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Fracture Mechanics Applications in Materials Selection, Fabrication Sequencing and Inspection

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Structural component failures associated with military aircraft have caused both operational limitations and concern over the current approach to assurance of structural integrity. Failures attributed to the presence of crack-like defects raised special concern over the probability of other defects existing in similar parts. For this reason, critical parts are being subjected to additional screening to evaluate their integrity when containing crack-like defects. This paper illustrates the use of fracture mechanics concepts to quantitatively compare the effects of various alternatives involved in design, manufacturing, assembly and quality assurance for a critical part.

Introduction

MATERIALS selection and structural design in the aircraft industry are based on five basic considerations of material response to loading,[1] namely:

1) Static ultimate strength and stiffness of undamaged, flaw-free material (σ_U, E, no flaws or damage).

2) Fatigue of undamaged, flaw-free material (safe-life design).

3) Residual static strength and stiffness of damaged structure (fail-safe, damage tolerance).

4) Fatigue of damaged, flawed material (inspection criteria, intervals and repair procedures).

5) Time dependent material behavior (creep, stress rupture, thermal fatigue, stress corrosion, etc.).

Safe-life design concepts require analysis or testing to show that the probability of any catastrophic fatigue failure is extremely remote for the assumed life of the structure. Increasing demands on materials to fill new or modified mission requirements, while maintaining high performance characteristics at minimum weight and cost, have brought about the use of high strength materials in safe-life designs for both military and commercial aircraft. The problems that are being experienced in fracture of aircraft components today are somewhat disturbing because they are now occurring in safe-life design components such as wing fittings and attachments, landing gears, and carry-through structure. Thus, use of high strength materials in components where safe-life design has traditionally and reliably been applied has produced service failures. Simply stated, this is because the increase in strength requirement very often brings a concomitant loss of toughness. Thus, where tough materials previously had been employed, materials that are subject to failure by small flaws are now being used. Often, the flaw sizes involved are not detectable by current nondestructive inspection (NDI) procedures.

The aircraft industry has effectively minimized the seriousness of the fracture problem by fail-safe design concepts for several years. Fail-safe design concepts require that a damaged structure continues to perform satisfactorily until discovery and remedial action can be accomplished. In employing this practice, the following items have been used in an attempt to prevent premature failure of components: 1) tough materials (large visible cracks prior to failure), 2) multiple load paths (redundancy), 3) crack stoppers, and 4) subcritical flaw growth prediction techniques (use of inspection intervals and crack propagation laws).

The fail-safe concept was developed as a result of failures encountered in the Comet series aircraft in the early 1950's. Current fail-safe design criteria specify the load requirements for structures having obvious damage and specify appropriate inspection intervals. In all cases the intent is to provide for sufficient damage tolerance so that fatigue cracks or other damage will be discovered before catastrophic failure becomes probable.

In the past, methods of calculating residual strength of damaged structures included effective width,[2] notch analysis[3] and fracture mechanics[4,5] techniques. In tough materials such as 2024-T3 aluminum, high residual strengths (near net section yield) have been obtained by proper use of crack stoppers such as properly placed straps, stringers or frames. When using materials that are lacking in toughness in relation to inherent flaw sizes and operating stress levels, it often becomes impractical to attempt arresting crack growth by adding structural reinforcements, since these reinforcements would need to be very close to one another, resulting in weight and cost penalties. When this is the case, redundancy and thus fail-safe conditions are most economically obtained by providing alternate load paths. However, in many applications even this tech-

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nique is not possible and the safe-life design approach must be employed.

Recent experience with aircraft structural failures has raised numerous questions regarding the current design approach for assurance of aircraft integrity, safety, and durability.[6] As an example, consider the failures during design verification testing of the F-111 aircraft and the F-111 crash at Nellis AFB, Nevada, on December 22, 1969. The cause of the crash was found to be a crack-like defect in the left wing pivot fitting.[7] The fact that the defect was generated during the manufacturing process and escaped detection in subsequent inspection steps raised concern over the probability of similar defects existing in other parts. For this reason a list of critical parts was defined and their processes, process controls, inspection techniques and service envelopes were evaluated by an interdisciplinary team of contractor, Air Force and university experts. This effort resulted in upgraded process controls and inspection techniques, and the use of a cryogenic proof test to screen flaws smaller than those detectable in NDI. The proof test and NDI criteria were in part based on application of fracture mechanics concepts.

The occurrence of such failures prompted the formation of a technology assessment group to: 1) reassess materials technology programs, manufacturing processes and inspection techniques to insure their adequacy, emphasizing increasing fundamental knowledge concerning fatigue, fracture mechanics, and high strength structural materials, 2) examine current structural integrity design philosophy including design criteria, margins of safety, verification and testing methods, and procedures for introducing new materials.

The activities of these groups have culminated in revision of military requirements for aircraft structural integrity,[8] in the form of a new military standard.[9] This revision introduced fracture mechanics, damage tolerance, slow crack growth, and quantitative flaw detection capability concepts into early design, specification, qualification and life cycle considerations.[9,10] These aspects are discussed in this paper, which outlines a procedure for using fracture mechanics and damage tolerant concepts for establishing the safety and reliability of critical aircraft structural components. This paper illustrates the use of fracture mechanics concepts to quantitatively compare the various alternatives involved in design, manufacturing, assembly and quality assurance and evaluate the possibility of premature failure for critical components.

Fracture Analysis

Identification of Fracture Critical Components

Since it is impossible to subject all of the components comprising a particular aircraft design to detailed scrutiny, it is necessary to devise a set of screening criteria to identify fracture critical components. These criteria will isolate for further study: primary structure loaded significantly in tension, components whose function is vital to safety of flight or landing, monolithic rather than redundant structure, components whose failure has large cost impact, either through down-time or cost of repair, and prior service experience problem areas.

Examination of available summaries of in-service failures[11-16] indicates the greatest number of failures occur in: aft fuselage, main frames, outboard wing plates and skins, inboard wing main frames, and inboard wing plates and skins.

Examination of failures that were very significant to fleet operations and maintenance[11] resulted in a different rating, with the most failures associated with: landing gear components, especially forgings, high strength materials such as 4340 steel, 7075-T6 aluminum, bolt and rivet holds, sharp corners and surface flaws, and, corrosion, stress corrosion, fit-up and residual stresses.

Based on these findings and the criteria for identifying fracture critical parts, examples used in this work will be main landing gear (MLG) components from a large bomber or transport aircraft.

Risk Analysis

The initial assumptions of operating stress spectrum, environment and material employed in preliminary design and previous service experience are used to conduct a risk analysis that identifies potential problem areas for each critical part. In this method, a profile sheet is used to rate potential dangers associated with factors involved in material selection, design, manufacturing, assembly and quality assurance. Each factor is assigned a numerical value for its corresponding risk, and the values for each factor summed to give the total risk for a critical part. The major drawback associated with this method is the subjective nature of the rating system. The severity of risk assigned to each factor depends on the background and experience of the personnel constructing the profile. In addition, it is difficult to quantify the impact of various potential problems, e.g., untempered martensite around holes vs improper cadmium-titanium plating for steel parts. It is also difficult to compare the relative risk associated with two different parts. This can be accomplished through use of fracture mechanics concepts to provide a means of quantitative definition of service life for various alternatives.

A sample risk profile for a main landing gear component from a large bomber or transport aircraft is presented in Table 1. Examination of the risk profile identifies potential problems related to: Low toughness and rapid subcritical flaw growth, leading to small critical flaws and shortened service lives. Sensitivity to processing operations such as welding, heat treatment and post heat treatment machining that produce degradation in fracture properties. Difficulty in detecting small flaws in large, hard-to-handle components. Ramifications for each of these items will be discussed in subsequent sections of this paper.

Fracture Mechanics Analysis

In this fracture mechanics analysis, it is assumed that flaws, discontinuities, inhomogeneities, surface scratches, corrosion pits, etc., are unavoidable in structural components.[5] These natural defects may nucleate cracks that grow in response to service loadings and cause fracture when a critical size is attained. The extension of these cracks is cyclic in nature, following fluctuations in applied stresses. The increment of crack extension (Δa) in a brief period of time representing a few cycles of stress (ΔN) is assumed to be related to the slope of a crack length vs cycles curve as given by $\Delta a = (da/dN)\Delta N$.

The rate of crack growth depends on the material, environment, stress/strain history of the component, crack geometry and component geometry. For a particular material in a specific environment a single parameter, the stress intensity range (ΔK), has been shown to describe time-dependent crack growth,[17]

$$\frac{da}{dN} = C(\Delta K)^n \tag{1}$$

In this expression C and n are empirically determined constants and ΔK is defined by

$$\Delta K = \Delta\sigma Y \sqrt{\pi a} \tag{2}$$

Table 1 Sample risk profile for a typical main landing gear component for a large transport or bomber aircraft

Item	Low	High	Reason for high risk value
Material[a]			
Characterization	X		
Limitations		X	Low toughness at 260–280 ksi
Manufacturing sensitivity	X		
Mechanical sensitivity		X	Notch sensitive at 260–280 ksi
Environmental sensitivity		X	Stress corrosion
Availability/development status	X		
Pertinence of specifications	X		
Design			
Part configuration	X		
Material adequacy	X		
Processing requirements	X		
Premature failure causes			
Static stresses	X		
Cyclic stresses		X	Highly stressed part
Fracture mechanisms		X	Subcritical flaw growth
Corrosion	X		
Processing induced damage		X	Post heat treat machining, plating
Part qualification			
Fail-safe/safe-life	X		
Damage tolerance		X	Monolithic part
Service record		X	High mortality rate in similar parts
Damage detection		X	Critical areas obscured in service
Repair/replacement ease	X		
Manufacturing			
Planning/process controls	X		
Processing methods			
Adequacy for part	X		
Potential for damage		X	Post heat treat machining, plating
Quality required	X		
Facilities/equipment	X		
Personnel training level	X		
Quality assurance			
Testing methods			
Development status	X		
Detection limits		X	Small critical flaw size
Inspection of raw material	X		
Processing/assembly controls	X		
Part inspection		X	Part size makes inspection difficult

[a] Material used is 4340 steel with σ_U = 260–280 ksi.

where $\Delta\sigma$ = cyclic stress range, a = flaw depth‡, γ = a correction factor for part and crack geometry. Allowance for varying range ratios ($R = \sigma_{min}/\sigma_{max}$) is achieved by modifications in this stress intensity range computation.[18,19]

The prediction of aircraft component service life requires knowledge of component and crack geometry, expected stress spectrum, crack growth rate data for the material in the environment to be encountered, and initial crack size. The initial crack depth (a_i) is taken to be just below the lower limit of detection for available nondestructive inspection techniques. The critical crack depth (a_c) is computed by use of

$$a_c = \frac{1}{\pi\gamma^2}\left[\frac{K}{\sigma_W}\right]^2 \tag{3}$$

using K = plane strain fracture toughness (K_{Ic}) and σ_W = maximum stress (σ_{max}) encountered during the ground-air-ground cycle (GAG), once each flight. Since crack propagation is a path dependent process, the number of cycles required for growth of the initial flaw to critical dimensions may be approximated by a process of iteration. The stress intensity range (ΔK) corresponding to the initial flaw depth (a_i) and maximum ground-air-ground stress range ($\Delta\sigma$) is computed using Eq. (2). This value of ΔK is used with constant-amplitude laboratory test data to determine the crack growth rate, da/dN, as shown in Fig. 1. The crack extension increment (Δa_1) during a period (ΔN_1) is calculated and added to a_i to give a new crack depth. This value of crack depth, $a_i + \Delta a_i$, is used to determine a new stress intensity range and a new crack growth rate. Application of a second group of fatigue cycles (ΔN_2) yields another increment of crack extension (Δa_2), and another new crack depth ($a_i + \Delta a_1 + \Delta a_2$). This interative process is continued until the growing crack attains the critical size previously computed to cause failure. The total number of cycles (ΣN) required to reach critical size defines the predicted service life. Since the maximum ground-air-ground stress occurs once each flight, the life in cycles is equivalent to the life in flights. Other stresses lower and higher than the peak ground-air-ground stress are experienced more and less frequently than once each flight, respectively. The iterative approach outlined herein can employ computer techniques to consider either crack growth due to peak ground-air-ground stress only or all stresses in the flight spectrum.[20,21]

Life values calculated by this method may be conservative if time is required for the flaw to attain the optimum shape and orientation for propagation. In addition, retardation of subsequent crack growth due to load sequencing in the use spectrum can extend fatigue life.[22-26] Empirical techniques have been devised to predict crack growth retardation following periodic application of high loads.[27,28]

‡This discussion is based on part-through-the-thickness or surface flaws, in which a represents the crack depth and $2c$ represents the crack length, or surface trace. Stress intensity determinations depend on crack depth, not length.

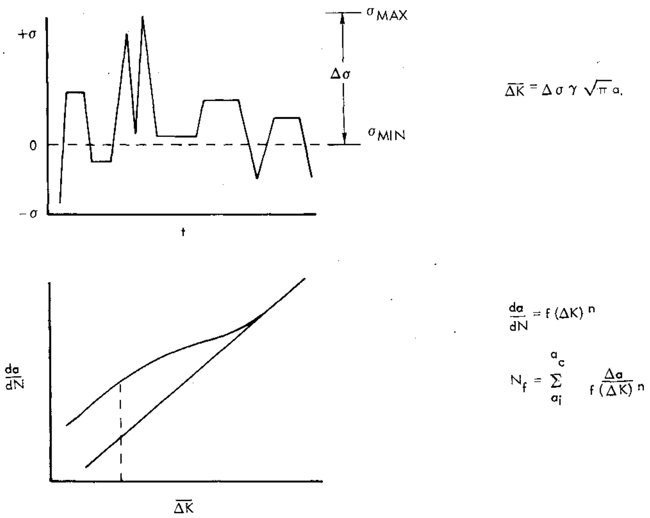


Fig. 1 Fracture mechanics analysis of fatigue data.

However, the amount of conservatism should be established for a particular spectrum by component testing, as calculations performed using basic fracture mechanics concepts have shown good agreement with observed fatigue life values.[29-31]

Design Applications

A fracture mechanics analysis using the stress level, environment and material assumptions used in preliminary design provides a baseline value of service life for each fracture critical component. If this analysis produces acceptable levels of service life and flaw detection ability, the normal iterative design process[32] can progress. However, if the crack growth rates lead to service lives below the desired value, or if critical crack sizes are so small as to be below current NDI detection levels, the designer must evaluate: reducing stress levels to lower crack growth driving force, expanding NDI capability to reveal small flaws, and changing material to obtain increased flaw tolerance and slower subcritical crack growth. Achievement of an acceptable level of service life and structural integrity may require one or all of these actions.

As an example, consider the fracture mechanics analysis summarized in Table 2. A typical 4340 steel main landing gear part, with a service life requirement of 6000 flights, was subjected to a simulated service spectrum whose peak stress was 78.5 ksi. Crack growth from an initial length of 0.075-in. to a critical length of 0.606-in. required 7550 flights, well above the target value of 6000 flights. Selection of the 78.5 ksi operating stress level was based on Fig. 2, which presents a relationship between GAG stress and service life. The graphical relationship was constructed by determining the expected life for a series of GAG stresses varying from 98.3 to 63.7 ksi. Since crack growth rate decreases with decreasing ΔK, lower GAG stresses produce longer service lives. Determination of the level of NDI detection required was based on Fig. 3, which presents a relationship between initial flaw length and service life. Attainment of the 6000 flight target service life required exclusion of flaws ≥0.150-in. long. The acceptability of this flaw detection requirement was judged by examination of previous experimental work,[33,34] such as the plot in Fig. 4, which indicated ≈85% probability of detection for 0.150-in. long surface flaws. In addition, examination of the B-1 damage tolerance requirements[35] revealed a 0.150-in. initial flaw length could be assumed in design.§ Thus, exclusion of flaw lengths ≥0.150-in. and use of a GAG stress value of 78.5 ksi produced the desired service life of 6000 flights. Use of a larger GAG stress or flaw detection limit would not produce the same level of structural integrity for the component geometry, service stress spectrum and environment assumed in this case. If a higher stress level is mandated by weight considerations, the designer must consider changing to a material with increased flaw tolerance and re-

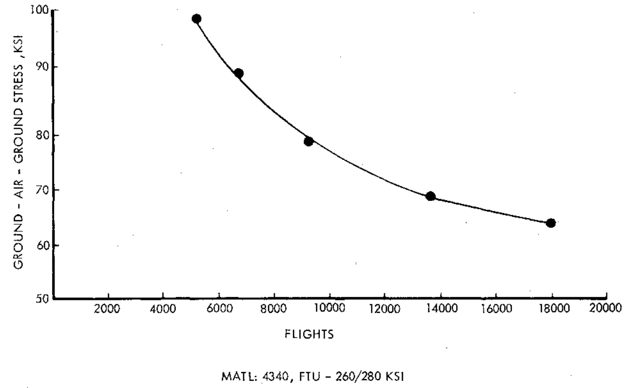


Fig. 2 Relationship between design stress and estimated service life for typical MLG component.

Table 2 Crack growth analysis for a main landing gear component from a typical large transport or bomber aircraft

Examine section 1.0-in. thick so that plane strain is achieved and a/t corrections are not needed.
Material is AISI 4340 steel with $\sigma_U = 260$ ksi, $K_{Ic} = 53$ ksi $(\text{in.})^{1/2}$, $K_{Iscc} = 27$ ksi $(\text{in.})^{1/2}$.
Flaw is part-through type, with $a/2c = 0.33$.
Initial flaw = NDI detection limit of $2c = 0.075$ in., $a = 0.025$-in.
Critical flaw: $a_c = 0.444\,(K_{Ic}/\sigma_{\max})^2 = 0.202$-in., $2c_c = 0.606$-in.
Crack growth rate data experimentally obtained in moist air.

a, in.	ΔK, ksi $(\text{in.})^{1/2}$	da/dN, in. × 10^{-6}/flt	N, flts	Δa, in.	$a + \Delta a$, in.	$2c$, in.
0.025	21.6	9	2680	0.025	0.050	0.150
0.050	31.0	18	2770	0.050	0.100	0.300
0.100	43.9	38	1320	0.050	0.150	0.450
0.150	53.5	66	780	0.052	0.202	0.606

ΣN = 7,550 flts

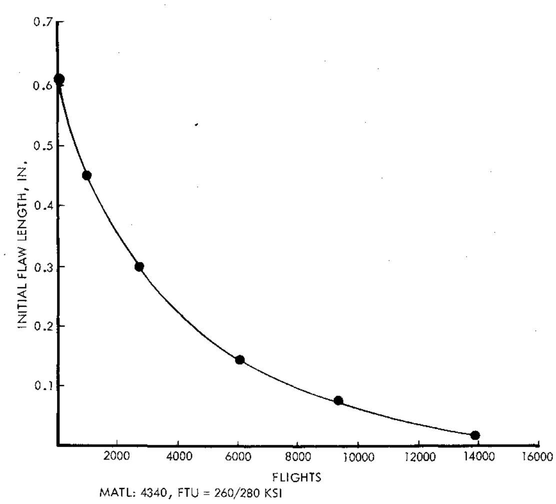


Fig. 3 Relationship between initial flaw size and estimated service life for typical MLG component.

§Smaller flaw sizes were actually assumed in design since flaw detection capability at a 90% probability, 95% confidence level was experimentally demonstrated.

Table 3 Flaw detectability limits for various ndt techniques[a]

Material	Visual[b]	Magnetic[b] particle	Penetrant[b]	X-ray[c]	Ultrasonic[c]	Delta ultrasonic[c]
0.25 in. extrusion						
7075-T6511	0.03	...	0.25	0.50	0.25	...
4340 V	0.03	0.30	0.35	0.50	0.20	...
0.020 in. plate						
7075-T6	...	...	0.040	...	0.050	...
Ti-6Al-4v	...	...	0.032	0.070	0.100	...
0.125 in. plate						
7075-T6	...	...	0.030	...	...	...
Ti-6Al-4v	...	...	0.050	0.130	0.090	0.034
0.500 in. plate						
7075-T6	...	...	...	0.460	0.290	...
Ti-6Al-4v	...	...	0.035	...	0.150	0.090

[a] 100% detection level in laboratory tests.
[b] Surface flaws of known location.
[c] Both known surface flaws and unknown embedded flaws.

duced subcritical flaw growth rates. Considering the MLG component used previously as an example, a GAG stress of 98.3 ksi results in a service life of 5400 flights (Fig. 2), below the target life of 6000 flights. The effect on service life of changing the material to steels with higher fracture toughness, D6ac or 300M, is shown in Fig. 5. Use of D6ac, which has higher toughness value but more rapid subcritical crack growth than 4340, actually resulted in a decrease in life expectancy, emphasizing the importance of da/dN for long life applications. Using 300M, with a higher toughness value and reduced subcritical crack growth, produced an estimated life of 6300 flights, above the 6000 flight target value.

Processing and Assembly Controls

In addition to flaws existing in raw materials and flaws induced during service, the possibility of flaw generation and property degradation during the fabrication cycle must be considered. Flaw size and fracture property levels after fabrication should be the same or superior to those assumed in design. A list of critical processing steps for the high strength steels assumed for the MLG component previously discussed would include:

Casting or forging—inclusions, porosity
Welding—incomplete fusion, heat affected zone
Machining quality—surface finish, stress concentrators
Heat treatment—quench rate, temper embrittlement, decarburization
Stress relief—straightening, residual stresses

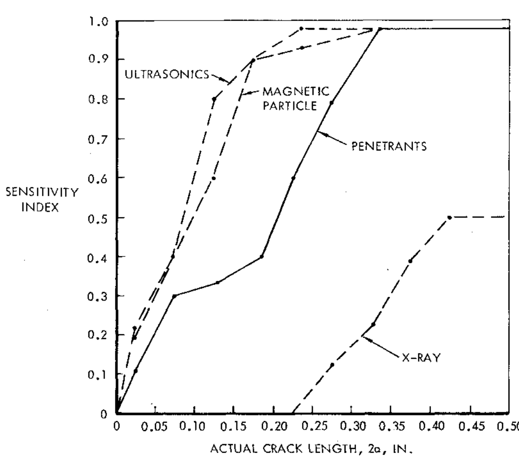


Fig. 4 Comparison of NDT inspection methods on steel cylinders[33].

Machining after heat treatment—hole drilling, grinding, untempered martensite
Plating—H_2 embrittlement, poor adhesion, coating flaws
Assembly—residual stresses, eccentric loads, interference fasteners, fretting

These critical processing steps should be identified during the risk analysis effort for each fracture sensitive part. Fracture mechanics analysis can be employed to determine the impact on structural integrity and relative risk associated with improper performance of each critical processing operation. Since improper processing will change fracture properties and induce flaws, performance of a second fracture analysis using modified values of those fracture parameters pertinent to the design application (K_{Ic}, K_{Iscc}, da/dN, da/dt, initial flaw size) will reveal the impact on service life. Operations shown to be significantly detrimental should be excluded if possible. If these possibly detrimental operations must be retained, a stringent fracture control plan must be implemented. In the past it has been common practice to take measures to ensure the fatigue quality of a structure by specifying surface finish, heat treatment, fastener type, design details, etc. For fracture sensitive materials, the designer must also consider flaw sensitivity by specifying on the engineering drawing fabrication methods, process controls, and inspection techniques that ensure flaw size and fracture properties following fabrication are at least equal in quality to those assumed during design. Procedural details associated with such an effort have been implemented by transferring process controls and intermediate inspection requirements from the engineering drawing to pertinent production documents that control shop operations.[7] The controls imposed on in-house organizations were also imposed on subcontractors.[7] Fracture control plans have required verification of fracture properties by testing of coprocessed specimens after critical process steps such as heat treating, welding and diffusion bonding.[35]

Table 4 Distribution of critical and initial flaws in Ti-6Al-6V-2Sn drag brace

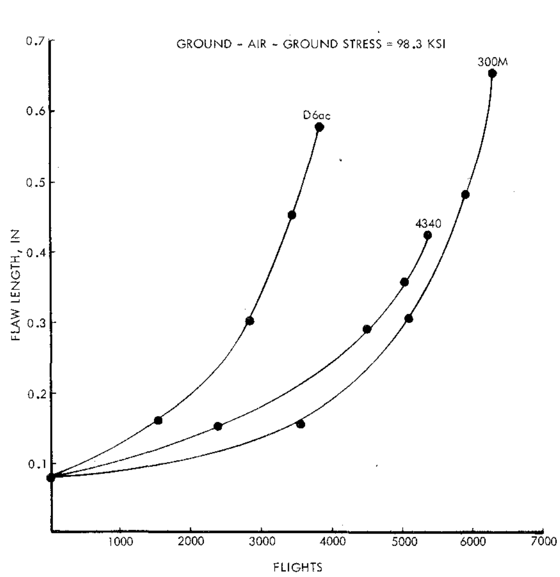
Section no.	Maximum GAG stress, ksi	Critical flaw length, in.	Initial flaw length, in.
1	22.6	1.910	1.230
2	26.4	1.460	0.780
3	31.7	1.030	0.330
4	35.6	0.800	0.120

Inspection Requirements

Following the determination of final stress spectra, service environment, desired life and processing sequence, the requirements for periodic inspection can be established. Just as in fatigue life estimations, there are a number of unknowns which preclude an absolute determination of life. These include: variation in material properties, distribution of flaw size, shape, location, actual load and environmental history for an operational fleet of aircraft, and uncertainties in theory and analysis methods. Thus, an equivalent to a fatigue life reduction factor is needed to account for these unknowns. This can be satisfied by establishing several inspection intervals within the estimated safe-life based on flaw size calculations. Structures are inspected frequently enough so that a crack that is not detected in one inspection will not grow to critical size due to normal working stresses during the inspection interval. One criterion requires inspections at one-quarter the time required for a crack to grow from the minimum inspectable size to critical size.[35] This allows for the possibility of not detecting a crack during an inspection and the statistical scatter in results. Another important parameter requiring definition is the maximum allowable initial flaw that permits attainment of desired life.

To demonstrate the calculation of maximum allowable initial flaw size and inspection intervals, consider a sample landing gear component fabricated from Ti-6Al-6V-2Sn alloy in the solution treated and annealed (STA) condition. Mechanical properties were σ_U = 165 ksi, σ_Y = 155 ksi, K_{Ic} = 40 ksi (in.)$^{1/2}$ At the zone of minimum thickness, 0.67-in., for an assumed maximum GAG stress of 35.6 ksi, the critical flaw length was 0.800-in. Crack growth analysis techniques (Table 2) were used to construct the service life versus initial flaw length plot shown in Fig. 6. The target service life of 20,000 flights was attained if the initial flaw length was less than 0.120-in. Assurance of service life required inspections to this level every 5,000 flights, as part of scheduled maintenance operations.

The flaw detection level required by the flaw growth analysis may be too stringent for the type of NDI specified. Table 3 presents flaw detectability limits for some common NDI methods. These results were obtained under laboratory conditions, with some surface flaw locations



MATL	FTU,KSI	KIC, KSI √IN	2c, IN	NF,FLTS
D6ac	233	65	0.579	3850
4340	260	53	0.390	5400
300M	270	69	0.606	6300

Fig. 5 Expected service life and critical crack size for typical MLG component fabricated from D6ac, 4340 and 300M steels.

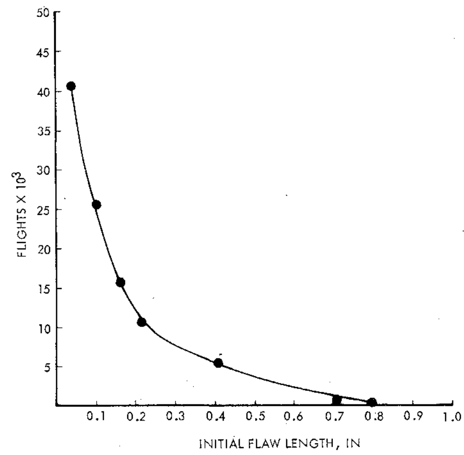


Fig. 6 Plot of predicted service life vs initial flaw size for Ti-6Al-6V-2Sn drag brace.

known before the test.[33,34] It has been estimated that flaw detection limits for production parts will be twice the levels shown, and that flaw detection limits for structure during maintenance inspections will be three times the levels shown.[33] Assuming inspection limits in Ti-6Al-6V-2Sn to be similar to those in Ti-6Al-4V, detection of open 0.120-in. long surface flaws by penetrant inspection should be possible. However, 0.120-in. long embedded flaws, especially those with small volume, will be difficult to detect, even by ultrasonic techniques.

This situation leaves two possible alternatives. First, the existing NDI capability can be upgraded through use of better personnel, equipment and techniques, or multiple inspections may be required. This may require the use of sophisticated procedures that permit detection of small flaws[36-39] or proof testing.[7,40]

Detection of very small flaws in large structures can be very costly. In order to reduce this cost impact, it is desirable to employ a selective NDI methodology, which subjects only critical areas to stringent or multiple inspections, allowing less sensitive areas to be processed normally. Four different regions of the Ti-6Al-6V-2Sn landing gear component were examined. Table 4 summarizes the maximum GAG stress, critical crack length and initial crack length required to yield the desired, 20,000 flight life for each stress region. Assuming a surface flaw detection limit of 0.150-in. long, only the peak stress region requires special attention.

Summary

The challenge of establishing the safety and reliability of critical aircraft structural components has been aggravated by the use of materials with higher strength but lower toughness and high subcritical crack growth characteristics. Consequently, procedures accounting for the presence of undetected flaws that are based on fracture mechanics and damage tolerance concepts have been developed for use in establishing design, material selection, fabrication and inspection requirements. This procedure included the following operations.

Fracture critical components were identified by application of screening criteria that isolate primary structure that is 1) loaded primarily in tension, 2) monolithic, 3) vital to flight or landing safety, and 4) identified as a problem area through examination of service experience and failure data. A risk profile that rates possible dangers associated with material design, manufacturing and quality assurance was used to identify potential problem areas for each critical part.

The initial predictions of operating stress spectrum and environment were used to estimate critical crack size and service life for candidate materials. If the critical crack size was so small as to be below current NDI detection levels, or if crack growth rates produced shortened service lives, alternatives of: reducing stress level, expanding NDI capability, and changing material; were evaluated to achieve an acceptable risk level at a minimum weight and cost penalty.

In addition to flaws existing in the raw materials, the possibility of fracture property degradation and flaw generation during the fabrication cycle were considered. Process controls, inspection procedures and frequency of inspection were developed to prevent flaw growth to critical size during operations.

Inspection intervals were calculated so that a flaw escaping detection at one inspection did not grow to critical proportions before the next inspection. A method for reducing inspection costs by determining the level of inspection needed for each section of a part was outlined.

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