

Buckling of a composite wingbox to achieve passive gust load alleviation

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ABSTRACT

Next-generation aircraft increasingly rely on slender, high aspect-ratio wings that improve aerodynamic efficiency but are more flexible and develop higher loads at the wing root during gusts. This study presents a design framework that exploits the local buckling of a composite wingbox skin as a passive gust load alleviation mechanism, using the post-buckling stiffness reduction to lower the peak Wing Root Bending Moment (WRBM). The framework couples two analyses so that this benefit can be assessed in early design without increased computational cost. First, the localized non-linearity induced by buckling is introduced into a linear aeroelastic stick model as a bilinear rotational stiffness, applied through a scalar point whose feedback moment reproduces the pre- and post-buckling behavior. A parametric study on this model identifies the spanwise location, critical buckling moment, and post-buckling stiffness reduction that maximize the WRBM reduction, indicating that up to 12% is achievable when buckling is induced near the wing root. Second, a three-dimensional finite element model of the wingbox is designed to buckle accordingly and to verify its structural integrity under the critical gust load. The resulting configuration buckles at 12.5% of the gust load with an average post-buckling stiffness reduction of 10%, yielding a 1.41% WRBM reduction and a 1.7% mass reduction. These results show that structural integrity constraints limit the load alleviation attainable in practice, confirming that peak gust loads can be reduced passively while the mass of the aircraft is decreased simultaneously.

1. Introduction

The aviation industry is under increasing pressure to design more efficient aircraft that consume less fuel, emit less CO₂, and operate at lower cost. Meeting these targets ultimately depends on reducing structural weight while preserving the capacity to withstand demanding aerodynamic loads, including those generated during gust encounters. This challenge is particularly critical for next-generation aircraft, which increasingly rely on slender, high aspect-ratio wings that improve aerodynamic efficiency but, at the same time, exhibit greater structural flexibility and elevated stress levels at the wing root. As a result, engineers must find a balance between creating lightweight structures and structures that are robust under critical dynamic loads.

Composite materials have become a key enabler of weight reduction in modern aircraft [1]. Among composite structures, stiffened panels exhibit a particularly valuable feature as they are able to sustain loads well beyond the initial skin buckling load through load redistribution in the post-buckling regime [2,3]. The buckling response of such panels is governed not only by the laminate but also by the stiffening layout. Kalhori et al. [4] analyzed stiffened composite plates and showed that the number, spacing, and cross-sectional dimensions of the stiffeners control the critical buckling load, reporting an increase of 41.8% in one

configuration for a mass penalty of only 10.3%. Research on composite stiffened panels has further shown that the force vs displacement curve in the post-buckling regime can be approximated as linear, with a lower slope, i.e., reduced stiffness, than in the pre-buckling regime, supporting a bilinear characterization of the panel response despite the high out-of-plane displacements registered in the post-buckling regime [5]. This reduced post-buckling stiffness can be exploited to modify aeroelastic responses and to induce beneficial geometric changes, leading to lighter structures and more efficient aircraft. Recent research on lightweight aerospace structures has therefore highlighted the use of buckling and post-buckling behavior of composite panels as a design feature rather than as a failure mode. This use of buckling by design is already established outside aeronautics, in components such as thermostats, micro-switches, and crash energy absorbers. Asemi and Shariyat [6] traced the post-buckling equilibrium paths of heterogeneous plates, showing that the stiffness retained beyond the onset of buckling depends on the material and boundary conditions and can therefore be tailored.

The deformation capacity offered by the post-buckling regime has been investigated in a range of composite concepts aimed at enabling flexible, shape-adaptive structures for next-generation aircraft. Arrieta et al. [7] induced the snap-through of bi-stable composite panels through dynamic excitation, simplifying the actuation of

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Nomenclature

Abbreviations

DOF	Degree of Freedom
FE	Finite Element
FI	Failure Index
GLA	Gust Load Alleviation
NABUCCO	New Adaptive BUCKling-driven COMposite aerospace structures
NeoCASS	Next generation Conceptual Aero Structural Sizing
NeOPT	NeoCASS Optimization Toolbox
SPOINT	Scalar point
WRBM	Wing Root Bending Moment

Latin Symbols

A	State matrix
B	Input matrix
C	Output matrix
D	Direct transmission matrix
E_1	Longitudinal Young's modulus
E_2	Transverse Young's modulus
M	Generic moment
$\hat{M}$	Buckling moment
G_{12}	In-plane shear modulus
H	Gust gradient
k_{pre}	Pre-buckling stiffness
k_{post}	Post-buckling stiffness
S_L	Maximum shear stress
U_1, U_2, U_3	Nodal displacements in the 1–2–3 directions
$\mathbf{u}_e$	External force input vector
$\mathbf{u}_g$	Gust input vector
v_g	Gust velocity input
X_C	Maximum longitudinal compressive stress
X_T	Maximum longitudinal tensile stress
$\mathbf{x}_{\text{ae}}$	Mechanical and aerodynamic state vector
$\theta_{A,B; i,j}$	Rotation of nodes A,B in the i, j directions
θ_{SPOINT}	Scalar point rotation
$\hat{\theta}$	Buckling rotation
Y_C	Maximum transverse compressive stress
Y_T	Maximum transverse tensile stress
$\mathbf{y}$	Output vector
y	Spanwise position

Greek Symbols

Δ_z	Vertical displacement
λ	Buckling eigenvalue
ν_{12}	Major Poisson's ratio
ρ	Mass density

morphing wings. Runkel et al. [8] introduced a shape-adaptation technique in which buckling at the rear shear web of the wingbox induces spanwise twisting deformations under specific aerodynamic loads. Nicassio et al. [9] exploited the "lever effect" of a bi-stable composite plate to develop a passive, lightweight morphing surface, integrable in an aircraft flaperon and passively activated by the pressure differential at a given altitude. Zhang and Bisagni [10] have shown that buckling of the rear spar of the wingbox results in a reduction of the torsional stiffness as well as a shift of the shear center toward the front spar, allowing for buckling-driven mechanisms to control wing twisting. More recently, Stammel and Bisagni [11] analyzed the pre- and post-buckling regimes of a composite wingbox under torsion, concluding that the post-buckling stiffness reduction induced by load redistribution can be advantageous for shape-adaptive aerospace structures. These and related efforts on bistable composite laminates for aerospace applications have been surveyed by Lemos et al. [12].

Recent studies on high aspect-ratio wings have demonstrated that aerodynamic drag can be reduced while improving the lift-to-drag ratio and range, thus increasing aircraft efficiency [13]. However, slender high aspect-ratio wings are inherently more flexible and develop higher stress levels at the wing root, introducing aeroelastic challenges related to gust-induced dynamic loads [14]. Mitigating these loads is the objective of Gust Load Alleviation (GLA) techniques, with both active and passive strategies having been proposed.

Passive GLA methods can reduce peak gust loads without additional active components by exploiting the displacement capability of composite flexible wings in the post-buckling regime, and have been shown to be effective in reducing the Wing Root Bending Moment (WRBM) in gust responses [15]. Several approaches leverage the high deformation capacity of bi-stable composite panels in the post-buckling regime [16]. Cooper et al. [17] designed a morphing wingtip using chiral-type internal structures that allow cant angle, camber, and twist changes across the flight envelope. Castrichini et al. [18] connected a folding wingtip to the wing through a piecewise-linear hinge spring, whose stiffness and threshold moment were selected so that the wingtip is released only under gust and does not deflect in cruise. Hahn and Haupt [19] explored the effect of non-linear wing behavior to trigger downward twist and camber line or profile change, and Hahn et al. [20] proposed a load-alleviation concept featuring a component inside the wing that buckles at a critical load, resulting in a reduction of the effective angle of attack and lift. Amoozgar et al. [21] attached a non-linear energy sink, consisting of a concentrated mass, a damper, and a non-linear spring, to a two-degree-of-freedom aeroelastic model of a wing subjected to gust and aerodynamic loads, and assessed its capability to passively absorb the energy introduced by the gust. In contrast to passive GLA approaches, active load-alleviation methods generally entail additional weight, power requirements, and system complexity associated with sensors, actuators, and control logic. Various active GLA approaches have been developed over the years, and a comprehensive overview is provided by Burgstaller and Galfy [22], covering developments from the earliest concepts in aeronautics to the most recent advances.

As the post-buckling behavior requires accurate simulation of the post-buckling regime, which typically demands non-linear analyses, incorporating this phenomenon into the early stages of aircraft design remains a challenge. The aeronautical industry must evaluate thousands of load cases to encompass the full flight envelope of an aircraft, which rules out the use of high-fidelity non-linear aerostructural models because of their high computational cost. It is nonetheless in these early stages that the benefit of load alleviation is best captured, as shown by Handojo [23], who embedded load alleviation into an aircraft pre-design process, coupling loads analysis, load alleviation, structural optimization, and fatigue analysis obtaining a reduction of the wingbox mass as well as an improvement in the fatigue life of the aircraft. Such an assessment is only feasible with computationally inexpensive tools. For this reason, dynamic aeroelastic responses are typically computed using simplified, linear aeroelastic analysis tools applied to so-called stick models, in which the structure is represented by a one-dimensional finite element model of beams and concentrated masses, and the aerodynamics is modeled with panel methods such as the Doublet-Lattice Method [24]. Standard tools used in the conceptual and preliminary design phases of aircraft are not suitable for analysis if the intent is to exploit non-linear phenomena such as the post-buckling regime of composite panels. Therefore, these phenomena are incorporated by resorting to more expensive, higher-fidelity simulations.

Beyond dynamic aeroelastic analysis, the inadequacy of linearized analysis for slender wings has also been examined in the context of detailed structural optimization. Mitrotta et al. [25] replaced the conventional linear buckling constraint with a non-linear structural stability constraint based on the positive-definiteness of the tangent stiffness matrix, allowing for a reduction of mass of an idealized composite wingbox, sized closer to its true load-carrying capacity. A subsequent study on a benchmark transonic wing [26] further showed that linear buck-

ling analysis is not always conservative. These works confirm that the non-linear buckling behavior of slender wings must be accounted for during design, yet they treat instability as a phenomenon to be avoided rather than exploited.

Rather than avoiding this instability, it can be deliberately induced and turned into a design advantage. It is within this context that the ERC Advanced Grant Project NABUCCO (New Adaptive BUCKling-driven COMposite aerospace structures) [27,28] pursues the development of adaptive buckling-driven structural concepts for the next generation of aircraft. It aims to control structural responses under critical conditions and to enable shape variation through the non-linear stiffness redistribution induced by the post-buckling regime. One of the project's targeted applications is the reduction of peak gust loads, enabling lighter wing designs and lower WRBM values. This work builds upon previous preliminary studies [29,30] where the aeroelastic and the structural sides of the problem were addressed separately: the sensitivity of the gust response to a concentrated non-linearity was mapped without a wingbox designed to reproduce it, and the wingbox was analyzed without the resulting load alleviation being computed. The current work presents the methodology and the procedure of calculating the WRBM reduction through a dynamic gust analysis on the stick model, showing that the configuration buckles under gust and not in trimmed cruise.

This paper presents a design framework that combines two analyses to reduce peak gust loads by exploiting the buckling of the skin of a composite wingbox. The first is a linear aeroelastic stick model, into which the local non-linearities induced by the post-buckling behavior are embedded, allowing the load-alleviation potential to be assessed already in the early design phases without compromising computational time. The second is a non-linear structural analysis of the wingbox, carried out in a commercial finite element software, used to create a wingbox which buckles under critical gust loads and to verify its structural integrity. The methodology is then demonstrated on a case study representative of a high aspect-ratio composite wing.

2. Methodology

Thousands of load cases have to be taken into account to properly encompass the whole flight envelope of an aircraft to analyze gust conditions. As such, employing high-fidelity non-linear aerostructural models is not always feasible as the computational cost would be too high.

Thus, to determine the complete dynamic load condition envelope, the aeronautical industry bases these calculations on simplified aeroelastic models. Such models are representative of the stiffness and mass distribution of the aircraft and allow for accurate dynamic load computation. The structure is usually represented through one dimensional Finite Element (FE) models implementing beams and concentrated masses, while the aerodynamics is represented by a panel method, like the Doublet-Lattice Method [24]. The coupling between the structure and the aerodynamics is realized with a set of splines, which exchange structural motion and aerodynamic forces between the two methods. These models are known as stick models.

In the present study, the Next generation Conceptual Aero Structural Sizing (NeoCASS) [31,32] software is used to develop the aircraft stick model and to perform the aeroelastic analyses. NeoCASS includes an automated identification procedure for unsteady generalized aerodynamic forces obtained via the Doublet-Lattice Method, which are calculated using the rational function approximation technique [33]. The rational function approximation provides a state-space, time-domain representation of the aerodynamic forces, which is a necessary prerequisite for coupling the aeroelastic model with a time-domain feedback force controller. In particular, the time-domain formulation allows for a simpler implementation of control laws and the introduction of non-linear effects that would be more difficult to work with if these were implemented within the original frequency-domain model.

The identified transfer functions between structural displacement and gust inputs, and the unsteady aerodynamic forces applied to the

structure are coupled with the mechanical state-space model, leading to an aeroelastic model in the form of Eq. (1).

$$\begin{cases} \dot{\mathbf{x}}_{ae} = \mathbf{A}\mathbf{x}_{ae} + \mathbf{B}_g\mathbf{u}_g + \mathbf{B}_c\mathbf{u}_c \\ \mathbf{y} = \mathbf{C}\mathbf{x}_{ae} + \mathbf{D}_g\mathbf{u}_g + \mathbf{D}_c\mathbf{u}_c \end{cases} \quad (1)$$

In Eq. (1), the aeroelastic state, represented by $\mathbf{x}_{ae}$, contains both the mechanical and aerodynamic states. The inputs include the gust profile, $\mathbf{u}_g$, and the external force, $\mathbf{u}_c$, while the output, $\mathbf{y}$, is the required structural response. The matrices $\mathbf{A}$, $\mathbf{B}$, $\mathbf{C}$, and $\mathbf{D}$ are the state-space governing matrices. Specifically, $\mathbf{A}$ is the state (system) matrix, $\mathbf{B}$ is the input matrix, $\mathbf{C}$ is the output matrix, and $\mathbf{D}$ is the direct transmission (feedthrough) matrix. With this linear time invariant model, aeroelastic gust responses can be directly calculated in the time domain.

In the present work, once the aeroelastic system has been reformulated in state-space form, a localized non-linear effect is introduced directly in the time domain of the aeroelastic model, in the region where buckling is expected to occur. A concentrated non-linear rotational stiffness is modeled in this region by modifying the structural kinematics. Specifically, the structural continuity is intentionally interrupted by introducing two coincident nodes at the selected location, as illustrated in Fig. 1.

The aeroelastic stick model represents the structure of the wingbox through a number of nodes connected by bars. In Fig. 1, five nodes and four bar elements are used as an example to help visualize the structural kinematics modification performed on the aeroelastic stick model. This modification results in a change of the original structure, which is transformed into a system with six nodes. The connection between the second and third bar elements is removed, yielding two kinematically independent substructures. The newly introduced nodes, denoted as A and B, share the same spatial coordinates but belong to different substructures, allowing a localized non-linear moment-rotation relation to be applied between them. $\hat{X}, \hat{Y}, \hat{Z}$ represent the axes of the new local coordinate system with origin at the coordinates of the new nodes A and B, and θ_{SPOINT} denotes the relative rotation between them. Thus, following the methodology demonstrated in this example, modifications to the structural kinematics of the aeroelastic model can be implemented in any region of the wing where buckling is intended to occur.

The kinematic continuity of the structure is restored by imposing linear algebraic constraints between the displacement and rotation Degrees of Freedom (DOFs) of the coincident nodes. These constraints are formulated in the form of multi-point constraints.

The structural DOFs of nodes A and B are partitioned into non-buckled DOFs, which remain linearly constrained, and the single DOF associated with the buckling mechanism, namely the relative rotation. In order to enable the introduction of a non-linear forcing term associated with the localized non-linear rotational stiffness, a scalar point is added to this buckling-related DOF. The scalar point does not have an independent spatial position in the structural mesh. Its location is implicitly defined by the two coincident structural nodes introduced at the selected buckling location, while its scalar degree of freedom represents their relative rotation. In the present implementation, the non-linear behavior is therefore introduced at a node of the stick model. If required, an additional structural node can be introduced at any desired spanwise position by locally subdividing the corresponding beam element. The scalar point rotation, θ_{SPOINT} , appearing in Eqs. (2) and (3), represents the relative rotation between the two coincident nodes (A and B in Fig. 1), defined with respect to the local coordinate system $\hat{X}\hat{Y}\hat{Z}$. The imposed constraints are expressed analytically in Eq. (2).

$$\begin{cases} \theta_{A,i} = -\theta_{B,i} & i \in \text{non-buckled rotational DOFs} \\ \theta_{A,j} = -\theta_{B,j} - \theta_{\text{SPOINT}} & j = \text{buckled (rotational) DOF} \end{cases} \quad (2)$$

To fully restore the kinematics while allowing for localized non-linear behavior due to buckling, the state-space aeroelastic model is augmented with a moment-feedback formulation. For the introduced scalar point, an associated moment is applied, whose magnitude depends on the corresponding scalar rotation. Thus, the applied moment

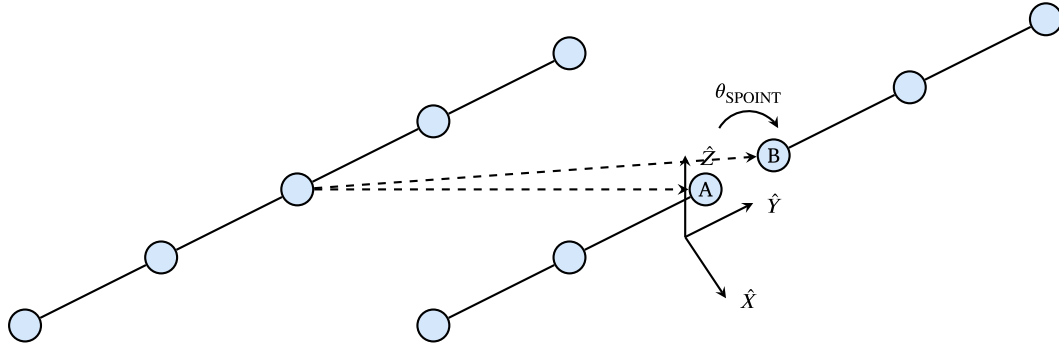


Fig. 1. Kinematics used for the introduction of the non-linear moment. Left: original structure. Right: structure modified with coincident nodes A–B.

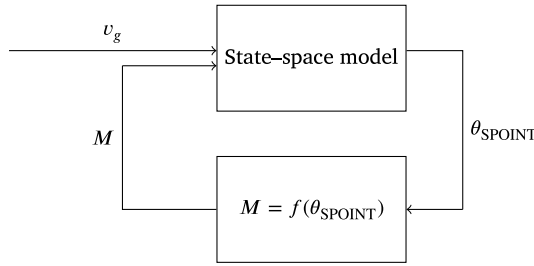


Fig. 2. Simulink feedback moment diagram.

M is defined as a function of the state variable θ_{SPOINT} , as shown in Eq. (3).

$$M = f(\theta_{\text{SPOINT}}) \quad (3)$$

where $f(\theta_{\text{SPOINT}})$ is a function of the structural response defined by the user. Within the scope of this work, it denotes the function that characterizes the moment-rotation relationship, accounting for both pre-buckling and post-buckling behavior.

This formulation can be implemented as a feedback controller in the time domain, where the controller input is the measured rotation θ_{SPOINT} and the controller output is the restoring moment M . In this way, the non-linear feedback moment M depends on the relative rotation between the coincident nodes and effectively represents a localized non-linear rotational stiffness, such as the one created by the buckling phenomena.

The implementation of the feedback controller as a Simulink block diagram is shown in Fig. 2. In the diagram, v_g denotes the gust velocity input, the state-space model represents the aircraft aeroelastic model, θ_{SPOINT} is the feedback signal, and M is the resulting moment applied to the system.

In this work, the buckling mechanism is represented by a rotational spring inserted at the scalar point, defined by a linear stiffness in the pre-buckling regime and a reduced linear stiffness in the post-buckling regime. Accordingly, the moment-rotation response of the composite structure that enters the post-buckling regime is approximated as shown in Fig. 3. The structure exhibits a bilinear moment-rotation behavior, mathematically described in Eq. (4), where k denotes the derivative of the moment-rotation relation, i.e., the rotational stiffness. Two distinct regions with constant stiffness can be identified: the first corresponds to loading below the critical buckling moment $\hat{M}$, governed by the pre-buckling stiffness k_{pre} , while the second represents the post-buckling regime, governed by the reduced stiffness k_{post} . In Fig. 3 and Eq. (4), $\hat{\theta}$ represents the rotation value at the critical buckling point.

$$\begin{cases} M = k_{\text{pre}} \theta & \text{for } \theta \leq \hat{\theta} \\ M = k_{\text{post}} \theta + \hat{M} - k_{\text{post}} \hat{\theta} & \text{for } \theta > \hat{\theta} \end{cases} \quad (4)$$

The bilinear moment-rotation law of Eq. (4) is evaluated from the scalar point rotation θ_{SPOINT} . Although each regime is individually gov-

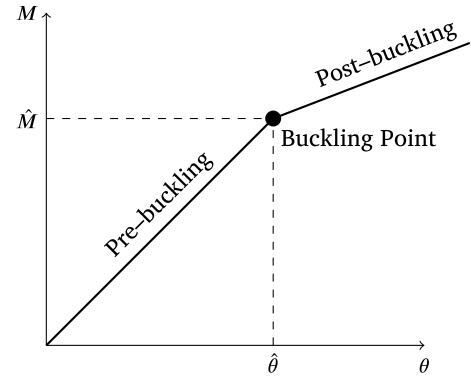


Fig. 3. Moment-rotation relationship in the non-linear behavior of the pre- and post-buckling regimes.

erned by a constant rotational stiffness, the change in stiffness at the buckling point makes the overall response non-linear. Introducing this stiffness reduction into the aeroelastic state-space model makes it possible to compare the maximum WRBM values obtained with and without the post-buckling regime. The bilinear law is used here to describe not the behavior of a spring or of an added component, such as a folding wingtip, but the bending stiffness of the composite wingbox itself in its pre- and post-buckling regimes. This bilinear moment-rotation behavior makes it possible to achieve gust load alleviation in a passive manner.

The design framework built on this capability is summarized in Fig. 4, which shows the models involved in each phase. First, an aeroelastic stick model of the aircraft under consideration is created in NeoCASS. Its structural kinematics are then modified as described above, so that buckling can be induced at a selected location through the coincident nodes and the scalar point carrying the bilinear rotational stiffness. A parametric analysis on this modified stick model identifies the buckling location, critical buckling moment, and post-buckling stiffness reduction that maximize the reduction in WRBM. These targets then guide the design of a three-dimensional finite element model of the wingbox, whose skin is made to buckle accordingly. Finally, the WRBM reduction achieved by the resulting configuration is verified on the stick model. Each phase is detailed below.

The parametric analysis on the modified stick model identifies the combination of critical buckling moment and post-buckling stiffness reduction that produces the greatest decrease in WRBM. Since the kinematics modification illustrated in Fig. 1 can be applied at any node along the wing, it is also possible to determine the buckling location that maximizes the reduction in WRBM. From this parametric study, carried out with recourse to aeroelastic dynamic analyses, a combination of desired critical buckling moment, post-buckling stiffness, and buckling location is obtained.

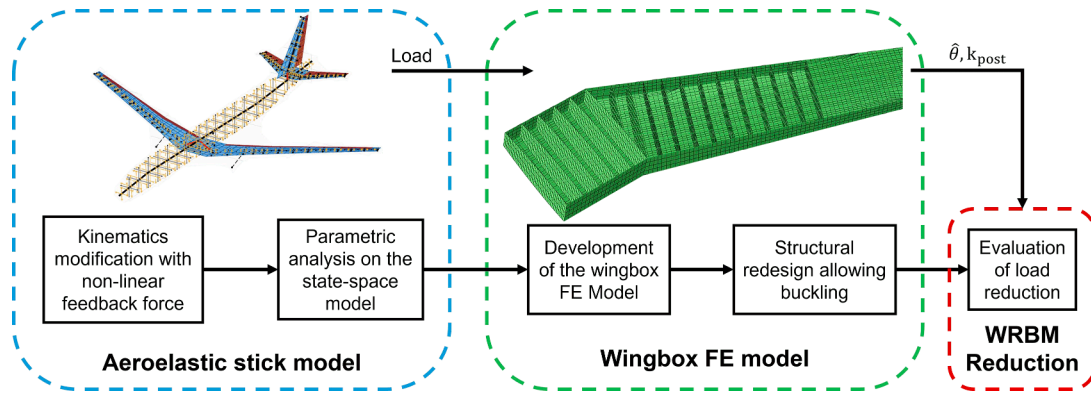


Fig. 4. Framework and models used in each phase.

The next step is to design a wingbox that follows the results obtained from the parametric analyses on the aeroelastic stick model. This means designing a wingbox whose skin will go into post-buckling regime at the preferred location specified by the aeroelastic stick model analyses, as well as following the critical buckling moment and post-buckling stiffness when subjected to the same gust load.

For this purpose, the aeroelastic stick model and the global three-dimensional FE model are generated from a common structural description provided by the NeoCASS Meta-Model [34]. At first, a Meta-Model is created from the geometry of the wing, defined in the graphical interface of NeoCASS. Here, macro parameters of the wing such as airfoils, span, dihedral and sweep angles, wingbox dimensions, and position are defined. The sets of planes that define front and rear spars, ribs, and stringers are created, and their intersections are identified. After this, the kinks are modeled, and material properties and thicknesses are assigned to the different structural components. With all this information, the Meta-Model is able to create a global three-dimensional FE model of the wingbox, then translated to ABAQUS [35].

The NeoCASS Optimization Toolbox (NeOPT) [34] is employed to compute a set of static loads that generates an internal forces distribution equivalent to the ones obtained at a given time instant during a gust excited dynamic response. The time instant that is selected is the one in which the highest value of WRBM is observed throughout the whole dynamic analysis. Then, NeOPT translates this load distribution into discrete load values composed of three forces and three moments which are applied at the coordinates of the nodes of the stick model. Since these nodes are also imported into the three-dimensional model, the calculated loads are then applied at these nodes and properly transferred to the mesh of the wing through kinematic and distributing coupling constraints.

An important aspect to highlight is that the structural analysis of the wing must account for the combined effect of loads arising from the aeroelastic dynamic gust response and the aircraft trim condition. However, NeoCASS performs these analyses separately, producing two distinct load sets: a time-varying load from the dynamic analysis and a constant load from the trim analysis. Since both analyses are linear, the corresponding load contributions can be superimposed. The resulting combined load history is then evaluated, and the time instant associated with the maximum load is selected and applied as input to the three-dimensional finite element model in ABAQUS.

FE analyses are then performed changing the structural components of the wing to obtain buckling phenomena at the preferred buckling location provided by the aeroelastic parametric analysis, along with the appropriate buckling moment and post-buckling stiffness reduction. This study is conducted with recourse to linear buckling analyses. The design variables include the upper panel composite skin thickness, the area of the stringers of the upper panel, and the number of stringers and ribs of the wingbox.

These linear buckling analyses are computationally inexpensive, which makes them well suited to exploring the large number of design-variable combinations required to identify a suitable configuration. The response of the selected design in the post-buckling regime is, however, characterized more accurately through a non-linear FE analysis. The buckling behavior and post-buckling stiffness are extracted from the model response: as the applied load is increased, the vertical (bending) displacements of selected nodes are monitored, and this raw measurement is then post-processed into the equivalent critical buckling moment and post-buckling rotational stiffness that define the bilinear moment-rotation law of the stick model.

After obtaining a wingbox design which is characterized by critical buckling moment and post-buckling stiffness as close as possible to those that resulted from the parametric analysis of the aeroelastic stick model, the value of the WRBM reduction obtained with this new configuration can be verified. For this last step of the methodology presented in Fig. 4, an aeroelastic dynamic gust analysis is carried out on the stick model with modified non-linear kinematics that reproduce the buckling moment and post-buckling stiffness characterized from the non-linear FE analysis.

3. U-HARWARD case study

In this research, a model of the aircraft developed in the U-HARWARD project [36] is used as a case study. This model is representative of a medium range, single aisle aircraft comparable to Airbus 321, with an aspect-ratio equal to 15, a half-span approximately equal to 21 m and a root chord equal to 2.2 m.

3.1. Aeroelastic stick model

Following the design workflow shown in Fig. 4, starting from the CAD model, the aeroelastic stick model is created using NeoCASS. Then, the stick model is modified to investigate a possible spanwise buckling location, buckling moment, and post-buckling stiffness. The WRBM is adopted as the key performance indicator for the wing structural sizing. Its minimization directly contributes to load alleviation and, consequently, to a reduction of the overall structural mass.

Fig. 5 shows the CAD model, the aeroelastic stick model, and the aeroelastic model modified with the kinematics described in Fig. 1 and Eq. (2) to introduce the non-linear feedback moment that models the buckling.

After implementing the non-linear structural kinematics modification to the stick model, a parametric analysis using dynamic gust analyses is performed on this model. This parametric analysis maps the sensitivity of the WRBM to the buckling moment (P1), the post-buckling stiffness reduction (P2), and the spanwise region where buckling occurs. Both P1 and P2 are defined with respect to the nominal model, that is,

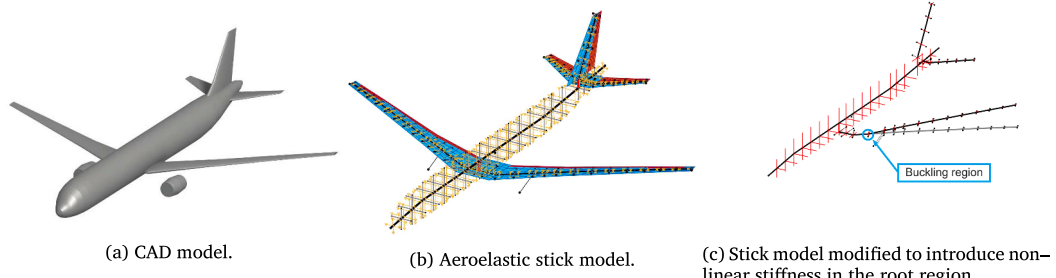


Fig. 5. U-HARWARD aircraft reference models.

the original stick model without kinematic modifications. P_1 is the critical buckling moment expressed as a percentage of the nominal moment, defined as the maximum incremental out-of-plane bending moment of the nominal model at the selected location during the gust response, that is, the moment produced by the gust alone, excluding the static contribution of the trim condition, so that a lower P_1 triggers buckling earlier. P_2 is the post-buckling stiffness reduction expressed as a percentage of the nominal stiffness, i.e. the pre-buckling stiffness of the nominal model, so that a higher P_2 denotes a larger loss of stiffness after buckling. P_1 and P_2 are varied independently. In an actual wingbox the two follow from the same design variables and are therefore coupled, so that the attainable combinations form a subset of the mapped domain. The maps are accordingly used to identify the buckling region and the range of parameters worth pursuing, while the combination actually realized is obtained from the three-dimensional model.

Three representative spanwise locations are selected to investigate the influence of the buckling position. These locations correspond to three regions characterized by structural and/or load-distribution discontinuities: the wing-root region, the engine-attachment region, and the region corresponding to the beginning of the aileron. For a wing semi-span of $b/2 = 21.337$ m, the selected positions are $y = 3.910$ m (18.33% of the semi-span), $y = 6.908$ m (32.38%), and $y = 16.704$ m (78.29%), respectively.

The purpose of this spatial investigation is to compare three regions and assess, for each of them, the sensitivity of the WRBM reduction to the critical buckling moment P_1 and the post-buckling stiffness reduction P_2 . The three selected locations are therefore used to identify the overall spanwise trend and the most promising region for the subsequent three-dimensional structural design.

Fig. 6 presents a top view of the wing, drawn from the node coordinates of the aeroelastic stick model: the horizontal axis y is the spanwise coordinate and the vertical axis x is the streamwise coordinate, both given in meters in the aircraft reference frame. The line and dots represent the stick model itself, where the dots are the structural nodes, carrying the concentrated masses and connected by bar elements to form the beam representation of the wing. The three red squares in Fig. 6 mark the nodes at which the kinematic modification of Fig. 1 is applied, thereby identifying the spanwise location of each of the three buckling regions considered.

In each of the three regions, buckling is modeled through the single rotational DOF introduced in Section 2. The instability is driven by the out-of-plane bending of the wing: the load associated with the buckling DOF is the out-of-plane bending moment M , and the corresponding kinematic quantity is the out-of-plane bending rotation θ . Under this load, the upper skin is in compression, and it is there that buckling is induced. Once a panel buckles, most of the additional load is redistributed to the stringers and to the portions of skin that remain stable. Therefore, the bending stiffness of the section is reduced and an increment of moment produces a larger increment of rotation. This is the reduction represented by k_{post} in Eq. (4). This rotation is defined as the relative rotation between the two coincident nodes A and B of Fig. 1,

taken about the local axis governing out-of-plane bending. The feedback moment associated with this DOF depends non-linearly on the rotation, as expressed in Eq. (3), and follows the bilinear rotational spring law of Fig. 3 and Eq. (4), so that the loss of bending stiffness caused by skin buckling reduces the moment transmitted across the region and, in turn, the WRBM.

Following the CS/EASA 25 [37] regulation for large transportation aircraft, all aeroelastic state-space models are subjected to 10 gust profiles with positive and negative amplitudes. Fig. 7 shows the positive gust profiles analyzed. The gust gradient, H , is equally spaced between 9 m and 107 m, and the flight point is representative of a cruise condition with a true air speed of 231 m/s at an altitude of 10,700 m. This flight point is selected as a representative cruise condition for demonstrating the proposed methodology and is not intended to represent the globally critical gust condition over the complete flight envelope. In a certification-oriented application, the procedure should be repeated over the relevant altitude-speed envelope to identify the critical gust condition. Among the gust profiles considered, the condition producing the maximum WRBM corresponds to a gust gradient of $H = 63.44$ m, with a gust amplitude of 16.8977 m/s and an associated gust frequency of 1.812 Hz. This case is therefore selected as the critical gust condition for the subsequent transfer of loads to the three-dimensional FE model.

The analysis on the stick model is conducted for each combination of the design parameters (buckling moment, post-buckling stiffness reduction and buckling region) and load gust profile. The maximum value of the WRBM is extracted as bar element internal force, output of the state-space model, and compared with the value obtained with the nominal model. As such, the reduction of the WRBM value can be calculated for each combination of the design parameters: buckling moment, post-buckling stiffness and buckling region. The WRBM reduction obtained for each combination of parameters can then be displayed with isolines, as shown in Fig. 8, for the three analyzed spanwise regions of the wing: the root, engine, and outboard.

From Fig. 8 it can be seen that the largest reduction of the WRBM is achieved when buckling is triggered in the engine region. The highest reduction is equal to approximately 15% and is obtained for a post-buckling stiffness reduction (P_2) of 50% and a buckling moment (P_1) equal to 20% of the applied moment. However, it can be noted that in this region buckling is characterized by a limited design space: for buckling moments greater than 60% of the applied moment ($P_1 > 60\%$), the WRBM reduction rapidly approaches zero, so that only a narrow range of buckling moment and post-buckling stiffness reduction values yields a meaningful WRBM reduction, leaving little freedom to adjust the design while retaining the load-alleviation benefit. It should be noted that this narrower design space is not the effect of a structural constraint, since no structural limitation is imposed in these analyses, but a feature of the aeroelastic response itself.

In the wing root region, instead, meaningful values of WRBM reduction are obtained up to buckling moments (P_1) of approximately 80% of the applied moment, without ever reaching zero. Furthermore, the

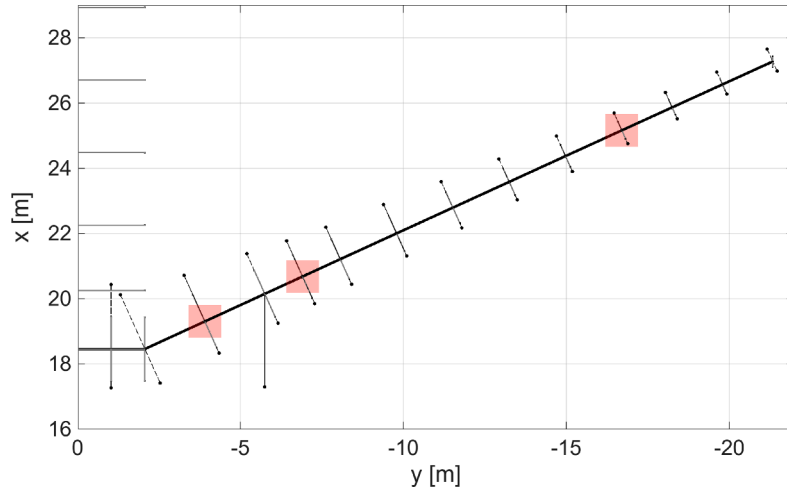


Fig. 6. Top view of the wing with the aeroelastic stick model, showing the spanwise location of the three buckling regions considered: root, engine, and outboard.

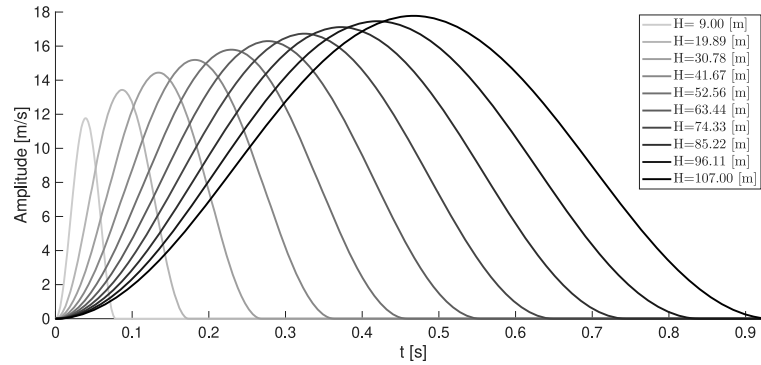


Fig. 7. Gust profiles used for the aeroelastic analysis at a true air speed equal to 231 m/s and altitude of 10,700 m.

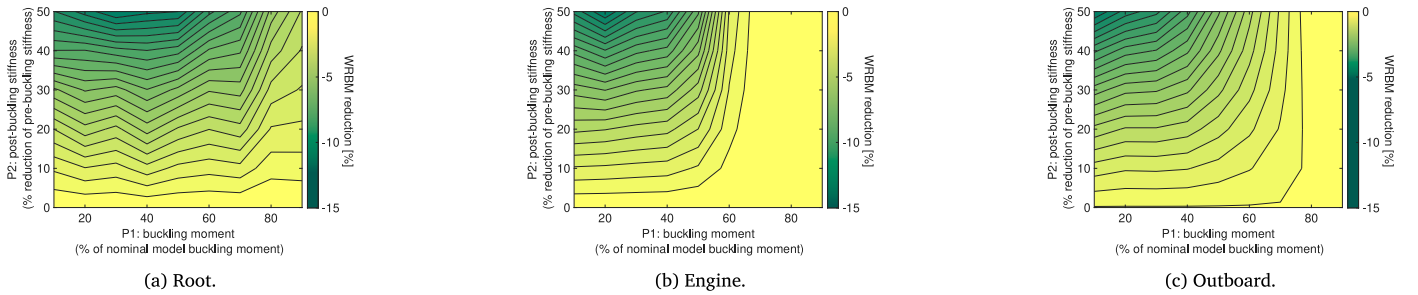


Fig. 8. WRBM reduction as a function of the buckling moment (P1) and the post-buckling stiffness reduction (P2), shown for the root, engine, and outboard regions.

sensitivity to the post-buckling stiffness reduction (P2) is comparable in the two regions, so that the wider range of buckling moments at the root is obtained without any penalty in this respect. For this reason, it is decided to analyze the wing root region. In this area, a WRBM reduction of approximately 12% can be achieved, while maintaining a wider design space for adjusting the buckling moment and post-buckling stiffness reduction. This makes the wing root a more practically exploitable configuration compared to the engine region.

Conversely, the outboard region is found to be the least effective in terms of WRBM alleviation, showing reductions of 5% across the explored parameter range.

In general, the analysis highlights that the parameter exerting the strongest influence on WRBM reduction is the post-buckling stiffness reduction (P2).

3.2. Wingbox FE model

As defined by the framework presented in Section 2, the next step is to design a wingbox that reproduces the results obtained from the parametric analysis conducted on the aeroelastic stick model. In other words, the wingbox skin must enter the post-buckling regime near the root region when subjected to the critical gust loads.

For this purpose, the aeroelastic stick model and the global three-dimensional FE model are generated from a common structural description provided by the NeoCASS Meta-Model [34]. The Meta-Model is a semi-analytical three-dimensional representation of the wingbox, described in detail in Ref. [34], which stores the geometrical, structural, and material information required to generate analysis models at different levels of fidelity.

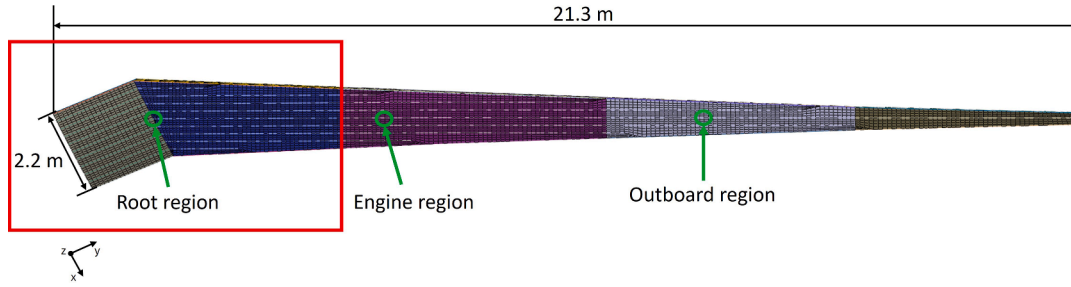


Fig. 9. Top view of the whole wingbox.

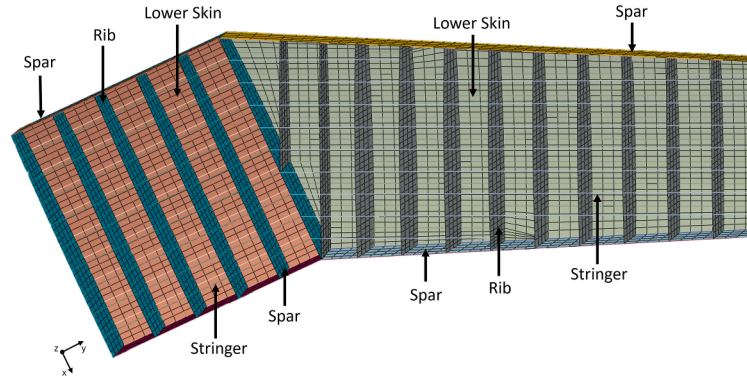


Fig. 10. Close-up view of the wingbox in the root region.

The wing geometry is defined through macro-parameters such as the airfoil distribution, span, dihedral and sweep angles, front- and rear-spar positions, and rib and stringer layouts. The corresponding structural properties, including skin and spar laminates, stringer cross-sectional areas, and spar-cap properties, are assigned directly to the individual wingbox components. From this common definition, the Meta-Model generates both the sectional representation required for the aeroelastic stick model and the explicit three-dimensional FE representation of the wingbox.

Therefore, the detailed structural properties of the three-dimensional model are not recovered from the equivalent beam properties of the stick model. Instead, both models are consistently derived from the same underlying wingbox definition.

The resulting three-dimensional FE model of the wingbox is developed in ABAQUS and composed of composite shell elements (S3R, S4R) to represent the skins and spars, while the stringers and spar caps are modeled with beam elements (B31), and the ribs are modeled with shell elements (S4R).

Fig. 9 presents a top view of the whole wingbox. The green arrows indicate the spanwise locations of the three regions analyzed in the parametric study performed on the aeroelastic stick model: the root, engine, and outboard regions.

Since the objective is to induce post-buckling behavior near the root region, Fig. 10 shows a close-up view of the wingbox in this region, highlighted by the red box in Fig. 9. Fig. 10 highlights the structural elements of the wing (stringers, ribs, spars, spar caps, and skins), and the upper skin panels have been removed to reveal the interior.

A cross-section view of the wing is presented in Fig. 11. This cross-section is taken at the outboard spanwise position, as shown in Fig. 9, and is composed of three upper and lower stringers, four spar caps, front and rear spars, and upper and lower composite skins. Due to the wing taper, the cross-section varies along the span. Consequently, the number of upper and lower stringers differs along the span, with a higher number near the root that progressively decreases toward the wing tip.

The wing's skins and spars are made of balanced and symmetric IM7/8552 carbon fiber composite laminate with a stacking sequence of

Table 1

Lamina properties and ply strengths of the wing's skins and spars.

Property	Unit	Value
E_1	[MPa]	161,000
E_2	[MPa]	11,400
ν_{12}		0.32
G_{12}	[MPa]	5200
ρ	[kg/m ³]	1570
X_T	[MPa]	2724
X_C	[MPa]	1690
Y_T	[MPa]	111
Y_C	[MPa]	264
S_L	[MPa]	120

Table 2

Material properties of the ribs and stringers.

Ribs			Stringers		
E [MPa]	G [MPa]	ρ [kg/m ³]	E [MPa]	G [MPa]	ρ [kg/m ³]
72,000	27067.7	1570	67,000	27006.5	1570

[0, 45, -45, 90]_s, in which the 0°, 45° and -45° orientations each account for 30% of the total thickness and the 90° orientation for the remaining 10%. The properties are reported in Table 1 [38,39]. The ribs and stringers are also made of carbon fiber, but unlike the skins and spars they are not modeled ply by ply. Instead, they are represented as an equivalent isotropic material whose stiffness is the average stiffness of the laminate, so no stacking sequence needs to be defined and the properties are described by the single Young's modulus, shear modulus, and density listed in Table 2.

As explained in Section 2, the load set applied to the three-dimensional FE model is obtained by superimposing the internal loads of the wing corresponding to a frozen time instant of the dynamic gust with the 1g trim load computed for the cruise condition. It corresponds

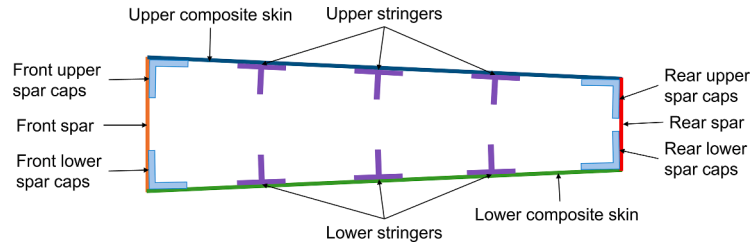


Fig. 11. Cross-section view of the wingbox.

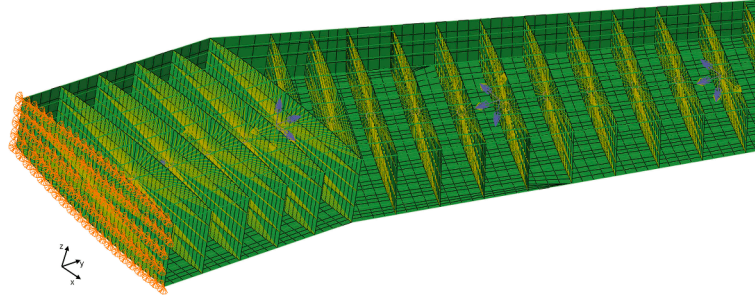


Fig. 12. Close-up view of the coupling constraints applied on each rib of the model (in yellow), loads (in yellow and purple arrows), and boundary conditions (in orange). (For interpretation of the references to color in this figure legend, the reader is referred to the web version of this article.)

to the gust response that produces the highest WRBM in the original model. This approach treats the problem in a quasi-static manner. As a result, inertial effects associated with the transient dynamic response, such as mass-related acceleration forces are not explicitly captured and do not dynamically evolve throughout the buckling response. Nonetheless, it is expected to provide a reasonable estimate of the critical wing loading condition during a gust event, while significantly reducing the computational cost compared to a fully coupled transient analysis [40].

The load set applied on the three-dimensional FE model is the most critical case which corresponds to the gust response that produces the highest WRBM. The forces and moments that compose this load set are applied at nodes in ABAQUS positioned in the same coordinates as the nodes that constitute the stick model and then distributed to all the nodes of the closest rib through distributing coupling constraints. Furthermore, pinned boundary conditions are applied to all nodes of the root rib, constraining all translational DOFs ($U1 = U2 = U3 = 0$). In Fig. 12 the coupling constraints, and the loads and boundary conditions are shown for the root region of the wing, but this method is applied across the entire wingspan.

Linear buckling eigenvalue and non-linear static analyses are performed on the original wingbox model, resulting in a maximum vertical displacement at the wingtip of $\Delta_z = 1.84 \text{ m}$, and a buckling eigenvalue of $\lambda = 3.46$, meaning that buckling occurs only under load values higher than three times those applied. Then, analyses are performed changing the upper panel skin thickness, the stringers area, the number of stringers, and the number of ribs at the root region to understand how they influence the structural response of the wing and to find a configuration that buckles near the root region, with a buckling eigenvalue lower than one.

Reducing the upper panel skin thickness significantly decreases the first eigenvalue obtained from the linear buckling analysis. A 50% reduction of the upper panel skin thickness leads to a first buckling eigenvalue equal to $\lambda = 0.85$. In this case, the buckling shape area is confined to a small area, located at the end of a stringer, which would lead to minimal stiffness reduction in the post-buckling regime, as shown in Fig. 13a.

When the stringers area is decreased by 50%, the first eigenvalue suffers a reduction of only 16%, totaling a value equal to $\lambda = 2.89$. The buckling region remains small, similar to that observed when varying the upper panel skin thickness.

When the skin thickness and the stringer area are both decreased by 40%, a buckling eigenvalue equal to $\lambda = 1.06$ is achieved. However, the buckling shape area still remains confined to a small region.

Then, a few topological modifications of the wingbox are performed. Since buckling occurs near the trailing edge, the stringers are progressively removed from the trailing edge to the leading edge. Removing the first two stringers near the trailing edge reduces the first buckling eigenvalue by nearly 68%, bringing it to $\lambda = 1.11$. In contrast, removing all remaining stringers results in only a 17% further reduction. The removal of all stringers expands the buckling area but confines it between ribs, forming elongated chord-wise buckling zones with a first buckling eigenvalue equal to $\lambda = 0.54$, as shown in Fig. 13b. This solution would lead to a loss of aerodynamic performance, which is not desired.

The effect of removing the ribs is analyzed. The influence of rib count on structural behavior reveals that removing adjacent ribs creates a buckling-prone region. As more ribs are removed, the buckling area expands and the first eigenvalue drops significantly, reaching a value equal to $\lambda = 1.08$ when six ribs are removed. However, this large buckling area may lead to other issues such as loss of airfoil shape. Fig. 13c shows the first buckling shape with $\lambda = 0.67$ where all ribs are removed.

These analyses indicate that ribs and stringers also govern the buckling mode shape rather than only the load at which it appears. Acting as internal supports, they subdivide the upper skin into panels whose dimensions determine the size of the region that can buckle: changing the skin thickness or the stringer area alters the stiffness of these panels but leaves their boundaries unchanged, so that buckling remains confined, whereas removing a support merges adjacent panels into a larger one. Since the post-buckling stiffness reduction is governed by the area of the buckled region, lowering the buckling load alone is not sufficient and the topology of the wingbox has to be modified.

The next step is to select a configuration. The design had to satisfy a first buckling eigenvalue below one, a buckled area large enough to produce a meaningful post-buckling stiffness reduction, the absence of ply failure, and the preservation of the airfoil shape. The last two of these are assessed from the buckling shapes and from the first-ply failure analysis rather than expressed as scalar constraints, which is why the design variables are examined one at a time and then combined. From the obtained results, a wing configuration is then studied in which four

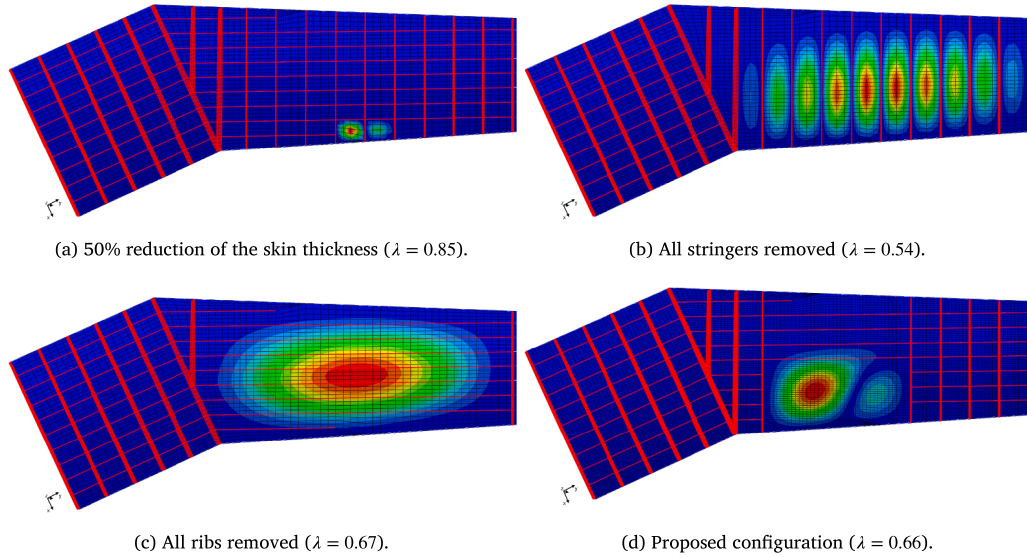


Fig. 13. First buckling shapes during the analysis of the wingbox root region: (a)–(c) individual structural modifications, (d) the proposed configuration combining them.

ribs are removed along with the four stringers closer to the trailing edge. Additionally, to avoid ply failure, the thickness of the upper panel skin in the region without ribs is increased by 35%. With this configuration, a reduction of 1.7% in mass is achieved compared to the original wing model.

The linear buckling analysis of this configuration results in a first buckling eigenvalue equal to $\lambda = 0.66$. The buckling shape is shown in Fig. 13d.

A first-ply failure analysis is carried out. The Tsai–Hill failure criterion reveals that this configuration is susceptible to ply failure, resulting in a maximum Failure Index (FI) equal to 1.00.

Thus, the laminate ply stacking sequence is changed from $[0, 45, -45, 90]_s$ to $[-45, 45, 90, 0]_s$ at the wing region where the ribs are removed, keeping the same thickness distribution among the orientations. The Tsai–Hill criterion for this final configuration results in a maximum FI value equal to 0.72, indicating that there is no failure. The contour plot of the maximum Tsai–Hill FI value recorded among all plies is shown in Fig. 14a.

It should be noted that the structural assessment presented here is limited to the critical limit-load condition. Demonstrating compliance with the corresponding ultimate-load requirement would require an additional non-linear post-buckling strength analysis, since the structure would operate beyond the onset of skin buckling and the associated load redistribution and failure mechanisms would need to be explicitly assessed.

With this design, the linear buckling analysis produces a first buckling eigenvalue equal to $\lambda = 0.71$, with the buckling shape represented in Fig. 14b. Comparing the buckling shapes shown in Figs. 13d and 14b, although the buckling eigenvalues are similar, the buckling shape of this configuration presents two half waves and a larger buckling area, while the previous configuration displayed only one half wave. The buckling shape has no counterpart in the aeroelastic stick model, in which buckling is introduced through the single rotational DOF of the scalar point, and it enters the framework only through the post-buckling stiffness reduction that it produces. A mode extending over a larger area yields a larger post-buckling stiffness reduction.

Both laminate sequences of the wingbox skin are symmetric and balanced and share the same thickness distribution among the orientations, so that the membrane stiffness of the laminate is unchanged and the in-plane load path is not altered. Changing the position of the plies through the thickness leads to a different bending response of the skin.

The buckled region is characterized by a large local curvature, so that the outermost plies carry the highest bending stresses. Therefore, the first proposed configuration shows a FI equal to 1.00 at the outer surface of the upper skin. In the sequence $[0, 45, -45, 90]_s$, those outer plies are the 0° ones, which are loaded transversely to the fiber direction and are therefore governed by the transverse strengths of Table 1, an order of magnitude lower than the longitudinal ones. Reordering the laminate as $[-45, 45, 90, 0]_s$ moves these plies toward the mid-plane, where the bending stress is lower, and brings the $\pm 45^\circ$ plies to the outer surfaces, where the load is carried partly along the fiber direction. The most stressed position and the weakest plies no longer coincide, which lowers the peak failure index. The same redistribution increases the twisting stiffness of the laminate, which is consistent with the slightly higher buckling eigenvalue and with the change of buckling shape reported above.

From the non-linear analyses, the values of the maximum vertical displacement along the wingspan are also evaluated. The displacement values are extracted from the same reference nodes of the aeroelastic stick model. Fig. 15 presents the nodes chosen, in red, where the node located on the rib of the root is identified as Node 0, and the node at the tip is labeled as Node 14. Table 3 reports the spanwise position for each reference node, and the maximum vertical displacement results obtained for each node for the non-linear analyses of the original wing model and the proposed design at the critical gust load.

Analyzing the results, it can be seen that the displacement variation is highest at Node 3, located at $y = 3.91$ m from the wing root. This coincides with the wingspan position where buckling occurs. Subsequent nodes located at an outboard position from this region also reveal high displacement variations, decreasing as the nodes are located closer to the wingtip.

Fig. 16 shows a lateral view of the root region of the wingbox, removing the rear spar, obtained from the non-linear analysis. It reports the contour plot of the vertical displacement. It is possible to note the buckling of the skin, deforming inwards, i.e. to the inside of the wingbox.

The stiffness at each reference node is obtained from the slope of its curve of applied load versus vertical displacement, extracted from the non-linear analyses of the original and proposed configurations. What is extracted is not a stiffness value, but the ratio between the slopes of the two configurations. Being dimensionless, this ratio is applied in the stick model as a percentage reduction of the pre-buckling rotational stiffness.



Fig. 14. Final proposed configuration after the stacking-sequence change: (a) maximum-envelope Tsai-Hill failure index across all plies (no ply failure), and (b) first buckling shape.

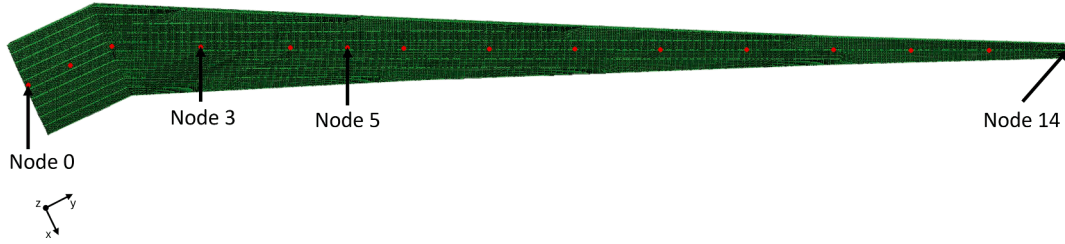


Fig. 15. Reference node positions along the wingspan.

Table 3

Spanwise position of each reference node and vertical displacement from the non-linear analyses of the original and proposed configurations at the critical gust load.

	Node number														
	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14
Spanwise position y [m]	0	1.04	2.07	3.91	5.75	6.91	8.07	9.79	11.52	13.25	14.98	16.70	18.25	19.79	21.34
Original config. [m]	0	0.00272	0.00957	0.0426	0.102	0.154	0.219	0.342	0.496	0.678	0.888	1.12	1.35	1.59	1.83
Proposed config. [m]	0	0.00274	0.00977	0.0479	0.110	0.167	0.237	0.367	0.528	0.717	0.934	1.17	1.41	1.65	1.90
Variation [%]	–	0.72	2.06	12.4	8.00	8.41	8.11	7.29	6.48	5.78	5.21	4.75	4.42	4.15	3.94

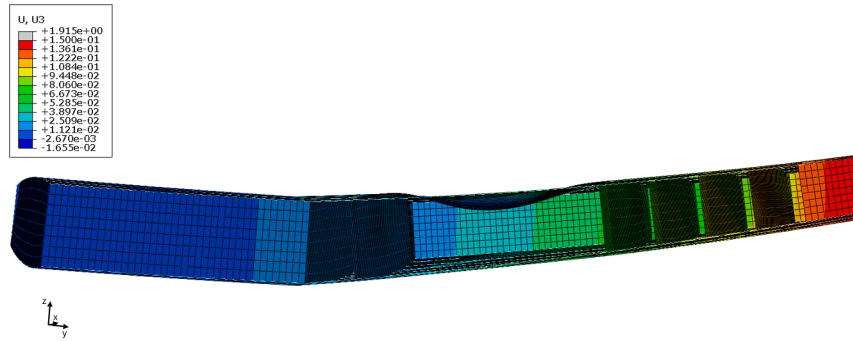


Fig. 16. Lateral view of the interior of the wingbox root region where it is shown the buckling of the skin, and the contour plot of the vertical displacement results obtained from the non-linear static analysis [m].

The response of the wing under the critical gust condition is dominated by bending, while the in-plane components of the local deformation of the skin have no counterpart in the bilinear law of Eq. (4). Therefore, the vertical displacement is used to calculate the stiffness reduction, condensing the effect of buckling into the bending response of the wing. Fig. 17 shows this curve for Node 5, located at $y = 6.91$ m from the wing root, just outboard of the region where buckling occurs. The vertical axis gives the applied load as a percentage of the total load, and the horizontal axis the vertical displacement of the node.

Node 5 is chosen as a representative example because it clearly shows the transition between the pre- and post-buckling regimes. Outboard of the buckling region, the pre-buckling response of the proposed configuration coincides with that of the original model, so the two curves over-

lap until buckling and separate only afterwards. This makes the buckling point, identified by the change in slope, easy to locate. At a node where the two curves already differed before buckling, this transition would be harder to identify.

When comparing the curves obtained from the non-linear analysis of the original model with the proposed configuration, it can be seen that the removal of the four ribs and four stringers produces a small reduction of the structure's stiffness in the pre-buckling regime. On the other hand, after the buckling load is achieved, the structural stiffness of the wingbox reaches a reduction equal to 15.1% when compared to the original model.

The values of the stiffness reduction calculated for every reference node of the wingbox are reported in Table 4. These reference nodes

Table 4
Spanwise stiffness reduction results for the proposed configuration.

	Node Nr.														Average
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	
Stiffness reduction in each node (%)	1.20	3.33	14.9	14.4	15.1	14.7	13.3	12.0	10.7	9.65	8.76	8.10	7.54	7.11	10.1

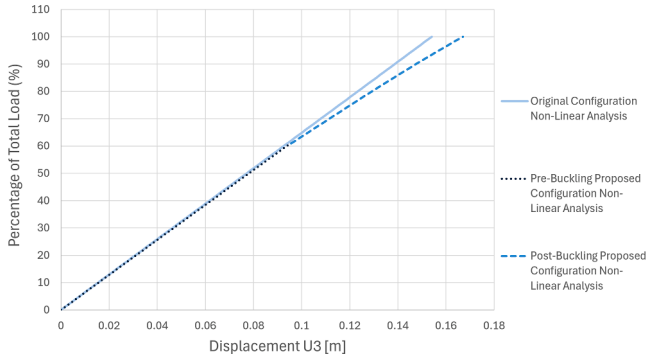


Fig. 17. Load as a function of the vertical displacement of a node at $y = 6.91$ m.

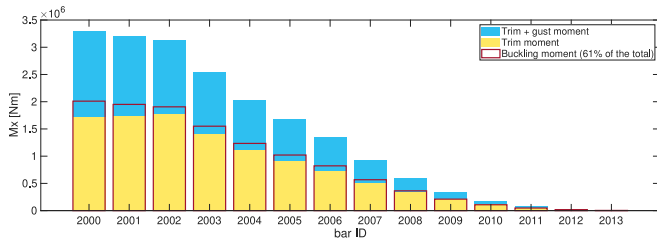


Fig. 18. Distribution of trim and gust moments across the reference bars of the wing.

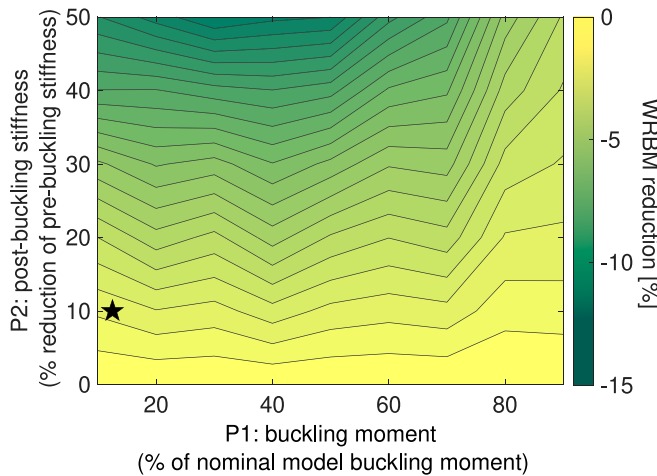


Fig. 19. WRBM reduction as a function of the buckling moment (P1) and the post-buckling stiffness reduction (P2) for the root region, with the marker indicating the solution obtained from the 3D analysis.

are placed at the same coordinates of the nodes of the aeroelastic stick model. An average of 10.1% stiffness reduction is obtained along the wingspan, registering higher values in nodes closer to where buckling occurs. The values reported in Table 4 show that the stiffness reduction depends on the spanwise position at which it is evaluated, which is why it is not extracted at a single node but averaged over all of them.

Two aspects of this procedure should be highlighted, as they constitute the main limitation of the present framework. First, as reported in Table 4, the stiffness reduction is not uniform, ranging from 1.20% at the nodes closest to the root to 15.1% near the buckling region, decreasing again toward the tip. This spanwise distribution cannot be reproduced by the aeroelastic model, in which the post-buckling behavior is concentrated at a single scalar point. Introducing a spatially varying stiffness reduction would require a multi-point constraint formulation, and the corresponding feedback moment, at every node of the stick model, significantly increasing its computational cost and complexity, forfeiting the advantage that motivates the use of a low-fidelity model. For this reason, the average value of 10.1% is adopted as the post-buckling stiffness reduction (P2) applied at the buckling location in the aeroelastic model. This value is conservative with respect to the local behavior of the buckling region, where the reduction lies between 14.4% and 15.1%, the average being lowered by the nodes inboard of the buckle and by those approaching the tip. Since the WRBM reduction increases with P2, adopting the average leads to an underestimation of the load alleviation rather than to an overestimation. Second, this stiffness reduction is derived from the vertical displacement of a single reference node, a one-dimensional quantity, whereas buckling develops over a three-dimensional region of the skin. The adopted procedure therefore captures the overall loss of stiffness caused by buckling, but not its spatial distribution.

Another consideration concerns the chord-wise distribution of the stiffness loss. The buckling field develops in the rear portion of the upper skin, near the rear spar. The reduction of the local torsional stiffness in this area leads to a shift of the elastic axis of the cross-section toward the front spar. The wingbox twist is proportional to the distance between the aerodynamic center and the elastic axis, and since the latter is positioned aft of the aerodynamic center, this forward shift reduces the lever arm, thus decreasing the local angle of attack and the incremental gust loads. However, the loss of torsional stiffness acts in the opposite sense, as its loss leads to a higher twist angle [41]. Which prevails cannot be established here, as the three-dimensional model has been characterized in bending only, and the single rotational DOF of the scalar point does not reproduce the twisting-bending coupling.

Further conclusions can be drawn from Fig. 17 regarding the load level at which buckling occurs. In the non-linear analysis the total load is applied incrementally, and the vertical axis of Fig. 17 represents the fraction of this total load reached at each increment, expressed as a percentage, while the horizontal axis is the vertical displacement of the node. Observing the curve of the proposed configuration, the slope changes when the applied load reaches 61% of the total, indicating that the skin enters the post-buckling regime at that global load level. This value is lower than the 71% predicted by the linear buckling analysis.

3.3. WRBM reduction

Having designed a wingbox that buckles near the root under the critical gust load, the final step of the framework is to quantify the resulting reduction of the WRBM. The buckling moment and post-buckling stiffness reduction characterized from the 3D FE analysis are used to define the bilinear moment-rotation law of the modified kinematics in the aeroelastic stick model, so that the stick model buckles at the same load level identified in the FE analysis. A dynamic gust analysis is then

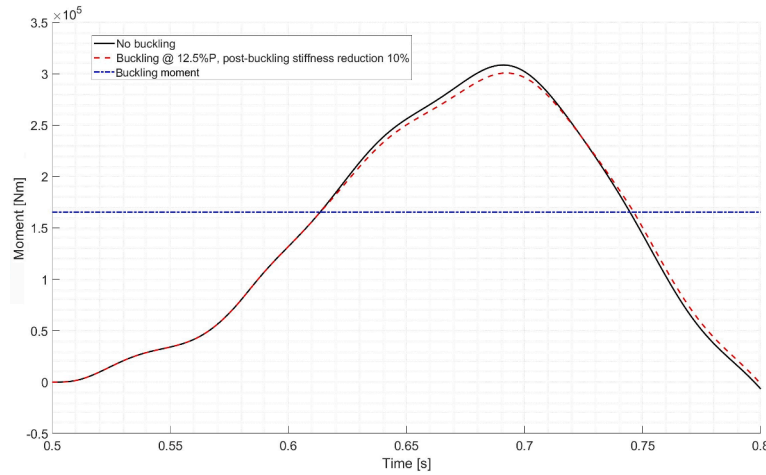


Fig. 20. Time response of the moment M at the scalar point of the aeroelastic stick model.

carried out on this model to evaluate the peak load alleviation produced by the proposed configuration.

As written in Section 2, NeoCASS performs the dynamic gust and trim analyses of the stick model separately, and since both are linear, their internal loads are summed for the structural analysis carried out in ABAQUS. For the buckling moment to be inserted into the stick model, it is first necessary to determine what fraction of the dynamic gust load it corresponds to. The buckling moment may be equal to a part of the static (trim) loads, or to the whole static loads plus a part of the dynamic loads. If the latter holds, the wing does not buckle during cruise, but only when subjected to a gust.

To identify this fraction, Fig. 18 reports, for each reference bar of the aeroelastic stick model, the moment about the aircraft x -axis, aligned with the symmetry line of the fuselage. In each column, the yellow column is the trim moment and the blue column is the sum of the trim and gust moments, so their difference gives the gust contribution at that node. The buckling moment is indicated by the red box: since the non-linear FE analysis showed that the skin buckles when the applied load reaches 61% of the total, the buckling moment at each node is taken as 61% of that node's total (trim + gust) moment.

Observing the results shown in Fig. 18, the buckling moment (61% of the total moment) is higher than the trim moment at every node, meaning that the proposed configuration only enters the post-buckling regime under gust conditions, not during trimmed cruise. At the bar where buckling occurs (ID = 2003), the buckling moment exceeds the trim moment by only a small margin. In Fig. 18, this margin is the distance from the top of the yellow column to the top of the red box, and it is the additional moment that the gust must produce to trigger buckling. It corresponds to 12.5% of the gust moment at that bar, so that once the aircraft is trimmed, a gust generating only 12.5% of its maximum moment at that location is already enough to make the skin buckle. Since the parameter P1 in the sensitivity map is defined relative to the gust moment, this 12.5% is the value inserted into the stick model. Combined with the average post-buckling stiffness reduction of 10% (P2), Fig. 19 gives the corresponding WRBM reduction: the marker shows the result for the proposed configuration, equal to 1.41%. The maximum WRBM is retained as the performance metric because the present analysis considers discrete gust encounters and is aimed at reducing the peak load governing structural sizing. RMS-based response metrics are therefore not considered, as they are more naturally associated with continuous-turbulence analyses in which the excitation and structural response are described statistically.

The reduction of the WRBM results in lower peak loads registered in the wing when subjected to gusts. This is evidenced by the time response of the moment M at the scalar point of the aeroelastic stick model

throughout the dynamic transient gust analysis. Fig. 20 shows the time window of the analysis in which the highest values of M occur. Once M reaches the buckling moment $\bar{M}$, the proposed configuration exhibits a decrease in M compared to the original model, resulting in lower peak loads.

4. Conclusions

In this study, a design framework was developed to exploit the buckling of the skin of a composite wingbox to passively reduce the peak loads experienced by an aircraft wing during gusts. The localized stiffness reduction induced by the post-buckling regime is introduced into a standard linear aeroelastic stick model as a bilinear rotational stiffness, while a three-dimensional FE model of the wingbox is used to design a structure that buckles accordingly and to verify its integrity. Because the framework relies on conventional low-cost aeroelastic tools, the load-alleviation potential of post-buckling behavior can be assessed in early aircraft design stages, where thousands of load cases must be evaluated.

The application to a high aspect-ratio composite wing leads to the following conclusions. The parametric study on the aeroelastic model shows that the Wing Root Bending Moment (WRBM) can be reduced by up to 12% when buckling is induced in the skin near the wing root.

The design of the wingbox demonstrates, however, that the structural integrity of the wing limits the post-buckling stiffness reduction that can be achieved in practice. The removal of ribs and stringers proves essential to enlarge the buckling region and further reduce the post-buckling stiffness, but it promotes ply failure of the skin and produces large buckling areas that may lead to a loss of airfoil shape.

The final wing configuration achieves a reduction in total mass of 1.7% compared to the original model, with the skin buckling at 12.5% of the gust load and an average post-buckling stiffness reduction of 10%. The resulting WRBM reduction is 1.41%, significantly lower than the 12% theoretical potential identified by the aeroelastic parametric study. This gap follows from the structural integrity constraints encountered during the wingbox design and from the modeling simplifications discussed below. Nevertheless, the result shows that the WRBM can be reduced passively while simultaneously decreasing the aircraft mass, and without introducing the complexity of active load-alleviation systems.

As discussed in Section 3.2, the computation of the post-buckling stiffness reduction represents the main limitation of the present study. The reduction varies along the span, but it is introduced into the aeroelastic model as a single averaged value applied at the buckling location, since reproducing its spatial distribution would require a multi-point constraint at every node of the stick model, negating the computational advantage of the low-fidelity approach. Moreover, it is extracted

from the vertical displacement of a single node of the three-dimensional model, whereas buckling develops over a three-dimensional region. Finally, the buckling field develops in the rear portion of the upper skin, resulting in a forward shift of the elastic axis which lowers the local angle of attack, while the concurrent loss of torsional stiffness acts in the opposite sense. Neither is captured by the single bending DOF of the scalar point, and quantifying their net effect on the local angle of attack requires extending the formulation with a torsional DOF.

A certification-oriented extension of the framework should also consider design load conditions other than gusts. Since the proposed buckling mechanism is triggered by the structural load level rather than by the nature of the excitation, the same critical buckling moment may also be exceeded during maneuver conditions. The resulting post-buckling stiffness redistribution could therefore modify the structural and aeroelastic response under maneuver loads. Consequently, the buckling threshold should ultimately be designed considering the complete set of relevant load conditions, including both gust and maneuver envelopes.

Despite these limitations, the proposed framework is considered adequate for preliminary aircraft design stages, where computational efficiency and rapid assessment of structural-aeroelastic effects are prioritized over a detailed representation of localized post-buckling phenomena.

CRedit authorship contribution statement

Pedro Farinha: Writing – review & editing, Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Data curation; **Francesco Toffol:** Writing – original draft, Visualization, Methodology, Investigation, Formal analysis, Data curation, Conceptualization; **Chiara Bisagni:** Writing – review & editing, Supervision, Resources, Project administration, Methodology, Investigation, Funding acquisition, Conceptualization.

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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Data availability

The data used for this research is available as open-access with the DOI link: <https://doi.org/10.5281/zenodo.21338532>.

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