THE APPLICATION OF FRACTURE MECHANICS METHODOLOGY TO THE DAMAGE TOLERANCE ANALYSIS OF THE BOEING 757 AIRPLANE

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ABSTRACT

Recent updates of airworthiness regulations require extensive use of fracture mechanics in damage tolerance analysis. The Boeing Company has developed new damage tolerance technology to satisfy these regulations. This new technology combines fracture mechanics methodology with accumulated service experience to develop rational procedures for the assessment of residual strength, damage growth and damage detection for aircraft structures. The developed technology has been applied successfully to the certification of new Boeing commercial airplanes. This paper presents an outline of the damage tolerance analysis of the Boeing 757 airplane, and discusses the application of fracture mechanics principles in that analysis.

KEYWORDS

Damage tolerance; structural integrity; airworthiness; fail-safe; residual strength; crack growth; fatigue; environmental deterioration; accidental damage; damage detection; inspection program; structural maintenance.

INTRODUCTION

Commercial airplane structures have been certified according to a "fail-safe" philosophy for over 20 years. According to this fail-safe philosophy, the structure is designed to have fail-safe strength for a failure or obvious partial failure of a single principal element. This design philosophy has resulted in an excellent safety record. However, experience has shown that actual cracking patterns of aircraft structure are frequently different from the single element failure assumed in the fail-safe design philosophy. Structure adjacent to a fatigue crack may contain a number of secondary cracks. Such multiple-site damage can significantly affect crack growth, and reduce residual strength and crack arrest capability of structure. These considerations plus the emergence of fracture mechanics capabilities to deal with complex damage configurations have resulted in recent updates of airworthiness regulations for new and maturing airplanes [1-3] where the traditional fail-safe philosophy is enhanced by "damage tolerance" philosophy. In the damage tolerance philosophy, structural safety is assured by requiring inspections with a high probability of detecting the damage in the fleet before the strength is reduced below regulatory fail-safe limits which must be maintained in the presence of multiple-site damage in the structure.

The Boeing Company has developed new technology to satisfy the current airworthiness regulations for damage tolerance assessment of new and maturing airplanes. This technology combines fracture mechanics methodology with accumulated service experience to develop rational procedures for assessment of residual strength, damage detection period and damage detection probabilities for various inspection techniques. This enables airline operators to develop flexible structural maintenance programs for timely detection of environmental deterioration, accidental damage and fatigue damage. This new technology has been validated by extensive damage tolerance testing. Detailed discussions on the development of this technology are found in other publications [4-6]. The objective of this paper is to show the application of this technology to the damage tolerance certification of the Boeing 757 airplane and to discuss the role of fracture mechanics.

DAMAGE TOLERANCE ANALYSIS OF THE BOEING 757 AIRPLANE

For the purpose of damage tolerance analysis, the entire 757 airplane structure is divided into two classifications: (i) Structurally Significant Items; and (ii) Other Structure. Any detail, element or assembly which contributes significantly to carrying flight, ground, pressure or control loads, and whose failure could affect the structural integrity necessary for the safety of the *airplane* is classified as a Structurally Significant Item (SSI). Thus, all primary structure is comprised of SSIs which are selected based on considerations of consequences of cracking, inspectability, stress and corrosion environment, multiple-site cracking and structural redundancy. The SSIs are numbered according to the Air Transport Association (ATA) numbering system, and grouped into major structural zones (e.g., wing, fuselage, empennage). For the damage tolerance analysis of the 757, a total of 312 SSIs have been selected. Table 1 shows the SSIs in the 757 outboard wing box (ATA 57-20).

For each SSI, structural safety is assured by either damage tolerance criterion or safe-life criterion. Damage tolerance criterion is preferred and is used for most of the 757 structure. Safe-life criterion is based on conservative fatigue life to crack initiation and is used only where an effective damage tolerant structure cannot be achieved within the limitations of geometry, inspectability or good design practice. An example of structure that is not conducive to damage tolerance design is landing gear which is certified according to a safe-life criterion.

In the damage tolerance design philosophy, structural airworthiness is substantiated by proper consideration of the following elements:

1. Residual Strength: The fail-safe load requirements are established by the regulatory agencies. Generally, the fail-safe load is the limit load. The maximum allowable damage including multiple-site secondary damage that the SSI can sustain at the required fail-safe strength is calculated. This allowable damage is the upper limit of the damage growth computation for the SSI.

2. Damage Growth: The damage growth period from a length below which there is a negligible probability of detection to the maximum allowable damage for fail-safe strength capability is calculated under operating loads. This period is available for detecting the crack before the strength is reduced to the fail-safe limit.

3. Inspection Program: A structural inspection program is required for the SSI. The inspection methods and frequencies are selected to ensure timely detection of corrosion, stress corrosion, accidental damage and fatigue damage. If damage is detected, the structure is repaired to provide ultimate strength capability.

The relationships of these elements are shown schematically in Fig. 1.

There are several instances when the maximum allowable damage for a SSI is mal function evident or obvious to airline personnel during routine interface with

TABLE 1 Structurally Significant Items in the 757 Outboard Wingbox

SSI NO.	TITLE
57-20-01	Upper Surface Typical Skin/Stringer and Skin/Rib Shear Tie Attachment
57-20-02	Upper Surface Spanwise Splice and Spar Chords to Skin Attachment
57-20-03	Upper Skin Attachment to MLG Beam and Trunnion Support Fittings
57-20-04	Upper Surface Hole at Fuel Filler Cap
57-20-05	Upper Skin to Nacelle Strut Attachments
57-20-06	Upper Surface Side of Body Splice
57-20-07	Lower Surface Typical Skin/Stringer Details
57-20-08	Lower Surface at Rib Shear Tie and Support Fittings
57-20-09	Lower Surface at Drain Installation
57-20-10	Lower Surface Spanwise Splice Stringers/Skin Attachment
57-20-11	Lower Surface Skin to Spar Chords Attachment
57-20-12	Lower Surface at Access Holes
57-20-13	Lower Surface Attachment to MLGB Support Fitting
57-20-14	Lower Surface Attachment to Nacelle Fittings
57-20-15	Lower Surface Typical Drybay Skin/Stringer Attachment
57-20-16	Lower Surface at Drybay Flame Arrestor Installation
57-20-17	Lower Surface Side of Body Splice
57-20-18	Front Spar Typical Details
57-20-19	Front Spar at Nacelle Fitting Attachment
57-20-20	Rear Spar Typical Details
57-20-21	Rear Spar at Trunnion Fitting Attachment
57-20-22	Rear Spar at MLGB Support Fitting Attachment
57-20-23	Rear Spar at Outboard Flap Support Fitting Attachments
57-20-24	Front Spar at Side of Body Splice and Terminal Fitting
57-20-25	Rear Spar at Side of Body Splice and Terminal Fitting
57-20-26	Non-Shear Tied Ribs
57-20-27	Shear Tied Ribs Typical Details
57-20-28	Shear Tied Ribs at Internal Back-up Fittings
57-20-29	Outboard Flap Forward Flap Track Support Back-up Fittings
57-20-30	Nacelle External Support Fittings
57-20-31	Side of Body Rib

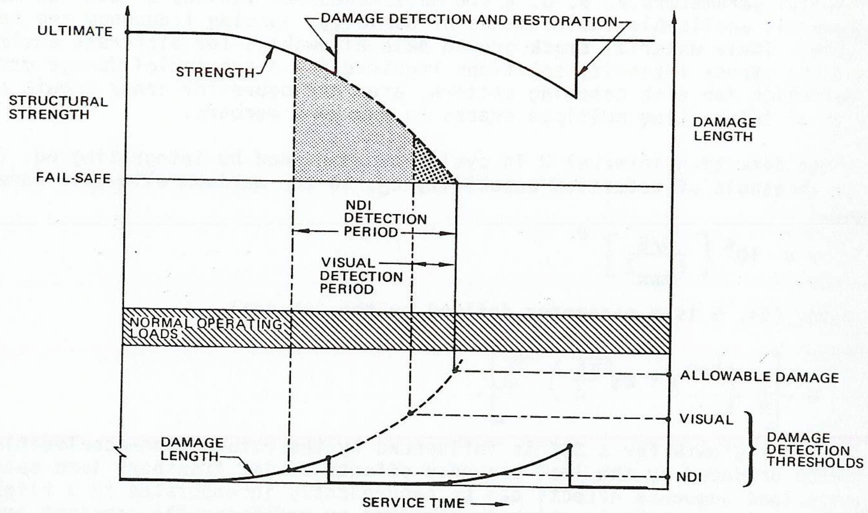


Fig. 1. Damage tolerance design of structures

TABLE 2 Structural Analysis Requirements

STRUCTURAL CATEGORY			TECHNIQUES OF ASSURING SAFETY	REQUIRED EVALUATION
OTHER (SECONDARY)	DAMAGE TOLERANCE DESIGN	1 SECONDARY STRUCTURE	DESIGN FOR SAFE SEPARATION OR LOSS OF FUNCTION	CONTINUED SAFE FLIGHT
STRUCTURALLY SIGNIFICANT ITEMS (SSIs) (PRIMARY)	DAMAGE TOLERANCE DESIGN	2 DAMAGE OBVIOUS OR MALFUNCTION EVIDENT	ADEQUATE RESIDUAL STRENGTH WITH EXTENSIVE DAMAGE THAT IS OBVIOUS DURING WALK AROUND OR BY MALFUNCTION	RESIDUAL STRENGTH
STRUCTURALLY SIGNIFICANT ITEMS (SSIs) (PRIMARY)	DAMAGE TOLERANCE DESIGN	3 DAMAGE DETECTION BY PLANNED INSPECTION PROGRAM	INSPECTION PROGRAM MATCHED TO STRUCTURAL CHARACTERISTICS	RESIDUAL STRENGTH CRACK GROWTH INSPECTION PROGRAM
STRUCTURALLY SIGNIFICANT ITEMS (SSIs) (PRIMARY)	SAFE-LIFE DESIGN	4 SAFE-LIFE	CONSERVATIVE FATIGUE LIFE	FATIGUE

the airplane. For example, excessive fuel leakage as a result of cracking of a lower surface wing skin, and significant changes in fuselage differential pressure or leakage rate due to cracks in a fuselage skin, can be cited as instances of obvious damage and malfunction evidence, respectively. For such SSIs the compliance with regulations is established by the verification of residual strength capability. Table 2 summarizes pertinent structural safety analysis requirements.

Fracture mechanics plays a vital role in the residual strength and damage growth elements of the damage tolerance analysis. State-of-the-art fracture mechanics principles have been applied in these two elements to develop rational procedures which are validated by extensive verification testing by The Boeing Company [4].

RESIDUAL STRENGTH

The structural configurations of the 757 SSIs vary from simple monolithic sheets with stress concentrations to complex built-up redundant load path structure. Structural materials of the SSIs exhibit varying amounts of ductility and strain hardening. The residual strength method must consider failure modes such as net section failure and plastic instability, in addition to fracture failure mode. The method must also account for stable crack growth prior to instability and load redistribution. For a built-up structure with multiple-site damage, each component of the structure must be checked for residual strength to determine the lowest value of the residual strength of the structure.

Linear Elastic Fracture Mechanics (LEFM) characterizes the fracture resistance of a material for the case of small scale yielding at the crack tip, by the stress intensity factor K. Plane strain fracture toughness, K_{IC}, characterizes the fracture resistance of the material for thick sections, where plane strain conditions prevail. For thin sheets with through-the-thickness cracks, the plane stress fracture toughness parameter, K_c, characterizes the fracture resistance of the material. In the case of a built-up structure, although it is possible to define the stress intensity factor at the crack tip, the failure strength of the structure is not always predicted by the fracture toughness of the material. Large number of residual strength tests of both unstiffened and stiffened panels conducted by The Boeing Company together with test data available in the literature [4] have led to a composite chart (Fig. 2) combining fracture mode with plastic instability failure mode for determining the residual strength of any component of a built-up structure.

In Fig. 2, L_Y is the crack length which defines the failure mode and is calculated from the stress intensity factor K, and material parameters, using the small scale yielding concepts of LEFM; F_{RS} is the residual strength at crack length L; F_{NY} is the stress which causes the net section to yield; C is a load redistribution factor.

The upper bound of the residual strength is determined by the stress which causes the net section to yield.

Stress intensity factor K, and load redistribution factor C, are catalogued in a handbook for all structural configurations required by the SSIs. In instances where the solutions of stress intensity factors or load redistribution factors were not available in the literature or from Boeing tests, they were developed by analytical/numerical methods.

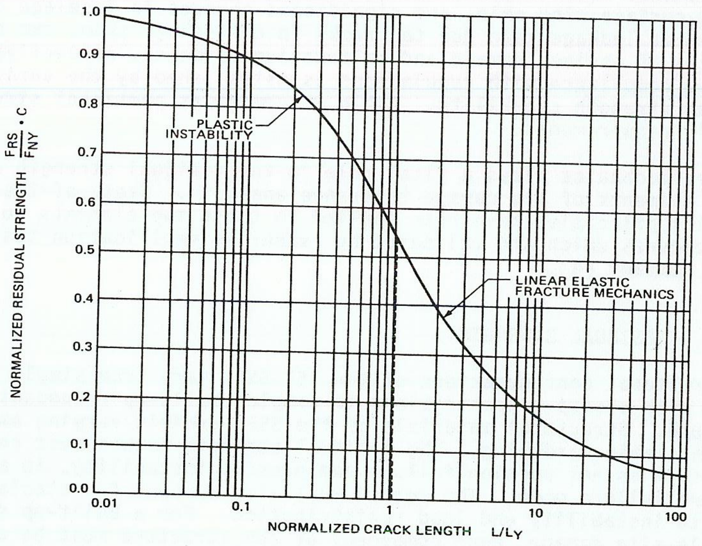


Fig. 2. Residual strength of a structure

DAMAGE GROWTH

The fracture mechanics method of characterizing crack growth by the stress intensity factor is followed in this analysis. The rate of crack propagation is a function of material properties, operating stresses, and structural geometry. Under constant amplitude loads, crack propagation rate per cycle, $\frac{dL}{dN}$ is given by

$$\frac{dL}{dN} = n\,10^{-4}\left[\frac{Z\,K_{max}}{M}\right]^{p} \qquad (1)$$

In eq. (1), K_{max} is the maximum stress intensity factor in the cycle; n denotes the number of crack tips; M, p are the material crack growth parameters; and Z is the "stress ratio term" given by

$$Z=\begin{cases}(1-R)^{q} & \text{for } 1.0>R>0.0\\ 1-uR & \text{for } R\leqslant 0\\ 1.1 & \text{for } R\leqslant -1.0\end{cases} \qquad (2)$$

where u, q are material dependent parameters and R is the stress ratio of the cycle given by

$$R = f_{min}/f_{max} \qquad (3)$$

Stress intensity factor K is defined in terms of stress f, configuration factor Y, and load redistribution factor C.

$$K = f\,Y\,C\sqrt{\frac{\pi L}{n}} \qquad (4)$$

Material parameters M, p, q, u are determined for various structural materials under all applicable environments of humidity, loading frequency and temperature. These material crack growth data allowables for different environments and the stress intensity solutions required for a sequential damage growth evaluation for each cracking pattern, are catalogued for crack growth computation of interacting multiple cracks in adjacent members.

Damage detection interval N in cycles, is obtained by integrating eq. (1) from the threshold of detection capability L_o, to the maximum allowable damage L_{AD}.

$$N = 10^{4}\left[\frac{M/G}{f_{max}Z}\right]^{p} \qquad (5)$$

In eq. (5), G is a parameter defined by the integral,

$$G = \left[\frac{1}{n}\int_{L_o}^{L_{AD}}\left(Y\,C\sqrt{\frac{\pi L}{n}}\right)^{-p} dL\right]^{-1/p} \qquad (6)$$

The crack growth for a SSI is influenced by the retardation-acceleration phenomena produced by the load sequence effects of jet transport load spectra. These load sequence effects can be conveniently incorporated in a flight-by-flight spectra analysis which is obtained by modifying the constant amplitude crack growth technique as follows.

Operating load conditions and spectra are defined for a segmented flight of the 757 Airplane based on anticipated usage [4]. Normalized spectrum crack growth calculations are performed for each flight segment, including the Ground-Air-Ground segment with a fixed value of M/G. These damage growth increments are added together to obtain the normalized damage growth per flight T, which can be used with the actual M and G values, to determine the number of flights N_F, to grow the crack from L_o to L_{AD}. In actual practice, a crack stress rating "S" is defined in terms of the normalized damage growth per flight T. The crack stress rating, S, may be viewed as a single parameter characterization of the cyclic stress environment of a flight, to achieve a standard performance. The damage detection interval N_F in flights, is given by

$$N_F = 10^{4}\left[\frac{M/G}{S}\right]^{p} \qquad (7)$$

A load sequence model has been incorporated in the crack growth procedure under spectrum loads. This model is based on the effective stress concept to simulate the retardation-acceleration phenomena of representative jet transport spectra [7]. In this procedure, the maximum and minimum stresses of a cycle are replaced by effective maximum and effective minimum stresses for crack growth purposes.

$$\begin{aligned} f_{max_{eff}} &= f_{max} - f_{red} + f_{inc}\\ f_{min_{eff}} &= f_{min} - f_{red} + f_{inc}\end{aligned} \qquad (8)$$

where

$$\begin{aligned} f_{red} &= \beta\left(f_{OL} - f_{max}\right) \leqslant F_{red}\\ f_{inc} &= \gamma\,\left|f_{UL}\right| \leqslant F_{inc}\end{aligned} \qquad (9)$$

In eq. (9), f_{OL} is the over-load stress responsible for crack retardation; f_{UL} is the under-load stress responsible for crack acceleration; β, γ, F_{red} and F_{inc} are parameters chosen by the reduction of test data for a material. In the description of operating loads by means of exceedance spectra used in this analysis, f_{OL} and f_{UL} are determined from the exceedance curve of the Ground-Air-Ground stresses for the flight at specific exceedance levels.

INSPECTION PROGRAM

Aircraft structural safety must be assured in the event of damage resulting from three independent sources; environmental deterioration (corrosion and stress corrosion), accidents and fatigue. Both accidental damage and environmental damage can be considered random events that can occur at any time during the operational life of an airplane. Therefore, inspection for timely detection of these two types of damage should be done at specified calendar time intervals. Fatigue damage growth following any form of initiation is a result of cyclic stresses. Therefore flight cycles are used as a basis for determining the frequency of inspections for fatigue crack growth.

An initial structural inspection maintenance program is developed for all of the 757 structure, by a Structures Working Group consisting of representatives from airlines, regulatory agencies and The Boeing Company. This initial program is based primarily on detection requirements for environmental deterioration and accidental damage. The program specifies the type of visual inspections (surveillance or detailed) and the frequency, for all possible inspections from structural zones [5].

Detection of fatigue damage before it becomes critical is the final control in the assurance of structural safety. Each SSI must be inspected during the damage detection period for that SSI. The inspection type and frequency are defined to provide a probability of detection greater than a requirement that is established from service experience. For the purpose of providing a direct quantitative measure of the probability of detecting the fatigue damage, The Boeing Company has developed a Damage Tolerance Rating (DTR) [5]. Using the DTR system, for each SSI, the adequacy of the initial structural inspection program is evaluated by Boeing for timely detection of fatigue damage. A brief description of this DTR system follows.

Damage detection is a function of the fleet size, the number and size of cracks and the number and type of inspections. The probability of detecting fatigue damage on a specific SSI in a fleet of aircraft is a function of three independent probabilities, P_1, P_2 and P_3:

1. The probability, P_1, of inspecting an aircraft with a damaged detail is a function of the number of aircraft inspected and their position in the fleet relative to a given fatigue life. For example, if it is assumed that damage has occurred, $P_1 = 1$ when the candidate fleet is inspected.

2. The probability, P_2, of inspecting the detail considered will generally be 1, or 0 for an individual airline with a given maintenance program for different check levels.

3. The probability, P_3, of detecting the damage in the detail, is a function of inspection type and the number of inspections. For a single inspection of the detail considered ($P_2 = 1$) on an aircraft with damage ($P_1 = 1$), the probability of detecting the damage, $\hat{P}_3$, is a function of crack length and the detection method, and is expressed in terms of a three parameter Weibull distribution,

$$\hat{P}_3 = 1 - \mathrm{EXP}\left[-\left(\frac{L-L_o}{\lambda-L_o}\right)^{\alpha}\right] \tag{10}$$

where L_o is the threshold length below which cracks are assumed undetectable; λ is the characteristic crack length of the distribution; α is the distribution shape parameter; and L is the inspectable crack length at the time of inspection.

$\hat{P}_3$ is derived from service cracking data for all the inspection types used in past jet transport maintenance programs. For each SSI, the inspection program specifies the number, type, and frequency of inspections, during the damage detection interval, N_F. A crack length and a $\hat{P}_3$ value are calculated for each inspection. Assuming $P_1 = P_2 = 1$, the cumulative probability of detection for a single airplane during the period N_F is obtained from the product of the probabilities of non-detection in the number of inspections m.

$$P_3 = 1 - \prod_{i=1}^{m} (1-\hat{P}_{3_i}) \tag{11}$$

If cracking should occur, it is likely that more than one aircraft in the fleet will have a crack at a given SSI. These multiple cracks in the fleet will increase the number of inspections performed on detectable cracks before the first crack reaches critical size. These additional opportunities increase the probability of detection which can be calculated in a manner similar to equation (11).

Considering the entire inspection program in the fleet, the cumulative probability of damage detection, P_D, is given by

$$P_D = 1 - \prod (1-P_{d_i}) \tag{12}$$

where $P_{d_i} = P_1 . P_2 . P_3$ for the i^{th} inspection.

The Damage Tolerance Rating, DTR, is defined by

$$\mathrm{DTR} = \frac{\log(1-P_D)}{\log(0.5)} \tag{13}$$

Alternatively, the probability of detecting a crack is given by

$$P_D = 1 - (2)^{-\mathrm{DTR}} \tag{14}$$

It may be seen from eq. (14) that the numerical value of DTR provides an equivalent number of safe opportunities for detecting a crack where each opportunity has an equal probability of success or failure.

Acceptable detectability levels, and consequently the required DTR values, have been established for each SSI based on the evaluation of previous service cracking history and engineering judgment of cracking circumstances. An acceptable inspection program for any SSI for structural safety is the one whose calculated DTR is greater than or equal to the required DTR. Thus, in this DTR system, flexibility is provided for defining a structural inspection program to assure structural safety at the required high level.

For most of the 757 structure, the initial inspection program established for detecting corrosion, stress corrosion and accidental damage also provided adequate opportunity for detecting fatigue damage. For those items where the initial program was inadequate to detect potential fatigue damage, the feasibility of various inspection options was evaluated by the DTR system. The options considered were (i) more intensive visual inspection, (ii) increased frequency of inspections, and (iii) use of NDI procedures. These options may be coupled with a sampling program for structural inspections. These feasible inspection programs were provided to the airlines as examples of adequate maintenance procedures.

The DTR system together with data provided for each item enables each airline to define its own modified inspection plan with suitable methods and convenient intervals, provided that the DTR from the defined inspection program equals or exceeds the required DTR for that SSI. The modified inspection program for detection of fatigue damage will begin after a suitable threshold which will be determined from the experience of similar inspection programs applied to mature Boeing airplane fleets.

EXAMPLE

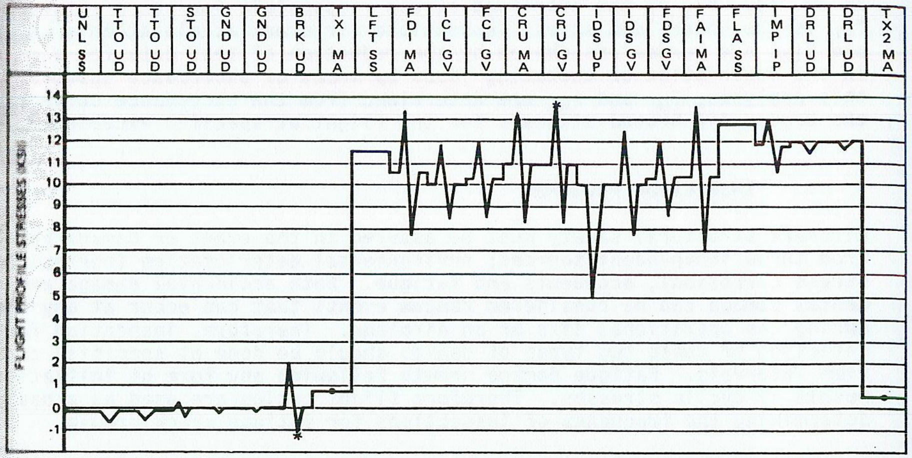
The damage tolerance analysis of the 757 wingbox lower surface is presented as an example to illustrate the application of the developed technology. The SSIs in the outboard wingbox lower surface are listed in Table 1. There are a similar number of SSIs in the center section wingbox lower surface. Each SSI can be classified into one of two categories (Table 2) for damage tolerance analysis: (i) category 2 structure which requires only verification of residual strength capability because damage is obvious during walkaround; (ii) category 3 structure for which safety is established by a planned inspection program. The residual strength analysis method described earlier is used to demonstrate fail-safe strength capability for all SSIs in category 2 and to calculate the maximum allowable damage for fail-safe strength capability for all SSIs in category 3.

The most critical cracking sequence in the wing box structure is obtained with an initial lead crack in a stiffening member (stringer or spar chord) at a fastener and a secondary crack in wing skin at the same fastener. Under operating loads, the crack progresses in the stiffening member and eventually fails it completely, thereby accelerating the growth of the skin crack.

Figure 3 shows the typical spectrum crack growth computations for a single flight and the 'S' rating for a location in the wing lower surface. The flight profile stresses in Fig. 3 are for once-per-flight occurrences. The contours shown in Fig. 4 provide a display of the relative severity of cyclic stress environment in the wing lower surface.

Damage tolerance at all locations in category 3 structure of the wing lower surface is substantiated by evaluation of a planned inspection program using the DTR system. This procedure is illustrated by an example given below.

The typical stringer to skin attachment (SSI NO. 57-20-07) is classified as category 3 structure for stringer 11 at WS 313.5 (Fig. 8). At this location, crack length is determined as a function of flights (eq. (7)) from an initial detectable crack length to the maximum allowable crack length, following the



FLIGHT CONDITION FOR FLIGHT NO. 4			MEAN STRESS	MANEUVER		GUST		GROWTH PER 10,000 FLIGHTS		
				ALT STRESS	CYCLES	ALT STRESS	CYCLES	MAN	GUST	TOTAL
UNL	SS	UNLOADED	-.20					0.000		0.000
TTO	UD	TURN TOW	-.44	-.29				.000		.000
TTO	UD	TURN TOW	-.36	-.20				.000		.000
STO	UD	STRAIGHT TOW	-.17	.34				.000		.000
GND	UD	GROUND TURN	-.25	-.09				.000		.000
GND	UD	GROUND TURN	-.22	-.07				.000		.000
BRK	UD	BRAKED ROLL	.38	1.66				.000		.000
TXI	MA	TAXI-PREFLIGHT	.71	-.04	11.9			.000		.000
LFT	SS	LIFT OFF	11.47					0.000		0.000
FDI	MA	FLAPS DOWN DEPARTURE	10.45	2.82	3.2			.125		.125
ICL	GV	INITIAL CLIMB	10.08			1.67	2.5		.019	.019
FCL	GV	FINAL CLIMB	10.13			1.76	2.9		.027	.027
CRU	MA	CRUISE	10.93	2.74	3.3	2.75	5.1	.129	.202	.331
IDS	UP	INITIAL DESCENT	7.45	-2.60	1.0	2.31	2.9	.013	.058	.071
FDS	GV	FINAL DESCENT	10.20			1.71	2.6		.023	.023
FAI	MA	FLAPS DOWN APPROACH	10.22	3.28	3.1			.180		.180
FLA	SS	FLARE	12.66					0.000		0.000
IMP	IP	LANDING IMPACT +SU/SB	11.75	1.32	2.0			.013		.013
DRL	UD	DRIFT LANDING	11.62	-.33	.5			.000		.000
DRL	UD	DRIFT LANDING	11.74	-.21	.5			.000		.000
TX2	MA	TAXI-POSTFLIGHT	.29	.09	8.6			.000		.000
						SUBTOTAL GROWTH		.460	.329	.789

STRESS MODE – TENSION M/G – 20.00	CRACK STRESSES – KSI			EQUIV CYCLES PER FLIGHT	GROWTH PER 10,000 CYCLES	GROWTH PER 10,000 FLIGHTS
	MINIMUM STRESS	MAXIMUM STRESS	STRESS RATIO			
GAG STRESSES	−1.27	13.68	−.09	1.27	(ONCE/FLT) .254	(ARRAY) .322
CRITICAL CONDS	BRK UD	CRU GV				

S = M/G REQUIRED FOR 10000 FLIGHTS = 20.6 CGR = CRACK GROWTH GAG RATIO = (.253564/ 1.110786) = .23 ANALYSIS INCLUDES A FLT AVG LOADS SEQUENCE FACTOR	1.111 TOTAL GROWTH

Fig. 3 Spectrum crack growth

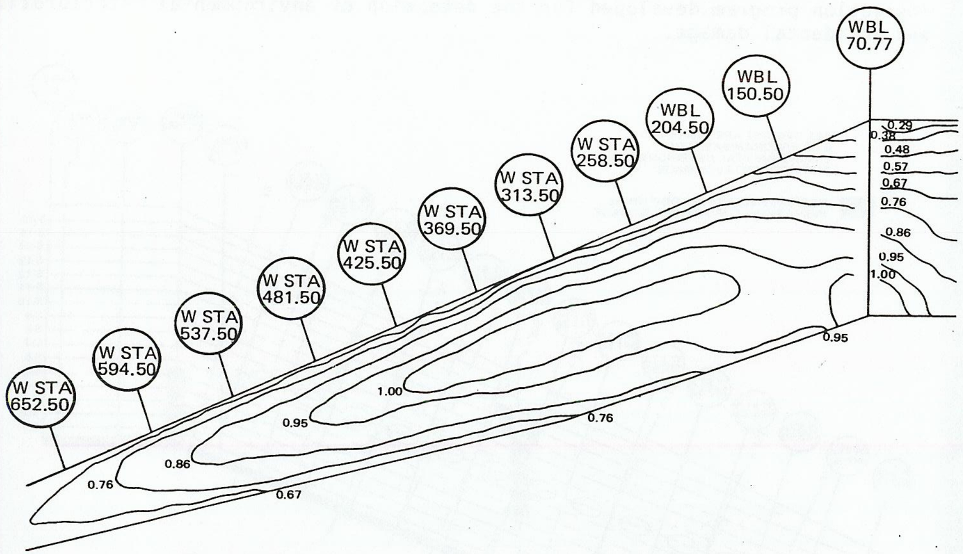


Fig. 4. Normalized contours of S ratings for the 757 wing lower surface

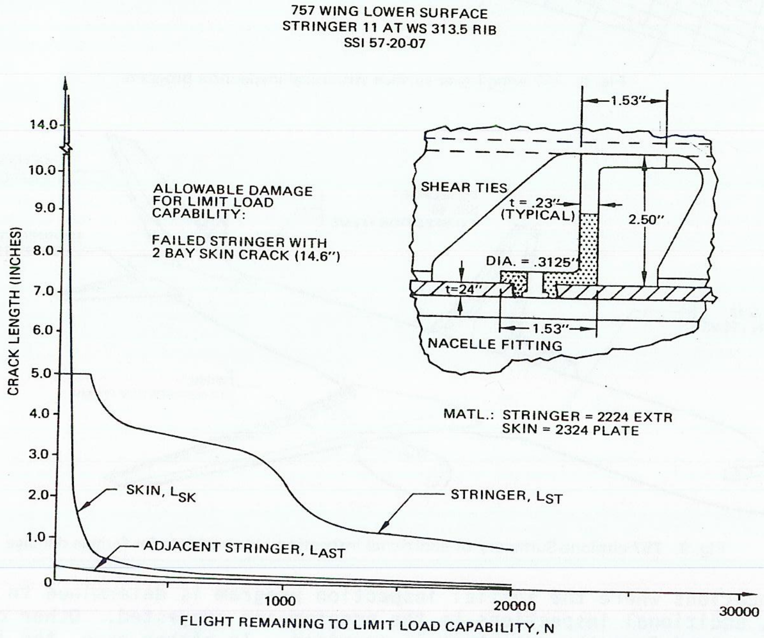


Fig. 5. Crack growth under spectrum loads

most critical cracking sequence described earlier. The results are shown in Fig. 5, where crack length is plotted as a function of flights remaining to the allowable damage. Independent cracks in adjacent stringers are considered to assess properly the residual strength for skin or adjacent stringer failure due to load redistribution. Inspectable crack length can differ from the actual crack length depending on the inspection direction. For each direction, the inspectable crack length is derived from the actual crack length as shown in Fig. 6.

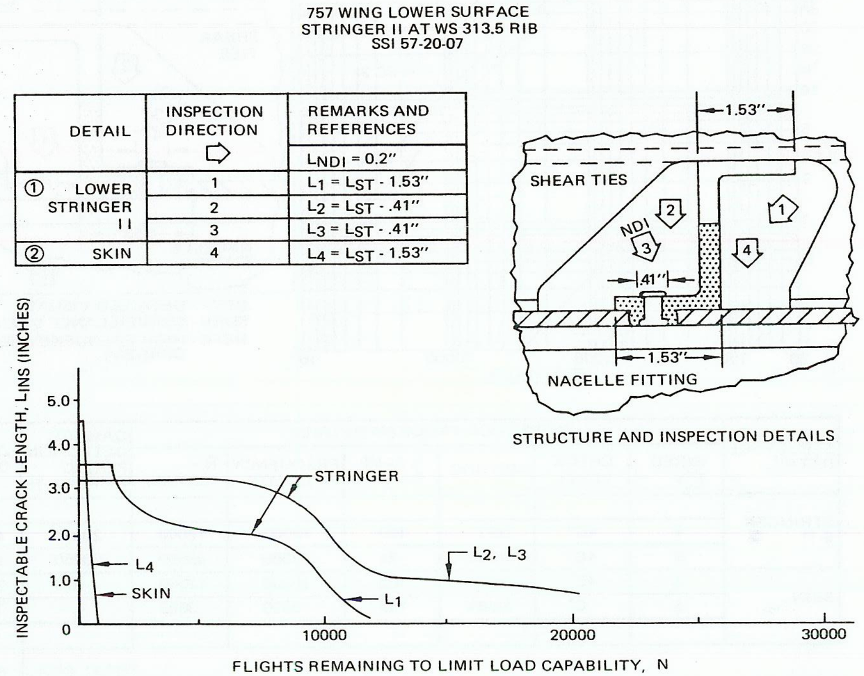
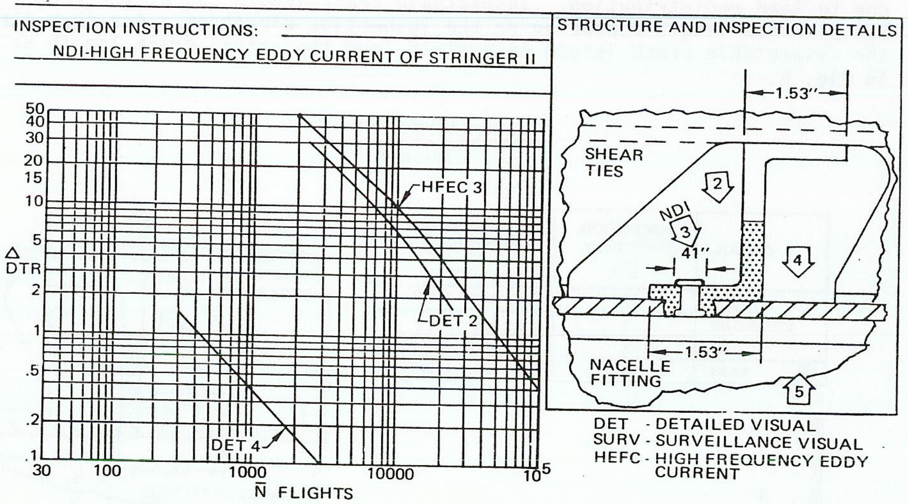


Fig. 6. Inspectable crack length for inspection directions

According to the initial structural inspection program based on detection requirements of environmental deterioration and accidental damage, the example location is inspected by visual surveillance externally at every C check interval (3,000 flights) and by detailed visual inspection inside the fuel tank at every 4C interval (12,000 flights). The DTR rating of this initial inspection program is calculated as 4.9. At this location, the required DTR rating is 6.0. Therefore, the initial inspection program is inadequate to detect fatigue damage and must be modified. The DTR requirement could be met by a detailed visual inspection internally at 10,000 flights, but this may not be practical because it would require access inside the fuel tank at a frequency different from that for the surrounding structure. One feasible approach is to modify the initial inspection plan by adding NDI high frequency Eddy Current inspection of stringer with a 25% rotational sampling which involves inspections of a quarter of the fleet with a frequency of 12,000 flights until all the candidate airplanes are inspected at least once. This rotational sampling provides an additional incremental DTR of 1.1. These results are shown in Fig. 7.

757 WING LOWER SURFACE
STRINGER 11 AT WS 313-5 RIB
SSI 57-20-07



STRUCTURE DETAIL	INSPECTION PROGRAM DETAILS						DAMAGE DETECTION PERIOD No FLIGHTS	Δ DTR
	DIREC ⇨	CHECK LEVEL	METHOD	% SAMP Ro	FREQUENCY F-FLIGHTS	N̄ = 100 F/Ro		
STRINGER S-11	2	4C*	DET	100	12000	12000	20250	4.9
	3	4C	HFEC	25	12000	48000	20250	1.1
SKIN	4	4C*	DET	100	12000	12000	355	0.0
	5	C*	SURV	100	3000	3000	— —	0.0
							TOTAL DTR	6.0
							REQUIRED DTR	6.0

*Initial inspection program for environmental deterioration and accidental damage

Fig. 7. A feasible structural inspection plan for fatigue damage detection

The damage tolerance evaluation of category 3 structure in the 757 wing lower surface showed that only a small portion of the highly stressed region required additional inspections for detection of fatigue damage as illustrated in Fig. 8. This fatigue sensitive region in the wing is covered by 13 inspection items which specify additional feasible inspection programs for timely detection of fatigue cracks.

The damage tolerance evaluation of the remaining major structural components of the 757 airplane also showed that small portions of the airplane structure in the highly stressed regions required additional inspections for fatigue cracks. Figure 9 summarizes the requirements of additional inspections for fatigue cracks, for the 757 airplane. For all of the remaining structure, structural safety in the event of fatigue damage is assured by the initial

inspection program developed for the detection of environmental deterioration and accidental damage.

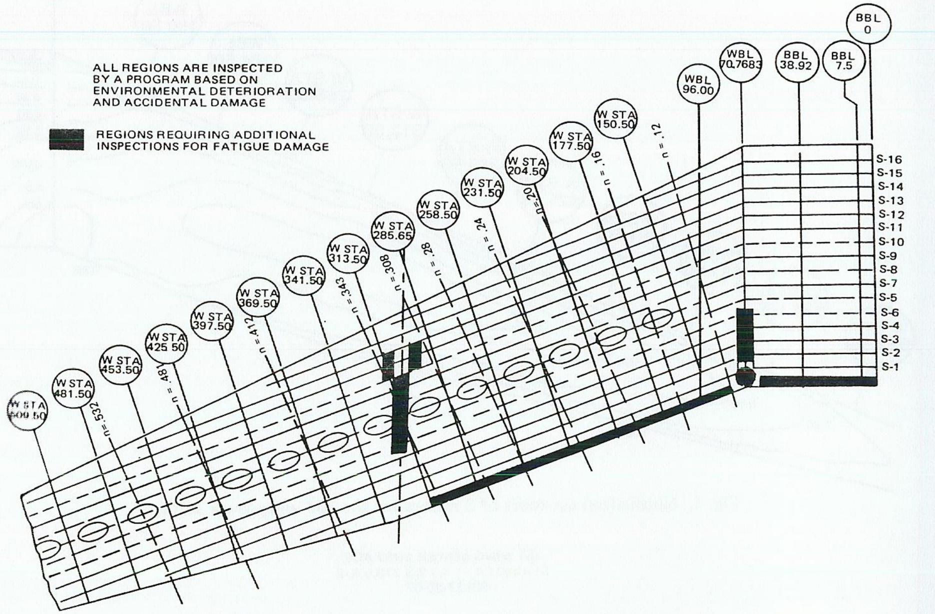


Fig. 8. 757 wing lower surface structural inspection program

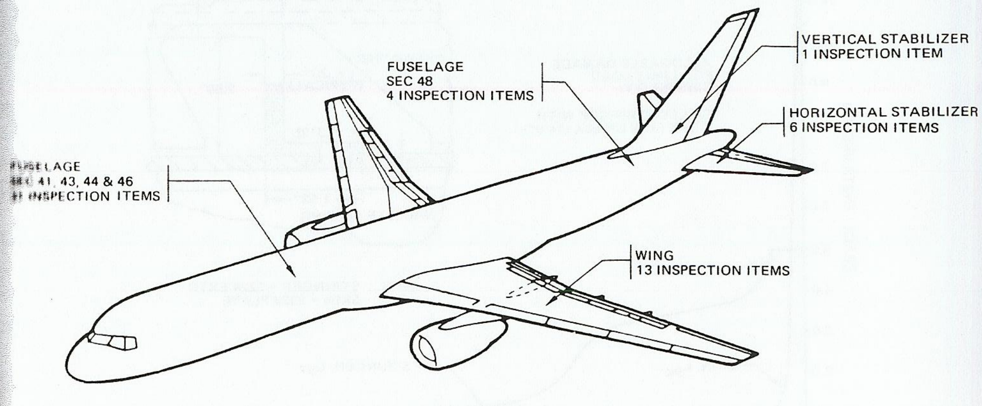


Fig. 9. 757 airplane-Summary of additional inspection requirements for fatigue damage

For locations where the initial inspection program is determined to be inadequate, additional inspections to the program are suggested. Other options may be determined from the data that is provided. In either case, the implementation of the inspection program for fatigue damage detection is required only after a suitable threshold, which will be determined from the experience of similar inspection programs applied to mature Boeing airplane fleets.

CONCLUSIONS

The application of fracture mechanics to the residual strength and damage growth elements of damage tolerance analysis of aircraft structures is challenging. These challenges are principally due to the complex geometry of the redundant load path aircraft structure and the variable amplitude stress spectra that the structure is exposed to under changing environments.

This paper has shown that fracture mechanics methodology can be combined successfully with accumulated service experience to develop rational procedures for assessment of residual strength, crack growth and damage detection for aircraft structure. The results of application of the developed procedures to the damage tolerance analysis of the Boeing 757 airplane showed the overall impact of the new airworthiness regulations on aircraft structural maintenance programs to be minor. It is anticipated that inspection time and cost will not increase substantially as a result of the new regulations. The new technology provides effective means to focus inspection efforts on the more critical structure and offers flexibility to airline operators to integrate the inspection requirements in their regular maintenance schedule with minimal economic impact.

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