



Effect of Composite Stiffened Panel Design on Skin-Stringer Separation in Postbuckling

11–15 & 19–21 January 2021
Virtual Event

Lucas J. Kootte and Chiara Bisagni

Delft University of Technology, Faculty of Aerospace Engineering, Netherlands

Vipul Ranatunga and Stephen B. Clay

Air Force Research Laboratory, Wright-Patterson AFB (OH), USA

Carlos G. Dávila and Cheryl Rose

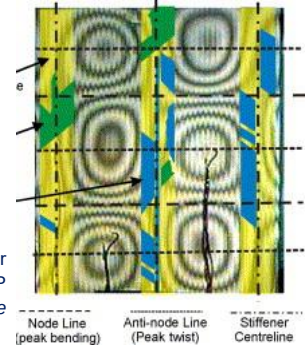
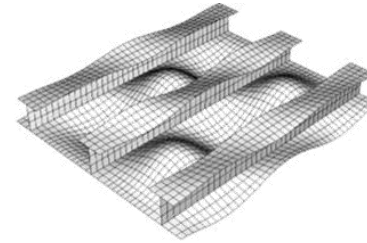
NASA Langley Research Center, Hampton (VA), USA

The authors gratefully acknowledge the financial support received from the European Office of Aerospace Research and Development (EOARD), United States Air Force (USAF), under the guidance of **David Garner**.

Introduction

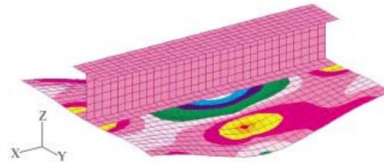
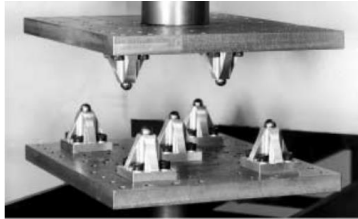
- Allowing postbuckling could lead to skin-stringer separation in composite stiffened panels.
- Critical locations are often at the buckling extrema.
- Difficult to identify initiation of skin-stringer separation.
- Expensive to model or test large multi-stringer panels.

Multi-stringer panel



Meeks, C., Greenhalgh, E. and Falzon, B.G., "Stiffener Debonding Mechanisms in Post-buckled CFRP Aerospace Panels", *Composites Part A: Applied Science and Manufacturing*, 36(7): 934-946, 2005.

Seven-point bending



Single-stringer specimen

Four-point twisting



Van Rijn, J.C.F.N., and Wiggendaad, J.F.M., "A Seven-Point Bending Test to Determine the Strength of the Skin-Stiffener Interface in Composite Aircraft Panels," *NLR Technical Report 2000-044*, 2000.

Kootte, L.J., and Bisagni, C., "A Methodology to Investigate Skin-Stringer Separation in Postbuckled Composite Stiffened Panels," *Proceedings of the AIAA Scitech 2020 Conference*, 2020.

Methodology

Objective: *Investigate skin-stringer separation in a stiffened composite panel due to postbuckling conditions using representative single-stringer specimens*

1. Simple panel model:

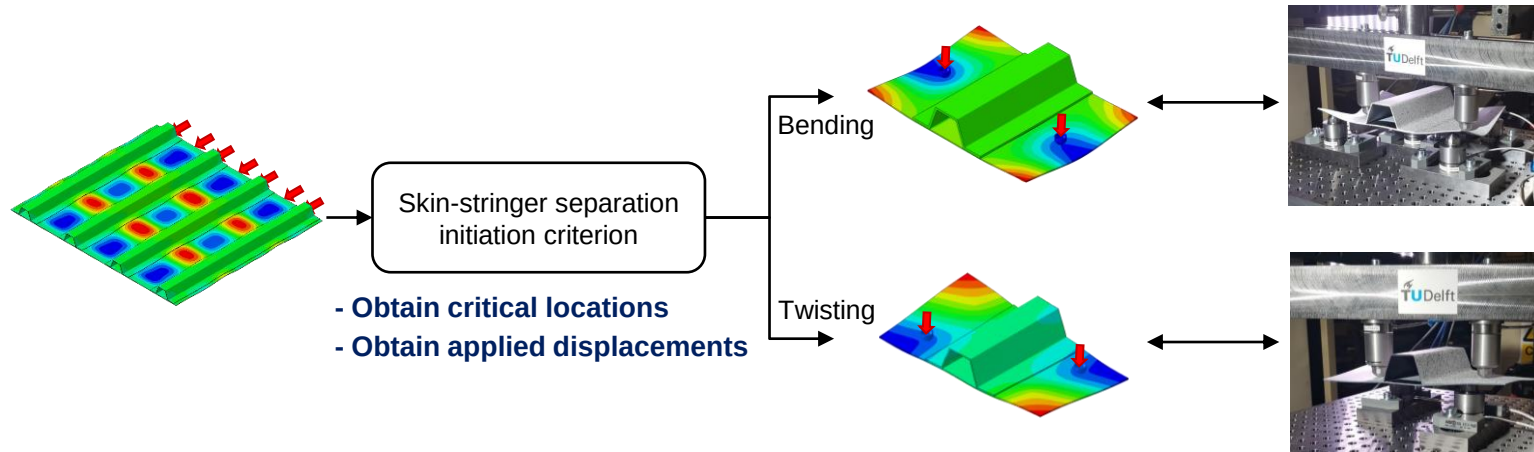
- Obtain postbuckling shape

2. Specimen numerical:

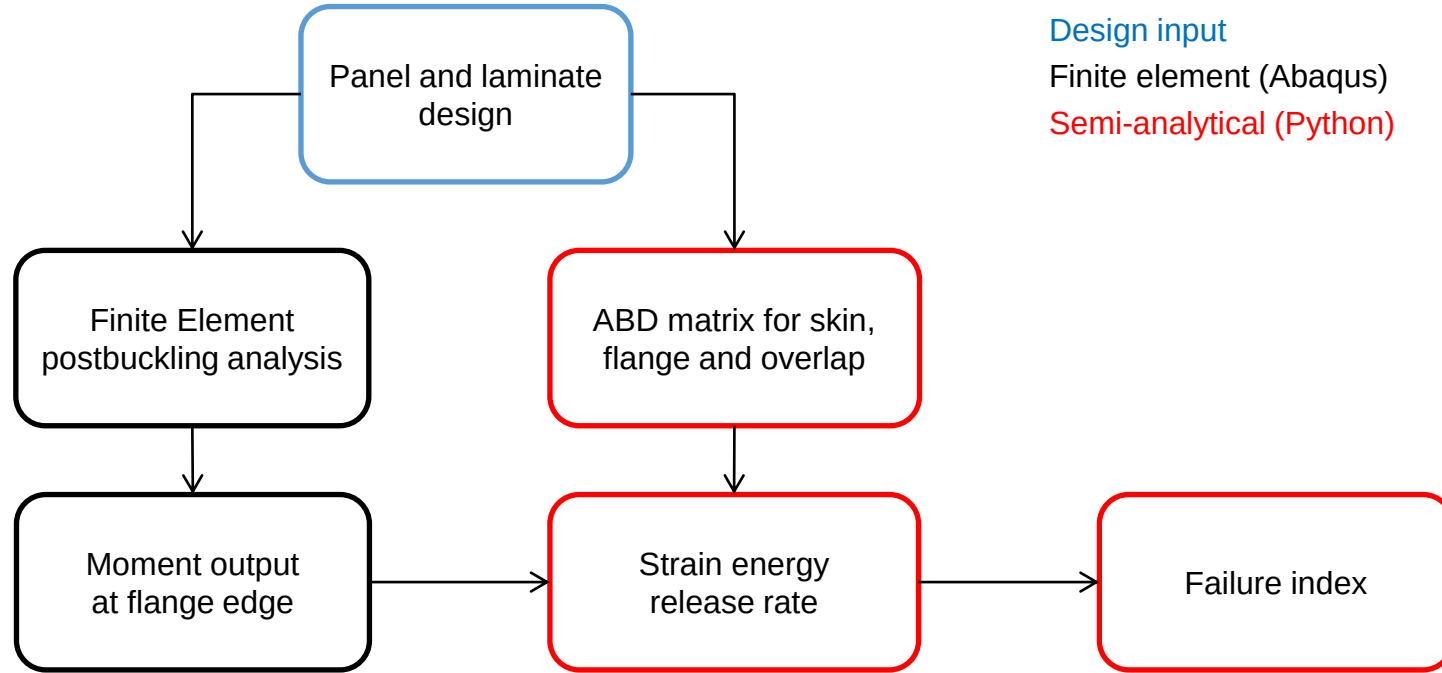
- Determine configuration
- Verify initiation
- Study propagation

3. Specimen experimental:

- Validate numerical models
- Study the physics



Skin-Stringer Separation Criterion Flow



Panel and Laminate Designs

→ Skin laminate (1.54 mm): $[-45/45/0/90/-45/45]_s$

→ Stringer Laminate

1. **Thin** (1.54 mm): $[45/-45/0/90/45/-45]_s$

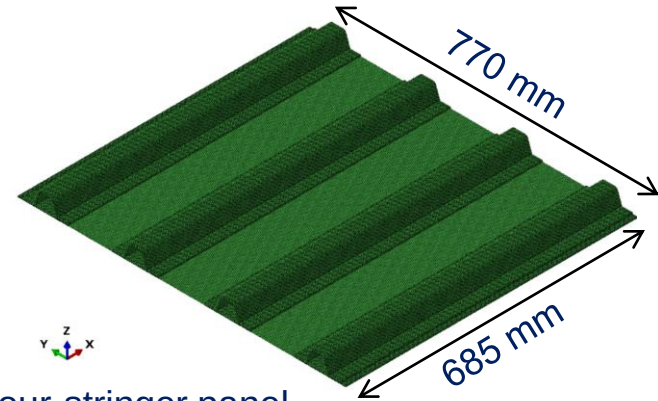
2. **Thick** (3.08 mm): $[45/-45/0/90/45/-45]_{s2}$

→ Stringer spacing of 102.5 mm

Material properties of IM7/977-3.

Longitudinal modulus	E_{11}	164000	MPa
Transverse modulus	E_{22}	8980	MPa
Shear modulus	G_{12}	5010	MPa
Poisson's ratio	ν_{12}	0.32	
Ply thickness	t	0.128	mm
Mode I critical strain energy release rate	G_{Ic}	0.256	kJ/m^2

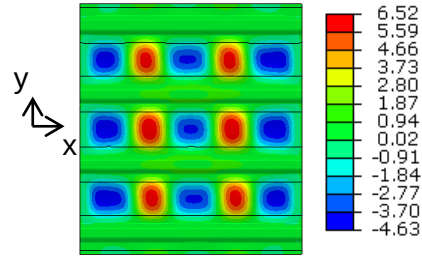
* Clay, S.B., and Knuth, P.M., "Experimental Results of Quasi-Static Testing for Calibration and Validation of Composite Progressive Damage Analysis Methods," *Journal of Composite Materials*, 51(10):1333-1353, 2017.



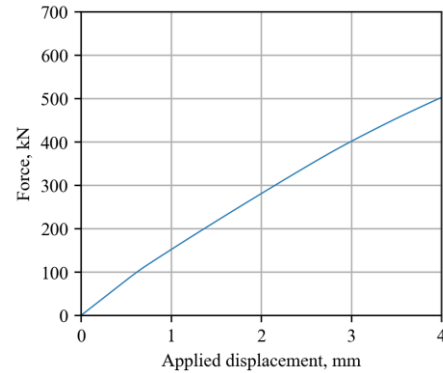
Four-stringer panel

- Clamped boundary conditions at the ends
- Free boundary conditions parallel to stringer
- Nonlinear dynamic implicit, 1 mm/s
- Mesh size of 1 mm

Postbuckling Analysis

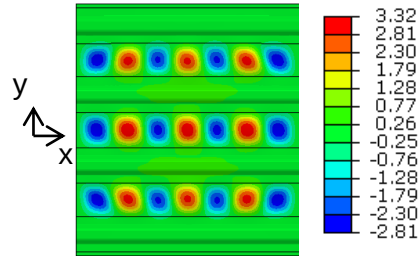


Postbuckling of panel with thin stringer at a displacement of 3.7 mm

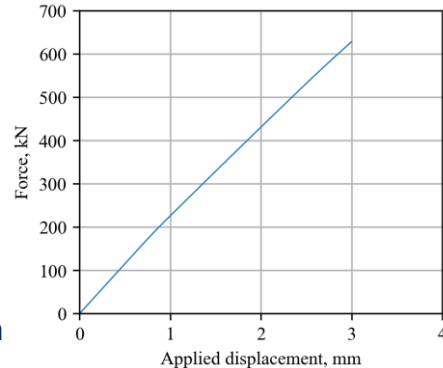


Thin stringer laminate

- Buckling point at ~0.6 mm or ~100 kN
- Five buckling half-waves (140 mm)



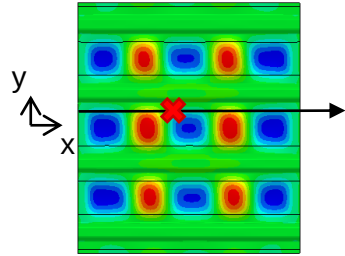
Postbuckling of panel with thick stringer at a displacement of 2.2 mm



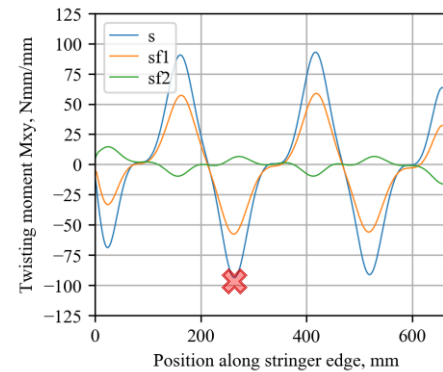
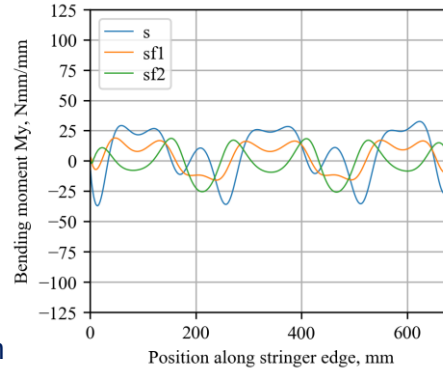
Thick stringer laminate

- Buckling point at ~0.8 mm or ~185 kN
- Seven buckling half-waves (100 mm)

Moment Output at Flange Edge

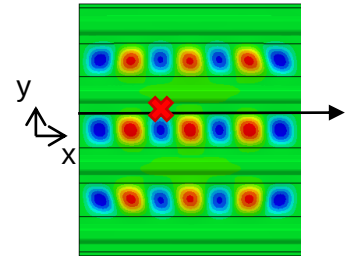


Postbuckling of panel with thin stringer at a displacement of 3.7 mm

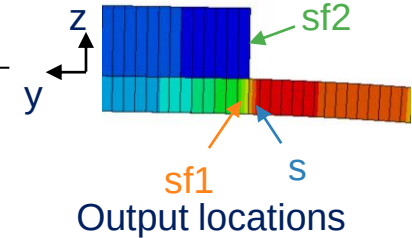
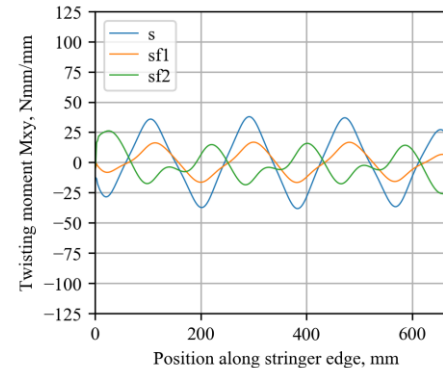
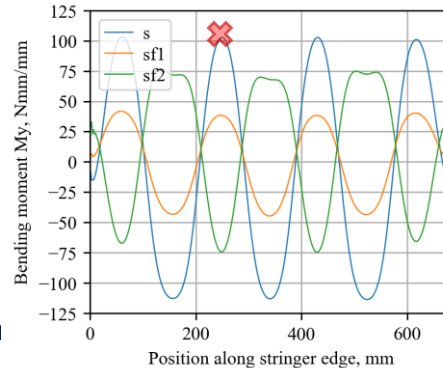


Thin stringer laminate

- Bending: 25 Nmm/mm
- Twisting: 100 Nmm/mm



Postbuckling of panel with thick stringer at a displacement of 2.2 mm

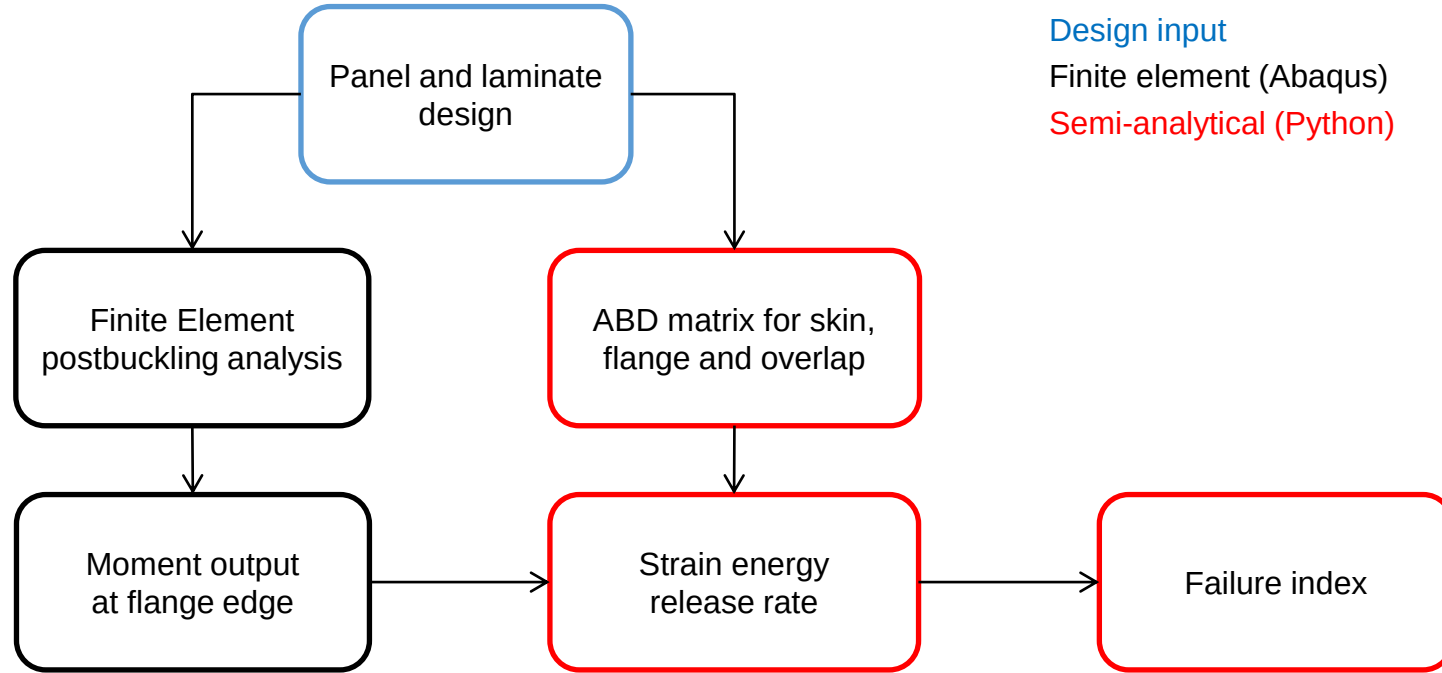


Output locations

Thick stringer laminate

- Bending: 100 Nmm/mm
- Twisting: 35 Nmm/mm

Skin-Stringer Separation Criterion Flow



Strain Energy Release rate I

The strain energy release rate is calculated locally*:

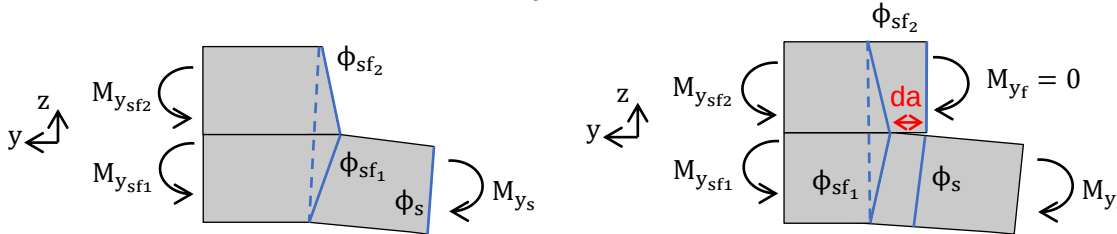
$$G = - \left(\frac{dU_e}{da} - \frac{dU_s}{da} \right)$$

Change in rotation:

$$\frac{d^2 w}{dy^2} = \frac{d\phi}{dy} = \kappa_y = M_y d_{22} = \frac{d\phi}{da}$$

The change in external work due the crack extending:

$$\frac{dU_e}{da} = \left(M_{y_s} \frac{d\phi_s}{da} + M_{y_f} \frac{d\phi_f}{da} \right) - \left(M_{y_{sf1}} \frac{d\phi_{sf1}}{da} + M_{y_{sf2}} \frac{d\phi_{sf2}}{da} \right) = 0$$



Variable	description
U_e	External work
U_s	Strain energy
da	Change in crack length
M_y	Bending moment normal to flange
d_{22}	Bending compliance normal to flange
ϕ	Rotation

*J.G. Williams, "On the calculation of energy release rates for cracked laminates", International Journal of Fracture 36:101-119, 1988.

Strain Energy Release rate I

The strain energy release rate is calculated locally*:

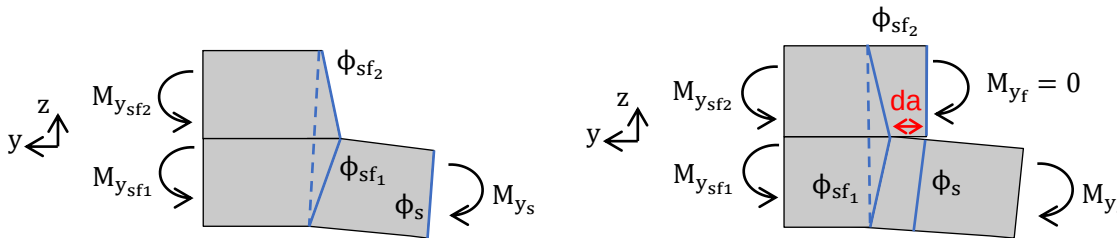
$$G = - \left(\frac{dU_e}{da} - \frac{dU_s}{da} \right)$$

Change in rotation:

$$\frac{d^2 w}{dy^2} = \frac{d\phi}{dy} = \kappa_y = M_y d_{22} = \frac{d\phi}{da}$$

The change in external work due the crack extending:

$$\frac{dU_e}{da} = M_{y_s}^2 d_{22_s} - (M_{y_{sf1}}^2 d_{22_s} + M_{y_{sf2}}^2 d_{22_f})$$



Variable	description
U_e	External work
U_s	Strain energy
da	Change in crack length
M_y	Bending moment normal to flange
d_{22}	Bending compliance normal to flange
ϕ	Rotation

*J.G. Williams, "On the calculation of energy release rates for cracked laminates", International Journal of Fracture 36:101-119, 1988.

Strain Energy Release rate II

The strain energy in a plate is given by*:

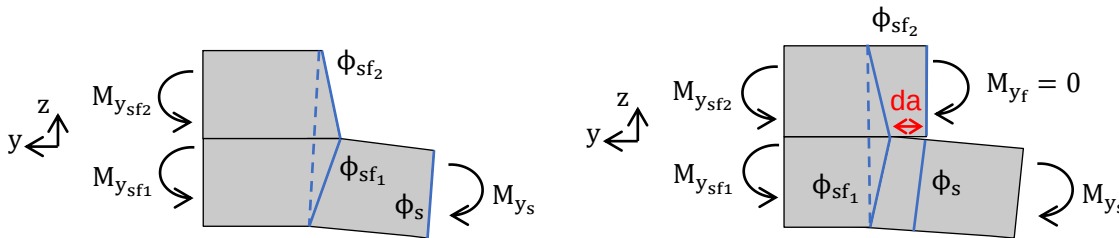
$$U_s = \frac{1}{2} [M_x^2 d_{11} - 2M_{xy}^2 d_{66} + M_y^2 d_{22}] = \frac{1}{2} M_y^2 d_{22}$$

Strain energy at the crack front before and after a crack extension:

$$U_s = \frac{1}{2} (M_{y_{sf1}}^2 d_{22s} + M_{y_{sf2}}^2 d_{22f}) \quad \text{and} \quad U_s = \frac{1}{2} (M_{y_s}^2 d_{22s} + \cancel{M_{y_f}^2 d_{22f}}) = 0$$

The change in strain energy due to an extension is:

$$\frac{dU_s}{da} = \frac{1}{2} (M_{y_s}^2 - M_{y_{sf1}}^2) d_{22s} - \frac{1}{2} M_{y_{sf2}}^2 d_{22f}$$



Variable	description
U_e	External work
U_s	Strain energy
da	Change in crack length
M_y	Bending moment normal to flange
d_{22}	Bending compliance normal to flange
ϕ	Rotation

*T.H.G. Megson, "Aircraft Structures for Engineering Students", Elsevier Science, 5th edition, 2012.

Calculate the Failure Index

After obtaining that:

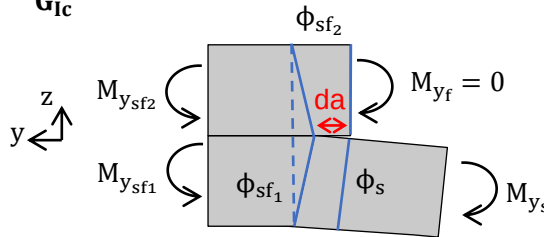
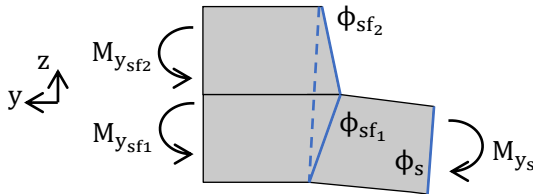
$$\frac{dU_s}{da} = \frac{1}{2} \frac{dU_e}{da}$$

The strain energy release rate becomes:

$$G_b = \frac{1}{2} (M_{y_s}^2 - M_{y_{sf1}}^2) d_{22s} - \frac{1}{2} M_{y_{sf2}}^2 d_{22f}$$

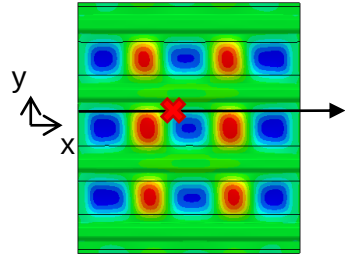
The strain energy release rate due to bending is compared to the critical strain energy release rate for mode I, resulting in the expression:

$$\text{Bending: } FI_b = \frac{G_b}{G_{Ic}}$$

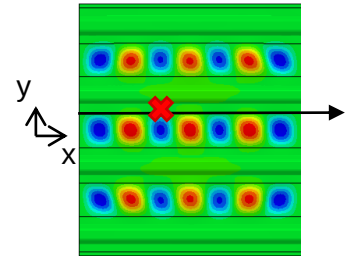
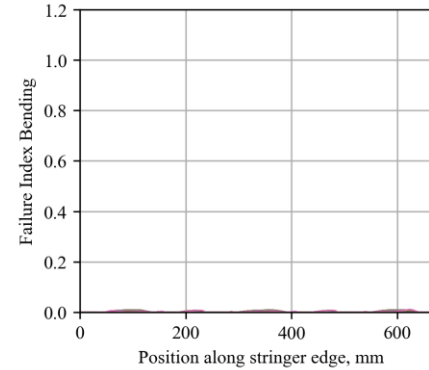
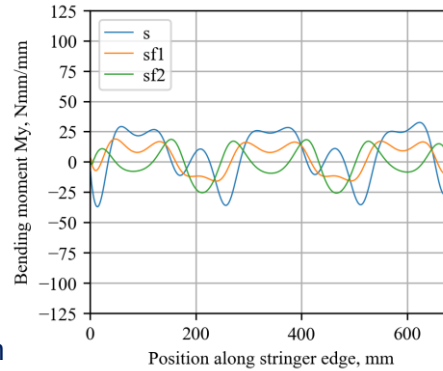


Variable	description
U_e	External work
U_s	Strain energy
da	Change in crack length
M_y	Bending moment normal to flange
d_{22}	Bending compliance normal to flange
ϕ	Rotation

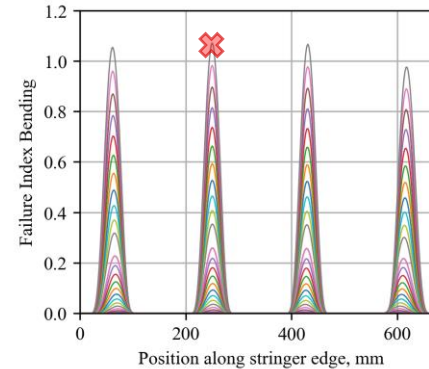
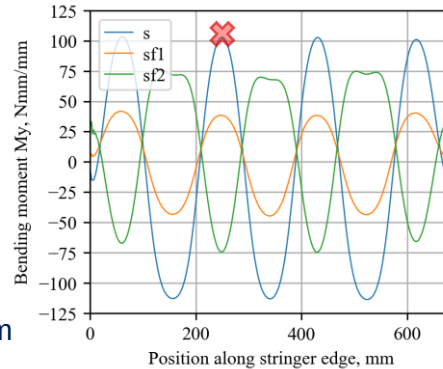
Failure Index in Panel



Postbuckling of panel with thin stringer at a displacement of 3.7 mm



Postbuckling of panel with thick stringer at a displacement of 2.2 mm



Methodology

1. Simple panel model:

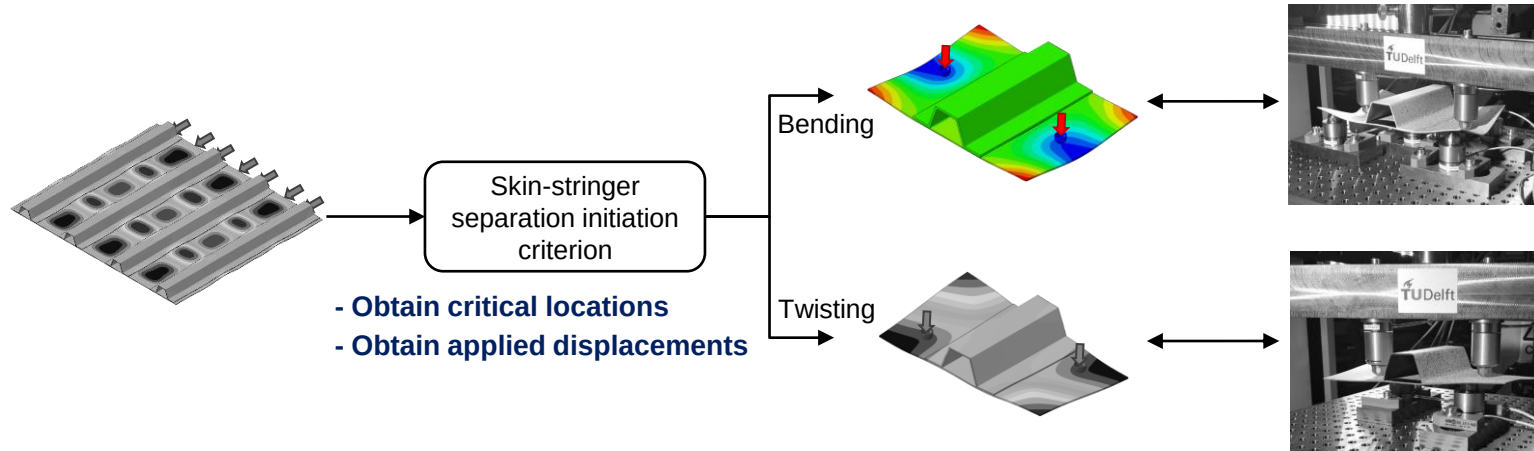
- Obtain postbuckling shape
- Obtain critical locations
- Obtain applied displacements

2. Specimen numerical:

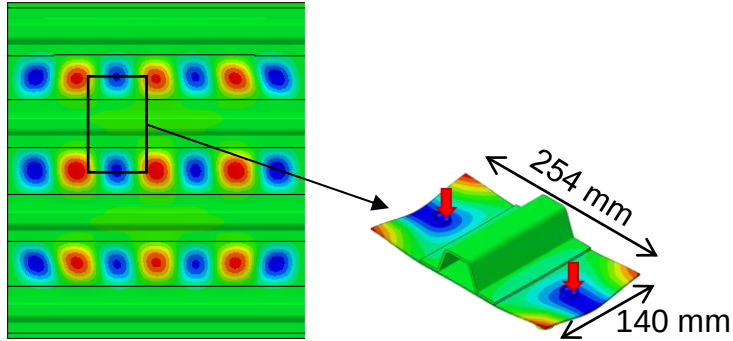
- **Determine configuration**
- **Verify initiation**
- Study propagation

3. Specimen experimental:

- Validate numerical models
- Study the physics



Single-Stringer Specimen Design



- Skin laminate: $[-45/45/0/90/-45/45]_s$
- **Thick** stringer laminate: $[45/-45/0/90/45/-45]_{s2}$
- Supports: 20 mm from both edges
- Loading points: 34 mm from the edge

Kootte, L.J., Bisagni, C., Dávila, C.G., and Ranatunga, V., "Study of Skin-Stringer Separation in Postbuckled Composite Aeronautical Structures", *American Society for Composites (ASC) 33rd Technical Conference*, Seattle, WA., 2018.

Skin-stringer interface

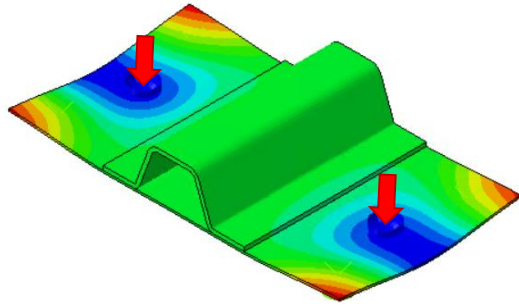
- Cohesive elements (COH3D8) (0.3 mm)
- Quadratic stress criterion
- Benzeggagh-Kenane (B-K) law

Interlaminar properties of IM7/977-3.

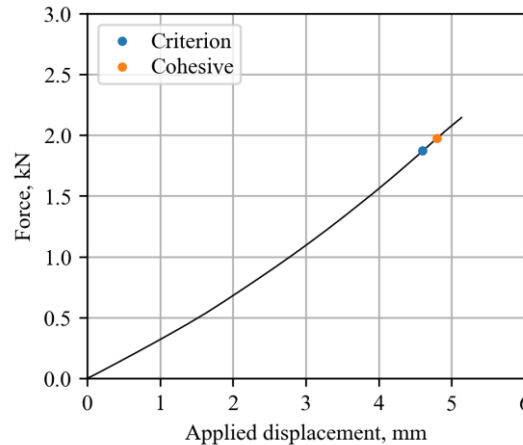
Mode I critical SERR	G_{IC}	0.256	kJ/m ²
Mode II critical SERR	G_{IIc}	0.65	kJ/m ²
Mode I strength	τ_{Ic}	78.9	MPa
Mode II strength	τ_{IIc}	99.4	MPa
B-K parameter	η	2.07	

Clay, S.B., and Knoch, P.M., "Experimental Results of Quasi-Static Testing for Calibration and Validation of Composite Progressive Damage Analysis Methods," *Journal of Composite Materials*, 51(10):1333-1353, 2017.

Cohesive Result of Single-Stringer

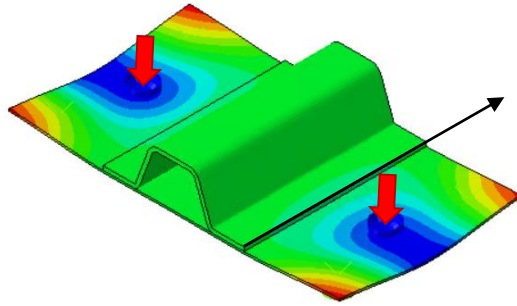


Single-stringer specimen at a displacement of 4.6 mm

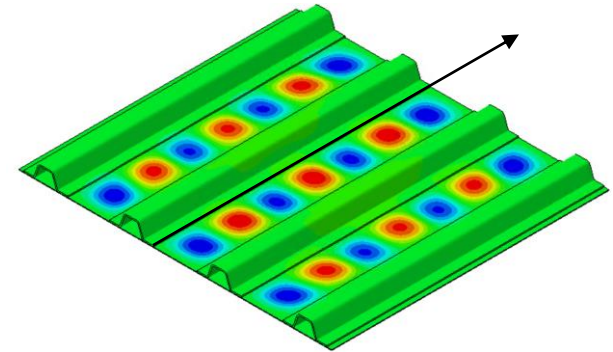
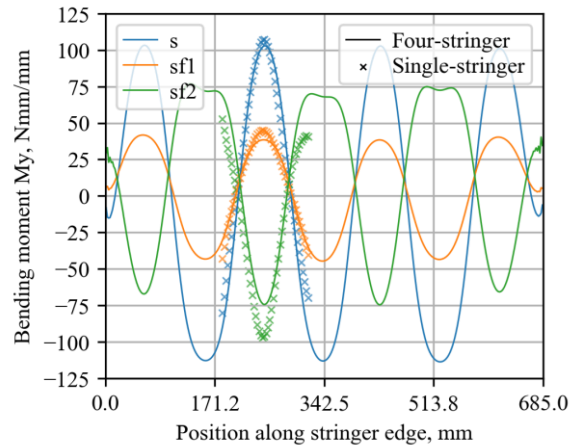


- Cohesive identified as damage variable (SDEG) equal to 1.
- Cohesive initiation is 4% higher compared to the criterion.

Single-Stringer compared to Multi-Stringer



Single-stringer specimen at a displacement of 4.6 mm



Postbuckling of panel with thick stringer at a displacement of 2.2 mm

Conclusions and Future Work

- Methodology to study skin-stringer separation in postbuckled composite structures is under development.
 - Identify critical locations through bending and twisting moments.
 - Estimate displacement through skin-stringer separation criterion.
 - Verify response through single-stringer specimen.
- Benefits:
 - Reduce dependency of cohesive elements on large scale models.
 - Allow to select the single-stringer specimen configuration.
 - Study the interaction between the skin-stringer separation and the postbuckling deformation.
- Future Work:
 - Validate response with experiments
 - Extend for Mode II + III and study with four-point twisting

