OPTIMAL WEIGHT DESIGN OF LAMINATED COMPOSITE PANELS WITH DIFFERENT STIFFENERS UNDER BUCKLING LOADS

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Abstract

The present investigation is devoted to development of new optimal design concepts that exploit the full potential of advanced composite materials in aircraft lateral wing upper covers. Three rib bays laminated composite panels with T, I and HAT-stiffeners are modelled with ANSYS and NASTRAN finite element codes to investigate their buckling behaviour in dependence on skin and stiffener lay-ups, stiffener height, stiffener top and root width. Due to the large dimension of numerical problems to be solved, an optimisation methodology is developed employing the method of experimental design and response surface technique. Weight optimisation problems are solved for laminated composite panels with three types of stiffeners, two stiffener pitches and four load levels taking into account manufacturing, repairability and damage tolerance requirements. Optimal results are verified using ANSYS shared-node and NASTRAN rigid-linked models.

1 Introduction

The European Aeronautics Industry's Strategic Research Agenda addresses the reduction in operating costs of relevant European aerospace products by 15%, through the cost effective application of carbon fibre composites to aircraft primary structure, taking into account systems integration. This can be achieved by realising the weight saving potential of advanced composite materials, by reducing the manufacturing costs of composite components and by reducing subsequent product maintenance costs.

Due to increasing application of advanced composites in aircraft structures, significant progress has been achieved recently in the optimisation of stiffened laminated composite panels [1-8]. The optimisation methodology based on genetic algorithms has been developed in paper [1] for the design of blade stiffened panels with stability and strain constraints. The optimisation problem is formulated as finding the stacking sequences of skin, stiffener blade and flange laminates, as well as the stiffener height in order to minimise the weight of stiffened panels.

Minimum weight design of T-stiffened and HAT-stiffened panels made of laminated composites is performed in paper [2] with the PANDA2 program. The panels are subjected to axial compression, in-plane shear and normal pressure loads, and designed for service in their locally post-buckled states. PANDA2 program has been used also to obtain an initial optimum structural design for a HAT-stiffened laminated composite panel used for the airplane upper covers in paper [3]. A refined optimum structural design has been obtained then by an optimisation using response surface technique.

HAT-stiffened panel is subjected in this study to internal pressure load and to combined internal pressure and in-plain loads. The structural optimisation problem is formulated using the panel weight as the objective function with stress and buckling constraints. As the design variables, the spacing of HAT-stiffeners, and thickness of skin and components of HAT-stiffener are taken into consideration. At the same time the stacking sequence of the preforms used in the skin and in all of the components of HAT-stiffeners is examined in this paper as a constant value. The response surface methodology has been applied also in paper [4] to optimise the dimensions of HAT-stiffened composite panels and stacking sequences under buckling constraints. In this case the optimisation method is based on modified efficient global optimisation with the multi-objective genetic algorithm and kriging response surface. As an advantage of the present methodology, a feasible optimal structure at a low computational cost could be examined.

A method to optimise long anisotropic laminated fibre composite panels with T-shaped stiffeners has been developed in paper [5], where the optimisation problem is solved in two steps. At the first step, continuous optimisation of lamination parameters with gradient-based techniques is used to get near the optimum discrete design. In this step the cross-sectional dimensions and values of the lamination parameters for an optimum superstiffener design are obtained, and strength, buckling and practical design rules are taken as the design constraints. At the second step, a genetic algorithm is used to identify the lay-ups for the superstiffener's laminates, which are the closest in the lamination parameter space to the continuous optima (minimum-distance approach) and satisfies to the discrete design constraints. However, sometimes the optimum discrete designs are not the closest in the lamination parameter space to the continuous optima. On this reason a new second-step optimisation that uses a genetic algorithm to find the safest design based on a linear approximation of the design constraints, instead of searching for the closest design in the lamination parameter space to the continuous optimum, has been successfully developed in paper [6].

An optimised design of laminated composite panels with other types of stiffeners, namely Z-stiffeners and squared tubes, is examined in papers [7] and [8]. A bilevel optimisation strategy for a fast design of composite stiffened panels, using VICONOPT and embracing practical composite design rules, has been developed and applied for the design of highly strained Z-stiffened composite panels in paper [7], where the stacking sequences satisfying laminate design rules are found using an optimisation at the laminate level. The objective of the design problem in paper [8] is to maximise the buckling loads of panels with squared tubes used as stiffeners by optimally oriented fibre plies.

In paper [9] a system for classifying complexity in optimisation problems based on model size, analysis procedure, and optimisation size and methodology is described for the composite stiffened shells and plates. At present time investigations on the optimal design of stiffened laminated composite panels continues taking into account more new design rules obtained from the airspace industry.

The present investigations are devoted to the methodology development based on the planning of experiments and response surface technique for optimal design of stiffened laminated composite panels with special emphasize on more close conformity of the developed finite element analysis and operational requirements for aircraft lateral wing upper covers. This study gives the possibility to compare optimal solutions obtained for the laminated composite panels with different stiffeners.

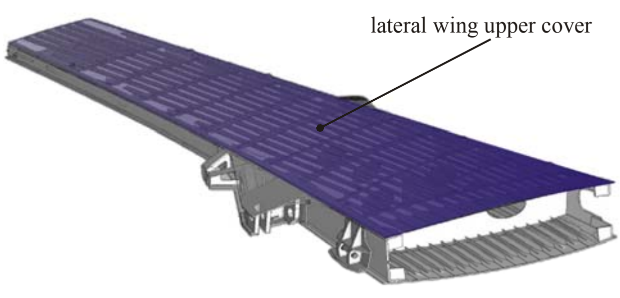

Fig. 1. Lateral wing of aircraft.

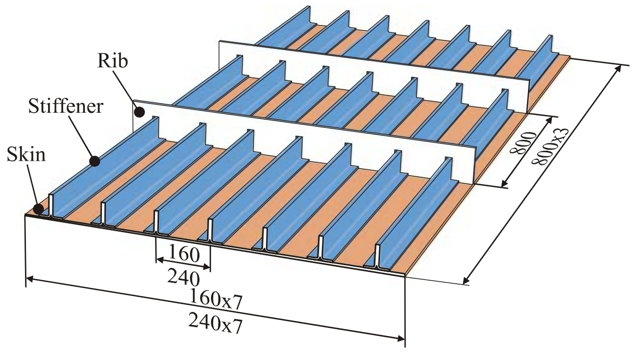

Fig. 2. Composite panel with T-stiffeners.

2 Construction of Stiffened Composite Panels

An aircraft lateral wing upper cover (Fig. 1) in the present study is examined as a plane stiffened panel (Fig. 2) consisting of skin, ribs and stiffeners made from unidirectional Intermediate Modulus carbon fibre prepreg tapes (T800/M21 or IMS/977-2) with the following material properties $E_1 = 157\ \text{GPa}$, $E_2 = 8.5\ \text{GPa}$, $G_{12} = 4.2\ \text{GPa}$, $\upsilon_{12} = 0.35$, $\rho = 1600\ \dfrac{\text{kg}}{\text{m}^3}$ and thickness of one cured ply $t = 0.25\ \text{mm}$. In the linear buckling analysis material properties are reduced by the value of safety coefficient $k = 1.2$. The stiffened panel is attached to the wing box structure by ribs (Fig. 3) perpendicular to stiffeners with the rib pitch 800 mm. The buckling analysis with a set of four load cases (100, 500, 1000 and 1500 kN per stiffener bay) is carried out for the panels with T, I and HAT-stiffeners (Fig. 4) and two stiffener pitches 160 and 240 mm in the present study to identify the global, skin and stiffener local buckling loads.

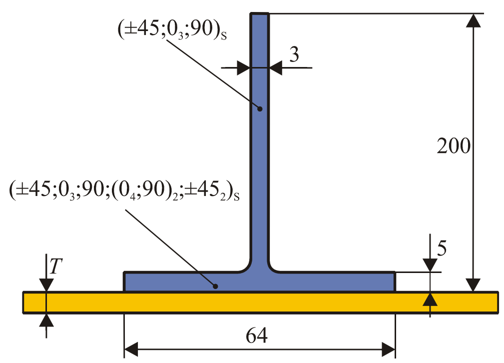

Fig. 3. Skin with rib.

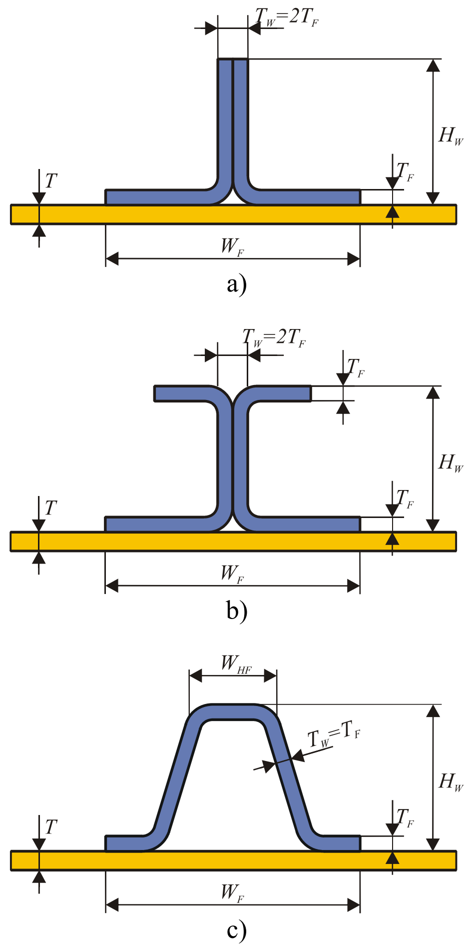

Fig. 4. Skin with stiffener:
a) T, b) I, c) HAT.

3 Finite Element Modelling and Buckling Analysis

Different finite element modelling approaches and boundary conditions are analysed for the stiffened composite panels to achieve the best accuracy with an acceptable computational time in optimisation process. For this purpose three reference configurations of three rib bays laminated composite panel with T-stiffeners (Table 1) are used. To investigate the buckling behaviour of these panels, several finite element models are developed with ANSYS using 8-node quadratic shell elements SHELL99 and NASTRAN using 4-node laminate finite elements CQUAD4.

3.1 Modelling of Skin-Stiffener Interface

Two basic approaches of skin-stiffener interface modelling: shared-node and element-linked

Table 1. Stiffened panel configurations.

Panel configuration	H_W, mm	W_F, mm	Skin lay-up	Stiffener lay-up
1	18.5	40	$(\pm45;0_4;90;0)_S$	$(\pm45;0_3;90;0_4;90)_S$
2	45	75	$(\pm45;(0_4;90)_4;0)_S$	$(\pm45;0_3;(90;0_4)_3;90)_S$
3	70	110	$(\pm45_2;(0_4;90)_7;0)_S$	$((\pm45;0_3)_2;(90;0_4)_5;90)_S$

Table 2. First buckling load for different skin-stiffener interface finite element models.

Panel configuration	NASTRAN	ANSYS			
	Rigid-linked model	Beam-linked model		Shared-node model	
	P, kN	P, kN	Δ, %	P, kN	Δ, %
1	169	167	1.2	167	1.2
2	4022	3982	1.0	3982	1.0
3	21151	21002	0.7	20923	1.1

(Fig. 5) are studied in the present investigation. In the first approach the ANSYS nodal plane offset option is used for a skin-stiffener interface modelling employing shared node technique, but in the second approach skin and stiffeners are bonded together by 3D beam finite elements in ANSYS and by rigid link elements in NASTRAN model. It is necessary to note that the beam stiffness of finite elements used as rigid links between skin and stiffener flange in ANSYS does not show a significant influence on the first buckling loads in the Young's modulus interval $E = 7,...,655\,\text{GPa}$.

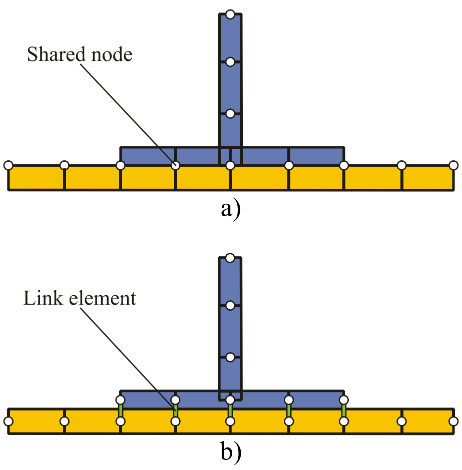

Fig. 5. Skin-stiffener interface:
a) shared-node model,
b) element-linked model.

The ANSYS shared-node and beam-linked models as well as the NASTRAN rigid-linked model demonstrate a good agreement in terms of critical buckling loads (Table 2) and mode shapes for three examined stiffened panel configurations. On this reason for optimisation the finite element model with less dimension, namely the ANSYS shared-node model will be used and results will be verified with the NASTRAN rigid-linked model.

3.2 Modelling of Panel Ribs

Two finite element models of panel ribs are developed in the present study: rib model, where rib is presented with finite elements and non-rib model, where rib is substituted by simply-supported boundary conditions along the nodal line presenting skin-rib flange interface nodes in the shared-node model and both skin-rib flange nodes in the beam-linked model. It is necessary to note that no significant difference is observed for the first buckling loads (Table 3) and corresponding mode shapes obtained using rib and non-rib finite element models. On this reason the non-rib model is used for calculations to decrease considerably the dimension of finite element problems to be solved repeatedly in time of optimisation.

Table 3. First buckling load for different rib finite element models.

Panel configuration	Model	ANSYS			
		Beam-linked model P, kN	Δ, %	Shared-node model P, kN	Δ, %
1	Non-rib	167	0.6	167	0.6
	Rib	168		168	
2	Non-rib	3982	0.5	3982	0.4
	Rib	4000		3997	
3	Non-rib	21002	0.4	20923	0.2
	Rib	21087		20962	

3.3 Modelling of Boundary Conditions

To follow operational requirements for lateral wing upper covers, the boundary condition set presented in Fig. 6 is used in the present study. In this case symmetry boundary conditions ($U_x = 0, R_y = 0, R_z = 0$) are applied on free edges of stiffened panel, and panel ends are simply supported and able to rotate around X-axis. Load is applied to the skin-stiffener interface nodes gathered by coupled set CP U_z. In this approach an eccentricity of load exists. To eliminate this phenomenon, it is necessary to introduce additional bending moment applied in the effective centroid of overall cross-section. Only in this case no bending occurs under static compression load. This approach is more correct than the boundary condition set used, but it requires additional calculations and does not give any reasonable benefits for analysis of critical buckling loads. This is why the boundary condition set presented in Fig. 6 is chosen for the buckling analysis of stiffened panels under compression load.

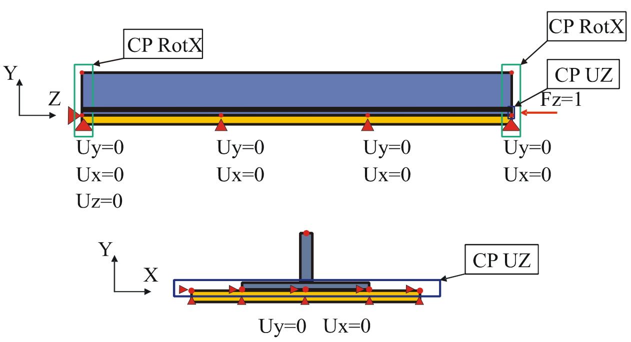

Fig. 6. Boundary conditions (CP – coupled).

4 Formulation of Optimisation Problem

An optimisation problem is formulated as the minimum weight design problem in term of cross-section area for one stiffener pitch

$$A(X_1,...,X_7) \Rightarrow \min \qquad (1)$$

The combined influence of skin and stiffener lay-ups (X_1, X_2, X_3, X_4), stiffener height (X_5), stiffener top and root width (X_6, X_7) on the buckling behaviour of composite panels is investigated. On this reason the design parameters are determined as follows

$$\text{Skin lay-up: } \left[\pm 45_{X_1}, (0_4, 90)_{X_2}, 0\right]_S$$

$$\text{Stiffener lay-up: } \left[\pm 45_{X_3}, (0_4, 90)_{X_4}, 0\right]_S \qquad (2)$$

$$H_W = X_5,\; W_{LF} = X_6,\; W_{HF} = X_7$$

The constraints on manufacture, repair and exploitation concerning to minimum fibre percentage in each direction, Poisson's ratio mismatch between skin and stiffener flange laminates, and damage tolerance are taken into consideration. From the design requirements for aerospace constructions the laminates should be symmetric and well balanced, and at least 8% of fibres should be in each direction

$$\frac{\sum 0^0}{\sum (0^0, 45^0, 90^0)} \geq 0.08$$
$$\frac{\sum 45^0}{\sum (0^0, 45^0, 90^0)} \geq 0.08 \qquad (3)$$

$$\frac{\sum 90^0}{\sum(0^0,45^0,90^0)} \geq 0.08$$

To avoid ply blocking, no more than 1 mm thickness of material in any orientation should be sequential or not more than 4 plies of the same orientation together should be presented in construction. A presence of two outermost plies on each side of the laminate oriented at $\pm 45^0$ is contributed to minimisation of impact effects. To minimise delamination effects, Poisson's ratio mismatch between skin and stiffener flange laminates should be less than 0.15

$$\left|\nu_{skin} - \nu_{flange}\right| \leq 0.15 \qquad (4)$$

The repairability requirements are checked for the bolted repairs on stiffener using the following repairability criteria

$$\begin{gathered} W_{LF} \geq k * d + R_W \\ 2 * W_{LF} + W_{HF} + 2 * H_W / tg\alpha \leq 180 \text{ mm} \end{gathered} \qquad (5)$$

where $d = \max\{3\text{ mm}, T_S/1.5, T_{LF}/1.5\}$, k=4.5, $R_W = 5\text{ mm}$. It is necessary to note that the residual strength after impact is checked for skin using AIRBUS methods [10] and the damage tolerance in this case is satisfied to the following condition: $\varepsilon \leq \varepsilon_{\max}$, where $\varepsilon_{\max}$ is allowable strains calculated by ACAT tool [10] developed in AIRBUS and approximated as the second order polynomial function to use it later in optimisation process.

To follow numerous manufacturing, repairability and exploitation requirements, the design parameters satisfy to the following constraints:

- for the panels with T and I-stiffeners

$$\begin{gathered} 1 \leq X_1 \leq 4,\ 1 \leq X_2 \leq 8, \\ 1 \leq X_3 \leq 4,\ 1 \leq X_4 \leq 8, \\ 18.5 \leq X_5 \leq 70\text{ mm}, \\ 41 \leq X_6 \leq 110(180)\text{ mm}, \\ 25 \leq X_7 \leq 110(180)\text{ mm} \end{gathered} \qquad (6)$$

- for the panels with HAT-stiffeners

$$\begin{gathered} 1 \leq X_1 \leq 4,\ 1 \leq X_2 \leq 8, \\ 1 \leq X_3 \leq 4,\ 1 \leq X_4 \leq 8, \\ 30 \leq X_5 \leq 70\text{ mm}, \\ 18.5 \leq X_6 \leq 34.5(69.5)\text{ mm}, \\ 25 \leq X_7 \leq 57(127)\text{ mm} \end{gathered} \qquad (7)$$

where in the brackets are given values related only for the panels with stiffener pitches 240 mm.

5 Solution of Optimisation Problem

Due to the large dimension of numerical problem to be solved, an optimisation methodology is developed employing the method of experimental design and response surface technique (Fig. 7). The basic idea of this approach is that simple mathematical models (response surfaces) are determined only using the finite element solutions in reference points of experimental design. The significant reduction in calculations is achieved in this case in comparison with conventional optimisation methods.

At the beginning of optimisation procedure the critical buckling loads and strains are calculated in 150 points of experimental design generated by the minimal square distance Latin Hypercube sampling method [11]. After finite element calculations, the second order polynomial functions are obtained for the critical buckling loads and strains using the conventional un-weighted least square method [12] with elimination of some points

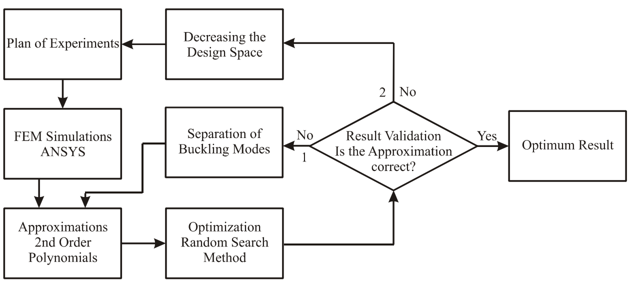

Fig. 7. Optimisation procedure.

Table 4. Optimal cross-section areas (A, mm^2) of stiffened composite panels.

Panel configuration		Load levels			
		100 kN	500 kN	1000 kN	1500 kN
160 mm stiffener pitch	T-stiffener	**898**	1807	2356	3254
	I-stiffener	902	1622	**2228**	**2871**
	HAT-stiffener	1148	**1392**	2398	3184
240 mm stiffener pitch	T-stiffener	**1454**	2528	3290	3908
	I-stiffener	1477	2507	3223	3819
	HAT-stiffener	1472	**2026**	**2526**	**3532**

$$\bar{F}(x) = \beta_0 + \sum_{i=1}^{K} \beta_i x_i + \sum_{i=1}^{K} \sum_{j=i}^{K} \beta_{ij} x_i x_j \qquad (8)$$

To increase accuracy, not more than 15 points with the maximal relative deviations are excluded from calculations of polynomial functions in each optimisation problem. This is less than 10% from the maximal points of the plan of experiments. Then non-linear optimisation problem is solved by the random search method [13] with the purpose to minimise the stiffened panel cross-section area for each specified load level. Optimal results are checked using ANSYS finite element solutions and analytical tool developed in AIRBUS [10]. If the difference in optimal point is higher than 5%, a separation of buckling modes or a decrease of design space for some parameters are executed before obtaining the final optimal solution. This is done to improve the correlation of approximating functions. As an example, the minimal weight design has been examined in detail for the laminated composite panel with T-stiffeners and stiffener pitch 160 mm in paper [14].

Final optimal results for three types of stiffeners, both stiffener peaches and four load levels are presented in Table 4. They are verified by the buckling analysis using ANSYS shared-node and NASTRAN rigid-linked models. Good coincidence of results is observed for different finite element models and approximations. The damage tolerance is checked additionally using allowable strains obtained with ACAT tool [10]. The constraints on this tolerance are satisfied also with the obtained optimal design parameters. Table 4 shows that the panel with HAT-stiffeners and stiffener pitch 240 mm has the best weight/design performance and by this way could be recommended for an application in the aircraft lateral wing upper covers. Optimal parameters for this panel are given in Table 5 together with the optimal results verification by the NASTRAN and ANSYS solutions. It is necessary to note that the optimal solution in this case comes on the lower boarder of the design parameters for the load level of 100 kN. On this reason it is considerably overestimated and the reserve factor is larger than 1 for the critical buckling load. The examined panel with the best weight/design performance has been used additionally for the following non-linear buckling analysis to study an effect of shear and fuel pressure on the performance of stiffened composite panels and their behaviour under skin post-buckling [15].

6 Conclusions

The methodology for weigh optimal design of laminated composite panels with T, I and HAT-stiffeners used as lateral wing upper covers has been developed based on the planning of experiments and response surface technique. Optimal solutions have been obtained for three types of stiffeners, two stiffener pitches and four load levels. Analysing the optimal results, panel with HAT-stiffeners and stiffener pitch 240 mm can be recommended from the weight point of view.

To considerably decrease the dimension of numerical problems to be solved, more attention should be paid for the optimal finite element mesh. Such mesh can be obtained from the

Table 5. Optimal parameters of laminated composite panel with HAT-stiffeners and stiffener pitch 240 mm.

Load, kN	100	500	1000	1500
Skin lay-up	$(\pm45,0_4,90,0)_s$	$(\pm45_2,0_4,90,0)_s$	$(\pm45,(0_4,90)_2,0)_s$	$(\pm45,(0_4,90)_3,0)_s$
Stiffener lay-up	$(\pm45,0_4,90,0)_s$	$(\pm45,0_4,90,0)_s$	$(\pm45,0_4,90,0)_s$	$(\pm45,(0_4,90)_2,0)_s$
H_W, mm	30	48.5	55.5	56
W_{HF}, mm	25	44.5	60	25.5
W_{LF}, mm	18.5	28.5	30.5	32
A, mm^2	1472	2026	2526	3532
$\varepsilon_{allowable}$, μstrain	3642	3646	3410	3891
NASTRAN				
P, kN	160	478	947	1463
RF	1.60	0.96	0.95	0.98
Mode	skin	skin	skin	panel
ANSYS				
P, kN	167	486	1007	1499
RF	1.67	0.97	1.01	1.00
Mode	skin	skin	skin	panel
ε, μstrain	602	2787	3407	3523
RF	6.05	1.31	1.00	1.10
RF – reserve factor				

convergence study, using shared-node finite element models instead of element-linked models and substituting the rib finite element model with the non-rib model, where ribs are presented with simply supported boundary conditions. Since the critical buckling loads are very sensitive to the boundary conditions, more attention should be paid also for an exact modelling of load application and displacement constraints.

Acknowledgements

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