PROBLEM NO. SIE-3

Title: Crack Growth Analysis of Main Cargo Door Surround Doubler Attachment to Fuselage Skin with Primary and Continuing Damage Cracks

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 a main cargo door surround doubler attachment to fuselage structure for the purpose of establishing inspection intervals for crack growth with primary and continuing damage cracks growing from opposite sides of a hole. The critical area includes the main cargo door surround doubler and the existing fuselage skin. The stresses acting at the doubler attachment at the upper edge are derived from a conservative loading spectrum based on pressure loading. The critical area was modeled using a standard NASGRO 3.0 stress intensity factor solution and crack growth model.

Topics Covered: Damage tolerance assessment, crack growth analysis, inspection intervals

Type of Structure: fuselage skin, main cargo door surround doubler

Relevant Sections of Handbook: Sections 2, 5, 11

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Overview of Problem Description

This problem focuses on the main cargo surround doubler attachment to the existing fuselage skin at stringers 2R and 26L. The skin is considered to be a single load path structure under the total hoop stress before the doubler attachment. The critical location is in the skin at the first row of fasteners because the skin sees both bypass and bearing stresses at this row, where as, at the other fastener rows the load is in both the doubler and the skin with each row having lower load transfer.

The fuselage skin was fabricated from 2024-T3 aluminum. The fasteners are 0.188 in diameter, and join the skin and surround doubler.

The specific area is shown in View A of Figure SIE-3.1, with the specific details and the primary and continuing damage cracks shown in Figure SIE-3.3. Note that the skin at this first row of fasteners is a single load path as shown in Figure SIE-3.2.

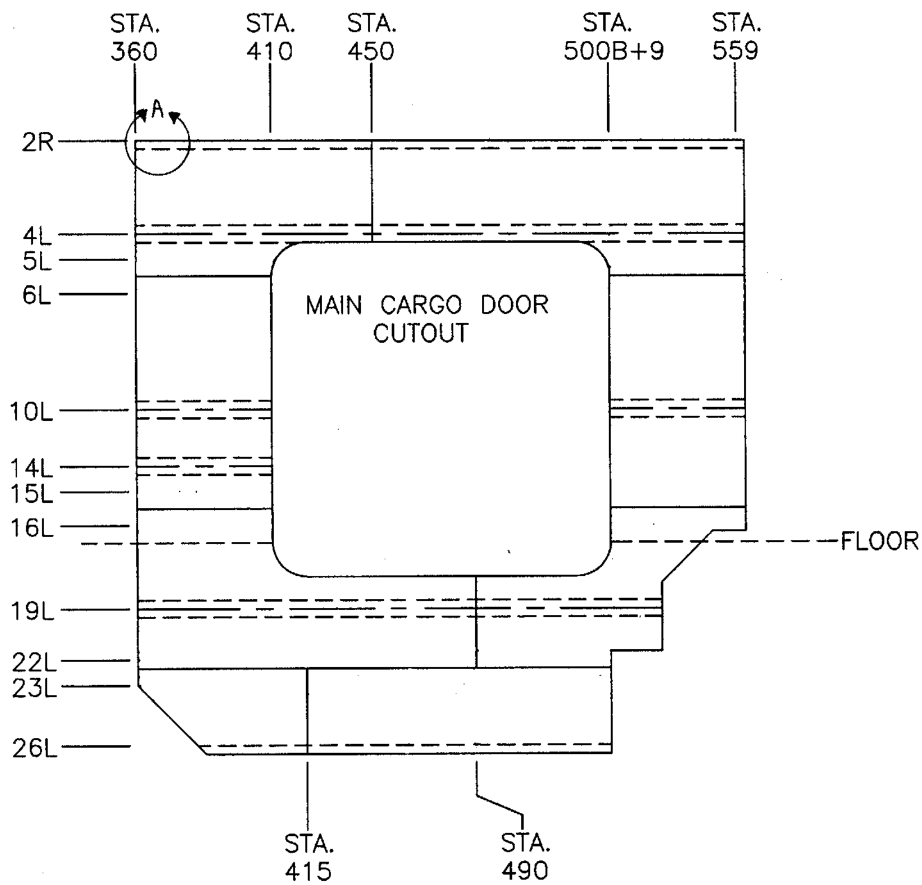

Figure SIE-3.1. Main Cargo Door Doubler Installation

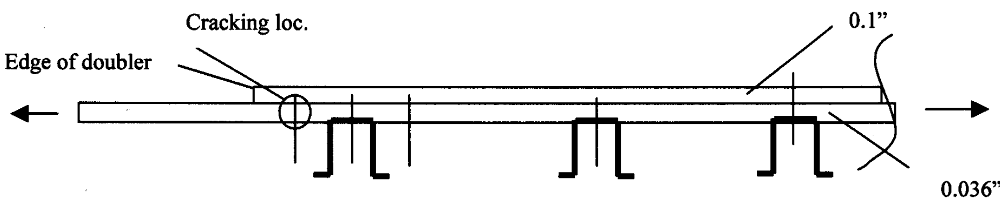

Figure SIE-3.2. Structural Detail for Critical Area

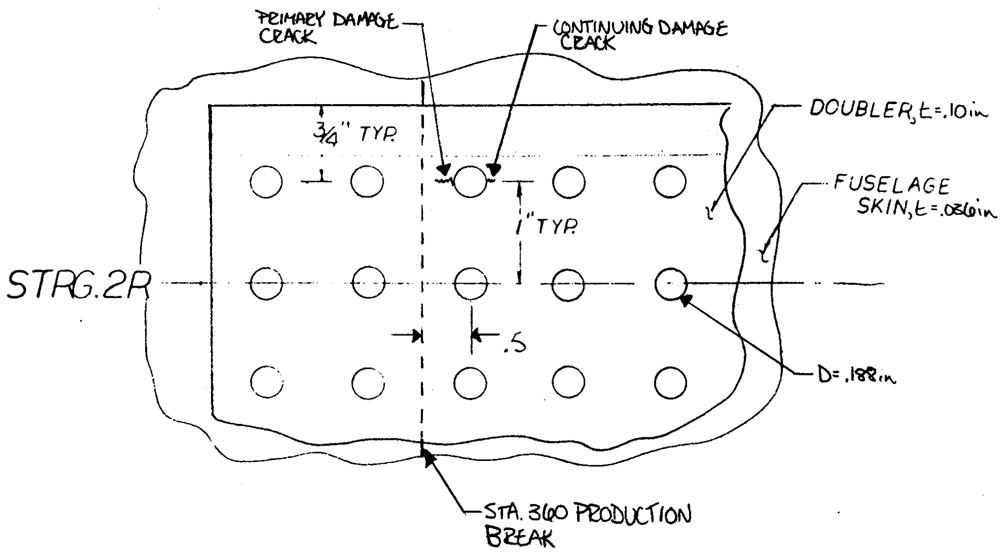

Figure SIE-3.3. Detail Geometry of Critical Location, View A.

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 with elements developed by Forman, Newman, de Koning, and Henriksen (see NASGRO reference manual). This is a modified Paris equation to account for fatigue crack closure, stress ratio effects, and upper and lower fatigue crack growth rate asymptotes for threshold and critical crack growth.

The analysis uses the NASGRO3.0 material libraries for the crack growth rate equation constants. Non-interaction of loads and constants for the Forman crack growth rate equation are used.

Since the standard crack models in NASGRO3.0 are for crack growth of single cracks, no influence of one crack upon another is calculated in NASGRO3.0 for these standard cases. The analysis presented here includes crack interaction effects between the primary damage crack and the continuing damage crack. This is accomplished by iterating though a series of NASGRO3.0 computer runs tracking the growth of both cracks and modifying the stress intensity factors appropriately. The increased stress intensity factors are based on the crack sizes of the interacting cracks from the previous iteration and correction factors based on the compounding of analytical stress intensity solutions.

This iteration procedure is accomplished in an Excel® Spreadsheet utilizing Visual Basic Programming to submit a NASGRO3.0 computer run for each crack at each iteration. The spreadsheet reads the NASGRO3.0 output files for cycles and current crack lengths. Based on these crack lengths, correction factors are calculated and input into the NASGRO3.0 input file for the next iteration, which is automatically submitted by the spreadsheet.

The correction factors are accounted for by increasing the stress scaling factors input into NASGRO3.0. These increased stress scaling factors can be input based on the following:

$$K = S\beta_{CF}\beta_N\sqrt{\pi a} = S'\beta_N\sqrt{\pi a}$$

where,

$\beta_N =$ Beta for the standard NASGRO crack model

$\beta_{\mathrm{CF}} =$ Beta calculated from the correction factor

$S' = S\beta_{CF} =$ increased stress scaling factor input into NASGRO

These correction factors for crack interaction account for interactions between the primary damage crack (rogue flaw) and the continuing damage crack. This is done assuming both cracks are in the same part.

Note that interactions between the rogue flaw and the continuing damage crack have historically not been done in crack growth analysis. This method of including these interactions from the onset of the crack growth of the rogue flaw is conservative since it does not account for any fatigue life due to the nucleation of the continuing damage crack.

As previously discussed, the correction factors for crack interaction are based on comparison of analytical stress intensity solutions. The correction factor for this analysis is termed “INT”, and is used for unequal length cracks growing from opposite sides of a hole.

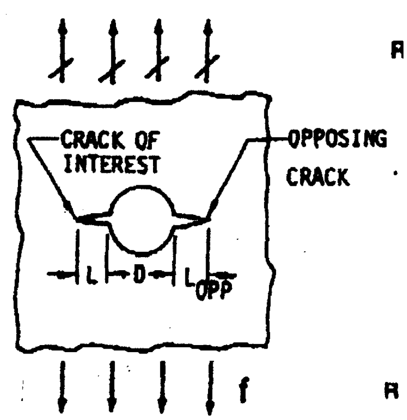

Figure SIE-3.4. INT correction factor

These correction factors are based on through the thickness cracks. They are used for part through cracks when defined with an equivalent crack length. The equivalent crack length is based on equating the area of a part through crack as a quarter ellipse to that of an equivalent through crack as a rectangular area with thickness, t:

$$A_{pc} = A_{eq} = \frac{\pi ac}{4} = a_{eq}t \Rightarrow a_{eq} = \frac{\pi ac}{4t}$$

The "INT" correction factor is derived based on comparing the stress intensity solution of a center cracked panel for two different crack lengths, a_1 and a_2. Including the diameter of the hole, D, in the total crack lengths, yields:

$$a_1 = D + a + a_{opp}$$

$$a_2 = D + a$$

$$INT = \frac{K_1}{K_2} = \frac{\sigma\sqrt{\pi a_1}}{\sigma\sqrt{\pi a_2}} = \sqrt{\frac{a_1}{a_2}} = \sqrt{\frac{D + a + a_{opp}}{D + a}} = \sqrt{1 + \frac{a_{opp}}{D + a}}$$

The crack growth model for the main cargo door surround doubler attachment to the fuselage skin at stringer 2R employs the NASGRO3.0 corner crack from a hole centered in a plate, CC02, with the correction factors for the influence of unequal length cracks growing from opposites sides of a hole. When the initial crack reaches the edge of the plate, the crack growth is continued in the opposite direction as a through crack from the edge of a plate, TC02.

The crack growth model CC02 was used with the following dimensional values.

$t = 0.036$ *in.*

$W_a = 220.0$ *in.*

$D = 0.188$ *in.*

$B_a = 0.5$ *in.*

$a_A = 0.050$ *in.*, primary damage crack

$a_B = 0.005$ *in.*, continuing damage crack

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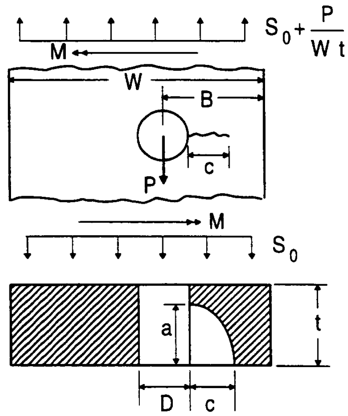


Figure SIE-3.5. NASGRO3.0 Crack Model, CC02.

Two NASGRO files are created for the primary damage crack and the continuing damage crack and submitted to the Excel interaction spreadsheet. The spreadsheet accesses NASGRO and grows both cracks for 100 flights. The β correction factors are calculated for the crack lengths at that time and the resulting increased stress scaling factors are plugged back into the NASGRO files. The interaction spreadsheet grows the two cracks until there is a 10% change in the primary damage crack length (this could also be done in increments of flights), recalculates the β correction and stress scaling factors, and continues to grow the cracks until the primary damage crack reaches the edge of the plate.

Once the primary damage crack reaches the edge and transitions into an edge crack growing in the opposite direction, the crack growth model TC02 was used with the following dimensional values.

$t = 0.036\ in.$

$W = 220.0\,in.$

$$c = B_a + \frac{D}{2} + c_B = 0.5 + .094 + 0.1729 = 0.7669\ in.$$

where, c_B = surface length of continuing damage crack when primary damage crack reaches the edge.

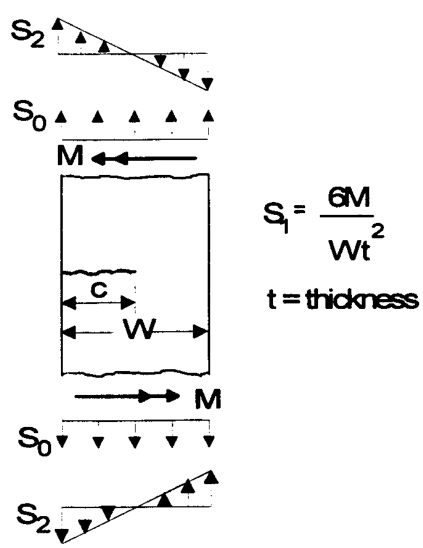


Figure SIE-3.6. NASGRO3.0 Crack Model, TC02

<u>Inspection Capabilities and Crack Limits</u>

The holes in the fuselage skin at the attachment of the first row of the surround doubler attachment at 2R (and 26L) are directly accessible from the inside. Therefore, these areas are inspected by HFEC surface probe. With a HFEC inspection, the minimum detectable crack size in the field is assumed to be a 0.0625 inch crack past the fastener head.

Structural Loading and Stress History Description

The stress spectrum is considered to have a remote stress due to cabin pressurization. Cabin pressurization primarily causes hoop tension in the fuselage. The GAG pressurization load is based on FAR25.571. The pressure condition is comprised of a 7.8 psi normal operating differential pressure and an additional 0.5 psi external aerodynamic pressure. A factor of 1.1 is only applied to the normal operating pressure for residual strength.

$$P = 7.8 + 0.5 = 8.3 \ psi$$

$R = 74 \ in.$ (radius of fuselage)

$$S_r = \frac{PR}{t_{skin}} = \frac{8.3(74)}{0.036} = 17.061 \ ksi$$

The bypass and bearing load at the critical fastener row is calculated using a displacement compatibility analysis as described by Swift ("Repairs to Damage Tolerant Aircraft," presented to the International Symposium on Structural Integrity of Aging Airplanes, Atlanta, Georgia, USA, 1990). Layer "a" is the fuselage skin and an existing bonded doubler. Layer "b" is the main cargo surround doubler. The surround doubler becomes fully effective after the first three rows. This analysis shows the most critical fastener location is the first row of fasteners.

Table SIE-3.1. Fastener Transfer Calculations.

DISPLACEMENT COMPATIBILITY ANALYSIS USING SWIFT'S FASTENER STIFFNESSES

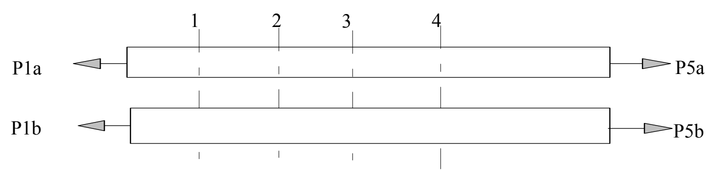
(Note: If there are n fastener rows, there are n+1 segments.
However, stiffnesses of last segment do not affect solution)

INPUT

See bottom for sketch (expand view for clarity)

Copy last two columns for additional fastener rows

SEGMENT	1	2	3	4	5	6	7	8	9	10
FASTENER										
STEEL? 1=yes, 0 = no	0	0	0	0	0	0	0	0	0	
ALUMINUM? 1= yes, 0= no	1	1	1	1	1	1	1	1	1	
D	0.188	0.188	0.188	0.188	0.188	0.188	0.188	0.188	0.188	
ta	0.072	0.072	0.072	0.072	0.072	0.072	0.072	0.072	0.072	
tb	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	
Esheet	10000000	10000000	10000000	10000000	10000000	10000000	10000000	10000000	10000000	
PLATES										
L	1	1	1	9.5	9.5	9.5	9.5	9.5	9.5	
Aa	0.072	0.072	0.072	0.072	0.072	0.072	0.072	0.072	0.072	
Ea	10000000	10000000	10000000	10000000	10000000	10000000	10000000	10000000	10000000	
Ab	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	0.1	
Eb	10000000	10000000	10000000	10000000	10000000	10000000	10000000	10000000	10000000	
CALCULATIONS										
Ca		1.389E-06	1.389E-06	1.319E-05	1.319E-05	1.319E-05	1.319E-05	1.319E-05	1.319E-05	
Cb	0.000001	0.000001	0.000001	0.0000095	0.0000095	0.0000095	0.0000095	0.0000095	0.0000095	
Cf	4.57069E-06	4.571E-06	4.571E-06	4.571E-06	4.571E-06	4.571E-06	4.571E-06	4.571E-06	4.571E-06	
Cf*Pf		7.322E-07	4.541E-07	6.663E-08	1.001E-08	3.067E-09	1.136E-08	7.605E-08	5.183E-07	
Pa	1	0.6952251	0.5350271	0.4356777	0.4210997	0.4189104	0.4182393	0.4157542	0.3991162	0.285714
Pb	0	0.3047749	0.4649729	0.5643223	0.5789003	0.5810896	0.5817607	0.5842458	0.6008838	0.714286
Pf	0.304774916	0.160198	0.0993494	0.014578	0.0021893	0.0006711	0.0024851	0.016638	0.1134022	
Cumulative load transfer a to b	0.304774916	0.4649729	0.5643223	0.5789003	0.5810896	0.5817607	0.5842458	0.6008838	0.714286	0.714286
% Pf load transfer of segment 1 Pa	30.48%	16.02%	9.93%	1.46%	0.22%	0.07%	0.25%	1.66%	11.34%	0.00%



Based on these results, 30% of the load is taken through bearing in the first row of fasteners. This first row of fasteners therefore has 30% as a bearing load and the remaining 70% as a bypass load.

The axial stress and bearing stress acting on this section are:

$$\sigma_{bypass} = S_0 = (1-0.305)S_r = 0.695S_r = 0.695(17.061) = 11.857\ ksi$$

$$\sigma_{brg} = S_3 = \frac{P}{Dt} = \frac{0.305S_rWt}{Dt} = \frac{0.305S_rW}{D} = \frac{0.305(17.061)0.94}{.188} = 26.018\ ksi$$

The limit stress used for residual strength purposes in this scenario is calculated, as stated earlier, according to FAR25.571.

$$P = 1.1*7.8+0.5 = 9.1\ psi$$

$$R = 74\ in.$$

$$S_r = \frac{PR}{t} = \frac{9.1(74)}{0.036} = 18.706\ ksi$$

The residual strength axial stress and bearing stress acting on this section are:

$$\sigma_{bypass} = S_0 = (1-0.305)S_r = 0.695S_r = 0.695(18.706) = 13.0\ ksi$$

$$\sigma_{brg} = S_3 = \frac{P}{Dt} = \frac{0.305S_rWt}{Dt} = \frac{0.305S_rW}{D} = \frac{0.305(18.706)0.94}{.188} = 28.527\ ksi$$

Material Property Description

The outer skin and doubler are made from 2024-T3 IAW QQ-A-250/5. The material properties from the NASGRO3.0 libraries are used for the fracture toughness and the crack growth rate properties. The material properties used are for 2024-T3; Clad, Plate and Sheet; T-L; LA & HHA NASGRO material code M2EA12AB1.

Table SIE-3.2. Material Properties and Growth Rate Data.

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

Material Properties:

:Matl:  UTS :  YS  :  K1e :  K1c :  Ak :  Bk :  Thk  :  Kc  : Keac :
: No.:      :      :      :      :     :     :       :      :      :
:----:------:------:------:------:-----:-----:-------:------:------:
:  1 :  66.0:  53.0:  46.0:  33.0: 1.00: 1.00:  0.036:  66.0:      :

:Matl:--------------- Crack Growth Eqn Constants -------------------:
: No.:     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 NASGRO3.0 with the crack growth interactions previously discussed. The input files for the equal length cracks growing from opposites sides of a hole are identical for the NASGRO3.0 analysis shown in Table SIE-3.3. The spectrum is included as a constant amplitude GAG cycle with 100 flights per block, with a single block applied per schedule.

Table SIE-3.3. NASGRO Input File for Problem SIE-3.

Data	Description
71fc1-2cout	Output file name
1	1=US units
D	D=direct
71fc1-2 skin at upper and lower doubler edges	Problem name
CC	Crack model type
2	Crack model no.
0.036	Thk, t
220	W
0.188	D
0.5	Hole center to edge
0.33	Poisson's Ratio
U	U=User defined crack

0.05	Initial a
1	Initial a/c
1	Number of materials
N	Non Interaction
1	Matl input choice
w	File input choice
M	Material Category
2	Material type
EA	Material alloy
1	Material heat treat information
Stress on skin at upper/lower edged	Spectrum name
N	Flag for identifying steps
100000	No. times to apply schedule
1	No. distinct blocks
N	Don't display spec blocks
1	Num steps/block
3	Schedule option
1	Load step number
100	Number of cycles
0	FMIN(1) t1 S_0
11.857	FMAX(1) t2 S_0
0	FMIN(2) t1 S_1
0	FMAX(2) t2 S_1
0	FMIN(3) t1 S_3
26.018	FMAX(3) t2 S_3
0	End manual input
1	Scaling Factor S_0
1	Scaling Factor S_1
1	Scaling Factor S_3
Y	Reference stress input
13	REFACT(1,1,1) S_0
2	Ref Stress at t2
0	REFACT(2,1,1) S_1
2	Ref Stress at t2
28.527	REFACT(4,1,1) S_3
2	Ref Stress at t2
N	Do not enter schedule from file
1	Sblock case
1	Number of times to apply
0	End Spectrum input

Results

Critical crack size/Residual Strength

The primary damage crack, crack A, is assumed to grow from a hole to the edge, during which the continuing damage crack, crack B, is growing from the opposite side of the hole towards an adjacent hole. Once crack A reaches the edge it transitions into an edge crack with the crack tip at the tip of crack B.

Life:

Based on the calculations for growing the crack in NASGRO and the crack growth interactions, the life from initial crack size to failure is determined to be 41,412 flights. The results of crack length versus life are shown in Figure SIE-3.7. The life is given in numbers of flights.

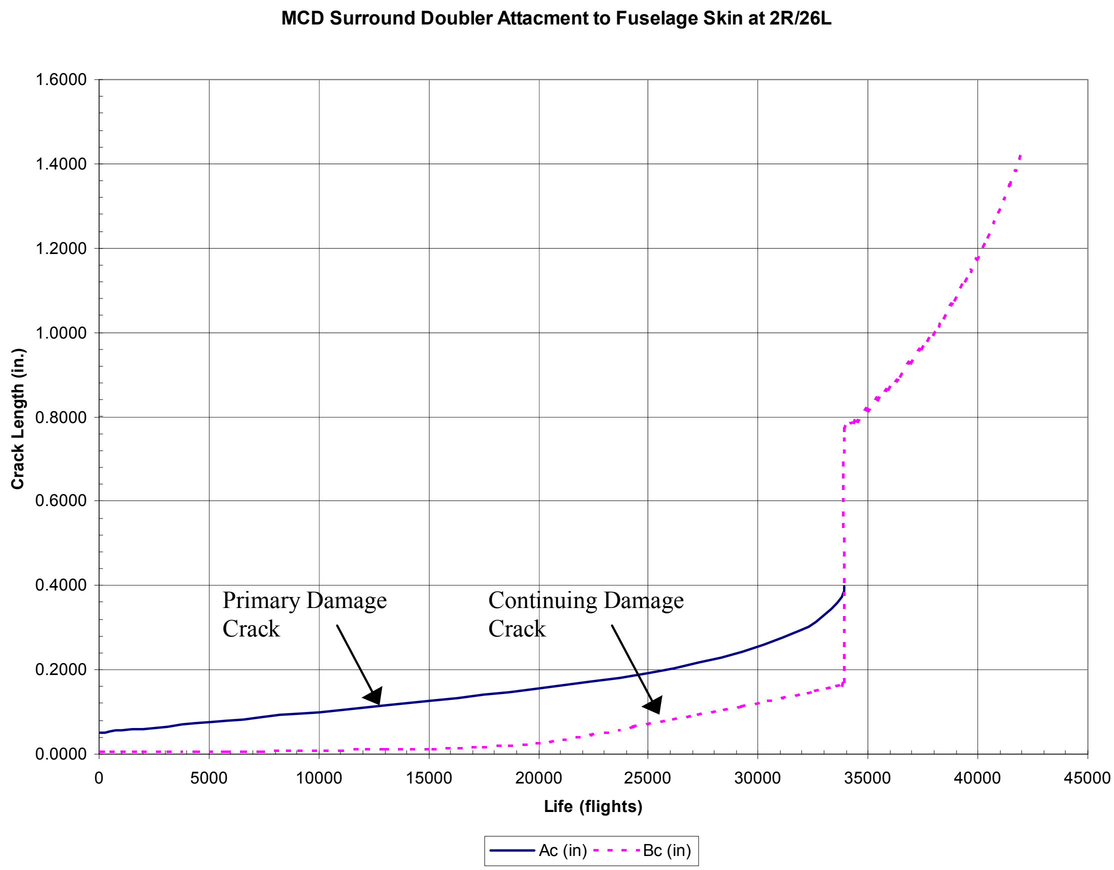


Figure SIE-3.7. Crack Growth Life for Problem SIE-3.

Inspection Intervals

The threshold and repeat intervals are calculated using the life reduction factors shown below.

Life Reduction Factors:
$K_1 = 2.0$
$K_2 = 3.0$

Detectable crack length (HFEC around fastener head):

$$c_{\det} = a_{\det} = \frac{(D_{head} - D)}{2} + 0.0625 = \frac{(0.3016 - 0.188)}{2} + 0.0625 = 0.1193\,in.$$

Number of flights @ detectable crack length, $N_{\det}$ = 13,758 flights

Critical crack length (distance to next adjacent hole):

$$c_{crit} = B_a + p - \frac{D}{2} = 0.5 + 0.94 - \frac{0.188}{2} = 1.346\,in.$$

Number of flights @ critical crack length, N_{crit} = 41,412 flights

$$\text{Threshold Interval} = \frac{N_{crit}}{K_1} = \frac{41412}{2.0} = 20,706\,\text{flights}$$

$$\text{Repeat Interval} = \frac{N_{crit} - N_{\det}}{K_2} = \frac{41412 - 13758}{3.0} = 9,218\ \text{flights}$$