Failure Prediction in Fiber Metal Laminates for Next Generation Aero Materials

H Jeevan Rao¹, Perumalla Janaki Ramulu², M Vishnu Vardhan³, CH Chandramouli³

¹Department of Aerospace Engineering, Vardhaman College of Engineering, Shamshabad, 501218, India
²School of Mechanical, Chemical and Materials Engineering, Adama Science and Technology University, Adama, Ethiopia, P.Box:1888
³Department of Mechanical engineering, Vardhaman College of Engineering, Shamshabad, 501218, India

Email: perumalla.janaki@astu.edu.et

Abstract. In aerospace industry, there is huge demand for low density and low cost materials with better mechanical properties. In this view, there are many researchers developed new materials in terms of composites. Similar manner, the present paper also aimed to produce a new approach for cost effective materials of 3D weaved glass fiber metal laminates (FML) with different compositions using a numerical study. A method for the simulation of progressive delamination based on de-cohesion elements has been presented. De-cohesion elements are placed between layers of solid elements that open and shear in response to the loading situation. The onset of damage and the growth of delamination are simulated without previous knowledge about the location, the size, or the direction of propagation of the de-laminations. A softening law for mixed-mode delamination that can be applied to any interaction criterion is also proposed. The constitutive equation proposed uses a single variable, the maximum relative displacement, to track the damage at the interface under general loading conditions. The material properties required to define the element constitutive equation are the inter-laminar fracture toughness’s, the penalty stiffness, and the strengths.

Keywords: Fiber metal laminates (FML); Epoxy resin, Metal matrix; Aluminum and Glass fiber

1. Introduction

For the past few decades, the application of fibrous composite materials in engineering applications has become increasingly popular, especially in the aeronautical and space sector. Their plethora of uses in both military and civil aircraft also extends to applications that are more exotic. Their growing utilities have arisen from a drive within the aerospace industry to produce light weight aircraft, as the cost of fuel increases and environmental awareness becomes an important consideration. Composites are preferred over the conventional materials, such as steel and aluminum, because of their strength/stiffness versus weight ratio and the ability to shape and tailor structures to produce more aerodynamically efficient structural configurations. However, reducing the weight and maintaining the structural integrity, affordability and durability continues to be a major issue in aircraft design. The manufacturing process, assembly process and performance of composites are well connected. However, metallic materials and their derivatives continue to have a fundamental role in applications where composites have yet to be exploited. This led to the development of a hybrid system partly made of fibrous composites, known as fiber metal laminates (FMLs). FMLs consist of alternating...
layers of thin metallic sheets and fiber-reinforced plastics. Composite materials have been subject of permanent interest of various specialists during the last decades. Firstly, military applications in the aircraft industry triggered off the commercial use of composites after the Second World War. The innovations in the composite area have allowed significant weight reduction in structural design. Composites offer many advantages when compared to metallic alloys, especially where high strength and stiffness to weight ratio is concerned. Additionally, they provide excellent fatigue properties and corrosion resistance in applications [1]. With all these advantages, composite structures have gained widespread use in the aerospace industry during the last decades [2–6].

The mechanical behavior of composite bolted joints has been extensively studied in the past by means of experimental, analytical and numerical approaches [7-11], mainly focusing on the determination of the load capability, the load and stress distributions and failure criteria of single- and multi-row bolted joints under the influence of varying laminate configurations and joint geometries. One of the most effective ways to improve the load capacity of composite bolted joints entails the local reinforcement of the composite laminate with high-strength metal layers [12-16], thus clearly improving its bearing and shear capabilities. The special feature of this reinforcement technique consists of only embedding the metal layers into the bolted joining area locally, which is accomplished either by ply-addition (metallic layers are inserted between the composite plies) or ply-substitution techniques (composite plies are replaced by metallic layers). Hence, the total load capability of this reinforcement approach depends not only directly on the load capability of the bolted joint, but also on the strength of the transition zone between the pure fiber composite material and the hybridized laminate region multiple layers of thin aluminum alloy sheets [17-20]. From the all the studies, it was understood that the essentiality of the new composites for structural applications is must. The present study aimed the new generation fiber metal laminates numerical simulation for aerospace industry.

2. Methodology

2.1 Code Development

Considering the dynamic nature of reiterations involved in the present paper, it has been decided to use Salome and Code Aster available as open source code in CAE Linux. As both were written in Fortran and python they provide end users with flexibility to define the boundaries in much detail. As first step of modeling, the SI unit system was followed for local and global systems. Geometrical configuration and boundary values were mentioned in the Table 1 and 2. Figure 1 shows the composite layer with mentioned geometrical and boundaries indicated in the Table 1 and 2.

| Property               | Value |
|------------------------|-------|
| Space dimension        | 3     |
| Number of domains      | 6     |
| Number of boundaries   | 29    |
| Number of edges        | 46    |
| Number of vertices     | 24    |

| Name       | Value |
|------------|-------|
| Width      | cl    |
| Depth      | wb/2  |
| Height     | Hb    |
| Layer 1    | hb/2  |
**Table 3** Composite layers details used for coding

| Name | Expression | Value | Description |
|------|------------|-------|-------------|
| lb   | 102[mm]   | 0.10200 m | Length |
| wb   | 25.4[mm]  | 0.025400 m | Width |
| hb   | 2×1.56[mm] | 0.0031200 m | Thickness |
| cl   | 34.1[mm]  | 0.034100 m | Initial crack length |
| Kp   | 1e6[N/mm^3] | 1.0000E15 | Penalty Stiffness |
| N_strength | 80[MPa] | 8.0000E7 Pa | Normal Tensile Strength |
| S_strength | 100[MPa] | 1.0000E8 Pa | Shear Strength |
| u_I_0 | N_strength/Kp | 8.0000E-8 m | Mode I failure initiation displacement |
| u_II_0 | S_strength/Kp | 1.0000E-7 m | Mode II failure initiation displacement |
| Glc | 0.969[kJ/m^2] | 969.00 J/m² | Mode I critical energy release |
| GlIc | 1.719[kJ/m^2] | 1719.0 J/m² | Mode II critical energy release |
| u_I_f | 2×Glc/N_strength | 2.4225E-5 m | Mode I ultimate displacement |
| u_II_f | 2×GlIc/S_strength | 3.4380E-5 m | Mode II ultimate displacement |
| eta | 2.284 | 2.2840 | Exponent of Benzeggagh and Kenane (B-K) criterion |

**Table 4** Cohesive Zone Definition

| Name | Expression | Description |
|------|------------|-------------|
| u_I | -solid.uspring3_tel1 | Normal relative displacement |
| u_II | sqrt(solid.uspring1_tel1^2 + solid.uspring2_tel1^2) | Total tangential relative displacement |
| u_m | sqrt(u_II^2 + u_I^2) | Mixed mode relative displacement |
| u_max | max(u_max_old, u_m) | Maximum mixed mode relative displacement |
| beta | if(u_I>0, u_II/u_I, 0) | Mode mixity |
| u_m_0 | if(u_I>0, u_I0×u_II_0×sqrt((1 - beta^2)/(1 + beta^2))^eta), u_II_0) | Mixed mode damage initiation |
| u_m_f | if(u_I>0,2/Kp/(GlIc/Glc×(beta^2/(1 - beta^2))^eta)), sqrt(2)×u_II_f) | Mixed mode total de-cohesion displacement |
| damage | u_m_fx(nojac(u_max)-u_m_0)/(nojac(u_max)×(u_m_f - u_m_0)) | Damage evolution function |
If $u_{\text{max}} < u_{\text{m}_0}$, $K_p$, if $u_{\text{max}} > u_{\text{m}_f}$, 0, $(1 - \text{damage}) \times K_p)$

**Table 5 Load Definition**

| Name   | Expression | Description               |
|--------|------------|---------------------------|
| $u_{\text{lp}}$ | $(3 \times l_b/2)/4(l_b/2) \times \text{intop1}(u_{\text{l}}) + ((l_b/2)/(l_b)) \times \text{intop2}(-w) + \text{intop1}(u_{\text{l}})/4)$ | Load point displacement |
| $F_{\text{lp}}$ | $\text{force} \times l_b/2/l_l$ | Load point force |

**Table 6 Mechanical Properties used for whole composite**

| Name         | Value                     | Unit          |
|--------------|---------------------------|---------------|
| Density      | 1570                      | kg/m$^3$      |
| Young’s modulus | [122.7e9, 10.1e9, 10.1e9] | Pa            |
| Poisson's ratio | {0.25, 0.45, 0.25} | 1             |
| Shear modulus | [5.5e9, 3.7e9, 5.5e9]    | N/m$^2$       |

Table 3 data indicates the complete data that used for the coding. It showed the entire the geometrical and boundary conditions of the composite layer. Similarly, cohesive zone details were showed in the Table 4. Table 5 and 6 load details and mechanical properties of composite layers and cohesive zone. Loss factor for both young modulus and shear modulus was considered as zero for both composite layers and cohesive zones.

Equation 1 was a characteristic equation used for the simulation

$$-\nabla \sigma = F$$  

The simulation settings were indicated in the Table 7. A quadratic displacement field was used by keeping boundary fluxes off during simulation. Table 8 indicated the variables used in coding.

**Table 7. Simulation settings**

| Description                                                   | Value               |
|---------------------------------------------------------------|---------------------|
| Displacement field                                           | Quadratic           |
| Compute boundary fluxes                                      | Off                 |
| Value type when using splitting of complex variables          | Complex             |
| Structural transient behavior                                 | Include inertial terms |
| Reference point for moment computation, $x$ component          | 0                   |
| Reference point for moment computation, $y$ component          | 0                   |
| Reference point for moment computation, $z$ component          | 0                   |
| Typical wave speed for perfectly matched layers               | $\text{solid.cp}$   |

**Table 8. Variables details in the coding**

| Name   | Expression | Unit | Description         | Selection       |
|--------|------------|------|---------------------|-----------------|
| solid.nX | nX        | 1    | Normal vector, X component | Boundaries 6, 10, 13, 15, 19, 22, 24 |
| solid.nY | nY        | 1    | Normal vector, Y component | Boundaries 6, 10, 13, 15, 19, 22, 24 |
| solid.nZ | nZ        | 1    | Normal vector, Z component | Boundaries 6, 10, 13, 15, 19, 22, 24 |
| solid.nX | dnX       | 1    | Normal vector, X component | Boundaries 1–5, 7–9, 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nY | dnY       | 1    | Normal vector, Y component | Boundaries 1–5, 7–9, 11–12, 14, 16–18, 20–21, 23, 25–29 |
| Field          | Description                                      | Components | Boundaries         |
|----------------|--------------------------------------------------|------------|--------------------|
| solid.nZ      | Normal vector, Z component                        | 1          | 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nx      | Normal vector, x component                        | 1          | 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.ny      | Normal vector, y component                        | 1          | 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nz      | Normal vector, z component                        | 1          | 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nXmesh  | Normal vector (mesh), X component                 | root.nXmesh| 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nYmesh  | Normal vector (mesh), Y component                 | root.nYmesh| 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nZmesh  | Normal vector (mesh), Z component                 | root.nZmesh| 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nxmesh  | Normal vector (mesh), x component                 | root.nxmesh| 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nymesh  | Normal vector (mesh), y component                 | root.nymesh| 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.nzmesh  | Normal vector (mesh), z component                 | root.nzmesh| 11–12, 14, 16–18, 20–21, 23, 25–29 |
| solid.refpntx | Reference point for moment computation, x component| 0          | Global             |
solid.refpnty 0 m Reference point for moment computation, y component Global
solid.refpntz 0 m Reference point for moment computation, z component Global
solid.cref solid.cp m/s Typical wave speed for perfectly matched layers Domains 1–6
xt d(x,TIME) m/s Mesh velocity, x component Global
yt d(y,TIME) m/s Mesh velocity, y component Global
zt d(z,TIME) m/s Mesh velocity, z component Global

Similarly for second layer also defined for data evaluation as shown in the Figure 2. Equations 2 to 5 were used for the simulation. Figure 3 shows the meshed geometry of the composite layers and the mesh details were indicated in table 9 including the boundary conditions.

![Figure 2. Composite layer 2](image)

\[0 = \nabla \cdot (FS) + F_v\]  \[\text{[2]}\]
\[F = I + \nabla u\]  \[\text{[3]}\]
\[s = s_0 + C : (\varepsilon - \varepsilon_0 - \varepsilon_{\text{inel}})\]  \[\text{[4]}\]
\[\varepsilon = \frac{1}{2} (\nabla u + (\nabla u)^T + (\nabla u)^T \nabla u)\]  \[\text{[5]}\]

Then we defined the load as shown in the Equations 6 and 7
\[F = F_L\]  \[\text{[6]}\]
\[F_L = \frac{F_{\text{tot}}}{L}\]  \[\text{[7]}\]

Using symmetry we define prescribed displacement as
\[w = u_{0z}\]  \[\text{[8]}\]

![Figure 3. Meshed composite layers along with boundary conditions](image)
Table 9 Details of the mesh and boundary conditions

| Name                          | Value         |
|-------------------------------|---------------|
| Maximum element size          | 0.00204/2     |
| Minimum element size          | 2.04E-5       |
| Minimum element size          | Off           |
| Curvature factor              | 0.2           |
| Curvature factor              | Off           |
| Resolution of narrow regions  | Off           |
| Maximum element growth rate   | 1.3           |
| Maximum element growth rate   | Off           |
| Predefined size               | Extremely fine|
| Custom element size           | Custom        |

The program was then prepped for computation by fixing the details shoed in the Table 10.

Table 10 Simulation conditions

| Property                                      | Value       |
|----------------------------------------------|-------------|
| Include geometric nonlinearity               | On          |
| Physics interface                            | Discretization|
| Solid Mechanics (solid)                      | physics     |
| Boundary ODEs and DAEs (bode)                | physics     |
| Geometry                                     | Mesh        |
| Geometry 1 (geom1)                           | mesh1       |
| Name                                         | Value       |
| Use study                                    | Study 1     |
| Use study step                               | Stationary  |
| Defined by study step                        | Stationary  |
| Solution                                     | Zero        |
| Field components                             | comp1.u_max_old |
| Field components                             | \{comp1.u, comp1.v, comp1.w\} |
| State components                             | comp1.force |

3. Results and Discussion

An extensive literature review has pointed out the existing approaches and methodologies used to evaluate Advanced Aero material numerically. In current approach, the authors applied the cohesive bond theory to evaluate the bond energy and stresses in the fatigued area to understand the de-lamination dynamics.

The model is computed for a mode ratio of 50%. The von Mises stress distributions are computed using parametric step method. The resultant crack initiated at this stage is represented in using red for de-bonded zone and green for healthy zone. Because of the nonlinearity and history dependence of the CZM, it is necessary to solve the model parametrically. The desired load is applied on the top edges of the beam. The force-displacement curve reveals that the applied forces are not monotonically increasing functions this can be used for the parametric solver.
To overcome the non-monotonicity in the interface of the joints, we define them using a simple code for the parameters which enable the Python binaries to focus the mesh deformation at the point of study using the formation functions.

One of the outputs of the mixed mode bending test is a load-displacement curve. Both load and displacement are measured at the end point of the lever that is used to apply the load to the test specimen. Since the layer is not explicitly modeled, the load-displacement data has to be deduced from the simulation results. Figure 5 shows the load vs displacement behavior of whole composite bar obtained from the simulation results. It shows, the maximum load is obtained as 255N corresponding 5.44 mm displacement. Finally using the above date, the resultant von Misses Stresses are shown in the Figure 6.

**Figure 4.** Displacement filed after simulation

**Figure 5.** Load vs Displacement behavior of composite bar
A method for the simulation of progressive de-lamination based on de-cohesion elements is presented in this work. De-cohesion elements are placed between layers of solid elements that open and shear in response to the loading situation. The onset of damage and the growth of de-lamination are simulated without previous knowledge about the location, the size, or the direction of propagation of the de-laminations. A softening law for mixed-mode de-lamination that is applied to any interaction criterion is proposed. The constitutive equation proposed uses a single variable, the maximum relative displacement, to track the damage at the interface under general loading conditions. The material properties required to define the element constitutive equation are the inter-laminar fracture toughness's, the penalty stiffness, and the strengths.

4. Conclusions
FMLs consist of metallic alloy and fibre reinforced prepreg. Mostly available GLARE, ARALL and CARALL consists various aluminium alloys. Many researchers have been trying to use possible metallic alloys such as magnesium, titanium, etc., instead of aluminium alloys. It is expected that this diversity lead optimum mechanical properties.
Same efforts have examined for engineering polymeric materials to replace fibre-reinforced prepreg. New processing methods suggested for improving the productivity of curing process and decreasing the labor costs of FMLs. These improvements will show FMLs very attractive to various industrial applications such as military, automotive and aircraft. By using thermoplastic matrix, FMLs will find new application areas. However, low compatibility of thermoplastic matrix with metal surfaces needs improved by surface modification methods. This study is a simple hypothetical phenomenon to understand the FMLs behavior under different conditions.

References
[1]. Sinmazcelik T, Avcu E, Bora M Ö, Çoban O, 2011 A review: Fibre metal laminates, background, bonding types and applied test methods, Materials & Design, 32(7), pp. 3671-3685.
[2]. Wu G, Yang J M, Hahn H T, 2007, The impact properties and damage tolerance and of bidirectionally reinforced fiber metal laminates, Journal of materials science, 42(3), pp.948-957.
[3]. Mathivanan N R, Jerald J, 2010, Experimental investigation of low-velocity impact characteristics of woven glass fiber epoxy matrix composite laminates of EP3 grade, Materials & Design, 31(9), pp. 4553-4560.
[4]. Morinière F, Alderliesten R, Tooski M, Benedictus R, 2012, Damage evolution in GLARE fibre-metal laminate under repeated low-velocity impact tests, Open Engineering, 2(4), pp. 603-611.

[5]. Seo H, Hundley J, Hahn H T, Yang J M, 2010, Numerical simulation of glass-fiber-reinforced aluminum laminates with diverse impact damage, AIAA journal, 48(3), 676-687.

[6]. Alderliesten R, Benedictus R, 2008, Fiber/metal composite technology for future primary aircraft structures, Journal of Aircraft, 45(4), pp. 1182-1189.

[7]. Seo H, Hundley J, Hahn H T, Yang J M, (2010, Numerical simulation of glass-fiber-reinforced aluminum laminates with diverse impact damage, AIAA journal, 48(3), pp. 676-687.

[8]. Botelho E C, Silva R A, Pardini L C, Rezende M C, 2006, A review on the development and properties of continuous fiber/epoxy/aluminum hybrid composites for aircraft structures, Materials Research, 9(3), pp. 247-256.

[9]. Bernhardt S, Ramulu M, Kobayashi AS, 2007, Low-velocity impact response characterization of a hybrid titanium composite laminate, Journal of Engineering Materials Technology, 129, pp. 220–226.

[10]. Villanueva GR, Cantwell WJ, 2004, The high velocity impact response of composite and FML-reinforced sandwich structures, Composites Science and Technology, 64, pp.35–54.

[11]. Beumler T, Pellenkoft F, Tillich A, Wohlers W, Smart C, 2006, Airbus costumer benefit from fiber metal laminates, Airbus Deutschland GmbH, Ref. no: L53pr0605135-Issue 1, pp. 1–18.

[12]. Alderliesten R C, Benedictus R, 2007, Fiber/metal composite technology for future primary aircraft structures, In: 48th Aiaa/Asme/Asce/Ahs/Asc structures, structural dynamics, and materials conference 15th; April 23–26, Honolulu, Hawaii; 2007, p. 1–12.

[13]. Cortes P, Cantwell W J, 2006, The prediction of tensile failure in titanium-based thermoplastic fiber–metal laminates, Composites Science and Technology, 66, pp. 2306–16.

[14]. Asundi A, Choi Alta YN, 1997, Fiber metal laminates: an advanced material for future aircraft, Journal of Material Processes and Technology, 63, pp. 384–94.

[15]. Vogelesang LB, Vlot A, 2000, Development of fiber metal laminates for advanced, Journal of Material Processes and Technology, 103, pp. 1–5.

[16]. Chang PY, Yeh PC, Yang JM, 2008, Fatigue crack initiation in hybrid boron/glass/ aluminum fiber metal laminates, Material Science and Engineering A, 496, pp. 273–80.

[17]. Alderliesten R, 2009, On the development of hybrid material concepts for aircraft structures, Recent Patents Engineering, 3, pp. 25–38.

[18]. Remmers J J C, Borst R D, 2001, Delamination buckling of fiber–metal laminates, Composites Science and Technology, 61, pp. 2207–13.

[19]. Schut J, Alderliesten RC, 2006, De-lamination growth rate at low and elevated temperatures in glare, In: 25th International congress of the aeronautical sciences, September 3–8; Hamburg, Germany, pp. 1–7.

[20]. Vlot A, 1996, Impact loading on fiber metal laminates, International Journal of Impact Engineering, 18(3), pp. 291–307.