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DURABILITY METHODS DEVELOPMENT

Volume I
Phase I Summary

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This technical report has been reviewed and is approved for publication.

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A summary is presented of an analytical methodology developed to meet USAF durability requirements. Details are given in four companion volumes. Analytical approaches that are currently used or have potential to be used to determine economic life are surveyed. Results of a structural survey of durability problems experienced at USAF ALC's are summarized. Durability critical parts criteria and economic life criteria are discussed. An analytical method to determine the crack population in a structure as a function of service time is

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presented. Fastener-hole cracking is the basic form of damage analyzed. Representations of initial quality and growth of small cracks are discussed. The effect of inspection and repair on the distribution of cracks is included.

F O R E W O R D

This program is conducted by General Dynamics, Fort Worth Division with George Washington University, (Dr. J. N. Yang) and Modern Analysis Inc. (Dr. M. Shinozuka) as associate investigators. This program is being conducted in three phases with a toal duration of 50 months.

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This report (Volume I) summarizes the highlights, progress, and accomplishments of Phase I of this program. Details are given in four supporting volumes. They are:

Volume II - Durability Analysis: State-of-the-Art Assessment
Volume III - Structural Durability Survey: State-of-the Art Assessment
Volume IV - Initial Quality Representation
Volume V - Durability Analysis Methodology Development

This report is published only for the exchange and stimulation of ideas. As such, the views expressed herein are not necessarily those of the United States Air Force or Air Force Flight Dynamics Laboratory.

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LIST OF SYMBOLS

a	=	Crack Size
a_e	=	Economical Repair Size Limit
a_f	=	Functional Impairment Size Limit
a_i	=	Initial Flaw Size
a_0	=	Crack Initiation Size
$a(0)$	=	Flaw Size At Time $t = 0$
A/P	=	Airplane
ACI	=	Analytical Condition Inspection
ALC	=	Air Logistics Center
b	=	Constant In Crack Growth Model: $\frac{da(t)}{dt} = Qa^b(t)$
CFA	=	Conventional Fatigue Analysis
c	=	$b - 1$
da/dn	=	Crack Growth Per Cycle
da/dt	=	Crack Growth Per Flight Hour
DCGA	=	Deterministic Crack Growth Approach
EIFS	=	Equivalent-Initial-Flaw-Size
EIQ	=	Equivalent Initial Quality
F	=	Fighter Spectrum
$F_{a(t)}(x)$	=	Distribution Function Of Crack Size $a(t)$ at time t
FCP	=	Fatigue Crack Propagation
$F_n(x)$	=	Empirical Distribution Based on Sample of Size n
$F_T(t)$	=	Time-To-Crack-Initiation Distribution Function

$F_x(x)$	=	EIFS Distribution Function
$F(x)$	=	Distribution Function
$F(x_i)$	=	Cumulative Probability Function
K	=	Equivalent Stress Intensity
K_t	=	Stress Concentration Factor
ΔK	=	Stress Intensity Range
LEFM	=	Linear Elastic Fracture Mechanics
m_i	=	ith Sample Moment About The Mean
m'_i	=	ith Sample Moment About The Origin
N	=	Number Of Observations
n	=	Sample Size
N_f	=	Design Life
n_i	=	ith Observation
$n\omega_n^2$	=	Goodness-Of-Fit Statistic
P	=	Proper Drilling Technique, Load Peak, or Probability
PCGA	=	Probabilistic Crack Growth Approach
PDF	=	Probability Density Function
PCF_B	=	Probability Distribution Of All Cracks Before Repair
PE_i	=	Probability Of Exceeding a_e or a_f at Operating σ
PF_D	=	Probability Of Detection

Q	=	Quackenbush Drill & Ream, or Constant In Crack Growth Model: $\frac{da(t)}{dt} = Qa^b(t)$
R	=	Stress Ratio
S	=	Unbiased Standard Deviation
SE 2	=	Squared Standard Estimate Of Error
S.O.A.	=	State-Of-The-Art
t	=	Time
t_o	=	Initial Time
TTCI	=	Time-to-Crack- Initiation
W	=	Winslow Drilled
X_i	=	ith Sample
X_L	=	Lower Limit
X_m	=	Mean
X_u	=	Upper Limit
$(X_\mu - X_L)$	=	Confidence Interval
$Y_1(t)$	=	Initial Flaw Size Corresponding To Size X at Time T
α	=	Shape Parameter For Weibull Distribution
β	=	Scale Parameter For Weibull Distribution
β_1	=	Standard Measurement Of Skewness
β_2	=	Standard Measurement Of Peakedness
γ	=	Confidence Level
ϵ	=	Location Parameter For Weibull Distribution
$\hat{\epsilon}$	=	Estimate Of Weibull Location Parameter

η	=	Shape Parameter
$\hat{\eta}$	=	Estimate Of Shape Parameter
λ	=	Scale Parameter
$\hat{\lambda}$	=	Estimate Of Scale Parameter
σ	=	Stress or Standard Deviation
σ_a	=	Alternating Stress
μ_i	=	Population Moment About Mean
μ'_i	=	Population Moment About Origin

S U M M A R Y

This report summarizes the essential details, recommendations, and conclusions of the Phase I effort. Details of the Phase I work are given in four supporting volumes [7, 8, 19, 40].

A durability methodology has been developed for implementing the U. S. Air Force's durability requirements for advanced metallic airframes. The methodology can be used during design to assure aircraft durability. Inspection and maintenance procedures can also be analyzed. These, along with maintenance costs, have potential application to fleet management and optimization of life-cycle costs.

For the first time, economic life can be analytically predicted for metallic airframes in physically meaningful terms which relate to life-cycle costs. The methodology includes economic life criteria and durability critical parts criteria. A state-of-the-art advancement has been made in durability analysis capabilities. Economic life design tradeoff options, such as life-cycle costs, weight, design stress levels, operational readiness, maintenance requirements, testing requirements, etc., can now be evaluated before the aircraft is committed to service.

The durability methodology accounts for initial fatigue quality, crack growth accumulation, loading spectra, material/structural properties, usage, etc. Economic life can be analytically predicted for a given detail, for a part, for a component, for an airframe, or for a fleet of aircraft. Two economic life criteria are recommended: (1) probability of crack exceedance and (2) cost ratio: maintenance cost/initial cost. In this report the "probability of crack exceedance" and the "percentage of crack exceedance" refer to the same concept but the results are expressed in different formats (e.g., a probability of crack exceedance of 0.15 is equivalent to a 15% crack exceedance).

The durability analysis methodology is a probabilistic crack growth approach. Durability damage is characterized by crack length and a fastener hole is the prototype used to develop the methodology. Other structural details, such as fillets and cutouts, can also be handled. The methodology also applies to functional impairment assessments of fuel leaks due to through-the-thickness cracks.

The durability methodology has been developed and verified using coupon data from the "Fastener Hole Quality" program [4]. Further verification and possible methodology refinements are required for different materials, spectra stress levels, and fastener load transfer levels. Also, the methodology must be verified for full-scale airframe applications. Further work is required to determine if the EIFS distribution is a generic material property. This will be accomplished during Phase II of the on-going program.

Initial fatigue quality is characterized using two concepts:

(1) equivalent initial flaw size (EIFS) and (2) time-to-crack-initiation (TTCI).

An equivalent initial flaw is a hypothetical crack assumed to exist in the structure prior to service. As such, it characterizes the equivalent effect of the actual initial flaws in a structural detail. EIFS's are not physically observable initial cracks. Therefore, EIFS's must be justified using applicable fractography.

An EIFS cumulative distribution $F_{a(o)}(x)$ is derived using the TTCI distribution, a deterministic crack growth law and a probabilistic crack growth format. The distribution of crack sizes as a function of time, $F_{a(t)}(x)$, is derived from the EIFS distribution using a statistical transformation. The resulting EIFS distribution is statistically compatible with the TTCI distribution and the distribution of crack sizes as a function of time, $F_{a(t)}(x)$. $F_{a(t)}(x)$ correlates very well with ranked observed crack sizes for two different specimen types. Thus, the derived EIFS distribution is indirectly verified using fractography and the distribution is linked to the fatigue wearout process. It is shown that a population of crack sizes can be statistically transformed from one time to another using a single

deterministic crack growth curve.

Several tasks were performed to develop the durability methodology. Phase I included the following tasks:

Task I - Durability State-Of-The-Art Assessment

- o Aircraft Structural Durability Survey
- o Durability Analysis

Task II - Durability Design Handbook Outline

Task III - Durability Analysis Methodology Development

- o Critical Parts Criteria
- o Economic Life Criteria
- o Initial Quality Representation
- o Durability Methods Development

Highlight accomplishments are summarized below:

- o Structural durability surveys of several in-service aircraft showed that cracking was the most frequent structural degradation problem, followed by corrosion and fastener related problems.
- o Conventional fatigue analysis (Palmgren-Miner Rule), the deterministic crack growth approach, and the probabilistic crack growth approach are useful in one way or another for implementing durability requirements at the design level. However, the probabilistic crack growth approach is the most promising for analytically quantifying economic life.
- o A preliminary durability handbook outline was developed.
- o A durability critical parts criterion was developed.

- Two economic life criteria were developed that can be implemented using the durability analysis methodology developed.

- Only four out of twenty-two statistical distributions were found suitable for characterizing EIFS. Of these, the Weibull-Compatible Distribution was found to be the most promising.

- A durability analysis methodology was developed for predicting the economic life of advanced metallic airframes. The methodology can be used to evaluate aircraft durability at the design level. It includes procedures for evaluating inspection and repair requirements, and life-cycle costs.

SECTION I

INTRODUCTION

Durability of aircraft structure is of primary importance, affecting the operational readiness of fleets of aircraft and the overall maintenance and operational cost. Current Air Force structural integrity (MIL-STD-1530A) and durability design specifications (MIL-A-8866B) require that airframe components be designed such that the economic life is in excess of the design service life. Specification MIL-A-8866B further requires that the economic life be analytically predicted. The conventional fatigue analysis, while capable of estimating design life, does not lend itself to predicting economic life, nor is it capable of providing a definition of economic life. The more recent approach is a single-valued deterministic method in which a small (0.005-0.01 inch) crack is assumed to be present in all fastener holes to simulate the possible existence of material and manufacturing defects. The F-16 durability analysis calculations were made by employing this deterministic approach which calculates the damage as a function of design life N_f.

This deterministic approach can provide a single value prediction of the crack growth damage accumulation, but it cannot predict the probable existence of other crack growth damages and this prediction is essential for economic life prediction. The next step in the refinement of the deterministic approach is the development of the probabilistic analysis method, where the distribution of damage or crack growth with time or design life can be calculated. The development of the probabilistic approach has been hampered by the lack of initial quality data associated with the materials, manufacturing processes, and structural usage variations.

However, from the damage tolerance and durability assessment programs of the F/RF-4C/D, F-4E (S), and A-7C aircraft, equivalent initial quality (EIQ) data representing crack growth from fastener holes have been generated {1, 2, 3}. More recently, Air Force Contract F33615-76-C-3113, "Fastener Hole Quality" to General Dynamics, generated a considerable amount of valuable EIQ data for 7475-T7351 aluminum alloy. These data are documented in AFFDL-TR-78-206, Vol. I and II {4}. These data plus the analytical durability methodology development by Shinozuka {5} and Yang [6,45] formed the basis and generated the impetus to develop a probabilistic durability analysis methodology for economic life prediction.

The objective of this program is to develop and verify a durability methodology for analytically quantifying the economic life of advanced metallic aircraft structures at the design level. The methodology will be documented in a handbook to provide guidelines for the design of advanced metallic structures which will have a minimum of structural maintenance inspection and downtime, costly retrofit, repair and replacement of critical structures due to fatigue cracking and/or structural or material degradation.

This program is conducted by General Dynamics, Fort Worth Division with George Washington University (Dr. J. N. Yang) and Modern Analysis Inc. (Dr. M. Shinozuka) as associate investigators. This program is being conducted in three phases with a total duration of 50 months. Both Phases I and II have three tasks each. The duration of each phase and the title and duration of each task are given in Figure 1.

This report (Volume I) summarizes the highlights, progress and accomplishments of Phase I of this program. Four other volumes are being written to describe, in detail, the progress made in Phase I. They are:

Volume II - Durability Analysis: State-of-the-Art Assessment
Volume III - Structural Durability Survey: State-of-the Art Assessment
Volume IV - Initial Quality Representation
Volume V - Durability Analysis Methodology Development

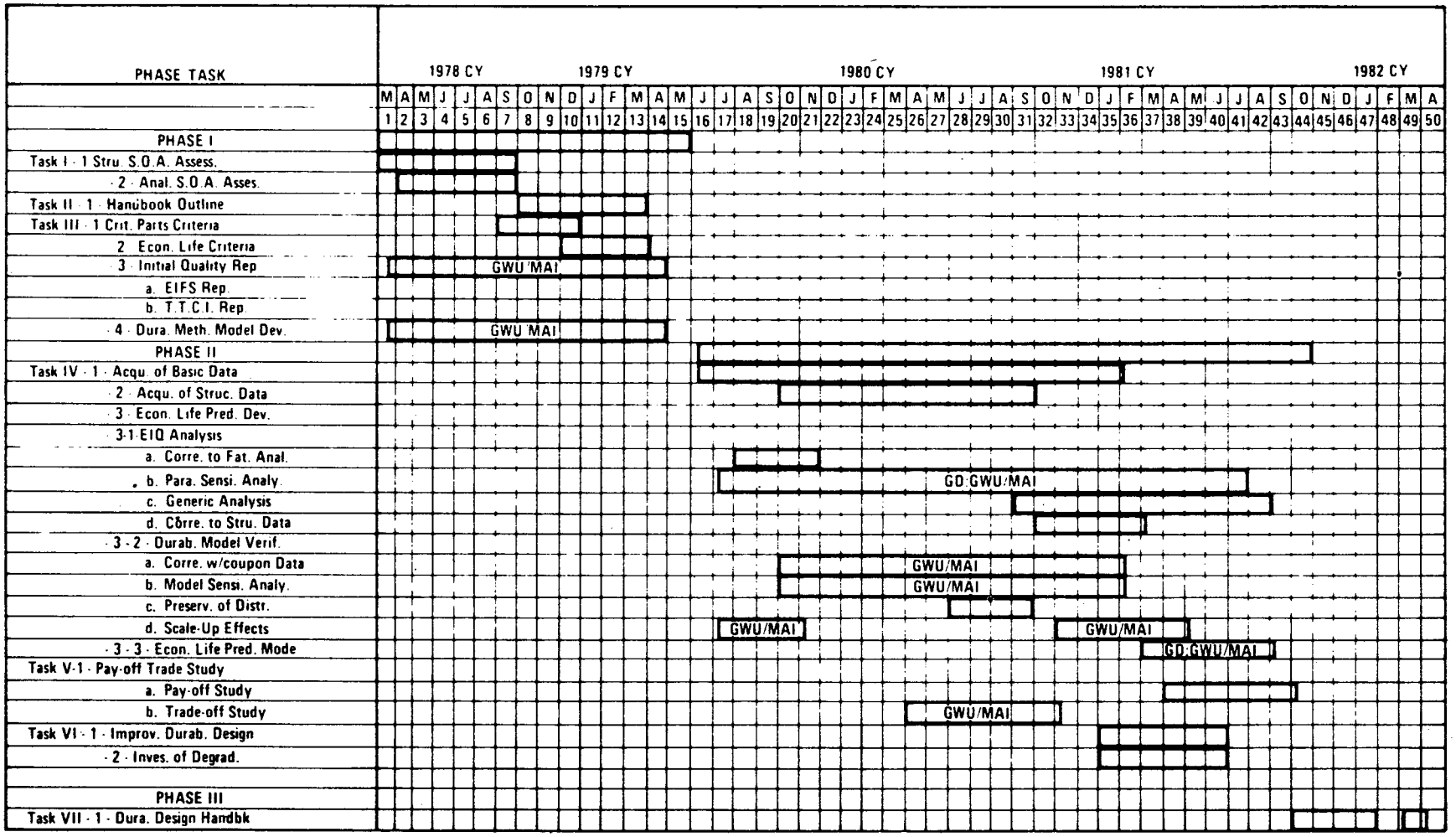


Figure 1 Durability Methods Development Program Tasks And Schedule

SECTION II

STATE-OF-THE-ART SURVEY

2.1 INTRODUCTION

Structural and analytical durability state-of-the-art (S.O.A.) assessments are summarized in this Section. Only essential results and conclusions are presented. Details are given elsewhere {7, 8}. The S.O.A. assessments provide the foundation for developing an overall durability methodology.

2.2 STRUCTURAL STATE-OF-THE-ART ASSESSMENT

The salient results and conclusions of the Structural S.O.A. assessment are summarized in this section.

2.2.1 Objectives

The overall objective of this task was to determine the types of damage occurring in aircraft structures, both previous and current, and to show the relationship between types of damage and the initial quality of the structures. The forms of damage considered were to include such degradations of structure or materials as cracking, corrosion and wear. The relative frequency of occurrence, original design practices and initial structural quality were to be surveyed, as well. Types of aircraft surveyed were to include service aircraft, full scale and component test articles as well as coupon specimens.

2.2.2 Approach

The structural S.O.A. assessment was performed in three basic stages; data acquisition, data analysis and documentation. Data acquisition consisted of an initial survey of "in-house" and open literature. This initial information was useful in determining the form and type of data to be gathered. The following Air Force Air Logistics Centers (ALC) were visited to obtain information, primarily on current systems, at the depot maintenance level:

San Antonio ALC	Kelley AFB, Texas
Oklahoma ALC	Tinker AFB, Oklahoma
Warner-Robins ALC	Robins AFB, Georgia
Sacramento ALC	McClellan AFB, California
Ogden ALC	Hill AFB, Utah

A wealth of information was obtained from the ALC visits. Aircraft Structural Integrity Plans, Analytical Condition Inspection Results, Fatigue Test Teardown Results, etc., and useful verbal information were obtained. Following these ALC visits other data sources, primarily those of other airframe manufacturers, were explored.

Results for the following aircraft systems are presented and evaluated in Reference 7:

Fighter Aircraft

F-100	F-105	F-111
F/RF-100	F-106	F-15
F-104	F-4C/D/E	F-16

Trainer Aircraft

T-37B/C
T-38
T-39

Bomber Aircraft

B-52
FB-111
F-111C

Cargo/Transport

C-130
C-141A
KC-135
C-5A

Attack Aircraft

A-7
A-10

Essential results are summarized in Section 2.2.3.

2.2.3 Results

Following the receipt of all printed matter, from both ALC visits and other airframe manufacturers, the information was compiled by type of aircraft. A majority of the ALC data were obtained verbally. Notes were kept on the verbal information obtained. This source of information was extremely valuable. Data from all sources were surveyed and evaluated. Results are summarized and documented {7}.

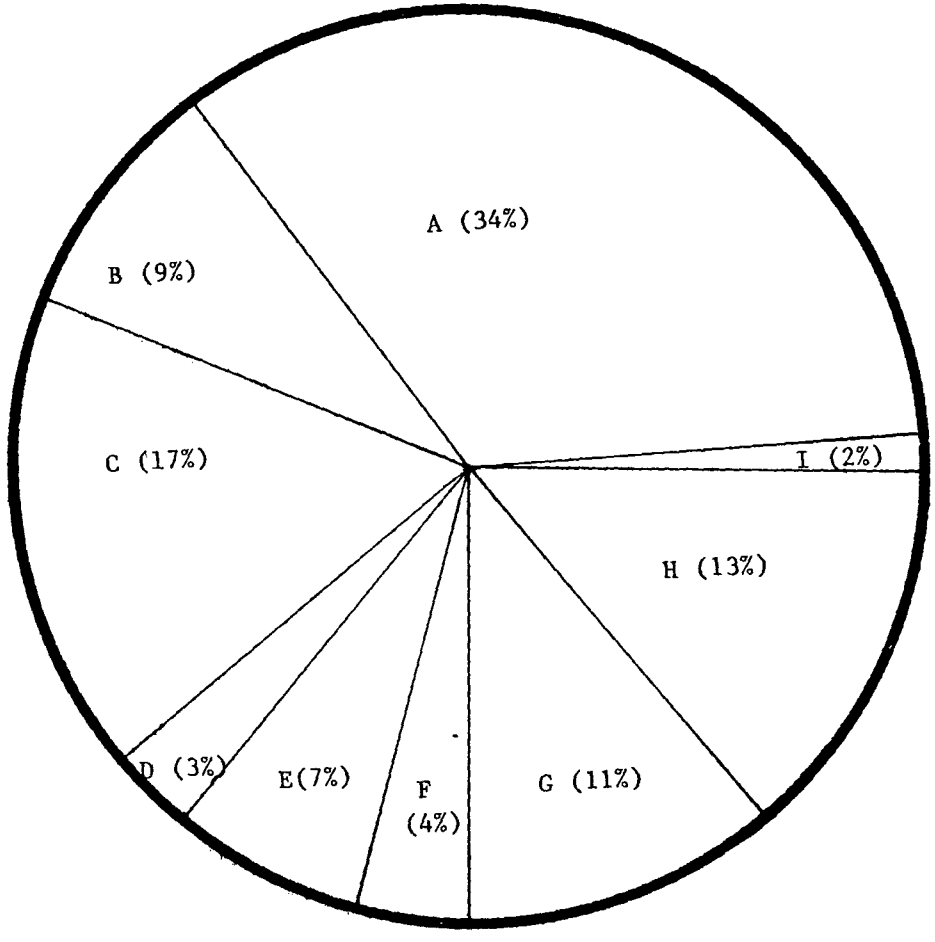
The Analytical Condition Inspection (ACI) reports were a good source for information. Such results were the most quantitative and best suited to satisfy the objectives of this task. ALC data were cataloged into the following incident categories:

A. Cracking: holes, plates, radii, fittings, etc.
B. Corrosion: stress and/or any other
C. Fastener Related: loose, missing, failed, etc.
D. Dents/Nicks/Scratches
E. Honeycomb: delamination/damage
F. Fastener Hole Related: out-of-round, etc.
G. Wear: chaffing, fretting
H. Maintenance: improper or faulty practices
I. Miscellaneous: specific system particularities

These categories were evaluated on a percentage basis for each system of a given aircraft type, as shown for the C-5A and F-15A in Figures 2 and 3, respectively. Percent occurrences for ACI incidents are summarized in Table 1. This table includes several combinations of unrelated categories. The consistency of these combinations clearly shows that the same type structural deficiencies occur at approximately the same rate in one system as another. A "pie" chart of the average percent of occurrence of cumulative incidents is shown in Figure 4. This figure shows the following relative numerical ranking of incidents:

1. Cracking
2. Corrosion
3. Maintenance
4. Fastener Related
5. Dents/Nicks/Scratches
6. Wear
7. Miscellaneous
8. Honeycomb Delamination/Damage
9. Fastener Hole Related

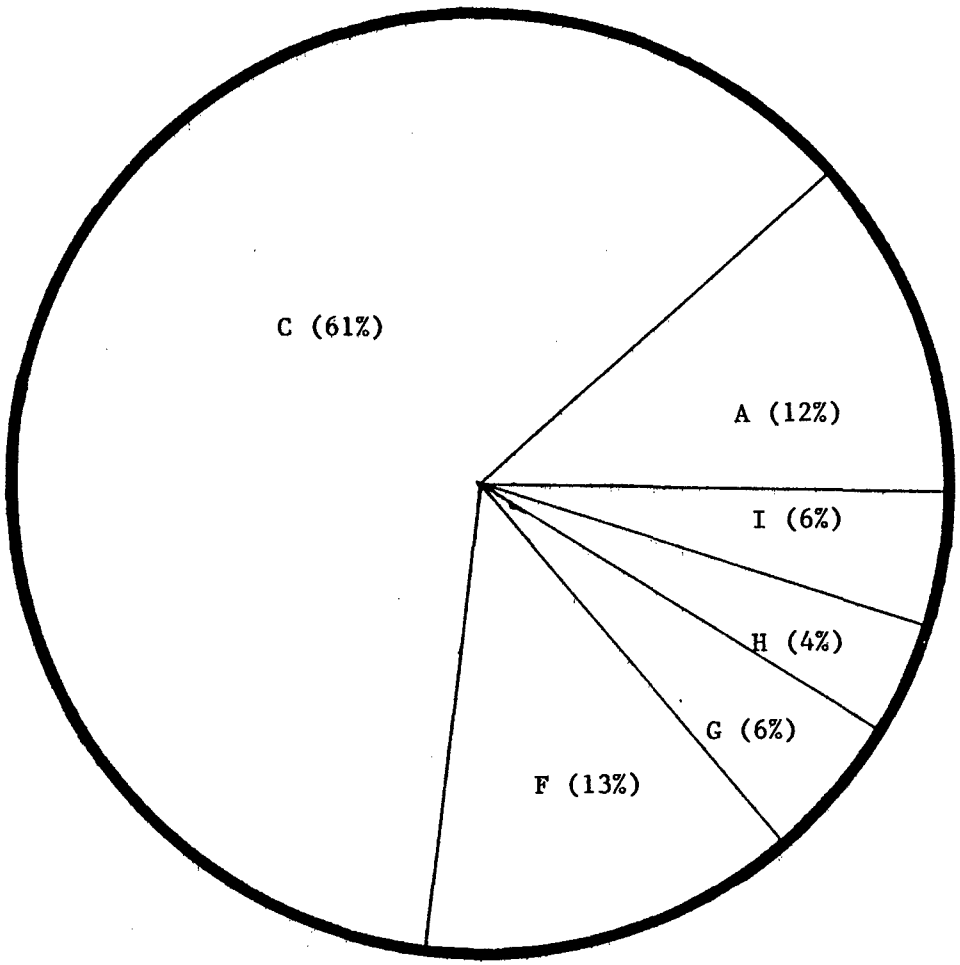
Although the percentages may vary from system to system, the above rankings are considered representative of the average airframe with an equivalent service life.



289 Occurrences

INCIDENT		CUMULATIVE OCCURRENCES
A	Cracking	98
B	Corrosion	25
C	Fastener Related	49
D	Dents/Nicks/Scratches	8
E	Honeycomb Damage/ Delamination	21
F	Fastener Hole Related	12
G	Wear	32
H	Maintenance	38
I	Miscellaneous, Pressure Seals, Etc.	6
	Total	289

Figure 2 C-5A Analytical Condition Inspection Cumulative Occurrences For CY76 And CY77



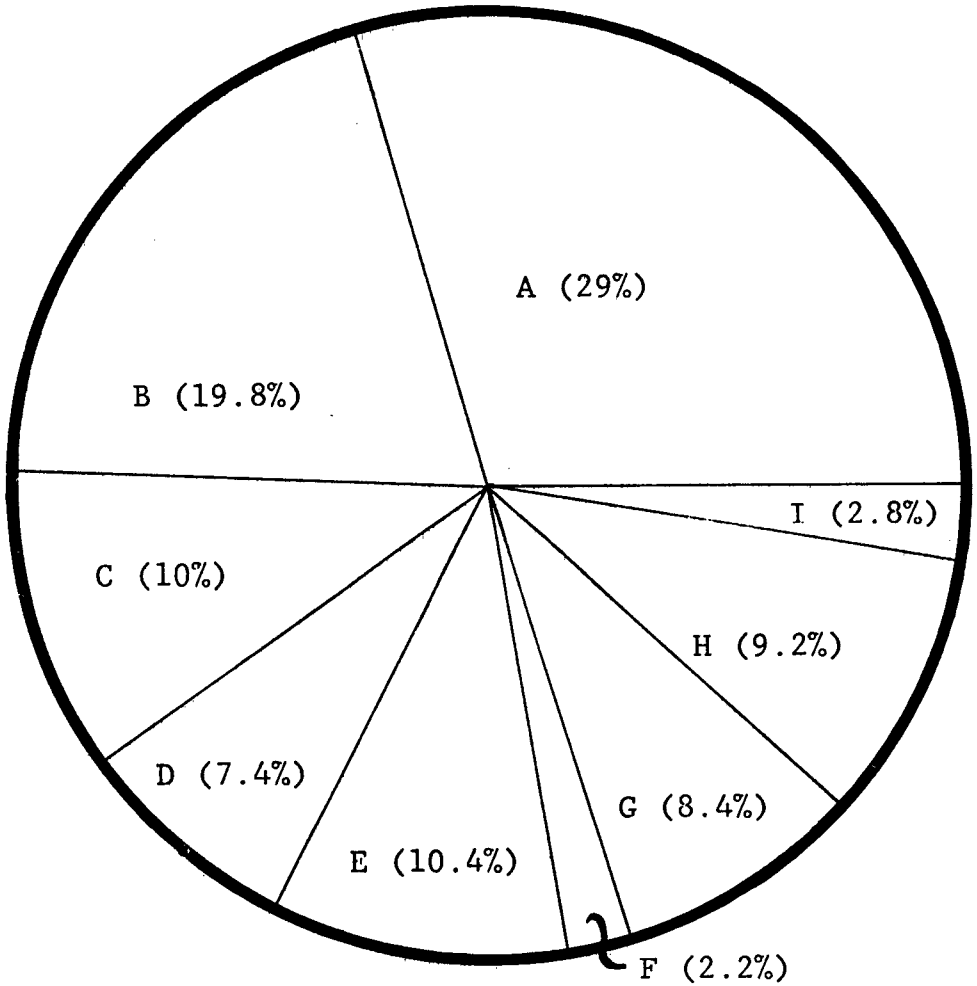
86 Occurrences

INCIDENT		CUMULATIVE OCCURRENCES
A	Cracking	10
B	Corrosion	0
C	Fastener Related	53
D	Dents/Scratches/Nicks	0
E	Honeycomb Damage/ Delamination	0
F	Fastener Hole Related	11
G	Wear	4
H	Maintenance/Manufacturing	3
I	Miscellaneous	5
	Total	86

Figure 3 F-15A Analytical Condition Inspection S/N 73-0085

TABLE 1 Percent Occurrences For ACI Recorded Incidents

INCIDENT	ACI % OCCURRENCES					
	T-39A	F-4	F/FB-111A	F-111C	C-5A	AVE
A. Cracking	44	28	17	22	34	29
B. Corrosion	13	23	28	26	9	19.8
C. Fastener Related	10	11	5	8	17	10
D. Dents/Nicks Scratches	11	0	16	7	3	7.4
E. Honeycomb Damage	1.2	30	12	2	7	10.4
F. Fastener Hole Related	1.2	3	3	0	4	2.2
G. Wear	9	4	14	4	11	8.4
H. Maintenance	11	0	0	22	13	9.2
I. Miscellaneous	0	0	3	9	2	2.8
A + B	57	51	45	48	43	48.8
C + D + F	22.2	14	24	15	24	19.8
C + D + F + I	22.2	14	27	24	26	22.6
D + H + F	23.2	3	19	29	20	18.8



INCIDENT		% CUMULATIVE OCCURRENCE
A.	Cracking	29
B.	Corrosion	19.8
C.	Fastener Related	10
D.	Dents/Nicks/ Scratches	7.4
E.	Honeycomb Damage	10.4
F.	Fastener Hole Related	2.2
G.	Wear	8.4
H.	Maintenance	9.2
I.	Miscellaneous	2.8

Figure 4 Average Cumulative Distribution By Percentage of Incidents From ACI For T-39A, F-4C/D/E , RF-4C, F-111, F-111C, And C-5A Systems

2.2.4 Conclusions/Observations

The following conclusions and observations are based primarily on the Analytical Condition Inspection:

1. For this assessment the Analytical Condition Inspection was the best source of data to rate structural durability.

2. The same type of structural problems tends to occur from system to system.

3. As the rate of corrosion occurrence decreases with improved material technology, other material-related problems occur, yielding a relatively constant percentage of "structural occurrences."

4. The F-15A fighter exhibits anomalous behavior compared to other aircraft systems, possibly establishing a trend of increased structural durability.

5. Improved methods are needed for tracking structural incidents at the maintenance depot level to allow a more accurate evaluation of structural performance.

6. Maintenance, the third most frequently occurring incident, should be given priority in that this area not only represents an incident category but also specifies the exact cause.

7. A voluminous data bank for durability was collected during the survey for most of the aircraft in the U.S. Air Force active inventory. Detailed assessment of these data is beyond the task, funds and schedule for this report. These data are on file at this contractor's facility.

2.3 ANALYTICAL STATE-OF-THE-ART ASSESSMENT

Essential results and conclusions from the durability analysis S.O.A. assessment report {8} are briefly summarized in this section.

2.3.1 Objectives

Two basic objectives of this task were: First, evaluate the applicability and potential of three different analytical approaches for implementing the U. S. Air Force's durability requirements for

advanced metallic airframes at the design level and second, evaluate the applicability of three approaches for analytically predicting the economic life of a metallic airframe prior to service.

2.3.2 Scope and Limitations

The scope and limitations of the original analytical S.O.A. assessment study (Ref. 8) are described in this section.

The following durability issues were reviewed and discussed:

1. Durability objectives
2. U. S. Air Force's durability requirements
3. Durability damage modes
4. Initial quality representation

   - o Equivalent Initial Flaw Size (EIFS)
   - o Time-to-Crack-Initiation (TTCI)

5. Economic Life

   - o Predictions
   - o Criteria

6. Critical parts criteria
7. Durability analysis requirements

A clear understanding of these issues is necessary to develop a proper perspective for the analytical S.O.A. assessment.

Durability analysis needs were assessed in terms of the U. S. Air Force's durability requirements at five levels:

1. Preliminary Design } Pre-Production Design
2. Interim Production Design
3. Durability Design Verification/Evaluation
4. Final Production Design
5. Force Management

An extensive literature survey was performed to determine the applicability of existing analytical methods for implementing durability requirements. Durability analysis methods were cataloged into three groups for the S.O.A. assessment:

1. Conventional Fatigue Analysis (Palmgren-Miner Rule) (CFA) {9, 10}
2. Deterministic Crack Growth Approach (DCGA) {11-14}

3. Probabilistic Crack Growth Approach (PCGA) [15-19]

Methods were loosely grouped according to the basic philosophy reflected. Cumulative damage oriented methods were grouped under the CFA approach, such as linear cumulative damage {9, 10}, nonlinear cumulative damage {20, 21} and local strain methods {22-25}.

Both the DCGA and the PCGA were considered as crack growth oriented approaches. The DCGA is concerned with the crack growth performance of a single crack. Whereas, the PCGA is concerned with the performance of a distribution of crack sizes as a function of time.

Several methods were grouped under the CFA approach. However, only the Palmgren-Miner Rule {9, 10} was considered in detail for the CFA S.O.A. assessment. Also, the Palmgren-Miner Rule was considered in the classical sense; i.e., fatigue failure results (part breaks) when $\Sigma n/N = 1.0$ without regard to crack size or crack growth.

Some analytical concepts may apply to one or more of the three approaches. For example, statistical and probabilistic concepts can be applied to any of the three approaches.

The durability analysis assessment was limited to durability design applications with particular emphasis on economic life (crack exceedance) prediction at the pre-production design level. There are several modes of durability damage, such as, fastener hole cracking, cracking in radii, cracking in cutouts and other stress risers, etc. Only one durability damage mode was considered in the durability analysis assessment--fastener hole cracking. This is one of the most frequent modes of durability damage.

The durability analysis S.O.A. assessment was performed as follows: Each of the three basic approaches was briefly described (what it is, how it works and what it will do to support durability analysis needs). The assumptions, limitations, advantages and disadvantages were described and discussed in terms of durability requirements and analytical needs. A numerical example was presented to illustrate the application of each approach. No attempt was made to completely document each approach. References were cited where further details are given.

Promising economic life concepts were reviewed to determine the best format for making analytical predictions. Analytical approaches (CFA, DCGA and PCGA) were conceptually compared in terms of economic life formats.

Durability design guidelines and recommendations will be presented in the final report for this program and in the durability design

handbook to be prepared. The analytical S.O.A. assessment is an essential step for developing an effective and comprehensive durability methodology for metallic airframe applications.

2.3.3 Results

Durability analysis requirements for five different levels are briefly summarized in Table 2 and details are given in Reference 8. A clear understanding of durability analysis needs is essential to evaluate the applicability and effectiveness of the three durability analysis approaches.

Key elements of the three analytical approaches are shown in the figures noted: CFA (Figure 5), DCGA (Figure 6) and PCGA (Figure 7). CFA, DCGA, and PCGA are philosophically compared in Table 3 . Details are given in Reference 8.

The three durability analysis approaches are conceptually compared in Figure 10 in a crack growth format. This format was used to judge the applicability and potential of the three approaches for economic life predictions. Economic life is characterized by the number of cracks exceeding a limiting crack size requiring repair.

2.3.4 Conclusions

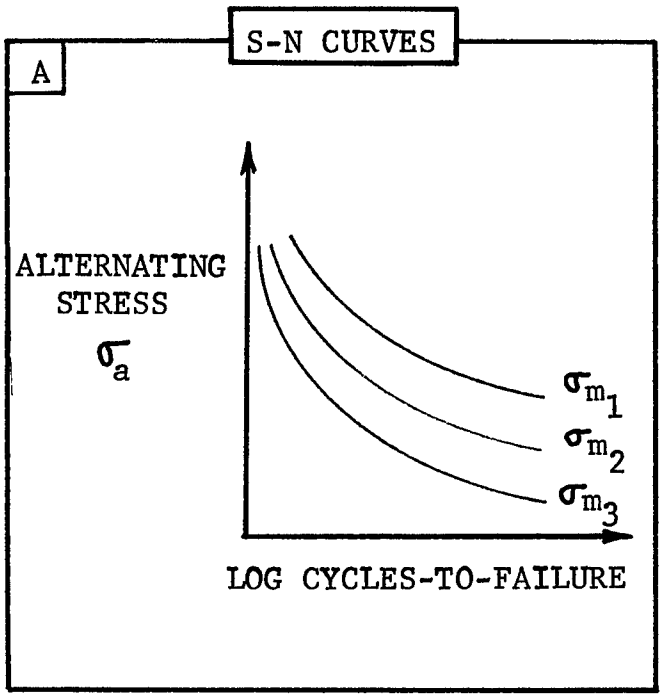
Essential conclusions from the durability analysis S.O.A. assessment are summarized below and in Table 4 . Details are given in Reference 8.

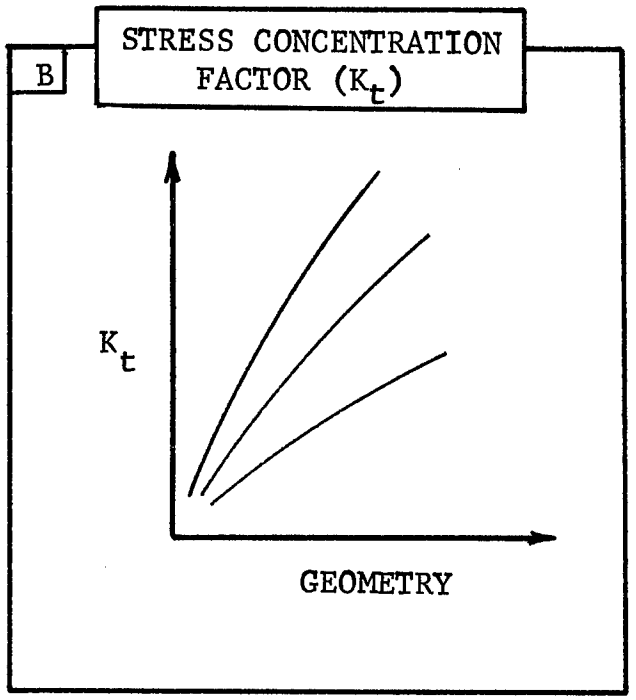
1. A crack exceedance format should be used to quantify the economic life of airframe parts,components, and assemblies based on fastener hole cracks. Crack exceedance refers to the percent of cracks $\geq$ the economical repair limit size (for example, Figure 11.) The crack exceedance concept provides a means for quantifying economic life in terms of maintenance cost and risk.

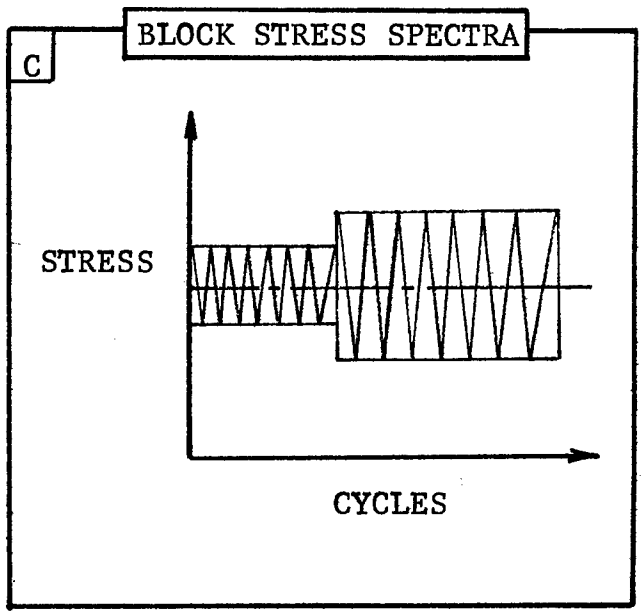
2. Of the three approaches considered, only the PCGA is capable of <u>directly</u> predicting "economic life" in terms of fastener hole crack exceedance and maintenance cost. The PCGA accounts for both the initial fatigue quality variability of the details (e.g., fastener holes) and the entire crack population as a function of time. Using the probabilistic approach, the average and possible extremes for the economic life prediction (e.g., number of cracks $\geq$ a specified size or repair cost) can be assessed. This provides an effective basis for evaluating durability design tradeoffs.

TABLE 2 DURABILITY ANALYSIS REQUIREMENTS

PRELIMINARY DESIGN	INTERIM PRODUCTION DESIGN	DURABILITY DESIGN VERIFICATION/ EVALUATION	FINAL PRODUCTION DESIGN	FORCE MANAGEMENT
o Apply durability data from various sources to particular application o Screen materials o Screen design concepts o Screen fastener systems o Screen manufacturing processes o Set preliminary design allowable stress levels o Define development test requirements o Set guidelines for applying data from one spectrum or environment to another o Identify fatigue "hot-spots"	o Set final stress allowable o Predict economic life o Evaluate design tradeoff options -structural performance -cost -weight -risk -design allowable o Evaluate test data o Define "hot-spots" for durability test article	o Evaluate "hot-spots" disclosed by the verification test -design changes required? -probable state of unrepaired flaw population for projected service usage -performance of structural repairs -further durability testing requirements (inspection, monitoring activities, etc.) o Evaluate economic life tradeoff options for redesign at "hot-spots"	o Procedures are needed to evaluate the durability performance of the final design using latest loads/environment spectra o Trouble spots and the economic life need to be reassessed in terms of new information	o Extend crack growth analyses to fleet o Evaluate effects of usage variations on durability o Risk assessments o Economic life limits for individual fleet aircraft o Assess durability status of repaired aircraft o Provide logistics support o Evaluate structural maintenance requirements and procedures o Structural repair/ rework requirements o Assess overall fleet management requirements







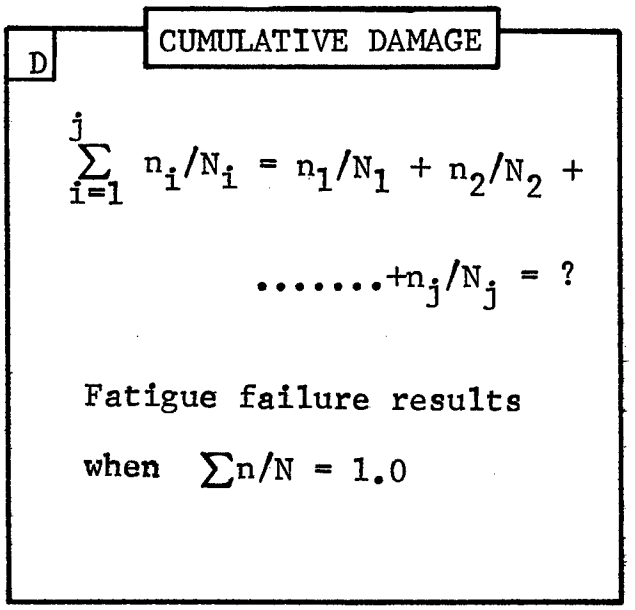


Figure 5 Key Elements of CFA Approach (Palmgren-Miner Rule)

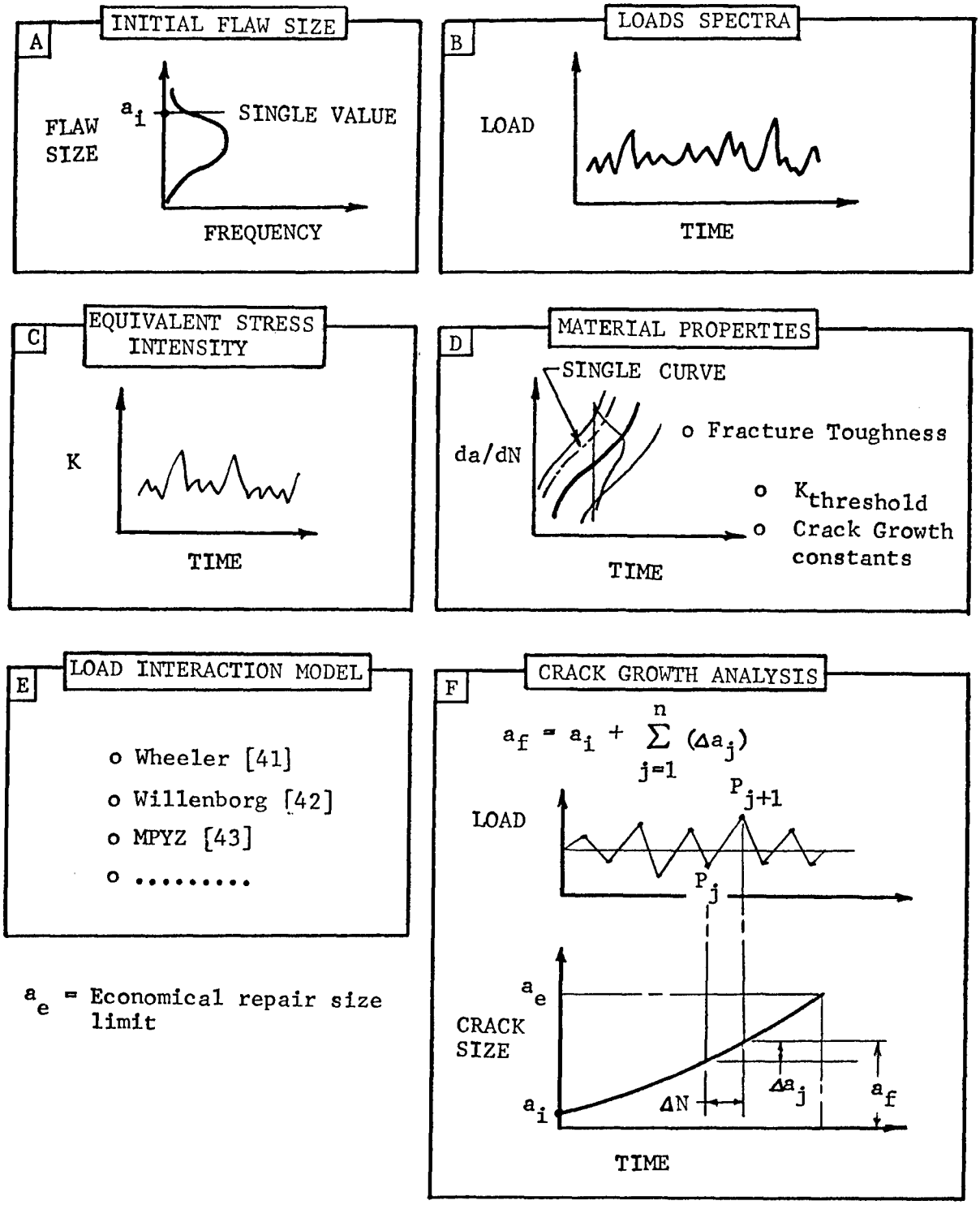


Figure 6 Key Elements of the DCGA

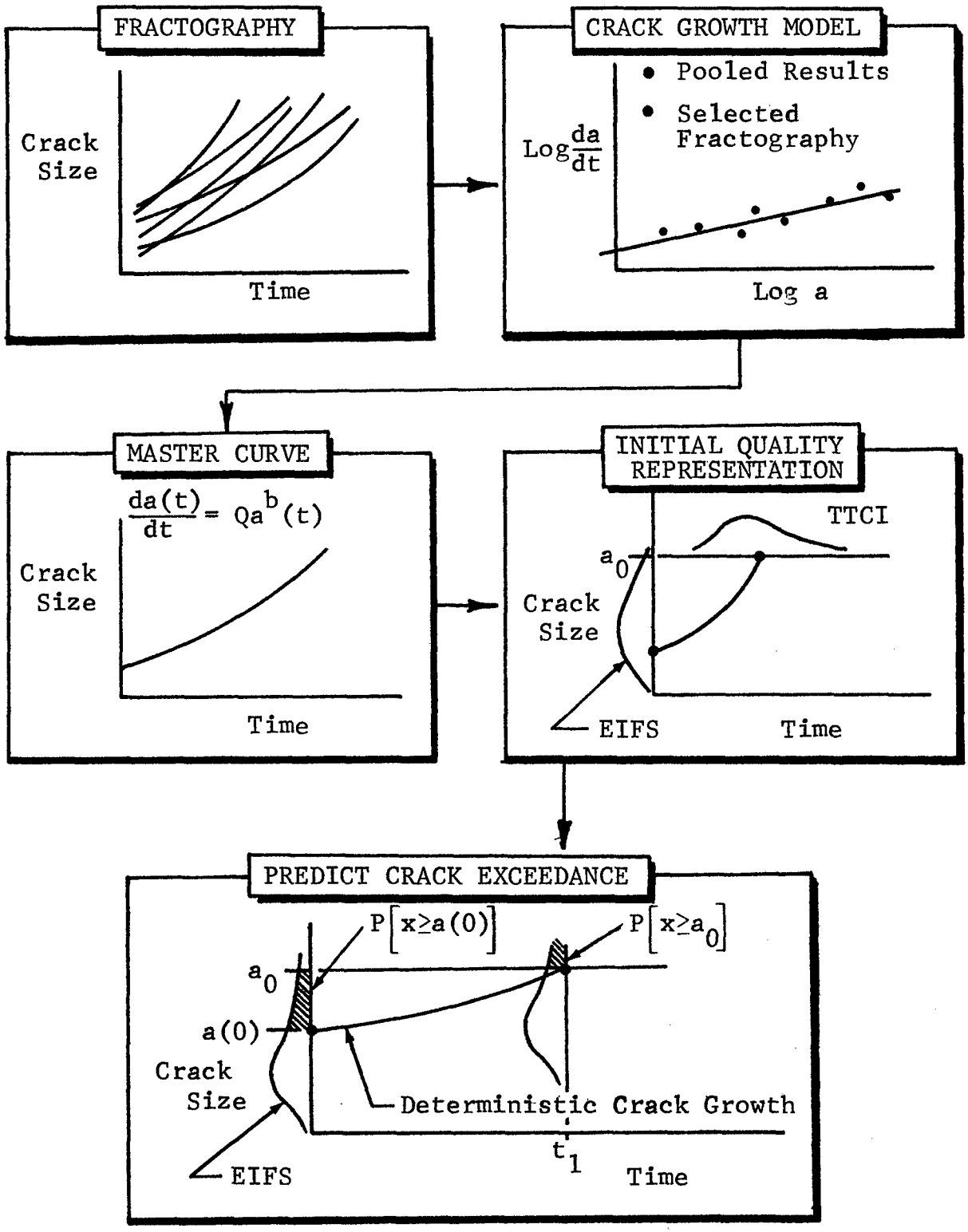


Figure 7 Key Elements of PCGA For Evaluating Economic Life

Table 3 Philosophical Comparisons of Analytical Approaches

ISSUE \ APPROACH	CONVENTIONAL FATIGUE ANALYSIS (PALMGREN-MINER RULE)	DETERMINISTIC CRACK GROWTH APPROACH	PROBABILISTIC CRACK GROWTH APPROACH
Normally Used Load History Simulation	Random or Ordered Block Spectrum	Random Flight-by-Flight	Random Flight-by-Flight
Load Interaction Effects on Life?	No	Yes	Yes
Fracture Mechanics Oriented?	No	Yes	Yes
Single Value Prediction	Yes	Yes	No
Initial Quality Flaw Size?	No	Yes	Yes
Crack Size Vs Time?	No	Yes	Yes
Distribution of Crack Sizes As Function of Time?	No	No	Yes
Economic Life Prediction? [1]	No	Yes	Yes[3]
Economic Life Prediction? [2]	No	No	Yes
Reliability Framework?	No	No	Yes
Type Test Data for Damage Integration	Constant Amplitude: S-N	Crack Growth	Crack Growth

[1]Like Fig. 8 [2]Like Fig. 9 [3]Use selected crack growth percentile curve for prediction.

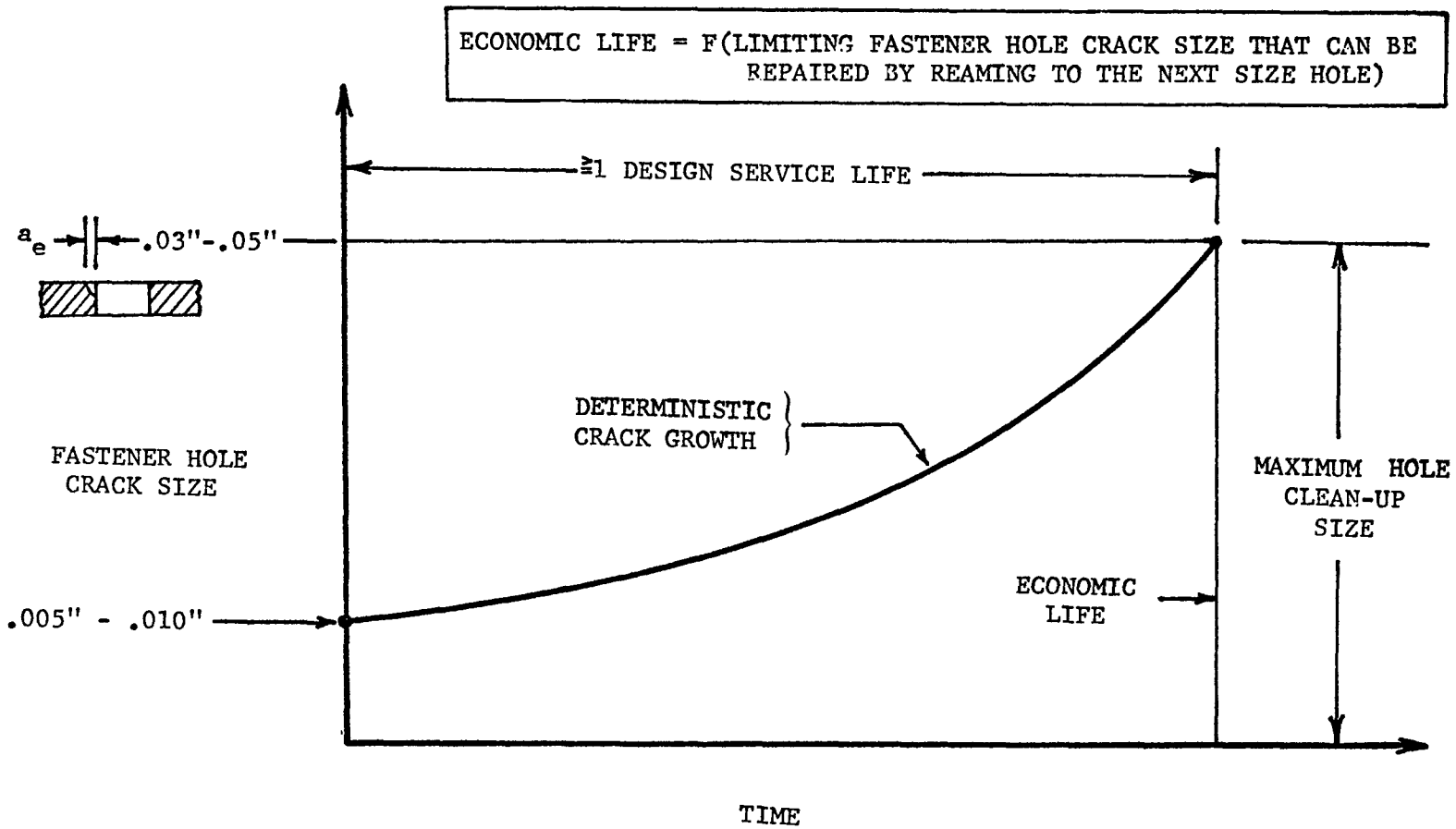


Figure 8 Economic Life Concept Based on Maximum Fastener Hole Clean Up Size

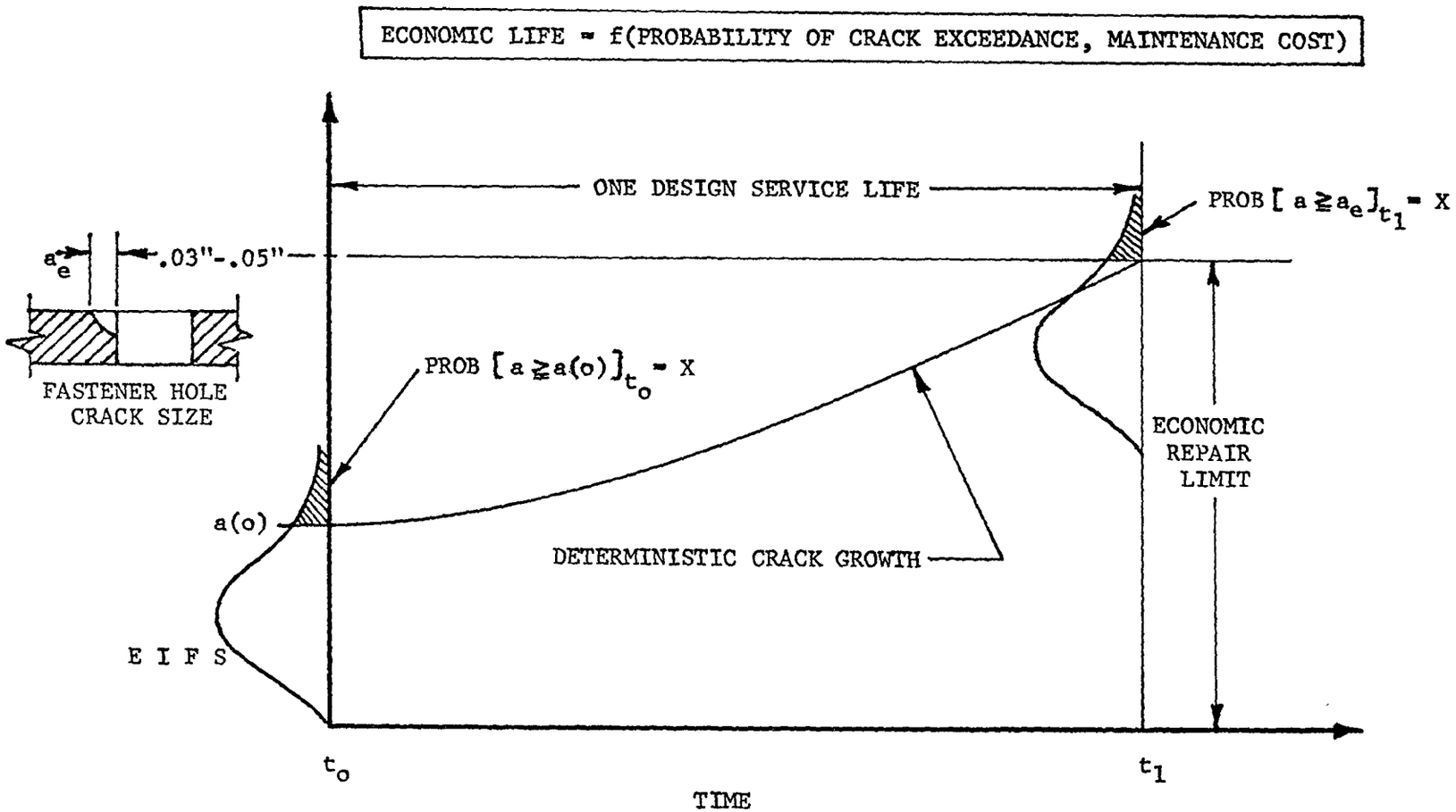


Figure 9 Economic Life Concept Based on Crack Exceedances and Maintenance Cost

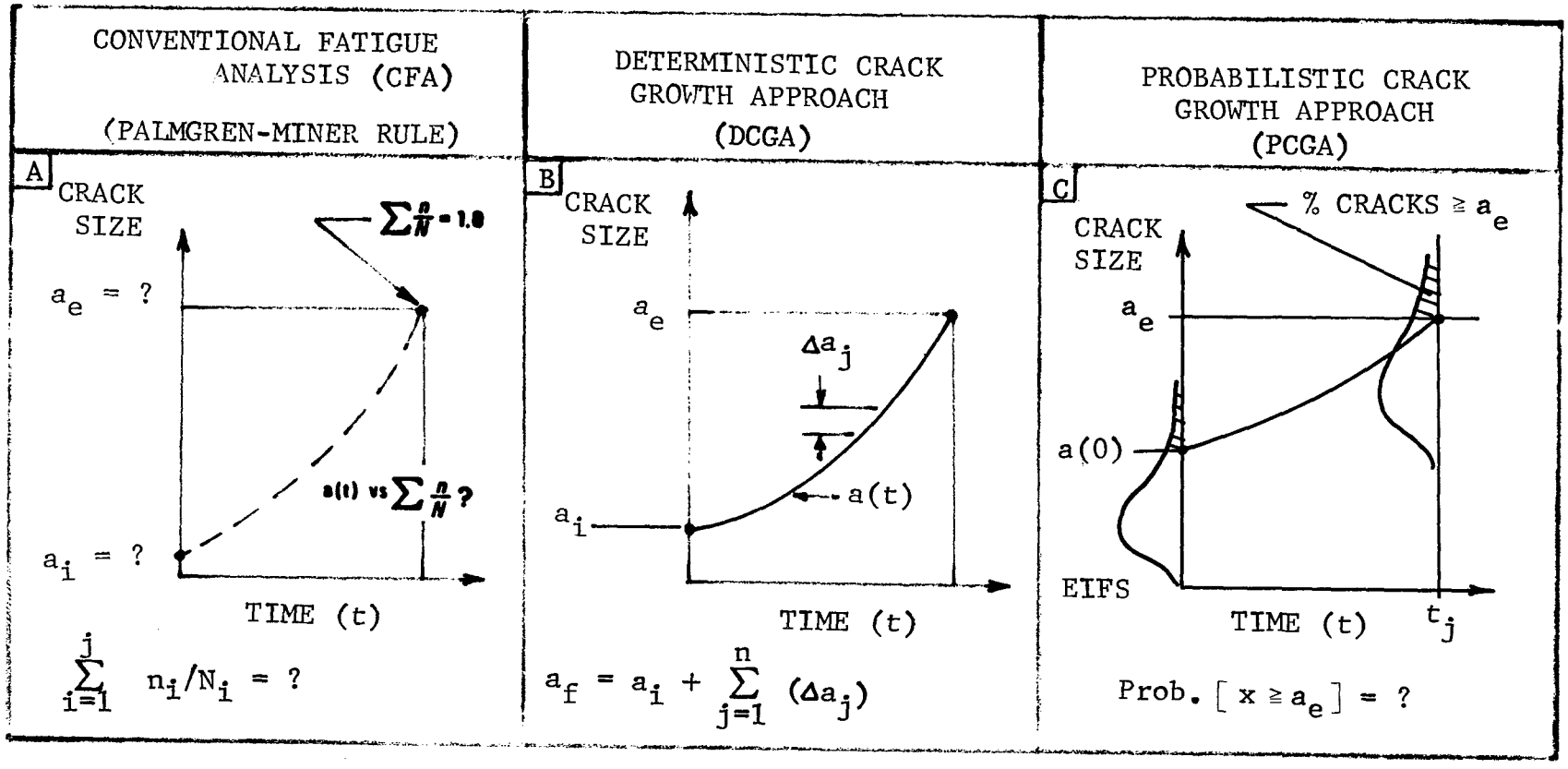


Figure 10 Conceptual Comparison of Durability Analysis Approaches From Crack Growth Viewpoint

TABLE 4 DURABILITY ANALYSIS S.O.A. ASSESSMENT-CONCLUSIONS

CONVENTIONAL FATIGUE ANALYSIS (CFA) (PALMGREN-MINER RULE)	DETERMINISTIC CRACK GROWTH APPROACH (DCGA)	PROBABILISTIC CRACK GROWTH APPROACH (PCGA)
1. Does not recognize initial quality flaw sizes 2. Not fracture mechanics oriented 3. Can estimate fatigue life but incapable of quantifying economic life 4. Uncertainties of the results may be no worse than the uncertainties of the design loads/environment at the preliminary design level 5. Use for preliminary design - Screen materials and design concepts - Set preliminary design allowables - Evaluate fatigue "hot-spots" - Study design trade offs	1. Fracture mechanics oriented 2. Can account for initial quality flaw sizes 3. Effective for predicting the performance of a single crack at a given location, geometry, material, design concept, and maximum stress level. Must repeat analysis for each new set of variables 4. No distribution of cracks as function of time without Monte Carlo simulations, etc. 5. Method applies if economic life is defined in terms of a limiting fastener hole crack size that can be cleaned up by reaming hole to the next size (Figure 11) 6. Not directly applicable if economic life is defined in terms of crack exceedance and maintenance cost 7. Useful for different durability analysis requirements	1. Fracture mechanics oriented 2. Accounts for initial quality flaw size distribution 3. Capable of quantifying economic life in terms of crack exceedance and maintenance cost. Considers distribution of crack sizes as a function of time 4. Provides format for estimating level of confidence in the prediction 5. Promising for various durability analysis levels 6. Need to unify methodology and gain applications experience 7. Need applicable data base to calibrate durability model

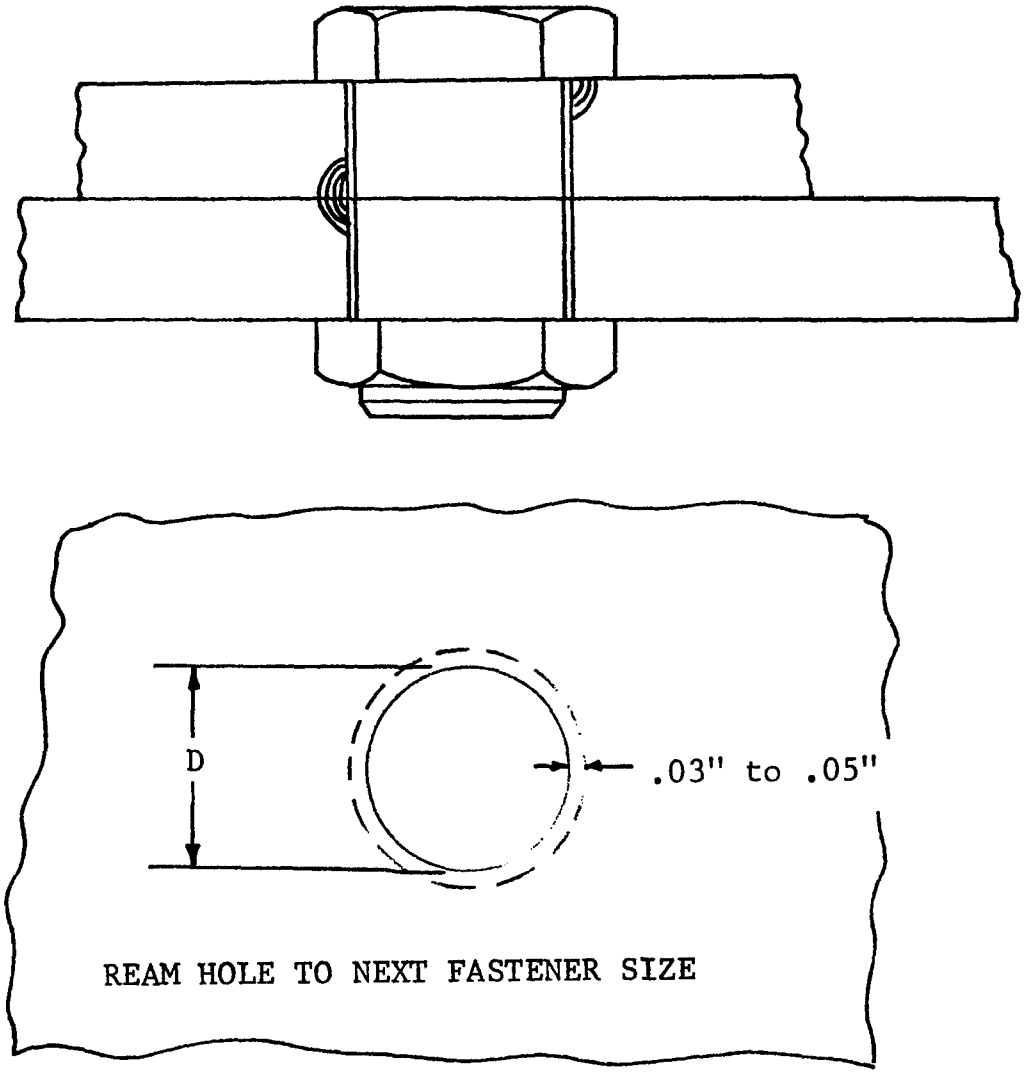


Figure 11 Economic Repair Limit

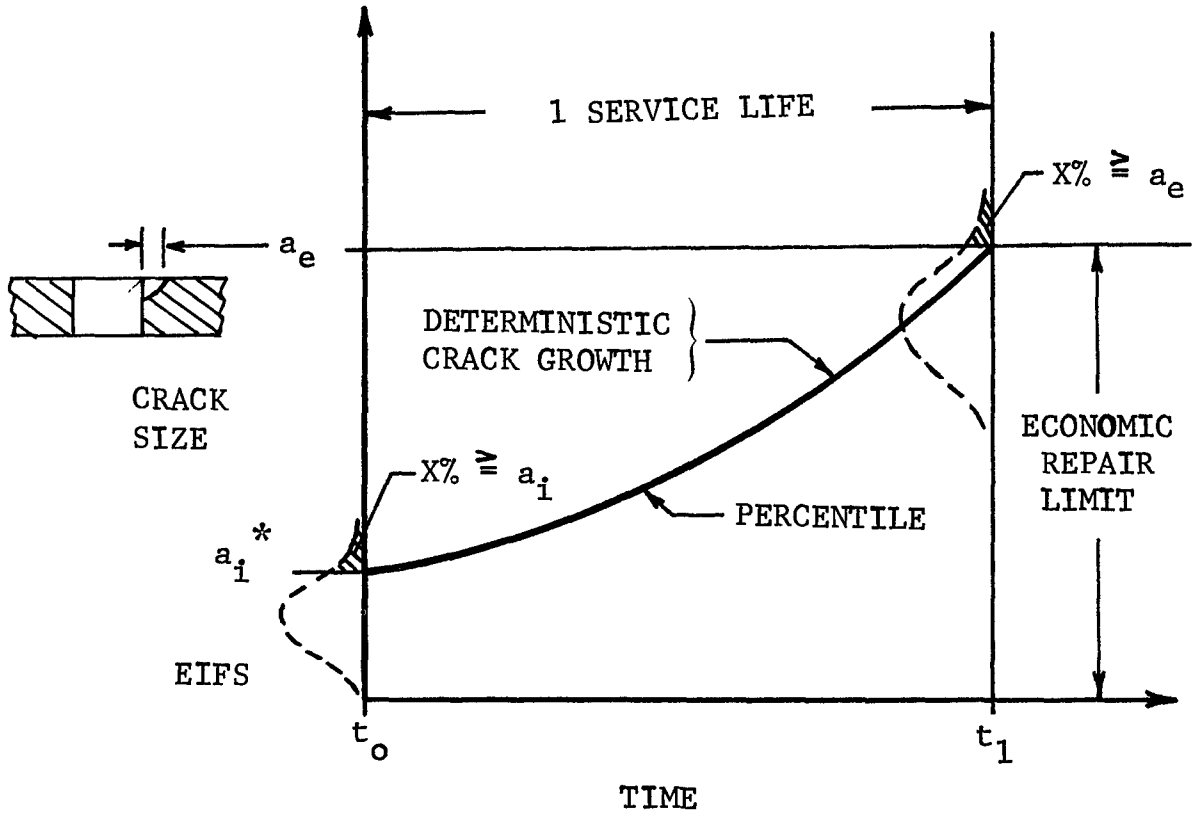
3. The PCGA uses a deterministic crack growth law to grow the EIFS population of cracks to some time, t. However, the results are in a probabilistic format.

4. Once the EIFS distribution has been defined, the difference between the DCGA and the PCGA is subtle. For example, the DCGA is performed using an assumed initial flaw size (a_i). The assumed a_i and resulting crack growth curve can be defined as a percentile of the EIFS population (Figure 12). This provides the same type of information as the PCGA.

5. The performance of a single crack at a given location and geometry and maximum stress level can be predicted using the DCGA (Figure 10 , Frame B). Economic life can be assessed using the DCGA by grouping details (e.g., fastener holes) into areas with similar stress levels and stress histories. An initial fatigue quality flaw size is assumed to exist in the most critical fastener hole in the most adverse position. The economic life is reached when the "worst-case" detail within the group reaches a limiting crack size at the end of a specified time (Fig. 13). This approach is generally conservative for durability analysis because "worst-case" values of the applicable variables are normally used to obtain a single value prediction. The variability of the results (e.g., number of cracks $\geq$ a specified size) can be studied using Monte Carlo techniques. However, Monte Carlo techniques require prohibitive amounts of computer time and they are inconvenient to implement at the design level, where efficient methods are essential for screening and evaluating design tradeoffs.

6. The DCGA is suitable for safety and reliability analysis, where a single conservative estimate of crack length is adequate. However, the DCGA is cumbersome for economic life analysis because the entire crack population must be considered rather than a single extremal crack size. Despite its shortcomings for economic life analysis, the DCGA is useful for screening materials for defining potential fatigue "hot spots" and for evaluating competing designs for durability requirements.

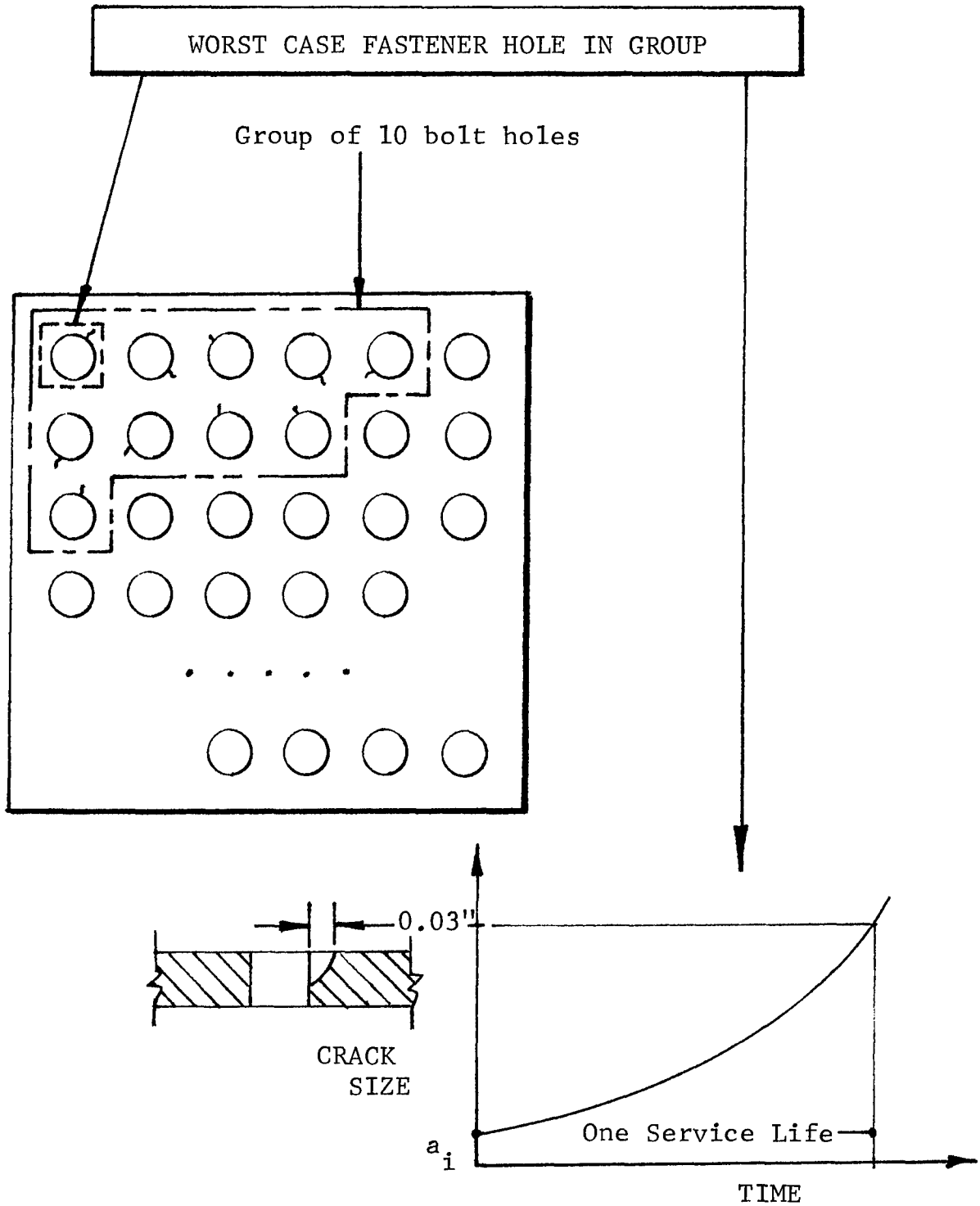
7. All three durability analysis approaches (CFA, DCGA and PCGA) are all useful for implementing durability requirements at the design level. Different analytical tools are needed to effectively implement the Air Force's durability requirements at different design levels. No single durability analysis approach has yet been developed and proven "best" for all situations.



*REQUIRED FOR DCGA

NOTE: X% OF THE CRACK POPULATION AT TIME t_o AND t_1 EQUAL OR EXCEED a_i and a_e RESPECTIVELY

Figure 12 Interpretation of DCGA Results When EIFS Is Specified



Economic Life Statement: One out of 10 holes within the group will reach an 0.03" hole crack at one service life.

Figure 13 Use of DCGA For Economic Life Prediction

8. Conventional Fatigue Analysis (CFA) does not recognize initial quality flaws and the analysis is not fracture mechanics oriented. CFA does not quantify economic life nor does it provide results for accurately making such judgments. It does not account for load sequence effects on fatigue life. Yet, it is still useful for implementing durability requirements at the preliminary design level. For example, it can be used for screening materials and design concepts, for setting preliminary design allowables, for evaluating potential fatigue "hot-spots", for making design trade-off studies, and for evaluating effects of load spectra variations. This method is simple and easy to use. The uncertainties of the results may be no worse than the uncertainties of the design loads at the preliminary design stage.

9. A complete durability methodology for analysis and design should be developed. The methodology should incorporate the effective features of existing approaches which are useful for implementing durability requirements. Also, it should include a "shopping list" of suitable analytical tools for implementing durability requirements at different levels and should include disciplined design procedures and guidelines. Such methodology will be developed under this program and will be presented in a final report and in a durability design handbook.

10. Analytical uncertainties associated with durability analysis predictions need to be considered. Several design variables affect the accuracy of an analytical prediction. Statistical and/or probabilistic principles should be used where feasible to assess the accuracy of a prediction.

11. The analytical tools for implementing durability requirements should be "design oriented". They should be practical. The level of sophistication should be compatible with accuracy requirements and the degree of uncertainty of the design variables. The designer should be free to select the methods for implementing durability requirements that best fit his needs, facilities, capabilities and personnel.

12. Fatigue and crack growth processes are very complex. These processes must be better understood in terms of the durability design variables. Until this understanding has been reached, components and structure will still have to be tested using realistic load spectra.

SECTION III

DURABILITY REQUIREMENTS AND CRITERIA

3.1 INTRODUCTION

The U. S. Air Force's durability requirements for metallic airframes and key durability issues are briefly reviewed and discussed in this section. Durability requirements and issues need to be properly understood so that they can be accounted for in the durability analysis methodology (Section V).

3.2 DURABILITY REQUIREMENTS

The U. S. Air Force's durability and structural integrity requirements are given in three specifications[26-28.] The overall objective of these requirements is to achieve a durable airframe design that will perform effectively under service conditions without excessive structural maintenance and operating costs. Structural degradation in service affects structural maintenance, operational readiness, operational risks and operational costs. Disciplined design, analysis, inspection and manufacturing procedures are essential to obtain a structural design that will resist excessive cracking and/or other structural degradation which would result in excessive maintenance or functional impairment.

The U. S. Air Force has two basic durability requirements for metallic airframes (Figure 14). First, the economic life of the airframe must exceed one design service life under expected service conditions. Second, no functional impairment such as: loss of stiffness, loss of control, loss of cabin pressure or fuel leaks, shall occur in less than one design service life.

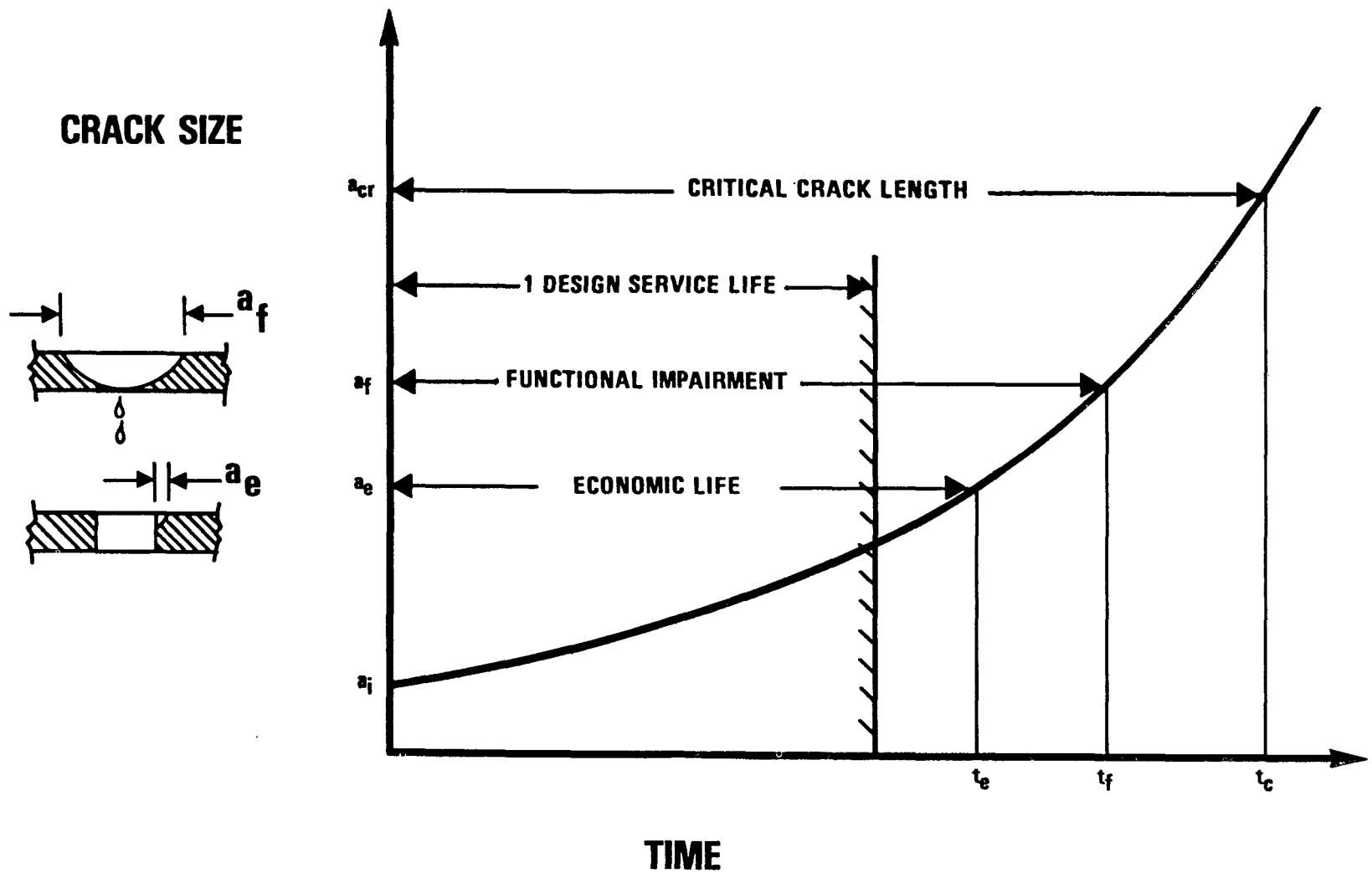


Figure 14 U. S. Air Force Durability Requirements

Durability affects airframe structural maintenance requirements, operational readiness, and life-cycle costs. Damage tolerance affects airframe structural safety. Durability, in general, is concerned with the growth of relatively small cracks which may require unscheduled structural inspection and repairs.

A clear understanding of the Air Force's durability requirements is essential to develop a responsive durability analysis methodology. Durability requirements and issues are discussed elsewhere [8, 11, 12, 18, 19, 29-32] .

3.3 DURABILITY DAMAGE MODES

Several durability damage modes affect aircraft structural maintenance. However, cracks in fastener holes are some of the most common structural maintenance problems. Tiffany [29] states that 90% or more of the fatigue cracks found in service aircraft have occurred at fastener holes. Similar conclusions have recently been reported [33] .

Cracking is only one form of in-service structural degradation that affects durability performance and structural maintenance. For example, corrosion, wear, galling, hole-out-of-roundness, lubrication, etc., are also important. Durability analyses are needed to reflect all durability damage modes. Durability is considered a structural maintenance problem. Thus, variables that affect structural maintenance also affect durability. These factors must be accounted for in airframe design.

The durability and economic life of a part, component, or airframe are currently evaluated considering the performance of cracks under expected service conditions. Cracks are convenient for characterizing structural damage. They are also suitable for applying fracture mechanics procedures. Such procedures are currently being used to assess economic life and crack performance.

3.4 ECONOMIC LIFE CRITERIA

Economic life criteria are guidelines for assessing airframe tradeoff options (e.g., design, maintenance requirements, life-cycle-costs, etc.) prior to service. Guidelines are needed for addressing the U. S. Air Force's durability and structural requirements [26-28] . Economic life criteria are described and discussed in this section. The proposed criteria can be implemented using the durability analysis methodology described in Section V and in Reference 19.

3.4.1 Introduction

Economic life is an elusive quantity referred to in MIL-STD-1530 [26] and MIL-A-8866B [27]. Economic life is currently defined in only general terms: "....occurrence of wide spread damage which is uneconomical to repair and, if not repaired, could cause functional problems affecting operational readiness" [26]. This definition is of limited use to designers because "wide spread damage" is currently subjectively defined based on the results of the durability tests and tear down inspection. This "after-the-fact" evaluation is useful for assessing the performance of the durability test article, for guiding final design changes and for evaluating structural maintenance requirments for service aircraft. However, the designer needs quidelines for designing durability critical parts and for analytically evaluating economic life tradeoffs before the durability certification test is performed. Economic life guidelines are needed for evaluating tradeoffs at four essential levels: (1) preproduction design, (2) durability certification test, (3) final production design, and (4) maintenance policy prior to service.

An economic life prediction concerns the number of cracks equalling or exceeding a specified crack size (limiting size for economical repair) for a part, component, or airframe prior to service. Several design variables affect the economic life of an airframe: material properties, design detail, initial quality, allowable stress level, material processing, manufacturing procedures, quality control, fabrication and workmanship, etc. Reliable and practical analytical procedures are needed to evaluate economic life tradeoffs for an airframe prior to service (Section V).

The objective is to assure, with a high degree of confidence, that the resulting airframe design will satisfy the durability certification tests with only minor final design adjustments required. After completing the durability certification test, the user needs analytical tools and guidelines for evaluating tradeoff options affecting life-cycle-costs, maintenance requirements, structural repairs, operational readiness, aircraft usage rotations, etc.

The proposed economic life criteria herein applies to both primary and secondary structure. Although secondary structure may not effect airframe safety, it can affect economic life. For example, repetitive structural maintenance (repairs, replacements, etc.) of secondary structure increases life-cycle costs and may effect operational readiness. A fastener hole in secondary structure will likely have looser tolerances and less stringent drilling requirements than comparable holes in primary structure. As a result, it may cost more to repair a hole in primary structure than it does for secondary

structure. If a fastener hole has to be repaired (e.g., reamed to next fastener size), it will effect maintenance cost no matter if the structure is primary or secondary. Thus, both primary and secondary structure should be considered when assessing the economic life of the airframe.

3.4.2 Philosophy

An economic life philosophy is proposed for evaluating trade-offs, such as design, life-cycle-costs, aircraft performance, and structural maintenance requirements, etc. prior to service. Elements of the proposed economic life philosophy are presented in Figure 15. Essential issues are discussed below and details will be presented in the durability design handbook at the conclusion of this program.

Two types of durability damage are emphasized for economic life: (1) cracks in fastener holes and (2) through-the-thickness cracks affecting functional impairment (e.g., fuel leaks; Figure 15, Frame B). These cracks are considered for two reasons: First, such cracks are commonly encountered in service [7,29,33]. Second, the limiting crack sizes for economic life analyses can be defined from geometric considerations. For example, the economic repair limits can easily be defined for fastener holes (Figure 11). If a through-the-thickness crack in a fuel tank causes a fuel leak, then the skin thickness provides the limiting geometry for assessing the probability of a fuel leak. Further, the repair cost/fastener hole can easily be estimated. Unfortunately, the limiting crack sizes (for economic life considerations) for other cracking sites, such as, radii, cutouts, and other stress risers are not well defined -- neither are the repair costs.

If economic life is to be related to structural maintenance cost (repair, inspection, replacement, etc.), as in the case of fastener holes, then one must first predict how many fastener hole cracks will likely exceed a specified crack size for a given service period (Figure 15, Frame G). The expected number of cracks is important because maintenance costs are directly proportional to the number of fastener hole cracks requiring repair. Fastener hole cracks are not the only mode of in-service durability damage.

When the cost to repair a part, component or airframe equals or exceeds the cost to replace, then one might say the economic life has been reached. Unfortunately, maintenance cost data for inspection, repair, replacements, etc., are limited. Also, such data may have to be extrapolated to a particular airframe situation considering such factors as inflation, type of tooling and processes involved, etc.

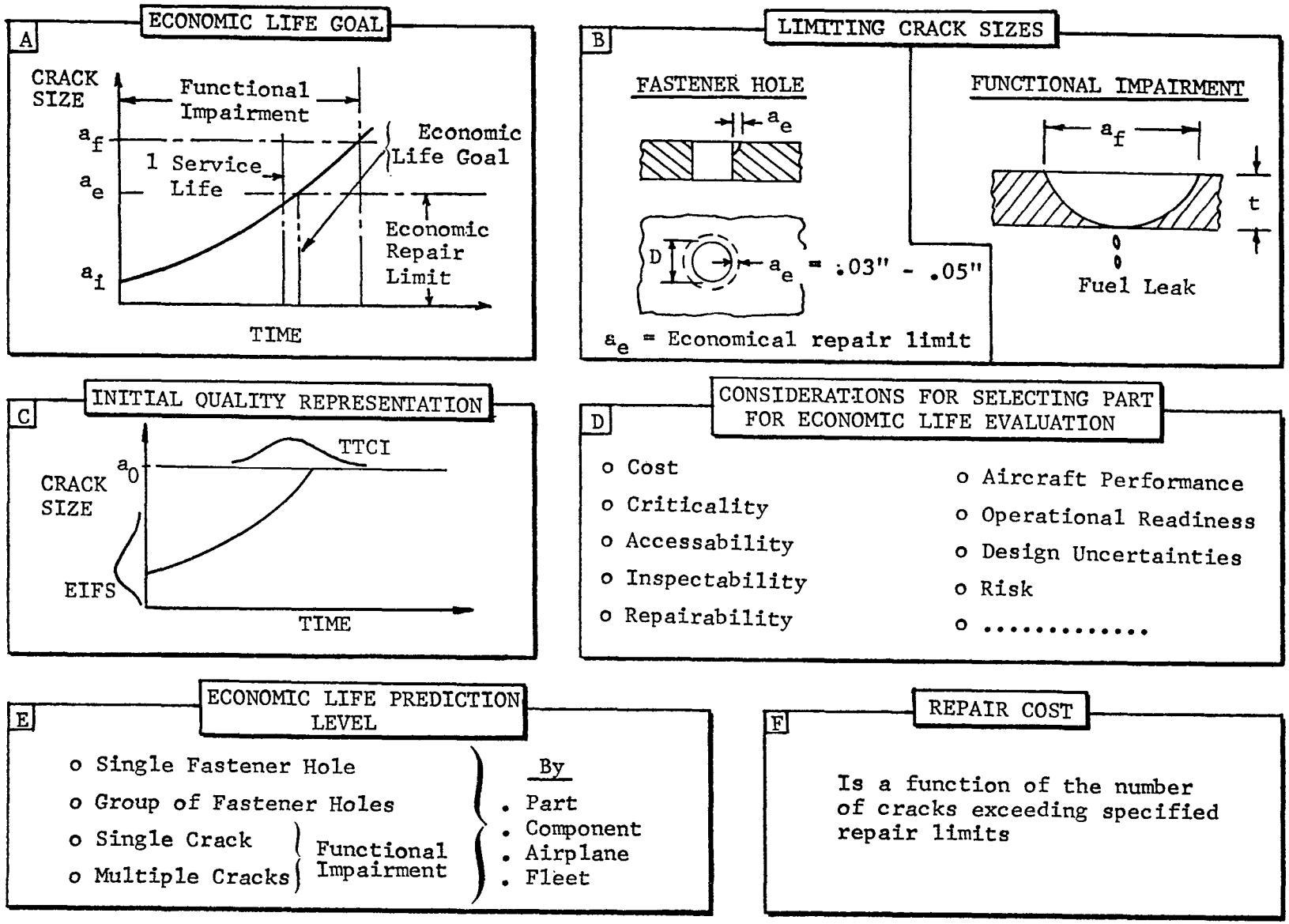
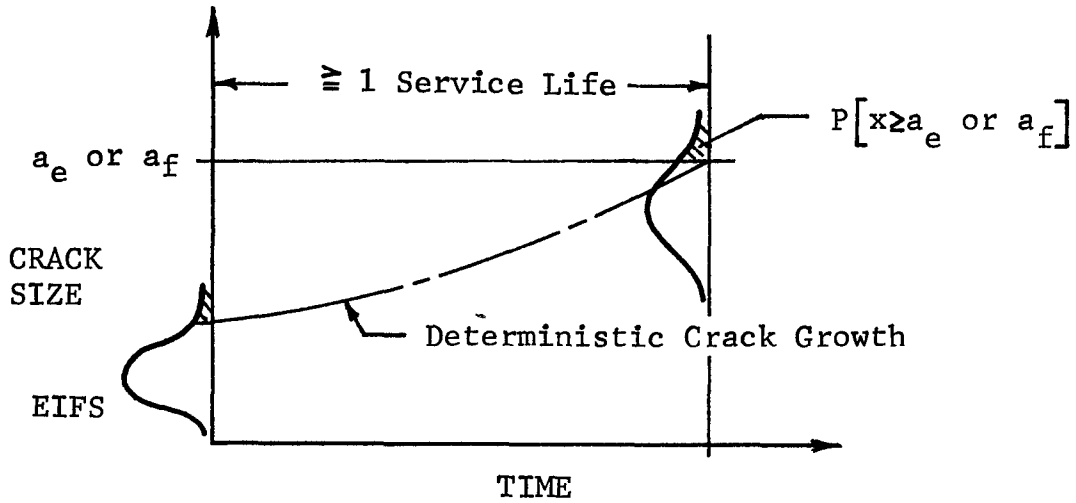


Figure 15 Elements of Economic Life Philosophy

G

PREDICT NO. OF CRACKS $\geqq a_e$ or a_f

The no. of cracks $\geqq a_e$ or a_f is an "indicator" of economic life and it provides the key for evaluating economic life tradeoff options prior to aircraft service



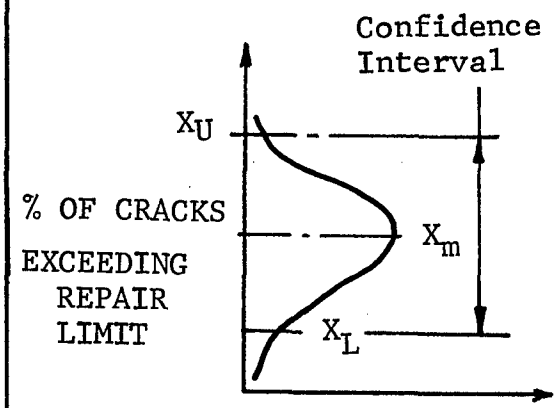
H

ECONOMIC LIFE TRADEOFFS

- COST
- LIFE
- OPERATIONAL READINESS
- DESIGN
- WEIGHT
- TESTING REQUIREMENTS
- MAINTENANCE REQUIREMENTS
- INITIAL QUALITY
- MANUFACTURING PLAN
- AIRCRAFT PERFORMANCE
- AIRCRAFT OPERATING LIMITS
- RELIABILITY / RISKS
- ....

I

STATISTICAL ANALYSIS



- Use Binominal or Poisson Statistics

Figure 15 Elements of Economic Life Philosophy (Continued)

3.4.3 Criteria

Two economic life criteria are proposed for metallic airframes: (1) probability of crack exceedance[1] and (2) cost ratio: Maintenance cost/Initial cost. Probability of crack exceedance predictions are needed for each criteria.

3.4.3.1 Probability of Crack Exceedance Criterion

The probability of crack exceedance criterion is recommended for assessing economic life and evaluating tradeoffs prior to aircraft service. This criterion is conceptually characterized in Figure 16. Durability damage is considered to be characterized by cracking.

The probability of crack exceedance provides an important indicator of economic life and for evaluating aircraft design tradeoffs prior to service. The objective is to predict the number of fastener hole cracks exceeding the economic repair limits or predict the number of cracks that will cause a fuel leak (functional impairment). Designers have the option to assess economic life at the level they want (e.g. for a single fastener hole, for a group of fastener holes, for a part, for a component, for an airplane, or for a fleet of airplanes; Figure 17).

The probability of crack exceedance is computed from the distribution of cracks which have been characterized as a function of the initial quality of the design detail and service time (Fig. 16). This calculation provides the average percentage of cracks equalling or exceeding a specified crack size (one requiring structural maintenance). In Figure 16 (Frame B) X% of the cracks $\geq a_e$ or a_f. There is 50% confidence in the X% value. Considering the number of details involved one can estimate the y% of crack $\geq a_e$ or a_f for a higher confidence level (e.g., 95%). This provides an upper bound limit for assessing economic life.

Crack exceedances, calculated for different areas or parts of a component, can be combined to assess economic life. If the crack growth of each detail (e.g., fastener hole) is statistically independent, binominal statistics [6,18,19] can be used to estimate the average number of fastener hole cracks $\geq a_e$. The statistical independence of hole cracks is a

---

1 Reference 19 details the concept of a certain percentage of the crack population exceeding a given size with a certain confidence level and a certain probability. In this report the terms "probability of crack exceedance" or "percentage of crack exceedance" refer to the same concept but the results are expressed in different formats. For example, if the probability of crack exceedance is 0.15, then the percentage of crack exceedance is 15%.

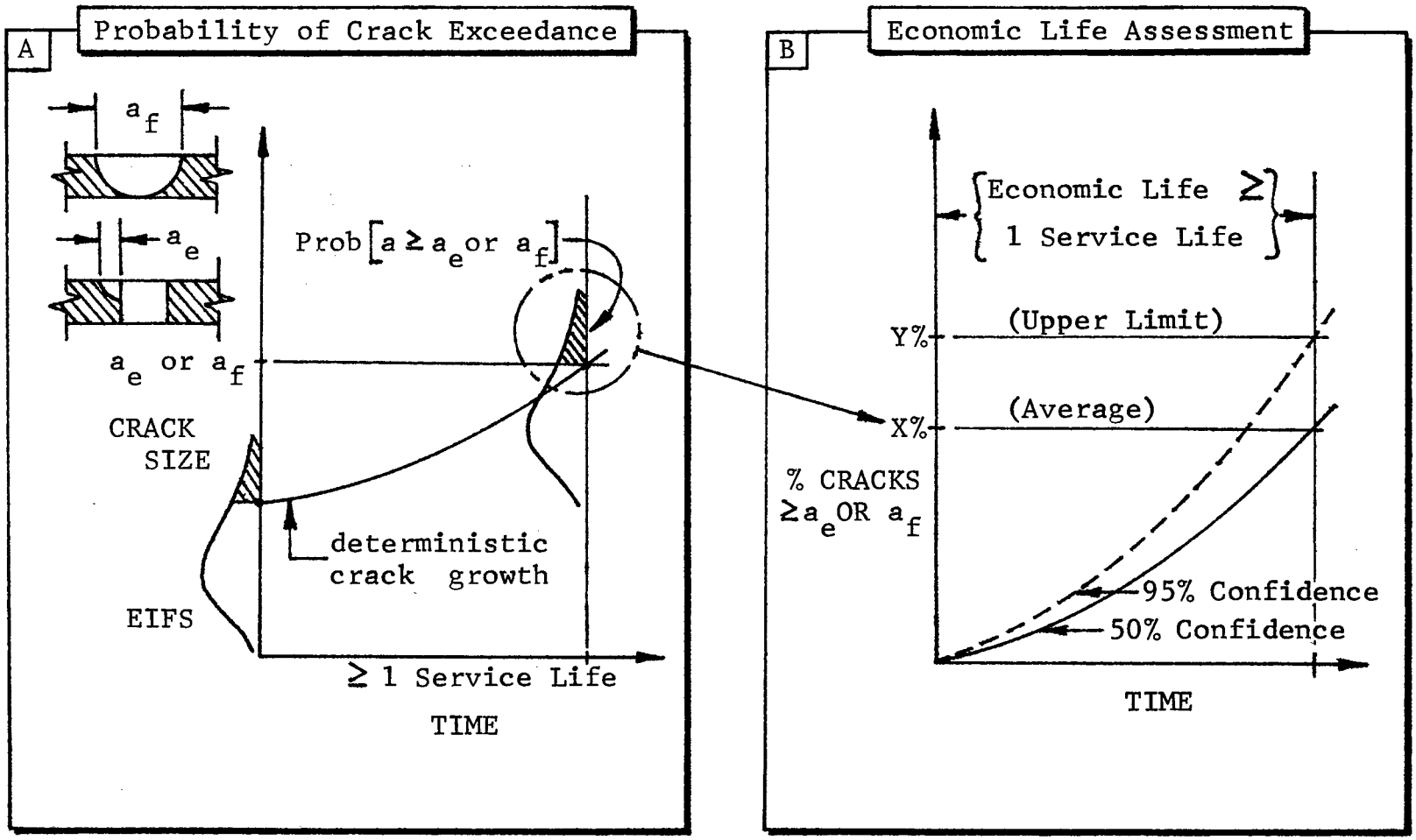


Figure 16 Economic Life Criterion: Probability of Crack Exceedance

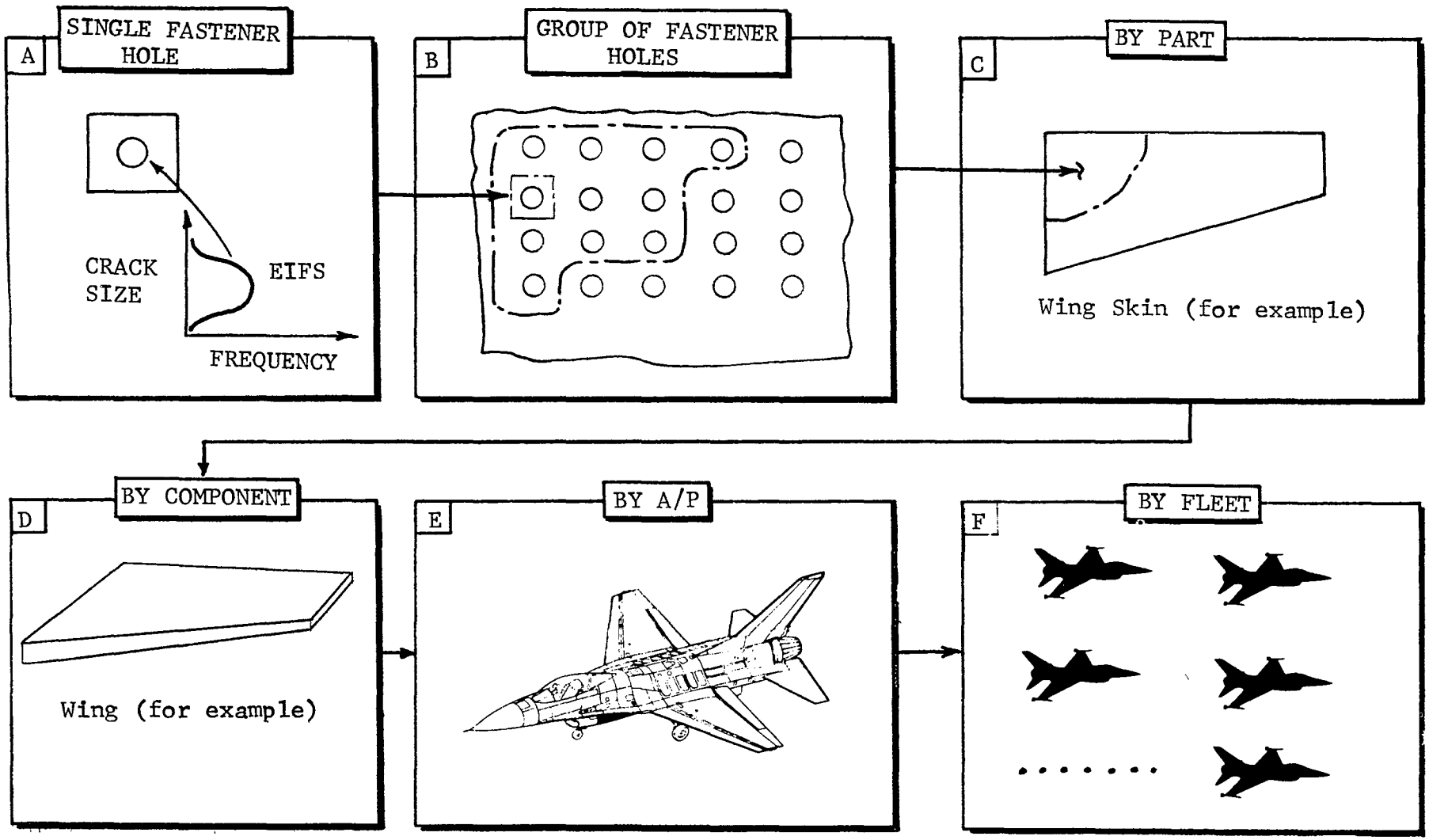


Figure 17 Assess Economic Life for Selected Level

reasonable assumption for small crack sizes associated with the economic repair limits for fastener holes (e.g., a_e = 0.03" to 0.05").

Binomial statistical concepts and applications will not be discussed in detail in this report. However, a simple example will be used to show how the probability of crack exceedance calculation for different levels can be combined. Suppose the economic life of a component is governed by two parts. Assume Part A has 100 fastener holes and $P[X \geq a_e] = 0.02$ at one service life. Part B has 200 fastener holes and $P[X \geq a_e] = 0.05$ at one service life. Using Binomial statistics, the average number of fastener holes cracks $\geq a_e$ for Part A and B is 2 (NP = 100 x 0.02 = 2) and 10 (NP = 200 x 0.05 = 10) respectively. The average total number of holes having cracks $\geq a_e$ for the component is simply the sum of the results of Part A and Part B; i.e., 12 holes (50% confidence). The number of hole cracks $\geq a_e$ can also be estimated for higher confidence levels by considering the variance and the number of details involved.

The number of cracks $\geq$ specified repair limits provides the key for estimating repair costs [6]. For example, the cost for oversizing a single hole can easily be estimated. With this information and the number of holes requiring repair, one can assess economic life and design tradeoffs in terms of costs.

The following factors should be considered for selecting structural details for economic life assessment and for setting probability of crack exceedance goals for airframe design:

- Cost
- Criticality
- Accessability
- Inspectability
- Repairability
- Aircraft Performance
- Operational Readiness
- Design Uncertainties
- Structural Life
- Risk
- .....

The probability of crack exceedance should not be a fixed value for all design applications. The value should be based on judgement and desired design objectives. For example, an expensive fracture critical part may be embedded into the wing understructure. The part is not readily accessible and it's difficult to inspect and repair. The bolt holes for this part govern its economic life. In this case, the designer may want to use a lower probability of crack exceedance goal than for an equally critical part that is more accessible and inspectible. For example, an average of 2% crack exceedance might be reasonable for this application. In another case, an average of 5% crack exceedance might be appropriate.

The probability of crack exceedance lèvel selected for design is a relative matter because $P[X \geq a_e \text{ or } a_f]$ can approach zero as a limit but it cannot theoretically equal zero. This means you can't design a highly critical part $P[X \geq a_e \text{ or } a_f] = 0$. A probability of crack exceedance level greater than zero should be selected consistent with the particular design situation. The probability of crack exceedance level affects the design allowable stress requirement. This in turn affects weight, life, life-cycle-costs, operational readiness, etc. In any case, the probability of crack exceedance goal for design should be based on the applicable design circumstances. Probability of crack exceedance goals can be set for specific areas of a part, for a complete part, for a component, for an airplane or for a fleet of airplanes.

3.4.1.2 Cost Criterion

The cost criterion for economic life is conceptually shown in Figure 18. In order to use this criterion one must have repair cost information available or estimated costs to evaluate tradeoffs at the design level. To use the cost criterion one must first predict the number of cracks exceeding specified repair limits at the desired economic life assessment level (by part, by component, etc.) The number of fastener holes that will have to be reamed to the next fastener size to clean up expected cracks must be estimated. Also, the number of fuel leaks must be predicted based on the estimated number of through-the-thickness cracks in a fuel tank. Maintenance cost can be estimated from the predicted number of details requiring repair or corrective maintenance.

The estimated maintenance cost divided by the initial production cost of the part, component, etc., provides a convenient ratio for assessing economic life and evaluating design tradeoffs [6]. The 50% and 95% confidence cost ratio (M & U, respectively) are shown in Figure 18. The Air Force should provide the contractor with "M" and/or "U" values compatible with his goal for life-cycle-costs.

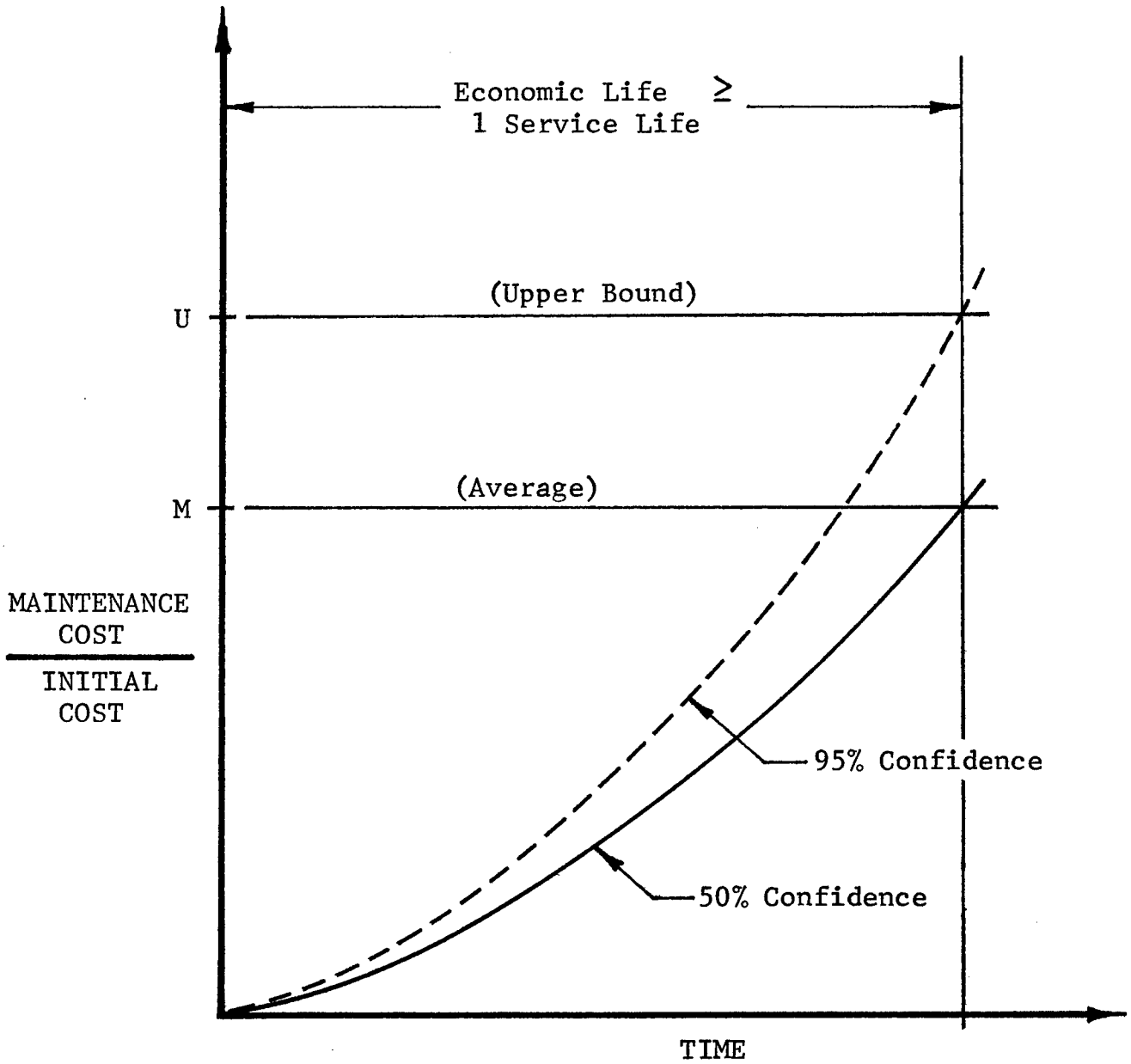


Figure 18 Economic Life Criterion
Based on Cost Ratio

3.4.4 Applications

The economic life criteria described are recommended for two different levels (Figure 19). The probability of crack exceedance criterion is recommended for both Levels I and II. However, the cost criterion is recommended only for Level II applications because limited cost data will likely be available at Level I.

3.4.5 Economic Life Tradeoffs

The two economic life criteria can be used for evaluating economic life tradeoffs prior to aircraft service. Some of the tradeoff options are shown in Figure 15 (Frame H). A conceptual example is shown in Figure 20 for evaluating the effect of the operating stress on economic life. This example illustrates the tradeoffs for two different stress levels ($\sigma 1 > \sigma 2$). Assume the same EIFS distribution and loading spectrum. In Figure 20 it is seen that the probability of exceeding a_e at t_1 for σ_1 is greater than the probability of exceeding a_e at t_2 for σ_2. The two different probabilities for exceeding a_e and the $t_2 - t_1$ life increment provides benchmarks for assessing the effects of the operating stress level used on economic life and design tradeoffs prior to service.

3.4.6 Conclusions

The formats of the proposed economic life criteria are compatible with the durability analysis methodology described in Section V and Reference 19. This methodology has been developed and tentatively verified using coupon data from the "Fastener Hole Quality" Program [4]. The proposed durability analysis methodology must be further verified for full-scale airframes. Possible methodology refinements may be required. This will be accomplished during the Phase II effort.

Economic life criteria will be presented in the durability design handbook. Any refinements to the criteria proposed herein will be reflected in the final report for this program.

3.5 CRITICAL PARTS CRITERIA

Central to any method for designing aircraft to meet durability, damage tolerance,and economic life is the development of critical parts criteria. That is, how to determine which of the design requirements controls the size, material, and processing of a part. Consequently, critical parts criteria are an integral part of the design process.

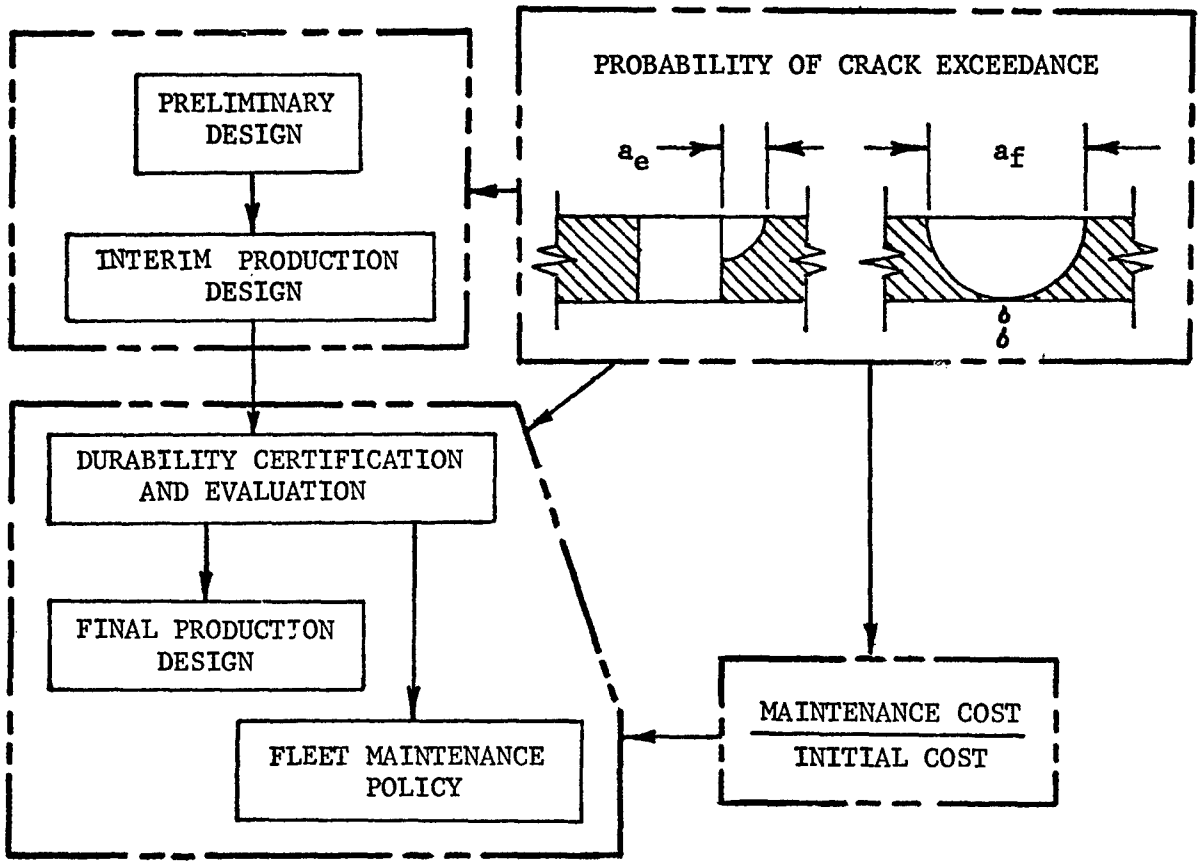


Figure 19 Economic Life Criterion Application

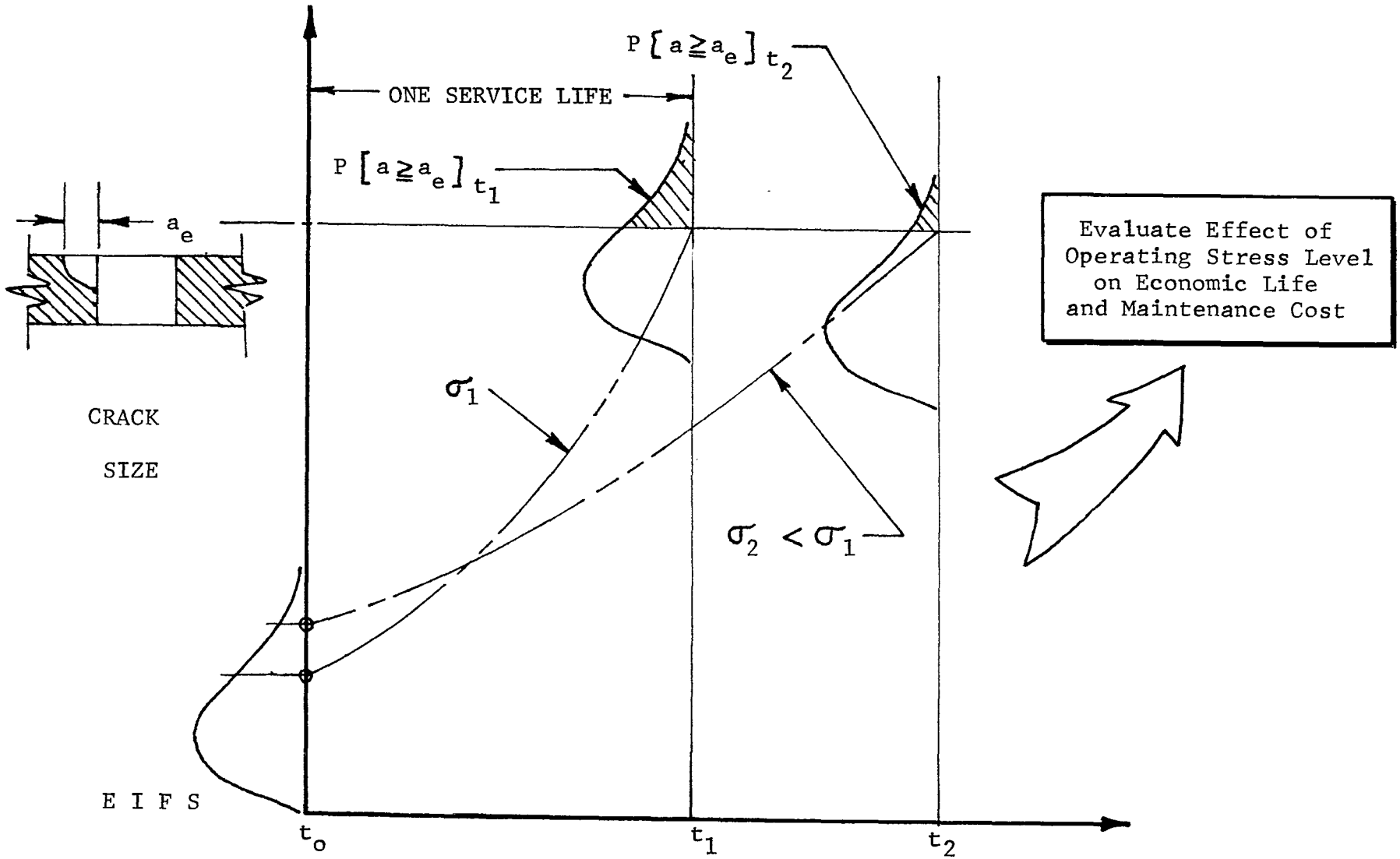


Figure 20 Example of Economic Life Tradeoff

Figure 21 depicts the steps involved in the design of a part to meet the requirements of static, durability, and damage tolerance criteria. Since both durability and damage tolerance affect economic life, this requirement is not a separate consideration.

Many decisions are required during the design of a part (Figure 21). The designer can evaluate design tradeoffs such as, stress level and life in terms of aircraft weight, performance, and cost. Undoubtedly, most aircraft will be so designed that both critical parts lists are as short as possible. Durability analyses are concerned with crack sizes affecting life-cycle costs and functional impairment (e.g., fuel leaks, loss of stiffness, etc.). Durability size cracks are typically very small (e.g., .03" to .05" crack in a fastener hole) compared to critical crack length.

In most instances, durability criteria can be met by using specially selected materials and processes in conjunction with judicious static stress levels. However, the possibility exists that the durability stress levels must be reduced as well. Such parts comprise the list of durability critical parts.

In contrast, it is difficult to meet damage tolerance, or fracture, criteria without controlling the stress levels. Consequently, the number of fracture critical parts can be large.

Figure 21 does not consider durability damage such as wear, galling, foreign object damage, etc. Such items, while part of the overall design process, only indirectly influence stress analysis and hence need not be explicitly treated.

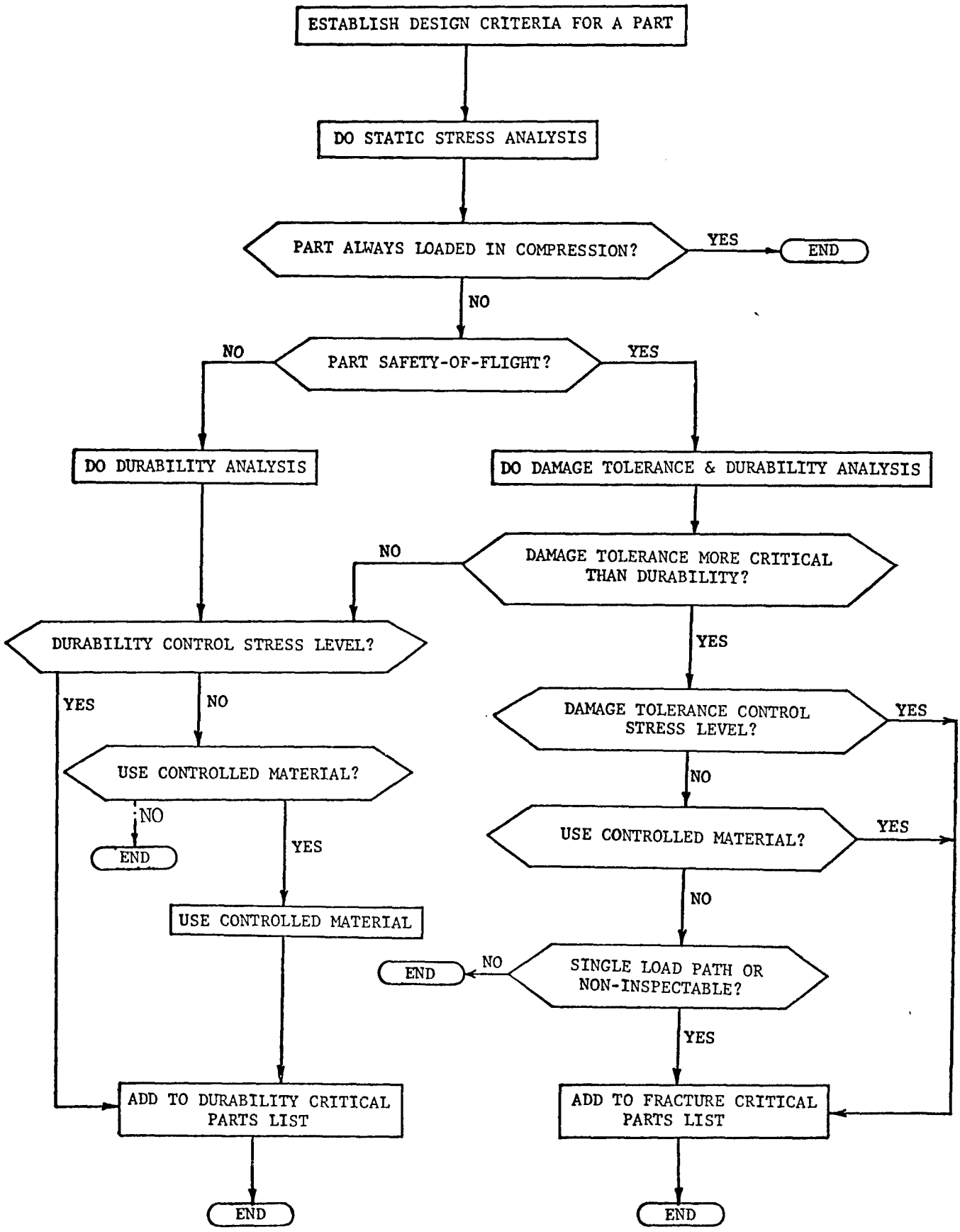


Figure 21 Flow Diagram For Selecting Durability Critical Parts

SECTION IV

INITIAL QUALITY REPRESENTATION

4.1 INTRODUCTION

In Phase I of the present program, approximate functional forms were investigated for the TTCI and EIFS distributions, explained below. Three data sets from the "Fastener Hole Quality" Program [4] were used. Statistical procedures were used to determine if any of several functional forms were applicable. These proposed distribution functions included several well-known types of functions:

- Johnson,
- Pearson,
- Weibull, and
- Asymptotic.

In addition the test procedures were applied to TTCI-compatible distribution functions which are derived by transforming the time variable in the TTCI distribution function into a crack length variable in the TTCI-compatible EIFS distribution. This transformation is based on the crack growth rate equation which was assumed to be of the power law form [5 ,6, 17-19,50].

The procedures and results are summarized herein and details are presented in Reference 40.

4.2 INITIAL FATIGUE QUALITY

An important input parameter to the durability analysis is the initial fatigue quality of the critical parts of the aircraft structure. For engineering analysis and design purposes, the initial fatigue quality has been characterized by either the time-to-crack-initiation (TTCI) or the equivalent-initial-flaw-size (EIFS), and attempts have been made in the literature [4, 12, 17-18, 48-61] to characterize the statistical distributions of both using available laboratory or field data. This section presents these concepts as they have been defined and developed during Phase I of the Durability Methods Development program.

4.2.1 Time-To-Crack-Initiation

The TTCI is defined as the time required for a crack to grow to a specified size, known as the "initiation crack size." In the present development of the durability analysis methodology, the initiation crack is always a physically observable crack size. That is, crack initiation is here defined so that the time at which it occurs can be positively identified. Since crack initiation is a readily observable event, independent observers can all agree on when that event actually occurs.

Definitions of fatigue crack initiation which are supposed to have some physical meaning but which are difficult to detect are not used. For example, sharpening of an intrusion formed from a persistent slip band might suffice for a definition of fatigue crack initiation from a metallurgical viewpoint. However, this definition of the initiation crack size for the durability methodology would be unacceptable because an intrusion is not readily detectable. Thus, a metallurgically initiated crack must undergo further growth before it reaches the initiation crack size as defined here.

Since the initiation crack size is defined so that it is easily observed, test data can be used to reliably measure the value of TTCI. For example, fractographic measurements can be made from a coupon specimen to reveal the time at which the initiation size is reached. When several specimens are tested, the TTCI will be a random variable, since both initial quality and small crack growth will depend on details of the manufacturing process and material structure which are random.

4.2.2 Equivalent-Initial-Flaw-Size

It is more convenient analytically to deal with a distribution of crack sizes at a given time than a distribution of times to reach a given crack size, such as TTCI. Usually it is desired to know the percentage of cracks exceeding a certain size at a particular time, such as at one service life. Thus, a distribution of crack sizes for any given time is the most convenient format. Also, inspection and repair/replacement will affect the crack population at the time it is done. It is necessary to describe the difference in the crack populations just prior to and just after such a procedure. Again, the distribution of crack sizes at a particular time is the proper format.

The distribution of crack sizes at time zero is a special case of this type of distribution. The apparent initial flaw size distribution at time zero, as computed from subsequent observed crack growth, is called the equivalent initial flaw size (EIFS) distribution. This distribution of flaw sizes can be transformed using the crack growth curve to obtain the distribution of flaw sizes at any other time, as explained in Section V.

4.3 STATISTICAL PROCEDURES FOR SELECTING DISTRIBUTION FUNCTION

4.3.1 Graphical

A large number of distribution functions will be considered to characterize the EIFS data. An engineering approach can be used to select from these distribution functions only those distributions that have the prospect of passing further tests of goodness-of-fit, while eliminating those that are obviously incompatible with the given EIFS data. The approach is to use the β_1-β_2 plane as shown in Fig. 22 [36] where each of these distributions may be identified either as a point or as a curve or as a region. The analytic relationships between β_1 and β_2 of some well-known distribution functions are shown in Fig. 22.

It will be assumed that these relationships hold for the point estimates of β_1, β_2 given by b_1, b_2. Similar figures have been used to single out those distribution functions which probably will fit well to the observed data. This is done by determining whether the point (b_1, b_2) plotted in the β_1-β_2 plane is inside the region (or close to the point or the curve) associated with the distribution function with respect to which the goodness-of-fit is to be considered.

4.3.2 Goodness-of-Fit Test

Although there are a number of possible ways in which a numerical test can be performed on goodness-of-fit (for example, X^2 test and the Kolmogorov-Smirnov test), for the present investigation the ω^2 method [35] is used. Let F(x) be the distribution to be tested and $F_n(x)$ be the empirical distribution based on a sample of size n, and form a statistic

$$n\omega_n^2 = n\int_{-\infty}^{\infty} [F_n(x) - F(x)]^2 dF(x) \qquad (1)$$

This statistic is distribution-free and some of the percentiles of its asymptotic distribution are listed below [35]:

$P[n\omega_n^2 \leq a]$	0.80	0.85	0.90	0.95	0.98	0.99
a	0.241	0.284	0.347	0.461	0.620	0.743
% Significance	20	15	10	5	2	1

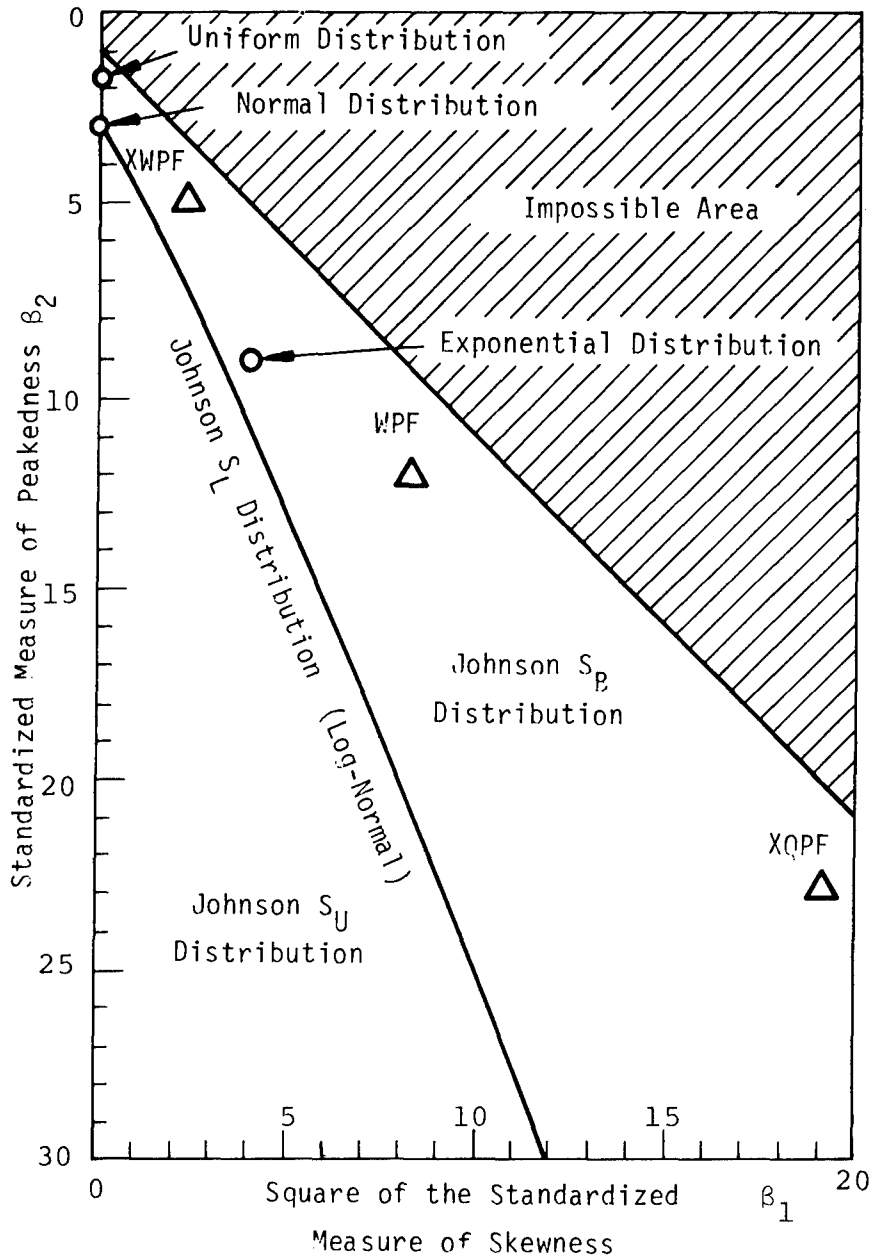


Fig. 22 Estimated Result of (β_1, β_2) for Data XQPF, XWPF, and WPF (Johnson Distribution Family)

The level of significance is the chance that $n\omega_n^2$ is as large as or larger than the observed value assuming that the observed value results from random sampling of a population having the proposed distribution. The proposed distribution is rejected if this significance level is small (conventionally 5 or 10%). Note that a high significance does not imply that the proposed distribution is the true distribution.

4.4 TEST DATA

Three sets of EIFS data were obtained from the "Fastener Hole Quality" Program [4] XQPF, XWPF and WPF. Specimen descriptions for these data sets are noted below:

X= Load transfer specimen
W= Winslow drilled holes
P= Proper drilling techniques
Q= Quackenbush-drilled and -reamed holes
F= Fighter spectrum

All EIFS data values are given in mils (10^{-3}inches). The cumulative probability of each EIFS value is calculated assuming equal probability of occurence, i.e.

$$F(x_i) = \frac{i}{n + 1} \tag{2}$$

where n is the total number of EIFS data values with the data arranged in ascending order. The EIFS values and cumulative probabilities are tabulated in Table 5 . Also point estimates of some fundamental statistics are listed. These estimates include:

Mean	- m_1'
Moments	- m_2, m_3, m_4
Standard Deviation	- σ
Skewness	- $\sqrt{b_1}$
Peakedness	- b_2

4.5 DESCRIPTION OF DISTRIBUTION FUNCTIONS

Several distribution functions were considered including:

- Johnson family
- Pearson family
- Weibull

Table 5 Ordered Observations, Plotting

Positions, and Sample Statistics

XQPF			XWPF			WPF		
NO.	EIFS (MILS)	CUM.PROB.	NO.	EIFS (MILS)	CUM.PROB.	NO.	EIFS (MILS)	CUM.PROB.
1	.026	.0263	1	.093	.0263	1	.140	.0256
2	.031	.0526	2	.145	.0526	2	.236	.0513
3	.047	.0789	3	.145	.0789	3	.280	.0769
4	.050	.1053	4	.160	.1053	4	.280	.1026
5	.058	.1316	5	.175	.1316	5	.320	.1282
6	.059	.1579	6	.180	.1579	6	.367	.1538
7	.060	.1842	7	.180	.1842	7	.367	.1795
8	.063	.2105	8	.190	.2105	8	.420	.2051
9	.063	.2368	9	.190	.2368	9	.450	.2308
10	.068	.2632	10	.210	.2632	10	.450	.2564
11	.068	.2895	11	.210	.2895	11	.450	.2821
12	.090	.3158	12	.240	.3158	12	.482	.3077
13	.096	.3421	13	.240	.3421	13	.518	.3333
14	.105	.3684	14	.240	.3684	14	.520	.3590
15	.110	.3947	15	.240	.3947	15	.520	.3846
16	.113	.4211	16	.250	.4211	16	.560	.4103
17	.120	.4474	17	.290	.4474	17	.560	.4359
18	.130	.4737	18	.295	.4737	18	.600	.4615
19	.150	.5000	19	.295	.5000	19	.647	.4872
20	.170	.5263	20	.295	.5263	20	.647	.5128
21	.200	.5526	21	.295	.5526	21	.650	.5385
22	.220	.5789	22	.330	.5789	22	.698	.5641
23	.270	.6053	23	.330	.6053	23	.698	.5897
24	.300	.6316	24	.330	.6316	24	.698	.6154
25	.536	.6579	25	.420	.6579	25	.698	.6410
26	.536	.6842	26	.420	.6842	26	.698	.6667
27	.611	.7105	27	.420	.7105	27	.698	.6923
28	.612	.7368	28	.470	.7368	28	.754	.7179
29	.650	.7632	29	.470	.7632	29	.754	.7436
30	1.090	.7895	30	.540	.7895	30	.817	.7692
31	1.090	.8158	31	.612	.8158	31	1.040	.7949
32	1.090	.8421	32	.700	.8421	32	1.140	.8205
33	1.100	.8684	33	.810	.8684	33	1.250	.8462
34	1.140	.8947	34	.810	.8947	34	1.250	.8718
35	1.240	.9211	35	.870	.9211	35	1.490	.8974
36	3.000	.9474	36	.940	.9474	36	1.640	.9231
37	7.700	.9737	37	1.280	.9737	37	2.730	.9487
						38	3.830	.9744

MEAN= .6233	MEAN= .3868	MEAN= .7986
M2= 1.7214	M2= .0694	M2= .4660
M3= 9.8704	M3= .0285	M3= .9179
M4=68.7061	M4= .0241	M4= 2.6276
STDV= 1.3120	STDV= .2634	STDV= .6826
$\sqrt{b_1}$= 4.3701	$\sqrt{b_1}$= 1.5598	$\sqrt{b_1}$= 2.8860
b_2=23.1853	b_2= 5.0060	b_2=12.1022

- Asymptotic
- TTCI Compatible

Most of these functions are well known. The Johnson family [36] is composed of three types: S_L (Log normal), S_B, and S_U. The Pearson family [37, 38, 39] is composed of twelve types: I-XII. There are two types of Weibull distribution functions [36], 2-parameter and 3-parameter. Three types of asymptotic distribution functions of largest values were considered - first, second and third asymptotic. The TTCI compatible distribution functions are the only special functions to be considered.

A distribution function for EIFS can be developed which is compatible to the TTCI distribution by transforming the time variable in the TTCI distribution into an equivalent crack length variable using the appropriate crack growth law. The compatible EIFS distribution is then defined as the equivalent crack length distribution at time zero. Shinozuka [5] and Yang [16, 17] have applied this technique to the generation of compatible distribution functions for TTCI distributions of two functional forms:

Weibull
Log-Normal.

Both of these TTCI compatible distributions will be considered.

First the parameters of the distribution functions must be estimated for each of the data sets. There are several techniques for determining these estimates including

- Maximum Likelihood,
- Moments,
- Least Squares, and
- Bayes.

The actual estimation procedures used for each of the distribution functions considered are detailed in Reference 40. In most cases, least squares estimates are used. However in some cases, the best estimate is defined as the estimate which produces the highest level of significance in the goodness-of-fit test.

4.6 RESULTS

Several of the proposed distribution functions are rejected based on Figures 22 and 23. Point estimators b_1, b_2 of β_1, β_2 are graphed for each data set. Also shown in Figures 22 and 23 are the characteristics for several of the proposed distribution functions and the allowed data domain for each distribution. These domains can be points, curves, or regions of the graph (Figs. 22 and 23). From Figure 22 all distributions are rejected except for the Johnson S_B and S_L distributions. All three data points fall in the S_B domain. While the three points are not particularly close to the S_L curve, their general proximity to the curve suggests that reasonable fits may be realized. In Figure 23 the three data points lie in the domain of the Pearson Type I distribution. One or more of the data points are outside the domains of all other Pearson distribution functions. Thus, all except the Pearson Type I distribution function are rejected based on Figure 23. Thus, graphically the following distribution functions have been rejected as possible distribution functions:

uniform
normal
Johnson S_U
exponential, and
Pearson Types II-- XII.

4.6.2 Goodness-Of-Fit Test

The results of the $n\omega_n^2$ test are given in Table 6. At 5% significance the following distributions can be rejected for the available data:

Johnson S_B & S_L
Pearson I
Asymptote 1, and
Asymptote 3.

Details of the computations are given in Reference 40. Note that the Weibull-compatible distribution has a consistently high significance level (i.e. $n\omega^2$ is low).

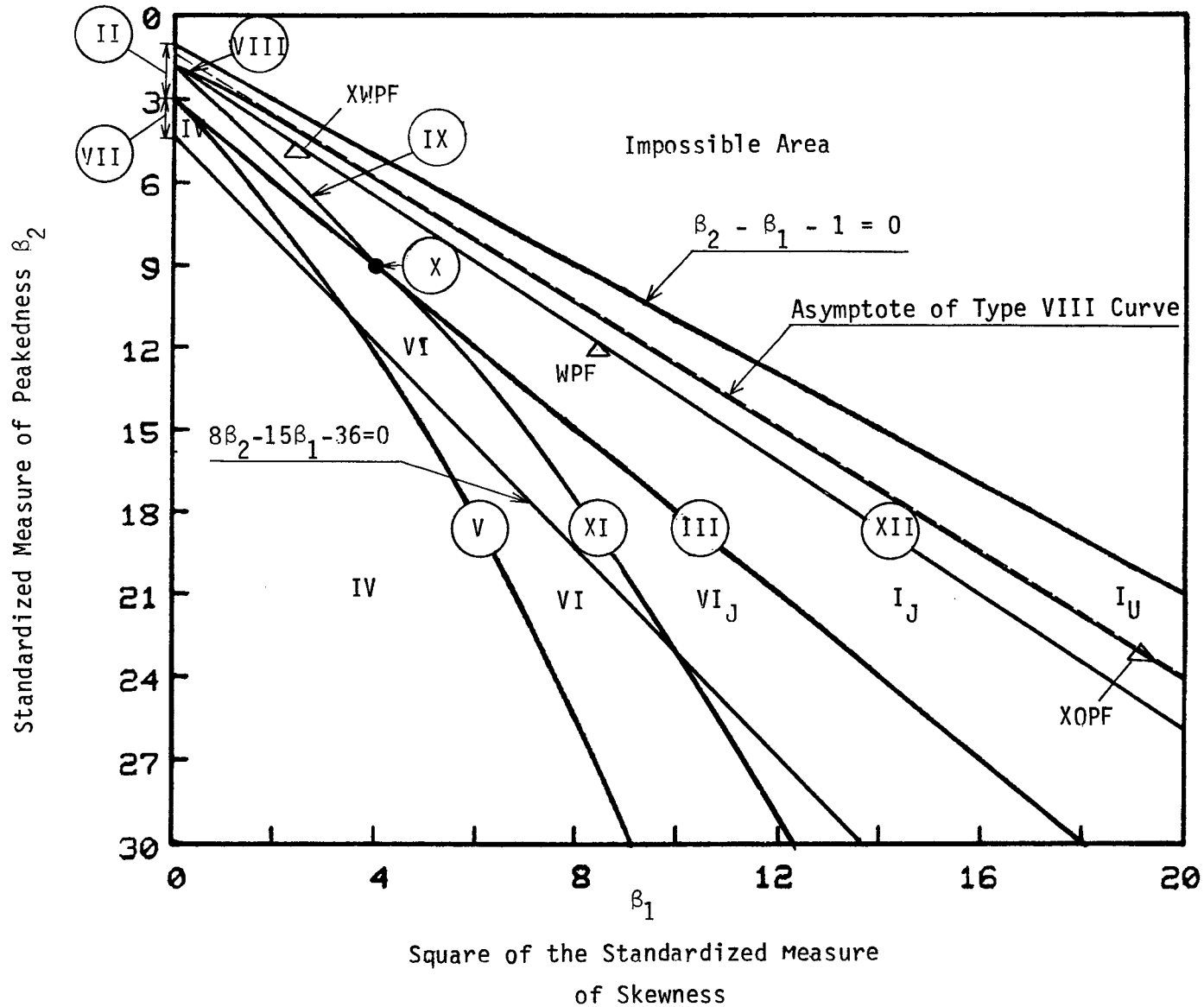


Figure 23 Estimated Result of (β_1,β_2) for Data XQPF, XWPF and WPF (Pearson Distribution Family)

TABLE 6

GOODNESS-OF-FIT RESULTS

$n\omega_n^2$

TYPE DISTRIBUTION \ DATA SET	XQPF	XWPF	WPF
Johnson S_B**	*	.664	.283
Johnson S_L**	1.65	.455	.333
Pearson I **	6.64	3.02	6.03
Weibull - 2	.248	.231	.304
Weibull - 3	.159	.125	.241
Asymptote 1**	1.48	2.63	.697
Asymptote 2	.0672	.0306	.140
Asymptote 3**	1.62	.370	.779
Compatible Weibull	.0869	.0345	.0151
Compatible Log-Normal	.184	.109	.181

* Unacceptable Parameter Fit

** Rejected at 5% Significance

4.7 CONCLUSIONS

Distribution functions of the Johnson family (including log-normal distribution functions), the Pearson family, Weibull, Asymptotic (the First, Second, and Third) and TTCI compatible EIFS distributions (Weibull and log-normal compatible EIFS distribution) have been examined by means of the ω^2 method for their acceptability in describing the statistical characteristics of EIFS. All of these proposed distribution functions were rejected except:

Weibull,
Asymptote II, and
TTCI Compatible
(Weibull & Log-Normal)

These results are based on three data sets and should not be considered as final until additional data is tested. However, it appears that the empirically derived compatible distribution functions provide acceptable descriptions of the actual EIFS distributions.

SECTION V

DURABILITY ANALYSIS METHODOLOGY DEVELOPMENT

5.1 INTRODUCTION

The durability methodology developed during Phase I is a probabilistic crack-growth approach for quantifying the economic life of advanced metallic airframes. The methodology is useful for structures containing many details which can be grouped according to common geometry and stress history. A fastener hole is the prototype used in developing the methodology, but other structural details such as fillets or cutouts could also be modeled.

Since damage is quantified as crack length, types of damage not easily characterized as cracks are not readily handled by the algorithm developed in Phase I. These other types will be included in subsequent phases of the program as time permits. By using crack length as the damage parameter, the durability methodology is consistent in philosophy with USAF damage tolerance requirements. The probabilistic approach used is the most responsive to durability analysis requirements and is coincidently compatible with assessments of airframe structural reliability based on damage tolerance concepts.

An analytical durability methodology that is capable of quantitatively predicting the economic life of advanced metallic aircraft structures has been developed. The methodology accounts for various factors, such as initial fatigue quality, loading spectrum, material/structural properties, usage changes, etc. Economic life can be determined using either of the following criteria: (1) a rapid increase in the number of crack damages exceeding the economic repair crack size, or (2) a rapid increase in the maintenance cost. The economic repair crack size a_e is defined as the crack size below which the least expensive repair procedure can be used, such as reaming the fastener holes to the next hole size. a_e is usually between 0.03" and 0.05" depending on the location and the fastener hole size.

The durability critical component may also be subjected to a scheduled (nonperiodic) inspection and repair maintenance procedure as shown in Fig. 24 , in which the component without a maintenance procedure is a special case. Within the framework of the first criterion of the economic life, the percentage (or numbers) of cracks exceeding a_e is obtained as a function of the service time. The percentage of cracks exceeding a_e is schematically shown in Fig. 16 . For the second criterion of the economic life, the average cost of maintenance, including the costs of inspection and repair, has been formulated as a function of service time, thus permitting

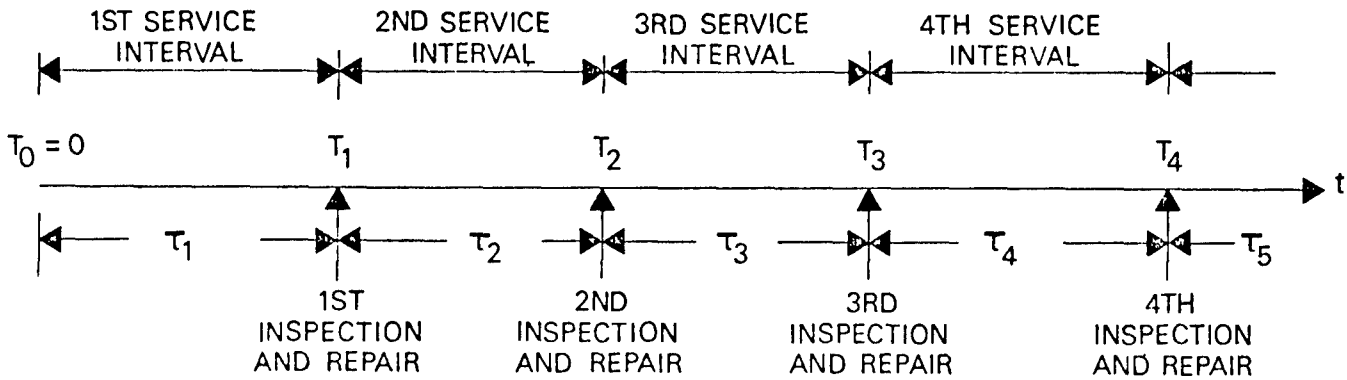


Figure 24 Inspection and Repair Maintenance Schedule

a determination of the possible economic life.

The analytical methodology is demonstrated herein by a numerical example. It is shown numerically that the number of cracks exceeding the economic repair crack size (for a given probability and confidence level) increases rapidly after a certain service time, thus determining the possible economic life of the durability critical component. While the inspection and repair maintenance procedures have a significant impact on the safety and reliability of aircraft structures [e.g., 5, 6, 15, 45-47], its effect on the economic life of a component is shown to be limited.

The development of the durability analysis methodology is summarized in this section. Details of the methodology, including a complete description of the methodology and derivation of all important results, are presented in Volume V [19].

5.2 Initial Fatigue Quality

The EIFS and TTCI distributions provide a measure of the initial fatigue quality, which is a necessary input parameter to the durability analysis methodology. The crack growth curve relates the two distributions. The considerations affecting the choice of these concepts and their inclusion in the durability analysis methodology are discussed below.

5.2.1 Relationship of EIFS Distribution to TTCI Distribution

Two natural concepts for describing initial fatigue quality have been presented. However, both have limitations. The TTCI distribution can be easily observed and measured, and can be tied directly to a physical event, but it is in an inconvenient format. The time-to-crack-initiation is a function of many variables including loading spectrum and stress level, since these affect the crack growth rate. If one of these variables changes then test data may have to be generated again, since the functional relation between TTCI and other influencing variables is not available to date. This situation becomes particularly critical in the present durability analysis where the crack growth damage at each location of the entire durability component has to be estimated. Since the maximum stress level of the loading spectrum varies from one location to another, it may not be economically practical to conduct laboratory tests for TTCI for all maximum stress levels which may occur at all the fastener holes. This problem is considered further below and will be studied extensively during Phase II of this program.

The distribution of crack sizes at time zero gives the desired information in a convenient format, but it cannot be measured. In fact, an initial flaw may not be a crack at all, but rather be an inclusion, corrosion pit, etc. In order to overcome these drawbacks, the "equivalent" initial flaw size distribution (EIFS) is defined. The EIFS distribution characterizes the distribution of crack sizes at time zero. However, the actual distribution cannot be directly measured because EIFS's are not physically observable cracks per se. Thus, the EIFS distribution must be justified using fractographic results. For example, assume that after 4000 hours a crack 0.020 inches long is observed. And assume that from the crack growth curve it is determined that a 0.020 inch long crack would have been just 0.003 inch long 4000 hours earlier. Then the "equivalent" initial flaw size for that structural detail is 0.003 inches. The distribution of all such equivalent initial flaw sizes for similar structural details is the EIFS distribution.

In the present development of the durability analysis methodology, the EIFS distribution is justified using TTCI results and observed crack size at a given time. The EIFS for each TTCI can be obtained using the crack growth curve to transform the observed TTCI value back through time. An example will be given in section 5.2.4.

The above method of defining the EIFS distribution ties the EIFS and TTCI distributions together intimately. The EIFS distribution may or may not represent the actual crack size distribution at time zero. Nobody knows for certain because the latter distribution cannot presently be determined. But the EIFS distribution does have physical significance in that it gives the same results as the TTCI distribution which is obtained directly from coupon specimen tests. That is, the EIFS is defined in such a way that it reproduces exactly the physically observed TTCI distribution after some crack growth has occurred. Thus the durability analysis methodology presented here does not distinguish the equivalent-initial-flaw-size approach from the time-to-crack-initiation approach.

5.2.2 Statistical Distribution of Time-To-Crack-Initiation

From a physical standpoint, it is generally agreed that the fatigue of metals is a wear-out process. For example, the longer a specimen has survived, the higher the probability the specimen will fail. Thus the failure rate of the fatigue life should increase monotonically. If the failure rate of the fatigue life is a positive power function of the service time, then it can be derived mathematically that the distribution function of the time to crack initiation is Weibull, i.e.,

$$F_T(t) = P[T \leq t] = 1 - e^{-(t/\beta)^{\alpha}} , \qquad t \geq 0 \qquad (3)$$

in which T is a random variable indicating the time to crack initiation, α is the shape parameter, and β is the scale parameter. In Eq. 3 $F_T(t) = P[T \leq t]$ is the probability that the time to crack initiation, T, is smaller than or equal to a value t.

The Weibull distribution given by Eq. 3 has been used to describe the time to crack initiation [50, 52-60], and methods of estimating the parameters α and β from test data have been available [53,54].

Observations of extensive specimen test data indicate that α is fairly constant for a particular material and it is not sensitive to specimen geometry, testing method, specimen size, etc. Compilation of coupon test data indicates that $\alpha = 4.0$ may be appropriate for aluminum [54]. Moreover, attempts have been made to apply the Weibull distribution for the time-to-crack-initiation to available service data for various types of aircraft; for instance C-130, C-141, F4, etc. [50, 57-60]. The improvement in predicting the time to service crack initiation has also been made by taking into account the statistical variability of service loads [60].

The lower bound of the two-parameter Weibull distribution given by Eq. 3 is zero. The lower bound, in effect, depends on the definition of the crack size a_0 at crack initiation. As a result, the following three-parameter Weibull distribution was suggested [17, 18]

$$F_T(t) = P[T \leq t] = 1 - \exp\left\{-\left(\frac{t-\varepsilon}{\beta}\right)^{\alpha}\right\} \quad ; \qquad t \geq \varepsilon \tag{4}$$

in which ϵ is the lower bound at TTCI. The lower bound ϵ increases as the defined crack size a_0 at crack initiation increases. A comparison of Eq. 4 and test data obtained during the Fastener Hole Quality program [8] is shown in Fig. 25. A reasonably good fit to test data is obtained.

5.2.3 Fatigue Crack Propagation In Small Crack Size Range

It has been observed previously that the crack propagation rate for very small cracks is different from that of large cracks at the same level of applied stress intensity factor [65, 66, 68]. This has been attributed to the effects of crack closure, plasticity, and difficulties in using continuum mechanics at a size scale where the material is heterogeneous. It also appears that the stress intensity factor calibration should be modified for short cracks [67].

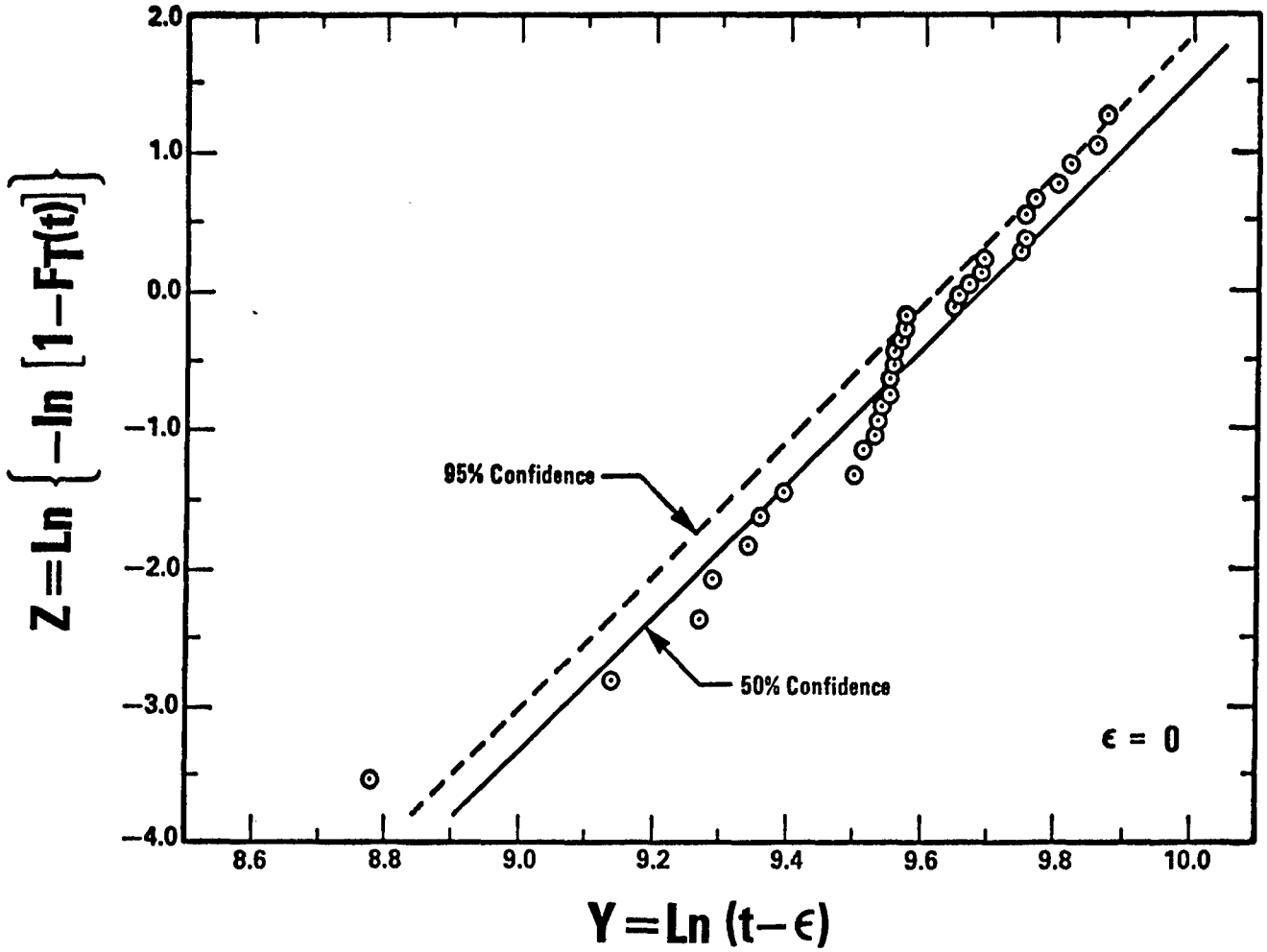


Figure 25 Weibull Best Fit Plot for WPF Data Set from Fastener Hole Quality Program (Ref. 4)

Other investigators have found no difference between small and large flaw fatigue crack propagation (FCP) behavior [69, 70]. However, these investigations did not consider cracks smaller than 5 or 10 thousandths of an inch long. Apparently large differences may not be apparent among flaws above this size.

Fractographic data similar to that which will be obtained in Phase II has already been obtained during the Fastener hole Quality program [4]. Data was taken for several sets of coupon specimens made of 7475-T7351 aluminum tested under a random loading spectrum. This data has been obtained for cracks as small as 0.0005 inches and indicates a significant deviation in FCP at the smallest sizes compared to the larger (.010 inch) cracks. While there is some variation from specimen to specimen of a given set, the FCP rate of small cracks is always observed to be of the form

$$\frac{da(t)}{dt} = Q\, a^b(t). \tag{5}$$

A typical plot of test data and Eq. 5 shows good agreement, as in Figure 26 . The relation given in Eq. 5 eventually becomes invalid as the crack length exceeds about .030 inches. The transition from small flaw growth to large flaw growth is complicated in these specimens because the transition from part-through to through-thickness flaw geometry occurs at about the same crack length.

As shown in Ref. 19, the form of Eq. 5 can be derived using linear elastic fracture mechanics (LEFM) concepts. Several approaches such as Gallagher's miniblock approach [62-64] or the well-known Paris equation can be used. However, the values of Q and b in Eq. 5 cannot be obtained from these derivations and large crack data. Instead, the values of Q and b are obtained directly from coupon specimen test data [17-19]. Specimen tests used to obtain the TTCI distribution can be used for this purpose.

The conclusion reached during Phase I is that Eq. 5 can be used to model the FCP rate of cracks up to a few hundredths of an inch, after which conventional linear elastic fracture mechanics concepts apply. Phase II will encompass many hundred more tests to verify this conclusion. During Phase II an effort will be made to relate small crack growth, i.e. the constants Q and b, to large crack growth predicted by LEFM concepts.

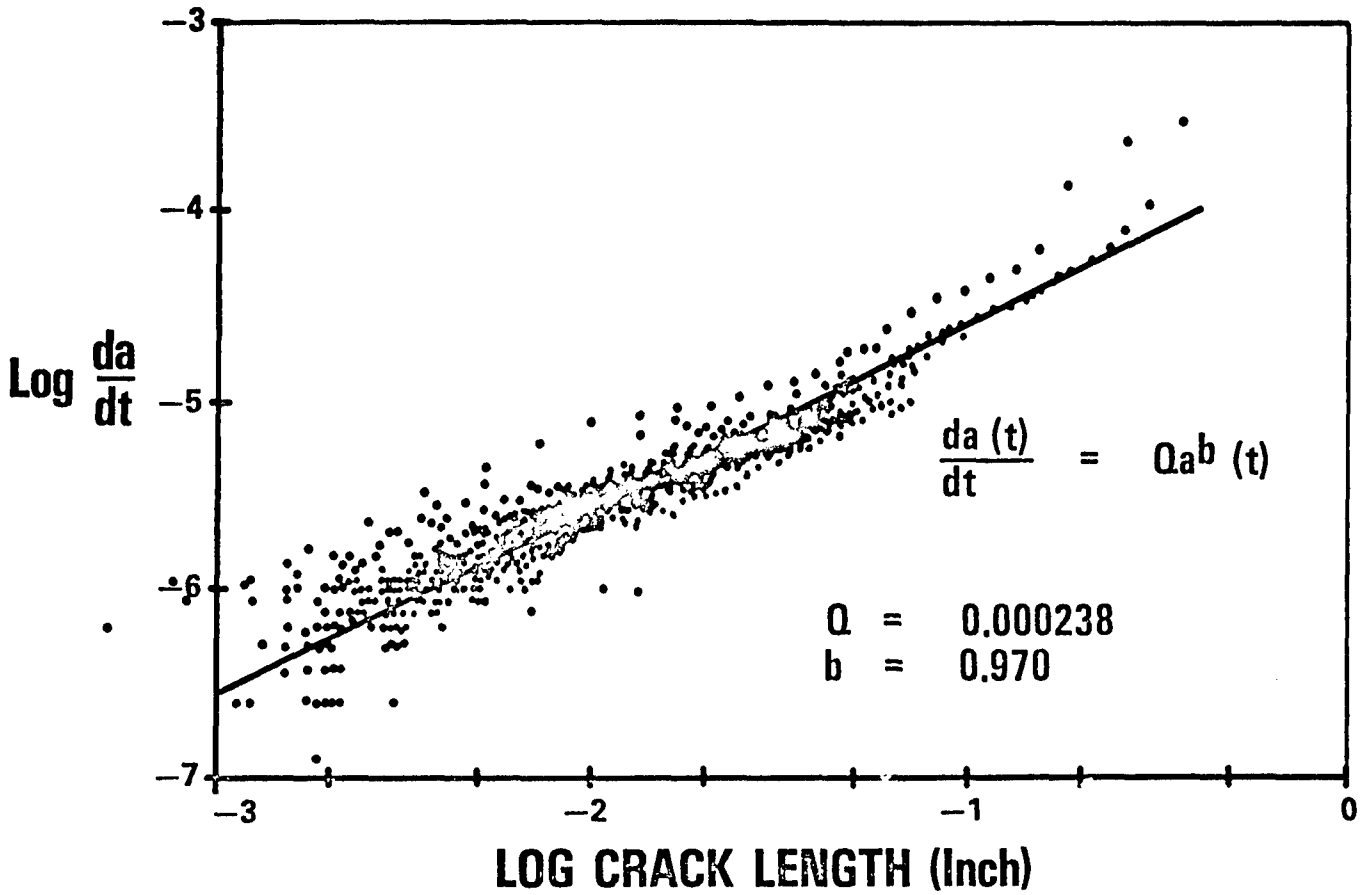


Figure 26 Log Crack Length Versus Log da/dt For WPF Data Set From Fastener Hole Quality Program (Ref. 4)

5.2.4 Statistical Distribution of Equivalent-Initial-Flaw-Size

If the crack growth curve and TTCI distribution function are known, the EIFS distribution function can be derived. Details are presented in Volume V [19] but a short example is given here to outline the method.

Let $a(T) = a_0$ be the crack size at crack initiation, where T is the time to crack initiation, and $a(0)$ be the initial crack size (at $t = 0$) prior to service. Integration of Eq. 5 from $t = 0$ to $t = T$ yields

$$a(0) = \frac{a_0}{(1 + a_0^c cQT)^{1/c}} \tag{6}$$

in which

$$c = b - 1. \tag{7}$$

The statistical distribution of time-to-crack-initiation is given by Eq. 4 . Thus, the statistical distribution of the initial crack size, $a(0)$ can be derived from Eq. 4 through the transformation of Eq. 6 as follows [Refs. 17-19]:

$$F_{a(0)}(x) = \exp\left\{-\left[\frac{x^{-c}-a_0^{-c}-cQ\varepsilon}{cQ\beta}\right]^{\alpha}\right\} \quad ; \quad 0 \leq x \leq x_u \tag{8}$$

$$= 1 \quad ; \quad x \geq x_u$$

where x_u is the upper bound of the initial crack size.

$$x_u = (a_0^{-c}+cQ\varepsilon)^{-1/c} \tag{9}$$

Such a transformation is schematically shown in Fig. 27.

Equation 6 has been used to obtain EIFS values from the TTCI values shown in Fig. 25. These values are compared in Fig. 28 to the theoretical distribution given by Eq. 8 . Also shown is the 95% confidence level curve, determined as in Vol. V [19]. The good agreement shown in Fig. 28 was also obtained for other data sets investigated and was generally better than for other distribution functions. Details are given in Vol. IV [40].

5.3 CRACK GROWTH DAMAGE ACCUMULATION

The EIFS population is grown analytically using the crack growth curve to model the growth of the actual initial flaw population which occurs in service. The resulting analysis can be used to find the percentage of cracks exceeding any size at any time. Results of such an analysis will be included in Section 5.4.

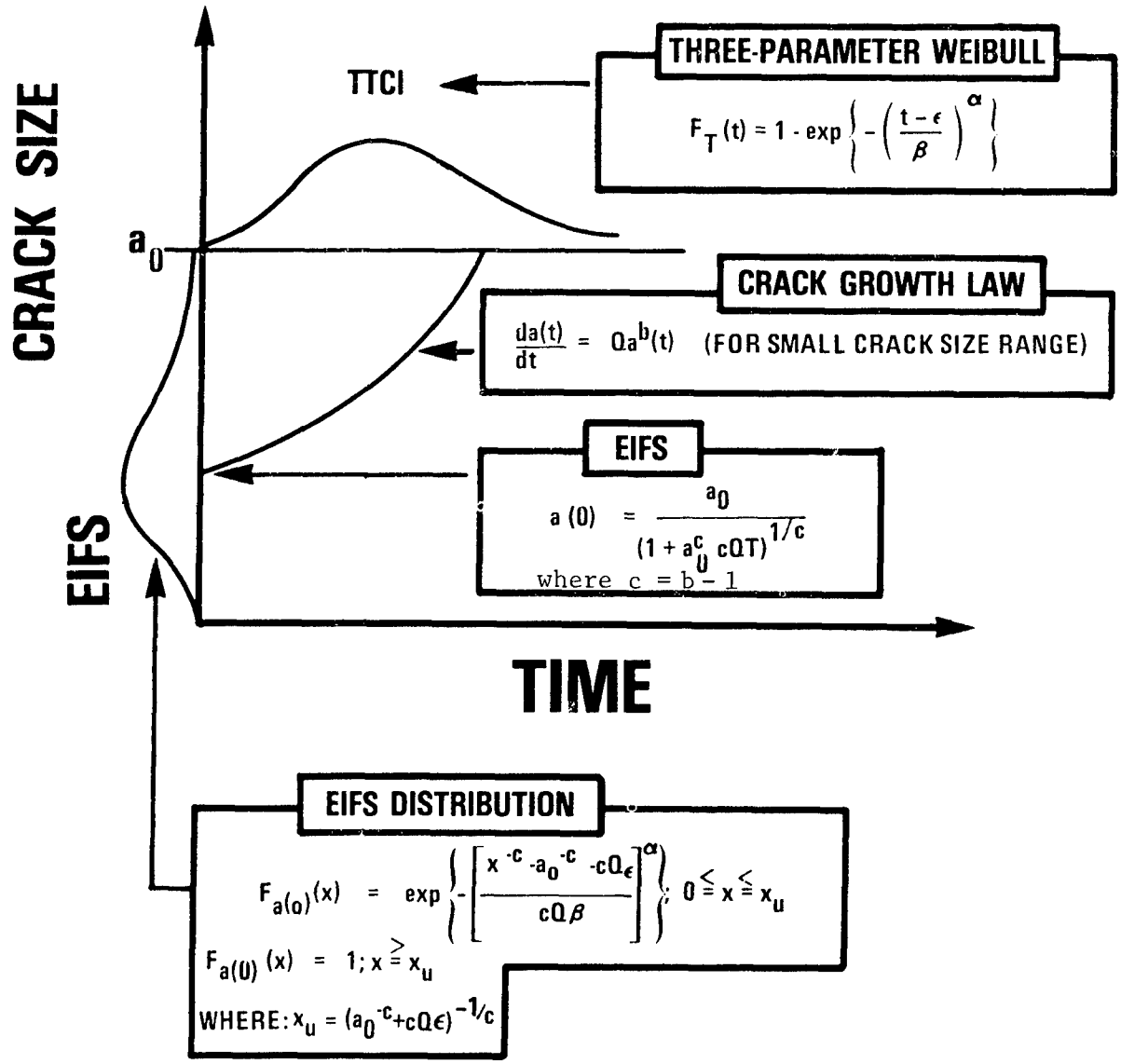


Figure 27 Elements of the EIFS Distribution Derivation

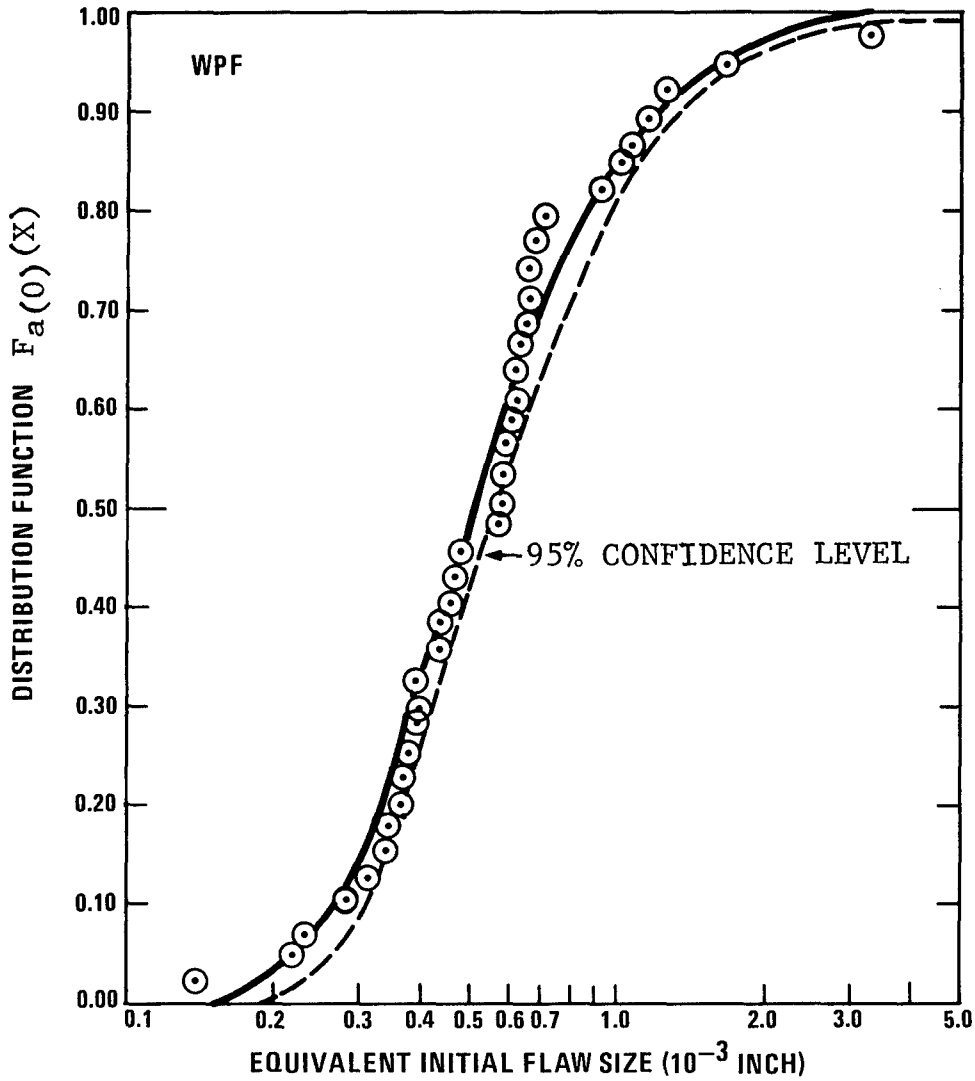


Figure 28 Cumulative Probability Versus EIFS for the Weibull-Compatible Distribution, WPF Data Set from Fastener Hole Quality Program (Ref. 4)

5.3.1 Crack Growth Curve

It is necessary to obtain a crack growth curve covering the entire range of crack sizes for which analytical predictions are desired. For crack sizes less than about .030 inch, Eq. 5 can be used with Q and b determined empirically. Because of changes in crack geometry, effective stress intensity factor changes, and possibly changes in material properties, Eq. 5 does not hold for all crack sizes.

At larger crack sizes, the crack growth curve can be found using LEFM concepts and a cycle-by-cycle integration procedure. That is, the crack length at any time, t, can be found from the initial crack length and integration of an equation of the form

$$\frac{da}{dN} = f(\Delta K, R). \tag{10}$$

Integration of Eq. 10 can be conveniently carried out using one of several available computer programs, such as the CGR program at General Dynamics.

Fractographic examination of coupon specimens can be used to reveal the actual crack growth rates with no need to rely on a possibly erroneous analytical prediction. In any event, each user of the methodology can use the crack growth curve which he feels is most accurate, whether it be derived analytically, experimentally, or a combination of the two.

5.3.2 "Master" Crack Growth Curve

Integration of Eq. 10 leads to a crack growth expression

$$a\ (t_2) = a(t_1) + \sum_j \Delta a\ (t_j) \tag{11}$$

in which $\Delta a(t_j)$ is the crack growth increment per flight hour between times t_1 and t_2. For a small time interval, the relation between $a(t_1)$ and $a(t_2)$ above will depend on t_1 and t_2. For example, the crack growth occurring in a single hour depends on whether the mission being flown is severe or relatively mild with respect to crack growth. Over somewhat longer time periods there will be many <u>repeated</u> missions and it appears reasonable to assume then that the relation between $a(t_1)$ and $a(t_2)$ will depend only on the time interval, t_2-t_1. Making this assumption simplifies the analysis since a single crack growth curve can be used for a crack, no matter what its initial length. With a little rearrangement, Eq. 11 then becomes

$$a(t_1) = W[a(t_2),\ t_2 - t_1)] \tag{12}$$

in which W is a general function representing the crack growth curve.

If several coupon specimens are manufactured and tested identically, it is commonly observed that there is some variation in the crack growth curve from specimen to specimen [7, 8]. This may be important for safety analyses, where a single fast-growing crack can control the structural integrity. However, for purposes of determining the overall economic life of a fleet of aircraft, or even a single component composed of many structural details, the average crack growth rate would seem to be a better measure of structural degradation. Accordingly, in the present development of the durability analysis methodology, differences in individual specimen crack growth rates are ignored. A single crack growth curve, represented by Eq. 12 , is used for all members of a crack population. This single curve is herein referred to as the "master curve."

It is worth noting that this procedure is consistent with the way the EIFS is defined from the TTCI distribution. The master curve is used in transforming the TTCI observations to EIFS values. It is to be emphasized that the methodology developed is quite general and allows the user to select any form of Eq. 12 desired. There is no requirement to use Eq. 5, LEFM concepts, or any particular techniques to obtain the master curve. Justification of the application of a single master curve is demonstrated with test results in Vol. V [19].

A preliminary examination of whether or not the predicted crack population matches the observed population has been performed using two data sets from the Fastener Hole Quality program [4]. Crack size versus cumulative distribution of crack size plots are shown in Figs. 29 and 30 for the WPF (no load transfer) and the XWPF (15% load transfer) data sets. The predicted cumulative distribution of EIFS is compared with the ranked EIFS predictions. Also, the ranked observed crack sizes are compared with the predicted cumulative distribution of crack sizes at two different times for each data set. Two preliminary conclusions can be drawn from these plots. First, a simple crack growth curve can be used to grow the EIFS population and the predicted crack populations correlate well with observed crack populations. Second, the derived EIFS population is justified because the EIFS distribution is statistically compatible with the TTCI distribution and the ranked observed crack sizes. These results give credibility to the derived EIFS distribution. Further evaluations (e.g., like Figs. 29 and 30) are required for different materials, spectrums, stress levels, etc. This work will be performed during Phase II of the program.

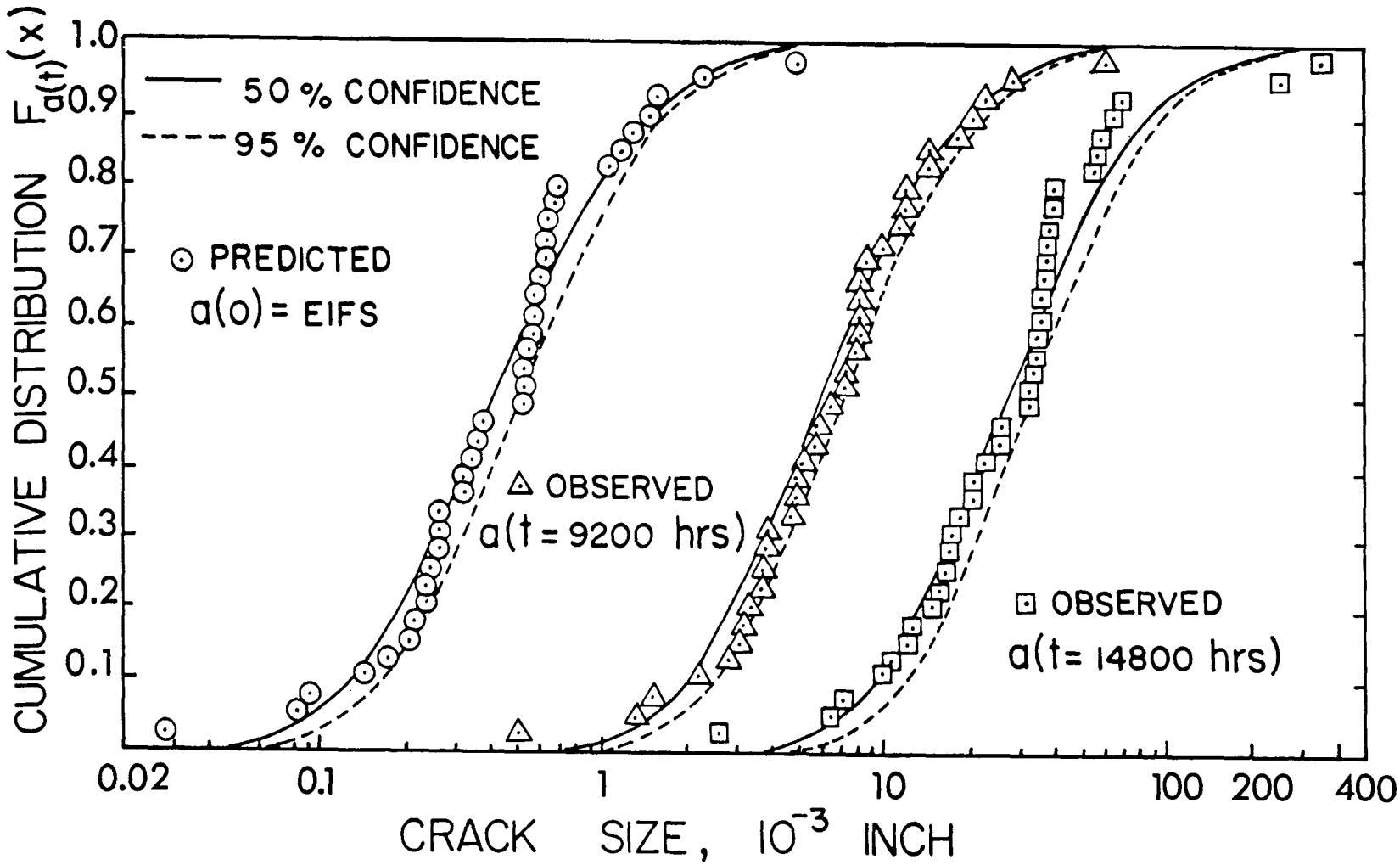


Figure 29 Crack Size Versus Cumulative Distribution $F_{a(t)}(x)$ For WPF Data Set

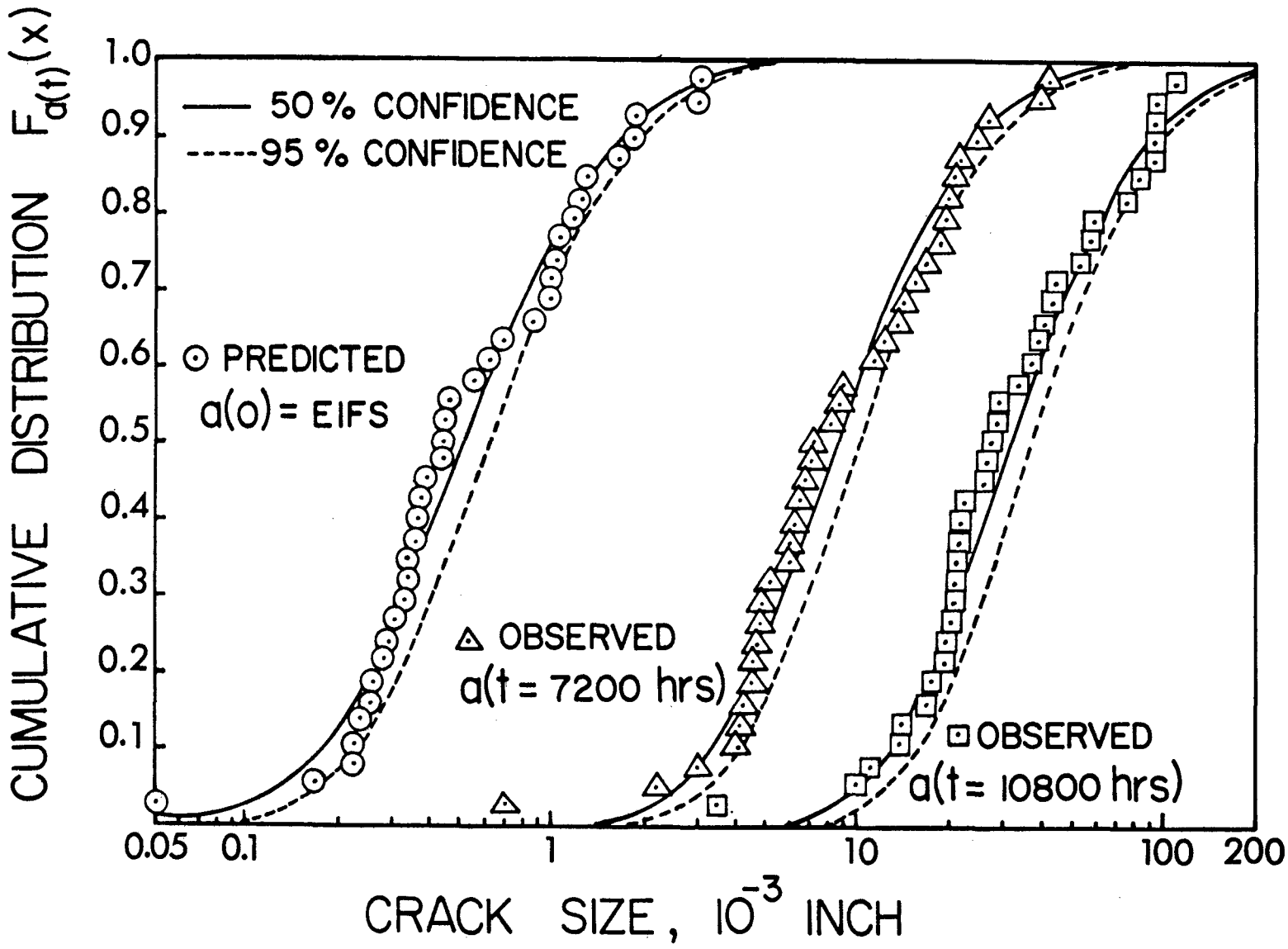


Figure 30 Crack Size Versus Cumulative Distribution $F_{a(t)}(x)$ For XWPF Data Set

5.4 CRACK EXCEEDANCE CALCULATIONS

The key quantity obtained from the durability analysis is the probability of exceeding a specified crack size. This quantity, along with the total number of structural details, can be used to estimate the number of crack occurences. Ultimately, the cost of repair or replacement can be calculated.

The probability of crack exceedance is directly related to the crack size distribution function. If $F_{a(t)}(x)$ is the cumulative distribution of crack sizes at any time t (flight hours), then the probability of a detail having a crack exceeding size x_1, at time t, is

$$p\ (x_1,\ t) = 1 - F_{a(t)}(x_1). \tag{13}$$

$F_{a(t)}(x)$ can be derived from the EIFS cumulative distribution, $F_{a(0)}(x)$, as follows. Since a single master crack growth curve is used, a flaw a_1 smaller than flaw a_2 at time zero must also be smaller at any other time, t. Crossover of the crack growth curves does not occur. The two distribution functions are therefore related simply by the master curve, and

$$F_{a(t)}(x) = F_{a(0)}\ [y_1\ (t)] \tag{14}$$

where $y_1(t)$ is a value of $a(0)$ corresponding to a value x of a(t).

For example, assume that the crack growth law given by Eq. 5 and the three parameter Weibull distribution fits the observed TTCI distribution. Then $F_{a(0)}(x)$ is given by Eq. 8 and the master curve is simply obtained from Eq. 6 Thus

$$F_{a(t)}(x) = \exp\left\{-\left[\frac{y_1^{-c}(t) - a_0^{-c} - cQ\varepsilon}{cQ\beta}\right]^{\alpha}\right\} \ ; \quad 0 \le y_1(t) \le x_u \tag{15}$$

where

$$= 1; \ y_1(t) \ge x_u$$

$$y_1(t) = \frac{x}{[1 + x^c cQt]^{1/c}} \tag{16}$$

For any crack size x, the probability of crack exceedance is given by Eqs. 13, 14, and 15.

Convenient design curves can be obtained by repeated application of these equations. Examples shown here are for the WPF data set of the Fastener Hole Quality program [4].

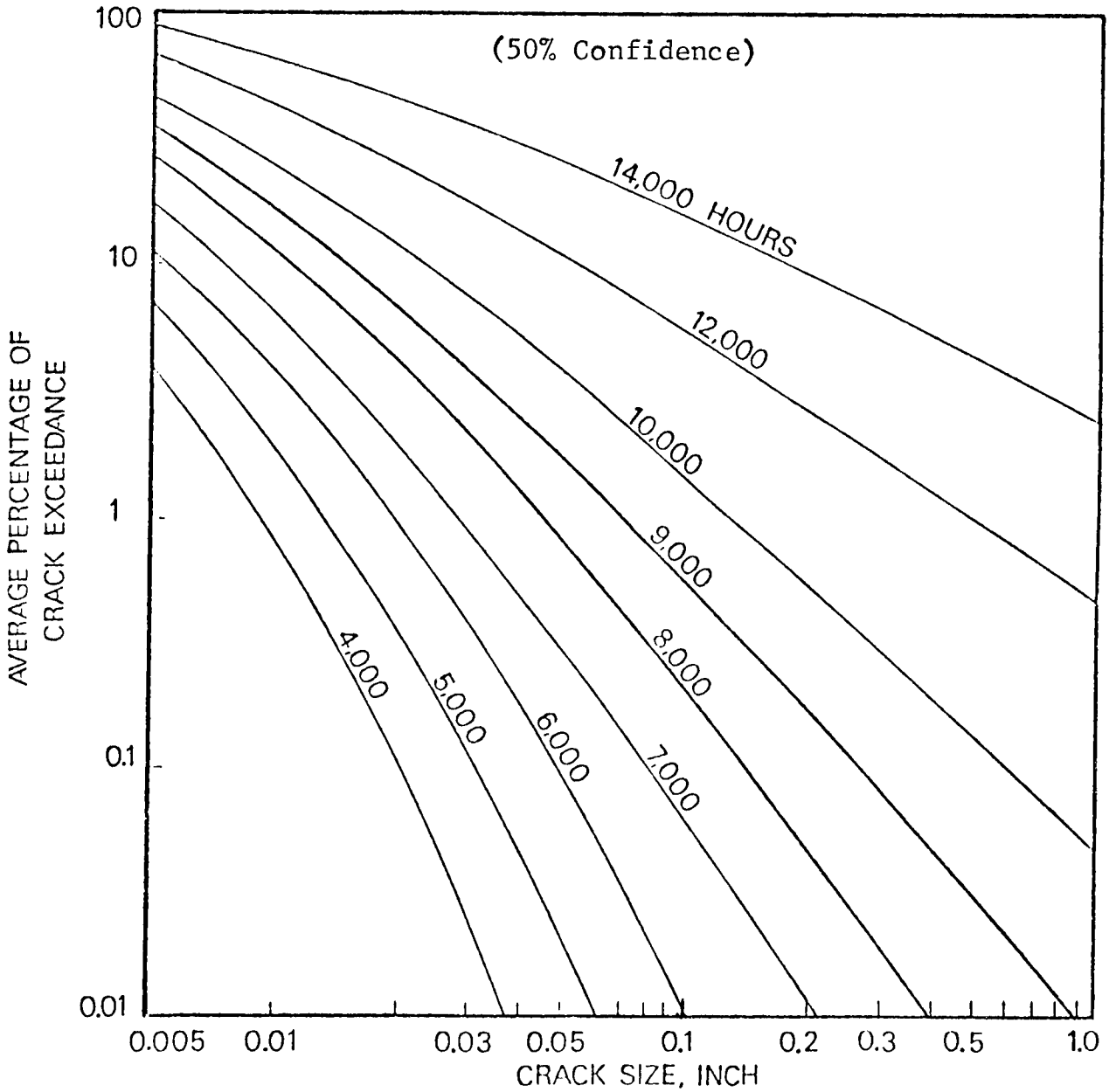
Figure 31 shows one way of plotting the basic information obtained. The abscissa shows various crack sizes while the ordinate represents the percentage of all fastener holes having cracks of length equal to or exceeding those crack sizes. The curves shown are crack exceedance curves for various service times. For example, the point marked by an "x" shows that at 6000 hrs., 0.01% of the fastener holes will have a crack size 0.1 inch or larger.

The average crack exceedance for 50% confidence and for 95% confidence are shown in Figs. 31 and 32, respectively. For the 95% confidence case, it takes 5800 hours on the average before 0.01% of the holes have cracks $\geq$0.1 inch. The 95% confidence level is based upon how well the distribution parameter β can be estimated. Methods to determine the confidence interval are included in Vol. V [19].

An alternate way of presenting the same information is shown in Figure 33. Here, crack exceedance percentage is plotted against service time. To eliminate confusion a curve for only one crack size (0.03 inch) is shown. (This represents the information lying on a vertical line through 0.03 in Figure 31.) The solid curve marked "50% probability" corresponds to the average crack exceedance. Thus, point A indicates that on the average, 5% of the holes will have cracks $\geq$0.03 inches at 9200 hours with 95% confidence. There is a 10% chance that as many as 8% of the holes would have cracks 0.03 inch or longer with 95% confidence. There is a 5% chance that about 8 1/3% of the holes could have cracks at least that large after 9200 hours with 95% confidence. It is observed that the percentage of cracks exceeding the economic repair crack size increases rapidly after a certain service life, thus determining the possible economic life.

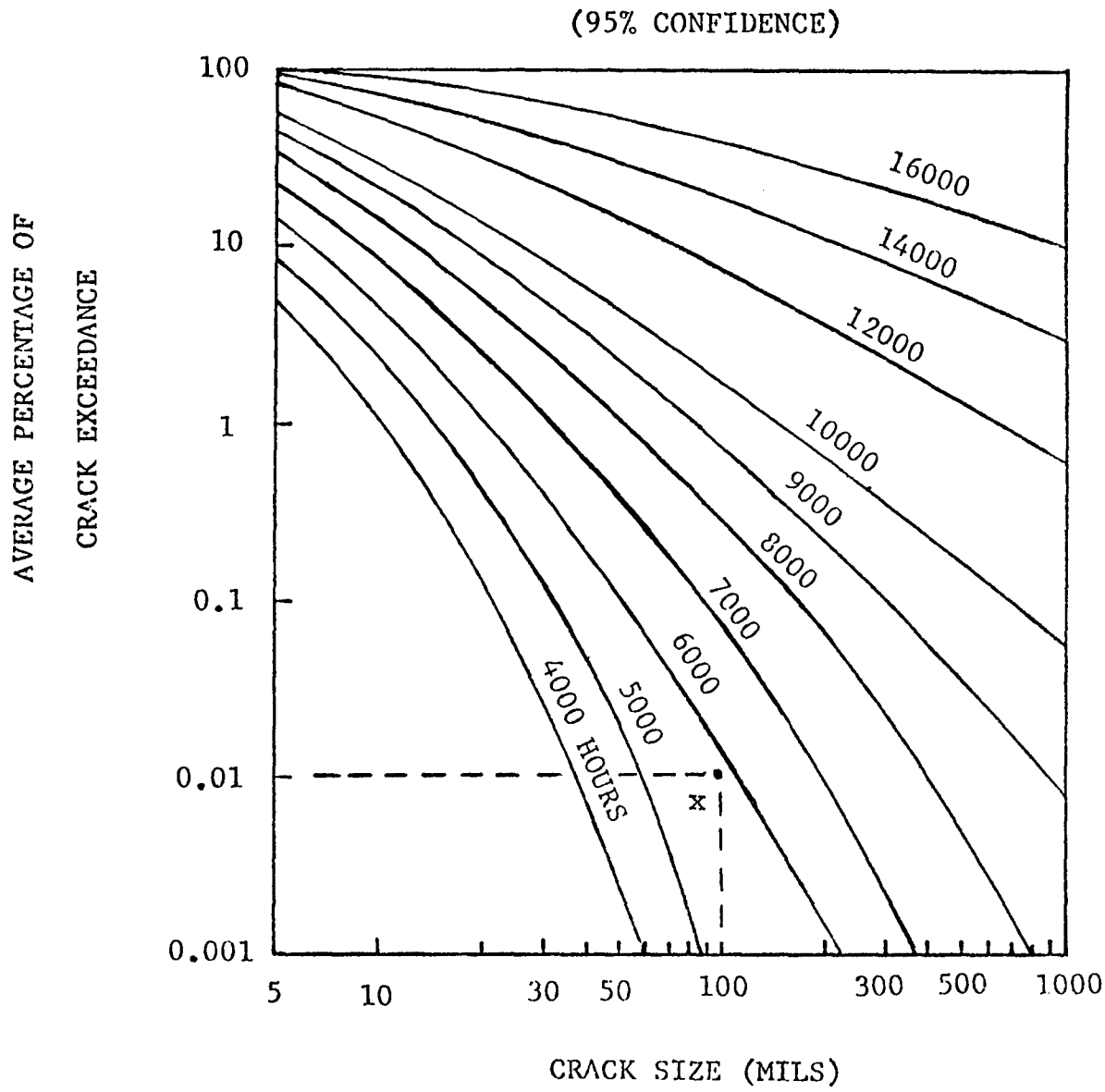
5.5 OTHER CAPABILITIES

Elements of the durability analysis methodology and example calculations included in this chapter represent a simple case of the overall durability methodology developed during Phase I. Important additional capabilities will be briefly summarized here.



Data for WPF data using F-16 Wing Root Bending Moment Random Spectrum with σ_{max} = 34 ksi

Figure 31 Crack Exceedance Curves for 50% Confidence Level



Data for WPF Data Using F-16 Wing Root Moment Bending Spectrum with σ_{max} = 34 ksi

Figure 32 Crack Exceedance Curves for 95% Confidence Level

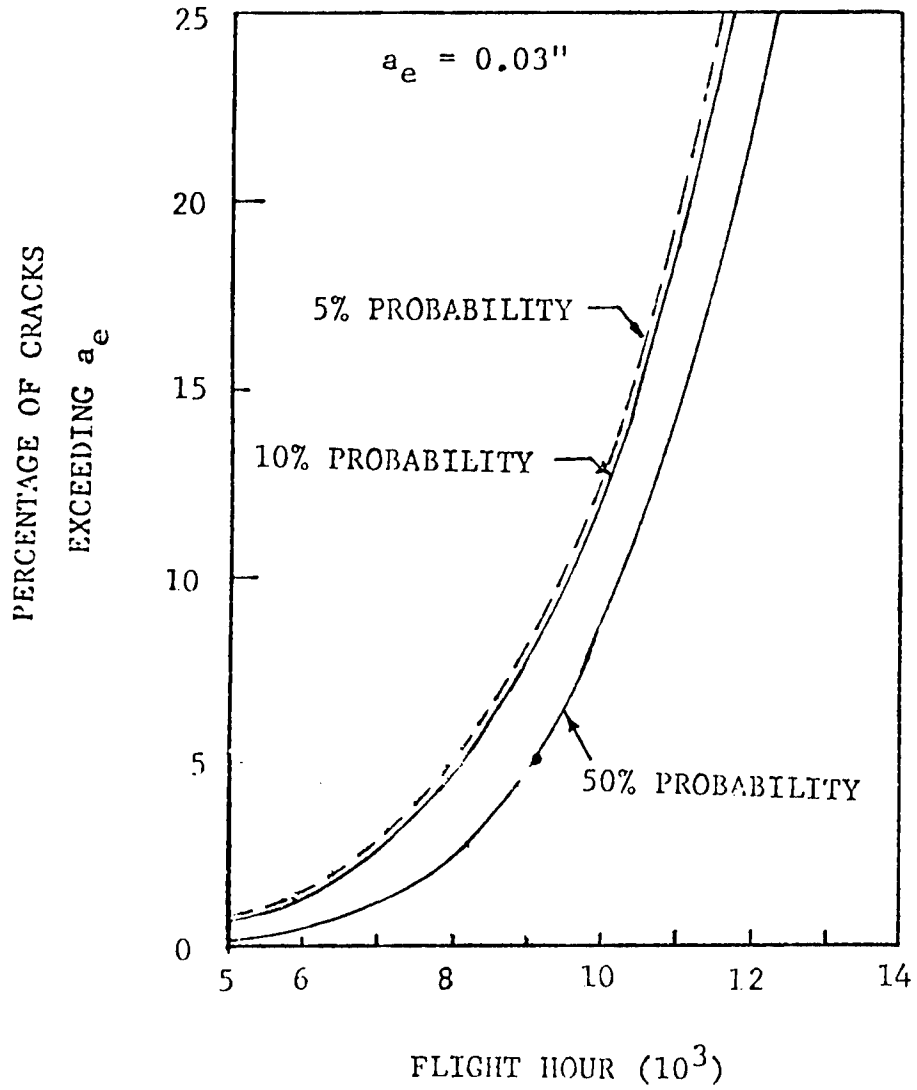


Figure 33 Crack Exceedance Versus Service Time

In a typical preliminary design problem, there will be a non-uniform stress distribution in a component containing a large number of structural details. Structural details can be grouped into a few regions so that details in each region are subjected to approximately the same stress spectrum. Each region can then be analyzed using a fairly simple procedure. However, the different regions may have significantly varying stress levels. Methods have been developed to compute overall component reliability by combining the results of several regions for economic life evaluation. Binomial statistics can be used to combine the expected number of cracks for different regions, parts, etc., if cracks can be assumed to be statistically independent (i.e. the crack growth from each fastener hole or other detail is independent of any other crack growth) [19]. This is probably a good assumption for the small cracks likely to control the durability of the structure. Dependent crack growth will be investigated during Phase II, so that large, safety-related cracks can also be considered.

Although the primary emphasis in this program is on a methodology for quantifying economic life of metallic airframes at the design level, the methodology is also responsive to fleet management. The effects of inspection and repair have been incorporated into the methodology during Phase I. The user can input a suitable function to represent the probability of being able to detect flaws of any size. Inspections can be performed (analytically) at any time. Multiple inspections can be considered. After each inspection, it is assumed that the portion of structural details found to be defective is repaired.

Using the same example data as in Section 5.4, the effect of inspection and repair is shown in Fig. 34. The probability-of-detection function used assumes any crack larger than 0.1 inch is always found and any crack smaller than 0.01 inch is always missed in the inspection. Referring to Fig. 34, assume that an inspection and repair is performed when the predicted average exceedance occurrence for an 0.03 inch crack reaches 5%. The average crack exceedance will follow curve 1 until point A is reached. The repaired crack exceedance is lower and follows curve 2 until point B is reached. A second repair reduces the crack incidence again, curve 3 is now followed, etc. In this case, the first repair allows for 1400 hours of service until a 5% crack exceedance is again reached.

Details of the durability analysis methodology are described in Vol. V [19]. Some features of the methodology have been omitted here for the sake of brevity. Analytical procedures have been developed for assessing the effect of inspection and repair. Although these procedures are not presented

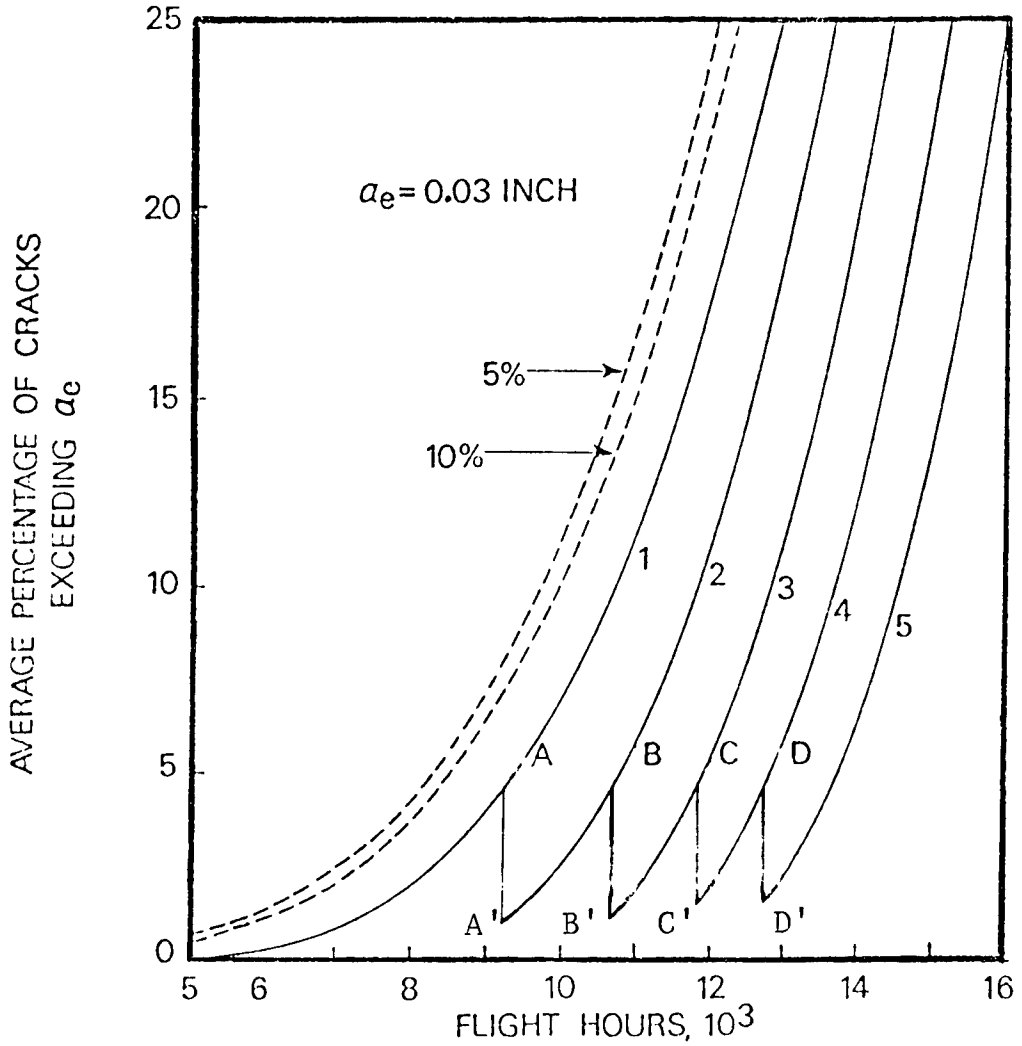


Figure 34 Crack Exceedance Versus Service Time Showing the Effect of Repairs

in this report, this does not diminish their importance or usefulness. Readers desiring a complete appreciation of the durability methodology developed during Phase I should refer to Volumes II, IV and V [8, 19, 40].

5.6 CONCLUSIONS

A statistical method for predicting the economic life of durability critical components of aircraft structures has been developed. It is shown that the number of cracks exceeding the economic repair crack size increases rapidly after a certain service life thus determining the possible economic life. The methodology can be used during design to ensure aircraft durability. Inspection and repair maintenance procedures can also be analyzed. These, along with maintenance costs, have potential application to fleet management and optimization of life cycle cost.

SECTION VI

CONCLUSIONS AND RECOMMENDATIONS

This report summarizes the work performed for Phase I of this three phase program which consists of three tasks. In Task I, a durability structural and analytical S.O.A. assessment was conducted. In Task II, a durability design handbook outline was completed. In Task III, an analytical durability analysis methodology was developed. Within this task, durability critical parts criteria and economic life criteria and guidelines were developed. The initial quality representation, either by the TTCI or the EIFS method, with the required statistical and confidence factors has also been developed. And finally, an analytical durability analysis methodology was developed by using both the EIFS and TTCI methods as initial quality representations and by using two different crack growth models.

All the analytical development of Task III was based on the EIFS and TTCI data generated by AFFDL's "Fastener Hole Quality" program [4].

CONCLUSIONS-TASK I

1. Five ALC's were visited and durability related data were collected for nine types of fighters, three trainers, three bombers, four cargo transports, and two attack aircrafts.

2. For the aircraft surveyed, it appears that cracking is the most frequently reported durability related problem with corrosion a close second and fastener related problems third.

3. Costs of repair and maintenance are not available for the different categories of durability related problems, such as cracking, for all the aircraft types surveyed.

4. A uniform format to record the durability related problems generally does not exist at the ALC's.

5. CFA, DCGA, and PCGA are presently available for design analysis. The DCGA is being used for the more recent aircraft, such as the F-16, and the PCGA is in the development stage.

6. The CFA cannot be used for economic life prediction; the DCGA can be used for economic life prediction if economic life is assumed to be reached when a fastener hole crack reaches a specified size at the end of a specified service period. The PCGA when fully developed, will be the most applicable method to perform economic life prediction.

RECOMMENDATIONS-TASK I

1. The five ALC's should be visited on a regularly scheduled basis to gather durability-related data, particularly for aircraft stationed in extreme environments.

2. A data bank should be developed to store, analyze, and retrieve durability-related data as "lessons learned" to benefit future aircraft design and usage, and to assist the ALCs to better maintain, schedule, and estimate cost-of-aircraft maintenance and repairs.

3. A format to record durability related problems should be developed and furnished to the ALCs for more uniform record keeping.

CONCLUSIONS-TASK II

The proposed outline of the durability design handbook is given in the Appendix of this report.

RECOMMENDATIONS-TASK II

This outline should be revised at the completion of Phase II of this program, if necessary to update this outline before the handbook is written.

CONCLUSIONS - TASK III

1. Economic life should be treated as a design goal rather than a fixed quantity for contractual compliance. Economic life criteria are general guidelines for evaluating economic life tradeoffs (e.g., design, life-cycle costs, weight, operational readiness, test requirements, maintenance requirements, etc.) prior to air-

craft service. A philosophy is recommended, based on predicted crack exceedances (fastener holes or through the thickness cracks causing fuel leaks), for addressing economic life issues and tradeoffs (Section 3.4). Repair costs are a function of the number of cracks exceeding specified repair limits. A probabilistic approach for evaluating economic life is recommended.

2. A durability critical parts criterion is presented in Section 3.5.

3. Initial quality representation can be characterized by either the TTCI or by the EIFS. The Weibull Compatible distribution gave a better fit than any of the other distributions considered for characterizing EIFS for the three "Fastener Hole Quality" [4] data sets (WFP, XQPF, and XWPF). The Weibull distribution was second best and the only other distribution considered acceptable for characterizing EIFS for all three data sets.

4. The Weibull Compatible distribution for EIFS is physically related to the fatigue wear-out process. The EIFS values are a function of the TTCI values and vice versa.

5. A durability analysis methodology is proposed in Section V for evaluating economic life based on crack exceedances. This general methodology is based on the TTCI and the EIFS concept. The proposed method will be further developed and evaluated during Phase II of this program.

6. Probabilistic and statistical principles are valuable for supporting durability analyses and for interpreting results. Binomial statistics are convenient for assessing economic life tradeoff options at desired complexity levels.

7. Fatigue and crack growth processes are very complex. The processes need to be better understood in terms of the durability design variables. Until this understanding has been reached, parts, components, and structure will have to be tested using realistic load spectra.

8. Economic life analyses are essential for evaluating airframe tradeoffs during the design stages and prior to service. Such analyses will increase the time and cost for evaluating designs. However,

the potential payoffs are substantial:

- improved confidence in the design prior to testing and service
- reduced life-cycle costs
- less down time for aircraft in service (improve operational readiness)
- simplify force management requirements
- improved reliability in service
- provide the procuring agency better information for assessing their tradeoff options before the airplane is committed to production and service.

RECOMMENDATIONS-TASK III

1. The proposed durability analysis methodology should be extended to cover damage tolerance size cracks. Durability and damage tolerance analyses should be incorporated into a unified methodology so that both requirements can be efficiently handled during the design and evaluation processes.

2. Procedures need to be further developed and illustrated for evaluating typical economic life tradeoffs.

3. EIFS values are currently characterized using fractography results for a given material, operating stress level, spectrum, manufacturing processes, etc. Is EIFS a generic material property? How sensitive are the economic life predictions to the EIFS distribution used? These questions need addressing and they will be studied during Phase II of this program.

4. Extend the proposed durability methodology to account for environmental effects and study impact on economic life predictions.

5. Criteria should be developed for defining economic repair limits for cracks in radii, cutouts, typical stress risers, etc. Such criteria should be incorporated into the proposed durability methodology of Section V.

6. The proposed durability methodology should be further developed and refined for evaluating depot maintenance requirements and aircraft user options for in-service aircraft. The methodology should account for considerations such as: safety, operational readiness, operational risks, cost, inspection/maintenance requirements, etc.

7. A Structural Maintenance Handbook should be developed for evaluat-

ing fleet management requirements and tradeoffs options. Guidelines and implementing procedures should be fully developed and proven.

A P P E N D I X

DURABILITY DESIGN HANDBOOK OUTLINE

A durability handbook is essential. It will briefly review specifications and contractual requirements for durability. Also, it will give guidelines and recommend suitable analytical procedures for satisfying the U.S. Air Force's durability and structural integrity requirements at the design level. The durability handbook will include practical information for structural design and for evaluating economic life tradeoff options for aircraft prior to service. Testing requirements and procedures for showing compliance with durability requirements will be described. Examples will be presented to aid the handbook user.

The information will be general in nature. A "loose-leaf" format will be used to facilitate updating of the handbook.

A proposed outline for the durability design handbook is presented on the following pages. This outline is considered preliminary. It will be updated as necessary during the course of the durability program to comply with new understandings and requirements.

DURABILITY DESIGN HANDBOOK OUTLINE

I. INTRODUCTION

- 1.1 Durability and Economic Life Considerations
- 1.2 Durability Analysis Objectives

II. DURABILITY REQUIREMENTS AND CRITERIA

- 2.1 U. S. Air Force Requirements
- 2.2 Critical Parts Criteria
- 2.3 Economic Life Criteria

III. DURABILITY ANALYSIS METHODS

- 3.1 Conventional Fatigue Analysis
- 3.2 Deterministic Crack Growth Approach
- 3.3 Probabilistic Crack Growth Approach
- 3.4 Recommended Applications

IV. ECONOMIC LIFE ANALYSIS AND GUIDELINES

- 4.1 Analytical Methodology
- 4.2 Design and Testing Guidelines
- 4.3 Example Applications

V. DURABILITY CONTROL PLAN

VI. MISCELLANEOUS DURABILITY CONSIDERATIONS

- 6.1 Stress Corrosion Cracking
- 6.2 Wear Endurance
- 6.3 Quality Assurance
- 6.4 Materials and Processes

APPENDICES

- I - List of Symbols
- II - Definitions
- III - Computer Programs for Implementing Durability Analysis
- IV - Durability Design Data

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