PROBLEM NO. SIE-2

Title: Damage Tolerance Analysis of Wing Main Spars for Residual Strength

Objective:

To illustrate the process of estimating crack growth behavior to set inspection limits.

General Description:

This problem focuses on a damage tolerance assessment of cracks in a main wing spar, emanating from stringer cutout hole #13 in the center web between W.S. 8.5 and W.S. 17.5. The methodology used to analyze these cracks utilized Franc2DL (a layered finite element program) to propagate the crack(s) to obtain 'K vs. a' values and fastener transfer loads, NASGRO to compute the life of the cracked structure and 2024-T3 Crack Growth Resistance curves to verify residual strength at ultimate load.

Topics Covered: Damage tolerance assessment, stress intensity solutions using finite element analysis, crack growth analysis, crack growth resistance curves, residual strength, and inspection intervals

Type of Structure: wing main spar webs

Relevant Sections of Handbook: Section 4, 5, 9, 11

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Overview of Problem Description

This problem focuses on the ability of a wing main spar to carry residual strength with the center web cracked at the stringer cutouts. The geometry of the main wing spar in the area between W.S. 8.5 and W.S. 27 is a built-up structure consisting of a center web, upper cap, lower cap, and a pressure plate fore and aft. The center web and pressure plates are connected via steel angles that are fastened to the upper and lower caps and a 'zee', both fore and aft, that is fastened directly to the web and pressure plates at W.S. 17.5. The specific area is shown in Figures SIE-2.1 and SIE-2.2 with the expected crack path marked. Note that Figure SIE-2.2 shows the crack trajectory as calculated by Franc2DL.

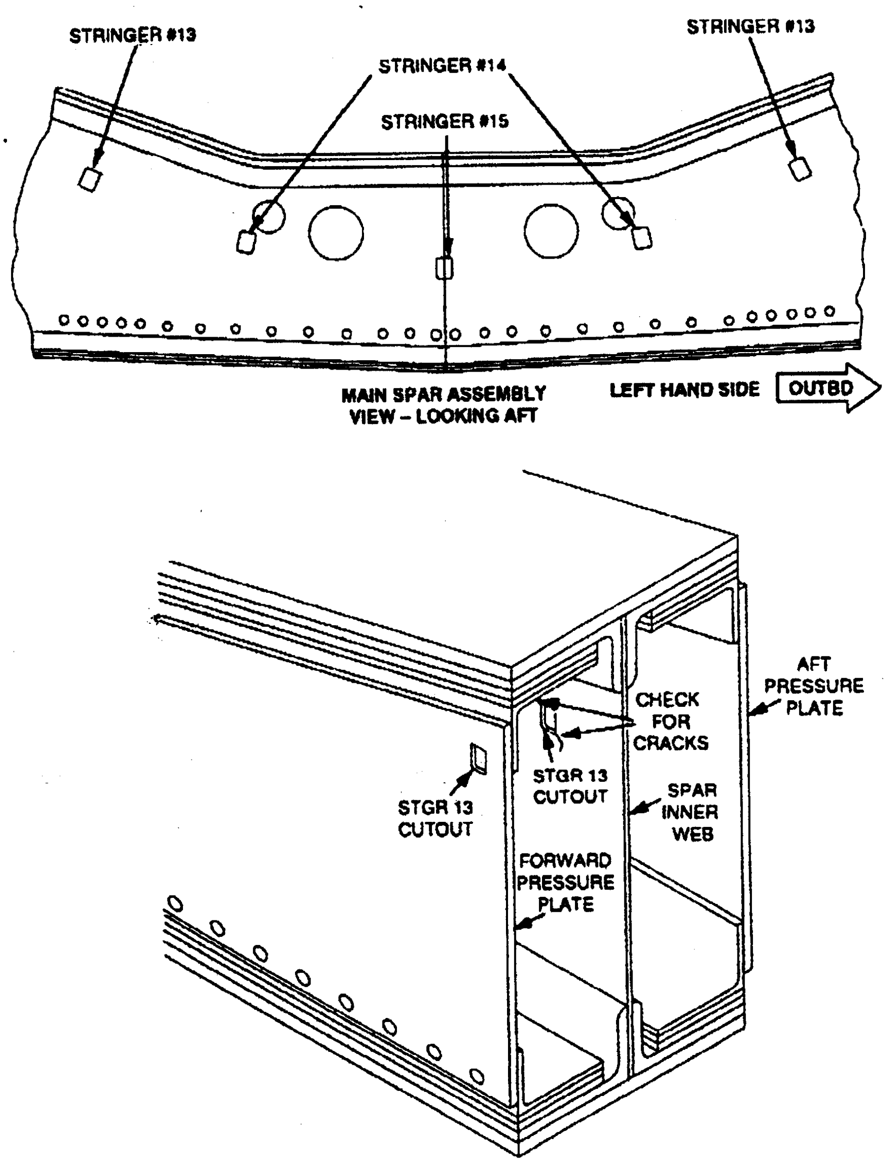

Figure SIE-2.1. Main Spar Assembly

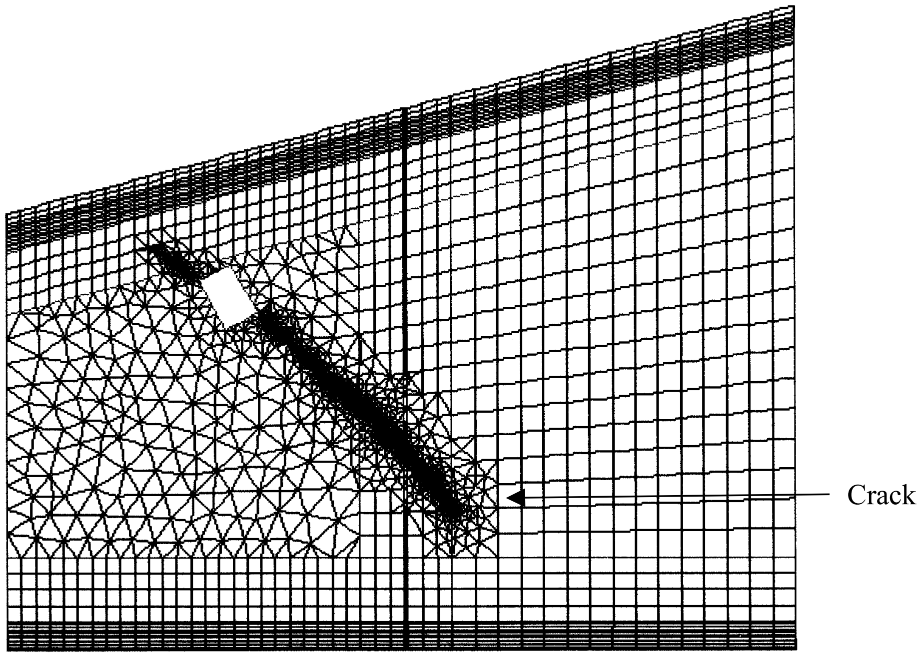

Figure SIE-2.2. Cracked Center Web as Calculated by Franc2DL.

Structural Model

Franc2DL models two-dimensional geometries with multiple layers fastened together. Therefore, the geometry of the Franc2DL finite element model involves the creation of multiple layers, each with equivalents areas as that of the structure being modeled. These layers and geometry are created in a meshing program, 'Casca', which are then incorporated via a conversion program, 'Casca to Franc', into Franc2DL.

Franc2DL limits the user to ten layers so a complex geometry such as this wing spar must be simplified in order to fit within that parameter. The final model has nine layers as shown in Figure SIE-2.3. These layers include a fore and aft pressure plate, a fore and aft, upper and lower steel 'reverse' angle which incorporate the adjacent straps, a center web which incorporate the steel angles along with the upper cap and lower cap and straps and a fore and aft 'Zee'.

The material properties for each element within a layer are defined individually to account for changes in material type and thickness. Layers are fastened together via rivets, which are actually finite element springs for which the user must define the stiffness.

Note that Franc2DL is used to propagate the crack(s) to obtain 'K' vs. 'a' values and fastener loads. This is input as tabulated data into NASGRO3.0 to compute crack growth life.

Idealized Franc2DL Finite Element Geometry

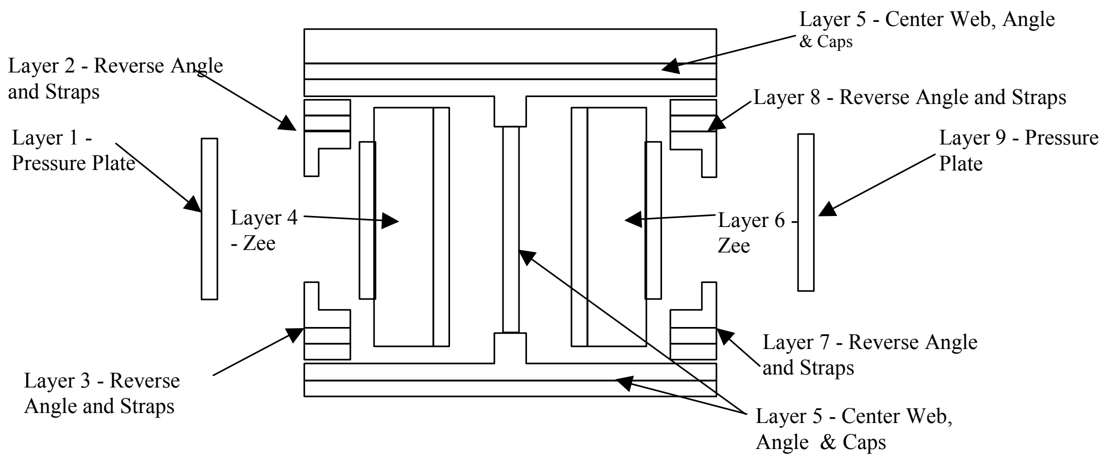

Figure SIE-2.3. Detail Geometry of Critical Location Shown in Figure 2.1.

Model Geometry Description

The crack growth analysis is based on the Fatigue Crack Growth Computer Program NASGRO3.0. This computer program calculates crack growth for a single crack for several standard crack cases. Crack growth rate calculations use the "NASGRO" equation. This is a modified Forman equation to account for stress ratio effects, and upper and lower fatigue crack growth rate asymptotes. The NASGRO3.0 material libraries are used for the material data in the analysis.

Stress intensity factors, K, and corresponding crack lengths are taken from the Franc2DL model and are used for the calculation of crack growth life. The K values from Franc2DL are converted into α values via the equation:

$$K = \beta\theta\sqrt{\pi a}$$

$$\alpha = \beta\theta$$

$$K = \alpha\sqrt{\pi a}$$

$$\alpha = \frac{K}{\sqrt{\pi a}}$$

The α and crack length values are input into NASGRO3.0 using the data tables option for a one-dimensional data table for through cracks, DT01, with a unit stress. Table SIE-2.1 shows the crack lengths, corresponding K values, and calculated α values for the outboard crack.

Table SIE-2.1. NASGRO Input Values for DT01.

a, in.	K, ksi√in.	α
0.05	11.45	28.89
0.1194	20.13	32.87
0.3194	21.48	21.44
0.5194	20.99	16.43
0.7194	22.03	14.65
0.9194	22.23	13.08
1.0194	22.94	12.82
1.2194	23.62	12.07
1.4194	24.47	11.59
1.6194	24.66	10.93
1.8194	25.91	10.84
2.0194	26.52	10.53
2.2194	27.14	10.28
2.5194	28.73	10.21
2.7194	28.26	9.67
3.0194	29.55	9.59
3.2194	29.95	9.42
3.5194	30.92	9.30
3.7194	31.65	9.26
4.0194	32.07	9.02
4.2194	32.77	9.00
4.4194	32.88	8.82
4.6194	33.41	8.77
4.8194	32.94	8.47
5.0194	34.41	8.67

Inspection Capabilities and Crack Limits

The wings will be inspected for cracking using visual techniques. To maintain a generous margin on the residual strength requirement, as well as avoiding the potential for alternate load path concurrent damage, crack lengths of two (2) inches or less were prudently selected to define tolerable limits. At the fastest crack growth rate, the proposed limitation of a 370 hrs/550 flights would allow for further propagation to a length of 2.5 inches. The proposed limitations furthermore require five intervals of repetitive crack monitoring inspections during this interval.

Structural Loading and Stress History Description

Initial loads reflect loads at W.S. 27 and 1.5g's for use in this analysis. The 1.5g loads represent an initial estimate at the simplified equivalent ground-air-ground flight spectrum for every flight. The 1.5g loads were then converted into shear stress using V/A and bending stress using the flexure formula Mc/I. The shear stresses were applied to layers 1,5, and 9 (layers with webs) while the bending stresses were applied to all layers except layer 4 and 6 (Zee's).

All loads were applied at W.S. 27. Subsequent tuning of these loads to bound the fatigue test crack growth rates in the 2-3 inch crack length regime led to factoring them up to 1.5g x 1.05 = 1.575g's or approximately 1.6g values.

An additional factor of 1.433 was applied to account for the apparent increase in the service spectrum severity beyond that of the test. This factor was derived by applying a fourth power law to the maximum difference between the test-to-field crack life. The calculation may be found in the Verification of Life Analysis Section.

The final GAG cycle used was then 1.433 x 1.575g = 2.26g's per flight. Since the K versus a data from the Franc2DL runs were already at 1.5g's, the combined total bump factor of 1.505 (1.505 x 1.5 = 2.26) was implemented via the stress scaling factor in the NASGRO runs.

In order to avoid operation at crack lengths promoting significant load redistribution and therefore possible concurrent damage or overloading of the secondary load paths, fastener transfer loads were monitored throughout the analysis. The change in fastener transfer loads was shown to be minimal, especially in tolerable crack length regime of less than or equal to 3.0 inches.

It is noted that the critical fasteners (lower spar cap tensile field) are somewhat relieved until the long crack lengths are generated. The subordinate fasteners (upper cap compressive field), which do feel the immediate detrimental effect of the cracking, are still well within their shear allowable at ultimate load; and furthermore, they transfer their load into the structure which is in overwhelming compression.

Material Property Description

In Franc2DL, materials can be assigned to each element individually. Material properties that are user defined for the models in this analysis are as follows; Young's modulus, Poisson's Ratio, and thickness. The values used are shown below for the various Young's modulus and Poisson's ratio.

Table SIE-2.2. Material Properties and Growth Rate Data.

Material	Young's Modulus	Poisson's Ratio
2024-T3 Aluminum	10.3E+06	0.35
Steel	29.0E+06	0.30
Titanium	17.4E+06	0.36

The material properties from the NASGRO3.0 libraries are used for the fracture toughness and the crack growth rate properties for the crack growth life determination. The material properties used are for 2024-T3; Clad, Plate and Sheet; T-L; LA & HHA NASGRO material code M2EA12AB1.

Table SIE-2.3. Material Properties and Growth Rate Data.

MATL 1: 2024-T3
Clad Plt & Sht; L-T; LA & HHA

Material Properties:

Matl No.	UTS	YS	K1e	K1c	Ak	Bk	Thk	Kc	Keac
1	66.0	53.0	46.0	33.0	1.00	1.00	0.050	65.9	

Matl No.	Crack Growth Eqn Constants: C	n	p	q	DKo	Cth+	Cth-	Rcl	Alpha	Smax/SIGo
1	0.829D-08	3.284	0.50	1.00	2.90	1.50	0.10	0.70	1.50	0.30

Solution Technique

This type of problem is conveniently solved using Franc2DL and NASGRO3.0 as previously discussed. The cycles for the crack growth life will be converted into hours with the assumption of 0.67 hours per GAG cycle.

Results

Critical crack size/Residual Strength

The crack lives calculated by NASGRO were found to be 8,539 cycles (5,721 Hours) for the outboard crack. These lives correspond to initial crack lengths of 0.05 inches propagating to 5.03 inches. The maximum applied stress intensity was found to be 51.77 ksi√in. The fracture toughness, K_c for the material was computed to be in the range of 115 ksi√in. This is a typical value for wide panel data. Since the stress corresponding to 2.26g developed these applied stress intensities, the residual strength at ultimate load is verified. To maintain a generous margin on the residual strength requirement, the tolerable crack length regime of less than or equal to 3.0 inches was established.

Residual strength at ultimate load has been shown for the 3.0 inch crack length by superimposing the crack growth resistance ('R') curve for the 2024-T3 web onto the 4.5g ultimate crack curve anchored at the 3.0 inch crack length. It can be seen that the applied K curve is well below the crack growth resistance curve for cracks larger than 3.0 inches. Thus, no crack will run catastrophically and cause failure under the ultimate load.

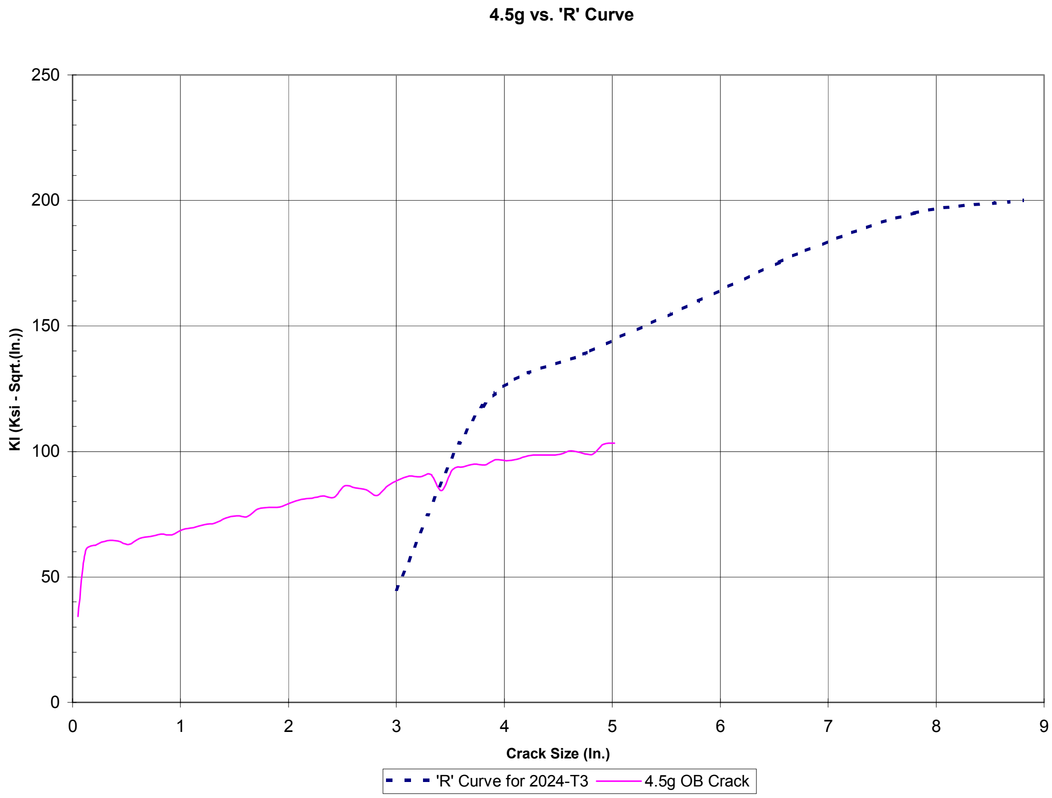

Figure SIE-2.4. Crack Growth vs. Resistance Curve.

Life:

Based on the calculations for growing the crack in NASGRO the life from initial crack size to failure is determined to be 8,539 cycles (5,721 Hours). The results are shown in Figure SIE-2.5. The life is given in numbers of cycles and hours.

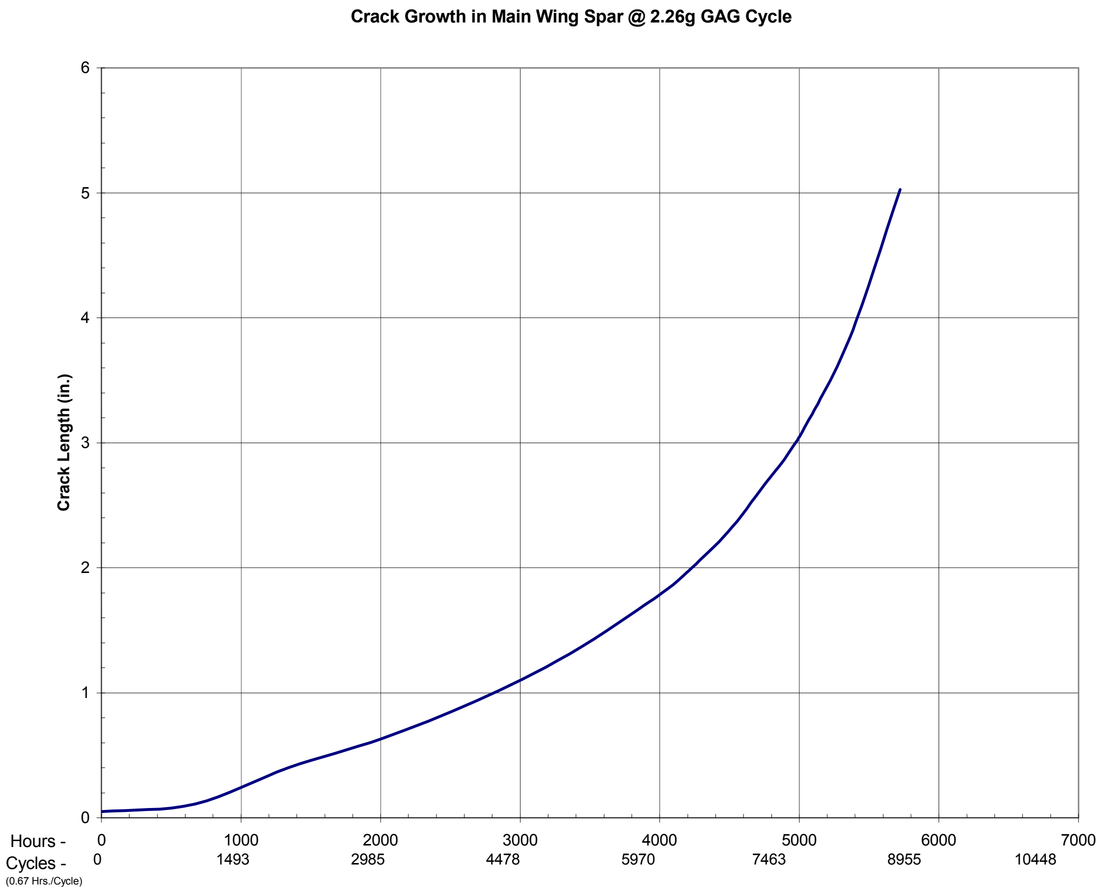

Figure SIE-2.5. Crack Growth Life for Problem SIE-2.

Verification of the Life Analysis

As mentioned previously, the following 4th power law calculation is used to derive a correlation factor between the fatigue test data and the field observations. The worst case (largest numerical correlation factor) is then used to scale the stress intensity factors input into NASGRO.

Table SIE-2.4. Test Data.

SWACO Test to Service Data Correlation Factor Derivation

226 Field Data	hrs	cycles	Field data crack size R IB	SWACO test cycles to field crack size	Test to Service correlation factor	Field data crack size R OB	SWACO test cycles to field crack size	Test to Service correlation factor	Field data crack size L IB	SWACO test cycles to field crack size	Test to Service correlation factor	Field data crack size L OB	SWACO test cycles to field crack size
TC 300	29769	46355	0.75	no data		4.00	130453	1.295	1.00	no data		4.25	74303
TC 358	23165	37219	1.75	63113	1.141	1.5	76184	1.196	2.5	68224	1.164	2.5	47276
TC 280	23577	30976	3.5	78362	1.261	4	130453	1.433	3.5	83749	1.282	4	66409

sample calculation of 4th power law correlation factor derivation:

test cycles/(field once-per-flt-stress/test once-per-flt-stress)^4 = field cycles

or (test cycles/field cycles) = (field once-per-flt-stress/test once-per-flt-stress)^4

or (130453/30976)^.25 = 1.433 for the worst case.

Worst Case : 1.433 (R. Outboard Crack from test and Outboard Crack from TC 280)

Inspection Intervals

A scatter factor of two (2) will be applied to all safe flight limits. The 2.0 inch crack would reach 3.0 inches in 740 hours/ 1100 flights; applying this scatter factor results in flight limits of 370 hours/ 550 flights (whichever occurs first). The repeat inspection intervals are determined according to the established limitation requiring five intervals of repetitive crack monitoring inspections. The repeat inspection intervals are calculated at 75 hours/110 flights, whichever occurs first.