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Parametric study on static and fatigue strength recovery of scarf-patch-repaired composite laminates

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ABSTRACT

Carbon–epoxy composite laminate recovery rates are analyzed with respect to their static and fatigue strengths following local-damage repair using the scarf-patch bonding method. The tensile strength recovery rate is tested for 1.9–11.3° scarf angles (scarf ratios: 1/30–1/5). Static tests are performed for three external-ply overlap lengths and two defect sizes. The fatigue characteristics are identified via fatigue testing of a defect-free specimen and two specimens with different scarf angles. The static test results indicate that the strength recovery rate is significantly lower for scarf ratios of 1/10 or greater. Conversely, 1/20 and 1/30 scarf-ratio specimens exhibit high recovery rates of 79.1% and 83.6%, respectively, compared to the defect-free specimen. The external-ply overlap length and defect-size effects are negligible. For a million-cycle fatigue test, the fatigue strength recovery rates of the 1/20 and 1/30 specimens are 39.7% and 52.1%, respectively, being lower than the static strength recovery rate.

1. Introduction

Composite materials exhibit excellent mechanical and chemical properties and, as a result, they are widely used in various fields, such as the automotive, aerospace, and wind turbine industries. In particular, such materials have diverse applications within the aerospace sector, in commercial aircraft, military aircraft, unmanned aerial vehicles, and launch vehicles. Although composite materials exhibit excellent mechanical properties and higher specific strength, specific stiffness, durability, and corrosion resistance compared to conventional metallic materials (which was even the case in the early 1980s), the application of composite materials in aircraft has been largely limited to aircraft secondary structures. This is because of the high price and production costs of the materials themselves, as well as a lack of information on design allowables. However, in accordance with an emphasis on the economic efficiency of aircraft fabrication and the stabilization of the related technology, composite materials are now being used more widely in primary structures such as fuselages and wings, in addition to aircraft secondary structures. Indeed, in some cases, entire aircraft have been designed using composite materials. Notable examples of the application of composite materials to the primary structures of aircraft include the Boeing 787, A350, F-35, and F-22 [1].

Composite materials in aircraft can sustain various types of damage, such as local damage experienced during the aircraft service lifetime. Typical causes of such damage include lightning strikes, tool drops, service vehicle collisions, hail, runway debris and birds, erosion, abrasion, manufacturing defects, excess heat exposure, and fluid infiltration [2]. Further, the effects of such damage can propagate as a result of the repeated loading applied to aircraft. Aircraft structures subjected to damage of this kind cannot maintain the integrity of the initial design phase; this results in a gradual decrease in strength or the occurrence and spread of fractures, which, in a worst-case scenario, can have a devastating impact on the safe operation of the aircraft. Therefore, it is important to maintain aircraft structural integrity through effective maintenance, and the development of appropriate maintenance technology that can take damage to composite aircraft structures into consideration is desirable.

In the past, damage to composite structures was repaired by replacing the structure or through use of mechanical fasteners such as bolts or rivets. However, with the increased use of integrated structures in recently designed aircraft, replacing the entire structure is not feasible in terms of both cost and time. Further, the use of fasteners for repair can cause protrusions on the aircraft surface, which may decrease the aerodynamic performance and

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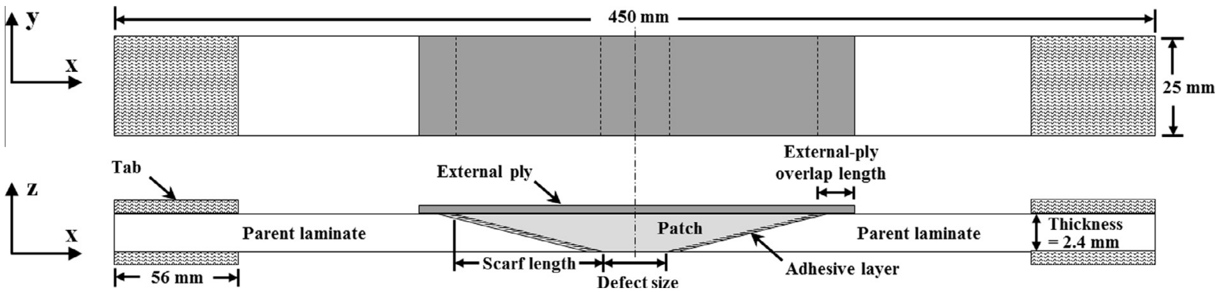

Fig. 1. Specimen configuration.

Table 1
Material properties of USN-125B and WSN-3K.

Property	Symbol	USN-125B	WSN-3KT
Elastic modulus (GPa)	E_1	142	70
	E_2	8.4	70
	E_3	8.4	9.6
Shear modulus (GPa)	G_{12}	5.34	3.59
	G_{13}	5.34	–
	G_{23}	3.06	–
Poisson's ratio	ν_{12}	0.298	0.058
	ν_{13}	0.298	–
	ν_{23}	0.47	–
Tensile strength (MPa)	X_T	2320	959
	Y_T	37.6	959
	Z_T	37.6	–
Compressive strength (MPa)	X_C	1400	–
	Y_C	130	–
Shear strength (MPa)	S_{12}	82.3	119
	S_{13}	82.3	–
	S_{23}	40	–
Thickness (mm)	T	0.120	0.160

Table 2
Material properties of adhesive used in this study [16].

Property	Symbol	Value
Young's modulus (MPa)	E	2400
Shear modulus (MPa)	G	840
Flatwise tensile strength (MPa)	σ_a	7.73
Shear strength (MPa)	τ_a	54.4
Poisson's ratio	ν	0.4

negatively affect the stealth capabilities of the aircraft. Therefore, repairs are now widely performed using adhesives instead of fasteners. To increase the convenience of such repair work, the overlap repair method, which involves adding additional composite layers to a damaged external part, is generally used for repairs to thin structures such as the fuselage skin. If the damaged structure is thicker, however, the scarf repair method may be a superior solution. This method has more advantages than the overlap repair method in certain scenarios, because it does not create an additional bending moment or stress concentration at the end of the adhesive layer, as is the case for the overlap repair method. The scarf repair method involves removal of the damaged area followed by application and assembly (using adhesives) of a cured laminate manufactured in scarf form with a gradual slope. Unlike an overlap repair, the delicately crafted scarf joints can withstand high loading, as the stress concentration in the bonded portion is minimal [3–5]. Further, an external ply is often applied following scarf repairs. Note that application of an external ply can reduce peak stresses, delay crack initiation at the scarf edge, and protect the repaired part from environmental damage or external impact [6].

In general, a scarf angle θ of below 3° is required for the scarf repair of structures comprised of carbon–epoxy composite materials [7]. However, a long scarf is necessary for repairs with a low θ (such as 3°), and long scarves are difficult to manufacture. Further, the use of a long scarf necessitates the removal of a large amount of undamaged parent laminate. There are also cases in which adequate repair areas cannot be secured on a structure for which a low θ is required. Therefore, research has been conducted to verify the effects of variations in the θ values on the efficacy of the repair, in order to both simplify the scarf production and reduce the parent laminate loss [6,8–14]. However, the majority of the existing research focuses on simplified scarf joints only, and very limited practical research has been conducted in which the parent laminate of an actual structure is removed, repairs using a composite patch are conducted, and an external ply is applied. Furthermore, it is difficult to accurately manufacture experimental specimens for repairs with θ lower than 3°, as noted above; therefore, the published studies on scarf repairs typically rely on numerical analyses.

In this paper, an experimental study is conducted in order to identify the static and fatigue strength recovery rates of a carbon–epoxy composite laminate specimen with local damage that has been repaired using a scarf patch. These strength recovery rates are measured by applying a tensile load. To identify the appropriate θ value from the static tests, testing is performed on specimens with four different scarf ratios (i.e., the ratio of the laminate thickness to scarf length), and the θ values and conditions of the repaired parts of the specimens are verified through microscope imaging. Furthermore, static tests are performed on specimens with two different defect sizes and three different external-ply-overlap lengths, which are additional variables that can affect the repair strength. Fatigue tests are then performed on the specimens with scarf ratios of 1/20 and 1/30, which are found to exhibit high static strength recovery rates. In addition, to assist understanding of the damage phenomenon, the stress distribution in the adhesive is analyzed via finite element analysis (FEA).

2. Test

2.1. Specimen preparation

Fig. 1 shows the designed specimen configuration. The geometry is a simplified approximation of a three-dimensional scarf repair with a cone-shaped model, and it is based on the ASTM D3039/D3039M-00 specification [15]. A scarf length of 450 mm was selected for this study based on the maximum scarf length (scarf ratio: 1/30). USN-125B, which is a carbon–epoxy unidirectional (UD) prepreg manufactured by SK Chemicals (Seongnam,