DURABILITY AND DAMAGE-TOLERANCE DESIGN AND ANALYSIS OF TITANIUM AIRFRAME STRUCTURES

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ABSTRACT

Due to its high specific strength, high toughness, good corrosion resistance, and excellent high-temperature performance, the use of titanium alloys for fabricating primary airframe structures such as the wing carry-through structure of an advanced aircraft was necessary. Because of the durability and damage-tolerance requirements, fatigue crack growth analyses were needed to be performed on these titanium components. This paper describes the procedure for performing such analyses.

KEYWORDS

Damage-tolerance; durability; fatigue-crack-growth; load interaction model; airframe structure; titanium alloy.

INTRODUCTION

The use of titanium alloys for building primary airframe structural components has been increased rapidly due to the fact that this family of materials offers several distinct advantages over other airframe materials such as aluminum alloys and steels. The advantages are:

1. High specific strength; i.e., strength-to-density ratio and comparable specific modulus

2. Excellent high-temperature performance

3. Good corrosion (including exfoliation, pitting, and stress corrosion) resistance

4. Good producibility, meaning it can be shaped, welded, formed, machined, etc, and still retain near-base-metal strength

5. Good compatability with advanced composite materials, both thermally and electrochemically

Ribs, spars, spar caps, and attachment lugs are typical examples of the structural parts of modern aircraft which are made of titanium alloys. A survey on an advanced aircraft shows that titanium parts contribute almost 20 percent of the total airframe structures by weight. Major titanium components of this aircraft system include the pivot lug and the cover of the wing carry-through structure, the nacelle support beam, the fuselage longeron, etc. All these structural components are classified as airplane safety-of-flight structures. The safety-of-flight structures are the structures whose failure would cause direct loss of the aircraft.

To protect the safety-of-flight structures from potentially deleterious effects of material, manufacturing, and processing defects, the U.S. standard Aircraft Structural Integrity Program (ASIP) (reference 1) specifies the damage-tolerance design approaches shall be used on such structures. Damage-tolerance designs are categorized into two general concepts based on airplane damage-tolerance requirements (reference 2):

1. The fail-safe concept where unstable crack propagation is locally contained through the use of multiple load paths or tear stoppers
2. The slow-crack-growth concept where flaws or defects are not allowed to attain the size required for unstable rapid propagation

Most of the titanium structures of this advanced aircraft are qualified as slow-crack-growth structures. Hence, the damage-tolerance design requirements for slow-crack-growth structures specified in reference 2 apply to these titanium structures. This paper presents a brief description of the damage-tolerance analysis methodology used in performing damage-tolerance analysis on these parts. A numerical example of such design analysis is also presented in this paper.

ASIP requirements also specify that durability design approaches shall be used on airframe structures in order to achieve complex structure systems with low in-service maintenance costs and improved operational readiness throughout the design service life of the aircraft. Durability analyses shall thus be conducted to demonstrate the economic life of the titanium structures of this advanced aircraft system is in excess of the design service lives (reference 3). This paper also presents a brief description of the durability requirements and analysis methods used for durability analyses on titanium structures.

DAMAGE-TOLERANCE ANALYSIS METHODOLOGY

Damage-tolerance analyses conducted on the titanium parts of this advanced aircraft system include the fatigue crack growth analysis and the residual strength analysis. All analyses were performed in the durability and damage-tolerance assessment (DADTA) task using a detailed level automated fatigue crack growth analysis computer code, CRKGRO, (reference 4). The following paragraphs briefly describe the analysis methodology implemented in CRKGRO.

CRACK GROWTH ANALYSIS METHODOLOGY

CRKGRO calculates crack growth in cyclic-loaded structures, based on linear elastic fracture mechanics (LEFM) principles on a cycle-by-cycle basis. The bi-slope Walker equation is used in the program as the baseline crack growth rate equation. The bislope Walker equation can be expressed as follows (reference 5):

For $\Delta K_p \geq \Delta K > \Delta K_{th}$

$$\frac{da}{dN} = C_1 \left[\Delta K/(1 - \bar{R})^{1-m}\right]^{n_1}, \quad 0 \leq \bar{R} > R^+_{cut}, \quad \bar{R} = R^+_{cut}$$

For $\Delta K_p < \Delta K > \Delta K_{th}$

$$\frac{da}{dN} = C_2 \left[\Delta K/(1 - \bar{R})^{1-m}\right]^{n_2}, \quad 0 \leq \bar{R} > R^+_{cut}, \quad \bar{R} = R^+_{cut}$$

where $\Delta K_{th} = (1 - R)\,\Delta K_{th0}$ is the threshold value of ΔK; C_1, n_1, C_2, and n_2 are the growth rate constants for lower slope (region I) and upper slope (region II), respectively; "m" is the Walker stress-ratio collapsing factor; R^+ is the cutoff value of the stress ratio R, above which no further stress ratio layering is shown in the da/dN versus ΔK plot; and ΔK_p is the ΔK value of the interception of the lower slope line to the upper slope line.

LOAD INTERACTION MODEL

The load interaction model used in CRKGRO is a combination of the generalized Willenborg retardation model (reference 6) with the Chang acceleration scheme (reference 7).

The generalized Willenborg retardation model can be expressed in the following mathematical form when there is a tensile overload:

$$(K_{max})_{eff} = K_{\infty max} - \Phi\left[K^{oL}_{max}\left(1 - \frac{\Delta a}{Z_{oL}}\right)^{1/2} - K_{\infty max}\right]$$

$$(K_{min})_{eff} = K_{\infty min} - \Phi\left[K^{oL}_{max}\left(1 - \frac{\Delta a}{Z_{oL}}\right)^{1/2} - K_{\infty max}\right]$$

and

$$\Phi = \frac{1 - K_{max_{TH}}/K_{\infty max}}{(K^{oL}_{max}/K_{\infty max}) - 1} = \frac{(1 - K_{max_{TH}}/K_{\infty max})}{R_{so} - 1}$$

where K_∞ is the stress intensity factor corresponding to the applied stress of the specific cycle which is being considered in the current calculation, K^{oL}_{max} is the stress intensity factor corresponding to the maximum stress of the overload cycle, Δa is the incremental growth following the overload, Z_{oL} is the overload interaction zone size, $K_{max_{TH}}$ is the threshold maximum stress intensity factor value, and R_{so} is the material overload shutoff ratio.

The effective stress-intensity-factor range and stress ratio are defined as:

$$(\Delta K)_{eff} = (K_{max})_{eff} - (K_{min})_{eff} = (\Delta K)_\infty$$

$$R_{eff} = (K_{min})_{eff}/(K_{max})_{eff}$$

Note that the effective stress-intensity-factor range $(\Delta K)_{eff}$ has the same value as $(\Delta K)_{\infty}$. Thus, the Willenborg model predicts the crack growth retardation by reducing the effective stress ratio below the remotely applied stress ratio.

For positive effective stress-ratio cases ($R_{eff} > 0$), the crack-growth-rate equation in CRKGRO takes the following form whenever there is an overload cycle:

$$\frac{da}{dN} = C_i[(\Delta K_{eff}/(1 - \bar{R}_{eff})^{1-m}]^{n_i}$$

$$i = 1,2$$

If the effective stress ratio is negative (i.e., $R_{eff} < 0$), the Chang acceleration scheme in conjunction with the Willenborg model is used in CRKGRO to account for the compressive load acceleration effect (reference 5):

$$\frac{da}{dN} = C_i[(1 + R_{eff}^2)^q (K_{max})_{eff}]^{n_i}$$

$$i = 1,2$$

where q is the acceleration index, which is determined from test data generated for a specific value of the negative stress ratio R and its R = 0 counterpart.

The effective overload interaction zone-size approach proposed by Chang (reference 7) has been incorporated into CRKGRO to account for the reduction of the overload retardation effect caused by a compressive spike load immediately following the oveload. The effective overload interaction zone size is defined in terms of the effective stress ratio (R_{eff}) as:

$$(Z_{oL})_{eff} = (1 + R_{eff})\ (Z_{oL}),\ R_{eff} < 0$$

where (z_{oL}) is the plastic zone size. In CRKGRO, the plane strain plastic zone size is used if the stress intensity factor at the maximum depth for a part-through crack is to be calculated. Otherwise, the plane stress plastic zone size is used. The plane strain and plane stress plastic zone sizes are (reference 8):

$$(Z_{oL})_{plane\ strain} = \frac{1}{6\pi}\left(\frac{K_{\infty\, max}}{F_{ty}}\right)^2$$

$$(Z_{oL})_{plane\ stress} = \frac{1}{2\pi}\left(\frac{K_{\infty\, max}}{F_{ty}}\right)^2$$

where F_{ty} is the material tensile yield strength.

DAMAGE ACCUMULATION SCHEME

The Vroman linear approximation method (reference 9) has been incorporated into this computer program as the damage accumulation scheme. The following paragraphs briefly describe the method.

For a given load spectrum, the Vroman damage accumulation scheme proceeds by considering a load step (i) and using σ_{max_i} and σ_{min_i} to calculate $(da/dN)_i$. The value of $(0.01a)/(da/dN)_i$ is then compared to N_i, where "a" is the instantanious crack size. If $(0.01a)/(da/dN)_i$ is greater than N_i, the crack growth for that particular load step i is $\Delta a = N_i \times (da/dN)_i$, the crack has grown from "a" to $(a + \Delta a)$, and the program proceeds to the next load step.

If $(0.01a)/(da/dN)_i$ is less than or equal to N_i, the crack size is $(a + 0.01a)$, and this load step is reexamined. This process continues with $(0.01a)/(da/dN)_i$ being compared to the remaining cycles in the step. When all load steps in the block or flight have been examined, the program then proceeds to the first step of the next block (or flight) and continues.

RESIDUAL STRENGTH ANALYSIS METHODOLOGY

The residual strength of a structural component containing a known size of crack can be automatically predicted by the CRKGRO program. The prediction methodology is based on the determination of the critical value of the stress intensity factor, K_{cr}, for a known crack configuration and sizes, under a given loading condition. This value is then equated to the fracture toughness of the material of the cracked body. Plane strain fracture toughness K_{IC} or the plane stress fracture toughness K_c of the material is used in the calculation, depending on the thickness of the cracked structure. For example, the residual strength σ_{res} of a panel containing a center-through crack is determined by using the following relationship:

$$\sigma_{res} = \frac{K_{cr}}{F_w\sqrt{\pi c}}$$

where c is the half-crack length and F_w is the width correction factor.

Since the damage-tolerance requirement specifies the need to calculate only the minimum residual strength of a structural component, the current practice is to calculate K_{lim} based on the design limit load or the maximum spectrum load, whichever is greater. The numerical value of K_{lim} is then compared to K_{cr} of the material at each calculated a_i. If $K_{lim}(a_i) < K_{cr}$, it indicates that the minimum residual strength of a structure with a size a_i is the design limit load or the maximum spectrum load, whichever is greater. When $K_{lim}(a_i) \geq K_{cr}$, instability occurs (i.e., the crack will propagate rapidly). The residual strength of the structure is thus reduced to zero.

DURABILITY ANALYSIS METHODS

It is necessary to conduct durability analyses on all primary airframe structures to demonstrate the economic life of the structural component of this advanced aircraft is in excess of the design service life when structures are subjected to the design service load spectra and the design chemical/thermal environment spectra. There are several durability analysis methods which can be used for performing the required analyses. The crack growth analysis code, CRKGRO, is being used as the primary analytical tool for predicting the economical lives of the structural components. Using this method, the economical life of a component is defined as the growth period of an initial flaw to its functional impairment size. The initial flaw size is assumed to be a_i = 0.01 inch, whether it is a surface crack, an edge corner crack, or a corner crack at a fastener hole, for most of the structural components. The 0.01-inch initial crack size was selected to represent the initial quality of a manufacturing part of the aircraft.

DURABILITY AND DAMAGE-TOLERANCE DESIGN ANALYSES

In the durability and damage-tolerance assessment task of the advanced aircraft system, durability and damage-tolerance analyses of the titanium structures were performed by CRKGRO. The following are the general procedures for performing the analysis:

1. Select baseline crack growth rate constants and load interaction parameters for service environment at which specific titanium part is exposed to: sump tank water (STW) environment or low-humidity air (LHA) environment.

2. Select proper crack code number which contains stress intensity factor solution for a specific crack required to be assumed to exist in the structure.

3. Input structure dimensions and initial flaw sizes. The assumptions of the initial flaw sizes shall meet the durability and damage-tolerance requirements.

4. Input fatigue spectrum loads.

To illustrate the preceding procedures, the durability and damage-tolerance analyses of the lower cover of the wing carry-through structure, which is made of recrystalized 6Al-4V titanium plate, is presented here as an example. In this example, an initial corner flaw was assumed to exist at the center fastener hole in the analysis. Since Taper-Lok fasteners are used, the beneficial effect of the interference-fit-fastener system was accounted for by reducing the initial flaw size to 0.01 from 0.05 inch, for damage-tolerance analysis. For durability analysis, the initial flaw size was assumed to be 0.005 inch.

The material properties, including the yield strength (F_{ty}), fracture thoughness (K_c), and baseline crack growth rate constants (C,n, and m), as well as the parameters used in the Willenborg-Chang load interaction model, were all based on the book value documented in Fracture Mechanics Design Analysis Material Properties Manual. The crack growth rate data for the STW and

LHA environments used in the design analyses are shown in table I . The sump tank water environment data were used in the analyses since the wing carry-through structure is closed to the fuel tank. Figure 1 shows the 6Al-4V titanium sump tank water da/dN data used to generate the baseline crack growth rate constants.

Table I

FATIGUE CRACK GROWTH RATE CONSTANTS AND PARAMETERS OF WILLENBORG/CHANG LOAD INTERACTION MODEL FOR 6Al-4V TITANIUM, RECRYSTALIZED CONDITION

MATERIAL AND PRODUCT FORM	Ti-6Aℓ-4V PLATE COND RA & COND DB		Ti-6Aℓ-4V Hand FORGING, COND RA
Environment	Low Humidity Air	Sump Tank Water	Low Humidity Air
Upper slope:			
C	4.745×10^{-11}	2.166×10^{-11}	6.201×10^{-11}
n	3.89	4.21	3.70
m	0.6	0.4	0.4
q	1.0	1.0	1.0
Lower slope:			
C	1.433×10^{-13}	5.779×10^{-14}	9.508×10^{-14}
n	6.22	6.73	6.45
m	0.6	0.4	0.4
q	1.0	1.0	1.0
Transition point:			
ΔK, KSI $\sqrt{\text{In.}}$	15.0	14.0	11.0
da/dN, Inch/Cycle	3.0×10^{-6}	3.0×10^{-6}	5.0×10^{-7}
ΔK_{tho} KSI $\sqrt{\text{In.}}$	4.5	4.5	4.5
R^+_{cut}	0.60	0.60	0.60
R_{cut}	-0.5	-0.5	-0.5
R_{so}	2.25	2.25	2.25

The fracture toughness value selected for the damage-tolerance analysis was K_c = 178 ksi$\sqrt{\text{in.}}$, for the 0.25-inch-thick plate, based on the published fracture toughness versus thickness plot for 6Al-4V titanium plate.

The wing carry-through-bending moment fatigue spectrum was used in the durability and damage-tolerance analyses. The spectrum is in the random cycle-by-cycle format which contains a unitblock of 81 flights. The maximum spectrum stress was σ_{max} = 36.5 ksi, while the design limit stress was σ_{lim} = 52.6 ksi.

All input data used in the sample analysis are shown in figure2 , which is a standard durability and damage-tolerance analysis input data sheet used for this aircraft project. Errors can be easily caught by reviewing this kind of input data sheet for a specific set of analyses. The results of the durability and damage-tolerance analysis of the lower cover of the wing carry-through structure are summarized in the standard table in figure3 . The table shows the current design meets durability and damage-tolerance requirements

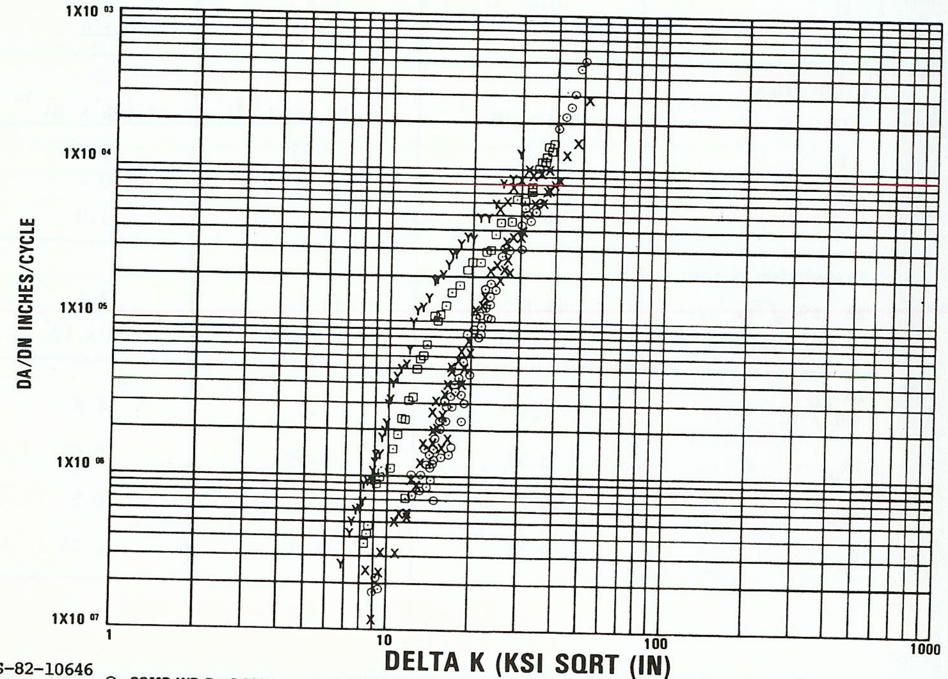
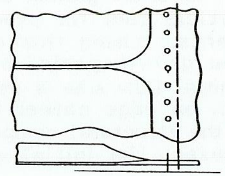


Figure1 . 6A1-4V Titanium, RA Condition, Sump Tank Water da/dN Data

DURABILITY AND DAMAGE TOLERANCE ANALYSIS - INPUT TO RASSP PROGRAM

Title: Wing-Carry-Through Structure
Drawing No.
Location: X_F 38.5 Y_F 982
Material: 6A1-4V Titanium , RA
F_{tu} 130 ksi F_{ty} 115 ksi
Gross Area=
Net Area =



Sketch

Spectrum I.D.: $DD540, LRCA 3B, WCT SPEC. DATA
Environment: ~~Sump Tank Water~~

Static Properties:
σ_{yield} 115 ksi
K_{Ic} 70 ksi$\sqrt{in}$
K_c 178 ksi$\sqrt{in}$

CRACK GROWTH PARAMETERS
c_1 1.052×10^{-18} c_2 2.126×10^{-11}
h_1 12.82 n_2 4.21
m_1 0.4 m_2 0.4
q_1 1.0 q_2 1.0
ΔK_{Trans} 8.0 ksi in da/dn_{Trans} 4×10^{-7}
R^+_{cut} +0.6 R^-_{cut} -0.5
R_{so} 2.25
ΔK_{TH} 4.5 A 0

STRUCTURAL ELEMENT LOAD AND STRESS DATA
Internal Loads Program
Element I.D. Type No. Node
Gross Stress Equation:

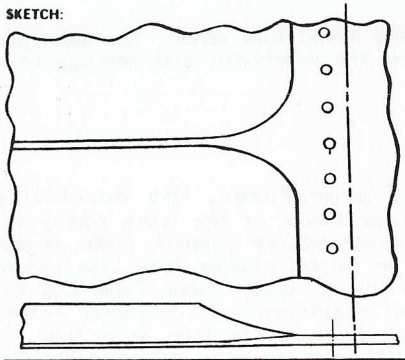
Static Design: Condition No.=
Limit Gross Stress=

CRACK GROWTH CALCULATION - DATA
Initial Crack Depth a_i 0.01 in.
Initial Crack Length c_i 0.01 in.
Effective Thickness 0.25 in.
Effective Width
Radius of Hole
Crack Library Code

FATIGUE DAMAGE CALCULATION - DATA
Dia. or Rad. Notch Depth
Plate Width
K_t Net/Gross Source

Figure2 . Typical Durability and Damage-Tolerance Analysis Input Sheet

SKETCH:



PROG. RUN I.D. $DD 709 Job 190
SPECTRUM I.D. $DD 540, LRCA 3B,WCT,SPEC.DATA
ANALYST DATE

SECTION OF AIRCRAFT: Wing-Carry-Through
LOCATION X_F38.5 Y_F 982 PART NO
NAME Lower Cover MATERIAL 6A1-4V Titanium,RA
F_{tu} 130 ksi F_{ty} 115 ksi
NOTES:

*Function inpairment size=plate thickness(0.25 inch.)

(1) Initial Crack Location	(2) Design Condition	(3) Initial Stress (KSI) Avg Gross	(4) Max Spect Stress (KSI) Avg Gross	(5) Crack Type	(6) K_{IC} AND K_c (KSI $\sqrt{In}$)	DURABILITY (7) Initial Crack Length (In)	DURABILITY (8) Final Crack Length (In)	DURABILITY (9) Predicted Life	DAMAGE TOLERANCE (10) Initial Crack Length (In)	DAMAGE TOLERANCE (11) Critical Crack Length at Limit Stress (In)	DAMAGE TOLERANCE (12) Predicted Life	(13) Crack Length After 1 Life (In)
		52.6	36.5	corner at hole	178	0.005	0.25*	6.81	0.010	6.90	9.27	0.012

Figure3 . Typical Durability and Damage-Tolerance Analysis Summary Sheet

CONCLUSION

This paper briefly describes the durability and damage-tolerance design requirements levied on the primary airframe structures of an advanced aircraft system in which titanium structures constitute almost 20 percent of the total airframe structures by weight. This paper also describes the durability and damage-tolerance analyses methodology and outlines the procedures for performing fatigue crack growth analysis. The durability and damage-tolerance analyses of the lower cover of the wing carry-through structure, which is made of recrystalized 6Al-4V titanium plate, is presented as an example.

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