3D Random Fiber Composites as a Repair Material for Damaged Honeycomb Cores

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The increased use of composites has rendered the need for development of reliable and efficient repair techniques. Like all other metallic structures composite structures are prone to variety of damage. Repair techniques developed for metallic structures are not directly applicable to composites and problem of composite repair has to be investigated in great detail. In the present paper application of 3D random fiber polymer composites, as a repair material for damaged high density honeycomb cores in sandwich structures is investigated. Based on Pan's theory the expression for matching fiber volume fraction of the repair material is derived and complete stress-strain field, of the repaired sandwich structure is determined using finite element approach. It is concluded that the 3D random fiber composites represent very good candidates as repair materias for damaged honeycomb structured composites.

Keywords: *Honeycomb core, composite repair, 3D random fiber composites .*

1. INTRODUCTION

Sandwich panels are used in applications where special properties, such as high flexural stiffness, high impact strength, high corrosion resistance, fatigue resistance, and low thermal conductivity are sought. Industries which make use of sandwich panels nowadays are transportation industries (such as aerospace, aeronautical, railway, automotive and marine) and construction industry.

Like any other load bearing structure, composite sandwich panels have been found to be liable to damage, arising from manufacture or service. Typical service damages (mechanical and environmental) are presented in the following table:

Table 1. Service mechanical and environmental damage

Damage	Cause
Scratch	Mishandling
Cut	Mishandling
Delamination	Impact, thermal
Disbond	Impact or overload
Dent	Impact
Edge damage	Mishandling
Penetration	High velocity impact
Abrasion	Erosion (rain,
Oxidation	Lightning strike,
Disbond	Impact, thermal
Core corrosion	Moisture penetration
Swelling	Use of solvents

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The common problem of damaged structure confronts the commercial user with considerable economic loss if no adequate repair procedure is known. The increasing use of composite sandwich panels requires the development of reliable repair methods that restore the integrity of the damaged structure, with minimum degradation in its functional capability and in aerospace applications with minimum weight addition. The implementation of onsite repairs calls for simple and effective procedures, without using sophisticated and expensive equipment in order to avoid excessive downtime of the component. Therefore, adequate and economically feasible repair techniques are required to further expand the use of composite sandwich panels.

A number of papers have been published in recent years trying to standardize repairs to composite parts and underline problems related to application of repaired composite components in service [1-4].

Unlike repairs to metal parts where the materials are supplied in the finished condition, composite repairs mean curing the material at the time of repair and outside the control of the component manufacturer. Also, when repair of composites is in question the storage and handling of matrix resins is serious matter. Fabrics, pre-pregs and fibers require storage at the exact storage temperatures to avoid their properties degradation and surface contamination [5-7].

The commercial aircraft composite repair committee (CACRC) developed wet-lay up method applicable to the repair of damaged honeycomb sandwich panels [7]. Adhesively bonded repairs have significant advantages over bolted repairs. Adhesively bonded repairs can restore a composite structure's original strength, are more fatigue resistant due to the absence of stress concentrations that occur at fasteners and are significantly lighter than bolted repairs due to the absence of fastener hardware.

Being one of the sandwich composites end users, the airlines have requested, at CACRC meeting that approval be given for any qualified repair material to be replaced by an alternative material that is qualified by the same OEM but under other specifications and another OEM for an equivalent or similar application. In this paper, the alternative approach to sandwich panel repair is investigated. 3D random fiber composite is used as the core repair plug.

The repair schema of this approach is presented in the following picture:

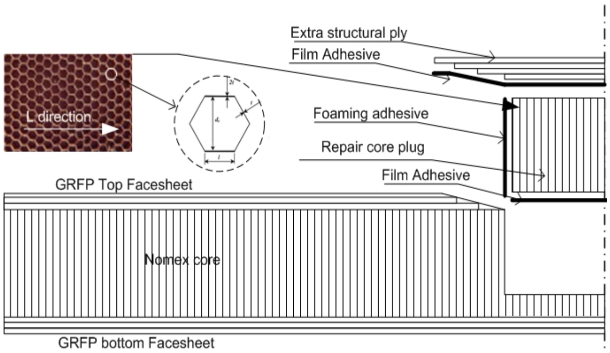

Fig. 1. Sandwich core repair : Standard repair procedure using honeycomb core repair plug

The investigated alternative repair method is depicted in the following figure

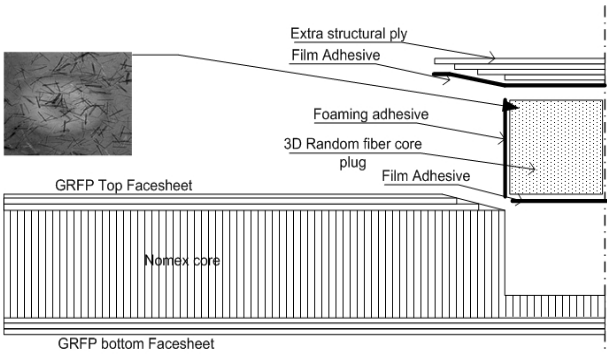

Fig. 2. Sandwich core repair : Investigated repair procedure using 3D random fiber composite material core repair plug

2. REPAIR METHOD

Honeycomb panel damage is usually found visually or during NDI [8]. Some instances of core debonding and delamination may be repaired through simple repair techniques, such as resin injection. However, with honeycomb core damage it is often difficult to ascertain the full extent of the damage until the skin over the damaged area has been removed.

Repairing honeycomb panels frequently entails cutting out a new piece of core to replace the old and splicing it firmly in place with foaming adhesives. Exterior skin plies are then built up in configurations which match the original skin. For rapid repair with minimal equipment, simply filling in a small damaged area with body filler can return a part to its original aerodynamic surface profile frequently with little or no performance penalty, and only a small weight penalty. The usual course of action for a very badly damaged honeycomb panel is to carefully remove the damaged portions of skin and core and sand away any paint and/or primer which may have been exposed. Replacement plies are cut out to exactly match the damaged area, and are laid up in the same orientation as the original prepregs. The repaired area is then covered with a vacuum-bag to apply pressure to the area during cure for maximum bond integrity and minimal voids. A common repair problem with honeycomb panel structures is ingression of moisture into the honeycomb, usually as a result of micro cracking in the composite skins. Skins must be peeled back, the honeycomb core dried out or replaced, and a repair patch applied. If moisture can get into the honeycomb core, it will become steam during the repair heating procedure, and can blow the part apart. The type of repair used depends upon the type and extent of the damage, as well as on the loads in the area.

In the present analysis following repair steps are taken:

- Locate damage area and assess the extent of the damage
- Remove the damaged area
- Prepare the repair plug, based on the removed area size (damaged zone)
- Fill in the film adhesive at the bottom of the damaged core
- Place the repair core plug
- Fill in the foaming adhesive
- Filler(s) are sanded flush with the surrounding structure
- Bond, vacuum bag and cure the external skin and inserted plug
- After curing process inspect the repaired area.

Key steps in the proposed repair methodology are presented in the following figures (Fig. 3 and Fig. 4).

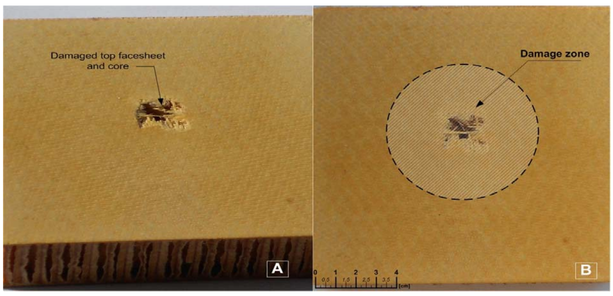

Fig. 3. a) Damaged sandwich beam (top face sheet and partially damaged core) b) damage zone

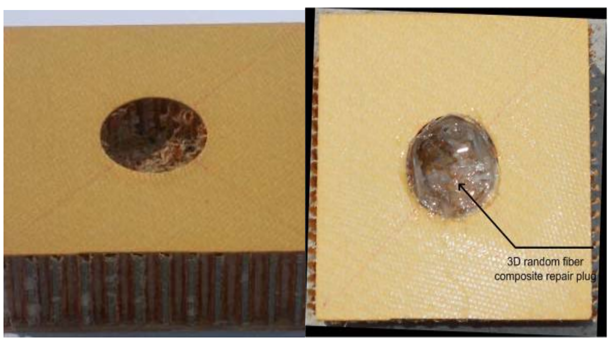

Fig. 4. a) Prepared panel for plug insertion, b) 3D random fiber composite core plug

3. GOUVERNING EQUATIONS

A sandwich panel consists of thin, stiff and strong facesheets made of dense material, separated by a thick low density core. Such a structure provides an analogy to I-beam where the facesheets are equivalent to the flanges and the core acts as a web.

The facesheets carry in-plane and bending loads, while the primary function of the core is to resist the transverse shear loads. The core should be stiff enough in the direction perpendicular to the facesheets to ensure that facesheets maintain the correct distance apart, while not sliding with the respect to each other in order to ensure composite action (Figure 5). Also, the core must be stiff enough to keep the facesheets as flat as possible to prevent them from local buckling under compressive loads .

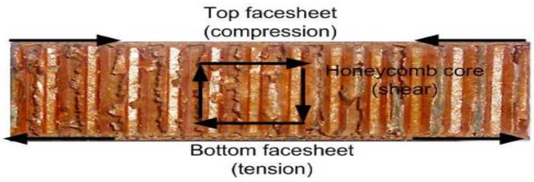

Fig. 5. Composite Sandwich plate under bending loads, stresses in the core and facesheets.

Adjusting the height of the core, the bending stiffness of this construction could be much greater than that of a single solid plate made of the same material as the facesheets [10-12].

To validate the effectiveness of the applied reparation method, one must determine the complete stress-strain field of both, virgin (undamaged) structure and repaired structure for the same service load conditions.

The displacement field of a rectangular laminated plate, based on the classical plate theory and including the effect of transverse shear deformations, can be expressed as (eq. 1).

$$u(x,y,z) = u_0(x,y) + z\left[\alpha\frac{\partial w_0}{\partial x} + \beta\theta_x\right],$$
$$v(x,y,z) = v_0(x,y) + z\left[\alpha\frac{\partial w_0}{\partial y} + \beta\theta_y\right], \quad (1)$$
$$w(x,y,z) = w_0(x,y).$$

In previous equation $u_0(x, y)$, $v_0(x, y)$ and $w_0(x, y)$ denote the corresponding midplane displacements in the x, y, z directions and θ_x and θ_y are the rotations of normals to midplane about the y and x axes. The above displacement field is the general displacement field which gives both the CPT and FSDT theories, as: Classical plate theory (CPT): $\alpha = -1$, $\beta = 0$ and First order shear deformation theory (FSDT): $\alpha = 0$, $\beta = 1$. Applying the principle of virtual displacements equilibrium equations are obtained and for the plate subjected to distributed loading (q) are given as :

$$\frac{\partial N_x}{\partial x} + \frac{\partial N_{xy}}{\partial y} = 0,\ \frac{\partial N_y}{\partial y} + \frac{\partial N_{xy}}{\partial x} = 0,$$
$$\frac{\partial Q_x}{\partial x} + \frac{\partial Q_y}{\partial y} = -q, \quad (2)$$
$$\frac{\partial M_x}{\partial x} + \frac{\partial M_{xy}}{\partial y} - Q_x = 0,\ \frac{\partial M_y}{\partial y} + \frac{\partial M_{xy}}{\partial x} - Q_y = 0.$$

where N_i ($i=x,y$) are membrane forces per unit length in i[th] direction, N_{xy} is the in-plane shear force per unit length, Q_i are shear forces in 'x' and 'y' directions and M_i (M_x, M_y, M_{xy}) are bending and twisting moments per unit length.

In order to fully define the stress strain relation, all elastic coefficients in elastic tensor have to be defined. Composite panel investigated in this paper consists of the honeycomb core, composite facesheets and core repair plug composed of 3D random fibers embedded in matrix.

2.1 Elastic Properties of 3D random composite

Several theories predict behavior of 2D and 3D random fiber polymer composites. Assuming isotropic properties of repair material Pan's theory [12] is used to determine elastic parameters of 3D random fiber polymer material.

Defining a relationship between the fiber volume fraction V_f and the fiber area function A_f when fibers are not unidirectionally aligned and introducing probability density function defined by a pair of angles (Θ,Φ) in the spatial curvilinear coordinate system, the composite modulus can be expressed as (1):

$$E_c(\Theta,\Phi) = E_f\cdot\Omega(\Theta,\Phi)\cdot V_f + E_m\cdot(1-\Omega(\Theta,\Phi)\cdot V_f) \quad (1)$$

In the preceding equation E_c is the modulus of the composite, E_f and E_m are moduli of constituents (fibers and matrix) respectively, V_f is fiber volume fraction and $\Omega(\Theta,\Phi)$ is the value of the probability density function in direction (Θ,Φ). In the case of three-dimensional random fiber orientation, the tensile modulus is given by :

$$E_c^{3D} = E_f\frac{V_f}{2\pi} + E_m(1-\frac{V_f}{2\pi}) \quad (2)$$

and, the Poisson's ratio by the equation :

$$\nu_c^{3D} = \nu_f\frac{V_f}{2\pi} + \nu_m(1-\frac{V_f}{2\pi}). \quad (3)$$

In equations (12 and 13) Ef and Em are moduli of elasticity of fibers and matrix, and νf, νm are their Poisson's ratios.

To determine optimal fiber volume fraction following equation is solved for Vf :

$$G_{12} = \frac{E_f\frac{V_f}{2\pi} + E_m\left(1-\frac{V_f}{2\pi}\right)}{2\cdot\left(1+\nu_f\frac{V_f}{2\pi} + \nu_m\left(1-\frac{V_f}{2\pi}\right)\right)} \quad (4)$$

and V_f is obtained in the following form:

$$V_f = \frac{2G_{12}\cdot(1+\nu_m) - E_m}{\frac{1}{\pi}\left[\left(\frac{E_f-E_m}{2}\right) + G_{12}\left(\nu_f-\nu_m\right)\right]} \quad (5)$$

In the previous equation G_{12} is the shear modulus of the core.

2.2 Elastic Properties of Honeycomb Core

Composite honeycomb cores are usually made of Nomex paper, which is aramid based paper. The initial paper honeycomb is usually dipped in a phenolic resin to produce a honeycomb core with high compression strength. The Nomex paper, basis of honeycomb Nomex core, is a non woven sheet made of short aramid fibers. It is calandrered before being impregnated with phenolic resin, its isotropy is assumed because of the arbitrary distribution of short fibers. The honeycomb core properties depend on the cell size, wall thickness and strength.

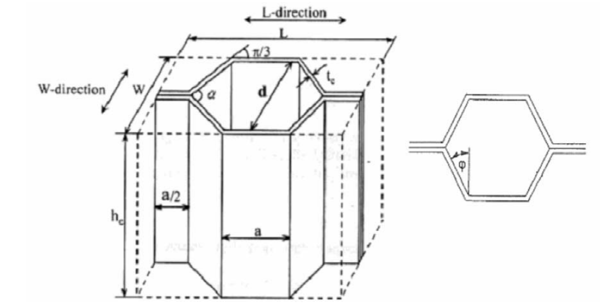

Fig. 6. Honeycomb unit-cell geometry

In the case of a honeycomb sandwich core, building a FEM representation for material and geometric properties is itself a difficulty. Detailed 3D model that accounts for the actual cell geometry was considered, but it reaches rapidly very high DOF numbers. To achieve efficiency in numerical analysis, the honeycomb cellular core is replaced with an equivalent continuum model. The sandwich core is analyzed in terms of its effective properties rather than by consideration of its real cellular structure. Therefore, the evaluation of effective elastic properties for this continuum core becomes important [13].

The orthotropic honeycomb core material law depends on 9 independent material parameters E_x, E_y, E_z, ν_{yx}, ν_{zx}, ν_{zy}, G_{xy}, G_{xz}, G_{yz}.

Based on the experimentally obtained mechanical characteristics of the constitutive material, and measured cell size and wall thickness (Fig. 7) the four in-plane moduli are calculated by causing the hexagonal cell wall to bend under loads in x and y directions. The Young's moduli, E_x, E_y, and shear modulus, G_{xy}, are calculated by standard beam theory, as the ratio of strain to stress. The Poisson's ratio, ν_{yx}, is deduced by the negative ratio of the strains normal to, and parallel to, the loading direction. It is assumed that the ν_{zx}, ν_{zy}, Poisson's ratios are equal to zero and the out of plane Young's modulus, Ez , is the core Young's modulus in z direction obtained from the manufacturer.

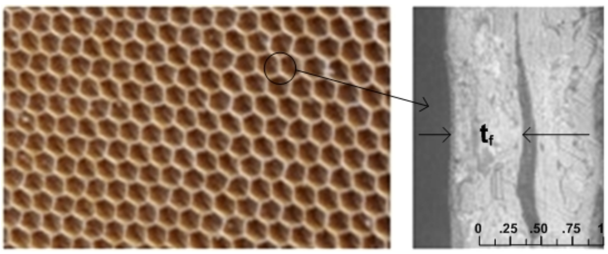

Fig 7. Honeycomb cell geometry and wall thickness

Based on these assumptions, elastic coefficients of honeycomb core can be expressed as [15,16]:

$$E_x = \frac{E_m}{\frac{\cos\varphi}{(1+\sin\varphi)} \cdot \left[\frac{\cos^2\varphi \cdot a^3}{t^3} + \frac{(2+\sin^2\varphi)\cdot t}{a} \right]}$$

$$E_y = \frac{E_m}{\frac{(1+\sin\varphi)}{\cos\varphi} \cdot \left[\frac{\sin^2\varphi \cdot a^3}{t^3} + \frac{\cos^2\varphi \cdot t}{a} \right]}$$

$$\nu_{xy} = \frac{\sin\varphi \cdot (1+\sin\varphi)}{\cos^2\varphi} \qquad (6)$$

$$G_{yz} = \frac{10\cdot t}{9a\cdot\cos^3\varphi\cdot(1+\sin\varphi)} \cdot G_m$$

$$G_{xz} = \frac{2t}{a\cdot\cos\varphi\cdot(1+\sin\varphi)} \cdot G_m$$

In the previous equations E_m and G_m represent elastic moduli of honeycomb core constitutive material (Nomex paper), obtained experimentally. Parameters t, a, and φ are geometric characteristics of the core unit cell as depicted in the figure 6 (Fig. 6).

2.3 Elastic Properties of Composite Facesheets

Facesheets are considered to be thin composite laminates with continuous fibers. The stress strain relation for composite facesheet lamina can be expressed as:

$$\begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \varepsilon_z \\ \gamma_{yz} \\ \gamma_{xz} \\ \gamma_{xy} \end{Bmatrix} = \begin{bmatrix} 1/E_x & -\nu_{yx}/E_y & -\nu_{zx}/E_z & 0 & 0 & 0 \\ -\nu_{xy}/E_x & 1/E_y & -\nu_{zy}/E_x & 0 & 0 & 0 \\ -\nu_{xz}/E_x & -\nu_{yz}/E_y & 1/E_z & 0 & 0 & 0 \\ 0 & 0 & 0 & 1/G_{yz} & 0 & 0 \\ 0 & 0 & 0 & 0 & 1/G_{xz} & 0 \\ 0 & 0 & 0 & 0 & 0 & 1/G_{xy} \end{bmatrix} \cdot \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \sigma_z \\ \tau_{yz} \\ \tau_{xz} \\ \tau_{xy} \end{Bmatrix} \quad (7)$$

Each lamina can be considered as orthotropic (three mutually perpendicular planes of material symmetry exist) and it can be regarded as transversely isotropic (one of the material planes is plane of isotropy).

Using micromechanical approach, elastic coefficients in relation 7 (eq.7), E_x and ν_{yx}, based on facesheet constituent properties ca be obtained with great accuracy, what is experimentally confirmed.

However, the same approach fails to predict G_{xy} and E_y moduli, especially in cases where fiber volume fractions are between 30% and 55%, which is the usual range for modern advanced plolymer matrix composites. Rule of mixture theory does not take into account the effects of fiber/matrix interface (Fig. 8). The experiments have shown that this interface has a significant role in prediction of in-plane shear and cross-fiber Young moduli. These moduli can be computed using Halpin-Tasai and Chamis theories.

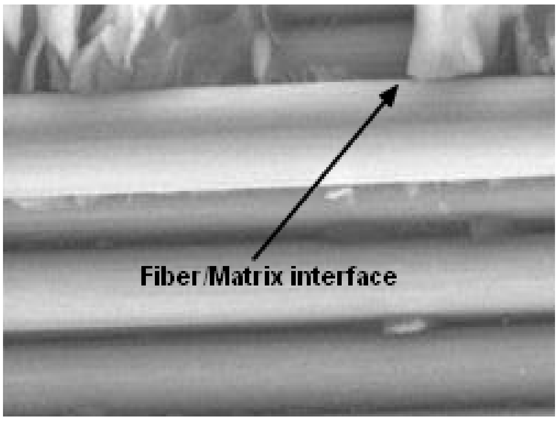

Fig. 8. Fiber/Matrix interface SEM

The formulation of Halpin-Tsai theory for E_y modulus can be expressed as:

$$E_y = \frac{\left(V_f + \eta_{22} \cdot (1 - V_f)\right) \cdot E_m E_f}{\left(E_m V_f + \eta_{12} \cdot (1 - V_f) E_f\right)} \tag{8}$$

And for the G_{xy} modulus, according to the same theory the expression is:

$$G_{xy} = \frac{\left(V_f + \eta_{12} \cdot \left(1 - V_f\right)\right) \cdot G_f \cdot G_m}{\left(G_m V_f + \eta_{12} \cdot \left(1 - V_f\right) G_f\right)} \tag{9}$$

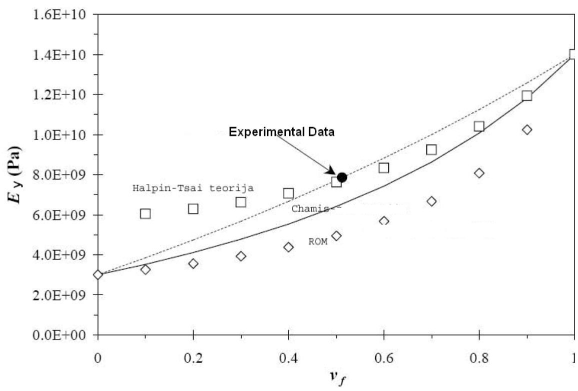

Fig 9. In-plane E_y modulus for composite facesheets (epoxy/carbon)

Values for η_{22} and η_{12} (stress partitioning coefficients) for common fiber types, used in equations 8 and 9 are given in Table 2.

Table 2 Fiber stress partitioning coefficients

	η_{22}	η_{12}
Carbon	0.500	0.400
Glass	0.516	0.316
Kevlar	0.516	0.400

The out of plane facesheet Young's modulus (E_z) is the same as E_y modulus for thin polymer matrix lamina with perfectly aligned fibers

3. NUMERICAL ANALYSIS

To determine complete three-dimensional stress-strain field of the composite panel under bending loads finite element approach was used. Three dimensional model was constructed and based on previous material models for each component in the structure the displacement, stress and strain fields were obtained.

In the present analysis, complete structure is modelled by means of 3D finite elements based on the 3D elasticity theory. Solid rectangular tri-linear finite elements were used to discretize the sandwich core, composite facesheets and the repair plug in the case of repaired sandwich panels.

In the present analysis eight-nodal hexahedron finite elements are used to model the complete sandwich beam structure. The finite element is presented in the following picture (Fig. 10).

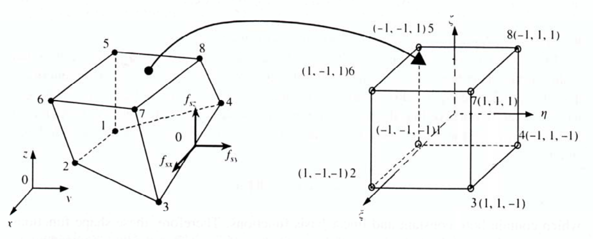

Fig. 10. eight-nodal hexahedron finite element and coordinate systems

The shape functions for the finite element used in the domain decomposition (in the local natural coordinate system) are:

$$N_i = \frac{1}{8}(1 + \xi\xi_i)(1 + \eta\eta_i)(1 + \zeta\zeta_i) \tag{10}$$

where (ξ_i, η_i, ζ_i) denote the natural coordinate of the i-th element node. From the previous equation it can be seen that the shape functions vary linearly in the $\xi, \eta,$ and ζ directions. The tri-linear finite elements used posses the delta function property, and since the shape functions (eq. 10) can be formed using the common set of eight basis functions ($1, \xi, \eta, \zeta, \xi\eta, \xi\zeta, \eta\zeta, \xi\eta\zeta$) they also posses linear reproduction property and partition of the unity property.

Due to symmetry only one quarter of the sandwich beams subjected to four point bend test was modeled.

Four point bending test was numerically simulated. Test set up is presented in Figure 11. Loading force was calculated using engineering bending theory assuming material properties of the panel listed in the table 3 and based on material models described in previous section. Maximal bending force for the composite panel under investigation was 4755 N. Under these conditions no failure of the panel occurs (panel buckling, shear crimping, skin wrinkling, intracell buckling and local top facesheet local compression). The finite element model consisted of 84613 hexahedron solid elements in order to achieve convergence.

The Nomex honeycomb core was modeled as solid continuum with 3D orthotropic elements. Each facesheet was modeled with 3 layers of solid elements since the high material anisotropy between facing structure materials was present. Applying symmetry boundary conditions only one quarter of the panel was modeled and the analysis model is presented in Figure 12. The applied boundary conditions are as follows: at the level of the supports the transverse displacement *w* is set to zero. At the first symmetry plane (mid-span *zy* plane) displacement in the *x* direction, rotations about *y* and *z* axis are set to zero and at the second symmetry plane (mid-width *zx* plane) the y direction displacement, and rotations about *x* and *z* axis are set zero.

Before finite element calculation was done, mesh sensitivity study has been performed to ensure that the finite element meshes of the core, transition regions, boundary condition regions and the facesheets are fine enough to yield satisfactory results.

The stress analysis results, for undamaged aand repaired panels, using this approach are presented in the following diagrams (Fig. 12 to Fig. 17).

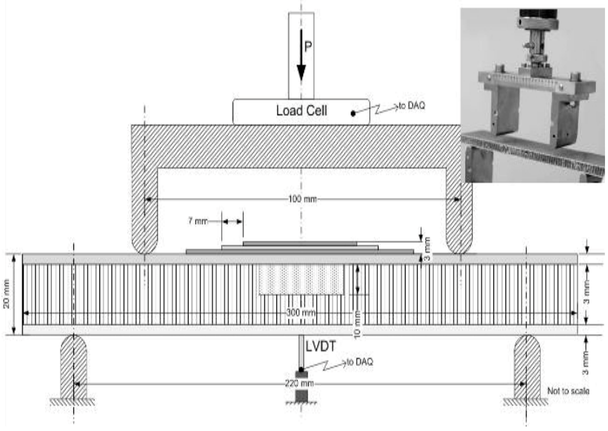

Figure 11. Repaired panel under four point bend test

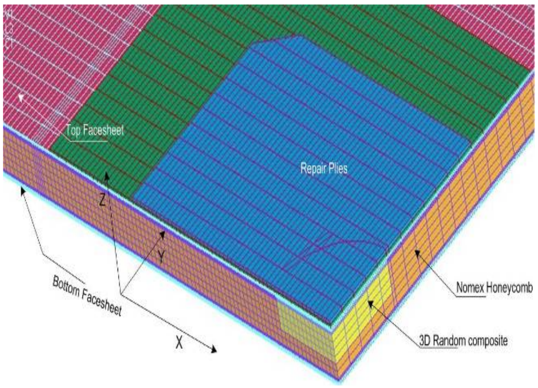

Figure 12. Finite element model (four point bending)

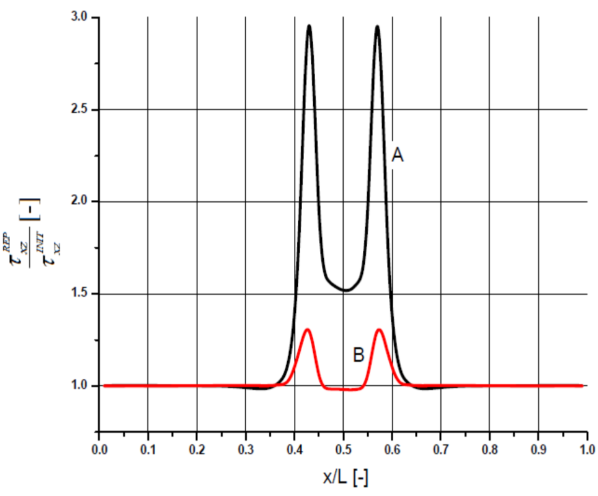

Figure 13. Shear stress in repaired composite core under bending load: A) Filler material with V_f=0.45 , unmatched shear modulus B) Filler material with V_f=0.07 , matched shear modulus (stresses relative to initial stress field)

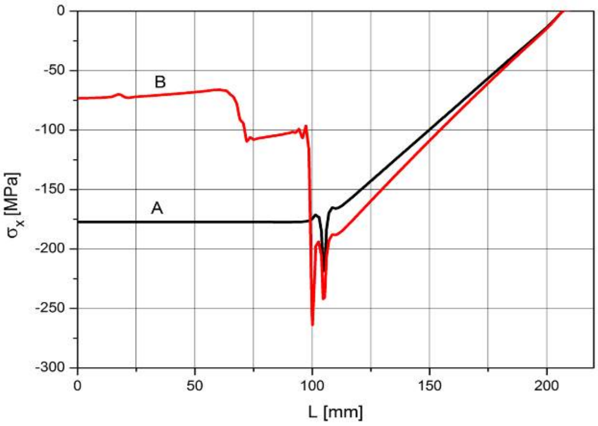

Figure 14. Upper Facesheet normal stress A) undamaged panel, B) repaired panel

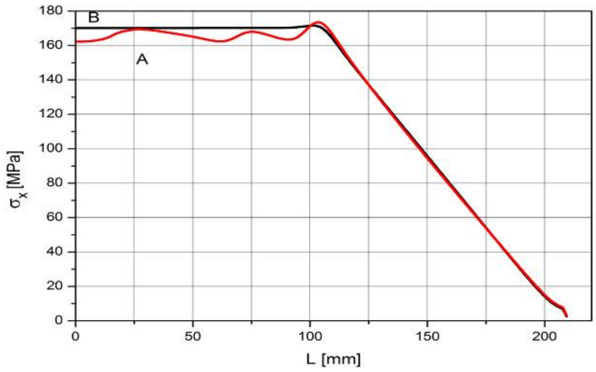

Figure 15. Lower Facesheet normal stress A) repaired panel, B) undamaged panel

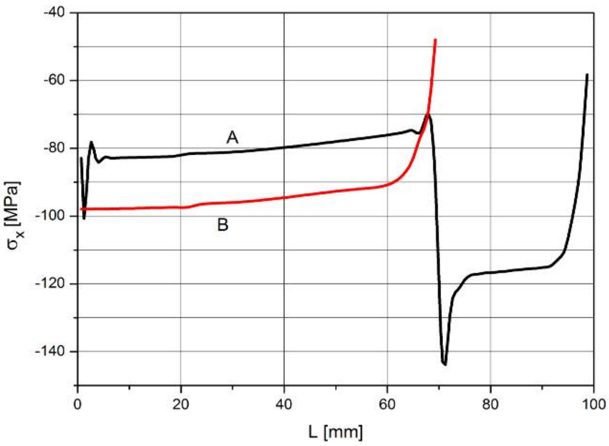

Figure 16. Repair plies normal stress A) Lower repair ply, B) Upper repair ply

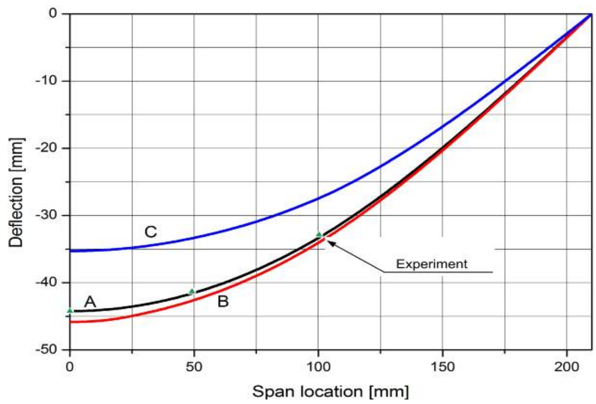

Figure 17. Panel deflection A) Undamaged panel, B) Damaged panel, C) Repaired panel

4. CONCLUSION

3D random fiber composites can be successfully used for damaged sandwich composite panel repair. In situations when the original core replacement material is not readily available, specially designed 3D random fiber composite plugs can be used. Mechanical properties of these inserts can be estimated based on existing theories and required volume fraction of reinforcement phase can be accurately calculated in order to match (primarily shear) mechanical properties of the damaged core. As a function of constituent properties and original structure core material shear properties, relation for the required fiber volume fraction of the replacement core plug is derived.

Finite element method can be successfully used in determining the complete stress, strain and displacement fields of repaired composite sandwich panels with 3D random composites. Using this approach complete three-dimensional stress, strain and displacement fields have been determined for the repaired honeycomb sandwich panels under bending loads.

REFERENCES

[1] Armstrong, K.,B.: Efforts to standardize aerospace composite repairs, 1988–2003 Proceedings of the Institution of Mechanical Engineers, Part G: Journal of Aerospace Engineering May 1, pp. 217-223,236, 2003.

[2] Caminero, M.A., Pavlopoulou, S., Lopez-Pedrosa, M., Nicolaisson, B.G., Pinna, C. and Soutis, C.: Analysis of adhesively bonded repairs in composites: Damage detection and prognosis, Composite Structures Vol. 95, pp. 500-517, 2013.

[3] Rachid, M., Serier. B., Bouiadjra, B. and Belhouari, M.: Numerical analysis of the patch shape effects on the performances of bonded composite repair in aircraft structures, Composites Part B: Engineering, Vol. 43, No. 2, pp. 391-397, 2012.

[4] Albedah, A., Bouiadjra, B., Mhamdia, R., Benyahia, F. and Es-Saheb, M.: Comparison between double and single sided bonded composite repair with circular shape, Materials and Design, Vol. 32, No. 2, pp. 996-1000, 2011.

[5] Dinulovic, M.: *Composite Structures Repair Methodology Development*, *Based on the Local Stress Strain Condition*, PhD thesis, Faculty of Mechanical Engineering, University of Belgrade, Belgrade, 2008, (in Serbian)

[6] Rašuo, B.: *Aircraft Production Technology*, Faculty of Mechanical Engineering, University of Belgrade, Belgrade, 1995, (in Serbian)

[7] Bonded Repairs of Aircraft Composite Sandwich Structures, FinalReport, DOT/ FAA/AR-03/74, February 2004.

[8] Office of Aviation Research, Guidelines for Analysis, Testing, and Nondestructive Inspection of Impact-Damaged Composite Sandwich Structures, Final Report, DOT/FAA/AR-02/121, March 2003.

[9] Rašuo, B.: Experimental study of structural damping of composite helicopter blades with different cores, Plastics, Rubber and Composites, Vol. 39, No. 1, pp.1-5, 2010.

[10] Rašuo, B.: An experimental methodology for evaluating survivability of an aeronautical construction from composite materials: An overview, International Journal of Crashworthiness, Vol. 12, pp. 9-15, 2007.

[11] Rašuo, B.: Experimental techniques for evaluation of fatigue characteristics of laminated constructions from composite materials: Full-scale testing of the helicopter rotor blades Journal of Testing and Evaluation, Vol.39, pp. 237-242, 2011.

[12] Pan, Y., Iorga, L., Assimina, A. and Pelegri, P.: Analysis of 3D random chopped fiber reinforced composites using FEM and random sequential adsorption, Computational Materials Science, Vol. 43, pp. 450–461, 2008.

[13] Garinis, D., Dinulovic, M. and Rašuo, B.: Dynamic analysis of modified composite helicopter blade, FME Transactions, Vol. 40, pp. 63-68, 2012.

[14] Masters, I.G. and Evans, K.E.: Models for the Elastic Deformationof Honeycombs, Composite Structures, Vol. 35, pp 403 – 422, 1996.

[15] Nast, E.: On honeycomb-type core moduli, AIAA/ASME/AHS Adaptive Structures Forum, Kissimmee, FL, Apr. 7-10, 1997,Collection of Technical Papers, Pt. 2 (A97-24112 05-39)

[16] Zhang, J. and Ashby, M.F.: The Out-of-Plane Propties of Honeycombs, International Journal of Mechanical Sciences, Vol. 34, No. 6, pp. 475–489,1992.

КОМПОЗИТНИ МАТЕРИЈАЛИ СА СЛУЧАЈНИМ РАСПОРЕДОМ ВЛАКАНА КАО МАТЕРИЈАЛИ ЗА ПОПРАВКУ ОШТЕЋЕНИХ САЋАСТИХ ЈЕЗГАРА

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Повећана употреба композитних материјала донела потребу за развојем поузданих и ефикасних техника поправки. Као и све друге носеће металне структуре и композитне структуре склоне су оштећењима. Развијене технике поправки металних конструкције нису директно примењиве на композите и проблем поправки композитних структура мора бити детаљно проучен. У овом раду анализирана је примена композитних материјала са случајним распоредом влакана, као материјалом за поправку оштећених саћастих језгара високе густине сендвич конструкција. Помоћу Панове теорије изведен је израз потребног запреминског удела влакана материјала поправке и одређено је комплетно напонско деформационо стање поправљене композитне стуктуре методом коначних елемената. Закључено је да композитни материјали са случајним распоредом влакана представљају веома добре кандидате за поправку оштећених композитних структура са саћастим језгрима.

Table 3: Materials properties

	Core	Facesheet /Repair Ply	Core plug matrix	Core plug matrix	Core plug fiber
	Nomex HRH 10	Epoxy UD glass V_f=55%	Corecell M130	DER 332 epoxy	E-glass
ρ [kg/m3]	144	2100	140	1205	2600
E_1 [MPa]	-	39	176	2861	7250
E_2 [MPa]	-	8.6	176	-	-
E_3 [MPa]	600	-	176	-	-
$\sigma_{c\ max}$ [MPa]	15	39	2.31	-	-
G_{13} [MPa]	$115_{L\ dir.}$ / $69_{W\ dir.}$	-	59	-	-
$\tau_{13\ max}$ [MPa]	$3.5_{L\ dir.}$ / $1.9_{W\ dir.}$		1.98	-	-
$\sigma_{t\ max}$ [MPa]	-	1080	416	73	3400