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Conceptual Redesign of a 90-Seater Battery-Electric Aircraft

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Recent research suggests that large battery-electric aircraft can achieve greater ranges than previously assumed and can therefore be a promising solution to decarbonize the aviation sector on short distances. However, several technical challenges need to be investigated for such aircraft to become technically and commercially viable. This paper summarizes the findings of a series of research projects investigating these technical challenges, which are then incorporated in the conceptual redesign of a 90-seater battery-electric aircraft. The top-level aircraft requirements are revisited and the main configuration trade-offs are discussed. The resulting design presents a maximum take-off mass of 82.5 t and achieves a battery-only range of 750 km for a pack energy density of 320 Wh/kg, while reserves are covered by a dedicated fuel-based reserve energy system (RES). The impact of using the RES for range extension is also investigated. A comparison to conventional aircraft configurations in terms of CO_2-equivalent and operating costs per passenger-kilometer demonstrates that such aircraft can become a cost-competitive solution to mitigate the climate impact of aviation on short ranges.

Nomenclature

AEO	=	All engines operative
ATC	=	Air traffic control
CASK	=	Cost per available seat-kilometer
CFD	=	Computational fluid dynamics
CS	=	Certification Specification
DC	=	Direct current
EASA	=	European Union Aviation Safety Agency
EM	=	Electric motor
eSAF	=	Synthetic (power-to-liquid) SAF
EU	=	European Union
EV	=	(Ground-based) electric vehicle
eVTOL	=	Electric vertical take-off and landing
FAR	=	Federal Aviation Regulations
EU	=	European Union
GT	=	Gas turbine
ICAO	=	International civil aviation organization
MTOM	=	Maximum take-off mass
OEI	=	One engine inoperative
RES	=	Reserve energy system
SAF	=	Sustainable aviation fuel
TMS	=	Thermal management system
TO	=	Take-off
A	=	Wing aspect ratio [-]
b	=	Wing span [m]
C_D	=	Drag coefficient, $D/(0.5\rho V^2 S)$ [-]
C_{D_0}	=	Zero-lift drag coefficient [-]
$C_{L,\max}$	=	Maximum lift coefficient [-]
M	=	Mach number [-]
P	=	Power [W]
R	=	Range [km]
S	=	Wing planform area [m^2]
s_{L}	=	Landing distance [m]
s_{TO}	=	Take-off distance [m]
t_{root}	=	Root thickness
V	=	Flight speed [m/s], volume [m^3]
V_{app}	=	Approach speed [m/s]
W_{TO}	=	Maximum take-off weight [N]
ρ	=	Ambient air density [kg/m^3]

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I. Introduction

Various pathways are being explored to reduce aviation's climate impact, such as the use of Sustainable Aviation Fuels (SAF), hydrogen, and batteries for hybrid or fully-electric propulsion [1]. In the process, it is often suggested that electric aircraft are only viable for short ranges (below 400 km) and small payloads, for both current and future battery technology (200–500 Wh/kg) [2–6]. As a result, developments in the field of electric aviation focus on smaller aircraft, while larger electric aircraft are generally dismissed as infeasible alternatives [5, 7]. However, two recent papers by the authors argue that, with the right design approach, battery-electric aircraft in the CS-25/FAR Part 25 category are technically feasible and can achieve ranges exceeding 1000 km for battery pack energy densities in the range of 200–500 Wh/kg [8, 9]. This can be achieved through four design principles: 1) exploiting the inherently low empty-mass fraction of battery-electric aircraft, 2) taking advantage of the high lift-to-drag ratio associated with low wetted-area-to-reference-area ratios, 3) using a fuel-based energy system for reserves, and 4) realizing that there are no predominantly negative scaling effects of aircraft size on maximum range [8]. In this research, ten technical challenges were identified at subsystem level which could significantly limit the (range) capabilities of such aircraft to the extent that the aircraft becomes commercially unattractive.

Research into such technical challenges supports a broader trend towards increased aircraft electrification. Although the aircraft described in this paper is labeled as "battery-electric" because the nominal mission is performed on battery energy only, the use of a fuel-based reserve energy system makes it, technically, a hybrid. Hybrid/electric large aircraft have been actively studied for over a decade (see e.g. Refs. [10, 11]), with different forms of "hybridization" being investigated (see e.g. Ref. [12]). In simplified terms, we can distinguish two extremes of propulsion system "hybridization". In *mild hybrids*, the main purpose of the batteries is to support the gas turbines in off-design conditions and provide electrical energy during specific (short) mission segments, thereby reducing the overall fuel consumption—though fuel remains the primary source of energy. For example, the electric motors can be used for emission-free taxiing, to down-size the gas turbine by providing a power boost during take-off, or to help the gas turbine cope with transient power requirements. This typically results in parallel hybrid architectures with a low degree-of-hybridization of energy, in the order of 10%–20%, and a degree-of-hybridization of power ranging from 10% to 50% or more, depending on the application [13, 14]. Thus, a mild-hybrid configuration generally cannot perform the entire climb and cruise segment on batteries. These concepts require a change in the design of the powertrain but do not necessarily require major changes in the overall aircraft layout and can even be applied as a retrofit [15]. Studies suggest that the reduction in fuel burn or emissions due to "mild hybridization" can be up to 10% or 20%, or in the case of clean-sheet designs that also incorporate other technological improvements, up to 30% [16].

In contrast, so-called *heavy hybrid* aircraft (or "plug-in" hybrids [17]) use batteries as primary source of energy. In other words, the aircraft can perform a significant part of the missions it is designed for on only batteries, using fuel only for some longer-range missions. Such configurations typically present serial hybrid architectures with a high degree-of-hybridization both in terms of energy (>50%) and power (100%). In a limit case, the fuel-based system can be kept exclusively for reserves, enabling fully battery-powered nominal missions [9]. Due to the large mass of batteries required, these aircraft are significantly heavier than fuel-based aircraft and require a clean-sheet design to be effective. However, since they can perform missions on batteries alone, they enable reductions in fuel consumption, and thus in-flight emissions, of up to 100%. While this seems promising, recent transitions from fully-electric propulsion to milder forms of hybridization in (startup) companies in both the CS-25/Part 25[1] and eVTOL[2] sectors raises the question of whether fully-electric or "heavy hybrid" large passenger aircraft are commercially or environmentally viable.

The aim of this paper is therefore to perform the conceptual redesign of a battery-electric 90-seater aircraft and assess how it compares to fuel-based alternatives in terms of operating costs and environmental impact. The "E9X" configuration as shown in Fig. 1 and published in Ref. [9] is taken as starting point for the redesign. To this end, Sec. II first summarizes the findings of different research projects investigating the ten technical challenges identified in Ref. [9]. Sections III and IV then describe the updated top-level aircraft requirements and configuration choices, respectively, illustrating how the lessons learned drive particular choices, and the alternatives that can be considered. The resulting design is described in Sec. V, which presents the range achieved as a function of the assumed battery technology scenario and compares the operating costs and CO_2-equivalent emissions per seat-kilometer to fuel (SAF)-based alternatives. Note that the design presented in this paper does not constitute a final detailed design, but a second "snapshot" in the evolution of the design.

[1] See e.g. articles about Heart Aerospace on Aviation International News or Maeve Aerospace on Aviation Week (accessed 15 June 2025).
[2] See e.g. articles about Vertical Aerospace on Forbes or Archer Aviation on Flight Global (accessed 15 June 2025).

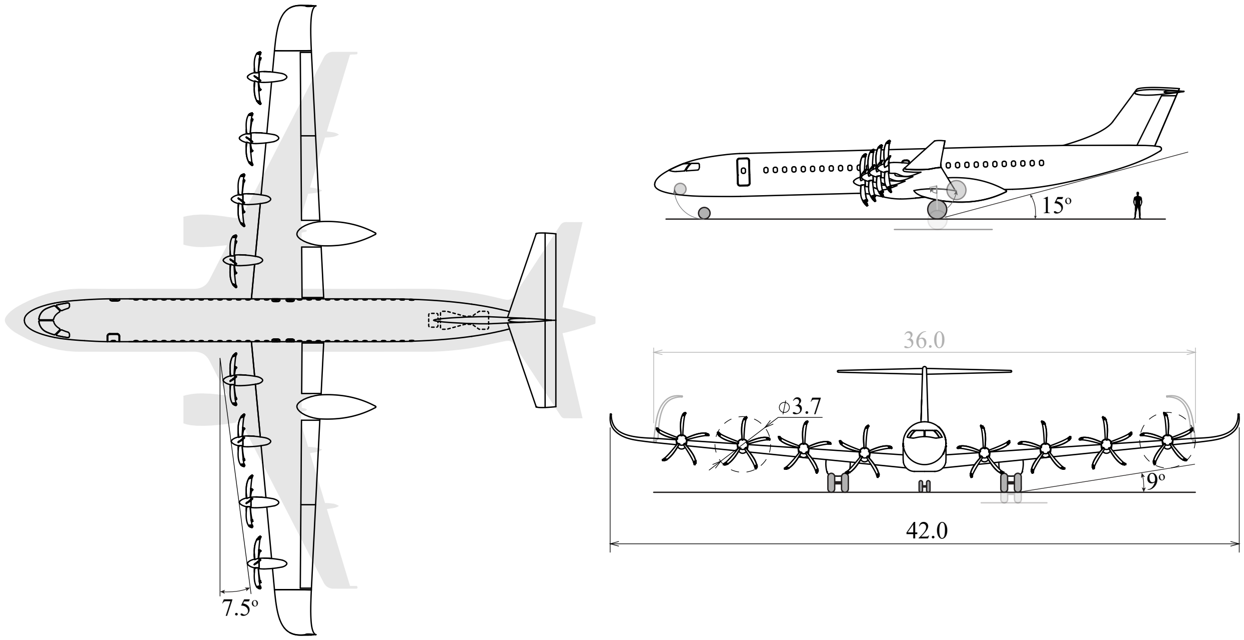


Fig. 1 Three-view of the original E9X design, as presented in Ref. [9]. Dimensions in meters. Top view includes A320 planform for scale in gray.

II. Summary of Technical Risk-Reduction Program

Reference [9] identifies ten technical challenges which may pose a risk for the feasibility of large battery-electric aircraft. These stem from the new technologies of the electrified propulsion system, and from the novelties or peculiarities of the associated aircraft configuration. Based on this list, Elysian started a two-year risk reduction program in collaboration with several key European research institutes to further investigate these so-called *Hot Potatoes*. The purpose of these research projects was not to create a detailed design of each subsystem, but to assess if there is *a* feasible solution, to identify potential showstoppers, and to reduce the uncertainty of the assumptions made at aircraft-design level. The main lessons learned in each of these projects can be summarized as follows:

1) *Wing/battery-pack integration:* in this project numerous ways of integrating and installing the battery inside the wing were explored. The main insights of this research were related to the volumetric requirements of the batteries, which were too rapidly discarded in the initial aircraft design phase since the wing presented multiple times more volume than the volume required for batteries. However, in practice, when the full cell-to-wing volume chain and safety requirements are considered, the ratio between cell volume and wing volume is found to be even lower, ranging from 5% to 20% depending on the design. This led to several changes in the design of the wing, most notably requiring a lower wing loading (see Sec. V).
2) *Wing structural sizing:* this study confirmed that significant wing weight savings are achieved by placing the batteries in the wing. The bending-moment alleviation provided by the batteries was found to decrease the wing structural mass by roughly 30%, compared to an empty wing with batteries in the fuselage (excluding additional snowball effects due to e.g. a larger and heavier fuselage required to house the batteries instead). First estimates of the wing mass penalty due to openings for access during battery replacement were made, showing no showstoppers but still presenting some uncertainty regarding the exact mass penalty. No unexpected limitations were found in terms of aeroelastic effects. Additional information is provided in Ref. [18].
3) *Design and certification of the Reserve Energy System (RES):* while the original design study assumed one gas turbine would be sufficient for safety since it is only needed in case of reserves—an already uncommon scenario—it rapidly became evident that a diversion to alternate is not an emergency and therefore the redundancy required by CS 25.1309 [19] still applies. The switch from one to two or more gas turbines drove changes in the placement and sizing conditions of the gas turbines. The main uncertainties regarding the RES were found to be mostly operational and certification-related (e.g. in-flight starting).

4) *High-voltage component and system design:* the increased weight estimated for the power distribution system in this project was found to compensate for the reduction in battery mass due to a higher-than-anticipated efficiency. This led to an overall system mass in line with the initial predictions. The avoidance of DC-DC converters and the development of high power-density circuit protective devices are found to be key drivers for low mass. Initial cable sizing studies are published in Ref. [20].
5) *Thermal-management system (TMS) sizing:* compared to the initial assumptions, a lighter TMS with lower power consumption was proven feasible. However, the initial assumptions regarding the drag of the heat exchangers and ram air ducts were found to be excessively optimistic. Some of the analyses are published in Ref. [21].
6) *Mass and energy consumption of non-propulsive systems:* based on early discussions with suppliers and experts, neither significant risks nor "double-digit" benefits in terms of partially or fully-electrified subsystems were identified, and therefore this project was de-prioritized. The most important systems in terms of mass and energy consumption, the environmental control and ice protection systems, were evaluated as part of the TMS project.
7) *Tail and control surface sizing with distributed propulsion:* an assessment of the longitudinal, directional, and lateral control and stability characteristics confirmed that the assumed 30% reduction in vertical-tail size by using differential thrust in OEI scenarios is feasible. However, in the end this benefit was not adopted due to the associated development and certification challenges, thus resulting in a larger vertical tail. The horizontal-tail size was found to be approximately correct. The ailerons were found to meet CS-25 performance requirements but lead to poor roll performance (partially due to the high roll inertia of the batteries in the wing), and thus larger ailerons are required for improved handling qualities.
8) *Propeller/wing aerodynamic performance:* The CFD analyses performed as a part of this study indicate the original aerodynamic-efficiency targets can be met, obtaining a cruise lift-to-drag ratio of 23 including propeller effects and heat exchangers. While this number is lower in practice once miscellaneous components such as leakage or excrescence drag is included, this reduction could be compensated by flying closer to max L/D. The modest $C_{L\max}$ target of 2.5 in landing was not possible with a plain flap system due to the negative impact of the nacelles and limited spanwise extension of the flap, and therefore a slotted Fowler flap was considered instead, comparable to current commercial transport aircraft. Additional information is presented in Ref. [22].
9) *Battery cell development:* cell requirements and technology roadmaps were investigated with suppliers and external experts. The targeted 450 Wh/kg cell energy density for an aircraft entering into service in the 2030s—which would require the cell technology to be mature prior to 2030—was found to be aligned with some cell supplier roadmaps, though given the significant uncertainty in such projections, it was decided to go for a more conservative assumption of 400 Wh/kg. This has a significant impact on the mass and/or range of the aircraft. For more information regarding the required cell specifications, the reader is referred to Ref. [23].
10) *Low-noise propeller design:* in this study different propeller designs were created to verify whether the aircraft concept could present acceptable noise levels while maintaining the propeller efficiency targets. Results suggest that the low disk loading and flexibility in rotational speed allow the aircraft to meet the ICAO Annex 16, Volume 1, Chapter 14 [24] noise levels and be significantly quieter than other aircraft of comparable mass, while meeting the initial assumptions of propeller efficiency. Although the cruise efficiency was found to be lower due to compressibility effects, the climb efficiency was found to be higher, overall leading to a net benefit. Several results of this assessment are presented in Ref. [25].

In general terms, no technical showstoppers were identified throughout the research program. The outcomes of these projects were incorporated step-by-step in the aircraft sizing loop, and the impact of the findings on MTOM is summarized qualitatively in Table 1. Overall, the increases and decreases in masses across the various subsystems and other smaller changes with respect to the original design were found to largely cancel each other out. The one exception to this was the reduction in assumed cell energy density, which evidently has a much more direct impact. This leaves the designer with three main choices, as sketched in Fig. 2: 1) reduce the payload, 2) reduce the range, and/or 3) maintain the payload/range requirements and accept an increase in MTOM. Reducing the cell energy density to 400 Wh/kg and incorporating the findings of the other *hot-potato* projects would cause the MTOM of the initial design increase to nearly 100 t, while maintaining MTOM would require a significant reduction in range or payload. Accordingly, the top-level aircraft requirements had to be re-evaluated iteratively and in the end a compromise of the three scenarios was selected, as described in the following sections.

Table 1 Qualitative outcome and relative MTOM impact of the *hot potato* projects. Legend: "+/-" = no significant impact; "+" = slight increase; "++" = significant increase; "-" = slight decrease.

#	Hot potato project	ΔMTOM
1	Wing/battery-pack integrated design, incl. battery replacement	+
2	Wing structural sizing for aeroelastic and landing loads with batteries in wing	+
3	Design and certification of the reserve-energy system	+
4	High-voltage component and system design	+/-
5	Thermal-management system sizing for powertrain components and batteries	-
6	Mass and energy consumption of non-propulsive systems	+/-
7	Tail- and control-surface sizing with distributed propulsion	+
8	Propeller/wing aerodynamic performance (cruise and high-lift)	-
9	Battery-cell development for large-aircraft applications	++
10	Low-noise propeller design for distributed propulsion	+/-

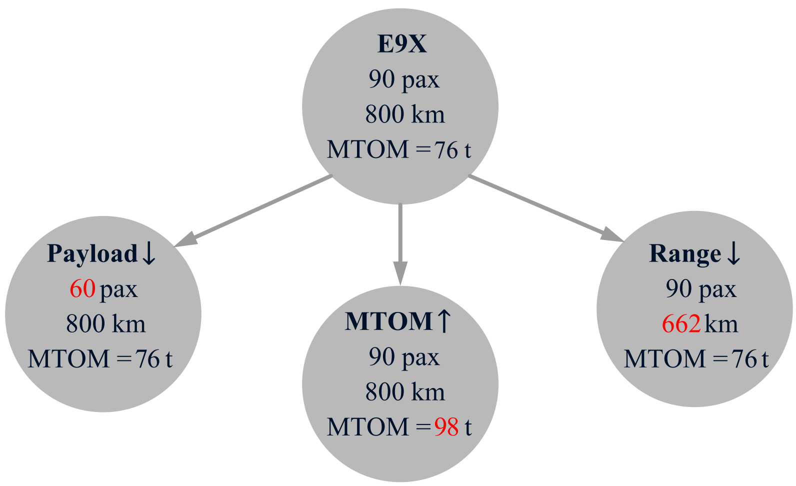


Fig. 2 Simplified schematic showing how the updated technology and subsystem assumptions, particularly on the battery, can be translated into a change in MTOM, payload, or range. Note this analysis was performed for an intermediate iteration of the E9X design without any major configuration changes applied yet.

III. Assessment of Top-Level Aircraft Requirements

In this section, a simplified market assessment is presented in Sec. III.A to investigate the market size for battery-electric aircraft, given that such aircraft present substantially lower range than fuel-based aircraft. Based on this, the design objective is formulated. The corresponding top-level aircraft performance, payload, and range requirements are subsequently presented in Secs. III.B and III.C.

A. Initial Market Assessment and Design Objective

Figure 3 gives an overview of the share of flights flown below 1000 kilometers versus the number of passengers carried over that distance, for several major international airlines. The figure shows that a majority of the flights operated by these airlines are shorter than 1000 kilometers. While 1000 km range is not considered feasible with current battery technology (see Sec. V), this is taken as target as battery technology progresses, since flights below 1000 km account for roughly 50% of all flights and nearly 20% of all CO_2 emissions of the passenger transport sector [26]. Some airlines such as ANA and Japan airlines operate more than 80% of their flights in the sub-1000-kilometer range. Furthermore, each pie chart in the graph shows the share of flight operated by turboprops, regional jets, narrowbodies or widebodies. From these pie charts it becomes evident that the majority of flights are performed with narrowbodies, followed by regional jets. Therefore, for an aircraft to compete on these distances and address a significant part of this market, it has to be cost-competitive compared to narrowbodies and regional jets. For many of the larger flag carriers, the short-range flights with regional jets often serve feeder routes to an international hub. Therefore, if the electric aircraft can break even on costs with regional jets, it can replace such aircraft on feeder routes. If the operating costs can be further reduced to the level of narrowbodies, then it can also open new routes which are too thin for narrowbodies to operate on and not profitable for regional jets. In that case, a new way of operation which connects secondary airports in a “web”-like network with stopovers as proposed in Ref. [9] could further enhance the market potential.

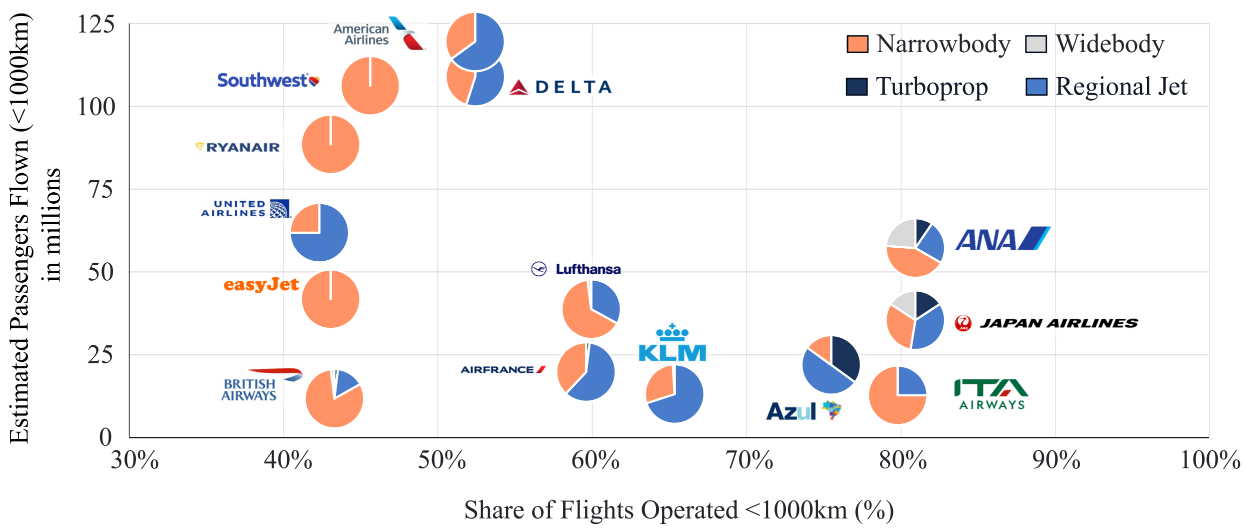


Fig. 3 Share of flights and absolute number of passengers flown below 1000 kilometers for several major airlines. Results based on Elysian internal analyses.

Based on this simplified market assessment, the design objective for the redesign of the E9X is stated as follows:

- The objective is to compete with the latest generation of narrowbodies in terms of Cost per Available Seat Kilometer (CASK), with zero in-flight emissions, and with a range as high as possible.
- The design must be forward-compatible with future battery cells, to increase range as battery technology progresses.
- The design must facilitate operations from secondary regional airports.

B. Payload-range requirements

The previous design objective favors a high passenger count and maximum range. However, due to the high mass of the aircraft and the infrastructure requirements discussed in Sec. III.C, the payload and range requirements were not fixed upfront, but evaluated iteratively as part of the sizing process. Based on the lessons learned in the first design iteration [9], the payload target was set between 84 and 100 passengers, assuming 100 kg mass per passenger on average, including luggage, of which 16 kg is check-in luggage. Furthermore, to allow for potential future growth in passenger count as other aspects of the aircraft evolve, the fuselage is sized for a max payload of 9600 kg and seating up to 100 pax, though payload is limited to 8800 kg at MTOM for the aircraft presented in Sec. V.

Moreover, a usable range between 500 and 1000 km was set as target, depending on the battery technology scenario (see Sec. V). Since the reserve energy system can act as a range extender to provide additional flexibility, the maximum fuel tank volume was set to twice the volume required to cover reserves. For reserves, a diversion to alternate of 200 km is considered, with 30 minutes loiter and a 5% additional contingency energy, as required by EASA Rules for Air Operations [27]. The resulting payload-range diagram is presented in Fig. 4 for a harmonic range of 750 km. These values are summarized in Table 2.

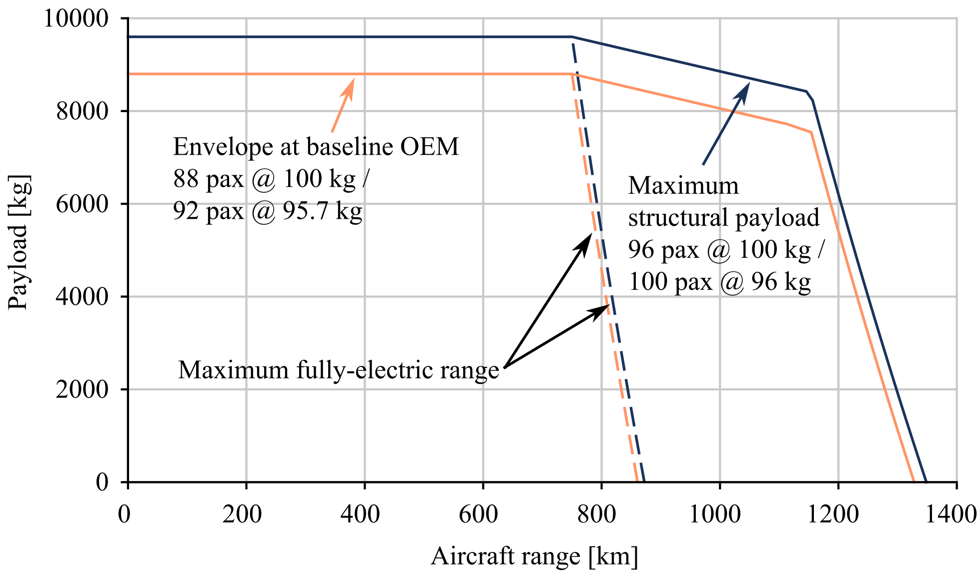


Fig. 4 Payload-range diagram of the aircraft. Note that the maximum structural payload can only be achieved without exceeding MTOM if the operating empty mass or battery mass is reduced.

Table 2 Selected top-level aircraft requirements.

Parameter	Requirement	Target
Number of passengers [-]	·	84 – 100
Mass per passenger [kg] (assumed)	100	·
Full-electric range [km]	≥ 500	1000
Take-off field length [m]	≤ 2000	·
Landing distance [m]	≤ 2000	·
Approach speed [kts]	≤ 145 (Cat. D)	≤ 140 (Cat. C)
Cruise Mach number [-]	·	0.4 – 0.6
Cruise altitude [ft]	·	21000 – 30000
Sea level AEO climb gradient [-]	≥ 8%	·
Contingency energy [-]	5%	·
Diversion range [km]	200	·
Diversion Mach number [-]	0.40	·
AEO Diversion altitude [ft]	12000	·
OEI Diversion altitude [ft]	6500	·
Loiter time [mins]	30	·
Wing span at gate [m]	≤ 52	≤ 36

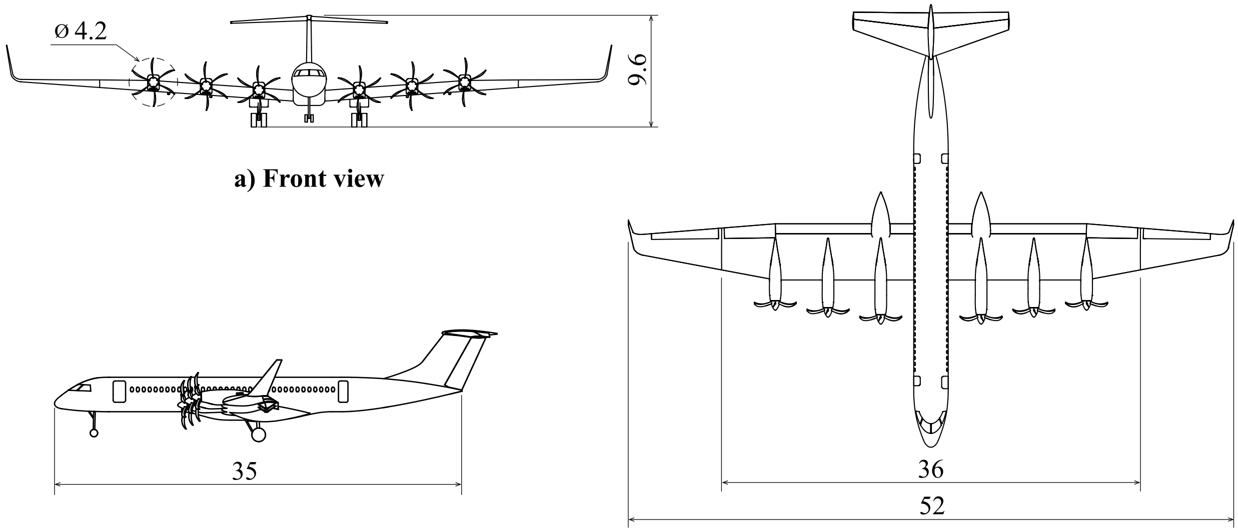
C. Performance requirements

The main aircraft performance requirements selected for the aircraft are given in Table 2. This table excludes performance requirements already specified by the EASA CS-25 regulations [19]. Take-off and landing distances of 2000 m are selected, such that the aircraft can operate on the same runways as most narrowbody aircraft. The approach speed category C (< 140 kts) is set as target, although exceeding this threshold is considered acceptable if needed. The cruise Mach number and cruise altitude are optimized to maximize lift-to-drag ratio. To ensure the aircraft can also climb sufficiently at sea level, an 8% climb gradient with all engines operative is imposed.

For the sizing of the reserve energy system, a diversion altitude and Mach number of 12000 ft and $M = 0.4$ are selected, respectively. In case of failure of gas turbine failure, a OEI[3] diversion altitude of 6500 ft is selected. Finally, a span constraint of 36 m is set as target to ensure the aircraft is compatible with ICAO Aerodrome Reference Code C, with 52 m (Aerodrome Reference Code D) being the extreme limit. The Aerodrome Reference Code affects the maximum span (particularly to fit at the gate), but also the width of the main landing gear track and compatbility e.g. with taxiway widths. This constraint is found to be particularly difficult to satisfy in combination with other requirements, driving the need for a more complex solution including a large folding wingtip (see Sec. V).

IV. Configuration Evaluation

The proposed aircraft configuration is shown in Fig. 5. In the following sections, main configuration choices and the rationale behind them are described. Different configuration options were narrowed down qualitatively before comparing the final options quantitatively. Several configuration choices made in the original design of Ref. [9] (e.g. T-tail configuration, four abreast, etc.) were not revisited, while many others were revisited but are not explicitly discussed here for brevity (e.g. number of gas turbines, type of high-lift devices, etc.). In some cases, uncertainty remains around the design choice until further information becomes available. Therefore, where applicable, the identified alternatives are listed.



a) Front view

b) Side view

c) Top view

Fig. 5 Three-view sketches of the updated E9X aircraft configuration. Dimensions in meters. Note wing fold location at 36 m span.

A. Fuselage Cross Section

Three alternatives were compared for the fuselage cross-section, for a same cabin diameter above the floor: circular (like the CRJ family), double-bubble with a flat bottom (like the ATR72), and double bubble with a larger cargo hold under the floor (like the E-jet family). The circular cross section was chosen since it is the lightest option and reduces wetted area compared to the larger double bubble.

→ *Alternative option(s):* If ditching requirements become challenging, the belly fairing creates excessive drag due to the thick wing extending below the bottom of the fuselage, and/or additional cargo volume is required, then the larger double-bubble may be considered.

[3] "One engine inoperative" is treated in this paper as the failure of a propulsor unit (electric motor, propeller, etc.) *or* the failure of a turbogenerator (gas turbine, generator, etc.). In other words, the aircraft must continue to fly regardless of where the failure occurs in the powertrain.

B. Number of propellers

The number of propellers affects the aircraft's control & stability characteristics, by how much each engine must be over-sized to satisfy OEI requirements, part count, nacelle drag, aerodynamic interference with the wing, and many other aspects. Decreasing the number of propellers, at constant diameter, also decreases total propeller disk area, a key parameter for propeller efficiency. Propeller diameter, however, cannot be increased indefinitely since it will affect the landing gear length, and could thus increase mass. Larger propellers also rotate slower, increasing motor mass in case of a direct-drive solution. When comparing 4, 6, and 8 propellers, fewer propellers were found to lead to a slight reduction in MTOM and operating costs, though the effect was relatively weak. However, from an electric motors perspective, more propellers are beneficial since the power per shaft reduces and thus the number of motors that are currently under development or available off-the-shelf with sufficient power rating increases. Likewise, from a power distribution perspective, an increased number of propellers may be beneficial for cooling and redundancy. Based on these considerations, six propellers were identified as the most promising compromise at this stage. Note that the thin blades shown in Fig. 5 are a result of the propeller being designed for low noise and low disk loading [25].
→ *Alternative option(s):* If electric motors and effective power distribution systems in the 2+ MW range "per lane" are developed, or if motors can be stacked to reach such power levels without a significant mass or complexity penalty, then going for a four-propeller configuration while maintaining constant disk loading can be beneficial in terms of maintenance costs and aerodynamic and mechanical integration of the nacelles and the systems inside.

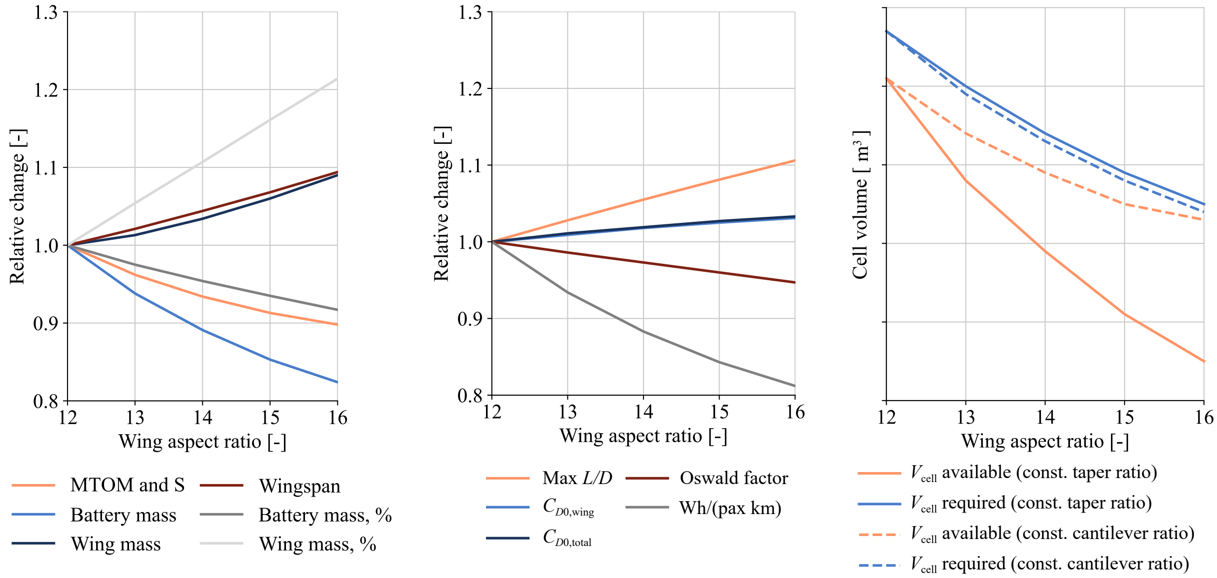
C. Reserve-Energy-System Placement

The position of the reserve-energy system (RES) affects both the mass of the high-voltage system (through cable length) and the center of gravity of the aircraft. Safety aspects such as rotor fragment release, fire protection, and crashworthiness also play a role. The placement options also depend on the number of turbogenerators. For this design, two turbogenerators driven by GE T08 turboshaft engines were assumed. Four positions were compared in addition to the initial position of the RES in the tail-cone: in nacelles ahead of or behind the wing, and in the belly fairing, ahead of or behind the center wing section. Placing the RES under the wing is challenging due to blade fragment release potentially penetrating the batteries. The RES was finally selected to be installed in the aft belly fairing, behind the center wing box, where a lot of volume is available due to the large wing relative to the fuselage. This allows for zonal segregation between high-voltage and fuel-based systems and is a low-drag solution with a good impact on center of gravity and inertias. An extended keel beam is placed between the two gas turbines to shield the two turbines from rotor fragment release. The main drawbacks of this configuration are related to fire protection, rotor fragment releases, and crashworthiness, which may require reinforcements or protection systems which come with an additional complexity and weight penalties.
→ *Alternative option(s):* If the integration of the RES in the belly fairing proves to be too challenging, the main alternative is to install the RES in the inboard motor nacelle, ahead of the wing. If the development of a multi-MW turbogenerator does not occur, then small, independent low- or medium-bypass ratio turbofans installed in nacelles behind the wing can be used as RES instead. In that case, the electrical power distribution system is simplified substantially because thrust is generated by the turbofans themselves, but the RES becomes less suitable for nominal range extension due to higher noise and reduced (propulsive) efficiency.

D. Wing Aspect Ratio

The wing aspect ratio has a pronounced influence on aerodynamic efficiency, and therefore on aircraft range, especially for electric aircraft. This is reflected in Fig. 6, which shows the sensitivity of aircraft size, mass, performance, and battery volume to aspect ratio for constant wing loading. Fig. 6a shows how, as the aspect ratio is increased from 12 to 16, the wing mass increases as expected, and the battery mass reduces due to the increase in aerodynamic efficiency (Fig. 6b). However, due to the large battery mass fraction of the aircraft, the reduction in battery mass far outweighs the increase in wing mass, and the MTOM reduces. Combined with the increased aerodynamic efficiency, this leads to a substantial reduction in energy consumption; in this case, a nearly 20% reduction for $A = 16$, compared to $A = 12$.

However, there are two main drawbacks which limit the achievable aspect ratio. First, whilst the battery mass, and thus required battery volume, reduces with aspect ratio, the volume available in the wing reduces even more rapidly, thus making it gradually more difficult to fit the batteries in the wing. This can be compensated by keeping a constant unswept cantilever ratio, $(b/2)/(t_{\text{root}})$, which is a key driver of both wing weight and wing internal volume. This is reflected in Fig. 6c, where a comparison at constant cantilever ratio, achieved by gradually reducing the taper ratio (i.e. less tip chord), shows how the required battery volume and the available battery volume converge at higher aspect ratios.



(a) Aircraft mass/size **(b) Aircraft performance** **(c) Battery volume**

Fig. 6 Sensitivity of top-level aircraft parameters to wing aspect ratio. Assessment performed at constant wing loading and taper ratio, except for the dashed lines, where a constant wing loading and cantilever ratio are maintained instead.

The second challenge is related to the wing span, which despite the reduction in MTOM, increases with aspect ratio. For the payload and range requirements listed in Sec. III.B and the battery energy density assumptions and volume requirements described in Sec. II, the wingspan was found to far exceed the 36 m span constraint of aerodrome reference code C gates, and even exceed the 52 m span constraint of D-gates at higher aspect ratios. Therefore, the wing presents winglets and a large folding wingtip with hinge at ±18 m span to fit in the C-gates, with an un-folded wingspan of 52 m. Note that this implies that the outboard folding wing tip is 8 m in span on each side, which is highly unconventional for a CS-25 aircraft and an additional development hurdle in itself—though large folding wings have been extensively used in military aircraft. Based on these trade-offs, a wing aspect ratio of 13 was selected. One could argue that, if the wing presents a fold mechanism anyway, it will be more effective to simply further enhance the span of the wing instead of using a winglet.
→ *Alternative option(s):* if the 52 m span is found to not be a limiting factor (e.g. if all variants of the aircraft incorporate a folding wing), then extending the wingspan and increasing aspect ratio is more beneficial than using winglets.

E. Dihedral and Vertical Propeller Position

In the configuration presented in Fig. 5, landing-gear length is mainly governed by the combination of wing dihedral and propeller vertical position, since the aircraft must maintain ground clearance in case of a banked landing. In this case, a roll clearance of 9° is maintained for positive pitch attitudes. A longer gear adds mass, hinders ground operations, and requires a larger fairing or cut-out to store the landing gear in retracted position.

Different dihedral angles were assessed from a flight-dynamics perspective and the required $C_{L,\max}$ performance was re-evaluated. The technical risk-reduction work performed in Ref. [22] highlights how a high nacelle position is detrimental for $C_{L,\max}$, triggering flow separation at relatively low angles of attack. However, since the updated battery volume constraints leads to a significantly lower wing loading, the required $C_{L,\max}$ is lower, and the nacelles were positioned above the wing.
→ *Alternative option(s):* If the installation of the nacelles above the wing is found to reduce maximum-lift capabilities more than anticipated, the propellers can be placed slightly below the wing chord, and the landing gear length must be increased.

V. Sizing Results

In this section, the configuration defined in Sec. IV is sized to determine the aircraft mass, dimensions and performance. The conceptual design method described in Ref. [28] is used for aircraft sizing. The method modifies the traditional preliminary sizing methods [29–32] to account for hybrid/electric powertrains and aero-propulsive interaction effects between the propellers and wing. The method is validated for Class-I sizing in Ref. [33], and is expanded to account for the Class-II weight breakdown of Torenbeek [29] and the drag component buildup of Raymer [31]. The weight estimates of conventional aircraft components are verified and calibrated against internal Fokker Aircraft data, and the operating costs model is verified and calibrated with airline/operator input. Tail size, drag polars, propeller efficiency, and powertrain technology parameters are calibrated with the results of the more extensive analyses performed in the risk-reduction research projects (see Sec. II).

A. Weight and Performance

The flight performance requirements which determine the power required from—and therefore the weight of—the propulsion system are shown in Figs. 7 and 8. The power-loading (or "matching") diagram of the electric motor (Fig. 7) illustrates how the 8% all-engines-operative climb gradient, which is not a hard certification requirement but a selected operational constraint, is the sizing case. This shows that, if we want to be able to climb at 8%, further increasing the take-off distance has no impact on the installed power. The graph also shows that, for the specific characteristics and assumptions of this aircraft, the OEI requirements listed in CS-25 are not limiting in the case of six engines. For this reason, reducing the number of engines from eight to six has no detrimental effect on the total required shaft power. The figure also shows how the selected "long range" cruise speed of $M = 0.45$ is far from being limiting, indicating that there is sufficient power to fly at higher speeds if desired for particular missions.

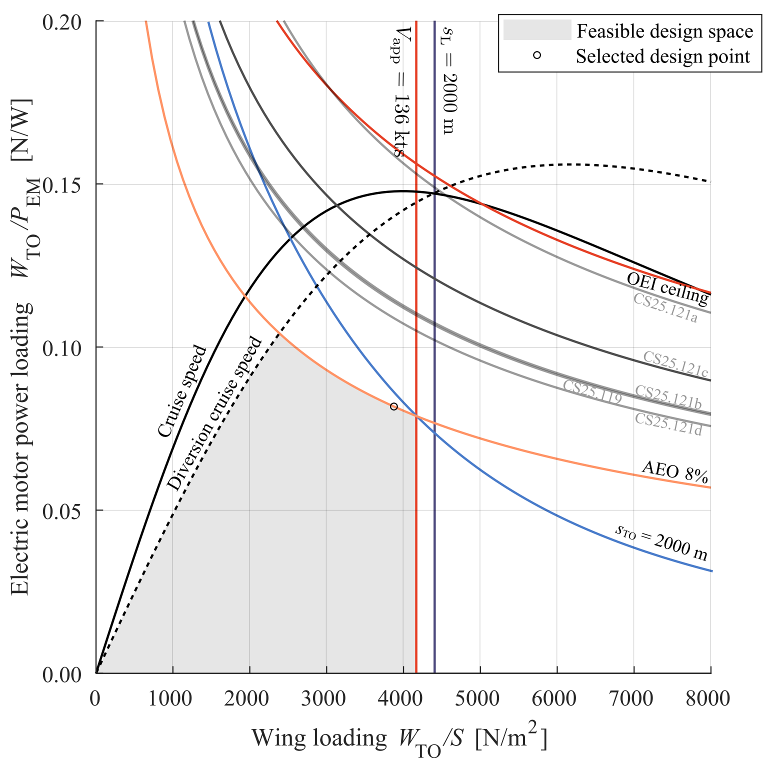


Fig. 7 Wing loading/power loading diagram of the electric motors.

Figure 7 also reflects how the landing distance and approach speed requirements can be met with some margin for the selected wing loading. The selected wing loading is lower than in the original design and slightly lower than modern high-speed turboprops such as the Dash 8. In this design, the wing loading is largely driven by the battery volume requirement (see Sec. II), with higher wing loadings making it difficult to acccommodate the battery volume. Note that for this design a lower wing loading is not detrimental, since it increases the allowable power loading and improves the $S_{\mathrm{wet}}/S_{\mathrm{ref}}$ ratio [8] (for a given aspect ratio). The main challenges of lower wing loading values are the associated high wingspan and lower optimum cruise speeds (for a given altitude), which in turn affects operating costs. Since there is some margin between the selected design point and the landing distance, approach speed, and take-off distance constraints, one could in principle allow a lower maximum lift coefficient. However, given the uncertainties with regards to high lift (see discussion on vertical propeller placement in Sec. IV.E), the margin is maintained.

The gas-turbine power loading diagram shown in Fig. 8 indicates how the selected one-engine-inoperative ceiling (interpreted here as "one gas turbine inoperative") is sizing for the gas turbines. The low ceiling in case of diversion with one gas turbine of the RES inoperative imposes operational restrictions (e.g. no landing at airports where an aborted landing and diversion to alternate is not possible without crossing a high mountain range). With both gas turbines operative, there is sufficient power to fly at higher altitudes and speeds, as reflected by the diversion cruise-speed constraint. For the OEI en-route climb constraint prescribed by CS 25.121c, the supplied power ratio of the battery is optimized such that the available gas turbine power is used without becoming a limiting factor.

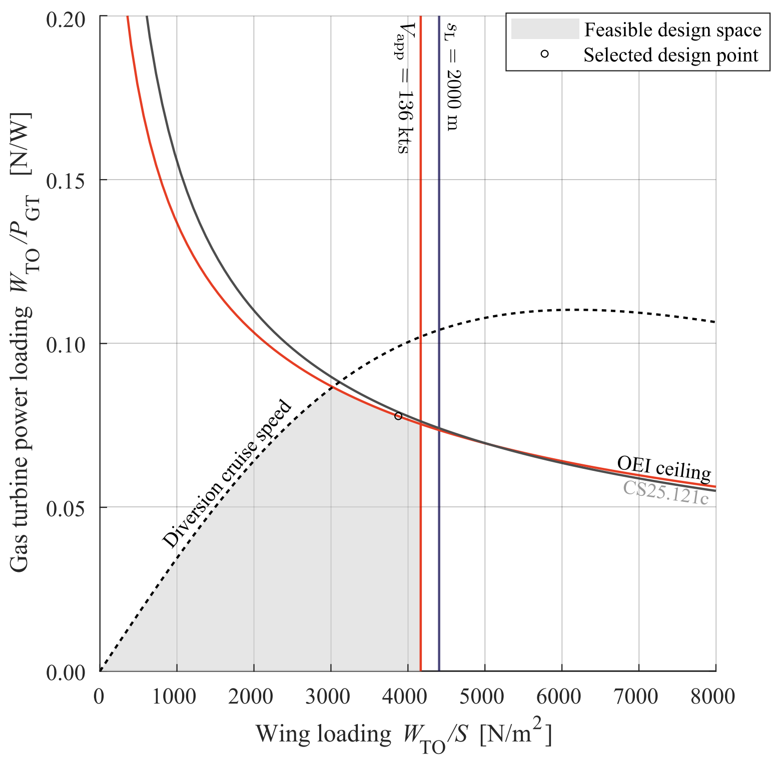


Fig. 8 Wing loading/power loading diagram of the gas turbines of the RES.

After selecting the design point, performing a mission analysis, and computing component masses, the mass breakdown of Fig. 9 is obtained. The MTOM of 82.5 t is similar to a Boeing 737 MAX 8. The aircraft is sized to reach 750 km with a battery pack energy density of 320 Wh/kg (see Sec. V.B). Compared to the original design shown in Fig. 1, the updated range and battery energy-density values lead to an increased MTOM (82.5 t versus 76.0 t), an increased empty mass fraction (45.5% versus 42.2%, if we include the fuel of the RES), and a reduced battery mass fraction (43.8% vs 46%). Overall, the scaling effects of large electric aircraft, characterized by a high battery mass fraction and a low empty mass fraction [8], remain evident in the mass breakdown.

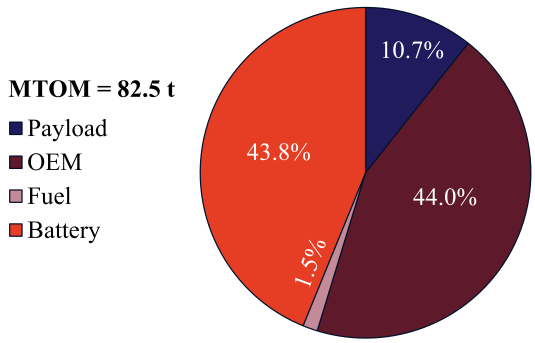


Fig. 9 Top-level mass breakdown of the aircraft.

B. Range Versus Battery Technology
Given that battery technology continues to evolve and that compatibility with future battery energy densities is part of the design objective (see Sec. III.A), Fig. 10 presents the aircraft range versus the assumed pack-level energy density. The aircraft range accounts for energy required for climb, descent, taxi, non-propulsive systems, etc., while the reserves are covered separately by the RES. Since there is an inherent uncertainty in the evolution of battery technology, three scenarios are defined.

The first scenario of 240 Wh/kg at pack level corresponds to high-performance batteries available today. With this, a usable range of approximately 540 km is achieved. For an aircraft entering into service in the 2030s, with the updated energy-density target of 400 Wh/kg at cell level (see Sec. II), a range of 750 km is obtained, assuming 25% pack mass ovehead. This corresponds to the maximum range that can be flown on batteries at max payload. However, as shown in Fig. 4, the fuel tanks are sized to be able to accommodate additional fuel to extend the mission range. The implications of this are described in the next sections.

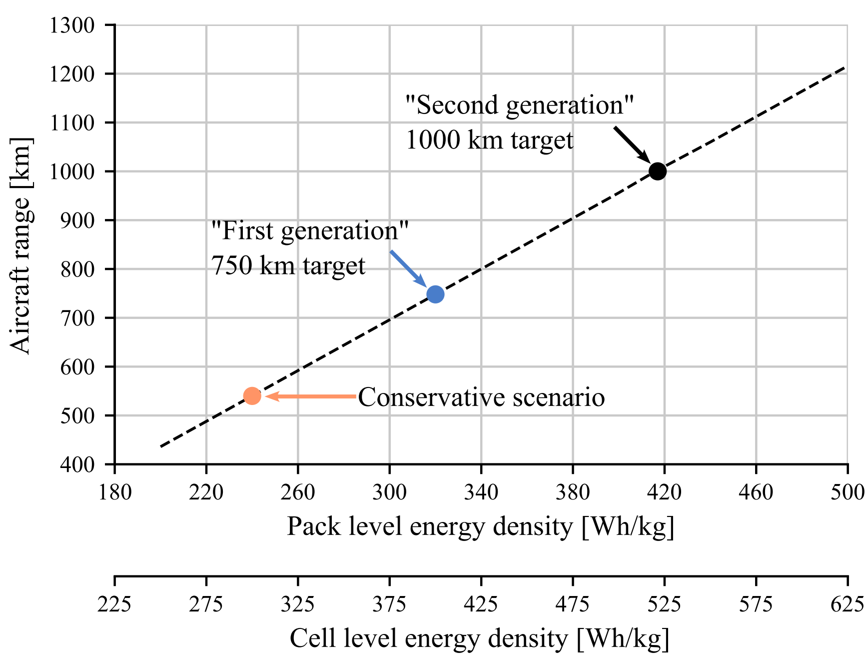


Fig. 10 Evolution of usable range as battery technology evolves. Required cell energy density calculated assuming a 25% mass overhead for packaging. Note the energy densities refer to 100% SOC at the start of life; the usable energy densities are lower.

Finally, the third scenario shown in Fig. 10 shows the energy density that would be required to cover 1000 km range on batteries. This corresponds to a pack-level energy density of roughly 420 Wh/kg which, to the knowledge of the authors, is beyond any near-term commercialization roadmap, but remains well within the 500 Wh/kg packs targeted in different research programs [23]. Note how, compared to the initial design presented in Ref. [9], the reduction in range (750 km versus 800 km) is due to the lower energy density assumed (320 Wh/kg pack versus 360 Wh/kg pack). In fact, for the same energy density, the updated design would present an increased range (850 km for 360 Wh/kg pack).

C. Operating Costs

In this section the operating costs per available seat-kilometer (CASK) of the aircraft are calculated to (1) understand how they compare to conventional aircraft and (2) investigate whether it is commercially beneficial to use the RES for range extension. The electric aircraft is compared to SAF-powered aircraft, which are currently considered the main pathway to "decarbonize" the aviation sector [16]. For SAF, a blend of biofuels and power-to-liquid synthetic fuels is assumed to be required to meet demand of the aviation sector as a whole. Hydrogen is omitted from this analysis for simplicity. We perform the assessment assuming a future scenario with widespread adoption of "zero-emission" technologies with mature supply chains. In other words, the effects of initial learning and low production scale are assumed to be over and the cost of SAF and batteries reach asymptotic values that are lower than their costs today. The reader is reminded that cost projections have an inherent uncertainty and that the analysis is performed here to show general trends, but the exact values will depend on how prices evolve.

The CASK calculations include fuel costs, electrical energy costs, battery replacement, maintenance, air traffic control (ATC) and landing charges, crew costs, asset costs, passenger costs, and commercial overhead costs. While not all numbers can be shared for confidentiality, the following key assumptions are made:

- ATC and landing charges are assumed to be equal for all aircraft on a per-passenger basis. Currently, these charges are MTOM-based in Europe, which penalizes battery-electric and other sustainable aircraft solutions which are heavier than aircraft today. Achieving this level playing field will require a change in policy.
- Maintenance costs of the electric components of the powertrain are assumed to be equal to 20% of gas turbine maintenance costs, due to the reduced number of moving parts, high-temperature parts, and fluids. Note that in this case, the maintenance costs of the propulsion system of the electric aircraft is still of the order of 40% of the total aircraft maintenance costs, if we neglect the gas turbines of the RES. If the RES is used, it is much higher.
- Electricity prices of 0.06 $/kWh (2025 US dollars). Current electricity prices are higher in regions such as Europe and a sensitivity study was performed to assess sensitivity to future price evolutions of electricity, but the following paragraphs focus on the assumed battery costs instead, since they have a bigger impact than the electricity costs.
- Battery cycle life is assumed to vary linearly between 1200 cycles and 3600 cycles at 100% and 0% depth of discharge, respectively, based on the ranges of values given in Ref. [34]. For the missions of interest, this leads to 1300–2100 cycles, depending on the payload and range combination. For the harmonic mission, it corresponds to approximately 1350 cycles.

The results of this analysis are shown in Fig. 11, which expresses the CASK as a difference with respect to a narrowbody such as the A320neo or B737 MAX. The analysis assumes that today's latest-generation narrowbodies become dominant on short-range markets in the 2035+ timeframe, given the current distribution (Fig. 3) and orderbooks, though a next-generation single aisle may be introduced by then. A new, high-passenger-count narrowbody (A321neo), and an older, low-passenger-count turboprop (Fokker 50) are selected as limit cases of the bandwidth of operating costs that can be expected from fuel-based aircraft. Two scenarios are assumed for both the fuel (SAF) prices and the battery prices. EASA estimates that current SAF prices range from approximately 3 to 10 times current Jet A1 prices depending on their nature (biofuel or synthetic) [35], while other sources suggest 3 to 5.5 times current jet fuel prices [36]. However, some sources expect SAF prices to drop to roughly 3 times the fuel prices [36]. Based on this, we define the two scenarios as: SAF at 1600 $/t (approximately 2 times current Jet A1 prices), and SAF at 3200 $/t (approximately 4 times current Jet A1 prices). A similar approach is taken for battery prices. Currently, prices for automotive batteries continue to decrease and are dropping below 100 $/kWh at pack level [37]. However, batteries for aerospace applications are substantially more expensive, potentially increased by a factor four [34] or more. Assuming wide-scale adoption of aerospace batteries for the 2035+ timeframe, the two scenarios are defined as 100 $/kWh and 250 $/kWh. This assumes that cell costs have dropped due to the economies of scale [38] and that a smart low-cost pack design is achieved.

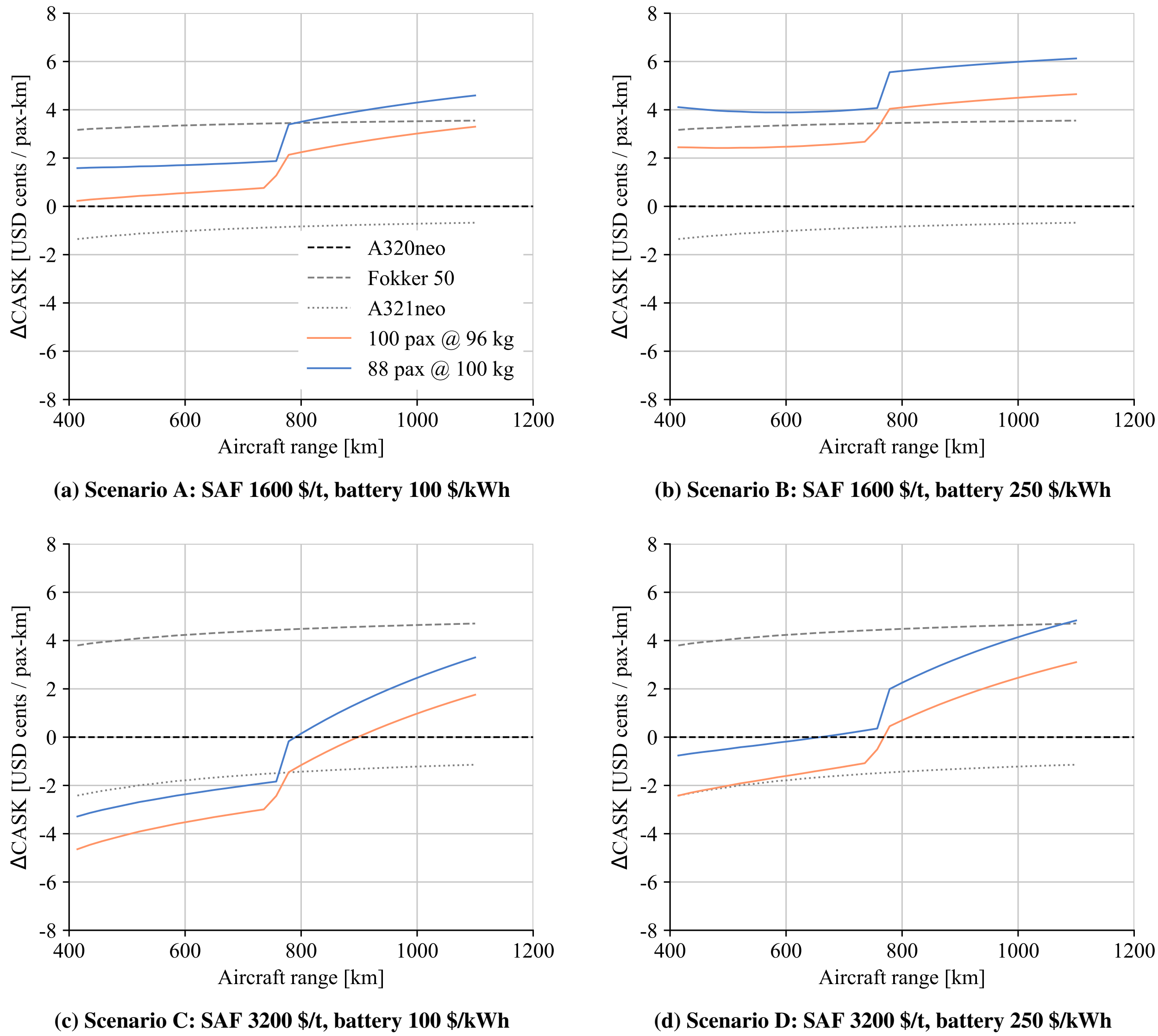


Fig. 11 Estimated change in cost per available seat-kilometer (CASK) of different aircraft configurations, relative to the A320neo. The discontinuity in the cost curves of the (hybrid) electric aircraft indicates the range beyond which the RES would have to be used for range extension.

The results of Fig. 11 show how the operating costs of the electric aircraft in full-electric mode (< 750 km) lie in the interval of operating costs of fuel-based aircraft, dropping below the costs of the A321neo in the case of 3200 $/t SAF and 100 $/kWh batteries. The electric aircraft becomes gradually more competitive as the range decreases, except in the case of high battery costs and low SAF costs. Beyond 750 km, fuel mass can be traded for payload mass and the range can be extended using the RES, as illustrated in Fig. 4. In this case, two things happen to the operating costs. First, there is a discontinuity since the gas turbine has to be switched on during the flight, and therefore maintenance costs increase. Second, the slope of costs versus range increases, due to the use of expensive fuels on an already heavy aircraft.

If the lines of Fig. 11 are extrapolated to the right-hand side of the graphs, we observe that the (hybrid) electric aircraft becomes more costly to operate than the most costly fuel-based aircraft for ranges above 1200 km, in all four scenarios. For this reason, the fuel tank capacity was limited to enable roughly 350 km of range extension. Based on this, we can draw two conclusions. First, the "heavy hybrid" aircraft operated on batteries only can be cost-competitive with large fuel-based passenger aircraft despite its lower passenger count. Second, using the RES as range extender enhances the payload-range envelope but is detrimental for operating costs. In other words, it can be used occasionally to enhance operational flexibility, but flying only long-range missions with the RES on is not competitive in most SAF/battery cost scenarios. This illustrates how hybrid aircraft can present “the worst of both worlds” if designed or operated suboptimally: in that case the aircraft has the increased mass of the electrified powertrain, but still has the high operating costs of a SAF-driven gas turbine.

D. Emissions

In this section we perform a simplified assessment of the climate impact of the aircraft in terms of CO_2-equivalent emissions. The purpose is to investigate the impact of using the RES for range extension and to place that in the context of other modes of transport. The comparison includes in-flight CO_2 and non-CO_2 emissions, emissions produced in the production of SAF or electricity, and in the production of batteries. For traceability, the same assumptions as in Ref. [9] are used. Note that a substantial uncertainty exists, particularly concerning the non-CO_2 effects, as indicated by the errorbars in Fig. 12. For a more comprehensive analysis including other life-cycle emissions such as airframe production or end-of-life disposal, see Ref. [39].

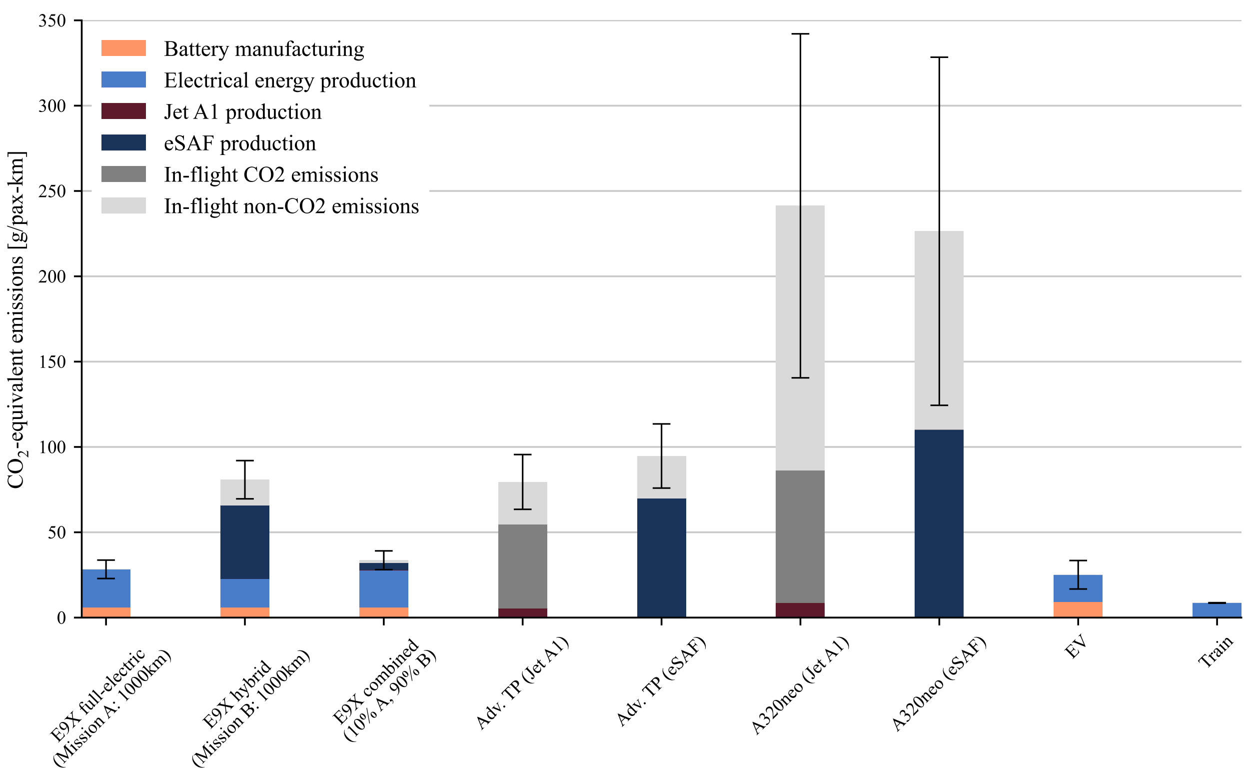


Fig. 12 Comparison of CO_2-equivalent emissions produced by different modes of transport, for a grid emission index of 114 gCO_2/kWh.

Figure 12 shows the CO_2-equivalent emissions of the electric aircraft (E9X), a hypothetical advanced fuel-based turboprop designed for similar payload–range capabilities (90 pax and 800 km; see Ref. [9]), a latest-generation narrowbody (A320neo), an electric car, and the train. The results are calculated for a “clean” EU 2030 target grid emission index of 114 gCO_2/kWh[4]. The emissions of the E9X are presented for three scenarios: for the design range of 750 km on batteries alone (“Mission A”), for a range of 1000 km using the RES as range extender (“Mission B”), and for a hypothetical fleet-level mix where the aircraft on average flies “Mission B” once in ten flights. The emissions of the advanced turboprop and the narrowbody are evaluated for an 800 km mission and are estimated using both Jet A1

[4]This corresponds to a grid which is roughly 75% renewable. For an indication of the sensitivity of CO_2-equivalent emissions to the assumed grid cleanliness level, see Ref. [9].

and eSAF as fuel. eSAF is taken in this comparison as scalable solution for the aviation sector. If biofuel is used instead, the "eSAF production" contribution to overall CO_2-equivalent emissions would likely be reduced compared to eSAF in this grid emissions index, though the non-CO_2 emissions remain. For all modes of transport, the average occupancy factor is assumed: 85% for aircraft, 1.2 passengers for the EV, and as provided by the railway operator for the train.

Comparing the three scenarios of the E9X, we observe that using the RES to extend the mission range from 750 km to 1000 km increases the emissions by a factor three. In that case, the emissions are comparable to an advanced turboprop designed for the same mission, though they remain well below those of a narrowbody operated on such short ranges. The figure also shows that if the RES is used as range extender only occasionally—in this case, one in ten flights—then the average climate impact remains comparable to fully electric operation and only slightly higher than the CO_2-equivalent emissions produced by traveling such distances with an electric car. In other words, analgously to the operating costs discussed in the previous section, the use of a heavy hybrid is highly beneficial from an environmental perspective as long as the majority of flights are performed on batteries alone. In all cases, CO_2-equivalent emissions of the train are significantly lower. However, this excludes the impact of infrastructure, which can skew this analysis in favor of the aircraft [39].

VI. Conclusions & Outlook

Previous research has indicated that large battery-electric aircraft can play a significant role in reducing the climate impact of aviation if they are able to cover ranges up to 1000 km in the long term. However, these aircraft must overcome several technical challenges, such as the integration of batteries or the thermal management of powertrain components. These technical challenges have been investigated in a two-year research program, providing insight into the various subsystems of such aircraft. The purpose of this paper is to summarize the findings of this research and perform the conceptual redesign of a battery-electric 90-seater aircraft.

Different configuration choices are described and the main alternatives are presented. The resulting aircraft is a so-called "heavy hybrid" configuration where the nominal mission is performed primarily on batteries alone, while a fuel-based Reserve Energy System (RES) is used to cover reserves. It presents a maximum take-off mass of 82.5 t, a wingspan of 52 m, six propellers, and 750 km full-electric range for a battery energy density of 320 Wh/kg at pack level. Two major configuration choices for which alternatives may be explored are the integration of the RES in the belly fairing and the use of large folding wingtips to fit in the 36 m span constraint. To assess the benefits or drawbacks of using the RES to enhance nominal mission range, the aircraft is compared to conventional fuel-based aircraft using Sustainable Aviation Fuel (SAF) in terms of operating costs and environmental impact. This simplified analysis shows how the aircraft can be competitive in terms of operating costs for different SAF and battery price scenarios, and have a substantially lower climate impact in terms of CO_2-equivalent emissions. Compared to a 750 km mission on batteries, if the RES is used for range extension up to 1000 km, the operating costs are found to increase significantly, and the CO_2-equivalent emissions are increased by a factor three. This shows that operating the aircraft only in "hybrid mode" is generally not beneficial from a cost or environmental perspective, but that occasional use of the RES (e.g. 1 in 10 flights) can provide operational flexibility without a major impact on the average costs or emissions across all missions.

The results of this study indicate that large battery-electric passenger aircraft, whether operated fully electrically or with occasional range extension, can provide substantial environmental benefits while breaking even on operating costs. The lessons learned and absence of showstoppers in the various research projects have led to a second conceptual design which serves as stepping stone in the evolution of the design of the aircraft. Further work into the development of such aircraft and the technologies that enable them are key steps towards decarbonizing air transport on short ranges, and with that, towards achieving climate neutrality of the aviation sector as a whole.

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