An Analysis Methodology to Predict Damage Propagation in Notched Composite Fuselage Structures

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Outline

• Background

• Laminate cohesive approach (LCA)

• Full-scale fuselage panel test

• Test and analysis results

• Concluding remarks
Motivation: Predict Damage Containment Behavior

Damage containment is achieved through:
1. Multiple load paths (e.g. Skin and substructure)
2. Damage arresting features (e.g. Rivets)

Objective: introduce an analysis methodology to predict damage propagation behavior in composite skin-stiffened structures with a notch

Current state-of-the-art:
- Metallic structures: Damage containment
- Composite structures: linear threshold
Simple Case: Center Notch Test Specimen

Notched Strength Prediction

Laminate assumed:
- Homogeneous
- Orthotropic (multidirectional)

\[ \sigma_n = \frac{K_{IC}}{(\pi a)^n} \]

Comments:
1. Classical linear elastic fracture mechanics (LEFM) does not scale accurately
2. Mar Lin is accurate, but requires large-scale testing to calibrate
3. Detailed, mesoscale progressive damage analysis is still being developed. Unresolved issues remain, e.g.:
   - Difficulties with interaction of matrix cracks and delaminations
   - Often computationally intractable for large structures

Analysis methods that can predict notched strength accurately reducing the number of large-scale tests will save time and cost
Strain Softening Approach

(Dopker et al. SDM Conference, 1994)

- Strain softening approach can predict notched strength accurately, but trial-and-error required to calibrate $\sigma - \varepsilon$ law.

- Strain softening law determined by trial-and-error for notch lengths of 1.25 in. and 2.5 in.

- Analysis using strain softening predicts excellent agreement for notch length of 8 in.

Legend:
- O Test
- + Analysis: Strain softening
- --- Analysis: Classical

Notched strength, $\sigma_n$

IM7/8551-7 $[\pm 45/0/90/\pm 30]_s$
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Actual Versus Idealization of (LCA)

Actual

Thin fiber reinforced polymer (FRP) laminate
• Multidirectional layup
• Thickness: \( t \)

Notch

Through-the-thickness damage

Damage propagates by evolution and interaction of micro- and mesoscale damage mechanisms

Idealization

Assume the damage can be represented with the cohesive zone model (CZM)

Laminate assumed:
• Homogeneous
• Orthotropic

Objective: Characterize the cohesive law for a laminate and crack orientation

References:
Characterization of LCA

1) Assume a trilinear cohesive law $\sigma(\delta)$

$$\sigma(\delta) = \begin{cases} \sigma_1(\delta) & 0 \leq \delta \leq \delta_k \\ \sigma_2(\delta) & \delta_k < \delta \leq \delta_t \\ \sigma_3(\delta) & \delta_t < \delta \leq \delta_c \end{cases}$$

Formulated $\sigma(\delta)$ in terms of $\sigma_c$, $G_c$, $m$, and $n$

$$\sigma_1(\delta) = K\delta$$
$$\sigma_2(\delta) = \frac{n\sigma_c(\sigma_t - \sigma_c)}{2mG_c} \delta + \sigma_c$$
$$\sigma_3(\delta) = \frac{\sigma_c^2(n-1)^2}{2G_c(m-1)} \delta + (1-n)\sigma_c$$

2) Integrate trilinear $\sigma(\delta)$:

$$G_{\text{fit}} = \int_0^{\delta_c} \sigma(\delta)\,d\delta$$

$$G_{\text{fit},2}(\delta) = \frac{n\sigma_c(\sigma_t - \sigma_c)}{4mG_c} \delta^2 + \sigma_c\delta + C_1$$
$$G_{\text{fit},3}(\delta) = \frac{\sigma_c^2(n-1)^2}{4G_c(m-1)} \delta^2 + (1-n)\sigma_c\delta + C_2$$

3) Fit expression for $G_{\text{fit}}(\delta)$ to test data: $G_R(\delta)$ using least squares

The fitting procedure determines: $\sigma_c, G_c, m,$ and $n$ which completely define the trilinear cohesive law

4) Compute cohesive law from fracture toughness & crack opening displacement

$$\sigma(\delta) = \frac{\partial G_{\text{fit}}}{\partial \delta}$$

Simple procedure to determine cohesive law for a through crack
Experimental Measurement of $G_R(\delta)$

**Compact Tension (CT) Specimen**

- $P, \delta_t$
- $0^\circ$
- $a_0$
- $\Delta a$
- $W$
- $\delta$
- Thin multidirectional laminate

- $W = 2.01$ in.
- $a_0/W = 2$

Measure $\delta$ between two green points using digital image correlation (DIC)

**Modified Compliance Calibration (MCC)**

$G_R = \frac{p^2}{2t} \frac{\partial C}{\partial a}$

Assume that $C(a)$ can be fit with:

$$C = \frac{\delta_i}{P} = (a_\alpha + \beta)^{-1/\chi}$$

Where $\alpha, \beta,$ and $\chi$ are fit parameters from a LEFM finite element (FE) model

Therefore:

$$G_R = G_R(P, \delta_t, t, \alpha, \beta, \chi)$$

From CT test

From linear FE model

$$G_R = \frac{p^2}{2t} \frac{\alpha((P/\delta_i)^\chi)^{-1+1/\chi}}{\chi}$$

*CT specimen with DIC can be used to measure $G_R(\delta)$*
Demonstration of LCA

Test specimens:
- AS4/VRM-34
- Warp-knit fabric
- $[\pm 45/90_2/0/90_2/\pm 45]_s$
- Thickness = 0.104 in.
- Two sizes:
  - Small: $W = 2.01$ in.
  - Large: $W = 4.02$ in.

FE model

Cohesive elements

LCA yields accurate predictions of through crack fracture propagation
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Test Objective: Assess damage containment capability by monitoring damage propagation ahead of the notch tips.

Full-scale integrally stitched composite fuselage panel
Pultruded Rod Stitched Efficient Unitized Structure

**Manufacturing Benefits:**
- React out-of-plane load without mechanical fasteners
- Single sided tooling
- VARTM process

*Promising technology for next generation airframes*
**Load Conditions**

**Full-scale Aircraft Structural Test Evaluation and Research (FASTER)**
(Bergan et al. *J Compos Struct*, 113, 2014.)

**FAA FASTER Fixture**

**Applied Loads**

- **Axial Tension**, $N_x$
- **Internal Pressure**, $p_i$
- **Hoop load on frame**, $N_\theta^F$
- **Hoop load on skin**, $N_\theta^S$

**Limit Loads:**
- $N_x^L = 4670$ lbf/in
- $p_i^L = 9.2$ psi

**Selected Load History**

*Flight loads simulated using FASTER fixture*
Post Test Damage Observations

Exterior

- Stitch rows

- Notch

- Damage path changes direction

- S-1, S-2, S-3, S-4, S-5, S-6, S-7

- F-2, F-3

Interior

- Notch

- Widespread damage
- Stiffeners disbonded

Damage path altered at stitch rows

Complex and extensive damage observed
Idealize damage at the structural scale:
- **Through crack** in skin
- **Delamination** between skin and stiffener

This idealization considers the interaction between damage in skin and delamination of stiffener interfaces.
Finite Element Modeling

Global Model

Local Model

~323,000 elements
~1.9M DoF
Refined mesh size: 0.025 in.

Green: Skin and frame flanges tied together

Stitch

Blue: Cohesive elements between skin and stringer

Notch

Ply drops

Through-the-thickness crack paths (skin)

Side View

Skin

Stitch

Cohesive elements

15 in.

21.8 in.

17
Stitched Skin/Stringer Interface Model

**Idealization**

Undeformed stitch

Deformed stitch

Delamination crack tip

**FE Representation: Superposed cohesive elements**


Input Parameters:
- **Delamination:**
  - Fracture toughness determined from ASTM standard tests
  - Mixed mode energy governed by Benzeggagh-Kenane (BK) criterion
- **Stitch behavior:**

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Strain Results: Indication of Damage Propagation

Local Model

45° Strain

Axial Strain

Consistent trend between test and analysis
Propagation of Skin/Stringer Delamination

Model predicts the delamination behavior inline with test observations
Crack Propagation

Test Measurements
- Exterior, A
- Exterior, B
- Interior, A
- Interior, B

Through crack half length, $a$ [in]

Good agreement between tests and analysis
Doubling the number of stitches increases damage containment load by 11%
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Concluding Remarks

- Introduced a new methodology to analyze damage propagation in a notched, stiffened composite fuselage structure

- Cohesive elements are used to represent:
  - Damage in the skin as it propagates from a notch
  - Delamination of skin/stiffener interface

- Good correlation between test and analysis observed for:
  - Damage initiation
  - Damage propagation
  - Strain redistribution

- Increasing the skin/stringer interface toughness can significantly improve the damage containment load
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Post-test NDI conducted at Sandia National Labs, NM
Questions?
Backup
Post Test Damage Observations

Damage along notch axis

1.60N^L_x

-45°, Exterior (and fiberglass)

-45°/+45°, Fiber orientation

-45°, Interior (and fiberglass)

-45°, Interior

Skin/stringer delaminated

Delamination & matrix cracks within: 90°

90°

Delamination surface Interface: -45°/+45°

-45°, Exterior (and fiberglass)

+45°

90°

0°

Damage turned

Fiber orientation

Inner stitch row

Center stitch row

Flange edge

Outer edge

Stitch row

Delamination & matrix cracks within: 90°

Damage in skin exhibited similar path through the thickness