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DAMAGE TOLERANCE EVALUATION OF AN AIRCRAFT FUSELAGE PANEL CONTAINING A BROKEN STRINGER

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ABSTRACT

The article presents two stage fatigue life prediction of a stiffened aircraft fuselage panel, subject to GAG pressure cycles, with a circumferential crack and a broken stringer in stage two. As a worst case scenario, it is assumed that double cracks start at the edge of a rivet hole both on the skin and the stringer simultaneously. The first stage involves the fatigue crack growth until the stringer is completely broken with the crack on the fuselage skin propagating. The first stage itself is treated in two sub-stages, and fatigue life prediction is performed by approximating the crack propagation process using the application defined models of AFGROW. It is shown that second stage, which starts after the stringer is completely is broken, stress intensity factor history, provided by the center through crack application defined model of AFGROW, has to be corrected to account for the broken stringer, and three dimensional effects such as bulging of the skin and panel curvature. For the second stage, a detailed three dimensional local finite element model of the fuselage skin with the broken stringer is prepared to calculate the variation of the normalized stress intensity factor with crack length. It is concluded that the skin curvature and bulging of the skin due to the internal pressure can have significant effect on the stress intensity factor and the fatigue life of the fuselage structure. It is also shown how fast the crack can propagate in stage two, after the stringer is completely broken, compared to stage one.

INTRODUCTION

The Aloha Airlines accident in 1988 showed that improved design methodology was necessary in the design of aircraft structures. Damage tolerant design method is introduced to ensure improved safety by assuming that the structure already has a damage induced in a critical location with appropriate measures taken to ensure that damage is well-behaved until it is detected in an inspection period. Damage tolerant design philosophy is based on the two fundamental requirements such that unstable crack propagation is locally contained through the use of multiple load paths or crack stoppers, and slow crack growth concept which requires that crack does not reach to a critical size suitable for unstable crack propagation before a pre-defined inspection schedule. According to the “two bay crack” criterion, fuselage structure must withstand a crack in the skin over two stringer bays or two frame bays, depending on whether the crack is longitudinal or circumferential [1].

Fatigue loading of pressurized aircraft due to ground-air-ground cycle (GAG) is probably the largest and most frequently applied single load cycle for most transport aircraft. In shorter flights, aircraft has more exposure to GAG cycle, thus the damage incurred is higher compared to longer flights [2]. Pressurized fuselage skins are subject to combined biaxial and internal pressure loads which inevitably make the evolution of cracks in pressurized fuselage structures a complex process. Fatigue crack growth evolution in pressurized fuselage structures generally requires the determination of the crack tip stress intensity factor by finite element analysis, since simplified analysis methods are usually not applicable to practical problems which involve complex loading and structural configurations.

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For stiffened panels several studies have been conducted for the computation of stress intensity factor. The original work on the calculation of the stress intensity factor for cracked sheets with intact riveted stringers is given by Poe [3]. On a follow-up study, Poe presented results on the effect of broken stringers on the stress intensity factor for a uniformly stiffened sheet containing a crack [4]. Rooke and Cartwight extended the compounding method for determining approximate stress intensity factors to stiffened sheets with cracks [5]. Swift presented a method based on displacement compatibility for the fracture analysis of cracked stiffened structure [6]. Palani et.al. performed fracture analysis of cracked stiffened panels under combined tensile bending and shear loads by computing stress intensity factors via parametric equations developed by using the numerically integrated modified virtual crack closure integral technique [7]. A detailed stress analysis of a single shear rivet lap joint was conducted by Moreira et.al. using three-dimensional elastic finite element analysis [8]. The stress intensity factors for this geometry with a through symmetric and asymmetric crack were determined. Moreira et.al. conducted a three dimensional finite element analyses to calibrate the stress intensity factor in a cracked stiffened plate subjected to remote uniform traction [9].

On the crack growth analysis of aircraft fuselage skins, various studies have also been conducted. Toor and Dagger evaluated various analytical and empirical approaches for the damage tolerance analysis of fuselage structures with emphasis on circumferential cracks resulting due to vertical bending of the fuselage [10]. Most of the studies performed on the bulging cracks deal with unstiffened shells. Because of the complexity in analyzing bulging cracks, stress intensity factor solutions for stiffened fuselage structures for a wide range of crack configurations are not available. A comprehensive study of bulging cracks is presented by Rahman et.al. [11] who compiled bulging factor solutions obtained at the Federal Aviation Administration William J. Hughes Technical Center for cracks in typical transport category aircraft. The solutions were obtained for longitudinal cracks in the critical rivet row of longitudinal lap splice joints. Both unstiffened, and frame and longeron-stiffened configurations were considered, and the effects of the parameters that were varied on the bulging factor are discussed. The report also addresses the few studies that analyze bulging cracks in pressurized stiffened aircraft fuselage structures.

Present study focuses on the two stage fatigue life prediction of an aircraft fuselage structure with a circumferential crack and a broken stringer. To demonstrate the two stage process of fatigue life calculation, simple fatigue loading is taken as the constant amplitude GAG pressure cycle. As a worst case scenario, it is assumed that double cracks start at the edge of a rivet hole both in the fuselage skin and in the stringer simultaneously. In the first stage, the number of GAG pressure cycles, until the stringer is completely broken, is calculated using the application defined models of AFGROW [12]. In stage 2, to determine the residual life of the cracked fuselage skin with a broken stringer, three dimensional finite element model of the fuselage skin with the broken stiffener is prepared. The effect of broken stringer and the three dimensional effects, such as bulging of the cracked skin due to pressure, on the crack tip stress intensity factor are taken into account by calculating the stress intensity factor along the crack line by the three dimensional finite element analysis. Stress intensity factors determined by the finite element analyses are then provided to the center through crack application defined model of AFGROW externally, and residual fatigue life assessment of the fuselage skin is made in accordance with the two-bay criterion. It is shown that compared to stage one, in stage 2 crack propagates very fast after the stringer is completely broken. In stage two, the effects of bulging of the skin and skin curvature on the normalized stress intensity factor are also studied. It is shown that bulging of the skin panels and panel curvature can have significant effects on the normalized stress intensity factors. Significant increases in the normalized stress intensity factors are determined due to curvature of the panel and bulging of the skin panels due to pressure. It should be noted that in the present study, linear three dimensional finite element analysis is performed to determine the stress intensity factors in stage 2 after the stringer is broken. It was reported by Rahman et.al. [11] that stress intensity factors determined by the geometrically linear finite element analysis of bulging cracks are higher than the ones determined by the geometrically non-linear finite element analysis. Although in order to appropriately characterize bulging cracks large deformations need to be considered, in the present study linear analysis is used since the main aim of the article is to demonstrate the two stage life prediction methodology of an aircraft fuselage structure with a circumferential crack and a broken stringer.

DAMAGE TOLERANCE ANALYSIS METHODOLOGY

Aircraft fuselage skin with a broken stiffener

During flight, fuselage skin is exposed to complex loading conditions as illustrated in Fig. 1. On a typical fuselage panel, axial stresses are generated due to bending and pressure effects, shear stresses are generated due to twisting and bending effects, and for the pressurized fuselage, the state of stress is three dimensional. For such a complex geometry and complex loading conditions, some simplifications are made for the preliminary life prediction calculations and sizing. In the initial phases of the design of fuselage structures, sizing is done considering the cabin pressure which causes one of the most critical types of loading for the stiffened panels [6,13].

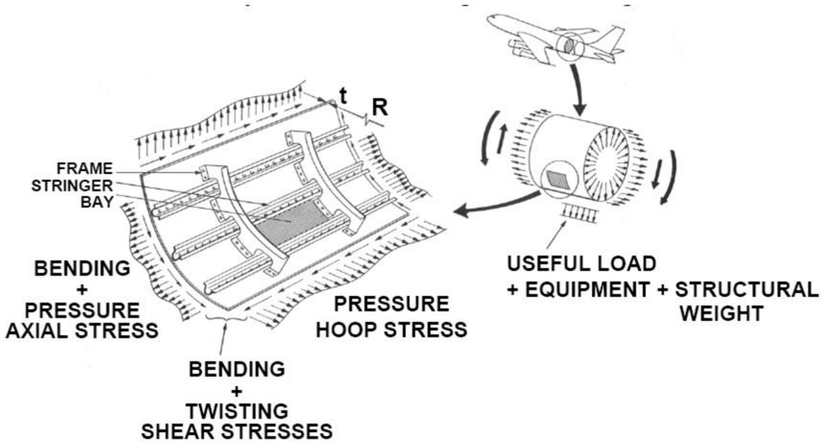


Fig. 1 Complex loading of fuselage skin [13]

According to damage tolerance design philosophy, the most critical scenario has to be chosen, and aircraft should be designed such that it must remain safe even if it experiences the worst case load. In fatigue life evaluation of fuselage structures, besides the pressure loading, other loads due to the gusts, turbulence, maneuvers, landing etc., are also taken into consideration. However, in the present study, to demonstrate the two stage fatigue life evaluation methodology of an aircraft fuselage structure with a circumferential crack and a broken stringer, loading is kept simple and cabin pressure is taken as the primary loading. It should be noted that cabin pressure triggers the crack opening mode which is the most critical crack propagation mode in aerospace structures.

For the demonstration problem involving the cabin pressurization loading only, it is assumed that the airplane climbs to the service ceiling from sea level in all of the fights, and cabin is assumed to be kept at the sea level pressure. For such a scenario, pressure difference between the cabin and the sea level atmospheric pressure is maximum, and such a pressure loading is conservative. In one ground-air-ground (GAG) cycle, the aircraft experiences a pressure difference which equal to the difference of the sea level and the service ceiling pressure. GAG cycle is a repeating load that the aircraft experiences in every flight, and it may cause the fatigue crack propagation in the fuselage panels of aircraft structure. It should be noted that the stress range of the GAG cycle for the skin of a pressurized fuselage is probably the largest and most frequently applied single load cycle for most aircraft.

The critical crack possibilities in the stiffened fuselage skin are longitudinal and circumferential cracks, as shown in Fig. 2.

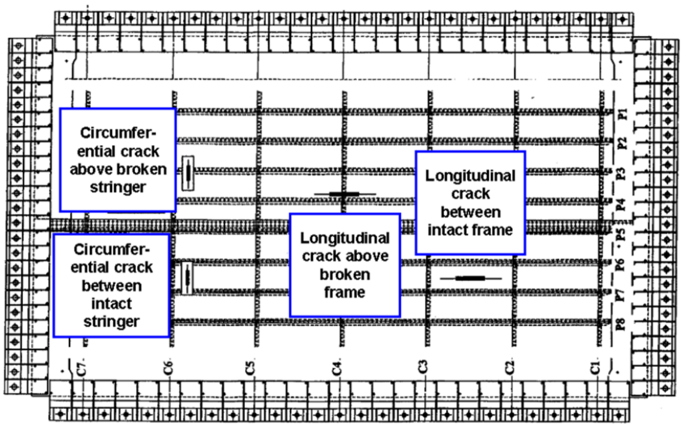
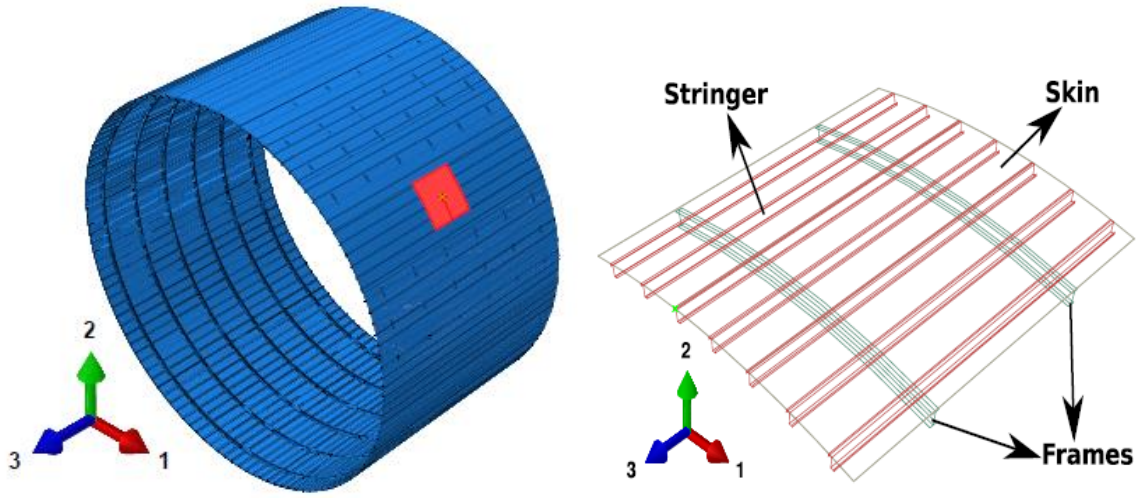


Fig. 2 Critical crack directions under bi-axial loading [13]

It is considered that the most critical crack configuration case in the stiffened skin is the crack under the broken stiffener [6,13]. Therefore, in the present study two stage fatigue life evaluation of the fuselage structure is performed for a circumferential crack under a broken longitudinal stringer. To present the complete process of crack propagation, it is assumed that double cracks start at the edge of a rivet hole both in the skin and the stringer simultaneously.

Figure 3 shows a section of the forward center fuselage model of an airplane which is assumed to contain double cracks which start in the skin and the stringer simultaneously. In the fracture mechanics analysis, a local model of the fuselage structure, which is highlighted in Figure 3a and magnified in Fig 3b, is used with the appropriate symmetry boundary conditions applied on the local finite element model. Double cracks are located in the symmetry plane 1-2 in the finite element model.



(a) The pressurized fuselage panel **(b)** Detailed local view of the fuselage panel

Fig. 3 Fuselage panel

Two stage crack growth analysis and life assessment

Two stage crack growth scenario is one of the most critical scenario to be considered in the design process for the stiffened fuselage structure. In this scenario, it is assumed that two double cracks with a crack length of 1.27 mm, which is the smallest crack length that NDT inspection can detect, start from the edge of the rivet hole in the stringer and in the skin at the same time. After the cracks develop, the crack growth process is treated in two stages which are described in detail below. In the crack growth calculations, fracture mechanics and fatigue crack growth analysis software tool AFGROW is used [12].

In the present article, the fatigue life calculation of the fuselage structure due to constant amplitude GAG cycle is treated in two stages.

<u>Stage 1:</u>

First stage is further divided into two sub-stages which are denoted as stage 1(a) and 1(b), respectively.

Stage 1(a) : Stage 1(a) is illustrated schematically in Fig. 4. Two double cracks start from the edge of the rivet hole on the stringer and on the skin simultaneously. The black fill color in Fig. 4 signifies the crack, whereas the white fill color signifies the pristine skin and the stringer. In stage 1(a), both cracks on the skin and the stringer propagate until the crack, which is closer to the free end of the stringer, breaks one portion the stringer with brittle fracture and reaches the free end. In stage 1(a), fatigue crack growth calculation is performed using the application defined model of AFGROW with the repeated longitudinal stress due to GAG pressure cycle, and Forman crack growth relation with plane stress option. Figure 7a shows the AFGROW model used in stage 1(a) which is the double through crack at the edge of a hole [12].

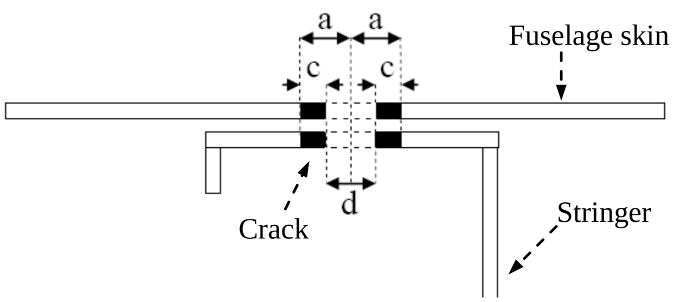


Fig. 4 Stage 1(a) - Cracks start from the edge of the rivet hole in the skin and the stringer

Stage 1(b) : At the end of stage 1(a), crack in the stringer can be treated as an edge crack which is schematically shown in Fig. 5. In stage 1(b), the crack in the stringer is modeled with an "edge through crack" application defined model of AFGROW, as shown in Fig. 7b [12]. It should be noted that at this stage, the fuselage skin is still modeled with "double through crack at the edge of a rivet hole" application defined model. Stage 1(b) ends when the edge through crack, shown in Fig. 5, completely breaks the stringer and reaches the other free end.

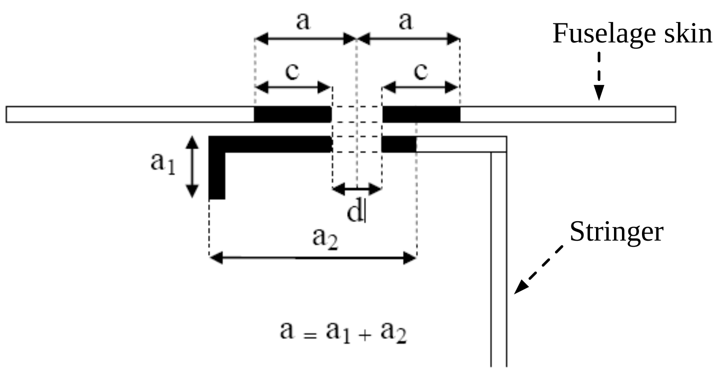


Fig. 5 Stage 1(b) - Crack in the stringer becomes an edge through crack

It should be noted that in stage 1, since the stringer is not broken completely, AFGROW models are used to approximate the fatigue crack growth process with an assumption of constant cyclic stress both in the skin and the stringer. In stage 1, stress intensity factor history provided by AFGROW is used in the fatigue crack growth calculations. The selection of effective widths of the stringer used in the AFGROW models in stages 1(a) and 1(b) is explained in section 4. In stage 1, only the longitudinal stress due to the GAG pressure cycle is considered, the fuselage skin is approximated by a straight panel and bulging effect of the pressure is not considered. In the numerical example given in section 4, it is shown that in stage 1 the crack growth in the fuselage skin is very small compared to the stringer spacing. Therefore, straight panel approximation of the fuselage skin is justified in stage 1. Three dimensional effects including the effect of curvature and bulging of the fuselage skin on the stress intensity factor, thus the crack growth process, is taken into consideration in stage 2 after stringer is completely broken.

Stage 2: Stage 2 starts when the edge through crack in stage 1(b) reaches its final length and the stringer is completely broken. This situation is shown schematically in Fig. 6. In stage 2, after the stringer is broken completely, "center through crack" application defined model of AFGROW, shown in Fig. 7c, is used with the stress intensity factor correction.

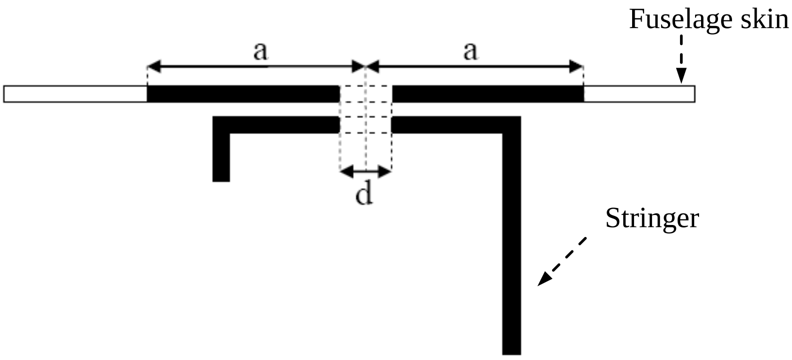
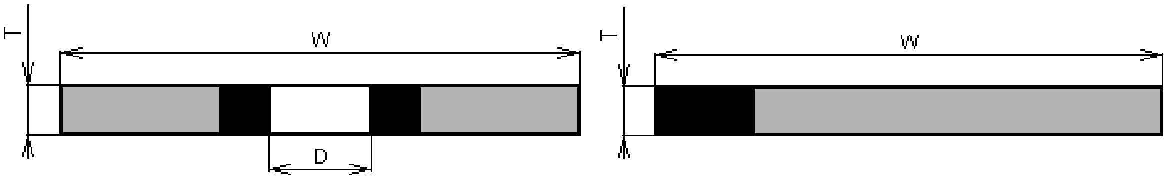
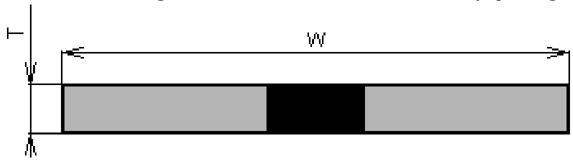


Fig. 6 Stage 2 - Stringer is completely broken, and the crack propagates in the skin

In stage 2, stress intensity factors provided by the “center through crack” application defined model of AGROW can not be used. This is because after the break of the stringer, it carries no load, and as it is shown in section 4, crack tip stress intensity factor in the fuselage skin increases significantly. Therefore, to analyze the crack growth process in stage 2, a detailed three dimensional local finite element model of the fuselage structure is prepared to determine the crack tip stress intensity factors for different crack lengths until the total crack length becomes two stringer spacing in accordance with the two bay criterion. Stress intensity factors determined by the three dimensional finite element analysis are then used in the “center through crack” application defined model of AFGROW to calculate the crack growth with the number of GAG cycles in stage 2. In all AFGROW calculations, Forman crack growth model is used.



(a) Double through crack at the edge of a hole (**b)** Edge through crack



(c) Center through crack model used with stress intensity factor correction

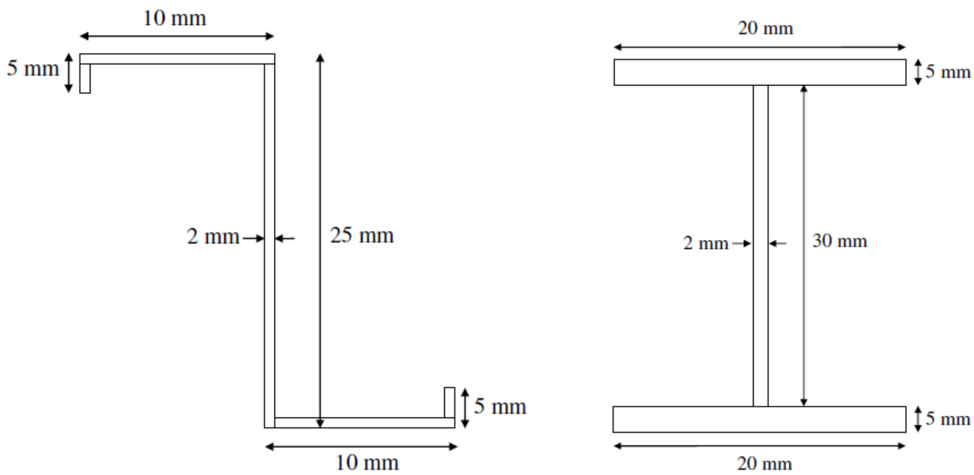
Fig. 7 AFGROW models used in the two stage fatigue life prediction of the fuselage skin

In stage 2, stress intensity factors are calculated by the three dimensional finite element analysis performed by Abaqus 6.10 [14], and they are used in the “center through crack” application defined model of AFGROW shown in Fig. 7c as β correction factors. Verification of the Abaqus implementation of stress intensity factor calculation and comparisons with the analytical and Franc2D/L [16] fracture analysis code results for stiffened cracked sheets are also performed. The verification study is given in detail by Sayar [17].

TWO STAGE CRACK GROWTH AND LIFE CALCULATION

In this part, a sample life calculation of a cracked fuselage skin with a broken stiffener is demonstrated for an airplane. To demonstrate the two stage process of fatigue life calculation, simple fatigue loading is taken as the constant amplitude GAG pressure cycle. In the sample life calculation, it is assumed that the airplane climbs to the service ceiling of 11300 m from the sea level in all flights.

Figure 8 gives the stringer and frame dimensions which are used in the finite element model of the cracked fuselage skin, and Table 1 summarizes the technical specifications and geometry of the airplane.



(a) Stringer C/S geometry (b) Frame C/S geometry

Fig. 8 Cross sectional geometry of the stringer and frame

Table 1 Technical Specifications and Geometry of the Cargo A/C

Service Ceiling	11300 *m*
Fuselage Diameter	5 *m*
Stringer Spacing	131 *mm*
Frame Spacing	500 *mm*
Rivet Diameter	2.5 *mm*
Rivet Spacing	25 *mm*
Skin Thickness	1 *mm*
Stringer C/S Area	110 mm^2
Frame C/S Area	260 mm^2
Skin Material	Al 2124 T-851
Frame Material	Al 2124 T-851
Stringer Material	Al 2124 T-851

The sample fatigue life evaluation is performed for a fuselage skin with double circumferential crack in the skin and the stringer. For the stiffened cylindrical fuselage, the longitudinal stress becomes

$$\sigma_L = \frac{\pi r^2 \Delta P}{2\pi r t + A_s N_s} \tag{1}$$

where, A_s is the stringer area and N_s is the number of stringers, and t is thickness of the fuselage skin. For the particular airplane there are 120 stringers, and for the pressure difference ΔP corresponding to the sea level and service ceiling altitude, the difference in the longitudinal stress in the GAG cycle becomes 68.06 MPa.

Determination of normalized stress intensity factor in stage 2

As explained in section II, to analyze the crack growth process in stage 2, normalized stress intensity factor must be calculated using a local finite element model of the stiffened fuselage skin. Figure 9 shows the local finite element model of the crack area which is prepared in Abaqus. It should be noted that local finite element model is taken as half of the highlighted fuselage skin, shown in Fig. 3a. Thus, the center crack is located in the symmetry plane, 1-2 in Fig. 9, of the highlighted fuselage skin shown in Fig. 3a. Figures 9a-9d show the position of the broken stringer, local model of the stiffened fuselage panel, the center crack with the broken stringer and the details of the crack, respectively.

For accurate evaluation of the stress intensity factor, fine mesh is used around the crack tip, as shown in Fig. 9d. Figure 10 shows the focused mesh used for modeling the crack tip region. Stress intensity factors are calculated by the contour integration, and square root singularity option is selected to perform linear elastic fracture mechanics (LEFM) based SIF calculations. For this purpose, 3D contour integral is defined at the crack tip with mid-side node parameter option set to 0.25 [8,15]. In the finite element model of the stiffened fuselage panel, stringers and the frames are modeled with S4R linear shell elements, and the fuselage skin is modeled with C3D20R quadratic brick elements with reduced integration, and single element is used through the thickness [14]. It should be noted that in the calculation of the stress intensity factor associated with the crack in the fuselage skin, solid elements are used in the finite element model of the skin of the fuselage since the skin is pressurized. The use of solid elements is also recommended by Abaqus [15], because out-of-plane loading contribution to the contour integral is taken into account with the use solid elements.

As specified in Ref. 15, in order to prevent the path dependence of the stress intensity factor calculation, stress intensity factors which are calculated at nodes *A*, *B* and *C* are averaged, as shown in Eq. (2).

$$K_{Average} = \frac{K_A + 4K_B + K_C}{6} \qquad (2)$$

In the finite element model of the stiffened fuselage skin, point based mesh independent fasteners are generated by the face to face attachment method, and finite element analysis is performed on the half symmetric model. To model the rigid riveted connection, rigid beam type connector section is defined [14]. To simulate the true effect of the internal pressure on the local finite element model of the fuselage skin, appropriate displacement and load boundary conditions are applied, as necessary, on the side edges, front edge and the back edge of the local finite element model shown in Fig. 9. On the front edge where the circumferential crack exists, symmetry boundary conditions are applied on the intact edge since the local model is taken as half of the highlighted fuselage panel in Fig. 3a.

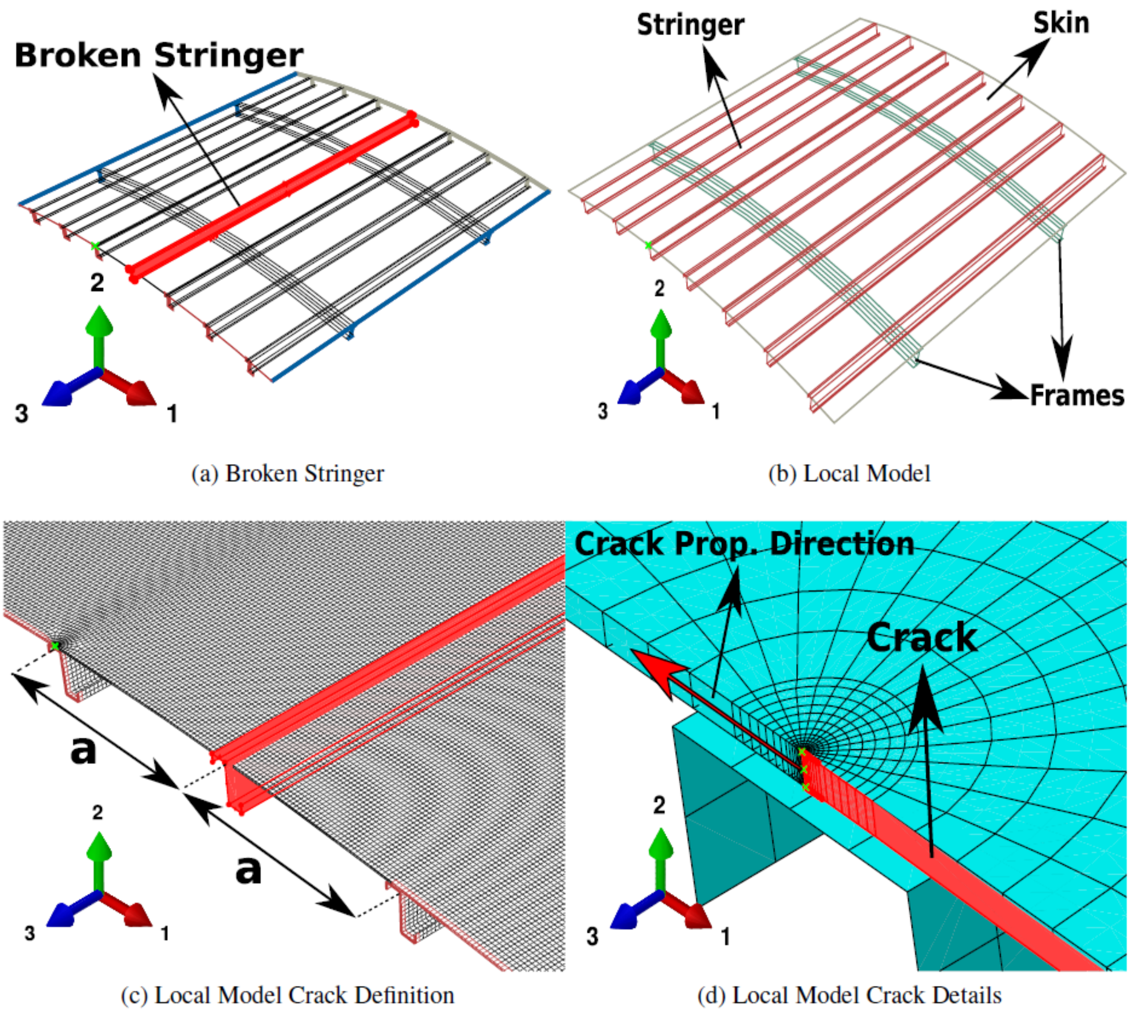


Fig. 9 Details of the local Abaqus model of the cracked fuselage skin with broken stringer

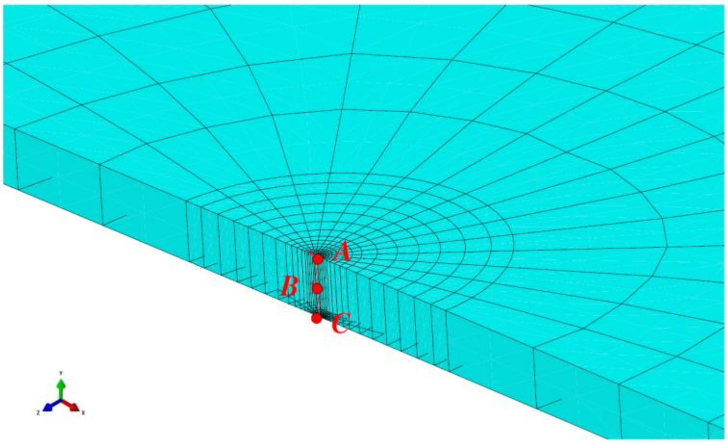


Fig. 10 Focused mesh around the crack tip; A-B-C nodes are on the crack tip element

The deformed local mesh under the internal pressure loading with the closed end condition is given in Fig. 11 which shows the bulging of the fuselage skin sections between the frames and stringers.

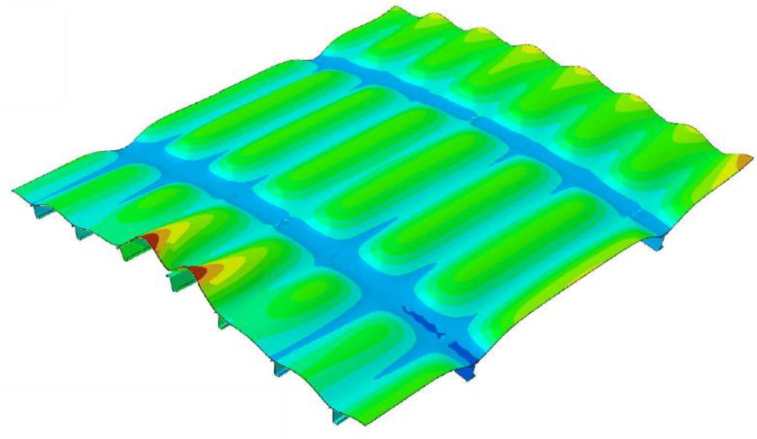


Fig.11 Deformed symmetric half crack containing 90 mm crack over broken stringer

In stage 2, after the stringer is completely broken, three dimensional local finite element analysis is performed, and stress intensity factors obtained are normalized with respect to $\sigma_L \sqrt{\pi a}$, where σ_L is the longitudinal stress given by Eq. (1) and a is the crack length. Figure 12 shows the variation of the averaged (Eq.(2) normalized stress intensity factor with the non-dimensional circumferential crack length (a/b) for the fuselage skin with a broken stringer. It should be noted that Fig. 12 is drawn for a single stringer width b in accordance with the two bay criterion.

Figure 12 shows that normalized stress intensity factor for the curved fuselage skin under internal pressure fluctuates around 2.5, and does not drop significantly when the crack approaches the stringer (a/b=1) unlike what has been observed in stiffened flat plates under in-plane loads with and without broken stringer [3,4,17]. Similarly, for the cracked straight panel with the broken stringer, Poe [4] showed analytically and Sayar [17] showed by the finite element analysis that normalized stress intensity factor continuously decreases as the crack approaches the intact stringer. Bulging of the fuselage skin due to internal pressure and panel curvature are considered to be the main reasons for the normalized stress intensity factor variation shown in Fig. 12.

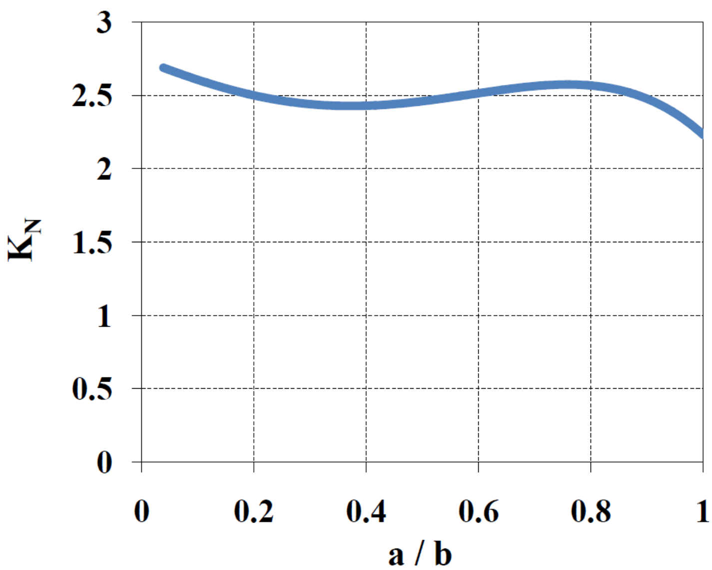


Fig. 12 Variation of the averaged normalized stress intensity factor with the crack length

Life assessment of the cracked stiffened fuselage skin

Crack growth model used

Life calculation of the cracked stiffened fuselage skin is performed by the Forman crack growth model, given by Eq. (3), due to the convenience of determining the Forman constants experimentally, and the success in modeling the secondary and tertiary region. It should be noted that in the present analysis crack retardation is not considered, since it is more conservative to ignore the existence of the reverse cyclic plastic zone in fatigue crack growth calculations for critical aircraft components.

$$\frac{da}{dN} = \frac{C(\Delta K)^m}{(1-R)K_{CR} - \Delta K} \tag{3}$$

where K_{CR} is the critical fracture toughness.

Al 2124 T-851, which has a plain strain fracture toughness of 24.3 $MPa\sqrt{m}$ in the T-L fracture plane [18], is used for the skin material of the curved fuselage skin, stringers and frames. Since the critical fracture toughness is a function of the plain strain fracture toughness K_{IC} and the thickness, for the 1 mm thick fuselage skin critical fracture toughness is calculated using the formulation given by Eq.(4) [19].

$$\frac{K_{CR}}{K_{IC}} = 1 + B_k \exp\left(-\left(A_k \frac{t}{t_0}\right)^2\right) \qquad t_0 = 2.5\left(\frac{K_{IC}}{\sigma_{ys}}\right)^2 \tag{4}$$

where A_k and B_k are fit parameters, σ_{ys} is the yield stress, and *t* is the thickness of the skin. Fit parameters A_k and B_k are recommended to be 1 in the NASGRO and AFGROW databases, thus for the 1 mm thick fuselage skin critical fracture toughness is calculated as 48.1 $MPa\sqrt{m}$.

In the present study, Forman constants for Al 2124 T-851 are determined by making use of the crack growth rate data obtained experimentally using the compact tension specimen. Figure 13 shows the crack length versus cycle data which is obtained in accordance with ASTM-E647 for the stress ratio R = 0 which simulates the GAG pressure cycle [20]. By making use of the crack length vs. cycle data shown in Fig.13, da/dN vs ΔK graph is prepared. Best fit to the experimentally determined crack length vs. cycle curve is determined to be the Inverse Bleasdale equation with offset which is shown by the solid line in Fig. 13. ΔK values corresponding to the crack length, at which the da/dN data is calculated, are determined utilizing Eq. (5) which is proposed by ASTM-E647.

$$\Delta K = \frac{\Delta P}{B\sqrt{W}} \frac{(2+\alpha)}{(1-\alpha)^{1.5}} (0.886 + 4.64\alpha - 13.32\alpha^2 + 14.73\alpha^3 - 5.6\alpha^4) \quad (5)$$

where *B* and *W* are the thickness and the width of the compact tension specimen which are taken as 12.7 mm and 50.16 mm, respectively, and α is the non-dimensional crack length given by *a*/*W*.

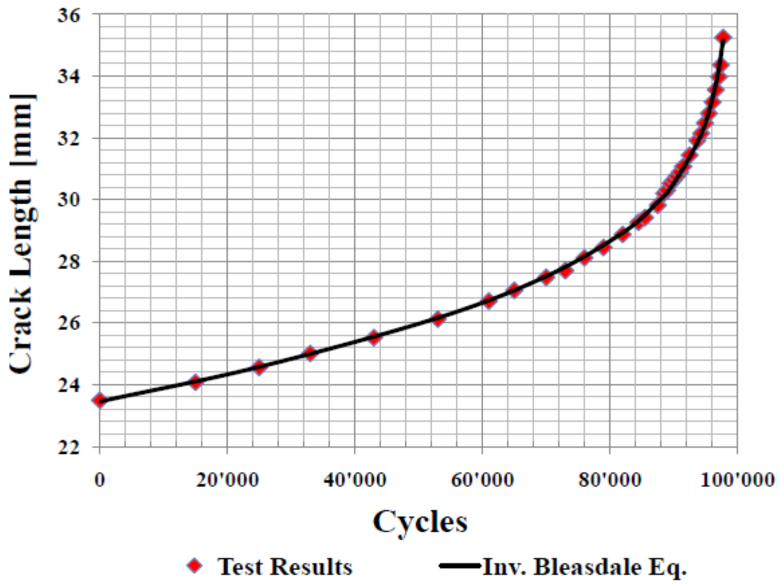


Fig. 13 Crack length vs. cycle curve obtained experimentally

Forman constants, *C* and *m* are obtained by picking up two points from the linear region on the *da*/*dN* vs. Δ*K* curve, and solving Eq. (3) simultaneously for these points. It should be noted that in Eq. (3) the critical fracture toughness K_{CR} is corrected, via Eq. (4), to account for the change of the fracture toughness with the thickness of the compact tension specimen used in the crack growth experiment. As a result of the experimental study, Forman constants *C* and *m* for Al 2124-T851 are determined as 8.097 10^{-10} and 3.368, respectively.

Crack growth in stages 1 and 2

Stage 1

It is assumed that in stage 1 two double through cracks of length 1.27 mm start from the edge of the rivet hole on the stringer and on the skin simultaneously. It is assumed that rivet holes exist at the center of the top flange of the Z stringer shown in Fig. 8.

Stage 1(a): Stage 1(a) is depicted schematically in Fig. 4. In stage 1, only the longitudinal stress due to the GAG pressure cycle is considered. Table 2 summarizes the status of the crack in the stringer at the beginning and at the end of stage 1(a) which is obtained by the AFGROW fatigue crack growth analysis using the Forman model, and the “double through crack at the edge of a hole” application defined model (Fig. 7a). AFGROW analysis showed that when the crack length reaches a value of 6.9 mm, measured from the edge of the rivet hole, net section yielding failure occurs. Therefore, in stage 1(a) it is assumed that the crack reaches to the closer end of the stringer.

Table 2 AFGROW results in stage 1(a) for the stringer

w	20 *mm*
d	2.5 *mm*
t	2 *mm*
c_i	1.27 *mm*
c_f	6.9 *mm*
a_i	2.52 *mm*
a_f	8.15 *mm*
N	174800 *Cycles*

The parameters given in Table 2 are defined below.

- w : Effective width used in AFGROW analysis; twice the distance from the rivet center line to the closer end of the stringer including the length of the vertical flange the stringer (Fig. 8)
- d : Rivet hole diameter
- t : Thickness of the stringer
- c_i : Initial crack length at the beginning of stage 1(a) measured from the edge of the rivet hole, as shown in Fig.4
- c_f : Final crack length at the end of stage 1(a) measured from the edge of the rivet hole
- a_i : Initial crack length at the beginning of stage 1(a), measured from the center of the rivet hole
- a_f : Final crack length at the end of stage 1(a), measured from the center of the rivet hole
- N : Number of cycles

Stage 1(b): Stage 1(b) is depicted schematically in Fig. 5. At the end of stage 1(a), crack in the stringer can be treated as an edge through crack, and in AFGROW analysis "edge through crack" application defined model is used, as illustrated in Figure 7b. In stage 1(b), after the initial phase of crack propagation, the stringer is completely broken with brittle fracture because SIF reaches the critical fracture toughness K_{CR}. Stage 1(b) ends when the edge through crack, shown in Fig. 5, completely breaks the stringer and reaches the other free end. Table 3 summarizes the status of the crack in the stringer at the beginning and at the end of stage 1(b) which is obtained by the AFGROW fatigue crack growth analysis. Since stage 1(a) ends with a final crack length of a_f = 8.15 *mm* and one side of the stringer, which is broken in stage 1(a), measured from the center of the rivet hole is 10 *mm*, the initial length of the edge through crack in stage 1(b) is taken as 18.15 *mm*. It should be noted that the total crack length includes the length of the vertical flange of the stringer, as shown in Fig. 5.

Table 3 AFGROW results in stage 1(b) for the stringer

w	55 *mm*
d	2.5 *mm*
t	2 *mm*
c_i	6.9 *mm*
c_f	-
a_i	18.15 *mm*
a_f	26.24 *mm*
N	1246 *Cycles*

In the "edge through crack" application defined model of AFGROW, the effective width w is defined as the total length of the Z stringer including all the horizontal and vertical flanges and webs, shown in Fig. 8a. Because the double through crack turns into an edge through crack at the end of stage 1(a), c_f is not applicable, and the final crack length a_f is defined as the sum of a_1 and a_2 in Fig. 5. In stage 1(b), the final crack length of 26.24 *mm* corresponds to the brittle fracture point when the SIF reaches the critical fracture toughness K_{CR}.

It should be noted that skin crack also propagates during stages 1(a) and 1(b). The total number of fatigue cycles in stage 1 is 176046 which is the sum of the fatigue cycles in stages 1(a) and 1(b). In stage 1, crack growth in the fuselage skin is analyzed by utilizing the "double through crack at the edge of a hole" application defined model of AFGROW, as illustrated in Figure 7a. In stage 1, only the longitudinal stress σ_L due to the GAG pressure cycle is considered, and bulging of the crack due to pressure is not considered since the stringer is not completely broken. Table 4 summarizes the status of the crack in the fuselage skin at the beginning and at the end of stage 1 which is obtained by the AFGROW fatigue crack growth analysis using the Forman model.

Table 4 AFGROW results in stage 1 for the fuselage skin

w	262 *mm*
d	2.5 *mm*
t	1 *mm*
c_i	1.27 *mm*
c_f	3.37 *mm*
a_i	2.52 *mm*
a_f	4.62 *mm*
N	176046 *Cycles*

For the fuselage skin, the effective width is taken as the length of the two stringer spacing in accordance with the two bay criterion. It is seen that in stage 1 crack growth in the fuselage skin is very slow, and at the end of stage 1 crack reaches to a final length of 4.62 *mm* which is very small compared to the stringer spacing. It should be noted that the use of "double through crack at the edge of a hole" application defined model of AFGROW, without stress intensity factor correction in stage 1, for the fatigue crack growth analysis of the fuselage skin is justified for the following reasons:

- In stage 1, crack length to stringer spacing ratio (a/b) is very low, and therefore normalized intensity factor is very close to unity for the pristine stringer case as shown by Poe [3] and Sayar [17] for configurations where crack in the skin starts right under the stringer.Therefore, the effect of rivets on the stress intensity factor can be neglected in stage 1, and the "double through crack at the edge of a hole" application defined model of AFGROW can be used with confidence to determine the crack growth in the fuselage skin in stage 1.
- In stage 1, since the crack is under the stringer, which is not completely broken, the length of the crack is small and the crack growth is slow, bulging of the crack can be neglected. Therefore, only the longitudinal stress σ_L due to the GAG pressure cycle is considered as the load acting on the fuselage skin.
- Since the crack length in the fuselage skin is very small in stage 1, curvature of the fuselage skin can be neglected. Thus, the use of the "double through crack at the edge of a hole" application defined model of AFGROW is justified in stage 1.

Stage 2

Stage 2 starts right after the complete break of the stringer. In Stage 2, since the broken stringer no longer carries axial load, stress intensity factor increases sharply. Figure 12 shows the variation of the normalized stress intensity factor with the crack length in stage 2. In the AFGROW database, there is no built in application defined model for simulating the crack growth process for such a complicated geometry. Therefore, in stage 2, after the stringer is broken completely, "center through crack" application defined model of AFGROW, shown in Fig. 7c, is used with the stress intensity factor correction. In stage 2, normalized stress intensity factor vs. crack length data, shown in Fig.12, is used as β correction factor to be used in AFGROW to calculate the fatigue crack growth in the fuselage skin. It should be noted that in stage 1, crack in the fuselage skin at the edge of the rivet hole has already propagated away from the hole. Therefore, in stage 2 "center through crack" application defined model of AFGROW is the right model to use for the fatigue crack growth calculations. AFGROW crack propagation results for the fuselage skin in stage 2 are presented in Table 5. Again, Forman model is used in the fatigue crack growth calculations. It should be noted that since the crack in the fuselage skin is center through crack in stage 2, c_i and c_f are not applicable.

Table 5 AFGROW results in stage 2 for the fuselage skin

w	262 *mm*
d	2.5 *mm*
t	1 *mm*
c_i	-
c_f	-
a_i	4.62 *mm*
a_f	25.75 *mm*
N	4360 *Cycles*

From Table 5, it is seen that in stage 2 crack in the fuselage skin propagates from an initial length of 4.62 *mm,* which is the final crack length in stage 1, to 25.8 *mm* in 4360 cycles. At a crack length of 25.8 *mm,* stress intensity factor reaches the critical fracture toughness K_{CR}, and the crack in the fuselage skin breaks the two stringer bays. Thus, in the demonstrative example given in this section, the fuselage structure can not withstand the crack in the fuselage skin over two stringer bays after a total number 180406 cycles, and the two-bay criterion is violated.

The crack propagation vs. flight cycles curve is given in Fig. 14 which clearly shows that crack propagates very fast in stage 2 after the break of the stringer. The demonstrative example shows that after the stringer is completely broken, as shown in Fig. 12, as the crack moves away from the broken stringer, normalized stress intensity factor fluctuates about 2.5, but does not drop. However, for the tensile loaded straight stiffened panels with broken stringer, normalized stress intensity factor drops to levels below 1.0 when the crack moves away from the broken stringer and approaches to the pristine stringer [4,17]. Bulging of the fuselage skin due to internal pressure and curvature are considered to be the main reasons for the higher normalized stress intensity factor in the curved fuselage skin compared to the straight panel case. Consequently, in stage 2 higher stress intensity factor causes fast crack growth in the fuselage skin.

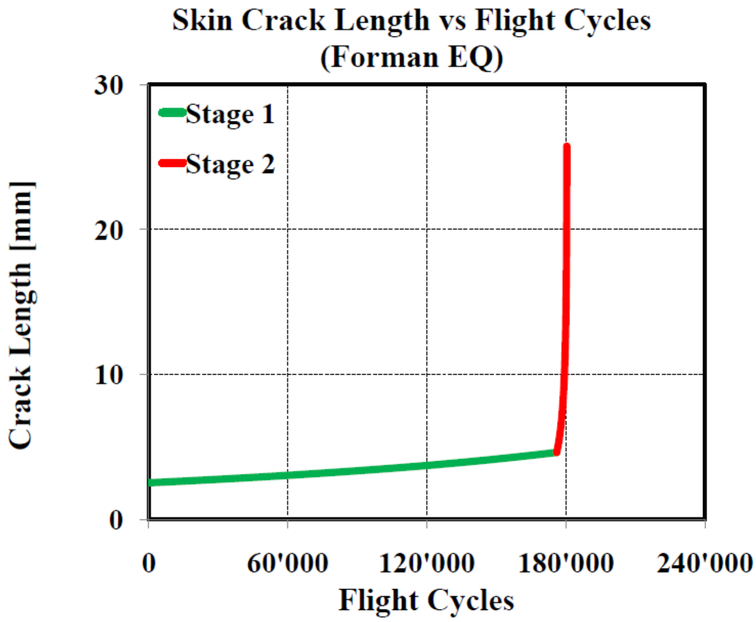


Fig. 14 Crack growth time history for the fuselage skin obtained by the Forman model with the experimentally determined Forman constants

CONCLUSION

Two stage damage tolerance evaluation of a stiffened aircraft fuselage panel with a circumferential crack and a broken stringer is presented. To demonstrate the two stage process of fatigue life calculation, simple fatigue loading is taken as the constant amplitude GAG pressure cycle. As a worst case scenario, it is assumed that double cracks start at the edge of a rivet hole both in the fuselage skin and in the stringer simultaneously. It is shown that in stage 2, after the stringer is completely is broken, stress intensity factor history, provided by the center through crack application defined model of AFGROW, has to be corrected to account for the broken stringer and other three dimensional effects. The effect of broken stringer and the three dimensional effects, such as bulging of the cracked skin due to pressure and curvature of the panel on the crack tip stress intensity factor are taken into account by calculating the stress intensity factor along the crack line by the three dimensional finite element analysis. It is shown that in stage 2, bulging of the skin panels and panel curvature can have significant effects on the normalized stress intensity factors. It is seen that unlike flat panels under uni-axial load, in the pressurized curved fuselage panel, stress intensity factor along the crack line remains almost constant, and does not decrease as the crack approaches to a pristine stinger. Curvature of the fuselage panel and bulging of the skin due to pressure are considered to be the main reasons for this behavior. For the demonstrative example, it is shown that after the stringer is completely broken in stage 2, crack propagates very fast. From a practical point of view, such a design can be considered poor since the very fast crack growth after the break of the stringer necessitates frequent inspection prior to stage 2. Although in order to appropriately characterize bulging cracks large deformations need to be considered in the finite element analysis, in the present article geometrically linear finite element analysis is implemented to demonstrate the two stage life prediction methodology of an aircraft fuselage structure with a circumferential crack and a broken stringer.

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