Supporting Information

Mass Summary		
Baseline A320 MZFW	60 500	kg
Net structural & systems modification	−651	kg
Dry electric MZFW (no energy system)	**59 849**	**kg**
Payload	(included in MZFW)	
Battery mass	27 376	kg
Liquid oxygen mass	1 613	kg
Energy system mass	**28 989**	**kg**
MTOW	**88 837**	**kg**
(1.14× baseline A320 MTOW of 78 t)		
Aerodynamic & Propulsion		
Wingspan	52.0	m
Wing area	140.4	m^2
Aspect ratio	19.27	—
Wing loading	633.0	kg/m^2
Zero-lift drag coefficient C_{D_0}	0.01479	—
Induced drag factor K	0.01540	—
Maximum lift-to-drag ratio $(L/D)_{max}$	33.1	—
Total installed power	**14.63**	**MW**
Power per motor (8 motors)	1.83	MW
Mission Performance		
Design range	700	nm
Cruise altitude	25 000	ft
Cruise Mach number	0.50	—
Block time	2:20	hr
Climb time	11.8	min

Energy & Oxidiser Budget (per flight)		
Battery pack capacity (with reserves)	20 848	kWh
Typical mission energy (no reserves used)	13 191	kWh
Mission O_2	982.7	kg
En-route reserve O_2	29.5	kg
Reserve mission O_2	600.8	kg
Total O_2 required	**1 613**	**kg**
Volume Allocation		
Outboard wing volume	26.58	m^3
Centre wing box volume	9.96	m^3
Total wing volume	36.54	m^3
Battery in wings + centre box	25 568	kg
Battery in fuselage	1 808	kg
Fraction of battery stored in wings	93.4	%
Fuselage volume consumed (battery + dewars)	4.25	m^3
A320 cargo volume (baseline)	39.54	m^3
Remaining cargo volume	**35.29**	m^3

Table S1 – Optimised aircraft metrics for 52m wingspan, 25000ft 0.5M cruise, 700nm range mission

Drag Polar Implications of Propulsion Architecture Change
Methodology
The propulsion-related drag modification was evaluated using a conventional parasite drag build-up approach based on equivalent skin-friction estimation[1,2]**.** The induced drag component was treated using the standard Oswald efficiency formulation, with sensitivity to engine count discussed by Howe[3]

Baseline Parameters
The Airbus A320 reference wing area is:

$$S_{ref} = 122.6\ \mathrm{m}^2$$

as reported in Airbus technical specifications [4]. The baseline propulsion installation consists of two Pratt & Whitney PW1100G-JM turbofan engines with take-off thrust ratings up to:

$$T_{TO} = 147.28\ \mathrm{kN}$$

depending on model variant[5]. These engines are mounted in under-wing nacelles with associated pylons.

Parasite Drag Estimation
Zero-lift drag is estimated using the equivalent skin-friction method:

$$C_{D,0} = C_{fe}\frac{S_{wet}}{S_{ref}} \qquad (1.1)$$

where C_{fe}represents the equivalent skin-friction coefficient. For transport-class jet aircraft, typical values lie in the range:

$$C_{fe} = 0.003 - 0.004$$

A nominal value of:

$$C_{fe} = 0.0035$$

is adopted. The propulsion-driven wetted-area change is:

$$\Delta S_{wet} = S_{wet,prop}^{(new)} + S_{wet,cool}^{(new)} - S_{wet,prop}^{(old)} \qquad (1.2)$$

Baseline Turbofan Installation
Using conceptual nacelle wetted-area estimation methods the combined nacelle and pylon wetted area per turbofan installation is assumed to lie within:

$$S_{wet,tf,unit} = 30 - 45\ \mathrm{m}^2$$

giving:

$$S_{wet,prop}^{(old)} = 60 - 90\ \mathrm{m}^2$$

Distributed Electric Propulsors
For the eight leading-edge propulsor pods, streamlined external-body approximations are used.
The wetted area per pod is estimated as:

$$S_{wet,pod,unit} = 4 - 7\ \mathrm{m}^2$$

giving:

$$S_{wet,prop}^{(new)} = 32 - 56\ \mathrm{m}^2$$

An additional cooling and integration allowance of:

$$S_{wet,cool}^{(new)} = 10 - 20\ \mathrm{m}^2$$

is included to account for heat exchanger inlets/outlets and electrical integration.

Resulting Zero-Lift Drag Change
The propulsion-driven zero-lift drag increment becomes:

$$\Delta C_{D,0} = C_{fe}\frac{\Delta S_{wet}}{S_{ref}} \qquad (1.3)$$

Substitution yields:

$$\Delta C_{D,0} = -0.0014 \text{ to } +0.0005$$

with a nominal midpoint of:

$$\Delta C_{D,0} = -4.5 \times 10^{-4}$$

The magnitude of this change lies within the conceptual limits of aerodynamic uncertainty.

Induced Drag Sensitivity

Induced drag is written as:

$$C_{D,i} = \frac{C_L^2}{\pi A e} \qquad (1.4)$$

The Oswald efficiency factor estimation methods indicate that engine count may influence span efficiency.

Increasing from

$$N_e = 2$$

to

$$N_e = 8$$

could reduce e. A suitable sensitivity band is:

$$e_{new} \approx (0.75 - 0.90)e_{old}$$

which captures the first-order effect without detailed aerodynamic redistribution modelling.

Conclusion of Drag Polar Analysis

Both the parasite drag modification and induced drag sensitivity fall within the uncertainty bounds associated with early-stage wetted-area and span-efficiency estimation. Consequently, the baseline drag polar was retained unchanged in the main sizing analysis.

Electrochemistry and Oxygen Storage Calculations

Li-O_2 Electrochemical Basis

Lithium-air batteries use oxygen as the cathode reactant.

Anode reaction

$$\text{Li} \rightarrow \text{Li}^+ + e^- \qquad (2.1)$$

Cathode reaction

$$\text{O}_2 + 2\text{Li}^+ + 2e^- \rightarrow \text{Li}_2\text{O}_2 \qquad (2.2)$$

Overall reaction

$$2\text{Li} + \text{O}_2 \rightarrow \text{Li}_2\text{O}_2 \qquad (2.3)$$

For the formation of lithium peroxide, four electrons are transferred per mole of oxygen.
The theoretical electrical energy released per mole of oxygen is:

$$E_{mol} = nFV \qquad (2.4)$$

where:

- $n = 4$
- $F = 96\,485 \text{ C mol}^{-1}$
- $V \approx 2.9 \text{ V}$

Therefore:

$$E_{mol} = 1.12 \times 10^6 \text{ J mol}^{-1}$$

Dividing by the molar mass of oxygen,

$$M_{O_2} = 32 \text{ g mol}^{-1}$$

gives the theoretical specific energy relative to oxygen mass:

$$\frac{E}{m_{O_2}} \approx 35 \text{ MJ kg}^{-1} \approx 9.72 \text{ kWh kg}^{-1} \qquad (2.5)$$

Full-Range Stoichiometric Oxygen Requirement
Using the fuel-equivalent energy benchmark:

$$E_{req} = 548\,765 \text{ MJ}$$

the ideal stoichiometric oxygen requirement is:

$$m_{O_2} = \frac{548\,765}{35} \approx 15.7 \text{ t} \qquad (2.6)$$

Applying a 10% utilisation and system margin:

$$m_{O_2,design} \approx 17.3 \text{ t}$$

Atmospheric Oxygen Extraction Penalty
Since atmospheric air contains approximately 21% oxygen by mass, the total air mass that must be processed is:

$$m_{air} = \frac{m_{O_2}}{0.21} \approx 82 \text{ t} \qquad (2.7)$$

For a representative 3-hour mission:

$$\dot{m}_{air} \approx 7.6 \text{ kg s}^{-1}$$

At cruise altitude (32,000-37,000 ft), where:

$$\rho \approx 0.35 - 0.40 \text{ kg m}^{-3}$$

the associated volumetric flow rate is:

$$\dot{V} \approx 20 \text{ m}^3 \text{ s}^{-1}$$

At 35,000 ft the ambient pressure is approximately:

$$P_1 \approx 0.25 \text{ atm}$$

The ideal specific compression work (Çengel and Boles, 2015) is:

$$w = c_p T_1 \left[\left(\frac{P_2}{P_1} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right] \qquad (2.8)$$

Using:

$$T_1 = 220 \text{ K}, \gamma = 1.4, c_p = 1005 \text{ J kg}^{-1}\text{K}^{-1}, \frac{P_2}{P_1} \approx 4$$

gives:

$$w \approx 80 \text{ kJ kg}^{-1}$$

The resulting compressor power is:

$$P_{comp} = \dot{m}_{air} w \approx 610 \text{ kW}$$

Allowing for realistic compressor efficiencies (70-80%), the electrical demand exceeds 750 kW

Constraint Diagram Equations
Drag Polars

$$C_D = C_{D0} + K C_L^2 \qquad (3.1)$$

where:

$$C_{D0} = 0.018$$

and

$$K = 0.039 \times 0.9 = 0.0351$$

The sharklet correction applied to the induced drag factor is:

$$k_{neo} = 0.9\, k_{baseline} \qquad (3.2)$$

The take-off and landing configuration drag polar is:

$$C_D = C_{D0,TO} + K_{TO} C_L^2 \qquad (3.3)$$

where:

$$C_{D0,TO} = C_{D0} + 0.017 = 0.035$$

and

$$K_{TO} = 0.039$$

Initial Cruise Altitude (ICA)

The cruise true airspeed at the initial cruise altitude is:

$$V_{cr} = M_{ICA}\sqrt{\gamma R T_0 \theta} \qquad (3.4)$$

where:

$$\gamma = 1.4, R = 287\ \mathrm{J\ kg^{-1}K^{-1}}, T_0 = 288\ \mathrm{K}$$

The lift coefficient required to sustain flight is:

$$C_L = \frac{(W/S)\ g\ f_W}{0.5\ \rho_0\ \sigma\ V_{cr}^2} \qquad (3.5)$$

The ICA power-to-weight constraint is:

$$\left(\frac{P_0}{W_0}\right)_{ICA} = \frac{V_{cr}\ g\ f_W}{\eta_{prop}\ f_{power}}\left(\frac{C_D}{C_L}\right) \qquad (3.6)$$

Climb with 20% Power Inoperative

The decision speed is:

$$V_2 = 1.2\ V_{stall,TO}$$

where:

$$V_{stall,TO} = \sqrt{\frac{2(W/S)g}{\rho_0 C_{L,max,TO}}} \qquad (3.7)$$

The required power-to-weight ratio with only 80% power available is:

$$\left(\frac{P_0}{W_0}\right)_{climb} = \frac{V_2 g}{\eta_{prop} \times 0.8}\left[\frac{C_{D0,TO} + K_{TO}C_L^2}{C_L} + \sin\left(\tan^{-1}\frac{2.4}{100}\right)\right] \qquad (3.8)$$

where:

$$C_L = \frac{W/S}{0.5\rho_0 V_2^2}$$

The climb-gradient term represents the CS-25 minimum second-segment climb requirement of 2.4%

Take-Off Field Length

The transition radius at $V_2$is:

$$R = \frac{V_2^2}{g_{rot}g} \qquad (3.9)$$

where:

$$g_{rot} = 0.1$$

The required power-to-weight ratio for the certified field length

$$S_{tot} = 1.15 \times 2180\ \mathrm{m}$$

is:

$$\left(\frac{P_0}{W_0}\right)_{TO} = \frac{\frac{V_2^3}{\sqrt{2}}}{S_{tot} - \frac{V_2^2\gamma_c}{g_{rot}g}}\ \frac{1}{1 - \frac{R\gamma_c + h_{sc} - R\gamma_c^2/2}{S_{tot}}} \qquad (3.10)$$

where:

$$\gamma_c = \frac{10\pi}{180}$$

and

$$h_{sc} = \frac{35}{3.048}\ \text{m}$$

Approach Speed
The approach-speed wing loading constraint is:

$$\left(\frac{W}{S}\right)_{app} = 0.5\rho_0 \left(\frac{V_{app}}{1.3}\right)^2 \frac{f_W g}{f_S} C_{L,max,land} \qquad (3.11)$$

Landing Field Length
The landing field length constraint is:

$$\left(\frac{W}{S}\right)_{land} = \frac{4.88243\left(0.6S_{land,ft} - 800\right)}{75 f_W} C_{L,max,land} f_S \qquad (3.12)$$

Constraint Envelope
The final design envelope is defined by the most demanding constraint:

$$\left(\frac{P_0}{W_0}\right)_{env} = \max\left[\left(\frac{P_0}{W_0}\right)_{ICA}, \left(\frac{P_0}{W_0}\right)_{climb}, \left(\frac{P_0}{W_0}\right)_{TO}\right] \qquad (3.13)$$

Mass Build-Up Equations
Motor Mass
The bare motor mass is determined from installed power and motor specific power:

$$m_{motor,bare} = \frac{P_0}{n_{motors}\, SP_{motor}} \qquad (4.1)$$

where:

$$SP_{motor} = 13.2\ \text{kW kg}^{-1}$$

and

$$n_{motors} = 8$$

An installation factor of 1.2 is applied:

$$m_{motor,installed} = 1.2\, m_{motor,bare} \qquad (4.2)$$

Empennage Mass Savings
Vertical tail saving:

$$\Delta m_{VT} = 490 \times 0.30 = 147\ \text{kg} \qquad (4.3)$$

Horizontal tail saving:

$$\Delta m_{HT} = 650 \times 0.15 = 98\ \text{kg} \qquad (4.4)$$

Structural Reinforcement

$$\Delta m_{struct} = k_{struct}\, m_{fuselage\ load} \qquad (4.5)$$

where:

$$k_{struct} = 0.10$$

Landing Gear

$$\Delta m_{LG} = 0.044\, MTOW - 3763 \qquad (4.6)$$

Thermal Management

$$m_{cooling} = 0.1\, P_0(\text{kW}) \qquad (4.7)$$

Modified Zero-Fuel Weight

$$MZFW_{new} = MZFW_{A320} - \left(\Delta m_{CFRP} + \Delta m_{APU} + \Delta m_{fuelsys} + \Delta m_{tail} + \Delta m_{wing}\right)$$

$$+\left(\sum m_{motors} + \Delta m_{struct} + \Delta m_{LG} + m_{cooling}\right) \qquad (4.8)$$

MTOW Closure

$$MTOW = MZFW_{new} + m_{batt} \qquad (4.9)$$

Electric Aircraft Drag Polar

$$C_D = 0.0148 + KC_L^2 \qquad (4.10)$$

where:

$$K = \frac{1}{\pi ARe}$$

is evaluated using the actual aircraft aspect ratio.

LOX Dewar Volume

$$V_{O_2} = \frac{m_{O_2}}{\rho_{LOX}\, f_{dewar}} \qquad (4.11)$$

Remaining Cargo Volume

$$V_{cargo,remaining} = V_{cargo,A320} - V_{batt,fus} - V_{O_2} \qquad (4.12)$$

Volumetric Energy Density

The cell-level volumetric energy density is calculated as:

$$\rho_{vol,cell} = \varepsilon_{grav,cell}\bar{\rho}_{cell} \qquad (4.13)$$

Using the assumed cell gravimetric energy density and average cell density:

$$\begin{aligned}\rho_{vol,cell} &= 1150\ \mathrm{Wh\,kg^{-1}} \times 0.826\ \mathrm{kg\,L^{-1}} \\ &\approx 950\ \mathrm{Wh\,L^{-1}} \qquad (4.14)\end{aligned}$$

The assumed average cell mass density is:

$$\bar{\rho}_{cell} \approx 1.0 - 1.1\ \mathrm{g\,cm^{-3}}$$

consistent with dense pouch-cell configurations supplied by an external oxygen source.

Cell-to-Pack Volumetric Efficiency

The volumetric packing efficiency is:

$$\begin{aligned}\eta_{vol,ctp} &= \frac{t}{t+g_v} \cdot \frac{W}{W+g_h} \\ &= \frac{10}{14.5} \times \frac{100}{105.5} \approx 0.65 \qquad (4.15)\end{aligned}$$

where:

- $t = 10\mathrm{mm}$ (cell thickness)
- $g_v = 4.5\mathrm{mm}$ (vertical spacing)
- $W = 100\mathrm{mm}$ (cell width)
- $g_h = 5.5\mathrm{mm}$ (horizontal spacing)

Liquid Oxygen Dewar Packing Efficiency

Liquid oxygen is assumed to be stored in vacuum-insulated Dewar vessels. Allowing for insulation, structural shell volume, ullage space and boil-off management, an overall volumetric packing efficiency of:

$$\eta_{vol,dewar} = 0.80$$

is adopted.

Wing Mass Estimation Methodology
Empirical Wing Mass Model
Following Kretov and Tiniakov[6], the wing mass fraction is:

$$\bar{m}_{wing} = n_{calc}\lambda\psi\, 0.022 m_{TO}^{1/4} p_0\, \bar{c}_{root}^{0.75} (\cos \chi_{0.25})^{1.5} \times \left(0.85 + \frac{\bar{c}_{root}}{\bar{c}_{tip}} + \frac{4.5 k_2 k_3}{p_0} + 0.01\left(\frac{1}{\eta} + 3\right) k_1\right) \qquad (5.1)$$

where:

- m_{TO}= maximum take-off mass
- n_{calc}= limit load factor
- λ= aspect ratio
- S= wing area
- $p_0 = m_{TO}/S$= wing loading
- $\bar{c}_{root}, \bar{c}_{tip}$= thickness ratios
- $\chi_{0.25}$= quarter-chord sweep
- $\eta = c_{tip}/c_{root}$= taper ratio
- ψ= bending-relief coefficient
- k_1= service-life coefficient
- k_2= high-lift complexity coefficient
- k_3= fuel-tank sealing coefficient

The absolute wing mass is:

$$m_{wing} = \bar{m}_{wing}\, m_{TO} \qquad (5.2)$$

Adapted Electric Aircraft Coefficients
Service Life

$$k_1 = 1.1 \qquad (5.3)$$

High-Lift Device Complexity

$$k_2 = 1.5 \qquad (5.4)$$

Fuel Tank Sealing
Since no wet-wing fuel storage is present:

$$k_3 = 1.0 \qquad (5.5)$$

Modified Bending Relief Formulation
The original regression defines the bending-relief coefficient as:

$$\psi = 0.92 - 0.5\bar{m}_{fuel} - 0.1k_{eng} \qquad (5.6)$$

For electric propulsion:

$$\bar{m}_{fuel} = 0 \qquad (5.7)$$

and the turbofan term is replaced by a distributed mass-moment formulation:

$$\psi_{elec} = 0.92 - c_\psi \sum_{i=1}^{8} \left(\frac{W_{mot,i}}{W_{TO}} \frac{y_i}{b/2}\right) \qquad (5.8)$$

where:

- $W_{mot,i}$= motor weight
- $W_{TO} = m_{TO}g$= take-off weight
- y_i= spanwise motor location
- b= wingspan
- c_ψ= calibration constant

Electric Propulsion Mass Model
Eight electric motors (four per wing) are assumed, with a specific power of:

$$\frac{P}{m} = 13.2\ \text{kW kg}^{-1} \qquad (5.9)$$

Wing Box Volume Estimation Methodology
Airfoil Scaling
The NASA SC(2)-0714 airfoil ordinates are scaled at each spanwise station by:

$$s_y(y) = \frac{(t/c)(y)}{(t/c)_{base}} = \frac{(t/c)(y)}{0.14} \qquad (6.1)$$

where the local thickness-to-chord ratio varies linearly from:

$$\left(\frac{t}{c}\right)_{root} = 0.152$$

at the wing root to:

$$\left(\frac{t}{c}\right)_{tip} = 0.108$$

at the wing tip.
The local chord also varies linearly from:

$$c_{root} = 7.0\ \text{m}$$

to:

$$c_{tip} = \frac{2S}{b} - c_{root} \qquad (6.2)$$

Gross Wing Box Area
The wing box is defined between:

- Front spar: 15% chord
- Rear spar: 60% chord

At each of 300 spanwise stations between:

$$y_{in} = 2.5\ \text{m}$$

and

$$y_{out} = \frac{b}{2} - 1.0\ \text{m}$$

the gross cross-sectional area is:

$$A_{gross}(y) = c(y)^2 \int_{x_{fs}/c}^{x_{rs}/c} \left[y_{upper}(x/c)\, s_y - y_{lower}(x/c)\, s_y\right]\, d(x/c) \qquad (6.3)$$

Structural Deductions
The following CFRP wing structural dimensions are adopted[7].

Upper & lower skin panels	$t_{skin} = 3\text{mm}$
Front & rear spar webs	$t_{spar} = 4\text{mm}$
Spar caps (4×)	$60 \times 20\text{mm}$
Stringers (20 total)	$A_{str} = 300\text{mm}^2$ each
Ribs	$t_{rib} = 2\text{mm}$, pitch $= 500\text{mm}$

The structural area at each station is:

$$A_{struct}(y) = 2t_{skin}w_{box} + t_{spar}(h_{fs} + h_{rs}) + 4w_{cap}h_{cap} + 2n_{str}A_{str} + A_{gross}\frac{f_{rib}t_{rib}}{p_{rib}} \qquad (6.4)$$

where:

- w_{box}= wing-box width

- h_{fs}, h_{rs}= front and rear spar heights
- f_{rib} = 0.70= rib solid fraction
- p_{rib} = 0.50 m= rib pitch

Cooling Duct Provision
Four spanwise cooling ducts are incorporated per wing-box bay.
Each duct has a cross-section of:

$$25 \times 40 \text{ mm}$$

giving a total duct area of:

$$A_{cool} = 4.0 \times 10^{-3} \text{ m}^2$$

at each spanwise station.
The duct sizing is informed by the CFD study of Koumakis et al. (2025), which showed that adequate battery cooling can be achieved when:

- inter-cell spacing exceeds 3 mm,
- air velocities are maintained between 1.5 and 3 m/s.

Net Usable Volume
After structural and cooling deductions, a 5% clearance allowance is applied for:

- wiring,
- BMS electronics,
- manufacturing tolerances,
- thermal expansion.

The net usable area becomes:

$$A_{net}(y) = \left[A_{gross}(y) - A_{struct}(y) - A_{cool}\right](1 - f_{clear}) \qquad (6.5)$$

where:

$$f_{clear} = 0.05$$

The total usable wing volume is therefore:

$$V_{net} = 2\int_{y_{in}}^{y_{out}} A_{net}\,(y)\,dy \qquad (6.6)$$

Limitations
The NASA SC(2)-0714 airfoil is representative but not identical to the proprietary Airbus wing section. Consequently:

- local differences in camber and curvature affect trailing-edge volume estimates;
- structural dimensions have not been validated by detailed finite-element analysis;
- battery-induced centre-of-gravity effects are not modelled;
- certification issues relating to crashworthiness, fire safety and maintenance access remain outside the scope of this study.

Flight Model Equations
This appendix presents the governing equations used in the time-stepping flight model described in Section 3.4. The model employs common aerodynamic and atmospheric relationships across all flight phases.

Atmospheric Model
ISA Troposphere ($h < 11\,000$ m)
Temperature is given by:

$$T(h) = T_0 - Lh \qquad (7.1)$$

Density ratio:

$$\sigma(h) = \frac{\rho(h)}{\rho_0} = \left(1 - \frac{Lh}{T_0}\right)^{\frac{g}{RL}-1} \qquad (7.2)$$

Pressure ratio:

$$\delta(h) = \frac{p(h)}{p_0} = \left(1 - \frac{Lh}{T_0}\right)^{\frac{g}{RL}} \qquad (7.3)$$

where:

$$T_0 = 288.15 \text{ K}$$
$$L = 0.0065 \text{ K m}^{-1}$$
$$g = 9.807 \text{ m s}^{-2}$$
$$R = 287.05 \text{ J kg}^{-1}\text{K}^{-1}$$
$$\rho_0 = 1.225 \text{ kg m}^{-3}$$

Stratosphere ($h \geq 11\,000$ m)
Temperature remains constant:

$$T = 216.65 \text{ K}$$

Density ratio:

$$\sigma(h) = \sigma_{11}\exp[-6.848 \times 10^{-5}(h - 11000)] \qquad (7.4)$$

Pressure ratio:

$$\delta(h) = \delta_{11}\exp[-6.848 \times 10^{-5}(h - 11000)] \qquad (7.5)$$

where σ_{11} and δ_{11} are the values at the tropopause.

The inverse ISA relation is used during cruise-climb:

$$h = \frac{T_0}{L}\left[1 - \sigma^{1/\left(\frac{g}{RL} - 1\right)}\right] \qquad (7.6)$$

to determine altitude from density ratio.

Airspeed Relationships
True airspeed and equivalent airspeed are related by:

$$V_{TAS} = \frac{V_{EAS}}{\sqrt{\sigma}} \qquad (7.7)$$

Mach number is:

$$M = \frac{V_{TAS}}{\sqrt{\gamma R T}} \qquad (7.8)$$

where:

$$\gamma = 1.4$$

and

$$T = T_0\theta$$

The calibrated airspeed approximation is:

$$V_{CAS} \approx V_{EAS}\left[1 + \frac{1}{8}(1 - \delta)M^2 + \frac{3}{640}(1 - 10\delta + 9\delta^2)M^4\right] \qquad (7.9)$$

This formulation is used during the deceleration phase to identify when the aircraft reaches 250 kt CAS.

Aerodynamic Forces
The lift coefficient at each time step is:

$$C_L = \frac{W/S}{\frac{1}{2}\rho_0\sigma V_{TAS}^2} \qquad (7.10)$$

The drag coefficient follows the quadratic drag polar:

$$C_D = C_{D0} + KC_L^2 \qquad (7.11)$$

where:

$$C_{D0} = 0.0148$$

and

$$K = \frac{0.9}{\pi ARe}$$

is the induced-drag factor evaluated using the current aspect ratio.
The drag force is:

$$D = C_D\left(\frac{1}{2}\rho_0\sigma V_{TAS}^2\right)S \qquad (7.12)$$

Propeller Efficiency Model
The propulsive efficiency is modelled as a function of density ratio:

$$\sigma = \frac{\rho}{\rho_0}$$

such that:

$$\eta_{prop}(\sigma) = \eta_{SL}\sigma^{\alpha_{prop}} \qquad (7.13)$$

where:

$$\eta_{SL} = 0.92$$

and

$$\alpha_{prop} = 0.05$$

This formulation captures the first-order effects of air density on propeller performance through changes in advance ratio, Reynolds number, and blade-element efficiency[8].
The exponent $\alpha_{prop} = 0.05$reflects an optimistic 2030+ technology scenario incorporating:

- variable-pitch propellers,
- contra-rotating open rotors,
- swept blade tips,
- active twist control,
- advanced aerofoil sections.

Propulsive Power Model
The available propulsive power at any altitude is:

$$P_{avail} = P_0 k_{pwr}\eta_{prop}(\sigma) \qquad (7.14)$$

where:

- P_0= installed shaft power,
- k_{pwr}= throttle setting,
- η_{prop}follows Equation (7.13).

The throttle setting varies by flight phase, as per table S2:

Phase	k_{pwr}
Climb (Phases 1-3)	1.0
Cruise acceleration	1.0
Cruise	DVtas/P_0 (Variable)
Deceleration	0.1
Reserve climb	0.75
Hold	DVtas/P_0 (Variable)

Table S2 – Flight phase versus throttle setting.

Oxygen Consumption

At each time step the electrical energy delivered is converted into oxygen consumption according to:

$$\Delta m_{O_2} = \frac{P\Delta t}{3.6 \times 10^6\, \varepsilon_{O_2}} \qquad (7.16)$$

where:

- P is power (W),
- $\Delta t = 1\text{s}$,
- $\varepsilon_{O_2} = 9.72\text{kWh kg}^{-1}$.

The factor 3.6×10^6 converts watt-seconds into kilowatt-hours.

Aircraft weight is updated as:

$$W(t + \Delta t) = W(t) - \Delta m_{O_2} g \qquad (7.17)$$

Flight Profile Stages

Altitude (ft)	σ	η_{prop}	Loss vs SL
0	1.000	0.920	—
5,000	0.862	0.913	0.8%
10,000	0.738	0.906	1.5%
15,000	0.629	0.898	2.4%
20,000	0.533	0.891	3.2%
25,000	0.449	0.883	4.0%
30,000	0.374	0.874	5.0%

Table S3. Propeller efficiency variation with altitude ($\eta_{SL} = 0.92, \alpha = 0.05$)

Phase 1: Constant-EAS Climb

At constant equivalent airspeed:

$$V_{TAS} = \frac{V_{EAS}}{\sqrt{\sigma}}$$

The climb rate is:

$$\frac{dh}{dt} = \frac{P_{avail} - DV_{TAS}}{W} \frac{1}{1 - \frac{V_{EAS}^2}{2g\sigma^2}\frac{d\sigma}{dh}} \qquad (7.18)$$

The density gradient in the troposphere is:

$$\frac{d\sigma}{dh} = -\left(\frac{g}{RL} - 1\right)\frac{L}{T_0}\left(1 - \frac{Lh}{T_0}\right)^{\frac{g}{RL}-2} \qquad (7.19)$$

or numerically:

$$\frac{d\sigma}{dh} \approx -4.2568\frac{L}{T_0}\left(1 - \frac{Lh}{T_0}\right)^{3.2568}$$

The correction term accounts for the kinetic energy required to maintain constant EAS as density decreases with altitude.

Phases 2-3: Accelerating Climb

The climb angle is:

$$\phi = \sin^{-1}\left(\frac{P_{avail}/V_{TAS} - D}{W}\right) \qquad (7.20)$$

The lift coefficient becomes:

$$C_L = \frac{W\cos\phi/S}{\frac{1}{2}\rho_0\sigma V_{TAS}^2} \qquad (7.21)$$

The climb rate is:

$$\frac{dh}{dt} = \frac{P_{avail} - DV_{TAS}}{W} \qquad (7.22)$$

The acceleration rate is:

$$\frac{dV}{dt} = \frac{P_{avail}/V_{TAS} - D - W\sin\phi}{W/g} \qquad (7.23)$$

Forward-Euler integration yields:

$$h(t + \Delta t) = h(t) + \frac{dh}{dt}\Delta t \qquad (7.24)$$

$$V_{TAS}(t + \Delta t) = V_{TAS}(t) + \frac{dV}{dt}\Delta t \qquad (7.25)$$

Phase 4: Cruise-Climb at Constant Mach
During cruise, the Mach number is held constant and the altitude increases gradually as oxygen is consumed. The cruise-climb condition is enforced by requiring that the density ratio tracks the weight:

$$\sigma_{cruise}(t) = \frac{W(t)}{W(t_{TOC})} \times \sigma_{TOC} \qquad (7.26)$$

where t_{TOC} is the top-of-climb time and σ_{TOC} is the density ratio at the initial cruise altitude.
The altitude corresponding to this density ratio is obtained from Equation (7.6).
At each timestep, the TAS is:

$$V_{TAS} = M_{cruise}\sqrt{\gamma R T(h)} \qquad (7.27)$$

The power required equals the drag power:

$$P_{req} = D\, V_{TAS} \qquad (7.28)$$

This phase is embedded in an outer range-matching loop that adjusts a distance offset parameter until the total mission distance (climb + cruise + descent) converges to the target range within 1 km.

Phase 5: Descent at Cruise Mach
The descent rate is prescribed as:

$$\dot{h} = -2000 \text{ ft/min}$$

The descent angle is:

$$\gamma = \arcsin\left(\frac{|\,\dot{h}\,|}{V_{TAS}}\right) \qquad (7.29)$$

The propulsive power required during descent is less than in level flight because the component of weight along the flight path assists the motion:

$$P_{descent} = (D - W\sin\gamma)V_{TAS} \qquad (7.30)$$

This expression is positive (power is still required) because at the relatively shallow descent angles used ($\gamma \approx 2 - 3°$), the drag exceeds the gravitational component.

Phase 6: Deceleration
At 10,000 ft, the throttle is reduced to:

$$k_{pwr} = 0.1$$

and the aircraft decelerates in level flight.

The deceleration is:

$$\frac{dV}{dt} = \frac{P_{avail}/V_{TAS} - D}{W/g} \qquad (7.31)$$

This is negative (the aircraft decelerates) because at 10% power the available thrust is less than drag.
The phase terminates when the CAS (Equation 7.9) decreases to 250 kt.

Phase 7: Descent to 1,500 ft
The aircraft descends at constant 250 kt EAS with:

$$\dot{h} = -1700 \text{ ft/min}$$

The equations are identical to Phase 5 except that the speed is expressed as EAS rather than Mach, and the descent rate is reduced to reflect the lower-altitude environment.

Reserve Mission
Phase 8: Climb to 12,000 ft
Identical in formulation to Phase 1, but at 75% power and climbing to 12,000 ft.

$$k_{pwr} = 0.75$$

Phase 9: Diversion Cruise (200 nm)
The diversion is flown at 12,000 ft at the long-range cruise speed, defined as the speed that minimises fuel (oxygen) consumption per unit distance. The optimal lift coefficient for best range is:

$$C_{L,br} = \sqrt[4]{\frac{C_{D0}}{K}} \qquad (7.32)$$

The corresponding TAS is:

$$V_{TAS,br} = \sqrt{\frac{W}{\frac{1}{2}\rho_0 \sigma C_{L,br} S}} \qquad (7.33)$$

The power required is:

$$P = D \times V_{TAS}$$

at this condition, and the phase terminates when 200 nm of additional distance has been covered.

Phase 10: Descent to 1,500 ft
Identical to Phase 7.

Phase 11: 30-Minute Hold
The hold is flown at 1,500 ft at the minimum-drag speed:

$$C_{L,md} = \sqrt{\frac{C_{D0}}{K}} \qquad (7.34)$$

This is the speed at which $L/D$is maximised, minimising power and hence oxygen consumption for a given hold duration.
The hold lasts 1800 s (30 minutes).

Battery Sizing
After all mission and reserve phases, the total oxygen consumed is:

$$m_{O_2,total} = m_{O_2,mission} + m_{O_2,reserve} + 0.03 m_{O_2,mission} \qquad (7.35)$$

where the 3% factor is the en-route contingency.
The required battery energy capacity is:

$$E_{batt} = \frac{m_{O_2,total} \times \varepsilon_{O_2}}{\eta_{DoD}} + E_{misc} \qquad (7.36)$$

where

$$\eta_{DoD} = 0.80$$

limits the usable state-of-charge window and E_{misc} accounts for taxi, take-off, approach, and overshoot energy (Section 2.2.4). The battery hardware mass is:

$$m_{batt,hw} = \frac{E_{batt}}{\varepsilon_{grav} \times \eta_{pack}} \qquad (7.37)$$

where

$$\varepsilon_{grav} = 1150 \text{ Wh/kg}$$

is the cell-level gravimetric density and

$$\eta_{pack} = 0.66$$

is the cell-to-pack gravimetric efficiency.
The total energy system mass fed back to the convergence loop is:

$$m_{batt} = m_{batt,hw} + m_{O_2,total} \qquad (7.38)$$

Horizontal Distance
At each timestep, the horizontal distance increment accounts for the vertical velocity:

$$\Delta x = \sqrt{\max\left(V_{TAS}^2 - \left(\frac{dh}{dt}\right)^2, 0\right)} \times \Delta t \qquad (7.39)$$

Reverse Thrust Implementation for Electric Propulsion
Reverse thrust is achieved using variable-pitch propellers operating in the beta/reverse range, consistent with turboprop practice and certification frameworks[5]. Although electric motors can reverse rotation, this is not preferred due to slower response and aerodynamic inefficiency. Variable-pitch systems enable rapid thrust reversal and effective blade incidence control, with reverse thrust levels of approximately 30–60% of take-off thrust reported. A conservative value of 40% is adopted in the present study.

Landing Distance Model with Reverse Thrust
Landing distance is evaluated using a time-marching simulation from approach speed to rest, comprising an airborne segment from the 50 ft screen height and a subsequent ground roll. The airborne distance is computed assuming a constant approach speed and descent rate, consistent with certification practice. The ground roll is modelled by integrating the longitudinal deceleration under the combined effects of aerodynamic drag, wheel braking, rolling resistance, and reverse thrust. Aerodynamic forces are calculated using a drag polar with a reduced lift coefficient to account for partial lift retention after touchdown, which reduces the normal force available for braking.Wheel braking is represented using a speed-dependent friction coefficient derived from empirical polynomial fits, capturing the reduction in braking effectiveness at higher speeds. Rolling resistance is included as a function of the normal load on the landing gear. Reverse thrust is modelled based on Equation 7.10. A thrust cap is applied to prevent non-physical behaviour at low speeds. The model is integrated forward in time until the aircraft comes to rest, and the total landing distance is obtained by summing the airborne and ground-roll components. This approach captures the key physical mechanisms governing landing performance for the electric propulsion configuration.

Atmospheric Oxygen Extraction Model
This appendix outlines the methodology used to estimate oxygen extraction requirements, associated energy demand, and system sizing for the atmospheric lithium-air propulsion architecture.
Oxygen Demand
Oxygen consumption is linked to electrical power via:

$$\dot{m}_{O_2} = \frac{P_{elec}}{\eta_{sys} e_{O_2}} \qquad (8.1)$$

where

$$e_{O_2} \approx 9.5 - 9.72 \text{ kWh/kg}$$

for Li_2O_2 formation [9,10].Integration over the mission yields a total oxygen requirement of: 1,949.9 kg (including reserves).

Air Flow Requirement
Assuming 21% oxygen content in air:

$$\dot{m}_{air} = \frac{\dot{m}_{O_2}}{0.21} \qquad (8.2)$$

giving a peak air mass flow of:

$$1.95 \text{ kg/s}$$

Energy Requirement
The extraction penalty arises primarily from separation inefficiencies rather than compression. Compressor power is estimated by:

$$P_{comp} = \dot{m}_{air} c_p T_1 \left[\left(\frac{p_2}{p_1} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right] \frac{1}{\eta_{comp}} \qquad (8.3)$$

Total extraction energy is:

$$134 \text{ kWh}$$

(1.1% of propulsive energy), with a peak power of:

$$206 \text{ kW}$$

System Sizing
ASU mass is estimated from specific power:

$$m_{ASU} = \frac{P_{comp,peak}}{\rho_{spec}} \qquad (8.4)$$

yielding:

$$2{,}065 \text{ kg}$$

and a volume of:

$$0.08 \text{ m}^3$$

Assumptions
- Constant extraction efficiency across operating conditions
- No altitude or thermal performance penalties
- Neglect of transient and thermal management effects

Atmospheric Oxygen Supply Alternative
An alternative propulsion architecture was evaluated in which oxygen is extracted from the atmosphere rather than stored onboard as LOX. This eliminates the need for cryogenic storage and associated tank volume, replacing it with an air separation unit (ASU) and compressor system supplying oxygen to the lithium-air cells as discussed within Section 2.2.4. Relative to the LOX baseline, the atmospheric architecture shifts the mass-energy balance: removing 1,613 kg of stored oxygen increases battery mass from 27,376 kg to 33,030 kg due to higher electrical demand.

Parameter	LOX Storage	Atmospheric O
Aircraft Performance		
Wing area (m²)	140.4	151.7
Aspect ratio	19.27	17.83
Total installed power (MW)	14.63	15.81

Power per motor (MW)	1.83	1.98
Mass & Energy		
Battery mass (kg)	27,376	33,030
O_2 storage mass (kg)	1,613	0
ASU mass (kg)	0	2,065
Battery capacity (kWh)	20,848	25,154
System Characteristics		
O_2 source	Liquid oxygen	Ambient air
Peak compressor power (kW)	0	206
Compressor energy (% mission)	0	1.1%
Volume & Payload		
Fuselage volume used (m³)	4.25	7.75
Remaining cargo volume (m³)	35.29	31.79

Table S4: Comparison of LOX Storage and Atmospheric Oxygen Architectures.

Eliminating LOX tankage reduces dedicated volume (ASU: 0.08 m³), but increased battery mass raises fuselage utilisation, reducing cargo capacity from 35.29 m³ to 31.79 m³. While useful for comparison, this configuration is not carried forward. The LOX architecture remains more energy- and mass-efficient, but introduces significant integration and certification challenges, as CS-25 regulations do not address propulsion-scale oxidiser systems and LOX presents established safety risks. The trade-off is therefore between efficiency (LOX) and integration feasibility (atmospheric).

Manufacturing Emissions: Supporting Calculations

Emission factors are cradle-to-gate and are taken directly from the cited literature or from the ecoinvent v3.8 database[11], no bottom-up scaling factors or overhead uplifts are introduced.
Where a source reports a range, the central value is used.

Li-Air Battery: Justification of the Adopted Factor

The adopted Li-air factor of 74 kgCO$_2$eq/kWh for pack-level NMC811. A qualitative comparison of the two chemistries supports this as an upper bound:

- **Cathode.** NMC811 active material contributes 20-30 kgCO$_2$eq/kWh, or 30-40% of the pack total A porous-carbon Li-air cathode replaces this with carbon black (2-5 kgCO$_2$eq/kg,) at approximately 0.17 kg/kWh, giving <1 kgCO$_2$eq/kWh even including MnO_2 catalyst loading.
- **Anode.** Synthetic graphite (4-8 kgCO$_2$eq/kWh,) is replaced by lithium metal. Ecoinvent gives lithium metal at 15 kgCO$_2$eq/kg; at 0.25 kg/kWh loading this yields 3.75 kgCO$_2$eq/kWh.
- **New emissions source.** The ceramic LLZO solid electrolyte membrane (ecoinvent: 15-30 kgCO$_2$eq/kg, thickness 50 µm) adds 1-2 kgCO$_2$eq/kWh.

The net effect is a reduction in the cathode and anode contributions of roughly 20 kgCO$_2$eq/kWh, partially offset by the new ceramic membrane. A plausible Li-air factor is therefore 55-65 kgCO$_2$eq/kWh. The more conservative 74 kgCO$_2$eq/kWh (the NMC811 median, with no chemistry benefit) is adopted to avoid overstating the environmental case. This choice adds no more than ~120000 kgCO$_2$eq to the estimated deficit, which corresponds to fewer than 8 additional break-even flights.

Liquid Oxygen Production

LOX emissions follow directly from ASU specific energy and grid carbon intensity:

$$e_{LOX} = E_{ASU} \times GI \times 10^{-3} \qquad (10.1)$$

with:

$$E_{ASU} = 0.42 \text{ kWh/kg}$$

(Smith and Klosek, 2001).
Per flight, with

$$m_{O_2} = 1613 \text{ kg}$$

LOX emissions range from 15 kgCO$_2$eq (France) to 479 kgCO$_2$eq (India).

Electric Motor: Material-Level Cross-Check
The adopted installed motor factor of 3 $kgCO_2eq/kW$ is taken from the PMSM LCA [12]report 4.1 $kgCO_2eq$ per kilogram of finished PMSM, Table S5. Dividing by aerospace specific power (13.2 kW/kg) and applying the installed mass factor of 1.2 gives 0.37 $kgCO_2eq/kW$ for the bare installed motor. Adding the power electronics contribution of 2.6 $kgCO_2eq/kW$ reported in the same study yields the rounded value of 3 $kgCO_2eq/kW$ used in the main text.
For the converged total installed power of:

$$14630 \text{ kW}$$

this gives:

$$43890 \text{ kgCO}_2\text{eq}$$

for the eight-motor system.

CFM56-5B: Cross-Check Against Material-Level Build-Up
The adopted value of 40000 $kgCO_2eq$ per engine is taken directly from the A320 LCA of Howe et al[3]..As an independent cross-check, the engine can be decomposed by material mass using ecoinvent v3.8 factors [11] accounting for aerospace buy-to-fly (BTF) ratios and closed-loop scrap recycling. Table S6 shows the result. The material-level cross-check gives ~70000 $kgCO_2eq$ per engine. This is higher than the 40000 kg figure from Howe[3], the discrepancy being attributable to assumed BTF ratios, scrap-recycling credits, and the allocation between manufacturing and facility overhead in the published LCA.   The main-text analysis adopts the published LCA value as the primary source; the material build-up is retained only as a plausibility check.

Break-Even: Worked Calculation and Grid Sensitivity

$$CO_{2,batt} = 20848 \times 74 = 1542752 \text{ kgCO}_2\text{eq} \qquad (10.2)$$

$$CO_{2,motors} = 14630 \times 3 = 43890 \text{ kgCO}_2\text{eq} \qquad (10.3)$$

$$\Delta CO_{2,mfg} = (1542752 + 43890) - 80000$$

$$= 1506642 \text{ kgCO}_2\text{eq} \qquad (10.4)$$

$$CO_{2,kero,ceo} = 5500 \times 3.16 = 17380 \text{ kgCO}_2 \qquad (10.5)$$

$$CO_{2,kero,neo} = 4675 \times 3.16 = 14773 \text{ kgCO}_2 \qquad (10.6)$$

(10.7)

Per-flight saving and break-even as a function of grid intensity, applied consistently to both battery charging (14338 kWh of grid energy from a typical mission draw of 13191 kWh at 92% charging efficiency) and LOX production (677 kWh for 1613 kg of LOX at 0.42 kWh/kg).The break-even result is now dominated by two competing effects: the very large manufacturing deficit (1.5 $MtCO_2eq$, over 95% of which is the 20848 kWh battery pack), and the substantial per-flight grid energy draw of 14338 kWh for charging. Against the A320ceo, break-even remains recoverable across all real-world grids, ranging from 87 flights on a fully renewable grid to 223 flights on India's coal-heavy grid. Against the more efficient A320neo, the range widens from 102 to 364 flights, and on a theoretically fully coal-fired grid (1000 gCO_2/kWh), per-flight grid emissions from charging alone exceed the neo's kerosene combustion emissions: the electric aircraft never breaks even in that limit.
The threshold grid intensities at which break-even becomes infeasible are:

$$\frac{17380}{15.015} \approx 1158 \text{ gCO}_2/\text{kWh}$$

(vs CEO) and

$$\frac{14773}{15.015} \approx 984 \text{ gCO}_2/\text{kWh}$$

(vs NEO).
No real-world national grid currently exceeds these thresholds, though several (Poland, India, China) are within a factor of two of the NEO threshold.
On the world average grid of:

$$473 \text{ gCO}_2/\text{kWh}$$

the electric aircraft pays back its manufacturing deficit within 196 flights against the A320neo, roughly one month of typical single-aisle utilisation, or ~0.5% of a 20-year service life. The environmental case therefore holds across all plausible electricity sources against both baselines, but it is considerably weaker than implied by the unrealistically small battery capacity assumed in earlier iterations, and is now quantitatively sensitive to grid decarbonisation.

20-Year Lifecycle CO_2: Detailed Calculation

The 20-year lifecycle CO_2 budget is constructed from three components:

1. Initial manufacturing.
2. Repeat battery manufacturing (electric only).
3. Cumulative per-flight operating emissions.

Material	Mass frac.	e_{mat} (kg CO_2/kg)	Source
Electrical steel (Si-Fe)	0.40	2.15	ecoinvent (Wernet et al., 2016)
Copper (primary)	0.20	4.75	ecoinvent (Wernet et al., 2016)
Aluminium (recycled)	0.20	1.35	ecoinvent (Wernet et al., 2016)
NdFeB magnets	0.03	30.0	Nordelöf et al. (2019)
Steel (shaft, bearings)	0.07	1.20	ecoinvent (Wernet et al., 2016)
Insulation, varnish	0.10	2.00	Nordelöf et al. (2019)
Finished motor total	**1.00**	**4.1**	Nordelöf et al. (2019)

Table S5: PMSM material composition and cradle to gate emissions.

Section	m_{fin} (kg)	Material	e_{mat}	R_{BTF}	CO_2 (kg)
HP turbine (blades + discs)	250	Ni superalloy	14	10	35 000
LP turbine (blades + discs)	350	Ni superalloy	14	7	34 300
HP/LP compressor	500	Ti-6Al-4V	10	6	30 000
Fan + case	450	Ti/composite	9	3	12 150
Combustor	150	Ni superalloy	14	4	8 400
Shafts, bearings	180	Steel	3	2.5	1 350
Accessories, FADEC	200	Mixed	4	1.5	1 200
Gross (virgin material)	**2 080**				**122 400**
With 50% closed-loop scrap recycling on ($R_{BTF}-1$) fraction:					~65 000
Less assembly, test-cell run, quality:					+5 000
Cross-check total					**~70 000**

Table S6: CFM56-5B component level cross check.

Parameter	Symbol	Value
Battery pack capacity	E_{batt}	20 848 kWh
Typical mission energy	$E_{mission}$	13 191 kWh
Battery mfg. factor	e_{batt}	74 kg CO_2eq/kWh
Total installed motor power	P_{total}	14 630 kW
Motor mfg. factor	e_{motor}	3 kg CO_2eq/kW
Two CFM56-5B (Howe 2013)	$CO_{2,CFM56}$	80 000 kg CO_2eq
A320ceo fuel burn (700 nm)	$m_{fuel,ceo}$	5 500 kg (Airbus, 2021a)
A320neo fuel burn (700 nm)	$m_{fuel,neo}$	4 675 kg (15% less (Airbus, 2021a))
Kerosene emission index	EI	3.16 kg CO_2/kg
LOX per flight	m_{O_2}	1 613 kg

Table S7 Break even calculation parameters 700nm design point.

Parameter	Symbol	Value
Service life	T	20 yr
Utilisation	N_{yr}	2 040 flights/yr
Total flights	N_{flights}	40 800
Battery cycle life	N_{cycle}	1 500
Pack replacements	N_{repl}	$\lceil 40\,800/1\,500 \rceil - 1 = 27$
Total packs manufactured	—	28
Battery pack mfg emissions	$\mathrm{CO}_{2,\mathrm{batt}}$	1 542 752 kg
Motor mfg emissions (one-time)	$\mathrm{CO}_{2,\mathrm{mot}}$	43 890 kg
Two CFM56-5B mfg (one-time)	$\mathrm{CO}_{2,\mathrm{eng}}$	80 000 kg
Per-flight CO_2 (ceo kerosene)	—	17 380 kg
Per-flight CO_2 (neo kerosene)	—	14 773 kg
Per-flight charging energy (grid)	E_{charge}	14 338 kWh
Per-flight LOX ASU energy	E_{LOX}	677 kWh
Combined grid energy per flight	E_{total}	15 015 kWh

Table S8 – 20 year lifecycle assumptions

Assumptions

Lifetime CO_2 Equations

$$CO_{2,20yr,conv} = CO_{2,eng} + N_{flights} \times CO_{2,kerosene} \qquad (10.8)$$

$$CO_{2,20yr,elec} = (1 + N_{repl}) \times CO_{2,batt} + CO_{2,mot} + N_{flights} \times (E_{total} \times GI \times 10^{-3}) \qquad (10.9)$$

Evaluating the fixed manufacturing contribution for the electric aircraft:

$$CO_{2,mfg,elec} = 28 \times 1542752 + 43890$$
$$= 43197056 + 43890$$
$$= 43240946 \text{ kg} \approx 43.24 \text{ Mt}$$

and for the two baselines:

$$CO_{2,20yr,ceo} = 80000 + 40800 \times 17380$$
$$= 708.98 \text{ Mt}$$
$$CO_{2,20yr,neo} = 80000 + 40800 \times 14773$$
$$= 603.08 \text{ Mt}$$

Grid Sensitivity

The per-flight operating CO_2 for the electric aircraft is:

$$(14338 + 677) \times GI \times 10^{-3} = 15.015 \times GI \text{ kg}$$

and the 20-year operating total is:

$$40800 \times 15.015 \times GI \times 10^{-3} = 612.6 \times GI \text{ kg}$$

Cycle-Life Sensitivity

Because battery manufacturing dominates the electric aircraft's lifetime footprint, the cycle-life assumption has a large effect on the final saving. For N_{cycle} between 750 and 3000, the required number of packs is:

$$\left\lceil \frac{40800}{N_{cycle}} \right\rceil$$

Table 10 shows the effect on the world-average-grid result. The cycle-life sensitivity is substantial but less extreme than the flight-count break-even result would suggest, because even at 750 cycles the electric aircraft's operating emissions (289.8 Mt on the world average grid) dominate its 55-pack manufacturing budget (84.9 Mt). The 20-year saving against the neo remains above 35% across the full plausible cycle-life range, and above 90% on a near-renewable grid regardless of cycle life.

Operating Cost Model: Detailed Calculations

Utilisation Assumptions

The assumption of 6 flights per day is conservative for a short-haul narrowbody, as typical operations range between 6 and 10 sectors depending on route length and network structure.The 700 nm mission, with a block time of approximately 3 hours, limits achievable daily utilisation relative to shorter routes. An operating year of 340 days accounts for maintenance downtime and is consistent with typical airline utilisation models.

Energy Cost Calculations
Conventional A320ceo

(10.10)
(10.11)
(10.12)

The fuel price of $0.85/kg corresponds to approximately $2.70/US gallon at the standard Jet-A1 density of 0.8 kg/L.

Electric Aircraft
CFM56-5B Engine Maintenance: Detailed Build-Up
The engine MRO reserve rate is built up from its constituent elements.
The adopted central value of $750/EFH (rounded from $752) is consistent with published Aircraft Commerce analysis of the CFM56-5B on short-haul operations and with the rate-per-flight-hour (RPFH) programmes offered by CFM International for fleet-wide coverage
At 3.5 block hours per flight:

$$C_{eng,conv} = 750 \times 3.5 = \$2{,}625 \text{ per flight} \qquad (11.1)$$

$$C_{eng,conv,20yr} = 750 \times 142{,}800 = \$107{,}100{,}000 \qquad (11.2)$$

Electric Motor Maintenance: Build-Up

$$C_{prop,elec} = 110 \times 3.5 = \$385 \text{ per flight} \qquad (11.3)$$

$$C_{prop,elec,20yr} = 110 \times 142{,}800 = \$15{,}708{,}000 \qquad (11.4)$$

The propulsion maintenance saving of the electric aircraft relative to the conventional is:

$$\Delta C_{prop,20yr} = 107{,}100{,}000 - 15{,}708{,}000 = \$91{,}392{,}000 \qquad (11.5)$$

This $91.4M saving over 20 years arises from the fundamental simplicity of electric motors versus gas turbines. The CFM56-5B contains approximately 25,000 parts including single-crystal turbine blades operating at temperatures exceeding 1,000°C, whereas an equivalent PMSM contains fewer than 100 parts with no combustion, no hot section, and no oil system.

Airframe Maintenance: Build-Up
The component overhaul rate includes hydraulic actuators, flight control computers, avionics, and other aircraft systems.

Parameter	Symbol	Value
Service life	T	20 years
Flights per day	n_{day}	6
Operating days per year	d_{yr}	340
Flights per year	$N_{\text{yr}} = n_{\text{day}} \times d_{\text{yr}}$	2,040
Mission time (from sizing model)	t_{miss}	3.0 hours
Turnaround / taxi time	t_{turn}	0.5 hours
Block time per flight	$t_{\text{block}} = t_{\text{miss}} + t_{\text{turn}}$	3.5 hours
Annual block hours	$\text{FH}_{\text{yr}} = N_{\text{yr}} \times t_{\text{block}}$	7,140 hours
Total flights (20 yr)	$N_{\text{total}} = N_{\text{yr}} \times T$	40,800
Total block hours (20 yr)	FH_{total}	142,800 hours

Table S9 – Fleet utilisation assumptions

Cycle life	Packs	Mfg (Mt)	Total (Mt)	Saving vs neo
750	55	84.89	374.65	37.9%
1 000	41	63.28	353.04	41.5%
1 500	28	43.24	333.00	44.8%
2 000	21	32.44	322.20	46.6%
3 000	14	21.63	311.39	48.4%

Table S10: Sensitivity of 20 year life cycle CO_2 saving to battery cycle life evaluated on the world average grid.

Parameter	Symbol	Value	Source
Fuel burn (700 nm)	m_{fuel}	5,500 kg	ICAO CO_2 databank (ICAO, 2024)
Jet-A1 price	p_{fuel}	\$0.85/kg	IATA Fuel Monitor, 2024 avg (IATA, 2024)

$C_{\text{fuel}} = m_{\text{fuel}} \times p_{\text{fuel}} = 5{,}500 \times 0.85 = \$4{,}675$ per flight

$C_{\text{fuel,yr}} = C_{\text{fuel}} \times N_{\text{yr}} = 4{,}675 \times 2{,}040 = \$9{,}537{,}000$ per year

$C_{\text{fuel,20yr}} = C_{\text{fuel,yr}} \times 20 = \$190{,}740{,}000$

Table S11: Fuel cost calculation – A320ceo

Parameter	Symbol	Value	Source
Battery pack capacity	E_{batt}	20,848 kWh	Sizing model (with reserves)
Typical mission energy	E_{mission}	13,191 kWh	Sizing model (no reserves used)
Charging efficiency	η_{charge}	0.92	Grid-to-battery (NREL, 2016)
Grid energy per flight	E_{grid}	14,338 kWh	$E_{\text{mission}}/\eta_{\text{charge}}$
Electricity price	p_{elec}	\$0.10/kWh	Eurostat industrial rate (Eurostat, 2024)
LOX consumption	m_{O_2}	1,613 kg	Sizing model (converged)
LOX price	p_{LOX}	\$0.15/kg	Bulk cryogenic (CryoGas International, 2024)

$C_{\text{elec}} = E_{\text{grid}} \times p_{\text{elec}} = 14{,}338 \times 0.10 = \$1{,}434$ per flight

$C_{\text{LOX}} = m_{O_2} \times p_{\text{LOX}} = 1{,}613 \times 0.15 = \$242$ per flight

$C_{\text{energy,elec}} = C_{\text{elec}} + C_{\text{LOX}} = \$1{,}676$ per flight

$C_{\text{energy,elec,yr}} = 1{,}676 \times 2{,}040 = \$3{,}419{,}040$ per year

$C_{\text{energy,elec,20yr}} = \$68{,}380{,}800$

Table S12: Energy cost calculation – electric aircraft

Item	Cost	Interval	Rate (\$/EFH)	Source
Performance restoration	\$3.5M	15,000 EFH	233	Aircraft Commerce (2018*a*)
LLP set replacement	\$1.7M	20,000 EFC*	113†	Aircraft Commerce (2018*a*)
On-wing line maintenance	—	Continuous	30	Industry average
Per-engine total	—	—	**376**	—
Two-engine total	—	—	**752**	—

*EFC = engine flight cycle; †converted to \$/EFH at 1.5 EFH/EFC (short-haul ratio)

Table 13: CFM56-5B engine MRO cost build up (per engine)

Item	Interval	Rate (\$/FH)	Basis
Bearing replacement (8 motors)	30,000 FH	20	\$75k per set, amortised
Winding inspection / rewinding	50,000 FH	15	Industrial motor practice
Power electronics servicing	10,000 FH	25	SiC inverter capacitor life
Propeller / fan blade inspection	5,000 FH	20	Composite blade NDT
Motor system total	—	**80**	—
LOX system (dewars, valves)	Continuous	30	Cryogenic plant data
Propulsion total	—	**110**	—

Table S14: Electric motor maintenance cost build up (all 8 motors)

Check	Cost (USD)	Interval (FH)	Amortised ($/FH)
A-check	20,000	750	27
C-check	225,000	7,500	30
D-check (heavy)	5,000,000	30,000	167
Component overhaul	Continuous	—	180
Landing gear OH	700,000	10 yr	10
APU overhaul	250,000	6,000	42
Miscellaneous	—	—	44
Total	—	—	**500**

Table S15 – A320 airframe maintenance cost build up (conventional)

The environmental control system. The D-check at $5M dominates the amortised rate at $167/FH.
For the electric aircraft, the APU line item is removed (no APU installed), fuel system inspections are eliminated, and battery thermal management / high-voltage harness inspections are added. The net rate is reduced to $400/FH:

$$C_{af,conv} = 500 \times 3.5 = \$1{,}750 \text{ per flight} \qquad (11.6)$$

$$C_{af,elec} = 400 \times 3.5 = \$1{,}400 \text{ per flight} \qquad (11.7)$$

Battery Replacement: Cycle Life Sensitivity

The battery replacement cost per flight is:

$$C_{batt} = \frac{E_{batt} \times p_{batt}}{N_{cycle}} \qquad (11.8)$$

where E_{batt} is the pack capacity (kWh), p_{batt} is the pack-level cost ($/kWh), and N_{cycle} is the cycle life.
The sensitivity to these parameters is: At the pessimistic extreme (800 cycles, $150/kWh), battery replacement costs $3909/flight and the electric aircraft's per-flight saving over the conventional reduces from $3418 to $1594, still a 12% saving. At the optimistic extreme (2500 cycles, $100/kWh), per-flight battery cost falls to $834 and the saving increases to $4669 or 35%. The electric aircraft maintains a cost advantage across all scenarios examined, but the range is substantially wider than with a smaller battery: cycle life is the single largest sensitivity in the cost model and the principal economic risk for the design.

Fixed Costs

The acquisition costs are $110M (conventional, A320ceo list price) and $140M (electric, reflecting a technology premium for a novel aircraft type and Li-air battery pack..The electric aircraft's higher insurance reflects both increased hull value and an additional risk premium for novel propulsion systems in early service, typically 1-2% of hull value per year (ICAO, 2019; IATA, 2020)..Both premiums are expected to decrease as the fleet matures.

Fuel Price Sensitivity

The break-even fuel price is the jet fuel price at which the conventional aircraft matches the electric aircraft's per-flight cost.
This can be found by equating the total per-flight costs:

$$\begin{aligned} m_{fuel} \times p^*_{fuel} + C_{eng,conv} + C_{af,conv} + C_{crew} + C_{apt,conv} \\ = C_{energy,elec} + C_{prop,elec} + C_{af,elec} + C_{batt} + C_{crew} + C_{apt,elec} \end{aligned} \qquad (11.9)$$

Substituting values and solving for p^*_{fuel}:

$$5500 \times p^*_{fuel} + 2625 + 1750 + 1400 + 2924 = 1676 + 385 + 1400 + 2085 + 1400 + 3010$$

$$5500 \times p^*_{fuel} + 8699 = 9956$$

$$5500 \times p^*_{fuel} = 1257$$

$$p^*_{fuel} = \$0.229/\text{kg} \qquad (11.10)$$

The electric aircraft is cheaper per flight at any fuel price above $0.23/kg.
Jet fuel has not fallen below $0.30/kg in the modern era (and has averaged closer to $0.85/kg in recent years), so the electric aircraft maintains a per-flight cost advantage across all historically observed fuel prices.

The margin is tighter than would be the case with a smaller battery pack: the dominant offsetting costs are battery replacement ($2085/flight at 1500 cycles, equivalent to roughly $0.38/kg of jet fuel) and electricity for charging ($1434/flight, equivalent to roughly $0.26/kg).
At $1.50/kg fuel (the approximate 2022 peak during the Ukraine crisis), the electric aircraft saves 41% per flight, a difference of $6,993, or $14.3M per year per aircraft.

20-Year Cost Build-Up: Complete Calculation
The 20-year total for each aircraft is the sum of variable costs (scaled by flight count or flight hours) and fixed costs (scaled by years).
Note that crew cost per flight differs between the two configurations because the electric aircraft's lower cruise Mach (0.5 vs 0.78) increases block time from 1.83 hr to 2.33 hr at the same $400/FH crew rate, giving $733/flight conventional and $933/flight electric.

$$
\begin{aligned}
C_{20yr,conv} &= N_{total} \times (C_{fuel} + C_{crew,conv} + C_{apt,conv}) \\
&+ FH_{total} \times (R_{eng} + R_{af,conv}) \\
&+ T \times C_{fixed,conv} \\
&= 40{,}800 \times (4{,}675 + 733 + 2{,}924) \\
&+ 142{,}800 \times (750 + 500) \\
&+ 20 \times 6{,}300{,}000 \\
&= 339{,}945{,}600 + 178{,}500{,}000 + 126{,}000{,}000 \\
&= \$644{,}445{,}600 \qquad (11.11)
\end{aligned}
$$

$$
\begin{aligned}
C_{20yr,elec} &= N_{total} \times (C_{energy} + C_{batt} + C_{crew,elec} + C_{apt,elec}) \\
&+ FH_{total} \times (R_{prop,elec} + R_{af,elec}) \\
&+ T \times C_{fixed,elec} \\
&= 40{,}800 \times (1{,}676 + 2{,}085 + 933 + 3{,}010) \\
&+ 142{,}800 \times (110 + 400) \\
&+ 20 \times 8{,}000{,}000 \\
&= 314{,}323{,}200 + 72{,}828{,}000 + 160{,}000{,}000 \\
&= \$547{,}151{,}200 \qquad (11.12)
\end{aligned}
$$

Cycle Life	Pack Cost ($/kWh)	Per Flight ($)	Per Year ($k)	20-Year ($M)
800	150	3,909	7,974	159.5
800	100	2,606	5,316	106.3
1,500	150	2,085	4,253	85.1
1,500	100	1,390	2,836	56.7
2,500	150	1,251	2,552	51.0
2,500	100	834	1,701	34.0

Table S16 – Battery replacement cost sensitivity

Item	Conventional ($/yr)	Electric ($/yr)
Insurance (hull + liability)	800,000	1,000,000
Depreciation (straight-line, 20 yr)	5,500,000	7,000,000
Total fixed	6,300,000	8,000,000

Table S17 – Annual fixed costs

$$\boxed{\begin{aligned}\Delta C_{20\text{yr}} &= 644{,}445{,}600 \\ &\quad - 547{,}151{,}200 \\ &= \$97{,}294{,}400 \\ &\quad (-15.1\%)\end{aligned}} \tag{.11.13}$$

Fuel Price ($/kg)	Conv. ($/flight)	Elec. ($/flight)	Saving (%)
0.50	11,449	9,956	13%
0.70	12,549	9,956	21%
0.85	13,374	9,956	26%
1.00	14,199	9,956	30%
1.20	15,299	9,956	35%
1.50	16,949	9,956	41%

Table S18 – Per flight cost comparison at varying fuel prices (electricity held at $0.10/kWh)

Parameter	Unit	Value	Source
Jet-A1 price	$/kg	0.85	IATA Fuel Monitor (IATA, 2024)
Electricity price	$/kWh	0.10	Eurostat (Eurostat, 2024)
LOX price	$/kg	0.15	Industrial gas market data (International Energy Agency, 2019; Air Products, 2023)
Charging efficiency	—	0.92	NREL (NREL, 2016)
CFM56-5B MRO rate	$/EFH	750	Aircraft Commerce (Aircraft Commerce, 2018*b*)
Electric motor MRO	$/FH	80	NASA electrified propulsion studies (NASA, 2023*a*)
LOX system maintenance	$/FH	30	Cryogenic systems studies (NASA, 2018)
Airframe MRO (conv.)	$/FH	500	EUROCONTROL (EUROCONTROL, 2025)
Airframe MRO (elec.)	$/FH	400	Assumed (reduced systems)
Battery pack cost	$/kWh	150	BNEF (BloombergNEF, 2024)
Battery cycle life	cycles	1,500	Luntz and McCloskey (2014); Lu et al. (2017*a*)
Flight crew	$/FH	400	EUROCONTROL (EUROCONTROL, 2025)
Landing fee	$/t	8	EUROCONTROL (EUROCONTROL, 2025)
Navigation charge	$/flight	800	EUROCONTROL (EUROCONTROL, 2025)
Ground handling	$/flight	1,500	IATA/industry data (IATA, 2020)
Acquisition (conv.)	$M	110	Airbus (Airbus, 2023*c*)
Acquisition (elec.)	$M	140	Estimated (technology premium)
Insurance (conv.)	$/yr	800,000	Typical 1–2% hull value (ICAO, 2019)
Insurance (elec.)	$/yr	1,000,000	Assumed higher hull value

Table S19 – Cost comparison with sources.

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