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ANALYSIS OF SELECTED PARAMETERS INFLUENCE ON FAILURE IN METAL-COMPOSITE MECHANICAL JOINT

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Abstract

The never-ending attempt to obtain as low mass as possible is the reason for using material of high specific strength (stiffness) in the aerospace industry. High strength titanium or aluminium alloys (e.g. 2024T3) and composite laminates (e.g. CFRP or Glare) are the examples of such materials. Despite a large number of composite types, fibre reinforced composites in the form of laminates are commonly used in aircraft structures. One-half of modern aircrafts is made of composites. However, the second one is still made of metallic alloys. The usage of different materials in aircraft structures results in the necessity of joining composite and metallic components. There are three connection methods for aircraft primary structures: mechanical (riveting, bolting), adhesive (bonding) and hybrid where both mentioned methods are used. The paper deals with metal-composite mechanical joint. Although fibre reinforced composites have high tensile strength, the load transfer in mechanical joints of such components is limited. Strength of composite laminates is dependent on the joint geometry; however, it is strongly influenced by laminate lay-up. There are five global failure modes for mechanically fastened composite laminates: tension, bearing, shear-out, cleavage and pull-through. The bearing failure mechanism is a safe progressive mechanism not leading to catastrophic failure and therefore it is acceptable. Problems with strength assessment of composite mechanical joints are drawn. Some geometrical (joint width), material (bolt material, stacking sequence) as well as numerical parameters (failure criterion) are analysed.

Keywords: *mechanical joint, CRFP, aviation, FEM*

1. Introduction

The attempt, to obtain as light aircrafts as possible forces, designers search new solutions with the usage of composite materials [1]. The newest Boeing and Airbus aircrafts are made of composite materials in approximately fifty percent. The fuselage and wing skins, together with the stiffeners are practically in the whole made of composites.

The rest is mainly metal alloys (aluminium, titanium, steel), which are used for highly loaded structural elements. The main parts of the aircraft structure are fuselage, wings and tail, consisting of the frame structure elements.

The usage of different materials in aircraft structures results in the necessity of joining composite and metallic components. There are three connection types concerning joining method:
- Mechanical e.g. riveting, bolting or pinning,
- Adhesive e.g. bonding, welding,
- Hybrid where both above-mentioned method are used.

In bonded joints, the load is distributed in a more uniform way. Additionally, the application of bonded joints leads to weight reduction. Main disadvantage of bonded joints is however higher cost determined by more rigorous assembly conditions, i.e. surface treating, moisture and temperature as well as the unfavourable tendency to voids nucleation between adhesive and adherent. Service conditions (atmosphere, service fluids) determine the strength of such joints. The ageing phenomenon is also important. Mechanical joints used for decades are proved to be reliable. They can be assembled and applied in very rough conditions since they are less sensitive to environmental effects.

Despite a large number of composite types, fibre (mainly carbon/graphite, glass or aramid) reinforced composites in the form of laminates are commonly used in aircraft structures [2].

Laminates consist of several layers. Each layer is usually a unidirectional fibre reinforced composite (Fig. 1). It means that it has a specific fibre orientation – θ – angle between fibres and assumed direction (mainly load direction). The laminate stacking sequence is usually describe by following code: [θ_n/ θ_{n-1}/… θ_2/ θ_1] where θ_n is the angle of top layer and θ_1 is the angle of bottom layer. The stacking sequence of laminate presented in Fig. 1 is therefore [0/45/90/-45/0]. If laminate has even number of plies and a symmetry plane the code is limited to one-half of layers with the subscript s. For example $[0/45/\text{-}45//90]_S$ means [0/45/-45//90/90/-45/45/0]. For symmetric laminates with odd number of plies, the middle layer is over lined: [0/45/-45/] S $[0/45/\text{-}45/\overline{90}]_S$ instead of [0/45/-45/90/-45/45/0].

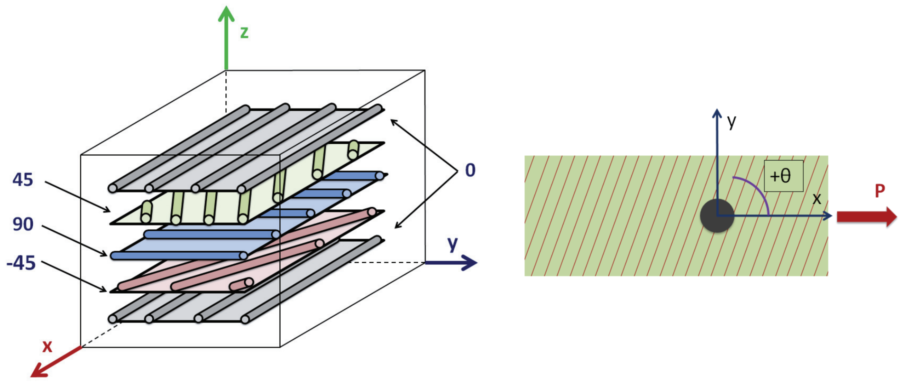


Fig. 1. Laminate stacking sequence determination

A unidirectional composite layer (lamina) has substantially different mechanical properties in different directions as its strength is mostly determined by fibre. The fibre direction is denoted by 1, the transverse direction is describes by 2, 3 means the perpendicular direction (Fig. 2).

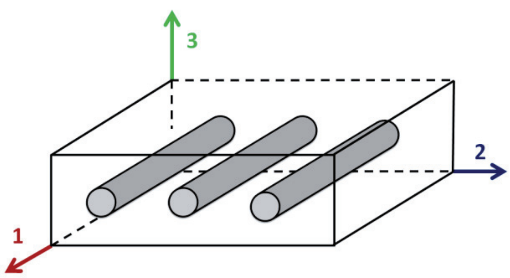


Fig. 2. Directions of lamina characteristic properties (lamina, material coordinate system)

Strength of composite laminates is dependent on joint geometry; however, it is strongly influenced by laminate configuration.

There are five global failure modes for mechanically fastened composite laminates [3]: tension, bearing, shear – out, cleavage and pull-through (Fig. 3). The bearing failure mechanism is a safe progressive mechanism not leading to catastrophic failure and therefore it is acceptable.

There are some hints for correct design of mechanical joint of composite panels [3]:

– appropriate geometry: Sheet width to hole diameter ratio W/d and edge distance to hole diameter ratio E/D should reach a high enough value specific to given material. The tensile failure is likely to happen for low W/D. Low E/D leads to shear-out,

– proper layer orientation: Composite should be quasi-isotropic that means that they should have at least 1/8 fibres but no more than 3/8 fibres in one of basic directions (0, +/–45, 90). If there are too many fibres in 0 direction and too few in 90 one, the shear-out is likely to occur. Composites with a small number of fibres in 90 direction and low E/D ratio are prone to cleavage.

If the above conditions are fulfilled, the occurrence of bearing failure mode is highly probable. In composite materials, it is more complex than in metal alloys

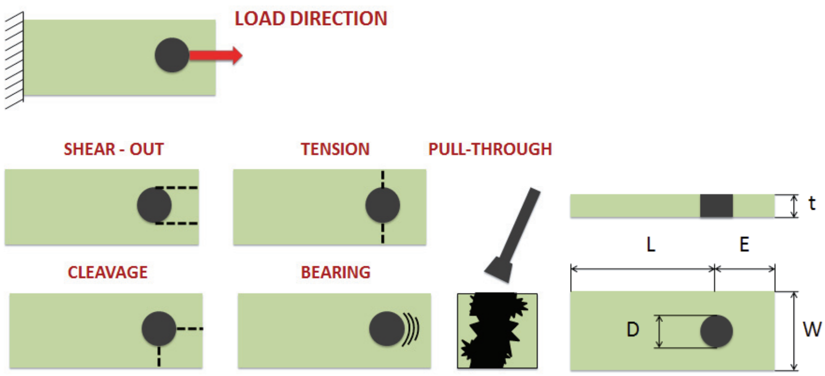


Fig. 3. Failure mechanism in bolted composites

2. Analysis

The analysis is performed on the specimen in the form of double-shear joint with two rows of two fasteners (Fig. 4) made of steel – ST or Titanium alloy – Ti. The outer element is made of 2024T3 aluminium alloy and the inner element is made of aforementioned laminate.

The stacking sequences used in this paper are [0/45/90/-45/0/45/90/-45]s – S1 and [0/0/45/45/90/90/-45/-45] s – S2. The selected material is HTA/6376 unidirectional prepreg.

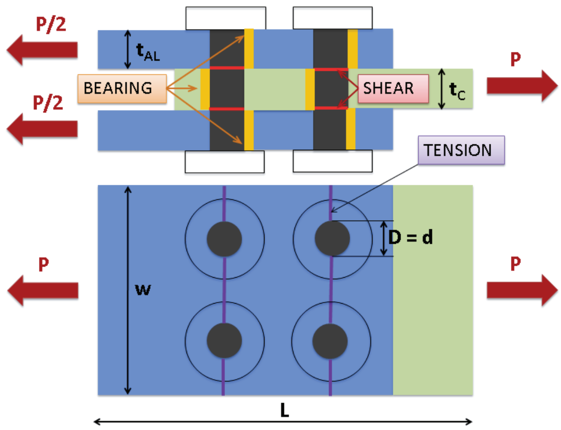


Fig. 4. Analysed joint

The joint length L is 300 mm. The bolt diameter d is assumed 6 mm. A selected pitch length is 5d which results in joint width of 60 mm (w = 60 mm) – W1. Additionally a joint with increased width (w = 70 mm) – W2 is analysed. The assumed composite configurations provides its thickness of about 3 mm (t_C = 3 mm). The aluminium sheet thickness is 2 mm (t_{AL} = 2 mm).

Solid element is used for all parts (aluminium sheet, composite and bolt). It is an eight-node element with linear interpolation functions, with three translational degrees of freedom at node. Due to symmetry, only a quarter of the joint is modelled. The boundary and symmetry conditions are presented in Fig. 5. The left grip edge is fixed and the right grip edge is pulled.

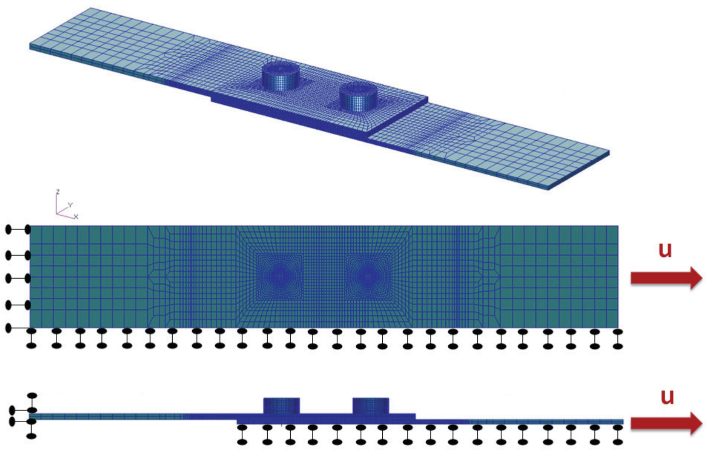


Fig. 5. Iso view, load, boundary and symmetry conditions in FEM analysis

The properties of metallic alloys used in analysis are presented in Tab. 1.

Tab. 1. Properties of metallic components [4]

	Young's modulus	Poisson's ratio
Aluminium alloy sheet [GPa]	70	0.33
Steel bolt	210	0.3
Titanium alloy bolt	110	0.29

A single lamina is described by means of one layer of 3D orthographic material which properties are presented in Tab. 2.

Tab. 2. Elastic properties of HTA/6376 lamina [6]

Young's modulus 1 [GPa]	E_1	140
Young's modulus 2 [GPa]	E_2	10
Young modulus 3 [GPa]	E_3	10
Poisson's ratio 1-2	v_{12}	0,3
Poisson's ratio 1-3	v_{13}	0,3
Poisson's ratio 2-3	v_{23}	0,5
Shear modulus 1-2 [GPa]	G_{12}	5,2
Shear modulus 1-3 [GPa]	G_{13}	5,2
Shear modulus 2-3 [GPa]	G_{23}	3,9

Node to segment contact [5] is applied between the contacting surfaces. Nonlinear analysis is performed using Newton-Raphson method with MSC.Marc code.

The composite element failure can be estimate with the usage of failure criteria. The failure criteria used in this paper are maximum stress (MS) and Hashin (H) failure criterion.

Failure Criteria compare the appropriate components of the stress tensor in the material (lamina) coordinate system ($\sigma_1, \sigma_2, \sigma_3, \sigma_{12}, \sigma_{23}, \sigma_{31}$) or their combination with the corresponding strengths values. The Tab. 3 shows the values of the laminate layers strengths, which were used for calculating failure indices.

According to maximum stress criterion (MS) the failure indices are calculated as showed in Tab. 4

Tab. 3. Strengths of HTA/6367 lamina [7]

Tensile strength 1 [MPa]	S_{1t}	2250
Compressive strength 1 [MPa]	S_{1c}	1600
Tensile strength 2 [MPa]	S_{2t}	64
Compressive strength 2 [MPa]	S_{2c}	290
Tensile strength 3 [MPa]	S_{3t}	94
Compressive strength 3 [MPa]	S_{3c}	290
Shear strength 1-2 [MPa]	S_{12}	98
Shear strength 1-3 [MPa]	S_{13}	98
Shear strength 2-3 [MPa]	S_{23}	30

Tab. 4. Maximum stress failure indices [5]

First index	Second index	Third index:
$\mathrm{FI\,no.1} = .\begin{cases} \frac{\sigma_1}{S_{1t}} & for\ \sigma_1 > 0 \\ -\frac{\sigma_1}{S_{1c}} & for\ \sigma_1 < 0 \end{cases}$	$\mathrm{FI\,no.2} = .\begin{cases} \frac{\sigma_2}{S_{2t}} & for\ \sigma_2 > 0 \\ -\frac{\sigma_2}{S_{2c}} & for\ \sigma_2 < 0 \end{cases}$	$\mathrm{FI\,no.3} = \begin{cases} \frac{\sigma_3}{S_{2t}} & for\ \sigma_3 > 0 \\ -\frac{\sigma_3}{S_{2c}} & for\ \sigma_3 < 0 \end{cases}$
Fourth index:	Fifth index:	Sixth index
$\mathrm{FI\,no.4} = \frac{\lvert\sigma_{12}\rvert}{S_{12}}$	$\mathrm{FI\,no.5} = \frac{\lvert\sigma_{23}\rvert}{S_{23}}$	$\mathrm{FI\,no.6} = \frac{\lvert\sigma_{13}\rvert}{S_{13}}$

The Hashin failure criterion (H) distinguishes between fibre failure and matrix failure [5]:

1. First index, fibre tension mode:

$$\mathrm{FIno.1} = \left(\frac{\sigma_{11}}{S_{1t}}\right)^2 + \frac{1}{S_{12}^2}\left(\sigma_{12}^2 + \sigma_{13}^2\right) \ or \ FIno.1 = \frac{\sigma_{11}}{S_{1t}}. \tag{7}$$

2. Second index, fibre compression mode:

$$\mathrm{FIno.2} = \frac{\lvert\sigma_{11}\rvert}{S_{1c}} \quad for \quad \sigma_{11} < 0. \tag{8}$$

3. Third index, matrix tension mode:

$$\mathrm{FIno.3} = \frac{1}{S_{2t}^2}\left(\sigma_2 + \sigma_3\right)^2 + \frac{1}{S_{23}^2}\left(\sigma_{23}^2 - \sigma_2\sigma_3\right) + \frac{1}{S_{12}^2}\left(\sigma_{12}^2 + \sigma_{13}^2\right) \ for \ \sigma_2 + \sigma_3 > 0. \tag{9}$$

4. Fourth index, matrix compression mode:

$$\mathrm{FIno.4} = \frac{1}{S_{2c}}\left(\left(\frac{S_{2c}}{2S_{23}}\right)^2 - 1\right)\left(\sigma_2 + \sigma_3\right) + \frac{1}{4S_{23}^2}\left(\sigma_2 + \sigma_3\right)^2 + \frac{1}{S_{23}^2}\left(\sigma_{23}^2 - \sigma_2\sigma_3\right) + \\ + \frac{1}{S_{12}^2}\left(\sigma_{12}^2 + \sigma_{13}^2\right) \ for \quad \sigma_2 + \sigma_3 < 0. \tag{10}$$

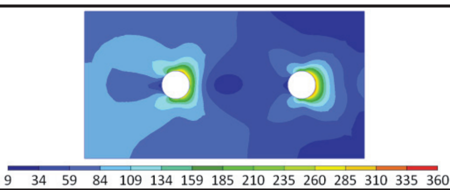
Table 5 shows performed analysis cases and their variances.

Tab. 5. Analysis cases and their variances

case	Stacking sequence	Joint width [mm]	Bolt material	Failure criterion
S1_W2_ST_MS	[0/45/90/-45/0/45/90/-45]s	70	Steel	Max stress
S1_W2_ST_H	[0/45/90/-45/0/45/90/-45]s	70	Steel	Hashin
S1_W1_ST_H	[0/45/90/-45/0/45/90/-45]s	60	Steel	Hashin
S1_W2_Ti_H	[0/45/90/-45/0/45/90/-45]s	70	Titanium alloy	Hashin
S2_W2_ST_H	[0/0/45/45/90/90/-45/-45]s	70	Steel	Hashin

All the results are presented for the displacement of the right grip equal to 0.375 mm. For this load level maximum von mises stresses in the vicinity of the hole (aluminium alloy sheet) equals yield stress (Tab. 6) for the S1_W2_ST_MS/S1_W2_ST_H case which is treated as base.

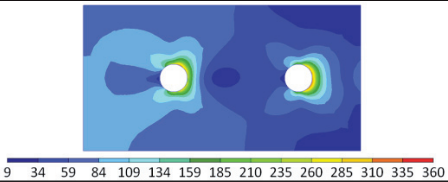
Tab. 6. Failure indices obtained according to maximum stress failure criterion

Aluminium alloy sheet		1=failure start 					
Laminate Layer	angle [°]	FI no. 1	FI no. 2	FI no. 3	FI no. 4	FI no. 5	FI no. 6
L1	0	0.614	1.230	0.368	1.190	2.586	0.540
L2	45	0.589	1.364	0.350	1.014	1.250	0.906
L3	90	0.535	1.630	0.238	1.000	1.829	0.811
L4	-45	0.708	1.042	0.225	0.864	1.630	0.529
L5	0	0.649	0.407	0.266	0.942	1.875	0.442
L6	45	0.563	1.339	0.207	0.967	1.103	0.778
L7	90	0.481	1.705	0.198	1.136	1.829	0.936
L8	-45	0.992	1.364	0.528	1.210	1.389	1.974

The Tab. 6 shows failure indices and their maximum positions with respect to the hole for S1_W2_ST_MS case. According to the maximum stress criterion the highest indices values are those corresponding to the tension/compression in direction 2 – FI no. 2 (determined by the properties of the matrix), the shear in 2-3 plane - FI no. 5 and Shear 1-2 plane – FI no 4. Large influence on large values of the index FI no. 5 has low shear strength in the plane of 2-3 as well as the proximity to the free edge (the hole edge).

Failure indices for S1_W2_ST_H are presented in Tab. 7.

Tab. 7. Failure indices obtained according to Hashin failure criterion

Aluminium alloy sheet		1=failure start  			
Laminate Layer	angle [º]	FI no. 1	FI no. 2	FI no. 3	FI no. 4
L1	0	0.993	0.580	2.500	6.250
L2	45	1.027	0.589	1.471	8.333
L3	90	0.996	0.535	2.055	8.333
L4	-45	0.910	0.708	1.829	6.818
L5	0	0.917	0.639	1.786	4.688
L6	45	0.986	0.563	1.389	10.714
L7	90	1.190	0.481	2.143	15.000
L8	-45	1.316	0.991	2.083	5.769

The analysis of the obtained Hashin criterion indices shows that the resin fails by compression and tension in each layer. The high values of fibre tension indices are probably overestimated [7].

The rest of the results are presented in the comparison to S1_W2_ST_H case. The sign: ▼ denotes decrease of failure index value. The sign: ▲ stadns for increase of failure index value.

Difference of failure indices for S1_W1_ST_H in comparison to S1_W2_ST_H are presented in Tab. 8. For the lower value of width the hole joint is more flexible, therefore there is decrease in almost all failure indices values in the vicinity of the hole for the same displacement of the grip.

Table 9 shows results obtained for S1_W2_Ti_H case.

Tab. 8. Difference of failure indices between S1_W1_ST_H and S1_W2_ST_H cases

Laminate Layer	angle [º]	FI no. 1 [%]	FI no. 2 [%]	FI no. 3 [%]	FI no. 4 [%]
L1	0	-3.6 ▼	-13.6 ▼	-11.8 ▼	-7.7 ▼
L2	45	+2.8 ▲	-9.5 ▼	-1.9 ▼	-5.3 ▼
L3	90	+6.0 ▲	-7.7 ▼	-3.9 ▼	-10.0 ▼
L4	-45	-4.6 ▼	-10.7 ▼	-10.9 ▼	-12.0 ▼
L5	0	-0.9 ▼	-12.5 ▼	-12.5 ▼	-11.1 ▼
L6	45	-6.3 ▼	-9.4 ▼	0.0	-12.5 ▼
L7	90	-11.3 ▼	-7.9 ▼	-5.4 ▼	-16.7 ▼
L8	-45	-5.0 ▼	-11.1 ▼	-13.3 ▼	0.0

Tab. 9. Difference of failure indices between S1_W2_Ti_H and S1_W2_ST_H cases

Laminate Layer	angle [°]	FI no. 1 [%]	FI no. 2 [%]	FI no. 3 [%]	FI no. 4 [%]
L1	0	+0.3 ▲	+8.1 ▲	+7.1 ▲	+9.1 ▲
L2	45	+2.8 ▲	-0.8 ▼	-1.9 ▼	0.0
L3	90	+6.0 ▲	-0.6 ▼	-1.4 ▼	-10.0 ▼
L4	-45	2.0 ▼	-4.9 ▼	-4.7 ▼	0.0
L5	0	+0.5 ▲	-4.2 ▼	0.0	11.1 ▼
L6	45	+0.6 ▲	-3.9 ▼	-6.9 ▼	-12.5 ▼
L7	90	3.1 ▼	-4.0 ▼	-2.8 ▼	-16.7 ▼
L8	-45	-3.4 ▼	-7.3 ▼	-2.7 ▼	0.0

The titanium alloy stiffness is nearly two times lower than stiffness of steel. The titanium bolt deforms more, which is profitable to most failure indices but more deformation create more difficult conditions for outer layer of laminate (relative displacement between top and ground surfaces of the layer is about 100% greater than for steel bolt) (Fig. 6).

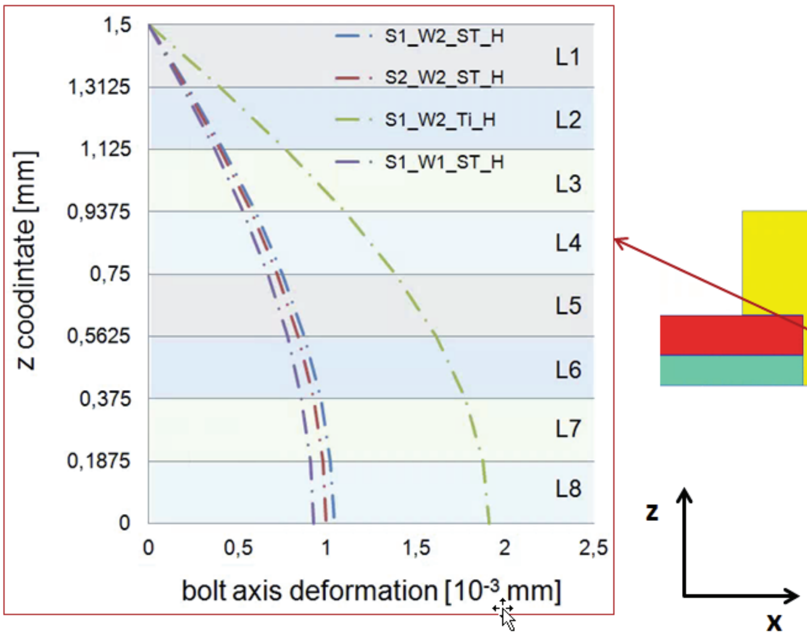


Fig. 6. Bolt axis deformations in laminate

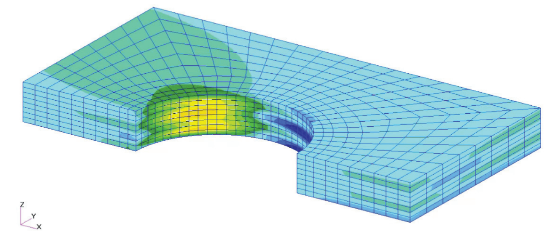
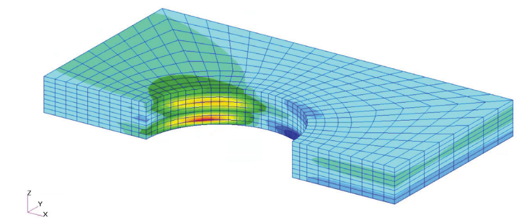
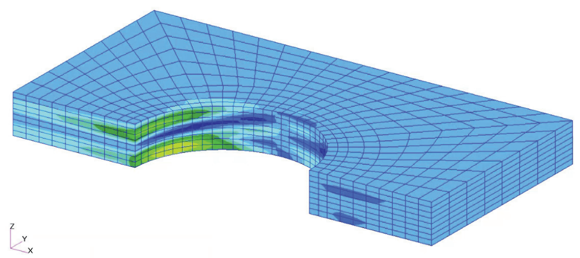
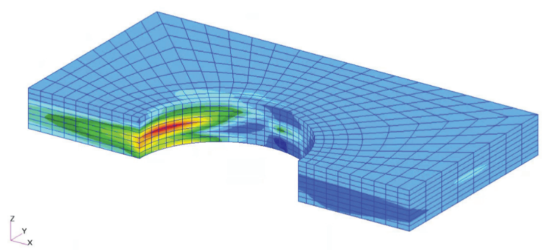
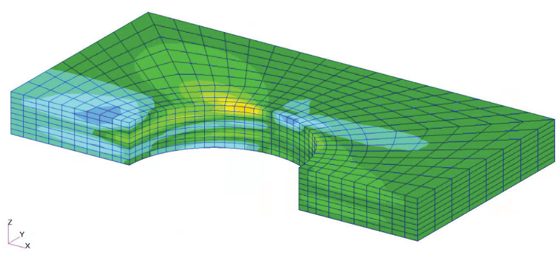
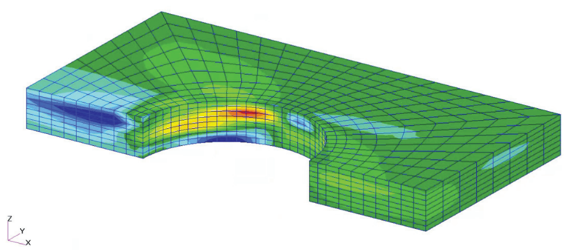
Difference of failure indices for S2_W2_ST_H in comparison to S1_W2_ST_H are presented in Tab. 10. For the clarity of interpretation, the laminate layers of S1_W2_ST_H were segregated and grouped with regard to orientation angle. The maximum and minimum failure indices values were determined and ordered for each group angle and compared with the corresponding values of S2_W2_ST_H failure indices.

For the case S2_W2_ST_H there is a noticeable increase in almost all failure indices values with respect to S1_W2_ST_H. Hashin failure indices are strongly dependent of shear stresses. The shear stresses distributions for these two cases are presented in table.

Tab. 10. Difference of failure indices between S2_W2_ST_H and S1_W2_ST_H cases

Laminate Layer angle [°]		FI no. 1 [%]	FI no. 2 [%]	FI no. 3 [%]	FI no. 4 [%]
0	max	+2.1	+1.4	+42.9	+20.0
		▲	▲	▲	▲
0	min	+8.3	+1.6	+44.8	+60.0
		▲	▲	▲	▲
45	max	+46.0	+4.8	-34.2	0.0
		▲	▲	▼	
45	min	+26.8	+6.3	+21.7	-28.6
		▲	▲	▲	▲
90	max	+14.5	+4.2	+29.6	-16.7
		▲	▲	▲	▼
90	min	+27.6	-3.7	+25.9	+28.6
		▲	▼	▲	▲
-45	max	+14.0	-8.6	+28.6	+83.3
		▲	▼	▲	▲
-45	min	+44.6	+0.3	+28.1	+44.4
		▲	▲	▲	▲

Tab. 11. Shear stresses distribution for S2_W2_ST_H and S1_W2_ST_H cases

S1_W2_ST_H	S2_W2_ST_H
xy shear stresses distribution	
	
-204 -158 -113 -67 -22 24 69 115 161 206 252 297 343 388 434	
yz shear stresses distribution	
	
-17 -12 -7 -1 4 9 14 19 24 29 35 40 45 50 55	
zx shear stresses distribution	
	
-36 -30 -24 -18 -12 -6 0 6 12 18 24 30 36 42 48	

The values of shear stress increased approximately in 9, 32 and 38% respectively in xy, yz, and zx plane for S2_W2_ST_H in relation to S1_W2_ST_H case. The stress distribution is therefore more severe which cases greater failure indices for this case.

3. Conclusions

In the case of metallic materials, strengths are clearly defined. For the composite materials, the situation is more complicated. Strength of composite laminates is dependent on joint geometry; however, it is strongly influenced by laminate configuration. The composite element failure can be estimate with the usage of failure criteria. The failure criteria used in this paper are maximum stress and Hashin failure criterion. The difference was measured in Hashin failure criterion between analysed cases.

For the lower value of width the hole joint is more flexible, therefore there is decrease in almost all failure indices values in the vicinity of the hole for the same displacement of the grip.

The titanium bolt deforms more, which is profitable to most failure indices but more deformation creates for outer laminate layer more difficult conditions.

Hashin failure indices are strongly dependent of shear stresses. The values of shear stress increased approximately in 9, 32 and 38% respectively in xy, yz and zx plane for S2_W2_ST_H in relation to S1_W2_ST_H case. The stress distribution is therefore more severe which cases greater failure indices for this case.

Although the failure indices are valid, only to first failure occurrence the presented values are recorded in the linear range of the aluminium sheet material. Therefore, the presented data gives insights on the behaviour of particular layer in selected case.

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