

**CURRENT PRACTICE ON ESTIMATING
CRACK GROWTH DAMAGE ACCUMULATION
WITH SPECIFIC APPLICATION TO STRUCTURAL
SAFETY DURABILITY AND RELIABILITY**

*STRUCTURAL INTEGRITY BRANCH
STRUCTURES DIVISION*

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AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio 45433

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Howard A. Wood

HOWARD A. WOOD
Principal Scientist
Structural Integrity Branch

R. M. Bader

ROBERT M. BADER, Chief
Structural Integrity Branch
Structures Division

FOR THE COMMANDER:

G. G. Leigh

G.G. LEIGH, Lt Colonel, USAF
Chief, Structures Division
Air Force Flight Dynamics Laboratory

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20. ABSTRACT (Cont'd)

(geometry), the material parameters (basic crack growth rate) and loading (usage patterns). Examples are cited to give indication of confidence in making life predictions. A summary of procedures for estimating service loads and chemical-thermal environments is included as they relate to simulating usage effects in the laboratory.

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FOREWORD

This report introduces the topic of predicting crack growth damage accumulation in metallic airframe structures. Although current, the data in this report is by no means complete. Discussions are intentionally brief and concise, emphasizing current practices of calculating growth rates and establishing confidence in the predictions.

For the most part the analyses, test results, etc., synopsized in the report were developed in-house by the authors. Guidance and relevance for the procedures have been established through close association with ASD and AFLC, and by active participation of the authors in solving USAF structural problems and in preparing USAF policy.

Reference is made to current USAF policy specification. Certain requirements are cited to illustrate the impact of damage growth technology on satisfying these specification requirements. The reader is encouraged to read MIL-STD-1530 for a complete description of USAF policy on Structural Integrity.

References are deliberately excluded from the body of the report. A bibliography is included for those who desire further knowledge on the subject.

Any additional information concerning this topic can be obtained from the authors by calling (513) 255-6104 (Autovon 785-6104).

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LIST OF SYMBOLS

a	crack size/length
a_{cr}	critical crack size/length
a_i	initial crack size/length
K	stress intensity factor
K_C	fracture toughness
ΔK	stress intensity factor range
K_{max}	maximum stress intensity factor
K_{min}	minimum stress intensity factor
P_{xx}	residual strength (load)
N	flights, blocks, hours, cycles
N_e	economic life
N_z	load factor
R	stress ratio
β	geometric correction factor
σ	stress
σ_{res}	residual strength (stress)

DEFINITION OF TERMS

1. Deterministic Analysis Methods/Approaches - Methods which predict life, level of damage (i.e., crack size) by considering all input data as discrete items. For a given set of data the prediction is a single value.
2. Probabilistic Analysis Methods/Approaches - Methods which predict distributions of lives or levels of damage (i.e., crack size population) by considering the statistical nature of one or more of the input variables. For a given set of data the result is presented in terms of probability of equaling or exceeding a given value.
3. Reliability (Structural) - The probability that a structure will perform its specified mission without failure when subjected to loads or other adverse environments.
4. Risk (Structural) - The probability that a structure will not perform its specified mission without failure when subjected to loads or other adverse environments.
5. Safety - The assurance that safety of flight structure of each aircraft will achieve and maintain a specified residual strength level (in the presence of undetected damage) throughout the anticipated service life.
6. Durability - The assurance that the fleet can operate effectively with a minimum of structural maintenance, inspection and downtime, costly retrofit, repair and replacement of major structure due to the degrading influence of general cracking, corrosion, wear, etc.
7. Damage - Flaws, cracks, voids, delaminations, etc. which may be present in structures as a result of manufacturing operations or service. In this report damage is considered as a sharp crack.
8. Damage Tolerance - The ability of a structure to successfully contain damage over a specified life increment without adversely affecting safety of flight.
9. Residual Strength (Required) - The minimum internal member load, P_{xx} , which the structure is required to sustain with damage present without endangering safety of flight.
10. Life (Fleet) Management - The actions required to maintain safety and durability throughout the service life of the fleet.

SECTION I

USAF PRACTICE SAFETY, DURABILITY, AND RELIABILITY

1. SUMMARY

Current USAF requirements for structural safety and durability are described in MIL-STD-1530A* and in the associated Military Specification system. Both safety and durability requirements can be satisfied by considering initial damage in the structure and assuring that the damage does not grow and reach specific limits in prescribed time intervals. Safety is associated with the extreme (i.e., the worst case) damage that could likely be missed during manufacturing inspection. This damage must not grow and degrade the strength of the structure below specified limits throughout the service life. Durability limits are associated with the economics of in-service repair, i.e., initial damage typical of common manufacturing operations is not allowed to grow to a size sufficient to require extensive rework or repair in less than one design service lifetime.

Current practice involves the use of deterministic methods to predict crack growth damage accumulation. Engineering experience with the use of these methods has allowed the proper emphasis to be placed on design decisions affecting such factors as allowable stress. Most of the input data for the deterministic methods are available in statistical format (e.g., initial flaw sizes) and thus it is

*in final coordination with Aerospace Industry Association (AIA).

possible to make estimates of the probability of damage growth exceeding a specified level. Both the deterministic and probabilistic approaches require a model of the damage accumulation process. Current practices described in this report are applicable to both approaches. Lack of essential data for important variables and inexperience with the use of reliability methods for making conventional design decisions have been the chief reasons for the absence of the use of these techniques in current practice.

Reliability techniques have been more effective in the assessment of older fleets where test results and in-service experience are available to establish the mean time to failure. USAF experience of reliability/risk analysis of older systems (e.g., F-111, B-52, C-5A) has been successful in illustrating the relative risks of operating the fleets under variations in usage, etc. The results have been informative and have established a level of confidence for operational and inspection times computed by deterministic approaches.

2. CURRENT POLICY

a. Safety and Durability

USAF policies and procedures for certification of flight structures are described in detail in MIL-STD-1530(A) and the associated Military Specification system. The current discussion will concentrate on the two major segments of the policy which deal with structural

damage, damage growth and damage containment. These are the criteria for:

- (a) Safety - The assurance that safety of flight structure of each aircraft will achieve and maintain a specified residual strength level (in the presence of undetected damage) throughout the anticipated service life.
- (b) Durability - The assurance that the fleet can operate effectively with a minimum of structural maintenance, inspection and downtime, costly retrofit, repair and replacement of major structure due to the degrading influence of general cracking, corrosion, wear, etc.

Safety and durability limits are specified, designs chosen to comply, tests conducted to verify and fleet management procedures established to maintain and/or adjust the limits in service. Each structure must meet both the safety and durability requirements.

b. Damage Growth Concept

Past experience with tests of structures under simulated flight loading has indicated that time to initiation of cracks from most structural details such as sharp corners or holes is relatively short and that the majority of the life (i.e., 95%) is spent growing the resultant cracks to failure. Likewise, analyses of in-service fractures, cracking instances, etc. have indicated that a major source of cracks is the occurrence of initial manufacturing defects such as sharp corners, tool marks and the like. Thus, it is now common practice to

consider the damage accumulation process as entirely crack growth with zero time to initiate the crack. Although this assumption may seem unduly severe, recent studies have shown the approach feasible, of minimal detriment to weight, cost, etc., but most important, the consideration of initial damage in the form of cracks or equivalent damage is absolutely necessary to ensure structural safety. Figure 1 includes a schematic of the damage model associated with the structural safety.

c. Safety Limits/Life-Design and Certification

Safety is ensured by designing to specific damage tolerance requirements in which initial damage is never allowed to grow and reduce the residual static strength of the structure below a prescribed level, P_{xx} , throughout the life of the aircraft. As indicated in Figure 1, P_{xx} is the greater of design limit load or the maximum load that might be encountered during the specified minimum period of unrepaired service usage. If in-service inspections are required to ensure safety (e.g., for fail-safe designs) then the residual strength level, P_{xx} , is the maximum load likely to occur during the inspection interval. For noninspectable structures, P_{xx} is the maximum load likely to occur during the design lifetime. Transport and bomber type aircraft rarely exceed design limit load during service life. In such cases, the maximum value of P_{xx} would be design limit load. Fighters and attack type aircraft frequently exceed limit load and are designed to sustain P_{xx} in excess of design limit. The damage assumed initially, a_i , is that associated with manufacturing inspection capability. For example, the values of a_i specified for slow crack growth structures in MIL-A-83444 reflect 95% confidence that 90% of the flaw sizes greater than

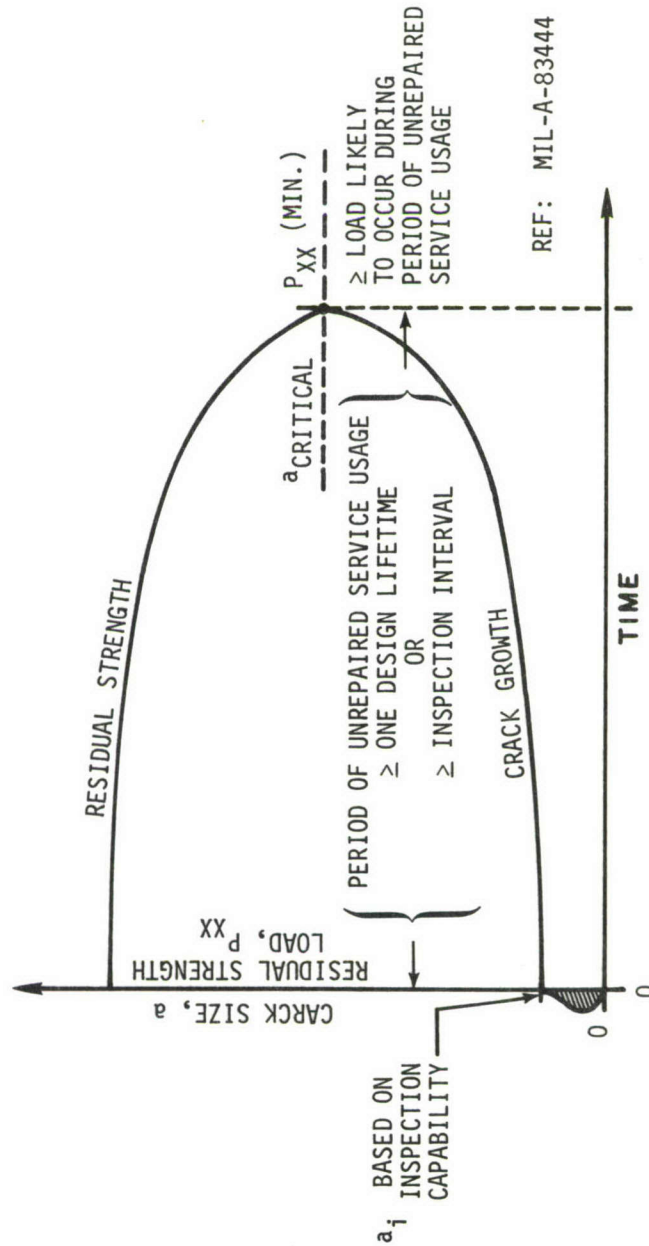


Figure 1. Schematic of Safety Goals for New Structure Based on Crack Growth Damage Accumulation

those specified will be found during inspection. The reader is referred to MIL-A-83444 for further details on the damage tolerance requirements, initial flaw sizes, safety limits, and lives associated with in-service inspection, etc.

Demonstration of compliance with safety requirements, i.e., slow crack growth rates and residual strength, is accomplished during the full-scale test phase, either by examining the results of the full-scale fatigue test or by conducting special tests. The first aircraft example of safety verification will be the B-1 bomber, as it represents the first system to be designed and qualified to a damage tolerance specification.

d. Durability Limits/Life-Design and Certification

A new concept, that of structural durability, has been developed and is now being implemented into the design specifications. This concept includes "fatigue" procedures as previously considered, but in addition it includes the requirements for protection against such degrading factors as corrosion, stress corrosion, wear, fretting, etc. To prevent widespread cracking, a flaw growth model is considered as shown in Figure 2.

In Figure 2 the limit, a_e , is not failure as in the safety requirement but is associated with a size of crack that can be repaired. The economic life, N_e , is defined as that time when the population of such cracks in each aircraft just reaches sufficient size that rework or repair is no longer cost-effective. The durability specification

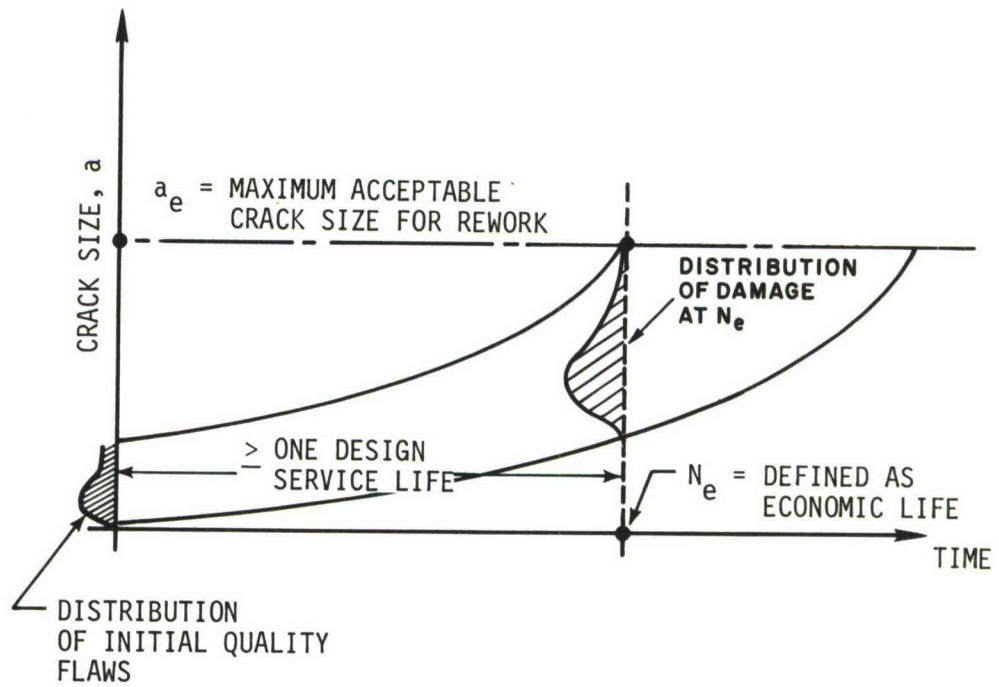


Figure 2. Durability Concept-Flaw Growth Model

requires that the "economic life" be in excess of the design service life. In the example shown, Figure 2, the initial population of flaws represents "apparent initial quality," or the condition of structural details following typical manufacturing operations such as hole drilling, etc. Typically, the economic limit on rework for fastener holes is of the order of 0.060 inch on the diameter, or the next nominal hole size. Thus, it is seen that the limit of growth for durability requirements is significantly smaller than for safety. Typical values of apparent initial defect size are discussed in Section II of this report but are generally in the neighborhood of 0.001 inch - 0.007 inch cracks emanating from the side of a fastener hole.

Durability is associated with total flaw population, or total number of flaws in excess of the economic rework limit for each aircraft in the fleet. Safety is concerned with the "worst case" of flaw size likely to be missed during an inspection. The demonstration of durability or a lower bound on the economic life is one of the major tasks of the full-scale fatigue test. The test article is examined to the extent that decisions can be made concerning the population of flaws at the end of one simulated lifetime. If the design appears to be adequate and no indications of general cracking are present at the end of the two lifetimes, then a decision is made to extend the test. A teardown is required at the completion of the testing program.

e. Safety and Durability/Life Management

In general, it is not possible to anticipate the extent of usage of a fleet of tactical aircraft. Design is accomplished with the best estimate of typical usage and variance in expected usage. This is in contrast to commercial aircraft experience where usage is fairly predictable. Thus, safety and durability limits of some individual fleet aircraft can be expected to be shorter than estimated in design. This places tremendous emphasis on life management and fleet tracking of individual aircraft. In practice, design estimates of safety and durability limits are adjusted to reflect usage severity based on analyses of crack growth damage accumulation conducted on individual aircraft.

3. RELIABILITY/RISK ANALYSES

a. Making Estimates

In current practice designs are chosen to satisfy the specific safety and durability life and strength requirements by performing deterministic damage growth and residual strength analyses. Decisions on materials, structural configurations, allowable stresses, etc., are based upon the results of these analyses. To illustrate the influence of analyses on one of these factors, allowable stress, consider the following. As will be noted in subsequent discussions, materials and structures respond to loading in a predictable manner. Considering response in the sense of life, N_F , the life to failure is directly influenced by the magnitude of the stress, that is the gross stress in the part. The higher the gross stress level, the shorter the life.

Other material and structural characteristics such as strength and stability are also directly related to the design stress level. The size and thickness of tension structures (i.e., lower wing skins) are related linearly to allowable stress, and thus weight is driven by the choice of allowable stress. Although it is true that only one design condition will govern the final stress selection (all others being of lesser influence), it is generally true that the specific requirements associated with crack growth damage accumulation and residual strength will largely contribute to the choice of allowable stress in aircraft tension structure.

Durability of metallic structure is influenced by the local details of joints, splices, fasteners, eccentricities, etc., but, for a given design, the lower the gross stress the more durable the design (in a crack growth-economic sense).

The crack growth damage analysis approaches cited in Section II of this document, while generally deterministic, rely on input data such as initial flaw sizes, material and usage variability, etc. which are available in a statistical format. Thus, in addition to predicting mean values of damage accumulation, the methods can be used to predict the distribution of damage with time. For example, under specified usage and assumed distributions of the other variables it is possible to estimate the distribution of lives, growth rates, final flaw sizes, etc., and thus the probability of equaling or exceeding a certain requirement. If only one variable is considered, the estimation can be rather straightforward. When joint probabilities are involved, the

procedure is more complex. The results of the probabilistic analyses are extremely sensitive to the initial distributions of the variables and the functions used to approximate them. Among the major variables to be considered are:

- material strength
- crack growth rate
- critical crack size
- usage and maximum operational loads
- crack detection capability/access
- inspection techniques
- frequency of inspection
- fleet size

Although published reliability methods can be found in the literature, current practice does not in general reflect their use. This is due to several factors as described in the following. Until recently, most approaches considered damage models based on time to crack initiation. It is anticipated that the inclusion of crack growth damage accumulation models will increase the usefulness of reliability techniques and bring them closer to current policy on safety and durability. The sparsity of data associated with the major variables listed above and the extreme sensitivity of the results to the distribution functions has limited the effectiveness of, and the confidence in, the results. To design to a specified reliability level requires that the acceptable failure rate be established in advance. It is also required that relationships be established between reliability and

normal design decision factors such as allowable stress, structural configuration, etc. Since overall fleet reliability can be significantly altered by factors other than those normally considered in design (e.g., inspection frequency), it would be difficult to assign the proper weighting function to any particular variable. Under current practice using deterministic techniques, experience with the use of crack growth damage methods, etc., has allowed the placement of proper emphasis on design factors such as allowable design stress.

Reliability techniques have been more successful in assessing older fleets where extensive test and service failure data is available to establish the mean time to failure with some degree of confidence.

Studies are now underway to evaluate the potential of reliability techniques in design. Under the concept of durability described in preceding paragraphs, probabilistic techniques may have the most direct impact, since the ultimate decision on the economic life is keyed to the total population of cracks just below the rework limit. Safety requirements, on the other hand, account for growth of the extremes of the initial flaw population and other variables (i.e., the worst case) and it is anticipated that the requirements will continue to be satisfied with conventional practices utilizing deterministic methods.

In summary, crack growth damage accumulation methods are required for both deterministic and probabilistic analyses. Current practices described in this report are applicable to both approaches. Lack of essential data for important variables and inexperience with the

use of reliability methods for making conventional design decisions have been the chief reasons for the absence of these techniques in current practice. As noted in the following, reliability techniques have been more effective in illustrating the effects of varying one or more of the major variables.

b. Examples

In application studies considering hypothetical fleet situations, reliability analyses have yielded results that are not surprising. On a relative basis it has been shown that reliability increases when:

- a. The frequency of inspection is increased, and damage is repaired,
- b. The thoroughness of the inspection improves,
- c. The critical crack size increases relative to the minimum detectable crack size,
- d. The time to achieve critical crack size increases,
- e. Variability in material behavior decreases,
- f. Mission severity or usage decreases.

Analyses have been conducted on several USAF fleets (e.g., F-111, C-5A, B-52) in order to examine the probability of survival, or risk of failure, for certain operational situations by assuming the location and mode of failure, the mean time to failure, and probabilistic models of the major variables. Figures 3 and 4 illustrate results for the F-111 following the cold proof test. Both figures indicate the

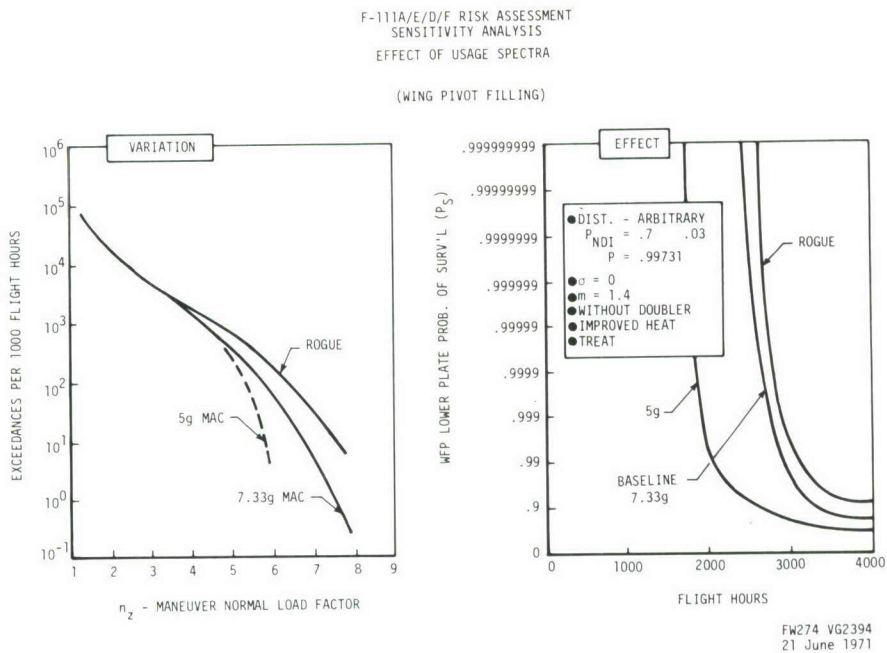


Figure 3. Risk Assessment - Effect of Usage Spectra

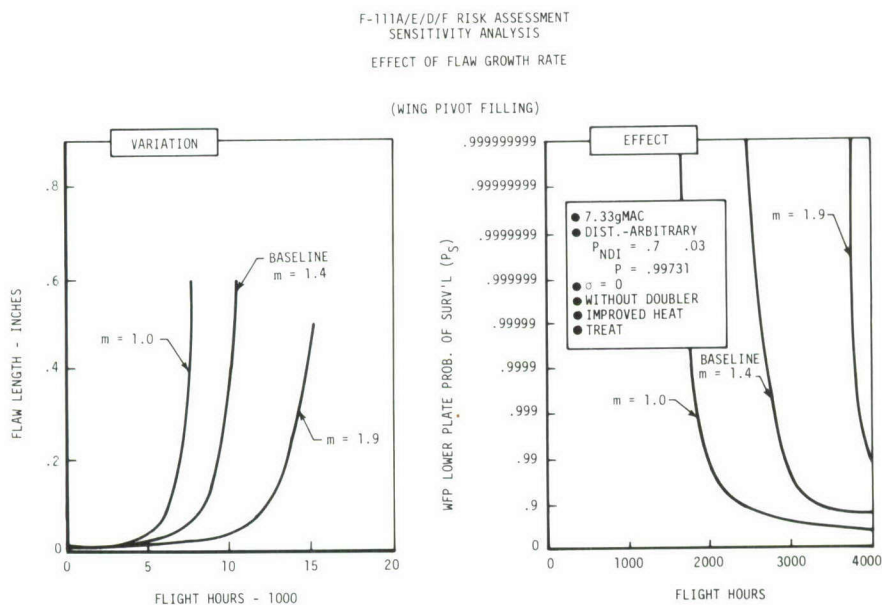


Figure 4. Risk Assessment - Effect of Flaw Growth Rate

significant effects of usage (Figure 3) and of flaw growth rate (Figure 4) on the probability of survival. As shown by Figure 3, increased maneuver loads from the 5g condition to rough usage (Rogue) condition cause a predicted increase in probability of survival for a given life. Figure 4 demonstrates that the fastest crack growth condition ($m = 1.$) gives the lowest probability of survival for a given life.

Sensitivity studies to examine relative risk of operation or the change in risk of failure associated with change in usage have been somewhat influential in deciding on inspection and proof test options for some older fleets. For the F-111, risk assessment placed a level of confidence on the operational limits and inspection intervals computed by deterministic means for the proof tested aircraft. Confidence, in this case, was achieved by comparing projected failure rates with actual in-service rates. Such data has been estimated for fighters and commercial aircraft.

SECTION II

LIFE PREDICTION METHODOLOGY

1. SUMMARY

Currently, with the Air Force, airframe life predictions are based on a crack growth damage integration package that uses a data base and analysis to interrelate the following elements: (a) initial flaw distributions, (b) aircraft usage, (c) basic crack growth material properties, (d) crack/structural properties, (e) damage model, and (f) fracture or life limiting criteria. To support evaluation of the damage integration package, laboratory tests are conducted which simulate the basic features of cracked hardware. Predictions are then compared to measured crack growth behavior. The confidence normally associated with life predictions using a damage integration package is derived from the ability of the package to predict the laboratory generated crack growth behavior. Only when cracking is evident from service inspections can there be the necessary information to verify that the damage integration package is performing satisfactorily. The difficulty of assessing the confidence level associated with the life prediction derived from the damage integration package results from extrapolating the use of the package from a simple data base to the more complex service hardware case.

2. INTRODUCTORY INFORMATION AND BACKGROUND

The only quantifiable measure of fatigue damage is a crack. Cracks impair the load-carrying characteristics of a structure. A crack can be characterized for length and configuration using a structural

parameter initially developed (independently) by Irwin and Williams in 1957. The crack parameter, termed the stress intensity factor, interrelates the local stresses in the region of the crack with (a) crack geometry, (b) structural geometry, and (c) level of load on the structure. In a manner similar to Irwin, who utilized the stress intensity factor for fracture studies, Paris and his colleagues at Lehigh University and at the Boeing Company developed a crack mechanics approach to handle the growing (fatigue) crack problem. The technology was sufficiently evolved in 1969 to make available the basic elements for a life analysis study at the time of the F-111 loss (A/C #94).

The concept of damage tolerance as applied to USAF aircraft structures developed as the result of the F-111 failure. The purpose of the Damage Tolerance or Fracture Control Plan is to prevent failures due to the presence of cracks in safety-of-flight structure. Damage Tolerance design concepts such as outlined in MIL-A-83444 preclude assumptions that safety of flight structure is crack free. Older structures (e.g., C-5A, F-4, and A-7) have been assessed or are currently being assessed by Independent Review Teams (IRT) whose purpose is to perform an airframe structural audit and thereby identify potential cracking problems. Major IRT functions are to: determine evidence of cracking incidences by reviewing field histories, define aircraft usage through use of aircraft load tracking records, establish potential critical part locations, and establish allowable crack growth limits. The results of such investigations directly utilize results of fatigue tests of full-scale articles for assessment of the scope of cracking problems.

This section will detail the fundamentals of life prediction based on crack growth. The crack will be the measure of damage and the crack growth rate will define the rate of damage accumulation. Figure 5 describes the type of information that defines the parameters basic to a life prediction.

The crack grows in response to the cyclic loading applied to the structure. Any crack (a) will grow a given increment (Δa) when subjected to a given number of cycles (ΔN), the rate being measured by $\Delta a/\Delta N$. When the crack length reaches a critical value (a_{cr}), the growth becomes unstable, thereby inducing failure. The life (N_F) is the measure of accumulated cycles required to drive the crack from its initial length a_i to the critical length a_{cr} . The interrelationship between crack length, loading, and residual strength of a structure is illustrated by Figure 6. As shown in Figure 6 the monotonic* increase in crack length is induced by a continuous sequence of cyclic loads. The residual strength (σ_{res}), the load-carrying capacity of the cracked structure, is shown to monotonically decrease with increasing crack length in the following manner:

$$\sigma_{res} = K_C / f(a) \quad (1)$$

where

K_C = material property, termed fracture toughness,
and constant for a specific geometry

*monotonic implies that the rate of change does not change sign.

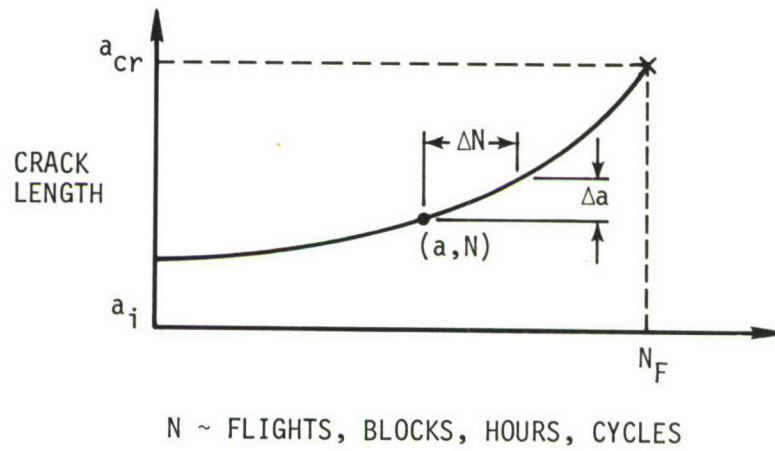


Figure 5. Schematic of Crack Growth Behavior

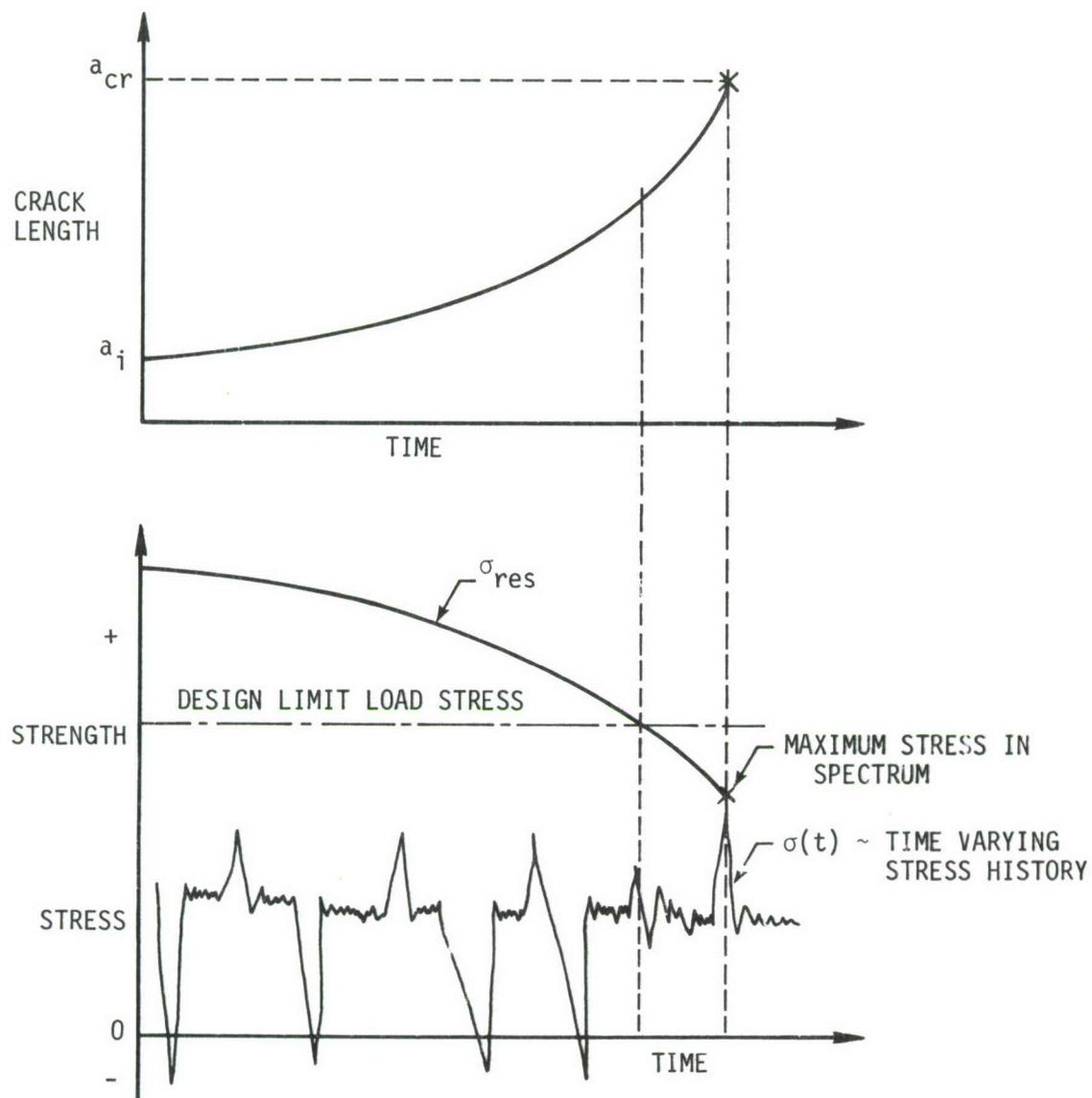


Figure 6. Effect of Crack Damage on Structural Integrity
(Transport type of spectrum is illustrated)

and

$$f(a) = \beta(a) \sqrt{\pi a}, \text{ a structure property, termed the stress intensity factor coefficient}$$

When the residual strength decays to the level of the maximum stress in the service load history, fracture of the structure occurs. The crack length associated with fracture (i.e., a_{cr}) is determined by solving Equation 1 for crack length, assuming that the residual strength equals the maximum spectrum stress or that it equals the design limit load stress level (whichever is greater). Note that the rate of growth of a crack is directly related to the rate of loss of residual strength through Equation 1, thus justifying the selection of the crack to quantify structural fatigue damage.

3. CRACK GROWTH BEHAVIOR/EFFECTS

A crack length a_i will grow to a_{cr} in some life, N_F . Experiments have shown that several parameters affect N_F ; the most important of these being (a) initial crack size, a_i , (b) loading, (c) material properties, (d) structural properties, and (e) critical crack size, a_{cr} . The isolated effect of each parameter on N_F will be discussed in turn. The interrelation of these parameters will be developed in the discussion of the prediction methodology Section II.4.

a. Initial Crack Size

The effect of initial crack size is significant. Given a configuration and loading, the smaller the initial crack size, the longer the life. This is illustrated in Figure 7. The shape of the a vs. N curve for a given configuration and loading remains essentially

constant for any given crack growth increment. Thus, given the crack growth curve from a_{i1} in Figure 8, it is possible to construct the crack growth curve from a_{i2} as shown, where N_i represents the life required to grow from a_{i1} to a_{i2} . Hence $N_{F2} = N_{F1} - N_i$.

b. Loading

The stress history experienced at each location on the aircraft will differ due to changes in bending moment, twisting moment, shear loading, etc., given a particular crack configuration (e.g., a crack growing from a fastener hole on a wing). The loading spectra for a lower surface location is typically more severe than a corresponding upper surface location. The effect on N_F is shown in Figure 9.

c. Material Properties

Experimentally, it has been shown that for the same loading condition (i.e., the same number and amplitude of stress cycles) cracks will grow faster in certain alloys than in others. The crack growth rate ($\Delta a/\Delta N$) can be derived experimentally for each material. Given the same load and geometric conditions, the alloy having the slower growth rate characteristics will have a longer life (N_F) as shown in Figure 10.

d. Structural Properties

The most complex of the parameters affecting crack growth behavior are the structural properties. The structural properties involve such things as crack configuration, load transfer through fasteners, fastener hole size, part thickness, etc. A substantial amount of experimental work has been performed to characterize the geometrical effects on life.

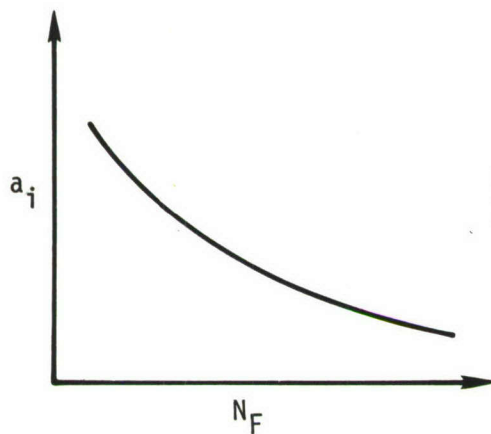


Figure 7. Effect of a_i on N_F

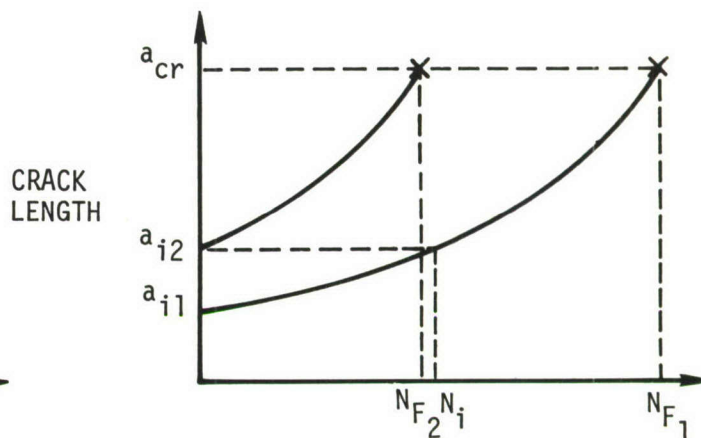


Figure 8. Effect of a_i on Growth

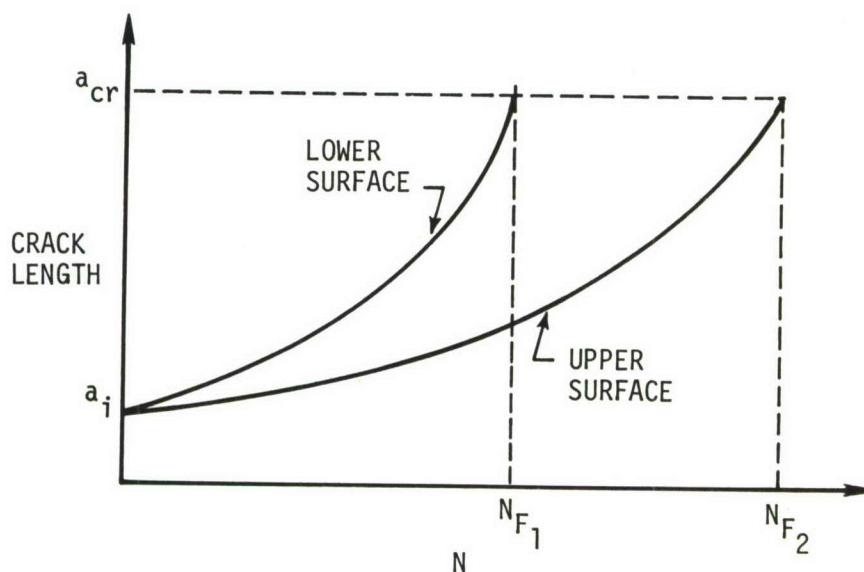


Figure 9. Effect of Spectrum Severity on N_F

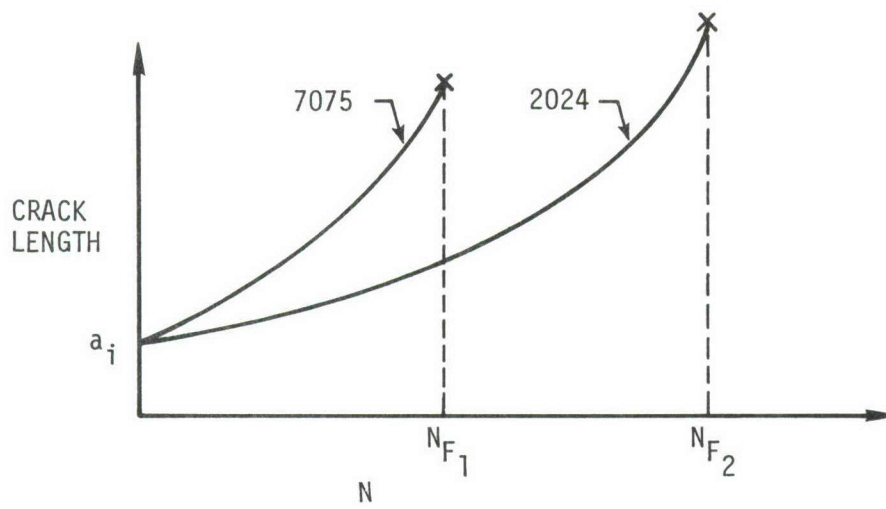


Figure 10. Effect of Material on N_F

The effects of some of these parameters are illustrated in Figure 11.

e. Critical Crack Length

The critical crack length (a_{cr}) is a function of material, structural geometry, and loading. As shown in Figure 12, the relative effect of a_{cr} on N_F is typically small (i.e., when $a_{cr}/a_i \geq 5$). The primary advantage of designing for a large critical crack length is the increased inspectability it provides. A large critical crack length increases the probability of locating the crack before it becomes critical, thereby enhancing aircraft safety.

4. CRACK GROWTH DAMAGE INTEGRATION PACKAGE

Life predictions are based on a crack growth damage integration package which interrelates the following elements:

- a) the initial flaw distribution which accounts for size variations and location of cracks in a given structure;
- b) aircraft usage describing the load spectra data base;
- c) constant amplitude crack growth rate material properties accounting for stress ratio and environmental effects;
- d) crack tip stress intensity factor analyses which account for crack size, shape, and structural interactions;
- e) damage integrator model which assigns a level of crack growth for each applied stress application and accounts for load history interactions; and

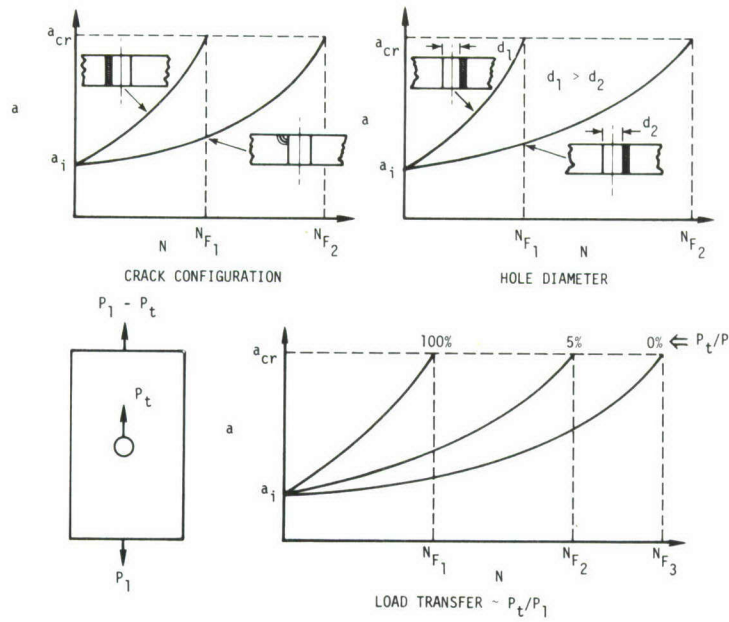


Figure 11. Effect of Structural Properties on N_F

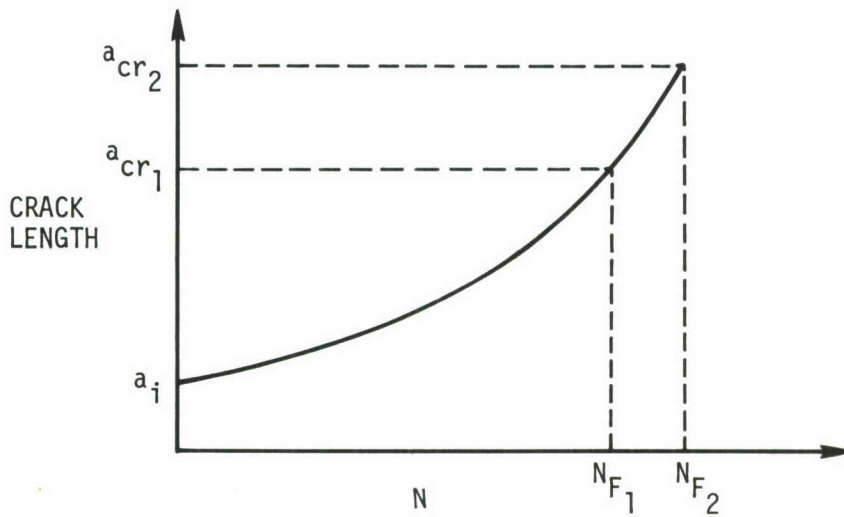


Figure 12. Effect of Critical Crack Size on N_F

- f) the fracture or life limiting criterion which establishes the end point of the life calculation.

Prior to describing each of the above itemized elements in separate subsections, the damage integrating equation will be introduced to show how the various elements interact. As expressed in a numerical form, the damage integrating equation is

$$a_{cr} = a_i + \sum_{j=1}^{N_F} \Delta a_j \quad (2)$$

where Δa_j is the growth increment associated with the j^{th} applied load. The purpose of Equation 2 is to determine the life N_F . The various elements affect the quantities in Equation 2 in the following manner:

1. a_{cr} is determined interrelating Elements b, d, and f.
2. a_i is determined using Element 1.
3. Δa_j is a function of interrelating Elements b, c, d, and e.

a. Initial Flaw Distribution

A measure of initial quality in a component of service hardware is given by the distribution of initial crack sizes as illustrated in Figure 13. For predictions of safety limits, the initial cracks larger than the nondestructive inspection (NDI) detectability limit are of principal concern. Present specifications detail NDI limits and

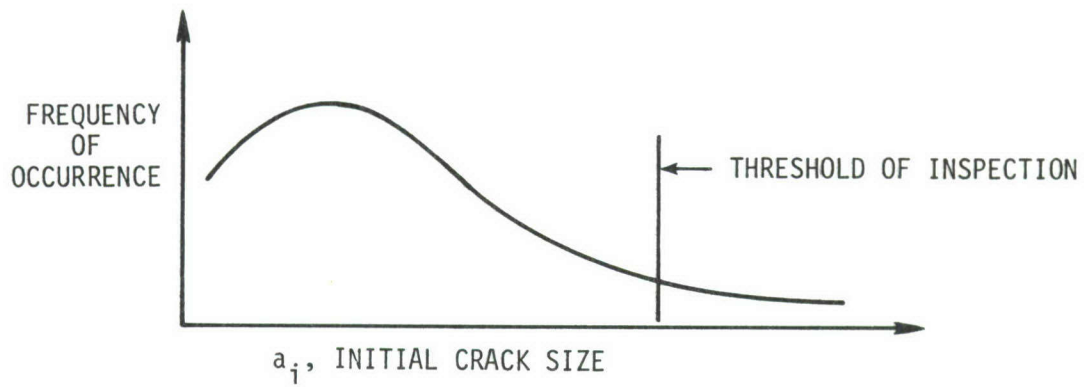


Figure 13. Distribution of Initial Crack Size for a Given Type of Crack (e.g., Radial Cracks Growing for Fastener Holes)

require verification/certification of contractor capability to detect cracks smaller than the specified NDI limits. Normally, such certification is demonstrated with curves of the type shown in Figure 14. The program of certification for a contractor's quality control inspector/inspection techniques allows the USAF to assess the probability and confidence limits associated with detecting a given crack.

Recent results generated by the F-4 Independent Review Team (IRT) provided a method of characterizing the initial flaw population (apparent initial quality) based on full-scale fatigue test-induced cracking behavior. Given the measurable flaw distribution in a structure at some time subsequent to test startup, the initial flaw population can be backtracked by analysis. The "back" extrapolation of the flaw population is conducted using the damage integration package. The process is schematically illustrated in Figure 15. Subsequently, the initial flaw distribution established as illustrated in Figure 15 can be used to estimate influence of load factors, mission profiles, and usage changes on the life of service hardware. The F-4 IRT study also provided an evaluation of statistical methods for describing the large crack length extremes for initial flaw distributions established in this manner. The resulting distribution of F-4 initial cracks is shown in Figure 16.

b. Usage

The sum of the load levels that a structure is expected to experience is determined by a projection of the amount of usage expected over the life in the various possible missions; e.g., hours in training,

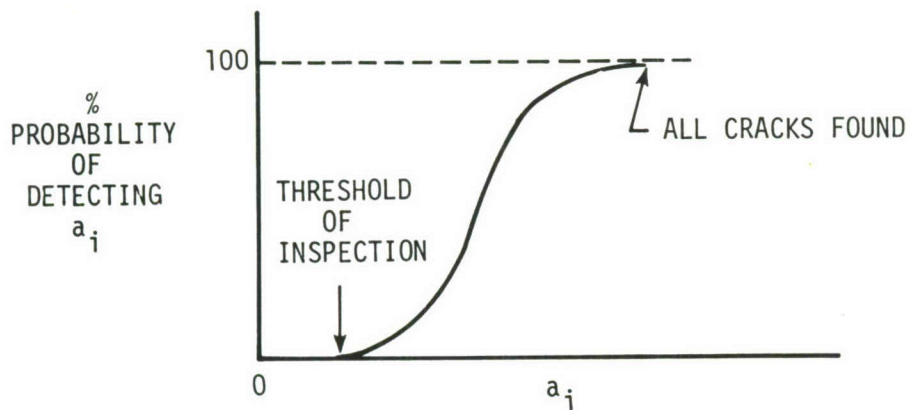


Figure 14. Certification of NDI Capability

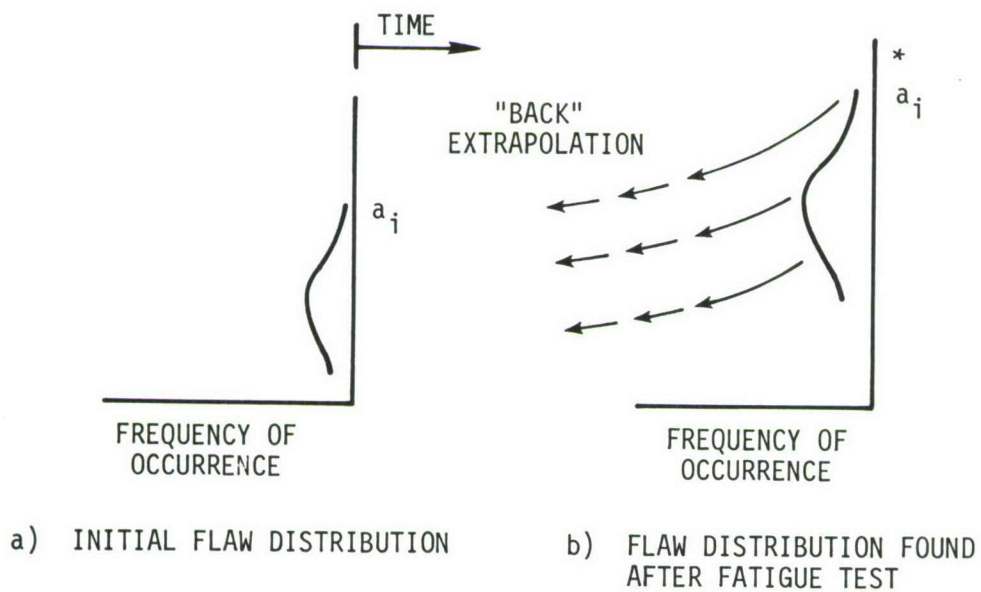


Figure 15. Determining Initial Quality by Back Calculation

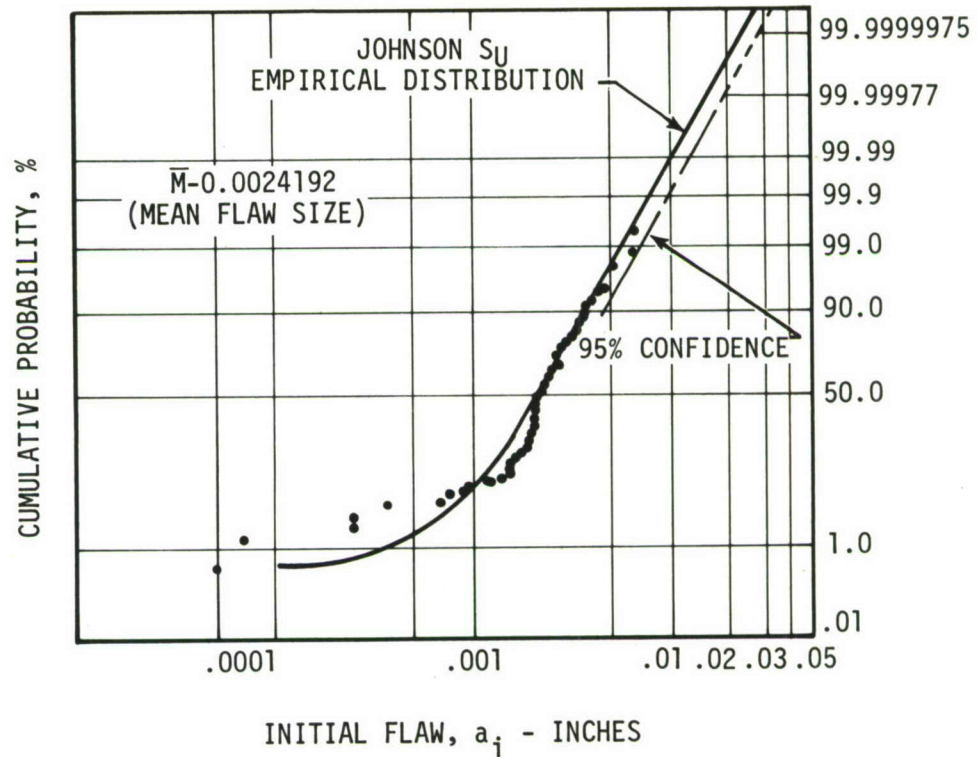


Figure 16. Initial Flaw Distribution for F-4 Based on Back Calculation

air-to-air combat, reconnaissance, weapons delivery, etc. The mission mix includes the relative amounts of time spent in each mission. The most basic information needed is the load factor exceedances at the center of gravity (CG) of the aircraft. This information is illustrated in Figure 17. For new designs this data is derived from actual measured exceedances from operational aircraft flying similar missions. The USAF specifications contain such data. The specific sequence of loads that is applied to the structure is necessary to the crack growth damage accumulation analysis. Current practice is to simulate the overall life on a flight by flight basis. Each flight in the design, analysis, or test load spectrum consists of a series of cycles that combine the deterministic and probabilistic events describing the type of mission. The deterministic events include takeoff and landing and certain basic maneuver loads during each flight. Probabilistic events such as gusts or rough field taxiing occur periodically. Although it is possible to estimate the number of times these events occur, their position in the load sequences is determined in a probabilistic manner.

In developing the load spectrum for crack growth damage analysis, it is necessary to determine the stress history for each critical area on the airframe. This is accomplished by determining the relationship between the load history derived above and the stress response. This is illustrated in Figure 18.

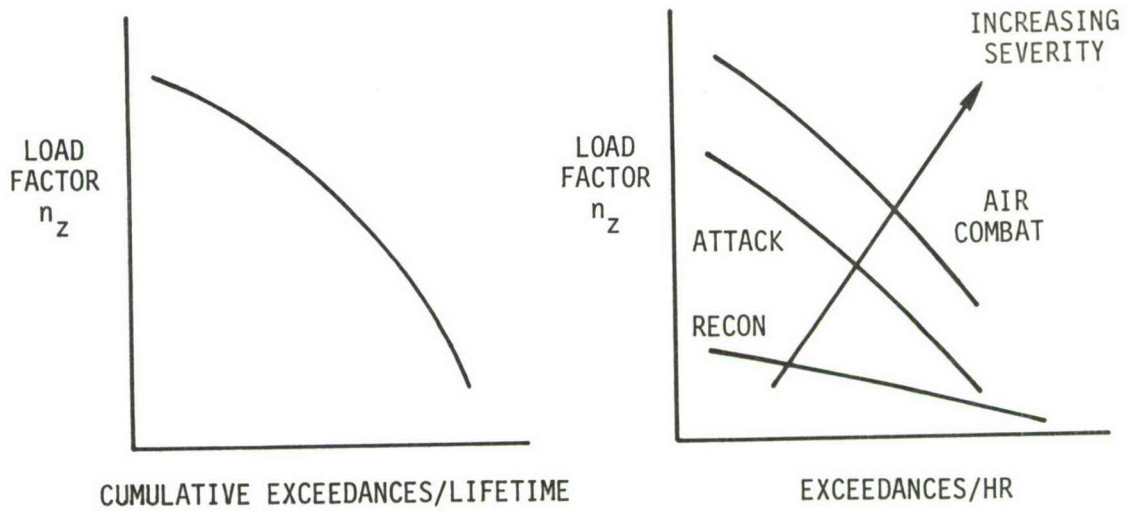


Figure 17. Typical Load Factor Exceedance Information Indicating Usage

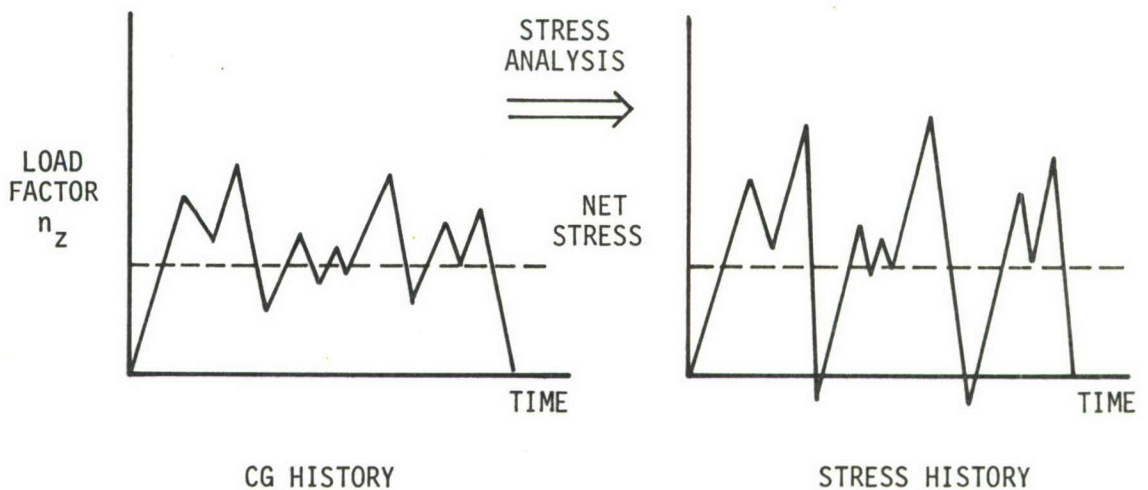


Figure 18. Load Factor to Stress History Transformation

Difference in crack growth resulting from mission mix can be significant. A fighter aircraft that is used primarily for air-combat or air-combat training typically accumulates more damage than one that is used for the same number of hours on a reconnaissance-type mission.

c. Material Properties

The material properties enter the damage integration package in the form of constant amplitude crack growth rate data. Crack growth data are generated in the laboratory under constant cyclic loading on simple specimens with accepted characterizing stress intensity factors. Crack growth rate data are developed and correlated on the basis of growth rate (da/dN) as a function of stress intensity factor range, ΔK , ($\Delta K = K_{\max} - K_{\min}$), as defined in Figure 19.

For a given ΔK , the crack growth rate increases with increasing stress ratio, R ($R = K_{\min}/K_{\max}$). Hence, the constant amplitude crack growth rate properties for a given material or alloy consist of a family of curves as illustrated in Figure 20. The crack mechanics approach described in Section II.2 considers that for a given ΔK , R combination, there is a da/dN that is independent of geometry. Thus, the damage integration package has available a growth rate for each ΔK determined for the given crack configuration and loading.

When necessary, thermal or chemical environment and time (frequency of loading) effects are also included in the crack growth rate data generated for use with the damage integration package.

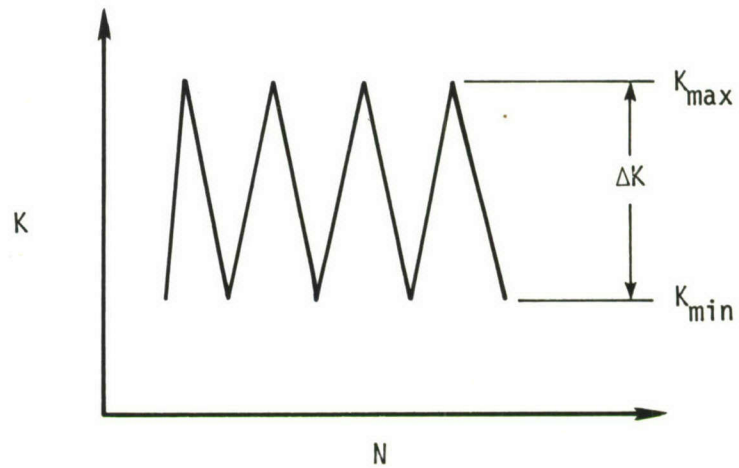


Figure 19. Stress Intensity Factors - Cyclic Loading

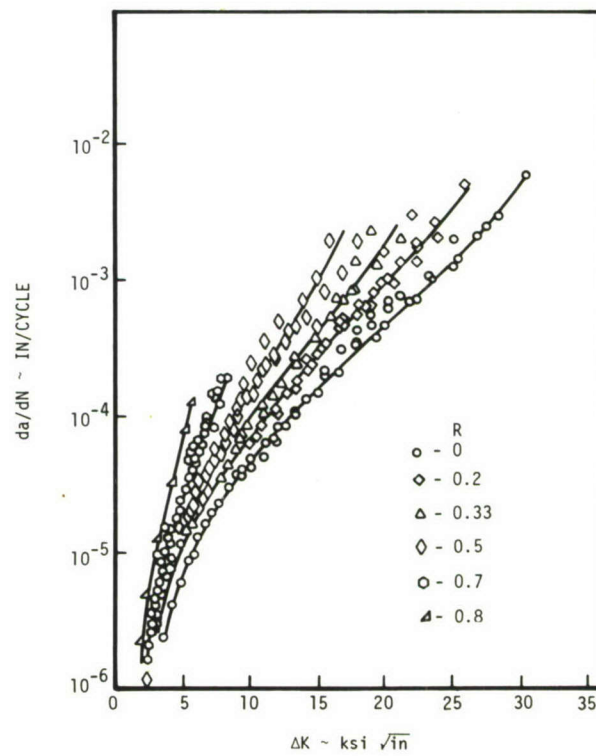


Figure 20. Constant Amplitude Crack Growth Rate Data for 7075-T6 Aluminum

d. Crack Tip Stress Intensity Factor Analysis

The crack tip stress intensity factor (K) interrelates the crack geometry, the structural geometry, and the load on the structure with the local stresses in the region of the crack tip. The stress intensity factor takes the form

$$K = \beta \sigma \sqrt{\pi a} \quad (3)$$

where

β = geometric term for structural configuration,
can be a function of crack length

σ = stress applied to the structure

a = crack length

It can be seen that any number of combinations of the parameters β , a , and σ can give rise to the same K. The crack growth analysis rests on the experimentally verified proposition that a given K gives rise to a certain crack growth rate, regardless of the way in which the parameters were combined to generate that K.

A considerable body of data exists which defines experimental and mathematical solutions for stress intensity factors for various structural configurations.

Since stress enters Equation 3 in a linear sense it is appropriate to express the geometrical part of the stress intensity factor by using the stress intensity factor coefficient, K/σ . Figure 21 illustrates two typical solutions expressed in this manner. For a through-the-thickness crack in plate of infinite extent, the value of β is unity and K becomes

$$K = \sigma\sqrt{\pi a} \quad (3a)$$

Equation 3a provides one way of normalizing more complex K solutions in terms of the infinite plate solution. Figure 22 depicts a typical solution of this type.

Through-the-thickness cracks are handled quite well analytically. However, for corner cracks and semielliptical part-through cracks, such as illustrated in Figure 23, K varies from point to point around the crack perimeter. This variation allows the crack shape to change as it grows, which leads to a very complex three-dimensional problem. For most of these cases, β and K/σ are estimated using simple cases which are amenable to solution.

e. Damage Integration Models

Rewriting Equation 2 such that the integration is conducted between the initial crack length (a_i) and any intermediate crack length (a_K) between a_i and the critical crack length results in

$$a_K = a_i + \sum_{j=1}^N \Delta a_j \quad (4)$$

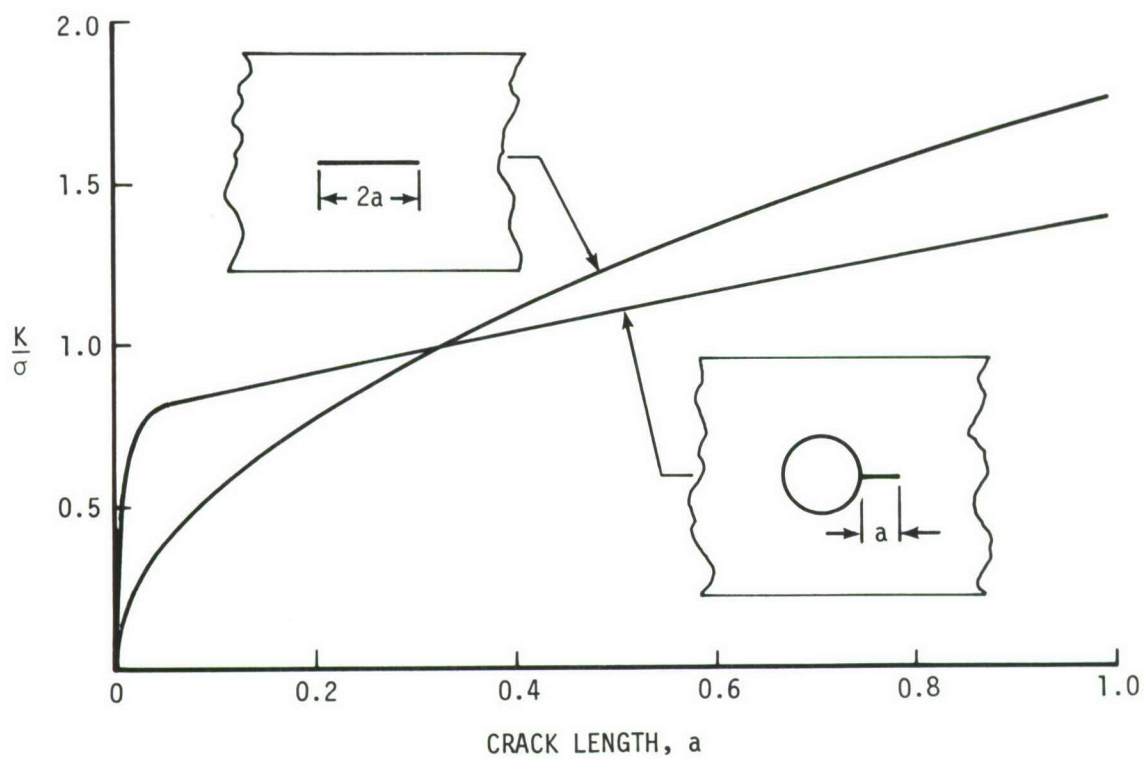


Figure 21. Stress Intensity Factor Coefficients Showing Influence of Hole on K

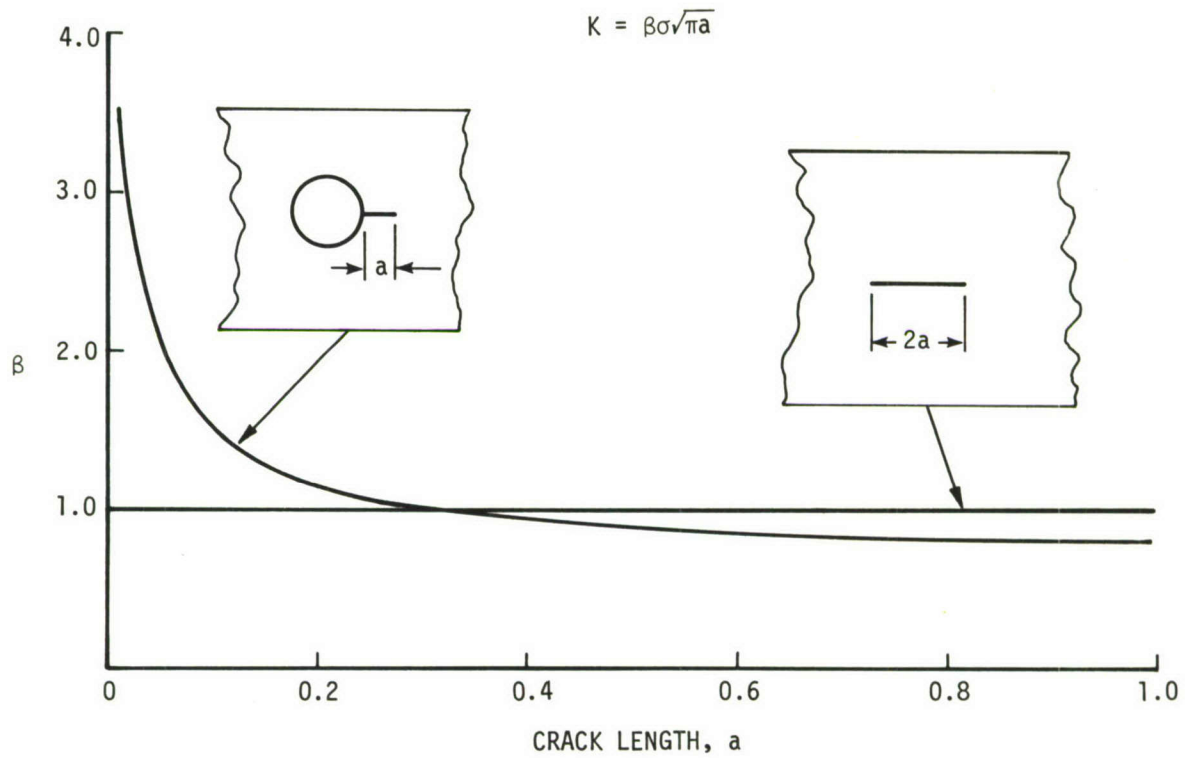


Figure 22. Influence of Hole on Geometric Correction Factor, β

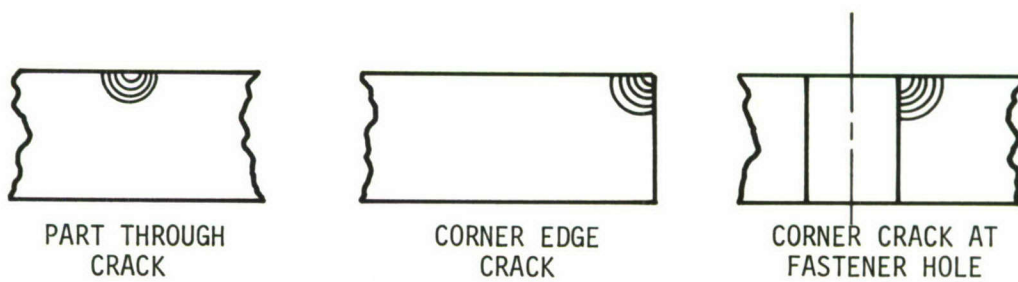


Figure 23. Complex Crack Geometries

where N is the number of cycles corresponding to the intermediate crack length a_K . The next cycle of the spectrum stress induces a crack length growth increment Δa_{N+1} . The damage integration model provides the analysis capability to determine this crack length growth increment. The growth increment Δa_{N+1} is set equal to the constant amplitude crack growth rate, which in turn is expressed as a function of stress intensity factor range (ΔK) and stress ratio (R), i.e.,

$$\Delta a_{N+1} = \left. \frac{da}{dN} \right|_{N+1} = f(\Delta K_{N+1}, R_{N+1}) \quad (5)$$

The stress intensity factor range and stress ratio in Equation 5 are determined by using the maximum and minimum stresses in the $N+1$ cycle of the given spectrum and evaluating the stress intensity factor coefficients associated with the given structural geometry at the crack length a_K . Subsequent to the direct calculation of the two crack tip parameters ΔK and R , and prior to their insertion in Equation 5, they are modified to account for the effect of prior load history using retardation models. Retardation models account for high-to-low load interaction effects, i.e., the phenomena whereby the growth of a crack is slowed by application of a high load in the spectrum. Failure to account for high-to-low load interaction via a retardation model leads to conservative crack life predictions. In the case of bomber and transport aircraft wing spectra, this life prediction will normally be conservative (~ 1.5 to 2.5 times shorter life), whereas for fighter spectra, the life prediction is even more conservative (~ 2 to 5 times shorter life). There are numerous functional forms of Equation 5 and numerous models describing

retardation. Figure 24 and the following list describe the general scheme of the crack growth calculation.

Step 1 - Knowing crack length a_k , determine the stress intensity factor coefficient, K/σ .

Step 2 - For the given stress cycle, $\Delta\sigma$, and the coefficient K/σ , determine the stress intensity factor cycle, ΔK , and stress ratio R .

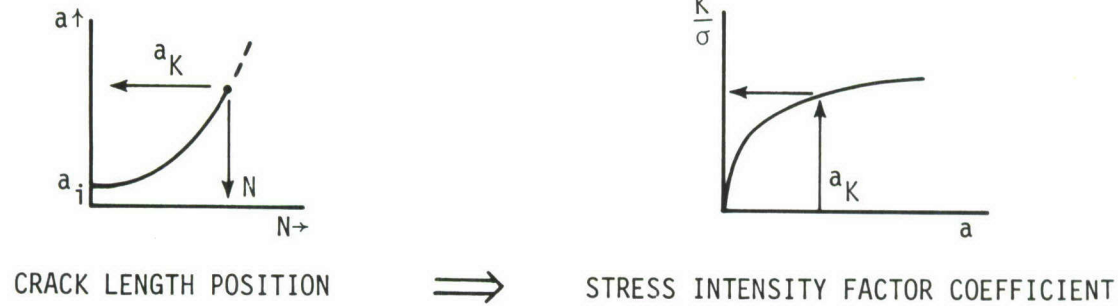
Step 3 - Utilizing the retardation model, modify the stress intensity cycle ΔK and R to account for previous load history.

Step 4 - Determine the growth rate for the stress intensity factor cycle to establish the crack growth increment.

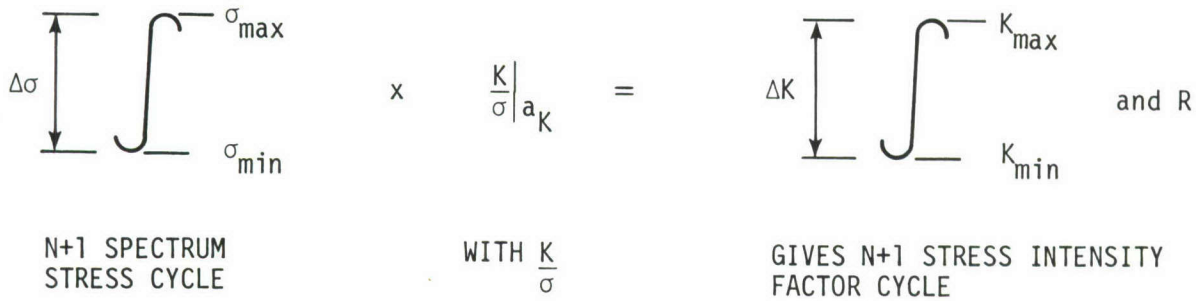
f. Final Crack Length

Consideration of the final crack length and its determination on the basis of residual strength was introduced in Section II.2 and in Figure 6. Such would ordinarily be sufficient for estimates of the safety limits; however, durability considerations often dictate that the final crack size, a_F , be chosen smaller than a_{cr} to represent re-work or repair limits. A choice of a_F along these lines is shown in Figure 25. The concept of economic repair is also discussed in Section I.2d in the context of design requirements.

STEP 1



STEP 2



STEP 3

$$\Delta K_{N+1} = g_1 (\Delta K, \sigma(N - N_0))$$

$$R_{N+1} = g_2 (R, \sigma(N - N_0))$$

STEP 4

$$\left. \frac{da}{dN} \right|_{N+1} = f(\Delta K_{N+1}, R_{N+1})$$

WHERE f MAY HAVE, FOR EXAMPLE THE FORM

$$f(\Delta K, R) = \frac{5 \times 10^{-7} \times \Delta K^3}{68 \times (1 - R) - \Delta K}$$

Figure 24. Sequence of Steps Required to Calculate Crack Growth Increment

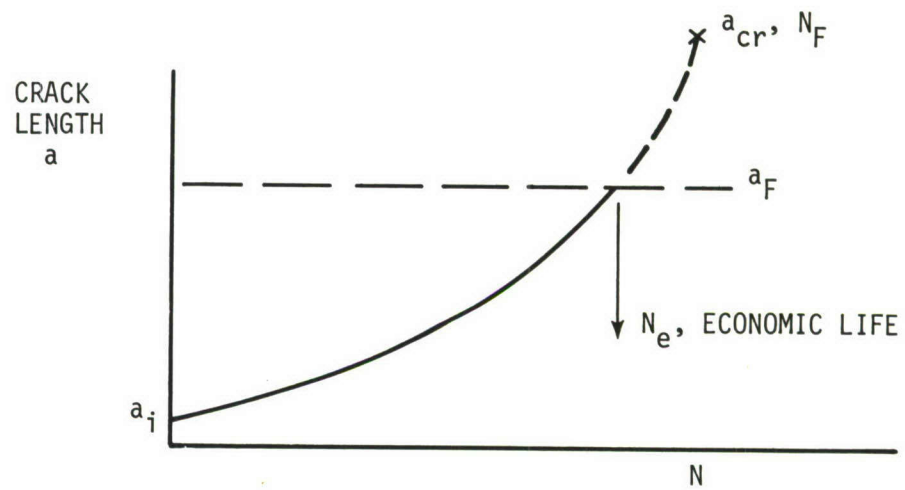


Figure 25. Economic Final Crack Size

5. ACHIEVING CONFIDENCE IN THE USE OF THE DAMAGE INTEGRATION PACKAGE

Since life predictions for service hardware are based on the crack growth damage integration package, the confidence in a life prediction value must be based on a measure of the ability for a given package to predict measured phenomena. A change of any fundamental element within the package (e.g., retardation model) generally requires a resubstantiation of this confidence for the revised package. An extension of capability, i.e., more complex geometry, would require only a substantiation for that level of complexity. We take this approach because of the substantiated influence of each of the variables associated with the individual elements. Verification of the package is normally conducted in steps progressing from predictions of laboratory-generated fatigue crack growth data (for which all test conditions are reasonably well characterized and documented) to predictions of service-experienced cracking behavior. Verifying the package in steps allows for immediate deletion of inaccurate or erroneous assumptions made in developing or improving a given element of the package. Since the package will be used to make life predictions where unknowns (e.g., spectra, structural load interactions) prevail, it is essential that confidence be established for each level of prediction capability that has been achieved.

Figures 26 through 28 are provided as examples to show how elements within a package are verified. All figures show the correlation between predicted and measured life. Figure 26 provides an evaluation of a new retardation model in which the data base was a measure of the cyclic delay subsequent to an overload. Figure 27 compares the predictions developed with the AFFDL-Willenborg-retardation model (damage integration

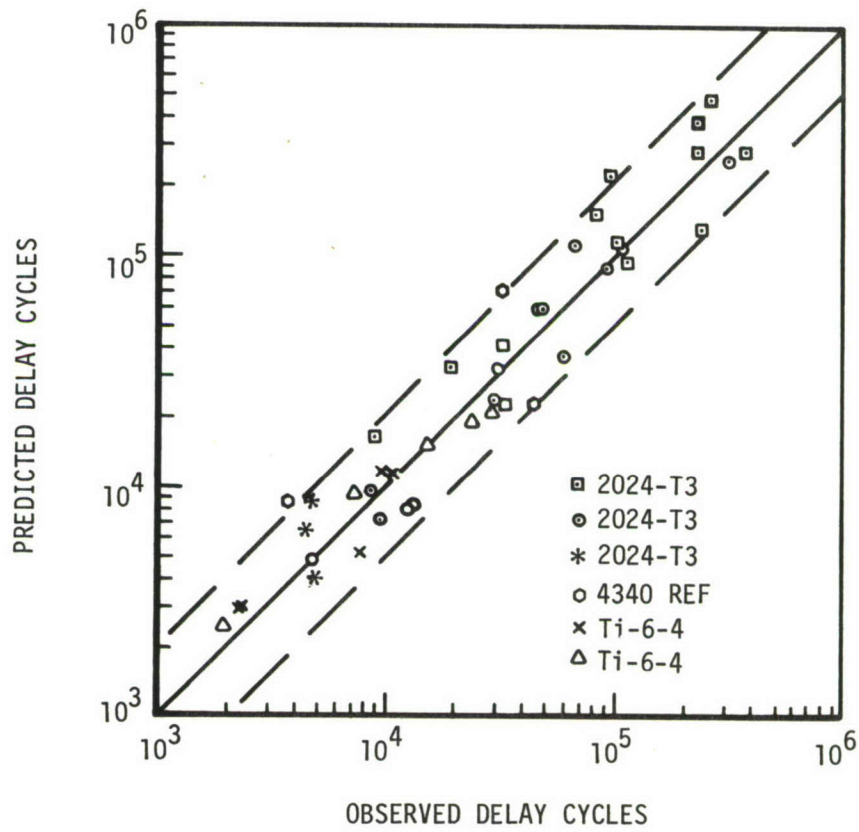


Figure 26. Single Overload Correlation with Modified Wheeler Retardation Model

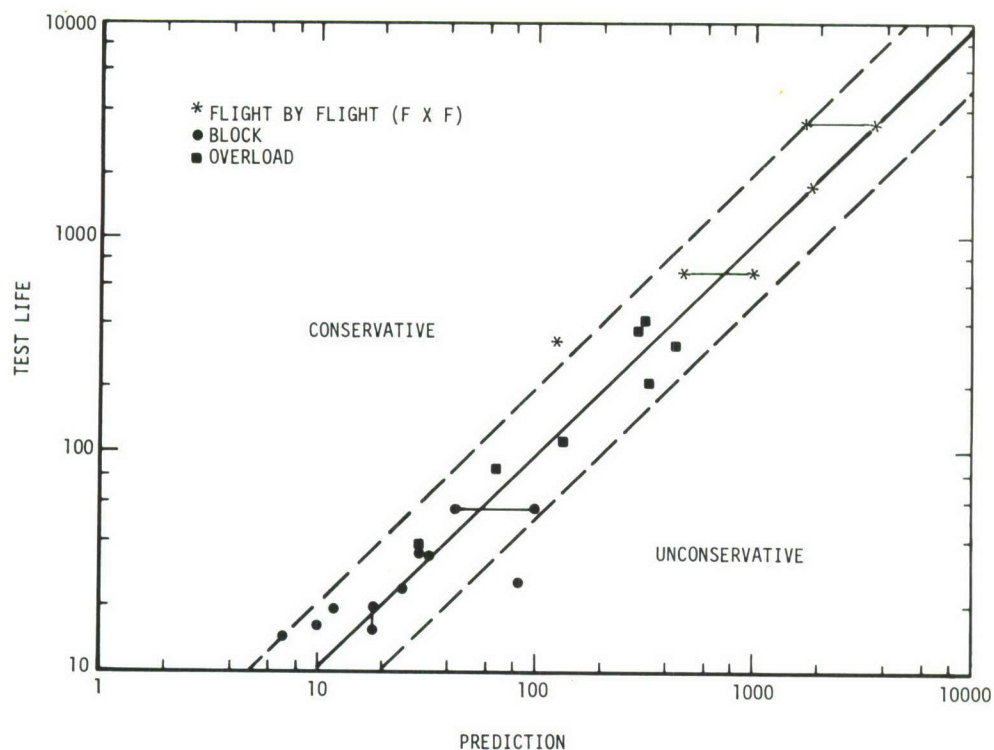


Figure 27. Spectrum Correlation Using the AFFDL Willenborg-Retardation-Model (Damage Integration Package)

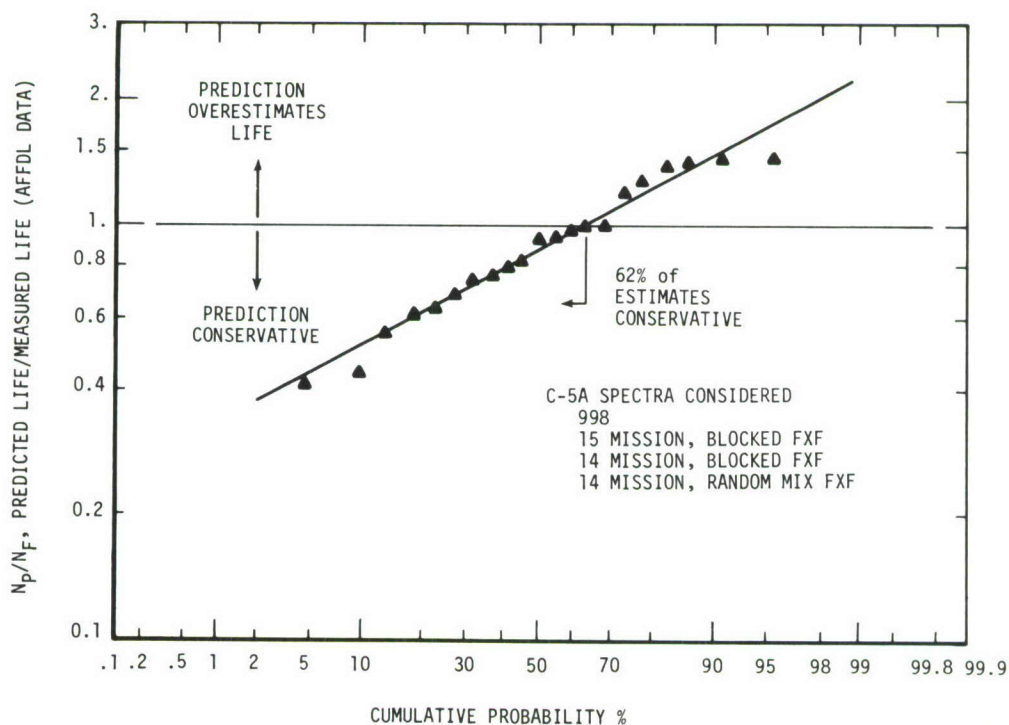


Figure 28. Prediction Capability of Damage Integration Package (Based on 21 Laboratory Tests Conducted at AFFDL)

package to laboratory test data) which show the influences of spectra and crack geometry changes. Figure 28 shows the evaluation of a new AFFDL damage integration package to account specifically for C-5A spectra changes on life observed when the crack geometry is a radial corner crack growing from an open or plugged hole.

SECTION III

SIMULATION OF SERVICE ENVIRONMENT FOR TEST

1. SUMMARY

The ground test of a full-scale aircraft is used to evaluate the durability and damage tolerance capability of a typical fleet aircraft when subjected to the simulated service environment. An aircraft that would have a useful life on the order of 10 to 20 years with 4,000 to 30,000 flight hours' capability can typically be given equivalent experience in a ground test lasting one year or less. The actual time required to apply the repeated loads is dictated by capability of the test equipment (i.e., the hydraulics, the loading devices, etc.), but the total time is governed by the size of the structure and the time required to shut down for inspection, repair, etc. For a small specimen, such as a coupon, it is possible to simulate one service lifetime in a few days.

Time shortening is accomplished by deleting sustained load times, truncation of low level cycles, and the application of loads at a faster rate than they occur in-service. Considerable analyses and associated tests must be accomplished to ensure that the shortened simulation adequately models the service behavior.

2. SUSTAINED LOADS

Most studies of load-time effects in metal structures for subsonic aircraft have shown that the crack growth damage accumulation behavior is mainly a function of the number and severity of the repeated ground and flight loads that are applied to the aircraft structure. The amount

of time that the structure experiences sustained loads, e.g., during cruise or on the ground, have not been found to significantly affect the damage accumulation. Thus, the time at a steady load is usually completely truncated for the ground tests as indicated by Figure 29.

3. REPEATED LOADS

Most studies have shown that gust and maneuver loads below a certain amplitude level do not significantly contribute to damage accumulation. These cycles are also truncated for the ground test. Figure 30 illustrates the type of behavior that is seen as increased cycles are truncated. Typically, as larger amplitude cycles are truncated, their damage contributions are deleted and the apparent structural life increases. The truncation level for the full-scale simulation must be chosen carefully to shorten the ground test time without affecting the damage accumulation rate.

4. ENVIRONMENTAL SIMULATION/FREQUENCY EFFECT

The requirement for simulation of environmental effects during ground tests is established in MIL-A-8866. Potential environmental problems such as corrosion are simulated when necessary in the full-scale test. Typically, there is considerable corrosion testing done for coupons and components to evaluate environmental effects and to qualify coatings and other protection methods. Normally, this approach eliminates the need to simulate the environment in the full-scale test.

In the event that time-dependent environmental effects can be neglected, the test duration is determined by the speed of the test

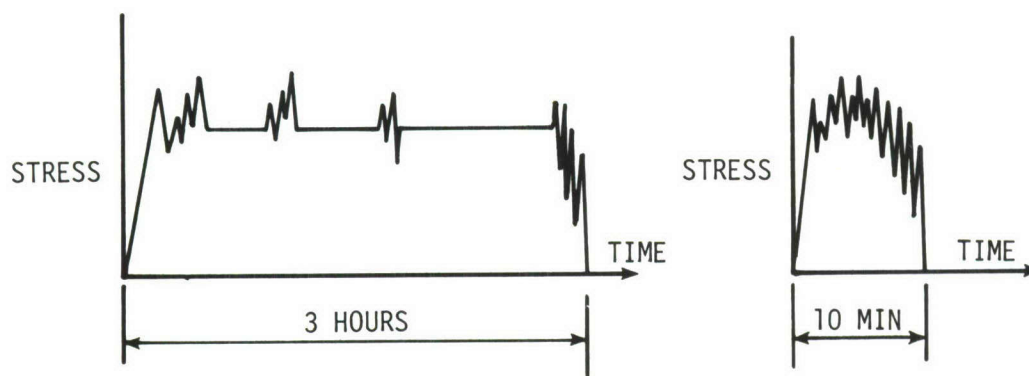


Figure 29. Simulation of Real Time Flight Loads by Leaving Out Cruise Time

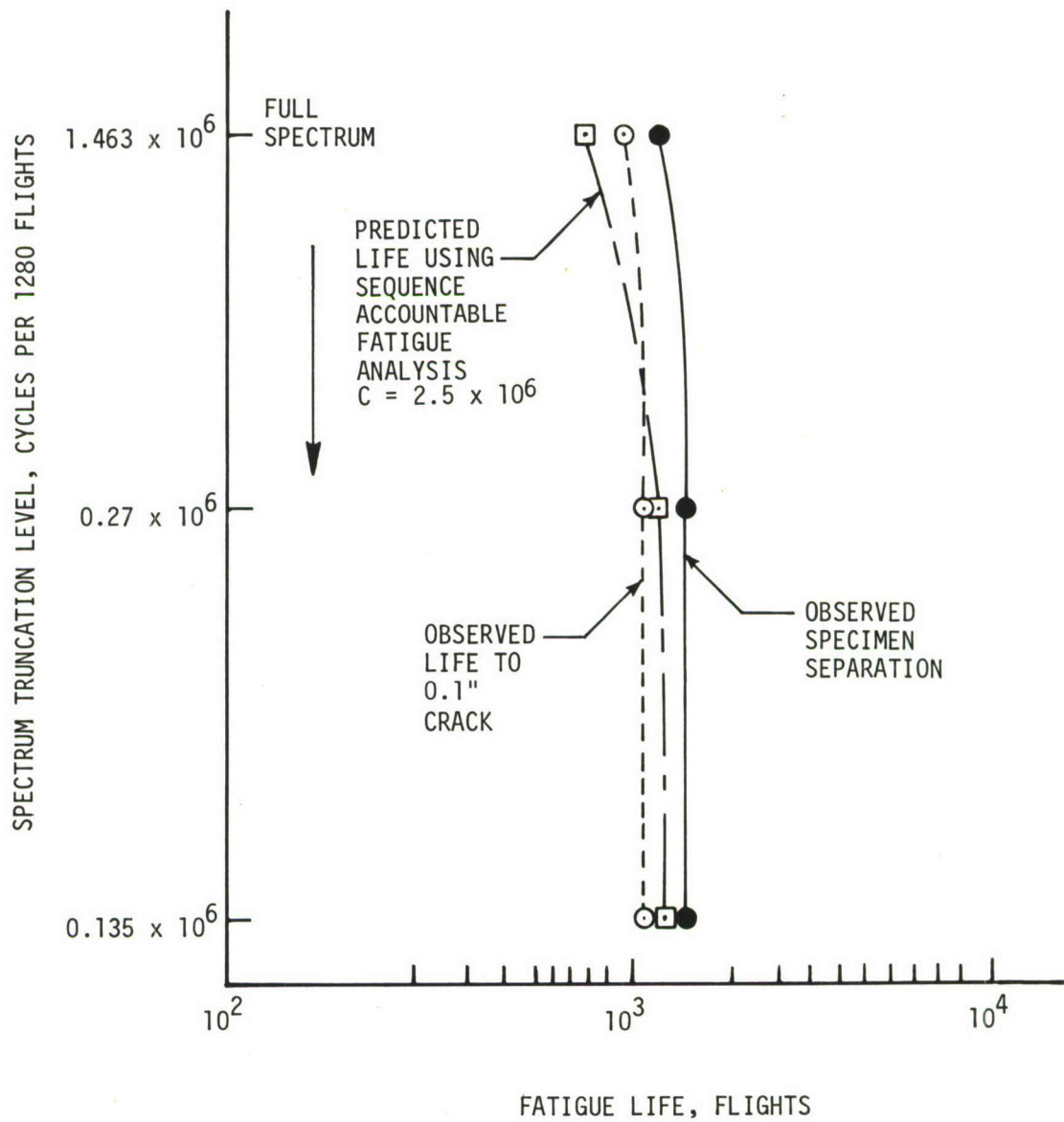


Figure 30. Effect of Truncation Level on the Life Determined in the Ground Test

equipment and the number and duration of the required detailed inspections. Through truncation of the time at a steady load and truncation of low-level cycles, it is possible to apply a lifetime of loading history on a full-scale ground test structure in less than a one year time span (multi-shift test operation). Tests of small components can be run at even faster rates so that it is often possible to accumulate a lifetime of the simulated history in a few days.

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