NATIONAL TRANSPORTATION SAFETY BOARD

Office of Aviation Safety
Washington, D.C. 20594

March 22, 2019

STRUCTURES

Group Chairman's Factual Report

ERA18FA120

Attachment 3 – USAF Wing Spar Analysis Results
(38 pages)

Piper Aircraft Mishap NTSB Support

A-10 ASIP
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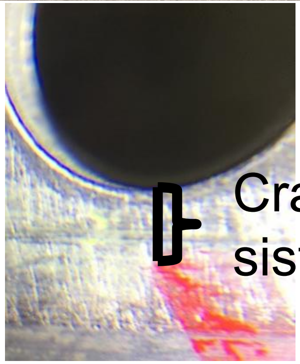
Acknowledgments

- Broad Application for Modeling Fracture (BAMF) assistance
  - Kaylon Anderson (USAF): Analytical support
  - Josh Hodges (Hill Engineering): BAMF Programming updates and analytical support
- Spectrum assistance
  - Luciano “Lucky” Smith (Southwest Research Institute): Maneuver and Gust randomization, exceedance data integration for spectrum

Problem

- **Why did mishap aircraft experience catastrophic failure while only one other aircraft inspected showed a crack indication?**



Crack on
sister aircraft

Analysis Overview

- Understand usage type and develop applicable spectrum
- Perform preliminary crack growth analyses in AFGROW to verify assumptions, spectrum, and stress
- Create StressCheck models for Broad Application for Modeling Fracture (BAMF) analysis
  - Evaluate influence of initial flaw size and fastener fit
- Compare models with mishap evidence
- Provide related observations

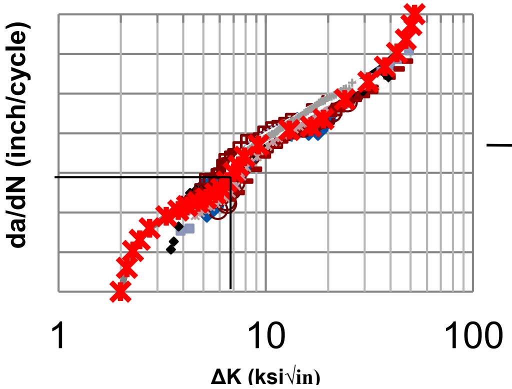
What is BAMF?

$$K_{min_{CN}} = a_{app_{CN}}\sigma_{min} + K_{res_{CN}}$$

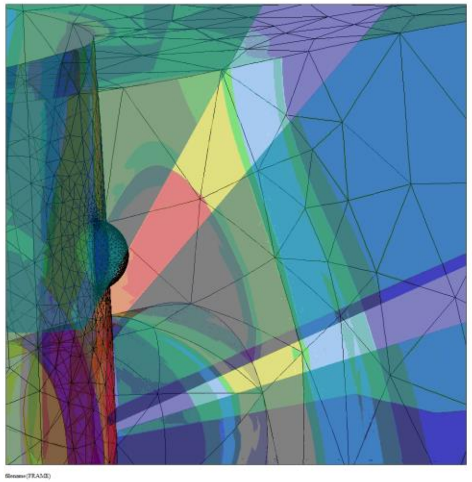
$$K_{max_{CN}} = a_{app_{CN}}\sigma_{max} + K_{res_{CN}}$$

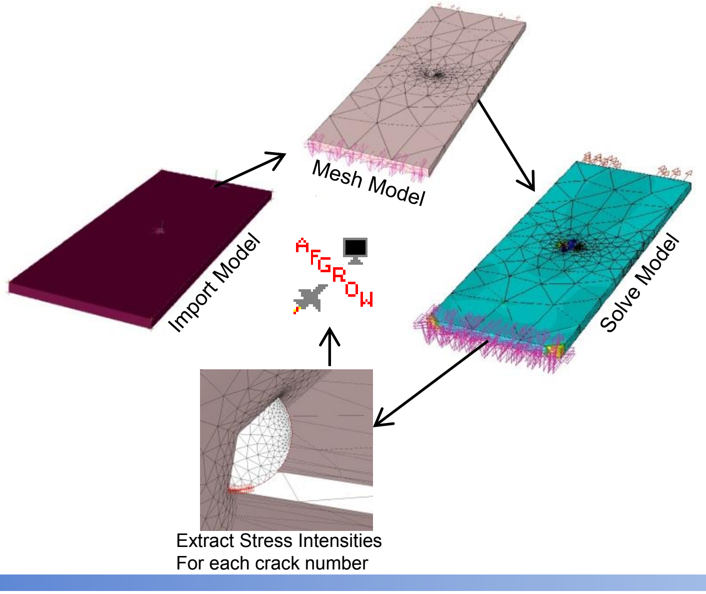
Where $\sigma_{min/max}$ is the AFGROW spectrum stress →

$$R_{CN} = K_{min_{CN}} / K_{max_{CN}}$$



→ New Crack Lengths

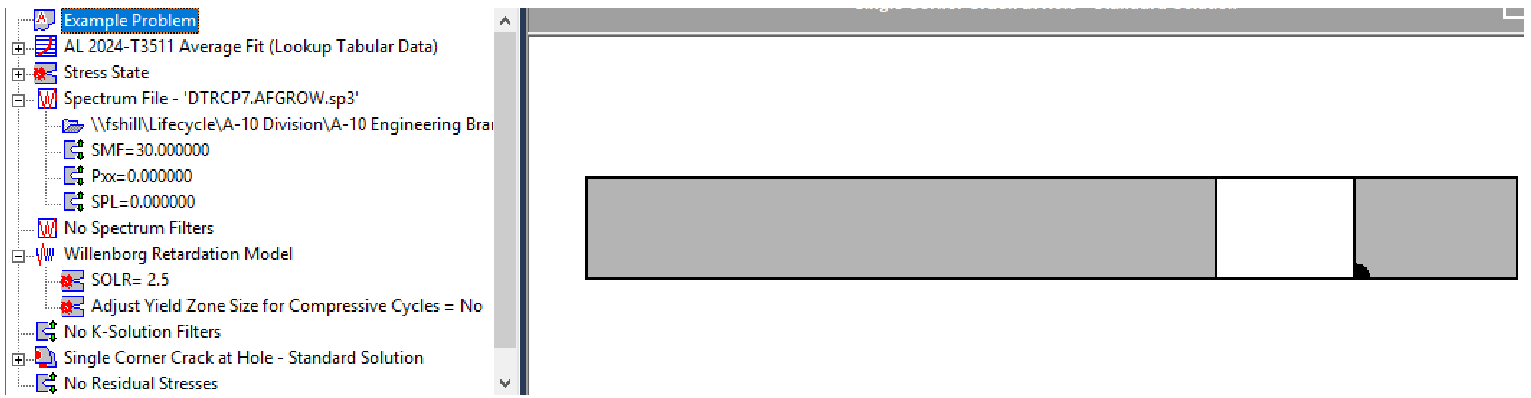


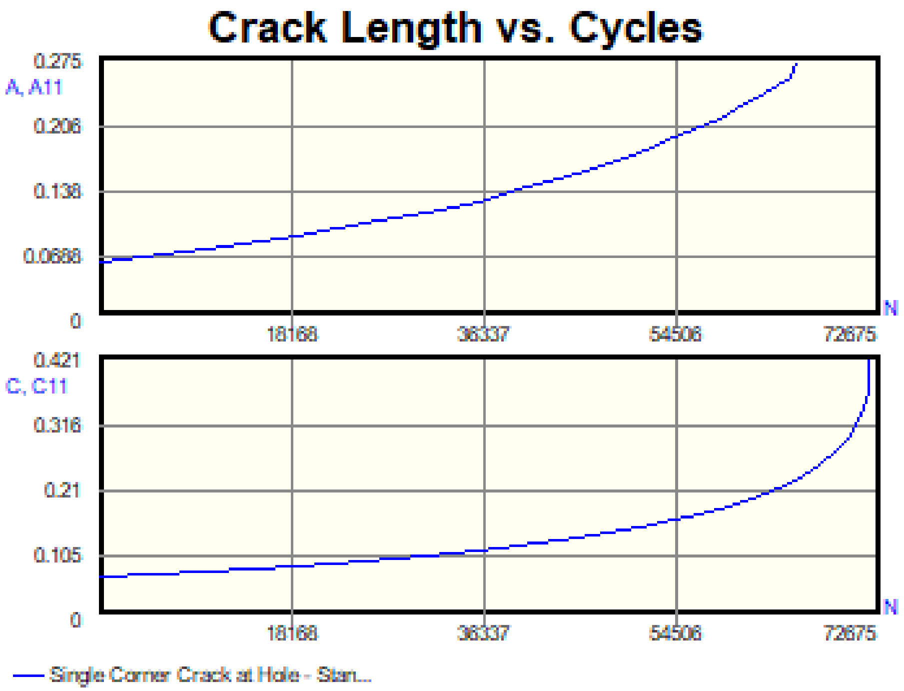


- Links StressCheck with AFGROW
  - Continuing damage solution determination
  - Multipoint stress intensity determination
  - Incremental crack propagation along crack front
  - Allows for inclusion of residual stresses
- Initially developed to support deep residual stress evaluations (CX, LSP, etc.)

AFGROW Analysis

- AFGROW analysis used to assess initial spectrum severity
- Started with A-10 wing root bending spectrum for comparison
- Iterated through 4 spectra developed for this effort
- Evaluated appropriate peak spectrum stress



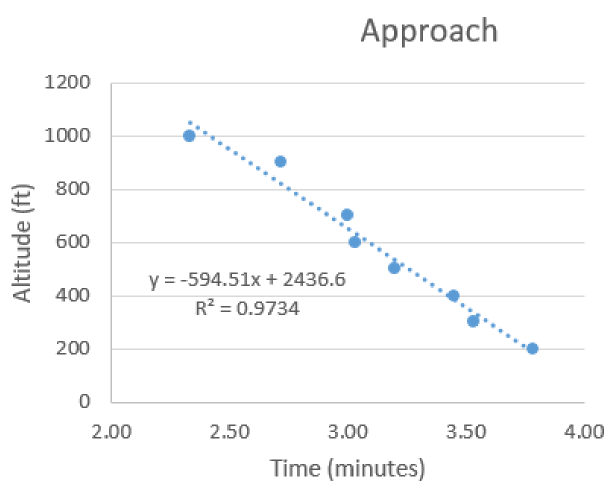


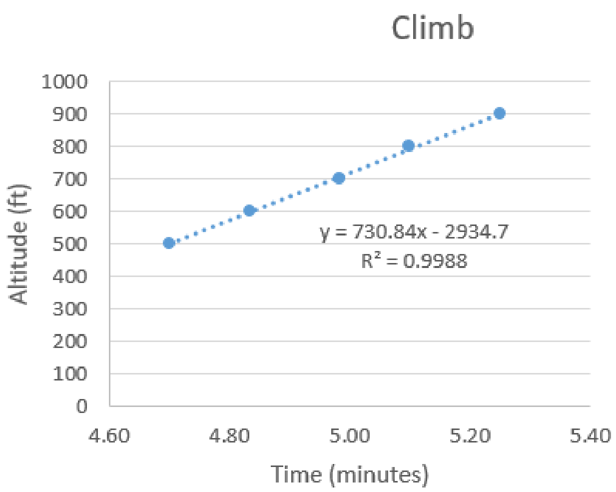
Usage Characterization

- Aircraft history indicated usage was predominantly training
  - S/N 2844137 had 7,690 total airframe hours for 33,276 takeoff/landing cycles (NTSB Investigative Update ERA18FA120)
    - S/N 2844137 flight length 13.87 minutes
    - S/N 2844135 flight length 14.26 minutes
      - Average training mission 14.06 minutes
  - Aircraft operational performance limited altitude attained for each mission
    - Subject to significant gust environment
- Radar approximates observed performance
  - Altitude and elapsed time

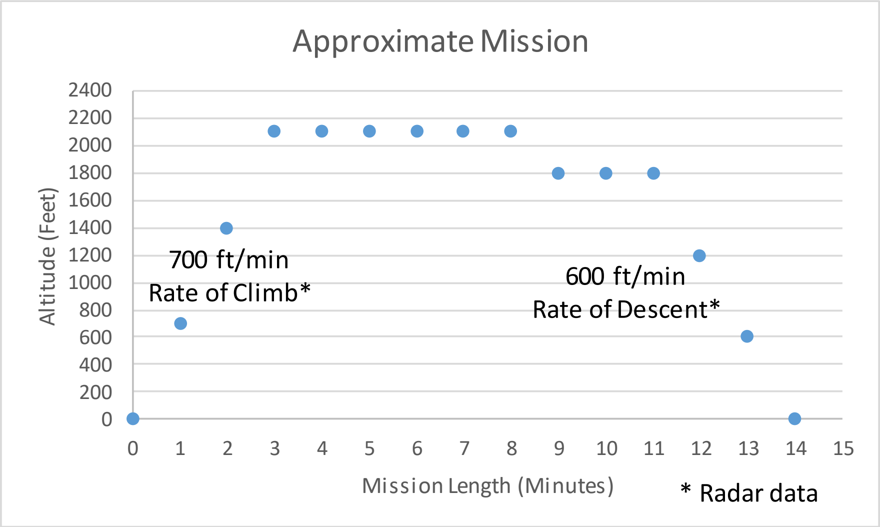
Usage Characterization (Continued)

- **Approximate descent and climb performance immediately prior to mishap (RDDL 106 Fatal.mp4)**





- **Descent and climb rates assumed to be representative of most descent/climb (touch and go) events**



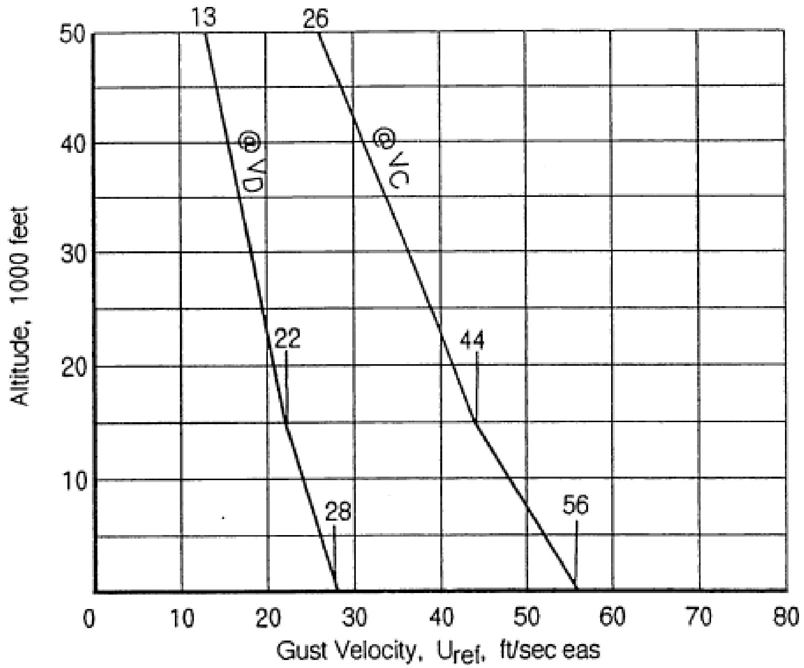
Spectrum Determination

- **Utilized data from LG88ER0016 as basis for modified spectrum (Piper Aircraft Model PA28-181 and PA32-300 Main Spar Fracture Analysis)**
- **Gust environment (Applicable)**
  - **Peak gust velocities are inversely proportional to altitude**
    - **Gust velocity variation between 0 and 5,000 feet is minimal (within 10%)**
    - **Aircraft performance limits mission altitude to less than 2,500 feet**
- **Landing and Taxi (Applicable)**

Manuever			Gust		Taxi		Landing	
Number of occ. / hr.	ΔG -	ΔG +	Number of occ. / hr.	±ΔG	Number of occ. / hr.	±ΔG	Number of occ. / ft.	G's
34.30	.11	.44	228.60	.35	320	.05	1.000	1.00
10.30	.36	.65	137.20	.51	190	.10	.350	1.25
9.14	.52	.85	57.15	.68	80	.15	.170	1.42
5.72	.58	1.054	22.90	.84	30	.20	.026	1.58
2.06	.69	1.26	9.4	1.00	8.5	.25	.003	1.80
.80	.88	1.46	1.26	1.17	2.1	.30	.0007	2.06
.37	.97	1.67	.46	1.33	.50	.35	.0002	2.3
.126	1.02	1.87	.16	1.49	.10	.40		
.05	1.13	2.07	.09	1.65	.018	.45		
.017	1.24	2.28	.046	1.82	.010	.465		
.006	1.41	2.48	.014	1.99				
.0033	1.55	2.65	.015	2.11				
.0013	1.66	2.86	.007	2.28				
			.0034	2.44				
			.0023	2.60				

Aircraft Model I.D.	Ground Stress (psi)	Stress / G (psi/g)	1 G Mean Stress (psi)	Design Limit Stress (psi)
PA - 28 - 181	2409.8	7816	7689	34,264
PA - 32 - 300	2552.0	8444	10,500	39,209

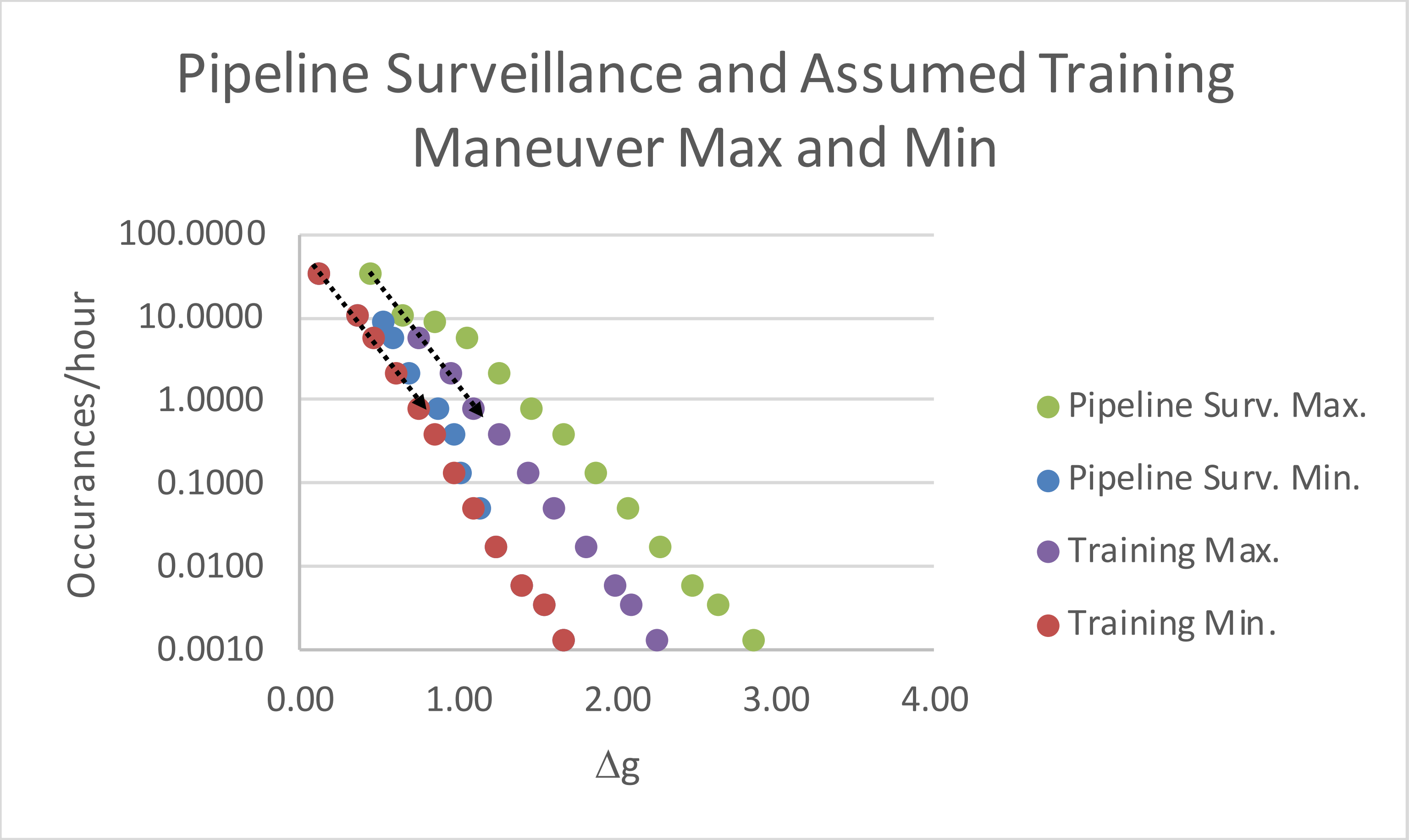
Figure 2.1 Exceedance Data and Stress Definitions as Supplied by Piper Aircraft Company



Lomax, TL, “Structural Loads Analysis for Commercial Transport Aircraft: Theory and Practice,” AIAA 1996.

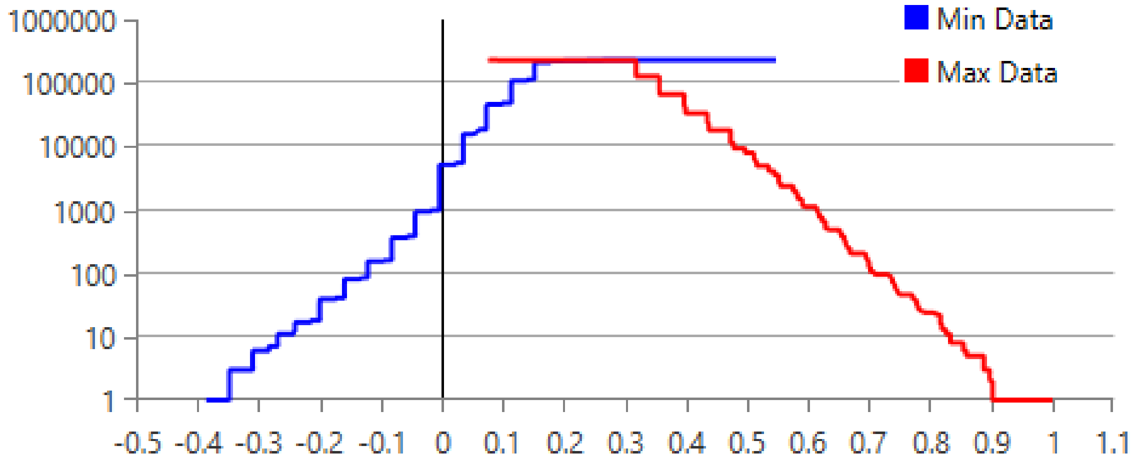
Maneuver Content Assumption

- **Pipeline Surveillance maneuver exceedance data considered aggressive for training mission**
- **Estimated training mission maneuver content to be consistent with initial slopes of the Pipeline Surveillance exceedance data**
  - **Estimated training content randomly combined with Gust content**



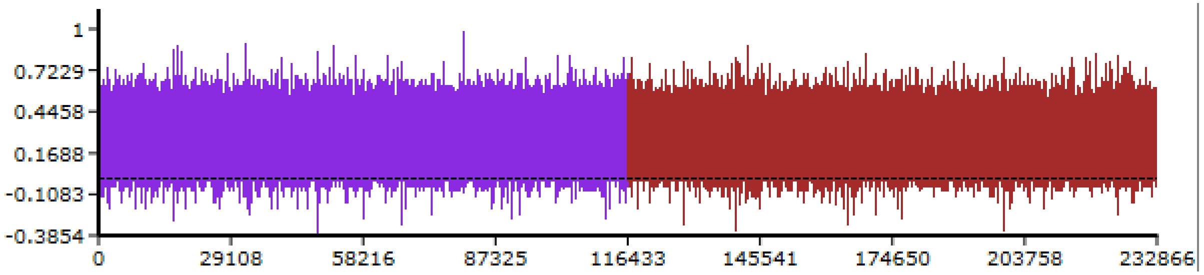
Spectrum Development

- Lockheed Pipeline Surveillance peak stress: 31 ksi
- New spectrum changes noted previously and to the right
  - Normalized to allow adjustment of peak stress
  - Approximate peak stress of 14 ksi
    - Roughly equivalent to 1.8-2.0 g peak



SEGMENT I.D.	CYCLES	STRESS MIN.	STRESS MAX.	
21	.4500	1898.55	2921.05	
21	.2500	1876.05	2943.55	
21	1.0000	1816.80	3002.80	Taxi
21	.3300	1721.90	3097.70	
21	.1180	1628.10	3191.50	
21	.0370	1534.30	3285.30	
21	.0110	1440.50	3379.10	
21	.0030	1346.70	3472.90	
13	655.0000	6189.00	11001.50	Gust, # of cycles divided by 2 for 1 hour mission
13	93.5000	1889.00	18626.50	
13	[illegible]	[illegible]	20314.00	
13	[illegible]	[illegible]	22339.00	
13	.1020	-5911.00	24639.00	
13	[illegible]	[illegible]	[illegible]	
13	.0085	-11411.00	29239.00	
13	.0025	-14811.00	31539.00	
24	.2800	2409.80	2550.00	Landing, # of cycles multiplied by 4 for training mission
24	.2900	2409.80	2850.00	
24	.2100	2409.80	3150.00	
24	.1400	2409.80	3450.00	
24	.0650	2409.80	3750.00	
24	.0105	2409.80	4050.00	
24	.0026	2409.80	4350.00	
24	.0009	2409.80	4650.00	
21	.4500	1898.55	2921.05	Taxi
21	.2500	1876.05	2943.55	
21	1.0000	1816.80	3002.80	
21	.3300	1721.90	3097.70	
21	.1180	1628.10	3191.50	
21	.0370	1534.30	3285.30	
21	.0110	1440.50	3379.10	
21	.0030	1346.70	3472.90	

Modified as described above

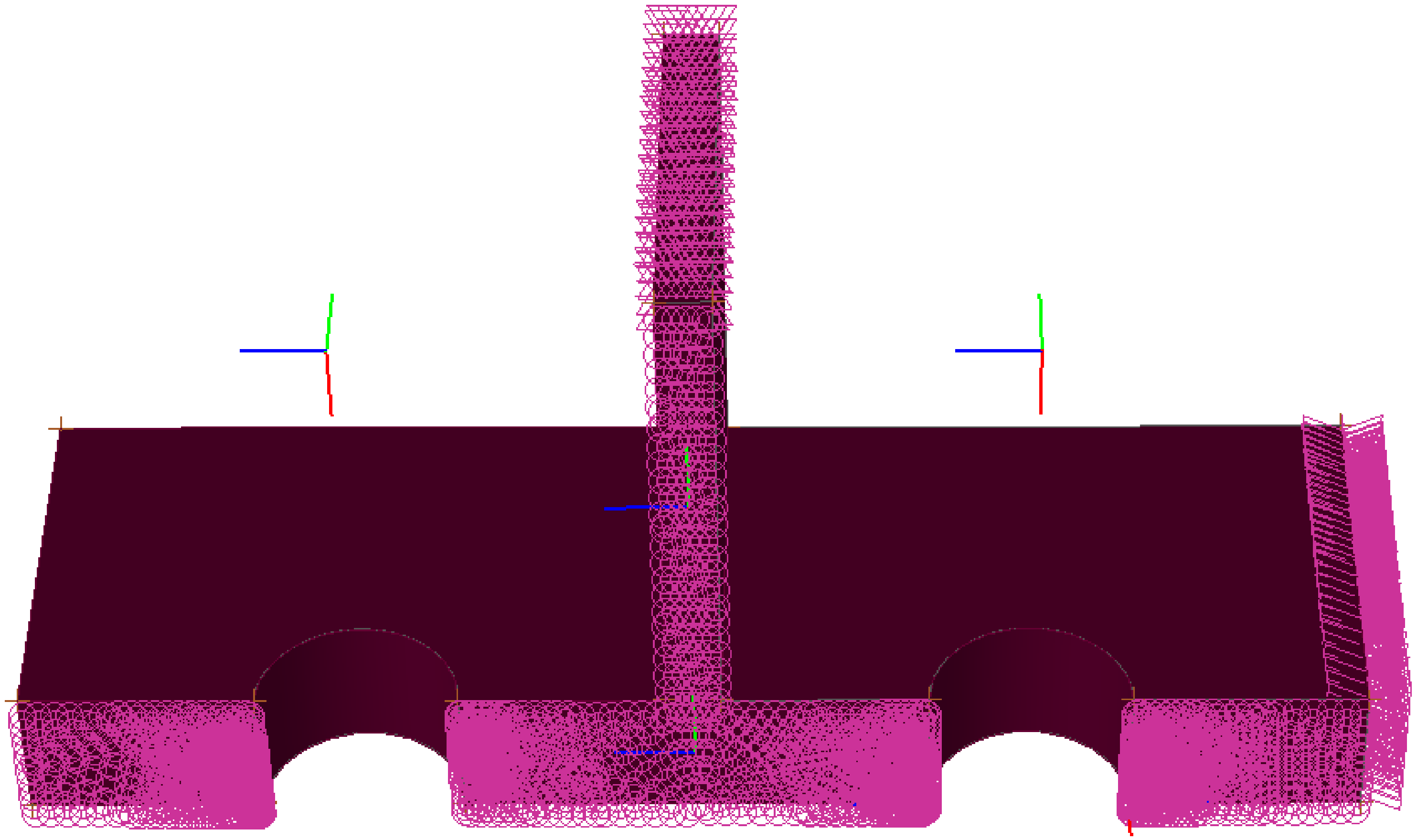


Damage Tolerance Analysis Assumptions

- Fracture mechanics material parameters
  - A-10 tabular look-up data were used
- No load history effects considered
  - Crack growth retardation parameters were not used
- Overall A-10 best practices used for analysis
  - Growth increments
  - Constraints
  - Etc.

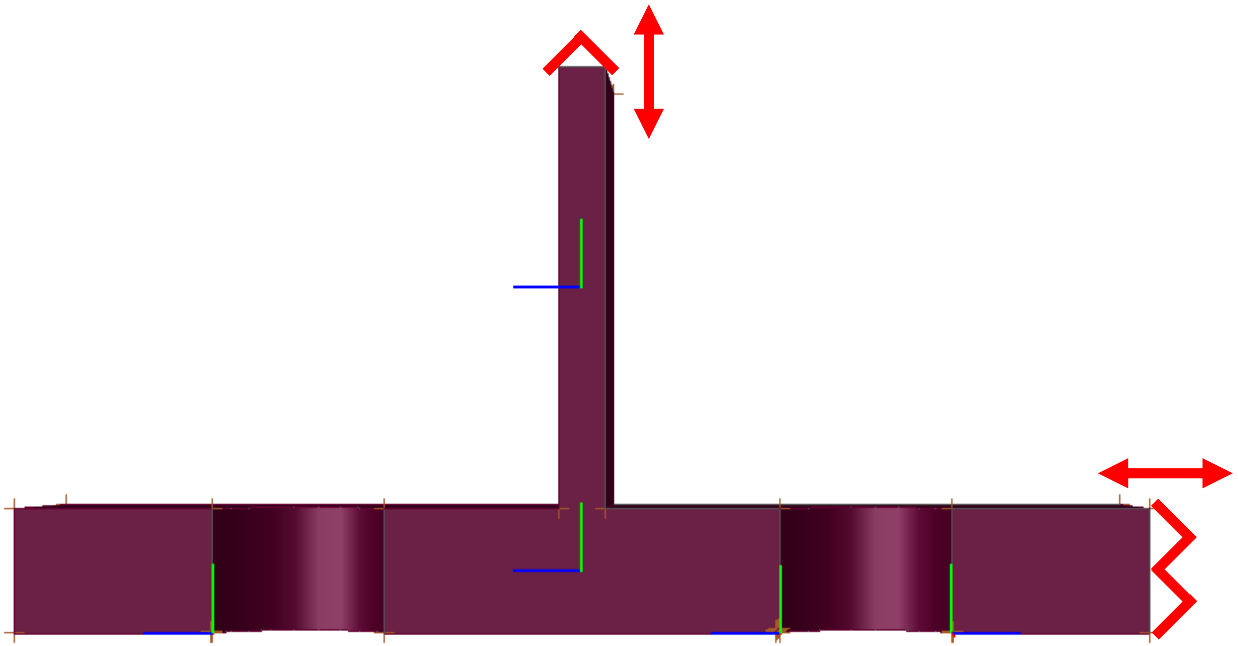
StressCheck Model

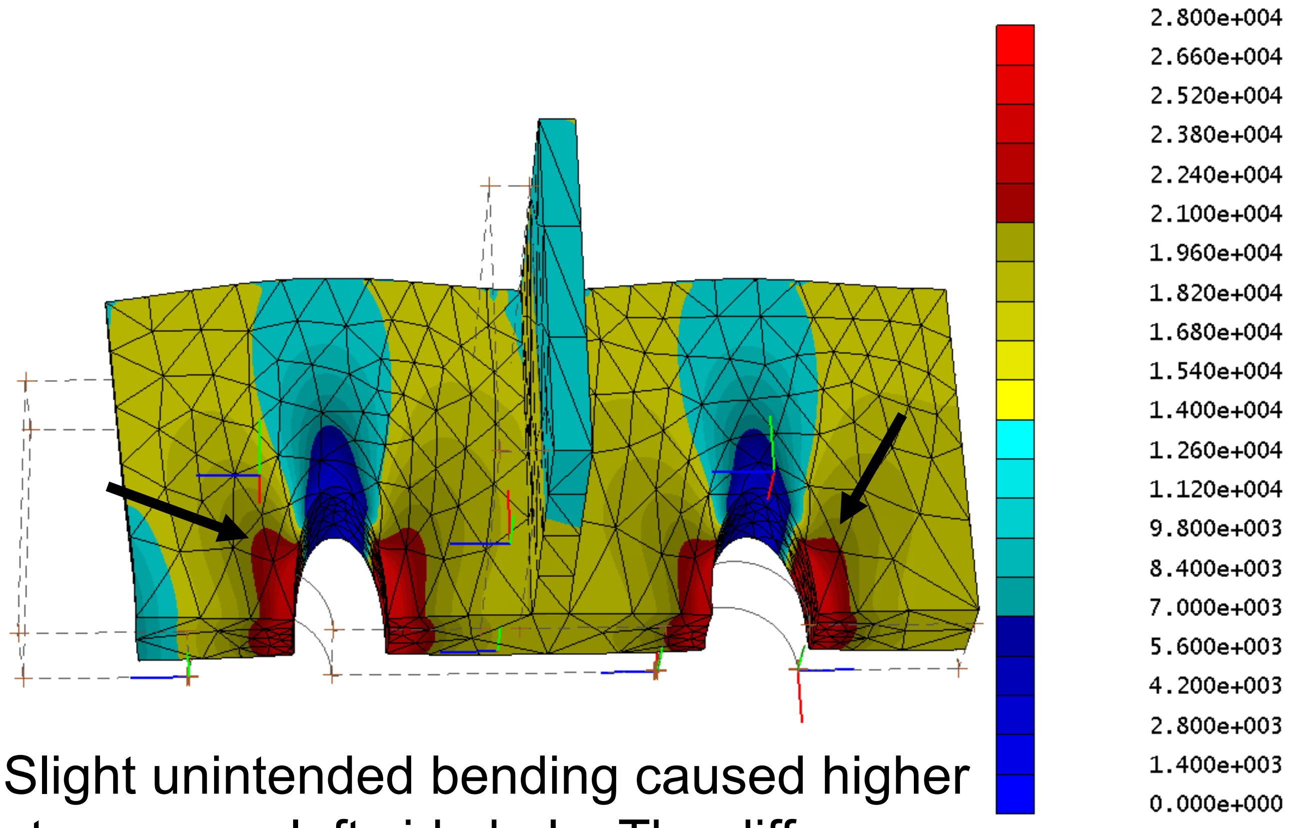
- **Initial constraint approach:**



StressCheck Model Initial Constraint

- **Initial constraint approach**

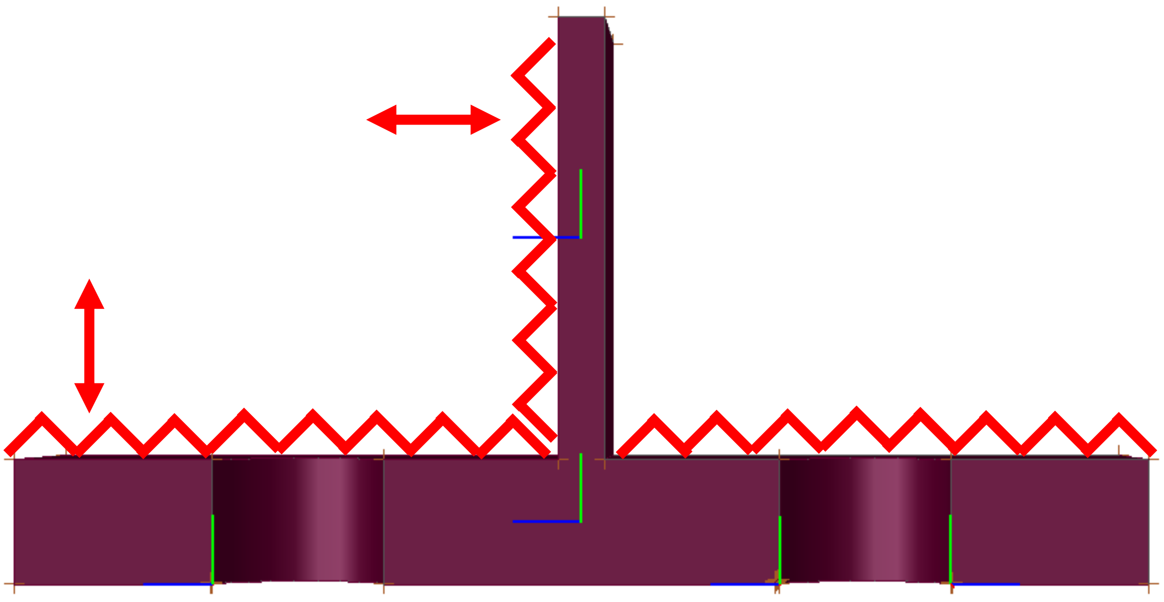


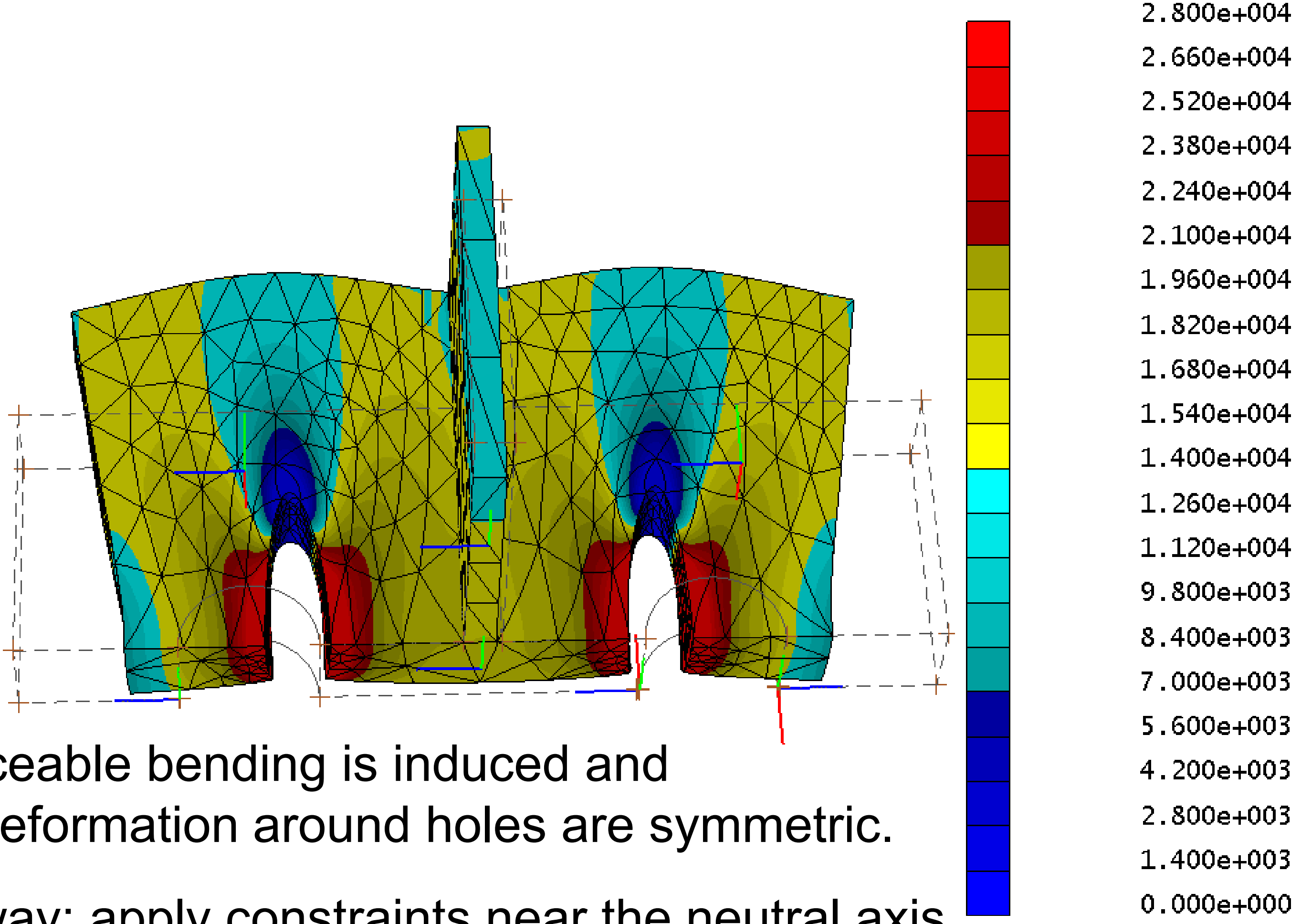


Slight unintended bending caused higher stresses on left side hole. The difference caused cracks to not grow analytically on the side closest to the constraint.

StressCheck Model Final Constraint

- Final constraint approach



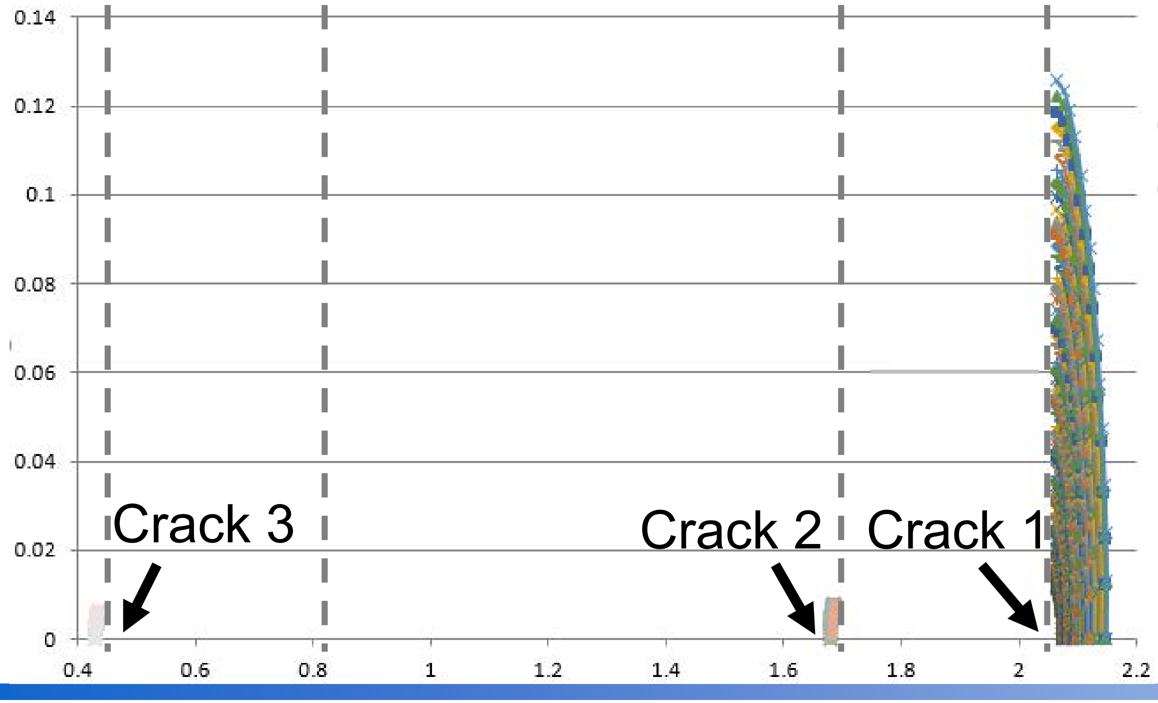


No noticeable bending is induced and stress/deformation around holes are symmetric.

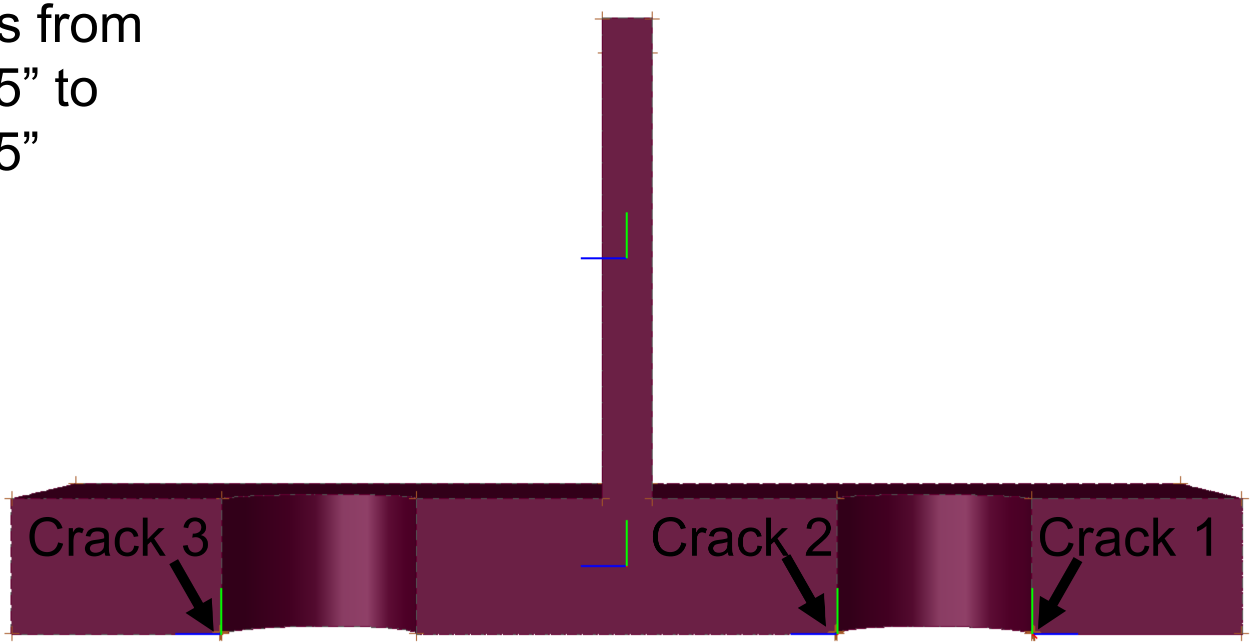
Take away: apply constraints near the neutral axis

Near Pristine Analysis (0.005” cracks) (14 ksi peak)

- USAF durability analysis typically assumes an initial crack of 0.01”
- Continuing damage analysis typically assumes a 0.005” crack in the secondary location
- Analysis was completed with three 0.005” cracks to estimate pristine open hole fatigue life

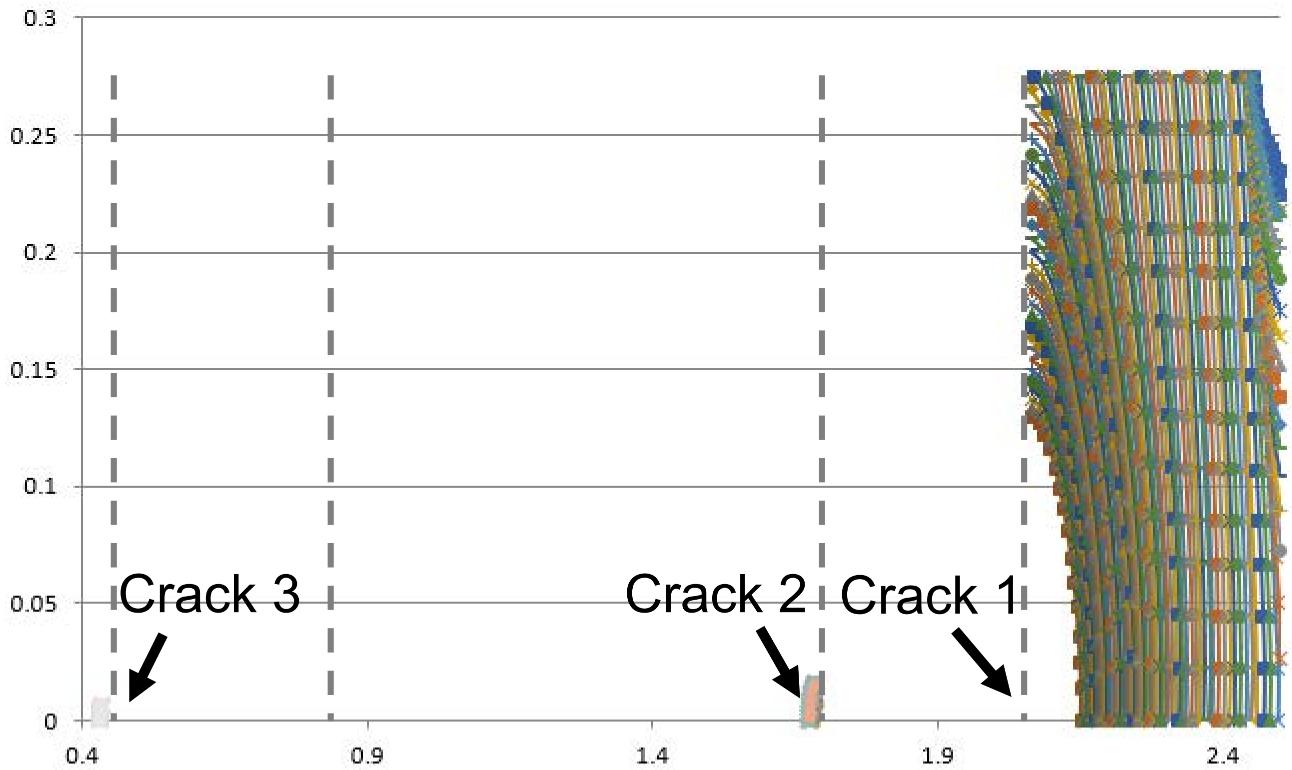


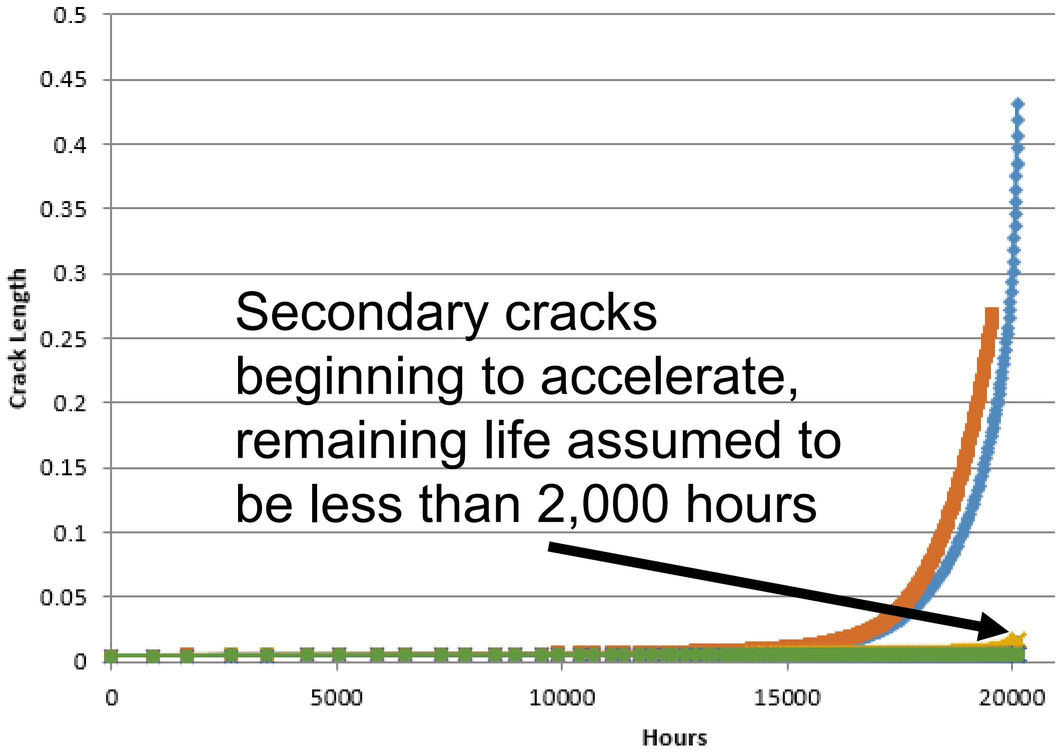
18,700 flight hours from 0.005” to 0.125”



Near Pristine Analysis (Continued) (14 ksi peak)

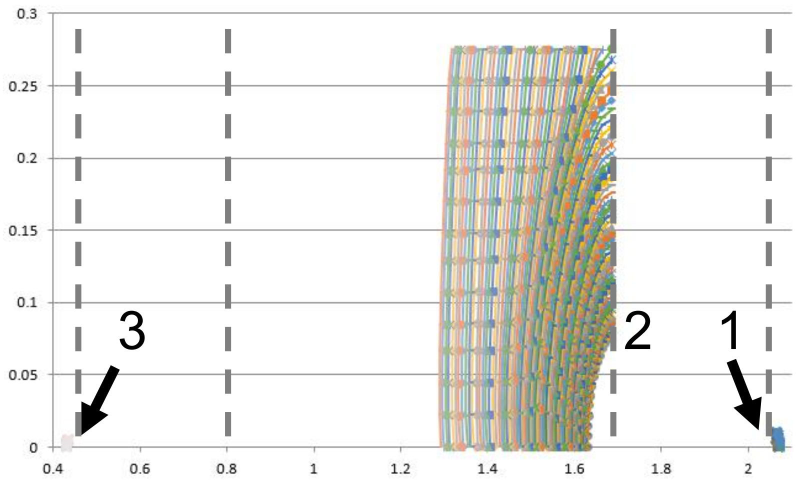
- Model refined/adjusted for transition to a thru crack
- ~1,400 flight hours from 0.125" crack to ligament failure
- ~20,100 flight hours total to ligament failure

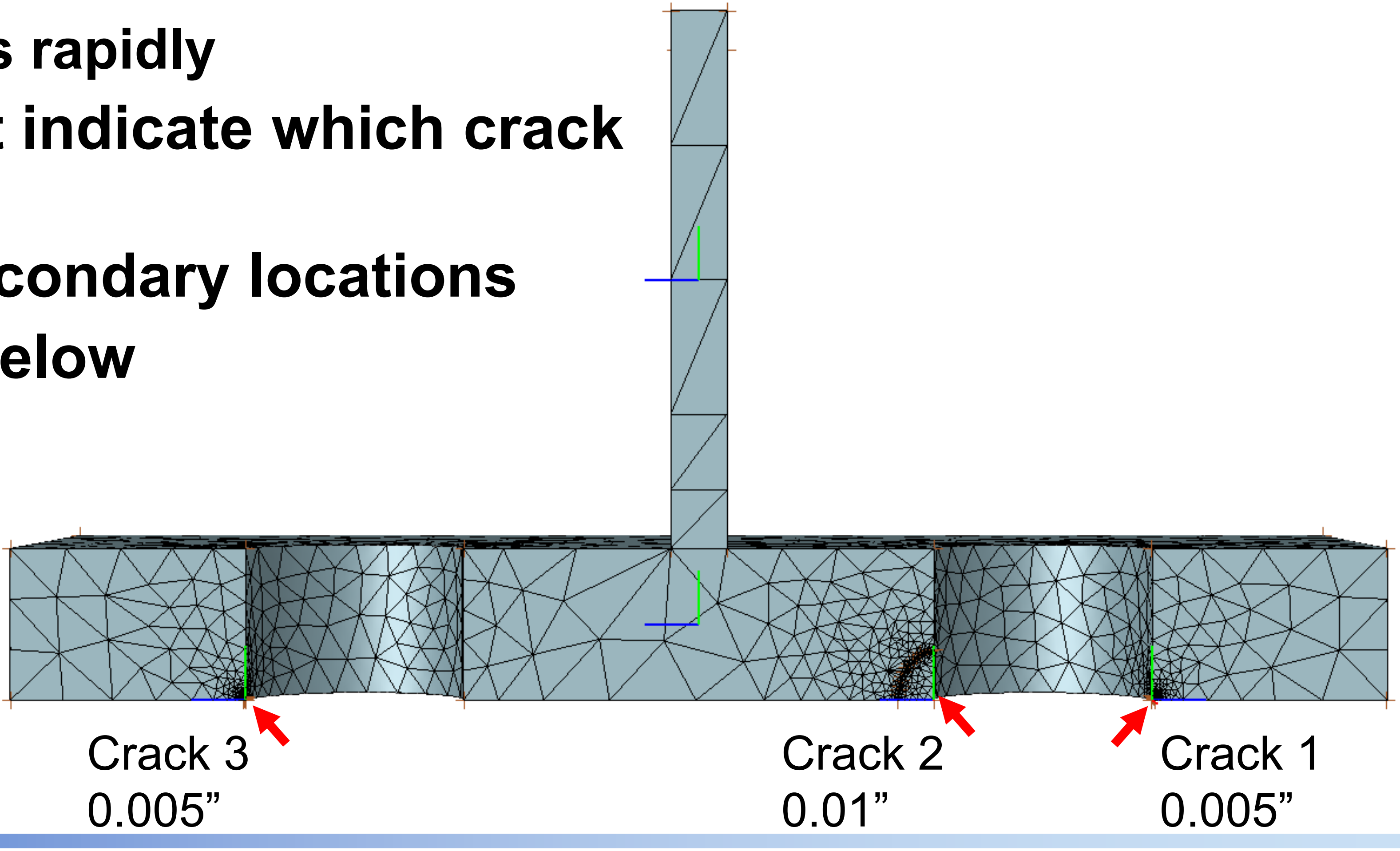




Durability Crack Analysis (14 ksi peak)

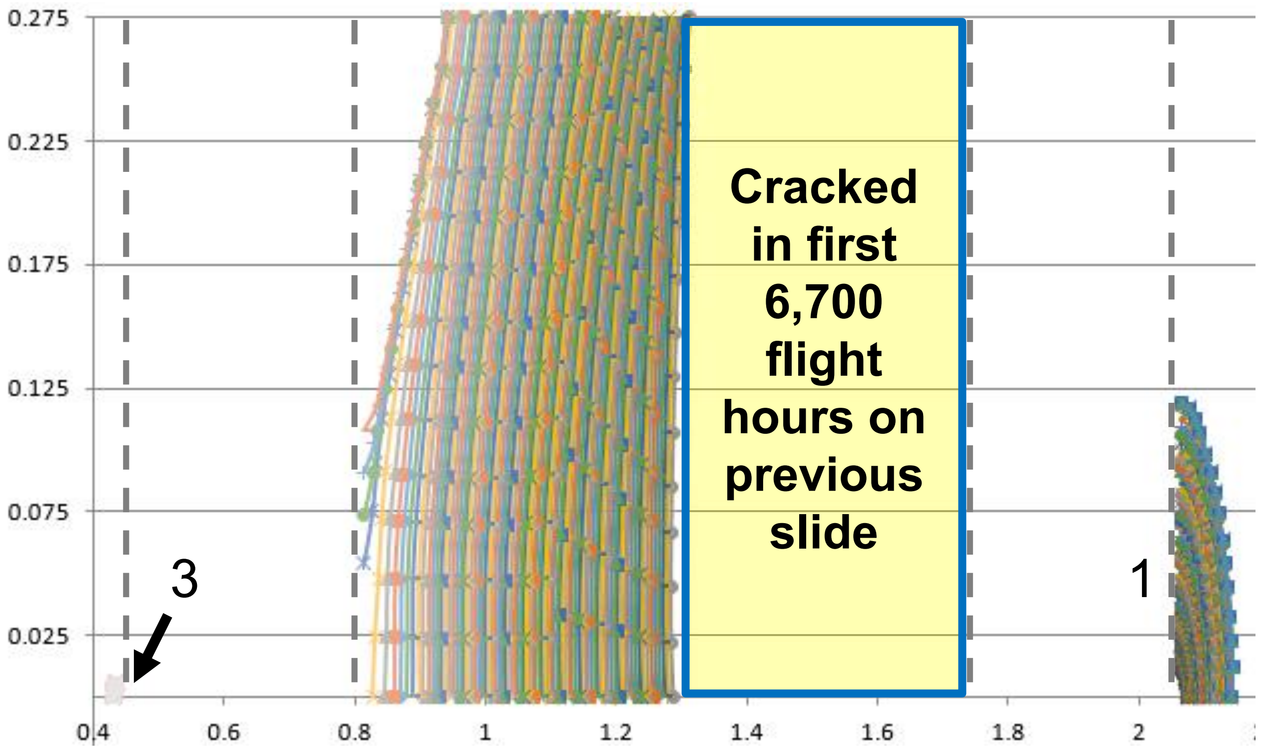
- Crack 2 selected to be primary flaw
  - Longest life scenario
  - Large flaw on ligament fails rapidly
- Metallurgical results did not indicate which crack was first
- Standard 0.005” crack in secondary locations
- ~6,700 flight hours shown below

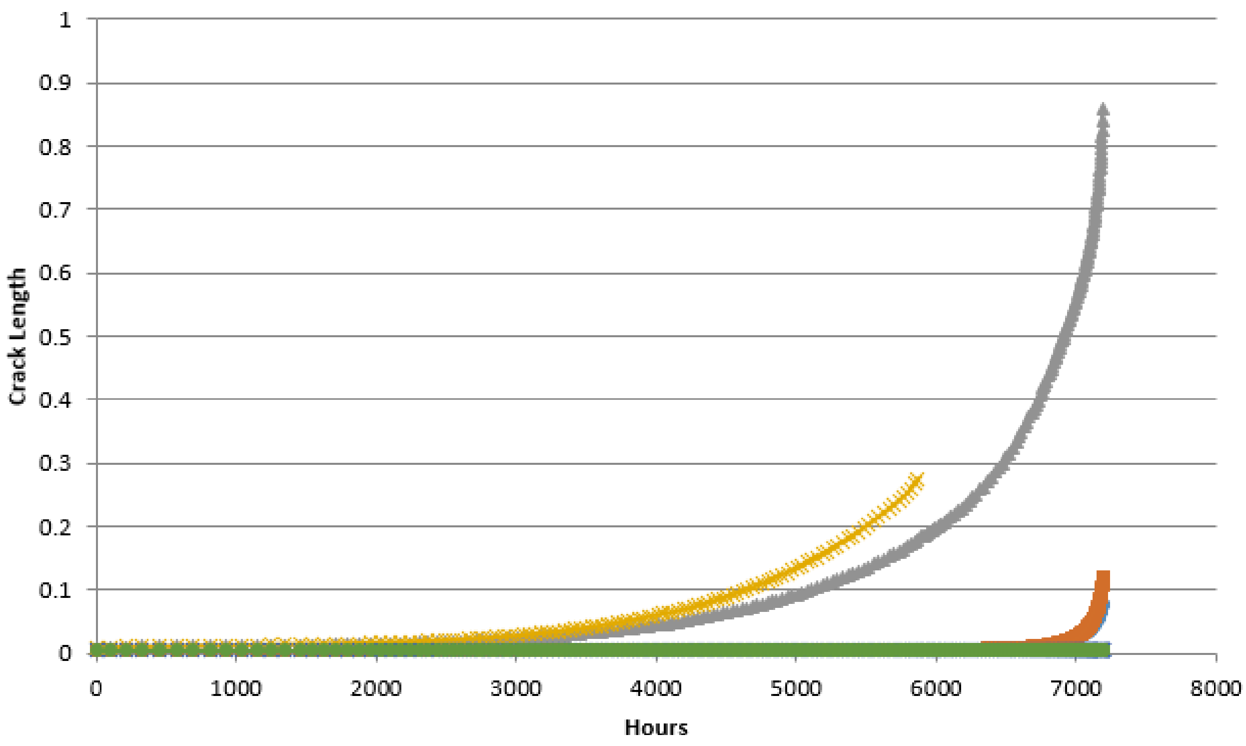




Durability Crack Analysis (Continued) (14 ksi peak)

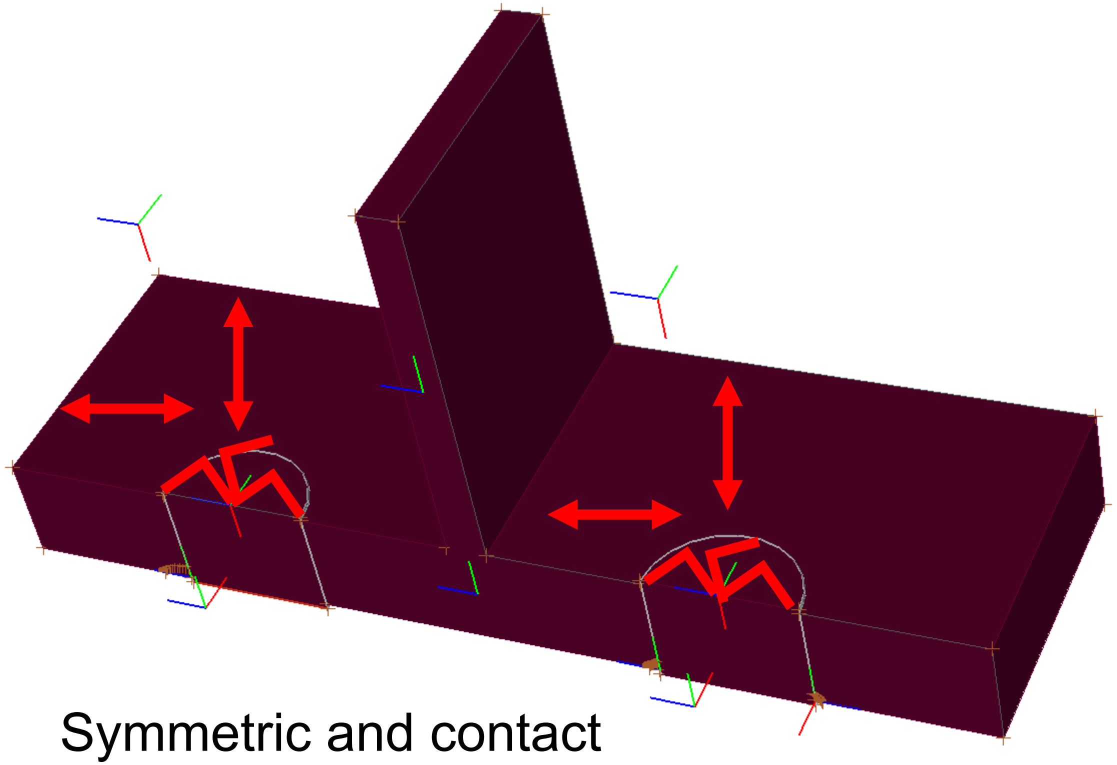
- Remaining center section of spar fails rapidly
- ~500 additional hours across remaining center spar section
- ~7,100 total flight hours



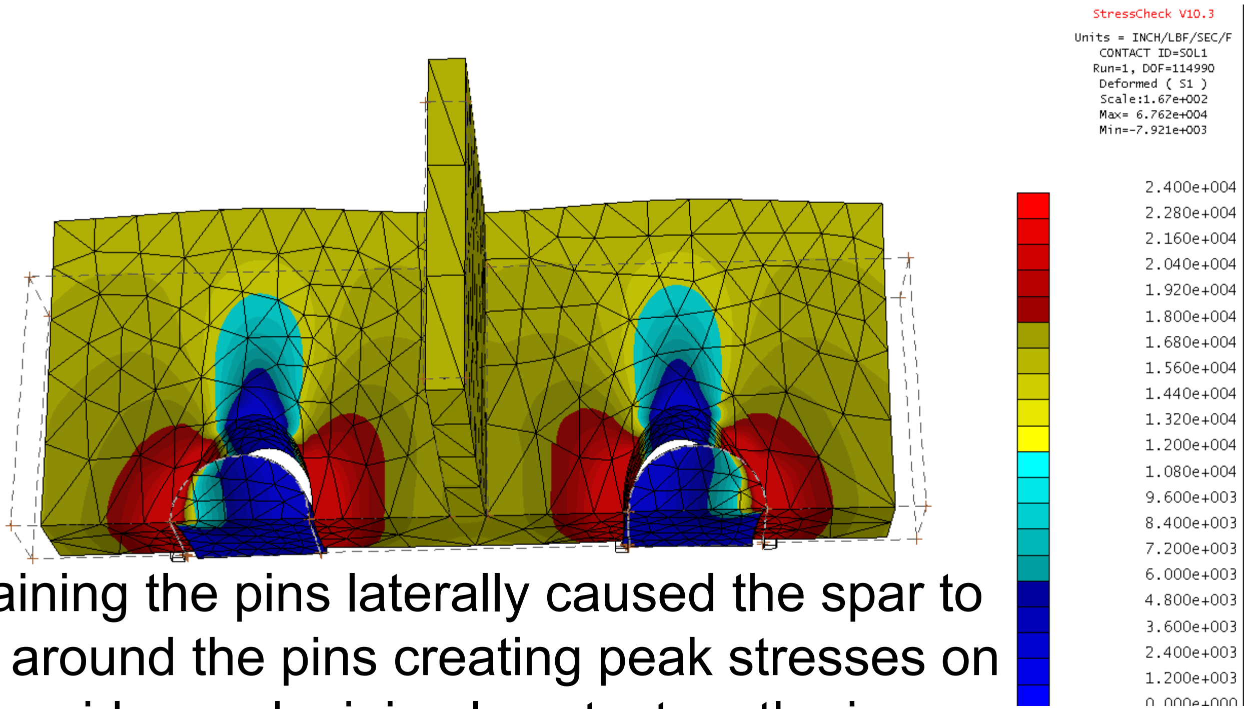


StressCheck Symmetric Pin Model

- This was the first analysis attempted using fastener contact in BAMF
- Initial pin constraint restricted translation on the upper face of the pin in two directions, up-down and fore-aft



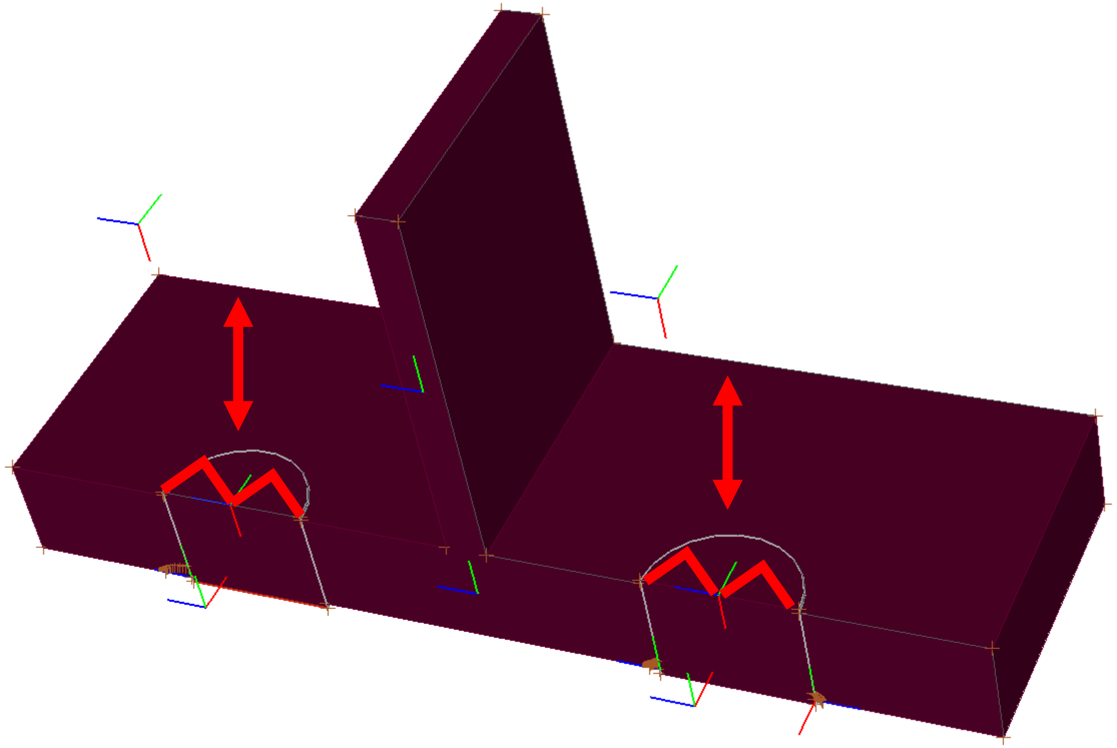
Symmetric and contact constraints not depicted

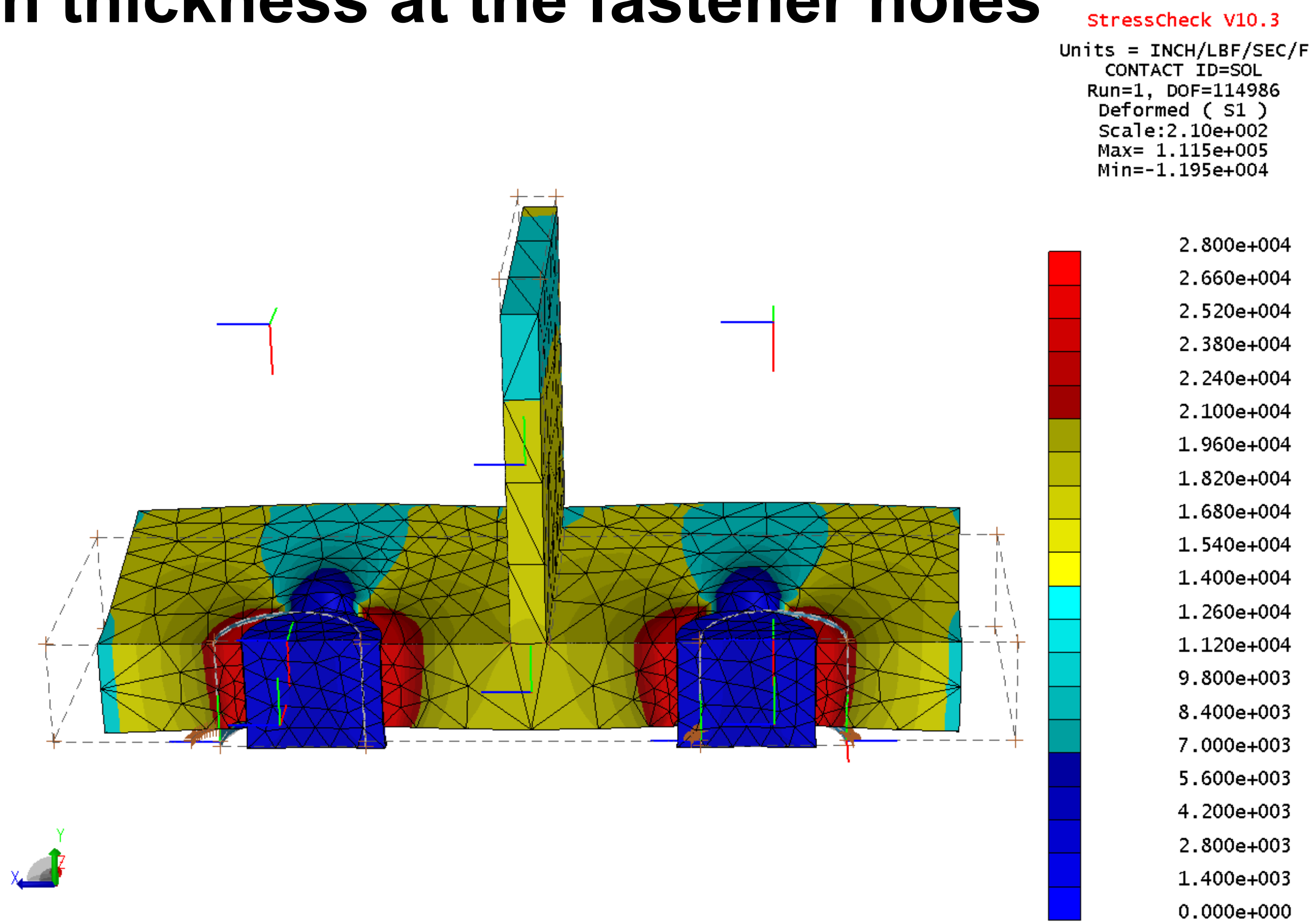


Constraining the pins laterally caused the spar to deform around the pins creating peak stresses on the outer sides and minimal contact on the inner. Furthermore, pin bending caused high stresses at the top of the pin with minimal contact at the bottom

StressCheck Symmetric Pin Final

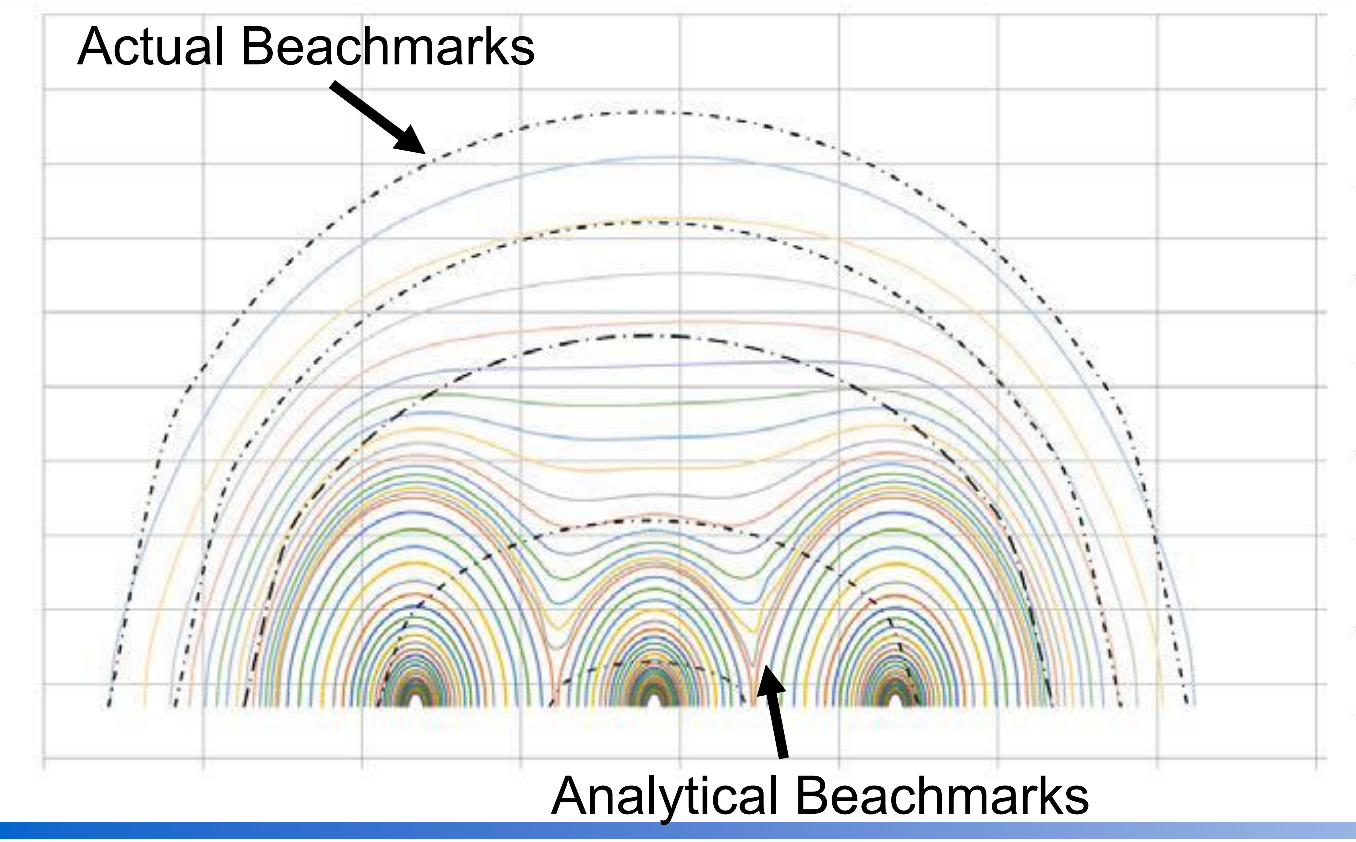
- Permitting the pin to translate laterally with spar deflection created the expected, uniform stresses through thickness at the fastener holes

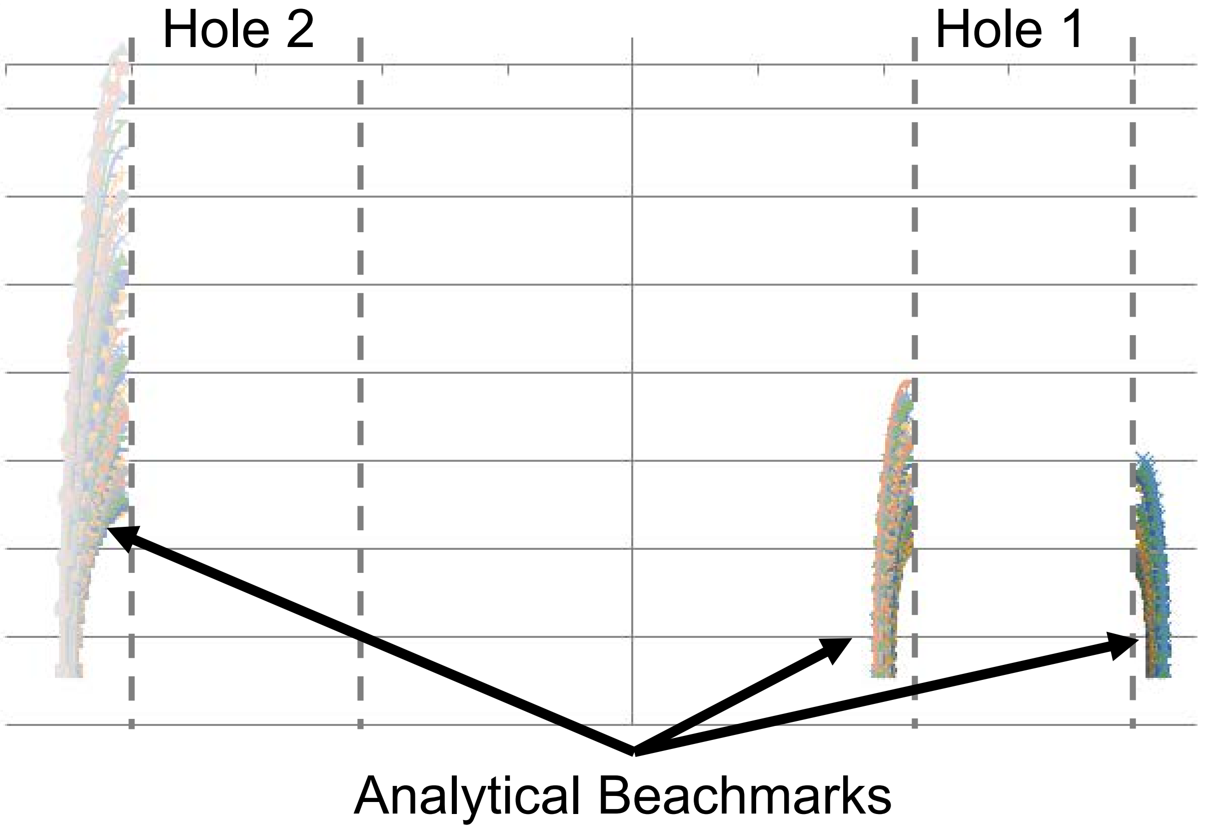




BAMF Multi-Crack Development (14 ksi peak)

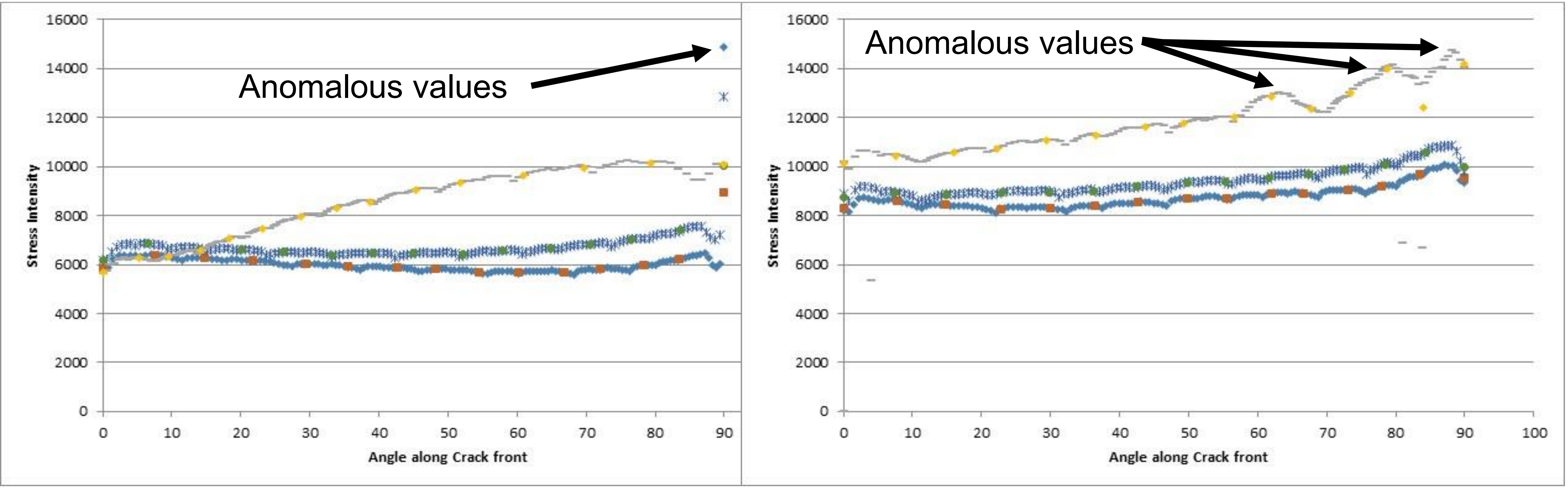
- Previous multi-crack analyses in BAMF used the same initial crack size for all cracks
- BAMF was enhanced to handle different crack sizes for each crack





BAMF K Extraction Enhancement

- Fastener contact creates a local analytical singularity at the contact face which creates anomalies with stress intensity extraction
- BAMF updated to omit anomalous K extraction from contact model

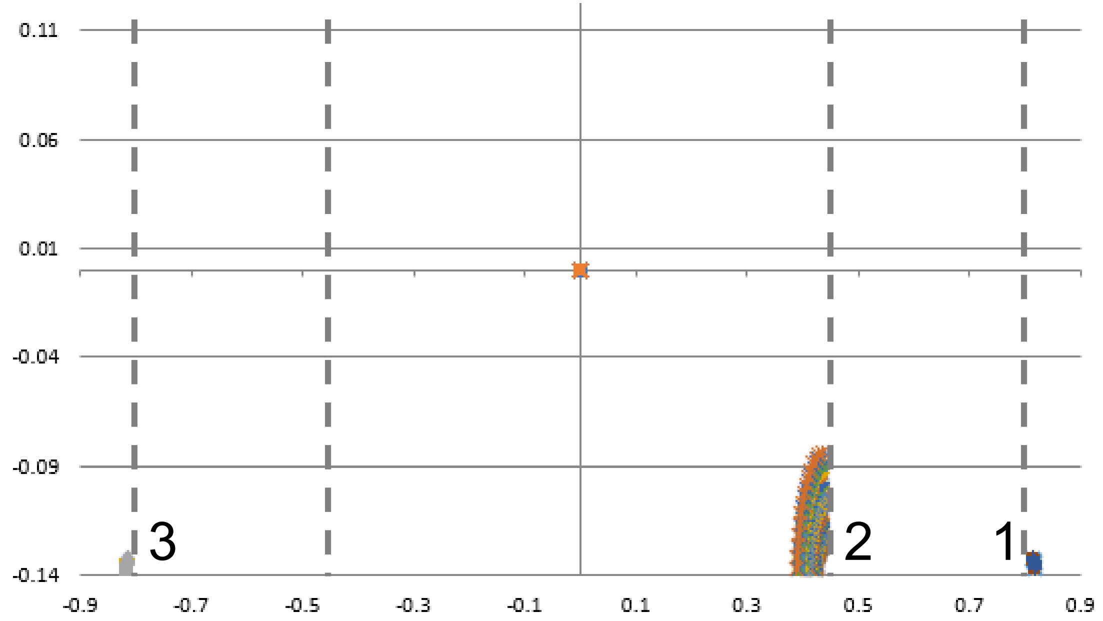


First iteration

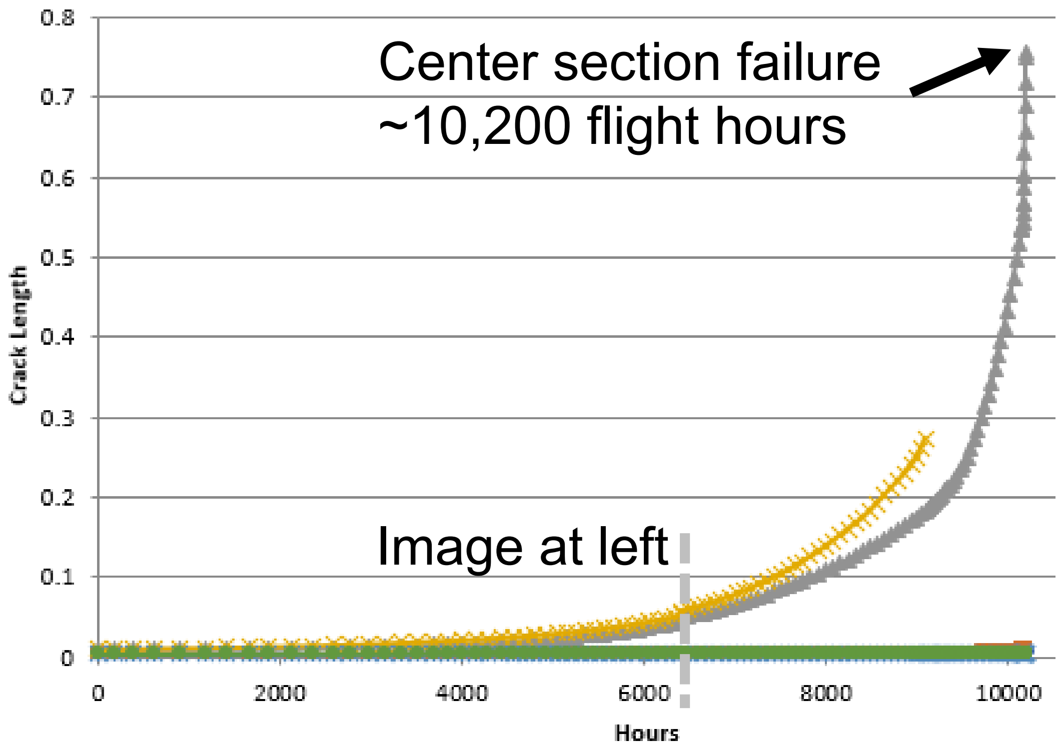
After a few iterations

Filled Hole Durability Analysis

- Filled holes can have significantly longer fatigue lives in tension dominated structure
- Steel pins were included in another model for comparison



~6,200 flight hours
Primary flaw .045” x .053” at this point
Open hole model had similar cracks at 3,900 hours



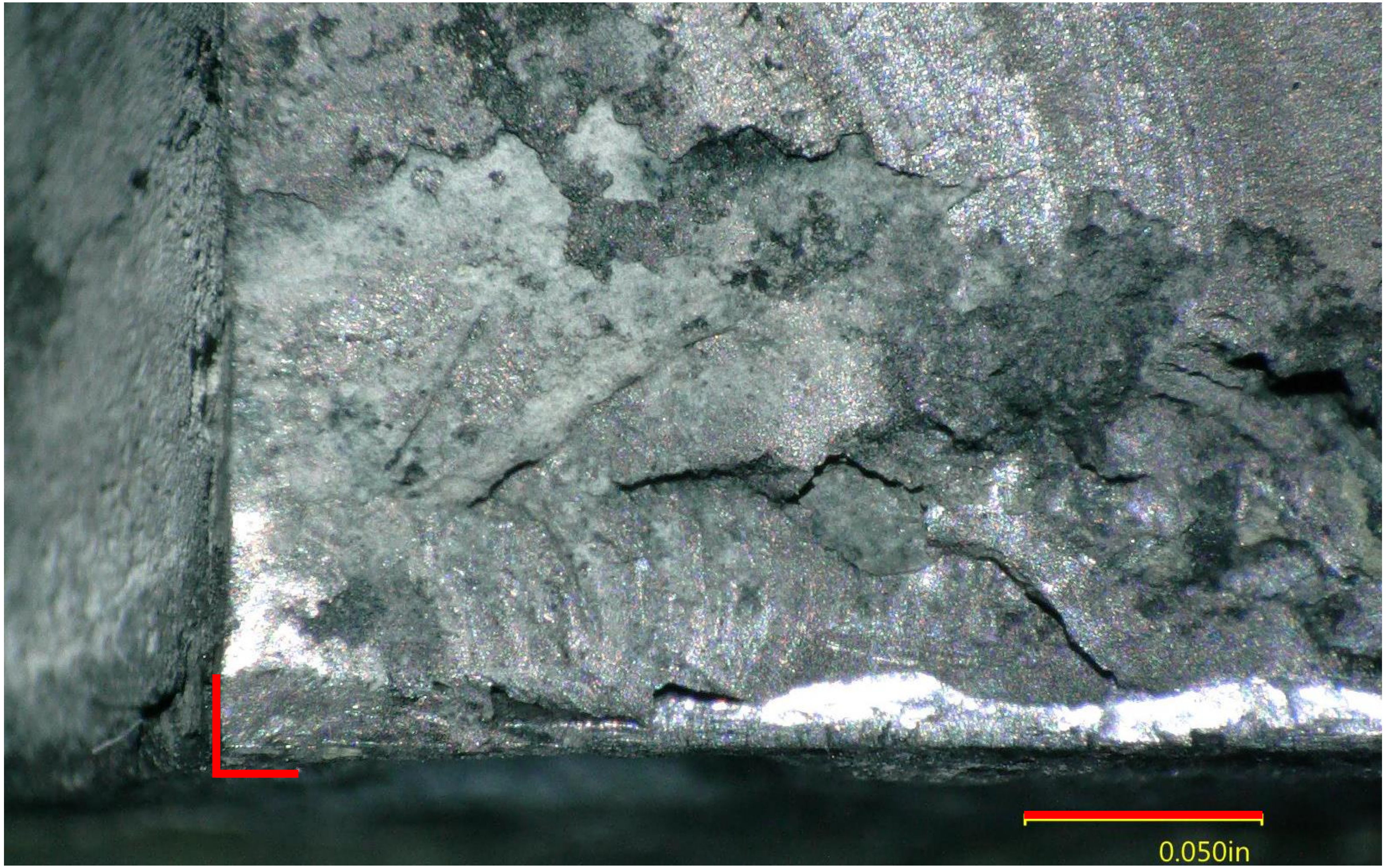
Life increase of ~1.5 with filled hole

Mishap Damage Sizes

- Fractography from mishap provides insight into initial damage state and initial crack sizes
- Images indicate initial damage may have been more severe than the durability case previously analyzed
- Additional set of models evaluated with crack sizes from fractography results
- Subsequent slides depict crack face images and assumed initial crack sizes from images

Crack 1

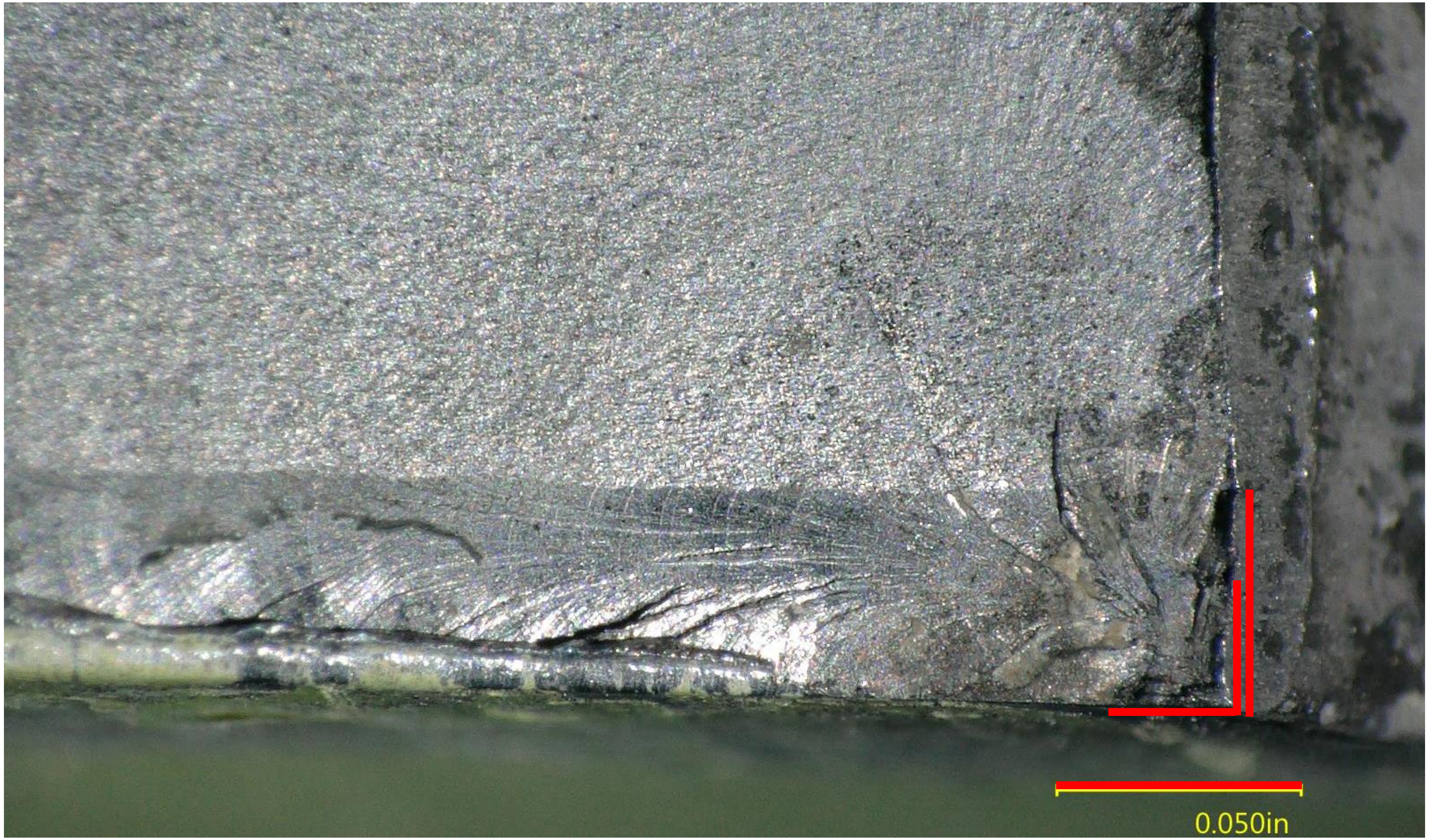
Surface length line: 0.6
Reference line: 1.73
Surface/Reference=x/0.05
a=0.6*0.05/1.73=**.017"**



Surface length line: 0.75
Reference line: 1.73
Surface/Reference=x/0.05
a=0.75*0.05/1.73=**.022"**

Crack 2

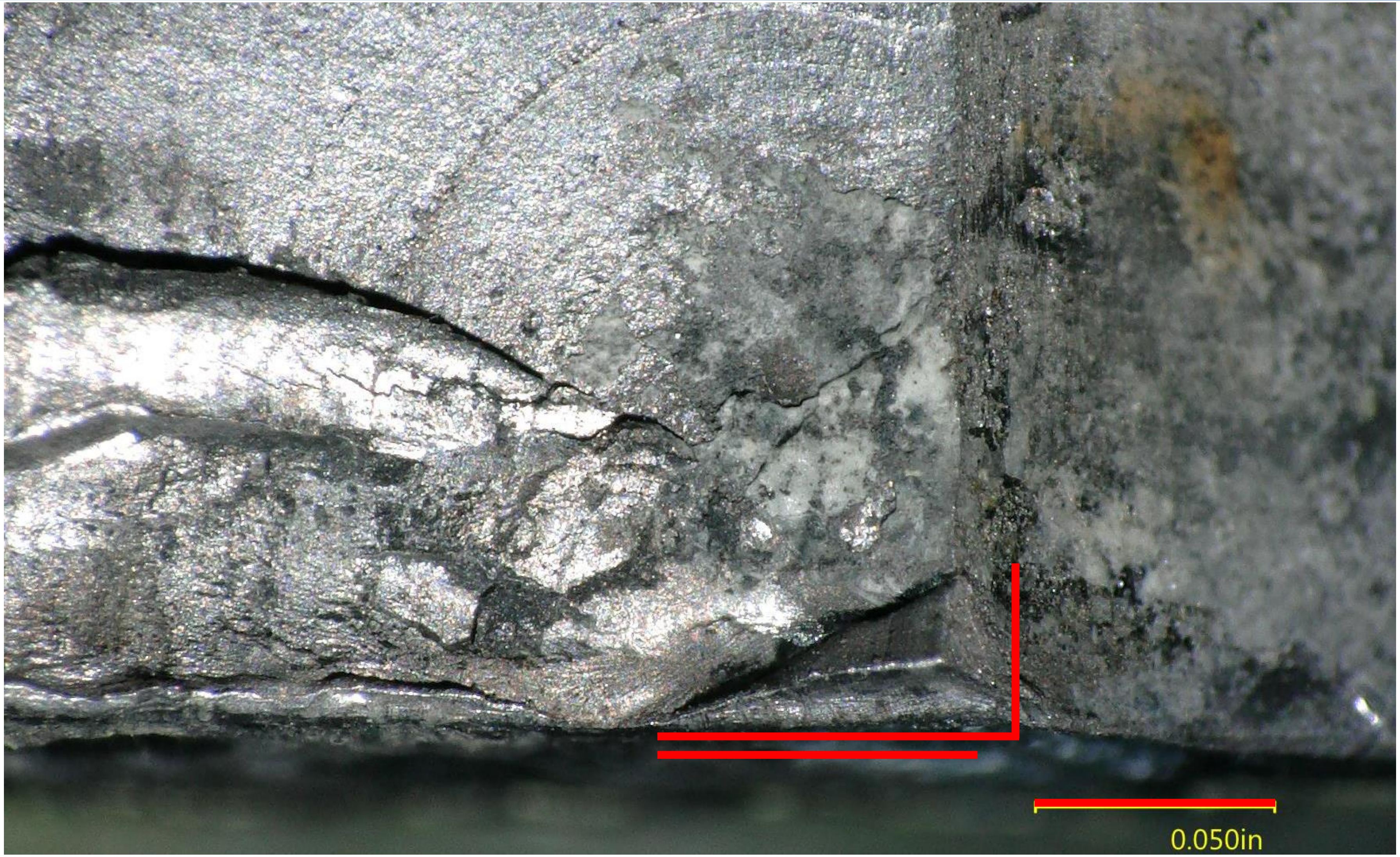
Surface length line: 0.9
Reference line: 1.73
Surface/Reference=c/0.05
c=0.9*0.05/1.73=**.026”**



Surface length line: 0.95
Reference line: 1.73
Surface/Reference=a/0.05
a=0.95*0.05/1.73=**.027”**

Crack 3

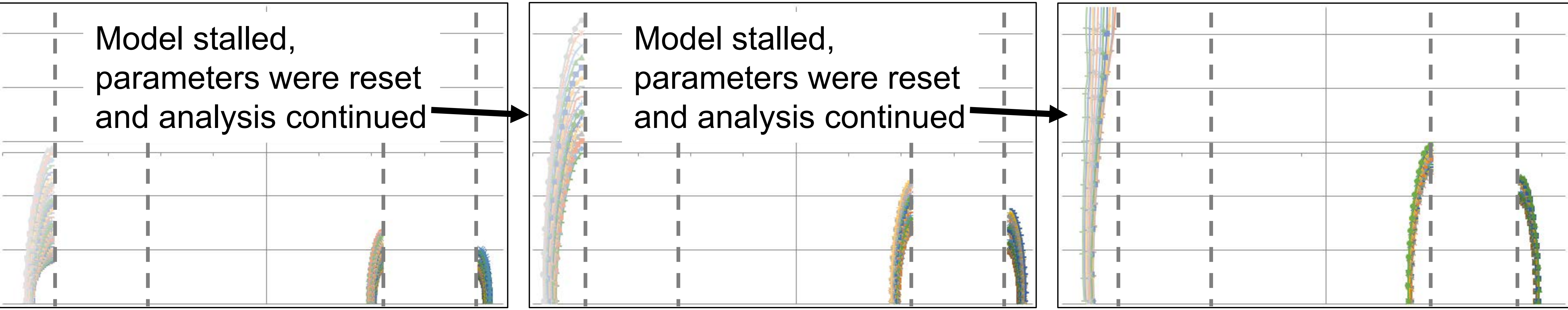
Surface length line: 2.6
Reference line: 1.73
Surface/Reference=c/0.05
c=2.6*0.05/1.73=**.075”**



Surface length line: 1.25
Reference line: 1.73
Surface/Reference=a/0.05
a=1.25*0.05/1.73=**.036”**

Mishap Damage Size Analysis
Open Hole

- Durability analysis showed minimal growth on secondary cracks
- This analysis shows significant growth for all cracks
- All cracks growing rapidly with minimal ligament remaining on dominant crack, estimate less than 1,000 hours remaining life
- Initial damage size has significant impact on fatigue life



~1,200 flight hours

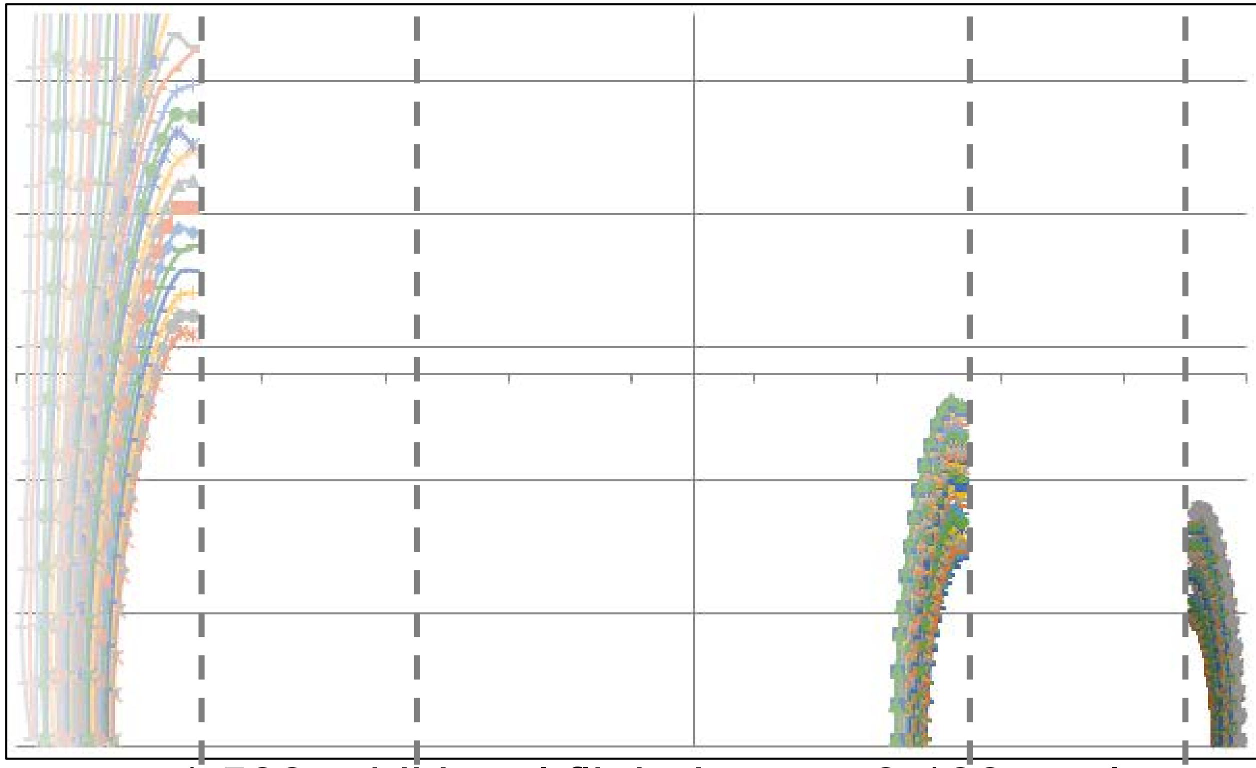
~700 additional flight hours
~1,900 total

~350 additional flight hours
~2,300 total

Mishap Damage Size Analysis
Filled Hole

- **Filled hole again showed a life improvement of roughly 1.5**
  - **Related observation in reduction of K for filled hole with small cracks**
    - **Bombardier, Y, Renaud, G, Li, G, “Prediction of Fatigue Crack Growth at Cold Expanded Fastener Holes with ForceMate Bushings,” AFGROW Workshop 2018, pages 49-50.**

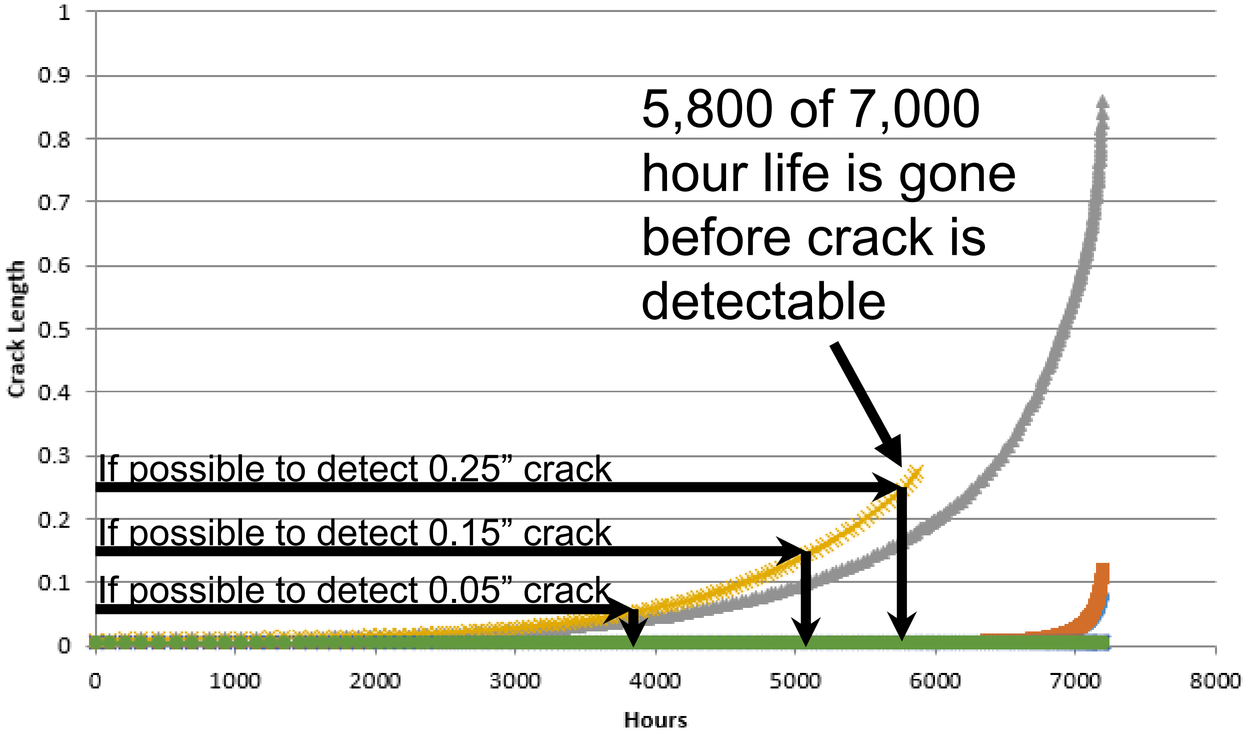
~1,700 flight hours



~1,500 additional flight hours ~3,100 total

Damage Tolerance Discussion

- The models herein only address the crack growth portion of the structural life, there is some life prior to detectable crack propagation
- Damage tolerance analysis is based on inspection capability
- Depending on NDI approach cracks of differing sizes can be detected
- Since most of the fatigue life is prior to a crack being detectable, caution must be used to determine appropriate actions necessary for continued safe operation
- Note, sister aircraft crack appeared to be ~0.18", indicating it was less than 2,000 hours from failure

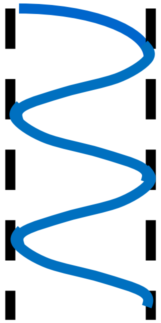


NDI Tools

- Three common NDI tools for bolt hole inspection
  - Manual right-angle probe
  - Manual bolt hole probe
  - Semi-automated bolt hole probe
- Eddy current signal response is highly sensitive to probe distance from material being inspected (lift off)
- Semi-automated bolt hole probes
  - Help ensure consistent contact
  - Improve full coverage during indexing

Right-angle probe has potential to lose contact with hole during inspection

Bolt hole probe ensures consistent contact during inspection



Manual bolt hole has potential to index prior to completing probe rotation, missing portions of the hole

NDI Approach

- USAF has not accepted manual bolt hole eddy current inspections for years due to:
  1. No established Probability of Detection (POD)
  2. Greatly reduced sensitivity
  3. Human Factor Variables
  4. Manual Probe Rotation Speed vs. current High Speed Scanning (>1200 RPM)
  5. Manual inspector depth increment adjustment.

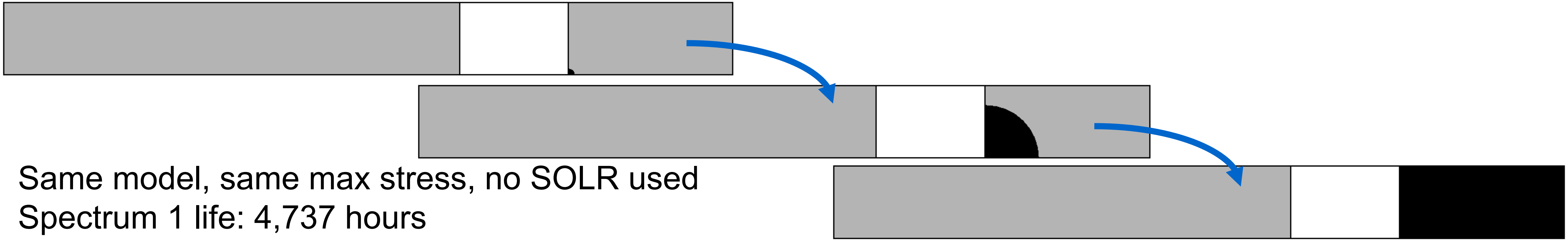
- USAF PoD data indicates 0.050” detectable flaw size for semi-automated bolt hole eddy current*

- USAF would assume 0.25” detectable flaw for manual bolt hole eddy current*

* USAF EN-SB-08-012

Flight Data Recorders

- Multiple load spectra were briefly compared to demonstrate prediction sensitivity
  - Generic corner crack model used for comparison
- Without flight data recorder multiple assumptions are necessary to generate a loading spectrum
- To accurately predict crack growth life, a loading spectrum based on recorded flight data is essential

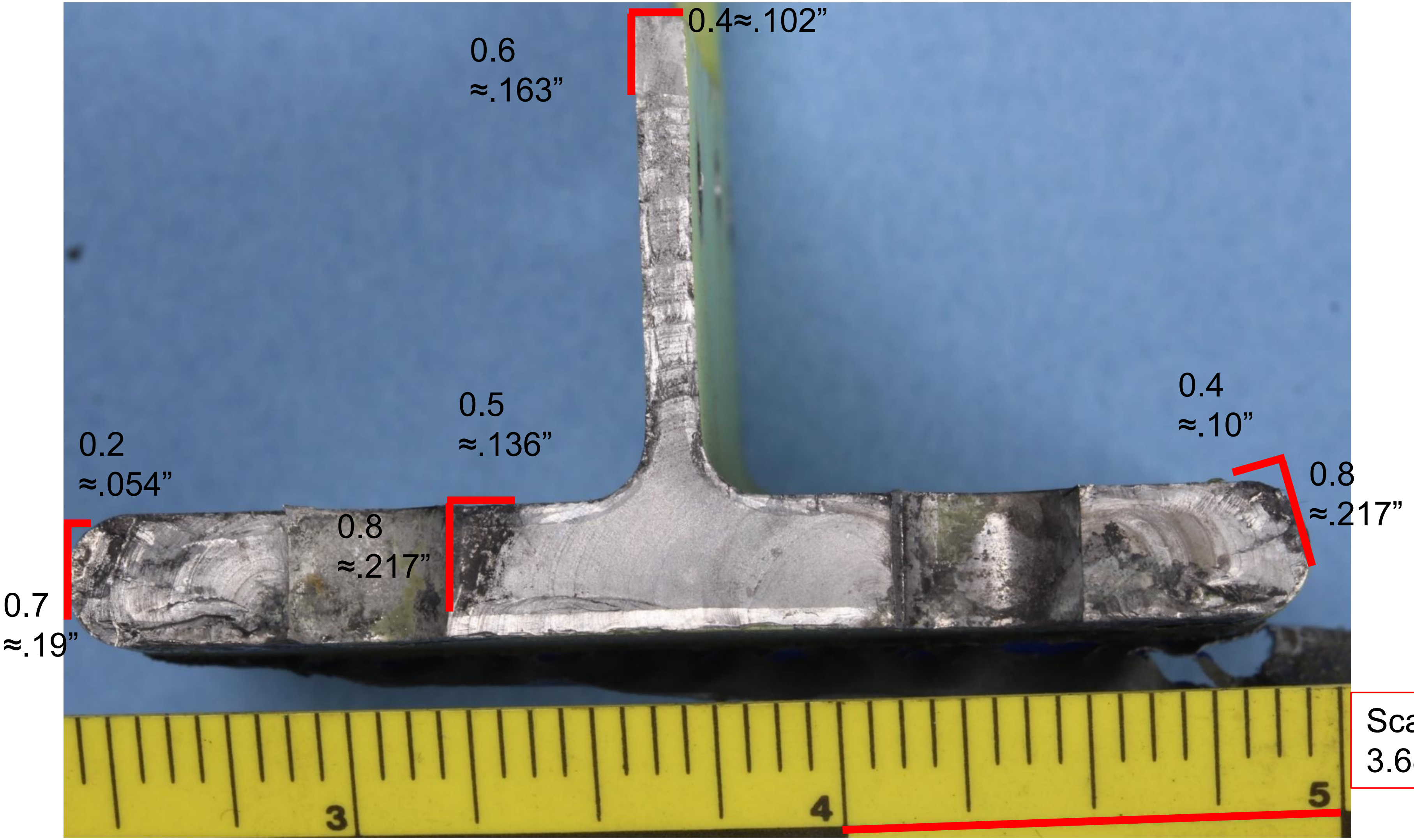


Same model, same max stress, no SOLR used
Spectrum 1 life: 4,737 hours
Spectrum 2 life: 37,967 hours
Falstaff life: 26,555 hours

Residual Strength Capability

Remaining Area:
.102” x .163” +
.054” x 0.19” +
.108” x .217” +
.136” x .217”/2 =
.065 in^2

Remaining load capability:
A=1.907 in^2
P=Sigma*A =
44 ksi*.065in^2=
2.86 kip



Scale:
3.68 = 1”

Load Redistribution

Remaining load capability of only 2.86 kip implies load re-distributed to other structure.

Image of angles confirms load did redistribute and grow cracks in secondary structure

Discussion

- Crack growth from a hole is extremely sensitive to subtle differences in surface finish, hole fill, residual stress state, etc.
  - Tightly controlled lab specimens can display significant scatter with small variations in manufacturing and fastener installation conditions
  - Hole cold expansion is a common example of this, though neat and interference fit fasteners can have a similar effect
- Damage tolerance analyses must consider the shortest reasonable life
  - Numerous inspections can result in no indications even though the analysis for the inspection is sound
- The analyses documented herein are based on a peak spectrum stress of 14 ksi suggesting no abnormal aircraft operations prior to failure
- Inspections without crack findings alone is insufficient to conclude fatigue damage does not exist
  - Analysis limitations and compounding factors should be considered

Thank you