmatrix}\text{applied designed methods, characteristics of the materials,}\\ \text{technology, production culture, etc.}\end{pmatrix} \;, \tag{49}$$

This is the reason, why the mass balance equation is strongly recommended to use in aircraft conceptual design, especially in case of developing revolutionary new aircraft and accelerated developments in sciences and technologies.

As it was stated by sub-paragraph 3.1., if the $\sum_i \bar{M}_i > 1$, than the aircraft with predefined performance can not be built in the given production environment (in given country) with planned performances. The $\sum_i \bar{M}_i < 1$ condition shows that even better aircraft with better performance might be built. Finally, the $\sum_i \bar{M}_i = 1$ means the best aircraft is existing.

It is easy to understand, with analogy to mass balance equation other balance equations as energy balances (see (5) or cost balances, etc. might be defined, too. For instant, the cost can be defined for total life cycle phases [111] as it was applied by [38, 39]

$$C_t = C_{cde} + C_{cd} + C_{eng} + C_{pr} + C_{op} + \cdots = C_t \sum_i \bar{C}_i \;, \tag{50}$$

or for the DOC (direct operational cost), only.

Implementing these balance equations into the conceptual design supports the following major objectives

- better understanding and more accurate preliminary definition of the mass and flight mission leg energy fractions,
- possible introducing and definition a series of new constraints (for mass, flight mission leg energy or cost fractions,
- definition of the new constraints with using the foresight (for example visions on future batteries technologies),
- catalization of developing the new solutions required for realization of the new, possible radically new solutions, technologies.

This last means, for example, if the $\sum_i \bar{e}_i > 1$ for $2-5$ % only, than the ground energy support the aircraft take-off might be a good solution (see Fig. 10).

Appendix V. Range equation.

This Appendix introduces the range equation adapted to calculation of hybrid aircraft range. This prove of range equation rather analogic to methods published by other references and it follows the approach described in [33, 65].

Breguet developed a well applicable method for determining the range of conventional (consumable energy support) aircraft from the weight loss that this experiences due to the fuel consumption:

$$R = \int_{W_{final}}^{W_{start}} \frac{V}{g \cdot PSFC}\frac{C_L}{C_D}\frac{1}{W}dW \;. \tag{51}$$

The integration of this equation depends on the flight mission, or flying strategies (like on gradual steady climb, applied cruise condition, descent strategy, etc.). Because it is based on weight reduction, it is cannot applied directly to the hybrid – electric aircraft, storing partly the energy in batteries, which weight is not changing during the flight.

Well-known general formula,

$$R = \int_{t_{start}}^{t_{final}} V dt \ , \tag{52}$$

starts deduction of the Breguet equation adapted to aircraft with hybrid propulsion system. The elementary time might be specified from the total energy reduction

$$\frac{de}{dt} = -\frac{de_{fuel}}{dt} - \frac{de_{bat}}{dt} \ , \tag{53}$$

$$R = \int_{e_{final}}^{e_{start}} \frac{V}{\frac{de_{fuel}}{dt} + \frac{de_{bat}}{dt}} de \ . \tag{54}$$

The conventional fuel (consumable) energy support provides energy

$$\frac{de_{fuel}}{dt} = -\frac{PSFC}{\eta_{pstce}} \frac{H_{fuel}}{g} (1 - PS) P_r \ , \tag{55}$$

while the batteries afford

$$\frac{de_{bat}}{dt} = -\frac{PS \, P_r}{\eta_{pste}} \ , \tag{56}$$

where PS is the power spilt (8), and the $\eta_{pst_{ce}}, \eta_{pst_e}$ are the total efficiency coefficients of the conventional and electric propulsion sub-systems.

The required power can be determined from assumption, the flying in quasi-steady, quasi-rectilinear modes, when like in cruise flight, thrust must equal drag and lift equals weight (37) and

$$P_r = TV = \frac{W}{k} V \ . \tag{57}$$

By substitution of (55), (56) and (57) into the (54)

$$R = \int_{e_{final}}^{e_{start}} \frac{\eta_{pst_{ce}} \eta_{pst_e}}{\left(PSFC \frac{H_{fuel}}{g} (1-S) \eta_{pst_e} + S \eta_{pst_{ce}} \right)} k \frac{1}{W} de \ , \tag{58}$$

and assuming the power split, aerodynamic goodness factor of aircraft ($k = C_L / C_D$), the propulsion coefficients and engine $PSFC$ are constant in cruise flight than

$$R = \frac{\eta_{pst_{ce}} \eta_{pst_e}}{\left(PSFC \frac{H_{fuel}}{g} (1-S) \eta_{pst_e} + S \eta_{pst_{ce}} \right)} k \int_{e_{final}}^{e_{start}} \frac{de}{W} \ . \tag{59}$$

Simplified mass (1) or instead of it the weight breakdown, weight balance equation

$$W = W_e + W_{pl} + W_f + W_b \tag{60}$$

allows to solve the integral. Calculating the weight of fuel and batteries as function of energies stored by them

$$W_{bat} + W_{fuel} = \left(\frac{e_{bat}}{H_{bat}} + \frac{e_{fuel}}{H_{fuel}} \right) g = \frac{e_{bat} H_{fuel} + e_{fuel} H_{bat}}{H_{bat} H_{fuel}} \tag{61}$$

and using the hybridisation factor, ψ,

$$W_{bat} + W_{fuel} = \frac{\psi H_{fuel} + (1 - \psi) H_{bat}}{H_{bat} H_{fuel}} g E_{total} \tag{62}$$

the range of aircraft supported by hybrid propulsion systems can be determined as

$$R_{hybrid} = \frac{\eta_{pst_{ce}} \eta_{pst_e}}{\left(PSFC \frac{H_{fuel}}{g} (1-S) \eta_{pst_e} + S \eta_{pst_{ce}} \right)} k \frac{H_{bat} H_{fuel}}{g (\psi H_{fuel} + (1-\psi) H_{bat})} \ln \left(\frac{\frac{\psi H_{fuel} + (1-\psi) H_{bat}}{H_{bat} H_{fuel}} g E_{total} + W_{empty} + W_{payload}}{W_{empty} + W_{payload}} \right). \tag{63}$$

This equations can be used for hybrid aircraft as Breguet range equation applying to conventional aircraft.

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Accronyms and Abbrevation

ATM – air traffic management
CO_2 – carbon dioxide
GT – gas turbine
EM – electric motor
eVTOL – electric vertical take-off and landing aircraft
ICE- internal combustion engine
IDEA-E – a project "Investigation and development of the disruptive technologies for e-mobility and their integration into the engineering education"
IATA – International Air Transport Association
NASA – National Aeronautics and Space Administration
NO_x – nitrogen oxides
TP – turboprop engine
PATS – personal air transportation system
PAX – passengers
pkm – passenger km
PS – propulsion system
PUKA - Platform for unmanned cargo aircraft
rpm – revolution per minutes
UAV – unmanned aerial vehicle

List of symbols

a, b, c, d, … m, n , … A, … - coefficients
AR – wing aspect ratio
C – non-dimensional aerodynamic coefficients,
C, C-rate – Coulomb's rate
D – drag (aerodynamic force)
e - Oswald (span) efficiency (its typical value equals to 0.7 – 0.85 for conventional aircraft), applied to definition the induced drag: $C_L^2/(e\pi AR)$
e – energy – [J], [Ws], [Wh] or [kWh]
f – regression model
g – gravitational acceleration – [m/s^2]
h – height – [m]
$\dot{h}$, $\dot{h}_z$, - rate of climb – [m/s]
H – altitude (determined from the sea level) – [m] - (height related to the sea level)
H – specific energy (heating values at consumable fuels or energy stored by batteries) [J/kg]
HF, ψ - hybridization factor ($\psi = \frac{E_{bat}}{E_{total}}$)
k – coefficient, aerodynamic goodness factor (k = L/D),
K_1, K_2 – drag polar coefficient for 1st and 2nd order term (using the Taylor series of approximation)
$k_{en_{CR}}$ – coefficient of reduction in thrust $k_{en_{CR}} = f(H, \bar{n})$
L – lift (aerodynamic force)
M – Mach number (depicts the speed)

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M – mass – [kg]

$\bar{M}_i$ – mass fraction (relative mass) - $\bar{M}_i = M_i/M_{TO}$

n – number (like n_{se} – number of seats)

n – load factor

n – engine rpm

$\bar{n}$ – engine relative rpm

P - power – [J/s] or [W] or [KW])

PS – power split – ($PS = \frac{P_{em}}{P_{total}}$)

$PSFC$ – power specific fuel consumption of the conventional heat engines – [g/kWh] or [g/kWs]

q – dynamic pressure - [N/m²]

R – range – [km]

s – distance - [m]

S - wing area – [m²]

SE – specific energy (specific power or power-to-weight ratio or power to volume ratio) – [J/s/kg], [W/kg], [Wh/kg]

SED – specific energy density [J/s/dm³], [J/s/l], [W/dm³], or [W/l]

SPR – supply power ratio ($SPR = \frac{\int P_{em}}{\int P_{total}}$)

T – thrust – [N] or often used as [kN]

$TSFC$ – trust specific fuel consumption (reciprocal of the specific impulse) of the conventional heat engines – [g/kN/s]

t or τ – time – [h] or [s]

V – speed (scalar) or velocity (vector quantity) – [m/s] or [km/h]

W – weight [N]

w_L - wing load [N/m²] - $w_L = W_{TO}/S$

α – power lapse ratio (actual power in the given flight mode related to the sea level power (prototype power – starting power at the decign cycle, reference and design power of engines)

β – weight function (ration of weigh in given flight mode comparing to the take-off weight

η – efficiency coefficient

μ – friction coefficient

ρ – density, air density - [kg/m3]

ρ_{en} - is the engine (mass /thrust) density – [kg/kN]

σ – ambient air density ratio ($\rho/\rho_0 = \rho/\rho_{SL}$)

ψ – HF, hybridization factor

Ω – turning (angular) speed – [1/s]

Subscripts and Superscripts

0 - initial (C_{D_0} – drag coefficient at zero lift)

a – average,

a – available (power or thrust generated by engines)

a – airframe

a – airborne (part of take-off)

aa – average in airborne

AL – approach and landing

b, bat - battery

C – climb

CR – cruise

cd – conceptual design,

cde – concept development

ce - conventional engine

cps – conventional propulsion system (uses fuel (consumable) energy source

D – drag (aerodynamic force)

de – designed

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DE - descent,
DD – door-to-door
e – equivalent,
e, elec - electric
e, empty – empty
e – extra (like extra drag)
efm – energy used during flight mission
en – engine
eng – engineering
f , fuel – fuel
f - friction
final – final condition
Flaps – flaps, wing mechanization
fm – flight mission (total flight mission)
fml – flight mission leg
fu – fuel specific energy used by aircraft
g - ground
Gear – landing gear
initial – initial condition
L – lift (aerodynamic force)
LO – loiter
LOF – lift-off
max - maximum
op – operation (generally it is includes the flight, ground operation, maintenance and repairs)
opt – optimum
pl, payload – payload, commercial load
pr - production
ps – propulsion system (including the conventional engines, propellers, fuel systems, electric motors, electric devices)
pst – propulsion system total
r – required (like required, directly used power for the given flight mode)
ref - reference
sy – systems (aircraft systems except the propulsion system)
se – seats
SL – sea level
start – start condition
t, total – total
TO , TOF - take-off
turn - turning
TA – taxiing
TO – take-off
w – wetted (area)
x,y,z – coordinates of the reference system

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