

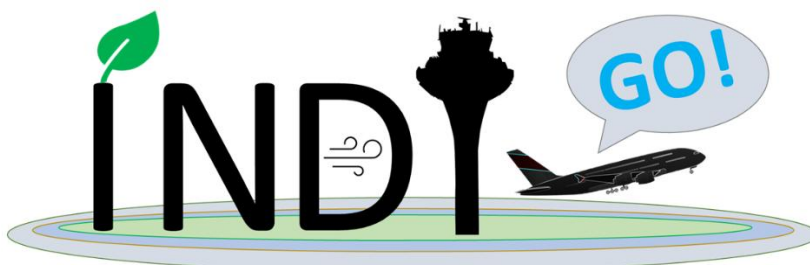
RESEARCH PROJECT MANAGEMENT

Document n°	INDIGO - D3.2
DHEP-LARW Aircraft performance model	
Issue date	A02 – 29/09/2026

Revision History Table		
Version n°	Issue Date	Reason for change
A01	17/09/2026	First draft
A02	29/09/2026	Final Version



Date of Issue: 29th September 2026



DHEP-LARW Aircraft performance model

Project: 101096055

INtegration and Digital demonstration of low-emission aircraft technoloGies and airport Operations

INDIGO

**Work package: WP3 - Models and simulations for airframe and propulsion
integration**

Deliverable: D3.2 - DHEP-LARW Aircraft performance model



INDIGO project has received funding from the European Climate, Infrastructure and Environment Executive Agency (CINEA) under the Horizon Europe programme under grant agreement No 101096055.

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List of abbreviations

Abbreviation	Definition	Abbreviation	Definition
BEMT	Blade Element Momentum Theory	LARW	Large-Aspect-Ratio Wing
CAA	Computational Aeroacoustics	LBM	Lattice Boltzmann Method
CC	Combustion Chamber	LEE	Linearised Euler Equations
CFD	Computational Fluid Dynamics	MDA	Multidisciplinary Analysis
CINEA	European Climate, Infrastructure and Environment Executive Agency	MDO	Multidisciplinary Design Optimisation
CO	Carbon Monoxide	NOx	Nitrogen Oxides
CO ₂	Carbon Dioxide	OB	Outboard
CPACS	Common Parametric Aircraft Configuration Schema	POD	Proper Orthogonal Decomposition
CRN	Chemical Reactor Network	RANS	Reynolds-Averaged Navier–Stokes
CS-25	Certification Specifications for Large Aeroplanes	RBE3	RBE3 interpolation element
DEP	Distributed Electric Propulsion	ROM	Reduced-Order Model
DHEP	Distributed Hybrid-Electric Propulsion	RUB	Ruhr University Bochum
DLM	Doublet Lattice Method	SAF	Sustainable Aviation Fuel
DLR	Deutsches Zentrum für Luft- und Raumfahrt	SBW	Strut-Braced Wing
EDLM	Enhanced Doublet Lattice Method	SPL	Sound Pressure Level
EPNdB	Effective Perceived Noise Level, expressed in decibels	TUBS	Technische Universität Braunschweig
FGM	Flamelet-Generated Manifold	UC3M	Universidad Carlos III de Madrid
FW-H	Ffowcs Williams–Hawkings	UHC	Unburned Hydrocarbons
GFEM	Global Finite Element Model	UST	University of Strathclyde
GT	Gas Turbine	VLES	Very Large Eddy Simulation
HC	Hydrocarbons	VLM	Vortex Lattice Method
HPC	High-Pressure Compressor	VPM	Vortex Particle Method
INDIGO	INtegration and Digital demonstration of low-emission aircraft technologies and airport Operations	WP	Work Package
LAQN	Local Air Quality and Noise	WSC	Wing + strut + connector structural mass



1. Executive Summary

This document constitutes Deliverable D3.2, DHEP-LARW Aircraft Performance Model, of Work Package 3, “Models and simulations for airframe and propulsion integration”, within the Horizon Europe INDIGO project. It presents the disciplinary models and analysis frameworks developed to assess the aerodynamic, aeroacoustic, aeroelastic, structural and mission-level performance of the INDIGO aircraft.

Task 3.2 establishes a hierarchy of aerodynamic methods, ranging from computationally efficient potential-flow approaches to high-fidelity CFD simulations and data-driven reduced-order models.

Task 3.3 develops a multi-fidelity aeroacoustic framework combining vortex-based methods, hybrid RANS-CAA simulations and high-fidelity LBM/VLES analyses. These methods support the prediction of installed propeller noise and the investigation of propeller-propeller and propeller–wing interaction effects.

Task 3.4 provides a coupled flight-dynamic and aeroelastic framework for trim, stability, manoeuvre response, dynamic-load recovery and discrete-gust assessment. The analyses show that the distributed propulsion installation must be represented as an integral part of the aircraft aeroelastic system. For the investigated conditions, static deflections remained sufficiently small for geometric nonlinearities to be neglected, although structural flexibility continued to influence the modal and transient response.

Task 3.5 addresses airframe sizing and weight estimation using structural models of increasing fidelity, while Task 3.6 integrates the disciplinary capabilities into the Aircraft Performance Model. This model supports digital mission evaluation and provides aircraft states, power requirements, fuel consumption, pollutant emissions and acoustic indicators. Together, these capabilities form the multidisciplinary-analysis core required by the aircraft optimisation activities conducted in WP5.



2. Introduction

INDIGO investigates a non-conventional mid-range aircraft combining a large-aspect-ratio strut-braced wing with distributed hybrid-electric propulsion. The concept is intended to improve aircraft efficiency while reducing pollutant emissions and noise, particularly during operations at and near airports.

The integration of these technologies creates a strongly multidisciplinary design problem. Large-aspect-ratio wings provide aerodynamic benefits but introduce greater structural flexibility and unconventional load paths. Distributed propulsive units modify the flow around the wing while adding concentrated masses, rotary inertias and gyroscopic effects along the span. Consequently, aerodynamics, propulsion, structures, aeroelasticity and acoustics cannot be treated as fully independent disciplines.

Deliverable D3.2 documents the modelling capabilities developed within WP3 to analyse these interactions and evaluate the performance of the selected aircraft configurations. A multi-fidelity strategy is adopted because the required balance between computational cost and physical accuracy changes throughout the design process. Lower- and medium-fidelity methods enable extensive parametric studies and repeated evaluations, whereas higher-fidelity methods provide detailed analysis, validation data and training datasets for surrogate and reduced-order models.

The document is organised according to the contributing WP3 tasks. Task 3.2 addresses aerodynamic modelling and surrogate generation. Task 3.3 presents the aeroacoustic methodologies developed for installed distributed propulsion. Task 3.4 describes the coupled aeroelastic, flight-dynamic and loads framework. Task 3.5 addresses structural modelling, sizing and weight estimation. Finally, Task 3.6 describes how the disciplinary models are integrated into the Aircraft Performance Model used for mission analysis and multidisciplinary optimisation.

3. Task3.2 - Aerodynamics models and simulations

3.1. Aerodynamic surrogate methodology overview & setup

The aerodynamic modelling strategy evolved throughout the INDIGO design process according to the level of fidelity and computational cost required at each stage. A computationally efficient surrogate was first developed to support the preliminary aircraft-level optimisation performed in WP1 and the definition of the IND11 configuration. The corresponding design-space exploration also enabled the most relevant geometric and propulsive parameters to be identified, providing the basis for the reduced parameter space subsequently adopted for the higher-fidelity aerodynamic campaign developed in Task 3.2. The preliminary model is briefly described below, while its development and application within the initial INDIGO multidisciplinary optimisation are documented in [1, 2, 3].

3.1.1. Preliminary aerodynamic model for IND11

The preliminary aerodynamic model was formulated for extensive design-space exploration at a computational cost compatible with aircraft-level multidisciplinary optimisation. The underlying aerodynamic database was generated using the VSPAero potential-flow solver, with the wing and strut represented through a Vortex Lattice Method (VLM). A reduced-order representation of this database was used to rapidly recover the aerodynamic coefficients and construct the corresponding lift and drag polars for arbitrary configurations within the prescribed design space [1, 2].



The basic potential-flow prediction was complemented by low-cost corrections required for the preliminary aircraft-design problem. High-lift-device effects were introduced through semi-empirical relationships, while the aerodynamic influence of distributed propulsion was represented through analytical propeller-slipstream corrections applied to the portions of the wing affected by the propeller wakes. The high-lift and blowing corrections were coupled so that their interaction could also be retained during low-speed operation. This provided a computationally efficient representation of the principal aerodynamic–propulsive coupling while allowing wing, strut and propulsion parameters to be varied within the multidisciplinary optimisation.

This aerodynamic model formed part of the minimal-fidelity MDA used in WP1 to generate the INDIGO baseline configuration. Within this framework, aerodynamic performance and propulsion were coupled through the propeller thrust requirement, such that changes in the propulsion operating point affected the aerodynamic state and aircraft trim, and vice versa [3]. The resulting preliminary campaign also supported the reduction of the aerodynamic design space adopted for the subsequent higher-fidelity studies described below.

3.1.2. Higher-fidelity surrogate setup

The objective of the campaign was to develop several separate aerodynamic surrogates for three representative operating regimes:

- Cruise
- Low speed operation with flaps retracted
- Low speed operation with flaps deployed

Separating the operating regimes allows each surrogate to be constructed using a fixed set of atmospheric and flight conditions. This simplifies the individual surrogate models and allows the aerodynamic behaviour associated with each regime to be represented without requiring further inputs to be included

The atmospheric and flight conditions assigned to each regime are summarised as:

Condition	Cruise	Flaps Retracted	Flaps Deployed
Altitude (m)	8000	1000	0
Flight Mach	0.6	0.25	0.25
Density (kg/m ³)	0.5252	1.1116	1.225
Speed of Sound (m/s)	308	336	340

Table 1: Operating Conditions & Atmospheric Properties

These conditions define three independent aerodynamic datasets from which the corresponding surrogate models are constructed. Within each operating regime, the aircraft geometry, propulsor operating state and angle of attack are varied according to the selected parameter bounds.

Parameter	Strut Twist	Outboard Diameter (m)	Normalised Thrust Coefficient
Upper Bound	1.64	4	1
Lower Bound	-3	3	0

Table 2: Condition Independent Parameters & Bounds

Condition	Cruise	Flaps Retracted	Flaps Deployed
Propeller tip Mach	[0.3 : 0.6]	[0.3 : 0.8]	[0.3 : 0.8]
Angle of attack	[-2.5 : 5]	[2 : 12]	[2 : 12]

Table 3: Condition Dependent Parameters & Bounds

The propeller thrust is defined using a normalised thrust coefficient rather than a direct definition of thrust. The minimum physical loading condition is defined using a thrust coefficient of ($C_t = 0.05$). This represents a low thrust operating condition for both the inboard and outboard propellers.

The maximum achievable thrust is not constant throughout the design space. It depends on the propeller tip Mach number, propeller diameter and atmospheric properties associated with the selected flight regime. The propeller map developed by TUBS is used to determine the corresponding maximum achievable thrust for each combination of these parameters. As this varies nonlinearly across the design space, prescribing thrust directly would produce parameter bounds that depend on the defined propeller variables.

A normalised thrust parameter is therefore used to describe the relative thrust coefficient. This provides a consistent definition of propeller thrust whilst ensuring that thrust applied to each actuator disc remains consistent with the propeller diameter, operating tip Mach number and atmospheric condition. Separate normalised thrust parameters are used for the inboard and outboard propeller groups, allowing their total thrust to be defined independently.

The surrogate input vector is defined as

$$\mathbf{V} = [Twist, D_{out}, M_{in}, M_{out}, Ct_{in}, Ct_{out}, \alpha]$$

Here $Twist$ defines the strut-twist parameter, D_{out} is the outboard propeller diameter, M_{in} and M_{out} are the tip Mach numbers of the inboard and outboard propeller groups, Ct_{in} and Ct_{out} are their respective normalised thrust inputs, and α is the aircraft angle of attack.

The completed surrogate is intended to accept an aircraft configuration vector and return the corresponding lift, drag and moment coefficients. It may be queried at an individual design point or repeatedly across a prescribed angle-of-attack range to generate complete aerodynamic polars.

3.2. Snapshot level fidelity and generation

The preliminary VSPAero-based aerodynamic model described in Section 3.1.1 was suitable for aircraft-design studies, where it enabled rapid prediction of aerodynamic coefficients and the parameters required to construct linear lift and quadratic drag curves. However, the underlying potential-flow formulation is inherently limited in its representation of compressibility, viscous effects and separated-flow behaviour. A higher-fidelity model was therefore developed, made computationally feasible by the reduction in parameter-space dimensionality achieved during the preliminary investigations.

For a design space of “ n ” independent parameters, the minimum number of required samples required to capture the corners and central point of the domain is $N_{min} = 2^n + 1$. For the initial campaign, 64 parameter combinations were evaluated at three angles of attack, producing 192 aerodynamic snapshots. An additional central case and face centred axial cases increased the initial snapshot database to 205 solutions per surrogate. These were generated using high-fidelity RANS simulations with a source-term actuator-disc representation of the propellers.

To improve computational efficiency, Euler simulations were first used to drive mesh adaptation. This increased resolution within the propeller wakes and other regions of strong aerodynamic interaction. The adapted inviscid mesh then formed the foundation of a shared viscous mesh for each strut-twist definition, retaining refinement in important flow regions while limiting unnecessary mesh detail elsewhere.

SU2 applies these forces as source terms within the governing flow equations, introducing the time-averaged influence of the propellers without explicitly resolving the blades or their rotation. This captures the principal effects of propeller loading and slipstream development at substantially lower cost than a geometrically resolved rotating-propeller simulation. Each simulation generated both volume and surface solutions. The principal quantities required for the reduced-order surrogate were the surface pressure coefficient and skin-friction coefficient vector, which contain the pressure and viscous information needed to reconstruct the aerodynamic loading.

3.3. Proper Orthogonal Decomposition

The 205 high-fidelity surface solutions form the snapshot database used to construct the reduced order model. The objective is to use the available RANS solutions to reconstruct new, out-of-sample aerodynamic surface solutions without requiring an additional CFD simulation at every queried point in the parameter space.

A non-intrusive, snapshot-based Proper Orthogonal Decomposition (POD) methodology is utilised. In the offline stage, the surface pressure and skin-friction fields from the available CFD snapshots are assembled into a common h5 database. POD identifies a set of orthogonal spatial modes representing the dominant variations present across these solutions. The original high-dimensional surface data can consequently be represented using a substantially smaller set of modal coefficients. For a new aircraft parameter vector, the modal coefficients are estimated within the sampled design space and combined with the retained POD modes to reconstruct the aerodynamic surface solution. The reconstructed field may be expressed as

$$u(x) \approx \hat{u} = \sum_{i=1}^{N_m} a_i \phi_i(x)$$

This constitutes the online stage of the reduced-order model and allows new surface solutions to be generated at substantially lower computational cost than the original RANS simulations. [4]

Importantly, the POD model reconstructs the aerodynamic surface quantities rather than directly predicting the integrated aircraft coefficients. The reconstructed pressure and skin-friction can therefore be integrated internally and the target aerodynamic coefficients returned. As the surface data retain the SU2 mesh-marker definitions, the integration can also be restricted to individual aircraft components or surface regions, allowing the aerodynamic contribution of features such as the wing, strut, or the nacelles to be extracted independently.

3.4. Initial Surrogate Results

The cruise surrogate can generate surface solutions for a variety of input design vectors; allowing the rapid evaluation of potential designs with C_L , C_D , and C_M being computed from the surface pressure and the skin friction. This can be used in a variety of forms, from producing polars of lift and drag, defining the maximum achievable aerodynamic efficiency, locating minimum drag under constrained conditions, understanding how sensitive lift and drag is to changes in parameters.



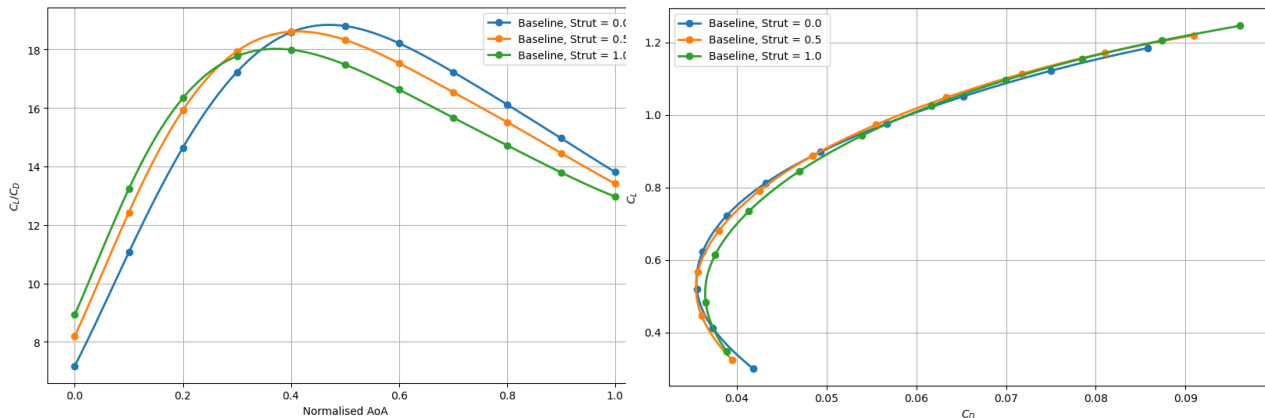


Figure 1: Demonstration of surrogate output curves for midpoint design with varying strut twist. Lift to drag ratio vs AoA (left) and design polars (right)

However, there is a need to understand the underlying uncertainty; how accurate the surrogate model truly is. Therefore, a series of testing points were simulated to use as a benchmark of the initial surrogate model

Test Case	Alpha	Strut Twist	Diameter	Mach In	Mach Out	Ct In	Ct Out
1	0.872	0.415	0.270	0.931	0.533	0.081	0.479
2	0.053	0.364	0.404	0.733	0.933	0.449	0.598
3	0.312	0.204	0.762	0.592	0.709	0.173	0.295
4	0.661	0.918	0.639	0.626	0.622	0.256	0.095
5	0.586	0.774	0.147	0.473	0.856	0.611	0.235
6	0.424	0.722	0.866	0.821	0.677	0.848	0.826
7	0.838	0.622	0.470	0.266	0.254	0.912	0.307
8	0.209	0.110	0.250	0.396	0.568	0.892	0.659
9	0.459	0.560	0.336	0.121	0.768	0.201	0.763
10	0.379	0.275	0.161	0.196	0.383	0.376	0.113
11	0.256	0.520	0.364	0.774	0.113	0.737	0.197
12	0.733	0.824	0.078	0.653	0.317	0.452	0.737
13	0.946	0.190	0.574	0.520	0.443	0.683	0.906
14	0.53	0.089	0.714	0.851	0.237	0.524	0.382
15	0.14	0.854	0.653	0.098	0.473	0.560	0.528
16	0.171	0.652	0.517	0.446	0.187	0.138	0.874
17	0.795	0.461	0.804	0.212	0.837	0.778	0.417

Table 4: Surrogate CFD test samples

The points outlined in Table 4 are evenly distributed across the domain, simulated using the described actuator disk models then coefficients of lift and drag calculated. From there the surrogate model is used to generate the corresponding surface results and compute the same coefficients. The absolute delta in lift and drag coefficients is noted, along with the relative distance of that point from the nearest neighbour. This is intended to correlate the observed error between the surrogate and CFD with the distance of that point to the nearest surrogate sample. An ideal surrogate will display errors of less than 10% with an exceptional result being a minimum error of no more than 5%.

Test Case	C_L CFD	C_D CFD	C_L POD	C_D POD	C_L Error	C_D Error
1	1.146	0.064	1.117	0.070	2.56%	8.18%
2	0.340	0.035	0.328	0.037	3.55%	7.37%
3	0.606	0.036	0.598	0.038	1.36%	4.89%
4	1.003	0.052	1.036	0.058	3.23%	12.06%
5	0.929	0.047	0.948	0.052	1.95%	10.02%
6	0.794	0.039	0.846	0.036	6.51%	8.76%
7	1.077	0.065	1.080	0.072	0.28%	10.50%
8	0.501	0.033	0.506	0.033	0.97%	1.11%
9	0.807	0.040	0.827	0.041	2.40%	3.44%
10	0.672	0.038	0.677	0.040	0.66%	4.83%
11	0.558	0.036	0.583	0.035	4.59%	2.71%
12	1.079	0.055	1.061	0.064	1.64%	15.62%
13	0.976	0.083	1.098	0.080	12.55%	3.95%
14	0.838	0.042	0.826	0.044	1.52%	5.08%
15	0.462	0.035	0.516	0.031	11.64%	12.13%
16	0.497	0.034	0.524	0.032	5.46%	6.01%
17	1.087	0.058	1.041	0.067	4.23%	15.07%

Table 5: Simulated lift and drag coefficients compared with surrogate predictions

Table 5 clearly indicates that the current level of sampling is inadequate for the cruise condition. With peak errors in lift and drag of 12.55% and 15.07% respectively. It is noted that the higher errors are seen with cases with a low angle of attack, and high total thrust for both inboard and outboard propellers. This is not unexpected as the propellers initiate complex flow behaviour and thus will need a higher sampling resolution to resolve.

3.5. LOO Cross Validation & LOO-Error Based Sampling

A secondary surrogate evaluation can be performed using a “Leave-one-out” (LOO) process. LOO is a method of evaluating surrogate performance without the direct generation of new sample points [5]. The core process is that the full surrogate data consist of n samples, a surrogate is then constructed with $n-1$ points, then the “ n th” point is predicted. This is repeated for all non-corner points, with our

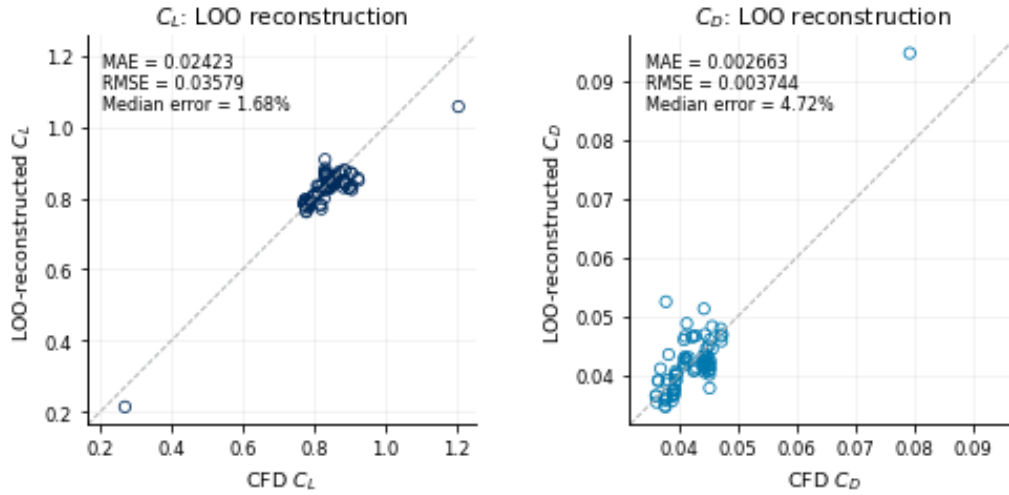


Figure 2: Leave-one-out cross validation of lift (left) and drag (right) coefficients

initial domain consisting of 205 samples, we have 128 defined corner points and thus 77 samples that are valid for the LOO process.

Figure 2 demonstrates the correlation between the CFD and LOO reconstructed error. Overall, the correlation holds well for the lift coefficient with median error of 1.68%. Meanwhile the drag coefficient construction is worse, with a median error of 4.72%. Results show that the cruise surrogate is broadly conservative when predicting configuration performance. Lift is underpredicted, and drag is broadly overpredicted, resulting in a reported aerodynamic efficiency lower than that seen in CFD.

The first enrichment stage is driven by the measured LOO error. Separate Gaussian Process models are fitted to the C_L and C_D errors. At any candidate point, the models provide both a predicted error magnitude and an uncertainty estimate. The two aerodynamic error fields are scaled by their observed maxima so that lift and drag errors contribute on a comparable basis.

An acquisition function is then applied with two major components. First, “exploitation” defining candidates with high LOO errors. Then “exploration” favouring candidate points with high GP uncertainty, allowing samples to be generated where the error field is poorly known. A tertiary component is also applied to account for the local density of the existing CFD database, a Latin hypercube cloud of points is generated within the domain to create an estimate of the Voronoi territory associated with each support point.

Candidate locations are then scored and assessed such that points are defined as exploitation dominant or exploration dominant. A weight λ is used across the batch so that the selection can be tuned to prefer one mode over the other. A further minimum distance constraint is applied to avoid point clustering

The batch is generated pseudo-sequentially. After each point is selected, it is temporarily inserted into the GP training set using a Kriging-believer update, where the current GP mean is used as the unknown error at that point. This allows subsequent selections to respond to the information that is expected to be gained from earlier members of the batch without requiring CFD to be run immediately. The resulting n points therefore represent a compromise between high predicted surrogate error, high epistemic uncertainty and insufficient local CFD coverage.

Objective	Lambda	Alpha	Strut twist	Diameter	Mach1	Mach2	Ct1	Ct2
Exploitation	0.15	0.013	0.6	0.518	0.027	0.711	0.099	0.497
Exploration	0.85	0.01	1.0	0.947	0.479	0.488	0.979	0.477
Exploitation	0.29	0.033	0.5	0.553	0.186	0.822	0.954	0.501
Exploration	0.71	0.001	0.4	0.895	0.49	0.01	0.084	0.484
Balanced	0.43	0.08	0.1	0.568	0.867	0.401	0.912	0.507
Balanced	0.57	0.075	0.1	0.516	0.833	0.748	0.09	0.514

Table 6: Samples proposed from LOO adaptive sampling process

Table 6 shows a resulting set from applying the LOO process upon the initial surrogate. The table shows that whilst a variation between the exploration and exploitation phases have been applied, the proposed samples remain clustered in specific parameters, with samples being proposed near to the minimum of the AoA domain. Whilst this is valid due to the complex flow behaviour at these points, the heavy weighting will lead to regions being potentially poorly supported.

3.6. Sparse Region Sampling

The second stage is deliberately independent of the aerodynamic error model. Once the LOO-driven locations have been proposed, their coordinates are added to the current CFD locations to form a planned-support database. At this stage, no C_L , C_D , POD reconstruction or GP prediction is required. The sparse sampler operates solely on the geometry of the sample locations in the same seven-dimensional design space.

A Latin-hypercube cloud is once again defined to characterise the surrounding domain, with each cloud point assigned to the nearest member of the domain after the LOO adaptive sampling process had occurred. This allows for the associated Voronoi territory for a given point to be determined, with a large volume corresponding to a sparse region. Within these regions candidate points are chosen to be furthest from the support set, providing the greatest reduction in volume for sampled points.

The resulting batch from this sampling process is also constructed sequentially. After each sparse point is selected, it is immediately inserted into the planned-support database, and the complete Voronoi partition is recomputed before the next selection. This is essential as the state of the design space changes after every added point. A region that was initially dominant may cease to be sparse after one or two additions, causing subsequent samples to move elsewhere.

3.7. Enhanced Surrogate Comparison

With the adaptive sampling process defined, the RANS samples were run until convergence and added to the original samples to train a new surrogate model. As before, a LOO error analysis is conducted along with a comparison of lift and drag errors between CFD testing points

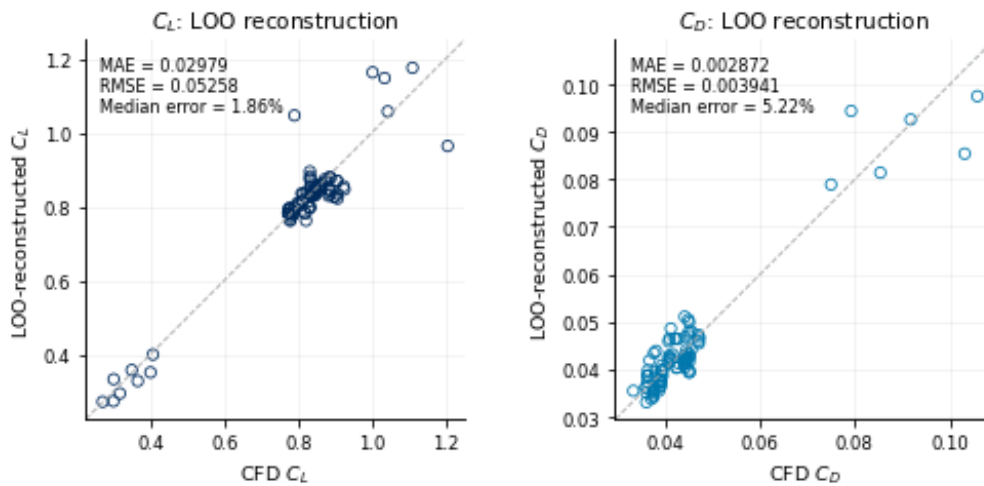


Figure 4: Leave-one-out cross validation of lift (left) and drag (right) coefficients, post-adaptive sampling.

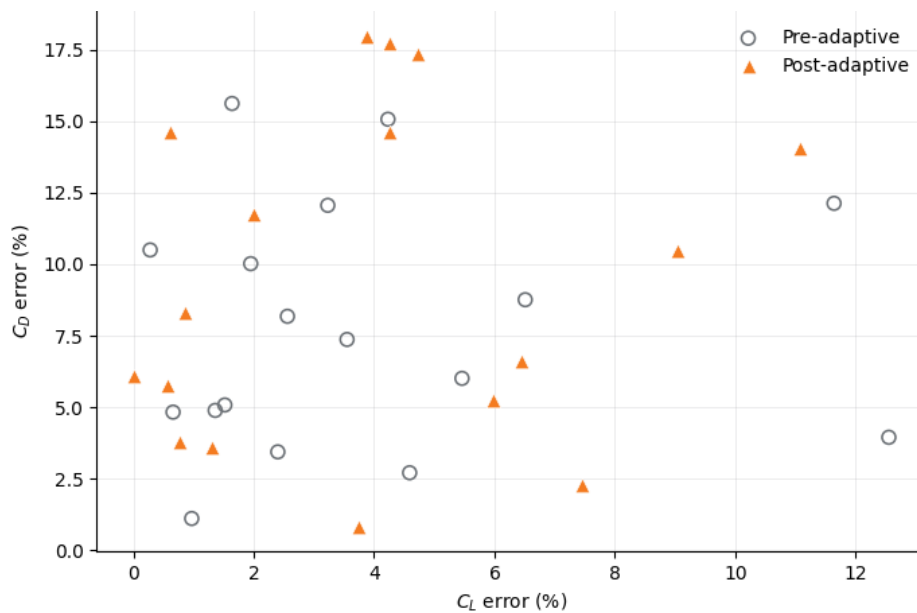


Figure 3: Comparison of lift and drag coefficient absolute errors pre and post adaptive sampling.

Figure 3 and Figure 4 provide a demonstration of how the adaptive process has impacted surrogate accuracy. The LOO analysis shows that the whilst the adaptive sampling process has provided better coverage of the domain, it is at the cost of destabilising the surrogate within the original LOO samples. The reported median error has increased by an absolute value of 0.18% and 0.5% respectively. When comparing the error across the testing set several changes can be seen. Firstly, there has been an improvement in the lift coefficient predictions; only a single sample is above the 10% error benchmark, along with the maximum and minimum errors reducing from 12.55% & 0.28% to 11.08% and 0.02%. However, drag accuracy has been marginally impacted, an additional case falls above the benchmark along with an increase in maximum error from 15.62% to 17.98%.

The increase in LOO error is expected, the initial samples that were included in the first LOO study were closely clustered. As the corner cases were unable to be used for the LOO study, the samples of interest were limited to the midpoint angle of attack cases and face centred axial samples. Essentially ensuring that all samples had a low distance between samples of interest and supporting samples. As such this is still evidence that whilst the results from adaptive sampling is positive, further sampling is required.

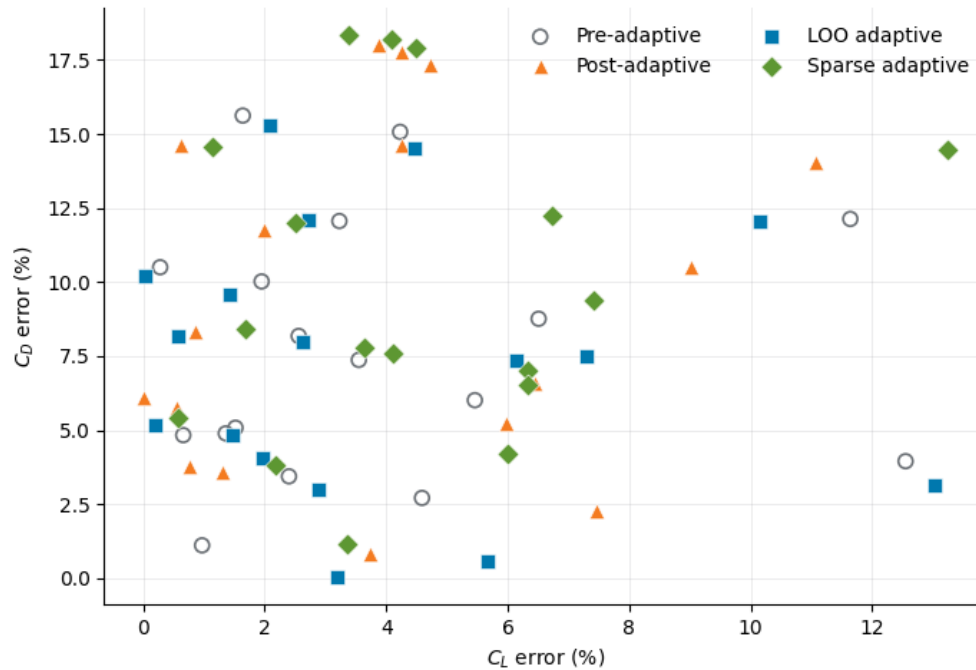


Figure 5: Comparison of lift and drag coefficient absolute errors; pre, post, solely LOO adaptive, solely sparse adaptive.

The first adaptive sampling phase utilised a balanced approach, where the number of LOO informed samples and the sparse sampling methodology were utilised equally. Figure 5 isolates the lift and drag errors of a surrogate enhanced solely with the LOO methodology and solely the sparse sampling methodology. The goal is to understand if there should be a level of biasing when choosing how many samples are generated with each method. The same destabilisation of the surrogate is seen, with a worse predictive performance seen with both lift and drag. It is important to note that as the number of adaptive samples have been halved, the overall predictive performance of the isolated methodology surrogates will have been reduced when compared to the full post adaptation surrogate. However, when focusing upon the comparative performance between the LOO adaptive and sparse adaptive approaches a clear advantage is seen. The maximum errors in both lift and drag coefficients with the LOO adaptive approach is lower than that of the sparse adaptive approach. The maximum absolute error difference between the two approaches is 0.20% and 3.05% in lift and drag respectively. Given the number of additional samples were the same for each approach, there is evidence that there should be additional bias towards the LOO adaptive approach when compared to the sparse adaptive approach.

3.8. Surrogate parameter sensitivity study

With the surrogates created for the different conditions it is useful to understand how the parameters vary across different operating conditions. Therefore, a parameter sensitivity study was conducted, centred on an optimised design for each operating condition. A global search of the design space was used to first determine the design parameters that maximised aerodynamic efficiency. This maximum configuration was then used to define the reference point around which design variables were varied across the full normalised range. For each parameter pair, the remaining five inputs were held fixed. The resulting heatmaps describe how the aerodynamic performance changes when the configuration is perturbed away from the identified optimum in specific directions through the design space. Within each map, aerodynamic efficiency is represented by the background field, while contours of lift and drag are included to distinguish changes in overall aerodynamic efficiency from changes in the underlying force coefficients. The locations of maximum lift to drag ratio, maximum lift, and maximum drag are also identified, allowing the optimum to be compared directly with conditions producing the greatest aerodynamic loading. Drag contours are indicated in green, with a green cross highlighting the maximum drag location. Lift contours and maxima are similarly displayed in orange.

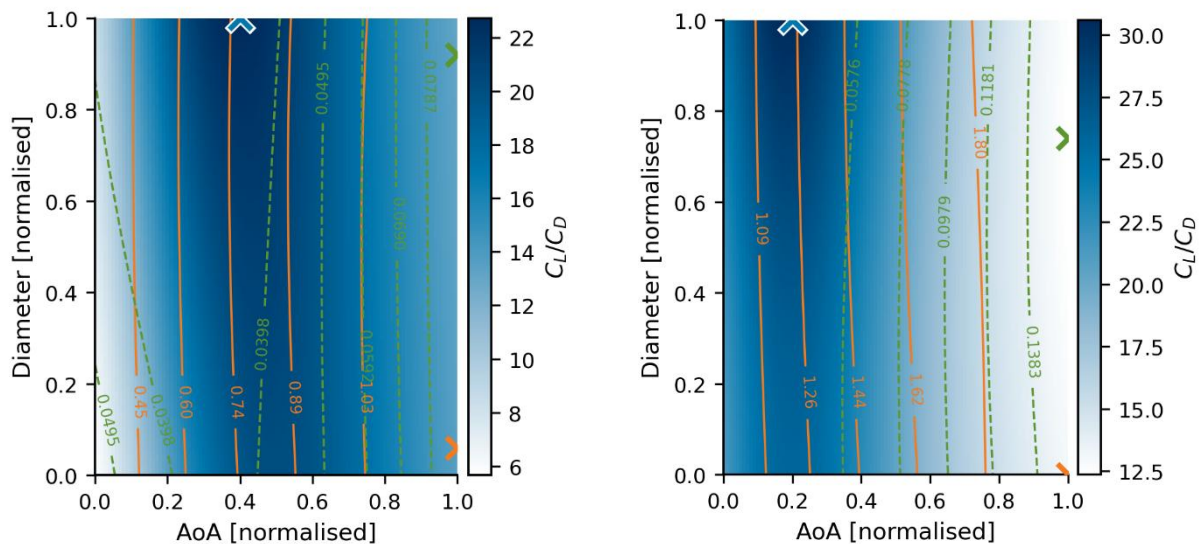


Figure 6: Combined sensitivity of angle of attack and propeller diameter for the high-speed condition (left) and low-speed condition (right)

The combined influence of angle of attack and propeller diameter is shown in Figure 6. Angle of attack provides the strongest and most clearly defined control over aerodynamic efficiency in both operating conditions. The aerodynamic efficiency field varies predominantly with incidence, producing a narrow high efficiency region, while changes in outboard propeller diameter modify the magnitude and shape of this response. The position of the optimum is condition dependent: the cruise case favours a normalised AoA of approximately 0.4, whilst the low speed condition demonstrates an optimum AoA of 0.2. In both cases increasing the AoA beyond the optimum continually increases lift, however, the corresponding increase in drag is proportionally greater than any lift increase. This is further demonstrated through the separation between the highest efficiency and the highest lift for both cases.

Diameter exhibits a consistent trend across both conditions. Increasing the propeller diameter generally shifts the response towards higher aerodynamic efficiency, with the maximum efficiency

positioned at the upper limit of the investigated range. The effect is especially present where the increase in diameter is accompanied by both an increase in lift and a reduction in drag. Although the precise form of this response differs between operating conditions; the repeated preference for increased diameter indicates that it represents one of the stronger propulsion sensitivities within the investigated design space.

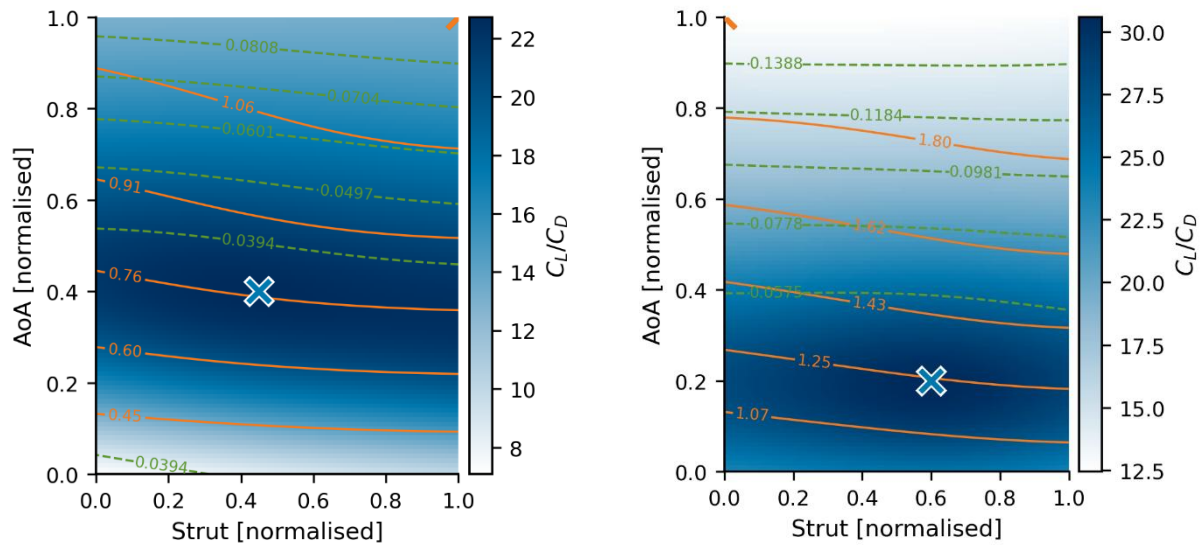


Figure 7: Coupled sensitivity of Strut setting and angle of attack for the high-speed condition (left) and low-speed condition (right)

The interaction between strut twist and angle of attack is shown in Figure 7. In contrast to diameter, the preferred Strut value remains within the interior of the design space and changes between the two operating conditions. The cruise condition favours a normalised strut twist of 0.4 to 0.5, whilst the low-speed condition shifts to approximately 0.6. The change occurs in tandem with a shift in optimum AoA and indicates the optimum geometry of the strut is dependent upon the flight operation rather

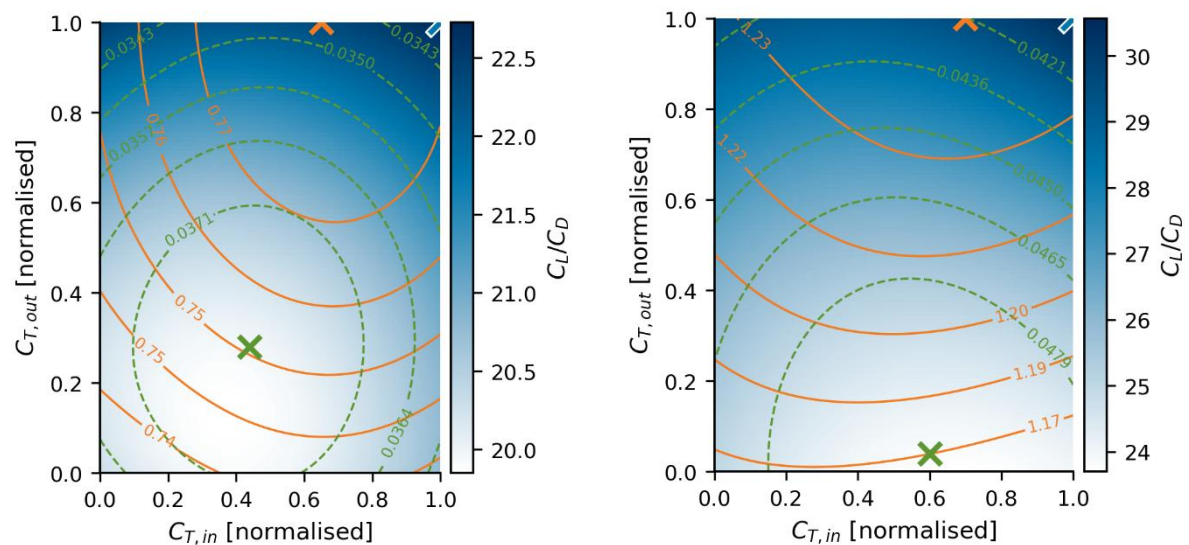


Figure 8: Sensitivity to inboard and outboard thrust coefficient for the high-speed condition (left) and low-speed condition (right).

than a single global optimum across the flight influence. Furthermore, the normalised strut influence

on aerodynamic efficiency is less pronounced than that of AoA. However, the curvature of the lift and drag contours indicates a coupled aerodynamic response. The preferred strut twist values are also notable when compared with several propulsion parameters, for which the maximum-efficiency solution is driven towards the limits of the surrogate domain. The influence of thrust allocation is considered through the efficiency maps displayed in Figure 8. Both operation conditions generally favour increased thrust for maximising efficiency. However, the individual lift and drag contours demonstrate that the inboard and outboard thrust coefficients do not produce equivalent aerodynamic responses. The locations of maximum lift, maximum drag and maximum efficiency remain separated, indicating that increasing propulsive loading does not solely move the configuration towards a common aerodynamic optimum. This bias becomes more pronounced at low speed, with efficiency gradients being highest. Variations in outboard thrust coefficient produce stronger changes in efficiency than comparable changes in inboard thrust coefficient. Furthermore, portions of the efficiency field show that outboard thrust variations can simultaneously increase lift whilst also reducing drag. The resulting improvement in aerodynamic efficiency demonstrates that the influence of propulsive loading depends on its spanwise distribution as well as its overall magnitude. This behaviour is less pronounced in the cruise condition, indicating that the value of thrust redistribution varies with operating condition.

Overall, the sensitivity maps identify both common and condition dependent behaviour of the design parameters. Angle of attack remains the dominant operating variable, while increased propeller diameter and higher thrust loading parameters are favourable to efficiency. The optimal strut twist along with the influence of inboard and outboard thrust change between the cruise and low speed conditions. The separation between maximum lift, drag, and efficiency demonstrates that the aerodynamic optimum results from the coupled balance between lift enhancement and drag growth rather than from maximising a single parameter independently.

The high-fidelity surrogates have been effective in investigating how design parameters effect operation in each condition, identifying unique behaviour across the different conditions.

3.9. Aerodynamic representation adopted for WP5 optimisation activities

The high-fidelity aerodynamic surrogate described in the preceding sections was developed within Task 3.2 to increase the physical fidelity available for aerodynamic assessment. In parallel, the aircraft-level optimisation activities in WP5 required an aerodynamic representation that could be evaluated repeatedly within large multidisciplinary optimisation problems. For these activities, a different reduced aerodynamic formulation based on VSPAero was adopted [6, 7].

The wing aerodynamic model was generated through a discipline-level aerodynamic optimisation performed at prescribed values of wing aspect ratio. For each aspect ratio, a parametric strut-braced-wing model was optimised by varying the wing twist at the root, mid-span and tip, together with the strut twist and the outer-wing taper ratio. The aerodynamic coefficients C_{L0} , $C_{L\alpha}$, C_{D0} , $C_{D\alpha}$ and $C_{D\alpha\alpha}$ obtained for the resulting optimal configurations were subsequently fitted across aspect ratio and used to reconstruct the wing aerodynamic polars within the aircraft-level MDA.

This formulation effectively introduced a bi-level aerodynamic design strategy: detailed wing-shape variables were treated within the offline aerodynamic optimisation, while only the reduced aircraft-level parametrisation was exposed to the multidisciplinary optimiser. The complete aircraft aerodynamic model combined the wing representation with separate contributions for the horizontal

and vertical tails, fuselage and additional parasite-drag sources, and provided the aerodynamic coefficients required for trim and mission-performance evaluation.

The approach was adopted for the refined WP5 aircraft-design studies and subsequently used in the single- and multi-trajectory optimisation campaigns. It should therefore be distinguished from both the preliminary WP1 aerodynamic model described in Section 3.1.1 and the CFD/POD surrogate developed within Task 3.2: the former was intended for broad coupled aerodynamic–propulsive exploration, whereas the WP5 formulation reduced the aircraft-level aerodynamic design dimensionality by embedding the detailed wing design within an offline aerodynamic optimisation.

4. Task3.3 - Low- to high-fidelity aeroacoustics models

4.1. Multi-fidelity aeroacoustic modelling strategy

The aeroacoustic assessment of Distributed Hybrid Electric Propulsion (DHEP) systems requires numerical methodologies capable of balancing computational efficiency and physical fidelity. Different stages of the aircraft design process demand different levels of modelling complexity, ranging from rapid preliminary assessments to detailed investigations of complex installation effects. Consequently, a progressive multi-fidelity modelling strategy was adopted throughout the project, in which complementary numerical approaches were employed according to the objectives of each stage of the analysis.

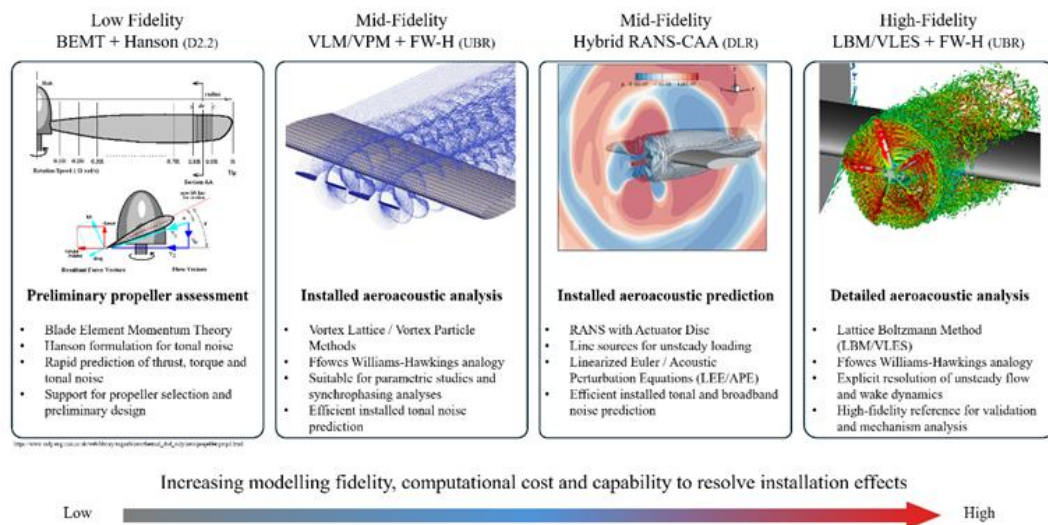


Figure 9: Multi-fidelity aeroacoustic modelling framework adopted throughout the project, illustrating the progression from low- to high-fidelity numerical methodologies and their respective roles in the aeroacoustic assessment of distributed propulsion systems

The modelling activities started with low-fidelity tools for isolated propellers based on Blade Element Momentum Theory (BEMT) coupled with a rotating frame Ffowcs-Williams Hawkins surface integral formulation, as documented in D2.1 [8]. These models provided a computationally efficient framework for preliminary propeller assessment, enabling rapid estimation of aerodynamic performance and tonal noise over a broad range of operating conditions.

Building upon this foundation, Task 3.3 extended the modelling capability towards installed propulsion systems through complementary medium- and high-fidelity numerical methodologies. Medium-fidelity approaches were developed to support efficient parametric investigations while capturing the dominant aerodynamic and aeroacoustics mechanisms associated with propeller-wing interaction. High-fidelity simulations were subsequently employed to resolve the unsteady flow field and aeroacoustics sources in greater detail, providing reference solutions for validation and detailed mechanism analysis.

Collectively, these methodologies constitute an integrated multi-fidelity framework supporting the aeroacoustics assessment of DHEP configurations across different stages of the design process. Besides providing physical understanding of installation effects, the resulting numerical datasets establish the foundation for the surrogate modelling activities developed within Task 3.3 and provide aeroacoustics inputs for the multidisciplinary design optimisation activities carried out in subsequent work packages.

4.2. Mid-fidelity methodologies

4.2.1. University of Bristol – VLM/VPM + FW-H

To investigate aeroacoustics installation effects over a broad range of operating conditions, a medium-fidelity numerical framework was developed at the University of Bristol by coupling a vortex-based aerodynamic solver with an acoustic propagation model. The methodology was specifically conceived to provide an efficient balance between computational cost and physical fidelity, enabling extensive parametric investigations that would be impractical using high-fidelity simulations alone.

The aerodynamic solution is obtained using a hybrid Vortex Lattice Method (VLM) and Vortex Particle Method (VPM) formulation. Lifting surfaces, including the propeller blades and wing, are discretised using vortex lattice panels, while the wake is shed from the trailing edges and subsequently evolved as Lagrangian vortex particles. This approach captures the dominant aerodynamic interaction mechanisms associated with distributed propulsion systems while maintaining a relatively low computational cost.



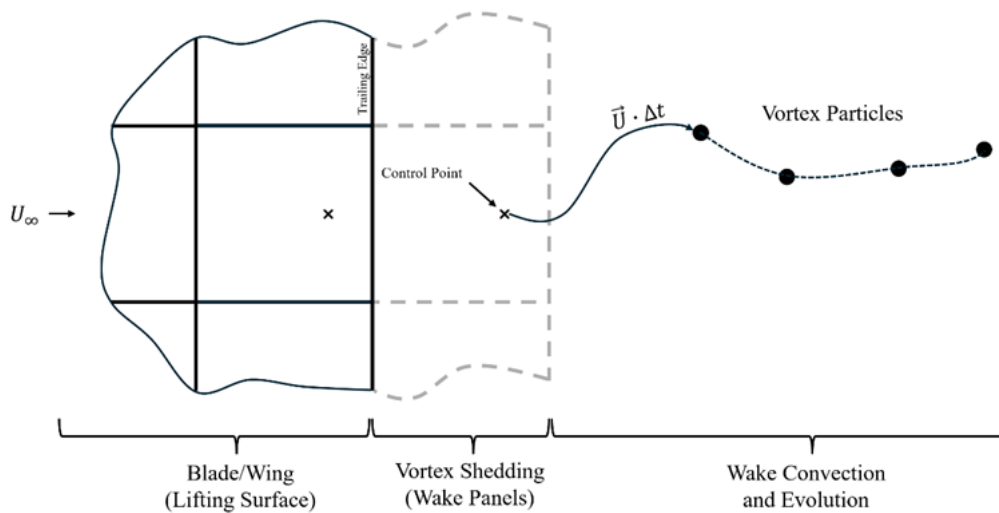


Figure 10: Schematic representation of the hybrid VLM–VPM aerodynamic framework. Lifting surfaces are discretized using vortex lattice panels, while the wake is initially represented by wake panels and subsequently converted into Lagrangian vortex particles for wake convection and evolution. Adapted from [9]

The unsteady aerodynamic loads predicted by the vortex-based solver are coupled with the Ffowcs Williams–Hawkins (FW-H) acoustic analogy to predict far-field tonal noise. Once periodic steady-state conditions are achieved, pressure fluctuations are propagated to prescribed observer locations, allowing the evaluation of sound pressure levels, acoustic directivity and three-dimensional radiation patterns.

The methodology was validated against aeroacoustics measurements obtained at the University of Bristol using synchronised propeller configurations under inflow conditions. Comparisons demonstrated good agreement with the experimentally observed tonal directivity trends over a range of relative phase angles, providing confidence in the capability of the framework to capture the dominant aeroacoustics mechanisms governing propeller interaction, see Sec. 4.4.1.

Following its validation, the framework was employed in a series of parametric investigations addressing relative phase synchronisation between propellers, allowing systematic assessment of installation effects, tonal noise redistribution and observer-dependent acoustic interference [9].

4.2.2. German Aerospace Centre (DLR) – Hybrid RANS–CAA

To augment the prediction capabilities and shortcomings associated with the vortex-based medium-fidelity framework developed at the University of Bristol, the German Aerospace Centre (DLR) contributed with a hybrid Reynolds-Averaged Navier–Stokes (RANS) and Computational Aeroacoustics (CAA) methodology specifically designed for the prediction of installed propeller noise. The approach provides an efficient compromise between computational cost and physical fidelity by combining steady aerodynamic simulations with dedicated acoustic source modelling and wave propagation techniques.

The aerodynamic flow field is first computed using steady RANS simulations in which the rotating propeller is replaced by an Actuator-Disc (AD) modelling, providing the necessary background mean-

flow input for the CAA perturbation approach. In the CAA approach, regularised propeller forces and displacement sources are defined based on the AD RANS results, replacing the geometry of the propeller blades. The sources obtained are prescribed to the Linearised Euler Equations (LEE), enabling the prediction of both the far-field and nearfield acoustic tonal response of a propeller. In this way, no rotating meshes are required, avoiding the excessive computational cost associated with fully resolved rotating-blades simulations. The developed workflow is implemented in the DLR CAA solver PIANO with immersed boundary representation of the non-rotating airframe components on hierarchical-Cartesian grids. With this method, sound propagation and refraction in non-uniform flow, as well as scattering, shielding and diffraction on solid surfaces, can be simulated, also taking into account the installation noise from the interaction of the blade tip vortices with the airframe.

The methodology was assessed using representative installed propulsion configurations developed within the project and compared against both experimental measurements and higher-fidelity numerical simulations [10]. These comparisons demonstrated the capability of the hybrid RANS–CAA framework to reproduce the dominant installation effects while providing substantially reduced computational requirements, making it particularly suitable for the aeroacoustics investigation of the INDIGO aircraft installed powertrain.

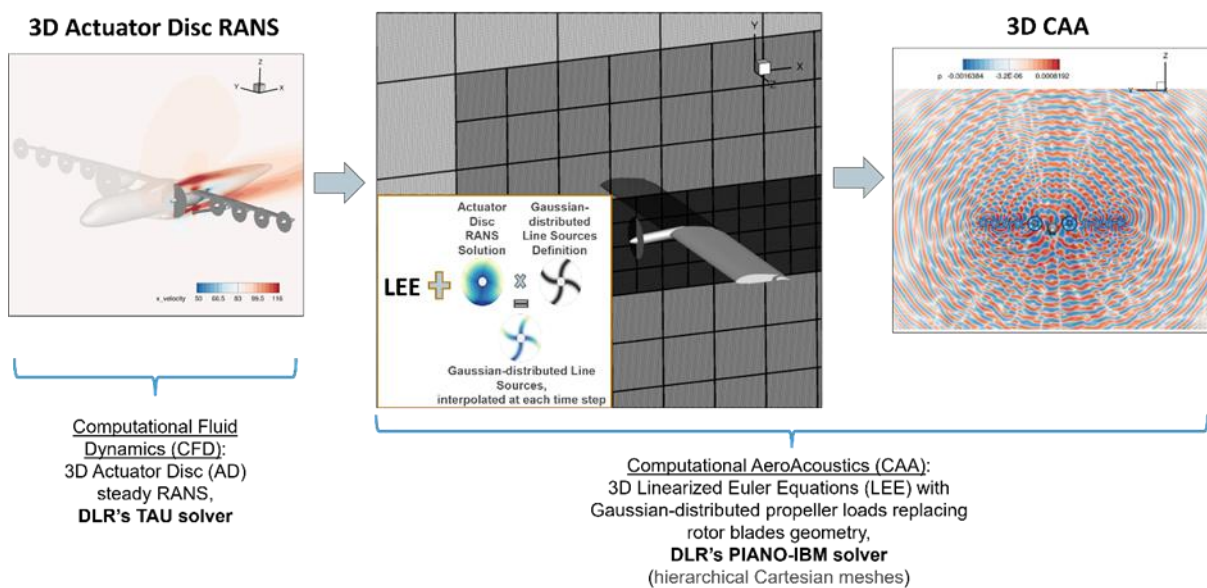


Figure 11: Schematic of the hybrid RANS–CAA workflow adopted by DLR.

4.3. High-fidelity methodology – LBM/VLES + FW-H

To complement the medium-fidelity methodologies developed within Task 3.3, a high-fidelity aeroacoustic framework based on the Lattice Boltzmann Method (LBM) coupled with Very Large Eddy Simulation (VLES) was employed to provide detailed resolution of the aerodynamic flow field and the associated aeroacoustic sources. Owing to its capability to explicitly capture unsteady flow structures and wake development, the methodology serves as the highest-fidelity numerical approach adopted within the project's aeroacoustic modelling framework.

The simulations were performed using the commercial solver PowerFLOW®, in which the compressible LBM formulation is combined with a VLES turbulence model to resolve the dominant turbulent structures responsible for aerodynamic interaction and noise generation. Far-field acoustic predictions were subsequently obtained using the Ffowcs Williams–Hawkins (FW-H) acoustic analogy based on pressure fluctuations collected over the relevant solid surfaces. This approach enables the prediction of both tonal and broadband noise while preserving the complex flow physics associated with propeller-wing interaction.

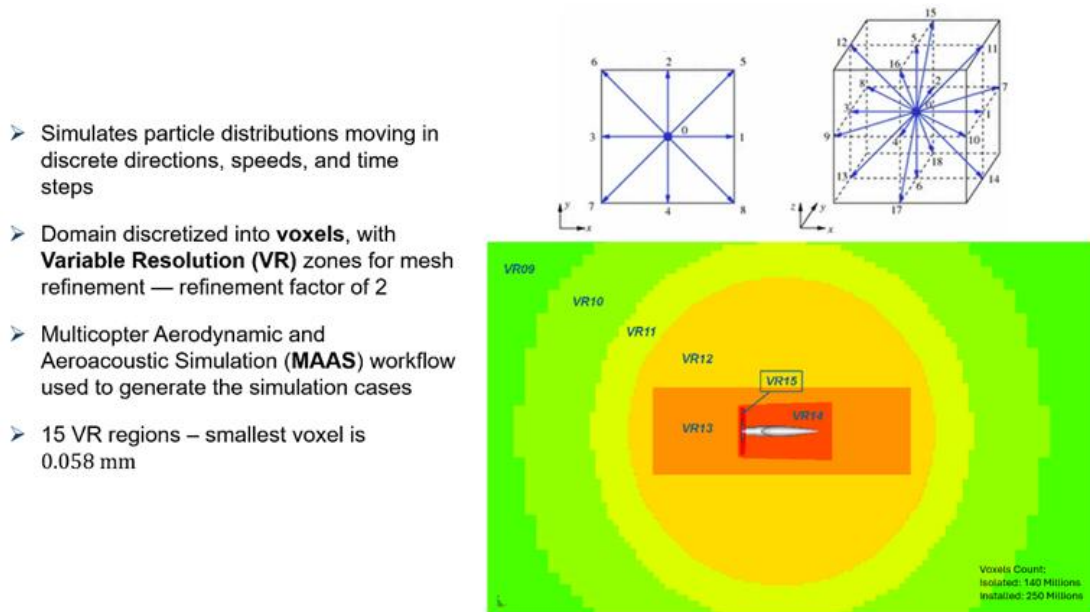


Figure 12: Computational framework adopted for the high-fidelity LBM/VLES simulations and the Variable-Resolution (VR) regions used. Adapted from [10]

The high-fidelity framework was applied to representative installed propeller configurations developed within the project, allowing detailed investigation of aerodynamic installation effects and the resulting aeroacoustic radiation. The numerical results were compared against dedicated experimental measurements and complementary medium-fidelity predictions, enabling a comprehensive assessment of the modelling capabilities across different fidelity levels. These comparisons demonstrated the ability of the LBM/VLES approach to reproduce the dominant aeroacoustic features observed experimentally while providing detailed insight into the underlying flow mechanisms.

Within the multi-fidelity strategy adopted in Task 3.3, the LBM methodology provides the reference level of numerical fidelity against which lower-cost approaches can be assessed [10].

4.4. Representative applications and validation

Representative examples of the different fidelity levels are presented below to illustrate the capabilities of the proposed numerical framework. The numerical methodologies described in the previous sections were applied to representative distributed propulsion configurations investigated throughout Task 3.3. Besides demonstrating their applicability to aeroacoustics analyses of installed propellers, selected cases were compared against dedicated experimental measurements and cross-

validated between different fidelity levels, providing confidence in the adopted multi-fidelity framework.

4.4.1. Mid-fidelity validation and application

The vortex-based mid-fidelity framework was first assessed against experimental measurements obtained at the University of Bristol for installed dual-propeller configurations operating under different relative phase angles. Particular attention was devoted to the prediction of tonal directivity and the redistribution of the acoustic field associated with synchrophasing strategies. The comparisons demonstrated that the methodology successfully reproduces the main trends observed experimentally, providing confidence in its capability to capture the dominant aeroacoustics mechanisms governing installed propeller interaction.

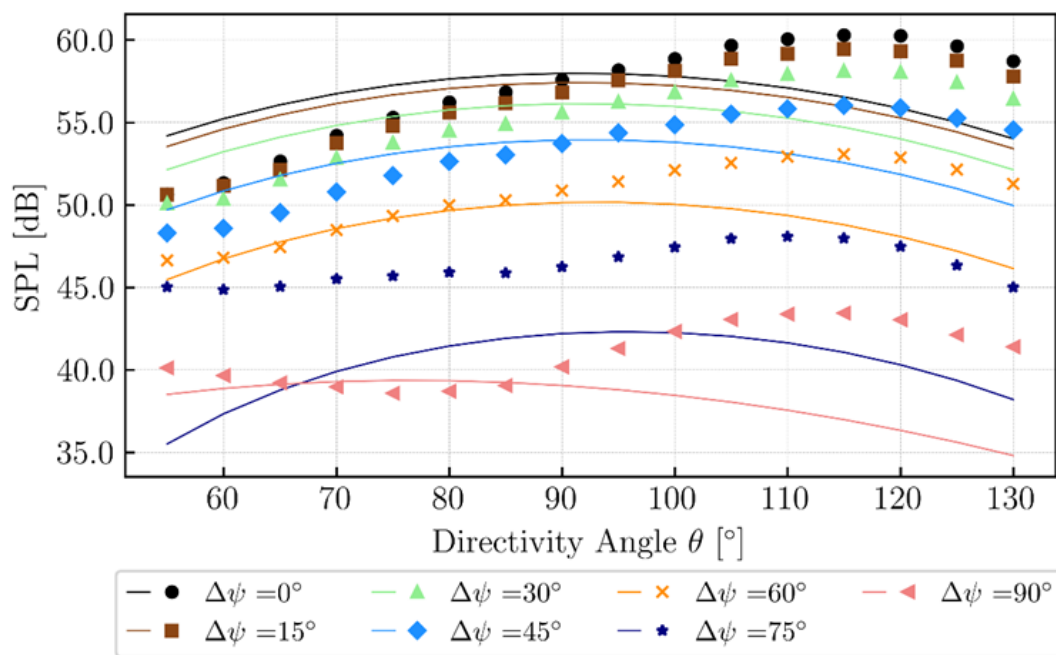


Figure 13: Comparison between measured and predicted tonal directivities for representative synchrophasing configurations, illustrating the validation of the VLM/VPM + FW-H framework against experimental measurements. Reproduced from [9].

Following its experimental validation, the vortex-based framework was employed to perform representative aeroacoustic investigations of installed propeller configurations. Owing to its computational efficiency, the methodology enables systematic parametric studies over a wide range of operating conditions while accounting for aerodynamic installation effects and propeller synchronisation. Representative tonal directivity predictions are presented in Figure 14, illustrating the capability of the methodology to efficiently evaluate the influence of different operating conditions and synchronisation states in multiple observer planes. These capabilities make the framework particularly suitable for design-space exploration and optimisation studies, where numerous operating conditions must be assessed within practical computational times. The full detail of the work is available in [9].

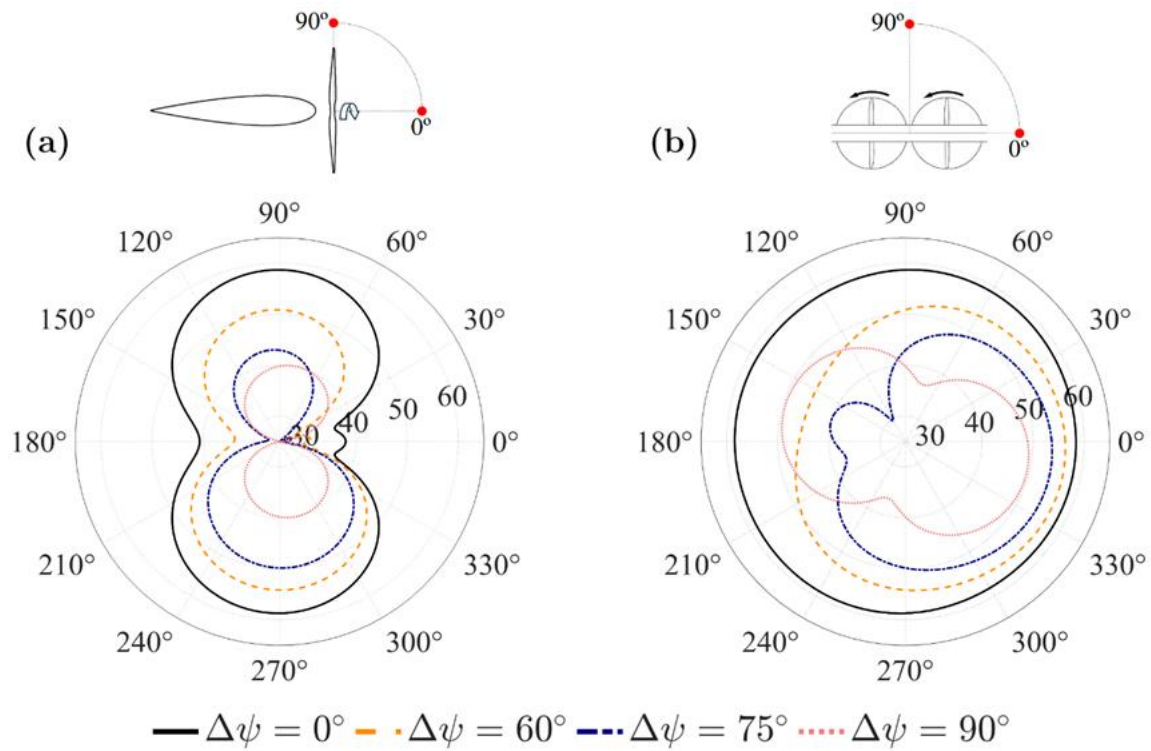


Figure 14: Representative tonal directivity predictions obtained using the proposed vortex-based framework for different propeller synchronisation conditions. The methodology enables efficient evaluation of acoustic directivity patterns in multiple observer plane supporting systematic parametric investigations of installed propulsion systems. Adapted from [9].

While directivity plots provide observer-based acoustic quantities, the same numerical framework is also capable of predicting complete three-dimensional acoustic fields surrounding the installed propulsion system. Representative three-dimensional acoustic fields are shown in Figure 15, demonstrating the ability of the methodology to resolve the global distribution of the radiated tonal noise for representative operating conditions and synchronisation states. Such global acoustic representations provide valuable physical insight into installation effects that would be difficult to obtain experimentally.

Together, the representative applications presented in Figure 14 and Figure 15 demonstrate that the proposed vortex-based methodology provides a practical compromise between computational efficiency and physical fidelity. Its capability to predict both observer-based acoustic quantities and complete three-dimensional sound fields makes it well suited for the aeroacoustics assessment of installed propulsion systems.

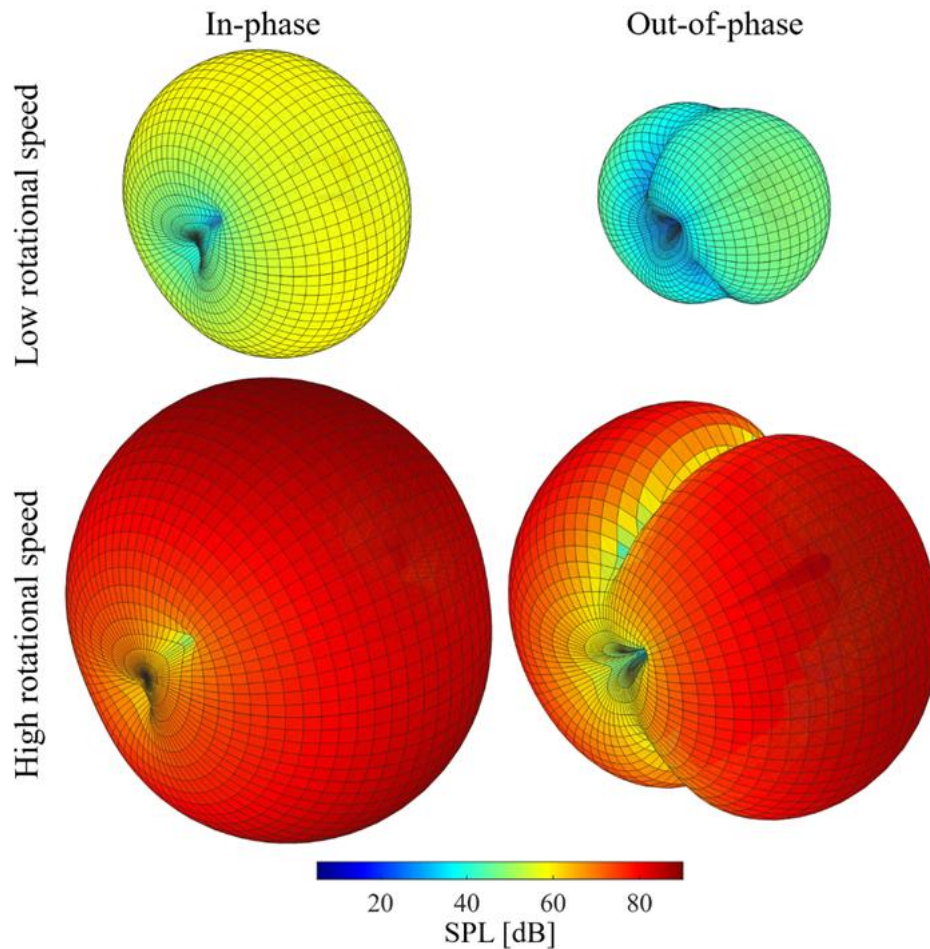


Figure 15: Representative three-dimensional tonal acoustic fields predicted using the proposed vortex-based framework for representative operating conditions and propeller synchronisation states. The figure illustrates the capability of the methodology to resolve global acoustic radiation patterns around installed propeller configurations while maintaining computational efficiency suitable for extensive parametric studies. Adapted from [9].

4.4.2. Cross-validation of medium- and high-fidelity methodologies

To further assess the applicability of the proposed modelling hierarchy, representative installed propeller configurations were investigated using complementary medium- and high-fidelity methodologies. Comparisons between the hybrid RANS-CAA framework, the high-fidelity LBM/VLES simulations and dedicated experimental measurements demonstrated consistent prediction of the dominant aeroacoustics features while highlighting the different levels of physical detail provided by each numerical approach.

These comparisons confirmed that the hybrid RANS–CAA methodology provides efficient engineering-oriented predictions suitable for parametric investigations, whereas the high-fidelity simulations offer detailed resolution of the underlying flow physics and serve as reference solutions for validation and mechanism analysis.

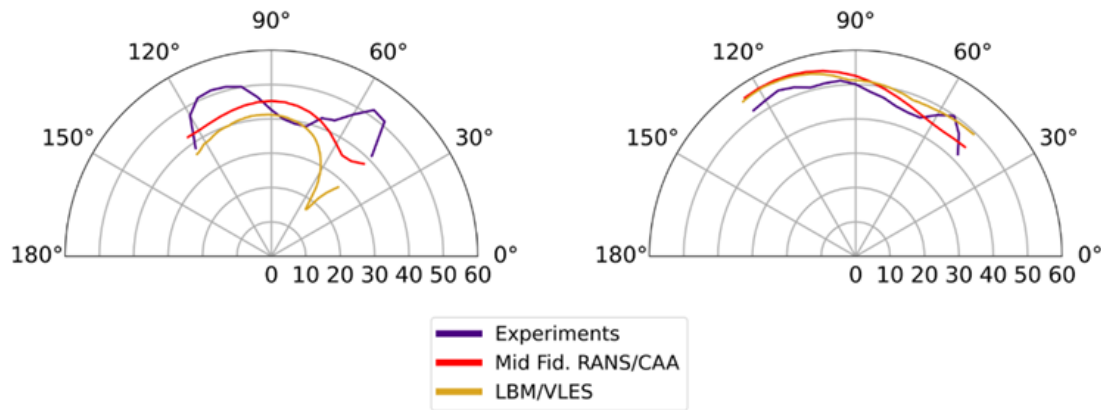


Figure 16: Comparison of experimental measurements, hybrid RANS–CAA predictions and high-fidelity LBM/VLES simulations for representative isolated (left) and installed (right) propeller configurations, illustrating the complementary capabilities of the adopted multi-fidelity framework. Adapted from [10].

To supplement the hybrid RANS–CAA methodology cross-validation results obtained in [10] for a single-element wing configuration, an additional high-lift configuration was considered in [11]. The wind tunnel model used in the experimental investigations consists in an array of three independently controllable propellers. The model was designed, fabricated and operated by TU Delft. It features an unswept, untapered wing with NLF-Mod22(B) airfoil designed by Boermans and Rutten. The geometry of the propeller used is the TUD-XPROP-S reference propeller, with the axis of rotation

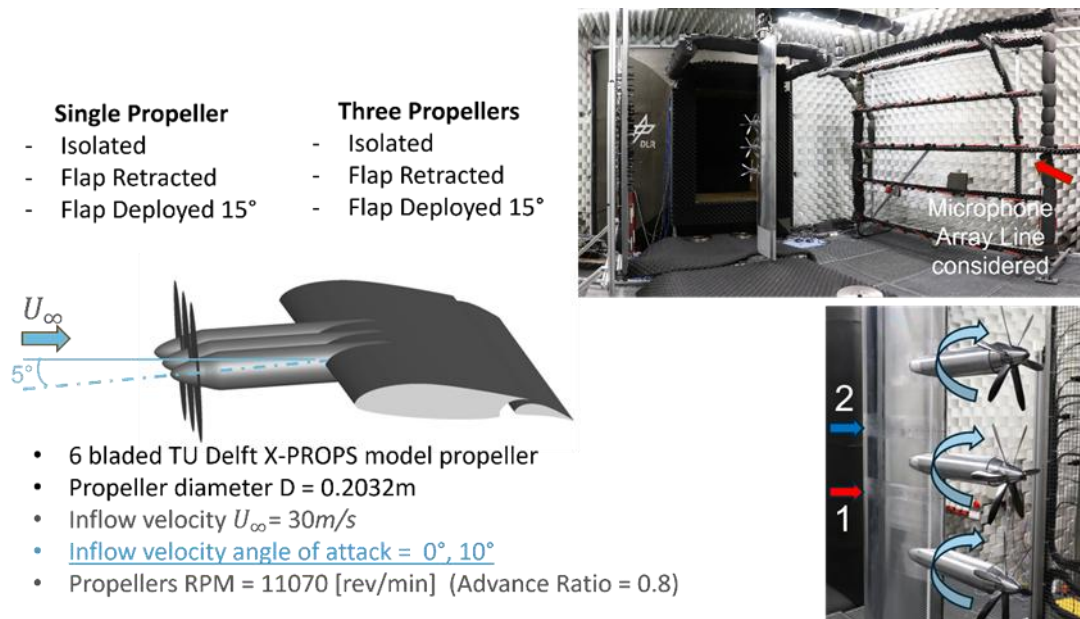


Figure 17: Experimental setup of high-lift aerodynamic/aeroacoustics setup used for validation of the hybrid RANS–CAA mid-fidelity approach [11]

angled with respect to the wing chord line. Both a single and three-propeller configuration in a co-rotating arrangement was selected. For each configuration, an isolated propeller arrangement is compared to two different installed propeller cases, one with flap retracted and one with flap deployed at an angle of 15° with respect to the flap chord line in the retracted position.

The comparison between experiments and simulations was performed first in terms of pressure coefficient, in which a close agreement can be seen for the flap retracted and flap deployed cases at 0° angle of attack, for both the single propeller and distributed propellers configuration. For the cases at angle of attack 10° , discrepancies are seen for the suction peak values of the simulations when compared to the experiments, possibly due to a laminar flow region near the wing leading edge, not considered yet in the RANS simulations performed.

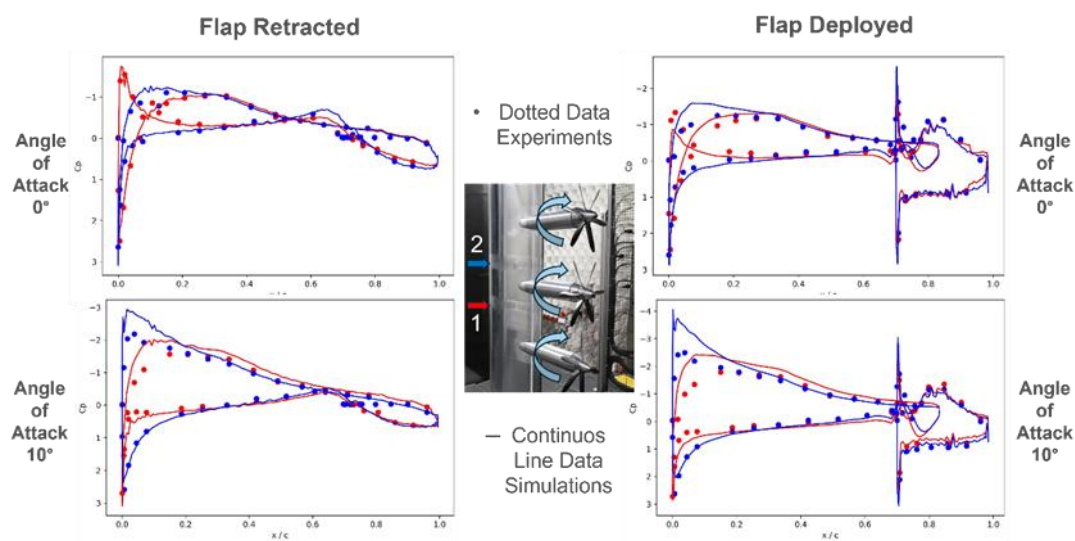


Figure 18: Aerodynamics validation of experiments and hybrid RANS-CAA mid-fidelity approach, in terms of pressure coefficient [11].

In addition to this, aeroacoustic validation was performed in terms of directivity plots, in which it can be seen that the simulation results are in agreement with experiments for all cases considered, concerning maximum SPL location and values.

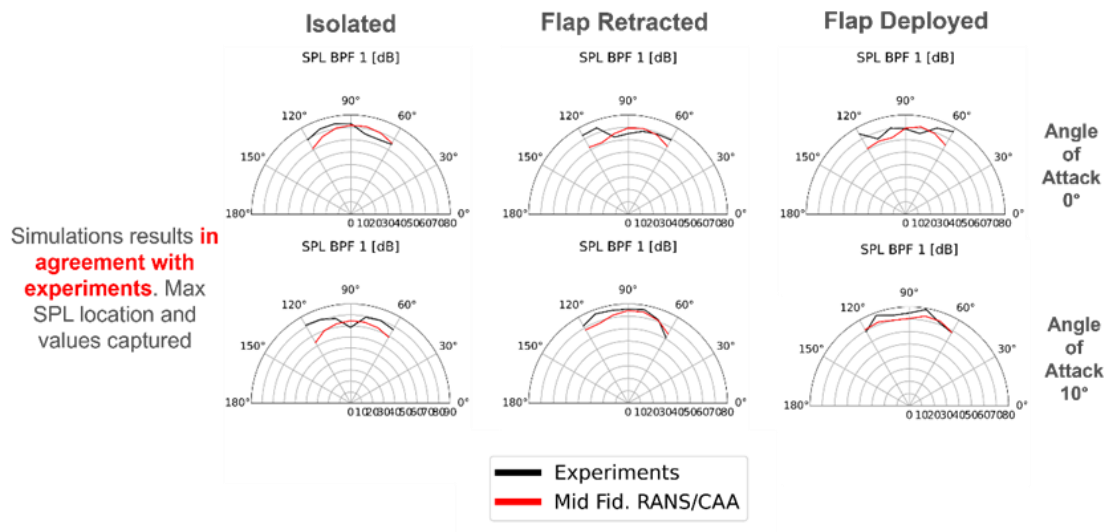


Figure 19: Aeroacoustics validation of experiments and hybrid RANS–CAA mid-fidelity approach, in terms of Sound Pressure Level (SPL) directivity, at the microphone array location highlighted in Fig.7 of [11].

Together, these representative applications demonstrate that the numerical methodologies developed within Task 3.3 constitute a consistent multi-fidelity framework capable of supporting aeroacoustics analyses across different stages of the design process. The complementary balance between computational efficiency and physical fidelity enables both large-scale parametric investigations and detailed analyses of installed propulsion systems, establishing the numerical foundation for the subsequent surrogate modelling activities.

4.4.3. Aeroacoustics Surrogate Model for design optimization

Due to the successful validation of the hybrid RANS-CAA medium fidelity approach, this methodology was chosen as the representative one within the INDIGO project to complete Task 3.3 with the generation of the necessary data for the definition of an aeroacoustics surrogate model, to be used in subsequent work packages for design optimization. For the surrogate model two aircraft configurations were selected:

- flap deployed, representative of a take-off operating point, characterized by air density 1.225 [kg/m³], speed of sound 343 [m/s] and flight Mach number of 0.25
- flap retracted, representative of a later climb operating point, characterized by air density 1.116 [kg/m³], speed of sound 336 [m/s] and flight Mach number of 0.25

The aircraft powertrain consisted of 10 five-bladed propellers, with the 2 inboard propellers having a diameter of 5 m, and the remaining outboard propellers having a diameter of 4 m. Figure 20 shows the arrangement of the propellers on the aircraft wings.

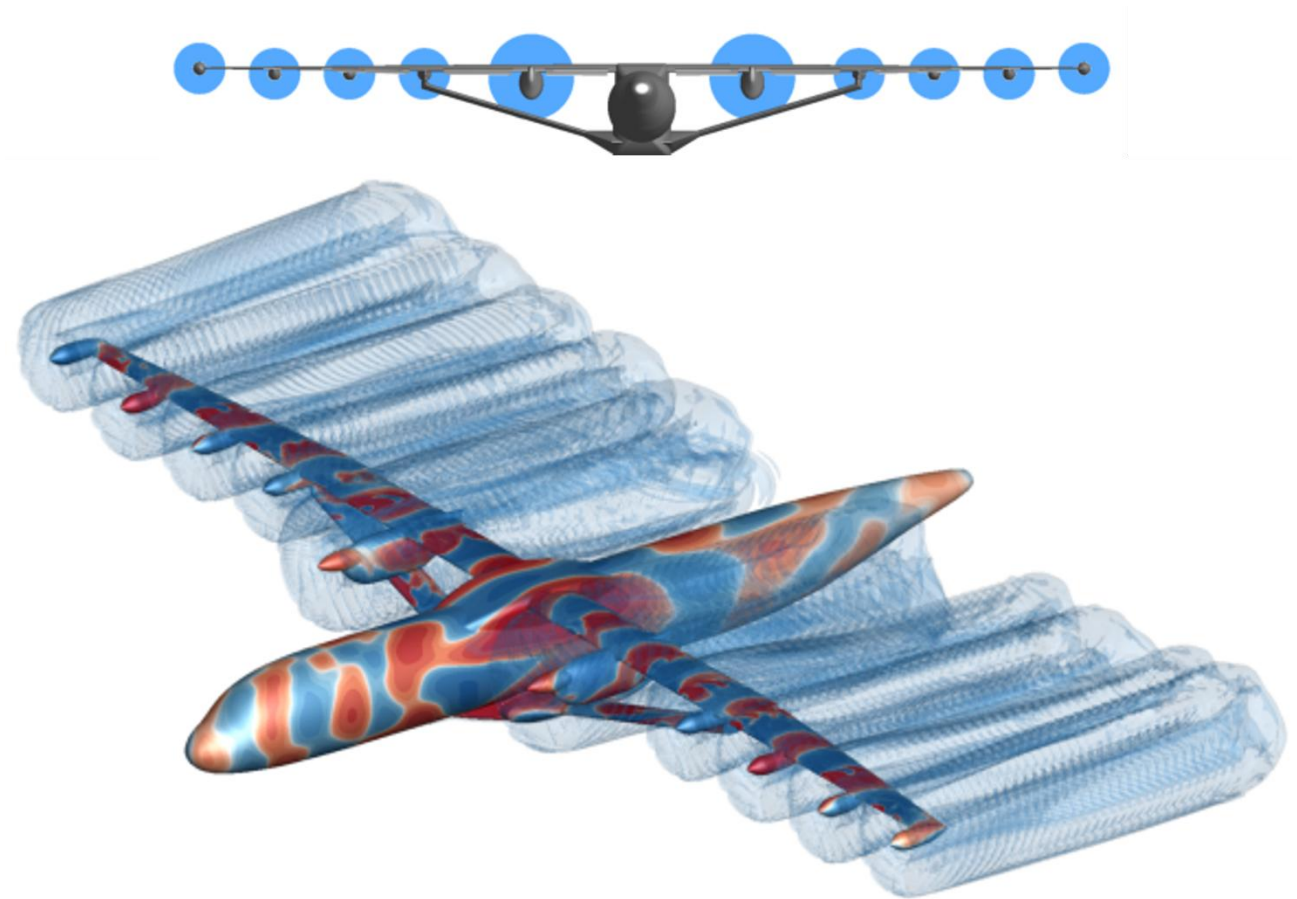


Figure 20: INDIGO aircraft propellers arrangement (top) and aeroacoustics simulation of installed propeller noise with resulting pressure fluctuations on the fuselage (bottom).

For each of the 2 configurations mentioned, 2 setups were considered, namely powertrain installed on the aircraft, and powertrain isolated (fictitious setup not installed on the aircraft). This allowed for the direct evaluation of the installation effects as a difference between installed and isolated setup, for each configuration. The parameters considered for the surrogate model are 2 different rotor tip Mach numbers, 2 different thrust coefficients, and 3 angles of attack, as described in Table 7. Considering the two aircraft configurations, and the setup and parameters considered for each configuration, a total of 144 simulations was performed.

Rotor Tip Mach nr.	Rotor Tip Mach nr.	Thrust Coeff.	Thrust Coeff.	
Inboard Prop.	Outboard Prop.	Inboard Prop.	Outboard Prop.	AoA [°]
0.8	0.3	0.22	0.05	2,7,12
0.8	0.8	0.22	0.05	2,7,12
0.3	0.3	0.4	0.05	2,7,12
0.3	0.8	0.4	0.05	2,7,12
0.3	0.8	0.05	0.23	2,7,12

0.8	0.8	0.05	0.23	2,7,12
0.8	0.8	0.22	0.23	2,7,12
0.3	0.8	0.4	0.23	2,7,12
0.3	0.3	0.05	0.42	2,7,12
0.8	0.3	0.05	0.42	2,7,12
0.8	0.3	0.22	0.42	2,7,12
0.3	0.3	0.4	0.42	2,7,12

Table 7: Aeroacoustics surrogate model parameters combinations considered.

For the generation of the necessary data for the surrogate model, the hybrid RANS-CAA methodology sketched in Figure 11 was expanded as sketched in Figure 21.

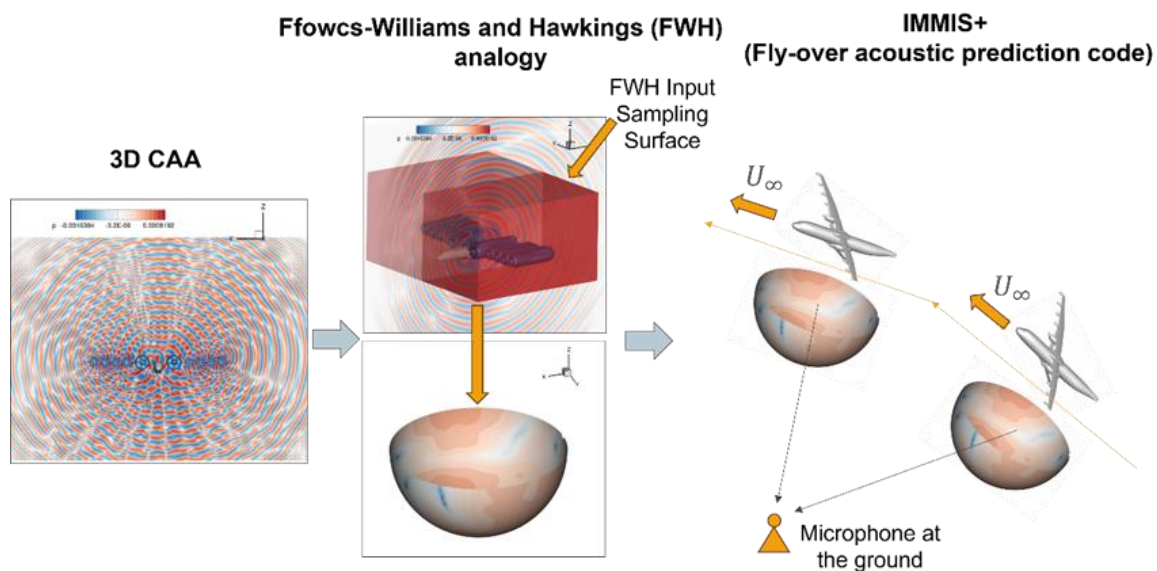


Figure 21: Aircraft aeroacoustic fly-over methodology extending the hybrid RANS-CAA methodology shown in Figure 11.

The CAA acoustic field results were sampled for multiple time steps and propagated using the Ffowcs Williams–Hawkings (FW-H) acoustic analogy to predict far-field tonal noise on a hemisphere placed below the aircraft. The results obtained from the FWH propagation step are then used in the DLR fly-over tool IMMIS+ as replacement for the closed-form solution model for isolated propellers implemented in previous versions of IMMIS+, to allow the inclusion of installation effects in the fly-over results. For the fly-over prediction, an exemplary trajectory segment representative of a climb phase was provided by UC3M with $[x,y,z]$ coordinates:

- Start point [m]: $[0, 0, 0]$
- End point [m]: $[7185.71, 0, 900]$
- Microphone location [m]: $[4500, 0, 0]$

The results obtained from the surrogate model were provided both as EPNLdB for the trajectory and microphone location considered, and also as Sound Power Level dB results. The Sound Power Level results were obtained by applying Myers Sound Intensity formula [12] to the FWH enclosing surface and integrating on the closed surface. Both EPNLdB and Sound Power Level results will be used in subsequent WP5 and WP6 tasks.

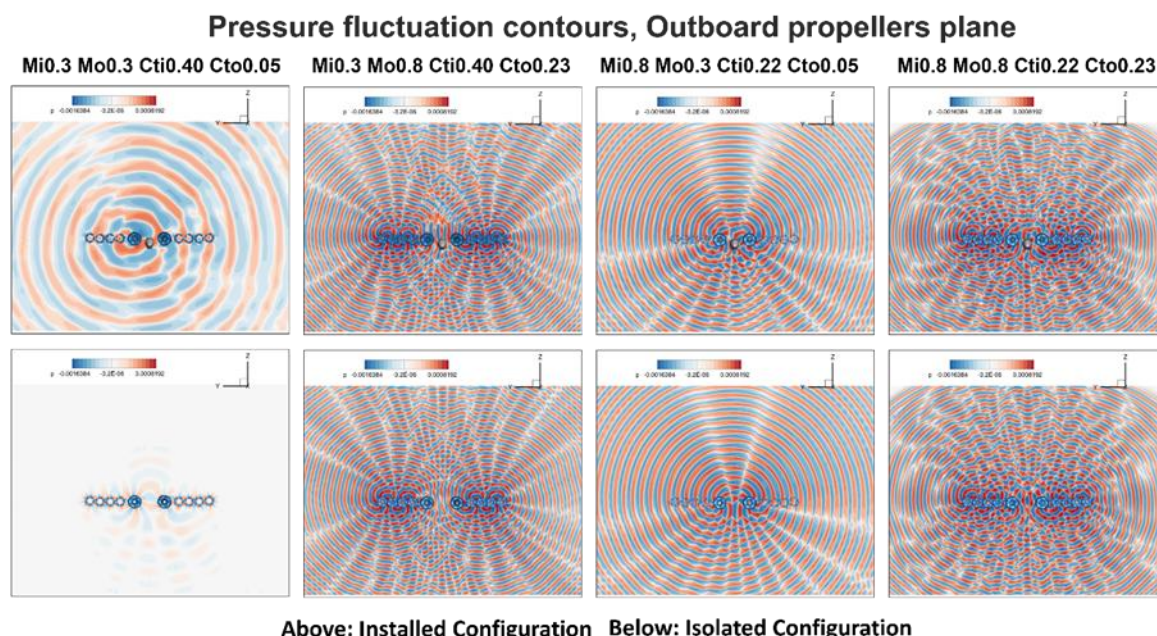


Figure 22: Aeroacoustics results in terms of pressure fluctuations contour plots, for the plane located at the outboard propellers.

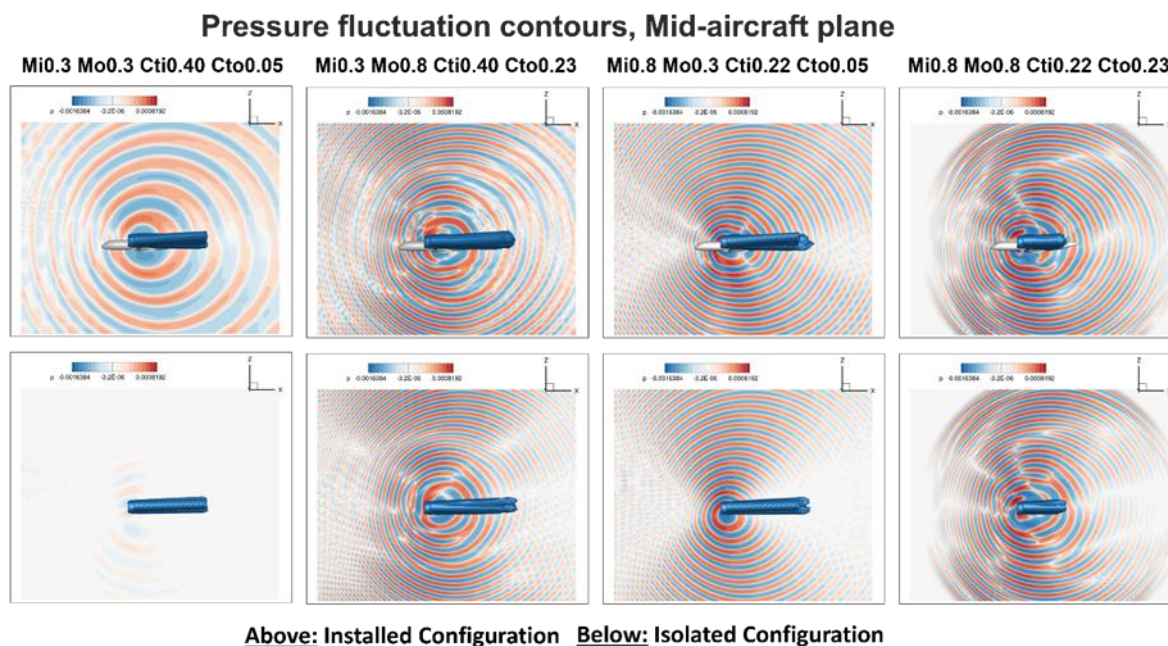


Figure 23: Aeroacoustics results in terms of pressure fluctuations contour plots, for the plane located at the middle of the aircraft fuselage.

Examples of the results obtained are included in Figure 22 and Figure 23, in which the pressure fluctuations contours for some exemplary cases of the surrogate model are shown in the propeller

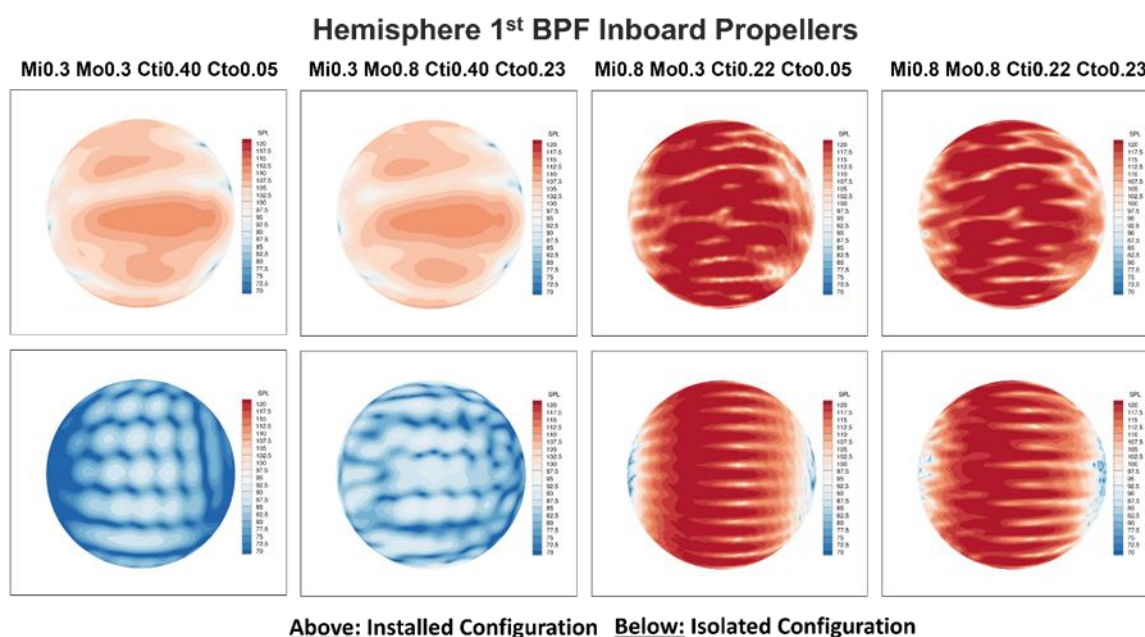


Figure 24: Aeroacoustics results in terms of contour plots of the 1st Blade Passing Frequency (BPF) of the inboard propellers on the hemisphere used in the fly-over tool IMMIS+.

plane and in the aircraft mid-plane, highlighting qualitatively the complex patterns arising from the powertrain considered, and its installation on the aircraft.

In Figure 24 and Figure 25 the results obtained for the same cases are shown in terms of Sound Pressure Levels on the hemispheres used as input for the fly-over tool IMMIS+, rescaled to a distance

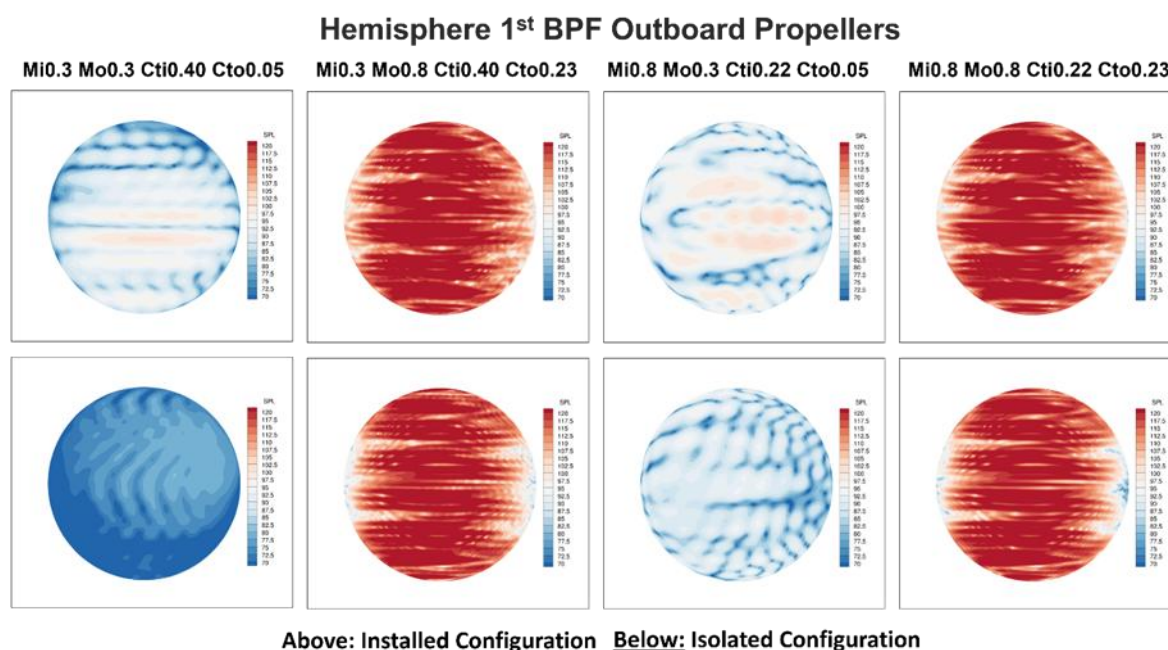


Figure 25: Aeroacoustics results in terms of contour plots of the 1st Blade Passing Frequency (BPF) of the outboard propellers on the hemisphere used in the fly-over tool IMMIS+.

of 30 m for visualization purposes. The non-trivial directivity patterns associated with each case considered can be appreciated in the results, even for the isolated powertrain, due to the design of the powertrain and the propeller locations.

5. Task3.4 - Low/mid -fidelity aeroelastic and loads models

5.1. Role inside INDIGO modelling framework

Task T3.4 addressed the development of low- and medium-fidelity methods for the aeroelastic and loads assessment of the INDIGO aircraft. Its original scope covered aeroelastic stability, manoeuvre and gust loads, structural geometric nonlinearities, and the aerodynamic and inertial effects of distributed propulsion. The analyses were to represent the aircraft as a free-flying deformable system and provide relevant information to the structural-sizing and multidisciplinary design activities.

The work resulted in an automated computational framework for aircraft combining large-aspect-ratio flexible wings and distributed propellers. The initial capability included nonlinear aeroelastic trim and coupled flight-dynamic/aeroelastic stability analysis, accounting for the in-flight deformed configuration, wing-propeller interaction and propeller dynamic effects. This methodology was presented at IFASD 2024 [13]. It was subsequently extended to cover transient flight manoeuvres, dynamic-load recovery and discrete-gust response [14] [15].

Although the aeroelastic capability was originally conceived to support the WP5 multidisciplinary design process, no dedicated aeroelastic surrogate was introduced in the final optimisation loop. The developed platform was instead applied as a detailed verification tool for selected INDIGO configurations, providing physics-based assessments of stability, transient response and structural loads.

5.2. Automated aeroelastic analysis framework

The T3.4 methods were implemented within a modular computational framework for the analysis of free-flying flexible aircraft with distributed propulsion. As shown in Figure 26, the workflow combines the aircraft structural, aerodynamic and propulsive models with the selected flight condition to generate the coupled rigid-elastic system. The same framework supports trim, stability assessment, transient manoeuvre simulation and recovery of internal structural loads.

The structural model is reduced to a modal basis, while the unsteady aerodynamic forces and propeller contributions are projected onto the same set of generalized coordinates. These elements are assembled into a state-space representation that can be used for eigenvalue analysis or time-domain simulation. The modular implementation allows the physical effects of flexibility, distributed propulsion and alternative aerodynamic formulations to be assessed consistently.

For configurations requiring the effects of large static deformation to be retained, the reference flight condition can be obtained through the nonlinear fluid-structure interaction procedure shown in Figure 27. The converged in-flight geometry and the corresponding structural properties then define the equilibrium condition about which the aeroelastic stability problem is linearised.

5.3. INDi2 configuration and analysed variants

The aeroelastic platform was applied to the **INDi2 aircraft configuration**, featuring a large-aspect-ratio strut-braced wing and a distributed hybrid-electric propulsion system. The configuration includes two main turboprop units and a set of electric propulsors distributed along the outboard wing. This arrangement combines structural flexibility, an unconventional wing-strut load path and



concentrated propulsive masses, making it a representative application for the coupled aeroelastic methods developed in T3.4.

The aircraft was represented through a full-aircraft free-free finite-element model comprising the

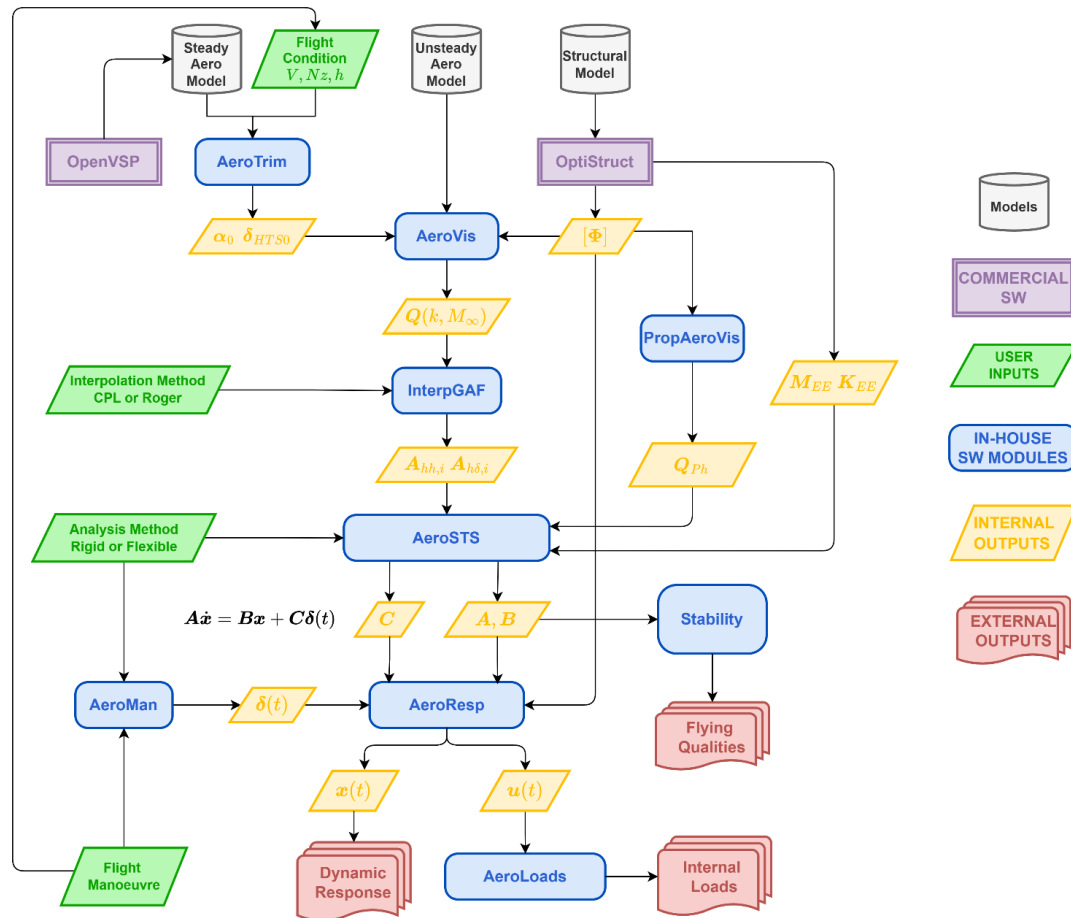


Figure 26: Computational framework for stability, manoeuvre-response and load assessment. Extract from Figure 3.1 on page 57 of [15].

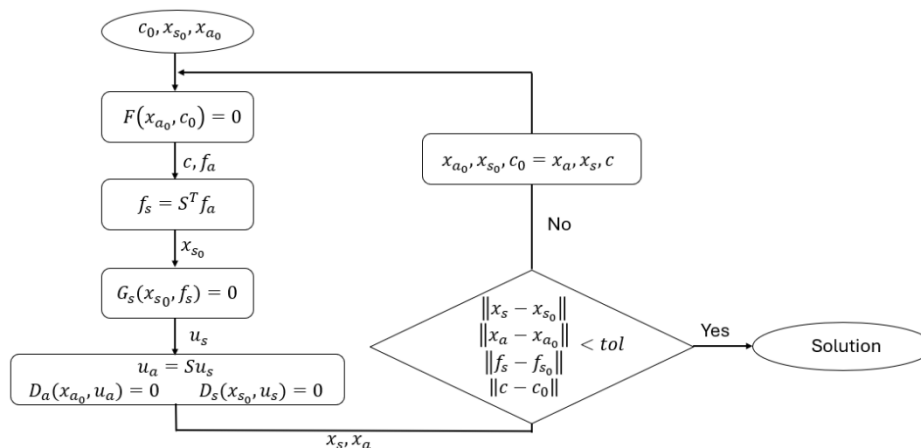


Figure 27: Nonlinear fluid-structure interaction trim workflow.

Extract from Figure 1 on page 9 of [13].

analyses. Rigid-aircraft variants were derived from the same configuration by suppressing the

fuselage, wing, struts, empennage and propulsive installations. The propulsive units were introduced through their masses, inertia properties and attachment flexibility. The structural model was reduced onto a modal basis for the coupled stability and transient-response

contribution of the elastic degrees of freedom, thereby allowing the effect of structural flexibility to be isolated without changing the underlying aircraft geometry.

The aerodynamic representation included the main wing, struts, horizontal and vertical tails, and the relevant control surfaces. Unsteady aerodynamic loads were computed using lifting-surface methods and projected onto the rigid–elastic modal basis. A complementary steady aerodynamic model was used for trim and for the calculation of stability and control derivatives. The distributed propellers were represented in this model to retain their steady influence on the surrounding aerodynamic surfaces.

Two propulsion layouts were considered, as shown in Figure 28. The baseline layout corresponds to

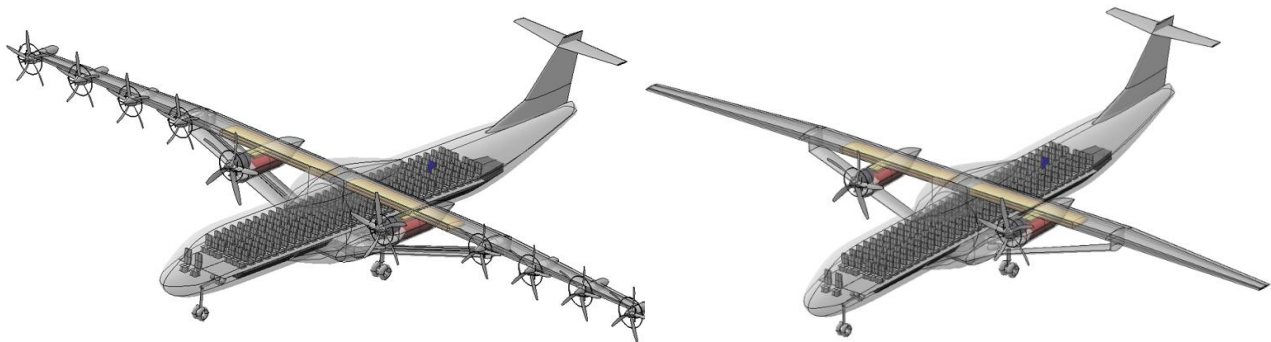


Figure 28: INDi2 propulsion layouts considered in the aeroelastic assessment: complete DHEP configuration and reference variant without the distributed electric propulsors.

the complete INDi2 DHEP configuration. A second variant retains the main turboprop units but removes the spanwise-distributed electric propulsors. The latter was not intended as an alternative project aircraft; it was introduced as a reference variant to isolate the effects of the distributed system on mass distribution, inertia, aerodynamics and aeroelastic response.

For the stability, manoeuvre-response and load analyses, the two propulsion layouts were combined with rigid and flexible structural representations:

Variant	Structural representation	Propulsion representation	Main purpose
INDi2 DHEP - flexible	Elastic modal basis	Main turboprops and distributed electric propulsors	Baseline coupled configuration
INDi2 DHEP - rigid	Rigid-body coordinates	Main turboprops and distributed electric propulsors	Isolate structural-flexibility effects
INDi2 without DEP – flexible	Elastic modal basis	Main turboprops only	Isolate distributed-propulsion effects
INDi2 without DEP - rigid	Rigid-body coordinates	Main turboprops only	Rigid reference case

For the gust analyses, the flexible variants were further evaluated with the rotation-dependent propeller terms either enabled or disabled. In the non-spinning cases, the propulsive hardware remained present in the structural model, so its mass and inertia were retained. The spinning cases additionally included the gyroscopic and propeller-aerodynamic contributions associated with

propeller rotation. This separation allowed the effects of the installed hardware to be distinguished from those arising specifically from rotating machinery.

5.4. Coupled flight-dynamic and aeroelastic formulation

The analyses were based on a unified representation of the aircraft as a free-flying flexible body. Rigid-body translations and rotations were combined with the elastic modal coordinates of the structural model, allowing flight-dynamic and aeroelastic modes to be evaluated within the same system. A mean-axis reference frame was adopted to minimise the inertial coupling between the rigid-body motion and the elastic deformation. The associated coordinates and the local representation of the propulsive units are illustrated in Figure 29.

The elastic deformation was expressed using a truncated set of free-free structural modes obtained from the finite-element model. Projection onto this modal basis reduced the size of the problem while retaining the wing, strut and propeller-support dynamics relevant to the frequency range of interest. The resulting equations were linearised about the selected equilibrium condition and can be represented schematically as:

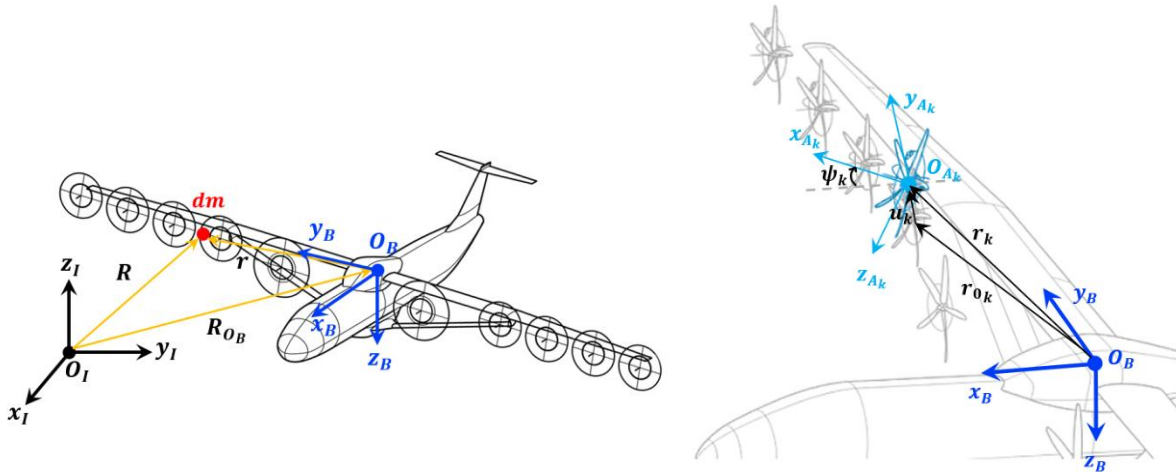


Figure 29: Reference frames and generalized coordinates used in the coupled free-flying rigid-elastic formulation: (a) inertial and aircraft mean-axis frames; and (b) local structural and rotational coordinates at a propulsive-unit attachment.

$$M\ddot{q} + (G + G_p)\dot{q} + Kq = Q_a + Q_c + Q_g + Q_p$$

where q contains the rigid-body and elastic coordinates. The right-hand side represents the aerodynamic forces generated by aircraft motion, control-surface inputs, atmospheric gusts and propeller-related effects, as applicable to each analysis.

The unsteady aerodynamic forces on the lifting surfaces were evaluated using an Enhanced Doublet-Lattice Method (EDLM). The formulation extends the classical DLM to account for the actual reference-flow incidence and for aerodynamic contributions associated with in-plane motion and loading, which are relevant for non-planar and highly flexible [16] [13]. The resulting frequency-dependent generalized aerodynamic forces were projected onto the rigid-elastic modal basis.

Rational-function approximations were then used to obtain finite-dimensional state-space models. Roger's formulation was adopted as the reference approach [17]. For the manoeuvre analyses, a Corrected Pure Lag formulation was also implemented. Building on established pure-lag approximations [18] [19], this formulation separates the quasi-steady aerodynamic terms from the

unsteady lag contribution, allowing the former to be corrected using independently calculated stability and control derivatives.

Distributed propulsion entered the formulation at several levels. The propulsive-unit masses and inertia properties were included in the structural model, while propeller rotation introduced additional gyroscopic coupling between pitch, yaw and elastic motion. Propeller aerodynamic forces were also projected onto the generalized coordinates. These contributions are particularly relevant for INDi2 because the distributed rotating masses can modify the modal basis and strengthen the interaction between low-frequency elastic modes and conventional flight-dynamic modes.

The final state-space system was used both for eigenvalue analysis and for direct time integration. The same underlying formulation therefore supported the stability, manoeuvre and gust assessments, with the excitation terms and post-processing adapted to the purpose of each analysis.

5.5. Trim definition and treatment of geometric nonlinearities

The coupled stability and transient-response analyses require an equilibrium condition about which the aircraft equations are linearised. The developed methodology supports a nonlinear aeroelastic trim procedure in which the structural deformation and aerodynamic loads are iterated until force and moment equilibrium is reached.

The converged in-flight geometry can then be used to update the structural modal basis and the unsteady aerodynamic model. Before applying the complete nonlinear procedure to INDi2, the magnitude of the expected static deformation was assessed. The resulting deflections were sufficiently small for the influence of geometric structural nonlinearities on the analysed conditions

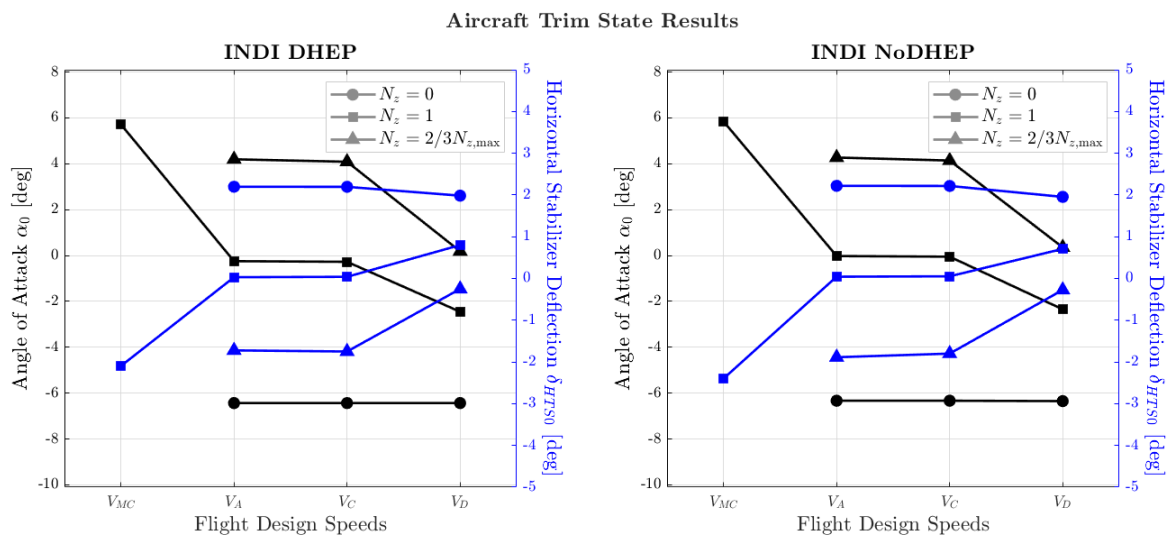


Figure 30: Trim angle of attack and horizontal-tail setting for the INDi2 DHEP and non-DHEP variants over the analysed flight conditions.

to be considered negligible. Consequently, the geometrically nonlinear structural update was not activated in the final INDi2 stability, manoeuvre and gust analyses.

The remaining elements of the methodology were retained. For each flight condition, the aircraft was trimmed by determining the angle of attack and horizontal-tail setting required to satisfy the prescribed vertical force and pitching-moment equilibrium. The steady aerodynamic model included

the wing, struts, empennage, fuselage and the influence of the distributed propellers, thereby providing a consistent reference state for the DHEP and non-DHEP variants.

The trim results obtained across the investigated airspeeds and load factors are presented in Figure 30. As expected, the required angle of attack decreased with increasing dynamic pressure and increased with the prescribed load factor. Differences between the two propulsion layouts were also observed because the distributed-propeller slipstream modifies the steady aerodynamic loading of the wing.

The coupled rigid-elastic equations were subsequently linearised about each trimmed condition under the small-deformation assumption. This preserved the flight-dynamic, aeroelastic and propulsion couplings required by the subsequent analyses while avoiding an unnecessary geometrically nonlinear treatment for the INDi2 configuration.

5.6. Dynamic stability and modal interaction

The dynamic stability of the INDi2 configuration and its variants was evaluated by solving the homogeneous state-space eigenvalue problem over the analysed range of flight speeds. The comparison included rigid and flexible representations of both propulsion layouts, using Roger and Corrected Pure Lag aerodynamic approximations. The modes were tracked between consecutive flight conditions using the Modal Assurance Criterion, allowing changes in damping, frequency and modal participation to be followed consistently.

The conventional short-period and Dutch-roll modes were first examined to assess the effect of structural flexibility and distributed propulsion on the aircraft flight dynamics. The results were interpreted using the flying-quality boundaries of MIL-F-8785C [20]. The predicted short-period mode remained within Level 1 throughout the investigated manoeuvring-speed range. Structural flexibility produced a moderate aeroelastic softening of the longitudinal response, while the DHEP slipstream modified the effective pitch stiffness and therefore the short-period frequency. The Dutch-roll mode showed a lower sensitivity to flexibility, with a small improvement in damping for the flexible variants.

The comparison between the aerodynamic approximations highlighted the importance of representing the low-frequency aerodynamic behaviour correctly. Roger's approximation reconstructed the quasi-steady and unsteady contributions simultaneously and produced more conservative predictions, characterised by lower modal damping and stronger proximity between the flight-dynamic and elastic modes. The Corrected Pure Lag formulation incorporated the independently calculated stability derivatives and captured the differences between the DHEP and reference propulsion layouts, including the steady influence of the propeller slipstream.

The main effect of the distributed propulsion system was observed in the elastic modal spectrum, as illustrated in Figure 31. The additional distributed masses and rotary inertias increased the number of low-frequency modes and reduced the frequencies of the principal wing modes. Gyroscopic coupling also introduced combined pitch-yaw motions associated with the propeller installations. The lowest elastic frequency decreased from approximately 2.5 Hz in the variant without distributed propulsors to approximately 1.2 Hz in the complete DHEP configuration.

Despite this downward shift, the INDi2 strut-braced architecture retained a frequency separation of approximately 0.5 Hz between the highest-frequency flight-dynamic mode and the lowest elastic mode. No modal coalescence between these two families was observed within the investigated range.



This indicates that the wing-strut system provides sufficient stiffness to limit the rigid-elastic interaction for the conditions considered, although the reduced separation introduced by the DHEP installation confirms the need to retain aeroelastic verification as the configuration evolves.

The assessment represents a comparative stability analysis of the selected INDi2 models and flight conditions. It should not be interpreted as a complete CS-25 flutter-compliance demonstration, which would require the evaluation of the full design envelope, mass cases, failure conditions and prescribed stability margins.

5.7. Manoeuvre response and dynamic loads

The transient response of the INDi2 configuration was assessed through a campaign of 20 representative manoeuvre cases covering maximum and checked pitch manoeuvres, roll manoeuvres and yaw manoeuvres at selected design speeds and initial load factors. The cases were defined with reference to the manoeuvre and structural-load conditions of CS-25, although the campaign was intended as a comparative aeroelastic assessment rather than a complete certification demonstration [21]. The four INDi2 variants introduced in Section 5.3 were simulated using the coupled time-domain formulation and the Corrected Pure Lag aerodynamic representation.



The maximum pitch-up manoeuvre was largely governed by rigid-body inertia, and only limited differences were observed between the rigid and flexible models. The checked pitch-up and pitch-down manoeuvres produced a clearer aeroelastic response because the oscillatory elevator input excited both the flight-dynamic and elastic degrees of freedom. Structural flexibility reduced the effective pitch stiffness and the control input required to reach the prescribed load factor. The complete DHEP variant required a lower angle of attack than the variant without the distributed

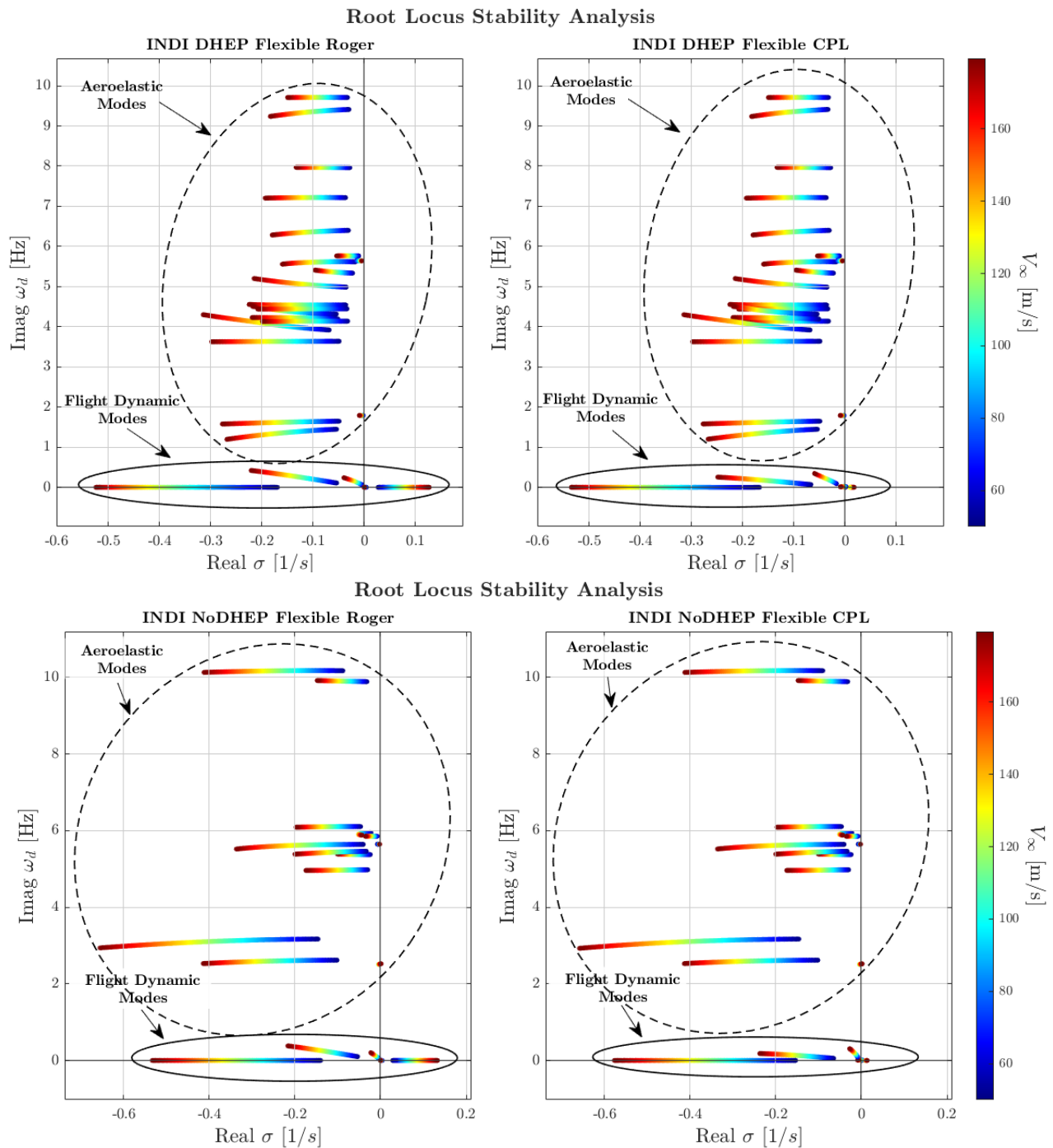


Figure 31: Root-locus comparison for the flexible INDi2 configuration with complete DHEP installation and for the variant without distributed electric propulsors, showing the effect of the aerodynamic approximation and propulsion layout on the flight-dynamic and elastic modal spectra.

propulsors, owing to the additional lift generated by the propeller slipstream. The checked pitch-up response used to illustrate these trends is presented in Figure 32.

The lateral-directional manoeuvres showed a stronger sensitivity to the selected aircraft representation. During the roll manoeuvre, abrupt aileron inputs generated short-period elastic oscillations in the angular accelerations of the flexible models. In yaw, structural flexibility reduced the transient sideslip overshoot, while the DHEP installation modified the directional stiffness and the resulting trajectory. These results show that flexibility and distributed propulsion do not act only as load corrections, but can also alter the control-surface effectiveness and transient aircraft motion.

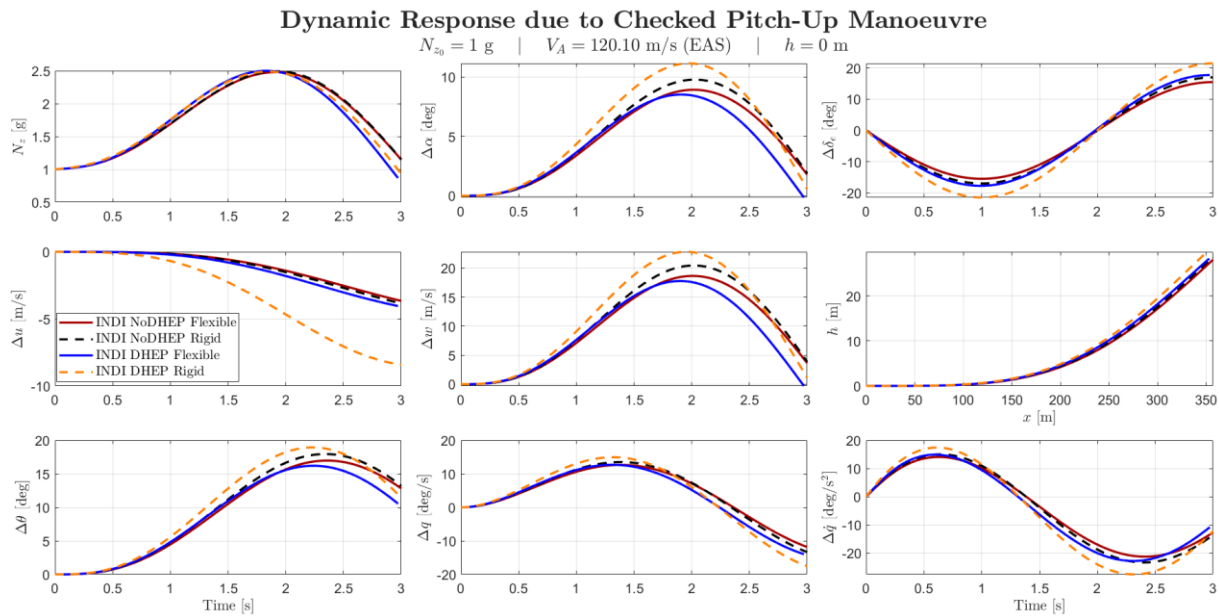


Figure 32: Coupled rigid-body and aeroelastic response of the INDi2 variants during the checked pitch-up manoeuvre.

Internal structural loads were recovered from the time histories of the elastic modal coordinates using the Mode Displacement Method. This method reconstructs the instantaneous structural deformation from the retained modal basis and derives the associated internal forces and moments [22] [23]. Since the method relies on elastic deformation, load recovery was performed only for the two flexible INDi2 variants. Three monitoring locations were considered: the wing root, the wing-strut interface and the strut root.

The resulting bending-torsion envelopes are shown in Figure 33. The complete DHEP configuration produced smaller load envelopes than the variant without the distributed propulsors at all three monitoring stations. This reduction was primarily associated with inertial relief: the masses distributed along the wing generated inertial forces that opposed part of the aerodynamic loading during the manoeuvres. The comparison also highlighted the effect of the strut-braced architecture on

the load path, with the largest bending demand occurring at the wing-strut interface rather than at the wing root.

Within the analysed campaign, the most critical bending loads were generally associated with high-speed roll manoeuvres at elevated initial load factor. The largest torsional loads were instead generated by checked pitch manoeuvres at the dive speed, where the rapid elevator input combined with the high dynamic pressure to produce significant transient wing twist. These critical cases refer specifically to the investigated INDi2 models, monitoring locations and manoeuvre set and should not be interpreted as a complete structural-sizing envelope.

5.8. Discrete-gust response assessment

The aeroelastic platform was extended to evaluate the time-domain response of the flexible INDi2 variants to discrete vertical gusts. Gust-induced aerodynamic forces were obtained from the Doublet-Lattice aerodynamic model, projected onto the retained rigid-elastic coordinates and represented in the time domain through Roger's rational-function approximation. The resulting state-space system retained rigid-body motion, elastic modal coordinates, aerodynamic lag states and the propeller-related dynamic terms introduced in Section 5.4.

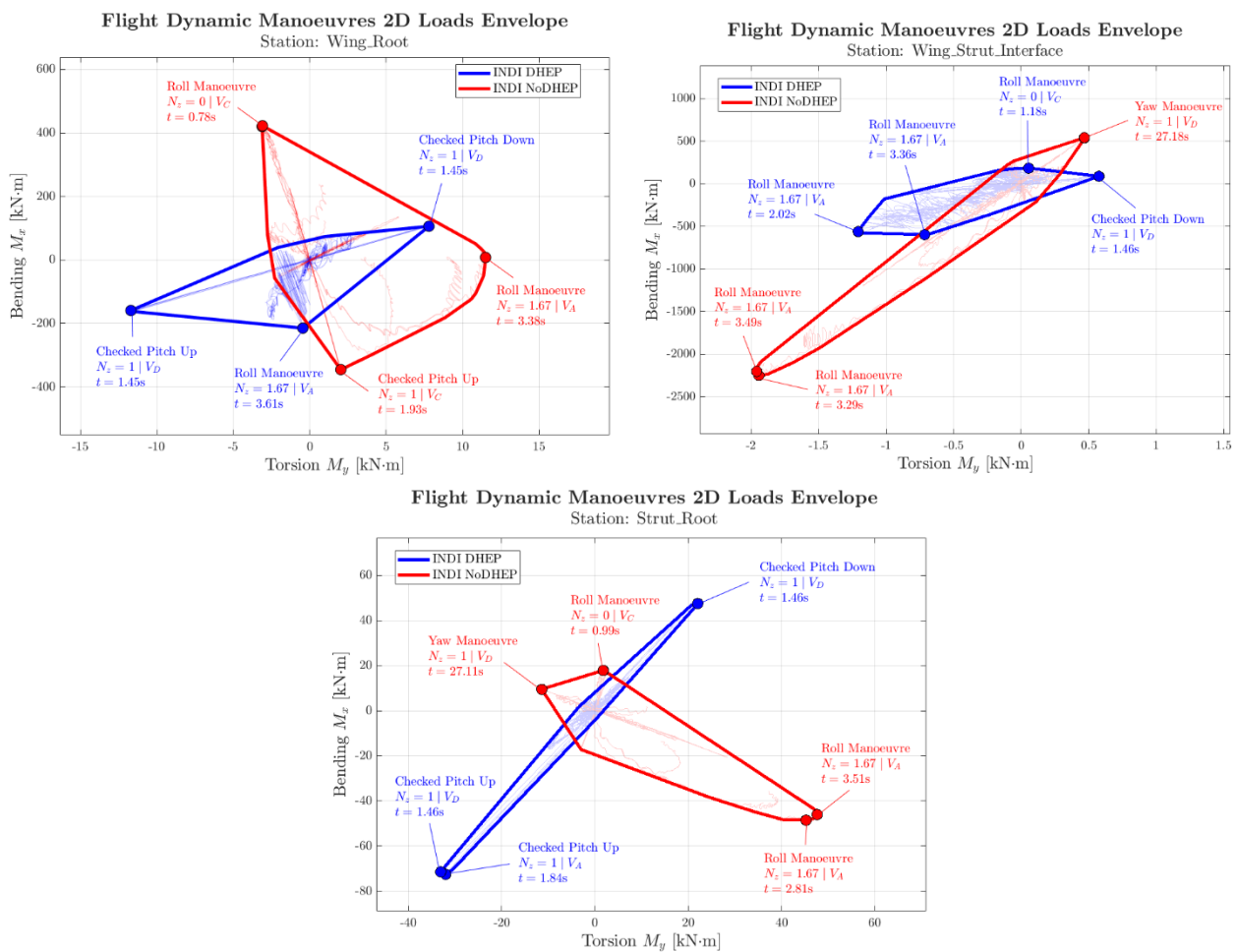


Figure 33: Dynamic bending–torsion load envelopes for the flexible INDi2 variants at the wing root, wing-strut interface and strut root.

The numerical campaign considered the aircraft at its reference cruise speed and maximum take-off mass under sea-level atmospheric conditions. Vertical one-cosine gusts were applied with gust-gradient distances of 30, 100, 200 and 350 ft, covering the range prescribed for discrete-gust assessment in CS-25.341 [21]. Because the gust amplitude and encounter duration vary with the gradient distance, the selected cases excited different parts of the coupled spectrum: short gusts contained stronger high-frequency components, whereas longer gusts primarily excited the rigid-body and low-frequency elastic response.

Before application to INDi2, the implementation was subjected to a code-to-code verification using a generic transport-aircraft model. The time-domain predictions were compared with an independent commercial code solution under matched structural, aerodynamic and gust assumptions. The comparison showed consistent trends in relative wing-tip displacement and vertical acceleration, including close prediction of the principal response timing. The remaining differences were mainly associated with the distinct time-domain and frequency-domain solution procedures.

The INDi2 campaign compared the complete DHEP model with the reference variant containing only the main turboprop units. Each structural configuration was analysed with the rotation-dependent propeller contributions enabled and disabled. The non-spinning cases retained the propulsive-unit masses, inertia and attachment stiffness, while the spinning cases additionally included gyroscopic coupling and the linearised propeller-aerodynamic terms. The comparison therefore separated the effect of the installed distributed hardware from the additional dynamics generated by propeller rotation.

The results confirm that the longest gust was not systematically the most critical, as illustrated in Figure 34. Short gusts produced sharper oscillations and increased the contribution of higher-frequency modes, while longer gusts generated larger low-frequency and rigid-body excursions. For the investigated flight condition, gust gradients around 200 ft generally produced the largest load-factor and relative wing-tip displacement responses. This represents a balance between a disturbance of sufficient duration to overcome aircraft inertia and an excitation sufficiently rapid to generate a significant elastic response.

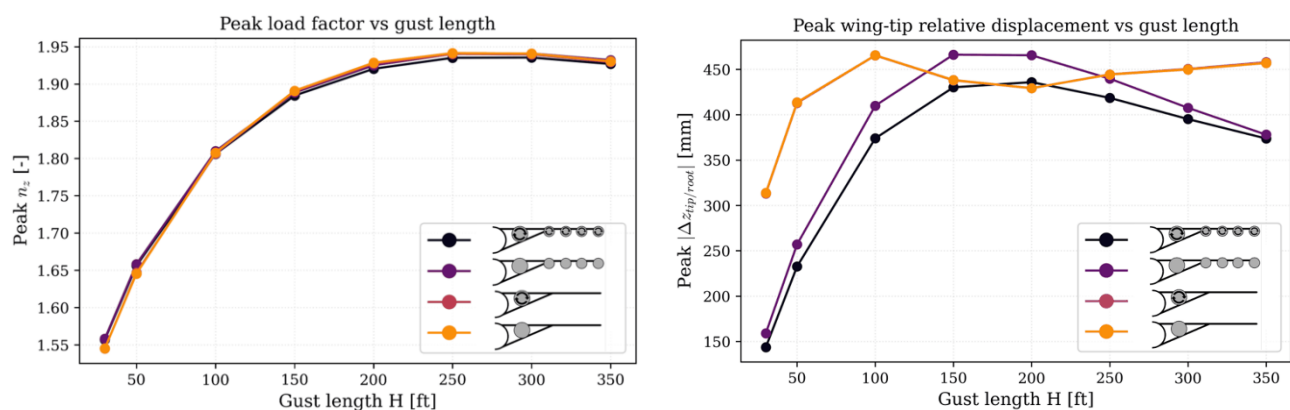


Figure 34: Peak load factor and relative wing-tip displacement as functions of gust-gradient distance for the flexible INDi2 propulsion variants, with and without rotation-dependent propeller effects.

The propulsion layout modified both the amplitude and phase of the gust response through changes in the structural modal basis and spanwise mass distribution. Activating the spinning-propeller terms introduced further pitch–yaw and rigid–elastic coupling, but its effect was dependent on the

configuration and response quantity considered. It cannot therefore be represented by a uniform stabilising or load-reducing correction. The analysis demonstrates the importance of retaining the distributed propulsion installation when identifying critical gust conditions for INDi2.

The campaign was limited to selected vertical discrete gusts at one reference flight condition. It does not constitute a complete certification gust-load envelope, which would also require additional speeds, masses, altitudes, lateral gusts, continuous turbulence and supplementary propulsion-support load cases.

5.9. Integrated influence of flexibility and distributed propulsion

The combined stability, manoeuvre and gust analyses show that structural flexibility and distributed propulsion cannot be treated as independent corrections to an otherwise conventional aircraft model. In INDi2, the propulsion installation affects the aeroelastic response through three main mechanisms: modification of the steady aerodynamic field, addition of distributed mass and rotary inertia, and introduction of rotation-dependent gyroscopic and aerodynamic terms. These mechanisms interact with the flexible wing-strut structure and influence the aircraft from the initial trim condition through to the transient structural loads.

A distinction must be made between static geometric nonlinearity and dynamic structural flexibility. The static deflections predicted for INDi2 were sufficiently small for geometric nonlinearities to be neglected in the final analyses. Nevertheless, the elastic degrees of freedom remained relevant to the dynamic response. Flexibility produced a moderate softening of the flight-dynamic modes and introduced short-duration elastic oscillations during manoeuvre and gust excitation. A linearised flexible-aircraft model was therefore appropriate for the conditions studied, whereas a rigid-aircraft representation would have omitted relevant dynamic effects.

The distributed propulsion installation had its strongest influence on the structural modal spectrum. The spanwise propulsive masses reduced the frequencies of the principal elastic modes and increased the density of modes in the low-frequency range. Propeller rotation introduced additional coupling between bending, pitch and yaw motions. Despite these changes, the analysed INDi2 configuration retained a separation between the flight-dynamic and elastic mode families, and no adverse modal interaction was identified within the investigated flight conditions.

The same installation produced both favourable and unfavourable effects depending on the response quantity. The propeller slipstream modified the trim state and control effectiveness, while the distributed masses provided inertial relief during manoeuvres and reduced the recovered bending-torsion envelopes relative to the reference configuration without the outboard propulsors. The strut also altered the structural load path, transferring a significant part of the wing loading through the wing-strut connection rather than concentrating it exclusively at the wing root.

The gust assessment did not indicate a uniform load-alleviation effect. Gust severity depended on the relationship between the gust-gradient distance and the coupled rigid-elastic modal content. The propulsion layout and the activation of the spinning-propeller terms changed both the amplitude and phase of the response, and the longest gust was not necessarily the most critical. Consequently, the influence of distributed propulsion on gust loads must be evaluated for each configuration, flight condition and response metric rather than represented through a single correction factor.

Overall, the results demonstrate that the INDi2 propulsion system should not be represented only as a prescribed thrust distribution. Its installed masses, inertia properties, aerodynamic interaction and



rotation-dependent dynamics form part of the aircraft aeroelastic system. Conversely, the benefits identified for particular manoeuvre loads should not be generalised to all stability or gust conditions. The developed platform provides the required coupled representation for this type of configuration-level verification.

5.10. Applicability, interfaces and conclusions

Task T3.4 delivered a modular aeroelastic platform for the assessment of free-flying flexible aircraft with distributed propulsion. The framework combines reduced structural models, steady and unsteady aerodynamics, control-surface inputs and propeller-related effects within a common formulation. It provides trim conditions, modal characteristics, stability trends, transient aircraft responses and internal structural loads, consistent with the original objective of supporting aeroelastic verification and structural-sizing activities.

The principal interface with the structural-design activities is provided through the recovered shear forces, bending moments and torsional moments at selected monitoring stations. These results can be used to identify critical manoeuvres and gust conditions, compare aircraft variants and define representative load cases for subsequent structural assessment. The platform can also quantify changes in modal frequency and damping caused by modifications to structural stiffness, mass distribution or propulsion installation.

Although originally conceived to support the WP5 multidisciplinary design process, no dedicated aeroelastic surrogate was introduced in the final optimisation loop. The platform was instead applied as a detailed verification capability for the INDi2 configuration and provides a reusable basis for future configuration updates. Such applications include identifying critical load cases for renewed structural sizing and repeating aeroelastic analyses following changes in geometry, mass distribution or structural stiffness.

For the conditions considered, the static deformation of INDi2 is sufficiently small for geometric nonlinearities to be neglected. The final analyses therefore used linearised flexible-aircraft models about trimmed equilibrium conditions. The scope was also limited to selected mass and flight conditions, representative manoeuvres and vertical discrete gusts. Accordingly, the results do not constitute a complete aeroelastic or loads certification campaign, which would require broader coverage of the flight envelope, mass cases, failure conditions, continuous turbulence and supplementary engine-support load cases.

Within this scope, the analyses demonstrated that the INDi2 distributed propulsion installation must be represented as an integrated part of the aeroelastic system. Its aerodynamic influence, spanwise mass distribution, rotary inertia and gyroscopic effects modify the trim, modal spectrum, transient response and structural loads. The developed platform therefore provides a reusable basis for subsequent configuration updates and for higher-fidelity verification.

6. Task3.5 - Mid/high -fidelity modelling for airframe sizing

6.1. Introduction

The growing demand for sustainable, efficient air transport is driving the development of unconventional aircraft configurations and advanced propulsion concepts. Among these, the strut-braced wing configuration can reduce structural weight and induced drag by enabling wings with



higher aspect ratios than those of conventional cantilever designs. When combined with distributed hybrid-electric propulsion, the resulting aircraft may deliver further gains in aerodynamic efficiency, emissions, and propulsion–airframe integration. However, installing multiple propulsion units along the wing introduces additional concentrated masses, local loads, and structural reinforcement requirements that can partially offset these benefits. The structural implications of this integration should therefore be assessed from the early stages of aircraft design.

Conventional airframe sizing and weight-estimation methods are largely empirical, relying on statistical correlations derived from legacy tube-and-wing configurations. While suitable for preliminary assessments, these methods may not accurately capture the distinctive load paths of a strut-braced wing, the interaction between the wing and its supporting strut, or the local structural modifications required by distributed propulsion integration. Consequently, a physics-based approach is needed to quantify the resulting weight penalties, internal load redistribution, and changes in structural mass distribution.

Within T3.5 a mid- to high-fidelity methodology for sizing and estimating the structural weight of a strut-braced wing aircraft equipped with distributed hybrid-electric propulsion has been developed. The proposed framework combines parametric finite-element modelling, structural analysis, and minimum-mass optimisation to assess the influence of the propulsion system on the wing, strut, and associated airframe structure. Particular attention is given to the additional masses and loads introduced by the distributed propulsors, the resulting modifications to the internal structural arrangement, and the use of advanced composite materials to enable lightweight designs.

6.2. Structural Design

The structural design of the INDIGO strut-braced-wing configuration was developed through a progressive modelling approach based on increasing levels of structural fidelity. The initial design phase employed a low-fidelity beam model to investigate the global structural behaviour of the wing-strut system and to support the definition of its main architectural features. In particular, this model was used to evaluate alternative hinge arrangements at the principal structural junctions and to identify the configuration providing the lowest structural mass while satisfying the prescribed strength and stability requirements.

The hinge arrangement selected from the low-fidelity study was subsequently adopted as the basis for the development of a shell-based Global Finite Element Model (GFEM). The higher-fidelity model provides a more detailed representation of the internal structural layout, the attachment regions, and the load-transfer mechanisms between the wing, strut, and fuselage. It also enables the application of more representative aerodynamic and inertial loads and supports component-level sizing optimisation. The following sections describe the low- and high-fidelity modelling approaches and their respective roles within the structural design process.

6.3. Low Fidelity Model - Beam Model

A common preliminary design approach consists of using a low-fidelity model to assess the feasibility of potential design solutions while maintaining the computational cost acceptable. With that in mind, the first study was conducted considering a beam model, in which lift forces considered are approximated using the Diederich method [24] and were applied to both wing and strut. The SBW cross sectional thicknesses were optimised to minimise mass, using stresses constraints in each



element to meet the von Mises criteria, while simultaneously avoiding buckling. To fully take advantage of the potential of this configuration, a study considering the placement of hinges between the different components was conducted: in the junction wing-strut connection (1), strut-connector-strut (2) and strut-fuselage (3), as shown in Figure 35. These hinge combinations were evaluated to minimise the mass of the SBW. The lowest mass was achieved with a hinge placed at position 3, located between the strut and fuselage and so, this hinge configuration was the one considered to develop a higher order model consisting of a shell-based GFEM.

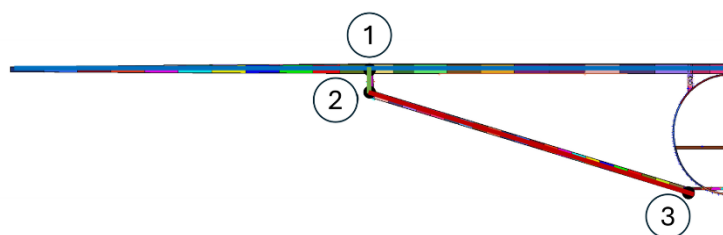


Figure 35: Beam model with hinges positions: 1 - between wing and strut connection; 2 - between strut connection and strut; 3 - between strut and fuselage.

6.4. High-Fidelity Model

Following the preliminary investigation performed using the beam model, a higher-fidelity shell-based Global Finite Element Model (GFEM) was developed to represent the structural arrangement and load-transfer mechanisms of the strut-braced-wing configuration in greater detail.

The development of the high-fidelity structural model history may be divided into two main configurations: The first configuration, denoted INDi1, established the initial structural architecture and the principal wing, strut, and fuselage attachment concepts. The second configuration, denoted INDi2, introduced the main landing gear into the structural model and incorporated geometric modifications required to align the wing-to-strut junction with the aerodynamic design. The connection region was also redesigned to ensure that its local thickness and geometry were compatible with composite manufacturing constraints.

6.4.1. INDi1: Initial Structural Configuration

The INDi1 configuration represents the first shell-based GFEM developed for the INDIGO strut-braced-wing aircraft. An isometric view of the model is presented in Figure 36. To account for the structural mass penalty associated with the introduction of the strut, the model includes the fuselage section containing both the wing-to-fuselage and strut-to-fuselage attachment regions.

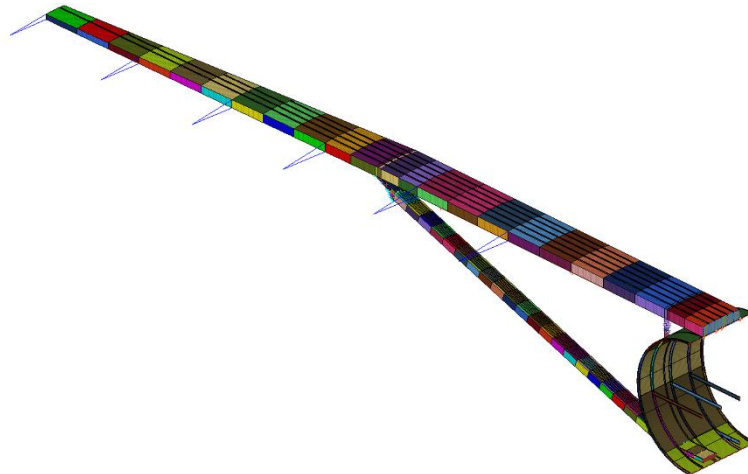


Figure 36: Isometric view of the INDi1 strut-braced-wing GFEM

Wing geometry and structural arrangement - The wing thickness-to-chord ratio is assumed to remain constant between the wing root and 50% of the semi-span, where it is equal to the prescribed root value. Beyond this position, the thickness-to-chord ratio varies linearly between the root and tip values.

The baseline aerodynamic profile is the NASA/Langley LS(1)-0413 airfoil, also designated GA(W)-2. Its thickness distribution is scaled at each spanwise position to match the prescribed local thickness-to-chord ratio while retaining the general shape of the original airfoil.

The wing primary structure comprises front and rear spars located at 20% and 70% of the local chord, respectively. Separate material properties and sizing variables are assigned to the spar webs and spar caps. The ribs are oriented approximately perpendicular to the leading edge, with an initial spacing based on the structural arrangement of the Airbus A320 wing.

The rib distribution is modified to ensure that a rib is located at each propulsion-system installation position. These ribs provide a simplified representation of the local pylon-support structure and distribute the concentrated propulsion-system loads into the surrounding wing box.

Propulsion-system integration - The distributed hybrid-electric propulsion architecture comprises one turboprop engine located at 30% of the wing semi-span and five electric motors positioned between the turboprop and the wing tip, as illustrated in Figure 36. The corresponding propulsion-system masses and loads are introduced at their respective spanwise locations and transferred to the wing structure through the supporting ribs.

Strut geometry - The strut has a rectangular planform with constant chord and uses a symmetric NACA0024 airfoil section. Consequently, its thickness-to-chord ratio remains constant along the strut span. The strut chord is treated as an independent geometric parameter and ranges from 0.906 to 1.597m.

The remaining strut dimensions, including its length, aspect ratio, and planform area, are determined automatically from the wing geometry and the prescribed attachment positions. For the configuration considered in this work, the wing-to-strut attachment is located at 50% of the wing semi-span.

Wing-to-strut attachment - The wing-to-strut junction, shown in Figure 37 and Figure 38 was designed to provide a continuous geometric and structural transition between the strut tip and the lower surface of the wing. The attachment contains a local wing-box-like structure that transfers the axial, shear, and bending loads carried by the strut into the wing primary structure.

Two additional ribs and three additional spars are introduced in the junction region to distribute the strut loads over a larger portion of the wing box. This reinforcement reduces local stress concentrations and prevents the transfer of the strut loads through an unrealistically small structural region.

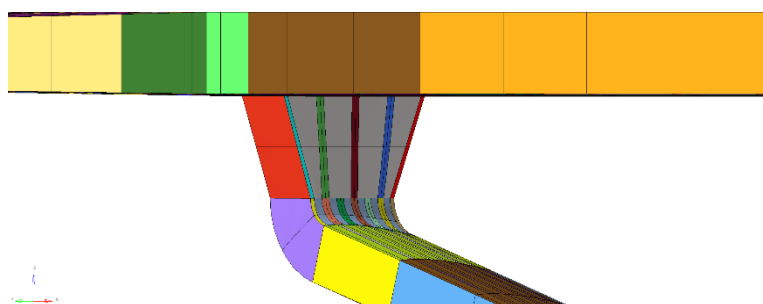


Figure 37: Wing to strut attachment

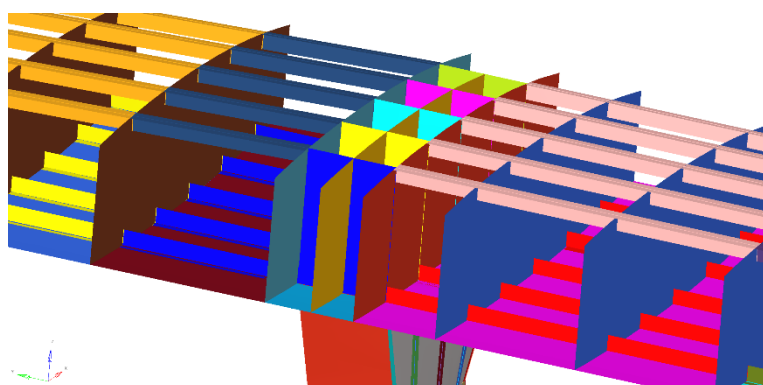


Figure 38: Detailed view of the wing-to-strut attachment region.

Wing-to-fuselage attachment - The high-wing arrangement requires a dedicated wing-to-fuselage attachment, shown in Figure 39. The central wing box is supported by the fuselage frames and reinforced through a lug-based connection representative of the structural arrangements used in high-wing transport aircraft, such as the ATR42.

The lugs connect the front and rear fuselage frames to the corresponding wing spars and transfer the wing shear forces and bending reactions into the fuselage structure.

Strut-to-fuselage attachment - The fuselage geometry is derived from that of the Airbus A320 and is locally modified to accommodate the strut attachment while minimising changes to the baseline airframe. As shown in Figure 40, the proposed arrangement uses a continuous structural member positioned within the lower fuselage, below the cargo floor.

This member has a wing-box-like architecture and is connected to the adjacent fuselage frames. The arrangement provides a continuous structural load path between the left and right struts while preserving the available cargo volume and limiting the structural mass penalty associated with the attachment.

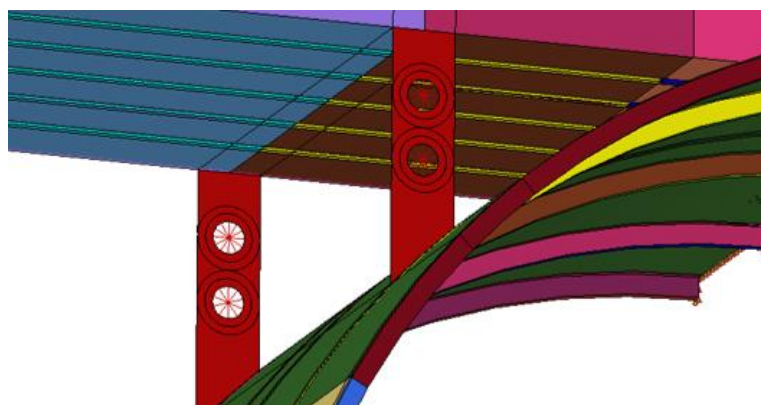


Figure 39: INDi1 wing-to-fuselage attachment.

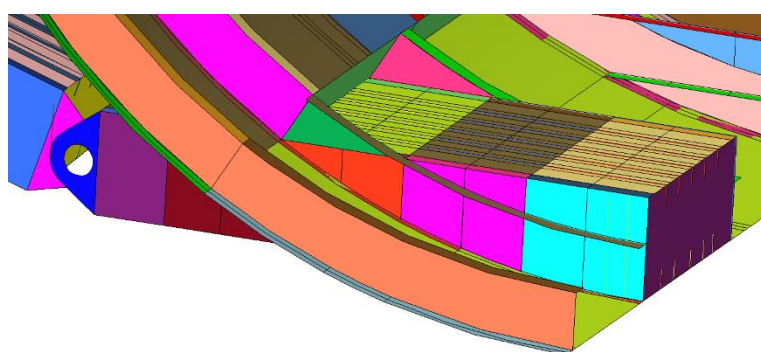


Figure 40: INDi1 strut-to-fuselage attachment.

6.4.2.INDi2: Landing-Gear Integration and Connection Redesign

The INDi2 configuration represents the second iteration of the high-fidelity structural model. It retains the general modelling and sizing methodology established for INDi1 while incorporating modifications arising from the development of the aerodynamic configuration and the overall aircraft architecture. The main changes concern the integration of the main landing gear, the redesign of the fuselage--strut attachment region, and the refinement of the wing-to-strut junction.

Landing-gear integration - The main landing-gear arrangement adopted for INDi2 was inspired by the levered landing-gear concept developed for the SUGAR High strut-braced-wing aircraft [25, 26]. In the SUGAR High configuration, the low aircraft ground stance is enabled by a levered landing gear attached to a dedicated sponson located near the strut-to-fuselage junction. This sponson acts as a short wing-box-like structure that transfers both landing-gear and strut loads into the fuselage.

A similar structural philosophy was adopted for INDi2, although the arrangement was adapted to the aerodynamic, structural, and packaging requirements of the INDIGO configuration. The lower



fuselage and strut-attachment region were redesigned to accommodate the landing-gear support and retraction mechanism while maintaining continuous load paths between the landing gear, strut, fuselage frames, and surrounding primary structure.

Particular attention was given to the geometry surrounding the landing gear. The supporting structure and external fairing were designed to enclose the main landing-gear system and its principal load-carrying components while limiting aerodynamic interference. The landing-gear kinematics were evaluated to ensure sufficient ground clearance, compatibility with the surrounding structure, and compact stowage within the available internal volume.

The corresponding landing-gear loads were introduced into the GFEM at the attachment locations. This allowed the redesigned fuselage-strut connection and the surrounding supporting structure to be assessed under the relevant ground load cases, in addition to the aerodynamic and inertial load cases associated with flight.

Wing-to-strut connection redesign - A further modification introduced in INDi2 concerned the geometry of the wing-to-strut junction. The connection was adjusted to conform to the outer aerodynamic surface defined by the aerodynamic design team. At the same time, the internal and external geometry of the junction was revised to avoid excessively thin sections and abrupt thickness variations that would be difficult to manufacture using composite materials.

The resulting design therefore represents a compromise between aerodynamic continuity, structural load transfer, geometric integration, and composite manufacturability. The revised junction preserves a continuous load path between the strut and the wing primary structure while satisfying the geometric constraints imposed by the updated aerodynamic configuration.

An isometric view of the resulting INDi2 GFEM is presented in Figure 41, while the integrated main landing gear is shown in Figure 42 and Figure 43.

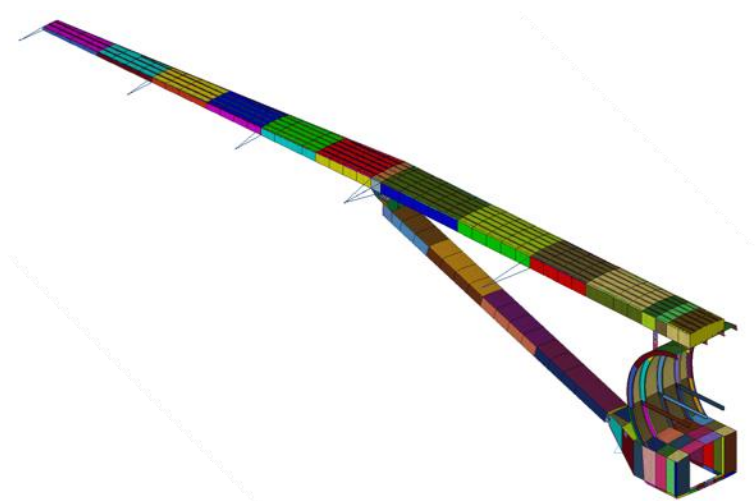


Figure 41: Isometric view of the INDi2 strut-braced-wing GFEM.

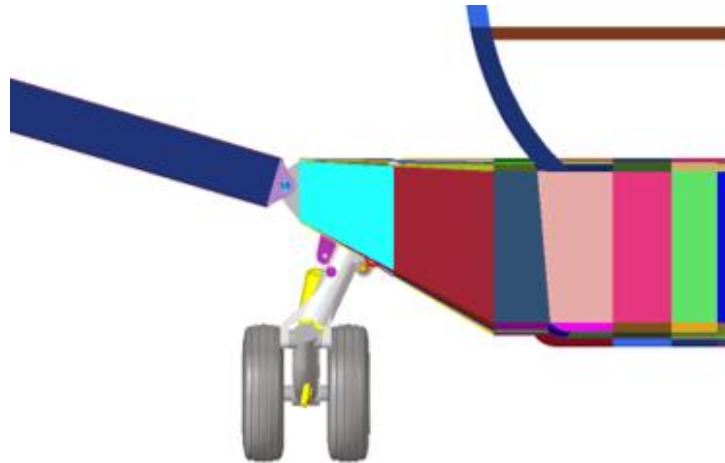


Figure 42: Front view of the integrated main landing gear.

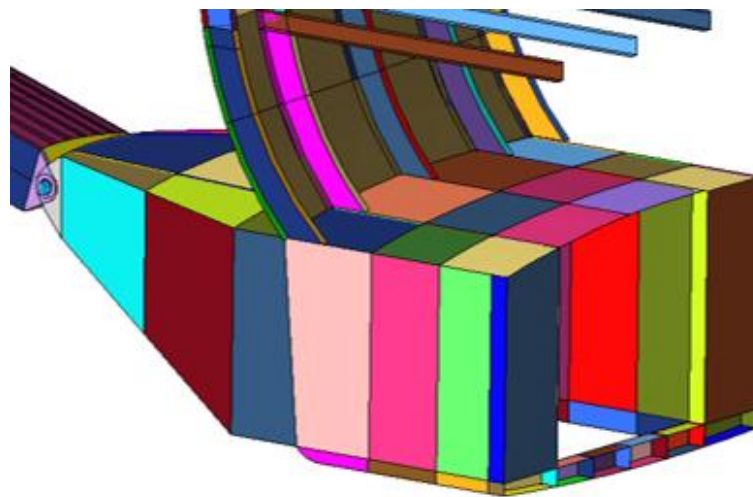


Figure 43: Isometric view of the integrated main landing gear.

6.5. Structural Sizing Methodology

The structural sizing methodology is based on a parametric GFEM generated automatically for each aerodynamic wing configuration. The framework combines structural modelling, material definition, aerodynamic-load transfer, boundary-condition assignment, and sizing optimisation within a consistent OptiStruct-based workflow. The methodology is applied to both the conventional cantilever-wing configuration and the strut-braced-wing (SBW) configuration, allowing their structural mass and material distributions to be compared under equivalent loading and design requirements.

6.5.1. Global Finite Element Models

A Tool Command Language (TCL) script is used to automatically generate the structural models in HyperMesh according to the OptiStruct finite element format. For each aerodynamic configuration, the script constructs the corresponding wing-box geometry, finite element mesh, material regions, structural properties, concentrated masses, and model connections.



The cantilever-wing GFEM contains the primary wing-box structure and the attachment region required to represent the high-wing connection to the fuselage. The SBW GFEM includes the wing and strut wing boxes, together with the fuselage section containing the wing-to-fuselage and strut-to-fuselage attachment regions. The inclusion of the fuselage section allows the additional structural mass and load-transfer mechanisms associated with the strut installation to be explicitly represented.

The skins, spar webs, spar caps, ribs, stringers, fuselage frames, and attachment structures are represented using CQUAD4 and CTRIA3 shell elements. The cantilever-wing model consists of 64,775 shell elements and 101,032 degrees of freedom, whereas the SBW model comprises 106,668 shell elements and 233,066 degrees of freedom.

Stringers are also represented using shell elements, allowing composite material definitions and independent sizing variables to be assigned directly in HyperMesh. The stringers have a T-shaped cross-section and are distributed with a nominal spacing of 250 mm.

The turboprop engines and electric motors are represented as concentrated masses located at their corresponding spanwise installation positions. These masses are connected to the supporting ribs using RBE3 interpolation elements, which distribute the associated inertial and propulsion loads over the surrounding structural nodes without introducing excessive local stiffness. An isometric view of the SBW structural assembly is presented in Figure 41.

6.5.2. Material Definition

Composite materials are assigned to the principal load-carrying components of the wing and strut, including the skins, stringers, spar webs, and spar caps. The ribs, attachment lugs, fuselage frames, and other local connection components remain metallic.

The metallic components are modelled using Aluminium 7010-T651. The composite components are based on a unidirectional carbon-fibre lamina. The elastic and density properties adopted for both materials are summarised in Table 8.

Table 8: Material properties used in the Global Finite Element Models

Property	Aluminium 7010-T651	Carbon-fibre lamina
E or E ₁₁ [GPa]	71.7	120
E ₂₂ [GPa]	-	8.5
G or G ₁₂ [GPa]	26.95	4.0
Poisson's ratio, ν , ν_{12}	0.33	0.30
Density, ρ [kg/m ³]	2810	1550

The balanced and symmetric composite laminates are represented using equivalent homogenised anisotropic properties. Each laminate is therefore defined by its total thickness and by a prescribed distribution of the $[0^\circ/\pm 45^\circ/90^\circ]$ ply-orientation families.

The relative percentage of each orientation family is predefined for every structural component and remains constant during the sizing optimisation. The total laminate thickness is used as the sizing variable. A minimum ply-percentage rule of 10% is imposed for each orientation family to ensure that the laminate retains sufficient capability to withstand secondary and unforeseen load paths.

6.5.3. Load Cases

The structural loads comprise aerodynamic forces, propulsion-system forces, and inertial loads associated with the structural and non-structural masses. The propulsion-system contribution includes the thrust generated by the turboprop engines and electric motors, together with the inertial forces resulting from their concentrated masses.

Symmetric Manoeuvres - Two symmetric manoeuvre load cases are considered in the structural sizing process. The positive manoeuvre corresponds to a pull-up condition with a limit load factor of $n_z = +2.5$, whereas the negative manoeuvre corresponds to a push-down condition with a limit load factor of $n_z = -1.0$.

An ultimate load factor of 1.5 is applied to the limit loads. The resulting ultimate manoeuvre conditions are therefore $n_z = +3.75$ for the pull-up condition and $n_z = -1.5$ for the push-down condition.

Aerodynamic-Load Calculation and Transfer - Aerodynamic loads are calculated for each prescribed manoeuvre condition using Open VSP panel method [27]. The aerodynamic and structural models use different discretisations, and a dedicated interpolation procedure is therefore required to transfer the aerodynamic loads to the GFEM.

A k-nearest-neighbours (k-NN) algorithm is used to map the aerodynamic forces from the aerodynamic discretisation to the structural nodes in OptiStruct. For each aerodynamic load-application point, the algorithm identifies the closest structural nodes and distributes the corresponding force among them according to their relative proximity.

This approach provides a consistent transfer of the aerodynamic loading over the wing and strut structures while preserving the global resultant forces and moments. The resulting nodal loads are subsequently combined with the propulsion and inertial loads before the structural analysis and sizing optimisation are performed.

Main Landing Gear Loads - The landing-load conditions prescribed by CS-25 [21] and FAR Part 25 [28] comprise several separate bookcases, including the maximum vertical reaction, lateral-drift landing, one-gear landing and asymmetric side-load conditions. In the present preliminary structural assessment, only one main-landing-gear load case is retained. A conservative component-wise envelope is therefore constructed by combining the maximum vertical, longitudinal and lateral force components obtained from the individual landing load cases.

The load components are defined relative to the maximum vertical ground reaction acting on one main landing gear, denoted by $F_{z,\max}$. According to the landing load cases, the maximum vertical component occurs during the maximum vertical reaction and one-gear landing conditions and is given by

$$F_z = F_{z,\max}.$$

The largest longitudinal force appearing in the landing load cases is associated with the lateral-drift level-landing condition. In this condition, the vertical force is equal to 75% of the maximum vertical reaction and the longitudinal force is equal to 40% of the corresponding vertical force. Consequently,

$$F_x = 0.40(0.75F_{z,\max}) = 0.30F_{z,\max}.$$

The largest lateral-force magnitude is obtained from the asymmetric side-load condition. Each main landing gear carries a vertical reaction equal to 50% of the maximum vertical reaction, while the



inward lateral force on one side is equal to 80% of the corresponding vertical reaction. For the right-hand main landing gear,

$$F_y = -0.80(0.50F_{z,\max}) = -0.40F_{z,\max},$$

where the negative sign indicates a force acting inboard, towards the aircraft centreline.

The resulting conservative envelope is therefore

$$\mathbf{F}_{\text{MLG,env}} = \begin{bmatrix} F_x \\ F_y \\ F_z \end{bmatrix} = \begin{bmatrix} 0.30 \\ -0.40 \\ 1.00 \end{bmatrix} F_{z,\max}.$$

The structural coordinate system is defined such that the positive x -direction acts from the wing leading edge towards the trailing edge, the positive y -direction acts outboard towards the right-wing tip, and the positive z -direction acts upwards from the landing gear towards the wing. Accordingly, the longitudinal drag force is positive in x , the critical inward lateral load on the right main landing gear is negative in y , and the ground reaction is positive in z .

For the preliminary analysis, the aircraft mass is taken as

$$m = 79\,000 \text{ kg}.$$

In the absence of a detailed dynamic landing simulation, the maximum aircraft-level vertical ground reaction is approximated as

$$F_{z,\max,\text{aircraft}} = 1.2mg.$$

Assuming equal sharing of the vertical reaction between the left and right main landing gears, the reference force acting on one main landing gear becomes

$$F_{z,\max} = \frac{1.2mg}{2} = \frac{1.2(79\,000)(9.81)}{2} = 464.99 \text{ kN}.$$

The corresponding force components are

$$F_x = 0.30F_{z,\max} = 139.50 \text{ kN},$$

$$F_y = -0.40F_{z,\max} = -186.00 \text{ kN},$$

and

$$F_z = F_{z,\max} = 464.99 \text{ kN}.$$

The single main-landing-gear load case applied to the right-hand side of the finite-element model is therefore

$$\mathbf{F}_{\text{MLG,env}} = \begin{bmatrix} +139.5 \\ -186.0 \\ +465.0 \end{bmatrix} \text{ kN}$$

This load vector does not represent a single physical landing event prescribed by CS-25 or FAR Part 25. Instead, it is a conservative component-wise envelope obtained by combining the maximum longitudinal, lateral and vertical forces from different certification landing load cases. Its use is therefore appropriate for preliminary structural sizing and comparative assessment, but it should not be interpreted as a substitute for the individual certification load conditions or for a dynamic landing-load analysis.



6.5.4. Boundary Conditions

The boundary conditions are defined using the global reference frame shown in Figure 41 and Figure 44. Symmetry boundary conditions are applied at the common symmetry plane of the fuselage, wing, and strut.

The upper attachment lugs connecting the wing to the fuselage, shown in Figure 44, permit rotation about the x -axis. This provides a hinge-like representation of the wing-to-fuselage attachment while transmitting the required loads to the fuselage structure.

The lugs connecting the strut to the lower fuselage wing box, shown in Figure 44, permit rotation about multiple axes and therefore approximate the behaviour of a spherical joint. This representation allows the strut to transfer predominantly axial and shear loads while limiting the transmission of artificial bending moments through the attachment.

Two additional RBE3 interpolation elements are defined in the yz -planes of the front and rear fuselage frames. Each RBE3 element connects the nodes belonging to the corresponding frame plane and distributes the applied boundary constraints over the complete frame section.

At the front fuselage frame, the displacements of the connected nodes are constrained in the x - and z -directions. At the rear fuselage frame, only the displacement in the z -direction is constrained. This arrangement stabilises the model and prevents rigid-body motion while minimising the introduction of non-physical reaction loads.

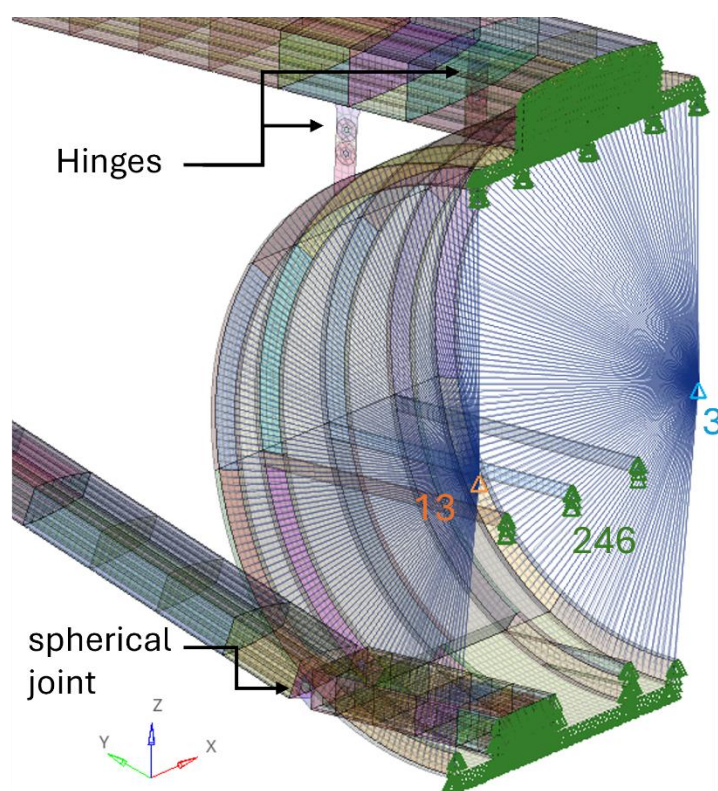


Figure 44: Isometric view of the SBW GFEM assembly and detail of the applied boundary conditions.

6.6. Structural Sizing Optimisation

Once the finite element model, materials, loads, and boundary conditions have been defined, a sizing optimisation is performed using OptiStruct. The objective is to minimise the structural mass of the wing and, for the SBW configuration, the strut and associated attachment structure.

The shell thicknesses of the metallic components and the total laminate thicknesses of the composite components are introduced as design variables. Independent design regions are defined along the wing and strut to allow the material distribution to adapt to the local load paths.

The optimisation is subject to strength, stiffness, and stability constraints evaluated for the prescribed manoeuvre load cases. This procedure determines the minimum-mass structural configuration that satisfies the required design criteria and enables a consistent comparison between the cantilever-wing and strut-braced-wing concepts.

6.6.1. Objective Function

The structural sizing problem is formulated as a mass-minimisation problem. The objective function is the total structural mass of the finite element model:

$$\min_{\mathbf{x}} M(\mathbf{x}),$$

where $\mathbf{x}$ denotes the vector of structural sizing variables and M is the total mass of the optimised structural components. The optimisation therefore seeks the lightest structural configuration that satisfies the prescribed strength, stability, manufacturing, and design constraints for all considered load cases.

6.6.2. Design Variables

The design variables are defined by dividing each structural component of the lifting surfaces into multiple spanwise sizing sections. An independent thickness design variable is assigned to each section of each component. This segmentation permits the material distribution to vary along the span in structural components such as the skins, stringers, spars, and ribs, while maintaining a computationally manageable optimisation problem.

For metallic components, the design variables directly control the thickness of the corresponding shell properties. For composite components, the total laminate thickness is varied discretely using a manufacturing ply thickness of 0.17 mm. The number of plies in each laminate is therefore obtained from

$$N_{\text{plies}} = \frac{t_{\text{lam}}}{t_{\text{ply}}}, \quad t_{\text{ply}} = 0.17 \text{ mm},$$

where t_{lam} is the total laminate thickness. The prescribed relative percentages of the 0° , $\pm 45^\circ$, and 90° orientations are maintained as the laminate thickness changes.

6.6.3. Constraints

Constraints are summarized in the following sections they can be grouped into these categories: (1) Metal Strength Constraints, (2) Composite Strength Constraints, (3) Stability Constraints and (4) Manufacturing Constraints.



Metallic Strength Constraints - For aluminium components, a non-yielding condition is imposed through a maximum von Mises stress constraint. The allowable stress is defined as 70% of the material yield stress to account for the reduction associated with damage-tolerance considerations:

$$\sigma_{VM} \leq 0.70 \sigma_{yield} = 289.58 \text{ MPa.}$$

This condition is evaluated on both the upper and lower surfaces of every shell element belonging to a metallic structural component.

Composite Strength Constraints - Composite laminate strength is assessed using a maximum-strain criterion. To account for barely visible impact damage and no-growth damage-tolerance requirements, the allowable tensile and compressive strains are limited to:

$$-3500 \mu\epsilon \leq \epsilon \leq 3500 \mu\epsilon.$$

Since the individual plies are not explicitly represented when equivalent homogenised laminate properties are employed, the strain constraint cannot be imposed directly in the local fibre directions of each ply. Instead, the maximum and minimum in-plane principal strains of each shell element are constrained using the prescribed strain allowable.

Stability Constraints - Structural stability constraints are included to prevent buckling before the ultimate load condition is reached. Linear eigenvalue buckling analyses are performed for both limit-load manoeuvre cases. A lower bound of 1.5 is imposed on the retained buckling eigenvalue has displayed in the equation below:

$$\lambda_i \geq 1.5, \quad i = 1, \dots, N_{modes},$$

where λ_i is the eigenvalue associated with the i^{th} buckling mode. Since the reference states correspond to limit loads, an eigenvalue of 1.5 indicates that the predicted instability load is at least equal to the corresponding ultimate load.

Manufacturing Constraints - For composite structures, the laminate thickness is restricted to discrete multiples of the manufacturing ply thickness. The balanced and symmetric character of the laminate is maintained, together with the prescribed ply-orientation percentages and a minimum contribution of 10 % from each required ply-orientation family. The prescribed orientation percentages for each structural component are presented in Table 9. In addition, the maximum allowable laminate thickness is limited to 75 mm. This upper bound is introduced as a practical manufacturability and modelling constraint, preventing the optimisation procedure from producing unrealistically thick monolithic laminates. At large thicknesses, the reduced heat dissipation during curing can generate significant through-thickness temperature and degree-of-cure gradients, excessive exothermic temperatures, residual stresses, distortion, porosity, and an increased susceptibility to interlaminar damage. The adopted value is also consistent with the approximate thickness range achievable for thick aerospace-grade carbon-fibre/epoxy laminates using specifically controlled autoclave cure cycles.

Table 9: Ply share constraints

Component	Orientation		
	0°	90°	±45°
Skin [%]	33.3	16.6	25
Spar Web [%]	22.25	22.25	27.75
Spar Cap [%]	57	12.5	15.25
Stringer [%]	50	16.6	16.6

6.7. Automatic Workflow

Wing geometry is parameterized in terms of aspect ratio, taper ratio, root and tip thickness-to-chord ratios, and projected wing area. The corresponding design ranges considered are $15 \leq AR \leq 25$, $0.22 \leq \lambda \leq 0.42$, $0.12 \leq t/c_{\text{root}} \leq 0.18$, $0.09 \leq t/c_{\text{tip}} \leq 0.14$, and $97.92 \leq S_W \leq 148.88 \text{ m}^2$.

Figure 45 summarizes the automated workflow used to generate and optimize the structural finite element model. The process is implemented through a Tcl/Tk based application embedded within the Altair environment and begins with the definition of the wing geometry, material properties, design load cases, boundary conditions, and structural internal topology.

Based on these inputs, the external geometry and wing box topology are generated. The structural components are then discretized, material properties are assigned, and composite layups are defined. All components are subsequently assembled into a common finite element model, after which the prescribed loads and boundary conditions are applied.

The assembled model undergoes a series of checks to verify mesh quality, connectivity, material properties, and thickness distributions. Once these checks are completed, the sizing optimization problem is defined in terms of its design variables, objective function, structural and manufacturing constraints, variable bounds, and convergence criteria. The objective is to minimize the structural mass while satisfying the required strength, stiffness, stability, and manufacturing constraints.

Finally, the optimized model is post processed to determine the structural mass, thickness distribution, and active constraints. The workflow can be applied to the main structural components of the strut braced wing aircraft, including the wing, strut, fuselage section, and attachment regions. This automated process enables the structural model to be regenerated and optimized efficiently for different aircraft configurations and input parameters.

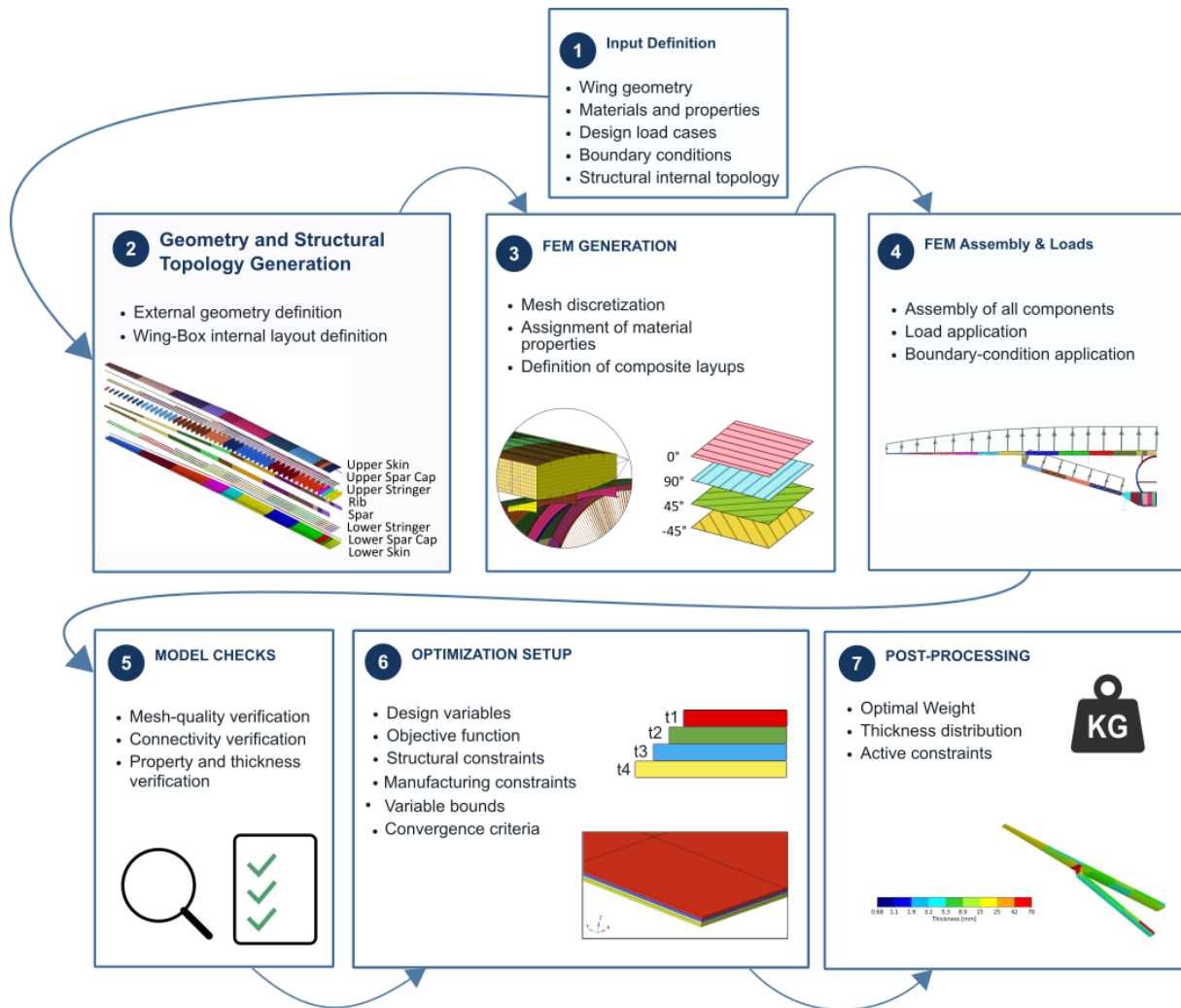


Figure 45 : Automation Workflow.

6.8. Weight Estimation Surrogate Model

Considering the physics-based structural model described above, a surrogate was generated to quantify the sensitivity of the wing-strut-connector structural mass (WSC) to outboard motor mass and to provide a compact interpolation of the sampled structural-sizing results. The surrogate was not used to update the lifting-system structural mass within the final WP5 multidisciplinary optimisation; in that implementation, the primary structural mass was treated as a frozen input. The present study is therefore retained as a structural-sensitivity and methodology result that can support future coupled structural/MDO developments.

In this optimization campaign, the linear static and buckling stability constraint are considered with the design variables, load cases, manufacturing requirements, and static-response constraints previously described. Buckling constraints were introduced to prevent the development of local and global instability modes within the prescribed design load envelope. Their inclusion generally requires additional structural stiffness and may therefore modify the thickness distribution, laminate definition, and overall structural mass compared with the solution obtained using linear static analysis alone.

The surrogate was generated varying the weight of the electric motors represented within the model by the four outboard (OB) point masses. An attempt to expand this design of experiments was done by also varying the inboard (IB) mass in the position of the turboshaft, but the results obtained did not show a clear variation trend, with the best results obtained showing that there was not a big variation when changing the mass of just the IB point. Therefore, this parameter was kept fixed.

6.8.1. Sampling

An initial sampling of eighteen points was generated considering OB masses between 0 and 0.4375 tons. Moreover, each of these eighteen points was tested by considering three random initial thickness distributions. Of the eighteen sampled values, sixteen converged to feasible designs, defined as having a maximum constraint violation of no more than 2%. However, among the sixteen converged solutions, the structural weight did not vary smoothly with the OB mass and since these solutions satisfy the imposed constraints, the observed variations cannot be attributed to failed convergence.

The irregular variation of WSC with OB mass can be explained by the behaviour of buckling eigenvalues during the optimization. As OB mass changes, the critical buckling modes of the structure also change. In some regions, two buckling eigenvalues approach each other very closely and undergo what is known as eigenvalue veering. Near such a region, the associated mode shapes exchange their physical characteristics. As a result, the eigenvalue identified by a given numerical index does not necessarily correspond to the same physical buckling mode for all values of OB masses.

This is important because the sizing optimization constrains the buckling eigenvalues while simultaneously modifying the panel thicknesses. When the identity of the governing buckling mode changes, the optimizer may follow a different design path and converge to a different local optimum. Two nearby values of OB mass can therefore produce noticeably different thickness distributions even though both final designs satisfy the same minimum buckling load factor of 1.5. One solution may require substantially more structural mass than the other. The sharp increases in WSC observed at some sampled values of OB mass can therefore be interpreted as convergence to heavier local optima associated with a different buckling-mode configuration, rather than as infeasible or not converged solutions.

For the subsequent physical analysis, only the converged points that exhibited a consistent and physically meaningful variation of WSC with motor mass were retained (Table 10). Specifically, the selected points follow the overall decreasing trend of WSC with increasing OB mass, while points showing abrupt deviations from this behaviour were excluded from the discussion. This filtering does not imply that the excluded solutions are invalid but rather avoids basing the interpretation on isolated solutions that may result from convergence to different local optima. The retained points therefore provide a clearer representation of the underlying influence of motor mass on the structural response.

Table 10: Optimized WSC as a function of OB motor mass.

Motor mass, m [ton]	WSC [ton]
0.03125	4.293
0.125	3.994
0.15625	3.341
0.1875	2.933
0.25	2.746
0.3125	2.662
0.4375	2.352

6.8.2. Empirical regression fit

To obtain a compact, quantitative description of the trend in Table 10, and to support interpolation between the sampled points, three regression models were fit to the seven-point subset by least squares: a linear model ($WSC = a m + b$), a quadratic model ($WSC = a m^2 + b m + c$), and a two-parameter exponential-decay model ($WSC = a \exp(-b m) + c$). The exponential-decay model is nonlinear in its decay-rate parameter b ; it was fit by a grid search over b , solving the remaining two parameters (a, c) by ordinary least squares at each candidate value of b and retaining the value of b that minimizes the sum of squared residuals. Goodness of fit is reported as the coefficient of determination, R^2 , computed on the seven fitted points. Table 11 reports the three fits and Figure 46 plots them against the data.

Table 11: Comparison of regression models fitted to WSC as a function of motor point mass.

Model	Fitted equation	R^2
Linear	$WSC = -4.989m + 4.258$	0.845
Quadratic	$WSC = 11.265m^2 - 10.352m + 4.719$	0.923
Exponential decay	$WSC = 2.983 \exp(-3.944 m) + 1.759$	0.915

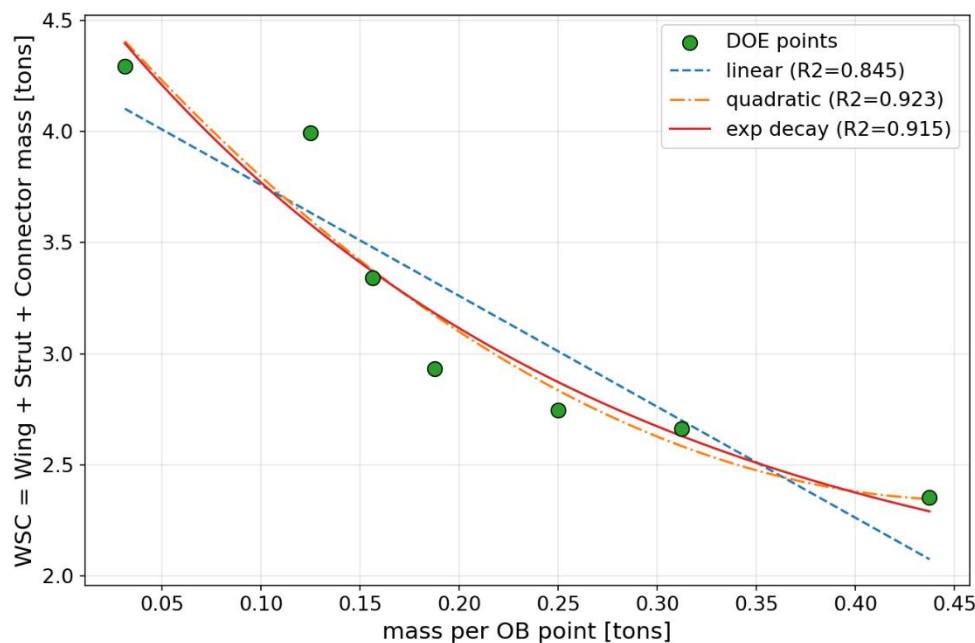


Figure 46: WSC mass with the variation each OB motor mass, with linear, quadratic, exponential decay regression fits.

Both the quadratic and exponential-decay models fit the seven points noticeably better than the linear model ($R^2=0.923$ and 0.915 , respectively, versus 0.845), reflecting a visible flattening of the WSC decrease at higher motor mass. The linear model overpredicts the mass reduction between $m=0.3125$ t and $m=0.4375$ t and underpredicts it between $m=0.03125$ t and $m=0.125$ t. Of the two curved fits, the exponential-decay form is the more physically defensible choice for extrapolation, because it has a strictly positive horizontal asymptote ($c=1.759$ t) representing a floor mass that persists no matter how large the motor mass becomes, consistent with the buckling-relief mechanism. The quadratic fit, by contrast, has no such floor and turns upward again beyond approximately $m=0.46$ t, just outside

the sampled range. This upturn is an artifact of the polynomial form rather than a predicted physical reversal of the trend, and the quadratic model should not be extrapolated past the sampled points. Within the sampled range, $m=0.03125$ t to 0.4375 t, all three models are close to indistinguishable, and any may be used for interpolation.

6.9. Full-Aircraft Mass Model

6.9.1. Aircraft-Level Mass Build-Up

The aircraft mass is evaluated using a component build-up approach. The take-off mass is expressed as

$$m_{TO} = m_{OE} + m_{PL} + m_F + m_{bat},$$

where (m_{OE}) is the operating empty mass, (m_{PL}) is the payload mass, (m_F) is the fuel mass, and (m_{bat}) is the battery mass.

The operating empty mass is further decomposed as

$$m_{OE} = m_{struct} + m_{sys} + m_{furn} + m_{oper} + m_{pp} + m_{elec},$$

where (m_{struct}) denotes the aircraft structural mass, (m_{sys}) the conventional aircraft-system mass, (m_{furn}) the furnishings and installations mass, (m_{oper}) the operational-items mass, and (m_{pp}) the conventional powerplant and associated installation mass. The term (m_{elec}) accounts for the additional electrical components required by the hybrid-electric propulsion architecture, excluding the battery mass, which is included separately in m_{TO} .

6.9.2. Mass-Estimation Methodology

The full-aircraft mass model combines physics-based structural sizing, semi-empirical component-weight equations, fixed reference-aircraft data, and power-specific scaling relationships. The adopted method depends on the component under consideration and on the level of fidelity available within the multidisciplinary analysis framework.

The primary structures of the lifting system, comprising the wing, strut, and central fuselage region containing their attachments, are evaluated using a Global Finite Element Model (GFEM). This physics-based approach is adopted because conventional statistical weight correlations may not adequately represent the structural behaviour and load paths of unconventional configurations such as the strut-braced wing.

- The remaining airframe, propulsion-system, aircraft-system, and operational masses are estimated using a combination of:
- fixed values derived from an A320-class reference-aircraft mass breakdown;



- power-based scaling relationships using assumed component-specific powers;
- semi-empirical equations for the empennage and propeller masses.

The primary-structure sizing is performed outside the MDAO framework considered in the present work. Consequently, the resulting lifting-system structural mass is treated as a frozen input during the aircraft-level optimisation and is not recalculated when the propulsion-system design variables are modified.

Although several of the source equations are conventionally referred to as component-weight equations and some implementation variables retain a weight-based nomenclature, the final quantities introduced into the full-aircraft model are component masses expressed in kilograms. Semi-empirical equations formulated in imperial units are therefore converted to SI units before their results are used.

Lifting-System Structural Mass - The primary structural mass of the lifting system is obtained from a GFEM-based sizing optimisation of the material distribution within the wing, strut, and central fuselage attachment region. The structural modelling and sizing methodology follows the approach described in [29] and in previous sections, in which the lifting-surface components are divided into spanwise design regions, with an independent thickness or laminate-sizing variable assigned to each region.

Secondary lifting-system structures and components not explicitly represented in the GFEM are estimated using handbook methods from Raymer [30] and Torenbeek [31]. Where applicable, these estimates are calibrated using the A320-200 mass breakdown reported by Obert [32]. The T-tail mass estimates are additionally adjusted using available ATR42 and ATR72 mass-breakdown data [32]

Propulsion-System Mass:

Turboshaft Mass - The mass of each turboshaft is estimated from its rated power and an assumed specific power. When the gas-turbine model is active, the sea-level-equivalent turboshaft rating is approximated from the top-of-climb rating as

$$P_{TS} = 2.5P_{TOC},$$

where P_{TOC} is the turboshaft power at top of climb and (P_{TS} is the equivalent rated power used for mass estimation.

A turboshaft specific power of

$$\sigma_{TS} = 5 \text{ kW, kg}^{-1}$$

is assumed. The installed mass assigned to one turboshaft is therefore calculated as

$$m_{TS} = \frac{P_{TS}}{\sigma_{TS}, 10^3} + \frac{m_{ECS}}{2} + \frac{m_{FD}}{2},$$

where the factor (10^3) converts the specific power from (kW, kg^{-1}) to (W, kg^{-1}). The engine-control and fuel-distribution system masses are prescribed as

$$m_{ECS} = 299 \text{ kg, } m_{FD} = 249 \text{ kg.}$$



Half of each system mass is allocated to each turboshaft installation, resulting in a fixed installation allowance of

$$\frac{m_{ECS}}{2} + \frac{m_{FD}}{2} = 274 \text{ kg}$$

per turboshaft. It can consequently be written as

$$m_{TS} = \frac{P_{TS}}{5000} + 274,$$

where P_{TS} is expressed in watts.

Gearbox Mass - The gearbox mass is estimated from its rated power using an assumed specific power of

$$\sigma_{GB} = 16.447 \text{ kW, kg}^{-1}.$$

The gearbox mass is then given by

$$m_{GB} = \frac{P_{GB}}{\sigma_{GB}, 10^3},$$

where P_{GB} is the gearbox rated power expressed in watts.

Electrical Propulsion-System Mass

When the aggregated electrical-system model is selected, the inboard and outboard electrical propulsion systems are represented using separate power-specific mass relationships. The assumed specific powers are

$$\sigma_{OB} = 4 \text{ kW, kg}^{-1}, \sigma_{IB} = 16 \text{ kW, kg}^{-1}.$$

For design powers expressed in megawatts, the corresponding masses are

$$m_{OB} = \frac{1000P_{OB}}{\sigma_{OB}}, m_{IB} = \frac{1000P_{IB}}{\sigma_{IB}},$$

where P_{OB} and P_{IB} denote the aggregated outboard and inboard electrical-system design powers, respectively. The total electrical propulsion-system mass is

$$m_{EPS} = m_{OB} + m_{IB}.$$

Individual-component model - When the aggregated electrical-system model is deactivated, the electric motors, inverters, and inverter--rectifier assemblies are evaluated individually. Their assumed specific powers are

$$\sigma_{EM} = 15 \text{ kW, kg}^{-1}, \sigma_{INV} = 23 \text{ kW, kg}^{-1}, \sigma_{IR} = 9.07 \text{ kW, kg}^{-1}.$$



The corresponding component masses are calculated as

$$m_{EM} = \frac{P_{EM}}{\sigma_{EM}, 10^3}, m_{INV} = \frac{P_{INV}}{\sigma_{INV}, 10^3}, m_{IR} = \frac{P_{IR}}{\sigma_{IR}, 10^3},$$

where the rated powers are expressed in watts.

Propeller Mass

The inboard and outboard propeller masses are estimated using a modified General Dynamics-type relationship reported by Roskam [25]. The relationship depends on the propeller diameter, transmitted power, and number of blades. A fixed value of five blades is assumed:

$$N_b = 5.$$

Inboard propeller: The mass of one inboard propeller is calculated as

$$m_{prop,IB} = 11.36 N_b^{0.391} (D_{IB} P_{GB,MW})^{0.782},$$

where (D_{IB}) is the inboard propeller diameter in metres and

$$P_{GB,MW} = 10^{-6} P_{GB}$$

is the gearbox power expressed in megawatts.

Outboard propeller: The mass of one outboard propeller is calculated as

$$m_{prop,OB} = 11.36 N_b^{0.391} \left[D_{OB} \left(\frac{P_{OB}}{8} \right) \right]^{0.782},$$

where (D_{OB}) is the outboard propeller diameter and (P_{OB}) is the aggregated outboard-system design power. The division by eight represents the equal distribution of the aggregated outboard power among the eight outboard propulsion units.

Empennage Mass

The horizontal- and vertical-tail masses are evaluated using modified Raymer-type component-weight equations [30]. These equations are evaluated internally using imperial units, after which the resulting component weights are converted into masses expressed in kilograms.

The design gross weight used by the implemented equations corresponds to a prescribed aircraft mass of 79,000 kg:

$$W_{dg} = 79,000(2.20462) \quad [\text{lb}].$$

The ultimate load factor is defined as

$$N_z = 1.5(2.5) = 3.75.$$

The tail arm is calculated as

$$L_t = x_{HTP} - (x_w + x_{ac,w}),$$

where x_{HTP} is the longitudinal position of the horizontal tail, x_w is the wing-root leading-edge position, and ($x_{ac,w}$) is the wing aerodynamic-centre position relative to the wing-root leading edge.



Vertical-Tail Mass: The vertical-tail reference area is converted from square metres to square feet according to

$$S_{VT}[\text{ft}^2] = 10.7639S_{VT}[\text{m}^2].$$

The implemented vertical-tail weight equation is

$$W_{VT} = 0.0026 \left(1 + \frac{H_t}{H_v}\right)^{0.225} W_{dg}^{0.556} N_z^{0.536} L_t^{-0.5} S_{VT}^{0.5} K_z^{0.875} \times (\cos 34^\circ)^{-1} AR_{VT}^{0.35} \left(\frac{t}{c}\right)_{VT}^{-0.5},$$

with the prescribed parameters

$$\frac{H_t}{H_v} = 1, AR_{VT} = 1.82, K_z = L_t, \left(\frac{t}{c}\right)_{VT} = 0.15.$$

The result of the above equation is obtained in pounds and converted to kilograms as:

$$m_{VT} = \frac{W_{VT}}{2.20462}.$$

Horizontal-Tail Mass: The horizontal-tail span and reference area are converted to imperial units using

$$b_{HT}[\text{ft}] = 3.28084b_{HT}[\text{m}], S_{HT}[\text{ft}^2] = 10.7639S_{HT}[\text{m}^2].$$

The implemented horizontal-tail weight equation is

$$W_{HT} = 0.0379K_{uht} \left(1 + \frac{F_w}{b_{HT}}\right)^{-0.25} W_{dg}^{0.639} N_z^{0.1} S_{HT}^{0.75} L_t^{-1} K_y^{0.704} \times (\cos 31^\circ)^{-1} AR_{HT}^{0.166} \left(1 + \frac{S_e}{S_{HT}}\right)^{0.1} \frac{6}{8},$$

where

$$K_{uht} = 1.143, F_w = 3(3.28084) \text{ ft}, AR_{HT} = 5, K_y = 0.3L_t, \frac{S_e}{S_{HT}} = 0.3.$$

The factor (6/8) is an additional empirical correction applied to the original relationship. The resulting horizontal-tail mass is

$$m_{HT} = \frac{W_{HT}}{2.20462}.$$

Fixed Airframe, System, and Operational Masses

Components that are neither explicitly sized by the GFEM nor recalculated using power- or geometry-dependent equations are represented using fixed masses derived from an A320-class aircraft mass breakdown. The prescribed values are summarised in Table 12.

Table 12: Fixed component masses used in the full-aircraft mass model.

Component	Mass [kg]
Wing reference mass	8457.43
Fuselage	8938



Complete landing-gear system	2275
Pylons and nacelles	907
Furnishings and installations	2743
Avionics	4788
Operational items	3215

The furnishings and installations mass is obtained from the sum

$$m_{FI} = 2431 + 103 + 200 + 9 = 2743 \text{ kg.}$$

Landing-Gear Mass Distribution

The total landing-gear mass is divided between the main and nose landing gears using fixed proportions:

$$m_{MLG} = 0.8m_{LG} = 0.8(2275) = 1820 \text{ kg, } m_{NLG} = 0.2m_{LG} = 0.2(2275) = 455 \text{ kg.}$$

Nacelle and Pylon Mass

The mass assigned to an individual nacelle or aircraft-side propulsion installation is one-half of the stored pylons-and-nacelles reference mass:

$$m_{nacelle} = \frac{907}{2} = 453.5 \text{ kg.}$$

This implementation effectively distributes the total reference value equally between the two aircraft sides or between two principal propulsion installations.

Wing Reference Mass

In the current aircraft-level implementation, the GFEM-derived wing reference mass is prescribed as a fixed value:

$$m_{wing} = 8457.43 \text{ kg.}$$

6.9.3. GFEM Full Model

The automatic structural-modelling workflow presented in Figure 47 was extended beyond the generation, sizing, and post-processing of the wing and strut finite element models. The original half-aircraft model, used primarily to estimate the structural mass and optimal material distribution of the lifting surfaces, was incorporated into a full-aircraft representation.



The full-aircraft model was assembled by combining the structural properties obtained from the wing and strut sizing process with the mass contributions of the remaining aircraft systems and components. These included the fuselage, empennage, propulsion system, landing gear, wing, and strut. Each contribution was introduced at an appropriate representative location, allowing the global aircraft mass distribution to be reproduced while retaining the structural mass estimated through the automatic workflow.

This extension enables the calculation of the overall aircraft centre-of-gravity position and inertial properties, including the mass moments and products of inertia. The main objective of the full-aircraft

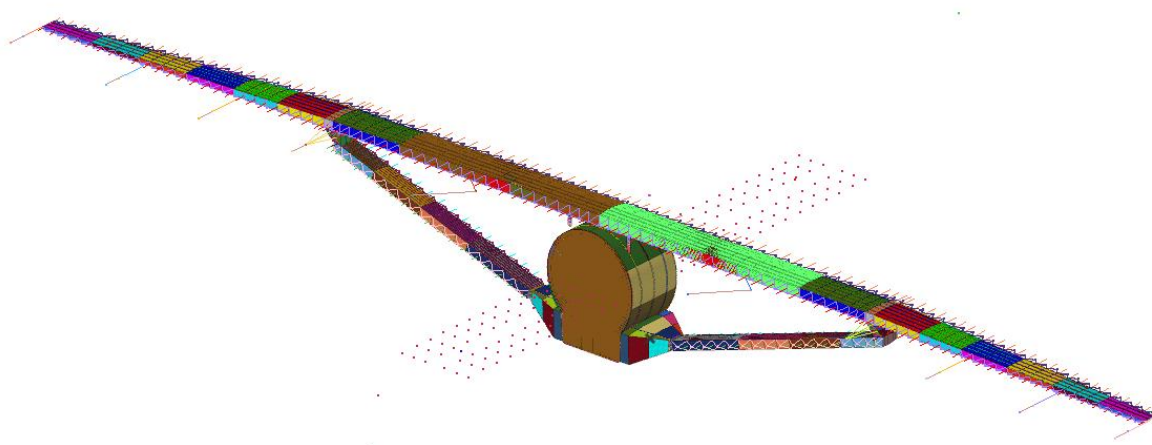


Figure 47: Aircraft full model: first extension of the structural sizing FE model

model was therefore to provide the centre-of-gravity location and inertia-related inputs required for the aeroelastic analyses performed within task T3.4. In this manner, changes in the structural sizing or aircraft configuration can be consistently reflected in the mass properties supplied to the aeroelastic model.

Figure 48 view isolates the lifting structure so that the relationship between the wing box, bracing strut and central fuselage can be seen clearly. The strut connects the lower central structure to an outboard wing location, providing an additional path for transferring wing loads. The shell mesh represents the geometry relevant to wing and strut bending and torsion. Five concentrated masses, connected at their structural attachment nodes, represent the WP3 propulsion units in the modelled

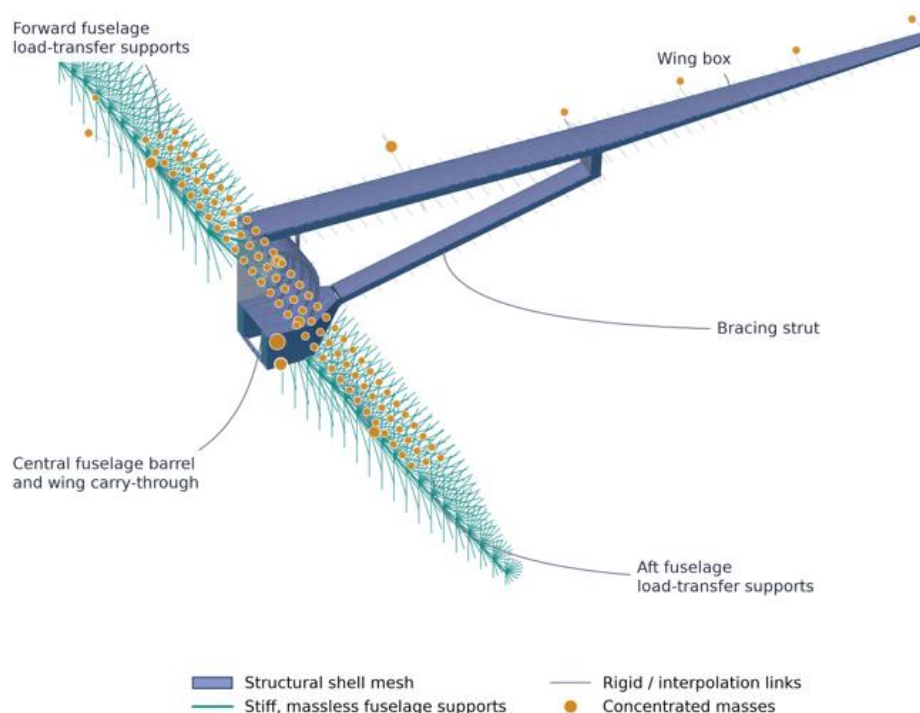


Figure 48: Augmented FE model for aerodynamic-load splining.

half. The markers show their positions; nacelles and propeller blades are not meshed in this representation.

The forward and aft fuselage regions use stiff, massless beam spines and radial support connections to provide stations for transferring aerodynamic loads to the central structural model. There are 32 support stations, each with 13 ring points, and 30 spine beam elements. The gold markers indicate concentrated masses in the fuselage region. This idealisation extends the load-transfer interface while the detailed shell model remains concentrated around the central barrel and lifting structure. The outer fuselage flexibility is therefore not resolved by these supports; the central barrel remains represented by structural shells.

The augmented FE model used for the aeroelastic activities in T3.4 was subsequently adapted for the T5.6 FSI activities, where the CFD and FE meshes must be interfaced. The resulting layout is shown in Figure 49.

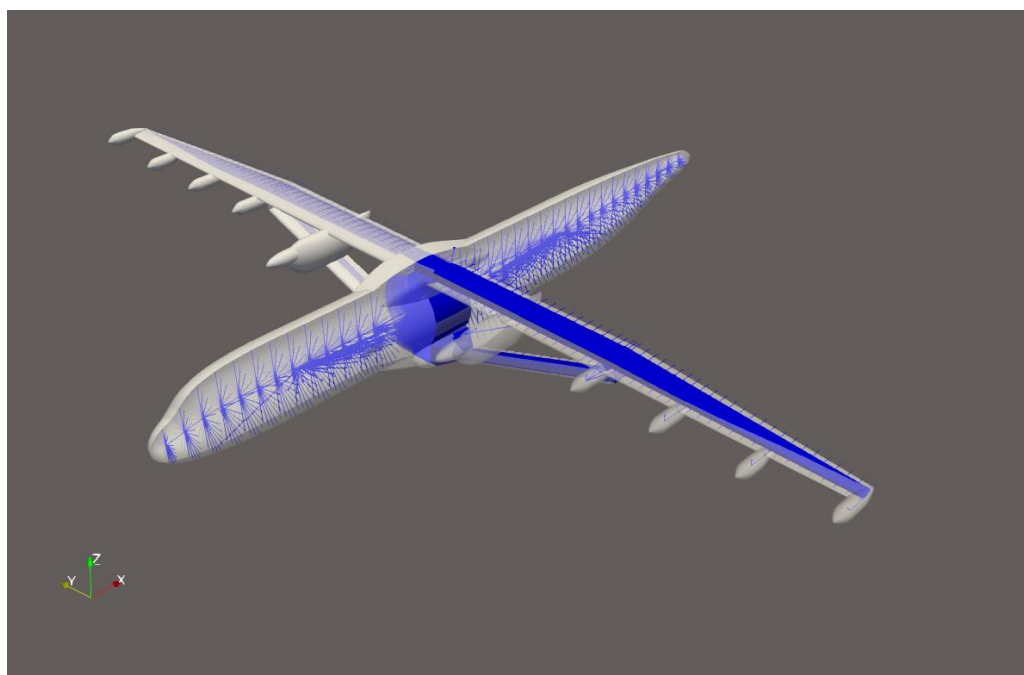


Figure 49: Layout of the structural FE model used for the WP5 Task 5.6 aeroelastic/FSI analyses.

7. Task3.6 - Aircraft performance model

Within INDIGO, the Aircraft Performance Model (APM) denotes the mission-level multidisciplinary analysis used to evaluate a prescribed aircraft configuration and mission. In the final WP5 implementation, the APM combines the aircraft-level representations for mass and balance, aerodynamics, trim, propulsion and energy, emissions and acoustic indicators to determine aircraft states, required power, fuel consumption and environmental metrics along the mission. The detailed CFD/POD aerodynamic surrogate, aeroelastic and loads framework, and high-fidelity structural and FSI models documented in the preceding sections provide complementary discipline-level modelling and verification capabilities. They were not all embedded directly in the final WP5 optimisation loop. In particular, the aerodynamic representation used in the WP5 MDO was the VSPAero-based model described in Section 3.9; the corresponding coupling and optimisation formulation is documented in D5.2.

To summarize, with this block it is possible to fly digitally the aircraft, obtaining, among others,

- pollution rates and noise impact for a given aircraft setting
- power settings to fly a certain mission
- state of the aircraft along a given mission

or conversely, with a inverse/design approach:

- designing
- operating



the aircraft to get certain performances and/or minimize certain quantities of interest.

7.1. Emission model

In gas turbine combustion chambers various pollutant species are produced and emitted to the atmosphere. The assessment of these emissions during the design and optimization process of the INDi2 configuration is performed using a combined gas turbine and combustion chamber surrogate model. These models were developed by the project partners RUB and TUBS as part of WP2 of the INDIGO project and were delivered to WP3 as part of D2.2 “Integrated Hybrid Propulsion System Model including Noise Source Modelling” [33].

The goal of the pollutant emission surrogate model is to predict CO₂ and non-CO₂ emissions as a function of the gas turbine design and operation. Non-CO₂ emission species comprise carbon monoxide (CO), nitrogen oxides (NO_x) and unburned hydrocarbons (UHC). To enable a cost-efficient pollutant emission simulation while maintaining the required predictive accuracy, a chemical reactor network (CRN) model has been developed by RUB which is derived from reactive 3D CFD simulations of a modern rich burn, quick quench, lean burn combustor (Figure 50). The formation of pollutant species such as CO and UHC is directly linked to the combustion efficiency resulting from the emission formation pathways influence on the global energy transformation in the combustion process. Accordingly, a direct implementation of the combustor model into the gas turbine model is required.

For this purpose, the CRN model is applied to a wide range of possible operating conditions defined by combustor inlet temperature and fuel-to-air ratio. The resulting data are stored in a multi-dimensional lookup table which is directly implemented into the gas turbine design and performance assessment model.

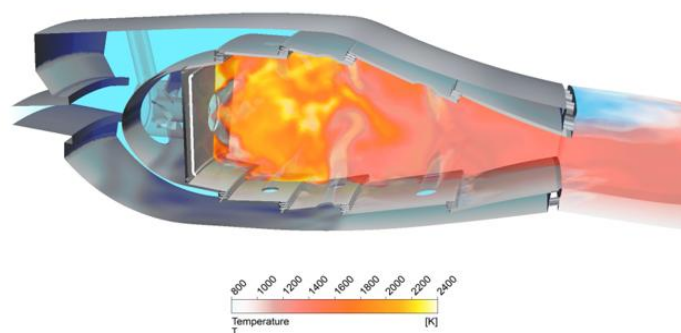


Figure 50: Snapshot of temperature field in the RUB CFD combustor model at the idle operating point, simulated using an FGM approach with liquid spray combustion and detailed chemical kinetics for sustainable aviation fuel (SAF) Jet-C1.

To generate the surrogate model for WP3, the combined gas turbine and combustion chamber model is used to carry out design calculations and performance assessments across different operational scenarios. The compressor pressure ratio π_{HPC} and the combustor outlet temperature $T_{t,4}$ are varied to account for various gas turbine designs. To cover different operating conditions, the altitude H , Mach number Ma and the high-pressure spool speed $N_{2,rel}$ are varied. The parameter ranges are summarized in the table below.



	On Design		Off Design		
Property	π_{HPC}	$T_{t,4}$ in K	H in m	Ma	$N_{2,rel}$
Range	[15, 45]	[1400, 1900]	[0, 8000]	[0, 0.6]	[0.4, 1.05]

An overview of the input and output structure of the combined gas turbine and combustion chamber surrogate model is shown in Figure 51. To enable a straightforward integration of the surrogate model into the MDO process, the input property $N_{2,rel}$ describing the throttle setting is converted to a power input at the given mission point, denoted with $P_{mission}$. Moreover, the maximum power of the gas turbine can be defined by an additional input property P_{Design} , which functions as a linear scaler to the absolute output properties. The conversion and the power scaling are performed within the surrogate itself. The output properties of the surrogate model include the compressor pressure ratio π_{HPC} , the power specific fuel consumption $Psfc$, the combustor outlet temperature T_{t4} as well as CO₂ and non-CO₂ emission species. These results are stored in a database that is accessed via a linear interpolation scheme to extract the output properties at any combination of input properties. During the MDO process, the gas turbine design is fixed at the beginning of each iteration. Subsequently, the surrogate is used to evaluate the fuel consumption and emission production along the entire mission, requiring low computational power.

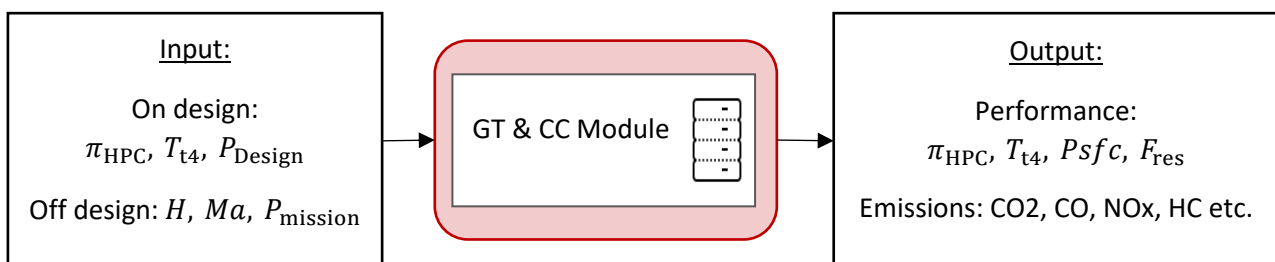


Figure 51. Input and output structure of the combined gas-turbine and combustion-chamber surrogate model.

8. Conclusions

Deliverable D3.2 establishes the multidisciplinary modelling framework required to evaluate the INDIGO DHEP-LARW aircraft. The developed capabilities cover aerodynamic performance, installed-propulsion noise, coupled flight-dynamic and aeroelastic behaviour, structural sizing, mission performance and pollutant emissions.

The aerodynamic and structural-sizing activities provide models of increasing fidelity together with reduced-order representations suitable for repeated design evaluations. The aeroacoustic framework combines complementary medium- and high-fidelity methodologies, allowing efficient parametric investigations while retaining the capability to analyse detailed installation and interaction effects.

The aeroelastic activities provide a common framework for trim, stability, manoeuvre response, dynamic loads and discrete-gust assessment. For the analysed INDi2 conditions, static structural deflections were small enough for geometric nonlinearities to be neglected. Nevertheless, structural

flexibility and distributed propulsion significantly influenced the modal characteristics, transient response and recovered loads. The propulsion system must therefore be represented through its aerodynamic effects, distributed mass, rotary inertia and gyroscopic contributions rather than only through a prescribed thrust distribution.

The mission-level disciplinary representations used in the aircraft-level analysis are brought together through the Aircraft Performance Model, which supports prediction of aircraft states, power requirements, fuel consumption, emissions and noise-related indicators. The higher-fidelity aerodynamic, structural and aeroelastic capabilities documented in D3.2 provide complementary verification and refinement capabilities.

Overall, the models documented in D3.2 provide the multidisciplinary-analysis capability required for the optimisation and subsequent verification of the INDIGO aircraft. The results remain configuration- and condition-dependent and do not represent a complete certification assessment, which would require a broader set of flight conditions, mass cases, failure scenarios and load cases.

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