

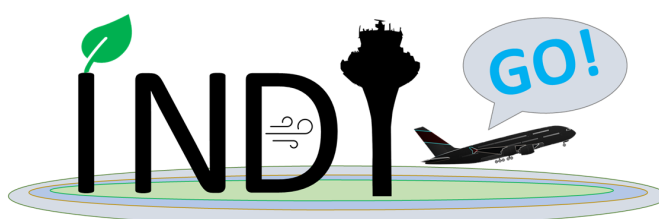
RESEARCH PROJECT MANAGEMENT

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Pareto-front of optimal DHEP-LARW aircraft and margins including selected nominal aircraft

Project: 101096055

**INtegration and Digital demonstration of low-
emission aIrcraft technoloGies and airport
Operations**

INDIGO

Work package: WP5 - Airframe-propulsion optimal integration
under uncertainty

Deliverable: D5.2



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Acronyms and abbreviations

Abbreviation	Meaning
AD	Actuator disc
AL	Actuator line
AoA	Angle of attack
AR	Aspect ratio
BF	Block fuel
BFL	Balanced field length
BPA	Basic probability assignment
BPF	Blade-passing frequency
CAA	Computational aeroacoustics
CCW	Counterclockwise
CFD	Computational fluid dynamics
DEP	Distributed electric propulsion
DHEP	Distributed hybrid-electric propulsion
DOE	Design of experiments
DST	Dempster–Shafer theory
EAS	Equivalent airspeed
ED	Energy density (battery specific energy in this report)
EI	Emission index
EPNL	Effective perceived noise level
EPU	Electric propulsion unit
EPS	Electric propulsion system (MDO notation)
ESC	Electronic speed controller
FB900	Fuel burned below 900 m
FC	Failure case
FEM	Finite-element model
FSI	Fluid-structure interaction
GFEM	Global finite-element model
HF	Hybridisation factor
HS	High-speed cruise case evaluated using the light-Euler model
HSV	High-speed cruise case evaluated using the RANS model
HTP	Horizontal tailplane
IB	Inboard
L_{AE}	A-weighted sound exposure level
LAQ	Local air quality
LARW	Large-aspect-ratio wings
LB	Lower bound
LFL	Landing field length
MDA	Multidisciplinary design analysis
MDAO	Multidisciplinary design analysis and optimisation
MDO	Multidisciplinary design optimisation
MTOW	Maximum take-off weight
OB	Outboard
OEW	Operating empty weight
OPR	Overall pressure ratio
PNWT	Pressure Neutral Wind Tunnel
POD	Proper orthogonal decomposition
PSD	Power spectral density
PSFC	Power-specific fuel consumption
RANS	Reynolds-averaged Navier–Stokes
RIX	Riga International Airport
RPM	Revolutions per minute
SA	Spalart–Allmaras turbulence model
SBW	Strut-braced wing
SLSQP	Sequential Least Squares Programming

Abbreviation	Meaning
SM	Static margin
SOC / SoC	State of charge
TOFL	Take-off field length
TOPAZ	Tool for Optimizing Powerplants and Aircraft with Zero-Emissions
TOW	Take-off weight
TRL	Technology readiness level
UB	Upper bound
UMDO	Uncertainty-based multidisciplinary design optimisation
URANS	Unsteady Reynolds-averaged Navier–Stokes
VTP	Vertical tailplane
WP	Work package
XDSM	Extended design structure matrix

1 Introduction

The development of distributed hybrid-electric propulsion aircraft with large-aspect-ratio wings (DHEP-LARW) requires the integration of several disciplines, including aerodynamics, propulsion, structures, aircraft performance, noise, and emissions. These disciplines involve competing objectives and important sources of uncertainty, particularly when low-TRL technologies are considered. The final aircraft configuration must therefore be selected not only according to its nominal performance, but also according to its robustness with respect to technological uncertainty.

This deliverable presents the Pareto front obtained from the multidisciplinary design optimisation process developed within Work Package 5. The Pareto front identifies the non-dominated aircraft configurations and illustrates the trade-offs between the selected environmental and performance objectives. It provides a basis for comparing powertrain solutions on the selected INDi2 reference airframe according to their fuel consumption, emissions, noise, aircraft performance and robustness. For the purposes of this deliverable, the INDi2 configuration is therefore retained as the selected nominal reference airframe and remains fixed throughout the optimisation campaigns, while the powertrain, including the propeller design, is re-optimised.

The uncertainty treatment focuses mainly on battery specific energy, which is represented using Dempster–Shafer theory. Monte Carlo sampling is used to propagate this uncertainty through Gaussian-process surrogate models of the aircraft responses. The resulting samples are used to estimate the expected value and standard deviation of quantities such as block fuel and fuel consumed below 900 m, which are then included in the uncertainty-based optimisation problem.

The Pareto-optimal solutions are subsequently examined to identify the main trade-offs between environmental performance, technological uncertainty, and aircraft-level constraints. The single- and multi-trajectory studies produce the INDi3s and INDi3m families, respectively, by re-optimising the powertrain, including the propellers, on the fixed INDi2 airframe. These families represent alternative solutions under the investigated mission, noise and battery-uncertainty assumptions, rather than a single final aircraft obtained by reintegrating all completed discipline models. The high-fidelity verification in Section 7 assesses the INDi2 reference configuration.

2 Aircraft Concept

Figure 1 provides a graphical overview of the aircraft concept adopted within the INDIGO project. The configuration is based on a strut-braced-wing (SBW) architecture, in which the primary wing is supported by a lower structural member, hereafter referred to as the strut [1].

The wing planform is characterised by a straight leading edge and a trailing edge that tapers from approximately 50% of the semispan towards the wing tip. The NASA Langley LS(1)-0413, also designated GA(W)-2, is adopted as the reference wing airfoil. The strut has a constant-chord planform and uses a symmetric NACA 0024 airfoil. The wing–strut junction is positioned at 50% of the wing semispan.

The aircraft employs a distributed hybrid-electric propulsion architecture combining inboard parallel-hybrid and outboard series-hybrid propulsion systems. The inboard system incorporates a turboshaft located at 25% of the wing semispan. Outboard of the wing–strut junction, four electrically driven

propulsors are distributed at equal intervals between the junction and the wing tip. A detailed description of the propulsion architecture and the corresponding powerplant modelling approach is provided in Section 4.3.4.

The main geometric and configuration parameters defining the lifting system are summarised in Table 1.

These parameters were established following preliminary conceptual-design optimisation activities performed during the initial stages of the project [2, 3].

Table 1: INDIGO wing and strut geometrical parameters.

Component	Variable [Units]	Meaning	Value
Wing	AR	aspect ratio	21.62
	λ_{wing}	taper ratio	0.42
	t/c_{root} [%]	thickness to chord ratio at root	18.0
	t/c_{tip} [%]	thickness to chord ratio at tip	9.0
	S_W [m^2]	wing planform area	122.4
Strut	c_{strut} [m]	chord	1.60



Figure 1: INDIGO project strut-braced wing configuration.

3 Noise mitigation techniques activities - UBR and DLR

Noise emitted from the DEP system is different from traditional aircraft configurations, usually considered to be quieter because of the multi-propulsor design dividing up the total thrust required and hence reducing the tip speed of the propeller [4, 5]. Understanding the physical mechanisms that govern noise generation is essential for UAM systems, particularly under hover and forward-flight conditions. Previous works have been examining the noise characteristics of all components in the DEP system, including the propeller-propeller interaction [6, 7] and propeller-wing interaction [8, 9, 10]. De Paola et al. [10] investigated the relationship between the tonal noise at the first blade passing frequency (BPF) and the angles of attack (AoAs) of the wing for a three-propeller DEP configuration. They observed that apart from the noise level at the first BPF, there is an apparent increment in both the tonal and broadband noise components on the wing pressure side when the wing angle of attack increases, primarily because of the interaction between the rotor wakes and the wing surface. Turhan et al. [11] found that, for a leading-edge-mounted two-propeller DEP configuration, the farfield noise depends strongly on the propeller advance ratio. At a constant rotational speed of 8000 rpm, increasing the advance ratio reduced the farfield noise. They also noted that a smaller tip clearance of 1% of the propeller diameter generated slightly higher sound pressure level (SPL) at the BPF than a larger clearance of 4%. Based on these studies, propeller phase-synchronisation has been chosen as an effective noise reduction method for the INDIGO aircraft. Examining the noise reduction effectiveness of phase-controlled propellers is achieved through lab testing inside an anechoic wind tunnel at the University of Bristol, and full aircraft numerical modelling using DLR's Computational AeroAcoustics (CAA) model.

3.1 Experimental works on propeller phase-synchronisation.

The aeroacoustic experiments were conducted in the Pressure Neutral Wind Tunnel (PNWT) at the Lawson Wind Tunnel Laboratory, University of Bristol. The PNWT is designed as an open-jet closed-loop facility with a nozzle exit measuring 1 m in length and 0.7 m in width. The nozzle and the test

section are located in an acoustically-treated chamber of $4.1 \times 4.5 \times 4$ m, with foam wedges to minimise acoustic reflections. This aeroacoustic facility has been utilised in numerous previous aeroacoustic studies, including [12, 13, 14]. The open-jet nozzle is capable of producing steady freestream velocities up to $V_\infty = 35$ m/s, with a turbulence intensity of approximately 0.2%.

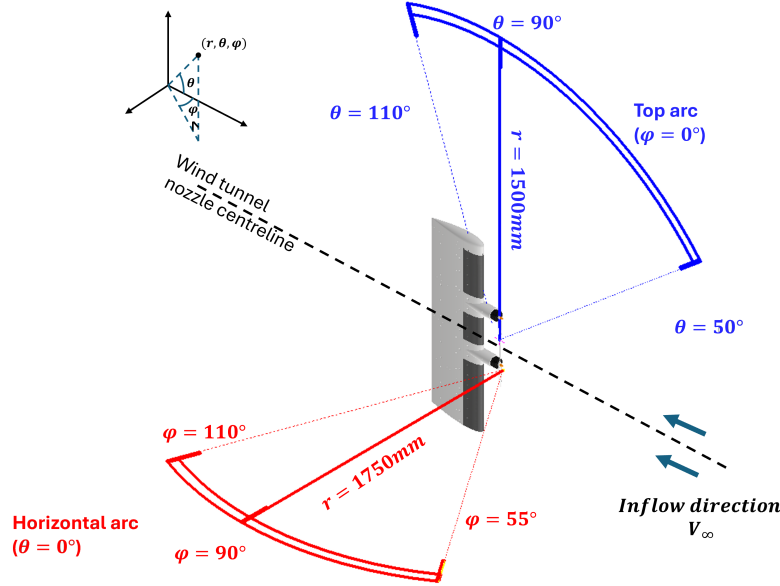


Figure 2: Schematic of the twin-propeller DEP setup and microphone arc layouts.

Figure 2 demonstrates the setup for the twin-propeller DEP configuration. Two microphone arcs were utilised in this work, with GRAS 40PL-10 microphones calibrated before and after each test using a GRAS 42AA pistonphone calibrator. The microphones featured a bandwidth of 10 Hz to 20 kHz, a dynamic range of 142 dB, and an accuracy of ± 1 dB. The top arc (indicated in blue in Fig. 2) measured the noise directivity in the spanwise plane of the DEP setup. Located in the plane of $\varphi = 0^\circ$ in the spherical coordinate system, it consisted of 13 microphones with observer angles between $\theta = 50^\circ$ and $\theta = 110^\circ$ in 5° increments. The microphone located at $\theta = 90^\circ$ was directly aligned with the propellers. The horizontal arc (indicated in red in Fig. 2) measured noise emission in the plane of $\theta = 0^\circ$ at the middle of the two propellers, covering observer angles between $\varphi = 55^\circ$ and $\varphi = 110^\circ$, also in 5° increments. Data were sampled at 2^{15} Hz for $t = 32$ s. The SPL is determined using the following equation:

$$SPL = 10 \cdot \log_{10} \left(\frac{\phi_{pp}}{p_{ref}^2} \right), \quad (1)$$

where ϕ_{pp} is the power spectral density (PSD) of the measured acoustic pressure, and the reference acoustic pressure p_{ref} is set as $20 \mu\text{Pa}$.

The centre of the wing was placed in the middle of the open jet nozzle and positioned approximately 500 mm downstream from the nozzle exit, with the middle of the two propellers aimed at the nozzle centreline. The wing was designed with a NACA 0018 profile and a chord of $c = 0.3$ m and a span of $L = 0.9$ m. It was manufactured from 6000-series aluminium alloy on a 4-axis CNC machine. The propulsion unit included an electronic speed controller (ESC) to drive a brushless motor with a propeller, and incremental encoders were used to monitor and control the motor's rotational speed and the relative phase angle between the two propellers. The phase control algorithm was built in-house in National Instruments LabVIEW, and readers are referred to [15] for further details on the phase controller design and performance. Two T-Motor 2.2 kW AT4125 540 kV motors were utilised. The tests employed two identical two-bladed constant-pitch propellers with diameters of $D = 9$ inches (228.6 mm) and a pitch-to-diameter ratio of $P/D = 1$. These propellers, manufactured by Mejzlik (<https://www.mejzlik.eu/>), were made from carbon fibre/epoxy composite material and featured a Clark-Y aerofoil profile.

The relative phase angle $\Delta\psi$ between the two propellers is defined in Fig. 3. The inboard propeller served as the reference propeller; different relative phase angles $\Delta\psi$ were achieved by controlling the outboard propeller's blade position. Both propellers were rotating counterclockwise (CCW) when viewed from the front. The wing AoA was selected at four different positions: 0° , 5° , 10° , and 15° , and the

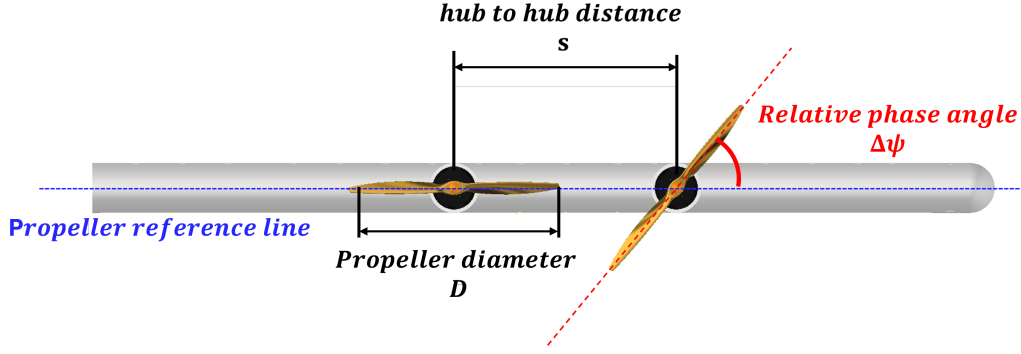


Figure 3: Propeller relative phase angle $\Delta\psi$ and the propeller separation s/D of the twin-propeller DEP system. Both propellers are rotating CCW when viewed from the front.

orientation of the AoA sweep is illustrated in Figure 4(a). The inflow speed, V_∞ , chosen for the current study was 12 m/s, while the propeller rotational speeds were 5000, 6000, and 9000 rpm. The corresponding advance ratio, J , can then be calculated as:

$$J = \frac{V_\infty}{nD} \quad (2)$$

where n is the rotational speed in revolutions per second, and D is the propeller diameter. The propeller separation is set to $s/D = 1.05$. Figure 4(b) illustrates the DEP test rig setup inside the aeroacoustic chamber.

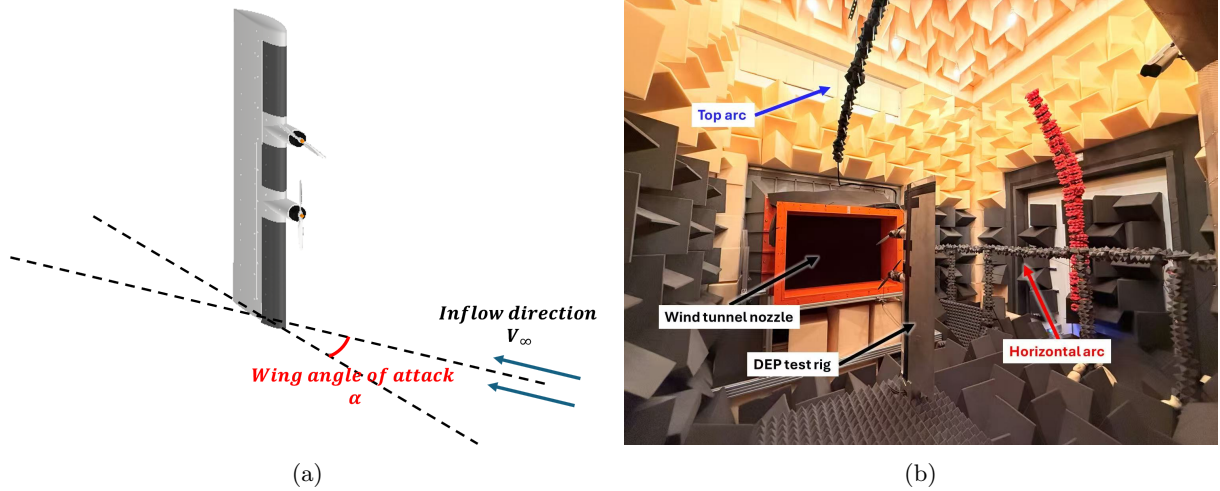


Figure 4: Twin-propeller DEP system utilised in this study: (a) wing angle of attack (AoA) α definition, and (b) setup inside the wind tunnel.

Figure 5 illustrates the tonal noise directivity patterns with propeller relative phase angles $\Delta\psi$ for all wing AoAs measured on the top arc at $J = 0.53$ and $J = 0.35$. At $J = 0.53$, as shown in the left column of Fig. 5, the tonal noise reduction of applying $\Delta\psi < 45^\circ$ is less than 5 to 7 dB compared to the 0° relative phase angle case at all four wing AoAs. The most promising tonal noise attenuation occurs at $\Delta\psi = 60^\circ$. At this relative phase angle, the effectiveness of tonal noise reduction does not significantly change with wing AoA sweeps, capable of providing around 12 to 15 dB reduction compared to the $\Delta\psi = 0^\circ$ case at all wing AoAs. For a relative phase angle of $\Delta\psi = 90^\circ$, a similar level of tonal noise reduction to $\Delta\psi = 60^\circ$ is achieved at AoA = 0° . However, as the wing AoA increases, the effectiveness of tonal noise reduction at $\Delta\psi = 90^\circ$ decreases compared to the $\Delta\psi = 60^\circ$ case, with noise levels becoming comparable to those at $\Delta\psi = 45^\circ$. On the other hand, as J decreases to 0.35, as shown in the right column of Fig. 5, the relative phase angle of $\Delta\psi = 90^\circ$ yields nearly the same noise level as $\Delta\psi = 0^\circ$. At higher wing AoAs, $\Delta\psi = 90^\circ$ provides almost no noise reduction compared to the $\Delta\psi = 0^\circ$ case. In contrast, both $\Delta\psi = 45^\circ$ and $\Delta\psi = 60^\circ$ provide greater noise reduction than the 0° phase configuration. At AoA = 0° , $\Delta\psi = 60^\circ$

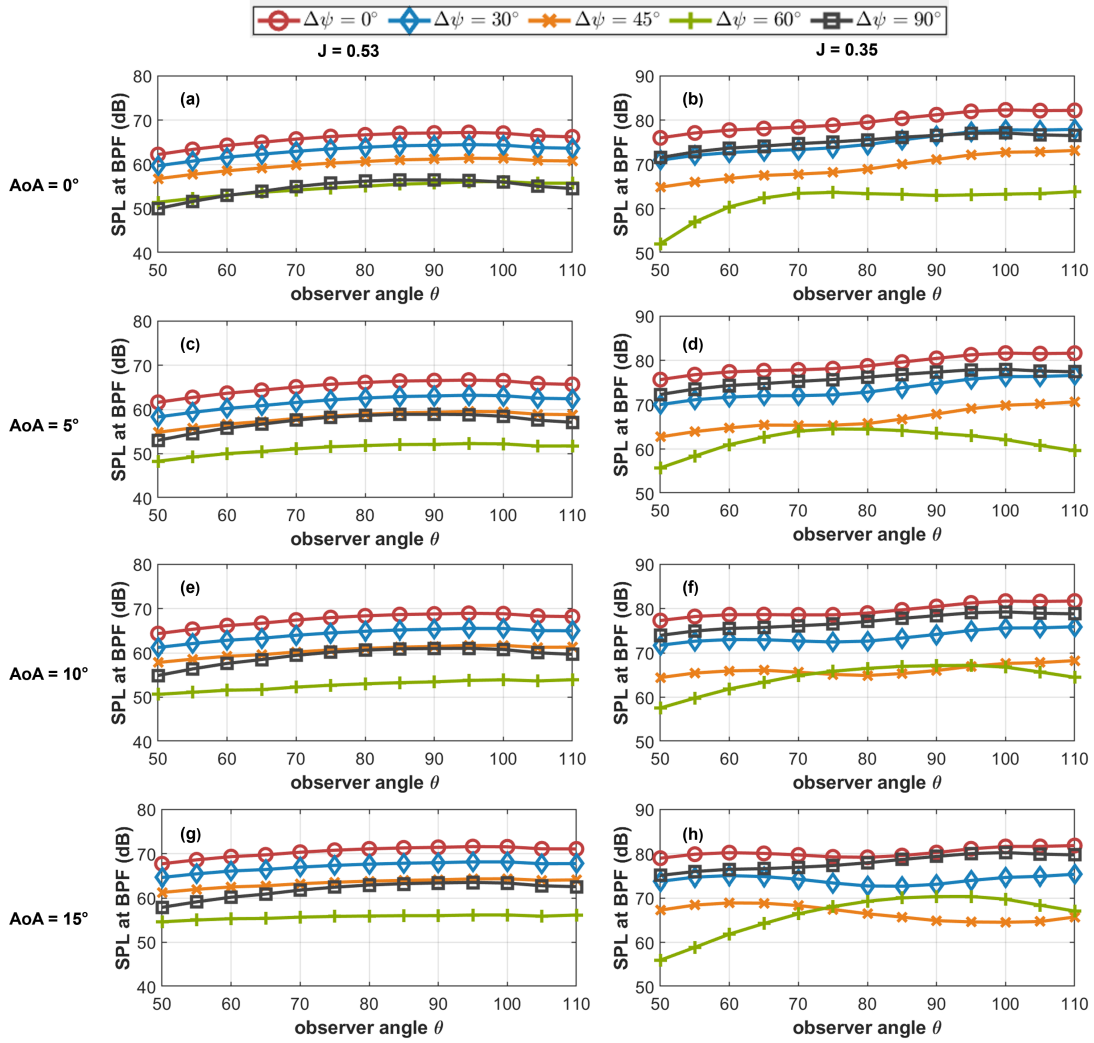


Figure 5: Directivity of the 1st BPF tonal noise for different propeller relative phase angles, $\Delta\psi$, measured on the top arc for $J = 0.53$ (left column) and $J = 0.35$ (right column). Results are shown for (a) and (b) AoA = 0° , (c) and (d) AoA = 5° , (e) and (f) AoA = 10° , and (g) and (h) AoA = 15° .

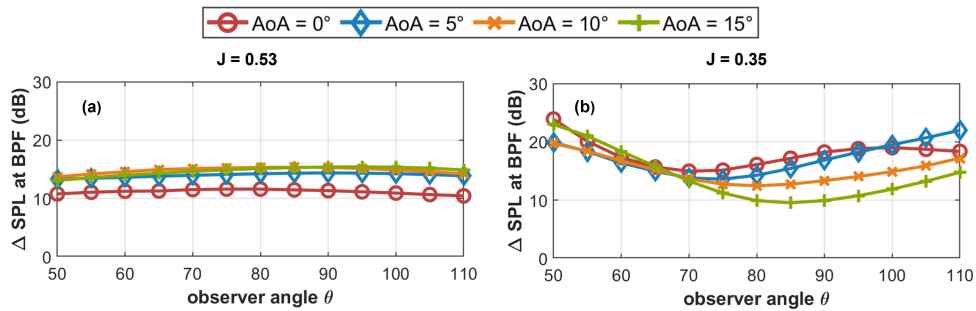


Figure 6: Directivity of the 1st BPF tonal noise difference, ΔSPL , between $\Delta\psi = 0^\circ$ and $\Delta\psi = 60^\circ$ measured on the top arc for different AoAs. Results are shown for (a) $J = 0.53$ and (b) $J = 0.35$.

achieves greater tonal noise reduction than $\Delta\psi = 45^\circ$ across all observer positions. However, as AoA increases, the effectiveness of tonal noise reduction for $\Delta\psi = 60^\circ$ gradually decreases. At AoA = 15° , $\Delta\psi = 60^\circ$ provides better noise reduction at observer positions $\theta < 75^\circ$, whereas $\Delta\psi = 45^\circ$ slightly outperforms $\Delta\psi = 60^\circ$ at larger observer angles.

Figure 6 presents a direct comparison of ΔSPL , defined as the 1st BPF tonal noise SPL difference of $(\Delta\psi = 0^\circ) - (\Delta\psi = 60^\circ)$, at $J = 0.53$ and $J = 0.35$ across all four wing AoAs. At the higher propeller advance ratio of $J = 0.53$, the effectiveness of tonal noise reduction achieved with $\Delta\psi = 60^\circ$ is largely

independent of both wing AoA and observer position along the microphone arc. At $\text{AoA} = 0^\circ$, the reduction in 1st BPF tonal noise is marginally lower than that observed at the other wing AoAs, while the remaining cases exhibit nearly identical noise reduction characteristics. As J decreases to 0.35, it is observed that for upstream observer positions ($\theta < 70^\circ$), the noise reduction achieved by applying $\Delta\psi = 60^\circ$ remains similar across all four AoAs. However, for observer positions beyond $\theta = 70^\circ$, the effectiveness of tonal noise reduction reduces at higher AoAs, exhibiting reduced tonal noise reduction capability relative to the lower AoA conditions.

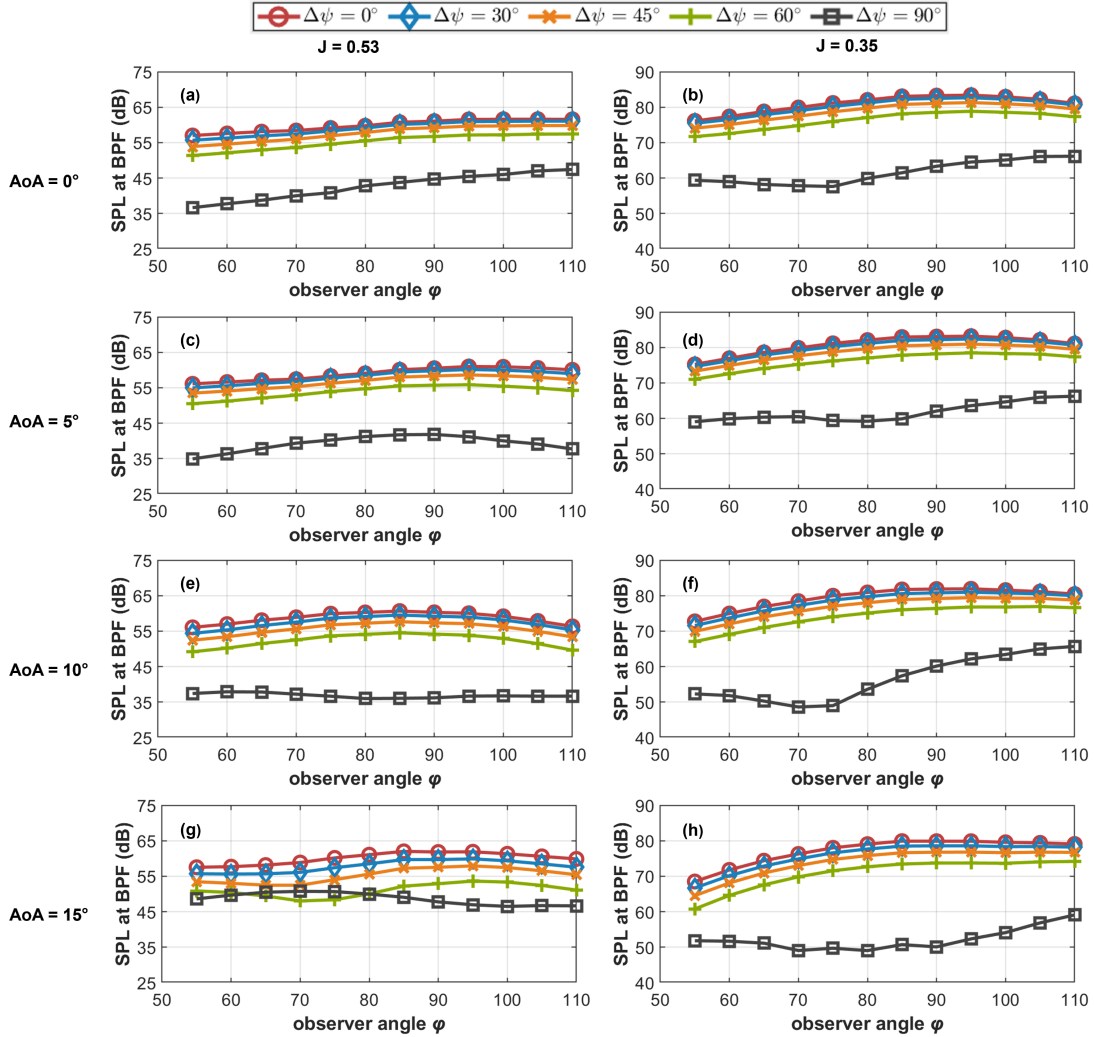


Figure 7: Directivity of the 1st BPF tonal noise for different propeller relative phase angles, $\Delta\psi$, measured on the horizontal arc for $J = 0.53$ (left column) and $J = 0.35$ (right column). Results are shown for (a) and (b) $\text{AoA} = 0^\circ$, (c) and (d) $\text{AoA} = 5^\circ$, (e) and (f) $\text{AoA} = 10^\circ$, and (g) and (h) $\text{AoA} = 15^\circ$.

Figure 7 shows the tonal noise directivity curves at the 1st BPF at different propeller relative phase angles at $J = 0.53$ and $J = 0.35$ on the horizontal arc. At $J = 0.53$, for wing $\text{AoA} \leq 10^\circ$, $\Delta\psi = 90^\circ$ exhibits outstanding performance in tonal noise reduction capability compared to other relative phase angles. The noise reduction of applying any phase angle of $\Delta\psi < 45^\circ$ is marginal compared to the $\Delta\psi = 0^\circ$ case, especially at wing $\text{AoA} = 0^\circ$. As the wing AoA increases from 0° to 10° , the SPL differences between $\Delta\psi = 90^\circ$ and $\Delta\psi = 0^\circ$ slightly increased for downstream observer positions $\phi > 90^\circ$. However, when the wing further pitches to $\text{AoA} = 15^\circ$, the effectiveness of applying 90° relative phase angle control drastically drops compared to lower AoAs. The right column of Fig. 7 shows the same concepts of tonal noise directivity at the lower advance ratio of $J = 0.35$. Similar to the high advance ratio case, tonal noise reduction for $\Delta\psi < 45^\circ$ is trivial compared to the $\Delta\psi = 0^\circ$ case. At this lower advance ratio, the effectiveness of applying 90° relative phase control to the two propellers enhances as wing AoA sweeps from 0° to 15° . The massive drop in noise-reduction effectiveness observed previously in the $J = 0.53$ case at $\text{AoA} = 15^\circ$ does not occur at this higher propeller rotational speed.

Figure 8 presents the directivity of the tonal noise difference, ΔSPL , at the first BPF between $\Delta\psi = 0^\circ$ and $\Delta\psi = 90^\circ$, measured on the horizontal arc for all wing AoAs at $J = 0.53$ and $J = 0.35$. At $J = 0.53$, the noise reduction at AoA = 0° and 5° remains comparable at upstream observer positions ($\varphi < 90^\circ$), with reductions of approximately 18–22 dB. As the observer angle increases toward downstream locations, the curves gradually diverge, with AoA = 5° achieving slightly higher reductions, reaching around 22–24 dB near $\varphi \approx 100^\circ$. The case at AoA = 10° exhibits the highest overall noise reduction across the mid-range observer angles ($\varphi \approx 65^\circ$ – 100°), peaking at approximately 24–25 dB. In contrast, AoA = 15° produces the lowest reduction among all AoAs at upstream locations, with values as low as 7–10 dB, which is consistent with the previously observed increase in SPL for $\Delta\psi = 90^\circ$ at AoA = 15° (see Fig. 7(g)). However, this case shows a gradual improvement toward downstream angles, reaching approximately 15 dB. When the advance ratio decreases to $J = 0.35$, the behaviour changes notably. For AoA = 0° and 5° , the noise reduction remains relatively uniform across all observer positions, with values around 15–23 dB. Unlike the higher advance ratio case, increasing the wing AoA to 15° leads to a substantial enhancement in noise reduction, particularly for observer positions beyond $\varphi \approx 70^\circ$, where reductions reach up to approximately 28–30 dB. Similarly, AoA = 10° also demonstrates strong reductions, with peak values of around 30–32 dB near $\varphi \approx 75^\circ$. These results indicate that the effectiveness of phase-based noise reduction is strongly dependent on both advance ratio and inflow conditions, with higher AoA becoming beneficial at lower advance ratios, particularly in upstream observer regions.

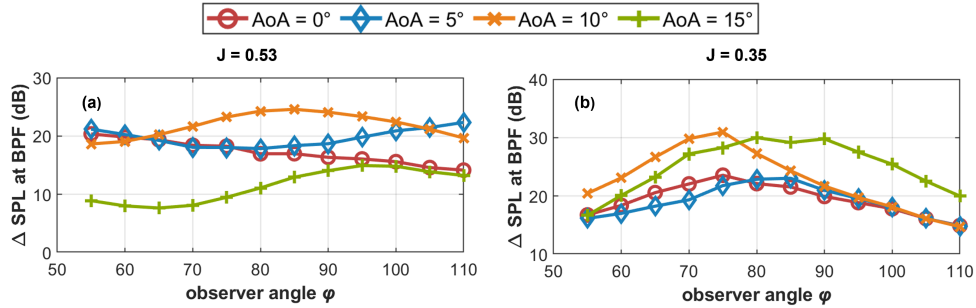


Figure 8: Directivity of the 1st BPF tonal noise difference, ΔSPL , between $\Delta\psi = 0^\circ$ and $\Delta\psi = 90^\circ$ measured on the horizontal arc for different AoAs. Results are shown for (a) $J = 0.53$ and (b) $J = 0.35$.

In summary, it has been proven experimentally that the 1st BPF tonal noise emission can be significantly reduced by applying appropriate relative phase angles under most operating conditions for two-bladed propellers. In the following section, the noise reduction effect of applying phase synchronisation to five-bladed propellers will be discussed experimentally and numerically. The validated numerical hybrid RANS-CAA model will then be utilised in computing the predicted noise reduction for the full scale INDIGO aircraft.

3.2 Numerical simulations for propeller phase-synchronisation on the full-scale INDIGO aircraft

Propeller phase-synchronisation was investigated as a potential noise mitigation strategy for the INDIGO aircraft. A preliminary validation of the hybrid RANS-CAA approach was first performed for two five-bladed phase-synchronised propellers to assess its capability to reproduce the associated change in SPL directivity. The cases considered are described in Fig. 9. Only the first BPF was considered for validation, since the levels of the second BPF in the experiments were much lower and close to the expected broadband-noise levels.

The results of the validation are shown in Fig. 10, Fig. 11, and Fig. 12. The experiments for five-bladed propellers show a noticeable SPL variation for adjacent directivity angles, which remains difficult to interpret. It can be seen that in general the hybrid RANS-CAA method can capture the directivity trends on average, which confirms the capability of the model of capturing the levels of interest for a subsequent assessment on the INDIGO aircraft.

The assessment of the benefits of the selected propellers phase-synchronisation angle of 36° on the INDIGO powertrain was performed for the mid-outboard propellers. The three mid-propellers were chosen since they are located on the same plane, at the same distance from the wing leading edge, and therefore most suited for phase control applications. A simulation along the trajectory identified for validation of the hybrid RANS-CAA approach with RIX data in WP6 was considered, for the most

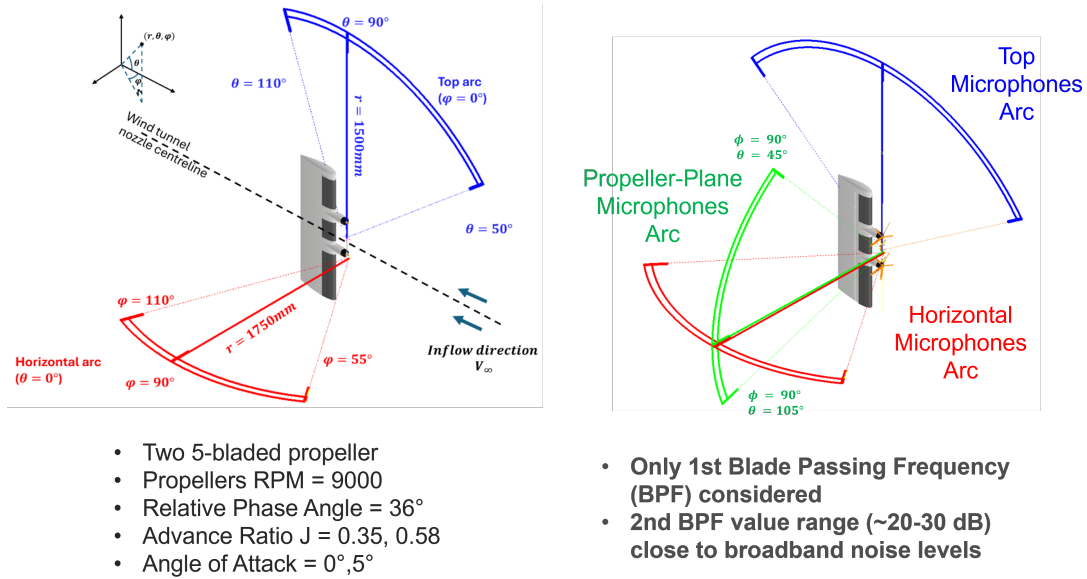


Figure 9: Validation cases of phase-synchronised propellers experimental results of University of Bristol with mid-fidelity hybrid RANS-CAA approach of DLR.

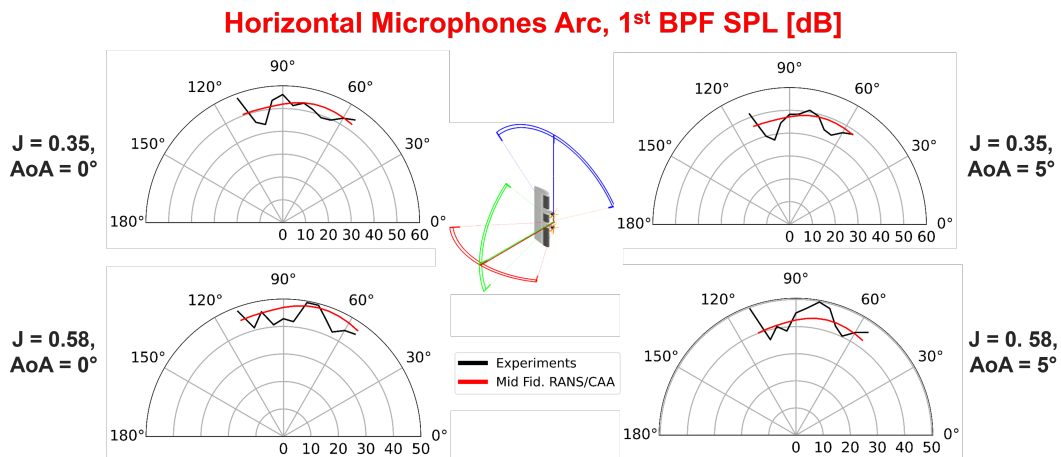


Figure 10: Phase-synchronised propellers validation of the hybrid RANS-CAA mid-fidelity approach of DLR with UoB experiments in terms of SPL directivity for the horizontal microphone arc.

promising noise-optimized operating conditions provided by UC3M. The trajectory considered is the one described in Fig. 13, that consists of two main segments (blue and cyan-colored segments), excluding ground roll (red-colored segment), with main parameters included in Table 2.

Table 2: RIX trajectory segments description.

Trajectory Segment	Altitude [ft]	Gradient [%]	Vertical Speed [ft/min]
Blue	0 - 1000	17.4	2562
Cyan	1000 - 3000	14.9	2220

A preliminary assessment with IMMIS+ was conducted, to select the most representative point of the trajectory for the corresponding CAA simulation, for which the mid-points of the blue-colored and cyan-colored segment were considered to be flown along the full trajectory to compute the EPN levels for a microphone located at the lateral location ‘NMT2’. Using the available analytical source models of IMMIS+, the inevitably simplified surrogate modeling approach pursued within the analysis of this work package can be estimated to provide prediction errors less than 0.5 EPNLdB for the cyan-colored segment, so the reference aircraft operational parameters for the CAA simulation were selected from the

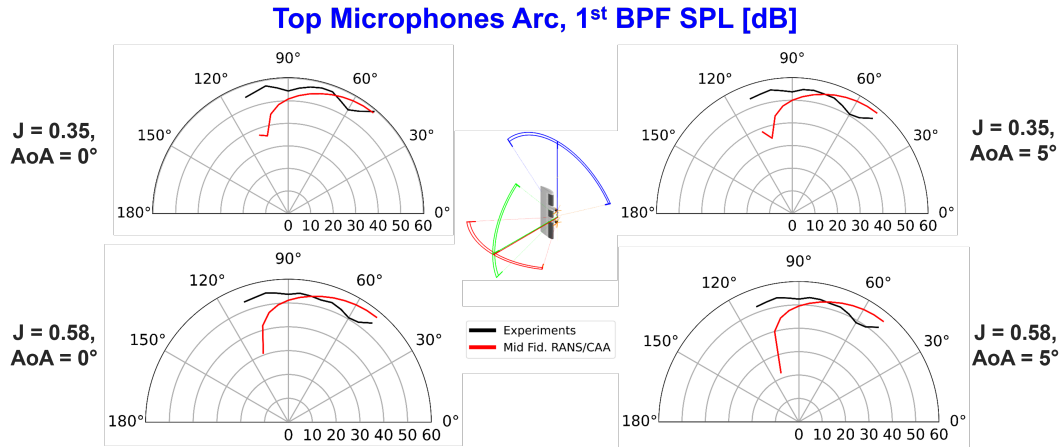


Figure 11: Phase-synchronised propellers validation of the hybrid RANS–CAA mid-fidelity approach of DLR with UoB experimental data in terms of SPL directivity for the top microphone arc.

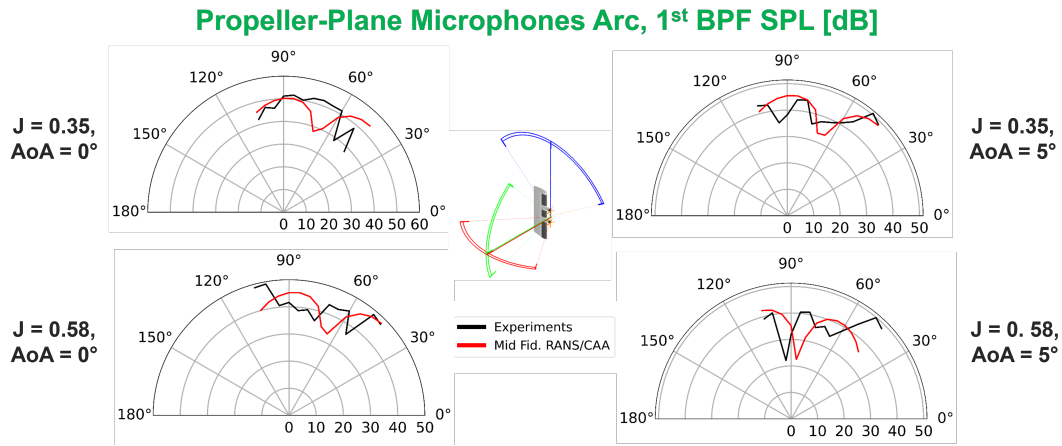


Figure 12: Phase-synchronised propellers validation of the hybrid RANS–CAA mid-fidelity approach of DLR with the UoB experiments in terms of SPL directivity for the propeller-plane microphone arc.

mid-point of the cyan-colored segment, taking into account also that the cyan-colored segment is the segment in which the aircraft is flying over the selected microphone. The aircraft operational parameters for the reference point are included in Table 3.

Table 3: Reference aircraft operational parameters of representative trajectory point.

Altitude [m]		True Air Speed [m/s]		Air Density [kg/m³]	
602.40		76.52		1.15	
Temperature [K]		Flight Path Angle [°]		Angle of Attack [°]	
284.23		8.47		10.17	
Air Speed of Sound [m/s]		Powertrain Thrust [N]		Inboard Propeller Thrust [N]	
338.55		146482.04		21911.63	
Outboard Thrust [N]	Propeller	Inboard Propeller Mach nr.	Tip	Outboard Propeller Mach nr.	Tip
12832.35		0.37		0.4	
Inboard Propeller RPM		Outboard Propeller RPM			
486.31		643.83			

The comparison of the INDIGO aircraft simulated results at the NMT2 microphone eventually was performed in terms of Sound Exposure Level (LAE), with a recorded A-Weighted value of 83.3 dBA corresponding to the most frequent Airbus A320 RIX trajectory. Two CAA simulations were considered,

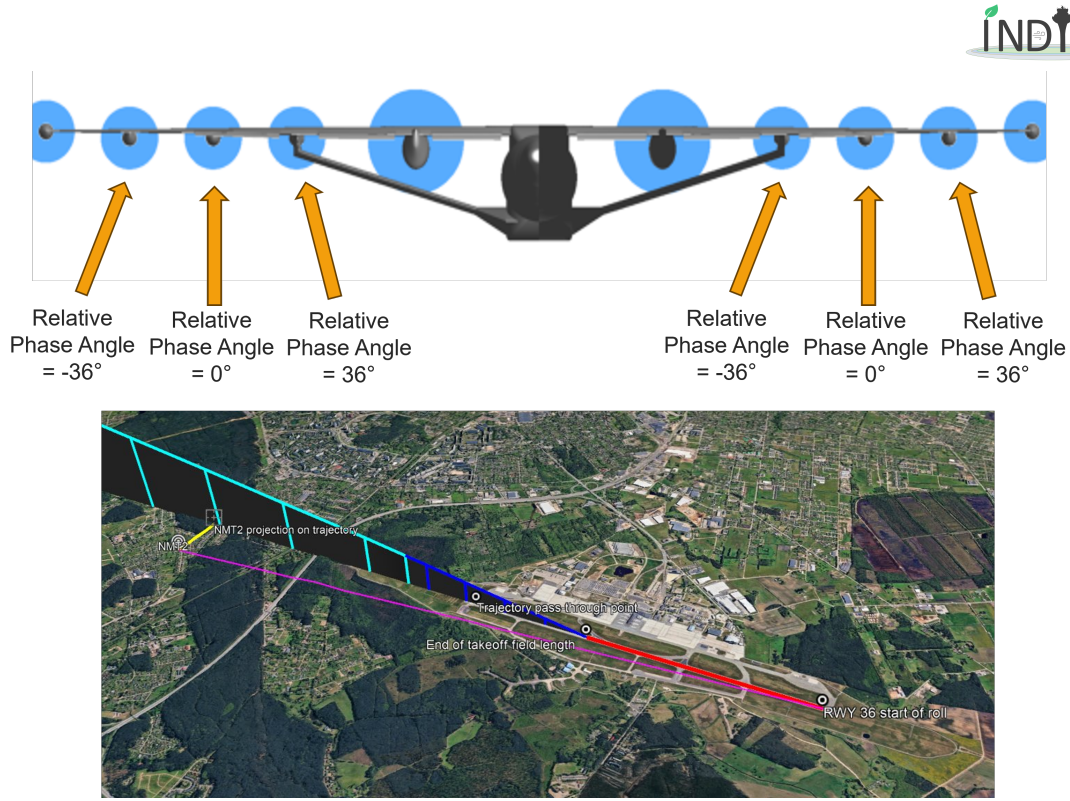


Figure 13: Phase-synchronised propellers arrangement for INDIGO aircraft, with trajectory description.

comparing the reference INDIGO aircraft operational parameters with the same reference parameters with propellers phase-synchronisation applied according to the findings of University of Bristol.

In Fig. 14 and Fig. 15, the results are shown in terms of fluctuating pressure contours, highlighting qualitatively the change in the acoustic field for the two cases considered, which can be seen more clearly when considering the results for the mid-plane of the aircraft.

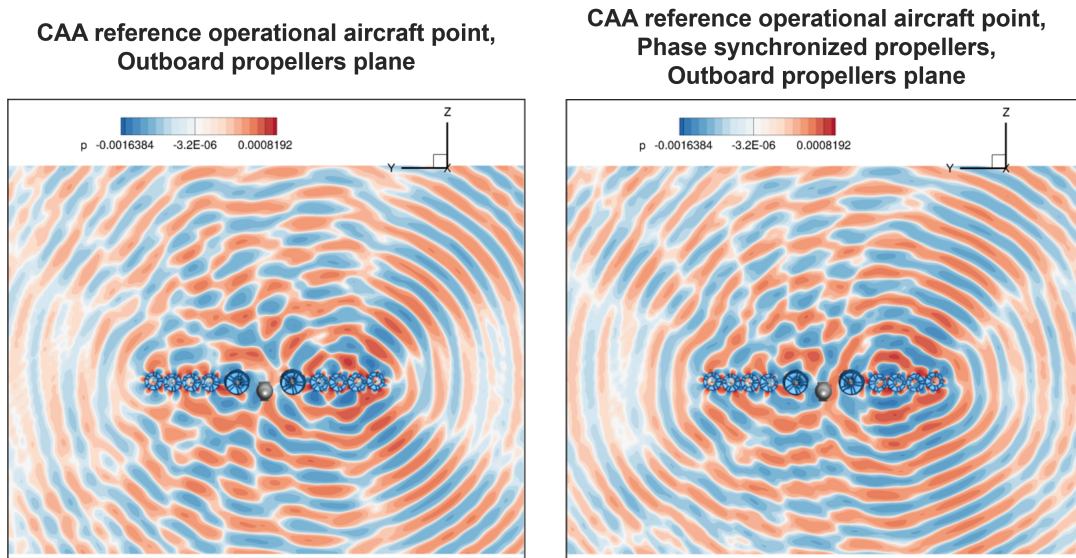


Figure 14: INDIGO aircraft phase-synchronization results in terms of pressure fluctuations contour plots, for the plane located at the outboard propellers.

The change in SPL recorded on the far-field hemispheres used as input for the fly-over tool IMMIS+, rescaled to a distance of 30 m for visualization purposes, highlights more clearly the change in directivity, as can be seen in Fig. 16 and Fig. 17, for the 1st BPF of both the inboard and the outboard propellers, in which a lateral shift in higher levels is detected. The corresponding propeller-tonal LAE levels resulting from the fly-over trajectory for the reference INDIGO configuration are approximately 6 dB(A)

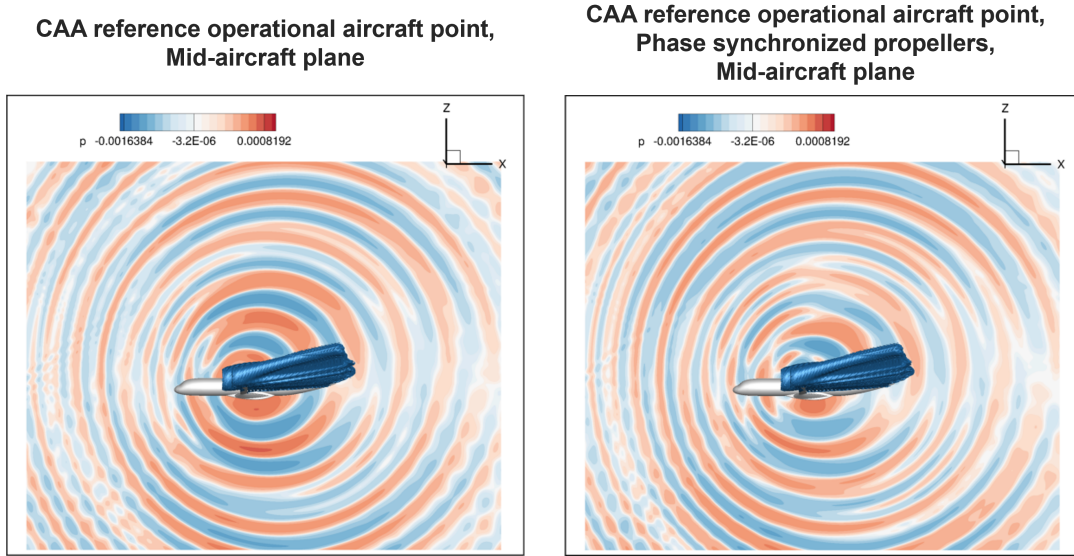


Figure 15: INDIGO aircraft phase-synchronization results in terms of pressure fluctuations contour plots for the vertical fuselage mid-plane of the aircraft.

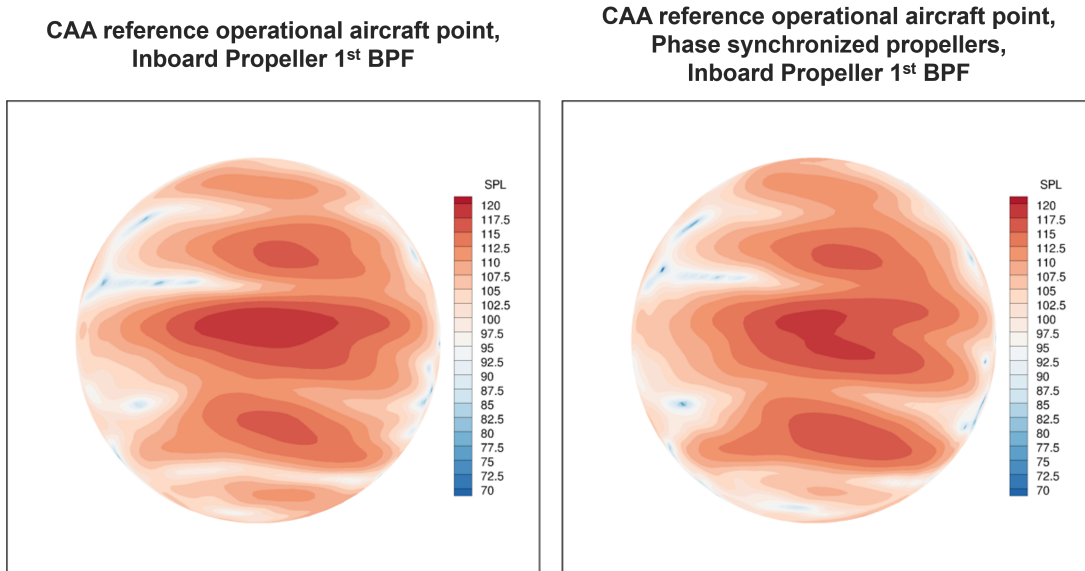


Figure 16: INDIGO aircraft phase-synchronization results, contour plots of the 1st BPF of the inboard propellers on the hemisphere used in the fly-over tool IMMIS+.

below the recorded LAE of the reference A320 flight from the RIX database, while the application of phase-synchronisation produces an additional reduction of approximately 2 dB(A) relative to the INDIGO reference configuration. The reason for the additional reduction due to phase-synchronisation could be attributed to the local reduction in SPL around the projected trajectory segment, which for visual reference would correspond to a horizontal intersection line passing through the centre of the hemispheres in Figs. 16 and 17, in the streamwise direction (from right to left). The comparison in terms of LAE was performed using only the propeller tonal noise component. For a more realistic noise assessment, the inclusion of broadband noise in the overall assessment was considered based on knowledge acquired over the past decades at DLR. For take-off operational conditions, the contribution to broadband noise due to landing gears is typically greater than approximately 3 dB with respect to other aircraft broadband noise type of sources, therefore only the inclusion of such a broadband noise contribution was considered within INDIGO [16]. For landing gears noise estimation, DLR in-house semi-empirical prediction tool PROFAN was used, summing the results energetically to the results obtained from the propeller tonal noise assessment. Only a preliminary estimation could be performed, since no landing

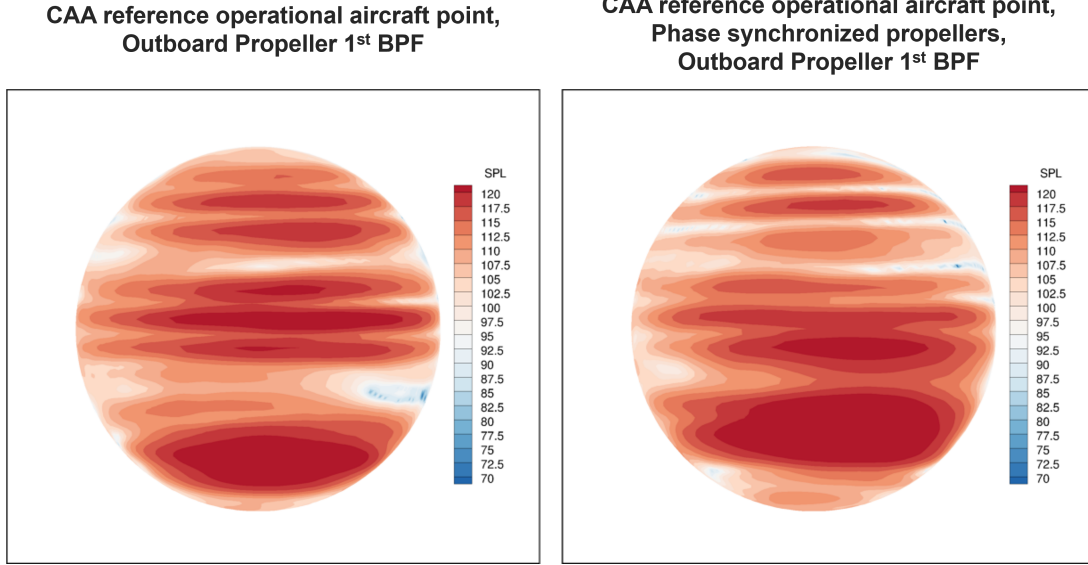


Figure 17: INDIGO aircraft phase-synchronization results, contour plots of the 1st BPF of the outboard propellers on the hemisphere used in the fly-over tool IMMIS+.

gear design was considered within the scope of the INDIGO project. A traditional three-landing-gears design was considered, with one nose landing gear and two main landing gears, based on Airbus A320 landing gears. The sizing of the main landing gears was chosen as follows:

- Number of wheels = 2
- Wheel width = 0.41144 m
- Wheel diameter = 1.21107 m
- Length single strut = 2.15610 m
- Diameter single strut = 0.20831 m

The sizing of the nose landing gear was chosen as follows:

- Number of wheels = 2
- Wheel width = 0.2098 m
- Wheel diameter = 0.7152 m
- Length single strut = 1.5909 m
- Diameter single strut = 0.1716 m

Since the trajectory considered is representative of a take-off and early climb procedure, it was considered in the IMMIS+ evaluation that the landing gears would be retracted after an altitude of 15 m. The addition of the landing gear evaluation contributed to an increase of 0.5 dBA in the levels detected for the propellers-only evaluation.

4 Multidisciplinary Design Analysis and Optimization Framework

The INDIGO framework evaluates each candidate aircraft through a sequence of coupled aircraft-level analyses. The process begins with the definition of the aircraft geometry, mass properties, propulsion-system ratings, and mission controls. These data are then propagated through the takeoff, airborne mission, landing, failure-case, emissions, and noise analyses. Because several quantities depend on one

another—most notably aircraft mass, required thrust, fuel burn, battery state of charge, and centre-of-gravity position—the disciplinary models are solved within a multidisciplinary design analysis before the resulting performance metrics and feasibility constraints are returned to the optimizer.

This framework is also applied deterministically in deliverables D6.1 and D6.2. Depending on the considered case, it is used either to optimize the aircraft powertrain while keeping the INDi2 airframe fixed (table 1), or to retain both the airframe and powertrain configurations fixed and optimize only the mission-control design variables (see Section 4.5.1 for the definition of the corresponding design variables). The optimization objective is defined either as the minimization of block fuel or as the minimization of a weighted objective function combining block fuel and fuel burned below 900 *m* during the nominal mission (see Section 4.5.2). In contrast, while the same aircraft-performance, fuel-consumption, and pollutant-emission metrics can also be evaluated, the MDO results presented in the following sections extend the framework by explicitly accounting for uncertainty within the aircraft design and optimization process, as described in more detail in Sections 5 and 6.

This section is therefore organised according to the path followed by a candidate design through the framework. The computational architecture is introduced first, followed by the definition of the aircraft and operating scenarios, the sequence of disciplinary evaluations, the treatment of coupled states, and finally the optimization problem.

4.1 Multidisciplinary Design Analysis (MDA) Framework

The Multidisciplinary Design Analysis (MDA) is performed using the in-house *TOPAZ* framework (*Tool for Optimizing Powerplants and Aircraft with Zero-Emissions*). *TOPAZ* was derived from the previously adopted MOTIVATION platform [17, 2] and is implemented using the open-source Python library GEMSEO [18]. Each disciplinary model is implemented as an independent module with explicitly defined inputs and outputs, while GEMSEO manages the execution sequence, data exchange, multidisciplinary coupling, and convergence of the coupled analysis. The framework also supports optimization and total-derivative computation. Selected disciplines additionally employ JAX automatic differentiation [19] through the GEMSEO-JAX plugin [20].

Figure 18 provides an aircraft-level overview of the MDA architecture. For each candidate design, the aircraft geometry and mass properties are established first. The analysis then evaluates the balanced field length, airborne mission, landing performance, failure conditions, and noise. Figure 19 expands the mission-analysis block and illustrates how aerodynamics, trim, propulsion, energy management, fuel burn, emissions, and mass updates are coupled at each flight point.

At each operating point, the framework evaluates the aerodynamic and propulsive states required to satisfy the prescribed flight condition. The resulting fuel-flow, battery-power, and pollutant-emission rates are integrated over the mission to determine the evolution of fuel mass, battery state of charge, and emitted pollutants.

Fuel consumption modifies the aircraft mass and centre-of-gravity position throughout the mission. These updated mass properties affect the aerodynamic forces, trim condition, and required propulsive power, which in turn influence the subsequent fuel and energy consumption. The MDA resolves these dependencies by repeatedly updating the shared disciplinary variables until consistency is achieved between the mission, mass, aerodynamic, and propulsion states.

Convergence of the MDA ensures internal consistency among the coupled disciplinary analyses. It does not, however, guarantee that the resulting aircraft satisfies all operational and design requirements. Field-length limits, component ratings, battery state of charge, geometric clearances, stability, and failure-case requirements are therefore enforced separately as optimization constraints.

4.2 Aircraft and Mission Definition

The first stage of the analysis establishes the aircraft state and the operating scenarios against which the design is assessed. This includes the takeoff weight, component masses, centre-of-gravity position, propulsion-system ratings, prescribed mission profile, takeoff and landing conditions, and failure cases. These quantities define the inputs required by the aerodynamic, trim, propulsion, emissions, and noise models.

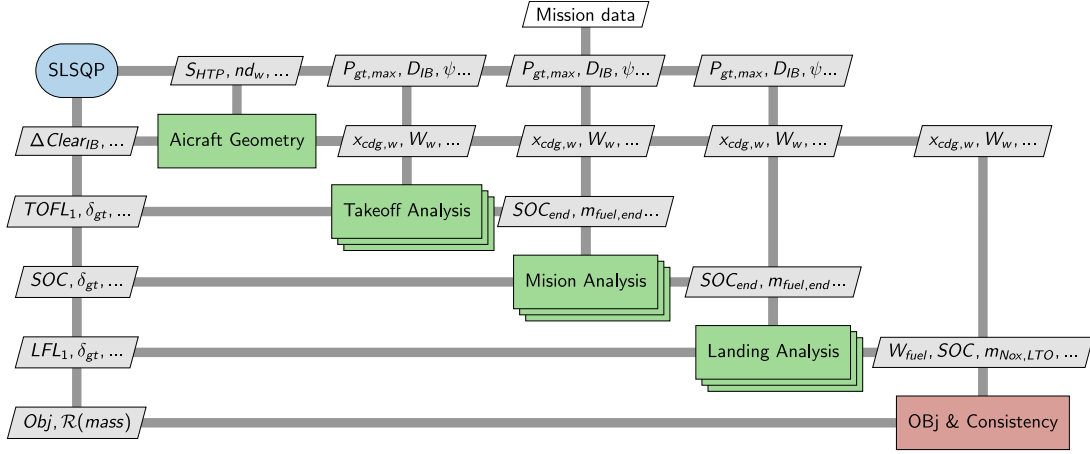


Figure 18: XDSM of the main optimization problem. Reproduced from [21].

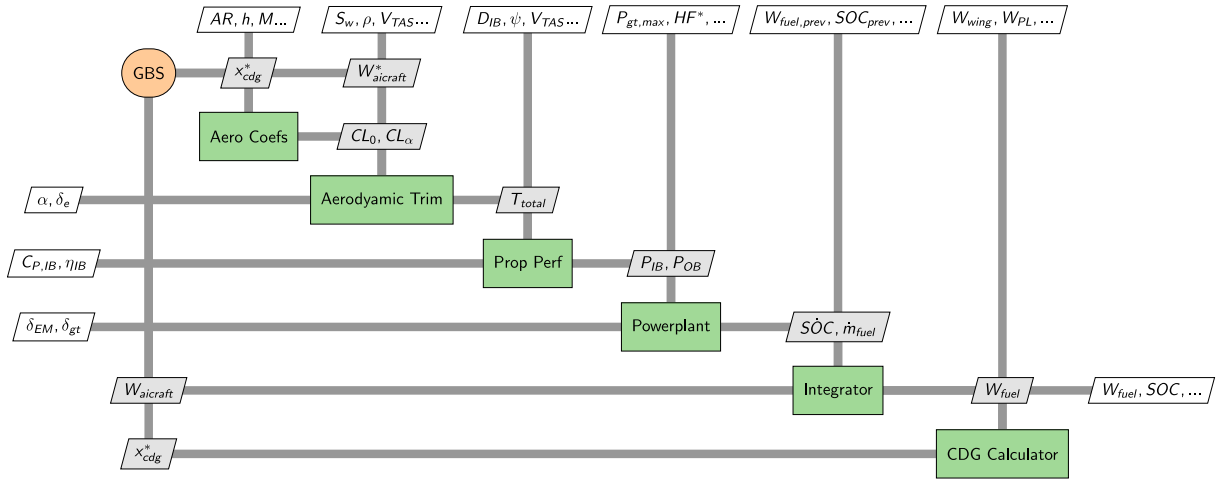


Figure 19: MDA executed for each flight phase. Reproduced from [21].

4.2.1 Aircraft Mass and Balance

Aircraft weight is evaluated using a component build-up approach, where the takeoff weight (TOW) is computed as:

$$TOW = OEW + W_{PL} + W_F + W_{bat} \quad (3)$$

where OEW is the operating empty weight, W_{PL} is the payload weight, W_F is the fuel weight, and W_{bat} is the battery weight. The operating empty weight is further decomposed as:

$$OEW = W_{struct} + W_{sys} + W_{furnishing} + W_{oper} + W_{ppsys} + W_{elec} \quad (4)$$

where W_{struct} is the structural weight, W_{sys} and $W_{furnishing}$ are the systems and furnishing weights, W_{oper} is the operational items weight, and W_{ppsys} is the weight of the powerplant and associated systems. Moreover, W_{elec} represents the weight of the additional electric components, battery excluded, needed to implement the hybrid-electric powertrain.

A mixed set of estimation methods is adopted: the lifting system primary structures (wing, strut, and central fuselage section) are sized using a finite element model, while the remaining components are estimated using handbook-based methods from the literature. This physics-based sizing approach is adopted due to the known limitations of experience-based weight correlations when applied to unconventional architectures (e.g., strut-braced wings).

The primary-structure sizing activity is performed outside the MDAO framework presented in this deliverable. Consequently, the resulting lifting-system structural weight is treated as frozen and is not updated during the optimization, even when powerplant units change.

Lifting system Wing primary structural weight is obtained using a GFEM-based sizing optimization of the wing-box material distribution (20), following the methodology in [22]. In the GFEM, the front and rear spars are located at 20% and 70% of the local chord, ribs are oriented perpendicular to the leading edge, and rib placement is aligned with propulsion-unit installation stations to represent pylon-to-wing attachments. Each propulsion unit is modelled as a concentrated mass connected to the corresponding rib through connector elements (20).

Sizing variables are defined by splitting each lifting-surface component into spanwise sections and assigning one thickness variable per section (20). All materials, layup constraints, load cases (limit and ultimate), strength and stability constraints, and boundary conditions are as defined in [22].

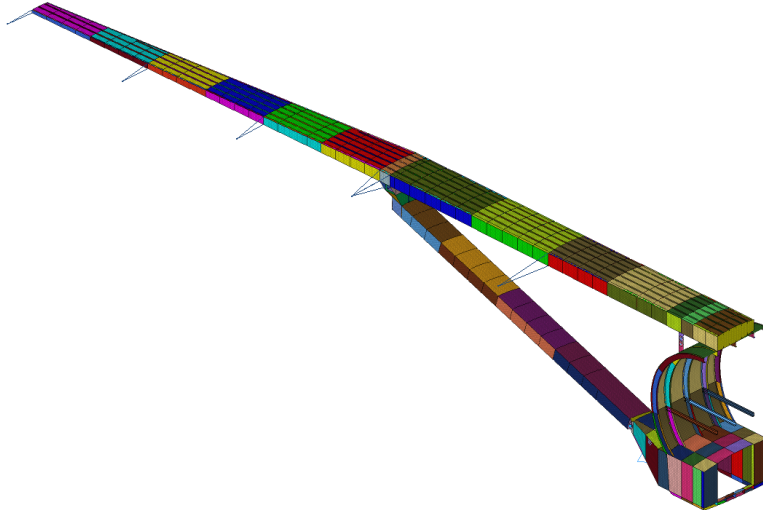


Figure 20: GFEM used to estimate the lifting system primary structural mass.

The remaining weights, including wing secondary structure, furnishings, nacelles, landing gear, and the horizontal and vertical tails, are estimated using the handbook methods of Raymer [23] and Torenbeek [24]. These estimates are corrected using the A320-200 weight breakdown reported in [25]. The T-tail weight is adjusted using weight breakdown data for the ATR42 [25] and ATR72 [26].

Powerplant masses Powerplant and electrical-system masses are estimated using adjusted specific-power values and surrogate models. A specific power of 5 kW/kg is assumed for the turboshaft dry core and auxiliary systems, and 16.4 kW/kg for the gearbox [27]. The turboshaft and gearbox design power ratings are optimisation variables, as specified in Section 4.5.1. Their masses are therefore updated during powertrain optimisation using the assumed specific powers.

For the electric system mass W_{elec} a surrogate model from INDIGO's partner UST (*University of Strathclyde*) [28] is used to estimate the mass of the main electrical powertrain elements (electric motors, wiring, and power converters) as a function of the rated power of the electric propulsors and the inboard electric machine.

Battery mass (W_{bat}) is estimated assuming a pack-level specific energy of 350 Wh/kg and a pack-level specific power of 1000 W/kg, consistent with future battery projections [29]. These battery properties are retained for all INDIGO reference cases. Alternative battery properties are introduced only in the dedicated battery-technology sensitivity and uncertainty-based optimisation studies.

Other masses Payload assumes all available seats are occupied and passengers are homogeneously distributed across the cabin, with an average passenger weight of 75.4 kg [30]. Luggage mass is set to 1800 kg in the overhead compartments (10 kg per passenger, consistent with airline specifications and slightly higher than the 7.7 kg surveyed by EASA [31]), with the remaining 4628 kg placed in the lower-fuselage cargo compartments.

Additional fudge weights for avionics, power-distribution subsystems, and operational items are introduced based on the A320 weight breakdown [25] to improve agreement in operating empty weight.

4.2.2 Mission Definition

Top level requirements The cruise altitude and Mach number are to $h_{cr} = 8000$ m and $M_{cr} = 0.6$, respectively. To support a more detailed representation of airport-vicinity operations, both climb and descent are divided into segments below and above 900 m.

Reference mission In addition to the airport-specific missions, selected case studies consider a separate reference mission with a prescribed range of 1000 nm (Fig. 21). The cruise altitude and Mach number are also fixed by the INDIGO top-level requirements. The low-altitude speed and flight-path-angle histories are defined from available operational flight-profile information, while the climb above 900 m is generated by connecting the end of the low-altitude segment to the prescribed top-of-climb condition using a smooth spline.

The intermediate control points of the spline are adjusted to obtain a representative regional-turboprop climb profile. For this 1000-nm reference mission, the profile shape is guided by available ATR72 operational data and ICAO/BADA performance information. The descent profile is generated using an analogous procedure, ensuring a smooth connection between cruise and the prescribed low-altitude descent segment.

CRIDA airport-specific trajectories For the flight segments below 900 m, INDIGO partner CRIDA (*Centro de Referencia I+D+i*) provided radar-derived trajectories for Riga. These data define the altitude, speed, and flight-path-angle histories during the initial climb and final approach, allowing the low-altitude portions of the mission to reproduce representative airport-specific operations.

The portions of the trajectories above 900 m are generated numerically to obtain complete missions. During climb, the final point of the CRIDA trajectory is smoothly connected to the prescribed top-of-climb condition. During descent, the cruise condition is connected to the initial point of the CRIDA low-altitude descent trajectory. Spline-based interpolation is applied in both cases to ensure smooth variations in equivalent airspeed and flight-path angle.

These reconstructed trajectories provide the operational mission set employed in the multi-trajectory UMDO formulation described in Section 6.2. Rather than sizing the aircraft for a single design mission, which could lead to a configuration excessively tailored to one operating condition, the optimization simultaneously considers a representative set of missions, $[M = M_{nom}, M_{short}, M_{long}]$.

The nominal mission, M_{nom} , corresponds to the statistically most frequent trajectory identified in the operational dataset (Most Frequent in table 4) and therefore anchors the optimization to observed aircraft operations. The remaining missions are selected to represent contrasting regions of the payload-range envelope. M_{short} (RIGA_TO_F78_LF137) corresponds to a high-payload, short-range mission, whereas M_{long} (RIGA_TO_F597_LF133) represents a high-payload mission at a comparatively longer range. Their inclusion exposes the common aircraft design to different combinations of mission range and payload that may drive different sizing requirements.

In addition, the 1000 nm reference mission is retained as a feasibility-only case. It is worth recalling that the INDIGO reference mission corresponds to a 1000-nm mission at maximum payload and maximum take-off weight.

All missions share the same fixed INDi2 airframe and the same optimised powertrain design variables, while mission-dependent quantities such as fuel load and battery energy are determined independently for each trajectory. The relevant feasibility constraints, including propulsion margins, mass compatibility, one-engine-inoperative take-off capability, noise, structural requirements, and centre-of-gravity limits, are enforced across the mission set. The robustness of the converged configuration is subsequently assessed by evaluating additional trajectories that are not included as active design points during the optimization.

Table 4: Selected Trajectories range and payload.

ID	Range [km]	Payload [t]
Reference	1852	20.0
Most Frequent	2111	12.5
RIGA_TO_F597_LF133	1670.39	18.7
RIGA_TO_F78_LF137	991.85	17.6

Diversion and holding profile The non-diversion portion of each mission, including the CRIDA airport-specific cases and the 1000-nm reference mission, is combined with the same fixed diversion and holding profile. The diversion and holding profile therefore remains unchanged across all trajectory cases.

For all mission definitions, continuity between the low- and high-altitude portions is enforced in equivalent airspeed (EAS) and flight-path angle γ . The horizontal distances covered during climb and descent are obtained by discretising the altitude profiles, calculating the transition time between successive altitude levels, and integrating the corresponding horizontal velocity. The cruise distance is then determined as the remaining distance required to satisfy the prescribed mission range. Atmospheric properties, including the freestream density ρ_∞ and speed of sound a , are evaluated using the ICAO standard atmosphere [32].

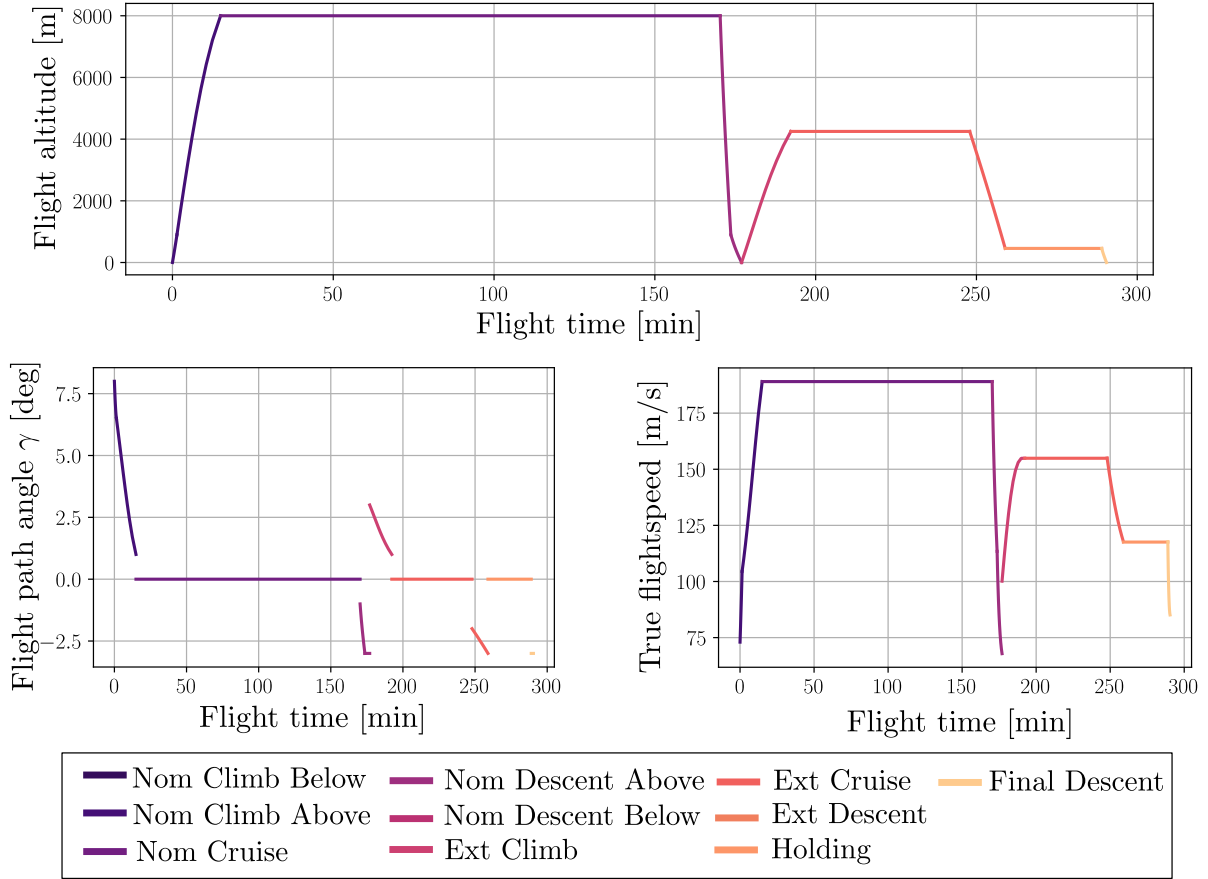


Figure 21: Altitude, flight-path angle γ , and flight speed along the 1000-nm reference mission. Reproduced from [33].

4.3 Aircraft-Level Analysis Sequence

Once the aircraft and operating scenarios are defined, the framework evaluates the candidate design in the same order as the physical flight sequence: takeoff, airborne mission, and landing. At each stage, the relevant disciplinary models are activated and their outputs are passed downstream to the next analysis.

4.3.1 Takeoff and Landing Analyses

The low-speed performance sub-module estimates the TOFL and LFL. For takeoff, the decision speed V_1 is determined such that, under a failure case, the accelerate-go distance (V_0V_1, V_1V_R, V_RV_2) equals the accelerate-stop distance (V_0V_1, V_1V_0). Here, V_0V_1 denotes acceleration from zero to V_1 , V_1V_R from V_1 to rotation speed V_R , and V_RV_2 from V_R to V_2 . In this study, $V_R = 1.1 V_{\text{stall}}$ and $V_2 = 1.2 V_{\text{stall}}$.

For all segments except V_RV_2 , the longitudinal dynamics are modelled as:

$$m \frac{dV}{dt} = T - D - \mu(mg - L) \quad (5)$$

where V is the airspeed, μ is the runway friction coefficient, L is the lift force, D is the drag force, m is the aircraft mass (assumed constant during all LTO phases), and g is the gravitational acceleration.

For the $V_R V_2$ segment, the airspeed is assumed to vary linearly from V_R to V_2 , with the excess thrust used to gain altitude:

$$\sin(\gamma) = \frac{T - D}{mg} \quad (6)$$

To compute the takeoff distances and V_1 , a doubly nested Newton algorithm is used. The outer loop identifies the value of V_1 that satisfies the balanced field-length requirement, while the inner loops integrate the acceleration profiles to obtain the time required to transition from the initial speed to the target final speed. For the $V_R V_2$ segment, the corresponding Newton loop determines the altitude required to reach $h_{obst, TO} = 35$ ft, given the γ profile, starting from $h = 0$.

For landing, LFL is simplified as the sum of three phases: (i) an initial glide at constant $\gamma_{approach}$ from the certification obstacle altitude $h_f = 50$ ft to a flare altitude h_{flare} , assuming a constant speed variation from $1.2 V_{stall}$ to $1.08 V_{stall}$; (ii) a flare segment from h_{flare} to $h = 0$; and (iii) a deceleration phase from the touchdown speed $1.08 V_{stall}$ to full stop V_0 .

During landing, the horizontal distances are computed as follows. For the initial glide, the distance is obtained geometrically using the constant approach angle $\gamma = -3$ deg. For the flare, the flare altitude h_{flare} and horizontal distance x_{flare} are computed using a simplified flare-radius approach (Eqs. 7 and 8) and a maneuver loading of $n_{loading}$. The horizontal distance of the deceleration phase is then obtained, as in takeoff, by numerically solving Eq. 5.

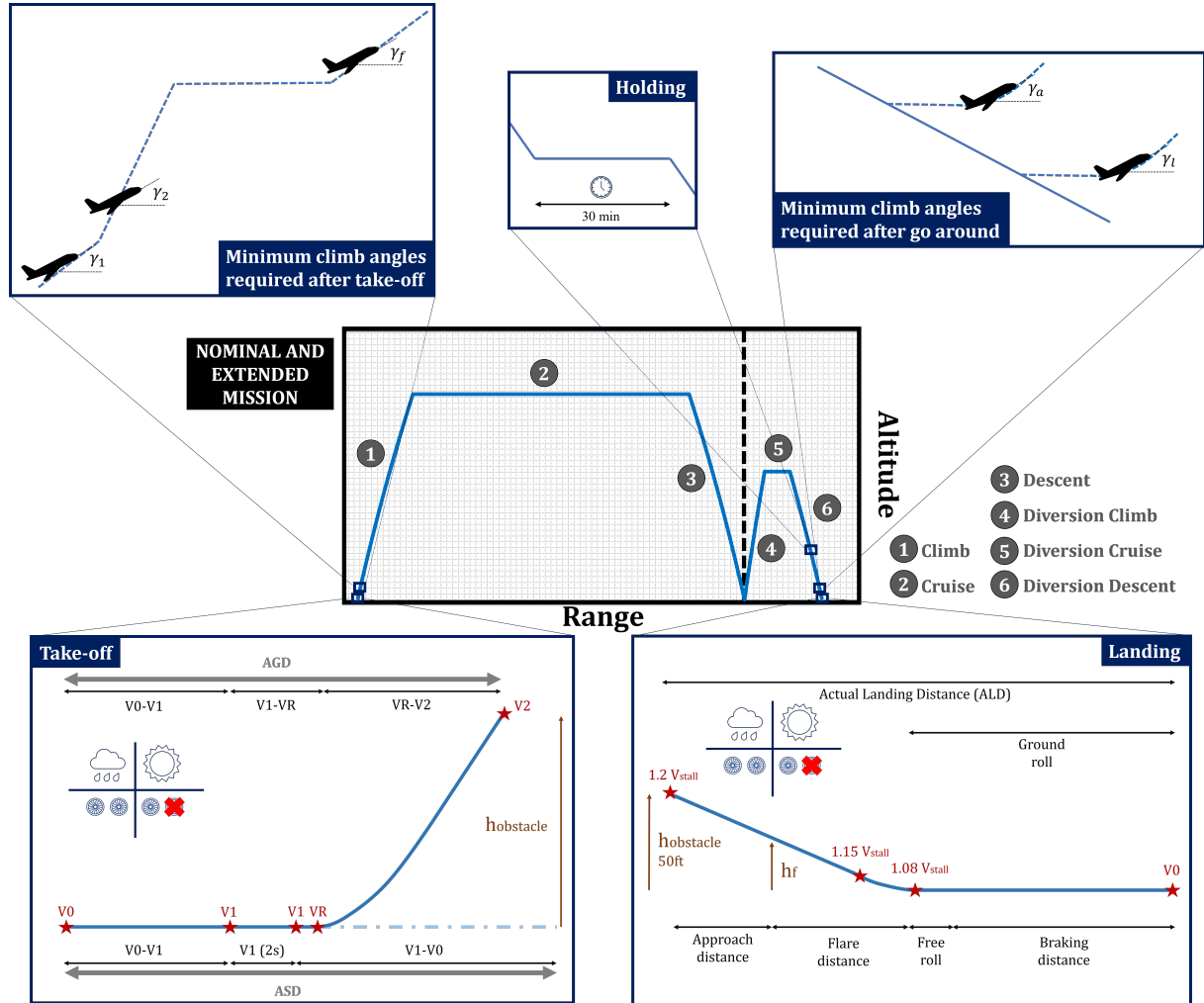


Figure 22: Schematic of the mission segments considered. Adapted from [17].

$$h_{flare} = \frac{V_{approach}^2 \cdot (1 - \cos(\gamma_{approach}))}{(n_{loading} - 1) \cdot g} \quad (7)$$

$$x_{flare} = \frac{V_{approach}^2 \cdot \sin(\gamma_{approach})}{(n_{loading} - 1) \cdot g} \quad (8)$$

Additional details are provided in [2, 3]. Figure 22, adapted from [17], provides a schematic overview of the mission-module segments considered.

4.3.2 Aerodynamic Model

Low Fidelity The aerodynamic model adopts a quadratic drag polar with linearized lift and pitching-moment coefficients, expressed as Eqs. 9. First- and second-order dependencies on angle of attack α and elevator deflection δ_e are retained as indicated by the corresponding subscripts. Overall aircraft coefficients combine wing and tail contributions for lift and drag, while the fuselage, fairings, and vertical tail contribute to the parasite-drag term.

$$C_L = C_{L_0} + \alpha C_{L_\alpha} + \delta_e C_{L_{\delta_e}} \quad (9a)$$

$$C_D = C_{D_0} + \alpha C_{D_\alpha} + \alpha^2 C_{D_{\alpha\alpha}} + \delta_e C_{D_{\delta_e}} + \delta_e^2 C_{D_{\delta_e\delta_e}} \quad (9b)$$

$$C_m = C_{m_0} + \alpha C_{m_\alpha} + \delta_e C_{m_{\delta_e}} \quad (9c)$$

Wing polar coefficients are provided by a surrogate model trained on strut-braced wing optimizations performed in VSPAero [34].

The surrogate provides $\{C_{L_0}, C_{L_\alpha}, C_{D_0}, C_{D_\alpha}, C_{D_{\alpha\alpha}}\}$. Figure 23 reports the reconstructed aerodynamic polar, with markers indicating direct VSPAero evaluations.

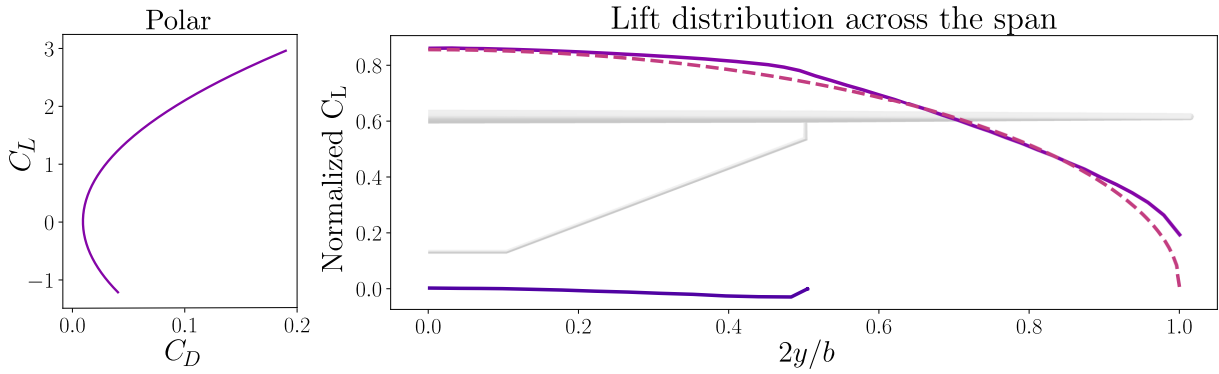


Figure 23: Left: aerodynamic polar of the optimised wing-strut configuration. Right: normalised lift distribution across the wing and strut. Reproduced from [33].

For the remaining surfaces, HTP induced drag is computed using conventional methods (e.g., DATCOM [35]), while parasite drag for the fuselage, HTP, and VTP is estimated using established drag-build-up correlations [23, 36, 37, 24]. Downwash is modelled assuming an elliptical lift distribution to separate wing and tail contributions, which is acceptable here given the near-elliptical optimized lift distributions shown in Fig. 23.

The neutral point (x_{np}), referenced to the leading edge, is computed at each mission point using Eq. 10. Contributions from the wing, HTP, and downwash are indicated by the subscripts w , HTP , and ϵ . The resulting x_{np} is then used in Eq. 11 to compute the static margin (SM), which is constrained to ensure longitudinal static stability.

$$x_{np} = \frac{(x_w + \Delta x_{wing,ac}) C_{L_{\alpha,wing}} + \frac{S_{HTP}}{S_{wing}} (x_{HTP} + \Delta x_{HTP,ac}) C_{L_{\alpha,HTP}} \left(1 - \frac{d\epsilon}{d\alpha}\right)}{C_{L_{\alpha,wing}} + \frac{S_{HTP}}{S_{wing}} C_{L_{\alpha,HTP}} \left(1 - \frac{d\epsilon}{d\alpha}\right)} \quad (10)$$

$$SM = \frac{x_{np} - x_{cg}}{\bar{c}_{wing}} \quad (11)$$

4.3.3 Aerodynamic Trim

For the remaining quasi-steady airborne mission phases, a dedicated trim module computes the aircraft trim by solving the residuals of vertical force, horizontal force, and pitching moment using a Newton-

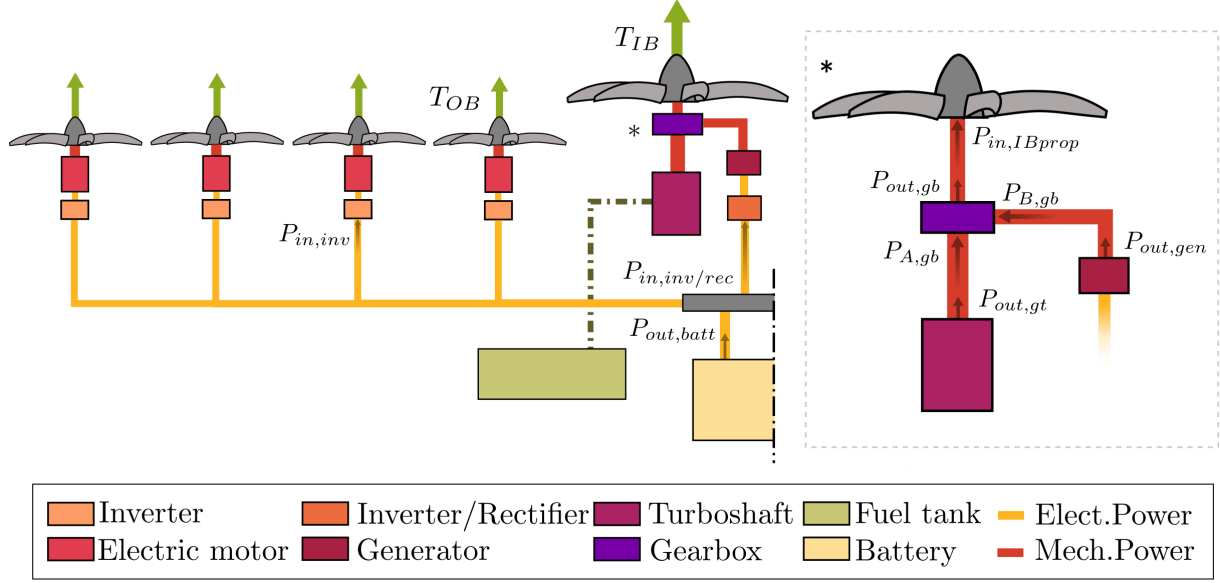


Figure 24: Schematic of the parallel-serial propulsion architecture for the INDIGO aircraft, reproduced from [21]. T denotes the thrust delivered by the inboard and outboard propellers; P denotes the power entering (*in*) or exiting (*out*) each component.

based algorithm:

$$\mathcal{R}_v = W \cos \gamma - q S_{ref} C_L(\alpha, \delta_e) = 0 \quad (12a)$$

$$\mathcal{R}_h = T - W \sin \gamma - q S_{ref} C_D(\alpha, \delta_e) = 0 \quad (12b)$$

$$\mathcal{R}_m = C_m(\alpha, \delta_e) = 0 \quad (12c)$$

where S_{ref} and c_{ref} denote the wing reference area and chord, q is the dynamic pressure, W is the instantaneous aircraft weight, C_L , C_D , and C_m are the lift, drag, and pitching-moment coefficients about the centre of gravity, and γ is the climb angle.

The residual system is solved for the angle of attack α and elevator deflection δ_e , together with the total required thrust T . Because the aircraft mass and center-of-gravity position evolve along the mission due to fuel burn, this module is called repeatedly during convergence of the MDA coupling states (see Subsec. 4.1).

4.3.4 Propulsion, Energy, Fuel, and Emissions

The trim solution provides the total thrust required at each quasi-steady operating point. This thrust is distributed between the inboard and outboard propulsors, converted into propeller power demand, and propagated through the hybrid-electric architecture. The resulting component powers determine throttle levels, fuel flow, battery-power demand, state-of-charge evolution, and pollutant-emission rates.

The hybrid-electric architecture adopts a parallel-serial configuration (Fig. 24) and comprises two subsystems: an inboard parallel system and an outboard serial electric system. The inboard subsystem includes a turboshaft coupled to an electric machine, which can operate as a generator (extracting mechanical power from the gearbox) or as a motor (supplying power to the gearbox that drives the inboard propeller).

The local hybridization factor (HF^*) characterizes the fraction of total power provided by the electric system. As defined in Eq. 13, $HF^* > 0$ corresponds to motor operation, while $HF^* < 0$ corresponds to generator operation:

$$P_{A,gb} = P_{out,gb} \cdot (1 - HF^*) \quad , \quad P_{B,gb} = P_{out,gb} \cdot HF^* \quad (13)$$

The outboard serial system incorporates eight EPUs, four per wing. Each EPU comprises an electric motor with its inverter driving an outboard propeller. All EPUs are connected to an electric bus that distributes power from the inboard electric machine; the battery power is therefore the sum of the bus-connected component power demands. Following the notation in Fig. 24:

$$P_{out,batt} = 8 \cdot P_{in,inv} + 2 \cdot P_{in,geninv} \quad (14)$$

A thrust split factor ψ regulates thrust allocation between the inboard and outboard propulsors. The thrust produced by each inboard propeller, T_{IB} , and by each outboard propeller, T_{OB} , are:

$$T_{IB} = T_{total} \cdot (1 - \psi)/2 \quad , \quad T_{OB} = T_{total} \cdot \psi/8 \quad (15)$$

To quantify the energetic impact of a thrust redistribution, the overall power-path efficiency is defined as

$$\zeta = \frac{P_{developed}}{P_{provided}} = \frac{8 \cdot P_{in,OBprop} + 2 \cdot P_{in,IBprop}}{P_{out,batt} + 2 \cdot P_{out,gt}} \quad (16)$$

Propeller model A surrogate model of propeller map efficiency is used, provided by TUBS (Technische Universität Braunschweig) [38, 39]. The model returns the propeller efficiency η_{prop} from maps defined in the J - C_T and J - C_P domains, where J is the advance ratio and C_T and C_P are the thrust and power coefficients. The surrogate is trained on maps spanning multiple Reynolds numbers Re and design thrust coefficients $C_{T,des}$, enabling the optimization of propeller diameter and rotational speed, as well as blade design, by trading disk loading against peak propeller efficiency. For the design variables, rather than directly optimizing the propeller diameter D and rotational speed n , the formulation is expressed in terms of the tangential component of the propeller tip Mach number at sea level (Eq. 17):

$$M_{tip}^* = \frac{\pi \cdot D \cdot n}{a} \quad (17)$$

This reformulation provides the additional benefit of recasting the conventional propeller operating coefficients,

$$J = \frac{V}{n \cdot D}, \quad C_T = \frac{T}{\rho \cdot n^2 \cdot D^4}, \quad C_P = \frac{P}{\rho \cdot n^3 \cdot D^5}, \quad (18)$$

into the following expressions:

$$J = \frac{V \cdot \pi}{M_{tip}^* \cdot a_{SL}}, \quad C_T = \frac{T \cdot \pi^2}{\rho \cdot (M_{tip}^*)^2 \cdot a_{SL}^2 \cdot D^2}, \quad C_P = \frac{P \cdot \pi^3}{\rho \cdot (M_{tip}^*)^3 \cdot a_{SL}^3 \cdot D^2}. \quad (19)$$

This transformation reduces the explicit power dependence to the propeller diameter while providing improved physical scaling of the rotational speed as the diameter varies.

Electric component model Powertrain component efficiencies are modelled as functions of the instantaneous operating power and the corresponding peak design power using a surrogate model developed by UST [28]. The surrogate is coupled to the system-level MDAO framework by taking as inputs the aggregated maximum operating power of all electric propulsion units, represented by a single variable P_{OB} , and the maximum operating power of the two electric machines mechanically coupled to the gas turbine, denoted as P_{IB} . The provided module internally evaluates the operating efficiencies and the combined component masses.

For the INDi2 reference configuration, a pack-level specific energy of 350 Wh/kg and a pack-level specific power of 1000 W/kg are assumed, based on projections in [29].

Gas turbine model A gas-turbine surrogate model developed by TUBS and RUB [38] predicts power-specific fuel consumption (PSFC) and pollutant emission indices (EI) as functions of power setting and flight condition. The model is trained on a database of turboprop configurations spanning overall pressure ratio (OPR) and turbine inlet temperature (T_{4t}). Each configuration is sized at top of climb for a prescribed peak gas-turbine power, ($P_{gt,peak}$), and evaluated across altitude, Mach number, and power setting to construct the (PSFC) response surface. Burner conditions are then supplied to a combustion-chamber surrogate, to estimate EI_{CO_2} , EI_{CO} , EI_{NO_x} , and EI_{UHC} , including non-ideal combustion effects [40]. More details can be found in [38].

Powertrain modelling flow For quasi-steady flight points, the required total thrust T_{total} is obtained by solving the trim residual system (Eq. 12). The thrust split ψ is then applied to determine the inboard and outboard propeller thrusts (T_{IB} and T_{OB}) via Eq. 15. For each propeller, the operating thrust is converted to C_T using the current flight conditions and propeller design point, and the propeller maps are used to obtain η_{prop} and the corresponding operating power.

Given the inboard propeller power, the local hybridization factor HF^* is applied at the gearbox (Eq. 13) to split the required gearbox output power between the turboshaft and the electric machine.

For the resulting turboshaft operating power and flight conditions, the gas-turbine surrogate provides the corresponding fuel-flow and pollutant emission rates.

Given the outboard propeller power and the efficiencies of the electric lines (EPU and generator path), the bus power demand is propagated to compute the battery power $P_{out,batt}$ and the battery discharge rate:

$$\frac{dSOC}{dt} = -\frac{P_{out,batt}}{3600 \cdot SPEC_{energy,batt} \cdot W_{bat}} \quad (20)$$

This explicit solve chain is summarized in Fig. 25. Component operating conditions are additionally checked against design peak-power limits using the throttle definition in Eq. 21. For the gas turbine, the available peak power depends on altitude h and Mach number M , and the corresponding throttle is referenced to $P_{gt,peak}(h, M)$.

$$\delta_i = \frac{P_i}{P_{i,peak}} \quad (a) \quad \delta_{gt} = \frac{P_{gt}}{P_{gt,peak}(h, M)} \quad (21)$$

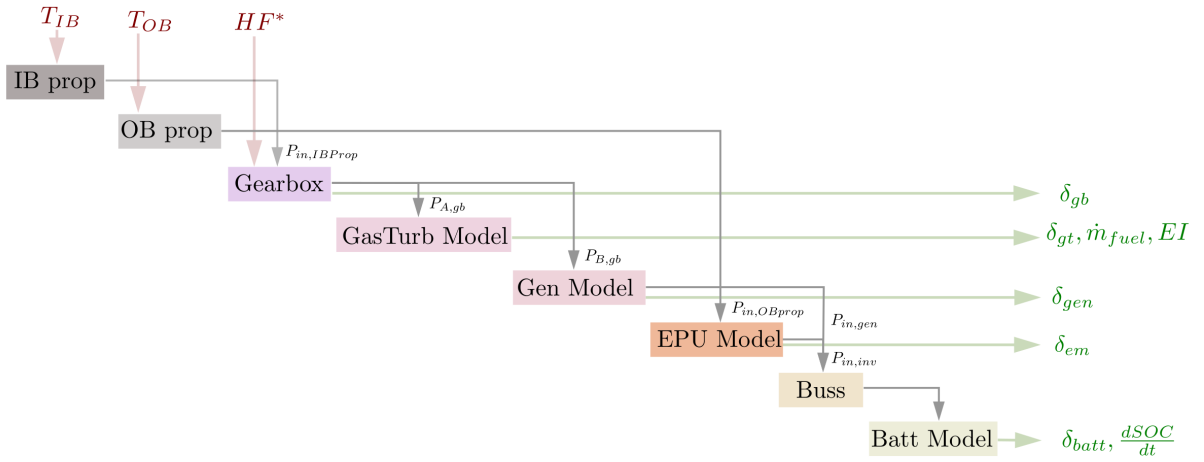


Figure 25: Logical flow of the powerplant state solve for quasi-steady trim conditions. Known values are highlighted in red and solved states in green. Reproduced from [33].

The control schedules for HF^* and ψ are defined as linear variations across mission segments. For takeoff and landing, HF^* is constant.

4.3.5 Failure Scenarios

Hybrid-electric propulsion introduces failure conditions beyond conventional engine-out scenarios, including the loss of distributed propulsors and electrical-system components. Five candidate failure cases were assessed during earlier optimisation activities [2, 3]. Two design-driving cases were retained in the current framework and are also both included in the balanced-field-length evaluation [3].

FC1: Inboard propulsion-unit failure FC1 represents the loss of one inboard propeller together with the adjacent propeller. The turboshaft associated with the failed inboard propeller remains available as a turbogenerator, and its throttle δ_{gt} is included among the optimisation variables. On the opposite side, the gearbox operates at full throttle, $\delta_{gb} = 1$, while the electric-machine throttle δ_{em} is optimised. The gas-turbine operating condition is determined from the failure-case hybridisation factor, which may differ from the nominal mission setting.

FC2: Outboard propulsion-unit failure FC2 considers the simultaneous loss of the two outermost outboard propellers on the same side of the aircraft. Both gearboxes operate at full throttle, $\delta_{gb} = 1$, while the throttles of the remaining electric motors are treated as optimisation variables.

Owing to the spanwise position of the failed propellers, FC2 produces the most demanding asymmetric-thrust yawing moment. Directional controllability is assessed by comparing the propulsive yawing moment

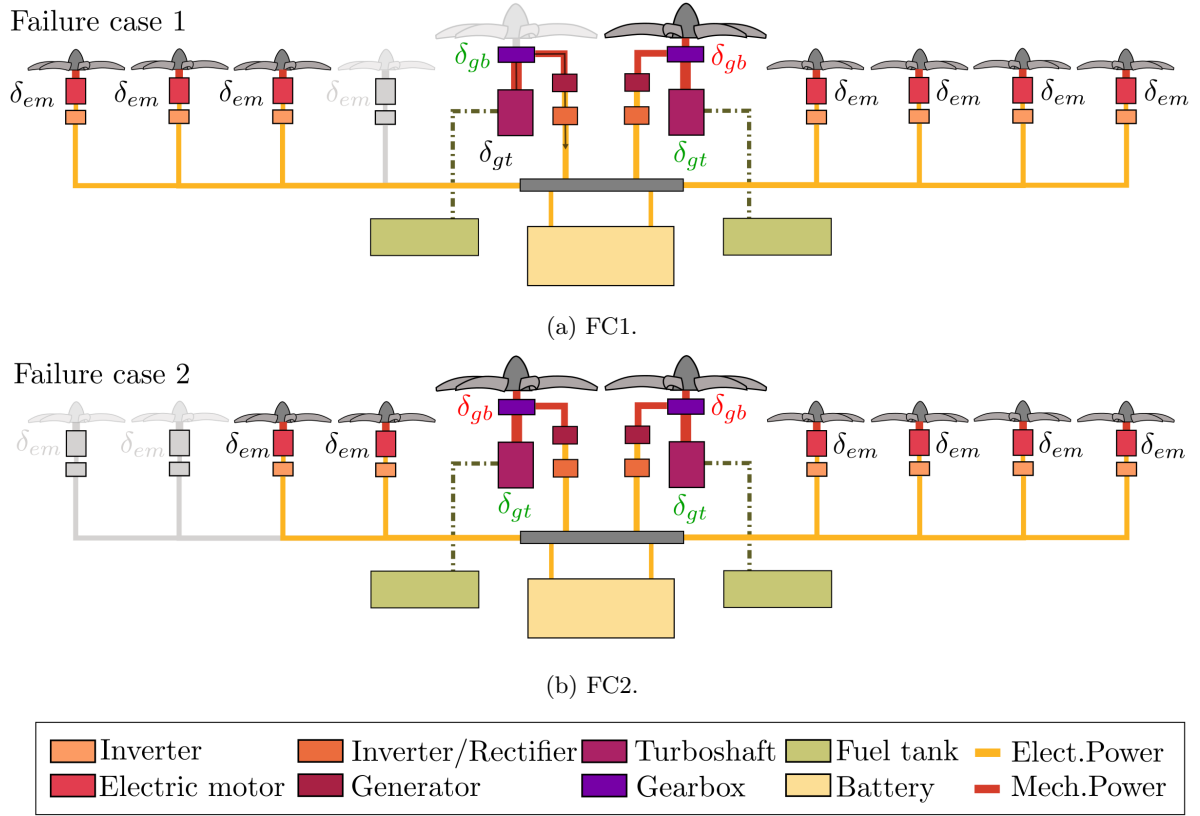


Figure 26: Failure-case scenarios for the hybrid-electric powerplant. Controls in red indicate fixed values, controls in black indicate optimization design variables, and controls in green indicate state variables. Adapted from [21].

during ground roll with the maximum available aerodynamic restoring moment. The latter is evaluated for $\beta = 10^\circ$ and $\delta_r = 10^\circ$ using a VSPAero-based regression model containing the fuselage, VTP, and HTP, with variable VTP area. Feasibility requires the available aerodynamic yawing moment to exceed the asymmetric propulsive contribution.

4.4 Environmental Evaluation

Pollutant emissions Pollutant-emission rates are provided by the gas-turbine model as a function of the engine operating condition and are integrated over the mission to obtain the corresponding emitted masses. Particular attention is given to the flight segments below 900 m, which are used to assess airport-vicinity emissions. A more detailed description of the emission model may be found in deliverable D2.2.

Aeroacoustic assessment Propeller noise is evaluated using a surrogate model derived from the DLR (*Deutsches Zentrum für Luft- und Raumfahrt*) in-house tool IMMISplus. The surrogate provides the acoustic contribution of a single propeller as a function of its diameter, thrust, rotational speed, trajectory, and receiver position. For multi-propeller configurations, each propeller is evaluated independently and the resulting levels are combined through logarithmic summation:

$$\text{EPNL}_{\text{tot}} = 10 \log_{10} \left(\sum_{i=1}^N 10^{\text{EPNL}_i/10} \right). \quad (22)$$

Since the trajectory used in the MDA may differ from the reference trajectory adopted to generate the acoustic surrogate, the predicted levels are corrected for the corresponding change in source–receiver distance:

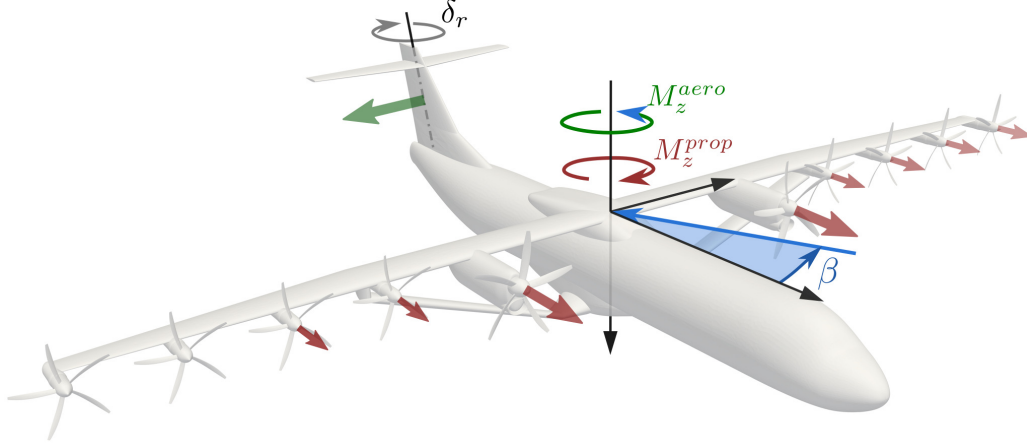


Figure 27: Directional-controllability constraint applied to FC2. Reproduced from [33].

$$EPNL_2 = EPNL_1 - 20 \log_{10} \left(\frac{h_2}{h_1} \right), \quad (23)$$

where $EPNL_1$ is the level obtained at the surrogate reference altitude h_1 , and $EPNL_2$ is the level corrected to the simulated altitude h_2 . A more detailed description of the aeroacoustic methodology and its integration within the MDAO framework is provided in [33].

4.5 Optimization Problem

After the MDA has converged, the framework returns the objective-function values and feasibility constraints to the optimizer. The optimisation process then updates the aircraft-level, propulsion-system, propeller, and control variables and repeats the complete aircraft-level analysis until the stopping criteria are met.

An optimization problem [41] can be formally stated as:

$$\begin{cases} \underset{x}{\text{minimize}} & f(x) \\ \text{subject to} & c(x) \leq 0 \end{cases} \quad (24)$$

where $f(x)$ is the objective function, x is the vector of design variables, and $c(x)$ denotes the set of equality and inequality constraints.

4.5.1 Design Space: Full powertrain optimisation

The aircraft considered in the optimisation studies used in WP5 and WP6 refer to the fixed INDi2 airframe specified in Table 1. In the full powertrain optimisation formulation considered here, the hybrid-electric powertrain is subject to optimisation. The design variables include the aircraft take-off weight (TOW), battery mass, component power ratings, propeller design parameters, and the control settings adopted during the different mission phases and failure conditions.

The corresponding design space is summarised in Table 5.

In this formulation, the turboshaft and gearbox are sized through $P_{TS,TOC}$ and $P_{Gearbox}$, while the electrical-system ratings are represented by $P_{IB,max}$ and $P_{OB,max}$. The electrical-component models subsequently determine the associated component masses and operating efficiencies.

The propeller design is defined through the inboard and outboard diameters, design thrust coefficients, and sea-level tangential tip Mach numbers. The rotational speed during takeoff and low-altitude operation may differ from the nominal design value through the multiplier $k_{RPM,LA}$.

The operational strategy is defined through the local hybridisation factor HF^* and thrust-split factor ψ for each mission phase. Separate throttle and hybridisation variables are introduced for the retained failure cases.

Table 5: Design variables considered in the full powertrain optimisation case.

Variable	Units	Meaning	LB	UB
Aircraft-level variables				
TOW	tons	takeoff weight	70.0	79.0
W_{bat}	tons	battery mass	0.7	20.0
Powerplant design				
$P_{TS,TOC}$	MW	turboshaft rating at top of climb	0.06	–
$P_{Gearbox}$	MW	gearbox rating	0.5	–
$P_{IB,max}$	MW	inboard electric-machine peak design power	0.0	–
$P_{OB,max}$	MW	outboard electric-propulsor peak design power	0.0	–
$\delta_{EPS,TO}$	–	electric-propulsion-system throttle at takeoff	0.1	0.9
Propeller design				
$k_{RPM,LA}$	–	low-altitude rotational-speed multiplier	0.85	1.0
d_{IB}	m	inboard propeller diameter	1.0	5.0
d_{OB}	m	outboard propeller diameter	1.0	4.0
C_{TIB}	–	inboard design thrust coefficient	0.2	0.5
C_{TOB}	–	outboard design thrust coefficient	0.2	0.5
M_{tipIB}^*	–	inboard sea-level tangential tip Mach number	0.3	0.5
M_{tipOB}^*	–	outboard sea-level tangential tip Mach number	0.3	0.5
Mission controls, defined for each flight phase				
HF^*	–	local hybridisation factor	–10.0	0.9
ψ	–	inboard–outboard thrust-split factor	0.1	0.9
Failure-case controls, defined for each failure case				
$\delta_{EPS,FC}$	–	electric-propulsion-system throttle	0.1	0.9
$\delta_{TS,FC}$	–	turboshaft throttle	0.1	0.9
HF_{FC}^*	–	failure-case hybridisation factor	–10.0	0.9

4.5.2 Objective Functions

Different objective functions are considered depending on the purpose of the optimisation. The adopted formulations focus on the fuel burned during operations below 900 m, which is an important indicator of the emissions in the airport-vicinity (and hence, a fuel-based proxy for airport-vicinity emissions and, hence, Local Air Quality), and on the block fuel, i.e., the mission fuel consumption. All fuel quantities entering the objective functions are evaluated over the non-diversion portion of the considered mission; the common diversion and holding profile is excluded from the objective-function calculation.

Block-fuel objective The first formulation minimises the total block fuel consumed during the nominal mission:

$$F_{BF} = BF, \quad (25)$$

where BF denotes the fuel consumed over the non-diversion portion of the considered mission. This formulation primarily favours solutions with reduced overall mission fuel consumption.

Combined block-fuel and low-altitude fuel objective The second formulation gives greater importance to fuel consumption in the airport vicinity by combining the block fuel with the fuel burned below 900 m, denoted by FB_{900} . The objective is defined as

$$LAQ = 0.35 \frac{BF}{BF_{ref}} + 0.65 \frac{FB_{900}}{FB_{900,ref}}, \quad (26)$$

where BF_{ref} and $FB_{900,ref}$ are reference values used to normalise the two contributions. The larger weight assigned to FB_{900} directs the optimisation towards reducing fuel consumption during the climb and descent portions below 900 m, while retaining sensitivity to the overall mission fuel requirement.

Multi-trajectory objective The framework also supports multi-trajectory optimisation, in which a single aircraft design is evaluated over a set of N_{traj} nominal trajectories. Depending on the selected formulation, the objective is calculated as the arithmetic mean of either the block-fuel objective or the combined block-fuel and low-altitude fuel objective across all trajectories:

$$\bar{F}_{BF} = \frac{1}{N_{\text{traj}}} \sum_{j=1}^{N_{\text{traj}}} BF_j, \quad (27)$$

or

$$\overline{LAQ} = \frac{1}{N_{\text{traj}}} \sum_{j=1}^{N_{\text{traj}}} \left[0.35 \frac{BF_j}{BF_{\text{ref } j}} + 0.65 \frac{FB_{900,j}}{FB_{900,\text{ref } j}} \right], \quad (28)$$

where the subscript j identifies the trajectory considered. This formulation prevents the optimisation from being driven by a single airport or mission profile and instead favours aircraft and operational settings that provide consistent performance across the complete set of trajectories.

Pollutant emissions and aeroacoustic metrics remain available as framework outputs and may be monitored or included in alternative optimisation formulations. In the cases documented here, however, the optimisation objectives are based on nominal-mission fuel consumption and fuel burn below 900 m.

4.5.3 Constraints

The constraints are summarized in Table 6; they can be grouped into three categories: (1) Performance and Airworthiness, (2) Geometric, and (3) MDA Feasibility.

For the *performance* constraints, TOFL and LFL must not exceed the corresponding A320neo approximate values. TOFL and LFL are evaluated for nominal and FC conditions, and for both dry and wet runway cases.

For the FC2 condition, an additional constraint is imposed on the residual thrust-asymmetry yawing moment (Eq. 29), accounting for the yawing moment contributions described in Sec. 4.3.5.

$$\mathcal{R}_{M_z} = M_z^{\text{prop}} - M_z^{\text{aero}} \leq 0 \quad (29)$$

To guarantee longitudinal static stability, the HTP is sized such that the static margin is not lower than 20%. Although this threshold may appear overly-conservative, it compensates for the limited scope of the stability check, which is restricted to a nominal full-payload mission and the high-speed configuration.

$$\mathcal{R}_{SM} = 0.2 - SM \leq 0 \quad (30)$$

The *geometric* constraints enforce clearances between propellers and between propellers and the ground, as shown in Fig. 28. The inboard propeller axes are fixed at 25% of the wingspan, whereas the outboard propellers are uniformly distributed between the strut-wing junction and the wingtip. Propeller-to-propeller clearance enforces a minimum separation equal to 10% of the propeller diameter:

$$\mathcal{R}_{cl} = 0.1 \max(D_i, D_j) - \text{dist}(\text{Prop}_i, \text{Prop}_j) \leq 0 \quad (31)$$

where D_i and D_j are the diameters of two adjacent propellers and $\text{dist}(\text{Prop}_i, \text{Prop}_j)$ denotes the minimum tip-to-tip distance between the corresponding propeller disks. The propeller-to-ground clearance constraint enforces a minimum roll-angle margin of 10° , thereby limiting the maximum allowable diameter of the outboard propellers.

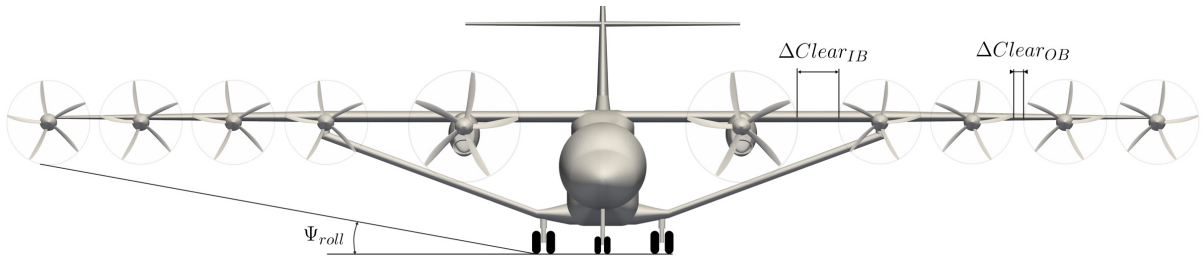


Figure 28: Definition of the inboard clearance constraint $\Delta C_{ler_{IB}}$, outboard clearance constraint $\Delta C_{ler_{OB}}$, and roll-angle clearance constraint Ψ_{roll} . Reproduced from [21].

As noted above, MDA convergence does not imply feasibility. Therefore, *MDA-feasibility* constraints are enforced at the optimization level. One subset ensures that each component remains within its

Table 6: Optimization constraints. Blue variables have a limit of 90% during nominal operation and 100% in failure scenarios.

Constraint	Description	LB	UB
$\mathcal{R}_{TOW} [kg]$	TOW consistency	–	0.0
$BFL [m]$	Balanced field length	–	2000.0
$LFL [m]$	Landing field length	–	1500.0
$\mathcal{R}_{M_z} [N \cdot m]$	Yawing-moment trim in FC2	–	0.0
$\mathcal{R}_{SM} [-]$	Static-margin constraint	–	0.0
$\mathcal{R}_{cl} [-]$	Propeller–propeller clearance	–	0.0
All flight phases:			
δ_{gt}	gas turbine throttle	0.0	0.9/1.0
δ_{gb}	gearbox throttle	0.0	0.9/1.0
δ_{em}	electric motor throttle	0.0	0.9/1.0
$\delta_{inverter}$	inverter throttle	0.0	0.9/1.0
$\delta_{Battery}$	battery throttle	0.0	0.9/1.0
SOC_{bat}	battery SoC	0.2	1.0

maximum allowable capacity: for FC conditions, the limit is set to 100% of capability, whereas for nominal operations it is restricted to 90%. This is checked through the so-called throttle level δ of each component, defined as the ratio of operating power to power rating. For gas turbines, throttle calculations account for operating power relative to the maximum available power at the given altitude and operating point.

Another key constraint is the battery SoC, which is required to remain above 20% to mitigate early degradation. This requirement is enforced over the full mission profile. Given the capability of the adopted powertrain architecture to recharge the battery, the optimizer may draw more energy in earlier phases and compensate by recharging during descent.

It is worth noting that constraints monitoring mission-dependent responses result in multiple scalar constraints. This arises because the mission is divided into segments and, within each segment, the governing equations are integrated using internal nodes (10 per segment) to improve accuracy.

A further MDA compatibility constraint is introduced to ensure a consistent value of the TOW. Since no top-level iterative loop is implemented to enforce consistency between aircraft weight and fuel burn (available only after simulating the full mission), the following residual is imposed:

$$\mathcal{R}_{TOW} = OEW + W_{PL} + W_F + W_{bat} - TOW \leq 0 \quad (32)$$

A strictly negative value indicates that additional payload or energy could be embarked.

Optimizer The optimization problem is solved using the Sequential Least Squares Programming (SLSQP) algorithm [42] as implemented in the `pyOptSparse` package [43]. Stopping criteria include a relative function tolerance of 10^{-5} and a maximum number of iterations of 400.

5 UMDO

The basic aspects of the methodology are described in the INDIGO deliverable D5.1 and, in greater detail, in [44], where we examine the risks of adopting emerging or developing technologies in projects that aim to deploy a product in the medium or long term.

5.1 Uncertainty Representation and Modelling

To represent this uncertainty, Dempster-Shafer Theory [45, 46] is adopted [44], as it is well suited to cases where the available information is insufficient to define a precise probability distribution. In the present work, the uncertainty model is implemented through a Basic Probability Assignment (BPA) table. For a given target year and optimizer-selected battery-specific energy requirement, the BPA table defines intervals of plausible battery-specific energy values and assigns a belief mass to each interval. These intervals describe possible future technology scenarios, while the associated belief masses express the confidence attributed to each range.

To integrate the imprecise probability into the robust design optimization loop, we **define a stochastic function that, for each selected year and required energy density, produces an output distribution of energy densities consistent with the Dempster-Shafer theory**. Rather than committing to a single precise probability for every possible outcome, the Dempster-Shafer evidential framework allows probability mass to be assigned to *sets* of outcomes. This captures the epistemic uncertainty that is unavoidable when forecasting the maturity of a future technology: an expert may be confident that a given energy density will be reached, but unable to say precisely *which* of several candidate values applies.

The quickest way to produce the probability distribution for the energy densities is to apply the **basic probability assignments (BPAs)** of the Dempster-Shafer theory to obtain the evidential representation. Each BPA distributes belief mass across the singletons and the composite hypotheses (unions of singletons), so that the resulting structure is a valid, self-consistent representation of the available evidence. Subsequently, this evidential representation is transformed into a probability distribution, using the pignistic approach sketched below [47], to be used in the robust decision-making process.

Defining a BPA table allows us to define the epistemic uncertainty of a given model. However, when a decision is necessary, we must transform the imprecise probability into a classical probability distribution. This can be achieved using Smets' Transferable Belief Model [47], as described below. Indeed, the BPA assigns belief mass to focal elements, which may be singletons or composite sets. Formally, the masses form a probability distribution over the power set of the frame of discernment, but the mass $m(A)$ should not be read as the probability of A . Rather, it is the portion of evidence that supports exactly the proposition "the true state lies in A ", without further discrimination among its elements. Mass assigned to composite sets thus expresses epistemic uncertainty: when experts (or the available evidence) cannot distinguish among certain alternatives, they assign mass to the set containing them, without committing to how that mass is distributed among its singletons. The probability of any event A is then bounded by the interval $[Bel(A), Pl(A)]$, whose width reflects the degree of ignorance. Because a concrete decision is required, the belief function is converted into a classical probability distribution through Smets' pignistic transformation [48], in which the mass of each focal element is distributed equally among its singletons. Following the TBM, this step is performed only at the decision (pignistic) level, after all evidence has been combined; the uniform split is therefore a decision rule, not an assumption that the singletons are equally likely at the credal level.

Table 7 reports the Basic Probability Assignments (BPA) table for the battery energy densities. Each entry is the belief mass assigned to the hypothesis (row) in the forecast year (column). Columns sum to one.

Table 7: Basic Probability Assignment (BPA) table for Battery Energy densities.

Focal Element (A)	2026	2028	2030	2035	2040
{300}	0.96618	0.00000	0.00000	0.00000	0.00000
{400}	0.00000	0.51282	0.04274	0.00000	0.00000
{500}	0.00000	0.28490	0.42735	0.07843	0.00000
{600}	0.00000	0.00000	0.21368	0.39216	0.07782
{700}	0.00000	0.00000	0.08547	0.23529	0.38911
{800}	0.00000	0.00000	0.00000	0.03922	0.27237
{300, 400}	0.02415	0.14245	0.00214	0.00000	0.00000
{400, 500}	0.00000	0.02849	0.10684	0.02941	0.00000
{500, 600}	0.00000	0.01425	0.05342	0.03922	0.00973
{600, 700}	0.00000	0.00000	0.01068	0.05882	0.04864
{700, 800}	0.00000	0.00000	0.01068	0.03922	0.09728
{300, 400, 500}	0.00966	0.01140	0.04274	0.03922	0.00778
{400, 500, 600}	0.00000	0.00570	0.00427	0.03922	0.01946
{500, 600, 700}	0.00000	0.00000	0.00000	0.00784	0.03891
{600, 700, 800}	0.00000	0.00000	0.00000	0.00196	0.03891
$\sum m(A)$	1.00000	1.00000	1.00000	1.00000	1.00000

5.1.1 Worked Example: Year 2028

To illustrate how the table is read, assume our year of interest is 2028. The only non-zero basic probability assignments in that column are:

Table 8: Non-zero basic probability assignments for the year 2028.

Hypothesis	Mass [%]
$E_d = 400$	51.3
$E_d = 500$	28.5
$E_d \in \{300, 400\}$	14.2
$E_d \in \{400, 500\}$	2.8
$E_d \in \{500, 600\}$	1.4
$E_d \in \{300, 400, 500\}$	1.1
$E_d \in \{400, 500, 600\}$	0.6

The masses assigned to the singletons (400 and 500) express precise belief, whereas the masses assigned to the composite hypotheses express imprecise belief: the evidence supports one of the listed values, but does not discriminate further between them. Note that, as required, the masses sum to unity:

$$0.513 + 0.285 + 0.142 + 0.028 + 0.014 + 0.011 + 0.006 = 1.000.$$

5.2 Uncertainty Propagation and Analysis

To integrate the Dempster-Shafer Belief and Plausibility estimations into a robust design optimization loop, an imprecise-probability sampler is implemented that turns a Dempster-Shafer basic probability assignment into an operational stochastic function implementing the pignistic distribution. Given a year and a required energy density, it returns draws consistent with the evidential structure of the forecast, honoring both the precise (singleton) and imprecise (composite) components of the belief mass. The scripts here described allow the designer to inspect the full sampled distribution for a given year, to study the effect of conditioning, and to visualize how the obtainable energy density tracks an arbitrary requirement as the technology matures over time.

5.2.1 Requirement-limited Conditional Sampler

The core routine, `draw_cumulative_sample`, draws a single realisation of the battery specific energy for a chosen year and a required energy-density value. Conceptually, the sampler (i) selects a hypothesis according to its basic probability assignment, (ii) resolves any composite hypothesis into a specific energy-density value, and (iii) limits the returned value to the energy density required by the aircraft design. Thus, sampled technology outcomes below the requirement are retained, whereas outcomes exceeding the requirement are capped at the required value. In other words, the battery specific-energy samples are drawn from the pignistic distribution and then limited by the required battery specific energy. In the stochastic function implementation, outcomes below the requirement are retained, while outcomes above it are capped at the required value.

Function Call The Python interface is:

```
1 sample = draw_cumulative_sample(
2     distribution_DST,
3     year,
4     cond)
```

where `distribution_DST` is the BPA table, `year` is the forecast year of interest, and `cond` represents the required energy density imposed by the aircraft design.

Minimal Test Snippet The following snippet calls the sampler for a single draw:

```
1 import DST_sampler as dst
2
3 if __name__ == '__main__':
```

```

4  bpa_df = dst.create_bpa_table()    # CREATE BASIC PROBABILITY ASSIGNMENT DATAFRAME
5  year = 2030                       # SPECIFY YEAR
6  cond = 500                        # SPECIFY REQUIRED ENERGY DENSITY
7  sample = dst.draw_cumulative_sample(bpa_df, year, cond) # DRAW SAMPLE
8  print(f"\nYear: {year:10.5f}, Required ED: {cond:10.5f}, Sample: {sample:10.5f}\n")

```

Note. To obtain a sample *without* any condition on the energy density, simply set the `cond` parameter to a value higher than the maximum energy density considered in the BPA table. In that case, the upper cap is never active and the sampler returns an unconditioned draw.

5.2.2 Sampling the Probability Density Function

By repeating the elementary draw many times, one recovers an empirical probability density function for the energy density in a given year, with or without a condition. The script below reads the year, the condition, and the number of samples from the command line, accumulates the draws, and plots the resulting histogram.

```

1  import DST_sampler as dst
2  import matplotlib.pyplot as plt
3  from sys import argv, float_info
4
5  if __name__ == '__main__':
6      print("DST Sampler Histogram Plotter")
7      if len(argv) < 4:
8          print("\nUsage: python DST_sampler_plot_histogram.py <year> <condition> <samples>\n")
9          exit(1)
10     year = float(argv[1])
11     if argv[2].lower() == 'false':
12         cond = float_info.max
13     else:
14         cond = float(argv[2])
15     nsamples = int(argv[3])
16     print(argv)
17     print("Generating histogram of samples...")
18     bpa_df = dst.create_bpa_table()
19     samples = []
20     for _ in range(nsamples):
21         sample = dst.draw_cumulative_sample(bpa_df, year, cond)
22         samples.append(sample)
23     plt.hist(samples, bins=30, edgecolor='black')
24     if(cond == float_info.max):
25         plt.title(f'Histogram of Samples for Year {year} without Condition')
26     else:
27         plt.title(f'Histogram of Samples for Year {year} with Required ED={cond}')
28     plt.xlabel('Sample Value')
29     plt.ylabel('Frequency')
30     plt.grid(True)
31     plt.show()
32

```

Passing the literal string `false` as the second argument sets the condition to `float_info.max`, which effectively removes it (as noted above), so the same script serves both the requirement-limited and the unrestricted case.

Figure 29 reports the empirical distributions of the sampled energy density for the year 2030. On the left is the unconditioned sampling, which reproduces the full BPA-implied spread from about 300 to 800. On the right, a required battery specific energy of $E_d = 600$ Wh/kg is imposed by the aircraft design. Sampled values below 600 Wh/kg are retained, while higher-energy outcomes are capped at 600 Wh/kg.

Comparing the two panels of the figure illustrates the effect of imposing the design requirement: imposing this design requirement accumulates the higher-energy outcomes at 600 Wh/kg, consequently producing the pronounced peak observed at this value.

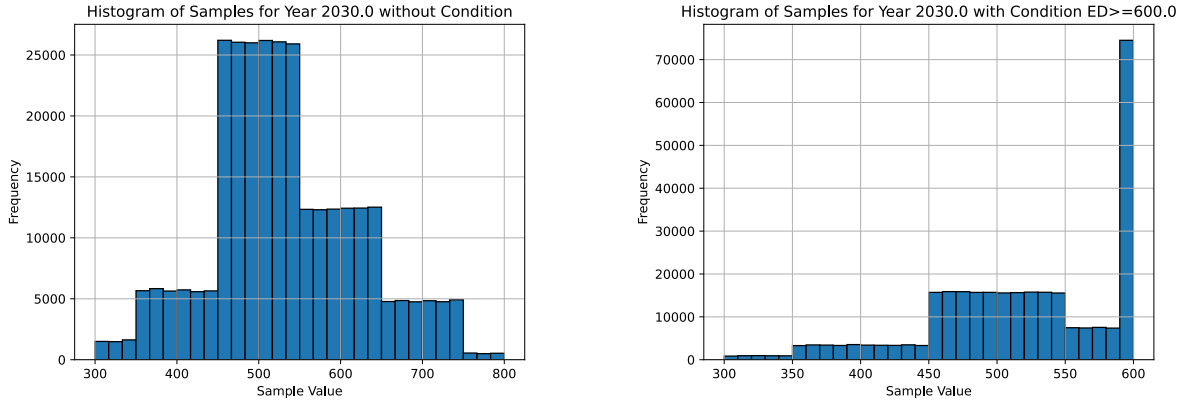


Figure 29: Unrestricted and requirement-limited empirical distributions of the sampled energy density for the year 2030.

5.2.3 Required versus Obtained Energy Density

A complementary diagnostic code snippet plots the obtained energy density against the required one, sweeping the required energy density uniformly over the range of interest. This reveals how frequently the sampler can meet an arbitrary requirement in a given year.

```

1 import DST_sampler as dst
2 import matplotlib.pyplot as plt
3 from sys import argv
4 import numpy as np
5
6 if __name__ == '__main__':
7     print("DST Samples as a Function of Condition Plotter")
8     if len(argv) < 3:
9         print("\nUsage: python DST_sampler_plot_histogram.py <year> <samples>\n")
10        exit(1)
11    year = float(argv[1])
12    nsamples = int(argv[2])
13    print(argv)
14    print("Generating conditions-samples plot...")
15    bpa_df = dst.create_bpa_table()
16    samples = []
17    conditions = []
18    for _ in range(nsamples):
19        cond = np.random.uniform(low=300, high=800, size=None)
20        sample = dst.draw_cumulative_sample(bpa_df, year, cond)
21        samples.append(sample)
22        conditions.append(cond)
23    plt.scatter(conditions, samples, s=5, alpha=0.1)
24    plt.title(f'Samples vs Conditions for Year {year}')
25    plt.xlabel('Energy Density Required')
26    plt.ylabel('Energy Density Obtained')
27    plt.xlim(275, 825)
28    plt.ylim(275, 825)
29    plt.grid(True)
30    plt.show()

```

The scatter plots in Figure 30 make the temporal evolution explicit. Points lying on the diagonal correspond to requirements that are exactly met; points below it correspond to requirements that could not be satisfied given the evidence for that year. As the forecast year advances, the diagonal segment extends to higher energy densities, reflecting the growing likelihood of reaching demanding targets. In particular, the technology is still immature in 2026 (Left), so the obtained density saturates at a low ceiling (around 300–350) regardless of the requirement. In 2029 (Right), the technology has matured, so the obtained density tracks the requirement along the diagonal $E_d^{\text{obtained}} = E_d^{\text{required}}$ up to a higher

ceiling, with a cloud of lower outcomes below it.

5.3 UMDO Workflow

Framework definition The UMDO workflow was developed to identify hybrid-electric powertrain designs with favourable expected performance while accounting for uncertainty in the battery specific energy available at battery-technology scenario year. The methodology is applied to the different trajectory formulations considered in the optimisation campaigns presented in the following sections.

The problem is parameterised by the battery-mass upper bound and the target battery specific energy. The battery-mass upper bound defines the maximum battery mass available during the powertrain optimisation, whereas the target battery specific energy represents the assumed technological performance of the battery. The former is treated as a deterministic design parameter, while uncertainty is introduced in the battery specific energy.

Deterministic database and surrogate construction For each considered trajectory formulation and prescribed noise constraint, a design of experiments is defined over the selected ranges of battery-mass upper bound and target battery specific energy. At each DOE point, *TOPAZ* is employed to optimise the hybrid-electric powertrain using the selected design space and objective-function formulation.

Each analysis provides the minimum attainable objective-function value for a prescribed combination of battery-mass upper bound and target battery specific energy. The resulting set of optimised solutions forms a deterministic database describing the variation of the optimal aircraft performance throughout the two-dimensional parameter space. The corresponding optimal design variables and relevant constraint values are also retained to support the subsequent interpretation of the optimisation results.

A Gaussian-process regression model is then trained using the deterministic DOE database. The resulting deterministic surrogate approximates the optimised objective-function response as a function of the battery-mass upper bound and battery specific energy, as illustrated in Fig. 31. A separate surrogate is generated for each trajectory formulation and prescribed noise constraint considered in the optimisation campaign.

The deterministic surrogate replaces the repeated execution of the complete multidisciplinary analysis and powertrain optimisation during uncertainty propagation. Directly embedding *TOPAZ* within the uncertainty loop would require a new powertrain optimisation for every sampled battery specific-energy value and every candidate UMDO design. The Gaussian-process surrogate instead provides computationally inexpensive predictions of the previously optimised response, making the repeated evaluations required by the UMDO process computationally feasible. For a given candidate battery-mass upper bound, each sampled battery specific-energy value is evaluated on a surrogate representing the optimum achievable after optimisation of the full powertrain design space. The uncertainty propagation therefore does not evaluate a single fixed powertrain design. Instead, the inner powertrain optimum is allowed to vary with the sampled battery-technology state, as represented by the deterministic optimal-response surface.

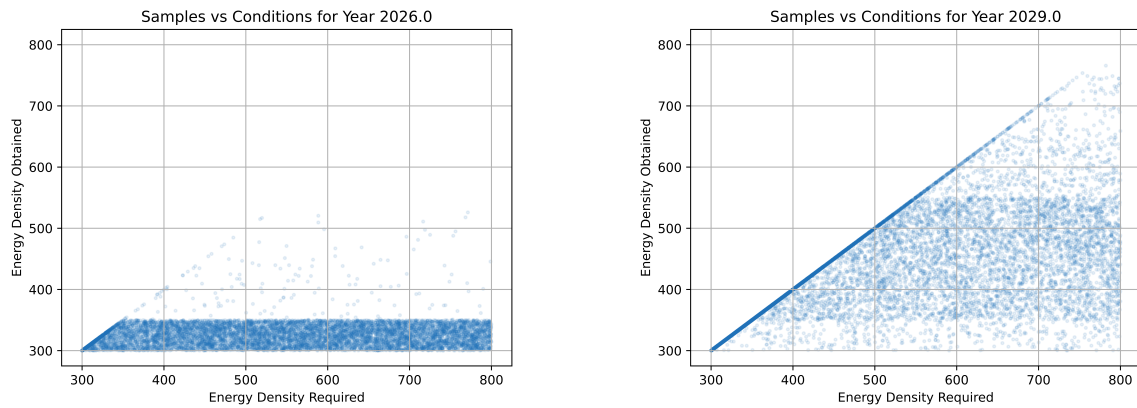


Figure 30: Obtained versus required energy density, with the condition drawn uniformly on $[300, 800]$ in 2026 and 2029.

Representation of battery uncertainty The battery specific energy attainable at battery-technology scenario year cannot be predicted precisely and is therefore represented as an uncertain technological parameter. Its uncertainty is described using the Dempster-Shafer framework presented in Section 5.1 and propagated using the sampling procedure described in Section 5.2.

For each considered battery-technology scenario year, this representation defines the range of plausible battery specific-energy values used within the UMDO process. In the present formulation, uncertainty is introduced only in the battery specific energy, whereas the battery-mass upper bound remains a deterministic variable selected by the UMDO optimiser.

When battery specific energy is varied, battery specific power is scaled consistently so that the assumed maximum discharge rate remains constant.

Uncertainty propagation Within the UMDO loop, each candidate battery-mass upper bound is evaluated under uncertainty in battery specific energy through Monte Carlo simulation. For each candidate design, battery specific-energy samples are generated from the uncertainty representation associated with the selected entry-into-service year using the DST sampling procedure shown in Fig. 31.

Each sampled battery specific-energy value is combined with the candidate battery-mass upper bound and supplied to the deterministic surrogate, which returns the corresponding optimised response over the full powertrain design space. The repeated Monte Carlo evaluations produce a distribution of possible objective-function values for the candidate design, representing the effects of plausible battery-technology outcomes on the optimised aircraft performance.

Statistical performance assessment The output distribution obtained for each candidate design is characterised through its sample mean and sample standard deviation. The mean represents the expected value of the objective function under uncertainty in battery specific energy, whereas the standard deviation quantifies the variability of the resulting aircraft performance.

A lower mean indicates improved expected performance, while a lower standard deviation indicates reduced sensitivity to uncertain battery-technology development. These statistical quantities allow the optimisation to distinguish between designs that perform favourably only for a restricted battery assumption and those that retain competitive performance over a broader range of plausible battery specific-energy values.

This general workflow is used throughout the optimisation campaigns described in the following sections. The individual cases differ in the trajectory formulation, objective function, design space, noise requirement, and battery-technology scenario adopted, while retaining the same overall procedure for deterministic database generation, surrogate construction, Monte Carlo uncertainty propagation, and statistical performance assessment.

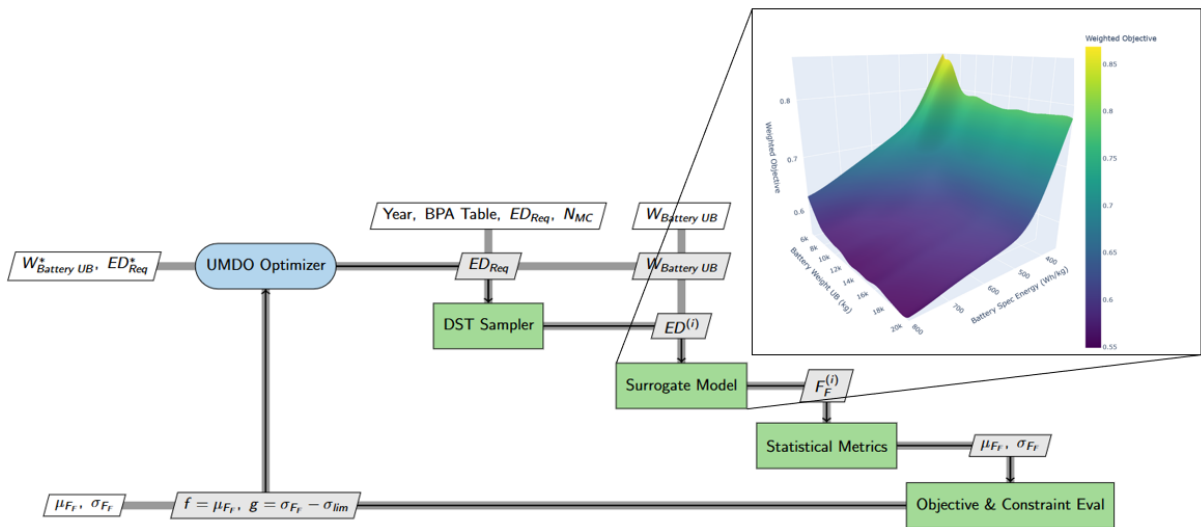


Figure 31: XDSM of the UMDO optimization problem.

6 Optimization Campaign

This section presents the optimisation results for the single- and multi-trajectory cases. In both cases, two maximum flyover EPNL constraints are considered: 77.5 and 75.5 dB.

6.1 Single-Trajectory UMDO Case

The single-trajectory case applies the uncertainty-based multidisciplinary design optimisation framework to the reference 1000 nm mission. A single representative trajectory is retained throughout the optimisation, allowing the effects of the considered technological uncertainties to be assessed independently of route or mission-profile variability.

The optimisation is conducted considering the design space reported in Table 5. The combined block-fuel and low-altitude fuel formulation defined in Equation 26 is adopted as the objective function. Two different maximum flyover EPNL constraint values equal to 77.5 and 75.5 dB are considered.

The use of a single trajectory isolates the effects of technological uncertainty from those associated with operational and trajectory variability. Differences between the deterministic and uncertainty-based solutions can therefore be related primarily to the representation of uncertain technology parameters and their influence on the optimal powertrain design. This case provides a controlled reference for the subsequent extension of the UMDO framework to multiple-trajectory formulations.

6.1.1 Deterministic Surrogate Analysis

Objective function The combined block-fuel and low-altitude fuel objective function variation with the design variables battery weight upper bound and battery specific energy is presented in figure 32 where the kriging surrogate contour generated with the aforementioned points is shown.

The contour shown in Figure 32 highlights that both block fuel and fuel below 900 m, and consequently the weighted objective, decrease as the battery specific energy increases. For a fixed battery specific energy, the objective initially decreases as the battery-weight upper bound increases, before reaching a plateau beyond a certain value. These results highlight the presence of two distinct design regimes in the battery-sizing process.

For low values of the imposed battery-weight upper bound, the optimal battery mass coincides with the maximum admissible value, indicating that the solution is directly constrained by this upper bound. In this regime, increasing the allowable battery weight enables the optimizer to exploit a greater degree of electrification, leading to a progressive reduction in block fuel. Beyond a certain threshold, however, the optimal battery mass no longer increases with the imposed upper bound and instead approaches an approximately constant value. The optimal battery weight evolution with the battery weight upper bound is shown in figure 33 and confirms what was mentioned.

Aircraft variables The analysis of the take-off weight (TOW) variation with the battery weight UB is consistent with the weighted objective function and battery weight variation. It suggests that, for increasingly higher battery weight upper bounds and once it reaches a sufficiently large value, the optimization enters a regime that is limited by the take-off weight rather than by the battery constraint itself. So, the battery weight plateau represents the optimal allocation of the available mass "budget" under an almost fixed take-off weight condition.

A particularly relevant outcome is that the plateau battery mass increases slightly with battery specific energy (figure 33). Although this may initially appear counter intuitive, it is in fact consistent with the nature of the optimization problem. If the battery were required to supply a fixed amount of energy, a higher specific energy would indeed imply a lower required battery mass. In the present case, however, the battery contribution is not prescribed a priori but is instead determined by the optimizer. As the battery specific energy increases, each unit of battery mass becomes more effective in providing mission energy, making the battery a more attractive option within the overall aircraft mass balance. Consequently, the optimizer allocates a larger share of the nearly fixed take-off weight to the battery, since doing so enables a greater substitution of fuel mass and therefore a lower block fuel and fuel below 900 m.

This shift toward greater electric-power utilization reduces the contribution required from the gas turbine. Accordingly, the turboshaft top-of-climb rating, $P_{TS,TOC}$, decreases monotonically with increasing battery specific energy for all battery-mass upper bounds. Conversely, for a fixed battery specific energy, $P_{TS,TOC}$ decreases as the battery-mass upper bound is raised while this constraint remains active. Once the optimal battery mass becomes independent of the imposed upper bound, $P_{TS,TOC}$ also approaches

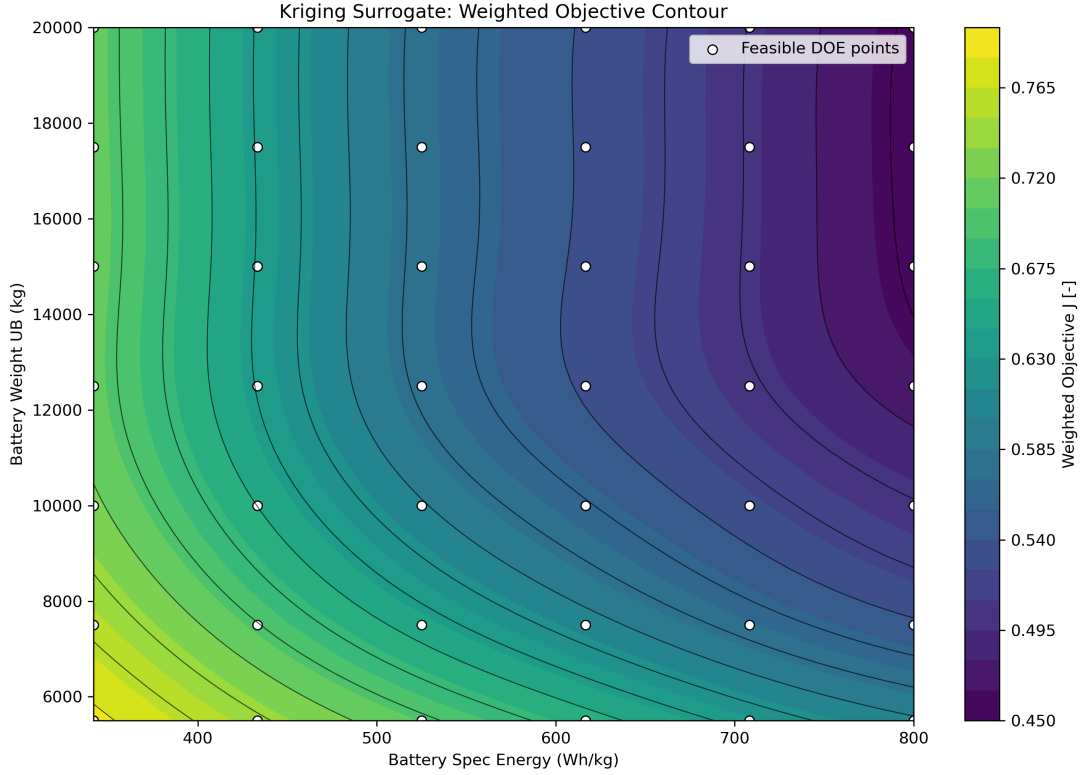


Figure 32: Single trajectory: objective function surrogate contour (flyover EPNL constraint equal to 77.5 dB).

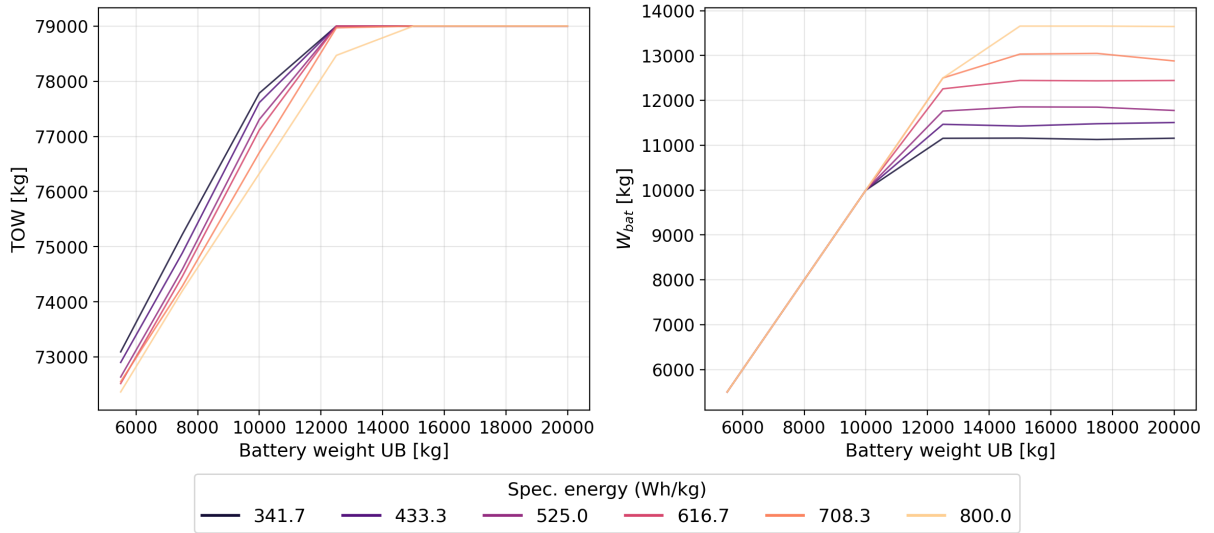


Figure 33: Variation of the TOW and Battery weight with the outer loop design variables (flyover EPNL constraint equal to 77.5 dB).

a plateau, following a trend similar to that observed for the objective function. A similar behaviour is observed for $P_{IB,Max}$, the inboard electric-machine peak design power.

The local hybridization factor shown in Figure 36 provides direct insight into the power transfer occurring within the hybrid-electric architecture. According to the adopted sign convention, negative values indicate operation of the IB electric machine in generator mode, with mechanical power from the turboshaft being converted into electrical power and supplied to the electrical network. During cruise, the local hybridization factor becomes progressively less negative as the battery specific energy increases, particularly for the larger battery-weight upper bounds. This indicates that the electrical branch becomes

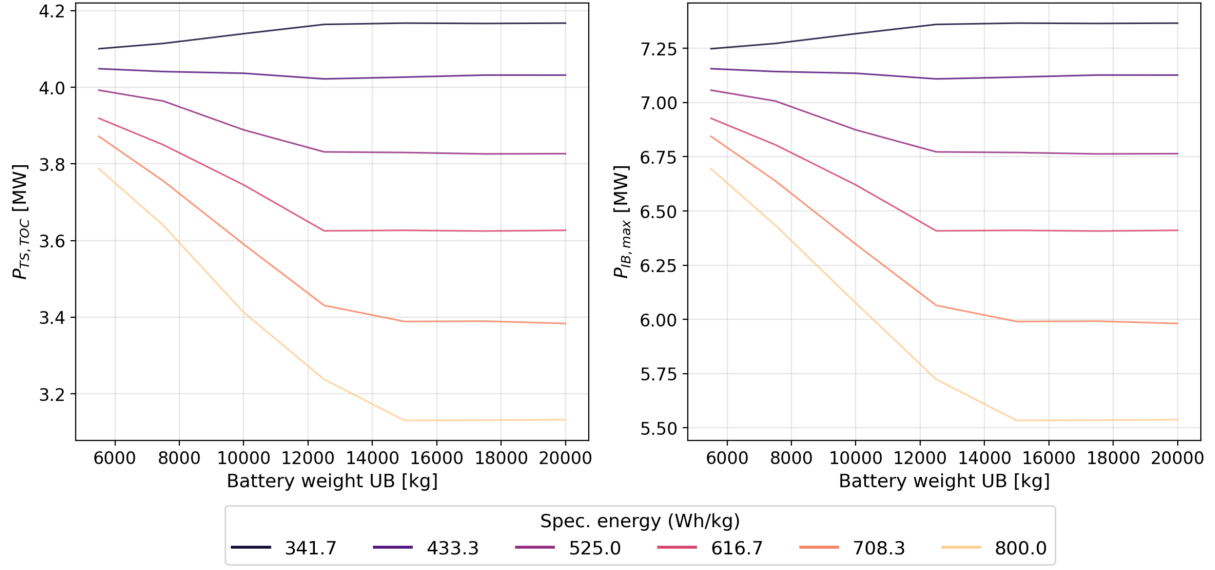


Figure 34: $P_{TS,TOC}$ and $P_{IB,max}$ variation with the outer loop design variables (flyover EPNL constraint equal to 77.5 dB).

less dependent on turboshaft-driven generation as battery performance improves, since a larger fraction of its power demand can be supplied directly by the battery.

This redistribution of power is also reflected in the inboard (IB), outboard (OB), and battery throttle histories shown in Figure 37. During cruise, the OB throttle remains relatively similar for a given battery specific energy, while the IB throttle becomes progressively less negative as the battery specific energy increases, particularly for the larger battery-weight upper bounds. Since the IB throttle is defined as

$$\delta_{IB} = \frac{2P_{in,IB,electric}}{P_{IB,max}},$$

this trend must be considered together with the simultaneous reduction in the rated IB power, $P_{IB,max}$. The combination of a less negative throttle and a lower $P_{IB,max}$ therefore results in a reduction in the magnitude of the negative power exchanged through the IB electric machine. This is consistent with the local hybridization factor and confirms a reduction in generator operation as battery specific energy increases. At the same time, the battery throttle increases during cruise, indicating a progressively greater utilization of the battery within the electrical propulsion system.

The resulting increase in battery participation is most clearly captured by the global hybridization factor shown in Figure 36, defined as the battery power expressed as a fraction of the total power supplied to the propulsion system. During cruise, the global hybridization factor increases systematically with battery specific energy, showing that the battery supplies an increasingly large fraction of the propulsion power. Therefore, while the local hybridization factor indicates a reduction in the electrical power generated from the turboshaft, the global hybridization factor simultaneously shows an increase in the contribution supplied directly by the battery. Together, these quantities demonstrate a progressive shift from turboshaft-generated electrical power toward battery-supplied electrical power as the battery specific energy increases.

The evolution of the thrust-split parameter ψ (equation 15), shown in Figure 35, complements the power-flow analysis by illustrating how the distribution of propulsive effort changes throughout the mission. Higher battery specific energies generally lead to larger values of ψ , indicating an increasing contribution of the OB propulsors to the total required thrust. This effect is particularly visible during the climb and descent phases, where the thrust split varies more markedly than during cruise. These phases are especially relevant to the adopted objective function, which assigns a larger weight to fuel consumption below 900 m through the FB_{900} contribution. The optimisation therefore has a stronger incentive to modify the distribution of thrust between the IB and OB propulsors in the airport vicinity, where such changes can contribute directly to reducing the more heavily weighted low-altitude fuel consumption. Together with the evolution of the local and global hybridization factors and throttle settings, these results show that increasing battery specific energy affects both the source of the electrical power and its distribution within the hybrid-electric propulsion architecture.

The redistribution of thrust between the IB and OB propulsors also changes their operating conditions. As the thrust split varies, the propellers move to different points on their performance maps, leading to small changes in propeller efficiency. As shown in the first and second rows of Figure 35, the efficiency decreases slightly as the battery specific energy increases. However, this effect remains limited, indicating that the main consequence of increasing battery specific energy is the redistribution of power within the propulsion system rather than a significant loss in propeller efficiency.

Results presented in previous paragraphs were obtained considering an EPNL constraint of 77.5 dB. To assess the effect of a more restrictive noise constraint, a second case with an EPNL limit of 75.5 dB was also investigated. The design trends discussed previously remain valid under this constraint. Nevertheless, as discussed in [33], the stricter noise limit leads to lower propeller rotational speed and tangential tip Mach number, M_{tip}^* . This shifts propeller operating points away from the peak-efficiency region, resulting in lower propeller efficiency, which explains an increase of both BF and FB_{900} with stricter noise constraints.

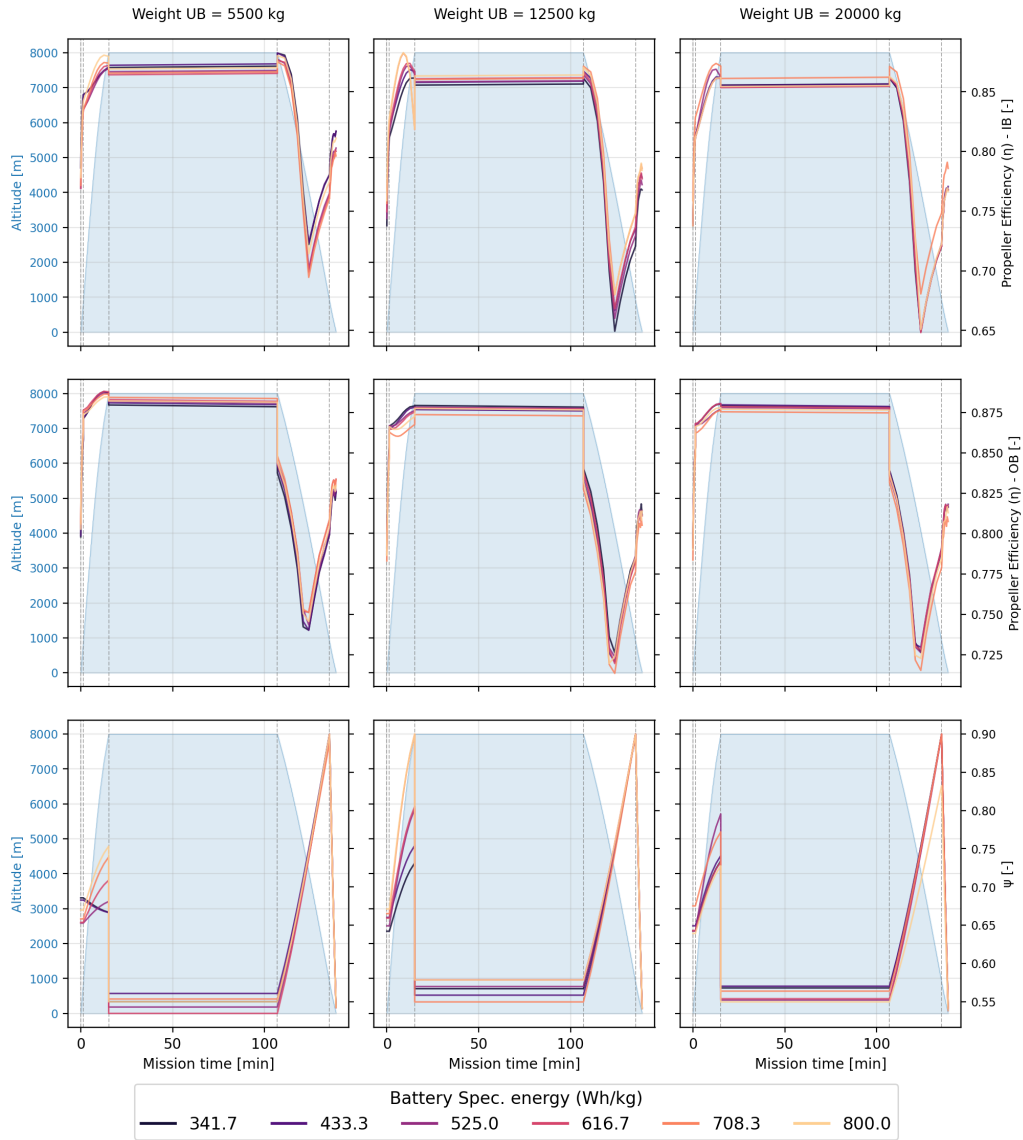


Figure 35: Propellers efficiency and ψ , along the mission (flyover EPNL constraint equal to 77.5 dB).

6.1.2 Uncertainty-Based Analysis

The robust single-mission optimization is assessed for two flyover-noise constraints, 75.5 and 77.5 dB, considering the 2030 and 2040 technology scenarios. The results are presented in Figure 38. The x -axis represents the imposed upper limit on the standard deviation, σ , while the left axes report the

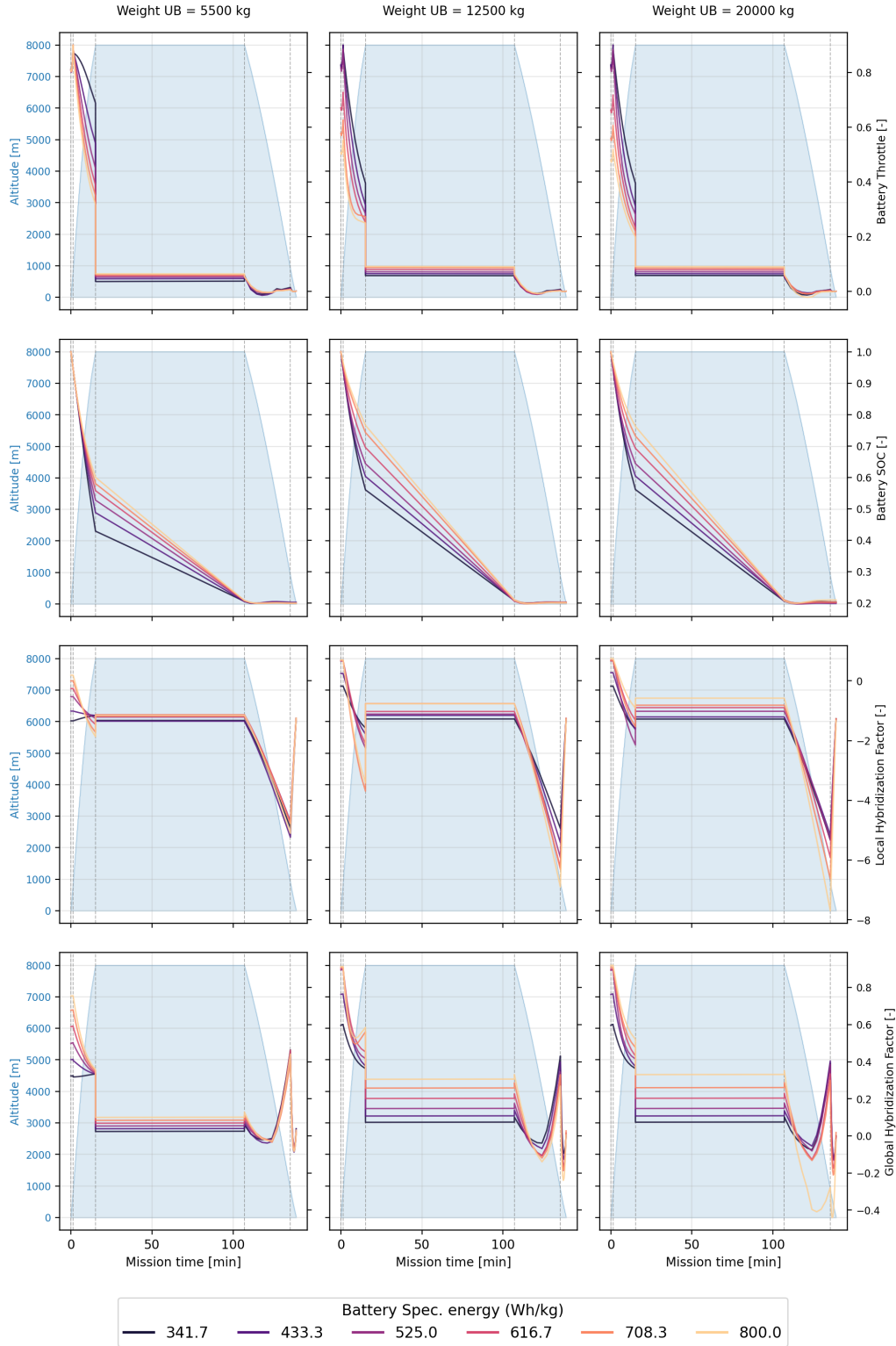


Figure 36: Battery throttle, battery SOC, local and global hybridization factor along the mission (flyover EPNL constraint equal to 77.5 dB).

expected values of the weighted fuel objective J , block fuel BF , and fuel consumed below 900 m, FB_{900} . The right axes show the battery energy density selected by the optimizer, ED_{target} , together with the corresponding energy density obtained after uncertainty propagation, ED_{achieved} . The 2030 and 2040 technology scenarios are shown in the left and right columns, respectively.

For both technology years and noise constraints, increasing the admissible standard deviation produces

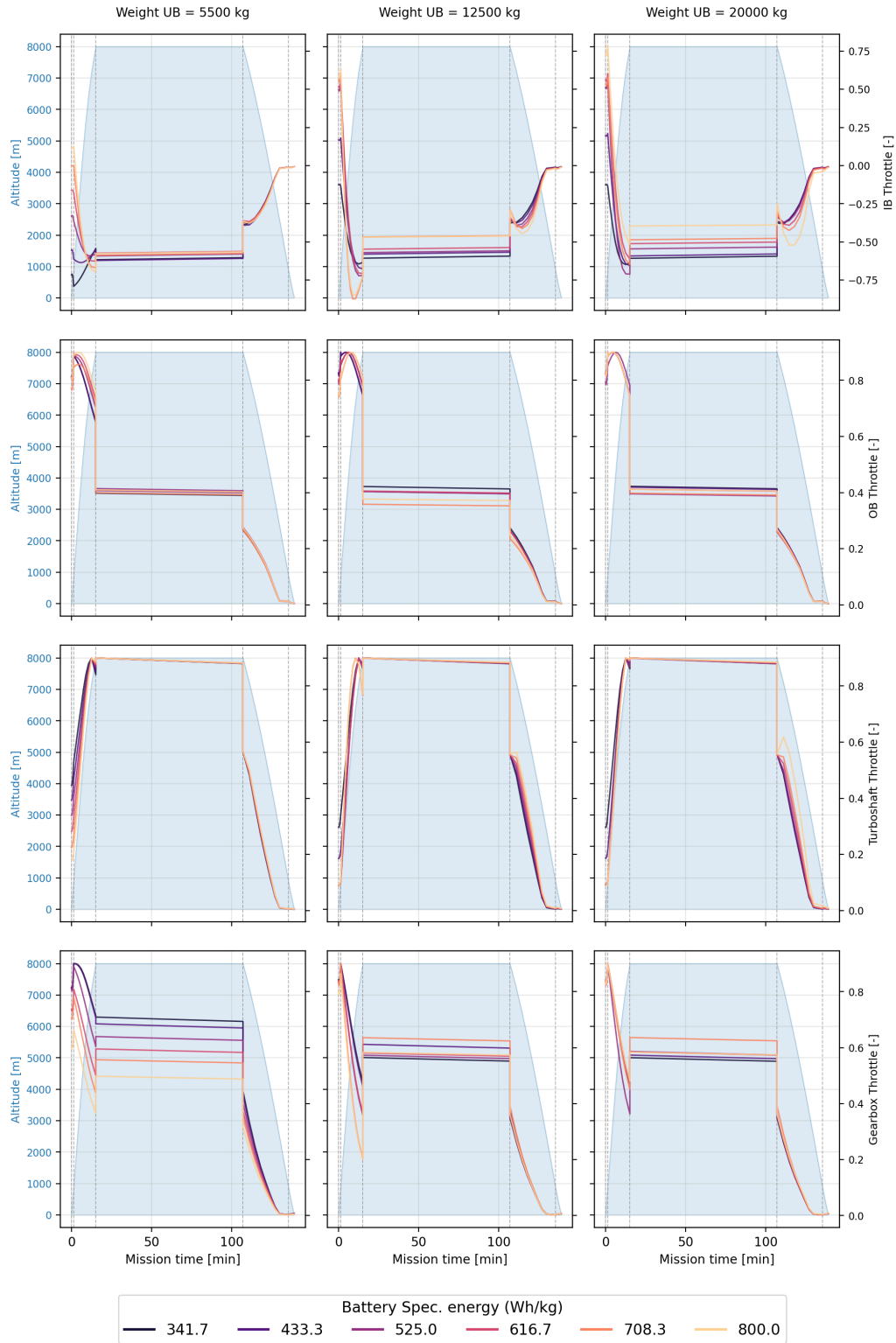


Figure 37: IB, OB, turboshaft and gearbox throttles along the mission (flyover EPNL constraint equal to 77.5 dB).

a clear trade-off between robustness and expected mission performance. At low values of σ , the optimizer is restricted to comparatively low battery energy densities, resulting in higher values of J , BF , and FB_{900} . Relaxing the standard-deviation constraint allows progressively larger values of ED_{target} to be selected and, as a consequence, all three fuel-related metrics decrease monotonically. Therefore, designs with lower expected fuel consumption are obtained at the expense of accepting greater uncertainty in the

solution.

This improvement is, however, not sustained indefinitely. A clear transition towards a saturation regime is observed at approximately $\sigma = 0.04$ – 0.05 for the 2030 scenario and $\sigma = 0.03$ – 0.035 for 2040. Beyond these values, the reductions in J , BF , and FB_{900} become negligible. This behaviour coincides with the rapid increase of ED_{target} towards the imposed technology ceiling of 800 Wh/kg and, more importantly, with the saturation of ED_{achieved} . Consequently, further relaxation of the robustness requirement permits the optimizer to request a higher nominal battery energy density, but does not provide a corresponding increase in the energy density effectively obtained under uncertainty, strongly limiting an additional fuel saving benefit.

The difference between ED_{target} and ED_{achieved} is particularly relevant in this regime. For small σ , the two quantities remain relatively close, indicating that the selected technology target can be largely retained after uncertainty propagation. As increasingly improved battery technologies are selected, the two curves progressively separate. In 2030, ED_{achieved} levels off at approximately 590 Wh/kg despite ED_{target} reaching 800 Wh/kg, whereas the 2040 scenario reaches approximately 700 Wh/kg. Accordingly, the nominal target alone does not fully represent the technological benefit, since uncertainty in battery performance limits the energy density that can effectively be exploited.

The comparison between technology years shows a substantial benefit from the more advanced 2040 battery scenario. The higher achievable energy density leads to lower asymptotic values of all fuel related quantities. Depending on the noise constraint, the block fuel decreases from approximately 3.35–3.39 t in the 2030 saturation regime to approximately 2.95–3.00 t in 2040, while FB_{900} decreases from approximately 60–61 kg to 51–52 kg. The weighted objective follows the same trend. Furthermore, the 2040 solutions reach their saturation regime at a smaller value of σ , indicating that most of the attainable performance benefit is already recovered while imposing a comparatively tighter uncertainty limit. Finally, the more restrictive 75.5 dB noise constraint consistently results in slightly higher values of J , BF , and FB_{900} than the 77.5 dB case.

6.2 Multiple-Trajectory UMDO Case

The multi-trajectory case applies the uncertainty-based multidisciplinary design optimisation framework to three missions associated with the RIX airport. The decision is to optimize the aircraft full powertrain (table 5) considering the statistically most frequent mission and two missions RIGA TO F78 L F137 and RIGA TO F597 L F133 as described in table 4. Moreover, the aircraft is constrained to fly the reference mission, that the aircraft was optimised to fly in the single trajectory case (section 6.1).

The optimisation is conducted using the design space reported in Table 5. The combined block-fuel and low-altitude fuel formulation defined in Equation 26 is adopted as the objective function. Two different maximum flyover EPNL constraint values equal to 77.5 and 75.5 dB are considered.

The multi-trajectory formulation introduces trajectory variability into the optimisation and favours a common powertrain design that provides consistent performance across different mission conditions, rather than one tailored to a single trajectory.

6.2.1 Deterministic Surrogate Analysis

Objective function The deterministic multi-trajectory surrogate shows the same general trends as the single-trajectory case. As shown in Figure 39, the weighted objective decreases as battery specific energy increases and as the battery-weight upper bound is relaxed. However, the benefit of increasing the battery-weight upper bound becomes progressively smaller once the optimal battery mass starts to level off. Figure 40 shows that, beyond this point, allowing a heavier battery does not lead to a significant increase in the battery mass selected by the optimizer, while the take-off weight also approaches an almost constant value.

Aircraft variables The powertrain sizing variables follow a similar trend. As battery specific energy increases, both the turboshaft top-of-climb rating and the peak inboard electric-machine power decrease, as shown in Figure 41. For a fixed battery specific energy, increasing the battery-weight upper bound initially leads to a reduction in both power ratings. Once the optimal battery mass reaches its plateau, however, both power ratings also tend to stabilize. This is consistent with the increased use of the battery and the lower dependence on turboshaft-driven electrical generation.

The mission profiles presented in Annex (section 10) help to show how this common powertrain is operated over the different trajectories. The same general behaviour observed in the single-trajectory case is also found here. As battery specific energy increases, the global hybridization factor increases, while the

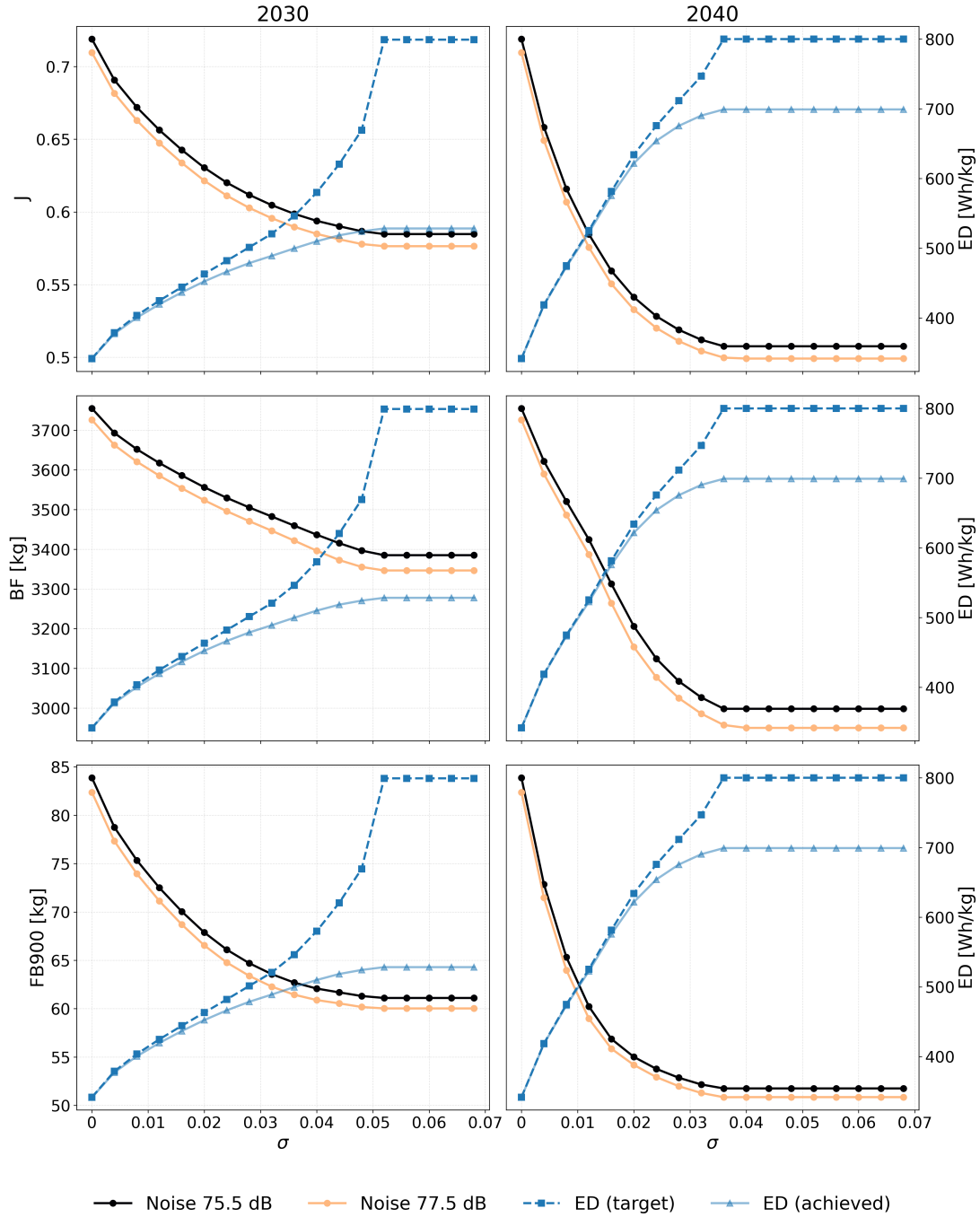


Figure 38: Expected values of J , BF , $FB900$, ED_{target} and $ED_{achieved}$ vs σ .

local hybridization factor generally becomes less negative. According to the adopted sign convention, this means that the inboard electric machine operates less in generator mode and that a larger share of the electrical power is supplied directly by the battery. The throttle histories show the same redistribution of power.

It is important to note that the local hybridization factor and the thrust-split parameter ψ are mission-control design variables and are optimized for each trajectory. Therefore, although the aircraft and powertrain design is common to all missions, the optimizer can adapt the degree of hybridization and the distribution of thrust between the inboard and outboard propulsors to the characteristics of each trajectory. Differences in the corresponding mission profiles are therefore expected and reflect the operating strategy selected for each mission rather than changes in the underlying aircraft design.

The Most Frequent trajectory and RIGA.TO.F597.L.F133 show relatively similar behaviour during cruise. The propeller efficiencies remain almost unchanged with battery specific energy, while clearer

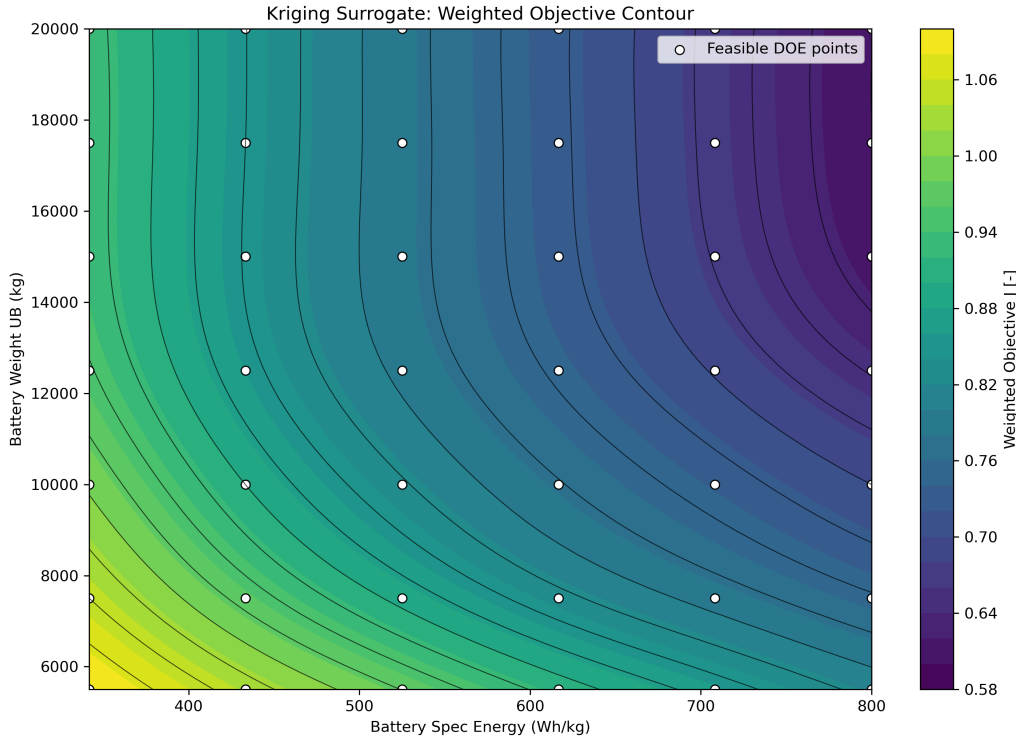


Figure 39: Multi trajectory: objective function surrogate contour (flyover EPNL constraint equal to 77.5 dB).

differences can be seen in the hybridization factors and in the electrical-system throttles. This suggests that the main effect of improving the battery is not a significant change in propeller operation, but rather a redistribution of the power supplied by the battery and by the turboshaft.

The RIGA_TO_F78.L_F137 trajectory shows a stronger dependence on battery technology. In this case, the cruise values of ψ , the hybridization factors, and some of the propulsion throttles vary more noticeably with battery specific energy, particularly for the larger battery-weight upper bounds. This indicates that the optimal power split and hybridization strategy for this trajectory are more sensitive to battery specific energy.

Looking across the three optimized trajectories, differences between the battery specific-energy cases are also visible during climb and descent, where ψ , the hybridization factors, and the resulting battery and propulsion-system operation vary more strongly. This is consistent with the single-trajectory results, where these phases were particularly affected by the inclusion of FB_{900} in the objective function. Since the mission-control variables are optimized independently for each trajectory, the optimizer can adapt the power split and hybridization strategy to the characteristics of each mission while maintaining the same aircraft and powertrain design.

The 1000 nm reference mission should be interpreted separately. This mission is included only as a feasibility requirement and does not contribute directly to the objective function. Its control histories should therefore not be interpreted as a power-management strategy optimized to minimize the objective for that mission. Instead, they mainly show how the common aircraft and powertrain can be operated while maintaining feasibility for the reference mission.

6.2.2 Uncertainty-Based Analysis

The multi-trajectory optimization presents the same general robustness–performance trade-off observed for the single-mission case. As the admissible standard deviation σ increases, the optimizer selects progressively higher values of ED_{target} , leading to higher ED_{achieved} and lower values of J , relative BF , and relative FB_{900} (Figure 42). The improvement is considerably stronger for the 2040 technology scenario, where ED_{achieved} approaches approximately 700 Wh/kg and allows a larger fraction of the requested battery capability to be exploited. At high values of σ , the increasing separation between ED_{target} and ED_{achieved} limits the additional performance benefit, particularly in 2030, where the target energy density reaches the imposed upper bound while the achieved value remains substantially lower.

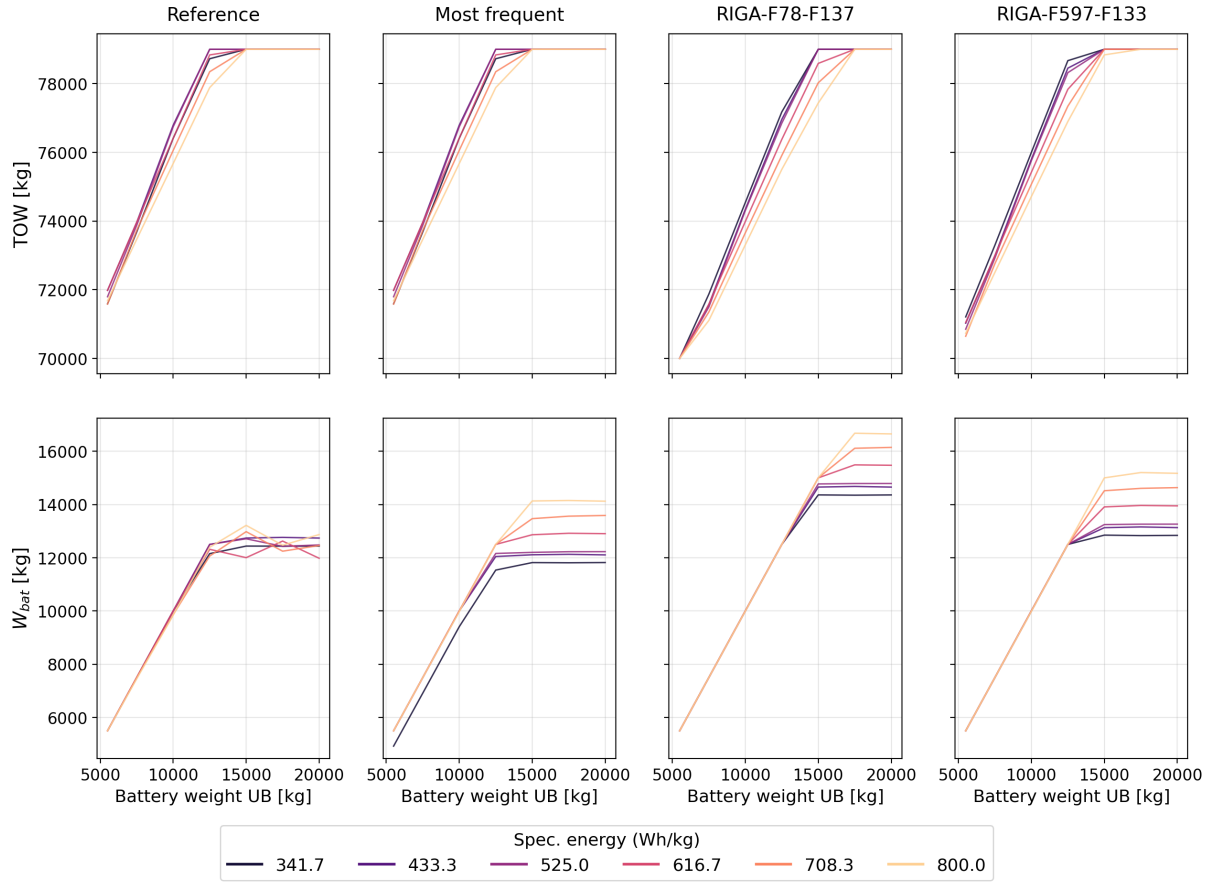


Figure 40: Variation of the TOW and Battery weight with the outer loop design variables (flyover constraint 77.5 dB).

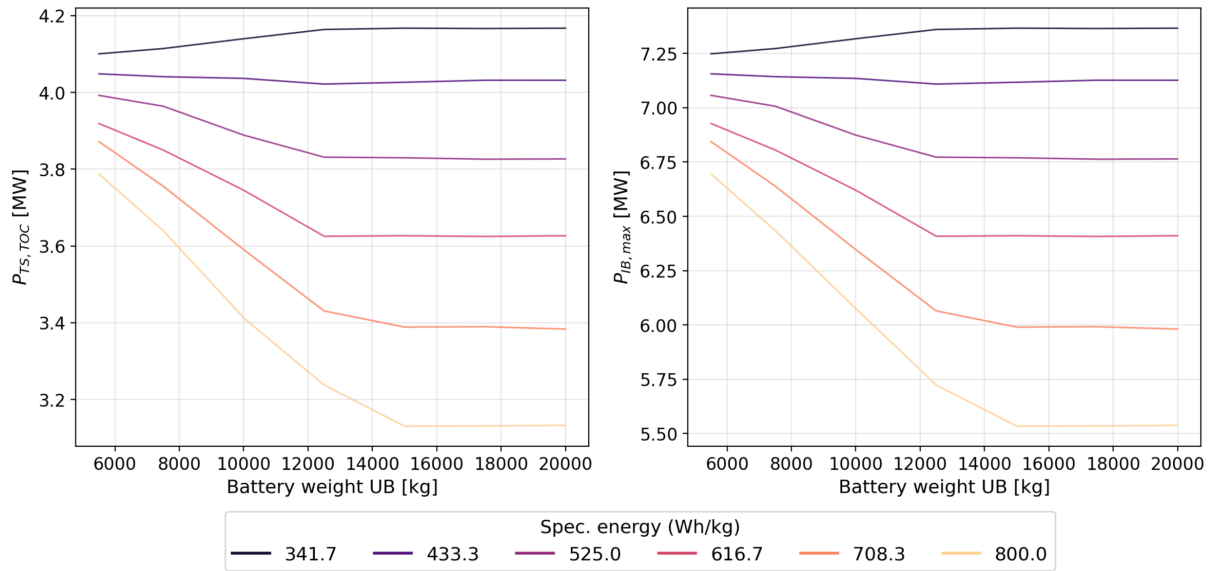


Figure 41: $P_{TS,TOC}$ and $P_{IB,max}$ variation with the outer-loop design variables for the multi-trajectory optimization (flyover EPNL constraint equal to 77.5 dB).

The more restrictive 75.5 dB noise constraint consistently produces slightly higher values of the fuel-related metrics than the 77.5 dB case. Nevertheless, the effect of the technology year and of the admissible uncertainty is considerably larger than the difference between the two noise constraints.

The trajectory-level results in Figure 43 show that the aggregate trends are reflected across the

different missions, although the magnitude of the benefit is trajectory dependent. Block fuel generally decreases with increasing σ , with considerably larger reductions observed in the 2040 scenario. This confirms that the benefit associated with the higher achievable battery energy density extends across different payload–range operating conditions rather than being restricted to a single trajectory.

The 1000 nm reference mission requires a different interpretation, as it is not directly incorporated into the optimization objective but rather imposed as a feasibility requirement. Consequently, because the optimizer was driven only to satisfy the prescribed constraints, the mission-control design variables (HF^* and ψ), and therefore their corresponding mission profiles, may not be optimal for flying this particular trajectory. This, in turn, affects the observed trends in BF and FB_{900} . The effect is particularly evident for FB_{900} , which exhibits a noisier evolution for this mission, especially under the 77.5 dB constraint.

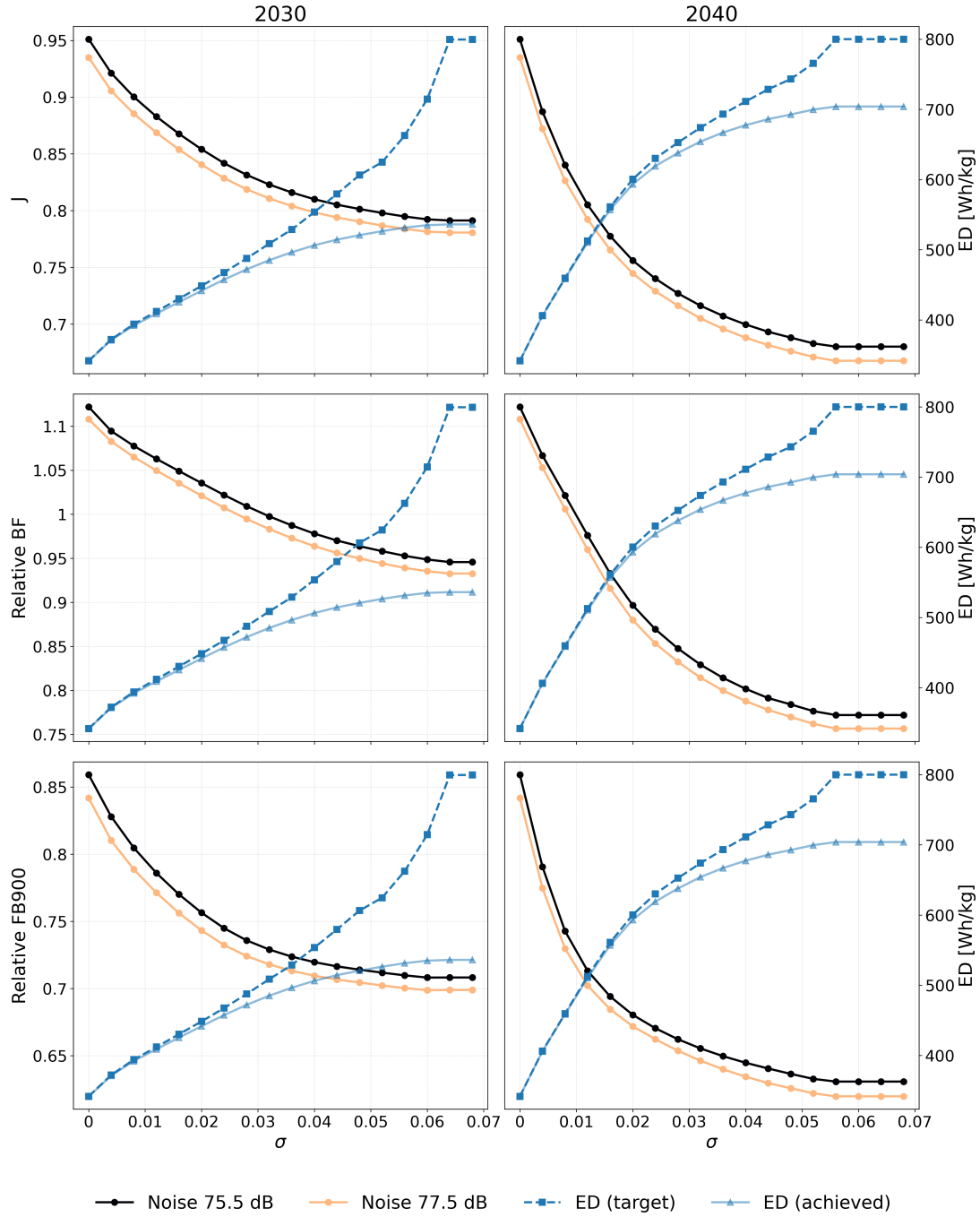


Figure 42: Expected values of J , BF , FB_{900m} , ED_{target} and $ED_{achieved}$ vs σ .

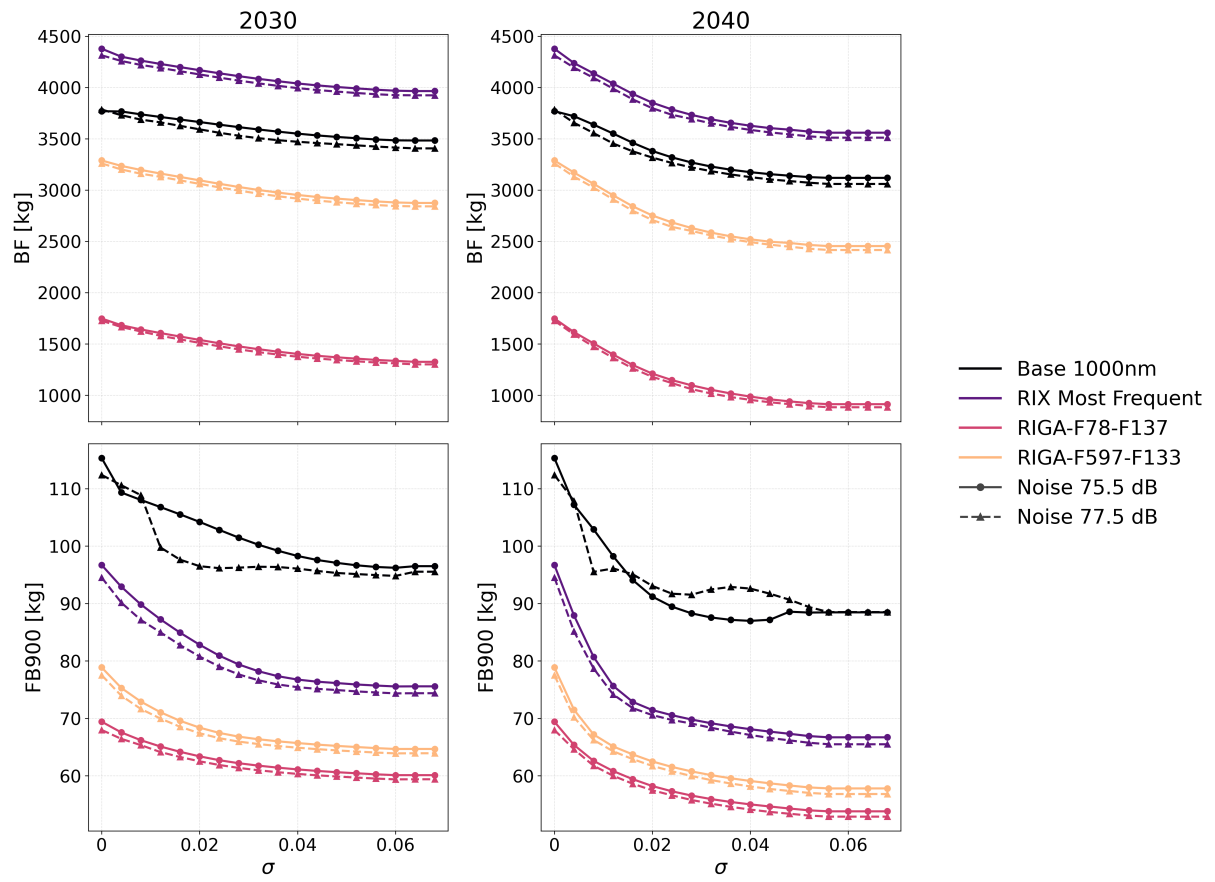


Figure 43: Expected values BF , FB_{900} for the trajectories considered.

7 High-fidelity Verification

7.1 High-fidelity Aerodynamic Verification

7.1.1 Comparison of MDO polars and surrogate prediction

The multidisciplinary optimisation (MDO) process utilised an aerodynamic surrogate constructed using aerodynamic coefficients generated with the panel-method solver VSPAero. This approach provided a computationally efficient basis for rapid design-space exploration and MDO, but ignored multiple important flow-physics effects captured by CFD. It is therefore useful to compare the aerodynamic representation employed during the MDO process with the new CFD-based surrogate, while accounting for the different component scope of the two models.

The VSPAero polar considered here represents the wing and strut. For the INDi-2 airframe, the corresponding coefficients are

$$\begin{aligned} C_{L,w+s} &= 0.576 + 6.82\alpha, \\ C_{D,w+s} &= 0.0161 + 0.1615\alpha + 0.976\alpha^2, \end{aligned} \quad (33)$$

where α is expressed in radians. These coefficients were obtained from the VSPAero-based surrogate evaluated at $AR = 21.617$ and are applied in the MDO with $S_{\text{ref}} = 122 \text{ m}^2$. The wing-and-strut polar does not include the fuselage, nacelles, HTP or VTP, and it does not depend explicitly on Mach or Reynolds number. Within the aircraft-level MDO model, the wing-and-strut contribution is supplemented by separate fuselage and tail terms. At the cruise condition considered here, $h = 8000 \text{ m}$ and $M = 0.60$, the complete aircraft-level polar with zero elevator deflection is

$$C_L = 0.561 + 7.51\alpha, \quad C_D = 0.0293 + 0.160\alpha + 1.118\alpha^2. \quad (34)$$

This polar is untrimmed. The constant aircraft parasite-drag contribution is $C_{D0,\text{aircraft}} = 0.0133$, of which 0.0089 is associated with the fuselage and 0.0044 with the HTP and VTP. No explicit nacelle-drag contribution is included in this aerodynamic build-up.

The CFD-based surrogate has a different component definition. The integrated aerodynamic coefficients include the wing, strut, fuselage and nacelles, but exclude both the HTP and VTP. Consequently, the closest available MDO-side representation for comparison with the CFD polar is obtained by adding the fuselage parasite-drag contribution to the wing-and-strut polar, without adding either tail:

$$\begin{aligned} C_{L,w+s+f} &= 0.576 + 6.82\alpha, \\ C_{D,w+s+f} &= 0.0250 + 0.1615\alpha + 0.976\alpha^2. \end{aligned} \quad (35)$$

This provides a substantially closer component match, although the two representations are still not fully equivalent. In particular, the CFD model includes the direct aerodynamic forces acting on the nacelles and their installation effects, whereas the MDO-side representation contains no explicit nacelle term and represents the fuselage through a parasite-drag estimate.

A further distinction is introduced by the propulsion state represented by each polar. The aerodynamic representation employed in the MDO does not contain explicit propeller-blowing effects, whereas the minimum operating condition represented by the CFD surrogate retains finite propeller loading. The minimum physical thrust coefficient in the CFD database is $C_t = 0.05$. Consequently, an exactly unblown CFD equivalent is not available. The CFD half-aircraft calculations use a reference area of 61.2 m^2 , corresponding to 122.4 m^2 for the complete symmetric aircraft, compared with 122 m^2 in the MDO. The resulting difference in coefficient normalisation is approximately 0.33% and is small compared with the aerodynamic differences discussed below. The discrepancy between the two aerodynamic predictions is substantial and varies systematically with aerodynamic loading rather than appearing as a constant drag offset. When the CFD polar is compared directly with the VSPAero wing-and-strut representation, the difference is approximately $\Delta C_D = 0.01568$ at $C_L \approx 0.669$ and increases to approximately $\Delta C_D = 0.04505$ at $C_L \approx 1.171$. Including the fuselage parasite-drag contribution reduces these differences to approximately 0.00678 and 0.03615, respectively. Thus, at the lower loading approximately 57% of the difference is associated with the component build-up, whereas at the upper end of the investigated range the corresponding fraction is approximately 20%.

The residual discrepancy remains strongly dependent on aerodynamic loading. Since the fuselage contribution introduces an approximately constant drag increment, while the remaining difference increases markedly with C_L , a simple additional constant drag correction would not adequately reconcile the two

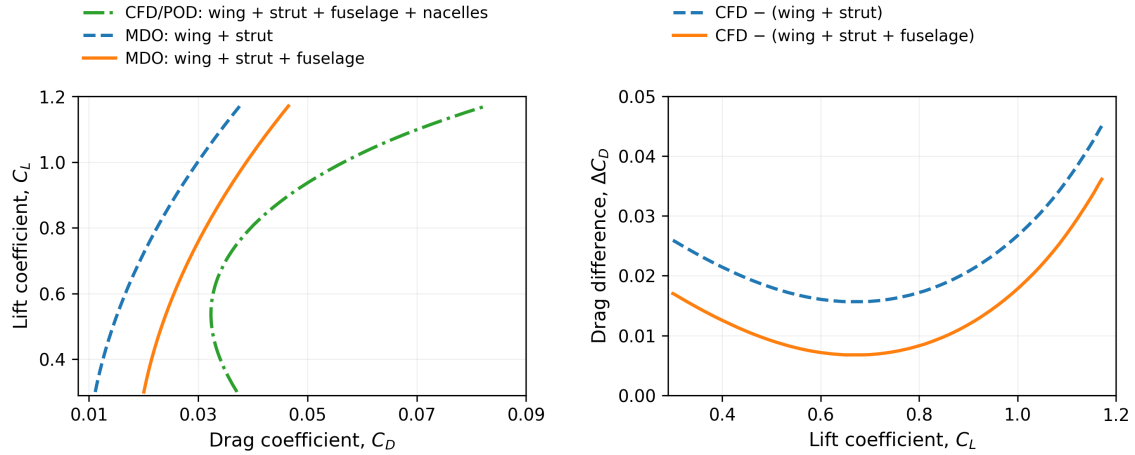


Figure 44: CFD-based and MDO aerodynamic polars (left) and drag-coefficient differences at matched lift coefficient (right).

representations over the complete investigated range. The remaining difference is consistent with the substantially different aerodynamic fidelity and component representation of the two modelling approaches, although the polar comparison alone cannot uniquely separate the individual contributions. The VSPAero model cannot resolve the viscous and compressible flow represented directly within the RANS calculations. The CFD surrogate incorporates skin-friction and pressure-drag contributions associated with boundary-layer development, separation and locally compressible flow, together with the effects of shock formation and shock-boundary-layer interaction where these occur. These additional physical mechanisms provide a plausible explanation for part of the increasing discrepancy observed at higher lift coefficients, where stronger aerodynamic loading can amplify viscous separation and compressibility-related penalties. The direct nacelle forces and nacelle-airframe installation effects represented in the CFD calculations can also contribute to the difference.

The finite propeller loading represented in the CFD surrogate introduces an additional contribution. The resulting flow acceleration and swirl modify the local velocity magnitude and effective incidence over regions of the wing immersed within the propeller slipstreams. These changes can alter pressure distributions, shock behaviour and separation, and therefore influence the resulting drag polar. Consequently, the observed difference should not be interpreted solely as a measure of VSPAero versus RANS model discrepancy, but rather as the combined effect of aerodynamic-model fidelity, component representation and propeller-airframe interaction.

The component build-up also has an important effect on the aerodynamic-efficiency prediction. Maximising the corresponding analytical polars gives a maximum C_L/C_D of approximately 36.2 for the wing and strut alone and 25.8 when the fuselage parasite drag is included. The corresponding lift coefficient at maximum efficiency increases from approximately $C_L = 0.67$ to $C_L = 0.93$. For comparison, the complete zero-elevator aircraft-level polar gives a maximum C_L/C_D of approximately 23.6 at $C_L \approx 1.09$. These values are maxima of the respective aerodynamic polars rather than the aerodynamic efficiency at a trimmed mission cruise point. The latter depends on the aircraft mass, longitudinal trim state and propeller operating condition.

The comparison assesses the aerodynamic representation of the selected configuration; it does not establish the performance attainable after further refinement of the installed geometry. The implications of the aerodynamic results for subsequent design refinement are discussed in Section 7.1.6.

7.1.2 Influence of thrust on aerodynamic performance

As the aerodynamic polar employed within the original MDO framework does not include the effects of propeller blowing, it is useful to examine how the aerodynamic performance of the configuration changes as propulsive loading is increased. The present analysis therefore considers the variation in lift, drag and maximum aerodynamic efficiency with increasing propeller thrust. The baseline propeller geometry is retained, with the inboard and outboard propeller diameters and tip Mach numbers held fixed, while the normalised thrust inputs are varied together across both propulsor groups. As noted above, the minimum normalised setting retains finite physical propeller loading rather than representing

an unblown condition. This isolates the aerodynamic influence of increasing global propulsive loading without introducing additional geometric or rotational-speed effects. The resulting trends provide an initial indication of how distributed propulsion modifies the aircraft-level aerodynamic response and establish a baseline against which the relative allocation of thrust between the inboard and outboard propellers can subsequently be assessed. The highest investigated propeller loading produces the largest

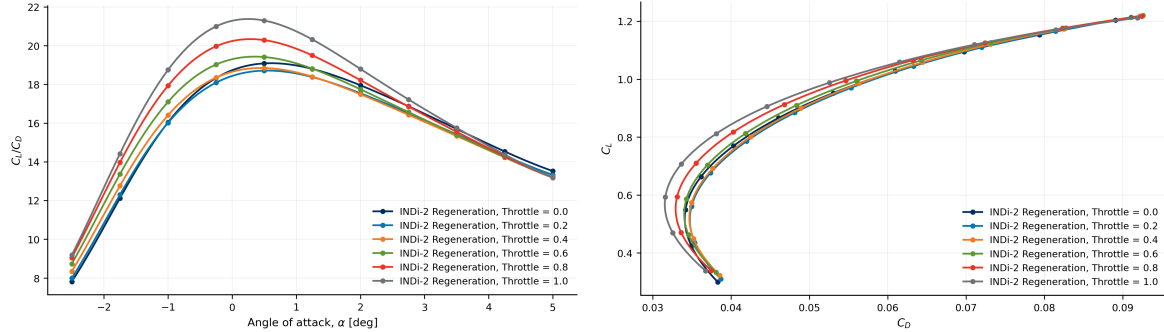


Figure 45: Aerodynamic efficiency curves (left) and lift-drag polars for different normalised propeller loadings/throttles (right).

peak aerodynamic efficiency. The resultant polar shifts towards a lower C_D over much of the moderate lift region. The influence of blowing is most evident near the maximum $\frac{C_L}{C_D}$. From Figure 45, the peak aerodynamic efficiency increases from approximately 19.1 at the minimum investigated loading to 21.4 at the highest loading, corresponding to an increase of just over 12%. However, the response is not monotonic across all intermediate loading settings. The angle of attack corresponding to peak aerodynamic efficiency remains broadly similar across the investigated loading settings. The lift-drag polar indicates that the improvement is non-uniform across a full sweep of lift coefficient. The largest separation occurs at the intermediate values of C_L , where the highest propeller-loading cases achieve set lift at a lower drag coefficient. However, as C_L increases towards the upper region of the polar, the curves approach one another and the drag differences diminish.

This behaviour is consistent with the distributed propellers modifying the local velocity and incidence over the wing through a combination of flow acceleration and swirl. Increasing the normalised loading of both the inboard and outboard propeller groups strengthens these effects and changes the spanwise loading. The resulting reduction in drag at comparable lift indicates that the propeller–wing interaction can provide an overall aerodynamic benefit when the propulsion system is operated at higher loading. These changes describe the aerodynamic response of the modelled configuration; their effect on aircraft energy consumption also depends on the required propulsive power and the corresponding trimmed operating state.

7.1.3 Sensitivity to nacelle surface-force contribution

The aircraft configuration necessitates the integration of nacelles to represent physical mounting and installation requirements of the propulsion systems. Retaining these components within the CFD is essential as the interaction with the strut-braced wing introduces installation effects that would otherwise remain unaccounted for. This is particularly relevant for the large inboard nacelle associated with the thermal-electric propulsor, which is positioned within the wing–strut junction region. During cruise, the nacelle and strut form a locally constricted flow passage. The reduced effective flow area accelerates the flow and can lead to the formation or strengthening of transonic shock structures. Removing the nacelles from the CFD analysis would therefore ignore an important aspect of true LARW-DHEP configurations.

Although the nacelle geometry is fixed within the CFD database, the POD-based surrogate methodology permits the aerodynamic contribution of individual surfaces to be isolated. The lift and drag associated specifically with the nacelle surfaces can therefore be extracted separately from the integrated coefficients of the modelled configuration. A nacelle factor f is applied to both contributions to assess their influence on aerodynamic performance. A factor of $f = 1$ retains the complete nacelle surface-force contribution, while $f = 0$ excludes the direct nacelle lift and drag from the coefficient calculation. The CFD geometry and flow field remain unchanged, so the nacelle-induced interference effects on the other surfaces are retained. Figure 46 demonstrates the resulting variation in total lift, drag and aerodynamic efficiency as the nacelle factor is varied at the cruise condition. The total lift coefficient remains com-

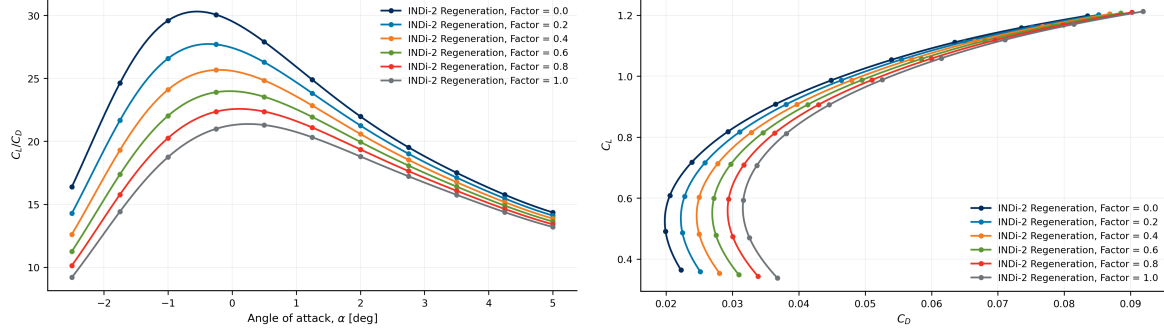


Figure 46: Aerodynamic efficiency curves (left) and lift-drag polars across a range of nacelle factors (right).

paratively insensitive to the nacelle contribution, exhibiting only minor variation across the investigated range. Drag, by contrast, shows a substantially stronger dependence. Reducing the retained nacelle-force contribution produces a noticeable reduction in the integrated drag and thus a corresponding increase in lift-to-drag ratio. The case used within this analysis is the maximum propeller-loading condition presented in Figure 45. Reducing the nacelle factor from one to zero increases the predicted maximum aerodynamic efficiency from 21.4 to 30.3, corresponding to a 41.6% increase. These values delimit the present post-processing exercise, not the performance bounds of a practical nacelle redesign, which would also modify the surrounding flow and the forces on other surfaces. Nevertheless, the magnitude of the variation demonstrates that nacelle integration remains an important contributor to the cruise aerodynamic performance of the configuration and a target for further refinement, as discussed in Section 7.1.6.

7.1.4 Actuator disk and actuator line methodologies

Throughout the presented simulations, the propellers were represented using actuator-based source-term methods. This avoided the requirement to resolve the physical blade geometry within the computational mesh. Both the actuator disc (AD) and actuator line (AL) formulations were implemented by introducing additional momentum and energy source terms into the compressible viscous RANS equations. The actuator forcing represents the full rotor loading generated by the propeller. This formulation allows the propulsive influence to be embedded directly within the existing unstructured CFD mesh without introducing rotating, sliding or overset mesh interfaces.

In the present implementation, the AL formulation provides the underlying representation from which both modelling approaches are constructed. Each propeller blade is replaced by a rotating one-dimensional line extending from the hub towards the blade tip and discretised into a series of radial actuator points. The prescribed propeller loading is defined using the radial gradients of thrust and power coefficient, $\frac{dC_T}{dr_n}$ and $\frac{dC_P}{dr_n}$, where r_n is the normalised radial coordinate. These distributions are integrated over the radial interval represented by each actuator point and converted into local thrust and power contributions using the conventional propeller-coefficient definitions. The power contribution is subsequently converted to torque, allowing the loading at each actuator point to be decomposed into axial and tangential force components. As the input loading describes the complete rotor, the resulting annular forces are divided by the number of blades before being assigned to the individual actuator lines.

The actuator forces cannot be simply applied directly to a single computational node without introducing strong mesh dependence and numerical discontinuities. Therefore the force for each virtual actuator point is distributed over a finite surrounding region using a Gaussian smoothing kernel. The weighting applied to each mesh point depends on its distance from the corresponding actuator point and is restricted by a finite support radius. The Gaussian weights are additionally scaled using a local connected-volume estimate and normalised over the support region so that the complete force generated by the actuator point is retained after projection within the mesh. Where the support regions of multiple actuator points overlap, their weighted contributions are collated. For the AL formulation, this process is repeated as the blades rotate, such that both the location of the actuator points and the associated tangential force directions vary with time. The resulting source term therefore retains the temporal variation associated with the prescribed rotating blade loading, without explicitly modelling the solid blade surfaces.

Within the unified formulation, the actuator-disc representation is constructed directly from the

actuator-line model through temporal averaging. The AL force field is evaluated at a prescribed number of angular blade positions over one complete revolution, and the resulting force vectors are averaged at each mesh point. This produces a time averaged source distribution representative of the mean rotor loading. The AD and AL approaches therefore share the same prescribed spanwise loading, force-projection, and source-term formulation; their principal distinction is whether the blade loading is retained as a time-dependent rotating distribution or averaged over the rotor.

This distinction defines their roles within the aerodynamic analysis. The AD model is used with steady RANS calculations to reproduce the average propeller influence at low computational cost, making it suitable for large design sweeps and the generation of aerodynamic datasets, such as the surrogate generation. However, by averaging the blade loading around the rotor plane, the method suppresses blade-passage effects and reduces the representation of discrete tip vortices. The AL model retains the individual rotating blade representations and is therefore employed with URANS when the temporal behaviour of the propeller–airframe interaction is required. It permits investigation of blade phase, local loading variation, tip-vortex development and blade-passage-induced aerodynamic fluctuations while retaining considerably lower geometric and meshing complexity than a comparable blade-resolved moving-mesh treatment.

7.1.5 Actuator disc and line simulations of INDi-2

The INDi-2 configuration provides a well defined benchmark for assessing the influence of propeller-model fidelity on the aircraft aerodynamic performance. Prior optimisation of the configuration set the propeller RPM and thus the corresponding blade tip Mach numbers for both cruise and low-speed operation. An independent aerodynamic study was conducted in which the thrust-coefficient parameters were adjusted to balance total propulsive thrust against the aerodynamic drag of the modelled configuration. These thrust-balanced conditions are distinct from the prescribed-loading sweeps discussed above. The resulting aerodynamic polars provide the reference performance envelope against which the subsequent actuator-model comparison is conducted. The aerodynamic database used throughout the surrogate training

Semi Span (m)	25.7	Taper Ratio	0.42
Wing Root Twist (deg.)	2.97	Wing Mid Twist (deg.)	2.4
Wing Tip Twist (deg.)	-1.07	Strut Twist (deg.)	1.64
Inboard Diameter (m)	5	Outboard Diameter (m)	3.97
Inboard RPM	461.39	Outboard RPM	569.65

Table 9: INDi-2 Aircraft configuration for wing, strut & propellers

process was produced using the AD formulation. This choice was driven by computational practicality: the AD model provides a steady, azimuthally averaged representation of the propulsion system and is therefore most suitable for the large number of CFD samples required. Extending the complete sampling campaign to an AL formulation would require URANS simulations to be conducted with sufficient temporal resolution to capture blade passage over several complete propeller revolutions. This would substantially increase the computational cost. The AD formulation therefore provides an appropriate basis for broad design space exploration, at the cost of removing the periodic aerodynamic behaviour associated with the individual rotating blades.

To understand the consequence of this modelling approximation, selected AD operating points were repeated using the higher-fidelity AL representation. Rather than attempting to reproduce the complete aerodynamic campaign, the comparison was focused on the operating condition corresponding to the maximum aerodynamic efficiency identified from the AD calculations. This does not imply that the selected points represent a global aircraft optimum; instead, they provide a useful reference for determining how sensitive the predicted peak aerodynamic performance is to the representation of the propulsion system. The cruise condition and the 10° flap-deployed condition were selected because they represent two contrasting interaction mechanisms of particular interest. The cruise case is dominated by the interaction between the propeller-induced flow and transonic features such as shock formation and shock-induced separation, whereas the flap-deployed case provides a means of examining the interaction between the periodic propeller wake and the high-lift system.

The two actuator formulations were treated as different fidelity levels within the same source-term framework. The AD model provides the time-averaged propeller influence and was first used to establish the mean aerodynamic behaviour of the aircraft. Spanwise lift and pressure drag distributions were examined to identify how the distributed propellers modified the local wing loading and where propeller–airframe interaction was strongest. In cruise, the AD results showed local increases in lift and

pressure drag across the propeller regions as the imposed swirl modified the local flow inclination. At the higher cruise incidence, the simulations additionally captured shock-induced separation shortly downstream of the leading edge, while the influence of the large inboard nacelle remained visible in both the skin-friction field and sectional aerodynamic loading.

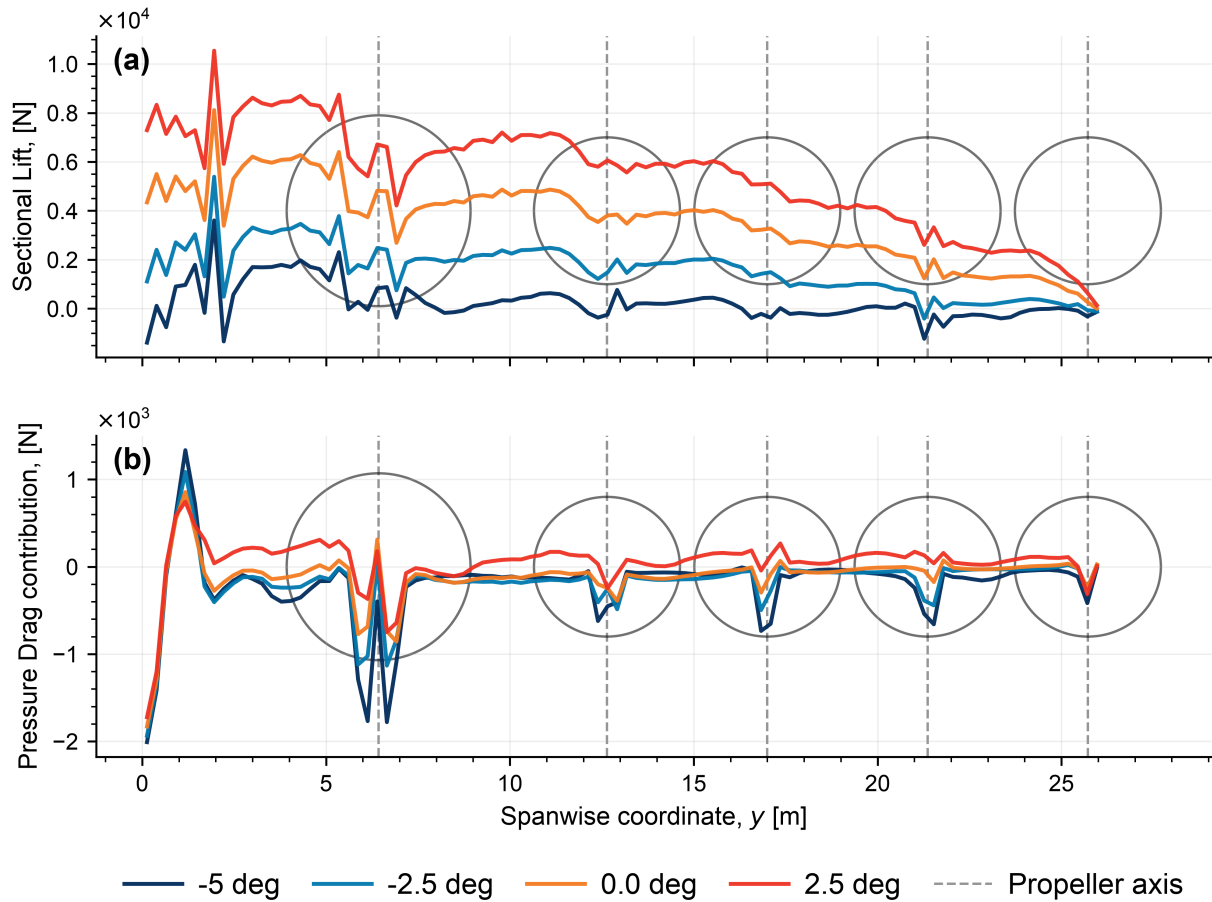


Figure 47: Cruise span-wise Lift (top) and Pressure Drag (bottom) profiles for 100 strips isolated along the wing span.

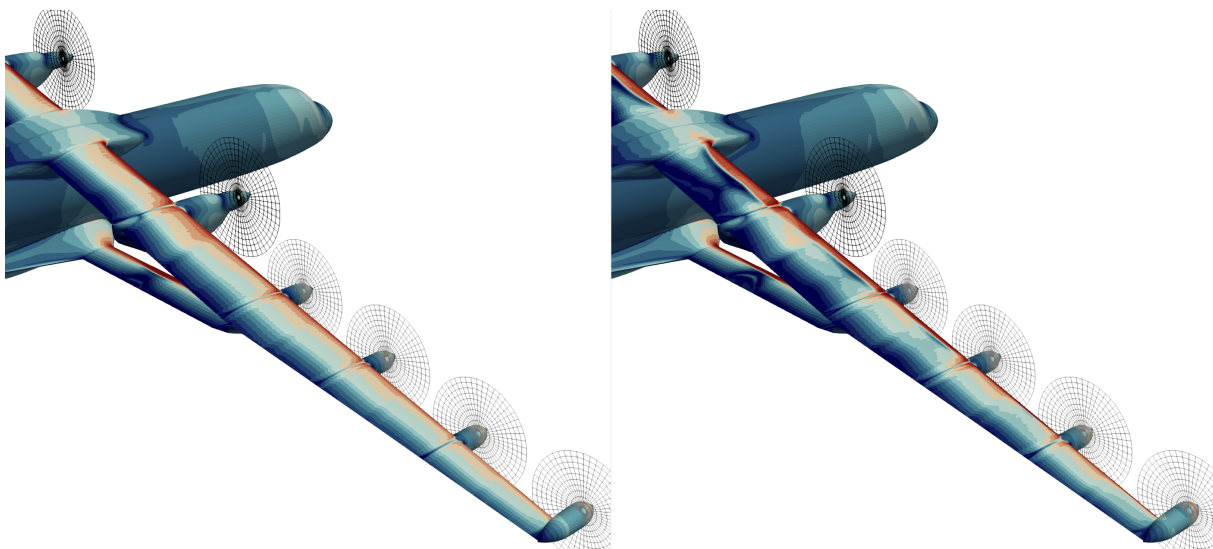


Figure 48: Cruise conditions. Skin friction coefficient for 0 deg (left) and for 2.5 deg (right).

The AL calculations were then initiated from the corresponding converged AD solutions, retaining

the same prescribed propeller loading and spatial discretisation while allowing the individual actuator lines to rotate through the computational domain. In accordance with the methodology described above, the source terms were recomputed at each time step as the blades advanced through their rotation. The simulations were continued for five complete propeller revolutions. This progression from AD to AL retains the prescribed propeller loading while introducing the blade-passage effects removed through azimuthal averaging.

The strongest distinction between the two approaches was observed for the cruise condition. The AL calculation revealed periodic fluctuations in both sectional lift and pressure drag. The largest pressure-drag variations occurred between the large hybrid propeller and the first outboard electric propeller, where blade passage perturbed the local flow inclination and interacted with regions of locally supersonic flow. The underlying cruise flow was particularly sensitive to this forcing because shocks on the wing suction surface and around the wing-strut junction were already associated with local separation. The rotating propeller wake therefore acted upon a flow field containing features that were intrinsically sensitive to comparatively small changes in local velocity and incidence. The flap-deployed condition exhibited

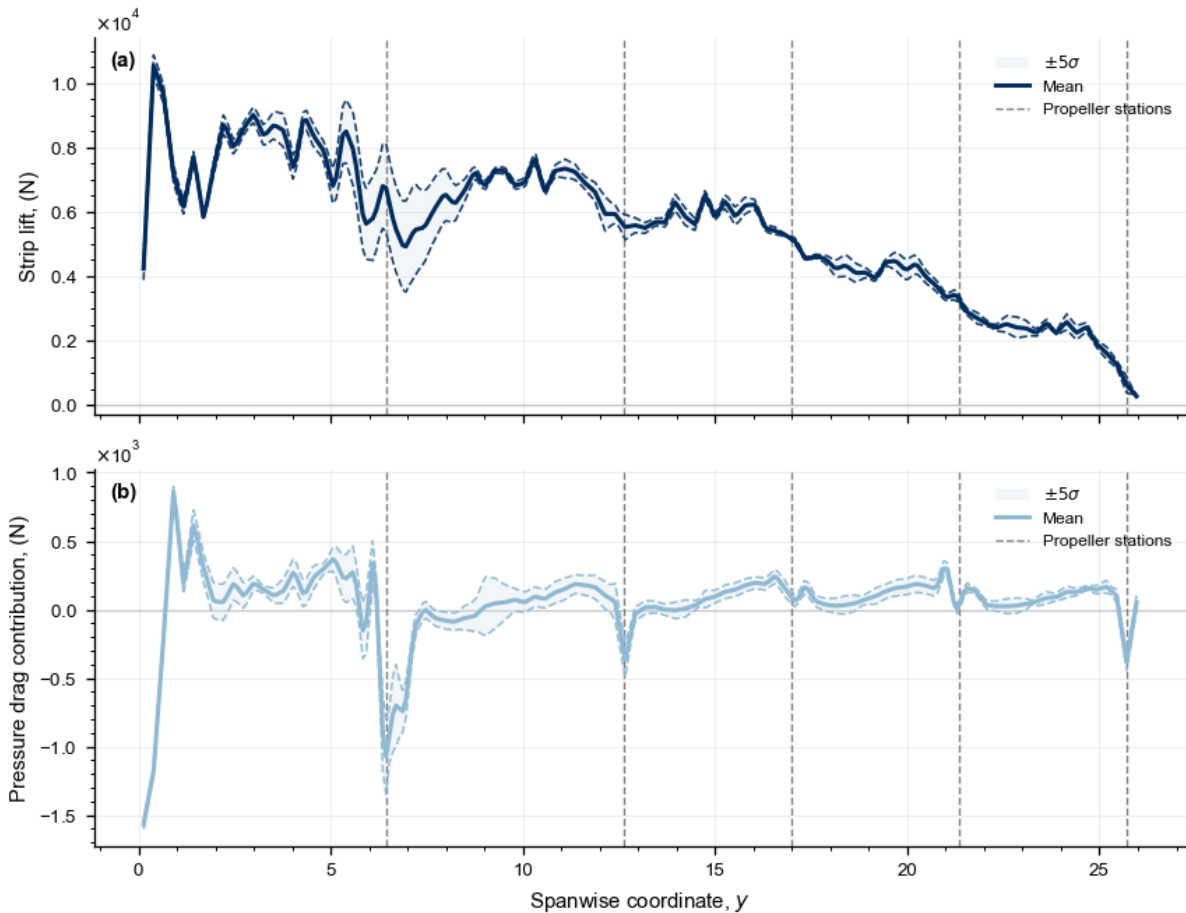


Figure 49: Span-wise distributions of Lift (top) and Pressure Drag (bottom) for the cruise configuration.

a noticeably weaker sensitivity to the transition from AD to AL. The sectional lift predicted by the AL calculation remained close to its time-averaged value over most of the wing, with comparatively small variations. Pressure drag displayed greater temporal variation than lift, although the magnitude of these fluctuations remained lower than those observed under cruise conditions. The largest variations again occurred away from the propeller centre lines, indicating that the changes in local flow inclination remained present, but interacted less strongly than with the cruise condition. The propeller wake structure varies significantly between the two operating conditions. During low-speed operation, the AL solution shows tightly wound helical wake structures. In cruise, the higher flight velocity transports the wake further downstream during each revolution, producing a more open structure. Principally the AL methodology is seen to provide insight to the flow physics of real propeller behaviour, demonstrating how complex structures can be initiated and influenced by blade passage effects.

For the investigated operating points, the AD and AL calculations produced comparable time-averaged

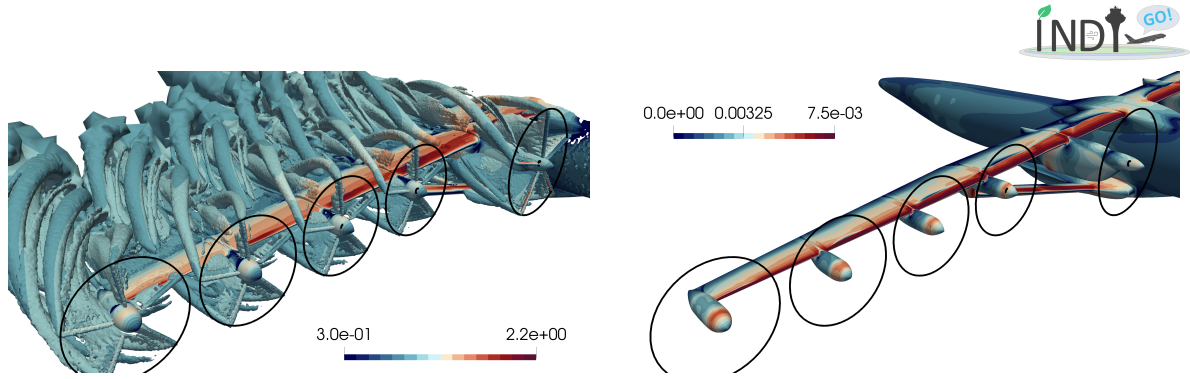


Figure 50: INDi-2 cruise configuration: AL solution after five propeller revolutions. Q-criterion coloured by normalised velocity (left) and skin-friction coefficient (right).

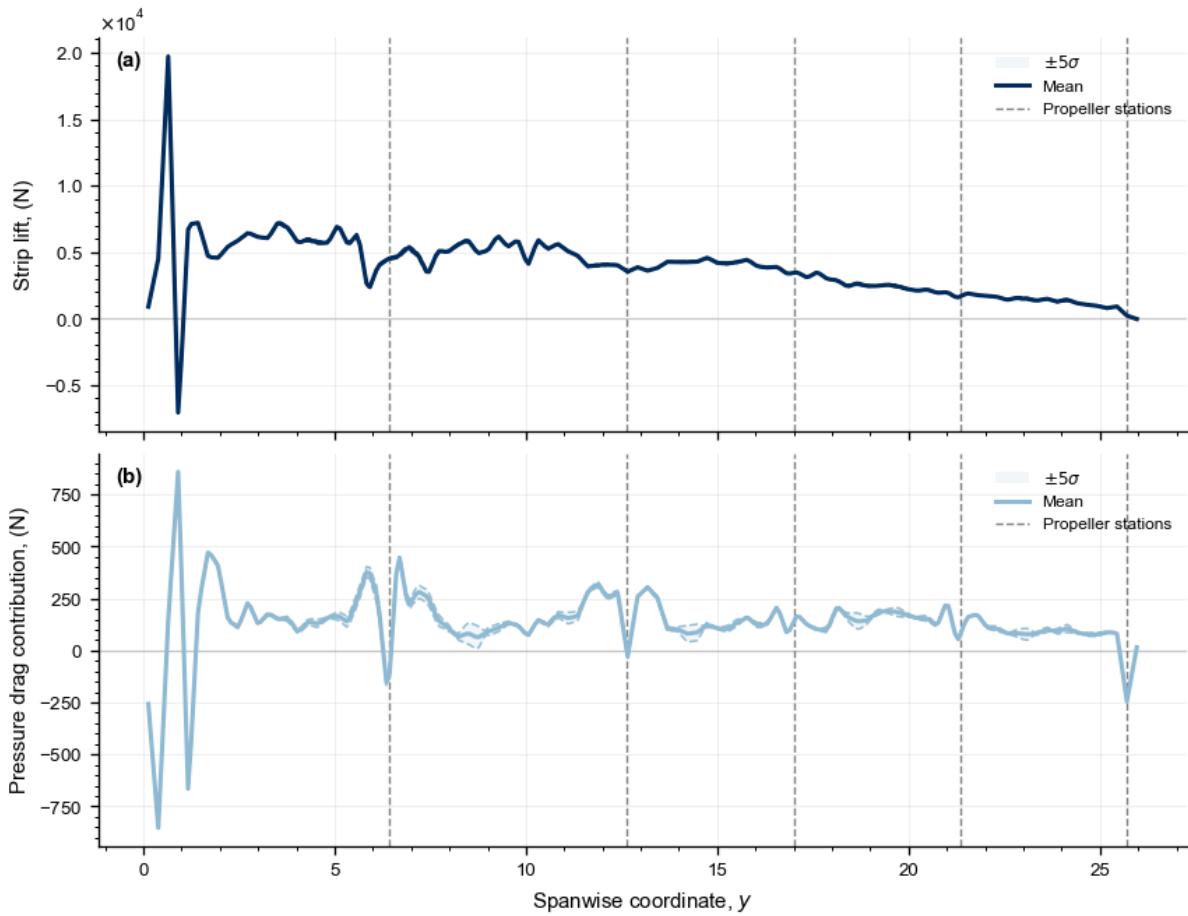


Figure 51: Span-wise Distributions of Lift (top) and Pressure Drag (bottom) for the high-Lift case.

global lift and drag coefficients, including in cruise, despite the local fluctuations resolved by AL. These results support the use of the AD formulation for generating mean aerodynamic data at the assessed conditions. The AL methodology provides additional information on blade-passage effects, local flow behaviour and unsteady wing loading.

7.1.6 Implications for aerodynamic design refinement

The aerodynamic verification assesses the selected INDi-2 configuration, not the maximum efficiency attainable by the DHEP-LARW architecture. The lifting system was optimised using VSPAero through changes in wing and strut twist and outer-wing taper. However, the CFD/POD parameterisation does not include optimisation of section shapes, fuselage and nacelle contours or junction fairings. The local shock, separation and interference effects identified by CFD therefore provide targets for further geometric refinement rather than necessarily representing irreducible penalties of the architecture.



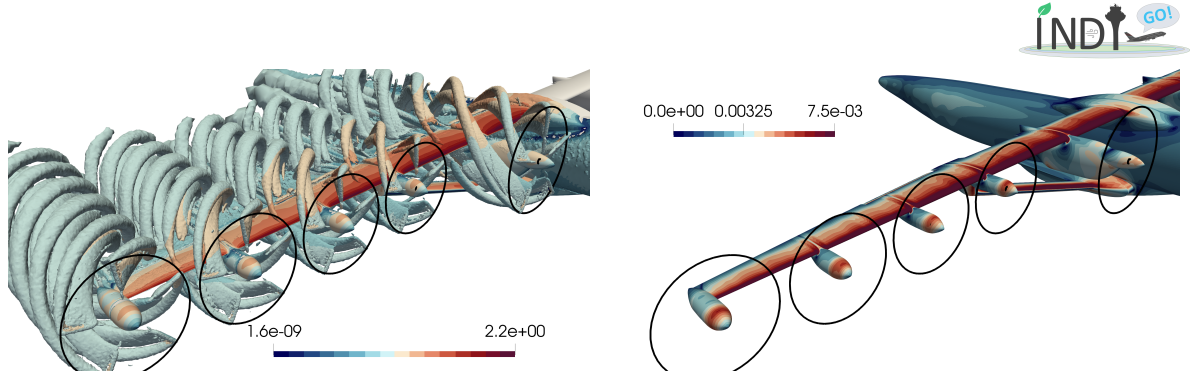


Figure 52: INDi-2 with 10° flap deflection: AL solution after five propeller revolutions. Q-criterion coloured by normalised velocity (left) and skin-friction coefficient (right).

Related studies demonstrate the potential of this approach. Secco and Martins [49] obtained a 15% drag reduction through RANS-based shape optimisation of a strut-braced-wing benchmark at fixed lift, associated with reduced shocks and separation near the wing-strut junction. Jones et al. [50] identified strong dependencies of drag on wing sweep and local junction geometry through changes in shock intensity and blockage. Chau and Zingg [51] demonstrated the value of multipoint optimisation: relative to their single-point optimum, off-design drag decreased by up to 11.6%, with a nominal-condition penalty of two drag counts. These results support the potential for material improvement, but their numerical gains are not predictions for INDIGO.

The propeller-loading and nacelle-force studies provide complementary guidance. Improvements in surface-force C_L/C_D with propeller loading must be assessed together with the required propulsive power. Similarly, the nacelle-factor study changes the force accounting without modifying the installed flow or its interference effects; it does not predict the benefit of a geometric redesign. These sensitivities should therefore guide a coupled investigation rather than be combined as additive performance improvements.

Further development should focus on constrained, multipoint refinement of the installed lifting system, propulsion geometry and relevant junctions. Performance should be evaluated over the mission-relevant lift range, with consistent component accounting, force balance and longitudinal trim, while retaining structural volume, thickness, installation clearances and low-speed requirements. Candidate improvements should be checked using direct CFD and propagated through the aircraft power and mission models. The results thus identify opportunities for aerodynamic improvement, rather than an intrinsic efficiency limit of the concept; the achievable gain and its aircraft-level consequences remain to be quantified.

7.2 Fluid–structure interaction

7.2.1 Objective and scope of the verification

The multidisciplinary design and optimisation activities presented in the previous sections rely on aerodynamic models in which the aircraft geometry is assumed to remain rigid. Accounting for structural flexibility directly within the complete optimisation and mission-analysis process would require a coupled aeroelastic solution for a large number of operating conditions and design evaluations, leading to a substantially higher computational cost.

The fluid–structure interaction (FSI) activity performed within Task 5.6 is therefore conceived as an *a-posteriori* verification of this modelling assumption rather than as an aeroelastic re-optimisation of the aircraft.

The main objective is to determine whether the static aeroelastic deformation of the INDIGO strut-braced-wing configuration introduces changes in aerodynamic performance or aerodynamic loading sufficiently large to question the aircraft-level conclusions obtained using the rigid aerodynamic representation. In particular, the analysis investigates the changes in the aerodynamic polar, the redistribution of the spanwise loading, and the magnitude of the resulting structural deformation. The aim is not to demonstrate that flexibility has no effect, but rather to establish whether the neglected aeroelastic effects constitute a first-order modification of the design obtained in WP5.

A complete aeroelastic assessment over all mission operating conditions is outside the scope of the present verification. Instead, the high-speed cruise condition, referred to as HS for the light-Euler case and HSV for the corresponding RANS case, is selected as the reference condition. It corresponds to $M_\infty = 0.6$, $h = 8000$ m and $q_\infty = 8971.2$ Pa, and represents the clean high-speed cruise configuration for

which a rigid RANS aerodynamic polar is available. Among the delivered aerodynamic cases considered in the present campaign, HS/HSV has the highest dynamic pressure and therefore provides a particularly relevant condition for assessing static aeroelastic effects.

Rigid and flexible aerodynamic polars are subsequently compared over their common verified lift range. The comparison is designed to isolate the effect of structural flexibility while retaining the same aerodynamic and propulsive modelling assumptions on both sides. The resulting analysis is used to assess the robustness of the rigid-aircraft representation adopted in WP5 and, more generally, to determine whether inclusion of static aeroelasticity in the design process would be expected to refine the resulting optimum or to lead to qualitatively different aircraft-level performance trends.

7.2.2 Numerical model and comparison strategy

The aeroelastic analyses were performed using a partitioned static fluid-structure interaction approach in which the aerodynamic and structural solvers retain their native discretisations. Aerodynamic loads computed on the INDIGO wetted surface are transferred to the finite-element model, while the resulting structural displacements are transferred back to the aerodynamic surface and used to deform the CFD mesh. The process is iterated until a stationary coupled state is reached according to the acceptance criteria adopted for the campaign.

Two aerodynamic models were employed with different purposes. The production model uses compressible RANS with the Spalart–Allmaras (SA) turbulence model and provides the reference results used to quantify the effect of flexibility on aerodynamic performance and aerodynamic load distribution. A lighter Euler model, based on the same external aircraft geometry, was used during development and verification of the coupling procedure and was subsequently employed to extend the investigation towards higher lift coefficients. Euler results are therefore used as supporting evidence, while the quantitative assessment of aerodynamic-performance changes is based primarily on the RANS calculations.

The structural model is a linear-static finite-element model of the complete aircraft structural assembly used for the FSI campaign. The lifting system, in particular the wing, strut and their primary load paths, is represented with the level of detail required for the present aeroelastic assessment, while the remaining aircraft structure is modelled primarily to provide the appropriate support and load-transfer paths. The fuselage representation therefore acts mainly as a structural support network, and the nacelle attachment regions are represented through dedicated load-transfer cages rather than detailed component-level structural models. The same structural model and coupling strategy are used for the Euler and RANS calculations. The coupled calculation does not include free-flight inertia relief or aircraft trim; these assumptions are considered explicitly when interpreting the high-load results in Section 7.2.5.

Figure 53 illustrates the non-matching CFD and structural discretisations and representative regions of the transfer interface. The CFD wetted surface contains approximately 2.96×10^5 unique nodes and 5.91×10^5 surface triangles. The Euler and RANS volume meshes contain approximately 3.45 and 12.9 million points, respectively. The coupled finite-element model contains 95 402 structural grids and approximately 114 000 shell elements.

Structural displacements are mapped to the CFD surface through a block-assembled rigid-mode moving-least-squares operator (pyMLS LRBM-TR), written as

$$\mathbf{u}_a = \mathbf{H}\mathbf{u}_s, \quad (36)$$

where each aerodynamic surface node follows a local rigid-body fit of the structural region supporting it. Aerodynamic nodal forces are transferred back through the transpose operator,

$$\mathbf{f}_s = \mathbf{H}^T \mathbf{f}_a. \quad (37)$$

This construction conserves virtual work and total force to machine precision. For the operator used in the present campaign, the total moment is preserved within approximately 0.6%.

To isolate the effect of structural flexibility, the propulsion forcing is kept unchanged between the rigid and flexible calculations and throughout the angle-of-attack sweep. The same prescribed volumetric force distribution is therefore applied at every point of the polar, and its magnitude and direction do not respond to the local elastic deformation or to changes in local inflow. The resulting calculations are consequently controlled rigid-flexible comparisons rather than independently trimmed or re-optimised flight states.

Comparison strategy Structural deformation modifies the local incidence of the lifting system and therefore changes the relationship between angle of attack and lift coefficient. A comparison at identical

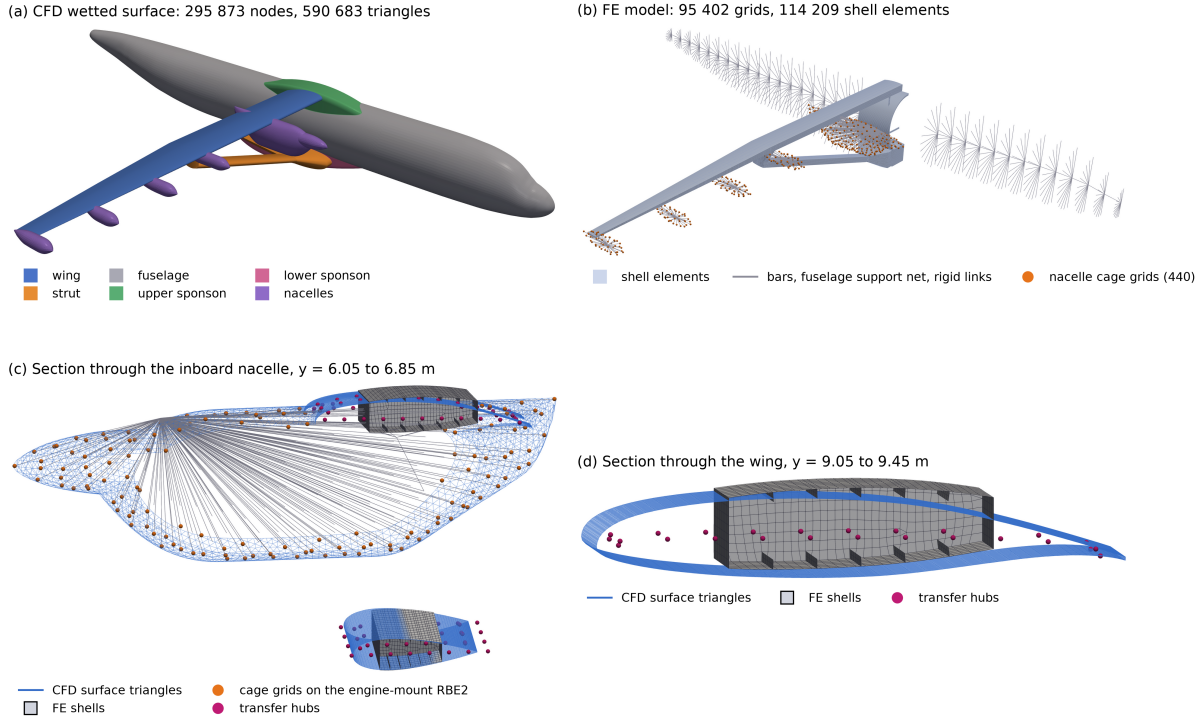


Figure 53: CFD and finite-element discretisations used in the FSI analysis, including representative local transfer regions.

angle of attack would consequently correspond to different total aircraft lift. The principal comparison is therefore performed at matched lift coefficient. For each accepted flexible RANS state, the rigid aerodynamic quantities are evaluated at the same C_L by piecewise-linear interpolation between neighbouring rigid solutions, without extrapolation. Differences in drag, aerodynamic efficiency and aerodynamic loading can therefore be interpreted as aeroelastic corrections at comparable aircraft lift rather than as the consequence of comparing different load levels.

7.2.3 Effect of flexibility on the aerodynamic polar

Figure 55 compares the rigid and flexible aerodynamic polars obtained with the Euler and RANS models at the selected HS/HSV condition. Structural flexibility produces a clear modification of the lift curve. Over the central range $-1.25^\circ \leq \alpha \leq +1.25^\circ$, the RANS lift-curve slope increases from approximately 0.150 to 0.197 per degree, corresponding to an increase of about 32%. The Euler calculations show the same qualitative steepening of the flexible lift curve.

The aeroelastic response is better interpreted as a rotation of the lift curve than as a uniform shift. At negative incidence, the flexible wing develops elastic wash-out and generates less lift than the rigid configuration. At $\alpha = -3.75^\circ$, C_L decreases from approximately 0.159 to 0.116, a reduction of about 27% at fixed incidence, while at $\alpha = -2.50^\circ$ the reduction is approximately 8%. As incidence increases, the deformation changes sign and the wing develops wash-in, leading to progressively higher lift than the rigid configuration. The rigid and flexible lift curves cross at approximately $\alpha = -1.8^\circ$. For the production RANS model, the quantitative aerodynamic-performance comparison is performed at matched lift as described in Section 7.2.2. At the lower end of the investigated range, the influence of flexibility on drag is essentially negligible. At $C_L \simeq 0.28$, the flexible drag differs from the rigid polar by only about +0.1%. With increasing lift, a modest drag reduction develops, reaching approximately 0.6% at $C_L \simeq 0.49$, 2.1% at $C_L \simeq 0.75$, and 4.4% at $C_L \simeq 0.98$.

The Euler calculations show the same qualitative aeroelastic behaviour and extend the comparison to higher lift coefficients, with drag reductions at matched lift reaching approximately 6%. These results are used as supporting evidence only, since the RANS model remains the production aerodynamic reference.

Overall, static flexibility produces a clear change in the incidence required to generate a prescribed lift. Its effect on aerodynamic efficiency is strongly dependent on lift level: it is negligible at the lower end of the verified range and increases to a few percent at the higher-lift conditions. The present campaign

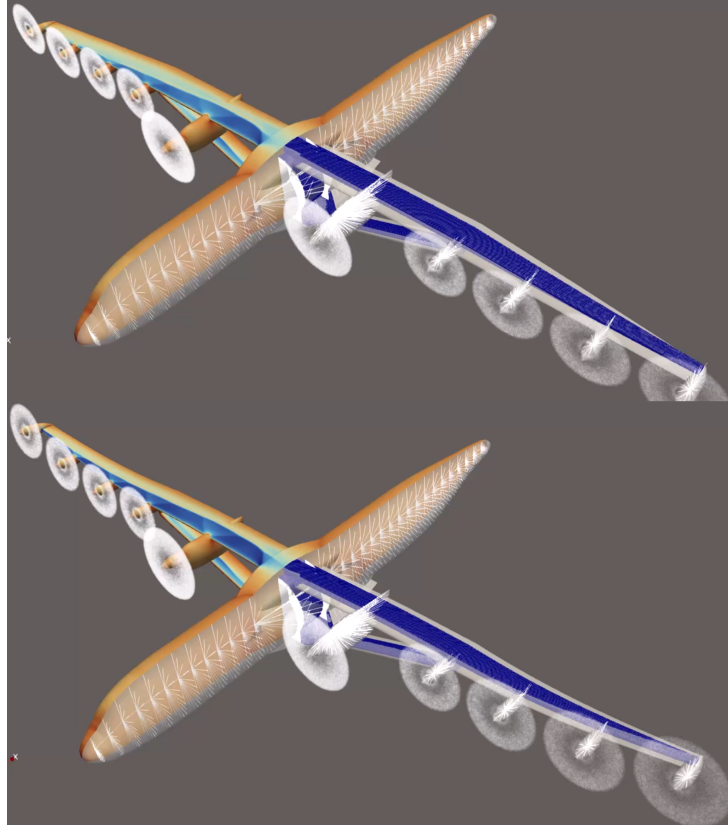


Figure 54: Undeformed (top) and FSI-deformed (bottom) aerodynamic and structural meshes; the aerodynamic surface is coloured by pressure coefficient C_p .

Table 10: Rigid–flexible RANS comparison at matched lift.

C_L	α_{rigid}	α_{flex}	$\Delta\alpha$	$\Delta C_D/C_{D,\text{rigid}}$	$\Delta(L/D)$
0.283	-2.707°	-2.50°	$+0.21^\circ$	$+0.08\%$	-0.08%
0.488	-1.120°	-1.25°	-0.13°	-0.55%	$+0.55\%$
0.747	$+0.593^\circ$	0.00°	-0.59°	-2.08%	$+2.13\%$
0.981	$+2.394^\circ$	$+1.25^\circ$	-1.14°	-4.43%	$+4.63\%$

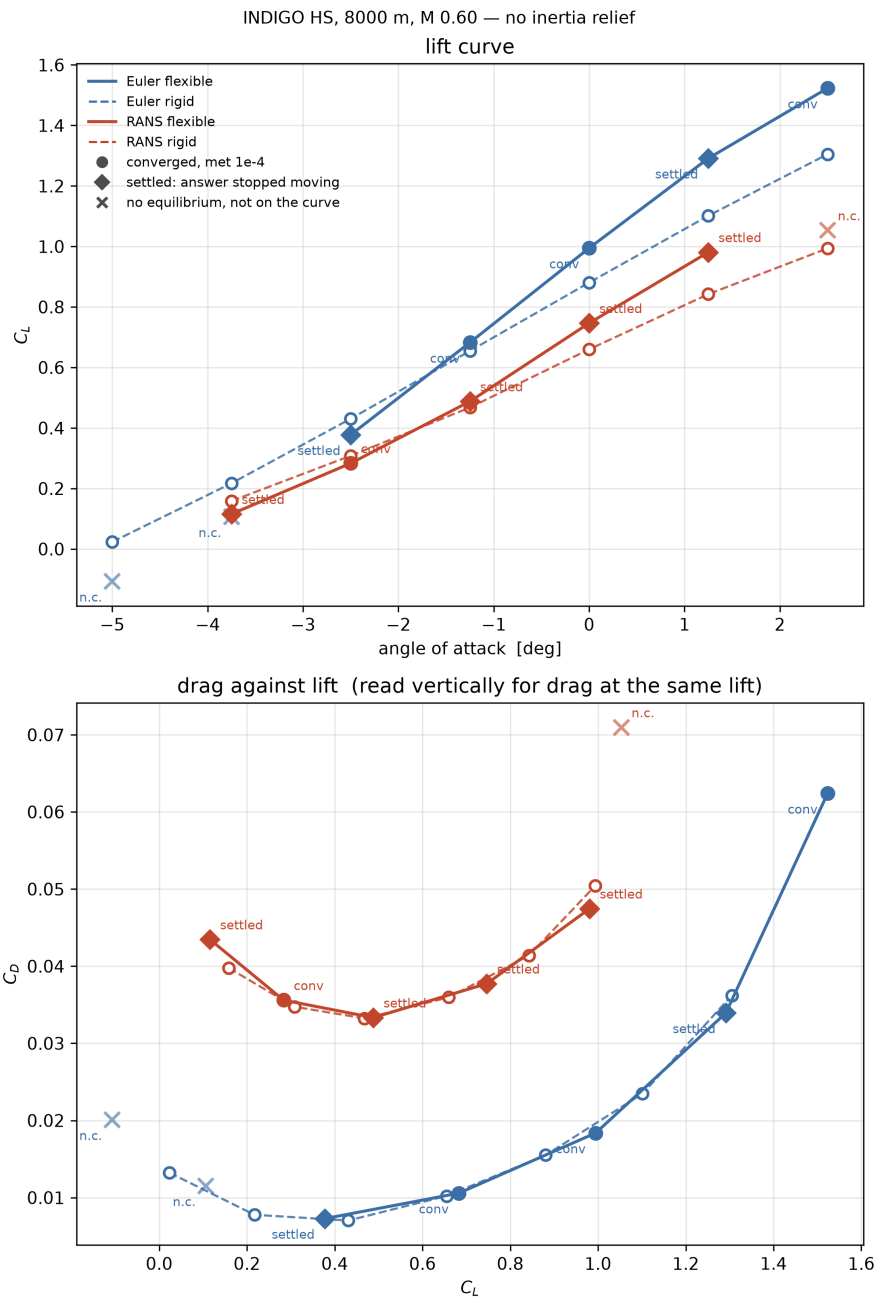


Figure 55: Rigid and flexible aerodynamic polars at the HS/HSV condition for the Euler and RANS models.

does not decompose the drag variation into induced, profile and interference contributions, and no specific mechanism is therefore assigned to the observed drag reduction.

Five RANS incidence conditions provide accepted flexible solutions. The solution at $\alpha = -2.50^\circ$ satisfies the nominal coupling tolerance, while those at -3.75° , -1.25° , 0° and $+1.25^\circ$ satisfy the adopted settled-state criterion. At $+2.50^\circ$, no static aeroelastic equilibrium is obtained because the increasing wash-in promotes flow separation. At -5.00° , the rigid RANS solution exhibits a persistent low-frequency oscillation and is therefore excluded from the quantitative rigid-flexible comparison.

7.2.4 Effect on aerodynamic load distribution

The matched-lift comparison was also used to assess whether structural flexibility modifies the aerodynamic loading of the lifting system. Since the total aircraft lift coefficient is matched by construction, the relevant question is whether flexibility produces a significant redistribution of that load over the wing and strut.

At global level, the differences remain limited over the investigated RANS range. The aerodynamic lift carried by the wing changes by less than about 2% relative to the corresponding rigid solution. Likewise, the first moment of the wing lift distribution about the aircraft centreline changes by only a few percent. These results indicate that the overall aerodynamic loading of the lifting system remains close to the rigid prediction.

The spanwise distribution nevertheless exhibits a systematic aeroelastic trend. Inboard of the wing-strut junction, the integrated wing lift remains essentially unchanged. Outboard of the junction, in contrast, the flexible solution carries progressively more lift than the rigid configuration, with the difference increasing with total lift coefficient. The fraction of the wing lift carried outboard of the strut junction changes only slightly, but the associated effect becomes more apparent when the spanwise lever arm is taken into account.

Figure 56 illustrates this behaviour at the highest matched RANS condition, $C_L \simeq 0.98$. The aerodynamic loading inboard of the wing-strut junction remains practically unchanged, whereas the outboard wing carries approximately 4–5% more integrated lift. The local difference increases progressively towards the outer wing and reaches values of the order of 10% close to the tip. Local variations are also visible at the nacelle stations, since the plotted distribution includes the wing surface only and excludes the nacelle surfaces.

The results therefore show that structural flexibility does not produce a major change in the overall aerodynamic load level. Its principal effect is a small but systematic redistribution of lift towards the outboard wing, consistent with the positive elastic twist generated by the structural deformation.

7.2.5 Structural deformation and high-lift extension

The magnitude of the structural response increases markedly towards the higher-lift portion of the RANS polar. At $C_L \simeq 0.488$, 0.747 and 0.981, the vertical displacement of the wing-box tip is approximately 0.14 m, 0.29 m and 0.44 m, respectively. The deformation is accompanied by an increasing positive elastic twist of the outboard wing, consistently with the modification of the lift curve and the outboard load redistribution discussed in the previous sections.

The light-Euler model extends the coupled analysis to higher lift coefficients than those for which a stationary flexible RANS solution could be obtained. The highest accepted Euler solution reaches $C_L = 1.523$ at $\alpha = +2.50^\circ$. At this condition, shown in Figure 57, the wing-box tip displacement is approximately 0.70 m, corresponding to about 2.7% of the wing semispan. The maximum elastic twist reaches approximately 3.6° at the wing-box tip, while the maximum bending slope remains below 5° .

The deformation remains moderate in geometrical terms even at this substantially higher loading. In particular, the deformed configuration in Figure 57 is shown at true scale, without geometrical magnification. The structural and aerodynamic twist estimates are also mutually consistent, confirming that the transferred deformation is reproduced coherently on both sides of the fluid-structure interface.

The coupled calculations do not include manoeuvre inertial loads. Their influence was therefore assessed separately by applying a $2.5g$ inertial load to the masses explicitly represented in the structural model, while retaining the converged aerodynamic nodal loading of the $C_L = 1.523$ Euler solution. With the aerodynamic load held fixed, the wing-box tip displacement decreases from 696 mm to 561 mm, corresponding to a reduction of approximately 19%. The maximum elastic twist decreases from 3.63° to 2.36° , and the maximum bending slope from 4.94° to 3.82° .

The same relieving effect is observed in the structural load path. The strut tensile force decreases from approximately 1.63 MN to 1.34 MN, while the spanwise compressive load transmitted to the inboard wing

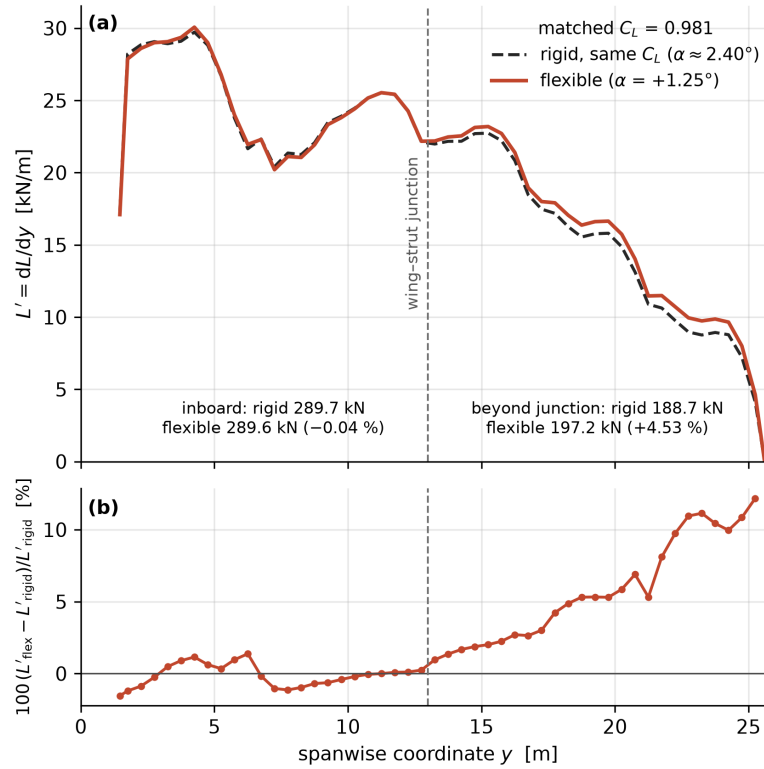


Figure 56: Spanwise aerodynamic lift distribution for rigid and flexible RANS solutions at matched lift, $C_L \simeq 0.98$.

is reduced by approximately 17%. For the masses represented in the structural model, inertial loading therefore acts to relieve, rather than amplify, the aerodynamic bending and twisting response.

This additional calculation is a structural sensitivity check and not a complete free-flight manoeuvre analysis: the aerodynamic load is kept frozen and only the masses represented in the coupled structural model are included. Nevertheless, it shows that the aerodynamic-only FSI solution provides an upper estimate of the global deformation for this structural model and prescribed aerodynamic load.

7.2.6 Implications for the WP5 design

The purpose of the FSI verification is ultimately to assess whether the rigid aerodynamic representation adopted in the previous optimisations introduces a first-order bias in the resulting aircraft design. The coupled analyses show that structural flexibility has a measurable effect on the aerodynamic response, but that its impact remains limited when the comparison is made at equivalent aircraft lift.

Over the verified RANS range, flexibility changes the incidence required to generate a prescribed lift and produces a systematic redistribution of loading towards the outboard wing. At matched lift, however, the variation in aerodynamic efficiency remains of the order of a few percent, while the overall aerodynamic load carried by the wing changes by less than about 2%. The structural response is likewise moderate at global level, with wing-box tip displacement remaining below 3% of the semispan even in the higher-load Euler extension.

These results suggest that explicitly including static aeroelasticity within the multidisciplinary optimisation would modify the detailed optimum, but would not be expected to produce a qualitatively different aircraft configuration. In particular, changes in structural stiffness, elastic twist and local aerodynamic loading could lead to adjustments in the detailed geometry and in the precise values of the optimum design variables. The present verification therefore supports interpreting aeroelasticity as a refinement of the WP5 design rather than as a mechanism that invalidates the rigid-aircraft optimisation.

Consequently, the principal aircraft-level performance trends and the order of magnitude of the performance potential identified in WP5 are expected to remain broadly representative when static aeroelastic effects are considered. This conclusion is an engineering inference from the present high-fidelity verification: the complete multidisciplinary optimisation and mission analysis have not been repeated with the

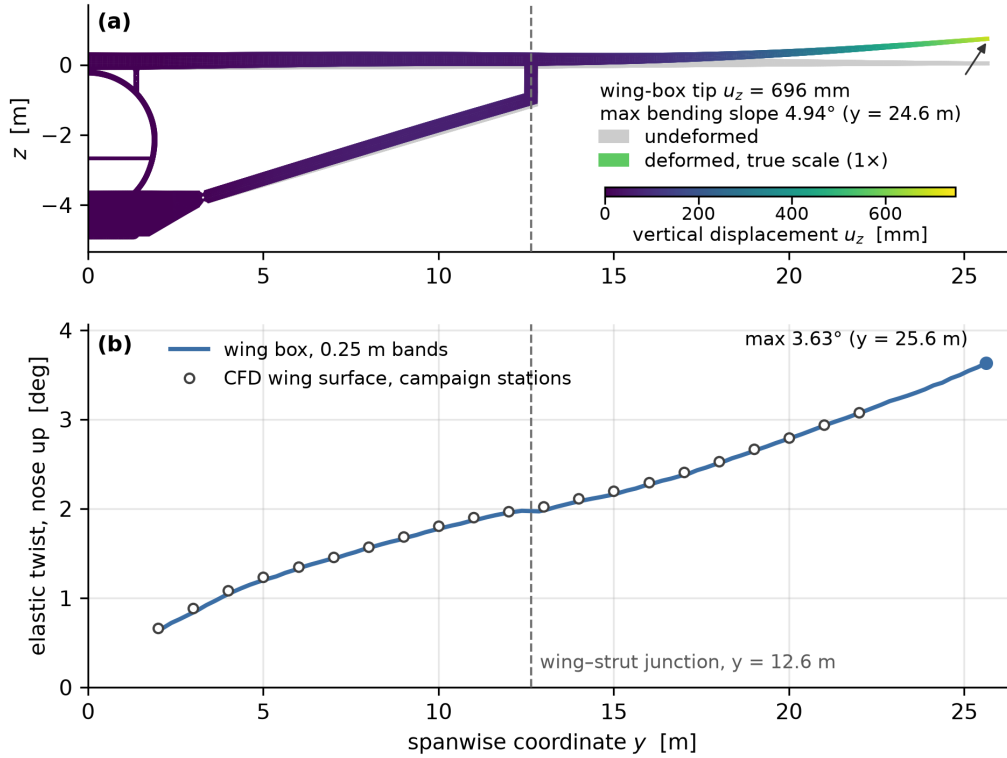


Figure 57: Structural deformation and elastic twist at the highest accepted Euler FSI condition, $C_L = 1.523$.

coupled aeroelastic model.

8 Conclusions

This deliverable has investigated the integration of distributed hybrid-electric propulsion and large-aspect-ratio wings through multidisciplinary optimisation under technological uncertainty, complemented by aeroacoustic studies and high-fidelity aerodynamic and static-aeroelastic verification. The TOPAZ framework couples aircraft mass, trim, propulsion, mission performance, fuel consumption and environmental models. The INDi-2 configuration is retained as the selected nominal reference airframe throughout the optimisation campaigns, while the powertrain, propeller design and mission controls are varied to examine the trade-offs between block fuel, fuel burned below 900 m, noise and sensitivity to battery-technology assumptions.

The deterministic results identify two battery-sizing regimes. When the battery-mass upper bound is active, increasing the available battery mass enables greater electrification and lower fuel consumption. Beyond a threshold, the design approaches a mass-limited plateau and further relaxation of the battery bound provides little benefit. Higher battery specific energy increases the contribution of battery-supplied power and reduces dependence on turboshaft-driven generation. The stricter 75.5 dB flyover EPNL constraint produces slightly higher fuel consumption than the 77.5 dB case, illustrating the trade-off between quieter propeller operation and propulsive efficiency.

The uncertainty analysis combines the adopted Dempster-Shafer battery forecasts with Monte Carlo propagation through Gaussian-process surrogates of the optimised responses. Relaxing the admissible variability improves expected fuel performance, but only until the usable battery capability reaches a saturation regime. For the single-trajectory 1000-nm mission, the reported energy density obtained after uncertainty propagation levels off at approximately 590 Wh/kg in the 2030 scenario and 700 Wh/kg in 2040. In the saturation regime at the loosest investigated robustness limits, expected block fuel decreases from approximately 3.35–3.39 t to 2.95–3.00 t, while fuel burned below 900 m decreases from 60–61 kg to 51–52 kg. These ranges span the two noise constraints and are conditional outcomes of the adopted $TRL \geq 7$ battery forecasts, rather than demonstrated battery performance.

The multi-trajectory study extends this assessment to a common powertrain design serving three

selected Riga missions. The general robustness–performance trends remain present, although their magnitude depends on the trajectory and the associated power-management strategy. The 1000-nm reference mission is retained as a feasibility requirement rather than included in the multi-trajectory objective. Its results therefore demonstrate the modelled capability to complete that mission, not a separately fuel-optimal operating strategy for it.

The aeroacoustic studies demonstrate the effectiveness of propeller phase synchronisation in the investigated laboratory configurations. For the two-bladed arrangement at $J = 0.53$, a 60° phase offset reduces the first blade-passing-frequency tone by approximately 12–15 dB over the tested wing incidences on the top microphone arc. At full scale, the numerical assessment predicts an approximately 2 dB(A) reduction in propeller-tonal L_{AE} for the investigated 36° phase-offset case relative to the INDIGO reference configuration on the selected RIX trajectory. This full-scale result remains a numerical prediction and is distinct from the EPNL constraint used in the optimisation.

The aerodynamic verification shows that component accounting explains a substantial part of the difference between the VSPAero wing-and-strut input and the CFD/POD polar. Including the fuselage contribution reduces this difference, while a lift-dependent residual remains. The propeller-loading and nacelle-force studies identify installation sensitivities, and the AD/AL comparison supports the use of AD for mean aerodynamic data at the assessed conditions. These findings motivate further refinement of the installed geometry rather than establishing an intrinsic efficiency limit of the architecture.

The static-FSI assessment finds a clear change in the lift curve and an outboard redistribution of loading, but a limited performance effect at matched lift. Over the verified RANS range, the drag change varies from negligible to a reduction of approximately 4.4% at $C_L \simeq 0.98$. These results support interpreting static aeroelasticity as a refinement of the assessed WP5 design, while remaining bounded by the investigated conditions and prescribed propulsion forcing. The complete optimisation and mission analysis have not been repeated with the coupled aeroelastic model.

Overall, the work provides a numerical basis for selecting powertrain and technology assumptions according to the required balance between fuel consumption, noise and robustness. Further development should combine the completed discipline models with constrained, multipoint refinement of the installed airframe–propulsion geometry, followed by renewed mission evaluation. This would quantify the achievable improvement and its effect on the absolute fuel and environmental predictions.

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10 Annexes

10.1 Multi-trajectory Optimization - 77.5 dB

10.1.1 Most Frequent:

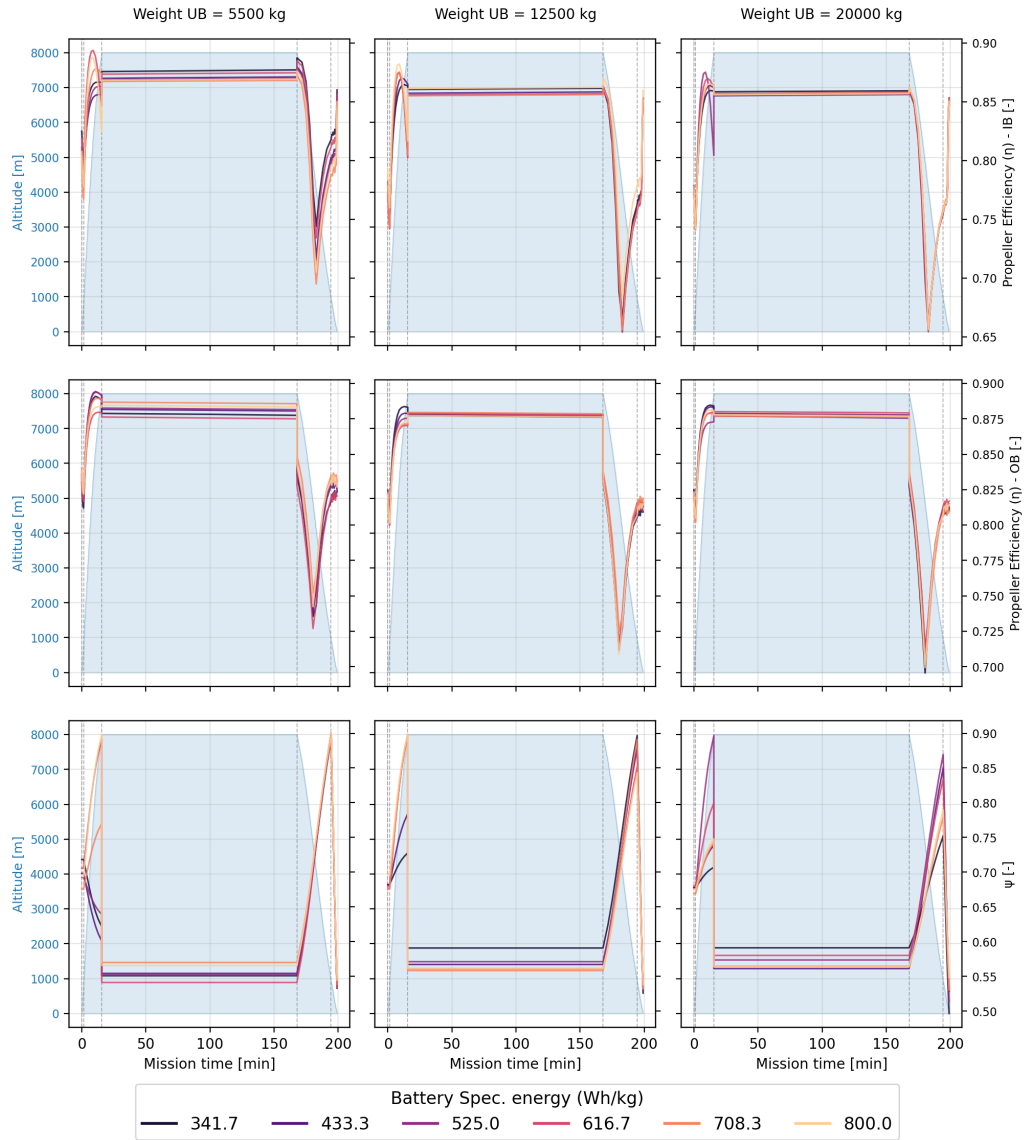


Figure 58: **Most frequent:** Propellers efficiency and ψ along the mission (flyover EPNL constraint equal to 77.5 dB).

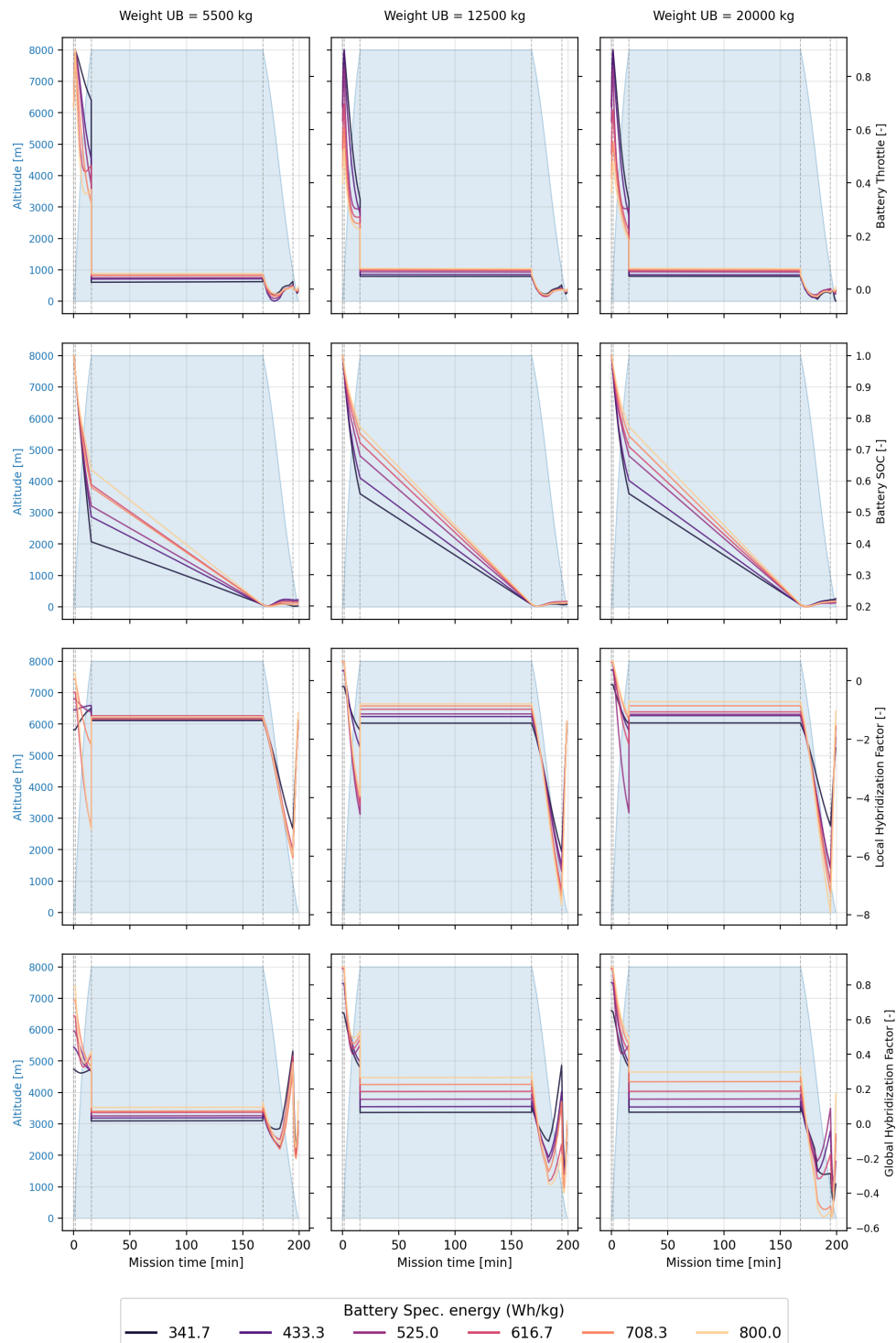


Figure 59: **Most frequent:** Battery throttle, battery SOC, local and global hybridization factor along the mission (flyover EPNL constraint equal to 77.5 dB).

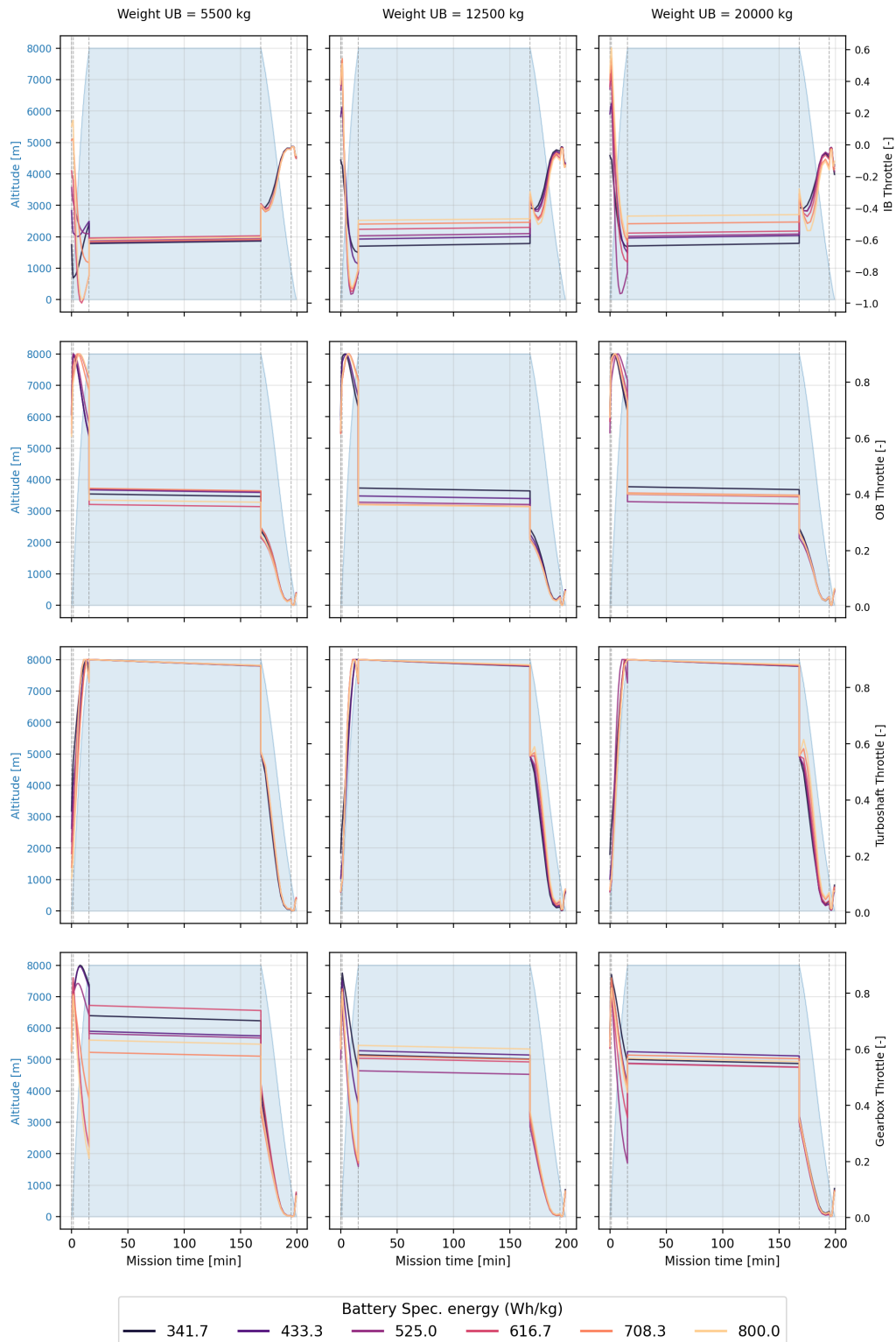


Figure 60: **Most frequent:** IB, OB, turboshaft and gearbox throttles along the mission (flyover EPNL constraint equal to 77.5 dB).

10.1.2 RIGA_TO_F597_L_F133:

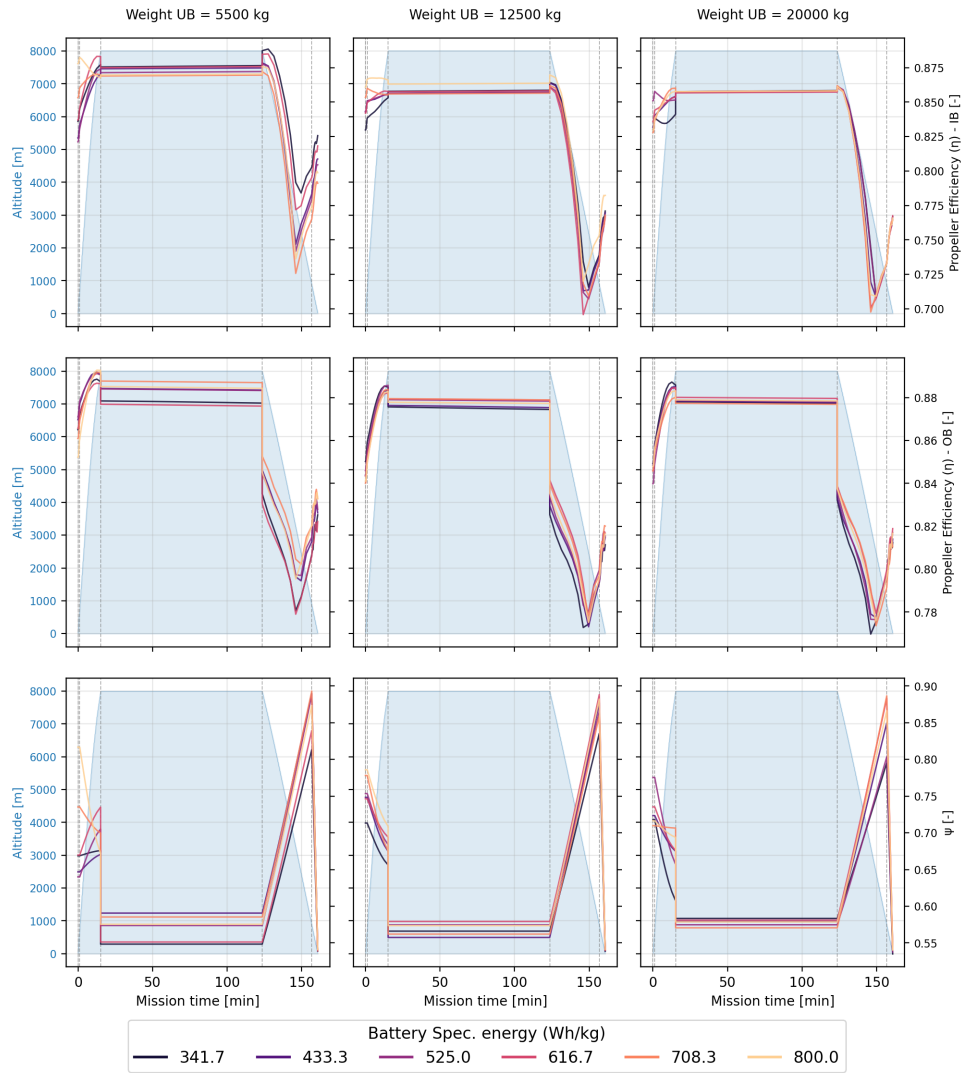


Figure 61: **RIGA_TO_F597_L_F133**: Propellers efficiency and ψ along the mission (flyover EPNL constraint equal to 77.5 dB).

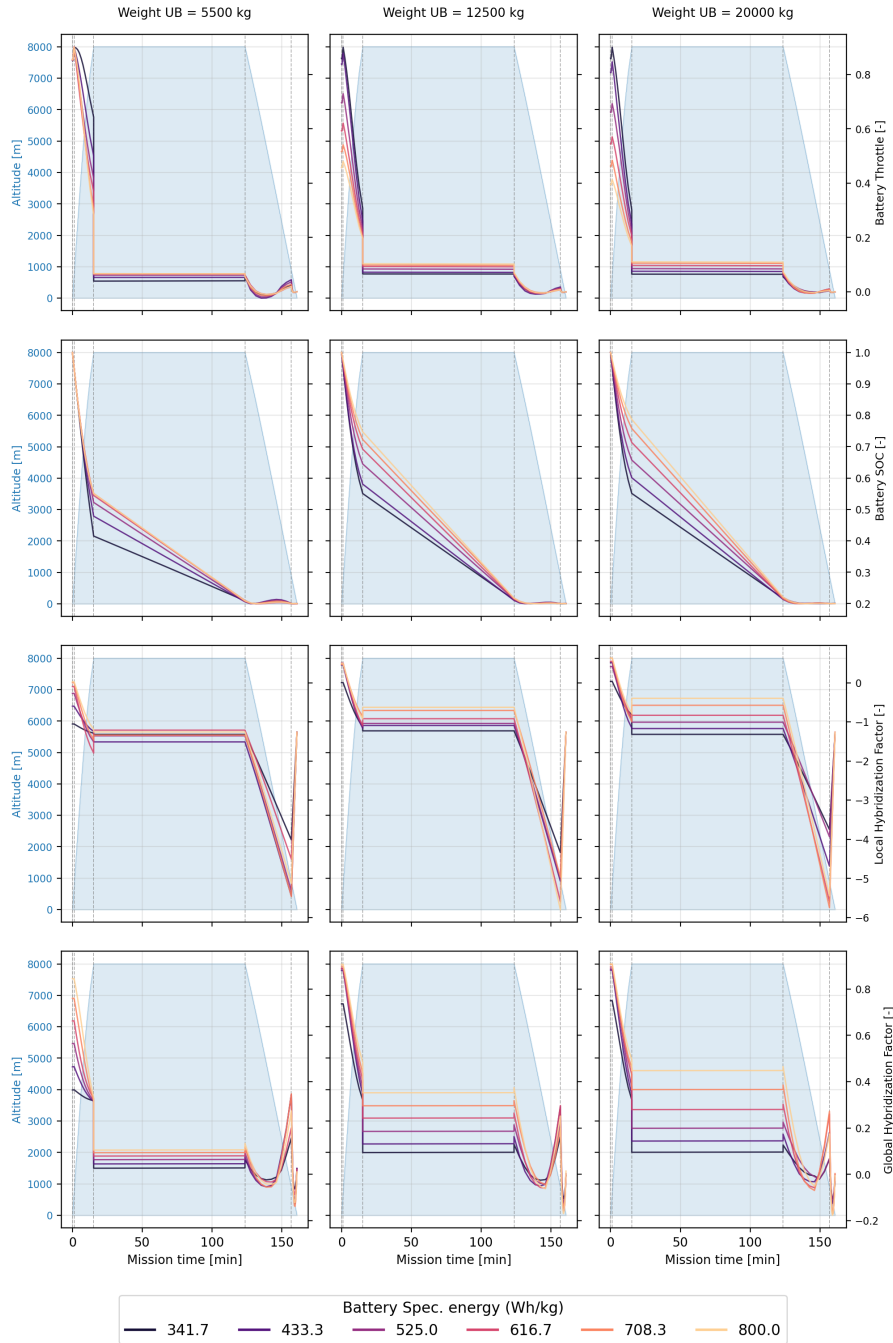


Figure 62: **RIGA_TO_F597_L_F133**: Battery throttle, battery SOC, local and global hybridization factor along the mission (flyover EPNL constraint equal to 77.5 dB).

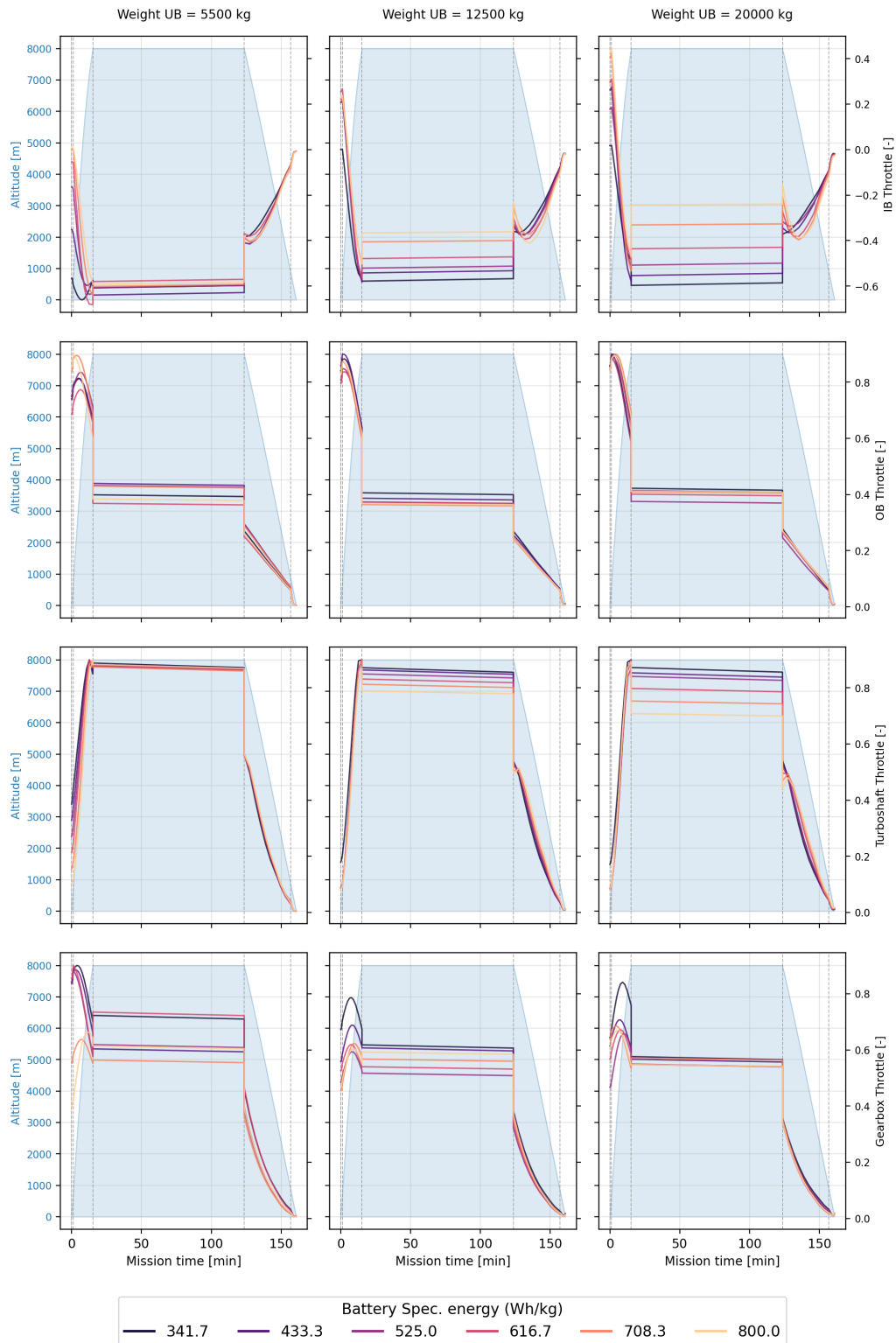


Figure 63: **RIGA_TO_F597_L_F133**: IB, OB, turboshift and gearbox throttles along the mission (flyover EPNL constraint equal to 77.5 dB).

10.1.3 RIGA_TO_F78_L_F137:

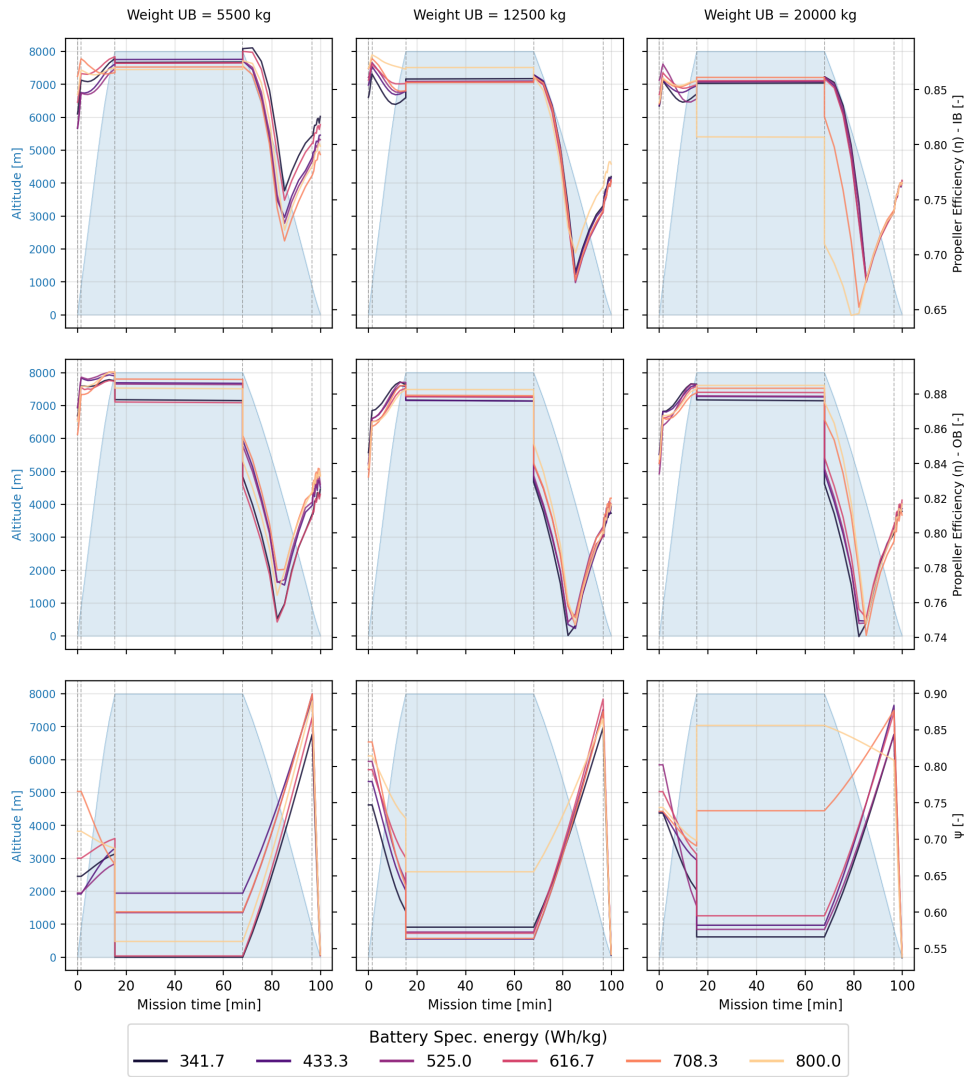


Figure 64: **RIGA_TO_F78_L_F137**: Propellers efficiency and ψ along the mission (flyover EPNL constraint equal to 77.5 dB).

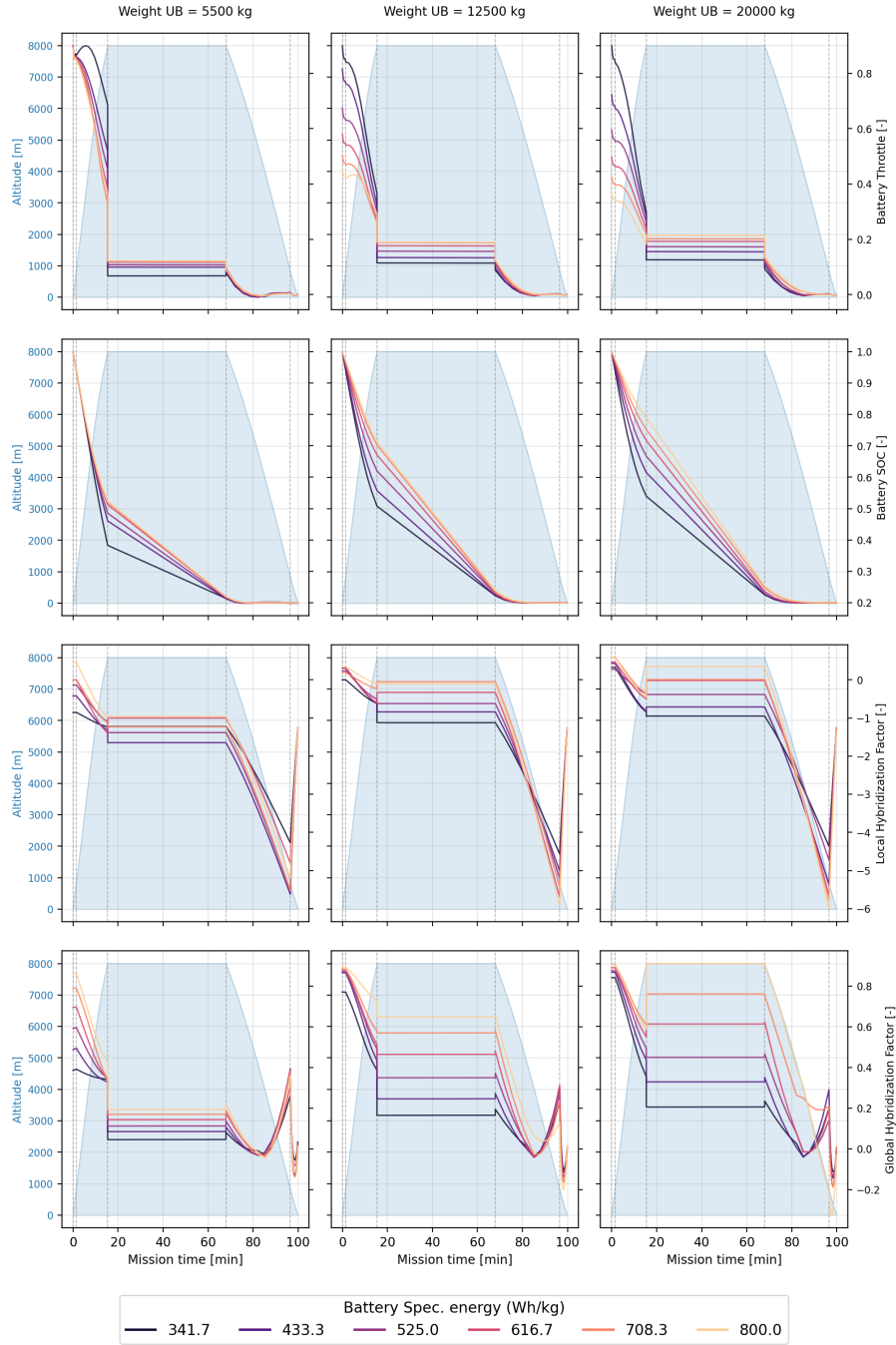


Figure 65: **RIGA_TO_F78_L_F137**: Battery throttle, battery SOC, local and global hybridization factor along the mission (flyover EPNL constraint equal to 77.5 dB).

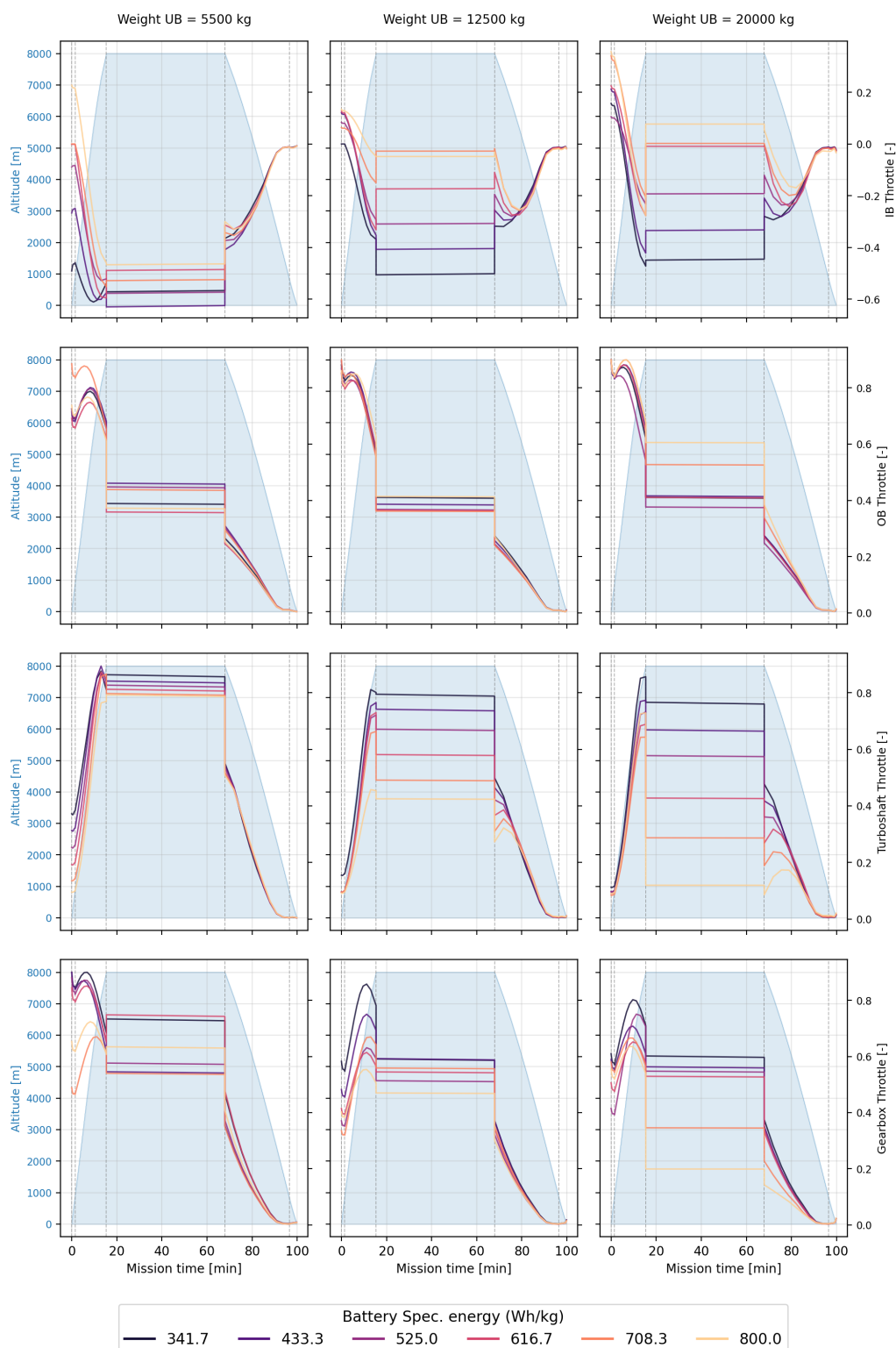


Figure 66: **RIGA_TO_F78_L_F137**: IB, OB, turboshaft and gearbox throttles along the mission (flyover EPNL constraint equal to 77.5 dB).

10.1.4 Reference Mission:

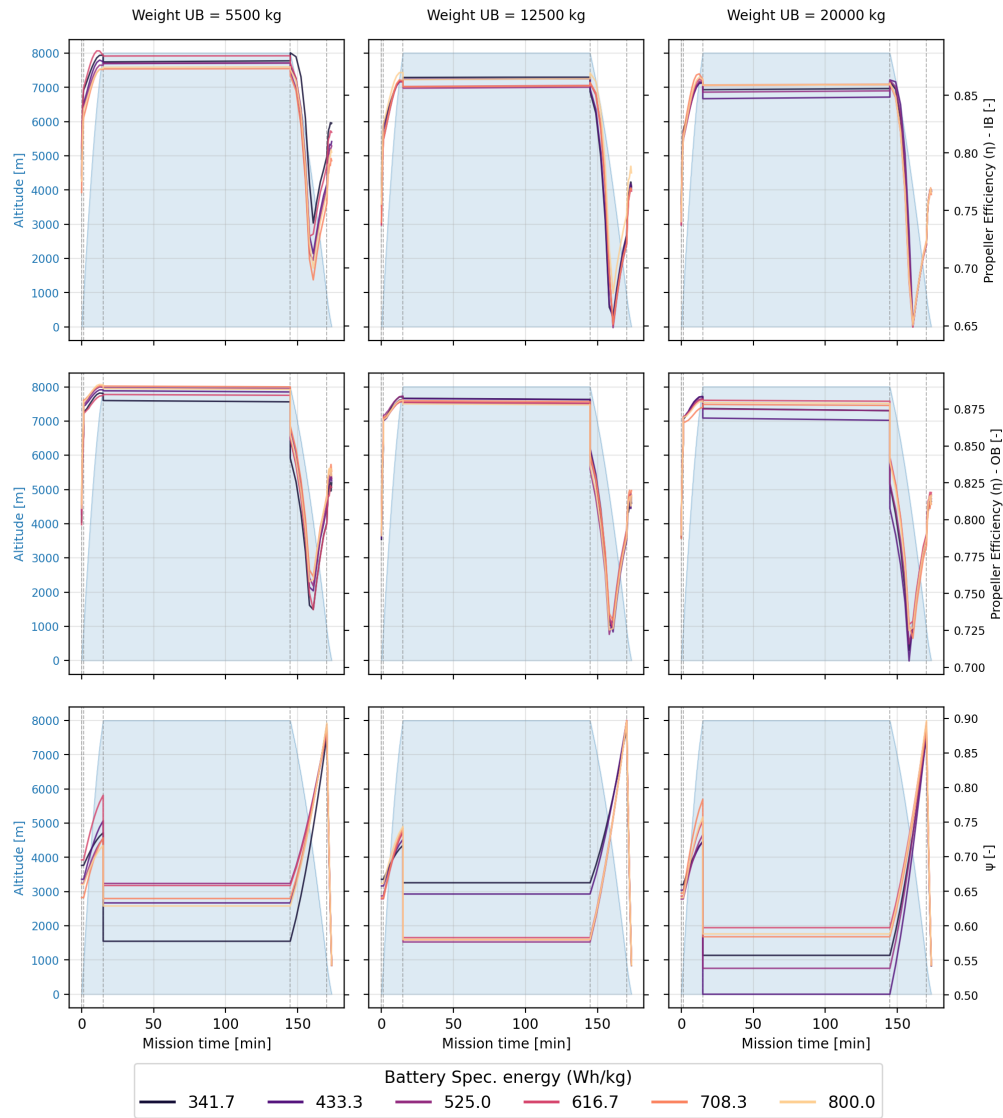


Figure 67: **Reference mission:** Propellers efficiency and ψ overall powertrain efficiency along the mission (flyover EPNL constraint equal to 77.5 dB).

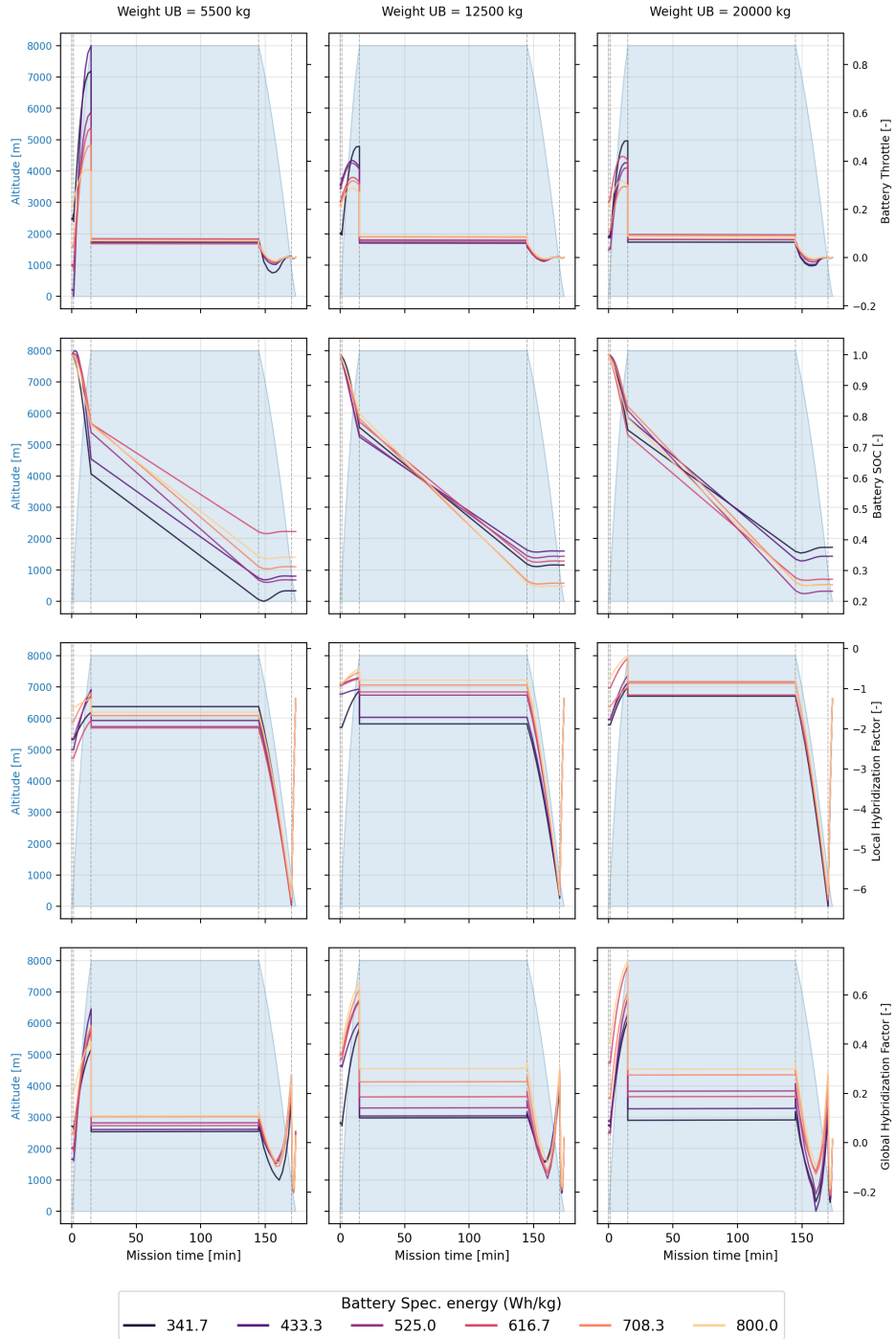


Figure 68: **Reference mission:** Battery throttle, battery SOC, local and global hybridization factor along the mission (flyover EPNL constraint equal to 77.5 dB).

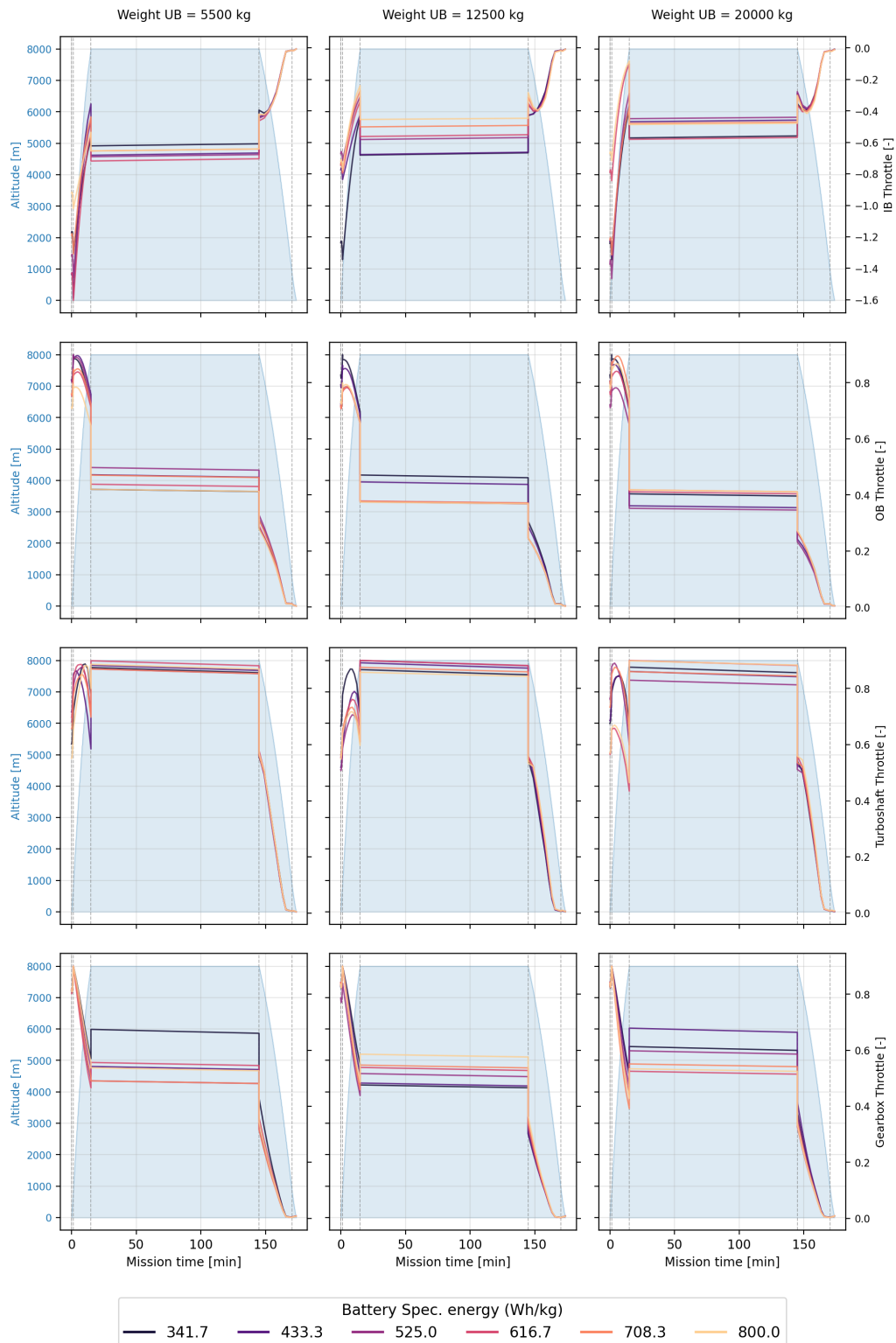


Figure 69: **Reference mission:** IB, OB, turboshaft and gearbox throttles along the mission (flyover EPNL constraint equal to 77.5 dB).