

GRADIENT-BASED OPTIMISATION OF A CARBON-FIBRE WING STRUCTURE FOR DEPLOYMENT IN A HYBRID-ELECTRIC SMALL-AIRCRAFT CONFIGURATION

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Abstract

As part of the DLR-Center for Small Aircraft Technologies, the project *D-LIGHT+* aims to develop a climate-neutral commuter aircraft by combining hydrogen fuel cells with distributed electric propulsion. The overall aircraft design process is supported by a multidisciplinary approach, bringing together specialists from conceptual design, aerodynamic optimisation, fuel-stack development, hydrogen-tank manufacturing, electric powertrain design, flight dynamics, and aeroelasticity. The weight and size of the fuel stack and powertrain systems have emerged as the key obstacles to this undertaking, prompting a search for creative solutions and technological breakthroughs.

One key aspect is the replacement of the current sizing process of the primary wing structures. The new approach implements a gradient-based optimisation process for carbon fibre laminates, tailoring the structural characteristics to the specific requirements of the configuration. For this purpose, a new optimisation model generator is implemented, capable of converting known topology models by identifying key assembly groups. The optimisation models are constructed using *lamination parameters*, which describe the structural properties of a laminate by a reduced set of variables in a continuous and convex space. This is complemented by a set of constraints, ranging from lamination parameter compatibility and global buckling simulations to the Tsai-Wu failure criterion in lamination parameter space. Finally, this work highlights the integration of this approach into an aeroelastic loads and optimisation framework and addresses the impact on key characteristics of the configuration.

Keywords

Structural Optimisation; Carbon Fibre; Lamination Parameter

NOMENCLATURE

Symbols

A	in-plane stiffness matrix	N/mm	I_2	maximum shear strain
α	step-size scaling parameter		λ	strain scaling factor
c	hyperplane offset		λ_{crit}	critical strain scaling factor
D	out-of-plane stiffness matrix	Nmm	n	normalized hyperplane coefficient
δ	conservatism tolerance		n_a	number of discrete fiber angles
E	hyperplane maximum violation		n_h	number of hyperplanes of reference
ϵ	laminate strain vector		$\tilde{n}_h$	number of hyperplanes of simplification
η	normalised wing span coordinate		n_l	number of constant thickness layers
F	stress interaction parameter $\text{mm}^2/\text{N}, \text{mm}^4/\text{N}^2$		n	hyperplane normal vector
G	strain interaction parameter		p	polynomial coefficient
g	acceleration due to gravity	m/s^2	Q	reduced ply stiffness matrix
I	material invariant matrix		$\bar{q}$	dynamic pressure
I_1	volumetric strain invariant		S	In-plane shear strength
			t	laminate thickness
			θ	ply fibre angle

U	material invariant	
u	material coefficient	
V^A	in-plane lamination parameter	
V^D	out-of-plane lamination parameter	
V_{hull}	reference convex hull volume	
$\tilde{V}_{\text{hull}}$	simplification convex hull volume	
$\tilde{V}_{\text{loss}}$	simplification convex hull volume loss	%
$\mathbf{v}$	lamination parameter vector	
x	hyperplane intersection vertex	
$\mathcal{X}$	set of hyperplane intersection vertices	
X	strength in x-direction	N/mm ²
ξ	aileron deflection	deg
Y	strength in y-direction	N/mm ²
z	laminate mid-plane distance	mm
$\bar{z}$	normalized laminate mid-plane distance	

Indices

c	compressive
t	tensile
x	x-direction
xy	in-plane
y	y-direction

Abbreviations

CFRP	carbon-fibre-reinforced plastic	
CG	center of gravity	
EIS	entry into service	
LFL	landing field length	m
MOEM	max. operating empty mass	kg
MTOM	max. take-off mass	kg
PAX	passenger	
RCH	reduced complexity hull	
RH	reference hull	
TAS	true airspeed	m/s
TOFL	take-off field length	m



FIG 1. Render image of the D-LIGHT configuration.

1. INTRODUCTION

As a contribution towards climate-neutral aviation, the DLR project *D-LIGHT+* aims to develop a hybrid-electric, commuter-class configuration. The target is to significantly reduce the carbon footprint of general aviation. Key aspects of the proposed nine-seater design include the ten distributed engines, paired with individual engine pods comprising electric motors, thermal management systems, and fuel stacks (Fig. 1).

Another distinctive feature is the placement of large hydrogen tanks next to the fuselage on top of the wing structure. The top-level aircraft requirements are listed in Tab. 1, of which the most challenging one emerged as the maximum takeoff mass of 5700 kg. Heavy fuel stacks limit the range of the aircraft and pose a complicated challenge for system integration.

To address these challenges, a multidisciplinary workflow has been established, featuring contributions by specialists in conceptual design [1], aerodynamic optimisation, fuel-stack development [2], hydrogen-tank manufacturing [3], electric powertrain design [4], and structural optimisation. Part of this workflow is the loads and optimisation software cpacs-MONA [5], generating a preliminary design of the primary wing structures. Based on the universal descriptive input format CPACS [6], this software generates simulation models, calculates a comprehensive set of elastic manoeuvre, gust, and ground load cases, optimises the load-bearing thin shell wing box structure, and provides an initial analysis of the static and dynamic stability characteristics of the elastic aircraft.

However, until now, the framework was limited to isotropic materials or static stiffness distributions. Consequently, this work proposes a method to update CSM models of wing box structures with an optimisation model for carbon-fibre-reinforced plastics (CFRP). In this context, an introduction to *lamination parameters* is given, suitable for continuous gradient-based optimisation. Furthermore, design space limits and failure criteria in lamination parameter formulation are introduced and details with respect to their implementation are presented. Additionally, the implementation of the conversion software *pyOptConv* is addressed, together with the integration into cpacs-MONA. Finally, the software is applied to the D-LIGHT configuration and

Requirement	Value	Unit	Origin
PAX	9		Project requirement
Payload	855	kg	95 kg per PAX
Range	600	km	Flight Data
Cruise altitude	10 000	ft	Fuel cell performance
Cruise speed, TAS	300	km/h	Similar A/C
TOFL at sea level	≤ 800	m	Airport size
TOFL at 6600 ft	≤ 1200	m	Assumption
LFL at sea level	≤ 800	m	Airport size
MTOM	≤ 5700	kg	Runway class
Wing span	≤ 24	m	Airport size
EIS	2030		Project requirement
Quarter chord sweep	4.74	deg	Distributed engines
Dihedral	7	deg	Distributed engines

TAB 1. Top level aircraft requirements

the result of the carbon fibre optimisation is compared to an aluminium reference design.

2. METHODOLOGY

2.1. Lamination Parameters

Since the structural optimisation results feed into a conceptual multidisciplinary design process, it is only necessary to capture the wing's structural properties with sufficient accuracy without defining a discrete, manufacturable ply layup. However, the approach shall still leverage the weight-saving potential of carbon fibre laminates tailored to the configuration's specific requirements during the early design phase.

In this context, the continuous lamination parameter space, introduced by TSAI and PAGANO [7] and TSAI and HAHN [8], offers several distinct advantages over discrete parameterisation: It reduces the design variables to at most twelve independent stiffness parameters and overall thickness [9]. This substantially lowers optimisation complexity for industrial laminates with hundreds of layers compared to discrete layup parameterisation [9]. Furthermore, the stiffness properties depend linearly and uniquely on the lamination parameters, and the feasible design space is convex [10]. These properties enable the use of efficient gradient-based optimisers and facilitate convergence toward a global optimum [10]. In the context of this work, the number of independent variables is further reduced to four by limiting the design to symmetric and balanced laminates.

The in-plane V_i^A and out-of-plane V_i^D ($i = 1, 3$) lamination parameters for these laminates are defined as the dimensionless integrals of the fibre orientation functions

across the total laminate thickness t [7]:

$$\begin{aligned} (V_1^A, V_3^A) &= \int_{-\frac{1}{2}}^{\frac{1}{2}} (\cos 2\theta(\bar{z}), \cos 4\theta(\bar{z})) d\bar{z} \\ (V_1^D, V_3^D) &= 12 \int_{-\frac{1}{2}}^{\frac{1}{2}} \bar{z}^2 (\cos 2\theta(\bar{z}), \cos 4\theta(\bar{z})) d\bar{z} \end{aligned} \quad (1)$$

where $\bar{z} = z/t$ is the thickness-normalized coordinate of the layer mid-plane z , and $\theta(\bar{z})$ is the fibre angle at that position. This allows the in-plane **A** and out-of-plane **D** laminate stiffness matrices to be expressed linearly in terms of the lamination parameters as follows [7]:

$$\begin{aligned} \mathbf{A} &= t(\mathbf{\Gamma}_0 + \mathbf{\Gamma}_1 V_1^A + \mathbf{\Gamma}_3 V_3^A) \\ \mathbf{D} &= \frac{t^3}{12}(\mathbf{\Gamma}_0 + \mathbf{\Gamma}_1 V_1^D + \mathbf{\Gamma}_3 V_3^D) \end{aligned} \quad (2)$$

where the $\mathbf{\Gamma}_i$ ($i = 0, 1, 3$) matrices are defined in terms of the material invariants U_i ($i = 1, \dots, 5$) as follows [7]:

$$\begin{aligned} \mathbf{\Gamma}_0 &= \begin{bmatrix} U_1 & U_4 & 0 \\ U_4 & U_1 & 0 \\ 0 & 0 & U_5 \end{bmatrix}, \mathbf{\Gamma}_1 = \begin{bmatrix} U_2 & 0 & 0 \\ 0 & -U_2 & 0 \\ 0 & 0 & 0 \end{bmatrix}, \\ \mathbf{\Gamma}_3 &= \begin{bmatrix} U_3 & -U_3 & 0 \\ -U_3 & U_3 & 0 \\ 0 & 0 & -U_3 \end{bmatrix}, \end{aligned} \quad (3)$$

with the material invariants U_i expressed in terms of the reduced ply stiffness components Q_{ij} as follows [7]:

$$\begin{aligned} U_1 &= (3Q_{11} + 3Q_{22} + 2Q_{12} + 4Q_{66})/8 \\ U_2 &= (Q_{11} - Q_{22})/2 \\ U_3 &= (Q_{11} + Q_{22} - 2Q_{12} - 4Q_{66})/8 \\ U_4 &= (Q_{11} + Q_{22} + 6Q_{12} - 4Q_{66})/8 \\ U_5 &= (Q_{11} + Q_{22} - 2Q_{12} + 4Q_{66})/8 \end{aligned} \quad (4)$$

2.2. Feasible Domain

Although the lamination parameters can be used to tailor specific properties of the laminate (cf. (2)–(4)), their values cannot be chosen independently. Since the trigonometric functions in equation (1) are integrated over the same stacking sequence, the V_i^A and V_i^D lamination parameters ($i = 1, 3$) are coupled both within and between the **A** and **D** matrices. Consequently, the set of all lamination parameter combinations physically realizable by an actual stacking sequence is restricted to a feasible domain. Furthermore, the feasible domain of the lamination parameters, which is required in practice for the simultaneous design of membrane and bending stiffness of a laminate, cannot be described analytically in a general manner, necessitating the use of approximation methods [10–13]. The approach proposed by SETOODEH [13] leverages the convexity of the feasible domain to approximate it as an intersection of half-spaces. These half-spaces are defined by a set of hyperplanes that are tangent to the boundary of the feasible domain and collectively enclose the feasible region, as illustrated in Fig. 11 in the . This linear constraint formulation has proven particularly relevant for practical applications, as it enables a general implementation that can be directly integrated into gradient-based optimisation procedures [10].

In general, these hyperplanes are expressed in the following vector form:

$$f_{\mathbf{n},c}(\mathbf{v}) = \mathbf{n} \cdot \mathbf{v} - c, \quad f_{\mathbf{n},c} : \mathbb{R}^{n_{LP}} \rightarrow \mathbb{R}, \quad (5)$$

where $\mathbf{v}$ represents the vector of lamination parameters, $\mathbf{n}$ denotes the outward-pointing unit normal vector, and c is the perpendicular distance from the origin of the lamination parameter space to the hyperplane. This expression can be reformulated as a normalized linear inequality:

$$\frac{\mathbf{n}}{c} \cdot \mathbf{v} \leq 1 \quad (6)$$

For the considered symmetric and balanced case, this yields the following set of linear inequalities for the lamination parameters V_i^m :

$$\sum_{i \in \{1,3\}} \sum_{m \in \{A,D\}} n_{ik}^m V_i^m \leq 1, \quad \forall k \in \{1, \dots, n_h\} \quad (7)$$

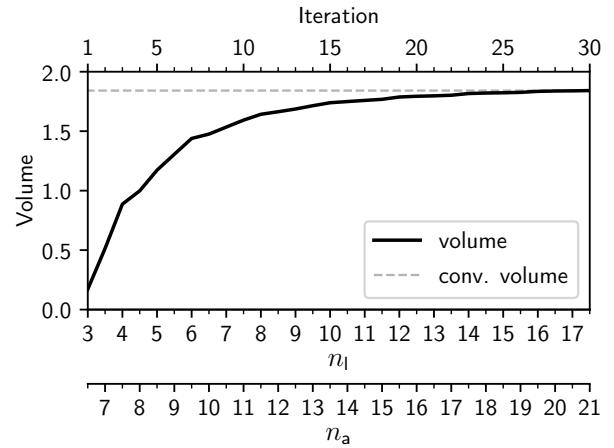
$$-1 \leq V_i^m \leq 1, \quad \forall i \in \{1, 3\}, m \in \{A, D\}$$

where n_h represents the total number of linear constraints.

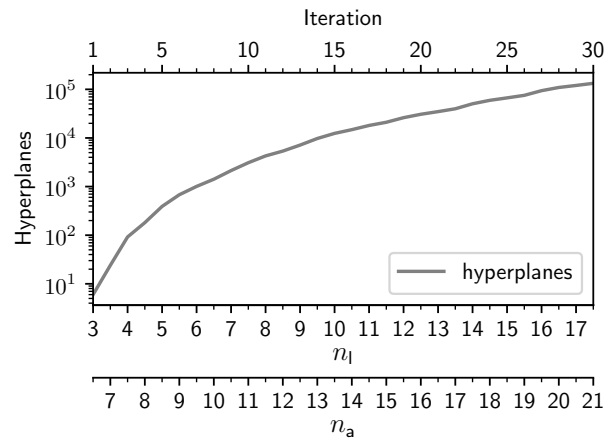
To implement this set of linear constraints, the coefficients n_{ik}^m must be defined, thereby specifying the set of hyperplanes. This is achieved through the following iterative process based on [13]: For symmetric and balanced laminates, all stacking sequences using n_a equidistant fibre angles ($\theta \in [0^\circ, 180^\circ]$) and n_l constant thickness layers define feasible points $(V_1^A, V_3^A, V_1^D, V_3^D)$ in the four-dimensional design space via Equation (1), provided the stacking sequence $\theta(\bar{z})$ is symmetric [11]. The balanced condition need not be explicitly enforced in the discrete stacking sequence, as any feasible symmetric laminate admits a symmetric and balanced sequence that preserves

$(V_1^A, V_3^A, V_1^D, V_3^D)$ [11]. Consequently, the feasible design space of symmetric and balanced laminates can be sampled for a given n_a and n_l by generating all possible permutations of the n_a discrete fibre angles across the $n_l/2$ layers of the upper half, symmetrising the resulting half-stacks to define the full stacking sequence, and calculating the corresponding lamination parameters. Subsequently, the *Qhull* algorithm [14] is used to construct the convex hull of the sampled feasible design space from the generated point set. This procedure is iteratively repeated by successively refining the sampling of the feasible domain with increasing values of n_a and n_l , until the volume of the convex hull converges [13]. The representation of the convex hull as a set of linear constraints can be directly obtained from the half-space output of the *Qhull* algorithm [14].

Fig. 2a illustrates the volume convergence achieved by the process of alternatively increasing the values of n_a and n_l for a symmetric balanced laminate. In comparison, Fig. 2b shows the trend in the number of hyperplanes over the same iterations.



(a) Volume convergence



(b) Hyperplane trend

FIG 2. Approximation convergence trends $(V_1^A, V_3^A, V_1^D, V_3^D)$

It can be observed that the number of linear constraints defining the convex hull approximation scales with sampling density even as the volume converges, while the final count depends primarily on the convergence criterion applied to end refinement. In this instance, the approximation was terminated after 30 iterations with $n_a = 21$ and $n_l = 17$ (cf. Fig. 2), resulting in a convex hull with an approximate volume of $V_{\text{hull}} = 1.8414$, described by $n_h = 132919$ hyperplanes in total. In the following, this convex hull will be considered the reference hull (RH).

The iterative nature of preliminary design optimisation in this work prioritises computational efficiency over high-fidelity model definition. Therefore, a complexity reduction of the RH is performed to balance approximation accuracy with the computational cost of constraints involved. The complexity reduction is implemented using the facet merging capability of the *Qhull* algorithm [14], which automatically combines near-coplanar facets during hull construction when their centrum proximity or normal-vector cosine satisfies a user-defined tolerance. However, a critical limitation of this approach is that the resulting reduced complexity hull (RCH) does not guarantee that all vertices will remain strictly within the boundaries of the RH. Consequently, conservatism must be enforced a posteriori by shifting any hyperplanes that violate the reference hull inwards.

To achieve this, a constraint-tightening algorithm is employed, which systematically minimizes the violation of the RCH relative to the RH. Let the RH be defined by a set of n_h hyperplanes and the RCH by $\tilde{n}_h$ hyperplanes. Each hyperplane is characterized by the inequality described in Equation (5). The objective of the iterative process is to adjust the perpendicular distances c_i ($i = 1, \dots, \tilde{n}_h$) of the RCH such that all vertices lie strictly within the RH. At each iteration k , the algorithm computes the intersection vertices $\mathcal{X}_i^k = \{\mathbf{x}_1, \dots, \mathbf{x}_l\}$ for each hyperplane i of the current RCH and then determines the maximum violation E_i^k for each vertex in $\mathcal{X}_i^k$ against the reference constraints ($j = 1, \dots, n_h$):

$$E_i^k = \max_{\mathbf{x} \in \mathcal{X}_i^k} \left(\max_j (\mathbf{n}_j \cdot \mathbf{x} - c_j) \right). \quad (8)$$

To enforce conservatism, the distances c_i of the RCH are updated to shift the boundaries inward:

$$c_i^{k+1} = c_i^k - \alpha \cdot \max(0, E_i^k) \quad (9)$$

where α serves as a step-size scaling parameter ($\alpha \ll 1$), avoiding unnecessary volume loss due to overcorrection. This iterative procedure continues until the global violation satisfies the termination criterion:

$$\max_i (E_i^k) < \delta \quad (10)$$

where δ denotes the convergence tolerance. Upon satisfying this condition, the tightened RCH constitutes a valid convex subset of the RH.

Fig. 3 illustrates the four-dimensional design space with different planes shown (cf. Fig. 3a–Fig. 3f) both for the reference hull (RH) and the conservative reduced complexity hull (RCH).

The RCH has a remaining volume of $\tilde{V}_{\text{hull}} = 1.5925$, which translates to a relative volume loss of $\tilde{V}_{\text{loss}} = 13.52\%$. But it is described by only $\tilde{n}_h = 450$ hyperplanes in total, providing high computational efficiency. Although the RCH excludes theoretical designs at the domain boundaries, in this work these edge cases are considered secondary during the conceptual design phase.

2.3. Tsai-Wu in Lamination-Parameter Space

Due to the continuous definition of the laminate properties when using lamination parameters, incorporating ply-level strength failure into the design process poses a challenge. Without a discrete stacking sequence, it is impossible to derive the individual ply stresses from the global strains of the laminate. IJSELMUIDEN [15] demonstrates how a conservative envelope for the laminate can still be defined within the lamination parameter space for the Tsai-Wu criterion, assuming a quasi-isotropic laminate. Within this envelope, no failure occurs according to the Tsai-Wu criterion, regardless of the fibre orientation [15]. Depending on the material properties, the authors define this envelope as a root-finding problem for a polynomial that depends on a scaling factor λ . As illustrated in Fig. 12, for the IM6 carbon fiber material system (Table 2) considered in this work, the second-order envelope is critical [15]:

$$f_2(\lambda) = p_{12}\lambda^2 + p_{11}\lambda + p_{10} \quad (11)$$

with:

$$\begin{aligned} p_{10} &= u_4^2 + 4u_1 - 4u_6 \\ p_{11} &= -4u_2I_1(u_1 - u_6) + 2u_4u_5I_1 \\ p_{12} &= 4u_6^2I_2^2 - 4u_3I_1^2(u_1 - u_6) - 4u_6u_1I_2^2 + u_5^2I_1^2 \end{aligned} \quad (12)$$

where I_1 is the volumetric strain invariant and I_2 is the maximum shear strain of the laminate strains ε_x , ε_y , and ε_{xy} :

$$I_1 = \varepsilon_x + \varepsilon_y, \quad I_2 = \sqrt{\left(\frac{\varepsilon_x - \varepsilon_y}{2}\right)^2 + \varepsilon_{xy}^2} \quad (13)$$

and the u_i depend on the material coefficients of the strain-based formulation of the Tsai-Wu criterion [15]:

$$\begin{aligned} u_1 &= G_{11} + G_{22} - 2G_{12}, \\ u_2 &= (G_1 + G_2)/2 \\ u_3 &= (G_{11} + G_{22} + 2G_{12})/4, \\ u_4 &= G_1 - G_2 \\ u_5 &= G_{11} - G_{22}, \\ u_6 &= G_{66} \end{aligned} \quad (14)$$

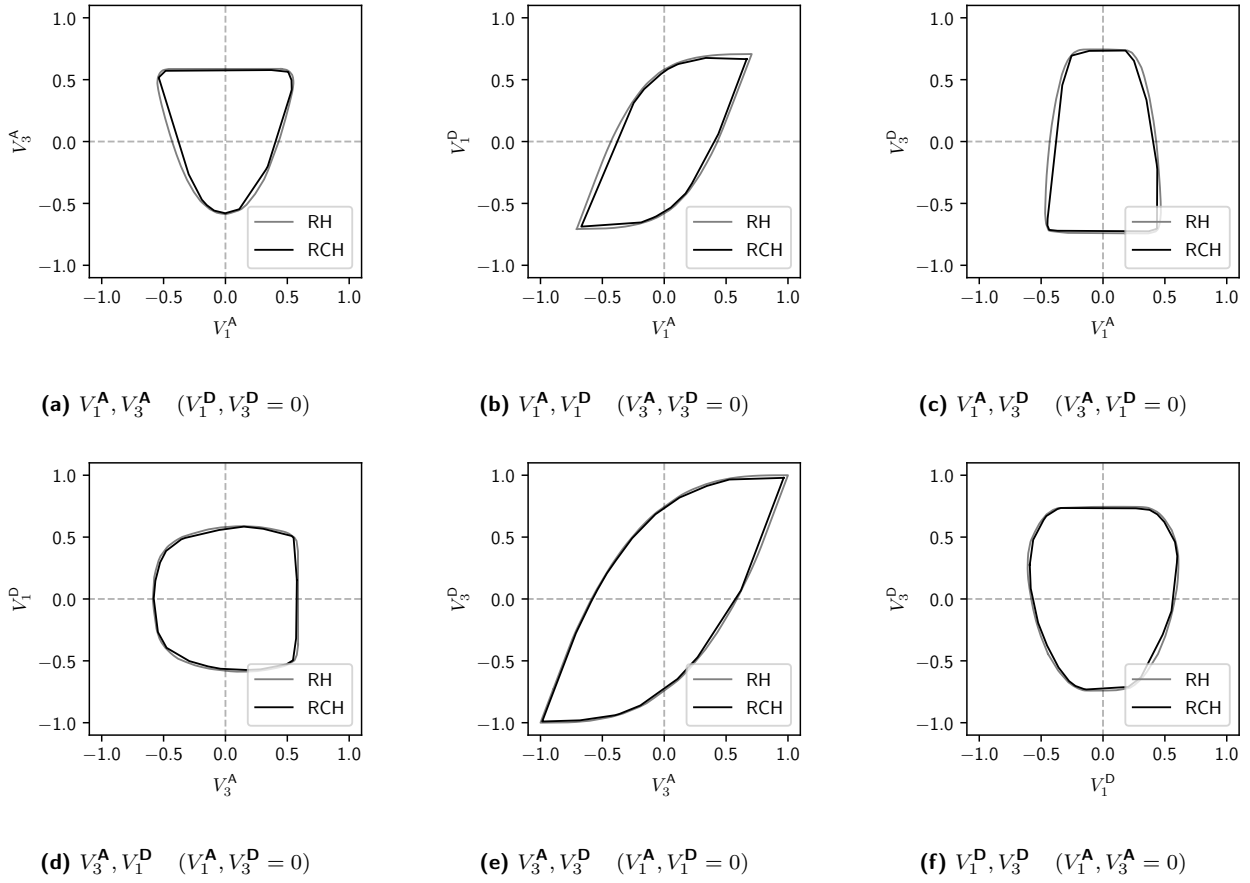


FIG 3. Feasible domain approximation and conservative complexity reduction ($V_1^A, V_3^A, V_1^D, V_3^D$)

with the strain interaction parameters G defined in the principal material coordinate system of the ply:

$$\begin{aligned}
 G_1 &= Q_{11}F_1 + Q_{12}F_2 \\
 G_2 &= Q_{12}F_1 + Q_{22}F_2 \\
 G_{11} &= Q_{11}^2F_{11} + Q_{12}^2F_{22} + 2F_{12}Q_{11}Q_{12} \\
 G_{22} &= Q_{12}^2F_{11} + Q_{22}^2F_{22} + 2F_{12}Q_{12}Q_{22} \\
 G_{12} &= Q_{11}Q_{12}F_{11} + Q_{12}Q_{22}F_{22} \\
 &\quad + F_{12}Q_{12}^2 + F_{12}Q_{11}Q_{22} \\
 G_{66} &= 4Q_{66}^2F_{66}
 \end{aligned} \tag{15}$$

where F_i are the stress interaction parameters in the principal material coordinate system of the ply:

$$\begin{aligned}
 F_1 &= \frac{1}{X_t} - \frac{1}{X_c} & F_2 &= \frac{1}{Y_t} - \frac{1}{Y_c} \\
 F_{11} &= \frac{1}{X_t X_c} & F_{22} &= \frac{1}{Y_t Y_c} \\
 F_{12} &= \frac{-1}{2\sqrt{X_t X_c Y_t Y_c}} & F_{66} &= \frac{1}{S^2}
 \end{aligned} \tag{16}$$

with X_t, X_c, Y_t, Y_c , and S being the material properties under consideration (cf. Table 2).

The parameter λ is a scaling factor of the strain vector ϵ in the strain space, where the roots of Equation (11) mark precisely those values of λ at which the strain vector ϵ is scaled either positively or negatively just enough to touch

the envelope boundary. When solving (11), up to two roots for λ can be found, which typically appear as a real pair consisting of one positive and one negative root. The objective is always to find the smallest positive real root (λ_{crit}), which can be interpreted as the safety factor of the strain vector ϵ with respect to the second-order envelope [15]. Therefore, using the quadratic formula, a closed-form solution for the positive real root can be derived and expressed as the following conservative strength failure inequality criterion, provided that the IM6 material system is used:

$$\frac{2p_{12}}{-p_{11} + \sqrt{(p_{11})^2 - 4p_{12}p_{10}}} < 1, \quad \epsilon \neq 0 \tag{17}$$

3. IMPLEMENTATION

3.1. pyOptConv

Despite best efforts of the German Aerospace Center to implement the descriptive standard *CPACS* as a method of choice for the exchange of information between aircraft design disciplines, collaboration in flight physics often reduces to the exchange of simulation models. Furthermore, for structural mechanics in aerospace applications, Nastran is still a well-established CSM software. In this context, the python library *pyOptConv* is developed in order to provide an abstraction layer between finite element

topology definitions and structural optimisation models of wing boxes. The target is to establish a flexible method of formulating optimisation models for already composed wing structures. Additionally, the new abstraction layer allows for dynamic specification and combination of various target functions, constraint sets, and material design spaces. This flexibility is achieved by employing an object-oriented inheritance architecture with abstract base classes for every major conversion phase. The process of generating a design model can be separated in to four phases.

Phase One - Input Data Processing

The initial phase processes all available information in the supplied Nastran deck file and aim to generate a blank sheet for further processing. In a first step, the supplied text file is parsed and converted to a tabulated data set, using the in-house software *pyBDF*. This allows for an initial analysis of the supplied state, if pre-defined design model components are to be reused. Consequently, any already defined optimisation related part of the data set is filtered out and a VTK [16] mesh representation of the input data is generated. If necessary, arbitrary supplied mesh functions or grid range filters are applied to extract the target wing. Afterwards, the significant dimensions of the wing are identified using a rotated bounding box. This is then used to temporarily rotate the mesh to a standard right wing position, allowing for a standardised normal vector correlation in order to identify and categorise major assembly groups.

Phase Two - Design Group Definition

pyOptConv provides several methods of combining elements to *design groups*. Although an optimisation model can be generated on a per-element basis, grouping is generally desirable, since it reduces the number of linearly dependent responses and accelerates the optimisation. The standard method is to reuse the grouping of the input data set, improving compatibility with already established processes. Another option is to generate standard *subsection* groups by splitting the meshes of major assembly groups by each other. For example, this causes the meshes of skins to be clipped by every rib and spar. In the third method, these newly generated groups can be clipped further, in order to retain a supplied limit aspect ratio of the element group. A subdivision of subsection may also be realised by clipping with defined stiffeners. The latter option can be beneficial for tasks incorporating stability constraints for individual cells of a stiffened plate. Lastly, a graphical user interface is developed, giving manual control over the grouping process, allowing dynamic combination or subdivision of design fields.

Phase Three - Design Space Definition

Generating the actual optimisation model requires a set of tasks, which may depend on the chosen material. In general, this includes the definition of design variables, the generation of relations between these variables and designed properties and materials, as well as a set of responses and constraints, varying by user choice and ma-

terials. In case of anisotropic materials, *pyOptConv* will define a material coordinate system by aid of a rotated bounding box. This detaches the material orientation from the element definition. For isotropic materials, constraints currently implemented only include stress limits. On the other hand, available anisotropic materials constraints include maximum strain, the Tsai-Wu failure criterion in lamination parameter space, as well as limits to the feasible domain, employing the method of Setoodeh. Independent of the material choice, stability constraints of stiffened cells can be generated.

Phase Four - Integration into Design Task

pyOptConv is developed in such a way that the original input file can be updated to compose an executable optimisation task in MSC Nastran. To achieve this, *pyOptConv* has to compose sets of constraint types and update the header of the input file accordingly. Furthermore, the target function has to be referenced, and additional parameterisation has to be added. Finally, the optimisation model itself is added in the Nastran specific card format and the deck is written to file.

3.2. Integration into Loads and Optimisation Process

Structural optimisation depends on model generation and loads simulation. In the context of this work, the aforementioned tasks are executed by the software *cpacs-MONA* [5], developed at the DLR-Institute of Aeroelasticity. Its integration with the CPACS standard enables its deployment in multidisciplinary workflows, e.g. exchanging information with aerodynamic or conceptual design. *cpacs-MONA* reads geometrical data, topological definitions or mass breakdowns. Based on the supplied information it calculates preliminary loads with conceptual methods, generates distributed mass models, composes necessary envelopes and simulation models and defines load cases used in loads simulation. Moreover, it may perform a preliminary cross-section sizing (PCS) before executing elastic loads calculations and structural optimisation in a loop until convergence. Finally, a flutter check is performed and the *cpacs-dataset* is updated.

This process is now complemented by two additional plugins. *pyOptConv* is first called when the *cpacs-MONA* structural optimisation deck is written. Only at this stage are all necessary information for updating the deck available. At this point it is possible to integrate *pyOptConv* as a plugin, without relying on additional information of the *cpacs-MONA* framework. However, this now results in inconsistencies between the portion of the structural model processed by the converter and that referenced by the main framework. To maintain consistency, the converter class is stored for later use and simulation composition calls are expanded by deck update hooks, which exchange components of the datasets not part of the optimisation model, such as elements, properties, coordinate systems, and materials. Furthermore, in case of symmetric wings *cpacs-MONA* will only run an optimisation task for a single wing. Subsequently, it becomes necessary to mirror changes to elements, added coordinate systems,

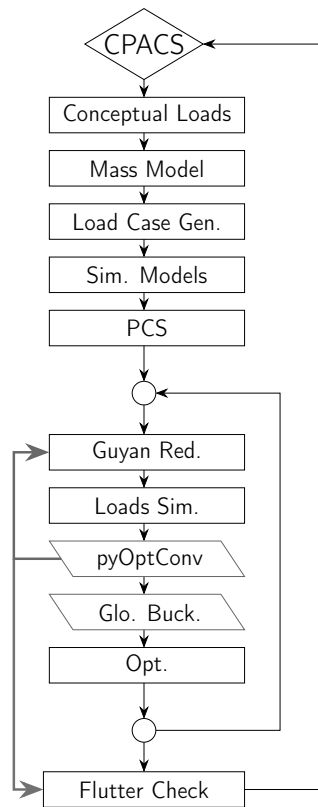


FIG 4. Simplified flow chart of adapted cpacs-MONA process

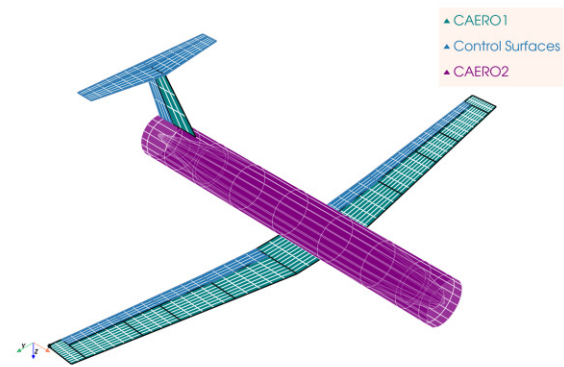
as well as adapted properties and materials, while running an extensive update of references. These hooks are then executed every time a simulation wrapper is called.

The integration of pyOptConv might be restructured in such a way that the converter is only called once after the PCS. However, if the plugin nature of the integration is to be preserved, this will limit the generation of the design space to the input specification of cpacs-MONA.

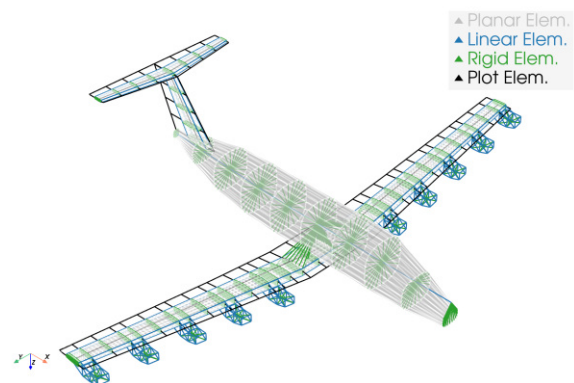
Another addition to cpacs-MONA is the global buckling plugin. Similar to pyOptConv, it constitutes a deck converter, which adds a global buckling analysis subcase and respective constraints for every linear static load case present in the deck. In contrast to previous publications on this configuration [17], the global buckling analysis is elevated from a post-processing step to an integral component of the optimisation model.

4. SIMULATION MODEL

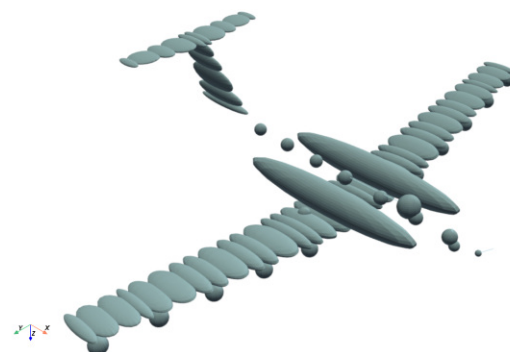
All simulations and the structural optimisation are carried out in the CSM-software MSC Nastran. Thus, the simulation model components, depicted in Fig. 5, reflect this context. The aerodynamic model (Fig. 5a) features a panel discretisation suitable for use together with the vortex lattice method. It features normal vector corrections for camber and twist. All significant lifting surfaces are present, and all control surfaces are addressable for trimming and manoeuvring. The fuselage is represented by a slender body. This body features additional formulation from potential theory and was found to improve the calcu-



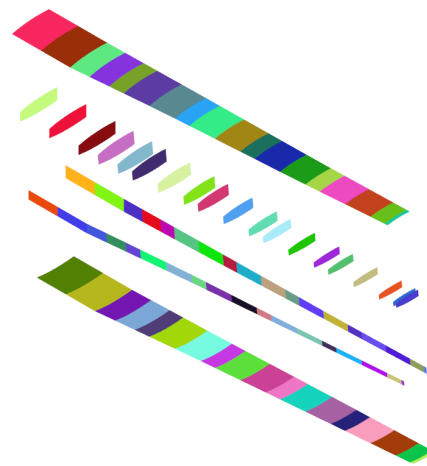
(a) Panel aerodynamics



(b) Structure



(c) Mass



(d) Design fields right wing box

FIG 5. Aeroelastic model components

lation of the pitching moment. The aerodynamic impact of the engine pods, as well as the propeller aerodynamics are neglected in the context of this study.

The structural model (Fig. 5b) remains largely unchanged from the studies presented at the German Aerospace Congress (DLRK) 2025 [17]. All load carrying structures of the primary lifting surfaces are represented as wing boxes consisting of skins, spars, and ribs. Inertia loads of the engine pods are introduced via a gondola comprised of rod elements. One key difference to the work presented in 2025 is the removal of plate stiffeners for ribs, spars, and skins. Since these elements are typically not part of the optimisation process in cpacs-MONA, they have been removed in favour of an increased design space of the optimised plate elements. Only the spar caps remain in the model, while being reduced to a minimum. The rest of the topology is the result of the studies presented in 2025.

cpacs-MONA is capable of distributing mass data provided via the cpacs-dataset along key assembly groups. Fig. 5c depicts a reconstruction of the inertia tensors of discrete mass points as ellipsoids. The bodies next to the fuselage represent the large hydrogen tanks. Mass data provided without inertia information, such as systems or pods masses, are visualized as spheres with the density of aluminium. The distributed masses and the translation of inertia tensors are significant for the fidelity of the elastic load calculations. The design groups of the right wing are visualised in Fig. 5d. Essentially, this represents the subsection standard discussed in section 3.2. Significant envelopes, such as mass and flight envelopes are depicted in Fig. 13 in the appendix.

5. APPLICATION

This section compares the results of two test applications of the extended loads and optimisation chain. Both test cases differ only in the utilised materials and consequently in the design model. The first test case composes an optimisation model for isotropic materials, utilising the properties of aluminium 2024-T3, a common alloy for airframe structures. In this test case, the design is constrained by strength limits and global buckling failure modes. The second test case constructs an optimisation model for anisotropic materials. In this context, the properties of IM6 are used together with Setoodeh and Tsai-Wu constraints. Global buckling failure modes are computed as well. Furthermore, the lamination parameter model is limited to describing symmetric balanced laminates. This reduces the number of design variables for the material from twelve to four per design group. The properties of both materials are listed in Tab. 2.

Despite the employed materials and the chosen set of constraints, the rest of the models and setup remains unchanged. Both test cases use the same topology. The set of load cases comprise static manoeuvre calculations, such as static pull-up and push-down, accelerated rolling and yawing, as well as Pratt-gusts, carried out for every mass case on the whole flight envelope. These manoeuvres are

complemented by quasi-static simulations for landing and taxiing loads. Both test cases are carried out in a loads and optimisation loop until convergence for mass, inertia tensors, and loads. The target function of the structural optimisation seeks to minimise the mass of the primary structure.

While the aluminium test case provides a reference, the focus of this analysis is on the lamination parameter optimisation.

5.1. Aluminium Reference

results of the structural optimisation of the aluminium reference case remain largely unchanged from the results published in [17]. However, since large portions of the stiffening bar elements were removed, the study is recalculated. In accordance with the results from 2025, the model is entirely limited by stability constraints. This is to be expected, since the removal of the stiffening elements further emphasises this tendency. The structural mass of the assembly converges to 204.82 kg, which is almost exactly the result of the analysis of the model featuring a comprehensive set of plate stiffeners. Fig. 6 depicts the plate thickness of the right wing box in an explosion plot. The upper skin features a distinct peak of 7.5 mm at the

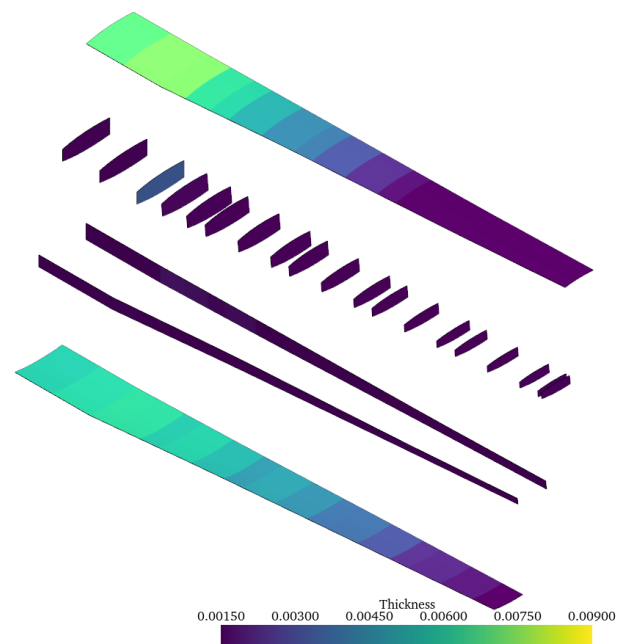


FIG 6. Explosion plot of the right wing box, isotropic optimisation

end of the inner wing box, before gradually decreasing to the minimum of 1.5 mm at $\eta = 60\%$. The lower skin peaks at the same inboard position at a value of 7 mm. However, the minimum of 1.5 mm is reached further outboard at $\eta = 80\%$. The thickness distribution reflects the shift of mass from stiffeners to upper and lower skins, since the thickness distribution of the skins in the analysis of 2025 reached a limit of 5 mm. Although the design space for the structural optimisation is increased by re-

Property	Carbon/Epoxy IM6G/3501-6, [18]	Aluminium 2024 T3, [19]
Density (ρ , g/cm ³)	1.62	2.77
Young's modulus (E , GPa)	169.0, 9.0	72.4
Shear modulus in-plane (G , GPa)	6.5	28.0
Poisson's ratio in-plane (ν)	0.31	0.33
Tensile strength (X_t , Y_t , MPa)	2240.0, 46.0	485.0
Compressive strength (X_c , Y_c , MPa)	1680.0, 215.0	485.0
In-plane shear strength (S , MPa)	73.0	285.0

TAB 2. Material properties, multiple values indicate respective properties aligned with, as well as transverse to the fibre direction

moving the static mass of the stiffeners, no reduction in total mass can be observed.

5.2. Lamination Parameter Optimisation

The thickness distribution of the lamination parameter optimisation, grouped by major assembly groups and plotted over the normalised half span, is depicted in Fig. 7. This diagram illustrates major differences to the aluminium reference calculation. Firstly, pyOptConv

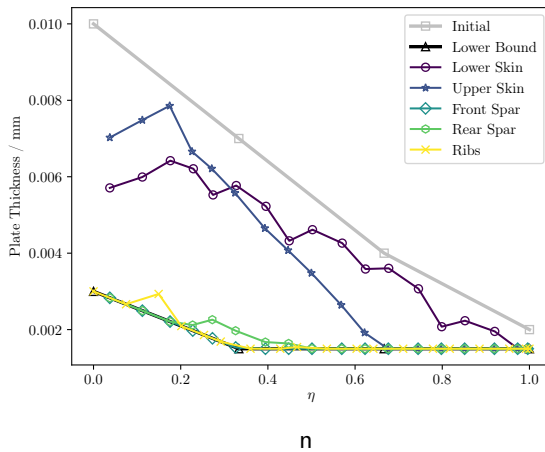


FIG 7. Thickness of planar elements over normalised span η

allows specifying minimal and initial thickness distributions as a linear function over the normalised half span. In case of the lamination parameter optimisation, the minimal thickness at the wing root has to be increased from 1.5 mm to 3.0 mm. The modal approach used for the global buckling analysis together with the nonlinear correlations in the lamination parameter optimisation model caused instabilities at the wing root. Setting the minimal thickness in such a way does not cause a penalty for the final structural mass, since the mass is redistributed from the skins to the spars.

Secondly, the peak in the plate thickness of the upper skin increases from 7.5 mm to 8 mm. Lastly, in contrast to the aluminium test case, the lamination parameter model exhibits a wavy distribution for the lower skin. These waves correlate with the positioning of the heavy engine Pods, causing relief in torsional moment (see Fig. 19, appendix).

Consistent with the aluminium model, the thickness of spars and ribs largely reach the defined minimum. Since these assembly groups feature a smaller geometrical moment of inertia relevant for the simulated load conditions, the sensitivity for fulfilling the specified constraints is generally lower for spars and ribs than for skins. In the structural optimisation this often leads to thin spars and ribs and respectively thick skins.

The convergence history of the lamination parameter loads and optimisation loop is visualised in Fig. 8. Despite the nonlinear nature of the global buckling analysis and lamination parameter optimisation model, structural mass converges nicely. Load updates, indicated by grey vertical lines, reset the structural optimisation and cause small disturbances, especially in the maximum constraint value. The latter value represents the maximum normalised constraint in one design cycle. The constraints are normalised with respect to the bound in such way that the constraint function goes from minus infinity to zero. Everything above zero is considered violated. In this context the final constraint violation amounts to 1.18 %, which is deemed acceptable considering a safety margin of 50 % is incorporated into every failure criterion. Similar disturbances caused by an update of the load data, may be observed in the convergence trends of design variables related to plate thickness and lamination parameters in Fig. 14 and Fig. 15, respectively. The mass of the optimised part of the primary wing box structure converges to 104.76 kg. Together with 20.36 kg for the aluminium spar stiffeners, the overall weight of the structure amounts to 125.18 kg. In comparison to the 204.82 kg for the full aluminium test case, this results in a reduction of 38.73% in primary structural mass. However, this result is attained without any further specification of manufacturing constraints. The necessity of the latter are illustrated in Fig. 9, which depicts the thickness of the assembly groups together with the resulting normalised membrane and bending stiffness per design group.

Especially the spars feature immense changes in stiffness from one design group to the other. Inter-design-group constraints, such as blending constraints [20–24] limiting the change in stiffness, or adjacency-constraints [25, 26] restricting the change in plate thickness, are currently

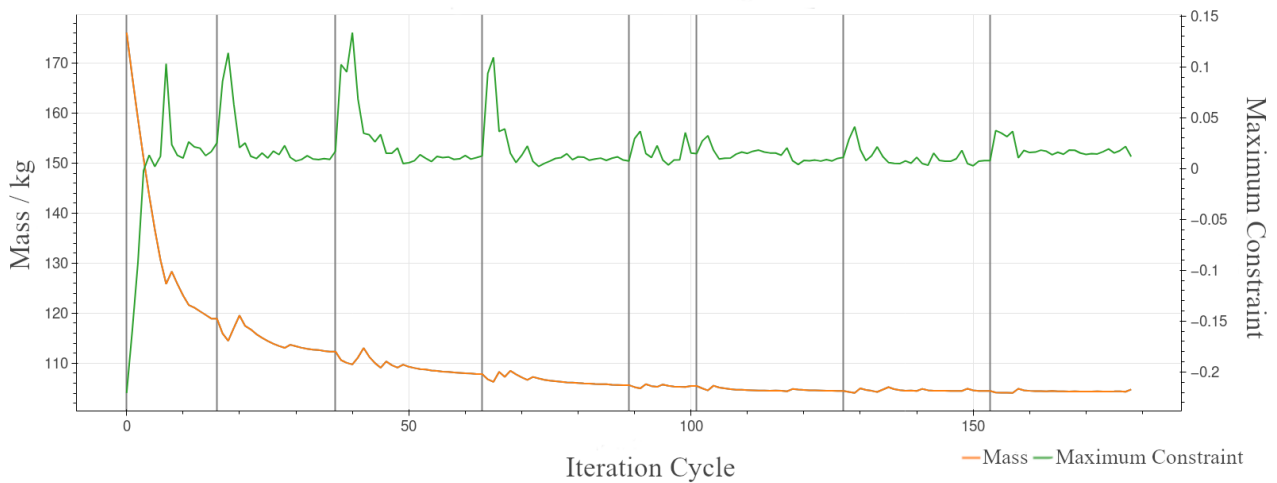


FIG 8. Optimisation convergence. mass, objective, and maximum constraint value over iteration cycle

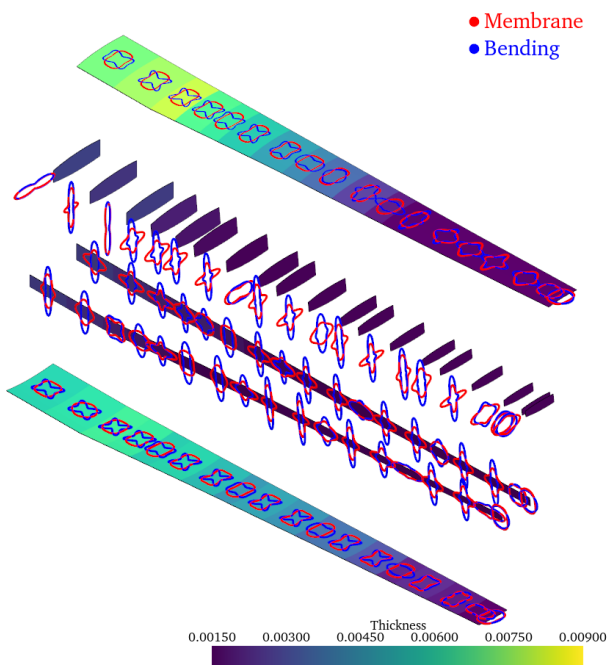


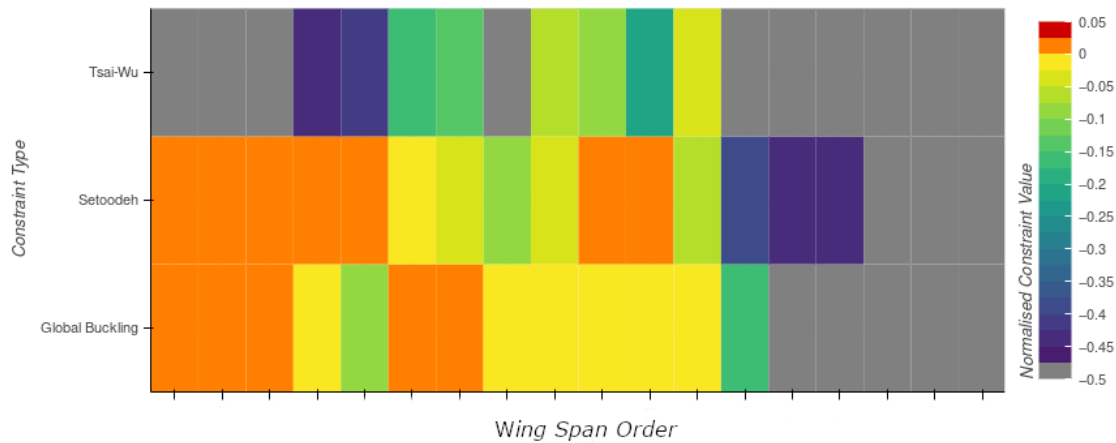
FIG 9. Explosion of right wing box, thickness and normalised membrane and bending stiffness

being integrated into pyOptConv. These constraints will support the reconstruction of discrete stacking sequences. However, the structural mass is expected to increase as a consequence.

The resulting stiffness distribution strongly correlates with the type of active constraints. Fig. 10 illustrates the maximum constraint per type and design field, ordered by increasing wing span from left to right. Both skins are largely characterised by global buckling failure modes. Additionally, the Tsai-Wu criterion is active in the wing centre of the upper skin, and at the wing root at the lower wing. Moreover, the design space of the lamination parameters is widely exhausted, indicated by the Setoodeh constraint values. The focus on stability failure modes manifests in the increased bending stiffness in $\pm 45^\circ$ di-

rections, while the membrane stiffness exhibits a slightly ovalised shape. The respective breakdowns for the other assembly groups are added in Fig. 16. However, the correlation between stiffness distribution and significant constraints is less distinct, since many of the design groups reach minimal thickness, and few of these groups actually have limiting failure constraints. Consequently, these design groups may support other assembly groups. Moreover, lamination parameters do not feature a direct sensitivity with respect to structural mass, but impact the design in an indirect manner via the structural constraints. If a design group lacks limiting constraints, the stiffness distribution can remain at the current state. Thus, some stiffness distributions may originate from a previous load or design cycle, in which the design group featured active failure modes. An exemplary design history of a stiffness distribution is illustrated by Fig. 17 for rib number seven, highlighting the dynamic change in the design throughout optimisation and load update.

To complete the analysis, several stability metrics are checked. Fig. 20 and Fig. 21 illustrate aileron effectivity and pitch stability respectively. Both metrics are calculated for the elastic aircraft, and for each combination of flight condition and mass case. Both figures indicate a broad reserve in the respective stability margins. Especially the pitch stability margin is always greater than $30\%_{mac}$. Such a large stability margin may also indicate a poor manoeuvrability. However, this has to be addressed in overall aircraft design. Lastly, Fig. 22 exemplary depicts the results of an unmatched flutter analysis for the MOEM (mOOee) mass case at a Mach number of 0.3 and an altitude of 2500 m. Neither for this specific mass case, nor for any of the for others of the considered mass configurations does the design exhibit tendencies for instability within the flight envelope. Nonetheless, it has to be mentioned, that the vortex lattice method in the form in which it is implemented in MSC Nastran is not capable of capturing T-tail flutter. Furthermore, the lack of propeller aerodynamics and the modelling of their interaction with the wing prevent capturing related instabilities such as whirl flutter.



(a) Upper skin



(b) Lower skin

FIG 10. Constraint heatmap, constraint types over design fields in wing span order

6. CONCLUSION AND OUTLOOK

The presented work covers the structural optimisation process in the DLR project D-LIGHT. The implementation of the optimisation converter *pyOptConv* represents a significant step towards the automated generation of CFRP optimisation models for wing structures. Throughout this work, it has been presented how a wing box structure can be segregated and complemented with the necessary components to describe anisotropic materials using laminations parameters. These models are completed with the formulation of several constraints types, such as Tsai-Wu in lamination parameter space, or compatibility constraints after Setoodeh. Furthermore, the integration into the loads and optimisation software cpacs-MONA was discussed. Finally, the application of the process on the D-LIGHT configuration was presented and analysed. The comparison to an aluminium reference test case revealed that an optimised anisotropic structure for the same topology can save 38.73 % in primary structural mass. The analysis of the optimisation results depicts a

design largely dimensioned by global buckling constraints, which is reflected by the anisotropic stiffness distribution.

On the other hand, these stiffness distributions highlight the necessity for of implementing blending constraints, since the abrupt change in stiffness from one design group to another impedes reconstruction of a discrete stacking sequence for manufacturing.

For full aeroelastic tailoring capability, *pyOptConv* will be paired with a loads and optimisation framework, featuring sensitivity calculations from elastic load simulations [27]. In combination with unbalanced laminates, this will enable specific designs targeting aeroelastic characteristics and applications, such as passive load alleviation or highly flexible designs.

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A. APPENDIX

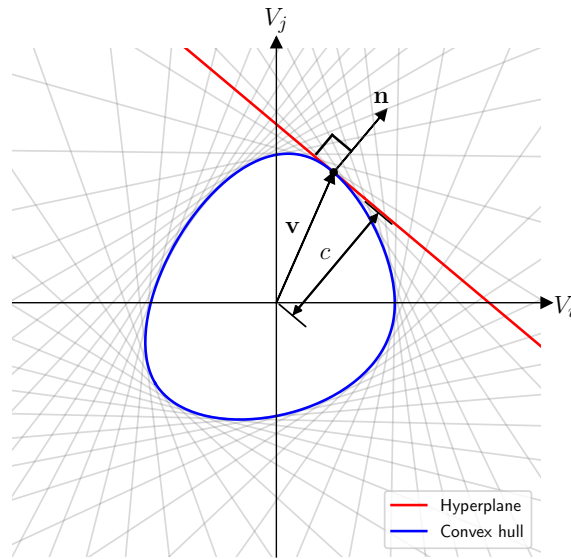


FIG 11. Conceptual illustration of the hyperplane approximation of the feasible domain

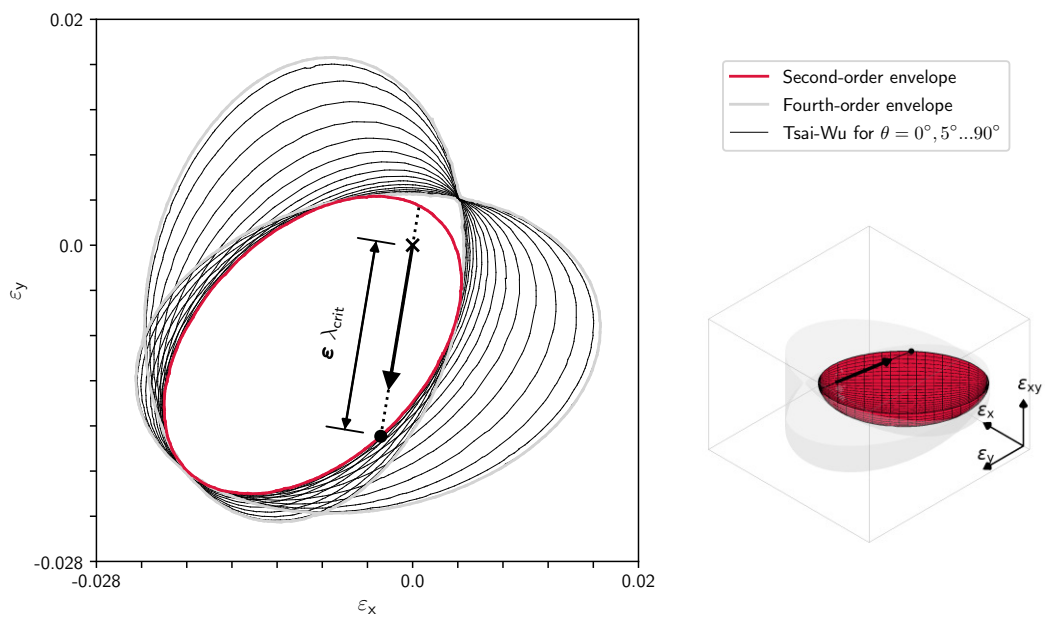
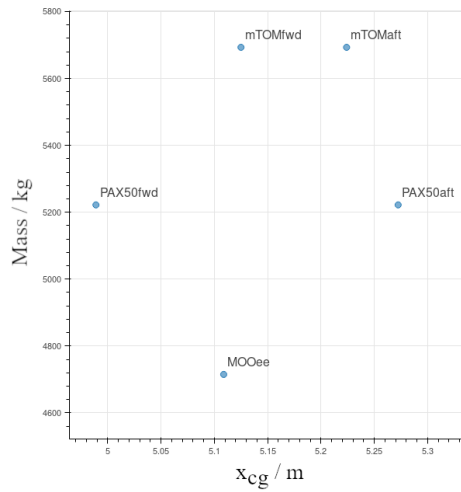
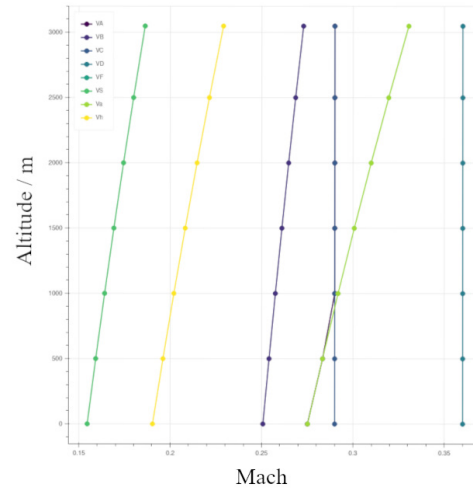


FIG 12. Critical second-order envelope according to Ijsselmuiden [15] (IM6 Table 2).



(a) Mass envelope



(b) Flight envelope

FIG 13. Significant envelopes of D-LIGHT configuration

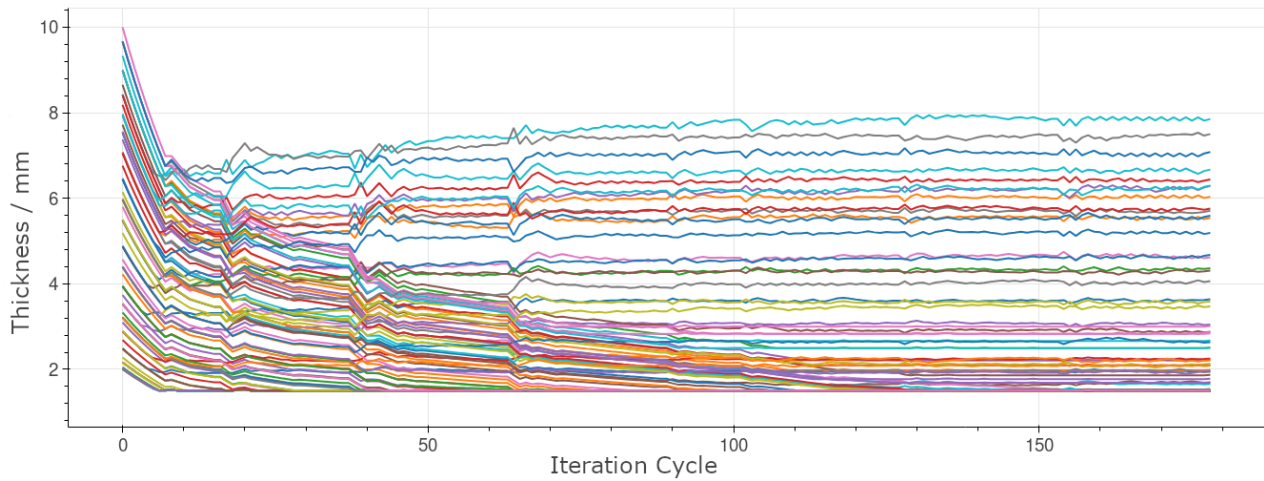


FIG 14. Trends of thickness design variables over optimisation cycle, lamination parameter optimisation

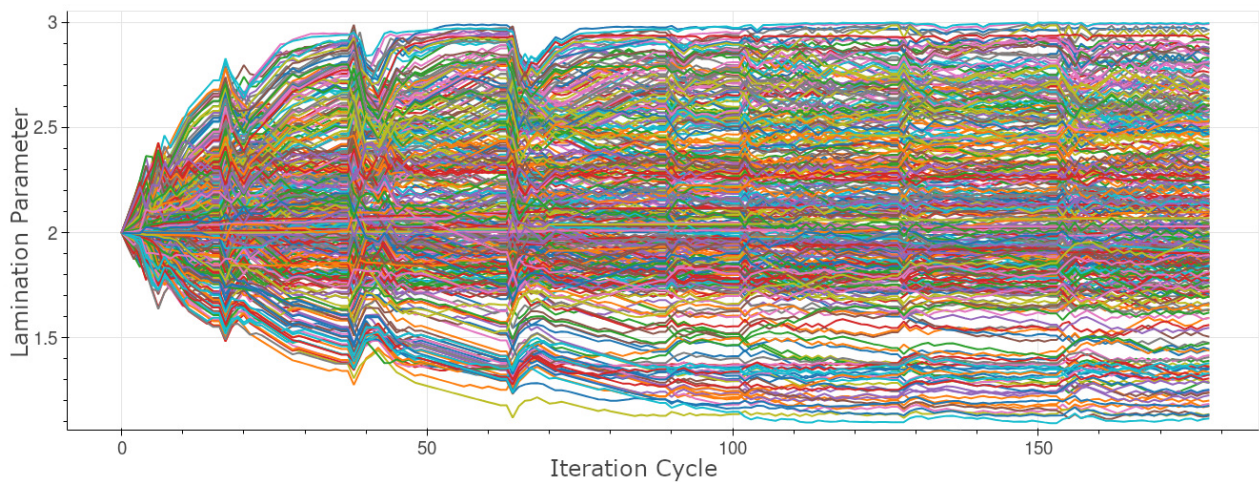
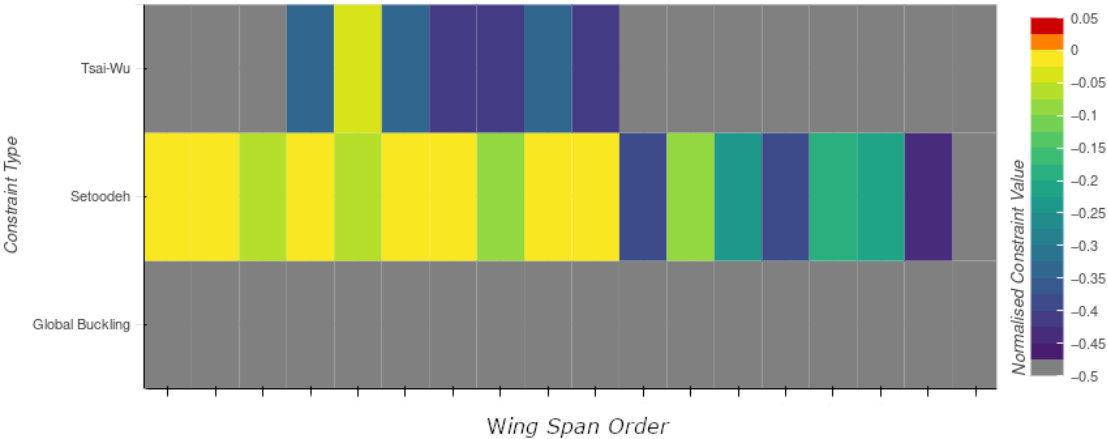


FIG 15. Trends of lamination parameter design variables over optimisation cycle, lamination parameter optimisation



(a) Front spar



(b) Rear spar



(c) Ribs

FIG 16. Constraint heatmap, constraint types over design fields in wing span order

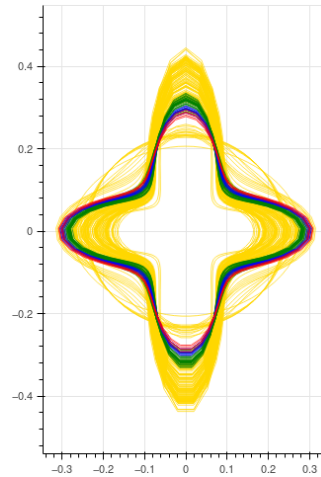


FIG 17. Optimisation History of membrane stiffness, rib 7

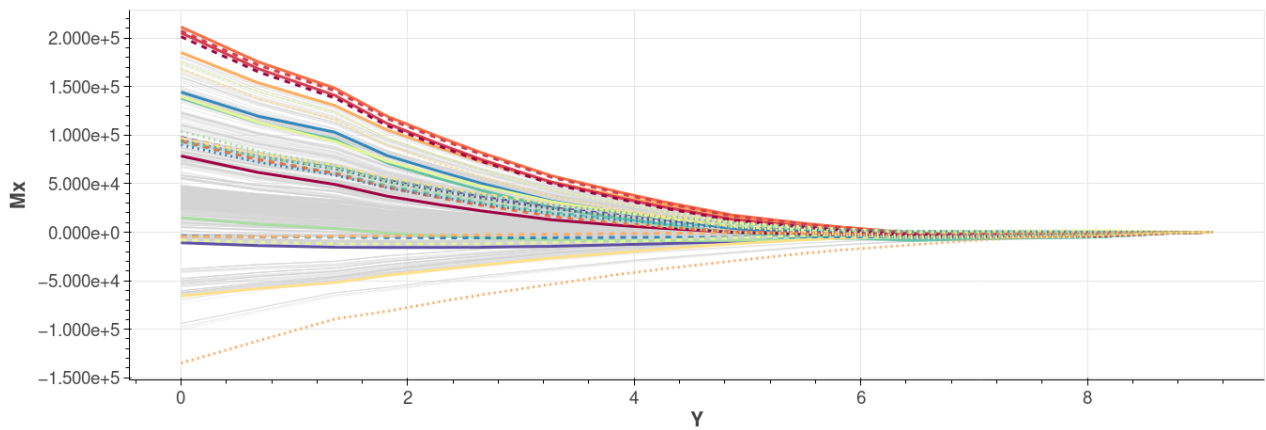


FIG 18. Wing bending moment w.r.t. half span

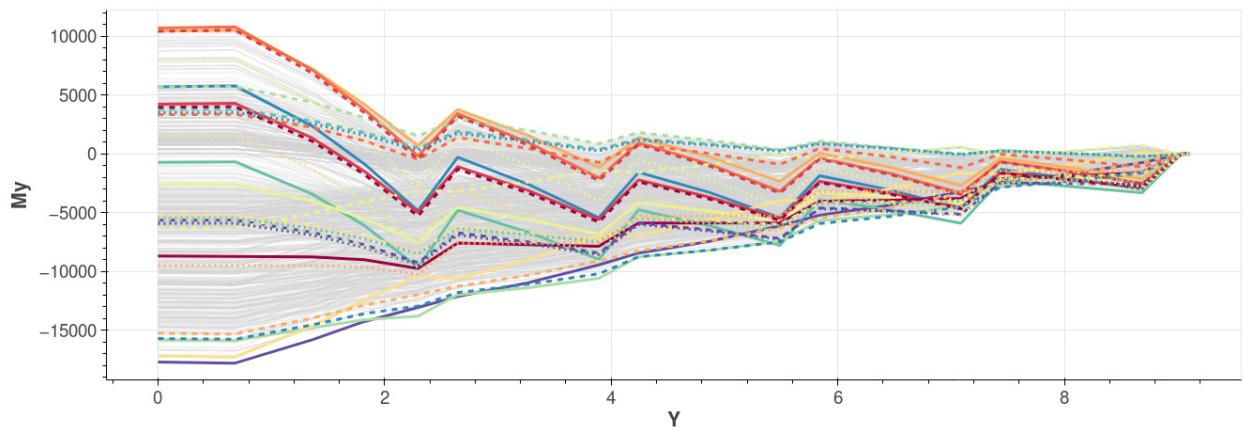


FIG 19. Wing torsion moment w.r.t. to half span, lamination parameter optimisation

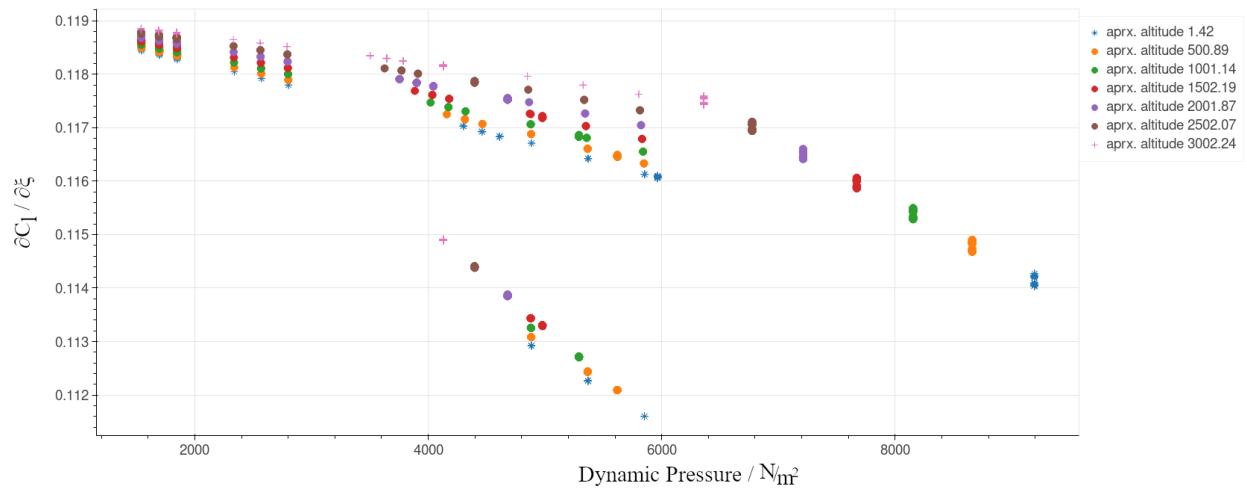


FIG 20. Rolling moment derivative w.r.t. aileron deflection, lamination parameter optimisation

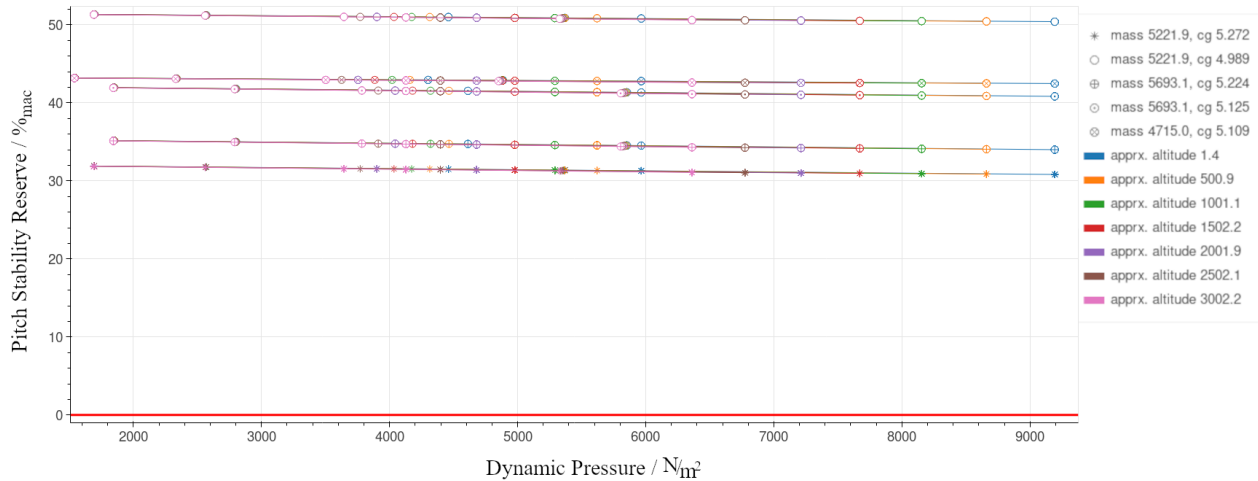


FIG 21. Pitch stability reserve, lamination parameter optimisation

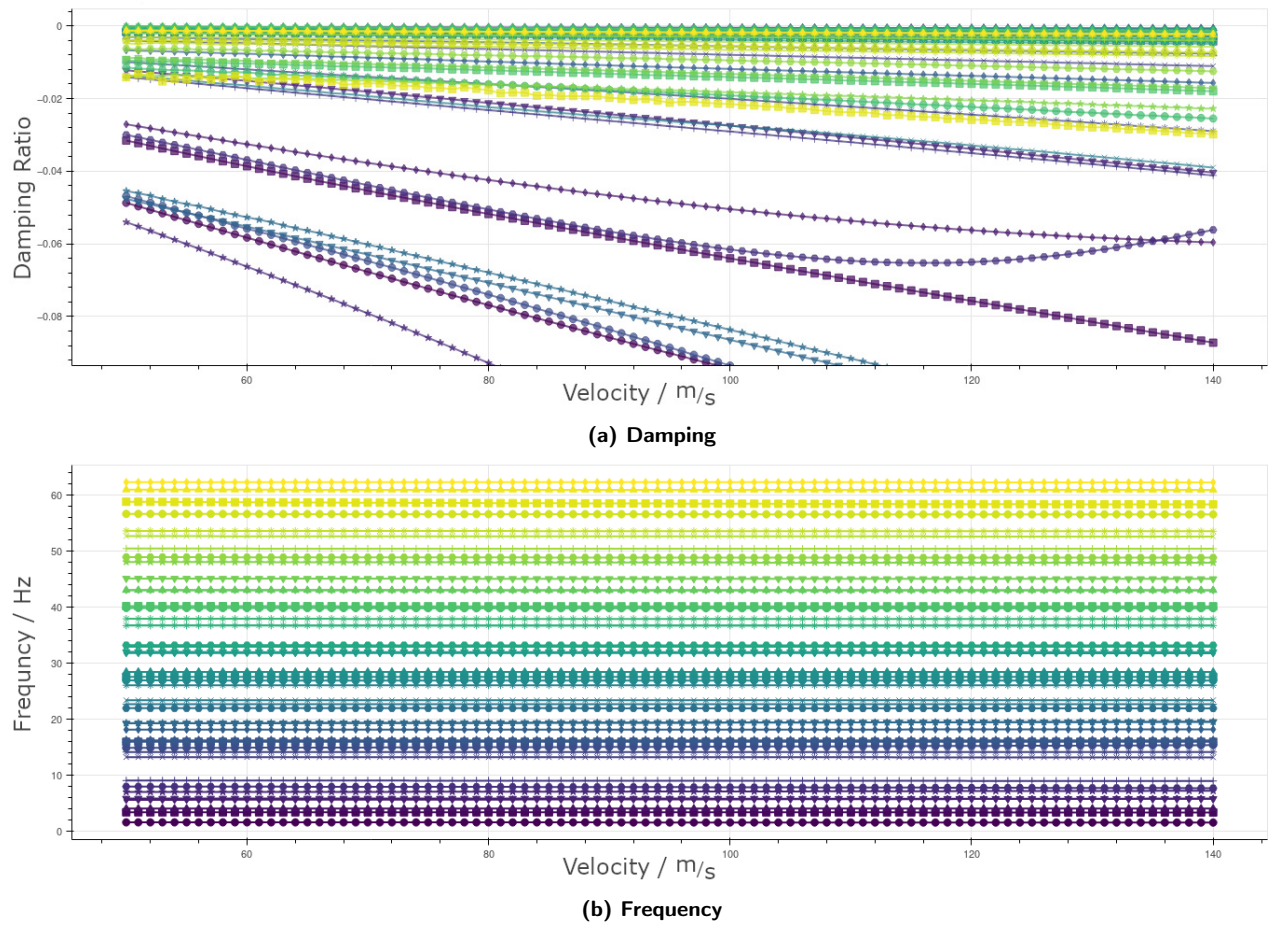


FIG 22. Flutter check, mTOMaft mass case, lamination parameter optimisation