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Mission Energy-Based Modelling to Assess the Performance and Feasibility of Battery-Powered Aircraft Retrofit: A Case Study of the BN-2B-26 Islander in Scotland

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Abstract

This study develops a mission energy-based framework to assess the performance and technical feasibility of retrofitting an existing commuter aircraft with battery-electric propulsion. The Britten-Norman BN-2B-26 Islander is selected as the case study aircraft for selected domestic routes in Scotland. A MATLAB-based parametric assessment framework was developed to evaluate aircraft and the feasibility of selected routes, and it integrates key design parameters including power, energy, battery system, operations and payload. The results show that battery specific energy is a dominant technology parameter, while specific power, available installation volume and retained payload determine whether the resulting configuration is practically feasible. For a 50 km mission, increasing battery specific energy from 400 to 1000 Wh/kg reduces the required battery mass fraction by approximately 60%. The energy available is around 57.6% of the nominal energy after deducting battery operational, regulatory and fixed mission energy requirements. Matching the baseline aircraft capability of approximately 774 kg of payload over 998 km would require about 6050 Wh/kg and 2840 Wh/L of battery technology. Route assessment indicates that high payload cases are infeasible, while reduced payload cases can support wider short-route coverage under future battery assumptions. Overall, the results indicate that battery-electric retrofitting is most feasible for short-range commuter operations with reduced payload demand.

Keywords: battery electric aircraft; mission energy modelling; design space; payload–range analysis; route feasibility



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1. Introduction

Greenhouse gas emissions play a key role in climate change. Commercial aviation emitted approximately 918 million tons of CO₂ in 2018, representing about 2.4% of global CO₂ emissions [1]. The ICCT projected that global passenger traffic would grow at an average annual rate of 3% between 2019 and 2050 [2]. Scientific reviews describe aviation as a hard-to-abate transport sector because aircraft require lightweight propulsion systems and energy sources with high specific energy, power density, reliability and safety [3,4]. Electric and hybrid-electric propulsion can improve energy efficiency and eliminate direct in-flight CO₂ and NO_x emissions; however, their practical application is limited by battery

specific energy, battery system mass and volume, operational requirements, regulatory requirements and payload–range penalties [5,6].

There are several promising technologies that are evolving to solve these problems like electric-, hybrid-, hydrogen- and SAFs-powered aircraft. For example, the battery-electric powertrains can reach up to 90% energy efficiency, reduce direct operating cost and reduce up to 10 dB noise, which is around 50% lower compared with conventional aircraft [7,8]. The battery-electric powertrain components like motor, gearbox, controller and other electrical distribution systems have higher efficiency. Battery mass specific energy and battery volume specific energy are remaining the main technical limitation for battery-electric aircraft. Current state-of-the-art lithium-ion battery packs typically provides about 180–300 Wh/kg at the pack level [6], which is far lower than the energy density of aviation fuel. Jet fuel has a gravimetric energy density of approximately 43 MJ/kg, while advanced lithium-ion batteries are around 1 MJ/kg on a cell or pack basis [6]. Recent battery developments show progress, including pouch-type rechargeable lithium batteries with a reported cell-level specific energy of 711.3 Wh/kg and volumetric energy density of 1653.66 Wh/L [9]. Commercially available battery cells have also been reported by Amprius, with values up to 520 Wh/kg and 1150 Wh/L [10]. Beyond lithium-ion technology, lithium–sulphur batteries offer a high theoretical specific energy of approximately 2654 Wh/kg [11], although practical cell and pack level values are expected to be considerably lower. Future battery projections indicate that optimistic aircraft relevant battery specific energy may approach about 1000 Wh/kg by around 2050 [12]. Figure 1 shows the aircraft using batteries with the specific energy density and required density in the future to meet the NASA projection on battery chemistry.

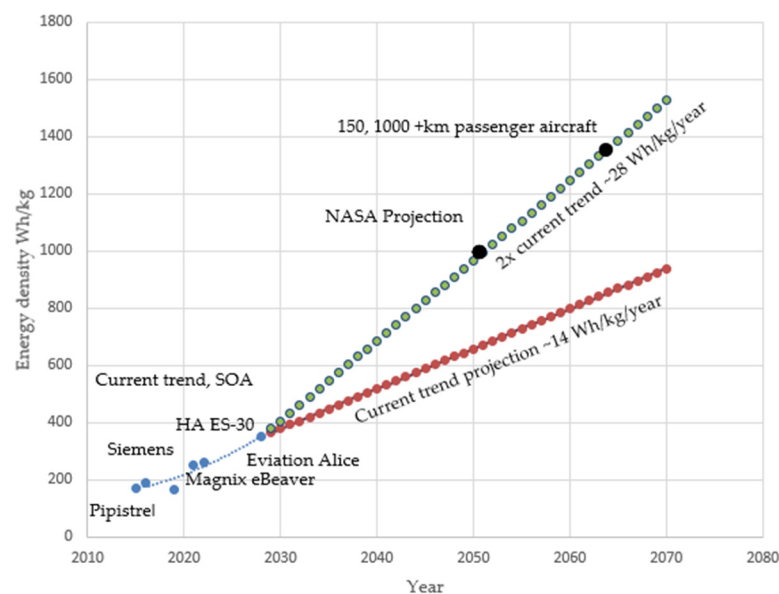


Figure 1. Battery specific energy SOA and future projection.

The energy density projection in Figure 1 shows that the current trend will not achieve the projected value by NASA, and battery chemistry may require a breakthrough to achieve such value in 2050. Clean-sheet battery-electric aircraft are underway. However, progress in key enabling technology metrics has yet to be improved. Novel aircraft design is costly and time-consuming. On the other hand, retrofitting with battery propulsion systems on an existing platform provides a pathway to enable battery-powered aircraft faster and with minimal certification and regulatory constraints compared with new designs. Currently, BN-2B-26 Islander is under operation in many domestic routes in Scotland, including in inter-island and mainland-to-island operations.

Previous studies have established that battery-electric regional aviation is strongly constrained by battery specific energy and battery mass fraction. Ludowicy et.al [13] showed that exact replacement of a larger regional aircraft mission, such as the ATR 42-500, would require a battery energy density (3450 Wh/kg) far beyond near-term technology; however, it does not include the battery installation volume constraint and payload replacement trade-off. Hepperle [14] highlighted that low battery volumetric energy density may become a significant design constraint when the available internal aircraft volume is limited, potentially requiring larger structural volumes or alternative battery integration arrangements. Anker et.al [15] analysed the 9-seat and 19-seat electric commuter aircraft for the Norwegian short-haul network. This study evaluated mass, power, energy and thermal management feasibility rather than physical battery packaging feasibility for electric aircraft. However, this study focused on existing electric aircraft, not retrofit. Bærheim et al. assumed that the existing fuel and fuel tank mass was replaced by batteries while maintaining the original aircraft mass; however, this study does not specify whether the batteries are installed in the wings, fuselage, cabin or cargo compartments. Mukhopadhyaya et al. [16] assessed evolutionary 9-, 19- and 90-passenger electric aircraft concepts under pack-level battery specific energies of 250 and 500 Wh/kg. Their findings identified short commuter missions as the most realistic near-term application, although detailed battery installation volume and systematic payload reduction effects were not assessed. Avanzini et al. [17] derived maximum range relationships for battery-powered aircraft, while Traub [18] investigated the influence of battery weight fraction on maximum range and endurance. Previous studies have shown that MTOW, battery mass and payload trade-offs are important constraints. Therefore, mission energy requirements alone are not sufficient for battery sizing, and the mass and physical integration limits of the baseline aircraft must also be considered [19–21]. Existing studies have investigated battery specific energy, battery mass fraction, maximum range and regional electric aircraft feasibility. However, fewer studies have simultaneously coupled mission energy demand, peak power requirements, retained payload, battery mass, physical installation volume and operational route requirements within the geometry and mass constraints of an existing commuter aircraft. This combined assessment is particularly important for retrofit applications because a configuration that satisfies the mission energy requirement may still be infeasible because of mass, power, payload or installation volume constraints. The present study therefore develops a mission energy-based aircraft retrofit feasibility framework and applies it to the Britten-Norman BN-2B-26 Islander. The framework evaluates mission energy demand, battery sizing, retained payload, installation volume requirements, payload–range capability and route feasibility. A configuration is considered technically feasible only when the mission energy, peak power, MTOW, retained payload and installation volume requirements are satisfied simultaneously. This study therefore aims not only to determine the achievable electric range, but also to identify which physical or operational constraint governs the feasibility of battery-electric retrofit such as route distance, payload and battery technology change.

2. Materials and Methods

A mission-based computational framework was developed in MATLAB 2025b to assess the technical feasibility and performance of converting an existing conventional commuter aircraft to battery-electric propulsion. The methodology was formulated as an aircraft-level retrofit assessment rather than a clean sheet aircraft design. Consequently, the baseline aircraft maximum take-off weight, operating empty weight, disposable load capability, mission characteristics and available installation volume were treated as principal boundary conditions.

2.1. Baseline Aircraft and Mass Balance

The assessment framework consists of six linked modules: (i) aircraft and retrofit input definition, (ii) mission energy calculations, (iii) battery energy and battery mass calculations, (iv) retained payload and installation volume assessment, (v) payload–range and design space analysis and (vi) route feasibility screening. The same set of mass, energy, reserve and volume constraints is applied throughout the model so that all outputs are internally consistent.

The performance specifications of BN-2B-26 Islander are provided in Table 1. The mass balance formulation for the battery-electric retrofit is based on fixed maximum take-off weight. The MTOW of the battery-electric aircraft cannot reach beyond 2994 kg. The reason behind this assumption is to avoid the complexities involved in the certification requirements. The baseline aircraft take-off mass can be written as

$$m_{\text{MTOW,b}} = m_{\text{OEW,b}} + M_{\text{PL}} + m_{\text{fuel}} \quad (1)$$

Table 1. BN-2B-26 Islander specifications [22].

Parameter	Value
Model	BN-2B-26 Islander
Maximum Take-off Weight (MTOW)	2994 kg
Empty Weight	1866 kg
Disposable load	1128 kg
Maximum Cruise Speed (TAS)	73 m/s
Economical Cruise Speed	65 m/s
Service Ceiling (both engines)	4145 m
Take-off Distance (ground roll)	189 m
Rate of Climb (both engines)	4.37 m/s
Range (with tip tanks/VFR)	1250 km
Range IFR	998 km
Maximum Passengers	up to 9 passengers
Take-off speed	36 m/s
Climb speed	43 m/s

The mass removed from the aircraft is the masses of the engines, fuel system and tanks. The electric propulsion system mass, m_{EPS} , includes the motors, inverters, gearboxes, high voltage distribution equipment, controllers, and thermal management system. The additional integration mass, $m_{\text{integration}}$, accounts for mounting provisions, battery containment, fire protection and any required structural reinforcement. The net propulsion system mass difference can be expressed as Equation (2).

$$\Delta m_{\text{prop}} = m_{\text{EPS}} + m_{\text{integration}} - m_{\text{removed}} \quad (2)$$

The take-off mass of the battery-electric aircraft is

$$m_{\text{MTOW,BE}} = m_{\text{OEW,b}} + m_{\text{battery}} + M_{\text{PL}} + \Delta m_{\text{prop}} \quad (3)$$

This study assumes that the removed mass from the baseline aircraft (m_{removed}) is equal to the electric powertrain (m_{EPS}) and integration masses ($m_{\text{integration}}$). The mass of the battery-electric aircraft can be expressed as

$$m_{\text{MTOW,BE}} = 1866 \text{ (kg)} + m_{\text{battery}} + M_{\text{PL}} \quad (4)$$

The retrofit study must satisfy the maximum mass constraint that is $m_{\text{MTOW, BE}} \leq m_{\text{MTOW, b}}$ so the maximum take-off weight is equal to the baseline aircraft, i.e., 2994 kg. From Equation (4), the maximum battery mass can be estimated as

$$m_{\text{battery, max}} = 1128 - M_{\text{PL}} \quad (5)$$

The mass constraint becomes a trade-off between payload and battery mass, as shown in Equation (6).

$$\Delta m_{\text{prop}} + m_{\text{battery}} + M_{\text{PL}} \leq 1128 \text{ kg} \quad (6)$$

This mass balance determines how much battery mass can be carried. The battery mass ratio can be determined using Equation (7).

$$\beta = \frac{\Delta m_{\text{prop}} + m_{\text{battery}} + M_{\text{PL}}}{m_{\text{MTOW}}} \quad (7)$$

The battery mass ratio (β) is assumed to be same as the baseline aircraft; in that case, the net propulsion system mass difference (Δm_{prop}) becomes zero (theoretical maximum possible mass).

2.2. Battery Technology, Efficiency Calculations and Operational Constraints

Battery technology is represented by battery specific energy. The model evaluates a range of battery specific energy assumptions to represent current, near-term, and future battery technology levels. The aircraft level usable energy is lower than the nominal battery energy because reserve, powertrain efficiency, SOC margin, and end-of-life degradation allowance are applied before cruise energy is calculated.

Section 2.3 discusses energy allowances and performance calculations. The reserve requirement was selected with reference to aviation safety requirements, while the SOC and EOL margins were treated as preliminary engineering assumptions for feasibility screening. Table 2 provides the assumptions and requirements for simulation.

Table 2. Battery operational requirements and modelling assumptions [23].

Parameter	Value
Battery specific energy, SE	200–1000 Wh/kg
Powertrain efficiency	0.80
SOC margin	20% [24]
EOL margin	10%
Reserve requirement	30 min at cruise power

The powertrain efficiency is calculated based on Table 3 and Equation (8). The total powertrain efficiency is around 0.8.

$$\eta_{\text{powertrain}} = \eta_{\text{bat}} \eta_{\text{HV}} \eta_{\text{inv}} \eta_{\text{mot}} \eta_{\text{gb}} \eta_{\text{prop}} \quad (8)$$

Table 3. Battery-electric powertrain component's efficiency.

Component	Symbol	Assumed Efficiency
Battery discharge	η_{bat}	0.98
High-voltage cables/distribution	η_{HV}	0.99
Inverter	η_{inv}	0.98
Electric motor	η_{mot}	0.95
Gearbox/mechanical drive	η_{gb}	0.98
Propeller	η_{prop}	0.90

Table 3 provides information to calculate the battery energy required after applying the powertrain efficiency, SOC margin, reserve and end-of-life requirements.

2.3. Mission Energy Model

The energy available for the battery-electric mission is determined from the installed battery mass, battery specific energy and the operational energy allowances. For a selected battery mass fraction, β , the battery mass is calculated as

$$m_{\text{battery}} = \beta m_{\text{MTOW}} \quad (9)$$

The corresponding nominal battery energy is

$$E_{\text{nominal}} = m_{\text{battery}} SE \quad (10)$$

For the protection of battery health, a minimum state of charge (SOC) of 20% is assumed as the modelling assumption, not a regulatory requirement. And to avoid reducing the battery operational cycle, an end-of-life limit of 90% is assumed [25].

$$E_{\text{usable}} = E_{\text{nominal}}(1 - \text{SOC})(1 - \text{EOL})(\eta_{\text{powertrain}}) \quad (11)$$

A previous study used a similar fraction for total efficiency in the modelling assumption for powertrain efficiency, which includes all the components in the powertrain and propeller [26].

Mission Phase Energy

The battery-electric mission was represented using a phase-based energy model comprising taxi-out, take-off, climb, cruise, descent, approach, taxi-in and reserve segments. For each prescribed mission phase i , the electrical energy requirement was calculated from

$$E_i = P_i t_i \quad (12)$$

Taxi power was estimated from the power required to overcome aircraft rolling resistance. For a simplified taxi-out equation at low speed (25 km/h), a higher rolling resistance value of 0.05 [27] and battery to traction efficiency of 0.5 leads to around 20 kW taxi-out power, according to Equation (13).

$$P_{\text{taxi out}} = \frac{\mu_r mg V_{\text{taxi}}}{\eta_{\text{taxi}}} \quad (13)$$

The flightradar24 [28] shows that most of the flights during taxi-out and taxi-in time were up to 5 min and 3 min, respectively. The change in altitude was estimated as the difference between the cruise altitude and the departure airport elevation. Most flights operate at cruise altitudes below 300 m, and many of the airports are located close to sea level. Therefore, an altitude change of approximately 300 m was adopted. The maximum rate of climbing of the reference aircraft is around 4.4 m/s. Using Equation (14), time to climb can be calculated under steady climbing conditions. Table 4 provides power, time and mission phase energy for various phases of the flight.

$$t_{\text{climb}} = \frac{\Delta h}{ROC} \quad (14)$$

Equation (15) was used to calculate the fixed mission energy, defined as the total energy required for taxi-out, take-off, climb, descent, approach, and taxi-in; cruise and reserve energy were calculated separately. The reference BN-2B-26 is powered by two Lycoming

O-540-E4C5 engines, each rated at 260 hp, giving a total rated take-off power of 520 hp or approximately 388 kW [29]. A climb power of 300 kW, corresponding to approximately 75% of the total rated power, was adopted as a mission model assumption [30]. The economical cruise power was assumed as 65% cruise power setting to the total rated power [31]. The fixed mission energy is 23.75 kWh.

Table 4. Mission profile and energy requirements.

Mission Phase	Power (kW)	Time	Energy (kWh)
Taxi-out	20	0.080 h (~4.8 min)	1.60
Take-off	388	0.017 h (~1.0 min)	6.60
Climb	300	0.019 h (~1.1 min)	5.5
Descent	60	0.080 h (~4.8 min)	4.80
Approach	90	0.050 h (~3.0 min)	4.50
Taxi-in	15	0.050 h (~3.0 min)	0.75

The descending flights take time up to a maximum up to 5 min, so the current assumption keeps the values which is closer to the maximum time required to descending. These values are from the real-time flight trajectory of the baseline aircraft and selected routes for the study.

$$E_{fixed} = E_{taxi\ out} + E_{take\ off} + E_{climb} + E_{descent} + E_{approach} + E_{taxi\ in} \quad (15)$$

The reserve energy is calculated using Equation (16).

$$E_{reserve} = P_{cruise} * t_{reserve} \quad (16)$$

The remaining energy after reserve and fixed mission energy is the cruise energy available to convert into range. It is calculated using Equation (17).

$$E_{cruise} = E_{usable} - E_{fixed} - E_{reserve} \quad (17)$$

The energy consumed during the actual mission is expressed as

$$E_{mission} = E_{fixed} + E_{cruise} + E_{reserve} \quad (18)$$

For the present preliminary model, cruise is assumed to occur at constant power, speed and altitude. The available cruise energy can be calculated as

$$E_{cruise} = \beta m_{MTOW} SE(1 - SOC)(1 - EOL)(\eta_{powertrain}) - (E_{fixed}) - (E_{reserve}) \quad (19)$$

Using the energy available Equation (19), cruise power and cruise speed, the range is calculated as

$$R = \frac{E_{cruise}}{P_{cruise}} V_{cruise} \quad (20)$$

The final range calculation is based on Equation (20); it is obtained by substituting Equation (18) into Equation (19).

$$R = \frac{\beta m_{MTOW} SE(1 - SOC)(1 - EOL) (\eta_{powertrain}) - (E_{fixed}) - (E_{reserve})}{P_{cruise}} V_{cruise} \quad (21)$$

Equation (21) provides the range calculation from available energy for cruising. The cruise phase is assumed to be at constant altitude.

2.4. Battery Sizing

Battery sizing was performed by considering both the mission energy requirement and the peak power requirement, with additional pack-level burden included. The pack burden assumption is supported by NASA X-57 battery pack modelling and cell-to-pack scaling studies [32]. Nominal energy is calculated using Equation (10). The corresponding battery mass is calculated using Equation (22),

$$m_{\text{battery}} = \max(m_P, m_E) \quad (22)$$

m_P is the power limited battery mass. Maximum power requirement was determined (take-off power), and a pack specific power of 1.0 kWkg^{-1} was assumed. Under this assumption, 388 kg is the minimum mass for power limit. m_E is the energy limited battery mass. The battery sizing (estimation of mass and volume) is an iterative process under operational, performance and certification constraints as shown in Figure 2. The battery system volume is

$$V_{\text{battery}} = \frac{\beta \text{ MTOW} * SE}{\rho_{V,\text{pack}}} \quad (23)$$

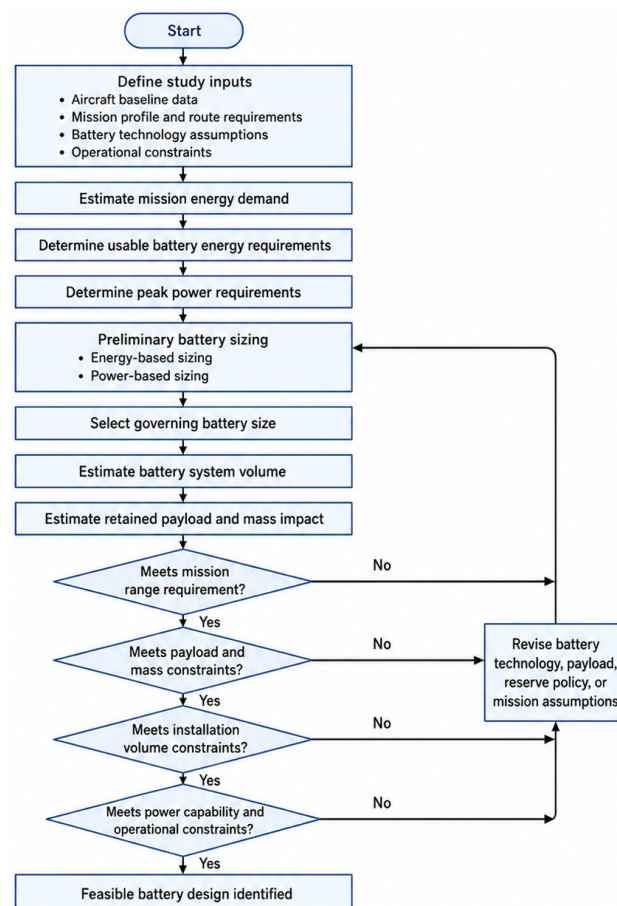


Figure 2. Battery sizing flowchart.

$\rho_{V,\text{pack}}$ is the pack-level volumetric energy density range of 180–580 Wh/L for 200–1000 Wh/kg battery specific energy, with a 1.35 installed system volume factor. High-energy lithium-ion cells have been reported with cell-level volumetric energy density approaching or exceeding 950 Wh/L, but aircraft pack-level values are lower after packaging, cooling, containment and battery management system integration [33].

Figure 2 provides the battery sizing workflow, and Section 2.6 further explains the aircraft battery volume calculations and available volume for passengers.

2.5. Sensitivity and Design Space Analysis

Sensitivity and design space analysis were performed to evaluate how battery technology and operational assumptions influence the battery mass requirement, retained payload and achievable cruise range of the battery-electric retrofit aircraft. The sensitivity analysis determines the minimum battery mass fraction required for a specified mission, whereas the design space analysis provide a design points and performance information.

The useable energy should be equal to the energy required for the mission and regulatory requirements.

$$E_{\text{nominal}} (\eta_{\text{powertrain}})(1 - \text{SOC})(1 - \text{EOL}) = E_{\text{mission}} + P_{\text{cruise}} \left(\frac{t_{\text{reserve}}}{60} \right) \quad (24)$$

The reserve energy is based on the cruise power required for 30 min. From the above equation, the nominal energy required can be written as

$$E_{\text{nominal}} = \frac{E_{\text{mission}} + P_{\text{cruise}}(t_{\text{reserve}}/60)}{(\eta_{\text{powertrain}})(1 - \text{SOC})(1 - \text{EOL})} \quad (25)$$

The battery mass required can be written as

$$m_{\text{battery}} = \frac{E_{\text{nominal}}}{SE} \quad (26)$$

The required battery mass fraction is the ratio between battery mass and maximum take-off mass. From Equation (7), the required battery mass fraction can be expressed as,

$$\beta_{\text{req}} = \frac{E_{\text{mission}} + P_{\text{cruise}}(t_{\text{reserve}}/60)}{SE m_{\text{MTOW}} (\eta_{\text{powertrain}})(1 - \text{SOC})(1 - \text{EOL})} \quad (27)$$

Based on the assumptions and requirements, the SOC, EOL and reserve time can be applied on the above equation. Therefore, the reference required battery mass fraction is

$$\beta_{\text{ref}} = \frac{E_{\text{mission}} + P_{\text{cruise}}(30/60)}{SE m_{\text{MTOW}} (\eta_{\text{powertrain}})(1 - 0.20)(1 - 0.10)} \quad (28)$$

At the reference condition, both battery mass fraction required and reference become equal. Hence, the required battery mass fraction can be written as

$$\frac{\beta_{\text{req}}}{\beta_{\text{ref}}} = \left[\frac{E_{\text{mission}} + P_{\text{cruise}}(t_{\text{res}}/60)}{E_{\text{mission}} + P_{\text{cruise}}(30/60)} \right] \left[\frac{(1 - 0.20)(1 - 0.10)}{(1 - \text{SOC})(1 - \text{EOL})} \right] \quad (29)$$

The main design space map expresses achievable cruise range as a function of battery specific energy and battery mass fraction. Payload boundaries are superimposed on the design space to identify whether a given range can be achieved while retaining a useful payload. The required battery mass fraction (β_{req}) is calculated using Equations (27) and (28).

For a required range, the required battery mass fraction is

$$\beta_{\text{req}} = \frac{\left[E_{\text{fixed}} + E_{\text{reserve}} + P_{\text{cruise}} \left(\frac{R_{\text{req}}}{V_c} \right) \right]}{SE m_{\text{MTOW}} \eta_{\text{powertrain}} (1 - \text{SOC})(1 - \text{EOL})} \quad (30)$$

The aircraft is assumed to be cruising at constant speed. Therefore, cruising time can be simplified as a required range over cruising velocity.

2.6. Battery Installation Volume and Available Aircraft Volume

In addition to battery mass and energy requirements, the feasibility of a battery-electric retrofit is constrained by the physical volume available for battery installation within the existing aircraft geometry. For the BN-2B-26 retrofit, the battery installation assessment considers the battery pack volume, additional system integration volume, and the usable volume available within the former fuel storage region, baggage compartment and passenger cabin. A retrofit configuration is considered volume-feasible only when the required installed battery system volume does not exceed the available installation volume. The subsystem volume (Equation (31)) accounts for the additional volume required by supporting battery system components such as cooling, battery management and high voltage equipment, containment and installation allowance. Figure 3 shows the workflow for feasibility assessment.

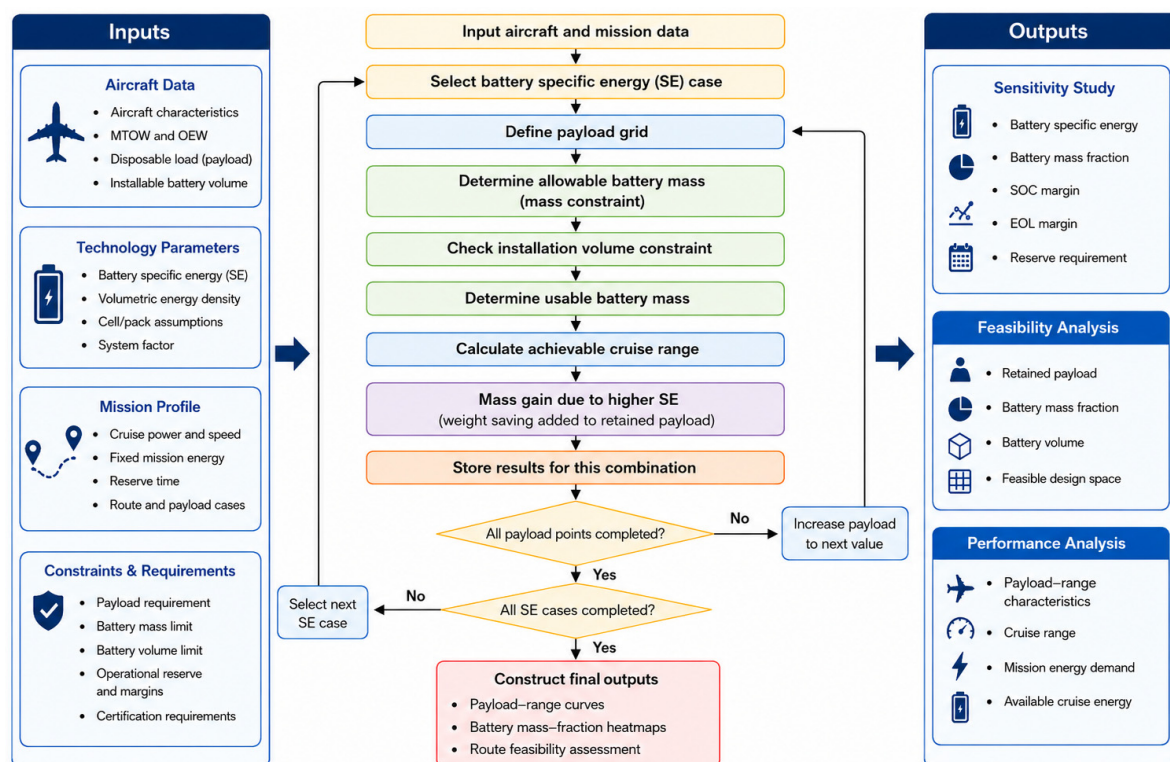


Figure 3. Integrated workflow for payload-range analysis and route feasibility assessment.

The installation volume is assumed to be 0.08, and the total volume required for the battery system is 1.35 times the battery volume at the pack level. A safety volume fraction of 0.35 is added. The factor for this additional volume is in Table 5.

$$V_{\text{subsystem}} = \left(f_{\text{cool}} + f_{\text{BMS}}^{\text{HV}} + f_{\text{contain}} + f_{\text{install}} \right) V_{\text{battery}} \quad (31)$$

The battery volume is given in Equation (23). The total battery system volume is calculated by Equation (32), and the battery volume is calculated based on volumetric energy density, referring to Equation (33).

$$V_{\text{system}} = V_{\text{battery}} + V_{\text{subsystem}} \quad (32)$$

Table 5. The battery cell to pack level factors.

Factor	Value	Reference
Cooling factor	0.14	[34]
BMS + HV hardware	0.05	[35]
Containment	0.08	[36]

The available cabin volume behind the crew zone is calculated from the usable cabin length, cabin width and cabin height, as shown in Equation (17).

$$V_{cabin\ behind\ crew} = (L_{cabin} - L_{crew\ zone}) W_{cabin} H_{cabin} \quad (33)$$

The passenger occupancy in the cabin is calculated as

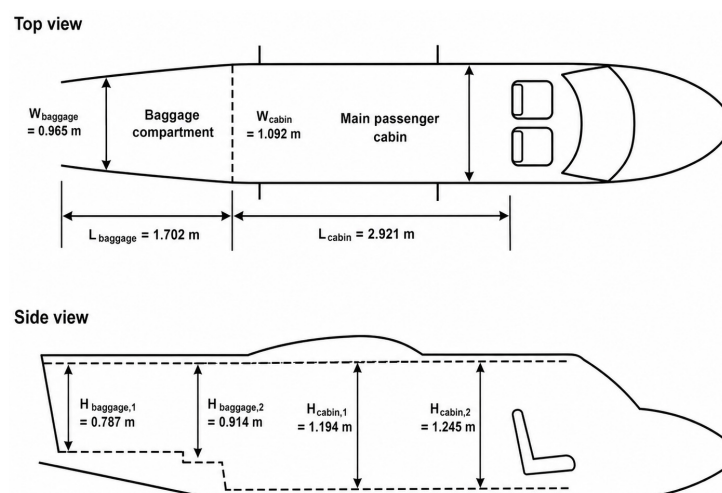
$$V_{occupied} = \frac{V_{cabin\ behind\ crew}}{N_{PAX}} \quad (34)$$

There are eight seats behind the crew members (if we assume that the two front seats are for crew members). Available battery installation volume (Equation (35)) is calculated by adding the cabin volume behind the crew zone, fuel volume in the wing and the baggage volume; subtracting the occupied volume; and applying a miscellaneous (Table 6). Figure 4 provides the dimensions for aircraft geometry.

$$V_{available} = (1 - f_{misc}) [V_{cabin\ behind\ crew} + V_{baggage} + V_{baseline\ fuel} - V_{occupied}] \quad (35)$$

Table 6. BN-2B-26 Islander cabin and baggage dimensions [22].

Parameter	Symbol	Value
Cabin length	L_{cabin}	2.921 m
Cabin width	W_{cabin}	1.092 m
Cabin height	H_{cabin}	1.194 m
Baggage volume	$V_{baggage}$	0.71 m ³
Payload specific volume	V_{occ}	0.486 m ³ /PAX
Miscellaneous/Safety volume allowance	f_{misc}	0.15

**Figure 4.** BN-2B-26 Islander cargo and cabin dimensions.

The baseline aircraft can carry a fuel mass of up to 354 kg at the full payload condition, and the equivalent volume is 0.442 cubic meters. The baggage and cabin dimensions are

given in Table 6. The available volume for battery installation must be higher than the battery system volume. It can be written as

$$V_{\text{system}} \leq V_{\text{available}} \quad (36)$$

2.7. Route Selection and Feasibility Screening

The selected route groups used for this feasibility screening are listed in Table 7. The routes are assessed under the constraints to check the feasibility at different battery specific energy, battery mass ratio and payload. The combined feasibility map is:

$$F = f(SE, \beta, R_{\text{req}}, m_{\text{PL, req}}, V_{\text{available}})$$

Table 7. Selected routes for finding the potential of converting BN-2B-26 Islander into battery-electric [37–39].

Route Group	Routes	Approximate Range Band
Orkney inter-island	Westray–Papa Westray, Kirkwall–Eday, Kirkwall–Sanday, Kirkwall–Westray, Kirkwall–Stronsay, Kirkwall–North Ronaldsay, Kirkwall–Papa Westray	3–60 km
Mainland/Northern Island routes	Kirkwall–Sumburgh, Kirkwall–Tingwall, Wick–Kirkwall, Inverness–Kirkwall	55–170 km
Mainland-to-near-island routes	Glasgow–Campbeltown, Glasgow–Islay, Glasgow–Tiree	90–170 km
Longer route	Glasgow–Barra	about 225 km

For each route i , battery scenario s and payload case p , run:

$$R_{\text{available}}(SE_s, \beta, p) \geq R_{\text{req}, i} \quad (37)$$

Then apply the retrofit constraints:

$$m_{\text{payload retained}} \geq m_{\text{payload required}} \quad (38)$$

$$\beta \leq \beta_{\text{payload}} \quad (39)$$

$$V_{\text{system}} \leq V_{\text{available}} \quad (40)$$

$$E_{\text{usable}} \geq E_{\text{fixed}} + E_{\text{cruise}} + E_{\text{reserve}} \quad (41)$$

A route is technically feasible only if all these Equations (38)–(41) are true.

For each selected route, the required route distance was compared with the cruise range predicted by the battery-electric retrofit model for each combination of battery specific energy, volumetric energy density, specific power and retained payload. A route was classified as technically feasible only when the required range was achieved while simultaneously satisfying the mission energy, peak power, MTOW, retained payload and battery installation volume constraints.

3. Results

This study investigates the potential of converting BN-2B-26 Islander into a battery-powered propulsion. The output of the framework provides a range of information under

assumptions and constraints. The results are presented in a sequence that follows the modelling framework developed in Section 2.

3.1. Mission Energy Budget

The mission energy budget shows that the energy requirement of the battery-electric retrofit is not dominated by take-off despite take-off being the highest power mission phase. The 388 kW take-off requirement acts primarily as a peak power constraint because of its short duration, contributing only approximately 6.6 kWh to the mission. In contrast, climbing requires approximately 5.5 kWh and represents the second largest fixed flight phase energy demand. The 30 min reserve requirement is more significant, requiring approximately 126.5 kWh at the cruise power of 253 kW. Consequently, approximately 150.25 kWh is allocated to fixed mission phases, and reserve before cruise range energy is considered. The resulting mission energy breakdown is shown in Figure 5.

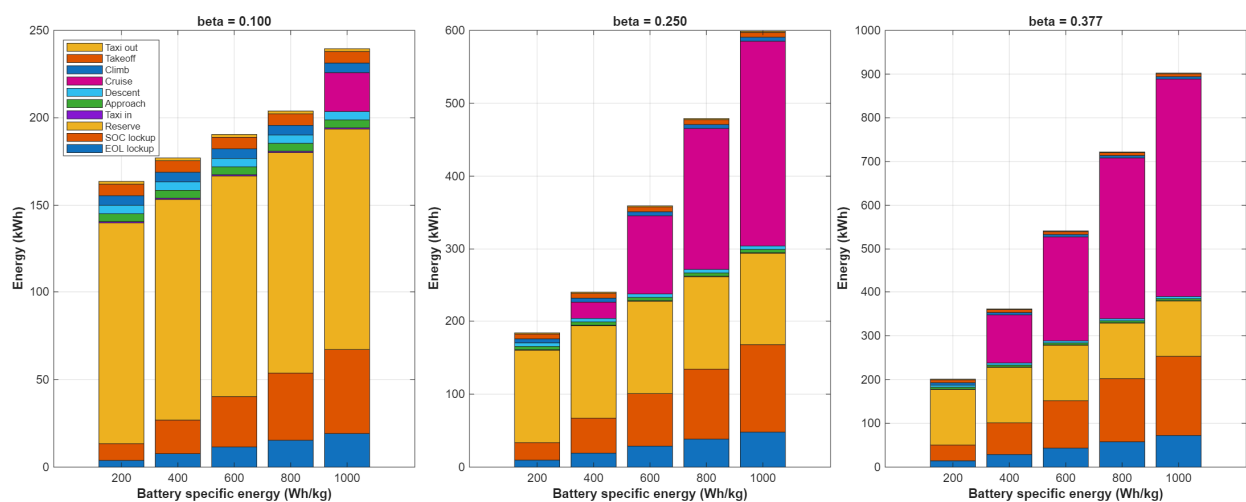


Figure 5. Energy budget model for a battery-electric BN-2B-26 retrofit.

The optimum battery mass fraction can be identified by locating the point at which the calculated cruise range reaches its maximum value. This occurs when the gradient of range with respect to battery mass fraction becomes zero; therefore, range increases with battery mass fraction until the aircraft reaches the allowable mass or payload boundary. For short missions, this fixed energy burden represents a substantial fraction of the total mission requirement, which explains why increasing route distance does not result in a proportional increase in required battery mass. After applying the powertrain efficiency and SOC and EOL allowances, only 57.6% of the nominal battery energy is available for satisfying the modelled mission energy requirement.

3.2. Sensitivity Analysis

The sensitivity analysis evaluates the influence of battery specific energy, state-of-charge margin, and reserve time on the minimum battery mass fraction required for a 50 km mission (Figure 6). The end-of-life allowance was maintained at 10% throughout the analysis, while the other parameters (SOC and reserve) were varied according to Equation (27) in Section 2.5. The results show that battery specific energy has the strongest influence on the required battery mass fraction. At the reference operating condition, increasing the battery specific energy from 400 Wh/kg to 1000 Wh/kg reduces the required battery mass fraction from approximately 29.65% to 11.9%, corresponding to a reduction of about 60%. This occurs because higher specific energy allows the same mission energy

requirement to be satisfied with a lower battery mass. β_P is the minimum battery mass fraction required for power limit.

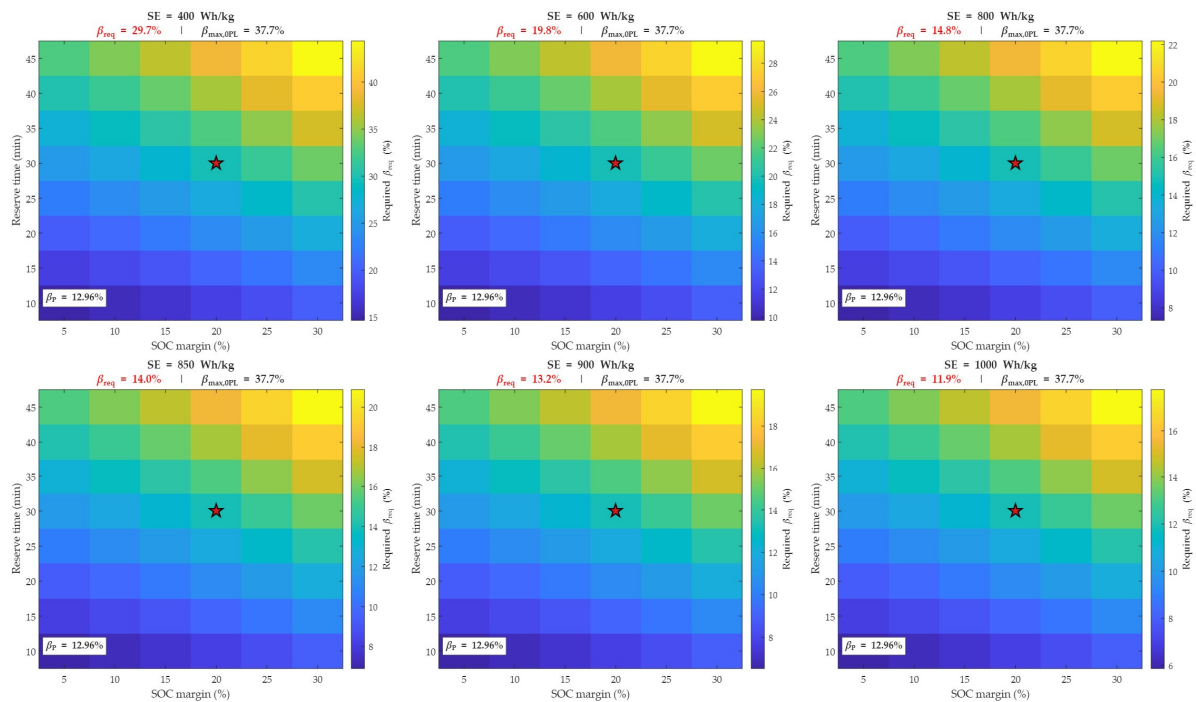


Figure 6. SOC, reserve and battery specific energy sensitivity to battery mass ratio at 10% EOL, 50 km range.

$\beta_{max,OPL}$ is the theoretical maximum battery mass fraction at zero retained payload, it is around 37.7% of the MTOW. The sensitivity map demonstrates that the required battery mass fraction is reducing for the same range as battery specific energy increases, which means the mass fraction reduction provides additional mass that can be used as payload.

3.3. Battery-Electric Retrofit Design Space

The battery-electric design space shows achievable cruise range as a function of battery specific energy and battery mass fraction (Figure 7). Range increases as either battery specific energy or battery mass fraction increases. However, the useful design region is bounded by retained payload limits. At lower battery specific energy, a high battery mass fraction is required to meet even short-range requirements, leaving little useful payload. At 400 Wh/kg, the indicated design point retains only about 240 kg of payload. Increasing battery specific energy to 1000 Wh/kg increases the retained payload to about 740 kg for the same design space condition. The corresponding retrofit design space is presented in Figure 7.

With 1000 Wh/kg and 580 Wh/L batteries, the baggage compartment plus former fuel tank volume is sufficient for approximately a 49 km range while retaining 774 kg of payload (satisfied both volume and mass for PAX). However, it became infeasible due to the power limited mass. At higher specific energy, the battery mass ratio reduces. But due to the specific power limitations, the minimum battery mass is maintained to satisfy the power limit (β_P), i.e., 12.96% (~13%) of MTOW.

At the maximum battery specific energy considered in this study, 1000 Wh/kg, the required battery mass fraction increases from approximately 11.86% for a 50 km cruise range to approximately 22.88% for a 225 km cruise range. A retained payload of 400 kg corresponds to an allowable battery mass fraction of approximately 24.3%, providing a maximum calculated cruise range of about 248 km. If the retained payload is reduced to

200 kg, the allowable battery mass fraction increases to approximately 31.0%, extending the calculated cruise range to about 354 km. The theoretical zero-payload upper limit corresponds to a battery mass fraction of approximately 37.7% and a calculated cruise range of about 460 km. Although this condition represents the maximum range within the fixed-MTOW mass balance, it is not operationally useful because no payload is retained.

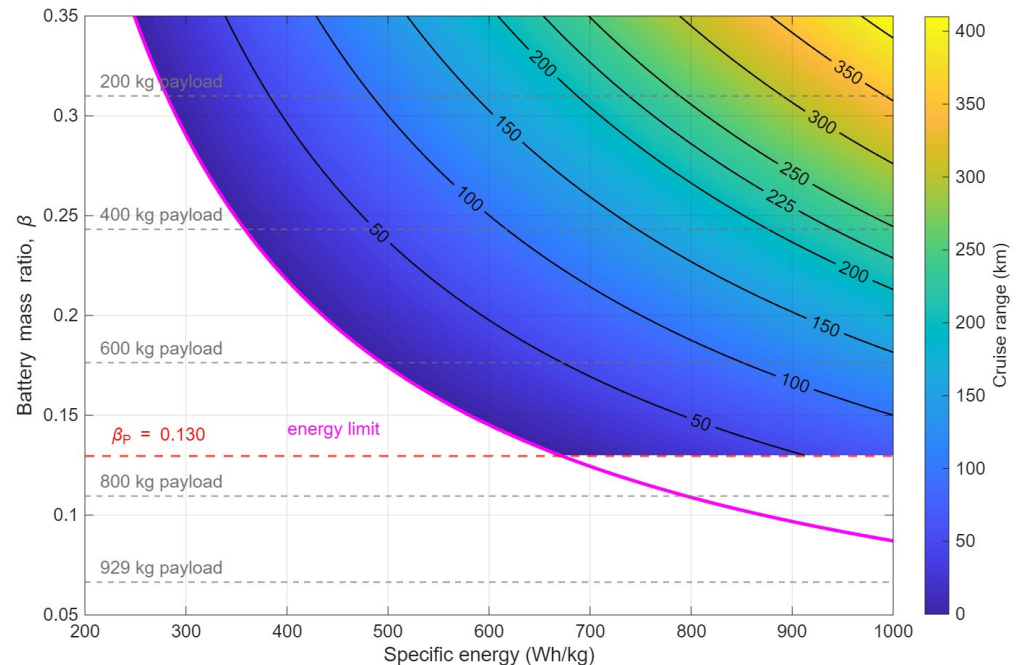


Figure 7. Design space for BN-2B-26 Islander.

3.4. Retained Payload–Range Performance Analysis

In Figure 8, each curve shows range and payload variation. Higher payload leaves less mass and volume for batteries, so range decreases. Higher battery specific energy shifts the curve outward because the same battery mass stores more energy. The retained payload–range analysis shows the trade-off between retained payload and achievable cruise range for the battery-electric BN-2B-26 retrofit. The maximum range at maximum payload for baseline aircraft is estimated based on the fraction of maximum fuel that can be carried while retaining maximum structural payload [29].

The baseline aircraft can carry approximately 774 kg of payload over about 998 km under the selected IFR range condition, whereas even the most optimistic battery case considered in this study remains substantially below this range. At 1000 Wh/kg, the maximum achievable electric range is 460 km under the theoretical zero-payload condition, and for a retained payload of 400 kg it is approximately 248 km, whereas reducing the retained payload to 200 kg increases the calculated cruise range to approximately 354 km.

3.5. Route Feasibility Assessment

The route feasibility assessment shows that route coverage depends strongly on both retained payload and battery specific energy. The 929 kg retained payload case, corresponding to nine passengers plus luggage, is not feasible for any of the selected routes across the battery specific energy range considered. The 750 kg retained payload case is also infeasible under all evaluated battery specific energy assumptions because only 378 kg is available for the battery, which is below the 388 kg minimum battery mass required to satisfy the 388 kW peak power requirement at the assumed battery specific power of 1.0 kW/kg.

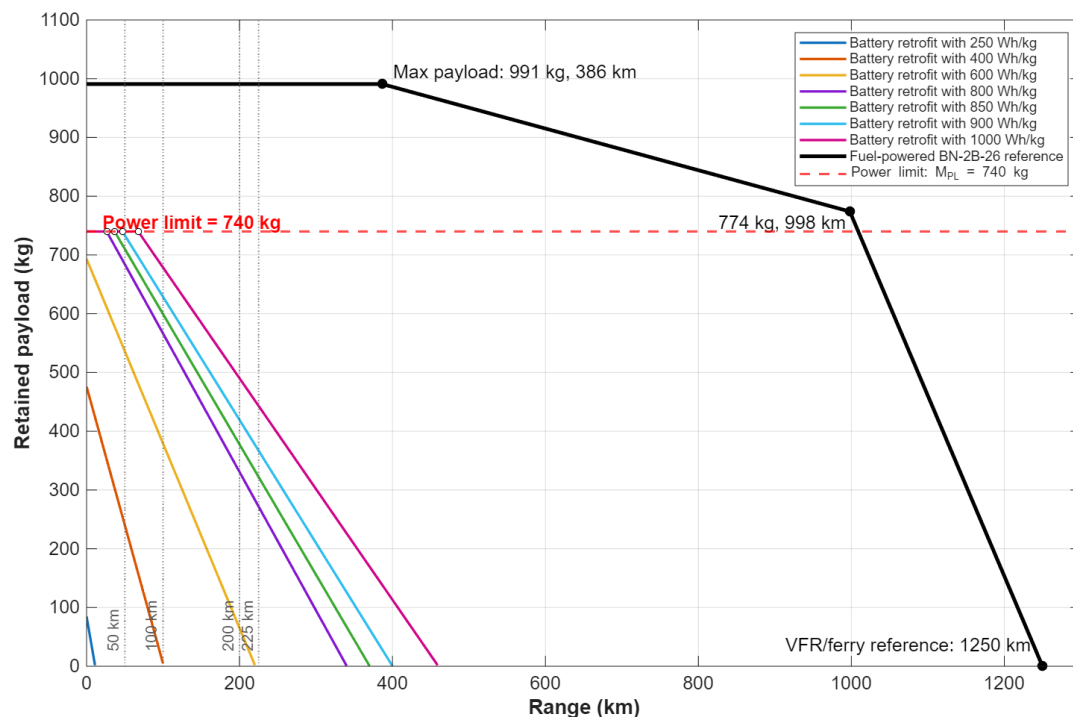


Figure 8. Payload–range comparison of battery-electric retrofit with different battery technologies and the baseline aircraft.

Retained payload was translated into passenger capability using an assumed allowance of 85 kg per passenger and 15 kg of luggage per passenger. Under this assumption, the 929 kg, 750 kg, 500 kg and 200 kg payloads correspond approximately to nine, seven, five and two passengers with luggage, respectively. The results show that high passenger loading sharply reduces feasible route coverage, while lower retained payload cases allow wider route coverage under future battery specific energy assumptions.

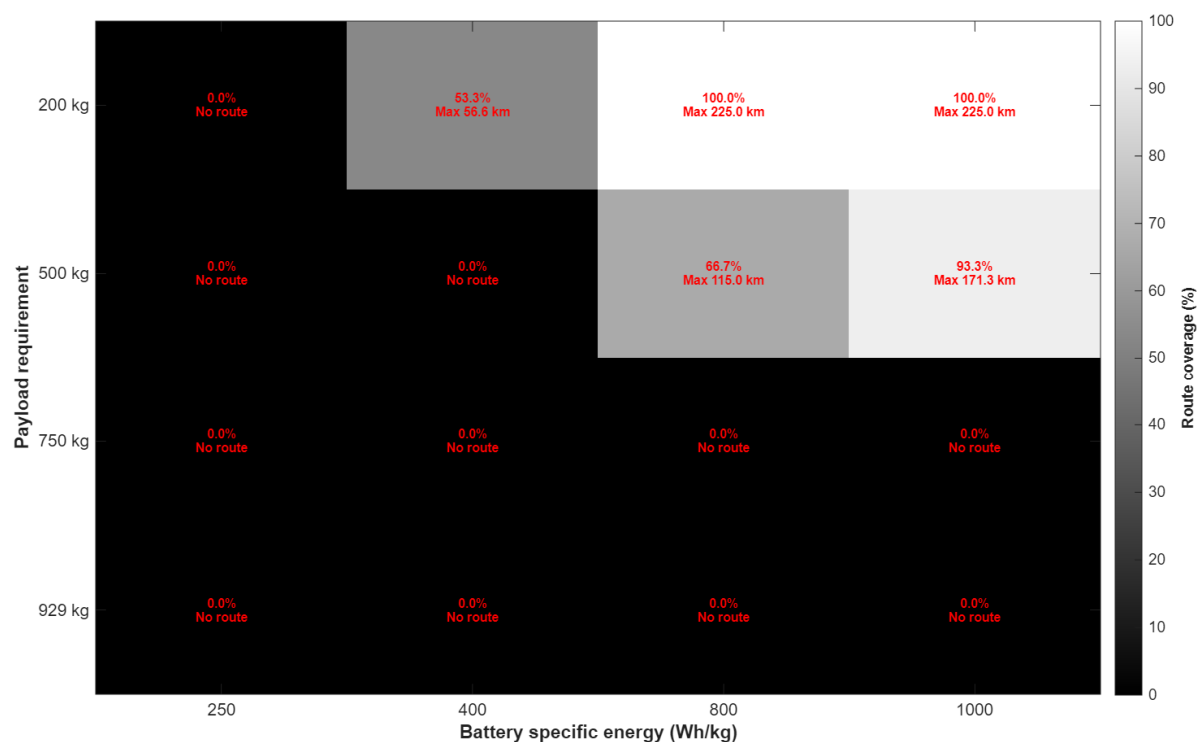
Table 8 summarizes the route level feasibility of the battery-electric BN-2B-26 retrofit for selected domestic routes based on retained payload and battery specific energy assumptions. A route was considered feasible only when the calculated cruise range satisfied the route distance while also meeting the retained payload, battery mass, battery volume, reserve energy and operational constraints.

The 500 kg retained payload case becomes feasible as battery specific energy increases. At 800 Wh/kg, approximately 66.7% of the selected routes are feasible, with a maximum covered route distance of approximately 115 km. At 1000 Wh/kg, route coverage increases to approximately 93.3%, with routes up to approximately 171.3 km becoming technically feasible (Figure 9). The 200 kg retained payload case provides the widest route coverage. At 400 Wh/kg, approximately 53.3% of the selected routes are feasible, with a maximum covered distance of approximately 56.6 km. These route coverage outcomes are reported in Table 8.

Overall, the battery-electric retrofit of the BN-2B-26 is not suitable for full payload replacement of the conventional aircraft. Instead, feasibility improves mainly through payload reduction and higher battery specific energy. The most operationally meaningful compromise is the 500 kg payload case under high specific energy battery assumptions, while the 200 kg case demonstrates maximum route coverage but with limited passenger number.

Table 8. Selected domestic routes in Scotland served by retrofitted aircraft under current, near-term and future battery technology.

Retained Payload	Passengers (+Luggage)	Battery Specific Energy	Feasible Route Range	Route Coverage	Covered Route Outcome
929 kg	9	250–1000 Wh/kg	—	0%	No selected routes are feasible
750 kg	7	1000 Wh/kg	—	0%	No selected routes are feasible because only 378 kg is available for the battery, below the 388 kg peak power minimum.
500 kg	5	800 Wh/kg	2.7–115 km	66.7%	All seven Orkney inter-island routes, plus Wick–Kirkwall, Glasgow–Campbeltown and Glasgow–Islay.
500 kg	5	1000 Wh/kg	2.7–171.3 km	93.3%	Feasible for all selected routes except Glasgow–Barra.
200 kg	2	400 Wh/kg	2.7–56.6 km	53.3%	All seven Orkney inter-island routes plus Wick–Kirkwall.
200 kg	2	800–1000 Wh/kg	2.7–225 km	100%	All selected routes are feasible.

**Figure 9.** Selected route coverage and maximum covered distance.

4. Discussion

The results show that the feasibility of a battery-electric BN-2B-26 Islander retrofit is controlled by the combined effect of battery specific energy, MTOW, retained payload, reserve energy requirement and available installation volume. Although increasing battery specific energy improves the calculated cruise range, the retrofit remains constrained by

the fixed MTOW. Therefore, the battery-electric conversion cannot be interpreted simply as replacing fuel mass with battery mass; instead, the retrofit creates a direct competition between battery mass and useful payload.

A significant fraction of the usable battery energy is consumed by a 30 min reserve requirement before any useful cruise range is produced. As a result, nominal battery capacity cannot be converted directly into range. And for short-range missions, reserve energy is a constraint for limiting the passenger.

The design space analysis shows that a technically possible design is not always useful in operation. At 1000 Wh/kg, the theoretical maximum range is about 460 km, but this occurs at the zero-payload limit, so it is not suitable for route operation. The conventional aircraft can retain approximately 774 kg of payload over about 998 km under the selected IFR range condition, whereas the most optimistic battery case considered in this study remains substantially below this envelope. The range increases further as electric powertrain weight reduces.

The route feasibility results show that the retrofit is strongly payload-dependent. The 500 kg payload provides the most practical balance between passenger capacity and route coverage, while the 200 kg case maximizes route coverage but offers limited passenger number. This supports the conclusion that battery-electric retrofits are more suitable for short, reduced payload route applications than for full payload replacement of the conventional aircraft. The main findings can be summarized in the following points.

- After SOC margin and end-of-life allowance are applied, only 57.6% of battery energy remains before fixed mission and reserve energy are subtracted.
- Matching baseline aircraft performance would require approximately 6050 Wh/kg and 2840 Wh/L, which is beyond current and near-term battery technology.
- Key technology parameters such as battery pack volumetric density, battery specific energy, battery specific power and battery mass ratio are contributing to deciding the feasibility of retrofitting an aircraft into battery-electric.
- Short routes are power limited and longer routes are energy limited (energy is limited by MTOW, payload and volume of the baseline aircraft).
- The results show that 3 to 60 km is the near-term, 60 to 150 km is a technology sensitive region and finally 150 to 225 km is a higher-energy and payload-constrained region.

Future work will focus on the model to include detailed battery pack architecture, thermal management design, CG analysis, structural integration, lifecycle emissions and operating cost. A comparative assessment of battery-electric, hydrogen-electric, hybrid-electric and conventional propulsion for the same route set would also be valuable. Such work would help identify whether battery-electric retrofit is a suitable solution for the short routes, while alternative zero emission propulsion systems may appear more suitable for longer routes.

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Abbreviations

The following abbreviations are used in this manuscript:

BMS	Battery Management System
BN-2B-26	Britten-Norman Islander BN-2B-26 Aircraft Variant
C-rate	Battery Charge/Discharge Rate Relative to Nominal Capacity
CG	Centre of Gravity
CTOL	Conventional Take-Off and Landing
DHC	De Havilland Canada
EASA	European Union Aviation Safety Agency
EOL	End of Life
ICCT	International Council on Clean Transportation
IFR	Instrument Flight Rules
IPCC	Intergovernmental Panel on Climate Change
Li-ion	Lithium-Ion
Li-S	Lithium–Sulphur
MTOW	Maximum Take-Off Weight
NASA	National Aeronautics and Space Administration
NO _x	Nitrogen Oxides
OEW	Operating Empty Weight
PAX	Passengers
SE	Battery Specific Energy
SOC	State of Charge
TAS	True Airspeed
VFR	Visual Flight Rules

Nomenclature

Notation	Definition	Unit
$m_{\text{MTOW,b}}$	Baseline aircraft maximum take-off mass	kg
$m_{\text{OEW,b}}$	Baseline aircraft operating empty weight	kg
M_{PL}	Retained payload mass	kg
m_{fuel}	Fuel mass	kg
$m_{\text{MTOW,BE}}$	Battery electric aircraft take-off mass	kg
m_{MTOW}	Maximum take-off mass	kg
Δh	Height change during climb	m
ROC	Rate of climb	m/s
P_{TO}	Take-off power	kW
P_{cruise}	Cruise power	kW
t_{reserve}	Reserve time	h or min
V_{cruise}	Cruise speed	km/h
R	Achievable cruise range	km
R_{req}	Required range	km
f_{cool}	Cooling system volume factor	-
$f_{\text{BMS/HV}}$	Battery management system and high voltage volume factor	-
f_{contain}	Battery containment volume factor	-

$L_{crew\ zone}$	Crew zone length	m
N_{PAX}	Number of passengers	-
$V_{baseline\ fuel}$	Baseline fuel storage volume	m ³
$R_{available}$	Available cruise range	km
$m_{payload\ retained}$	Retained payload mass	kg
$m_{payload\ required}$	Required payload mass	kg

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