THICK PLY VERSUS THIN PLY COMPOSITE LAMINATE STIFFENED PANEL BUCKLING AND POST-BUCKLING BEHAVIOR

Thaís B. C. Braz[1], Carlos A. Cimini Jr.[1], Per Wennhague[2], Dan Zenkert[2] and Tonny Nyman[2,3]

[1]Department of Structural Engineering, School of Engineering, Federal University of Minas Gerais
Av. Antônio Carlos, 6627 Pampulha, 31270-901, Belo Horizonte, MG, Brazil
thais-bcb@ufmg.br, carlos.cimini@gmail.com
[2]Department of Aeronautical and Vehicle Engineering, Royal Institute of Technology
Teknikringen 8, 5th floor, SE-100 44, Stockholm, Sweden
wennhague@kth.se, danz@kth.se
[3]SAAB AB, Broderna Ugglas gata, SE-581 88 Linköping, Sweden
tonny.nyman@saabgroup.com

Keywords: stiffened panel, thin ply, post-buckling analysis, finite element analysis

ABSTRACT

For their weight reduction, thin web panels are often used in airspace industry. In this study, a basic fuselage section is represented by a stiffened panel composed by skin, frame, stiffeners and attachments, as conceived by Arakaki and Faria [1]. In this panel design, the structure withholds large loads after the buckling of the web. This study presents a comparison between thick and thin ply composite laminates behavior under shear buckling loads. The two concepts were modeled using the commercial finite element software Abaqus®. Buckling and post-buckling nonlinear behavior for both thick ply and thin ply laminates was analyzed and results were compared.

1 INTRODUCTION

Weight reduction has always been an issue in aerospace industry, and for that the use of slenderer structures is mostly advantageous. Almost a century ago, Wagner [2] introduced a metallic panel design concept of very thin web with upper and lower flanges and vertical stiffeners. The weight reduction was clearly very advantageous and the core of that concept was his finding that although the web would easily buckle, the panel could also endure much higher loading after buckling. The buckling of structures, as well as the post-buckling behavior, has so been increasingly the subject of research and study.

The study of the behavior of structures on post-buckling regimen relied on well-established semi-empirical equations which are losing space with the development of alternative materials end geometric designs. In order to facilitate this study, the use of Finite Element (FE) methods is becoming increasingly common. On these grounds, Murphy et al [3] developed a few guidelines for the non-linear computational analysis of post- buckling response of stiffened fuselage panels under shear loading.

The classical orthotropic or anisotropic plate theory can be used when the plates are symmetrically laminated, but for unsymmetrical laminates more complex theories are needed [4]. Chamis [5] made a numerical formulation for the problem of buckling of laminated composites or fiber reinforced homogeneous materials using the Galerkin method. For laminated composites, the buckling and post-buckling response are sensitive to a variety of factors besides the structure's geometry and loading conditions. Reddy and Khdeir [6] studied the effects of thickness ratio, lamination schemes and aspect ratio on composite plates critical buckling loads.

Orifici et. al. [7] made a study that compared NASTRAN®, Abaqus® and LS-DYNA® solvers with experimental results for buckling and post buckling response of stiffened composite structures. It concluded that all codes somewhat overestimated panel post-buckling stiffness and comparing the software codes, the implicit solvers all gave very similar results, though Abaqus® and NASTRAN® SOL 106 use a more robust and efficient approach to handle convergence difficulties. LS-DYNA® predictions showed very good comparison prior to global buckling, which may be due to material degradation that was not taken in account. It also showed that the buckling shape varied between the solvers and the

strain data, when compared to experimental results, showed to be not well predicted by all software codes used.

More recently, Arakaki and Faria [1] used NASTRAN® and Abaqus® as well as experimental tests to study different conceptions of a reinforced composite panel with aeronautical application. The Wagner [2] panel concept was used to design a stiffened composite panel and the influence of attachments in its behavior under a diagonal tension field as well as skin and stiffener interface alternatives were studied.

Advancements in laminated composites fabrication open up new paths in composite research. Tsai et al. [8] introduced a new thin ply laminate fabrication method that spreads large tows of fiber without damage, which allows laminate plies as thin as 1/6 of conventional thickness. The work also showed that thin ply laminates can be a quite favorable alternative as micro damage such as delaminations can be avoided with the use of those. In later work, Sihn et. al. [9] measured mechanical properties of quasi isotropical thin and thick ply laminates and found that the thin ply specimen had both Young's modulus and longitudinal tensile strength higher than the thick ply version.

In this study, Arakaki and Faria [1] panel concept was modeled and used as base for further analysis of the buckling and post-buckling behavior of the structure, introducing a different configuration of the web with the use of thin ply laminate.

2 METHODS

Wagner [2] introduced a panel design concept of very thin web with upper and lower flanges and vertical stiffeners. The weight reduction is very advantageous for aerospace applications and although the web would easily buckle, it could also endure much higher loading after buckling. The same concept was used in Arakaki and Faria [1] to design a stiffened composite panel design for pressurized fuselage, which was used as foundation for this work and is shown in Figure 1.

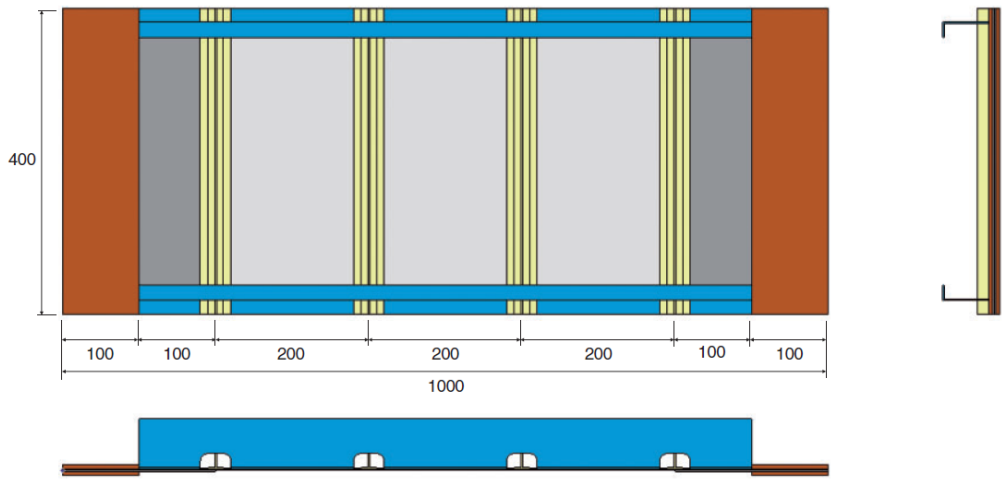

Figure 1: Panel design and dimensions [1]

The panel is composed by skin, with smaller thickness in the center part than the edges, two longitudinal "Z" shaped frames, which have small holes at the base to fit the stiffeners, and four stiffeners, where the two at the center are "T" shaped typical stiffeners and the two at the borders are transition stiffeners with longer flanges towards the edges of the plate. The materials used were woven and tape carbon/epoxy, and their properties are displayed in Table 1.

Table 1: Material properties

Material	E_1 (MPa)	E_2 (MPa)	ν_{12}	G_{12} (MPa)	Thickness (mm)
Woven Carbon/Epoxy	60,800	58,250	0.07	4,550	0.21
Tape Carbon/Epoxy	125,450	9,450	0.32	4,700	0.19

A finite element model of the test plate was made using the commercial software *Abaqus*®. The model was made considering the skin and attachments were considered integrated. As shown in Figure 2, one edge of the panel is clamped, while the other one is attached to a rigid loading tab that assures that the loading is applied vertically and distributed through all of that edge of the panel. A midpanel point (point A in red, Figure 2) was selected to compare simulation predictions to experimental results. In the experiment of Arakaki and Faria [1], back-to-back rosette strain gages were attached at this point to evaluate point strain in both faces of the skin.

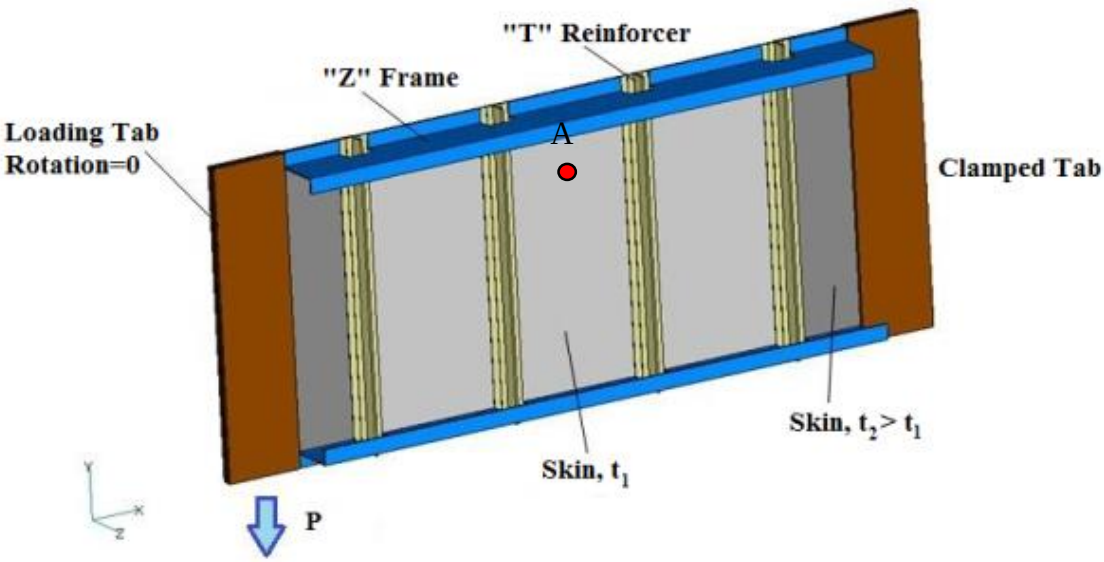

Figure 2: Panel design, boundary and loading conditions [1].

The first step performed on the model is an Eigenvalue buckling analysis. The *Abaqus* buckling step is a linear perturbation procedure, where the applied load is increased until the model stiffness matrix becomes singular. It turns to a simple eigenvalue problem involving the stiffness matrix:

$$\left(K_0^{NM} + \lambda_i K_\Delta^{NM}\right) v_i^M = 0 \tag{1}$$

Where K_0^{NM} is the stiffness matrix corresponding to the base state; K_Δ^{NM} is the differential initial stress matrix and load stiffnes matrix due to increment loading pattern; λ_i the eigenvalues; v_i^M are the buckling mode shapes; *M* and *N* refer to the whole model's degrees of freedom and *i* to the *i*th buckling mode [10].

The post-buckling response is predicted using the Riks method. This method can provide complete information about the structure collapse as it considers the nonlinear geometry to predict the unstable response. The Riks procedure assumes that the load magnitude varies with a single scalar parameter, the *load proportionality factor* (*LPF*). The response path is assumed to be smooth and it is calculated from the nodal variables and the loading method with an algorithm similar to the Newton method. To limit the increment size the variable "arc length" along the static equilibrium path in load-displacement space is defined. The total loading during a Riks step P_{total} is always proportional and can be described by

the following equation, where the first term P_0 is the "dead load", defined in the beginning of the step and is fixed, P_{ref} is the reference load vector and λ is the "load proportionality factor" [10].

$$P_{total} = P_0 + \lambda(P_{ref} - P_0) \tag{2}$$

The analysis was performed in two models with the same geometry and loading conditions. The conventional thick ply model was compared to the one from Arakaki and Faria [1] and then a thin ply model was then developed in order to verify the influence of this parameter on the buckling behavior of the stiffened panel. Thin ply thickness was considered ¼ of thick ply thickness. Panel components were kept with the same thickness, therefore including 4 times more layers. The panel thicknesses and layups for each situation are specified in Table 2.

Table 2: Panel layup definitions

Panel Region	Total Thickness (mm)	Original Layup	Thin Ply Layup
Central Region	t_1=1.47	$[\pm45/(0,90)_2/\pm45(0,90)_2/\pm45]$	$[\pm45/(0,90)_2/\pm45(0,90)_2/\pm45]_S$
Outer Region	t_2=3.99	$[\pm45/(0,90)_2/\pm45(0,90)_2/\pm45(0,90)_2/\pm45]_S$	$[\pm45/(0,90)_2/\pm45(0,90)_2/\pm45(0,90)_2/\pm45]_{4S}$

3 RESULTS

Firstly, Arakaki and Faria [1] model for thick ply laminates was reproduced, in order to verify how it compared to both their simulation and test results. The linear buckling analysis was performed. The critical buckling load for the thick ply laminate was 6665 daN and the first mode shape is shown in Figure 3. The post buckling analysis was made using the Riks method. The diagonal waves formed can be observed in Figure 3.

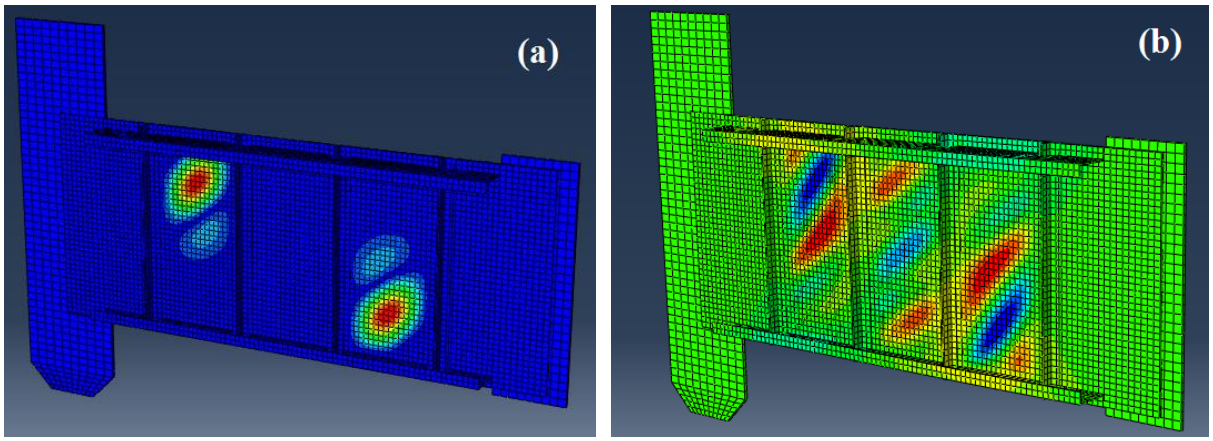

Figure 3: First buckling mode (a) and postbuckling behavior (b)

The thick ply laminate strain versus load plot was obtained and in Figure 4 the results were plotted together with Arakaki and Faria [1] results for comparison.

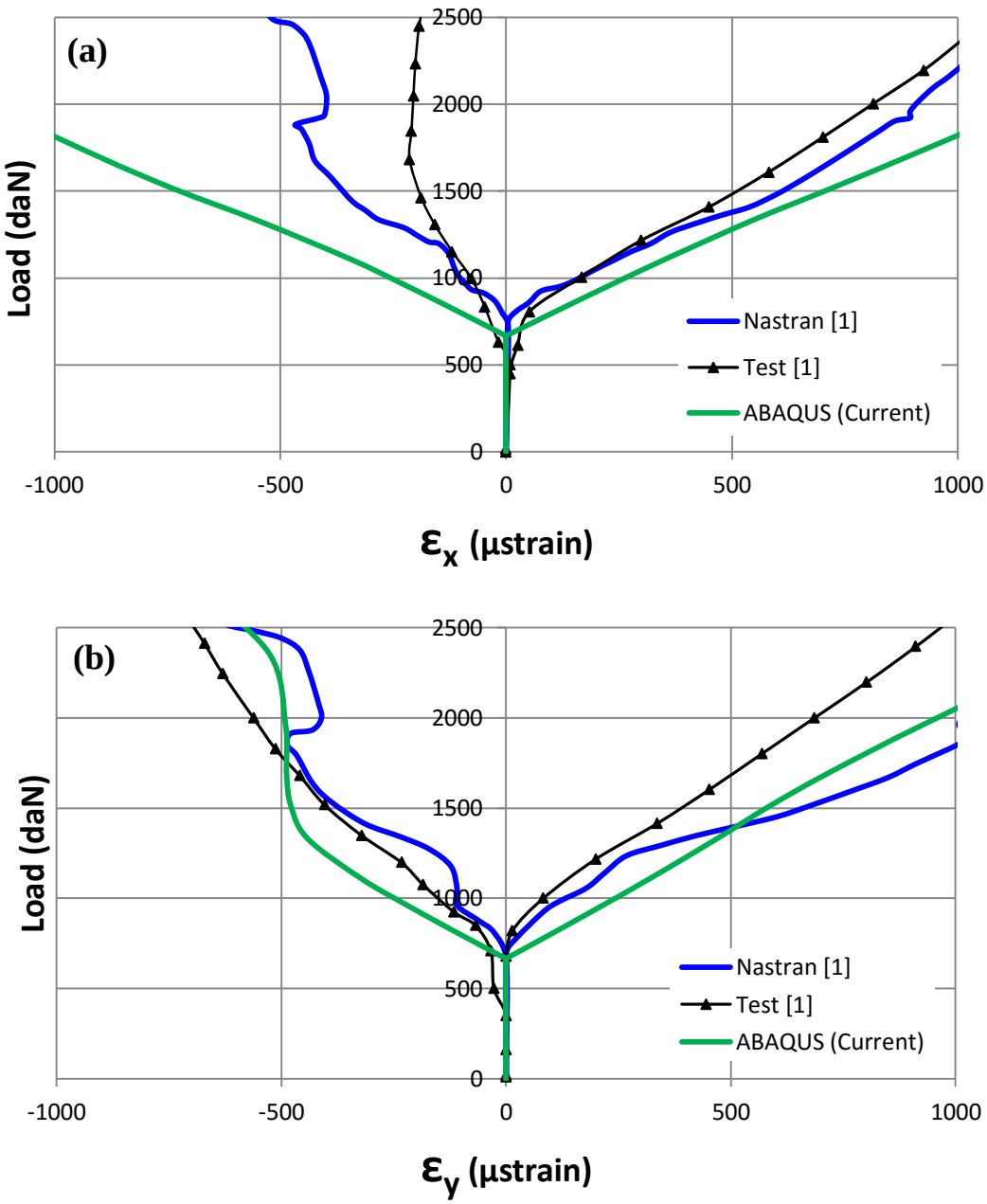

Figure 4: Strain results for thick ply case in (a) longitudinal and (b) vertical directions

Finally, a thin ply model was analyzed using the same methods. The strain results in the two plate directions were plotted along with the thin ply results in Figure 5 for comparison.

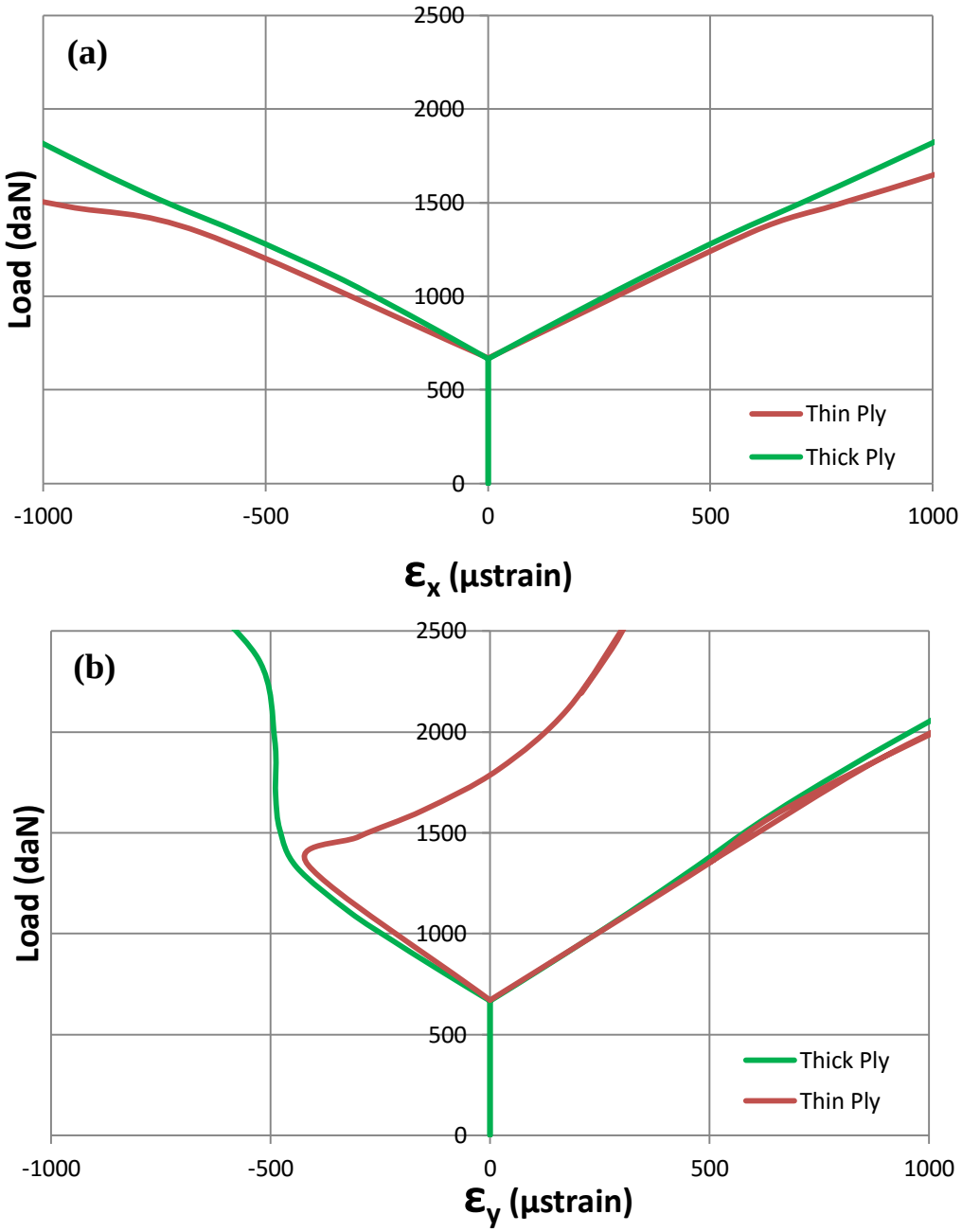

Figure 5: Strain results for thick ply case in (a) longitudinal and (b) vertical directions

4 DISCUSSION

The critical buckling load found for the thick ply panel is within the range of experimental results found in [1]. The diagonal waves on the web at buckling and post-buckling in Figure 3 demonstrate that buckling was caused by shear from the diagonal traction field. It can be seen in Figure 4 that the strain prediction for the nonlinear regimen follow a close path to the experimental results, it is not very accurate. This outcome upholds Orifici [7] findings over finite element solvers post-buckling predictions.

The results in Figure 5 show little difference in critical buckling loads and indicate that the thin ply laminate will have higher strain than the thick ply which signals a higher stress bearing capacity as the elastic properties are the same. As the model does not predict interlaminate developments, the benefits of the thin ply design compared to the thick ply were probably underestimated, as they have superior performance concerning micro-damage [9], which was not considered.

5 CONCLUSION

Abaqus® finite element analysis offers good critical buckling load prediction but the strain path in post-buckling response is not accurate to predict experimental results. The use of thin ply laminate does not significantly add to the stiffness of the plate, as the critical buckling load is very similar to the one found for the thick ply layup. To obtain a more precise comparison between the two laminate concepts a more complex model considering all the inter-laminar effects should be made.

ACKNOWLEDGEMENTS

The authors are thankful for the assistance of the Structural Engineering Department of UFMG and the Department of Mechanical Engineering of Hanyang University. This work was supported by CAPES, CNPq, and FAPEMIG.

REFERENCES

[1] F.K. Arakaki and A.R. Faria, Composite-stiffened panel design under shear postbuckling behavior, *Journal of Composite Materials*, **50**, 2016, pp. 3643-3662 (doi: 10.1177/0021998315623625)

[2] H. Wagner. *Structures of thin sheet metal, their design and construction*, National Advisory Committee for Aeronautics, Technical Memorandum no. 490, 1928.

[3] A. Murphy, M. Price, C. Lynch an, A. Gibson, The computational post-buckling analysis of fuselage stiffened panels loaded in shear. *Thin-walled structures*, **43**, 2005, pp. 1455-1474 (doi: 10.1016/j.tws.2005.03.010)

[4] A.W. Leissa, An overview of composite plate buckling, *Composite structures*, **4**, 1987, pp. 1-29 (doi: 10.1007/978-94-009-3455-9_1)

[5] C.C. Chamis, Buckling of anisotropic composite plates. *Journal of the Structural Division* **95**, 1969, pp. 2119-2140.

[6] J.N. Reddy and A. Khdeir, Buckling and vibration of laminated composite plates using various plate theories, *AIAA Journal*, **27**, 1989, pp. 1808-1817 (doi: 10.2514/3.10338)

[7] A.C. Orifici, R.S. Thomson, A.J. Gunnion, R. Degenhardt, H. Abramovich and Bayandor, J, Benchmark finite element simulations of postbuckling composite stiffened panels, *Proceedings of the 11th Australian International Aerospace Congress, Melbourne, Australia, March 13-17, 2005*, Engineers Australia, 2005, pp. 1-13.

[8] S.W. Tsai, S. Sihn and R.Y. Kim. Thin ply composites, *Proceedings of the 46th AIAA/ASME/ ASCE/AHS/ASC Structures, Structural Dynamics and Materials Conference, Austin, Texas, USA, April 18-21, 2005*, American Institute of Aeronautics and Astronautics, 2005. (doi: 10.2514/6.2005-2005)

[9] S. Sihn, R.Y. Kim, K. Kawabe, and S.W. Tsai, Experimental studies of thin-ply laminated composites. *Composites Science and Technology*, **67**, 2007, pp. 996-1008 (doi: 10.1016/j.compscitech.2006.06.008)

[10] Dassault Systèmes (2011) *Abaqus 6.11 Online Documentation*. Available at: <http://abaqus.ethz.ch:2080/v6.11/>

[11] E.J. Barbero, *Finite Element Analysis of Composite Materials Using Abaqus*. Boca Raton: CRC Press, 2013.