Patch materials selection for ageing metallic aircraft structures using digital quantitative materials selection methods

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Abstract

When aircrafts reach the end of their service life, fatigue cracks are found to have developed along rivet holes and other highly stressed regions of aircrafts. In order to extend the life of these aircrafts, repairs should be made to arrest these cracks. Composite doublers or repair patches provide an innovative repair technique, which can enhance the way which aircrafts are maintained. Bonded repair of metallic aircraft structure is used to extend the life of flawed or under-designed components at reasonable cost. Such repairs generally have one of three objectives: fatigue enhancement, crack patching or corrosion repair. Repair of cracked structure may be performed, by bonding an external patch to the structure, to either stop or slow crack growth. The selected material must be able to withstand the expected conditions in the damaged area. The material selected for the patch will almost always be either metallic or composite and within these classes, there are many different materials with different advantages and disadvantages associated with their use. Applying materials selection methods would make it possible to select the best material considering every important participating factors and their level of importance. Materials selection is a multidisciplinary research, which integrates a large number of knowledge fields. It can help to have a good choice in design patch for damaged components. Here, two new digital logic methods will be examined to the above task.

1. Introduction

In the area of repair application, research is required to develop new methods of preparing metallic surfaces for adhesive bonding. While current methods are extremely effective, improvements would lead to further reductions in repair time (and hence cost) as well enabling repairs to be applied with reduced levels of quality assurance. This would assist for field-level repairs or perhaps battle damage type repairs. Adhesive bonding technology, particularly bonded composite repairs, has been successfully applied by several nations to extend the lives of aircraft by bridging cracks in metal structure, reducing strain levels, and repairing areas thinned by corrosion. Bonded composite reinforcements are highly efficient and cost effective when compared to conventional mechanically fastened approaches. In some cases, bonded repair technology is the only alternative to retiring a component.

Repairs can be broadly divided into non-patch procedures for minor damage and patch (or reinforcement) procedures to restore structural capability. The technique of repairing cracked metallic aircraft structures using high strength advanced composite materials, is commonly known as ‘‘Crack Patching’’ and was pioneered by the Aeronautical and Maritime Research Laboratories (AMRL), for the Royal Australian Air Force (RAAF) [1]. The composite reinforcement, also known as the patch, can be attached to a damaged or weakened structure either by a mechanical fastener or adhesive bonding. The use of adhesively bonded composite patches as a method of repair has several advantages over mechanically fastened repair methods, which include reduced installation cost, increased strength and fatigue life and hence effective crack retardation, reduced repair down time, elimination of unnecessary fastener holes in an already weakened structure and stress concentrations at fasteners, corrosion resistance, high stiffness, and lightweight. Three critical steps in implementing a repair are design, choice of materials and application. The material selected for the patch will almost always be either metallic or composite and within these classes are many different materials with different advantages and disadvantages associated with their use [2].

Till now, many of choices of materials made in the industry for this purpose are only the result of the person knowledge and his/her experiences. Material selection is a logical consequence of weighting the advantages and disadvantages. Applying materials selection methods would make it possible to select a material considering every important participating factors and their level of importance.

1.1 Patch criteria

The CRMS Guidelines (1998) indicates that considerations in selection of a patch material include stiffness, strength, thickness, conformability, service temperature, and product form. Repair materials may be conventional metals, fiber metal laminates, or composites. Factors that may dictate patch material selection include thickness, weight, stiffness, thermal expansion coefficient, ability to inspect the damage through the patch, and operating temperature requirements. Thinner patches can be designed with higher modulus materials. Composite materials have higher stiffness to weight ratio. Metals and fiber metal laminates have CTEs more compatible with the metal structure being repaired and are more capable of enduring elevated temperatures [3].

1.2 Objectives and organization of the paper

For the patch material, there are three main options generally considered: the fiber composites boron/epoxy and carbon/epoxy and the aluminum alloy-glass/epoxy laminate GLARE. Most Australian and U.S. repairs have used boron/epoxy as the reinforcement rather than graphite/epoxy by virtue of its superior properties. However, compared to carbon/epoxy, boron/epoxy is much more costly, less readily available and because of the large fiber diameter less formable. Thus carbon/epoxy is used whenever it is more cost effective or where very high formability is required. GLARE (aluminum/fiberglass laminate) is an alternative patch material that has high fatigue resistance and important benefits where minimizing residual stresses is important; however, it has limited formability and relatively low stiffness so is best suited to the repair of thin-skinned fuselage components [4].

Raizenne [5] investigated Fatigue cracks, in the CF-116 upper wing skin fastener holes. Crack initiation was believed to be the result of high compressive loading. He considered Boron and graphite for this purpose, so boron 5521/4 prepreg was selected for this application because of its superior strength and stiffness under compressive loading conditions and for its low (250°F) cure temperature. Boron has a coefficient of thermal expansion closer to that of aluminum than graphite. Boron's low electrical conductivity reduces galvanic corrosion with aluminum and allows eddy current inspection of the substructure.

Guijt and Verhoeven [6] successfully modeled two bonded repairs were applied as a prototype repair on a C-5A which performed on boron and Glare panels. The repairs are technically a viable solution for the crown cracking problem of the C-5A. With respect to the remaining service-life of the C-5A, the high initial cost of bonded repairs vs. the lower cost of mechanically fastened repairs, might prohibit a more durable (bonded) repair option. They consider the results of tests for an unpatched panel and both Glare and Boron patched panels. Both patch materials extend the fatigue life of the panels considerably. A Glare patch combines the advantages of composite materials with a higher CTE and is therefore a good candidate for this type of repair.

Chester [7] selected boron/epoxy composite in the wing pivot fitting (WPF) of the F-111 aircraft, as the reinforcement material because this material offers the highest strength, stiffness and expansion coefficient of available composites although at a high cost premium. Also he had careful attention to the reduction of residual stress which has been an important part of the overall development process.

Baker et al. [8] applied BFRP crack-patching to the field repair fatigue cracks in the aluminum alloy wing skin of Mirage III fighter aircraft. This material selected between two choices, boron/epoxy and carbon/epoxy. Although this material is more expensive for its better stiffness, fatigue resistance, and higher thermal expansion coefficient.

Baker et al. [9] chose boron/epoxy for the patch material an F-111 lower wing skin because of its high stiffness and strength which would minimize the aerodynamic profile of the patch. Because of its low electrical conductivity, through-the-patch eddy current NDI can readily be used to detect growth in the patched crack. Additionally, the low conductivity of boron/epoxy eliminates the danger of galvanic corrosion, which is a potential problem with carbon/epoxy patches.

Ratwani et al. [10] considered three different bonded materials namely aluminum, graphite, and boron for patching the T-38 lower wing skin. High stresses at the location of machined pockets precluded the use of aluminum due to very thick reinforcement design. A graphite/epoxy reinforcement design was thicker compared to boron/epoxy design. A thick reinforcement, bonded on one side, causes excessive out-of-plane bending and there by reduces the effectiveness of the reinforcement. Thus, relatively thin boron/epoxy reinforcement and better thermal compatibility with aluminum compared to graphite reinforcement provided the best load transfer and life extension prospect.

Schweinberg and Fienbig [11] expressed that the C-141 aircraft was experiencing primary structure fatigue cracking due to age, and increased and expanded mission requirements. Application procedures were developed and necessary equipment identified. Both boron and graphite were identified as candidate repair materials, and two demonstration repairs were applied using both materials. For this application it was found that graphite would work as well as boron. However, boron was selected due to material availability and the need to eliminate the problems of galvanic corrosion when fastening through the doublers was required.

Harkless et al. [12] identified corroded region on the forward cargo hook beam of a CH-47 aircraft. Graphite was chosen as the repair material because of its ability to conform to the necessary repair geometry. Some potential concerns arose because of the potential of galvanic action between the repair and the aluminum substrate.

British aerospace [13] has investigated the use of bonded graphite/epoxy patches for the repair of metallic of components and carried out design studies for the repair of primary and secondary structures on military aircraft. The assessment criteria used to identify suitable candidates for bonded composite patch repair are fatigue life enhancement, stiffness matching, reducing cost, suitable access for repair application and inspection, and reducing the number of mechanical fastener.

This paper presents the materials selection for composite patch repairs in aging metallic aircraft. The objective of this research is to rank the candidate materials which can be used in patch repairs and analyze the result accordance with real application. The processes by which the adhesive and patch materials are installed on the aircraft have a direct influence on the final properties and long-term durability of the repair. The material properties considered for design should take into account the effects of these processes, such as the cure cycle (time/temperature) and pressure application method used for adhesives and cocured patches. The need is for high strength and stiffness, fatigue and environmental durability and formability. The composites satisfy most of the requirements; however, their main disadvantage is their low thermal expansion coefficient which gives rise to undesirable residual tensile stresses in the repaired component [14].

2. Materials selection

There are numerous materials selection methods. Ashby is one of the pioneers in this field by introducing some novel methods. In recent years, Digital Logic (DL) and Modified Digital Logic Methods (MDL) have been proposed [15,16]. Additionally, due to some of the limitation of these methods, a new method or in a better word a basic modification to the existing methods was introduced which is so called Z-transformation method [17]. In all of the last three mentioned methods, each material property is assigned a certain weight depending on its relative importance to the others. It is called weighting factor (α). Then, the scaled value of each property of a material (Normalized material property, *Y)* with respect to the other candidate materials is calculated. Please see [15-17] for details and equations used in Dl, MDL, and Z-transformation methods to calculate weighting factors and normalized material properties. Finally, the performance index for each candidate material (γ) is found by using equation $\gamma = \sum_{i=1}^{n} \alpha_i Y_i$. The material that obtains the highest performance index is believed to be the best choice for the specific application. Table 1 presents the properties of the candidate materials for patch repair. Also the scaled property values performed by both Z-transformation and MDL method are given in tables 2 and 3, respectively.

Table 1
Candidate materials properties [2,3,18,19,20, 21].

Material	1 Coefficient of thermal expansion ($\times 10^{-5}$ /degree)	2 Approximate relative material cost	3 Young's modulus (GPa)	4 Shear modulus (GPa)	5 Elongation (%)	6 Density (g/cm^3)	7 Tensile strength (GPa)
Al 2024-T3	2.32	1	73	27	18	2.78	0.483
Al 7075-T6	2.3	1	72	27	11	2.81	0.572
Titanium alloy 6 AL/4V	0.9	12	110	41	14	4.5	0.95
Aramid/Epoxy	-0.8	2	82.7	2.07	2.5	1.38	2.9
Glass/Epoxy	0.61	1	72.5	3.52	4.8	2.58	4.03
HM Carbon/Epoxy	-1	18	390	7.1	1.8	1.81	6.9
Boron/Epoxy unidirectional	2.3	40	208	7	0.8	2	3.4
Graphite/epoxy unidirectional	2.8	13	148	5	1.3	1.6	2.17
Aluminum laminate ARALL	1.6	7	68	17	0.89	2.3	1.282
Aluminum laminate GLARE	2	5	65	14.72	0.52	2.5	0.717
Electroformed Nickel	1.31	2	207	76	30	8.88	0.317

Table 2
Scaled property values performed by Z-transformation method [17].

Materials	Y1 CTE	Y2 Relative cost	Y3 Young's modulus	Y4 Shear modulus	Y5 Elongation	Y6 Density	Y7 Tensile strength
Al 2024-T3	0.798	-0.721	-0.648	0.518	1.685	-0.279	-0.866
Al 7075-T6	0.783	-0.721	-0.658	0.518	0.531	-0.243	-0.820
Titanium alloy 6 AL/4V	-0.317	0.238	-0.267	1.663	1.025	1.783	-0.624
Aramid/Epoxy	-1.652	-0.633	-0.548	-1.522	-0.871	-1.958	0.385
Glass/Epoxy	-0.545	-0.721	-0.653	-1.404	-0.492	-0.519	0.969
HM Carbon/Epoxy	-1.810	0.760	2.611	-1.111	-0.987	-1.442	2.454
Boron/Epoxy unidirectional	0.783	2.676	0.740	-1.119	-1.152	-1.214	0.643
Graphite/epoxy unidirectional	1.175	0.325	0.123	-1.283	-1.069	-1.694	0.007
Aluminum laminate ARALL	0.233	-0.198	-0.699	-0.301	-1.137	-0.855	-0.452
Aluminum laminate GLARE	0.547	-0.372	-0.730	-0.487	-1.198	-0.615	-0.745
Electroformed Nickel	0.005	-0.633	0.730	4.528	3.664	7.035	-0.951

Table 3
Scaled property values performed by MDL method [16].

Material	Y1 CTE	Y2 Relative material cost	Y3 Young's modulus	Y4 Shear modulus	Y5 Elongation	Y6 Density	Y7 Tensile strength
Al 2024-T3	69.53	100.00	-18.28	19.71	13.49	0.71	-85.62
Al 7075-T6	68.23	100.00	-19.03	19.71	-32.71	-0.35	-82.97
Titanium alloy 6 AL/4V	-29.27	-83.33	5.96	50.10	-13.25	-37.44	-71.77
Aramid/Epoxy	-91.78	0.00	-11.30	-83.41	-85.23	100.00	-14.53
Glass/Epoxy	-51.29	100.00	-18.66	-73.27	-71.38	8.40	18.22
HM Carbon/Epoxy	-91.78	-88.89	100.00	-52.01	-89.39	53.51	100.00
Boron/Epoxy unidirectional	68.23	-95.00	50.26	-52.55	-95.30	39.21	0.00
Graphite/epoxy unidirectional	100.00	-84.62	25.68	-63.91	-92.36	73.17	-35.84
Aluminum laminate ARALL	21.18	-71.43	-22.10	-9.85	-94.77	21.37	-61.95
Aluminum laminate GLARE	48.45	-60.00	-24.46	-18.06	-96.95	11.81	-78.67
Electroformed Nickel	0.72	0.00	49.89	100.00	100.00	-68.16	-90.55

3. Case study

3.1 Case I : *Wing pivot F-111*

In this case, the aim is to provide high performance reinforcement. Thus it is needed (a) to provide high durability in the reinforcement and adhesive systems, (b) to minimize adverse residual stresses, (c) to avoid corrosion, mechanical or metallurgical damage to the wing when applying the reinforcement, and (d) to avoid stiffening other than in the desired directions [22]. According to these requirements the candidate material should have highly directional Young's modulus to ensure effective reinforcement in desired directions; high strength in particular high shear and peel strength to cope with in-plane shear stresses and out-of-plane tensile stresses; formability at relatively low pressures and temperatures to allow shaping and fitting to the complex curvature of the wing; the ability to incorporate inserts such as a softening strip; and a relatively high thermal expansion coefficient in the reinforcement direction to match that of the metallic structure and thus minimize residual stresses during the adhesive cure cycle. Also, higher values of elongation are preferred that minimize the danger of patch failure at even quite high elastic strain levels in the parent metal structure. Finally, lower density and cost is preferred.

Table 4 shows the calculations for α. It should be noted that MDL and Z-transformation have the same weighting factors. Table 5 presents the calculated performance index and the corresponding ranking of the candidate materials using three of the applied methods. Also, table 6 shows this ranking only for composite materials

Table 4

Application of modified digital logic method to wing pivot (case I)

Goals	Number of possible decisions																					Positive decisions	Weighting factors
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21		
CTE	3	2	1	3	3	3																15	0.179
Cost	1						1	1	1	1	1											6	0.071
Young's modulus		2					3					1	3	3	3							15	0.179
Shear modulus			3					3				3				3	3	3				18	0.214
Elongation				1					3				1			1			3	2		11	0.131
Density					1					3				1			1		1		1	8	0.095
Tensile strength						1					3				1			1		2	3	11	0.131

Table 5

Performance index and ranking of candidate materials for wing pivot using MDL and Z-transformation method (case I).

Materials	DL method [15]		The method of manshadi et al. [16]		Z-transformation method [17]	
	Performance index	Rank	performance index	Rank	performance index	Rank
Al 2024-T3	39.47	4	11.14	2	0.194	3
Al 7075-T6	37.21	6	4.97	3	0.093	4
Titanium alloy 6 AL/4V	36.12	7	-14.08	7	0.019	6
Aramid/Epoxy	18.07	11	-39.82	11	-0.501	11
Glass/Epoxy	22.31	10	-27.21	10	-0.239	9
HM Carbon/Epoxy	45.65	2	-9.54	6	0.234	2
Boron/Epoxy unidirectional	41.56	3	-5.63	4	-0.02	7
Graphite/epoxy unidirectional	38.98	5	-7.12	5	0.033	5
Aluminum laminate ARALL	27.22	9	-25.87	9	-0.231	8
Aluminum laminate GLARE	27.39	8	-25.75	8	-0.242	10
Electroformed Nickel	61.04	1	25.21	1	0.663	1

Table 6

Ranking of material selected between composite materials for wing pivot.

Materials	DL method [15]		The method of manshadi et al. [16]		Z-transformation method [17]	
	Performance sindex	Rank	performance index	Rank	performance index	Rank
Aramid/Epoxy	18.07	7	-39.82	7	-0.501	7
Glass/Epoxy	22.31	6	-27.21	6	-0.239	5
HM Carbon/Epoxy	45.65	1	-9.54	3	0.234	1
Boron/Epoxy unidirectional	**41.56**	**2**	**-5.633**	**1**	**-0.023**	**3**
Graphite/epoxy unidirectional	38.98	3	-7.12	2	0.033	2
Aluminum laminate ARALL	27.22	5	-25.87	5	-0.231	4
Aluminum laminate GLARE	27.39	4	-25.75	4	-0.242	6

3.2 Case II: *Fuselage crown cracking*

These cracks are possibly caused due to the usage of stress corrosion sensitive aluminum 7079-T6. For low temperature applications, analysis and experiments have shown that composite material with moderate coefficient of thermal expansion (CTE) can perform better as fuselage skin repair materials than the more traditional low CTE composite patch materials because the highest loads in a pressurized fuselage typically occur at low temperatures. The difference in CTE of the patch material and the fuselage plays an important role in patching effectiveness. In general, a moderate- or high- CTE material will cause the crack to be in compression. A low CTE material can actually put the crack in tension. Because of this effect, for this application a moderate- or high-CTE patch material is favorable over a low CTE patch material [6].

In order to match patch stiffness to that of the repaired structure**,** a boron patch will be much thinner than a GLARE patch. For very thick sections, or aerodynamic critical areas, the thickness of the patch could play an important role. On a relatively thin fuselage skin, in an area where the boundary layer is large, the thickness of the patch will play a minor role; therefore both patch materials are candidate. Therefore, the best material in the list must have higher Young's and shear moduli. In addition higher coefficient of thermal expansion is favorable because it reduces the severity of the residual stress problem. Also, higher values of elongation are preferred. They minimize the danger of patch failure at even quite high elastic strain levels in the parent metal structure. Finally, lower density and cost are preferred.

Table 7 performs the calculations for weighting factors. It should be noted that again weighting factors for MDL and Z-transformation are the same. Table 8 shows the calculated performance index and the corresponding ranking of the candidate materials using three of the applied methods. Also, Table 9 shows this ranking only for composite materials in fuselage crown cracking.

Table 7

Application of modified digital logic method to fuselage crown cracking (case II).

Goals	Number of possible decisions																					Positive decisions	Weighting factors
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21		
CTE	3	3	2	3	3	3																17	0.202
Cost	1						1	1	1	2	1											7	0.083
Young's modulus		1					3					1	1	3	3							12	0.143
Shear modulus			2					3				3				3	3	3				17	0.202
Elongation				1					3				3			1			3	3		14	0.167
Density					1					2				1			1		1		1	7	0.083
Tensile strength						1					3				1			1		1	3	10	0.119

Table 8

Performance index and ranking of candidate materials for fuselage crown cracking using MDL and Z-transformation method (caseII).

Materials	DL method [15]		The method of manshadi et al. [16]		Z-transformation method [17]	
	Performance index	Rank	performance index	Rank	performance index	Rank
Al 2024-T3	47.01	2	15.90	2	0.291	1
Al 7075-T6	42.46	3	8.05	3	0.045	6
Titanium alloy 6 AL/4V	38.76	5	-15.75	6	-0.180	10
Aramid/Epoxy	15.01	11	-44.67	11	-0.346	11
Glass/Epoxy	20.32	10	-28.57	10	-0.126	9
HM Carbon/Epoxy	32.10	7	-20.76	7	0.252	3
Boron/Epoxy unidirectional	38.29	6	-10.18	5	0.160	4
Graphite/epoxy unidirectional	39.04	4	-9.64	4	0.267	2
Aluminum laminate ARALL	27.68	9	-28.21	9	0.031	7
Aluminum laminate GLARE	28.87	8	-26.88	8	0.050	5
Electroformed Nickel	67.52	1	27.72	1	-0.068	8

Table 9

Ranking of material selected between composite materials for fuselage crown cracking.

Materials	DL method [15]		The method of manshadi et al. [16]		Z-transformation method [17]	
	Performance index	Rank	performance index	Rank	performance index	Rank
Aramid/Epoxy	15.01	7	-44.67	7	-0.346	7
Glass/Epoxy	20.32	6	-28.57	6	-0.126	6
HM Carbon/Epoxy	32.10	3	-20.76	3	0.252	2
Boron/Epoxy unidirectional	38.29	2	-10.18	2	0.160	3
Graphite/epoxy unidirectional	39.04	1	-9.64	1	0.267	1
Aluminum laminate ARALL	27.68	5	-28.21	5	0.031	5
Aluminum laminate GLARE	**28.87**	**4**	**-26.88**	**4**	**0.050**	**4**

4. Conclusion

In this paper three materials selection methods used for choosing the best material for repairing metallic aircraft. Each one of the methods gives a different ranking for candidate materials. The advantages of crack-patching over conventional repair procedure are: (1) no mechanical damage to surrounding structure, no fastener holes, (2) crack can be inspected through the patch by conventional eddy-current NDI, (3)the cracked area is protected from further external corrosion, (4) reinforcement is only in the direction required, no undesirable stiffening in other directions, and (5) patches can be removed and replaced with no damage to the surrounding structure.
Table 5 and 8 presents the ranking of all candidate materials. According to these tables, metallic repairs are better choices to help simplifying the design process; however, there are also very good reasons to consider the use of composite materials.

Composites make exceptionally good repair materials due to their resistance to fatigue stresses and corrosion. When selecting composite patch materials, the two most important physical properties are strength (Uniaxial ultimate strength) and stiffness (Young's modulus). These are two properties allow the patch to be manufactured much thinner than metallic patches, providing a lighter, more aerodynamic and desirable repair. Therefore, the chosen material can also be composite materials. Finally in Tables 6 and 9 composite materials are ranked with all three methods.
As it is seen in table 6, boron epoxy is placed among the first three materials used for RAAF cases. These materials are highly suited for use as a patching or reinforcing material for defective or degraded metallic structure.
Briefly the attributes of these composites include:
· High Young's modulus and strength, which minimizes the required patch thickness.
· Highly resistant to damage by cyclic loads.
· Immunity to corrosion, forms excellent protective layer.
· High formability, which allows easy formation of complex shapes.
· Low electrical conductivity, which facilitates use of eddy current NDI for monitoring the patched cracks and eliminates concerns with galvanic corrosion.

Also, in Table 9 GLARE which has been used in RAAF for crown cracking fuselage has the rank fourth among other composites. GLAREs are offered as an alternative to boron/epoxy for this special crack patching application. Extending the lives of aging transport fuselage structures, however, may involve repairs to large areas of thin fuselage skins and lap joints. These structures often see their highest mechanical stresses (due to pressurization) at the low temperatures encountered at cruise altitude. Hence, more attention to the thermal properties of composite materials may be needed when fuselage structures are being repaired. The results showed several advantages of GLARE over boron/epoxy patches in fuselage skin repairs due to improved thermal expansion compatibility between GLARE and aluminum. The results predict GLARE to be an effective, damage-tolerant fuselage repair material [13].

The main disadvantage of composites as patching materials results from their relatively low coefficient of thermal expansion compared to the parent material which results in residual tensile mean stresses in the repaired component. Although relatively costly, boron/epoxy is chosen as the patch or reinforcement for most Australian bonded composite repair applications, mainly because of its excellent mechanical properties, low conductivity and relatively high coefficient of thermal expansion. However, graphite/epoxy because of its better formability is chosen for regions with small radii of curvature and sometimes because of its low cost and much higher availability [23]. Finally, RAAF has had extensive experience in boron patching and many of repairs in there provide with this material however another material could have better advantages. As mentioned in the Sec. No 1 in all patch repair cases in RAAF (Australia), U.S and British, there are three main options generally considered and used: the fiber composites boron/epoxy, carbon/epoxy and graphite/epoxy. Also, according to table 6 and 9 these materials have best ranking among other composites.

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