Effect of Bonding Technique on Debonding of Complex Composite Skin/Stringer Configuration

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Abstract

Delamination/Debonding is one of the predominant forms of failure in laminated composite structures, especially when there is no reinforcement along the thickness. To utilize composite structures that are more damage tolerant, it is necessary to understand how delamination or debonding develops, and how it can affect the residual performance of the parent structure. A number of factors such as residual thermal stresses, matrix-curing shrinkage and manufacturing defects affect damage growth in a composite structure. It is important to assimilate which manufacturing or assembling technique fits the best. The objective of the current work is to identify the best joining technique for an un-symmetric lambda shape complex stringer, in order to avoid or delay debonding of skin-stiffener composite specimens. Adhesive bonding and Co-curing/ Co-bonding is investigated through three-dimensional finite element analyses. Commercially available software HYPERMESH and ANSYS were used to do the 3D meshing and analysis. Adhesive bonded structure was analysed by contact analysis and co-cured structure was analysed by cohesive zone modelling technique (CZM). Under multi-axial loading, adhesive bonded configuration showed larger stresses compared to co-cured configuration. The former skin-stringer configuration de-bonded 83.8 % more than later in normal direction and 61.04 % in x-direction. It is concluded that for unsymmetrical lambda shape stringer, under given multi-axial, co-curing is the best method of manufacturing.

Key Words: Composite Stringer, Complex Stringer, Joining, Debonding, Adhesive Bonding, Cocuring

1. INTRODUCTION

The future generations of aircraft structures is reckoned to be made of all-composite material keeping the same design composed of stiffeners, frames and thin skins, i.e. the semi-monocoque construction. However, the stiffeners of traditionally used metal structures are riveted whereas advances in composite technology facilitate the stiffeners and skin to be manufactured together. Stiffened aeronautical structures are intended to allow post-buckling, thus permitting a lighter structure. The composite stiffened panels generally have much higher post buckling hence these are useful structural design candidate for the aerospace vehicle.

In the post-buckling phase, the appearance of buckle and redistribution of the forces in the stiffener lead to localized stresses at the stiffener/skin interface. Thus, the final failure of these structures may occur by stringer debonding between stiffener flanges and the skin of the wing [1]. The skin-stiffener debonding failure reduces the post buckling strength of the composite stiffened panels [2].

Conventionally used open stringer section offer good bending strength and accessibility for joining and reparability but are not very effective in torsion. On the other hand, closed section stringer offer good torsional strength but are poor in bending and have distinct reparability and corrosion problem. The type of section chosen greatly influences the torsional stiffness of a component or structure.

This paper focuses on the evolved stiffened skin-stringer composite structure and analyses the prevailing bonding techniques to propose the optimum technique to avoid debonding of complex stringer to skin. Fig. 1 shows failed de-bonded composite skin-stringer structure [3]. Unsymmetrical lambda shape stringer [4], as shown in Fig. 2, is chosen for analysis as it offers high resistance in flexural torsional buckling. Additionally, it offers better reparability and joining accessibility with rib castellation when compared to conventional “I”, “J”, hat section stringers [4]. Adhesive bonding and Co-curing joining technique were analyzed by three dimensional finite element methods for the unsymmetrical lambda stringer. Cohesive Zone modeling (CZM) technique has been extensively used to model the debond interface. It is an effective methodology to study and simulate fracture in solids in comparison to the conventionally used Linear Elastic Fracture Mechanics (LEFM). It is a phenomenological model instead of an exact physical representation of material behavior in the fracture process zone, where distributed micro cracking or void formation would takes place [5].

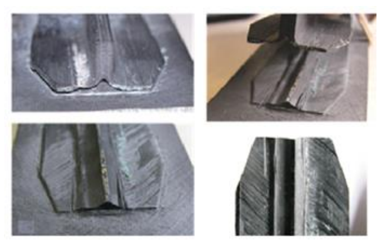

Fig. 1 Debonding failure [3]

The delamination response of composite laminates is formulated with a traction-separation law in the CZM, which is described with the following factors: stiffness, strength, fracture energy and its shape. CZM has been used in the prediction of delamination of composite structures; good correlation between experiments and simulation confirms the applicability of CZM in the delamination analysis, [6].

Stiffened skin construction, additionally if composite; provides good load bearing capabilities with less weight. The use of composite stiffened panel comes with an associated disadvantage of skin-stringer delamination or debonding, this might be either in the bonding interface or in the structure itself. The damage mechanisms in composite bonded skin/stringer constructions under uni-axial and bi-axial (in-plane/out-of-plane) loading conditions as typically experienced by aircraft structural panels were investigated [7]. Two-dimensional non-linear plane strain finite element (FE) analysis was performed a tapered composite flange, representing a stringer or frame, bonded onto a composite skin. Multilevel analysis of composite skin/stiffener debonding were performed [1]. The "pyramid of test" approach is used both for experimental and numerical analysis. Simple T shaped stiffeners, tapered at the ends were used for 7 point-bending tests and the approach was finally validated on real panels of an ongoing program during the time of research at Airbus. This paper presented a detail debonding assessment of open section stringers and concluded that open section stringer are very effective in flexural loading. A step-by-step simulation of ply damage based on strain softening decohesion elements; precisely the cohesive zone modeling technique was presented [8]. The methodology was applied to the prediction of the debond loads of skin and stringer, a problem clearly dominated by delamination. The methodology clearly described the evolution in application of traditional fracture mechanics for analyzing co-cured structure. Unlike isotropic materials, the composites bear a complex response to loadings which can be analyzed now by FEM because of CZM. A study on the global behavior of a composite stiffened panel with integrated structures in two directions undergoing buckling were presented [9]. Stiffened skin with hat shaped stringers infused was selected as the specimen and it was concluded that modifying connections between several structures considerably influences the collapse of the panel. The drawbacks associated with multiple joints, repairs or design and behavior of closed section stringer under in-plane loads was presented. A damage model used for investigating the deformation and interfacial failure behavior of an adhesively bonded single-lap thick joint was proposed [10]. Cohesive Zone Modeling (CZM) [7] provides an effective methodology to study and simulate fracture in solids. It does not represent any physical material, but describes the cohesive forces which occur when material elements are being pulled apart.

The failure behaviours (including skin stiffener debonding) and strength may be influenced by bonding methods and other design parameters. So, it is very important to predict the load bearing capability of the composite stiffened panels considering the skin-stiffener debonding failure, especially delamination between stringer foot and skin in thin-walled stiffened structures have to be taken into account.

2. GEOMETRICAL MODEL

The CAD modelling of the skin-stringer model was done in CATIA V5R19 and the meshing was done in HYPERMESH 12.0. The dimensions of the stringer and skin considered for the analysis are in-line with available literature on typical aircraft wing stringer and skin [11]. Fig. 2 shows the adhesive bonded skin-stringer model.

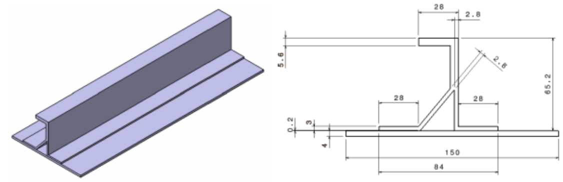

Fig. 2 Unsymmetrical lambda shape stringer [4]

3. NUMERICAL MODEL

The CAD model of the single lap joint was meshed in HYPERMESH 12.0. Two elements were meshed in skin and stringer flange incorporating 10 and 7 plies of 0.2 mm thickness in one element layer. Element type SOLID185 was used to mesh the composite parts. Cohesive material properties were assigned to the contacts between skin-adhesive and adhesive-stringer. Fig. 4 shows the FE model of the adhesive bonded skin-stringer model.

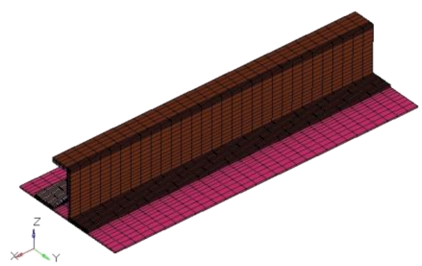

Fig. 4 FE model of adhesive bonded skin-stringer model

3.1 Material Properties

Skin and stringer material considered for three dimensional static non-linear analysis is IM6/3501.6 unidirectional graphite/epoxy and for adhesive its CYTEC 1515 (Assumed Isotropic). The material properties are provided in Table 1.

Table 1. Material properties

IM6/3501-6 graphite/epoxy					
E_1 =	144.7 GPa	v_{12} =	0.3	G_{12} =	5.2 GPa
E_2 =	9.65 GPa	v_{13} =	0.3	G_{13} =	5.2 GPa
E_3 =	9.65 GPa	v_{23} =	0.45	G_{23} =	3.4 GPa

Adhesive – CYTEC 1515	
Young's modulus, E	1720 MPa
Poisson's ratio, v	0.30

3.2 Loads

In-plane mechanical loads arising from wing bending and brazier loading and out-of-plane mechanical loads generated by fuel, payload and air pressures are considered in the analysis. Details of applied loads are tabulated in Table 2.

Table 2. Description of applied load

Nature	Magnitude
Aerodynamic Distributed load	23805
Axial Compressive Load	2500

3.3 Boundary Conditions

The integrity criterion is set to de-bonding of either the skin-stringer interface (Adhesive failure) or within the composite laminate itself (Cohesive failure). Fig. 5 shows the constrained FE model.

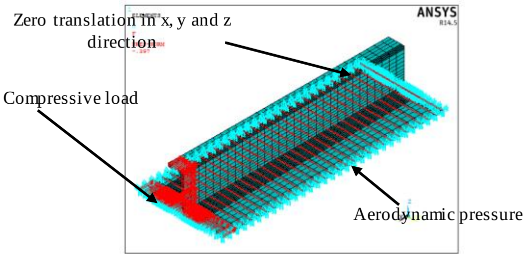

Fig. 5 Constrained FE model

4. VALIDATION STUDIES

The validation of procedure followed for FEA of stiffened skin with lambda shape stiffener is presented in this section. Single lap joint [12] and skin-flange model [13] are considered as benchmark model for FEA of adhesive bonded stiffened skin and co-cured stiffened skin respectively. The material of adherend in single lap joint is CFRP and for adhesive, it is DP-80051. Properties of CFRP and DP-80051 are presented in Table 3. Skin-stringer specimen has the material properties of IM6/3501-6.

Table 3. Material properties of adherend and adhesive [10]

CFRP					
$E_1 =$	109 GPa	$v_{12} =$	0.342	$G_{12} =$	4.315 GPa
$E_2 =$	8.81 GPa	$v_{13} =$	0.342	$G_{13} =$	4.315 GPa
$E_3 =$	8.81 GPa	$v_{23} =$	0.38	$G_{23} =$	3.2 GPa

Adhesive – DP80051	
Young's modulus,E	590 MPa
Poisson's ratio, v	0.35

4.1 Single lap joint

Fig. 6 shows the constrained FE model of the single lap joint.

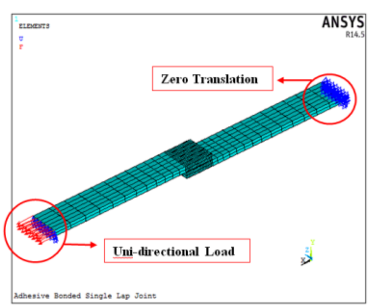

Fig. 6 Constrained single lap joint

As observed in Fig. 7 the results from the finite element analysis (a) are in agreement with the generic trend (b) [3]. The stress values are not compared here, and only the variation is compared.

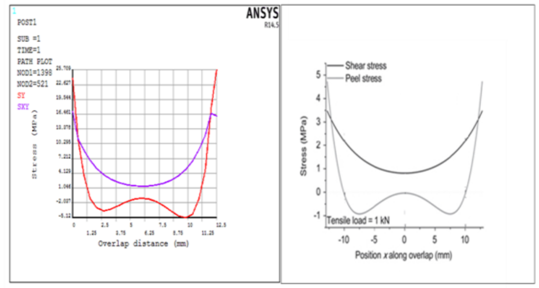

Fig. 7 Comparison on FE and Generic result trends

4.2 Skin-Flange Specimen

The model considered for the analysis is taken from [13]. Both skin and flange are made from IM6/3501.6 graphite/epoxy pre-preg tape with a nominal ply thickness of 0.188 mm. The delamination was assumed to occur at the interface between the skin and the flange, because of geometrical non-linearity and was in agreement with the results presented [13]. Cohesive zone modeling (CZM) technique is used to model the failure at the skin-stringer interface. Co-curing process is assumed to be carried out at room temperature and hence, thermal stresses are not considered for the analysis. The load and boundary condition as depicted [13] are applied to the model. Constrained model is shown in Fig. 8.

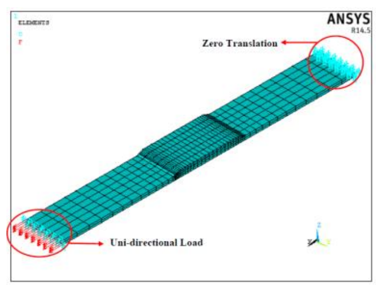

Fig. 8 Constrained FE stiffened skin-flange model

The results obtained are presented in terms load-strain relation. The debond growth is not symmetric across the width; the debond initiates on the left corner of the flange due to the lack of symmetry introduced by the terminated plies at the flange tapered ends as shown in Fig. 9.

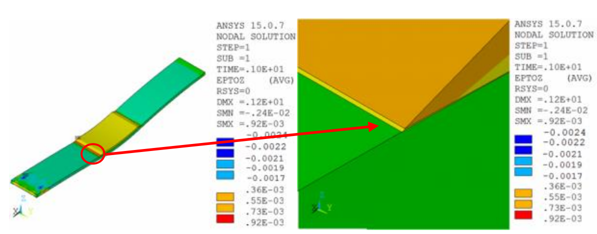

Fig. 9 Total mechanical strain in Z-direction

The analytical results were found to be in agreement with the result [13].

5. DISCUSSION AND CONCLUSION

As depicted in Fig. 10 and Fig. 11, for the same configuration and loading conditions, Co-cured configuration shows less debonding when compared to adhesive joined structure. The de-bonded sections show displacement in lateral and longitudinal directions, which were shown to be multiple times higher in adhesively, joined structure.

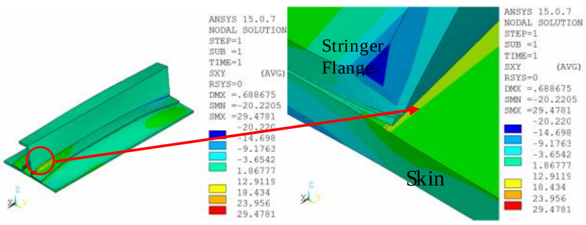

Fig. 10 Deformed shape and debonded adhesive bonded skin-stringer

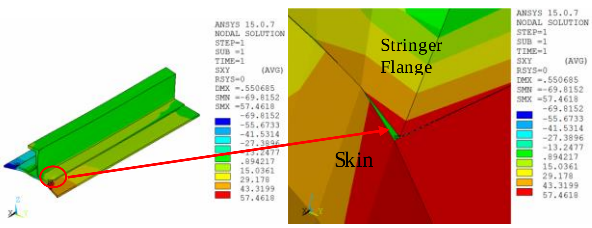

Fig. 11 Deformed shape and debonded occurred skin-stringer

Same loading condition shows different responses in both the cases. The compressive loads, responsible for the in-plane shear is largely relieved by the uniformly distributed aerodynamic load in the adhesive bonded joint. This is because, as soon as the strain energy release rate of the adhesive adherend interface reaches the threshold, adhesive separates from the skin/stringer flange and the flange deforms independently. On a contrast, as co-cured structure has no material nonlinearity, debond magnitude is less. This in turn holds back the structure and restricts further debond or delays the debond process. Hence, larger shear stresses are noted at extreme end of skin, Fig. 12.

Adhesive bonded and co-cured lambda stringer/skin structure were analyzed and it was observed that former de-bonded 83.8 % more than later in normal direction and 61.04 % in x. direction.

It is concluded that co-cured structure have more resistance to debond for same static loading. Hence, the subject structure must be co-cured to form assembly.

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