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

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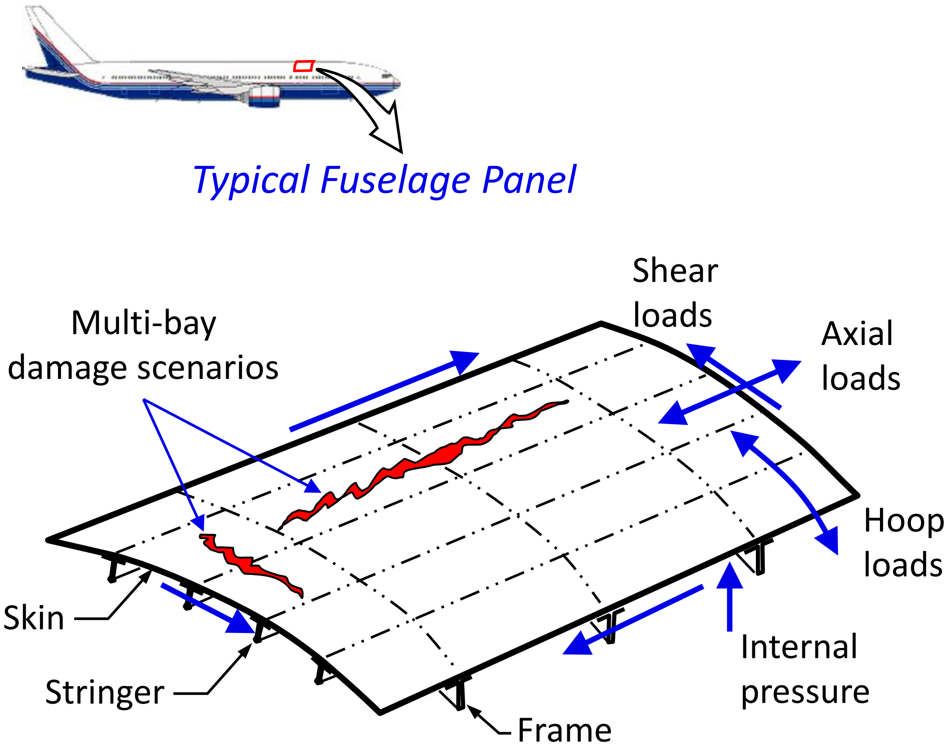
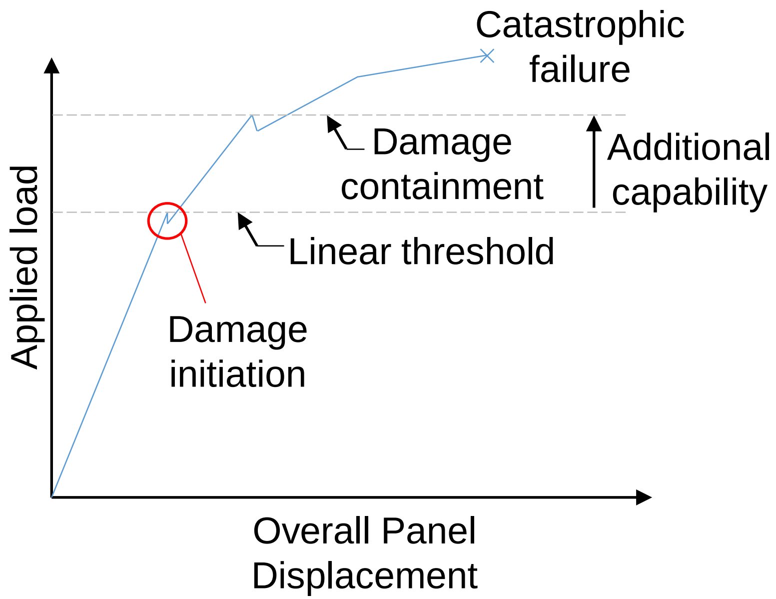
Outline

- Background
- Laminate cohesive approach (LCA)
- Full-scale fuselage panel test
- Test and analysis results
- Concluding remarks

Motivation: Predict Damage Containment Behavior



Residual strength of fuselage panel



Damage containment is achieved through:

1. Multiple load paths (e.g. Skin and substructure)
2. Damage arresting features (e.g. Rivets)

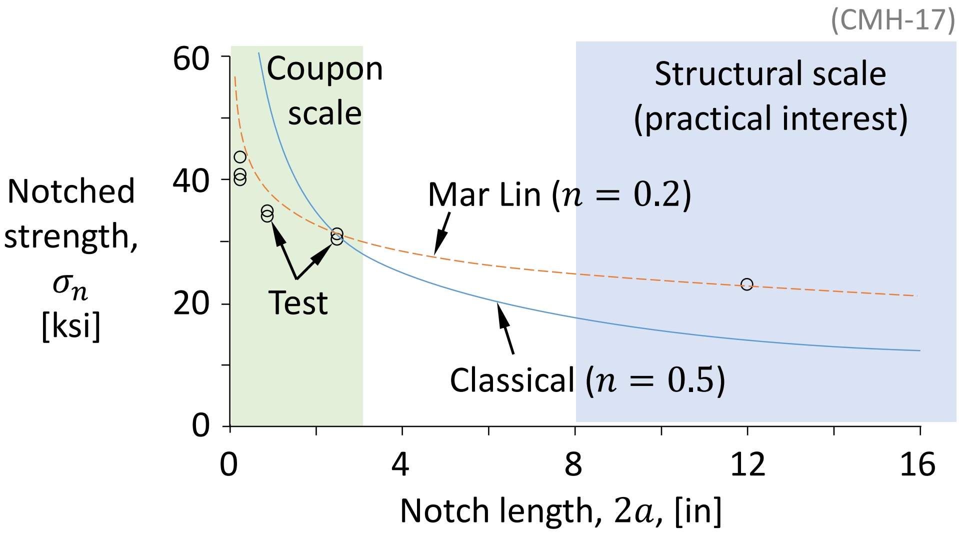
Current state-of-the-art:

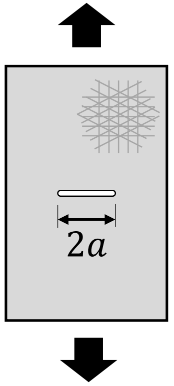
- Metallic structures: Damage containment
- Composite structures: linear threshold

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

Simple Case: Center Notch Test Specimen

Notched Strength Prediction





Laminate assumed:

- Homogeneous
- Orthotropic (multidirectional)

$$\sigma_n = \frac{K_{Ic}}{(\pi a)^n}$$

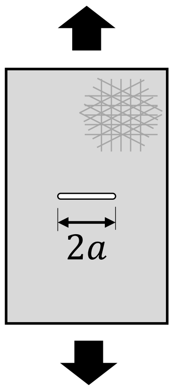
Comments:

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

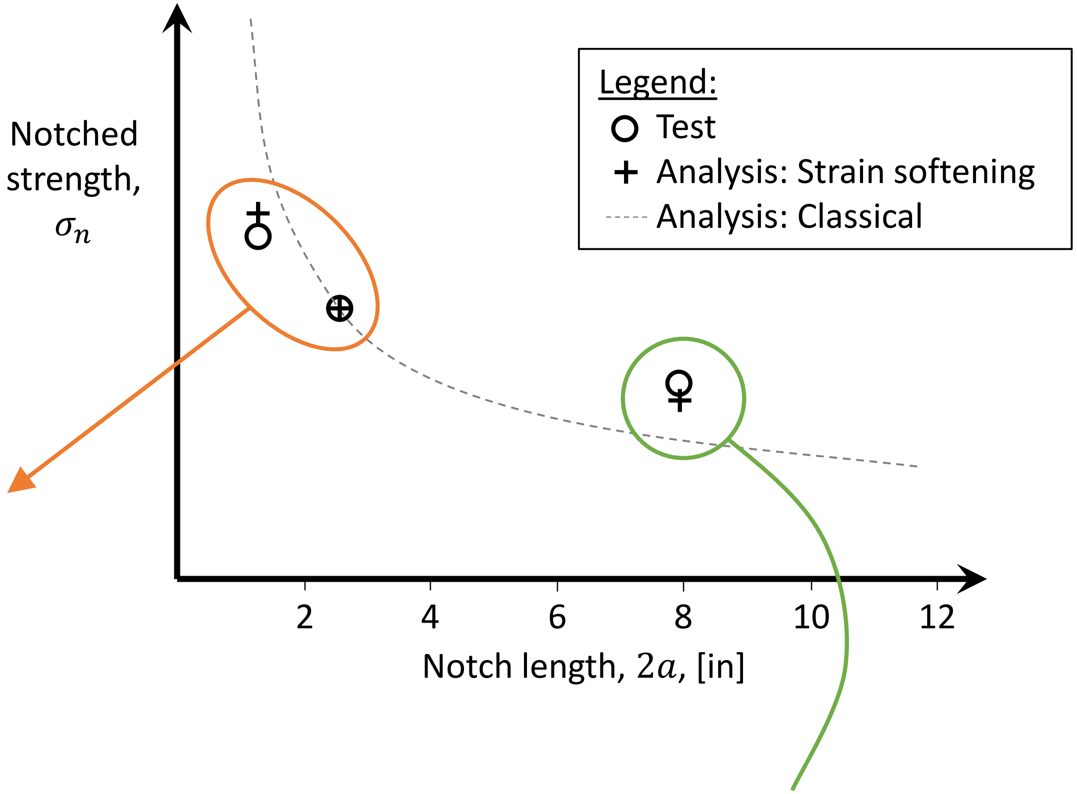
Analysis methods that can predict notched strength accurately reducing the number of large-scale tests will save time and cost

Strain Softening Approach

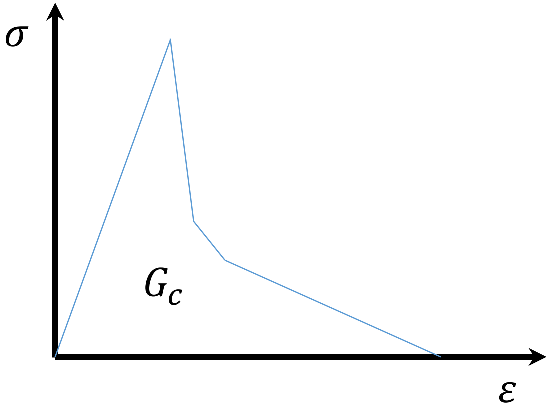
(Dopker et al. SDM Conference, 1994)



IM7/8551-7
$[\pm 45/0/90/\pm 30]_s$



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.

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

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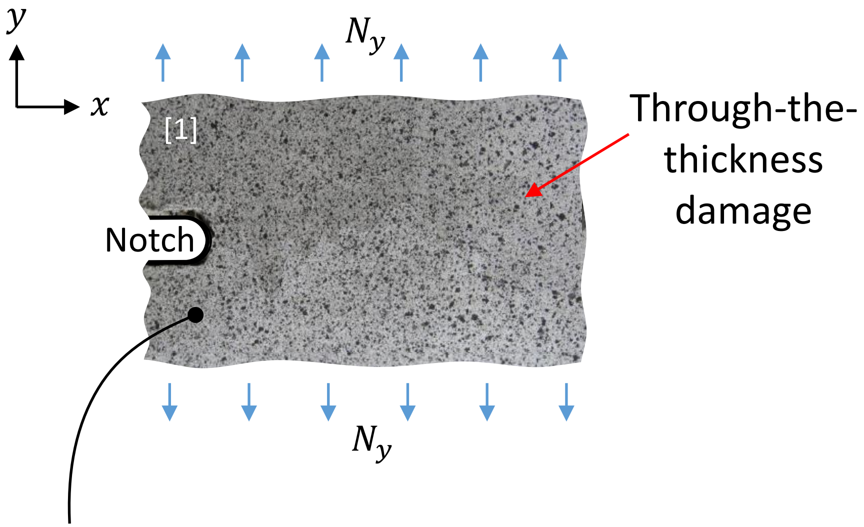
Actual Versus Idealization of (LCA)

Actual

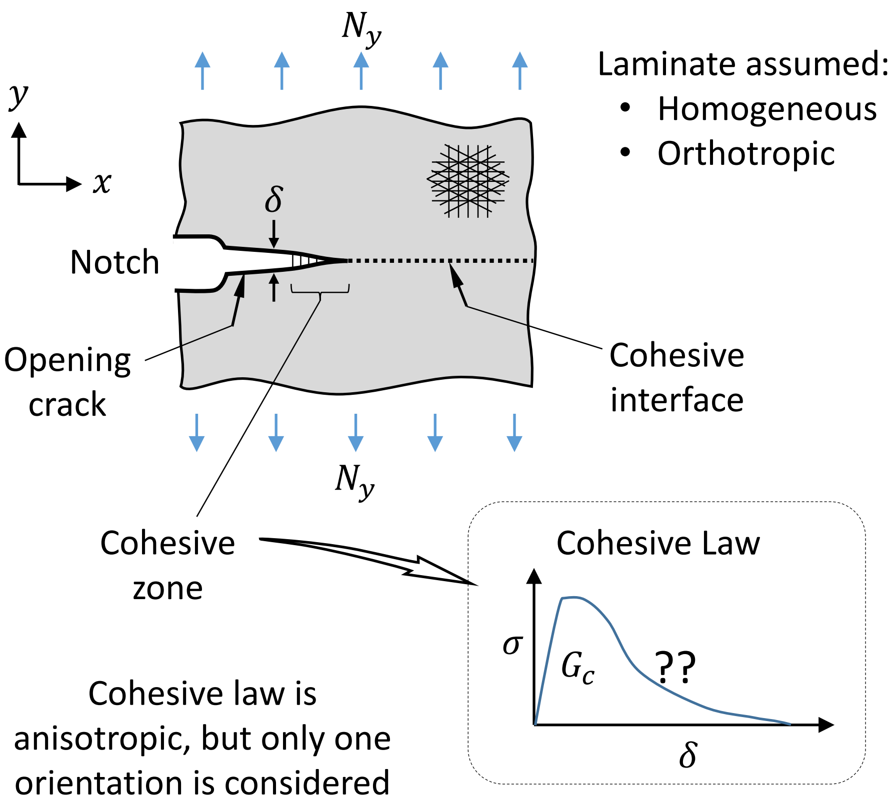


Thin fiber reinforced polymer (FRP) laminate
- Multidirectional layup
- Thickness: t

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

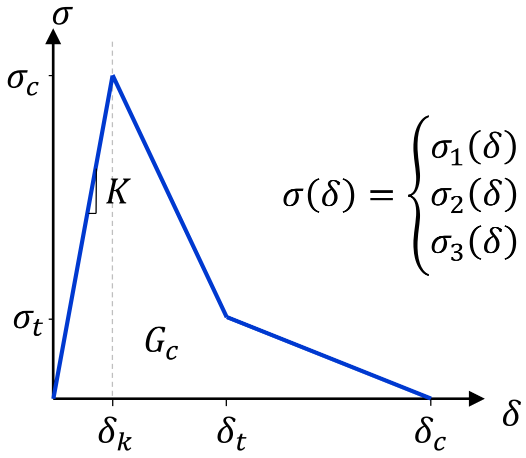
Cohesive law is anisotropic, but only one orientation is considered

<u>Objective</u>: *Characterize the cohesive law for a laminate and crack orientation*

References:
1. Leone. PhD Thesis 2010
2. Rose et al. NASA/TM–2013-218024, 2013.

Characterization of LCA

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



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

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

$$\sigma_1(\delta) = K\delta$$

$$\sigma_t = \frac{\sigma_c(n-1)(n-m)}{n(m-1)}$$

$$\sigma_2(\delta) = \frac{n\sigma_c(\sigma_t - \sigma_c)}{2mG_c}\delta + \sigma_c$$

$$\sigma_3(\delta) = \frac{\sigma_c^2(n-1)^2}{2G_c(m-1)}\delta + (1-n)\sigma_c$$

2) Integrate trilinear $\boldsymbol{\sigma(\delta)}$: $G_{\text{fit}} = \int_0^{\delta_c} \sigma(\delta)d\delta$

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

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

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

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

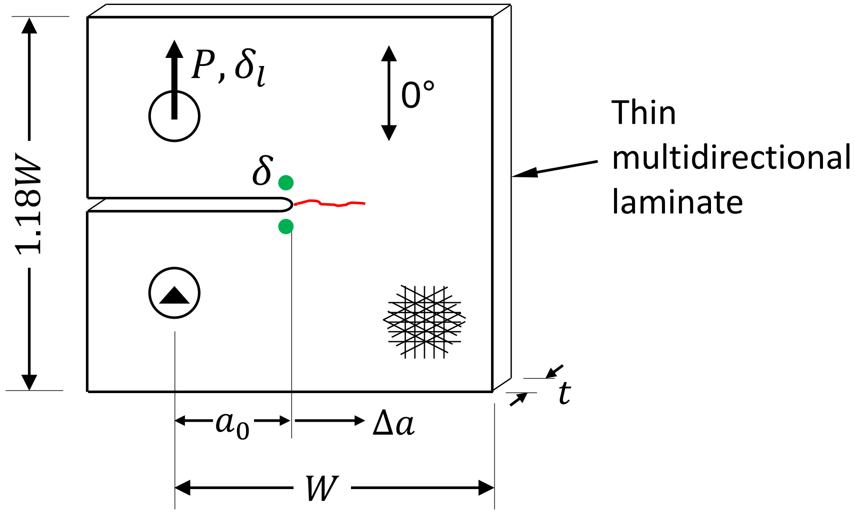
4) Compute cohesive law from fracture toughness & crack opening displacement

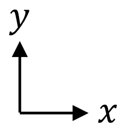
$$\sigma(\delta) = \frac{\partial G_{\text{fit}}}{\partial\delta}$$

Simple procedure to determine cohesive law for a through crack

Experimental Measurement of $G_R(\delta)$

Compact Tension (CT) Specimen





W = 2.01 in.
a_0/W = 2

Measure δ 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:

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

Where α, β, and χ are fit parameters from a LEFM finite element (FE) model

Therefore:

$$G_R = G_R(\underbrace{P, \delta_l, t}_{\text{From CT test}}, \underbrace{\alpha, \beta, \chi}_{\text{From linear FE model}})$$

$$G_R = \frac{P^2}{2t}\frac{\alpha\left((P/\delta_l)^{\chi}\right)^{-(1+\frac{1}{\chi})}}{\chi}$$

CT specimen with DIC can be used to measure $G_R(\delta)$

Demonstration of LCA

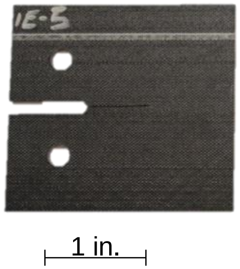
Test specimens:

- AS4/VRM-34
- Warp-knit fabric
- $[\pm 45/90_2/0/90_2/\pm 45]_s$
- Thickness = 0.104 in.
- Two sizes:

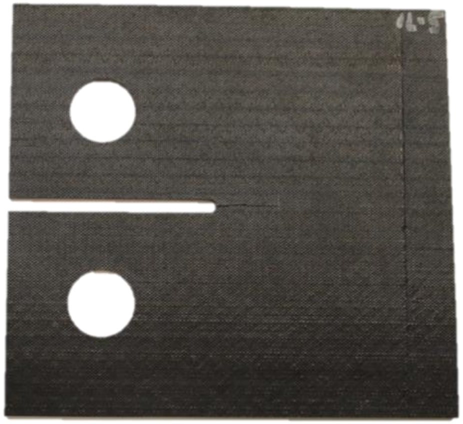
  Small: $W = 2.01$ in.

  Large: $W = 4.02$ in.

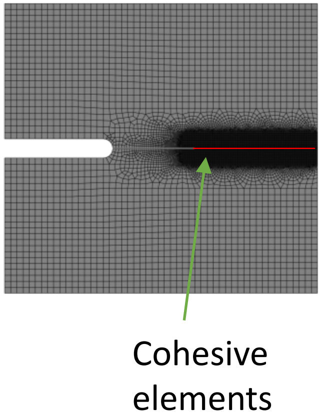
Small CT

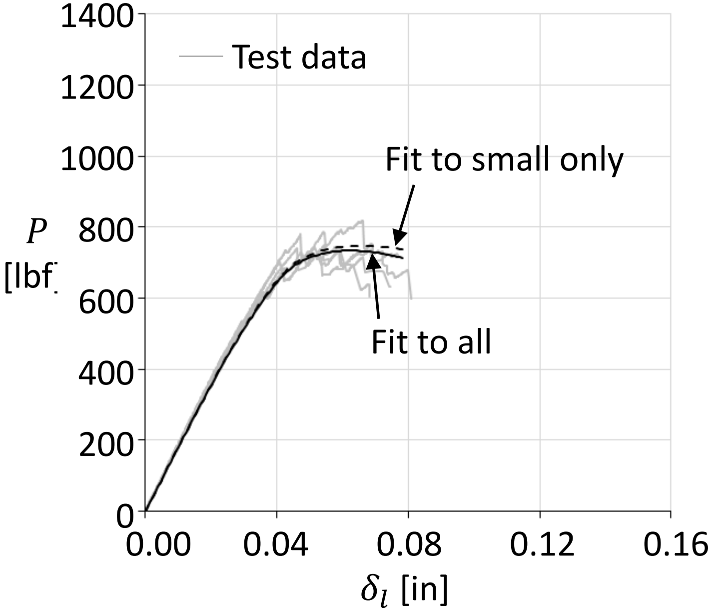


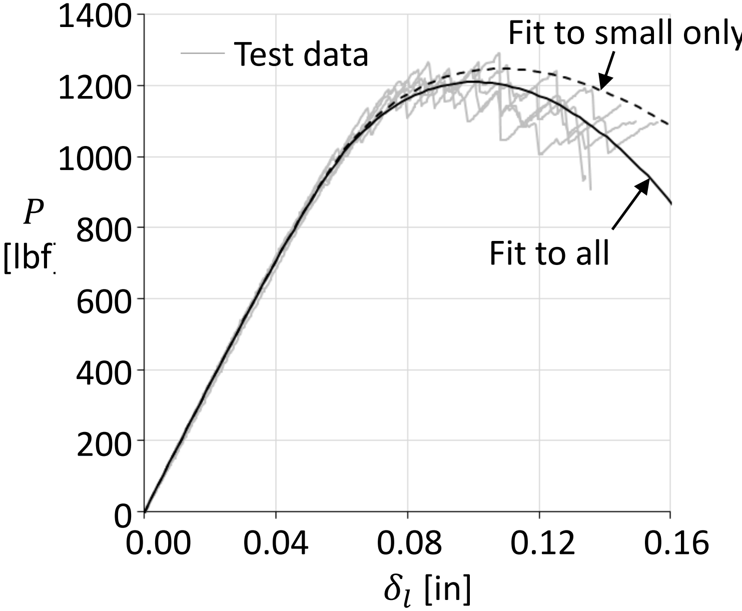
Large CT



FE model







LCA yields accurate predictions of through crack fracture propagation

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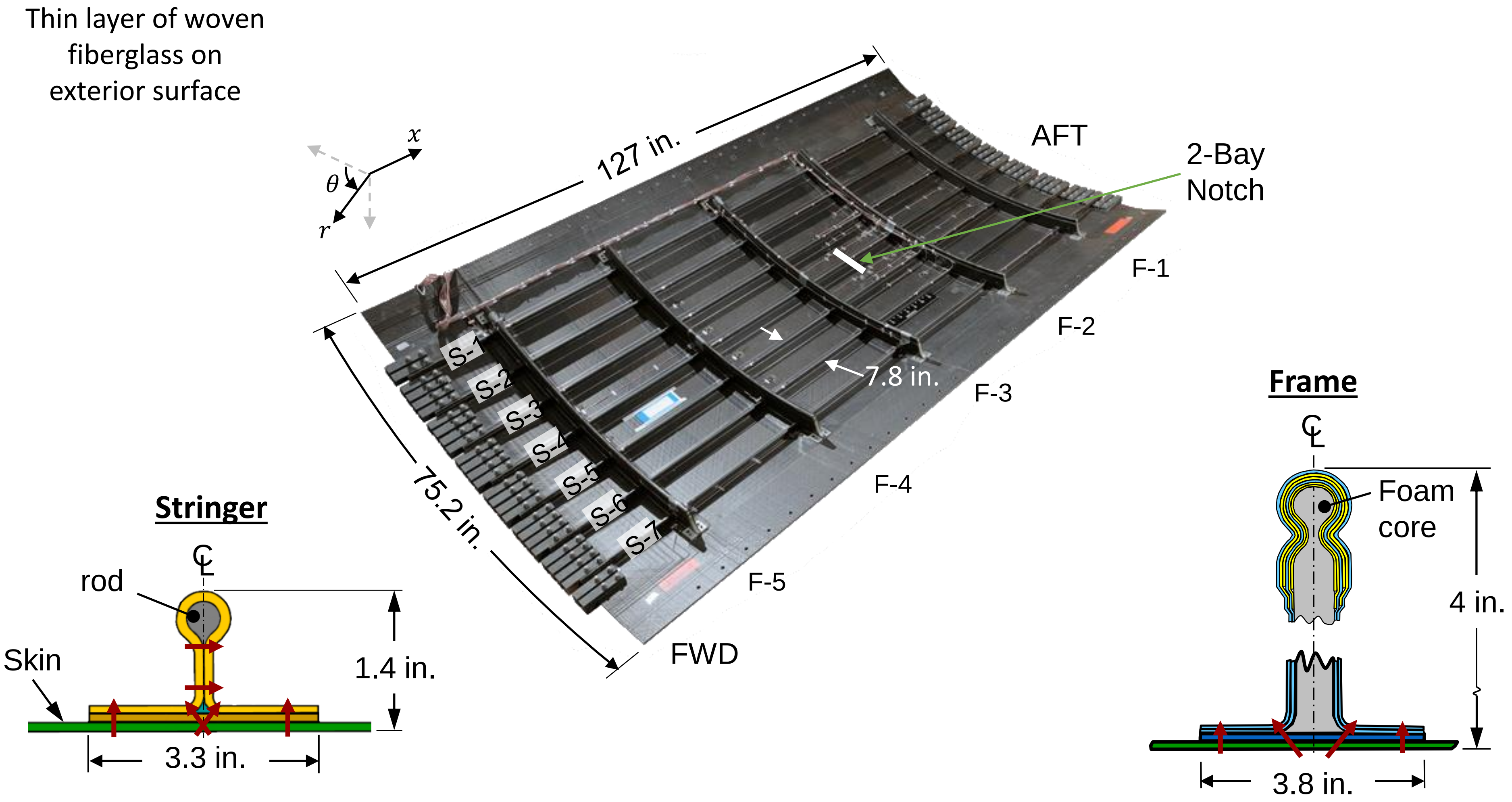
PRSEUS Fuselage Panel

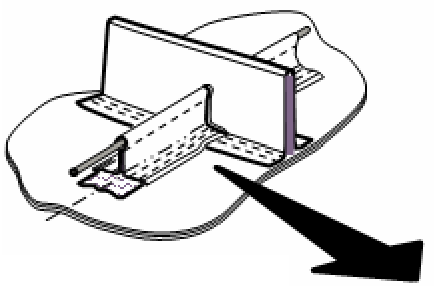


Thin layer of woven fiberglass on exterior surface

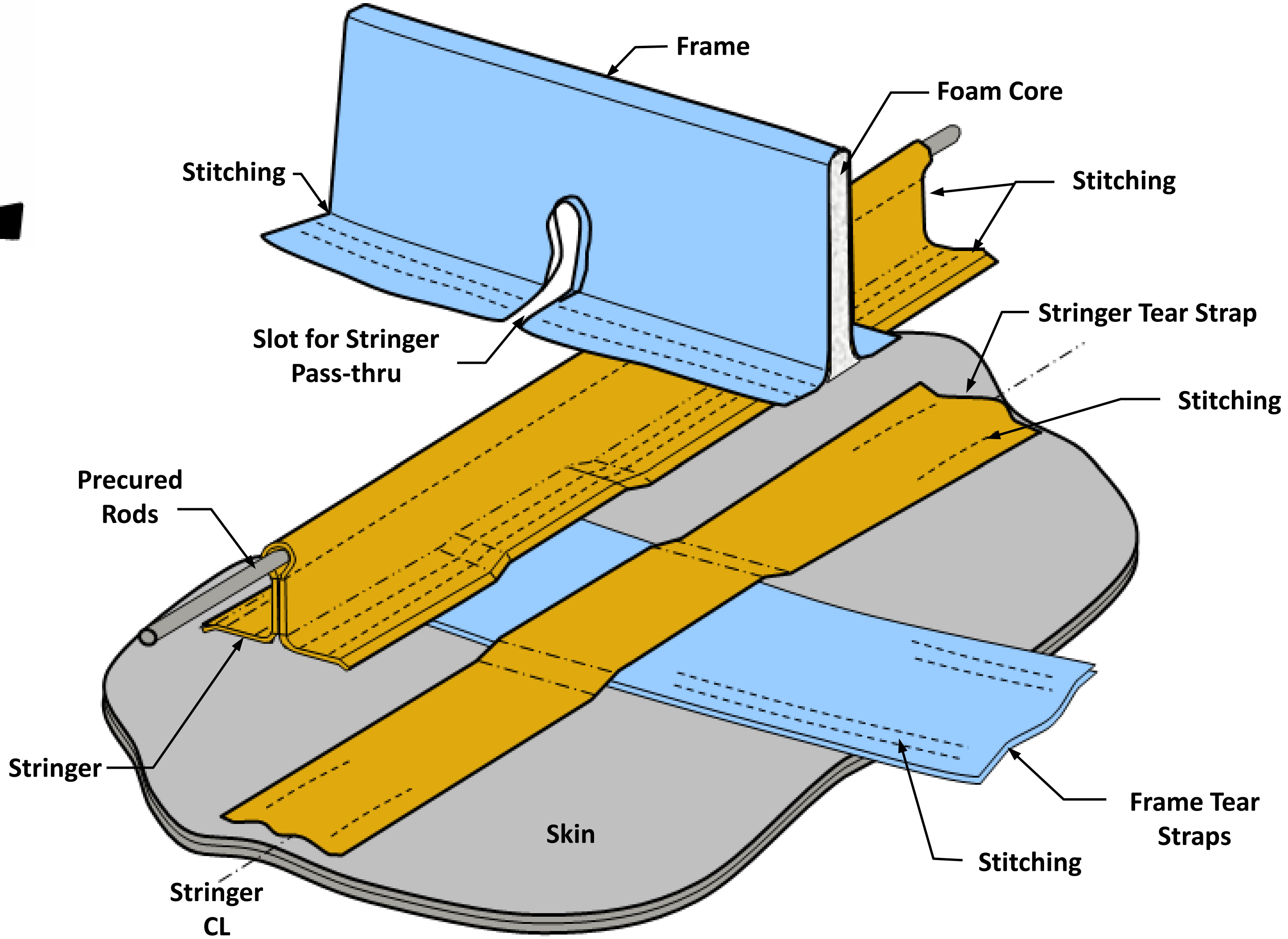
<u>Test Objective:</u> Assess damage containment capability by monitoring damage propagation ahead of the notch tips

Full-scale integrally stitched composite fuselage panel

Pultruded Rod Stitched Efficient Unitized Structure



PRSEUS
Integrated
Structure



Exploded View of Preform Assembly

Manufacturing Benefits:

- React out-of-plane load without mechanical fasteners
- Single sided tooling
- VARTM process

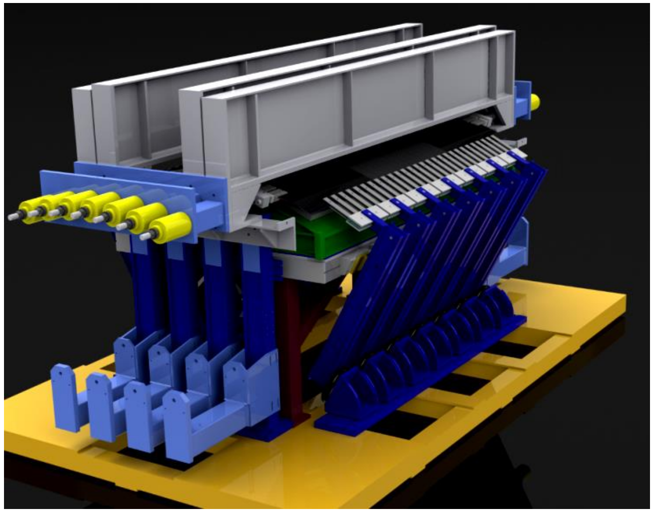
Promising technology for next generation airframes

Load Conditions

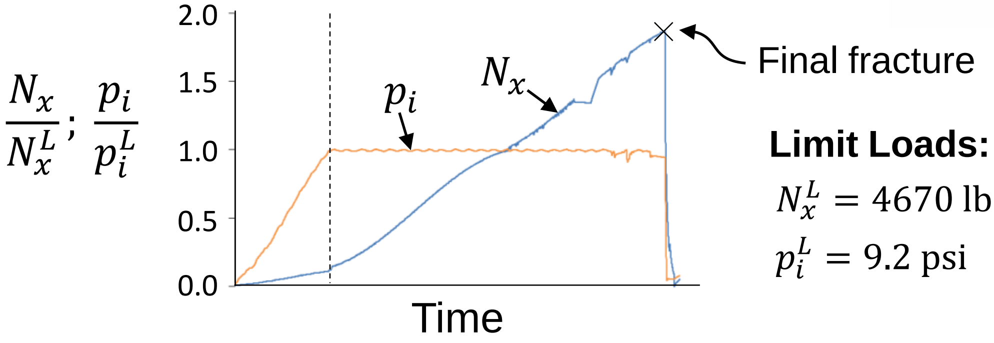
Full-scale Aircraft Structural Test Evaluation and Research (FASTER)

(Bergan et al. *J Compos Struct*, 113, 2014.)

FAA FASTER Fixture



Selected Load History

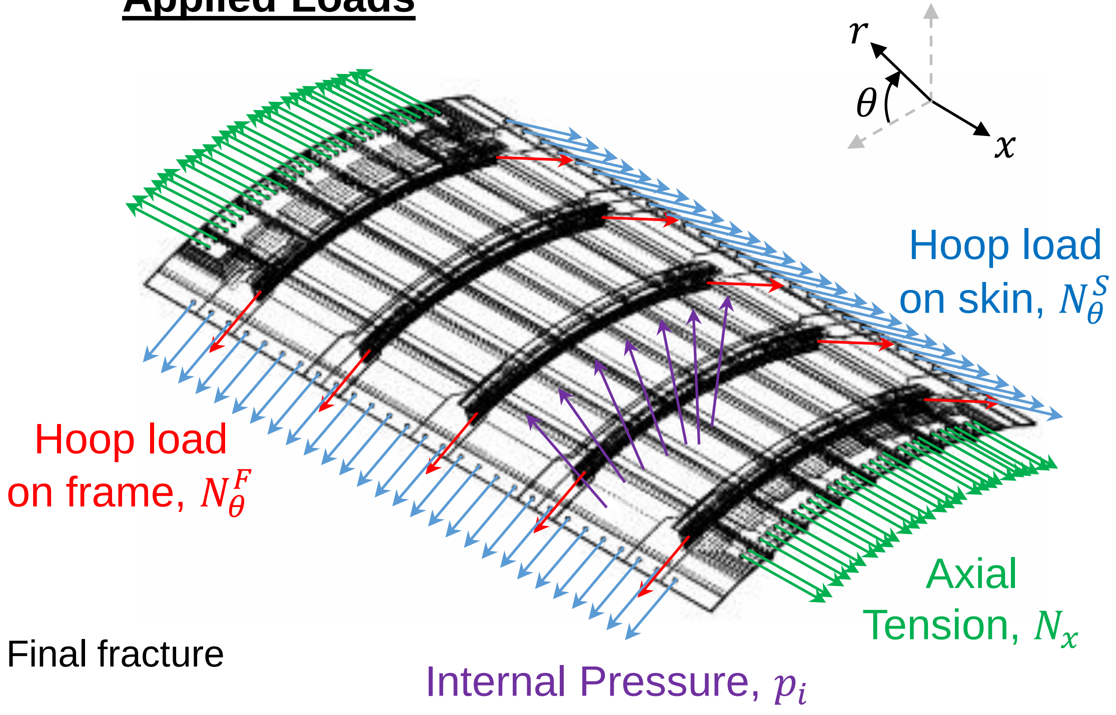


Limit Loads:

$N_x^L = 4670\ \mathrm{lbf/in}$

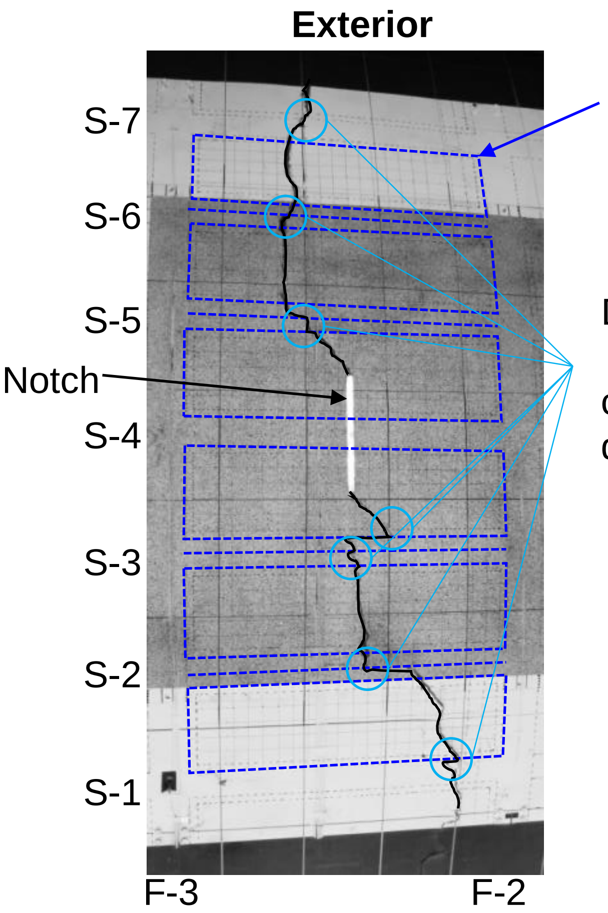
$p_i^L = 9.2\ \mathrm{psi}$

Applied Loads



Flight loads simulated using FASTER fixture

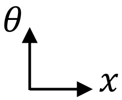
Post Test Damage Observations

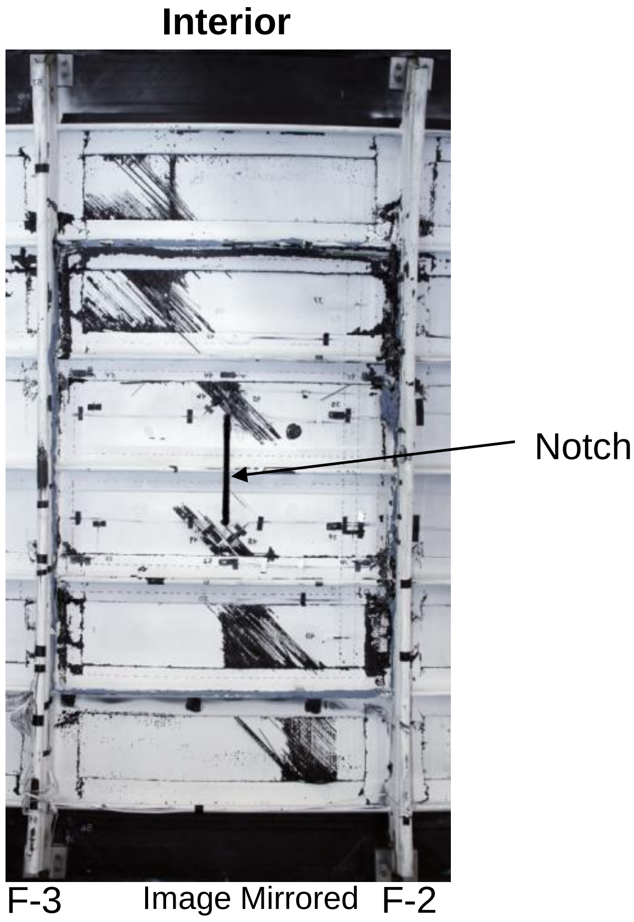


Stitch rows

Damage path changes direction

Damage path altered at stitch rows





Widespread damage Stiffeners disbonded

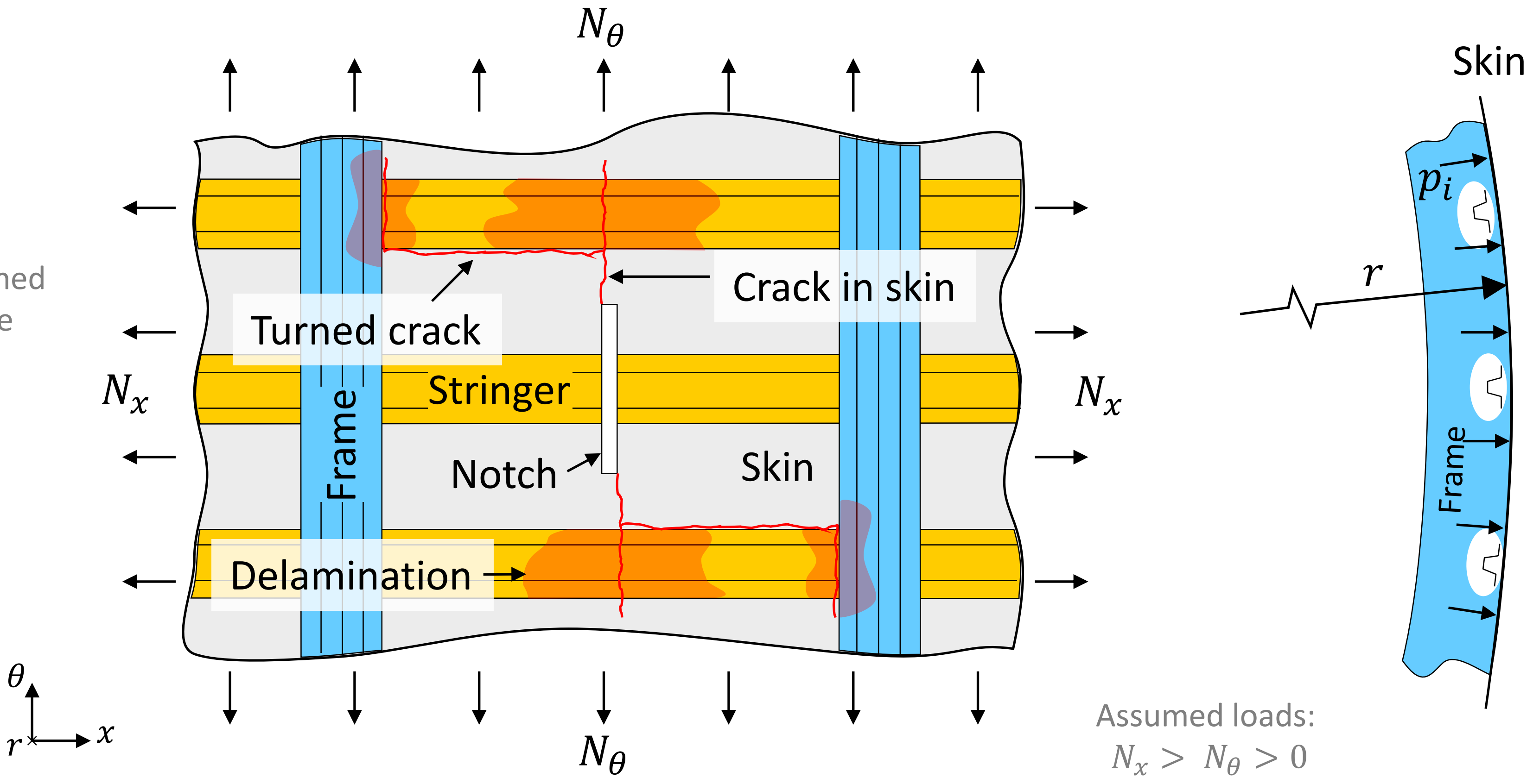
Complex and extensive damage observed

Methodology: Residual Strength Prediction of Stiffened Panels

Idealize damage at the structural scale:

- **Through crack** in skin
- **Delamination** between skin and stiffener

General stiffened FRP fuselage structure



Assumed loads:
$N_x > N_\theta > 0$

This idealization considers the interaction between damage in skin and delamination of stiffener interfaces

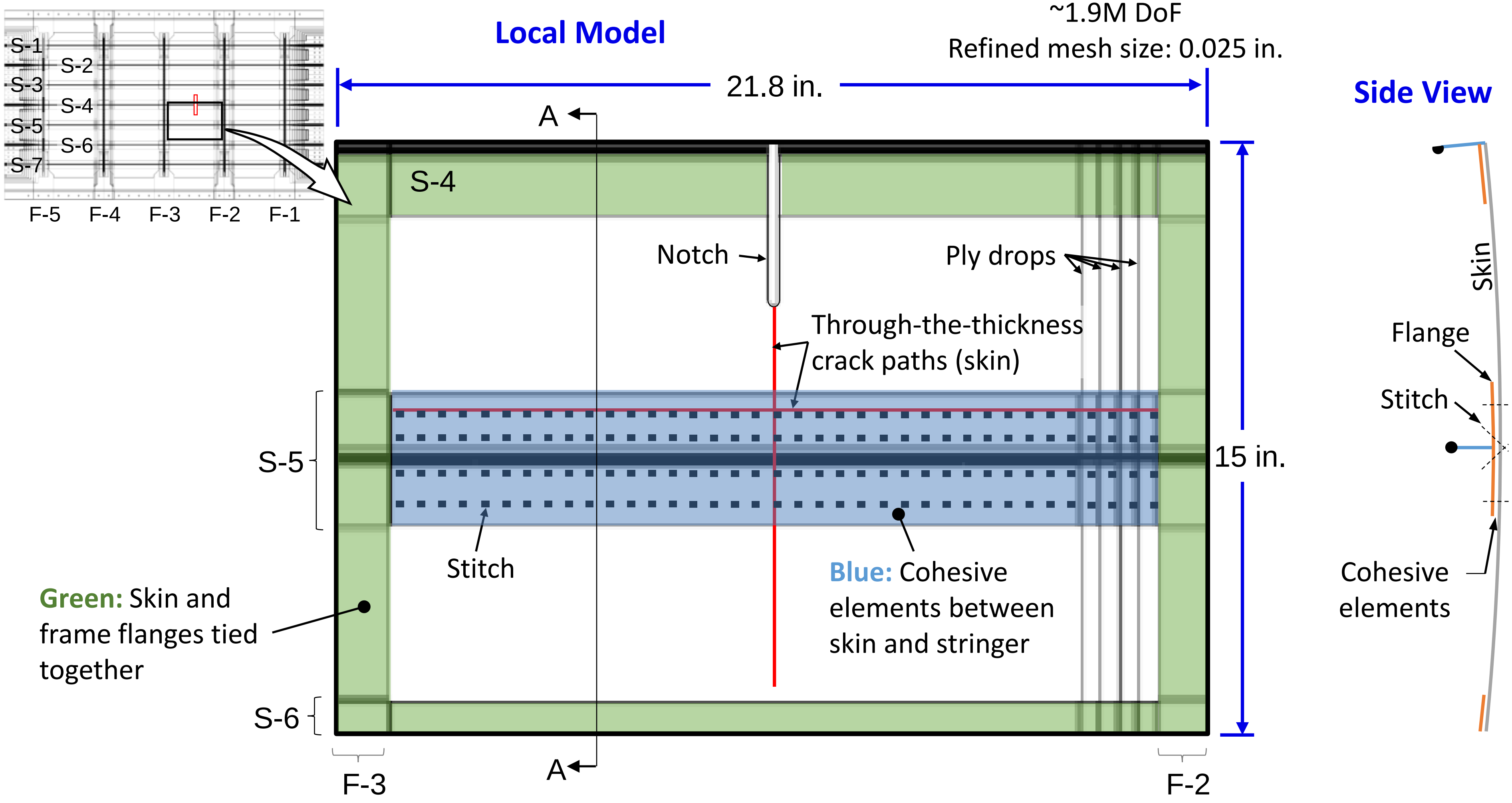
Finite Element Modeling

Global Model

Local Model

Side View

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

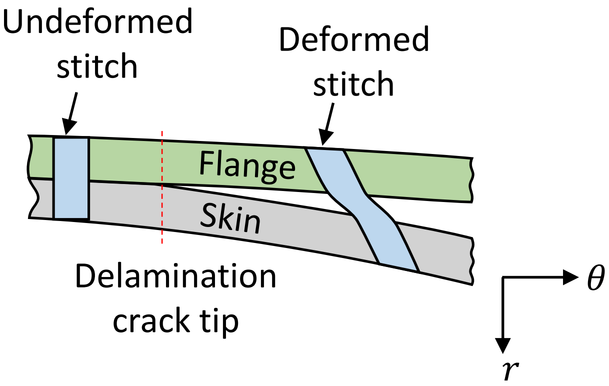


Green: Skin and frame flanges tied together

Blue: Cohesive elements between skin and stringer

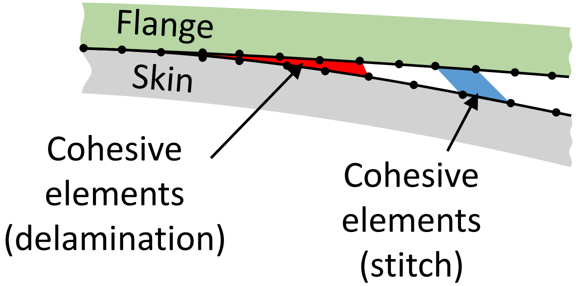
Stitched Skin/Stringer Interface Model

Idealization



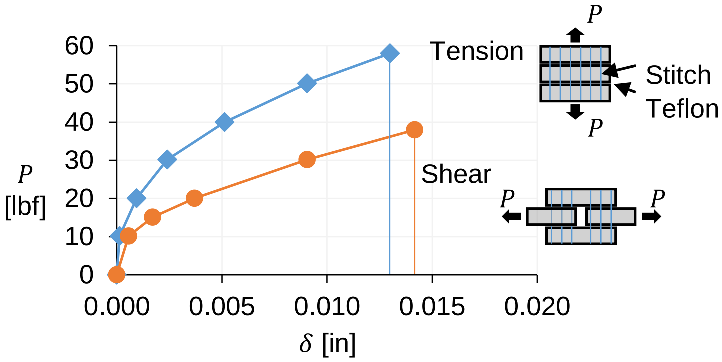
FE Representation: Superposed cohesive elements

(Bianchi and Zhang. *Compos Sci Technol*, 71(16), 2011;
Bianchi and Zhang. *Compos Sci Technol*, 72(8), 2012.)



<u>Input Parameters:</u>

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



(Glaessgen et al. *J Compos Mater*, 36(23), 2002.
Adams. *J Reinf Plast Compos*, 19(14), 2000)

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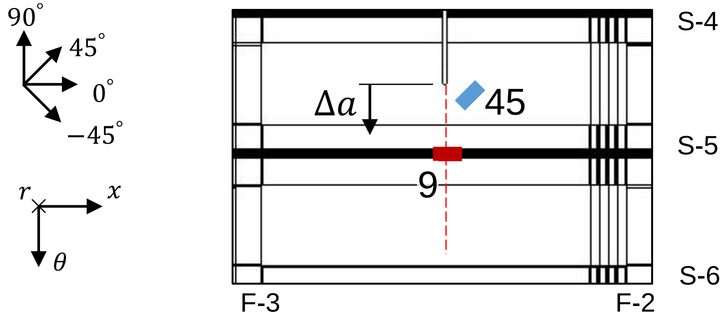
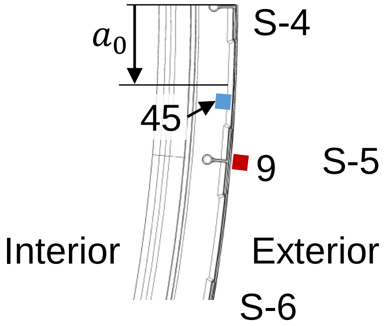
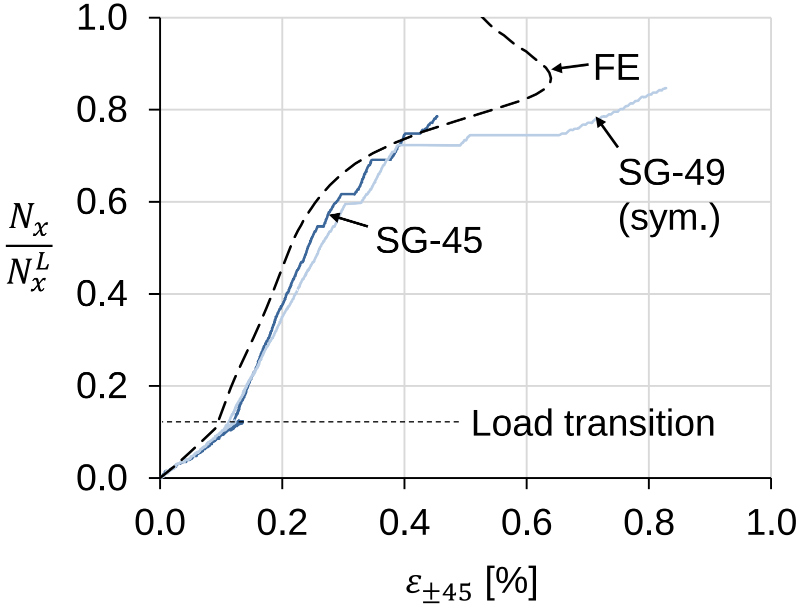
Strain Results: Indication of Damage Propagation

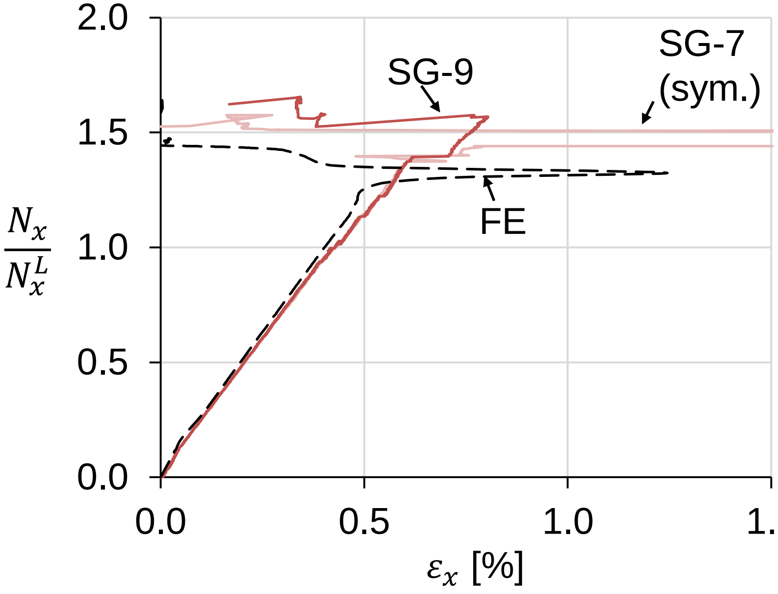


Local Model





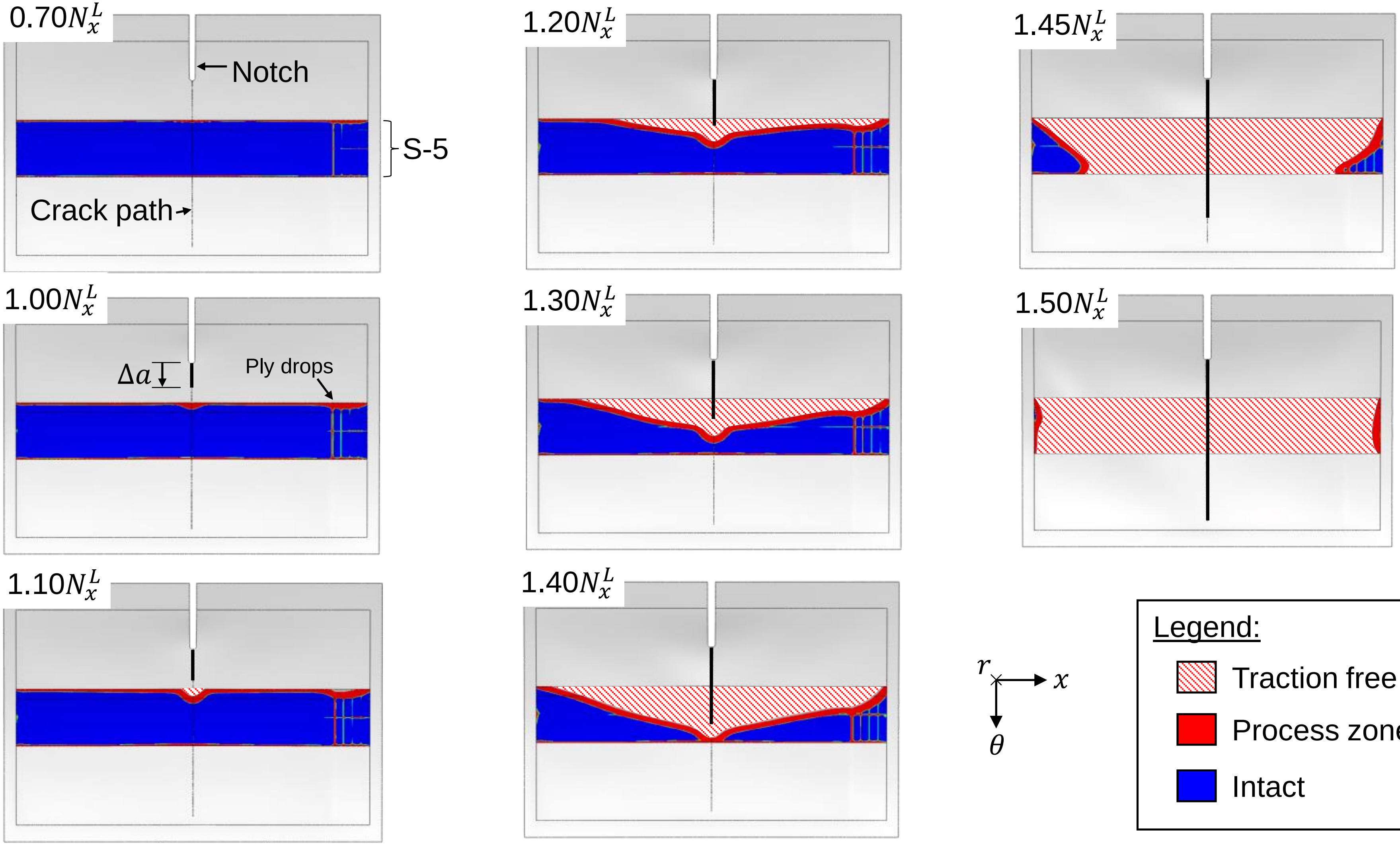
45° Strain



Axial Strain

Consistent trend between test and analysis

Propagation of Skin/Stringer Delamination

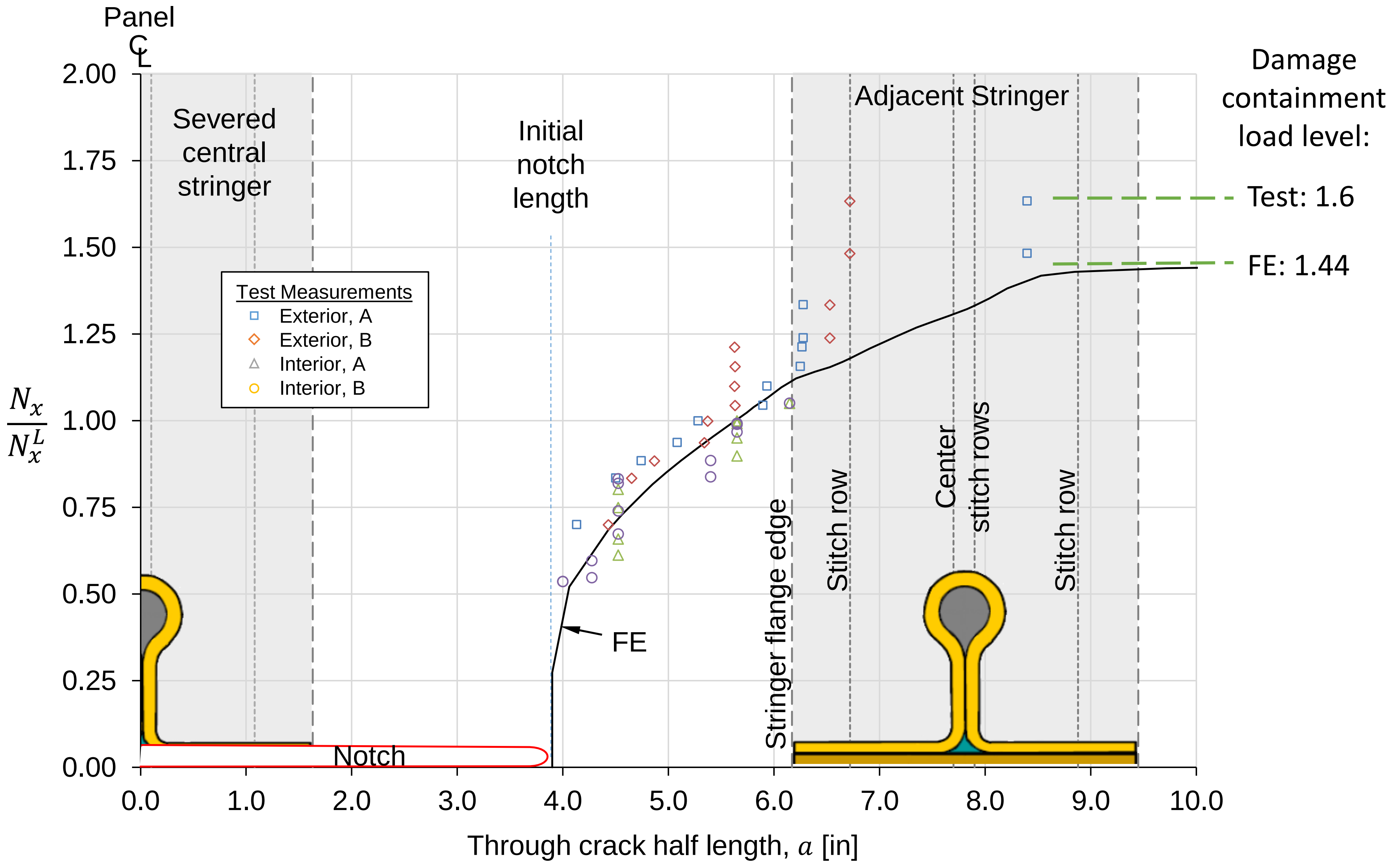


Legend:

- ▨ Traction free
- 🟥 Process zone
- 🟦 Intact

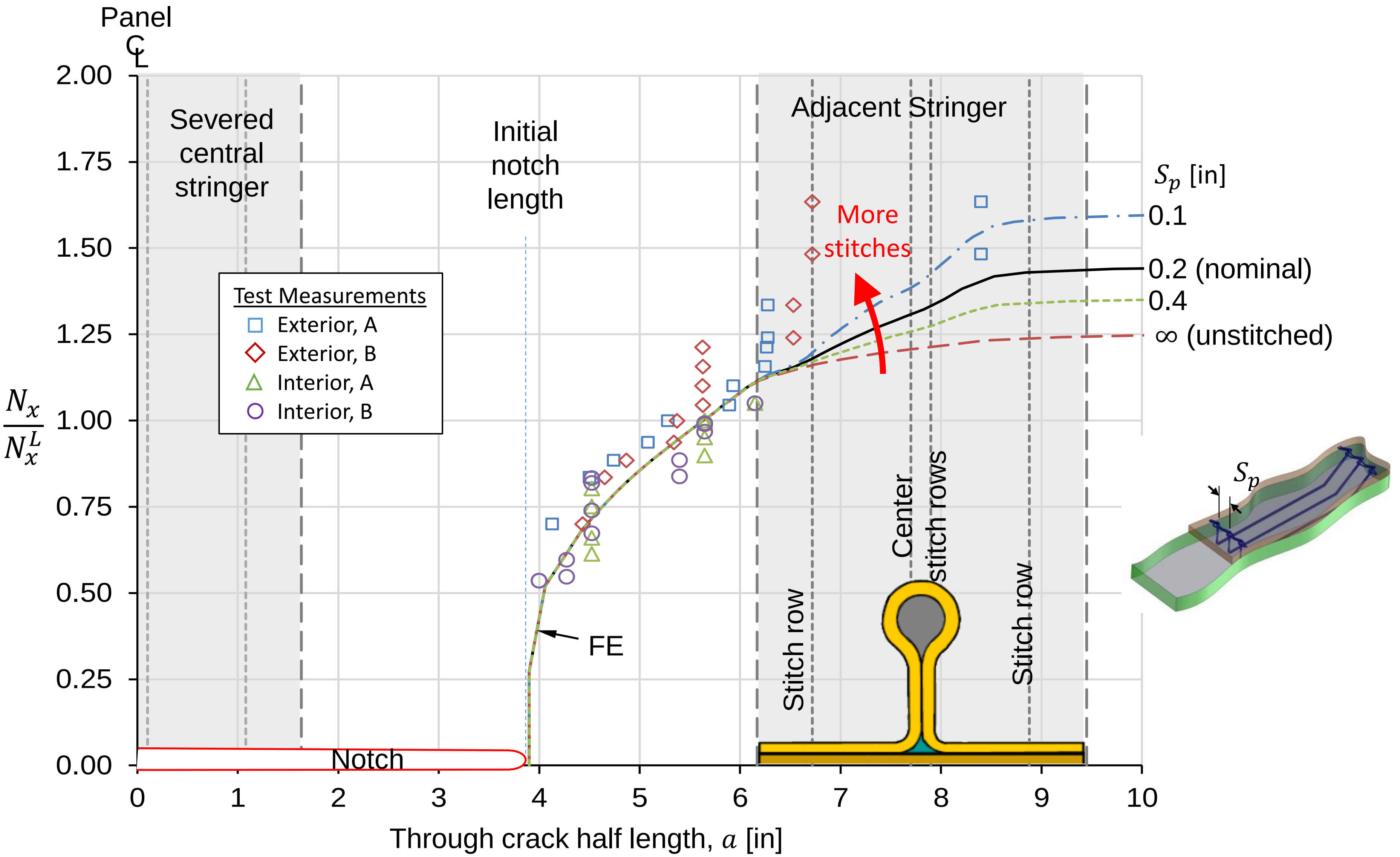
Model predicts the delamination behavior inline with test observations

Crack Propagation



Good agreement between tests and analysis

Effect of Stitching Pitch



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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- Pat Sheehan
- Curt Davies

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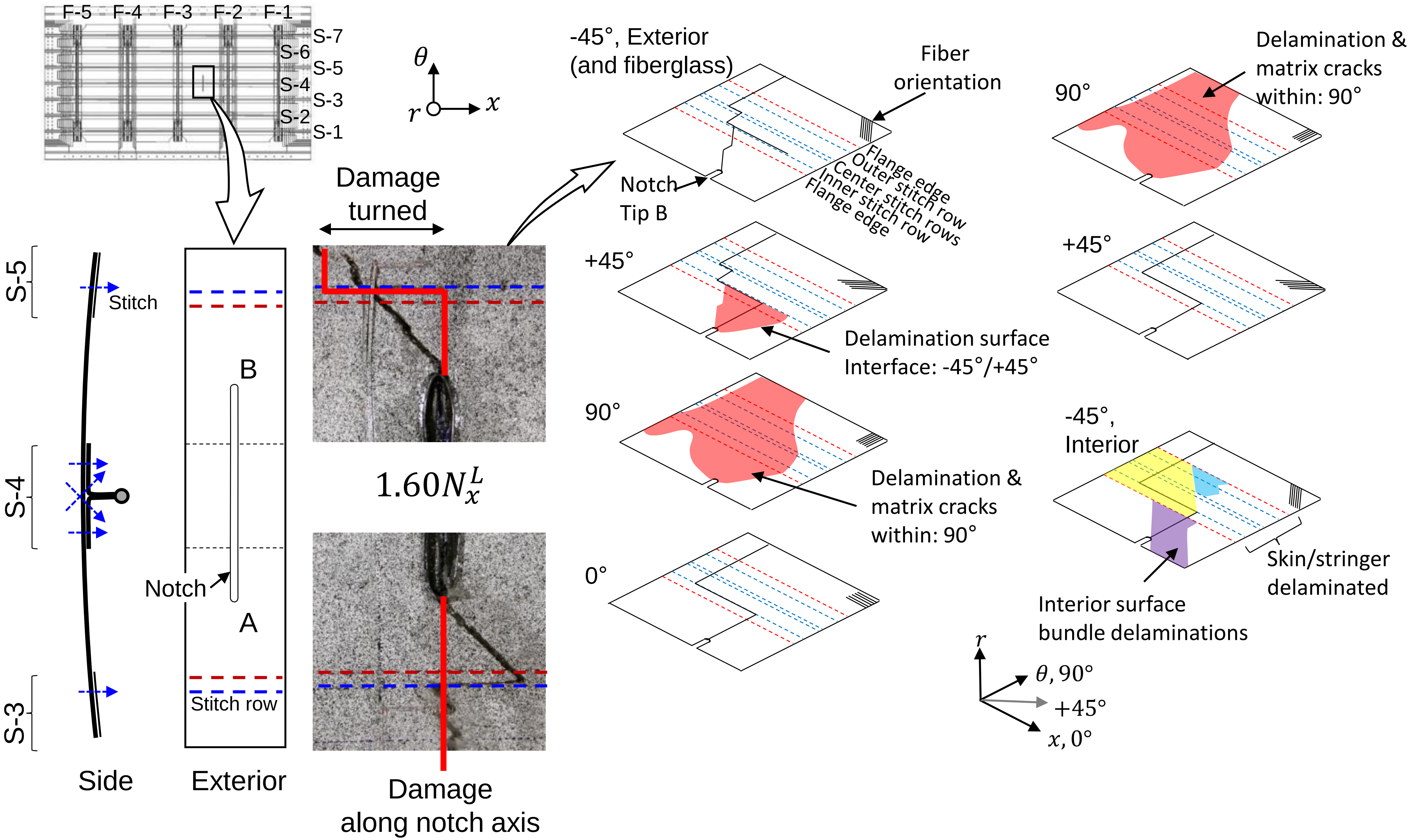
- Amey Khonalkar
- Reewanshu Chadha

Post-test NDI
conducted at
Sandia National
Labs, NM

Questions?

Backup

Post Test Damage Observations



Damage in skin exhibited similar path through the thickness