A-10 Nose Landing Gear (NLG) Door Hinge Failure Assessment

Mike Worley
Southwest Research Institute

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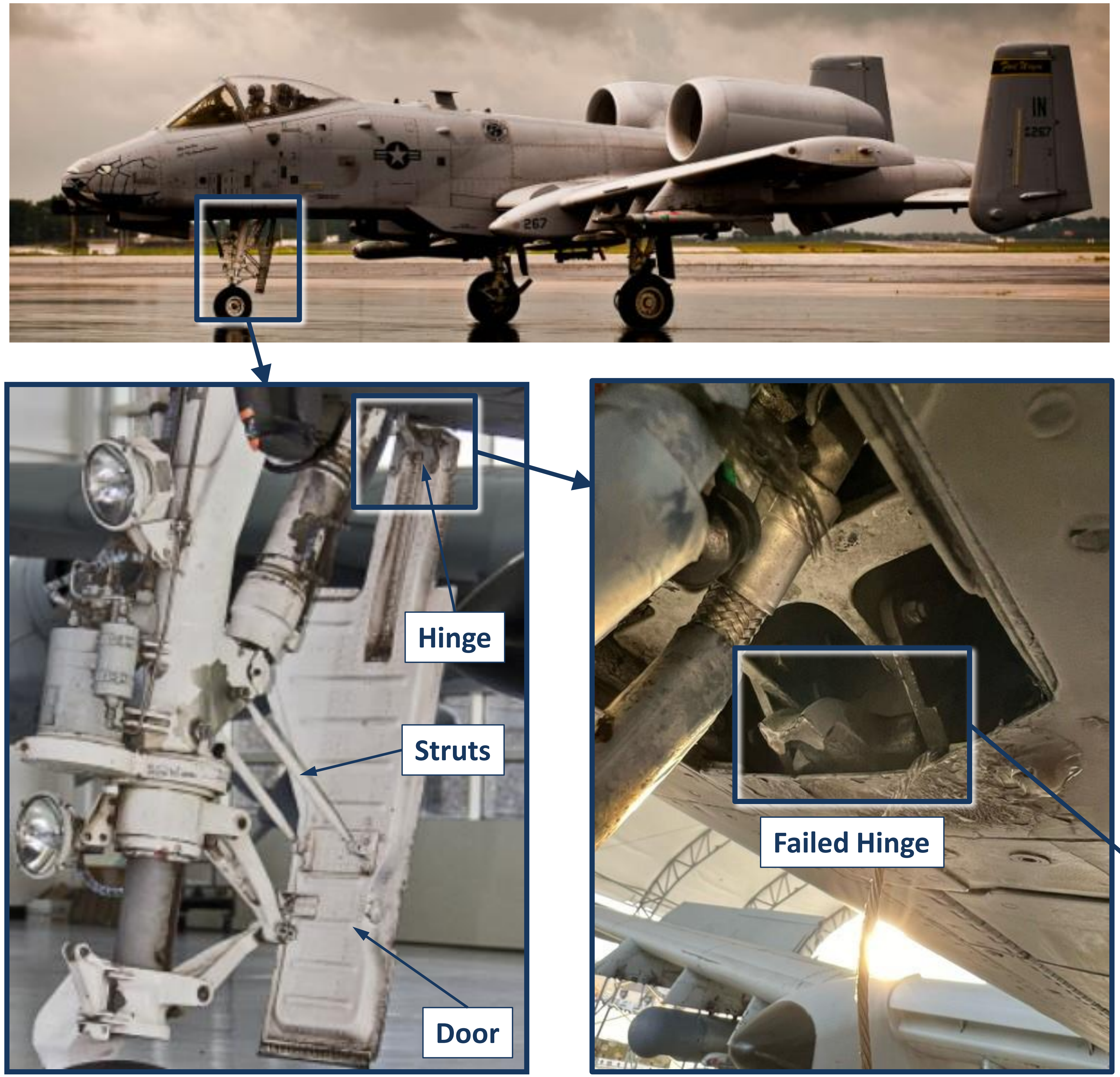
- Evan Ross and Cody Hone (HAFB Materials Lab)

Background

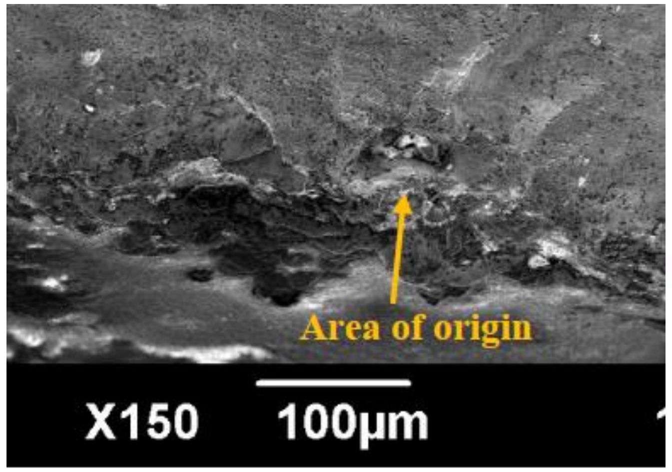


- Late 2023, the nose landing gear (NLG) aft door departed the aircraft in-flight during routine training sortie operating inside normal operating envelope of the A-10
- The door, struts, and forward half of the hinge were lost but the aft portion of the hinge remained and was available for evaluation
- The hinge is a 4340 steel forging

Fracture Face Findings

- **The side of bracket labeled “Right” (but actually LHS on aircraft) exhibited significant corrosion damage indicating it cracked first and had been open to elements for some time.**

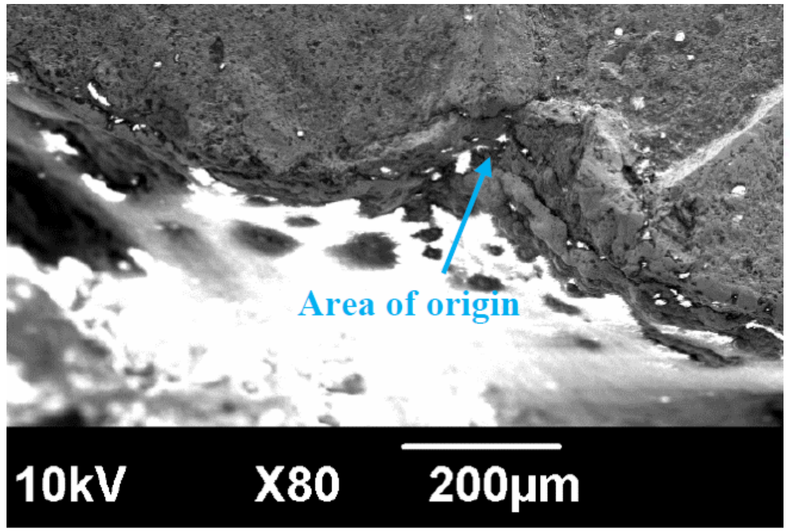
  The crack originated from a localized surface defect roughly 0.010” in size.



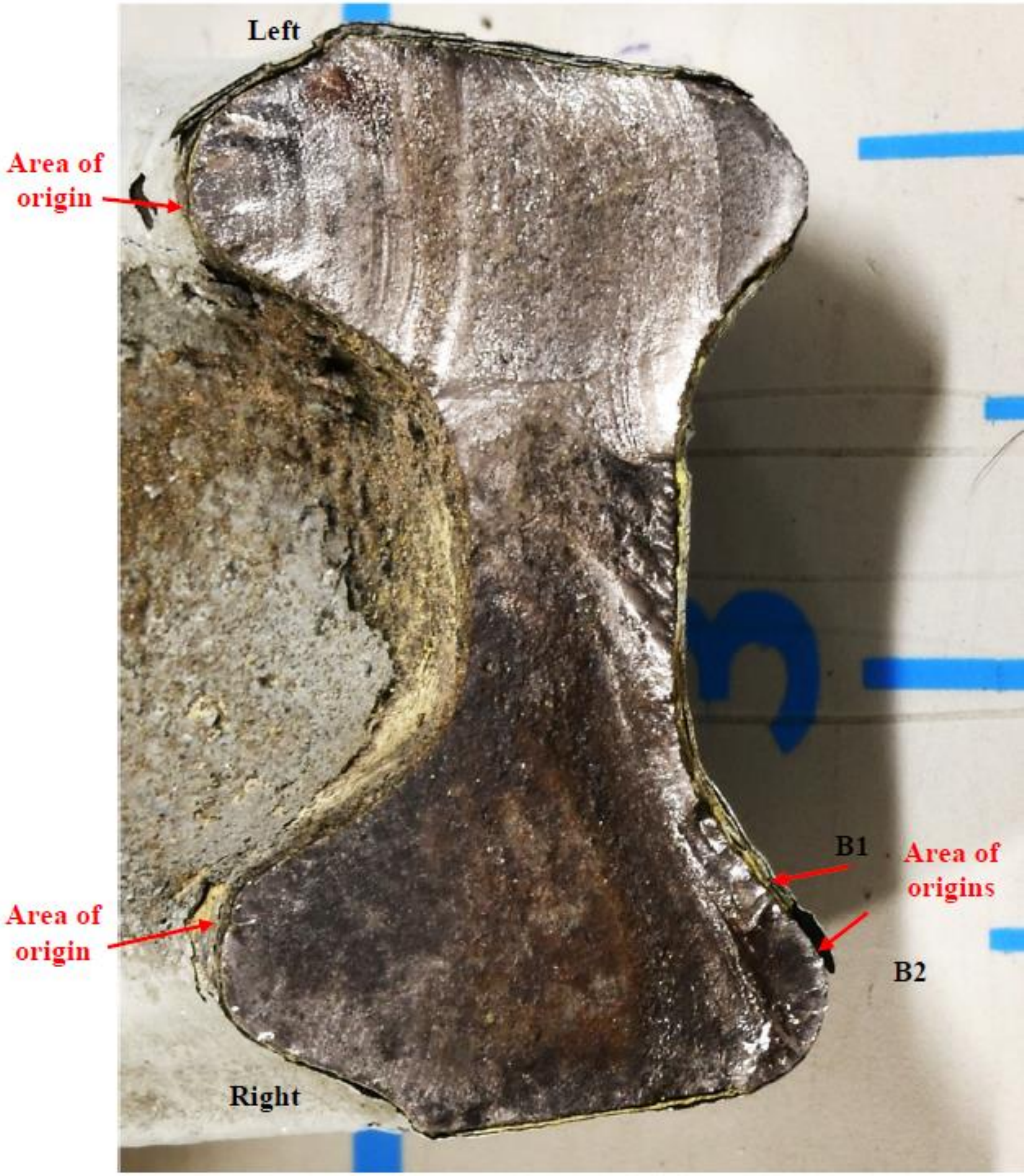
Also observed on the “Right” side top was reversed direction cracking, originating at locations labeled “B1” and “B2”, indicating crack growth due to reversed bending during service.

- **The side of bracket labeled “Left” (but actually RHS on aircraft) exhibited minor corrosion damage indicating it cracked more recently.**

  The crack originated from a localized surface defect roughly 0.005” in size.



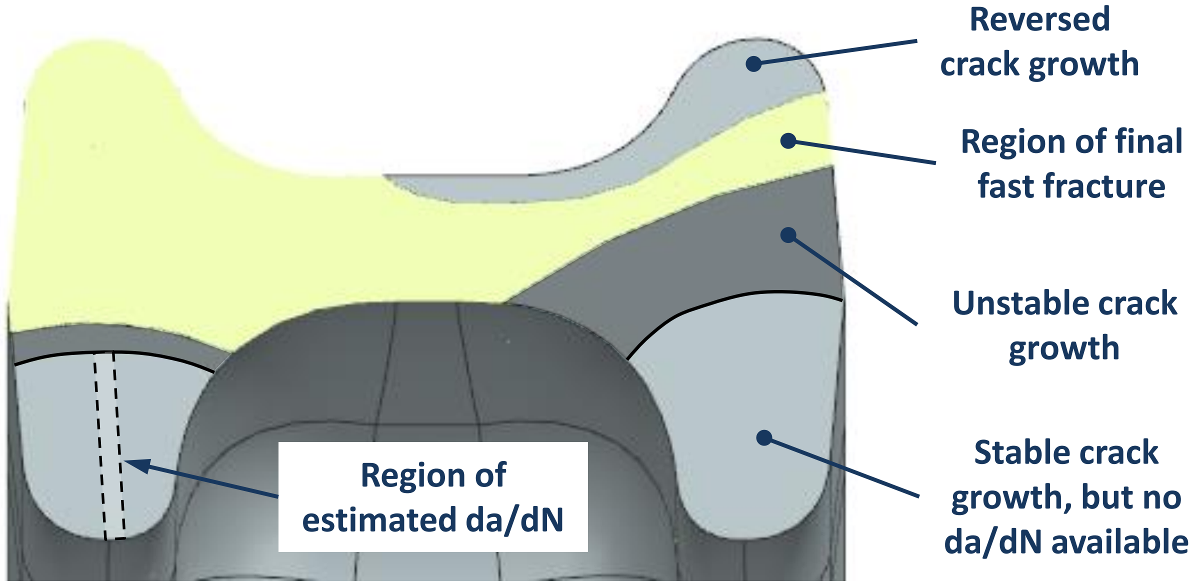
(Labeled “Left”, but is RHS *on aircraft*)



(Labeled “Right”, but is LHS *on aircraft*)

Fracture Face Findings

- The RHS of bracket was too corroded to detect any striations
- Zones of dimpled rupture indicative of fast fracture were visible and identified
- Crack growth rates for the LHS of bracket were estimated using visible striation-like features at six locations along propagation direction
- The figure at right provides a crude map of fracture surface regions



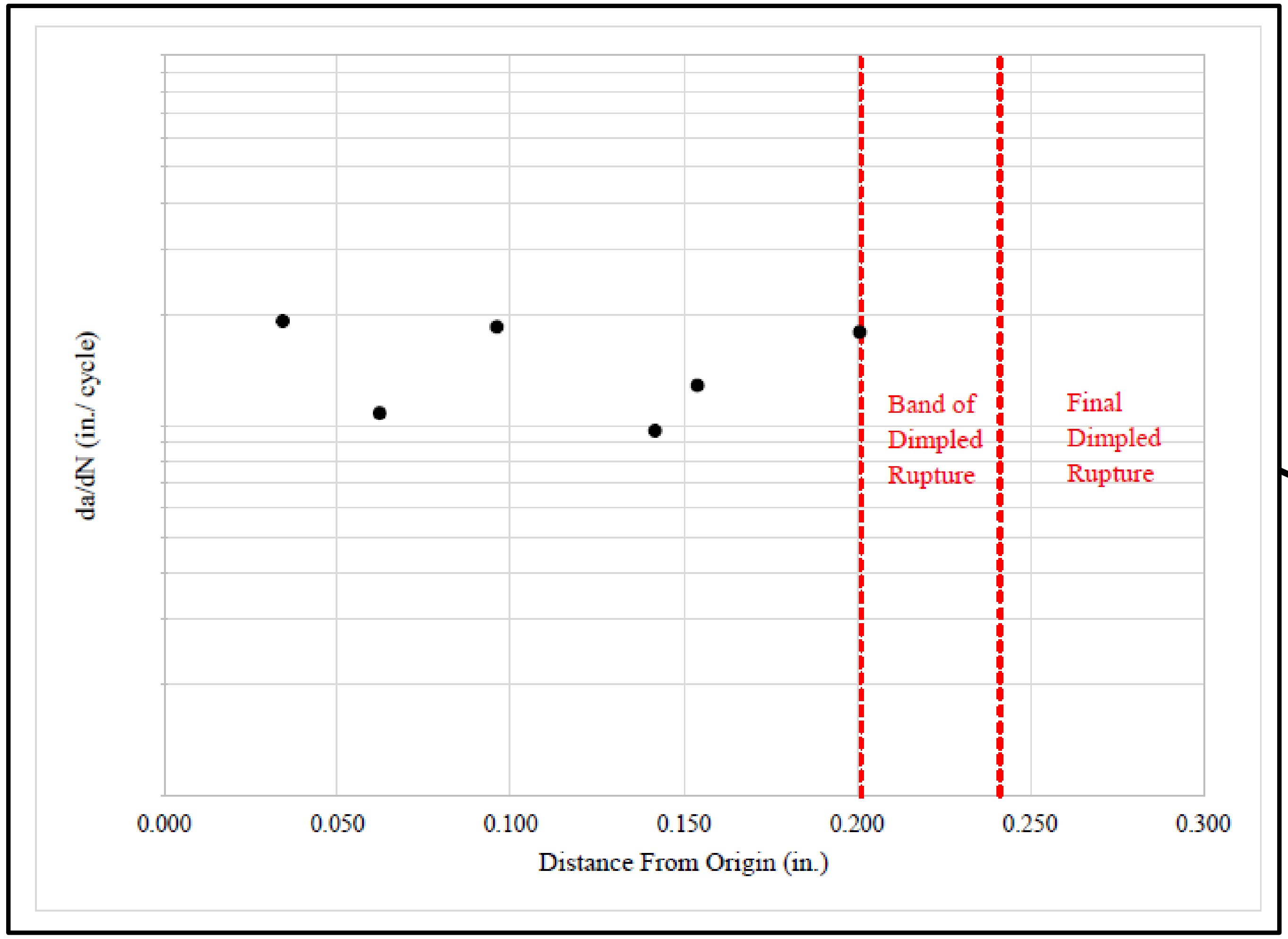
Reversed crack growth

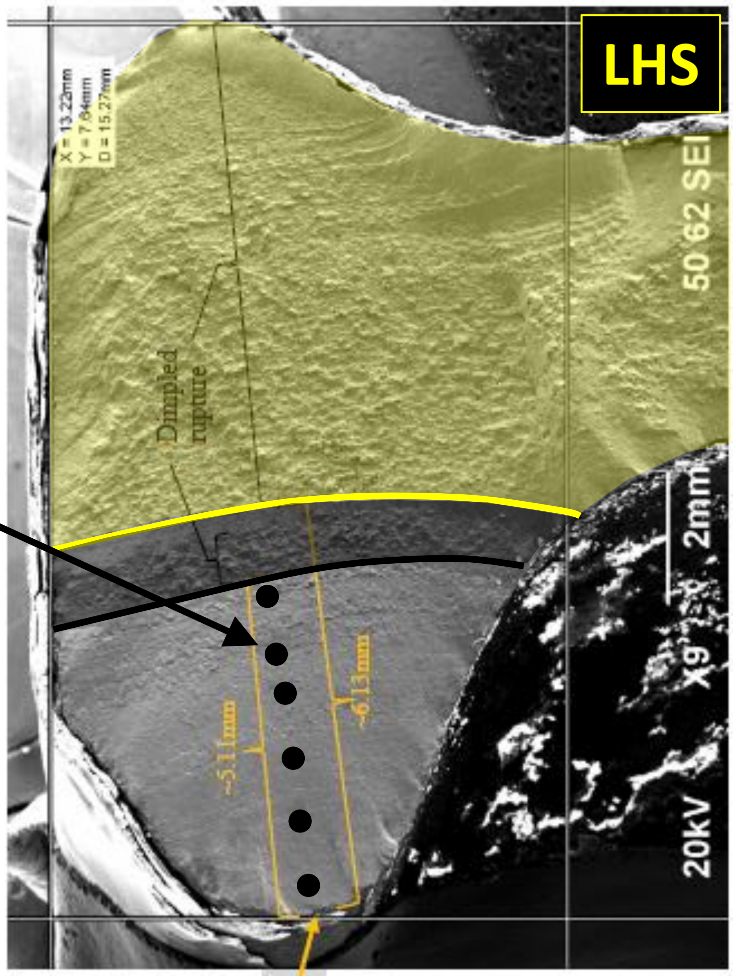
Region of final fast fracture

Unstable crack growth

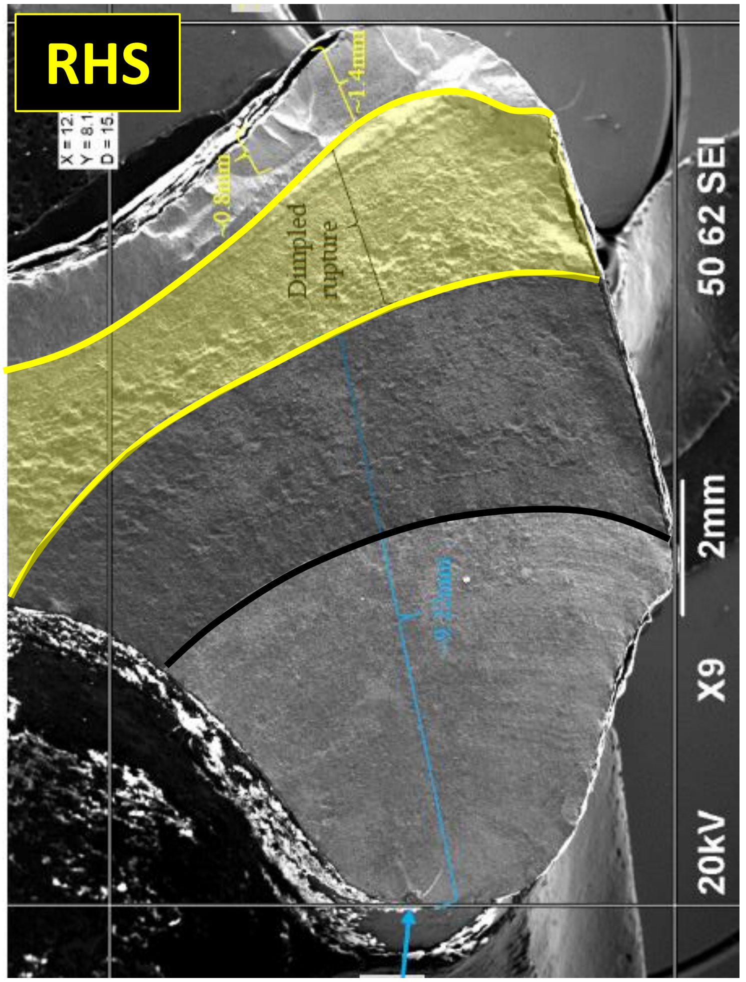
Stable crack growth, but no da/dN available

Region of estimated da/dN





(RHS *on aircraft*)



(LHS *on aircraft*)

Initial Questions and Puzzles

- What are the loads on the bracket? Are load estimates from legacy A-10 analyses appropriate?

- Why is nearly half of the bracket corroded?!
  - What is the history of this bracket?
    - Did it sit in storage for some time?
    - Do we know how many flight hours this bracket has flown?
  - Can loads and stress analysis confirm continued operation up to the observed loss of section?

- Why are measured da/dN nearly constant right up to final failure?

- What inspection interval is required to find similar cracking in the future?

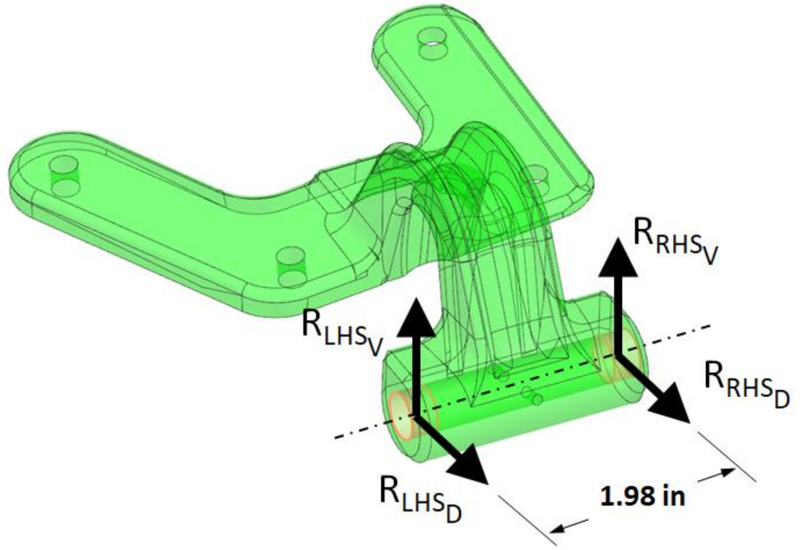
Part History

- Failed hinge was installed as a replacement part in Summer 2006
- No record if this was a CANNd (cannibalized) part from storage
  - However, the timeframe and maintenance records for similar hinge replacements on 3 other aircraft at same base during same time indicate that it most likely was a CANNd part
- No history available on hinge prior to 2006
  - Did the part sit in storage? For how long? Does this explain the corrosion?
- Unknown how many total flight hours the hinge has flown
  - Crack growth analysis will be based on cycles but need conversion to flt-hrs to set time-based inspection criteria. Actual spectrum is unknown.

➢ Decision made to assume fleet average flight hours up to date of mishap for conversion from cycles to hours

Initial Loads Investigation

- Two fatigue loading conditions, documented in the original aircraft design analyses, were scaled to limit loads and applied to a finite element model (FEM) of the hinge
  - COND #1: In-flight pressure on the door tries to open the “gooseneck”
  - COND #2: Air drag on door during landing tries to close the “gooseneck”
- Peak stress areas in FEA aligned with assumption that corroded side of hinge had higher stresses and cracked first

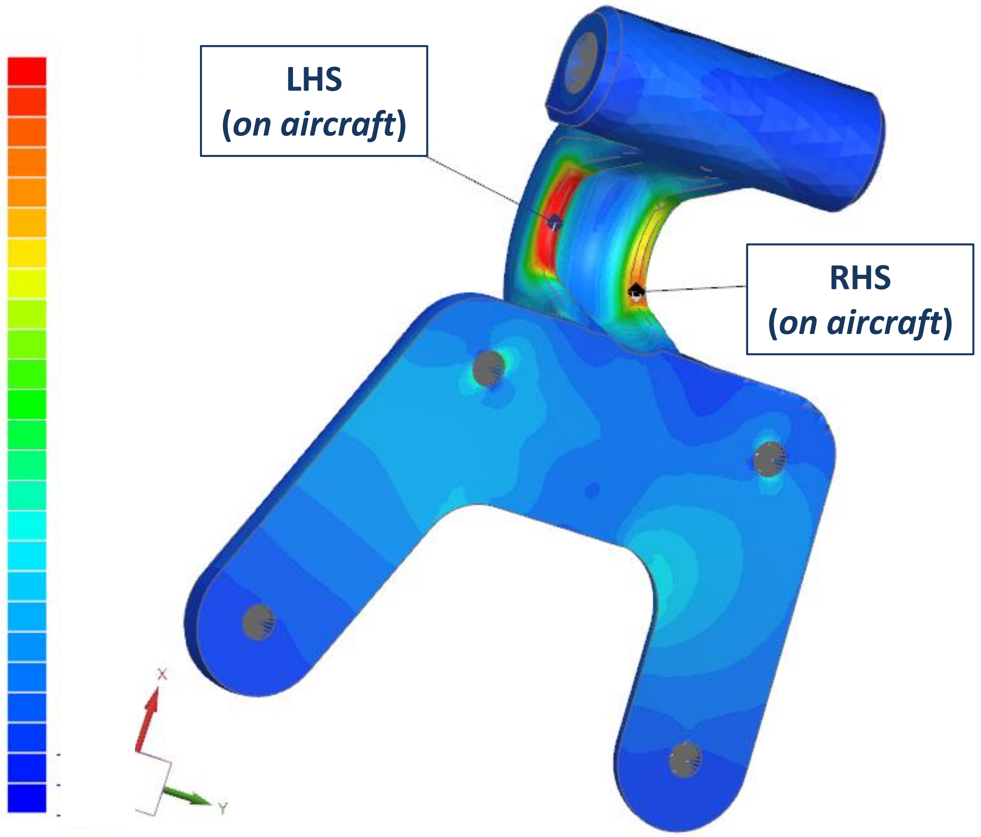


Hinge *Limit* Loads to Apply to FEM

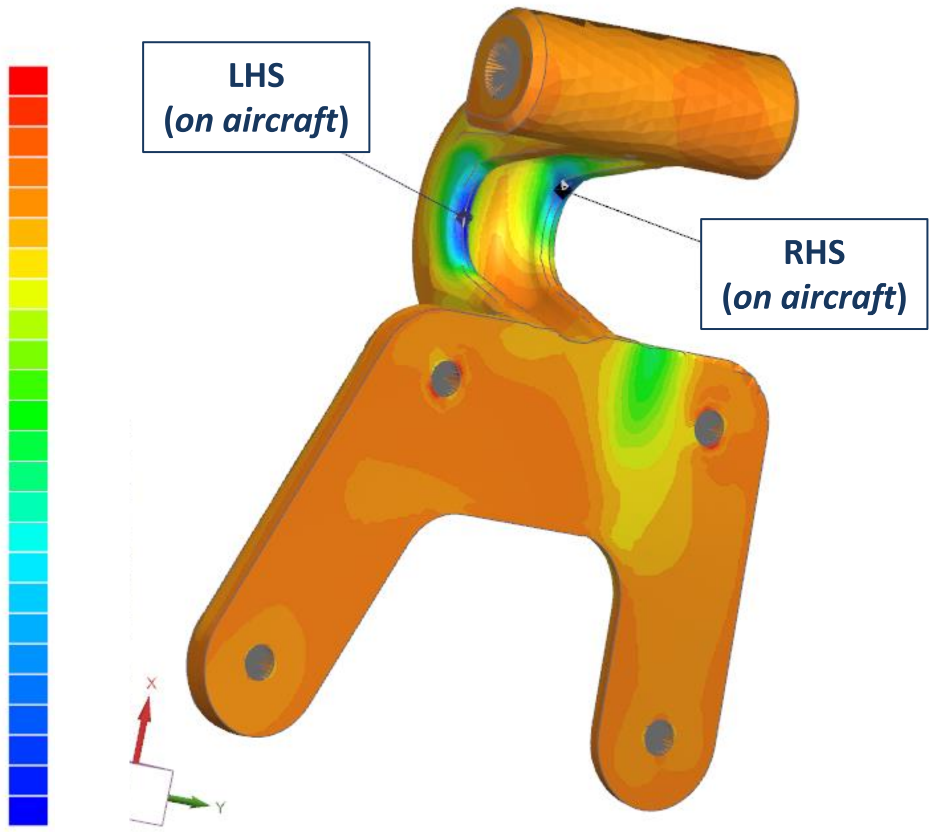
Hinge RXN Spread = 1.98

		COND #1	COND #2
RHS Vertical Reaction	R_{RHS_V}	155.4	104.8
RHS Drag Reaction	R_{RHS_D}	72.5	-34.6
LHS Vertical Reaction	R_{LHS_V}	16.0	-17.3
LHS Drag Reaction	R_{LHS_D}	698.8	-147.4

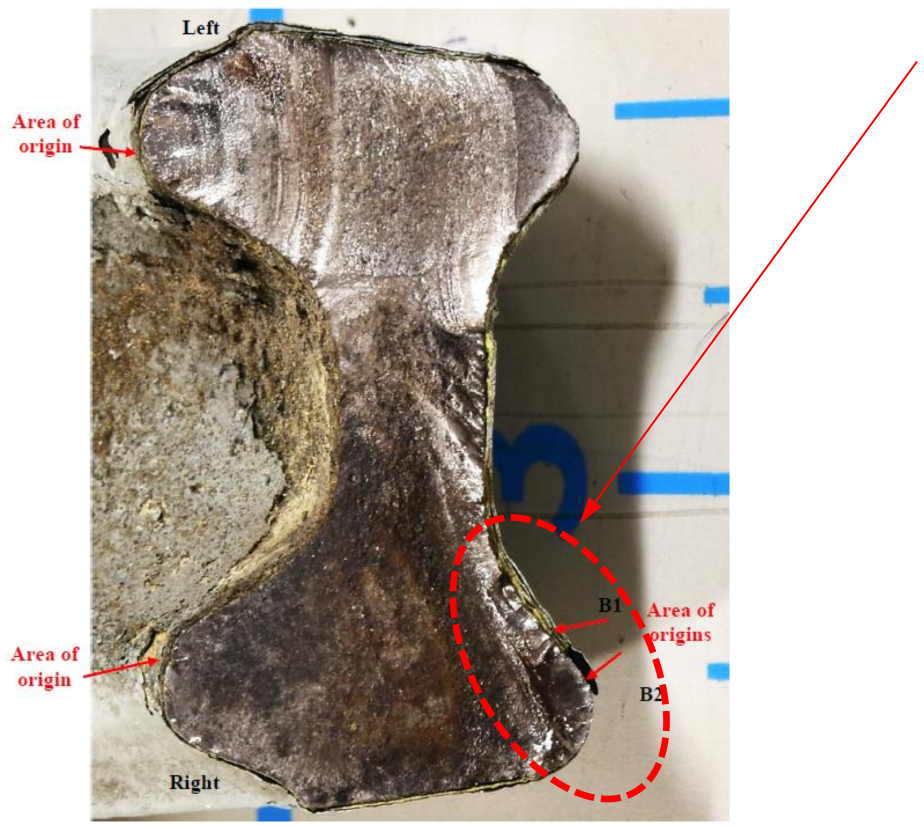
COND #1



COND #2

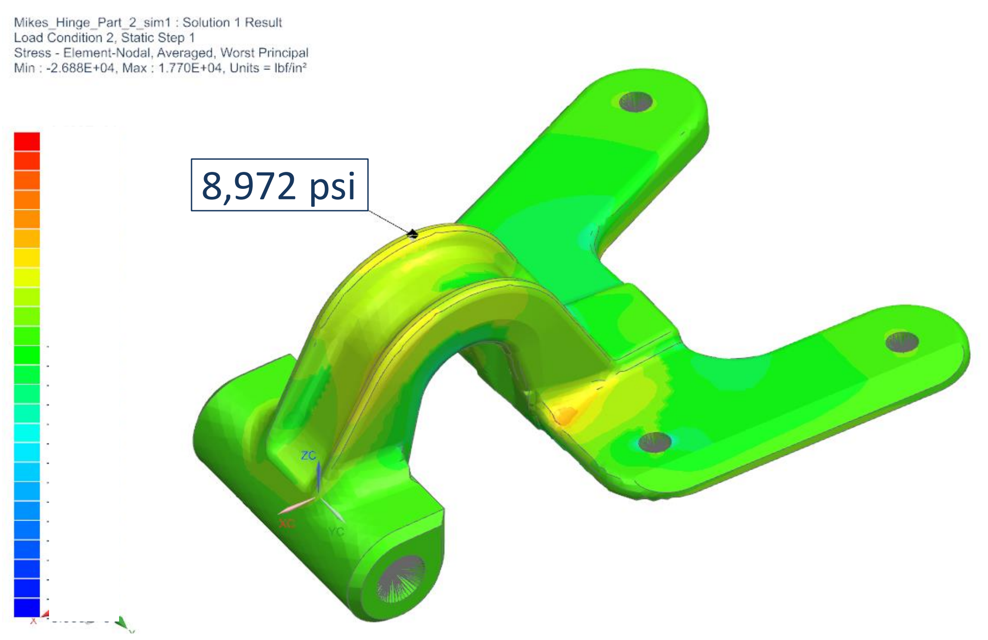


Initial Loads Investigation



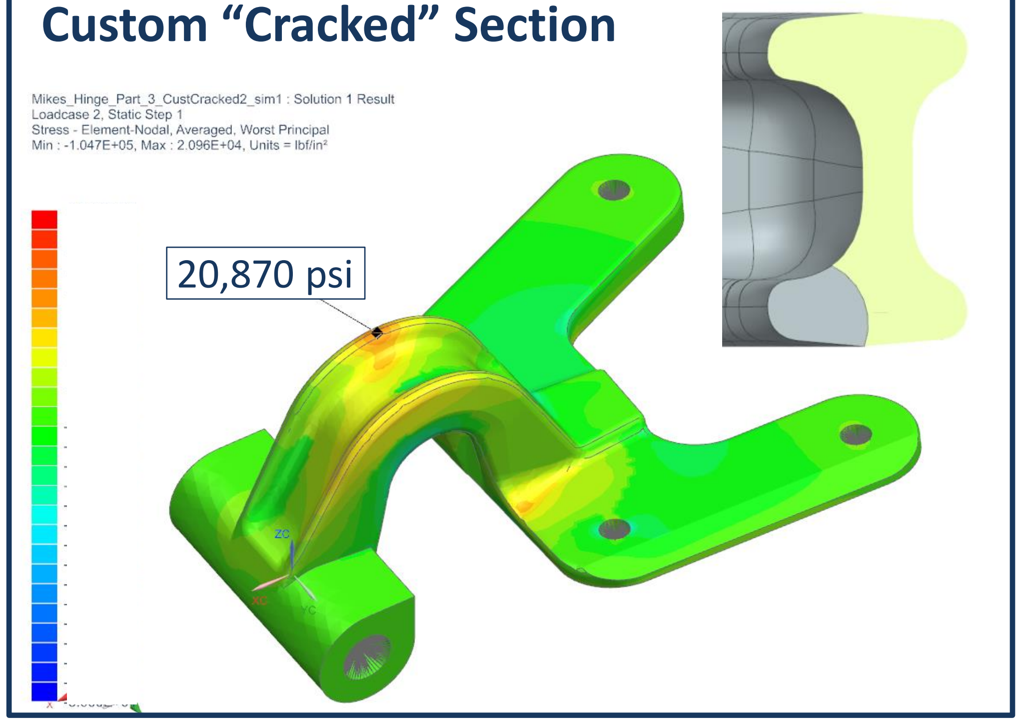
- The fractography report indicated that reverse-direction cracking was observed in the outer radius of hinge:
- Loading COND#2 (landing air drag) puts the outer radius in tension, but its magnitude is low—on the order of 9 ksi—for a pristine part
- However, as the inner radius crack grows larger and larger, the magnitude of this tension rapidly increases with loss of section, peaking at the correct location
- Though not explicitly modeled, it's conceivable the tensile stresses at some point may be high enough to initiate the observed reversed crack growth from existing stress risers.

Pristine (no pre-crack)



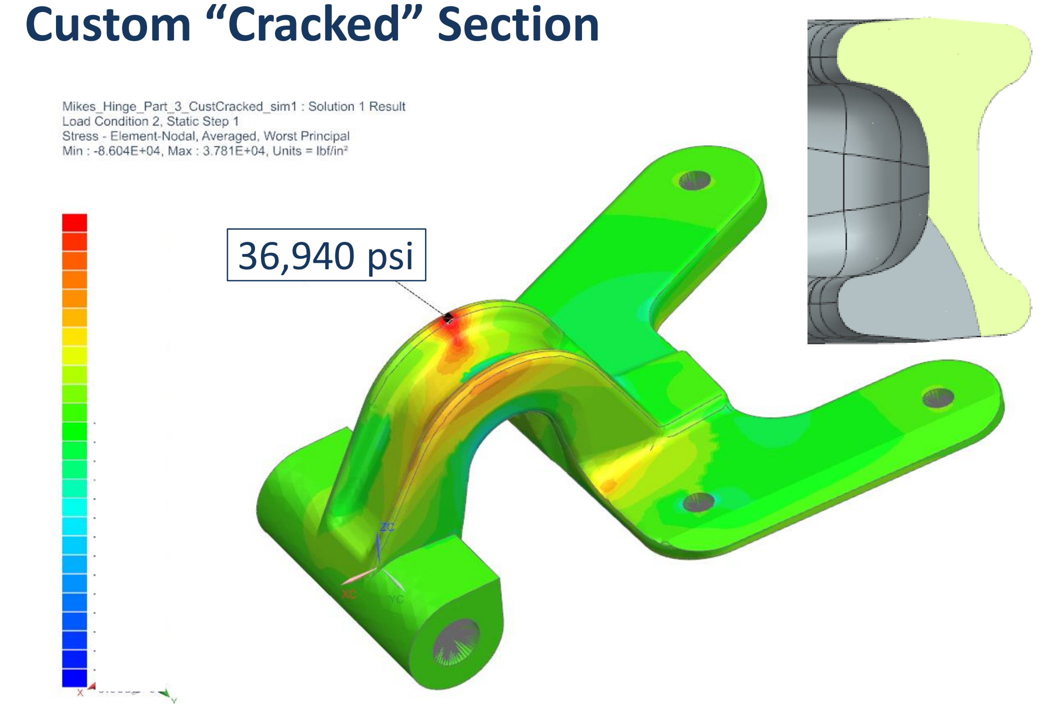
Mikes_Hinge_Part_2_sim1 : Solution 1 Result
Load Condition 2, Static Step 1
Stress - Element-Nodal, Averaged, Worst Principal
Min : -2.688E+04, Max : 1.770E+04, Units = lbf/in²

Custom "Cracked" Section



Mikes_Hinge_Part_3_CustCracked2_sim1 : Solution 1 Result
Loadcase 2, Static Step 1
Stress - Element-Nodal, Averaged, Worst Principal
Min : -1.047E+05, Max : 2.096E+04, Units = lbf/in²

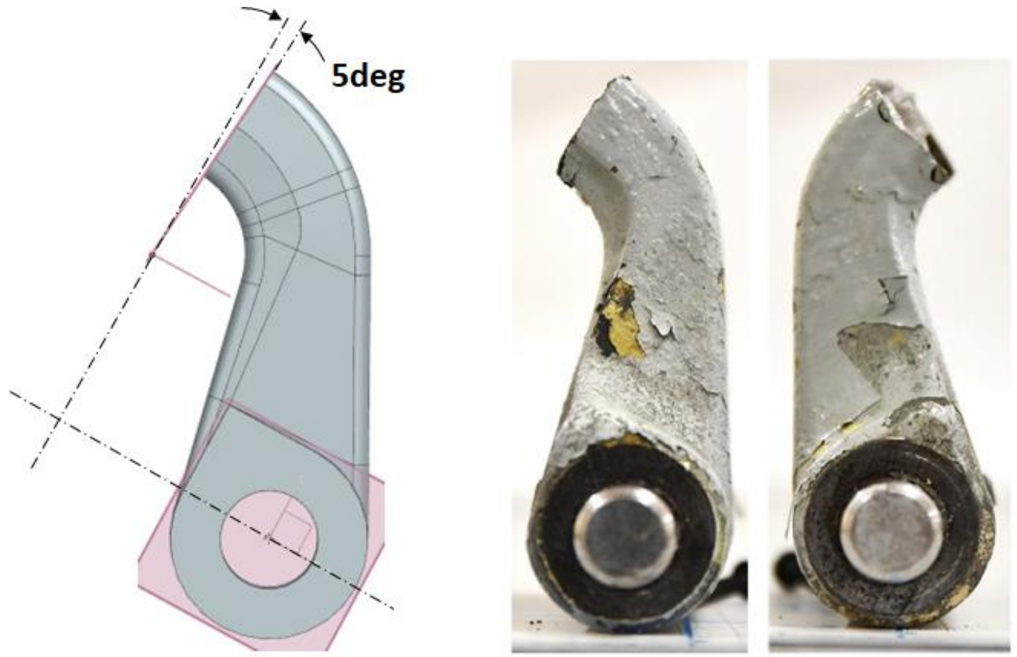
Custom "Cracked" Section

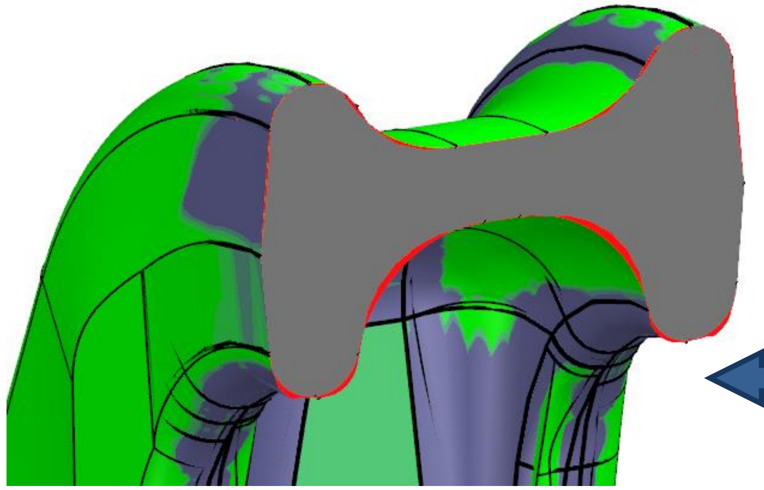


Mikes_Hinge_Part_3_CustCracked_sim1 : Solution 1 Result
Load Condition 2, Static Step 1
Stress - Element-Nodal, Averaged, Worst Principal
Min : -8.604E+04, Max : 3.781E+04, Units = lbf/in²

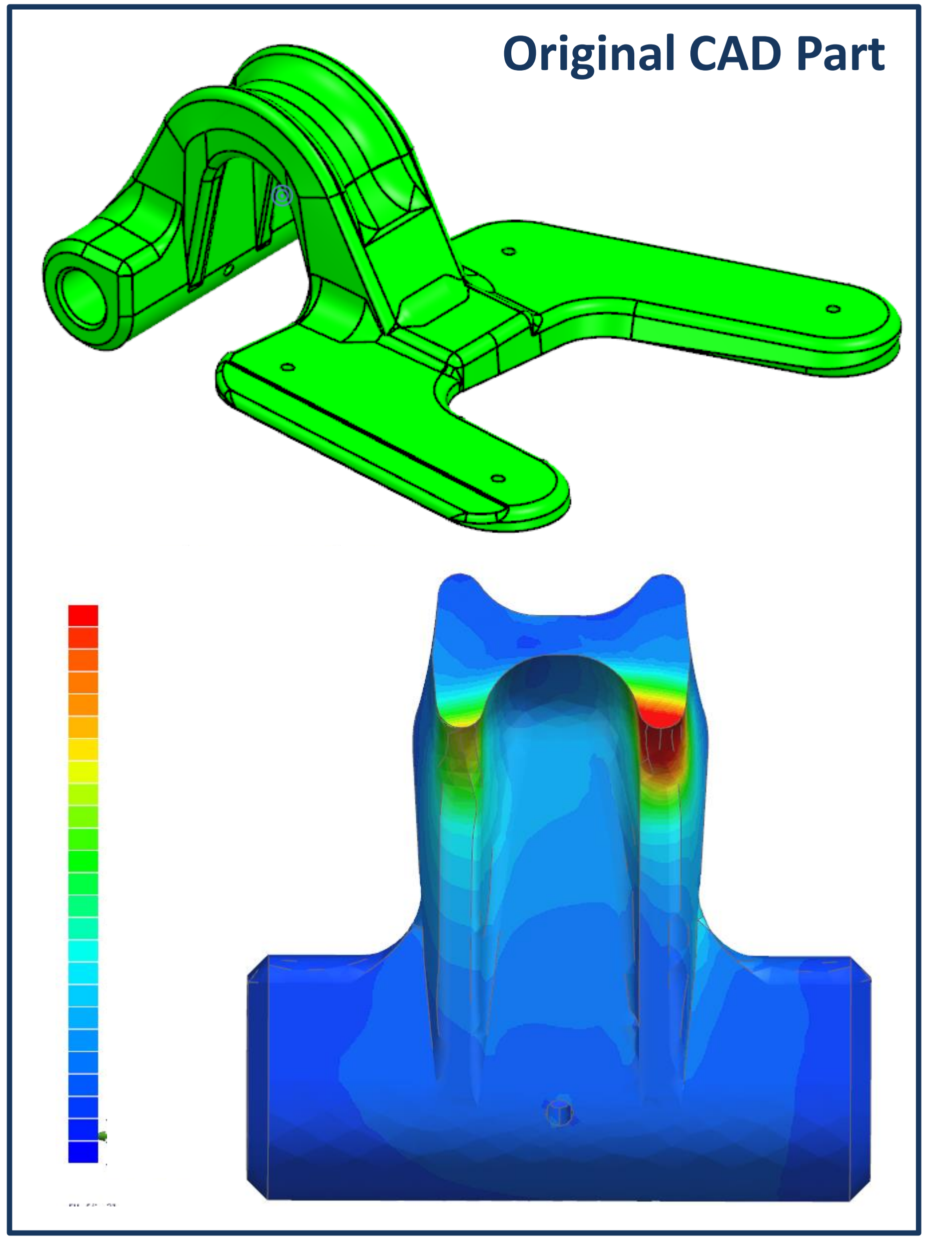
Conversion to StressCheck

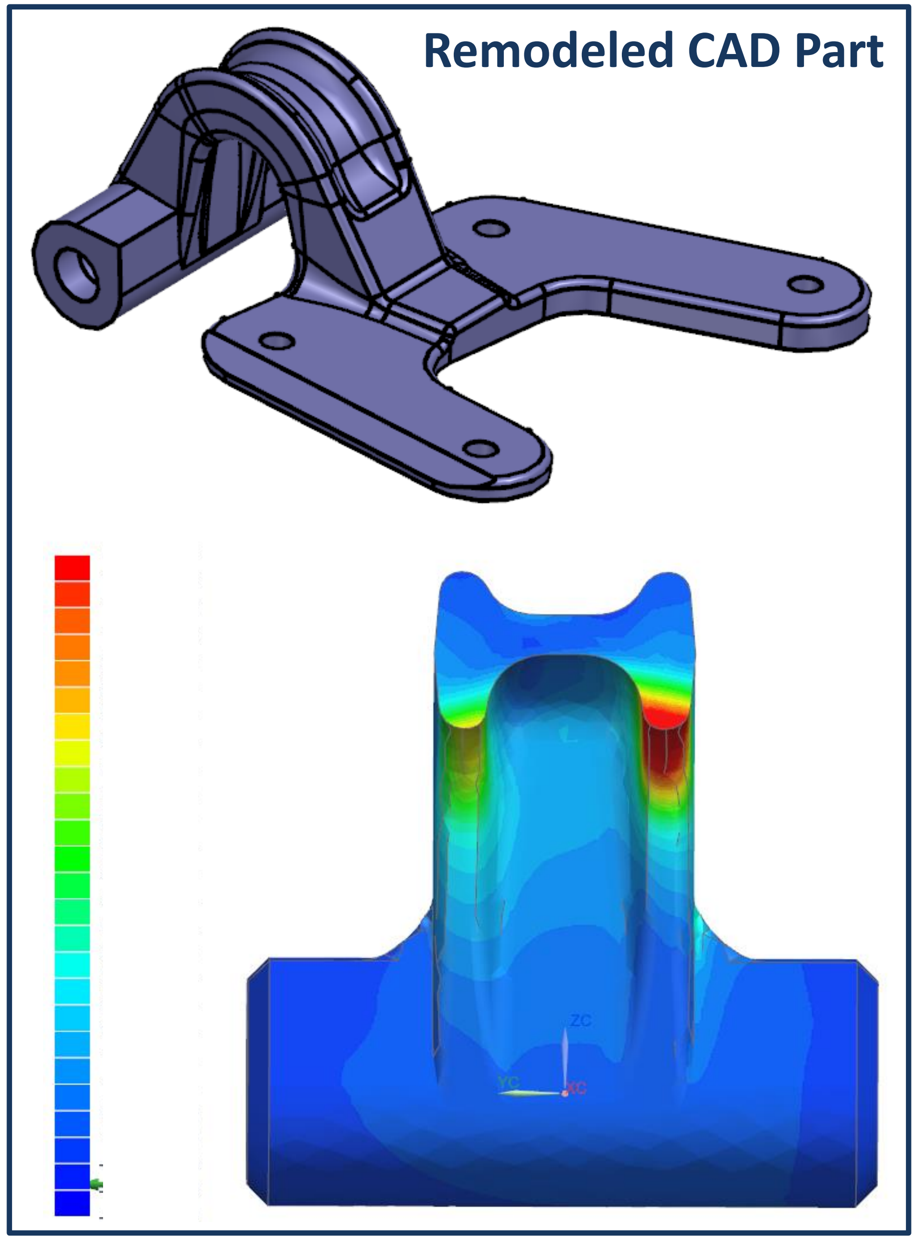
- The initial FEA was run using Siemens SimCenter 3D software
- When converting to StressCheck (for crack K solution capability) the original CAD part would not mesh and attempts to clean up or modify the CAD geometry were unsuccessful
- The CAD part was remodeled for meshing with primary focus on matching section at crack plane near the center of “gooseneck”





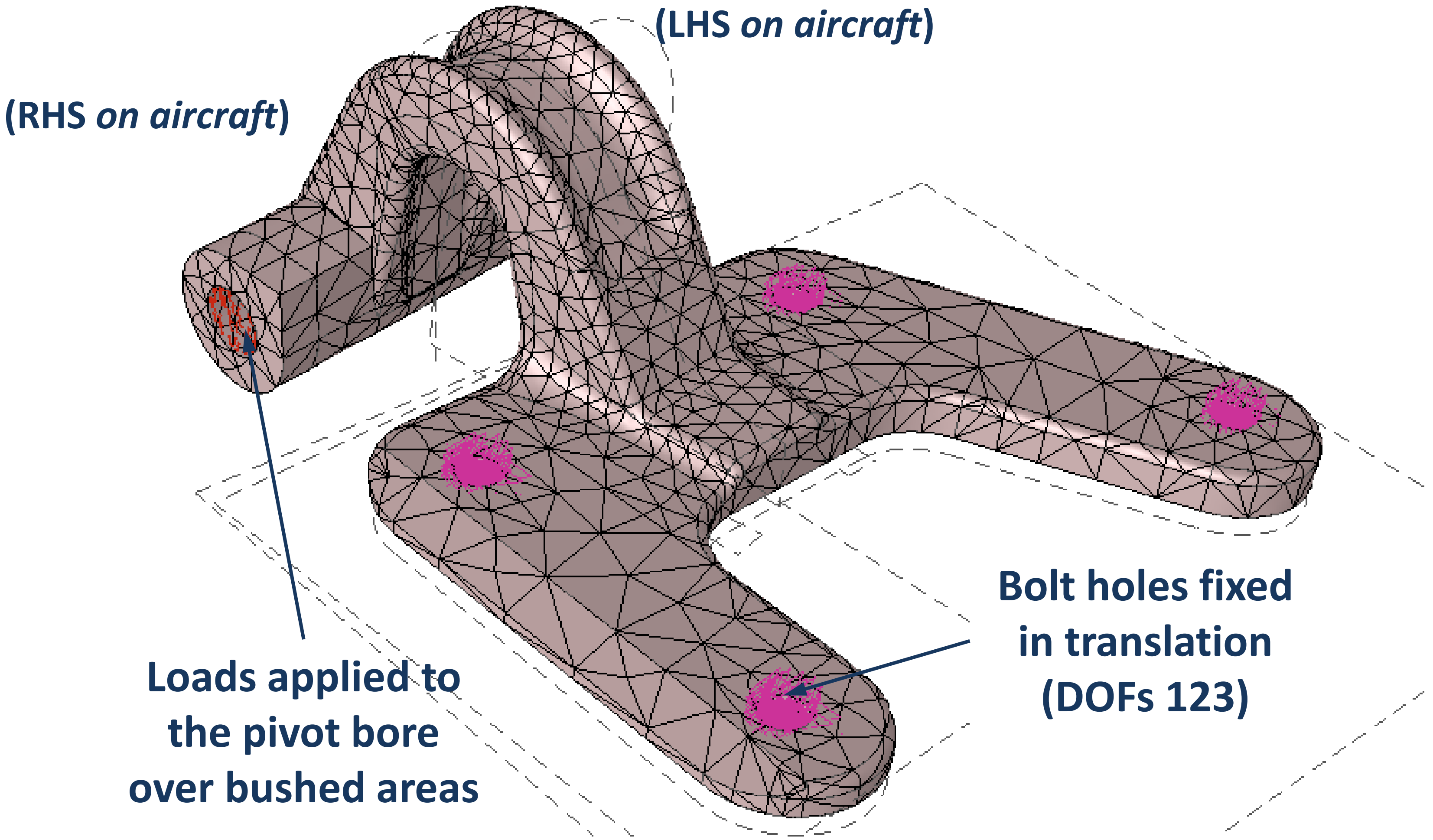
Section overlay of Original and Remodeled CAD Parts


Original CAD Part


Remodeled CAD Part

StressCheck Finite Element Model

- **Loads and boundary conditions set up to mimic initial FEA done in SimCenter 3D**
- **Stresses at crack origins pulled from “pristine” geometry FEA and referenced for peak stress and R-ratios:**
  - **Near crack origin on LHS (on aircraft): σ_{max} = 169.7 ksi; R = -0.153**
  - **Near crack origin on RHS (on aircraft): σ_{max} = 81.3 ksi; R = -0.188**

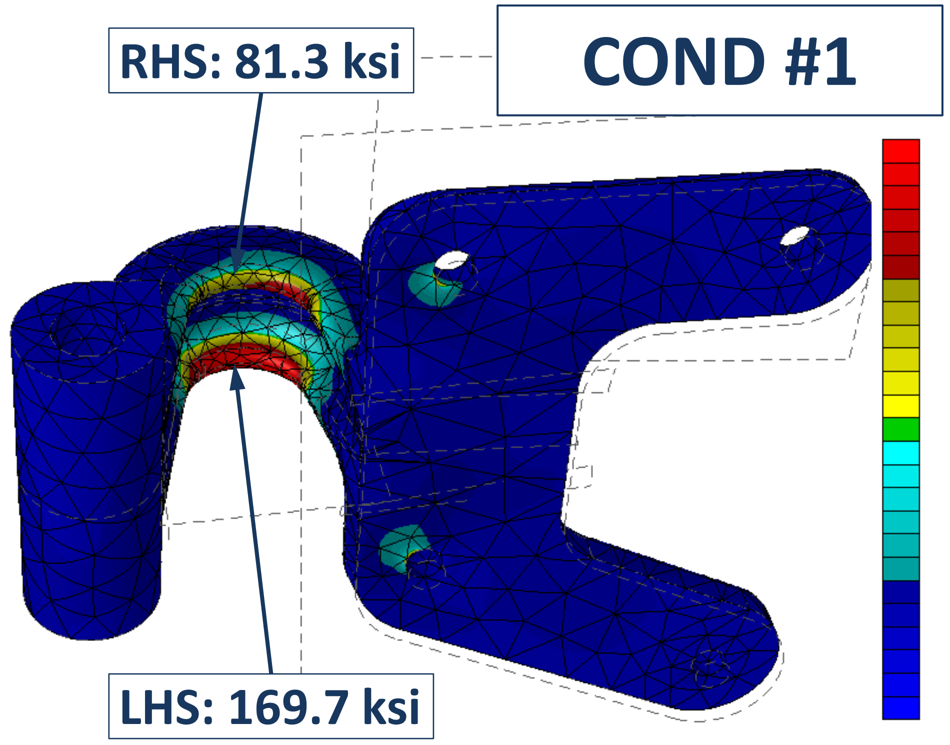


(LHS *on aircraft*)

(RHS *on aircraft*)

Loads applied to the pivot bore over bushed areas

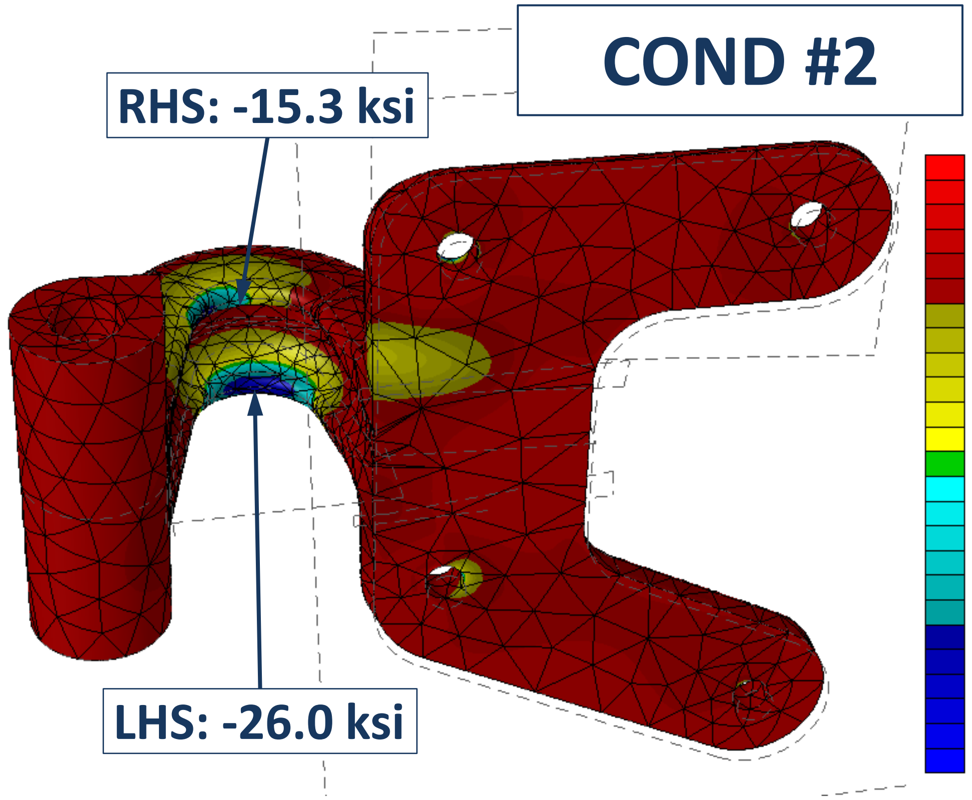
Bolt holes fixed in translation (DOFs 123)



COND #1

RHS: 81.3 ksi

LHS: 169.7 ksi



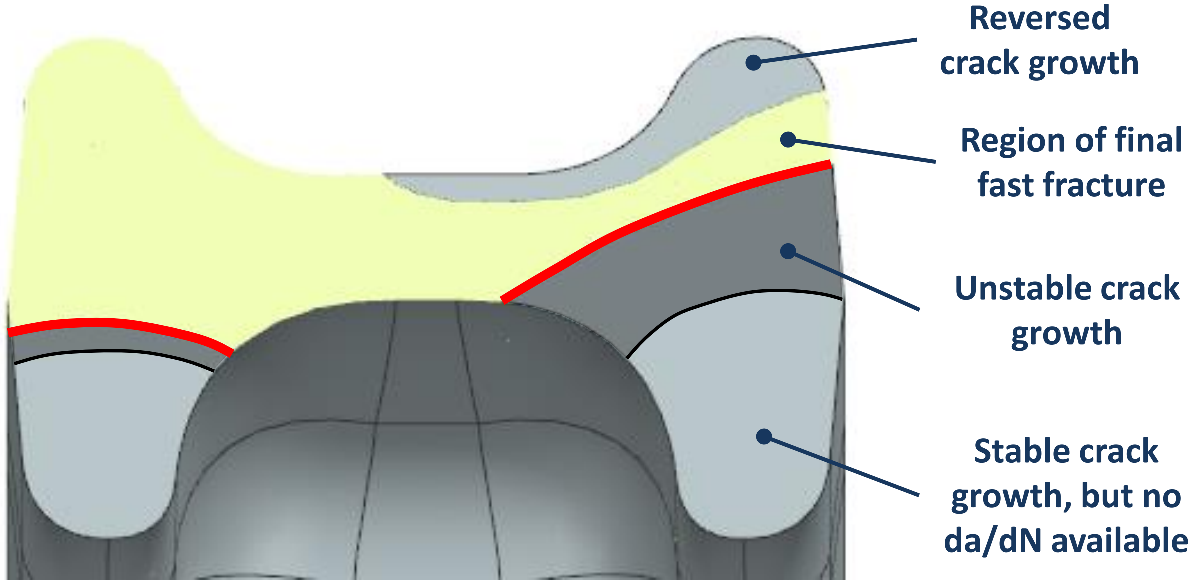
COND #2

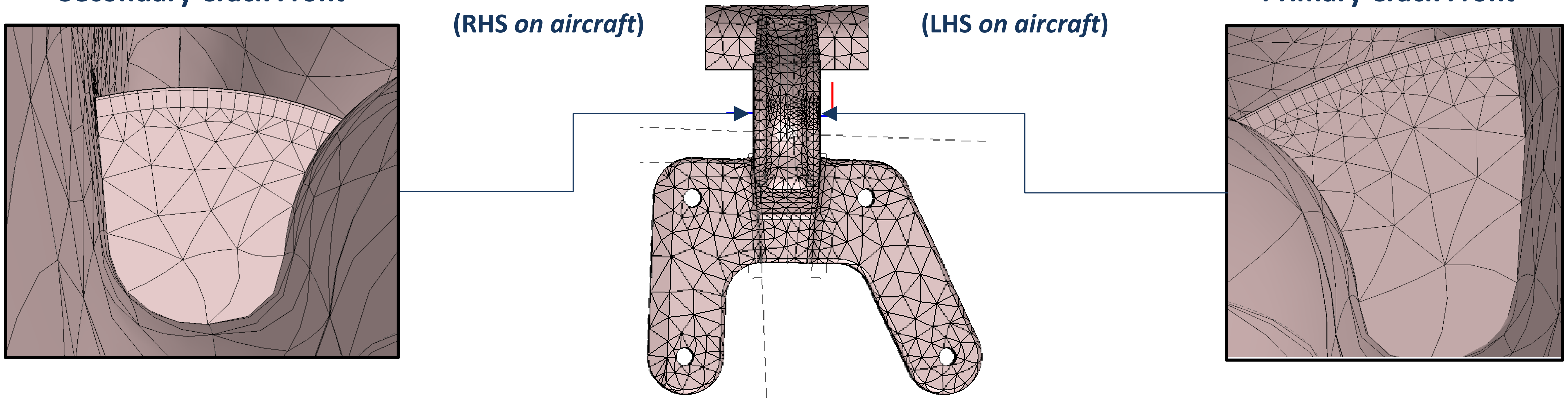
RHS: -15.3 ksi

LHS: -26.0 ksi

StressCheck Finite Element Model

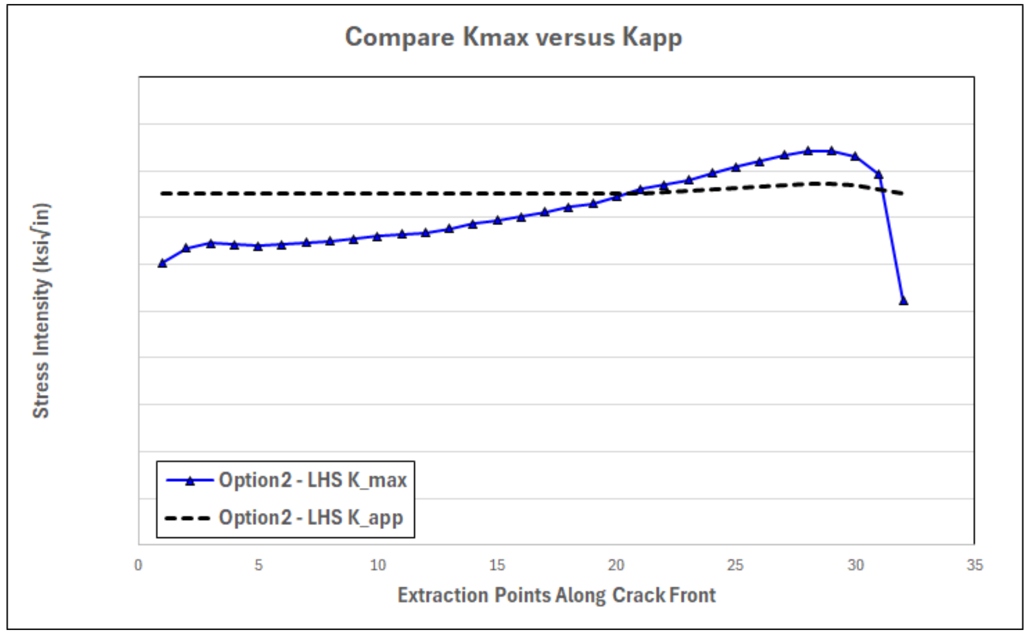
- Various configurations of the StressCheck FEM utilized for several different analyses/checks:
  - Residual strength checks
  - Crack growth rate correlation
  - DTA model for generating inspection interval
- Typically modeled one or both cracks on LHS and RHS of the hinge



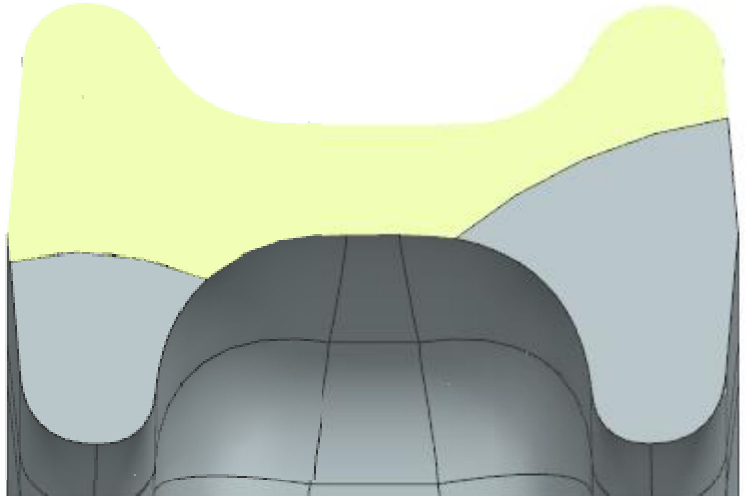

Example 2-Crack Front Configuration

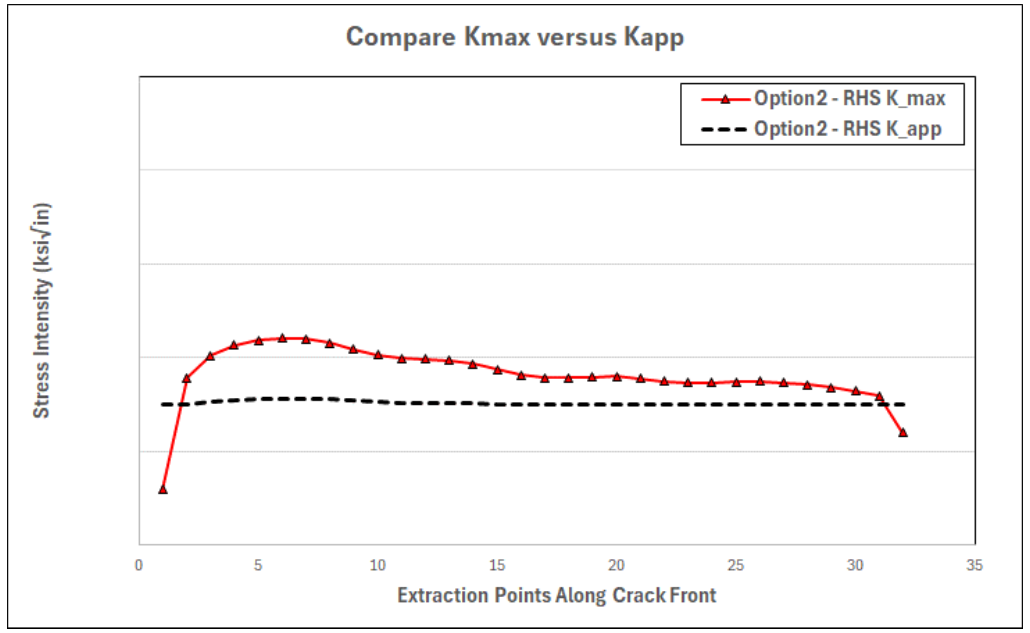
K_{max} versus K_{app} (Residual Strength)

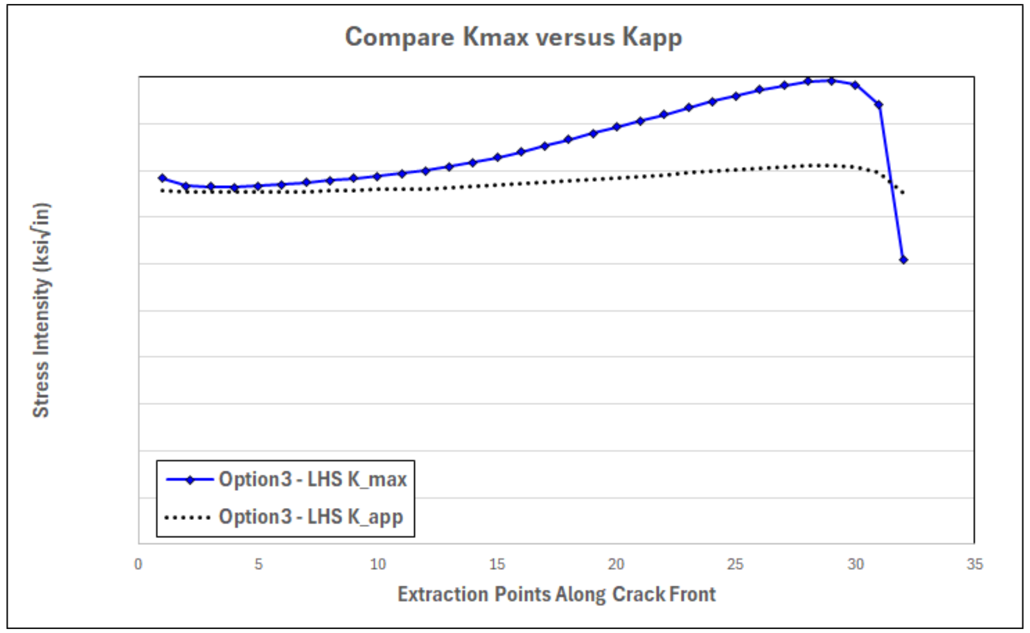
- Fractography images were used to estimate crack front geometries at transition to dimpled rupture zones and build StressCheck models having those crack front geometries
- Assumed peak load condition (COND #1) was applied to the model
- Comparison of K_{max} to K_{app} along crack front points lends credence to assumed load level



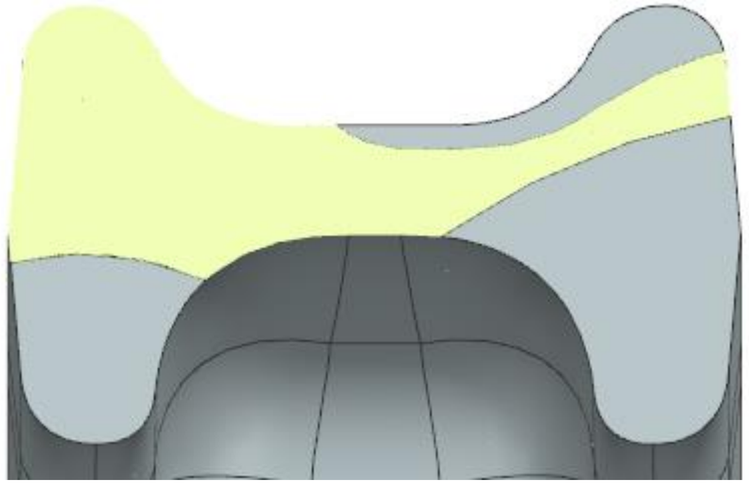
Fracture State Check #1

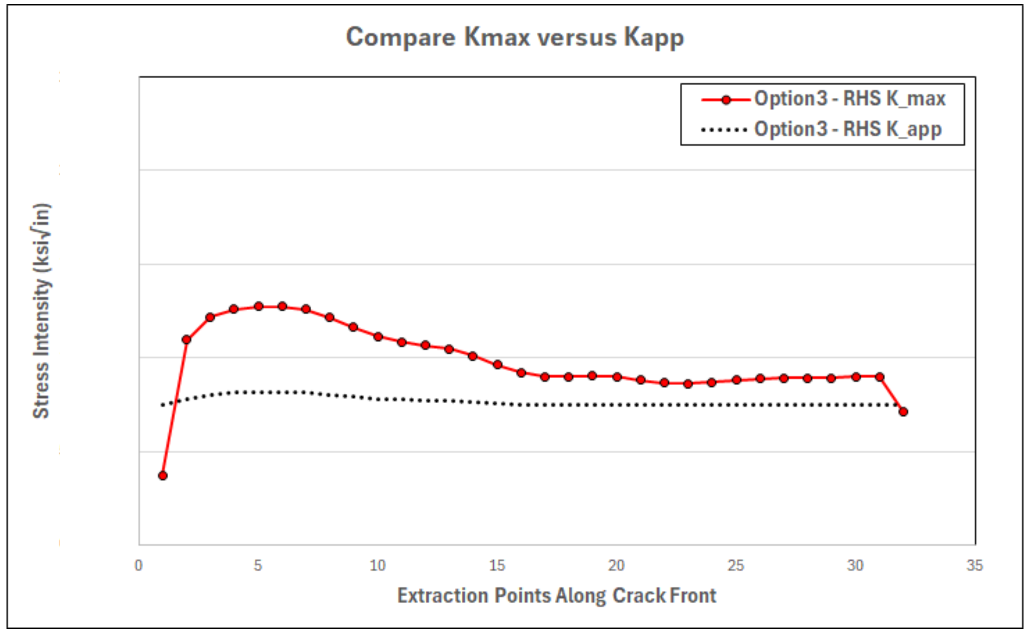






Fracture State Check #2





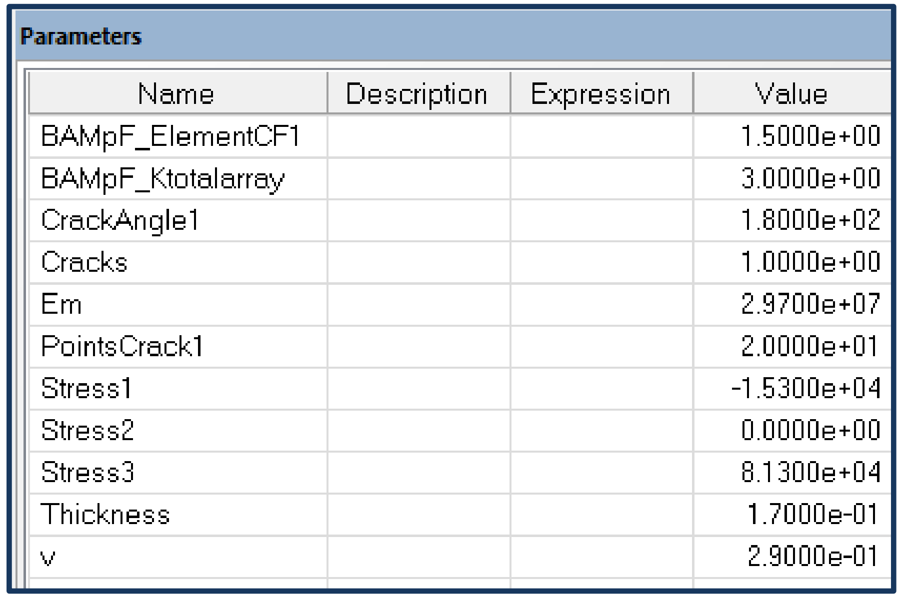
Spectrum

- **Actual loading spectrum unknown but initial FEA using design limit conditions demonstrated those conditions to be reasonable guesses for enveloping (i.e. min-to-max) conditions**

- **A constant amplitude spectrum was assumed where the stresses for a load cycle at each crack origin range from σ_{max} and σ_{min} for COND #1 and COND #2, respectively**
  - **Near crack origin on LHS (on aircraft): σ_{max} = 169.7 ksi; σ_{min} = -26.0 ksi; (R = -0.153)**
  - **Near crack origin on RHS (on aircraft): σ_{max} = 81.3 ksi; σ_{min} = -15.3 ksi; (R = -0.188)**

- **Because the stress range is defined by independent load conditions, the BAMpF "Karray" feature was utilized to provide independent SIF's for each condition**
  - **Stress1 (BAMpF) == SMF*R (AFGROW)**
  - **Stress 2 == 0**
  - **Stress3 (BAMpF) == SMF (AFGROW)**

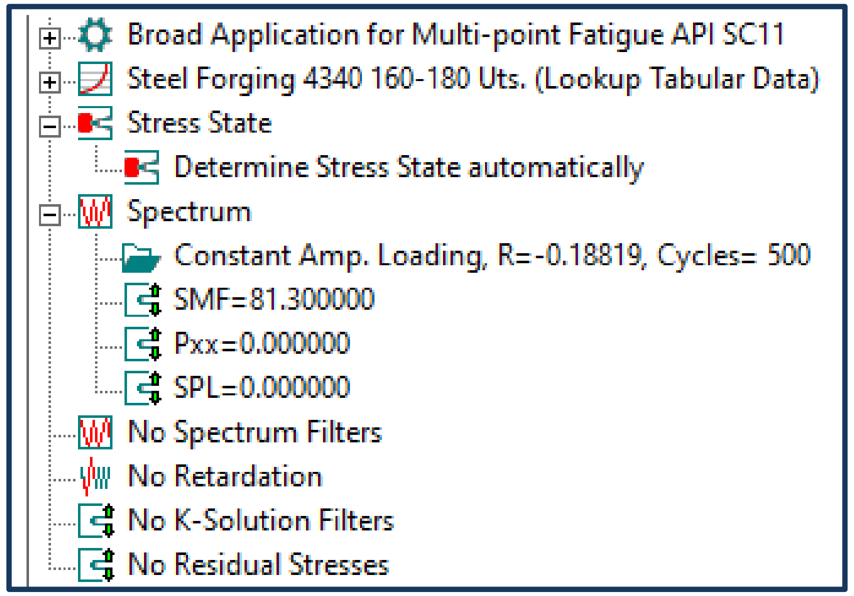
BAMpF Parameters Defined in StressCheck



Parameters

Name	Description	Expression	Value
BAMpF_ElementCF1			1.5000e+00
BAMpF_Ktotalarray			3.0000e+00
CrackAngle1			1.8000e+02
Cracks			1.0000e+00
Em			2.9700e+07
PointsCrack1			2.0000e+01
Stress1			-1.5300e+04
Stress2			0.0000e+00
Stress3			8.1300e+04
Thickness			1.7000e-01
v			2.9000e-01

AFGROW Model Inputs

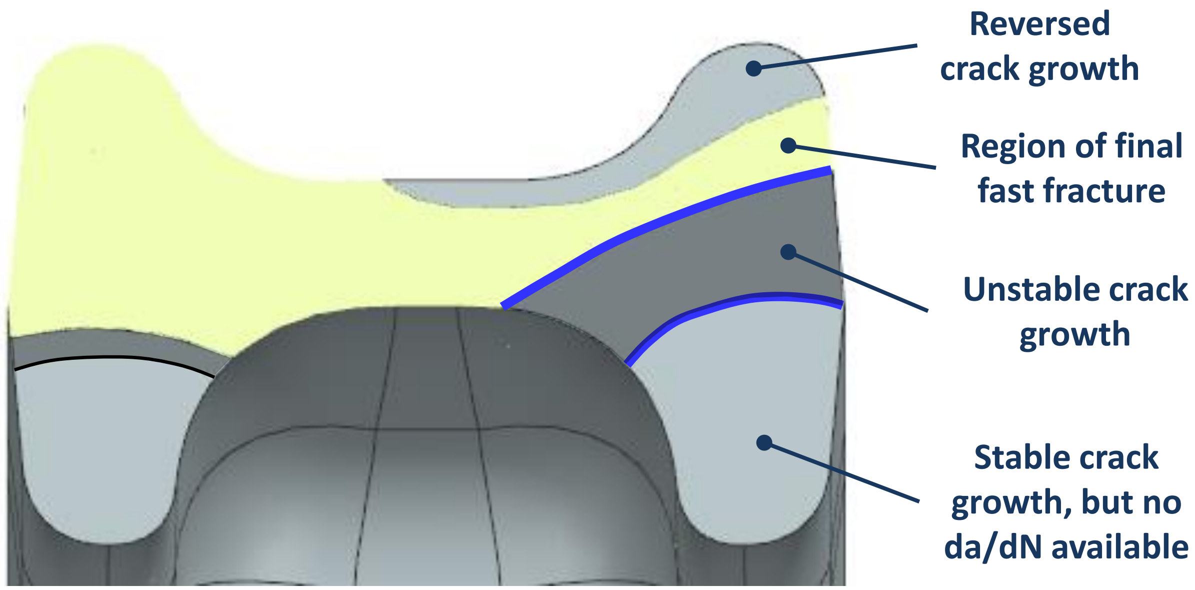


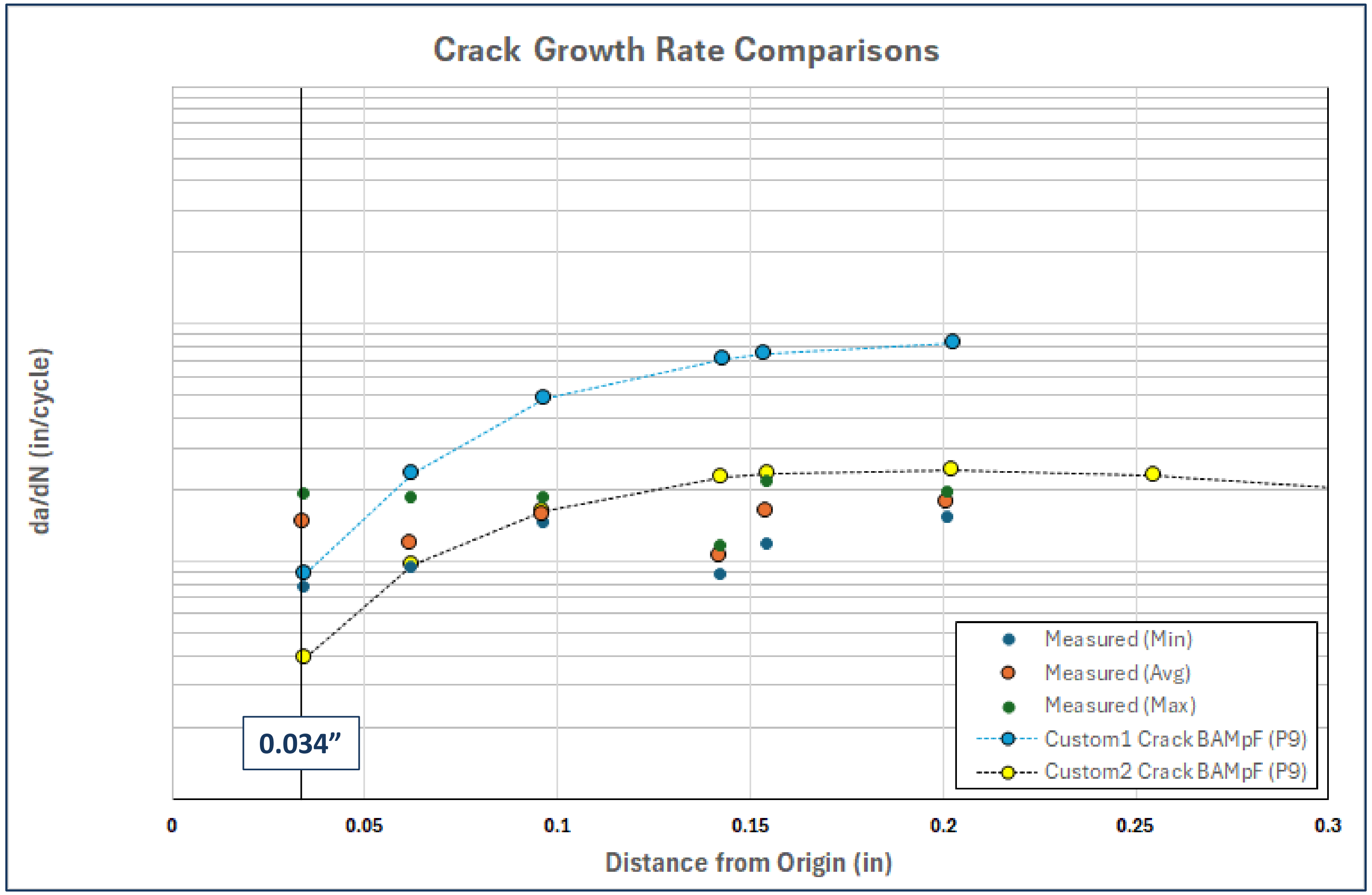
- Broad Application for Multi-point Fatigue API SC11
- Steel Forging 4340 160-180 Uts. (Lookup Tabular Data)
- Stress State
  - Determine Stress State automatically
- Spectrum
  - Constant Amp. Loading, R=-0.18819, Cycles= 500
  - SMF=81.300000
  - Pxx=0.000000
  - SPL=0.000000
- No Spectrum Filters
- No Retardation
- No K-Solution Filters
- No Residual Stresses

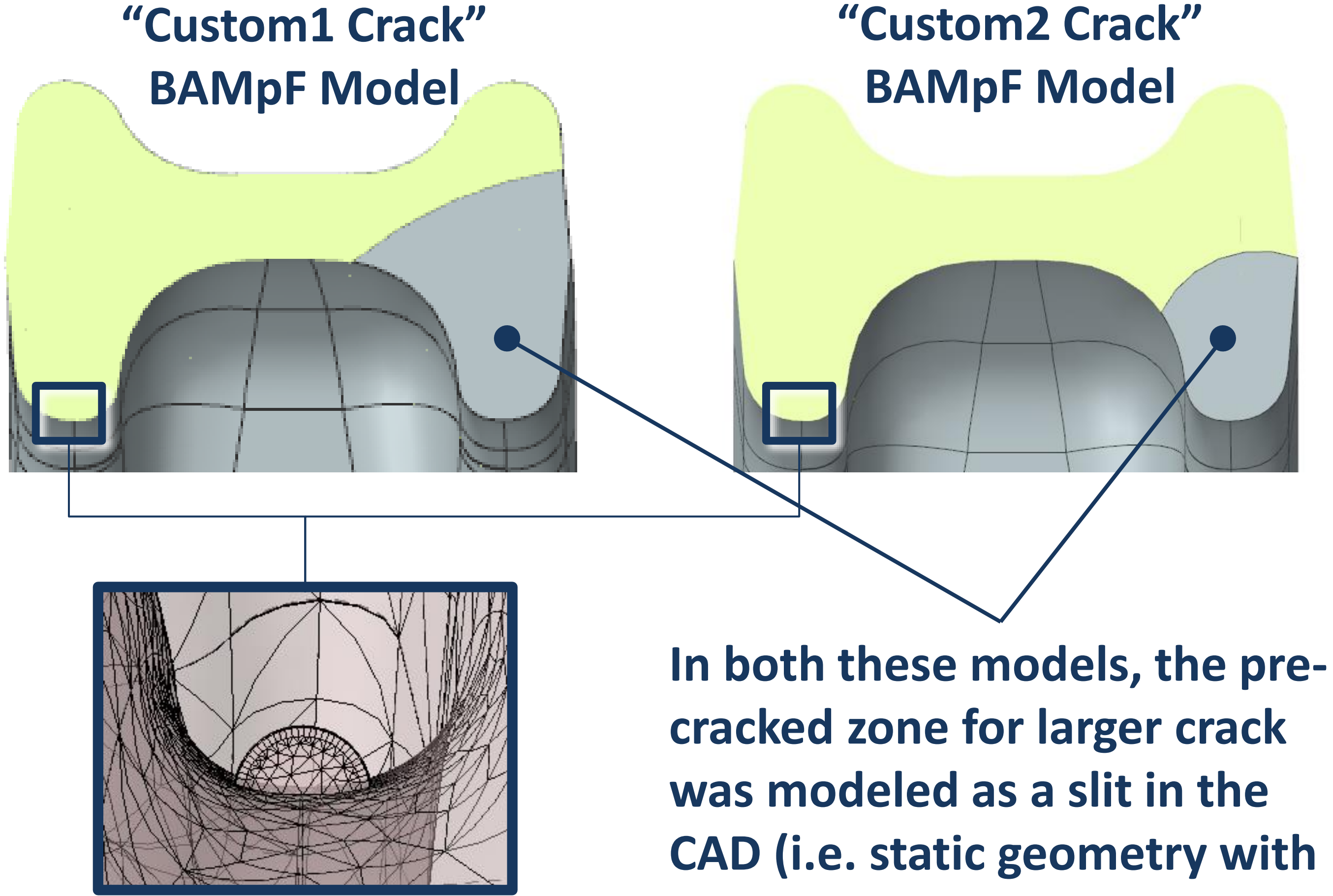
- **As long as the stress parameters defined for BAMpF match the far field stress definition(s) in AFGROW, then the actual SIF's extracted from StressCheck FEM for *any crack size* will be properly recovered in AFGROW for the pointwise da/dN calculations**

da/dN Correlation

- BAMpF models were created having the two custom primary pre-crack geometries shown on the right (“Custom 1 Crack” & “Custom 2 Crack”)
- At secondary crack origin, each assumed an 0.034” semicircular IFS which was based on the starting dimension of available measured da/dN data
- Each used σ_{max} = 81.3 ksi and R = -0.188





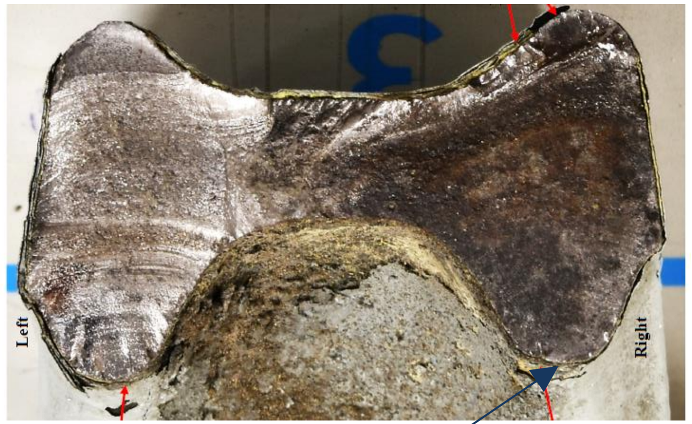


0.034” Semicircular crack

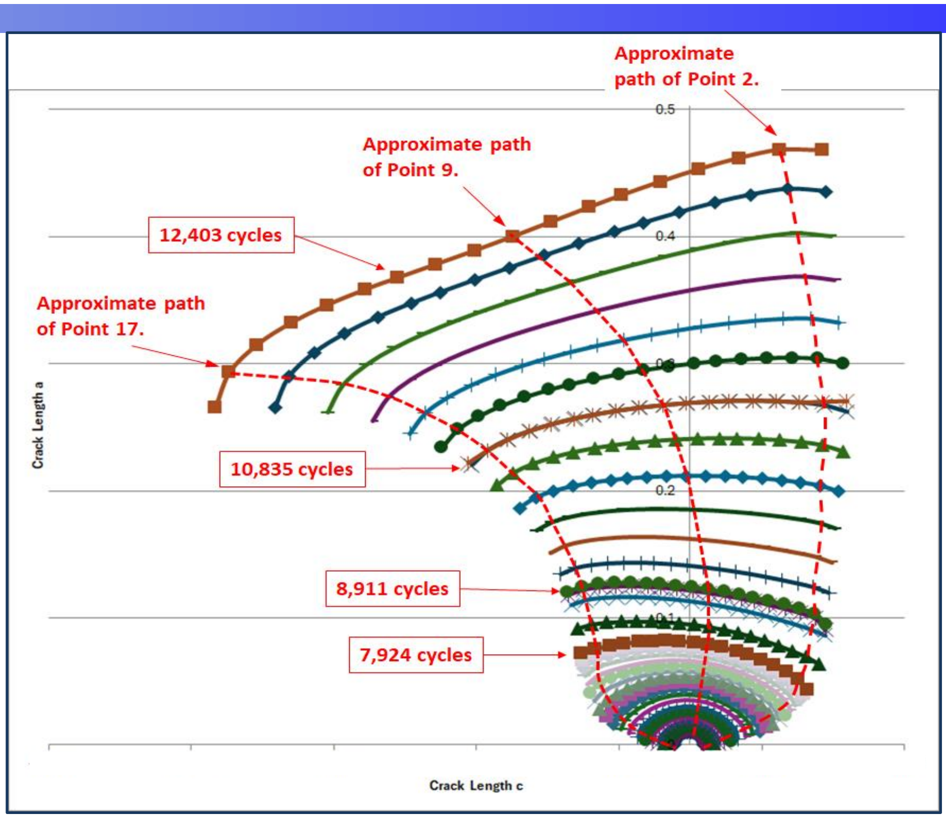
In both these models, the pre-cracked zone for larger crack was modeled as a slit in the CAD (i.e. static geometry with no crack front modeled)

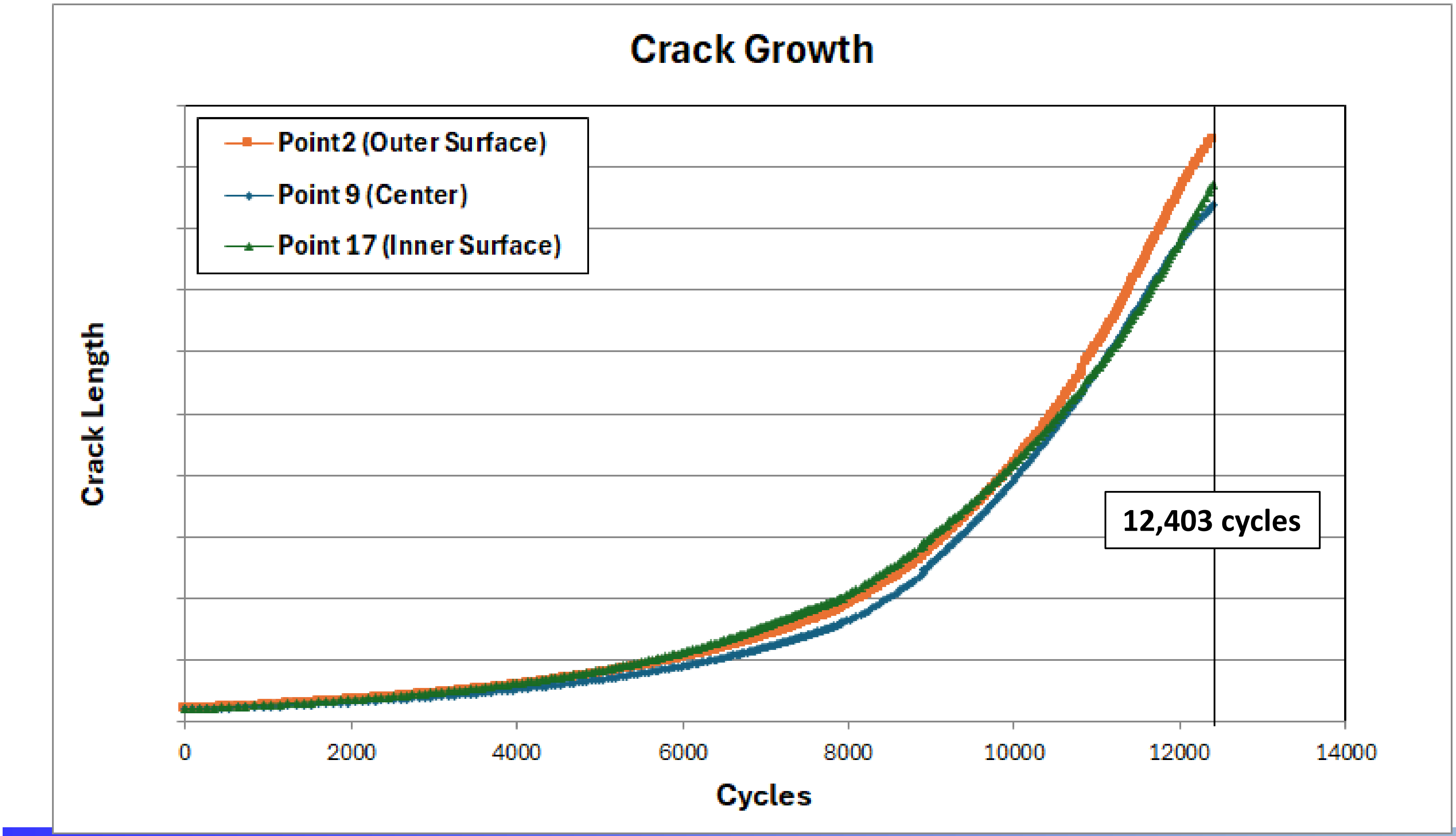
1-Crack Model DTA

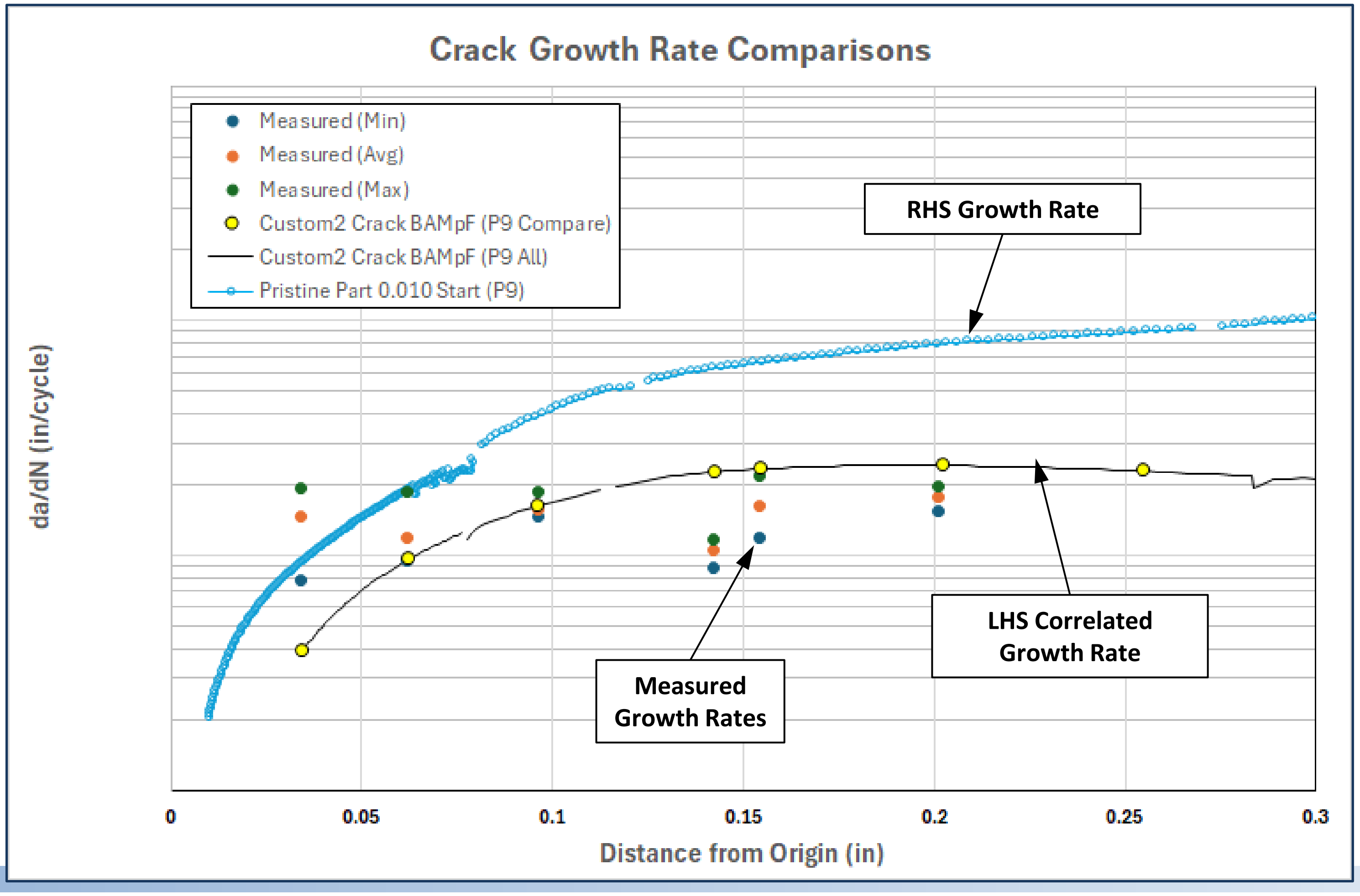
- A new BAMpF model was created having a 0.010” semicircular crack on the higher-stressed RHS of the hinge.
- The model predicted ≈12,400 stress cycles to grow the crack to the shape shown at right, which nearly matches the extent of the corroded RHS region on the failed hinge.



DTA model results shown here assume a 0.010” semi-circular initial flaw at this higher stress location on RHS.

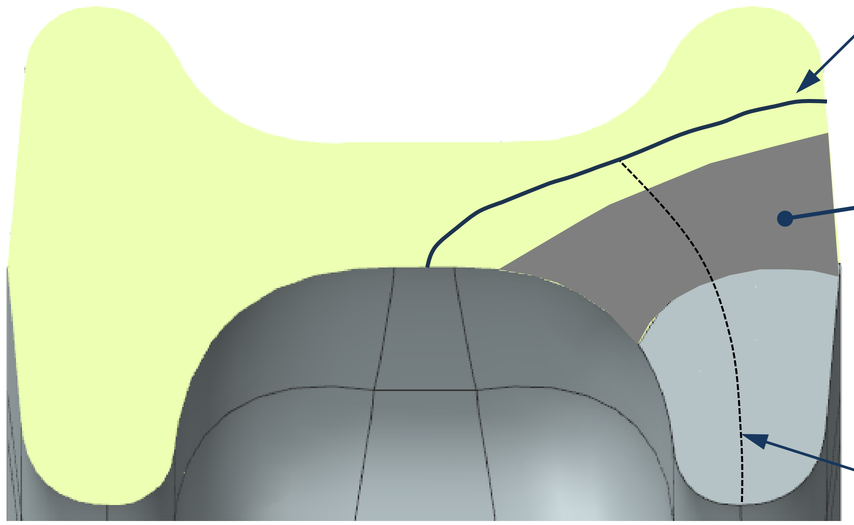






1-Crack Model DTA

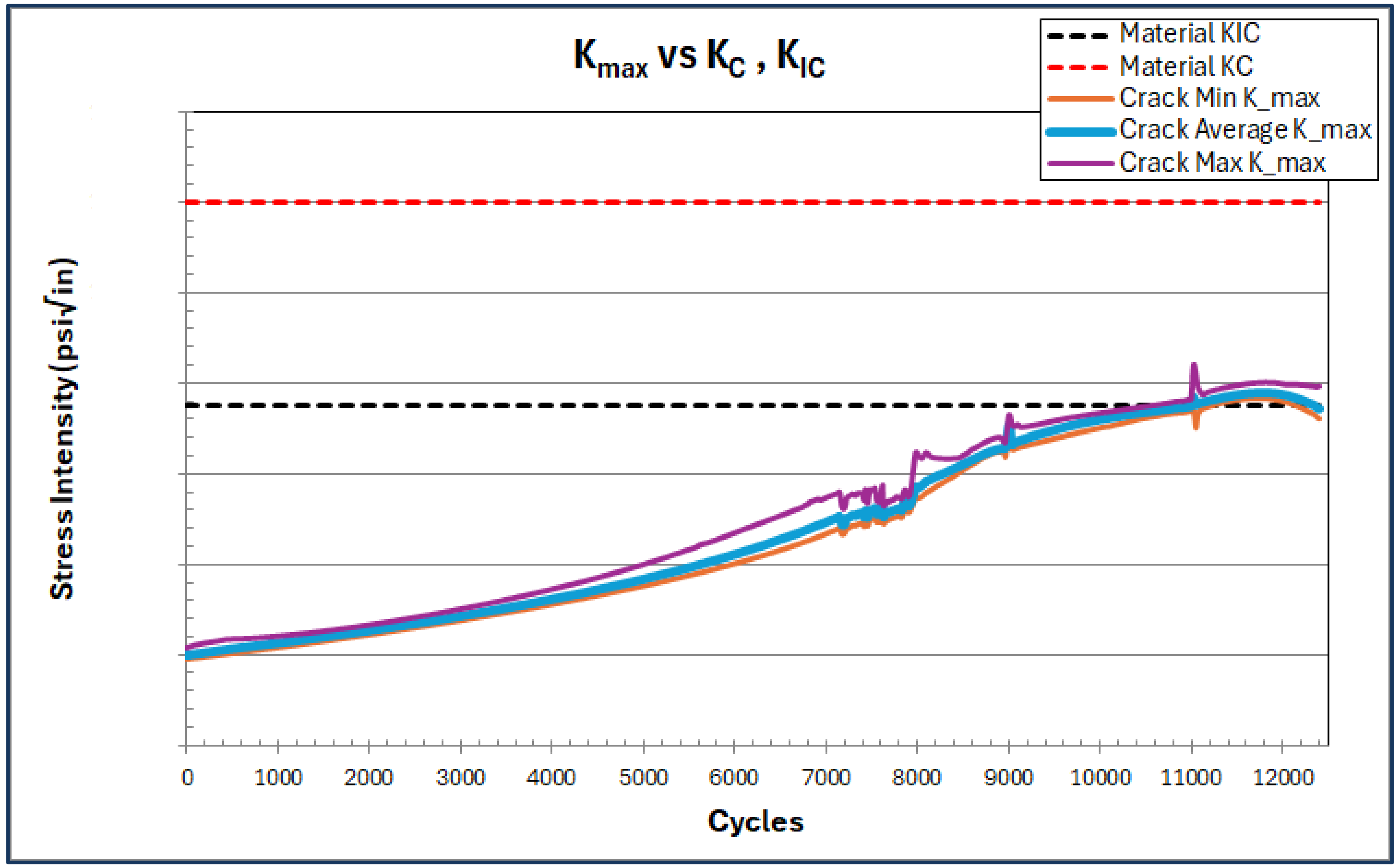
- Plots below show comparison of crack front K_{max} data versus K_C and K_{IC} over the cycle life and center crack point path length
- The region where predicted crack K_{max} begins to exceed K_{IC} agrees reasonably well with observed region of unstable crack growth on fracture surface
- (Possibly more cycle capacity based on previous residual strength checks that included LHS crack)

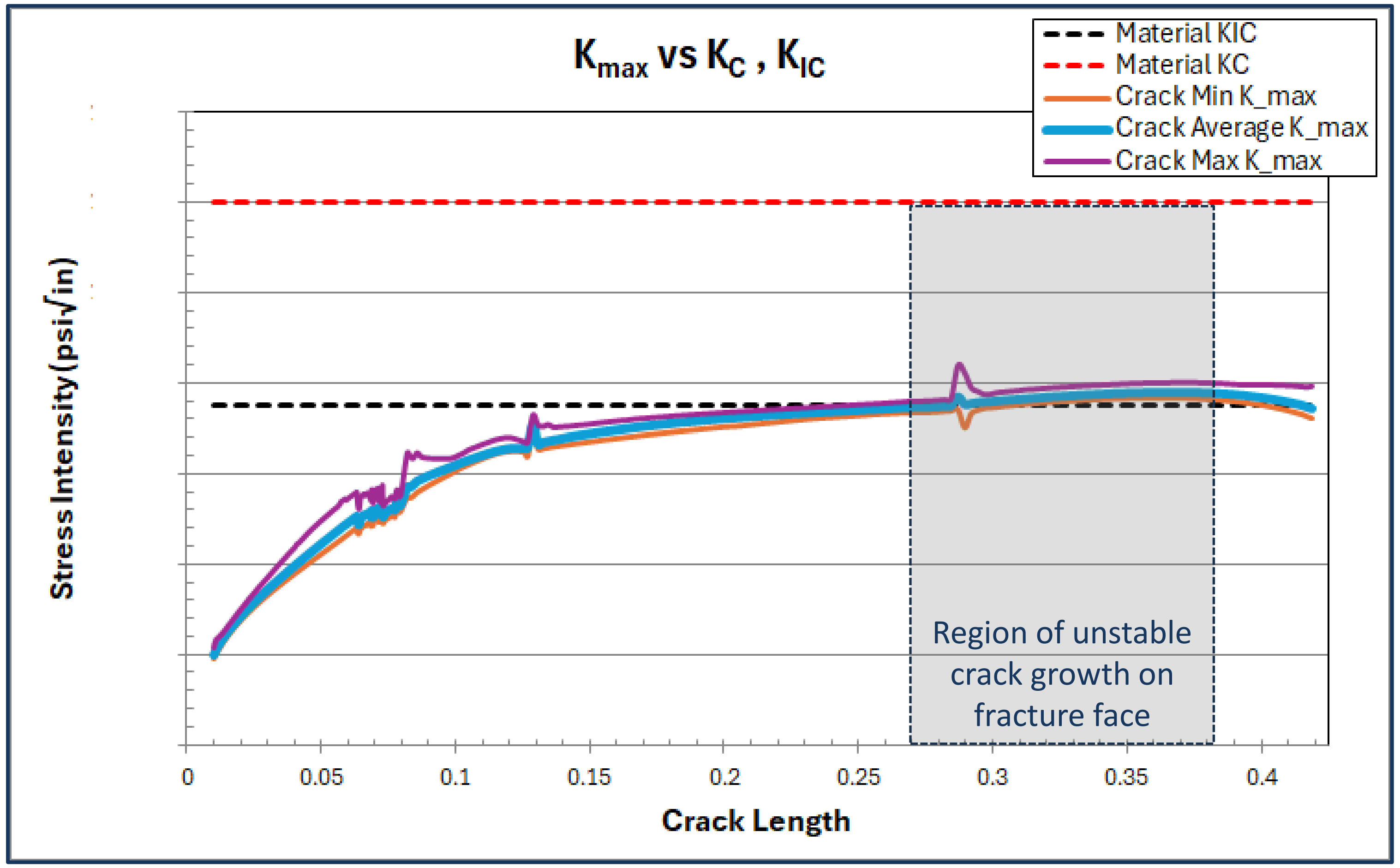


BAMpF crack model run to approximately this shape

Unstable crack growth

Center crack path length used in chart below

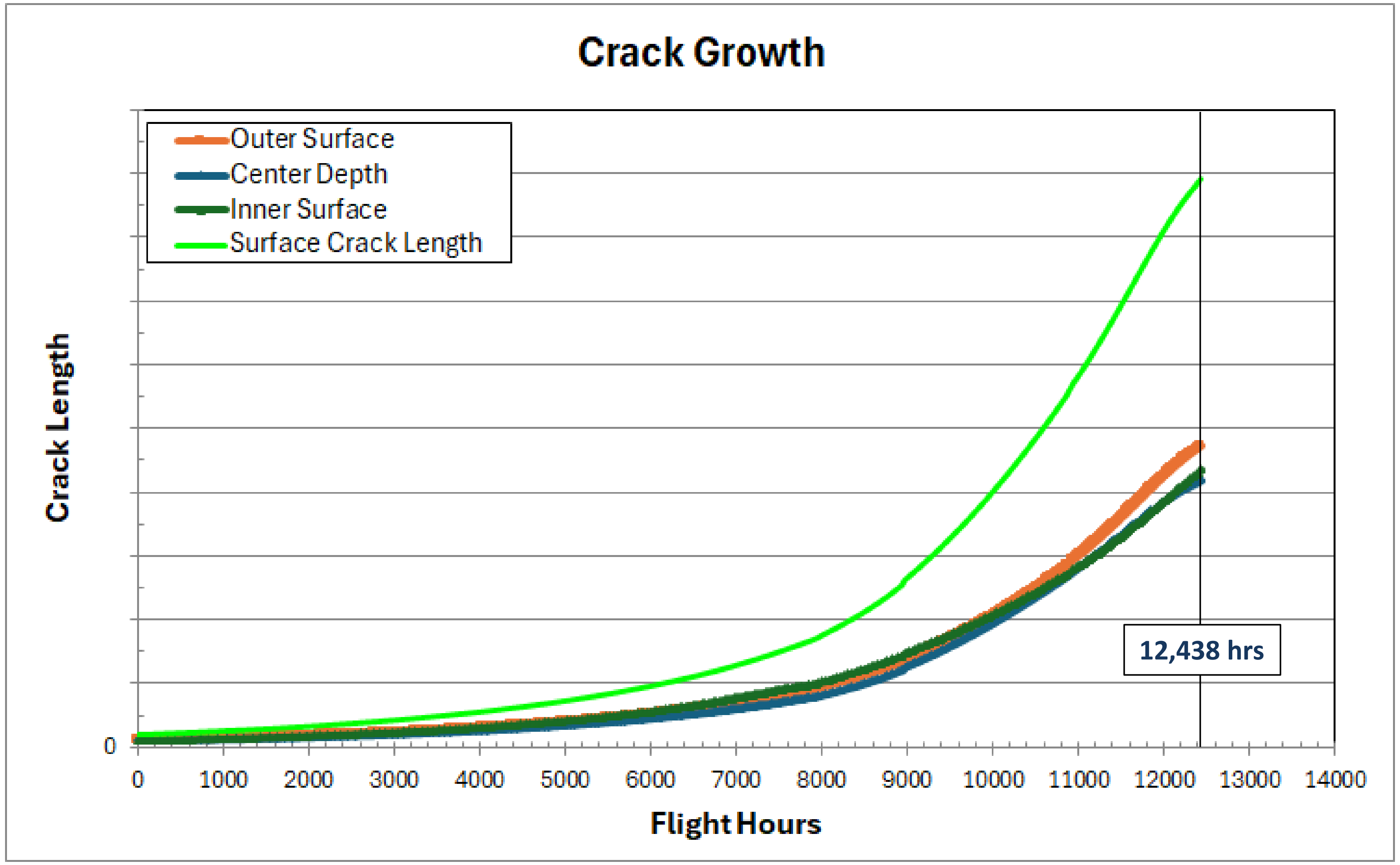




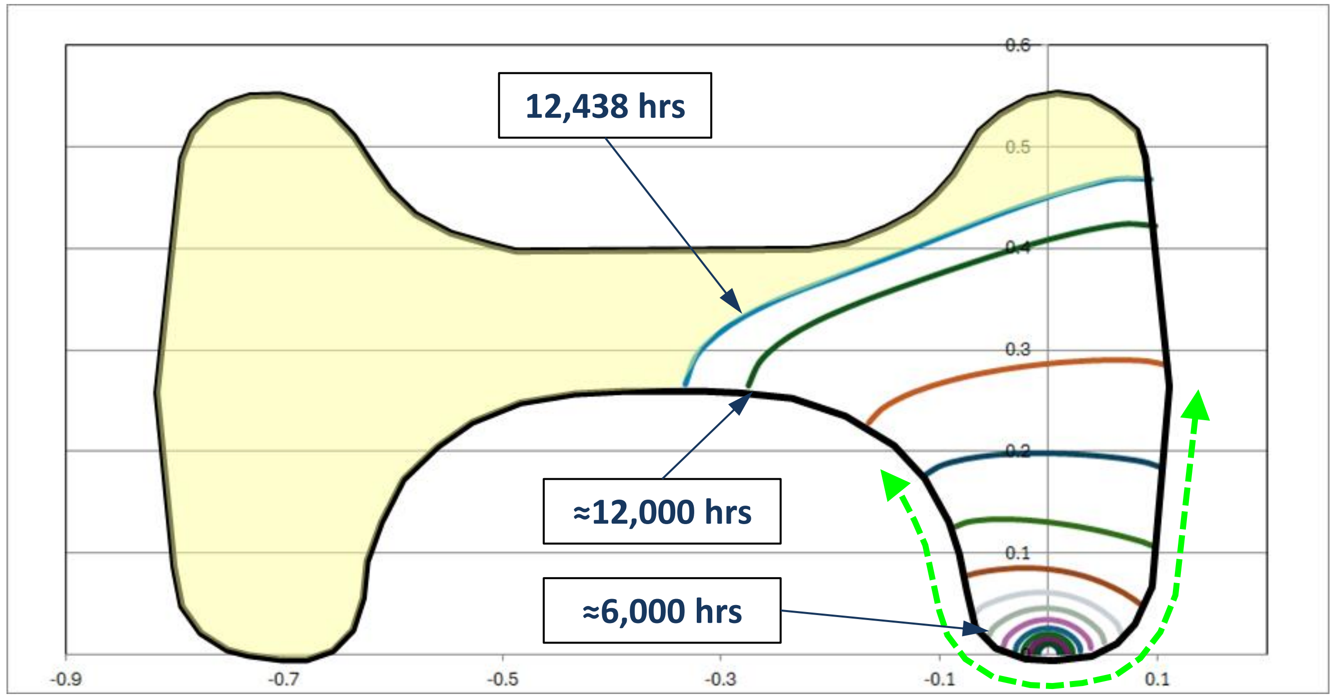
1-Crack Model DTA

- The fleet average total flight hours is 12,438 hrs. *If we assume the hinge cycle count correlates to this average flight time,* then the cycle-to-hours rate required to correlate to failure for this hinge is 12,403/12,438 ≈ 0.997 cycles/hr

➢ The BAMpF-predicted crack growth rates and shapes as a function of flight hours are shown below

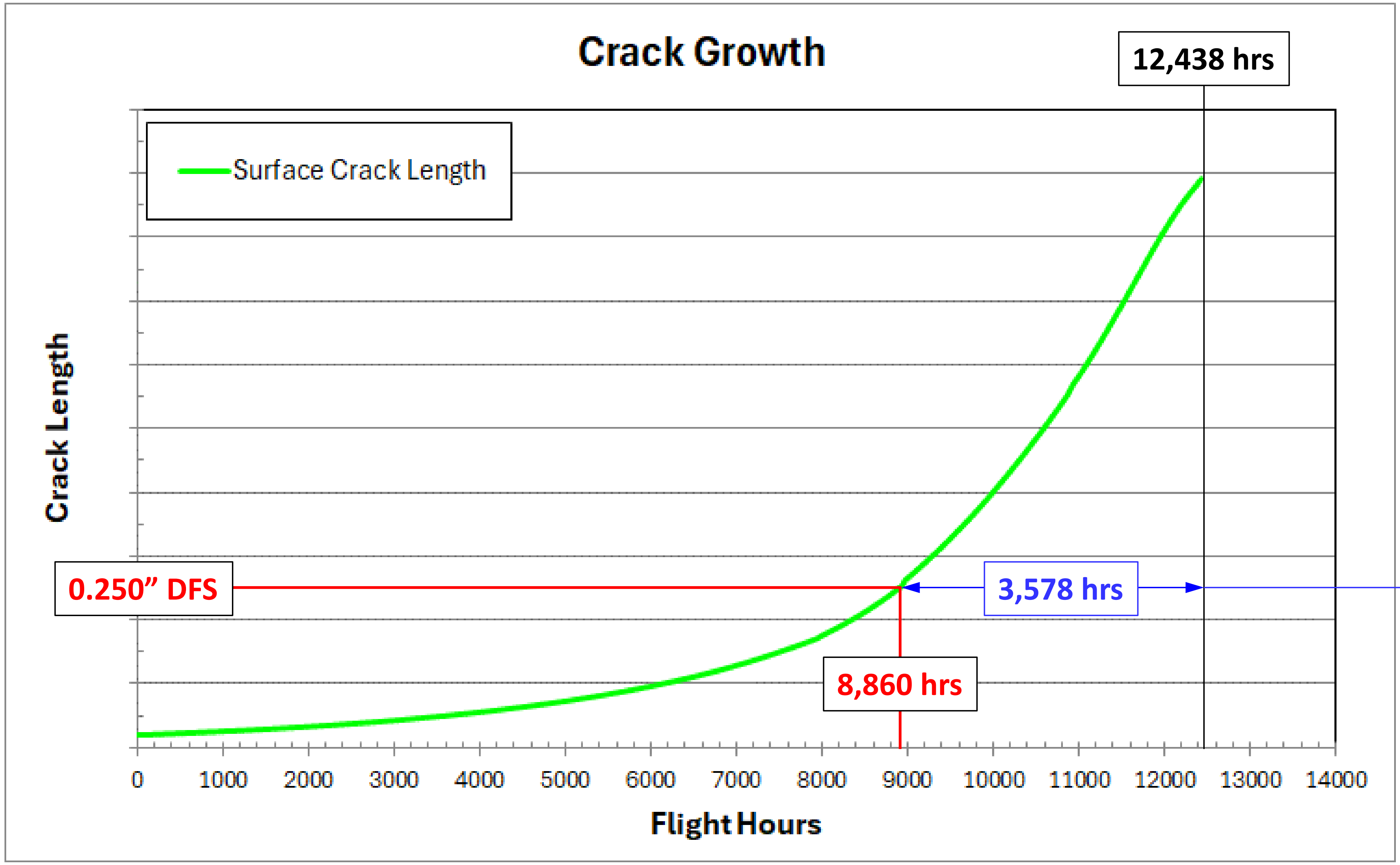


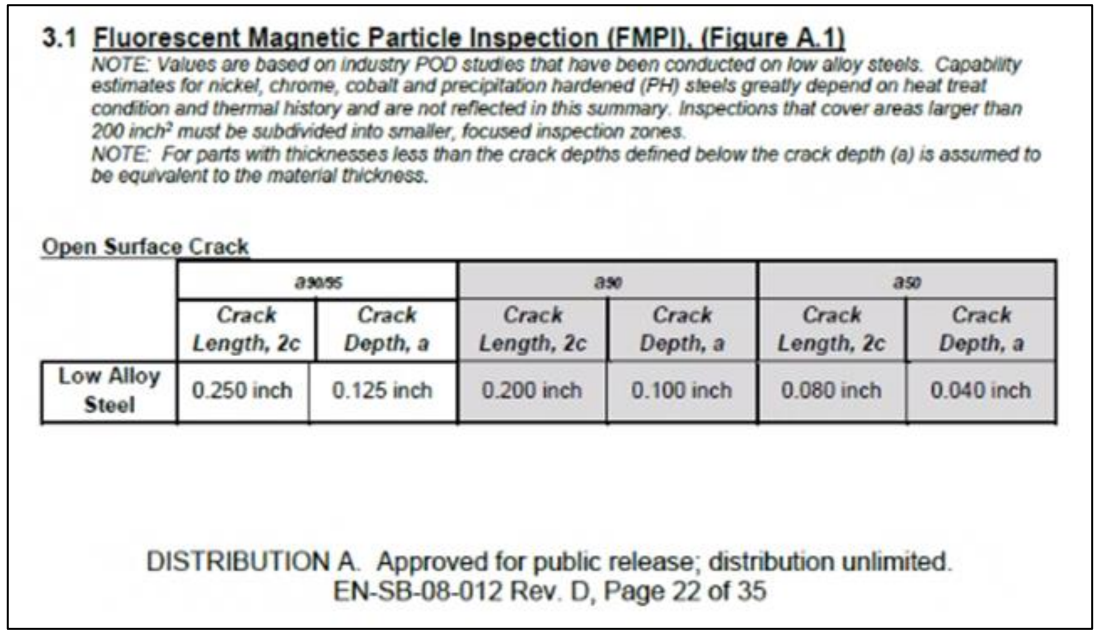
- Contour lines below show the BAMpF model crack front geometry for approximately every 1,000 flt-hrs.



1-Crack Model DTA

- $a_{90/95}$ detectable flaw size (DFS) for Fluorescent Magnetic Particle Inspection assumed to be 0.250” (Source: EN-SB-08-012 RevD)
- Recurring inspection interval does not meet desired 2,000 EFH to handle this inspection during depot induction





3.1 Fluorescent Magnetic Particle Inspection (FMPI), (Figure A.1)

NOTE: Values are based on industry POD studies that have been conducted on low alloy steels. Capability estimates for nickel, chrome, cobalt and precipitation hardened (PH) steels greatly depend on heat treat condition and thermal history and are not reflected in this summary. Inspections that cover areas larger than 200 inch² must be subdivided into smaller, focused inspection zones.

NOTE: For parts with thicknesses less than the crack depths defined below the crack depth (a) is assumed to be equivalent to the material thickness.

Open Surface Crack

	$a_{90/95}$		a_{90}		a_{50}	
	Crack Length, 2c	*Crack Depth, a*	*Crack Length, 2c*	*Crack Depth, a*	*Crack Length, 2c*	*Crack Depth, a*
Low Alloy Steel	0.250 inch	0.125 inch	0.200 inch	0.100 inch	0.080 inch	0.040 inch

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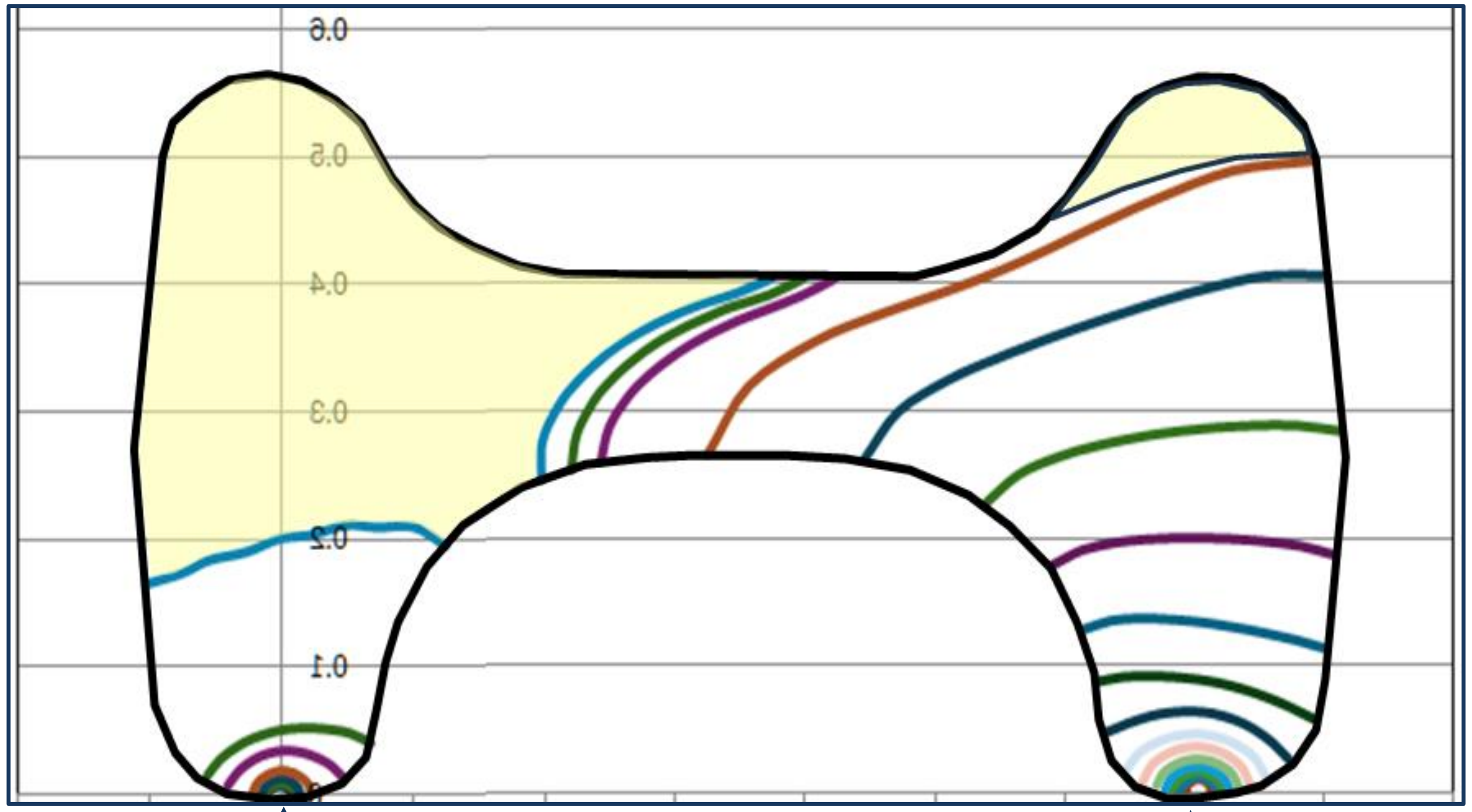
Life from 0.250” DFS:	3,578 hrs
➔ Recurring inspection interval:	1,193 hrs

2-Crack Model DTA

- A new 2-crack BAMpF model was created having semicircular cracks of sizes observed on failed part
  - 0.010” on the higher-stressed RHS
  - 0.005” on the secondary LHS

- Goal: grow the primary crack past the ≈12,400 cycles from single crack DTA model but include effects, if any, due to the presence of a secondary crack

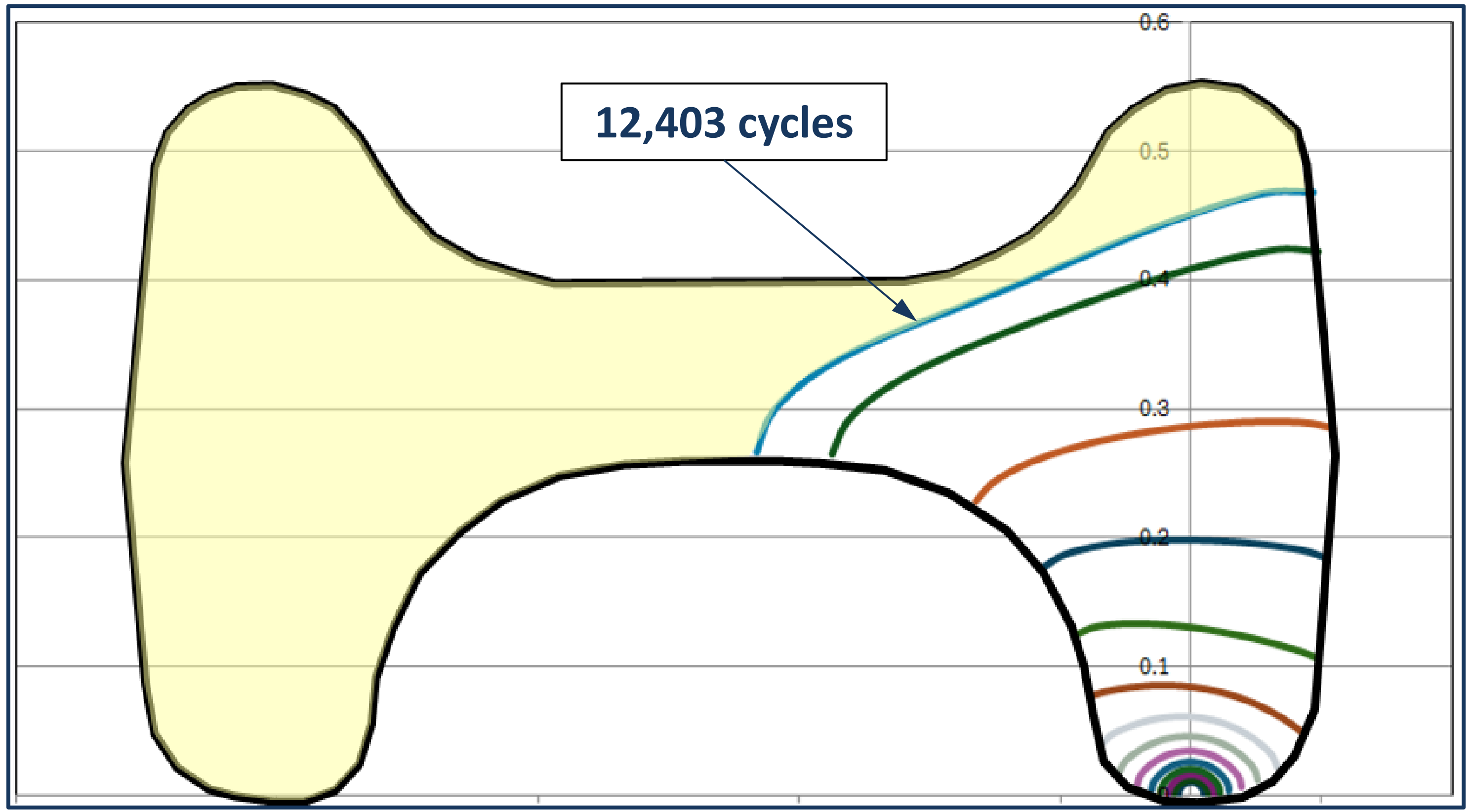
2-Crack BAMpF Model



0.005” semi-circular initial flaw

0.010” semi-circular initial flaw

Previous 1-Crack BAMpF Model

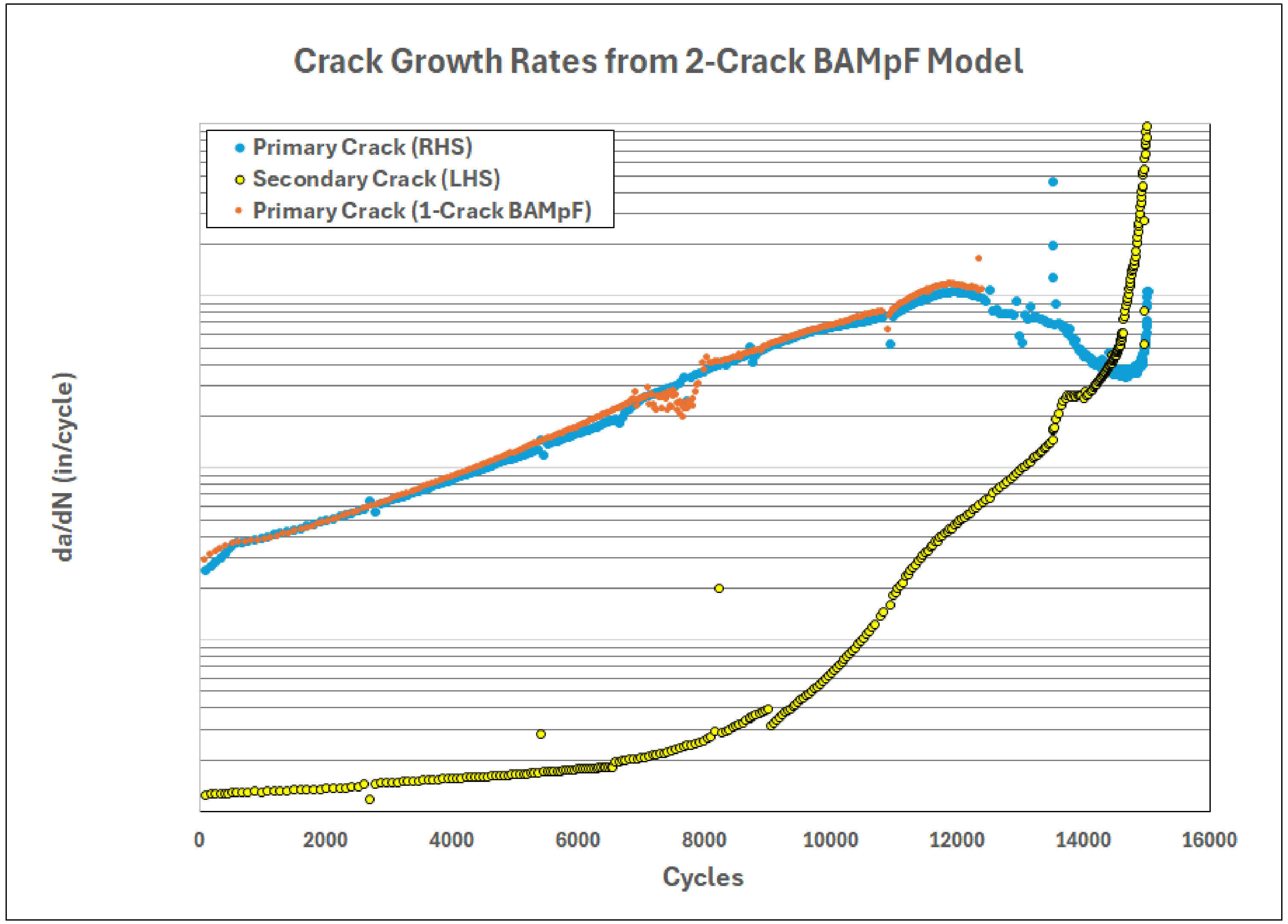


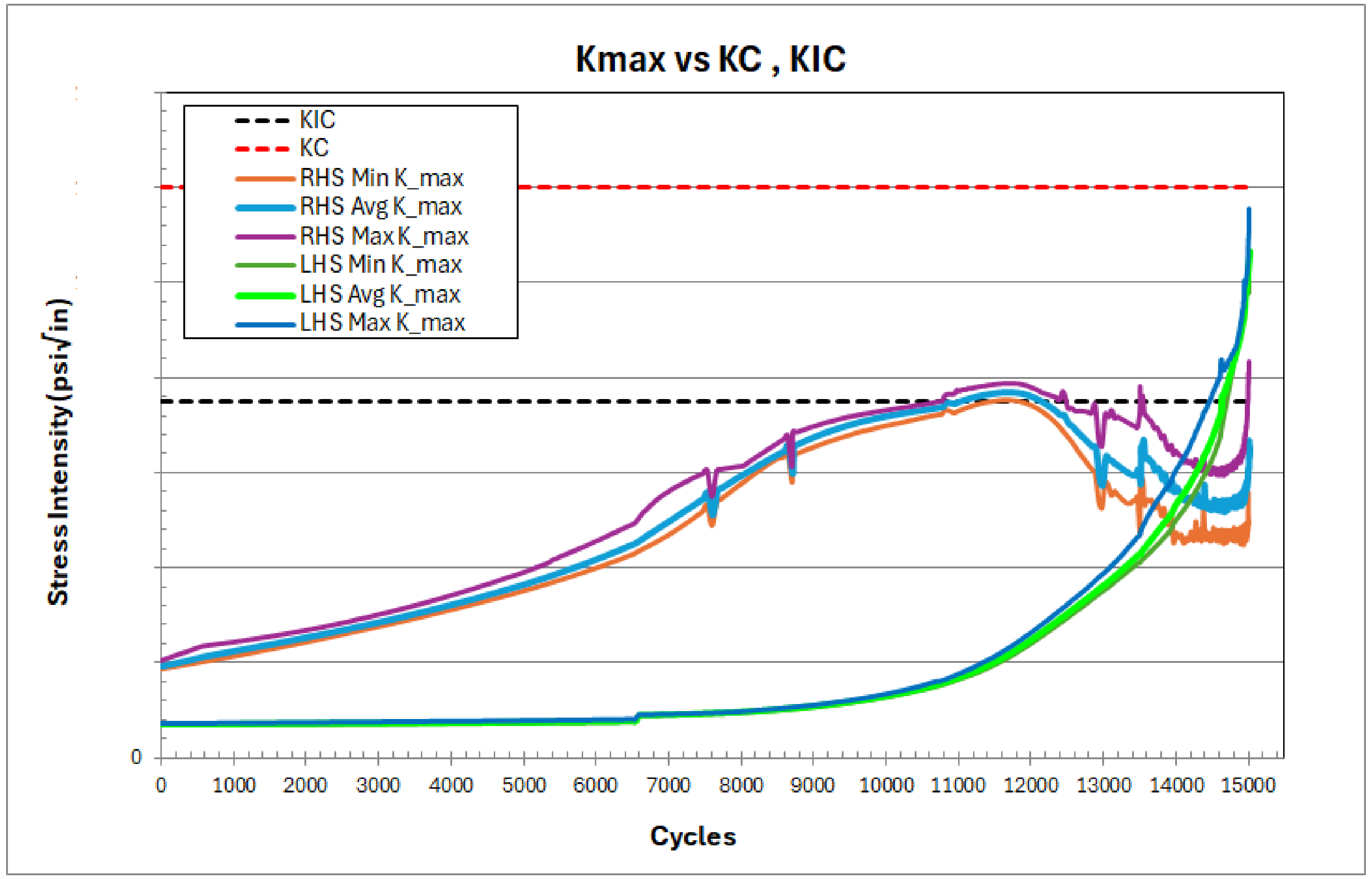
12,403 cycles

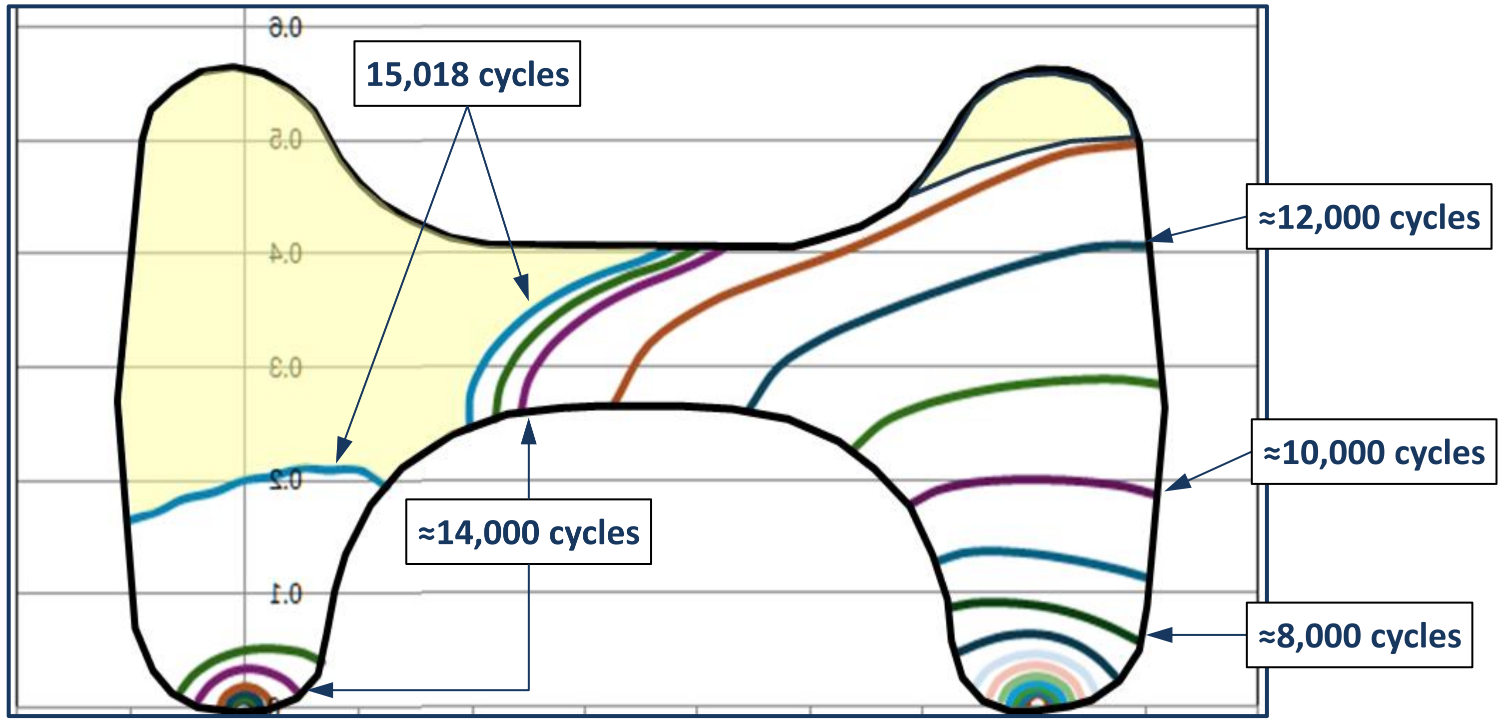
0.010” semi-circular initial flaw

2-Crack Model DTA

- Primary crack growth rate relatively unchanged up to 12K cycles between the 1-crack vs 2-crack BAMpF models
- Significant primary crack growth allowed before secondary crack begins growing
- Fracture assumed imminent near ≈15,000 cycles
    - Accelerating crack growth rates for both cracks
    - Both crack front K_{max} trending over K_{IC}

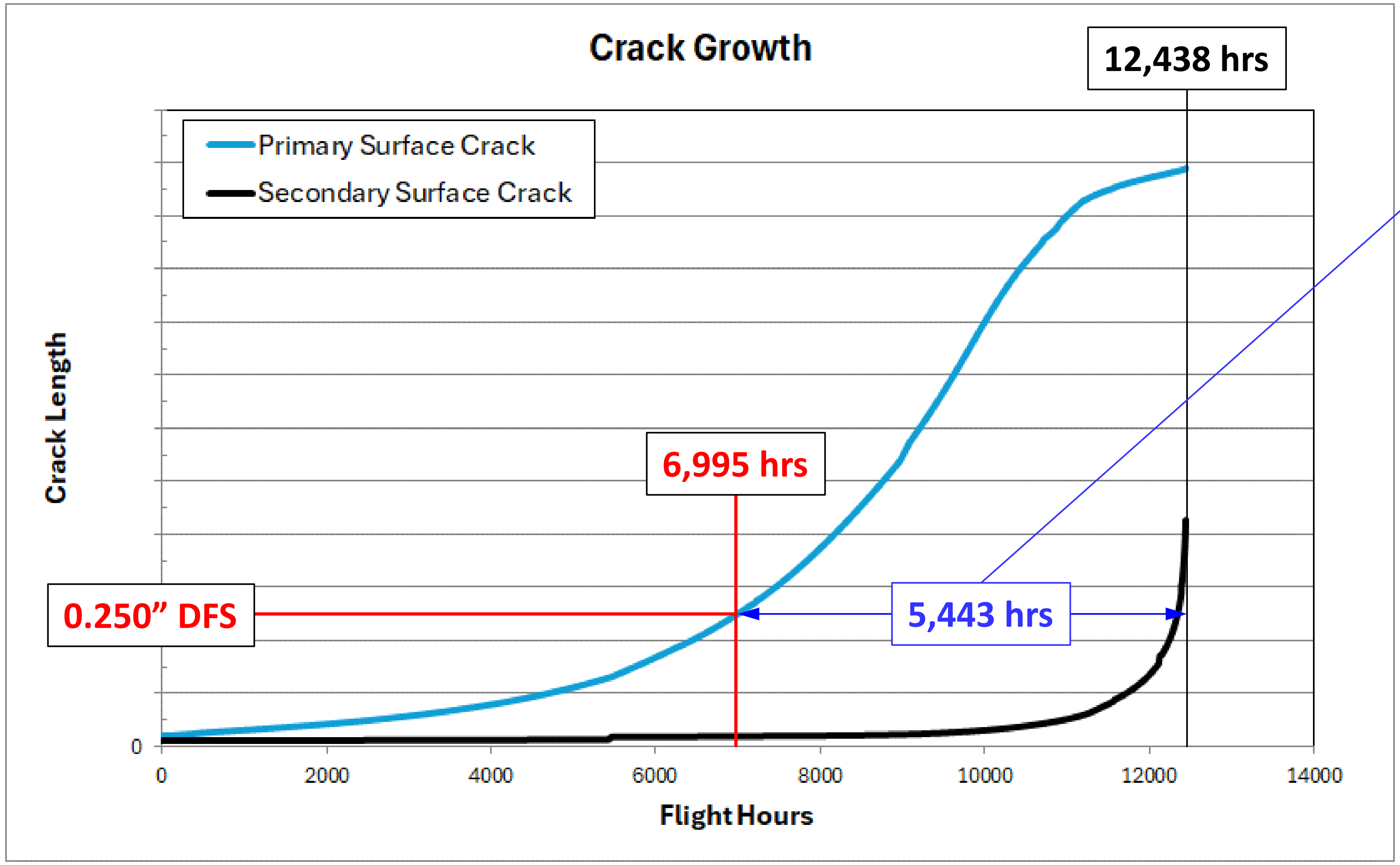






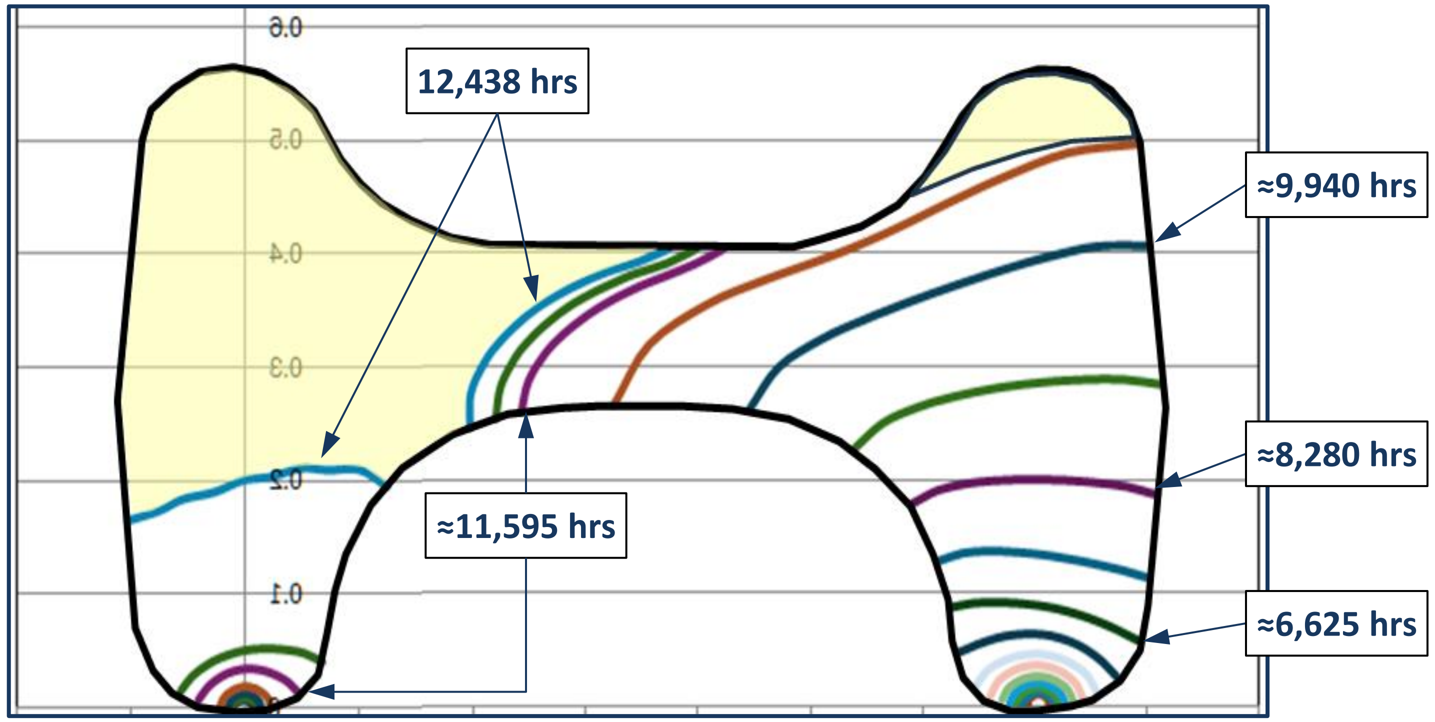
2-Crack Model DTA

- Scaling new total cycle prediction to the fleet average total flight hours requires a new cycle-to-hours rate of 15,018/12,438 ≈ 1.207 cycles/hr
- Prediction *may* explain corrosion seen over large area of RHS of fracture face—primary crack shape is large long before final fracture
- Recurring inspection interval has increased but still less than desired 2,000 EFH



Life from 0.250" DFS: 5,443 hrs

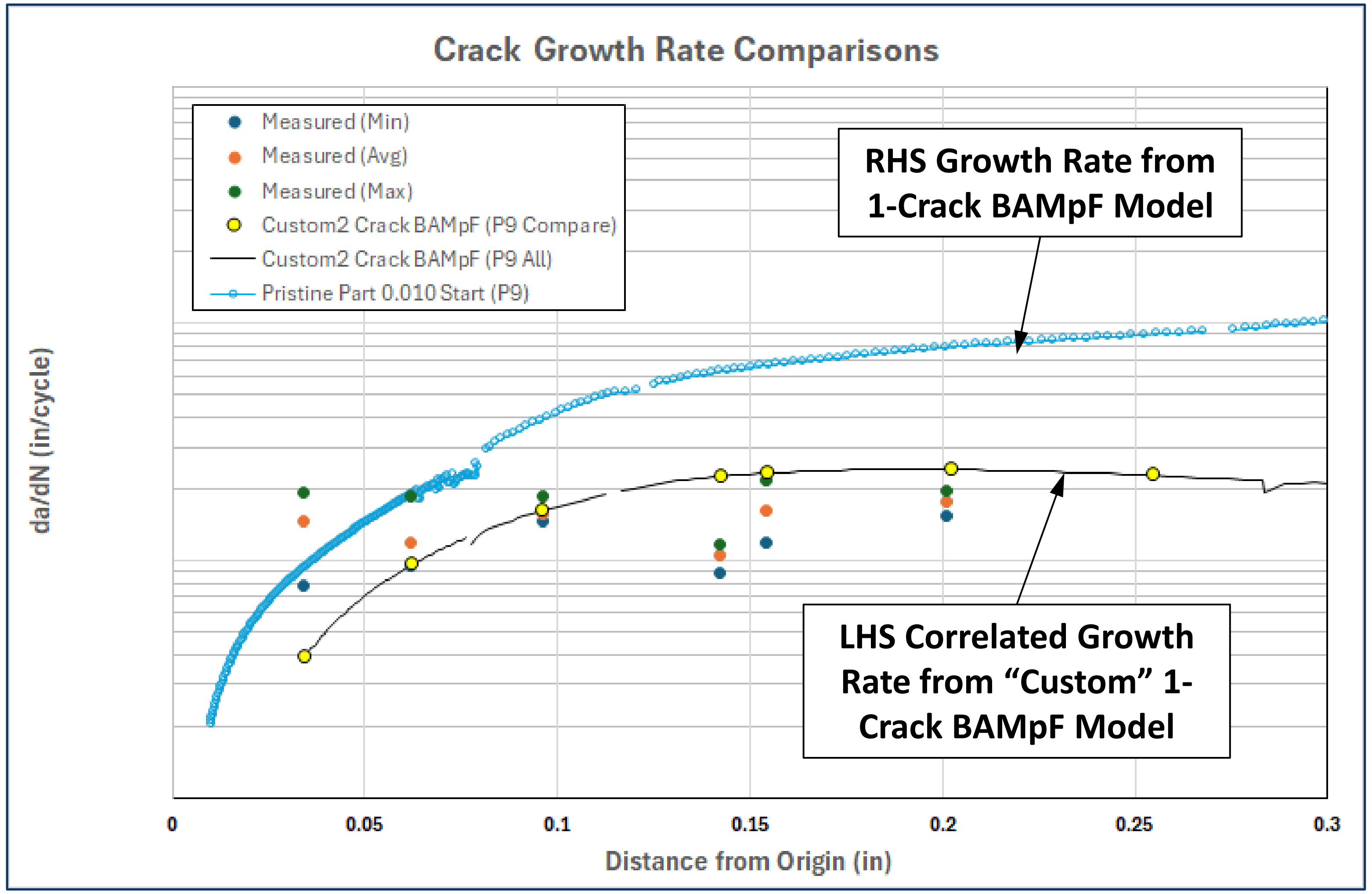
➔ Recurring inspection interval: 1,814 hrs



2-Crack Model DTA

- A couple discrepancies with the 2-crack model results should be addressed:
  1. Crack growth rate correlation
     - Load spectrum justified in part* by correlation to measured LHS crack data
     - LHS growth rates predicted for the single-crack analysis agreed with measured striations, whereas LHS growth rates predicted for the 2-crack analysis were much higher by comparison

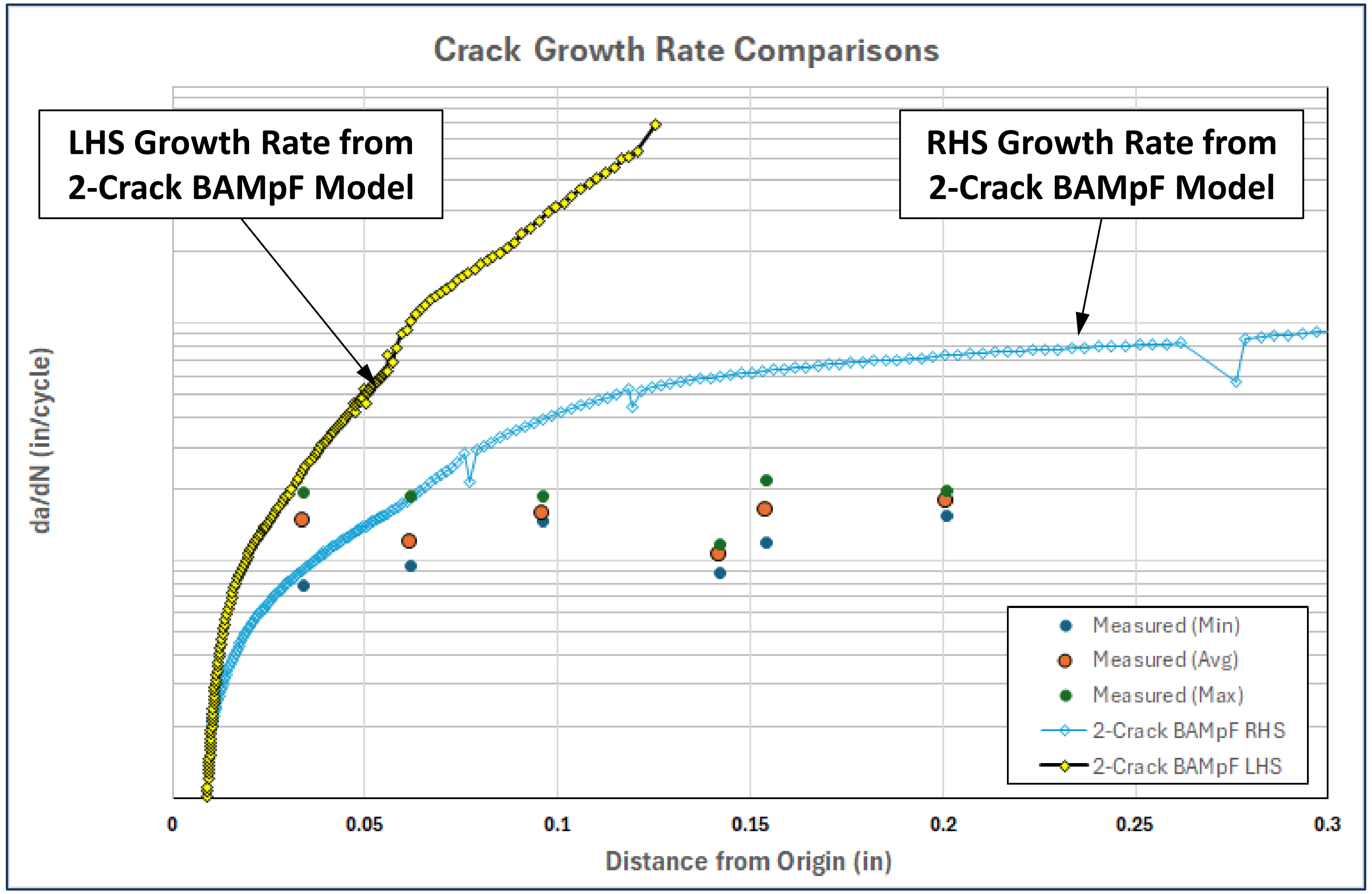
* Also used comparison of K_{max} vs K_{app} to justify peak spectrum load.



Crack Growth Rate Comparisons

RHS Growth Rate from 1-Crack BAMpF Model

LHS Correlated Growth Rate from “Custom” 1-Crack BAMpF Model



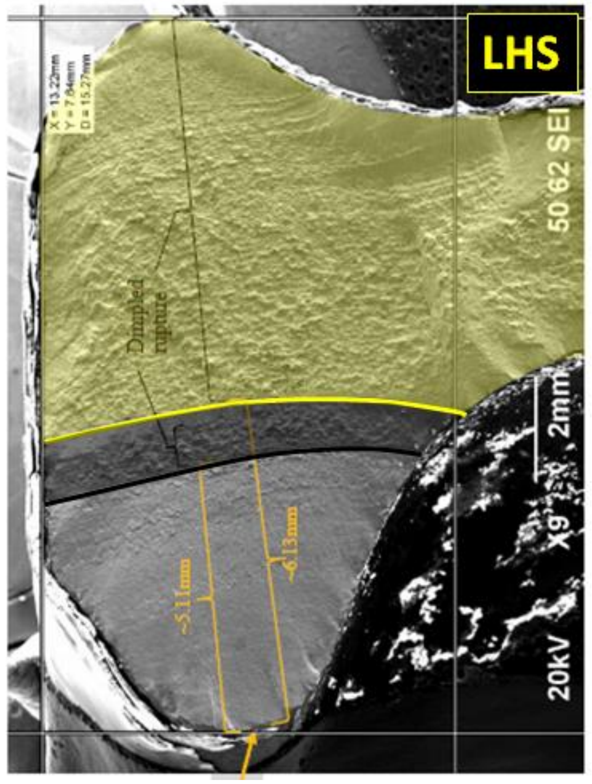
Crack Growth Rate Comparisons

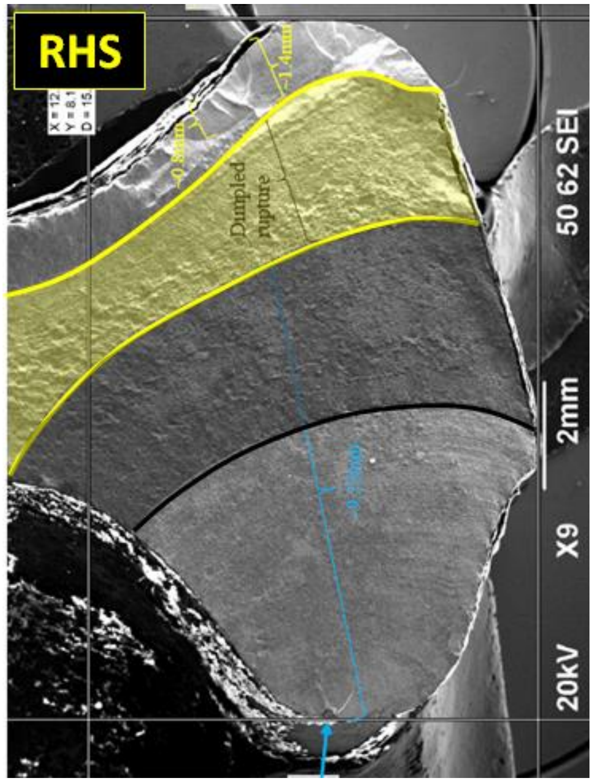
LHS Growth Rate from 2-Crack BAMpF Model

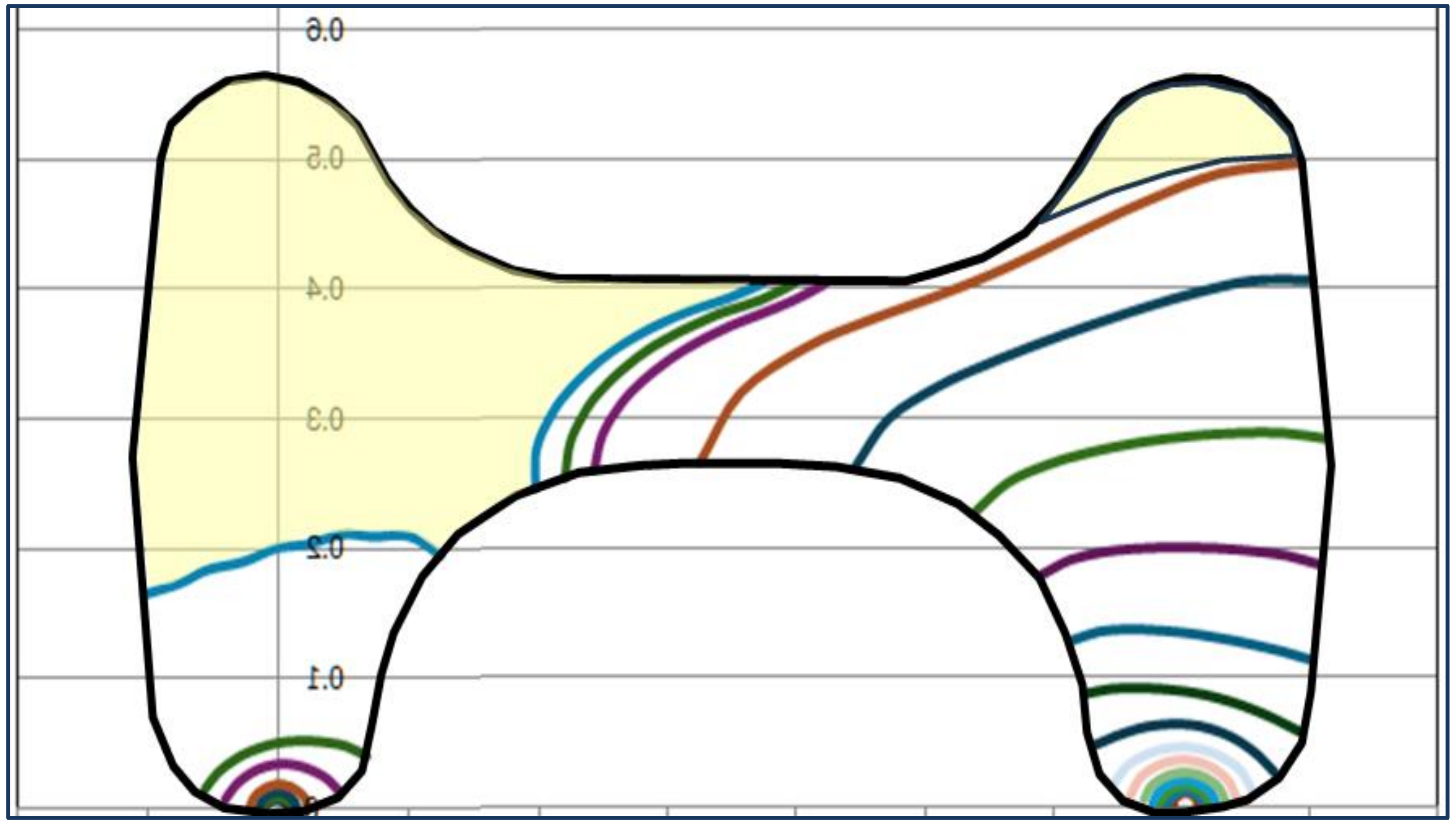
RHS Growth Rate from 2-Crack BAMpF Model

2-Crack Model DTA

- A couple discrepancies (continued):
  2. Crack shape predicted at final fracture does not match that originally assumed based on fractographic images
     - But analysis did not include reversed bending crack at top RHS
     - The corrosion on fracture face suggests nearly all of the RHS may have been cracked prior to final fracture







Concluding Remarks

- Failed NLG aft door hinge presented a unique challenge for assessing failure
  - Limited part history and unknown total time in flight
  - Unknown loading spectrum
  - Corrosion damage over nearly half the fracture face
    - Masked FCG data
    - Initially eluded explanation (did not make sense)

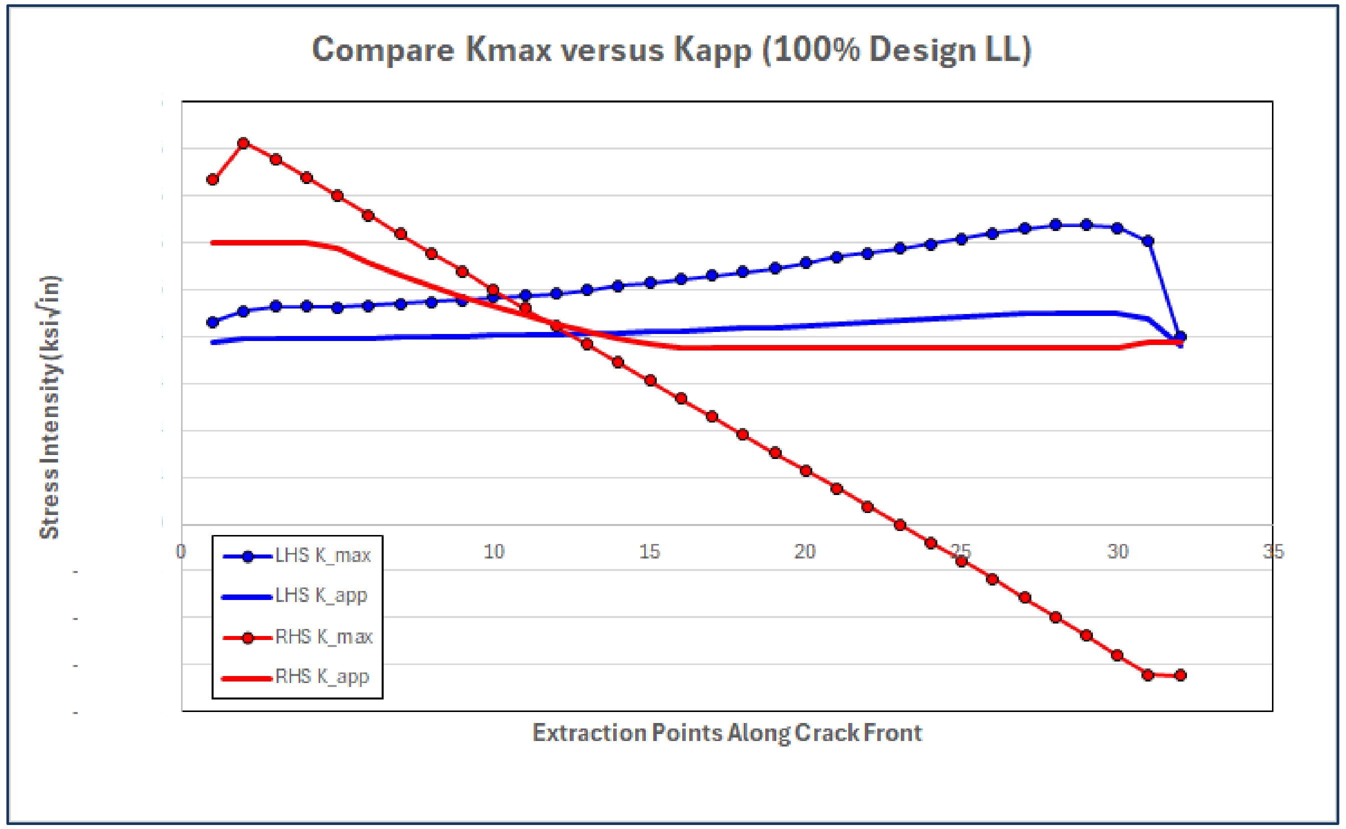
- Existing models and legacy design documentation enabled quick evaluation of relevant loading on hinge

- Multi-point fracture mechanics models (BAMpF) used to capture complex stress distribution and crack evolution

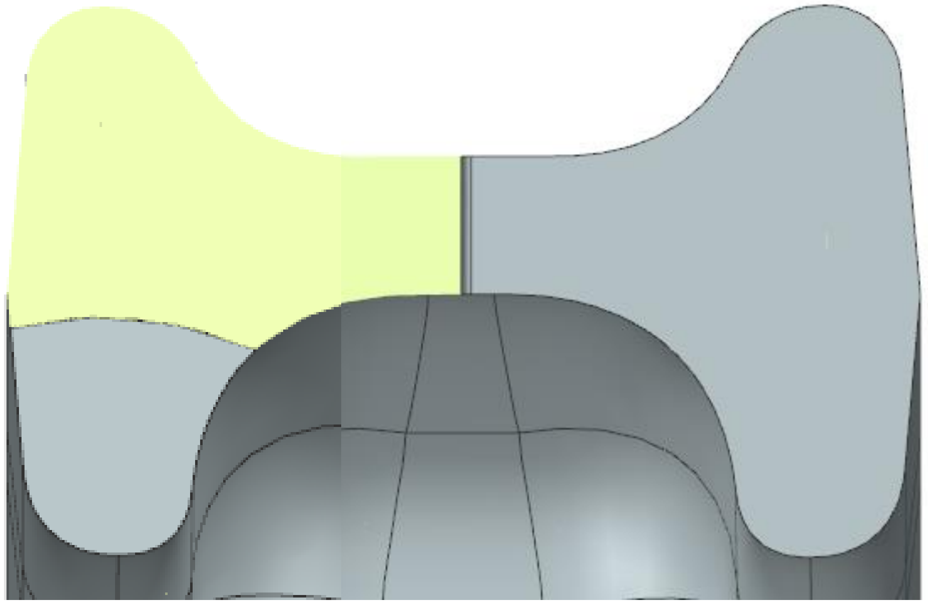
- Final results support the following conclusions:
  - Part retains capability even with a large primary crack → may explain the observed extensive corrosion
  - Predicted recurring inspection interval of 1,814 hrs less than the depot level requirement of 2,000 hrs, but may be minimal risk to extend inspections to depot cycle

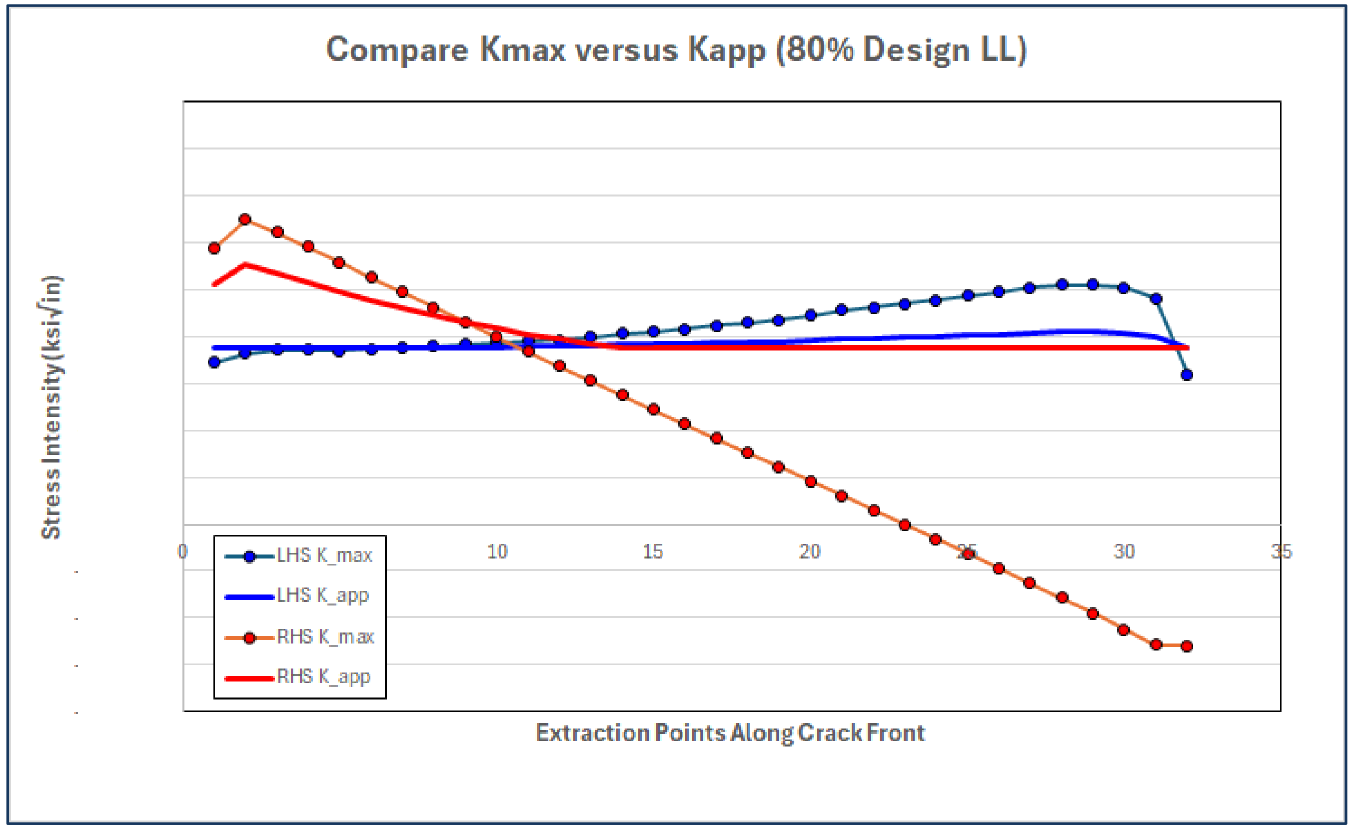
Questions?

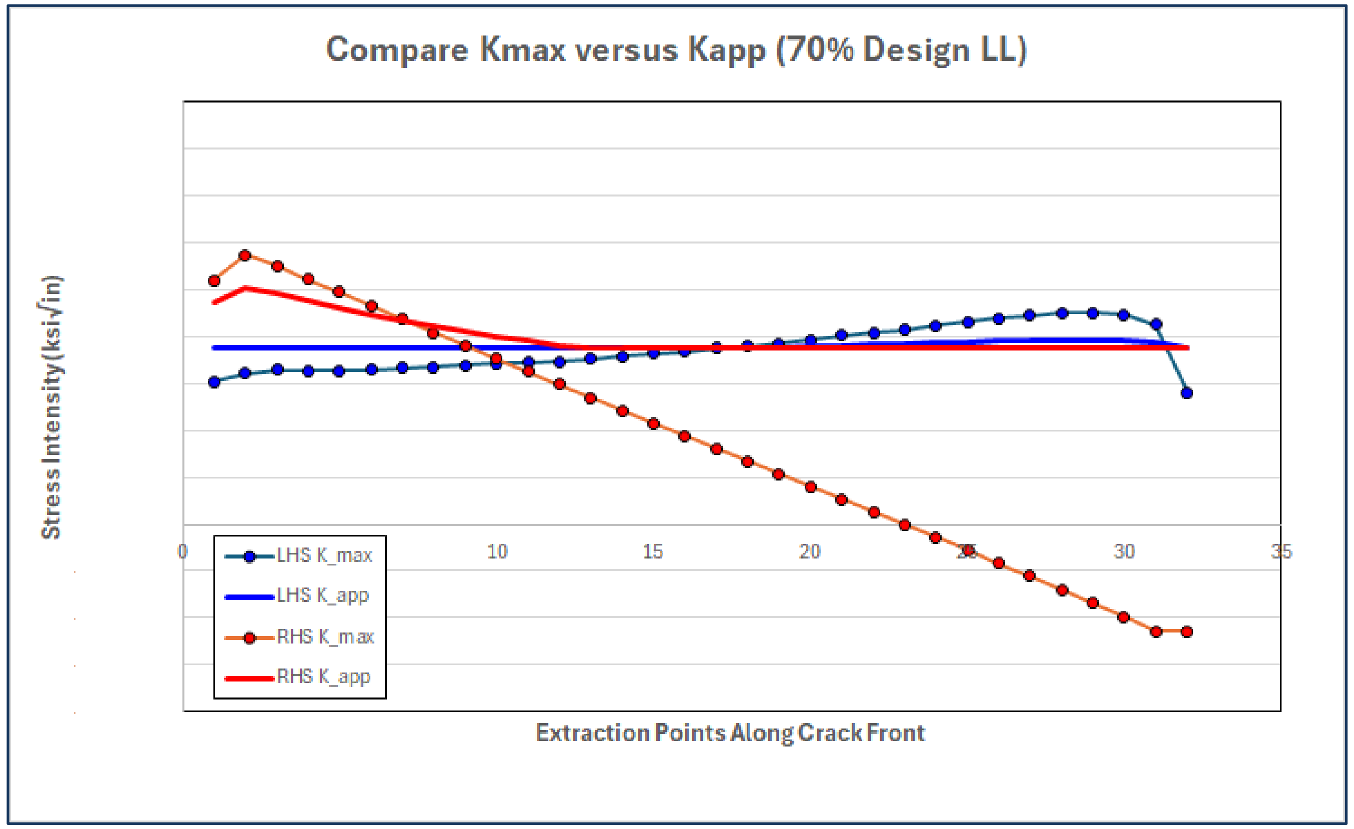
K_{max} versus K_{app} (Residual Strength)



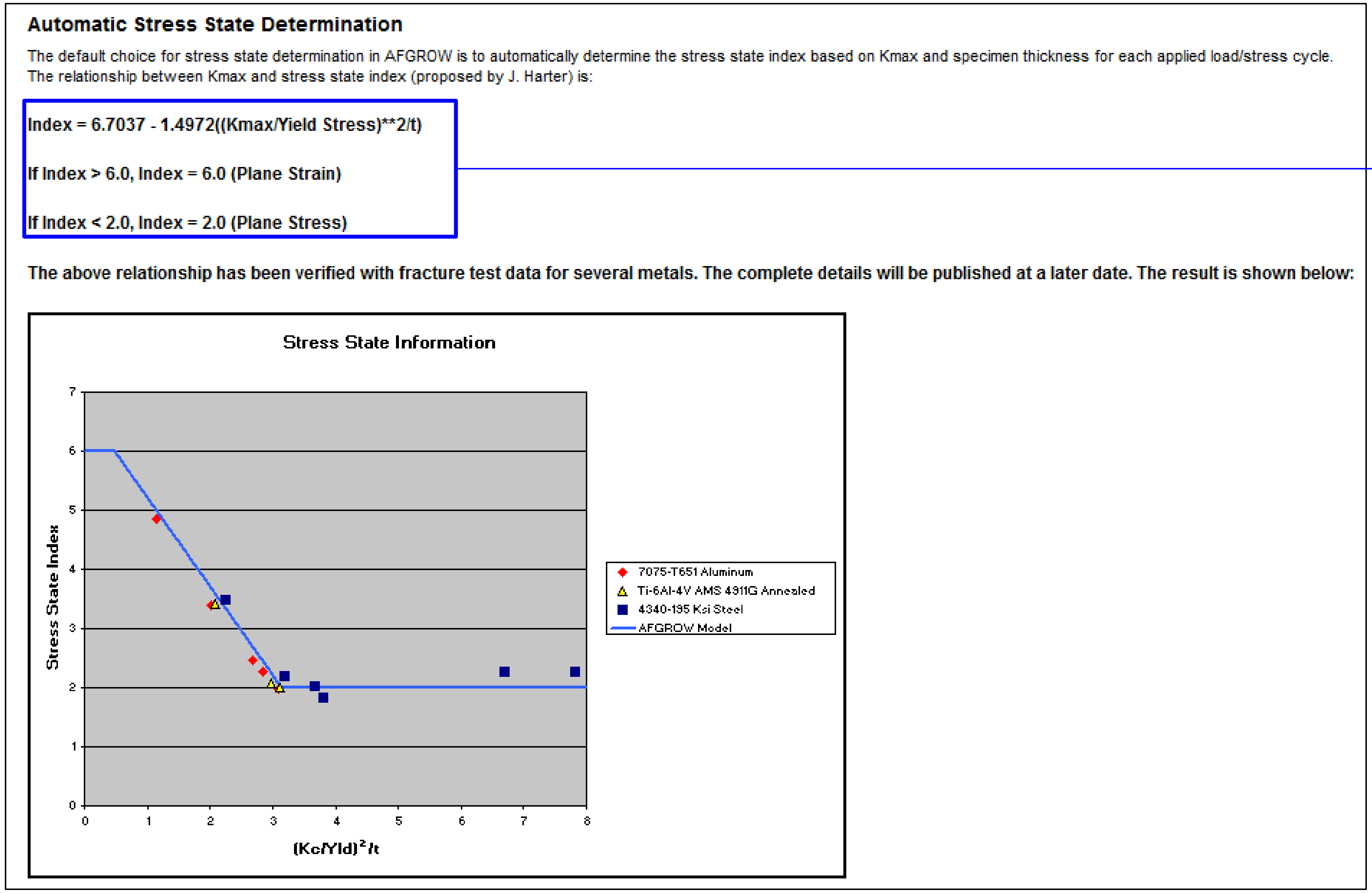
- **Half-cracked checks**







Calculation of K_{app} Using AFGROW Method



Automatic Stress State Determination

The default choice for stress state determination in AFGROW is to automatically determine the stress state index based on Kmax and specimen thickness for each applied load/stress cycle. The relationship between Kmax and stress state index (proposed by J. Harter) is:

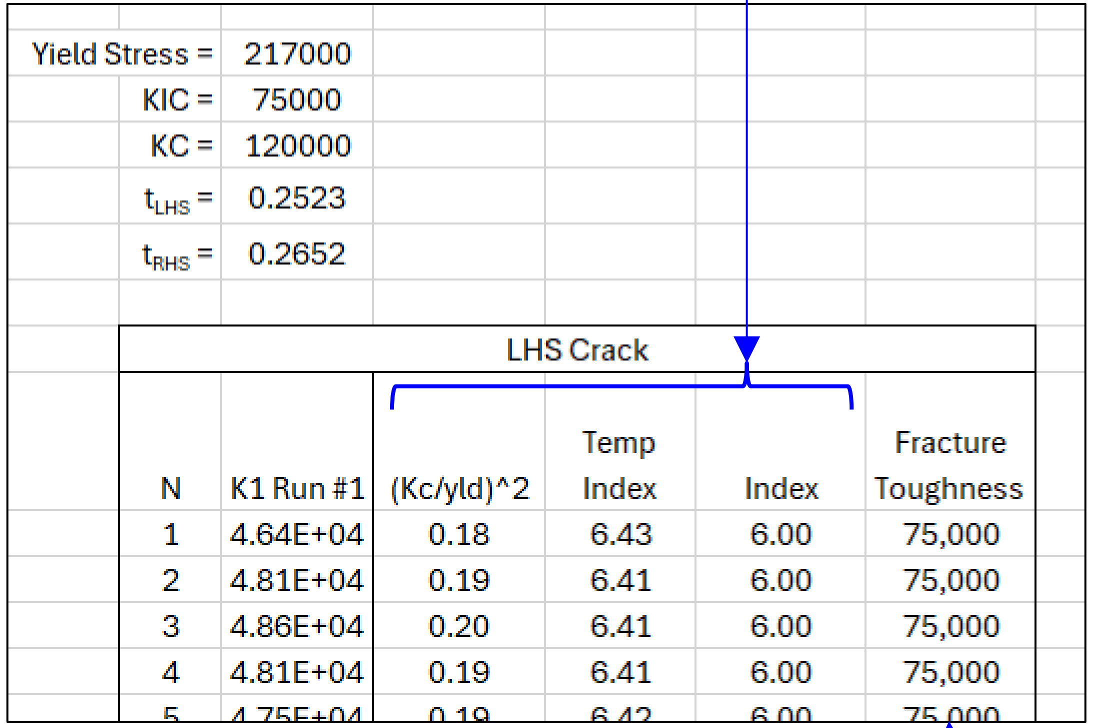
Index = 6.7037 - 1.4972((Kmax/Yield Stress)2/t)**

If Index > 6.0, Index = 6.0 (Plane Strain)

If Index < 2.0, Index = 2.0 (Plane Stress)

The above relationship has been verified with fracture test data for several metals. The complete details will be published at a later date. The result is shown below:

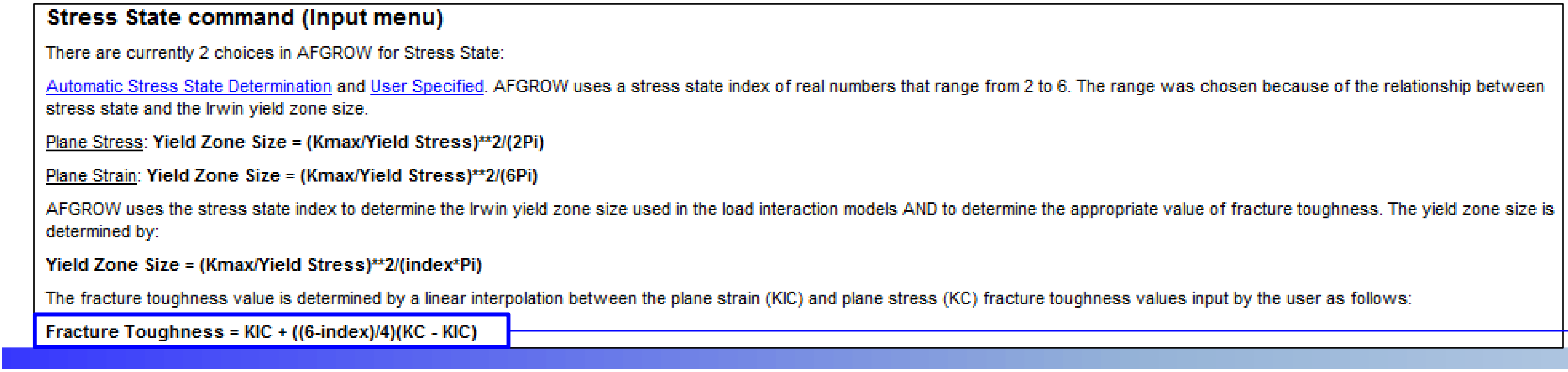
Stress State Information



Yield Stress =	217000				
KIC =	75000				
KC =	120000				
t_{LHS} =	0.2523				
t_{RHS} =	0.2652				

LHS Crack

N	K1 Run #1	(Kc/yld)^2	Temp Index	Index	Fracture Toughness
1	4.64E+04	0.18	6.43	6.00	75,000
2	4.81E+04	0.19	6.41	6.00	75,000
3	4.86E+04	0.20	6.41	6.00	75,000
4	4.81E+04	0.19	6.41	6.00	75,000
5	4.75E+04	0.19	6.42	6.00	75,000



Stress State command (Input menu)

There are currently 2 choices in AFGROW for Stress State:

Automatic Stress State Determination and User Specified. AFGROW uses a stress state index of real numbers that range from 2 to 6. The range was chosen because of the relationship between stress state and the Irwin yield zone size.

Plane Stress: **Yield Zone Size = (Kmax/Yield Stress)**2/(2Pi)**

Plane Strain: **Yield Zone Size = (Kmax/Yield Stress)**2/(6Pi)**

AFGROW uses the stress state index to determine the Irwin yield zone size used in the load interaction models AND to determine the appropriate value of fracture toughness. The yield zone size is determined by:

Yield Zone Size = (Kmax/Yield Stress)2/(index*Pi)**

The fracture toughness value is determined by a linear interpolation between the plane strain (KIC) and plane stress (KC) fracture toughness values input by the user as follows:

Fracture Toughness = KIC + ((6-index)/4)(KC - KIC)