14 Cyclic Loading Effects

The purpose of this chapter is to provide general information to the stress analyst on the subject of fatigue, durability and damage tolerance, and aircraft life. Each program has its own requirements and tools for durability and damage tolerance analysis but the general concepts are the same. Refer to specific program guidance for requirements.

Cyclic Loading effects for laminated composite material is covered in Reference 14.5, PM4056 Section 15.

Contents

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14.2 Symbols and Nomenclature

Symbol or Acronym	Definition	Units
a	Crack Length	in
a_0	Initial Crack Length	in
a_{crit}	Critical Crack Length	in
A	Amplitude Ratio	--
c	Crack Length (alternative)	in
D	Fastener Diameter	in
D	Damage Ratio	--
GAG	Ground Air Ground Cycles	--
K	Stress Intensity Factor	ksi√in
k_t	Stress Concentration Factor	--
K_{SF}	Stress Severity Factor	--
K_{TB}	Bearing Stress Concentration Factor	--
K_{TG}	Bypass Stress Concentration Factor	--
L/ESS	Loads and Environmental Spectrum Survey	--
NDI	Non-destructive Inspection	--
P	Load	lbs
S, σ	Stress	psi
S-N	Stress-Number of Cycles	--
SDC	Structural Design Criteria Document	--
t	Thickness	in
T	Time	hours
α	Hole Condition Factor	--
β	Hole Filling Factor	--
θ	Fastener Tile Factor	--

14.3 Introduction

Cyclic loading in general refers to the calculation of life of aircraft structure. There are a number of different theoretical and philosophical approaches to this calculation and the science of life analysis has evolved over time in response to knowledge gained from aircraft structural incidents or accidents and fleet maintenance concerns.

There are four terms in common use: fatigue, durability, damage tolerance and safe-life..

Fatigue is a general term describing weakening or failure of a material as a result of applied cyclic loads. Repeated loading and unloading of the material will start microscopic cracking, typically at stress concentrations, grain boundaries, or metallic flaws which will, under continued load cycling, result in the crack growing to a critical size resulting in failure.

Durability is a measure of the structure's ability to resist or avoid cracks or damage that results in excessive, untimely, or costly actions in-service such as repeated inspections, required repairs or invasive modifications. It is also concerned with resistance to functional problems such as fuel leaks or control system binding as well as any structural behavior which would result in adverse impacts to fleet readiness. Durability is concerned with the economics of operating the aircraft or the functional consequences of cracking and, near end of life, safety from widespread fatigue damage. Durability will be discussed in Section 14.7.

Damage tolerance is a measure of a structure's capability to safely contain cracks or damage for some period of aircraft usage during which it is able to sustain normal operational load levels. Thus damage tolerance is concerned with the safety of flight with a stated goal of preventing catastrophic failure due to the presence of a rogue defect. Damage Tolerance will be discussed in Section 14.8.

There is one life management strategy, no longer in common use but which may be applicable to older, legacy aircraft (prior to 1956). It is called **safe-life** or safety by retirement. It is a fatigue management strategy based on the proactive replacement or modification or retirement of parts or aircraft at a pre-established time, regardless of condition. It still may be in use on aircraft components such as landing gear and older commercial aircraft.

14.4 Requirements and Governing Specifications

The Aircraft Structural Integrity Program (ASIP), as defined in Reference 14.7, Mil-Std-1530 governs the requirements for structural integrity for the USAF. Its stated scope is

"This standard describes the USAF Aircraft Structural Integrity Program (ASIP) which defines the requirements necessary to achieve structural integrity in USAF aircraft while managing cost and schedule risks through a series of disciplined, time-phased tasks. It provides direction to government personnel and contractors engaged in the development, production, modification, acquisition, and/or sustainment of USAF aircraft."

It requires that all new aircraft programs address all sections of the standard and come up with an ASIP Master Plan for their specific platform. This is required because the effectiveness of the military force is dependent on the operational readiness and safety of the aircraft and, in particular, the airframe structure. The ASIP plan governs structural planning, part classification, material and fastener selection, requirements for testing from coupon level to full scale, structural certification and force management development and certification. This is codified in what have been called the Five Pillars of ASIP, shown in Figure 14.4-1. The activities are arranged into five tasks which are sequenced throughout the program.

Task I	Task II	Task III	Task IV	Task V
Design Information	*Design Analyses and Development Tests*	*Full-Scale Testing*	*Certification and Force Management Development*	*Force Management Execution*
ASIP Master Plan	Materials and Joint Allowables	Static Tests	Certification Analyses	Individual Aircraft Tracking Program
Design Service Life and Usage	Loads Analysis	First Flight Verification Ground Tests	Strength Summary and Operating Restrictions	Loads/Environment Spectra Survey
Structural Design Criteria	Design Service Loads Spectra	Flight Tests including Flight Vibration Tests and Flutter Tests	Force Structural Maintenance Plan	ASIP Manual
Damage Tolerance and Durability Control Plans	Chemical/Thermal Environment Spectra	Durability Tests	Loads/Environment Spectra Survey	Aircraft Structural Records
Corrosion Prevention and Control Program	Stress Analysis	Damage Tolerance Tests	Individual Airplane Tracking Program	Force Management Update
Nondestructive Inspection Program	Damage Tolerance Analysis	Climatic Tests		Recertification
Selection of Mat'ls, Processes, and Joining Methods	Durability Analysis	Interpretation and Evaluation of Test Results		
	Vibr/Sonic Analysis			
	Aeroelastic and ASE Analysis			
	Mass Properties Analysis			
	Design Development Tests			
	Production NDI Capability Assessment			

Figure 14.4-1 The Five Pillars of ASIP: USAF ASIP Program Tasks (Reference 14.7)

The structural integrity program is more than just a requirement to calculate the life of the aircraft. It is holistic in its approach to make sure anything that could affect the life is examined, planned-for, analyzed and tested. Task I is the planning phase where the requirements for design information are identified and written into a program plan. Task II is the heart of the development program where the design is matured, drawings and models are developed, all initial analyses are performed, material and joints allowable testing is performed and manufacturing and inspection development is completed. Task III is the full scale testing of the first articles built for ground testing such as static or durability tests, climate tests, ground vibrations testing, and flight testing. More information on testing can be found in PM4057 Section 17 Structural Certification and Testing or PM4056 Section 18 Structural Certification and Testing (Composites). Task IV is where the final deliverable reports which represent the correlated results from test, finite element models and analysis are presented. Additionally the flight restrictions are defined in the Strength Summary and Operating Restrictions report, based solely on static strength analysis. The Force Structural Maintenance Plan defines the inspection and maintenance requirements and limited life parts based on durability and damage tolerance analysis. Task V is a predominantly USAF task to maintain and fly the aircraft and, if desired, they will contract with LM to perform specific tasks.

The U.S. Navy has a different approach, which is not written into a specification. Although, in principle, they use many of the same elements, the specifics of the NASIP plan are proposed/negotiated with the Navy customer at the beginning of the aircraft program. Additionally there are differences in the analytical approaches used by the U.S. Navy and Air Force. Reference 14.11 may provide some insight.

One of the requirements for any new program is typically a full scale durability test. This test has a number of objectives:

- Correlate to and verify the durability analysis
- Find “Hot Spots” where unexpected early cracking occurs and provide basis for corrective actions to production aircraft

- Evaluate economic life of airframe and the late-in-life crack locations which provides downstream data for life extension programs
- Verify damage tolerance analysis at selected critical points
- Verify or modify inspection intervals for fracture critical parts
- Verify in-service inspection methods and field repairs

More information on full-scale testing can be found in Section 17.7.

14.4.1 Life Requirements

The Structural Design Criteria document will specify a required life for the aircraft structure. This may be in flight hours or in numbers of years. Analytically the life is typically in terms of the number of flight hours or cycles. So if the aircraft life is specified in numbers of years then it will have to be converted using some average number of flights per year. For new aircraft platforms, this will likely be estimated from similar aircraft flying similar missions. For existing aircraft platforms, it may come from fleet usage data.

There can be a number of definitions of “life” throughout the program based on how the prediction was made. The most basic is the **analytical life** which is the mathematically calculated predicted mean life. **Factored Analytical Life** is the predicted (mean) life divided by a scatter factor. For example, suppose the predicted (mean) life of a part is 100,000 hours and the program requires a scatter factor of 4 on analytical durability predictions. The factored analytical life is then 100,000 hours divided by a scatter factor of 4 or 25000 flight hours.

The scatter factor or uncertainty factor is a reduction factor to account for test scatter and inaccuracy of analysis methods. Table 14.4-1 gives some typical scatter factors; however these may vary by program and customer and other values may be used. Scatter factors are used for both durability and damage tolerance predictions.

Table 14.4-1 Typical Scatter Factors

Scatter Factor	Applicability
4.0	Analytical durability life
2.0	Test-based durability life (also may be called Life Reduction Factor)
2.0	Analytical damage tolerance life
1.0	Life predictions based on actual flight hours

Life Reduction Factor or Scatter Factor is the ratio of test-demonstrated life of a design to the certified fatigue life. **Certified Life** is the test-demonstrated life divided by the life reduction factor. For example is a full scale article has a test life of 60,000 flight hours, and a life reduction factor of 2.0, the article then has a certified life of 30,000 hours.

It is recommended that when a life is quoted in an analysis it is a factored analytical life or a certified life rather than the unfactored values and the scatter factor is clearly stated.

14.4.2 Parts Classification

Per Reference 14.7, all critical structural parts have to be identified and classified. Figure 14.4-2, from Reference 14.7 identifies a flow chart to be used in the classification of parts. Based on the part design and function in conjunction with analysis assumptions, a part is evaluated as to whether its failure could cause loss of aircraft, crew or inadvertent release of stores. If the answer is yes, the part is a safety of flight (or safety critical) part. If not, then there are a series of other choices to determine if the part is mission critical, maintenance critical or normal controls. Maintenance critical parts are also referred to as durability critical on some programs.

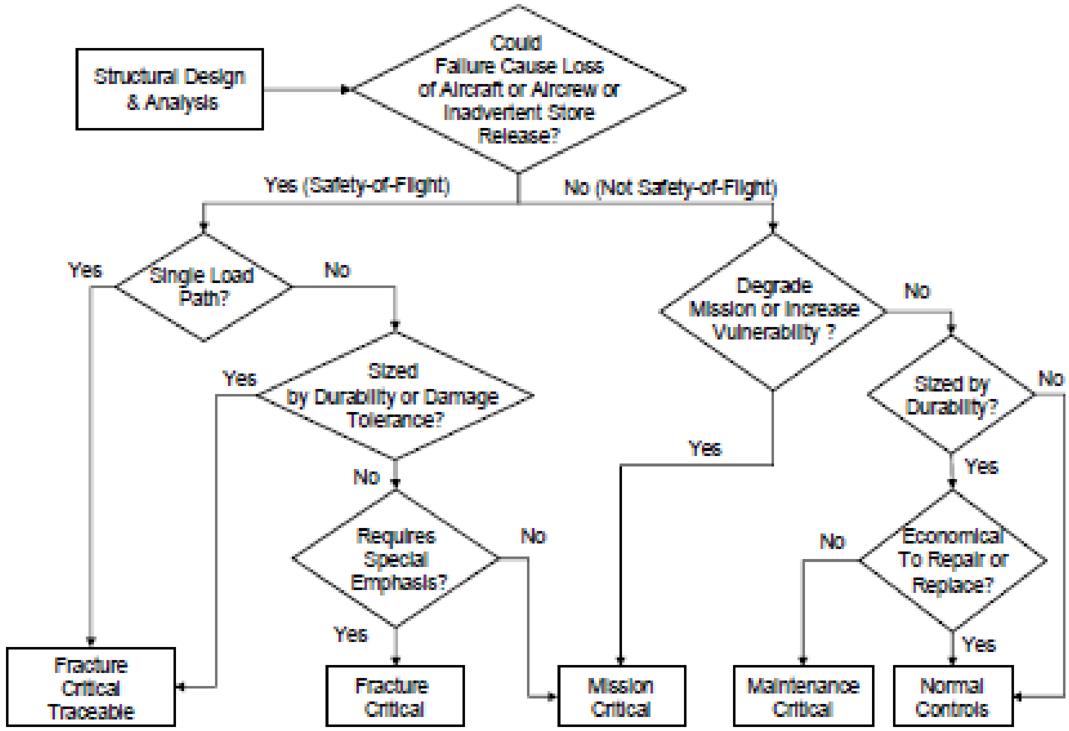

Figure 14.4-2 Critical Parts Selection Flowchart (Reference 14.7)

For all programs since the early 1990's the classification has to be reflected on the drawing. Prior to that, how the classification was documented varied by program. The classification of the part is determined during the design phase as structural margins of safety and analytical life predictions are made. Typically the DADT analyst, in consultation with the stress analyst will recommend a classification. The program fracture control board will make the final determination.

Parts which are fracture critical or maintenance critical or mission critical may have restrictions on the types of manufacturing processes or materials that can be used in their manufacture. For example, on some programs parts classified other than normal controls cannot be cold formed. This is to minimize residual stresses which can affect the life.

Additionally, the type of analysis is also a function of classification. All parts have a static strength analysis and must have a margin of safety greater than or equal to 0.0. Durability analysis is conducted on all mission critical maintenance critical and fracture critical parts. Durability analysis or screening may be conducted on the normal controls parts depending on program guidelines. Damage tolerance analysis is conducted on all fracture critical parts and on other parts as judged necessary. See Section 14.8 for further discussion.

Fracture critical traceable (also on older aircraft, fracture critical I) parts require cradle to grave traceability which means every process throughout the life of the part has to be documented and these parts have to be serialized to enable this tracking. This makes these part very expensive and this classification is reserved, typically, for the most critical single load path parts.

By far the largest number of structural parts on the airplane are normal controls. Mission and maintenance critical make up the second largest block and the smallest category is fracture critical.

14.5 Cyclic Loading

The external input of interest in fatigue is the set of cyclic loads experienced by the aircraft. The external cyclic loads give rise to cyclic stresses throughout the airframe structure, which drive the fatigue process. A single stress cycle has a mean stress, a valley, and a peak. These two stress values are needed to fully describe the cycle. This is illustrated in Figure 14.5-1. A cycle can be described in terms of its peak and valley or its mean and range. Half of the range is the amplitude. The "fatigue load sequence" is the sequence of peaks and valleys, depicted in Figure 14.5-1 as a function of time. Typically, as shown below, the peaks and valleys vary.

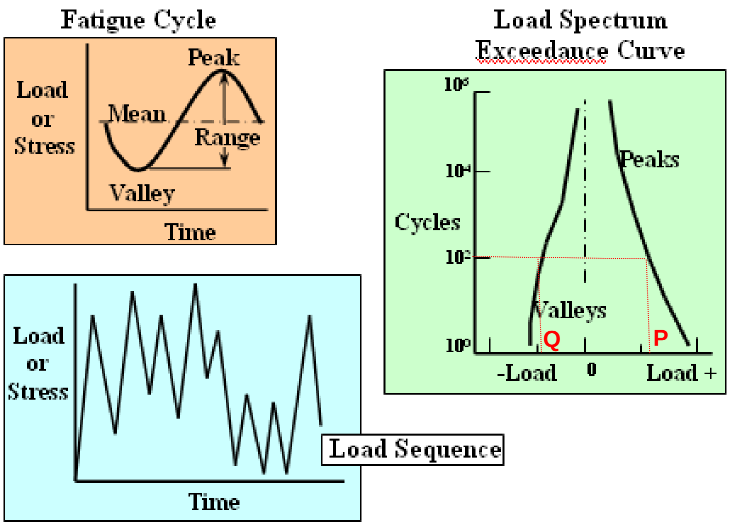

Figure 14.5-1 Fatigue Cycle Definitions

Aircraft fatigue stress sequences are rarely simple constant amplitude sequences. Variable amplitude cyclic loading, also called spectrum loading, is used for aircraft design. A spectrum load sequence can consist of millions of cycles. Thus the load sequence plot becomes an impractical engineering tool.

An exceedance curve or cumulative occurrence curve can be more practically used to view the character and severity of a particular sequence of fatigue loads or stresses. An exceedance is a cumulative occurrence of a peak, a valley, or a range of parameters greater than a given value. So, the exceedance curve tells how frequently each given peak (or valley) load is exceeded. For example on the exceedance plot shown in Figure 14.5-1 (right), 100 tensile peaks or load magnitudes (per 1000 flights) are of larger magnitude than load magnitude P. The sign convention for valleys is reversed, thus, an exceedance of valley Q is a higher magnitude compressive load. Therefore, here, 100 valleys per 1000 flights are more compressive than Q.

Also note the magnitudes, on the tensile side the peaks are much higher than P, as evidenced by the right tail extending much further to the right. On the compressive side, while there are the same number of exceedances the magnitudes are reduced as is evidenced by the left tail being much closer to Q.

Figure 14.5-2 illustrates the nomenclature associated with constant amplitude fatigue cycles.

- The maximum peak is S_{MAX} and the minimum valley is S_{MIN}
- The stress range, ΔS, is the difference between the maximum and minimum stress values.
- The stress amplitude, S_A, is half the stress range or $\Delta S/2$.
- The mean stress, S_M, is half the sum of the maximum and minimum stresses or $(S_{MAX} + S_{MIN})/2$.
- The stress ratio, R, is S_{MIN}/S_{MAX} and may be positive or negative. A stress ratio of -1 means the compression stress is just as large as the tensile stress. This is also referred to as fully reversed. A stress ratio of 0 means the minimum stress is 0.

- The amplitude ratio, A, is S_A/S_M.

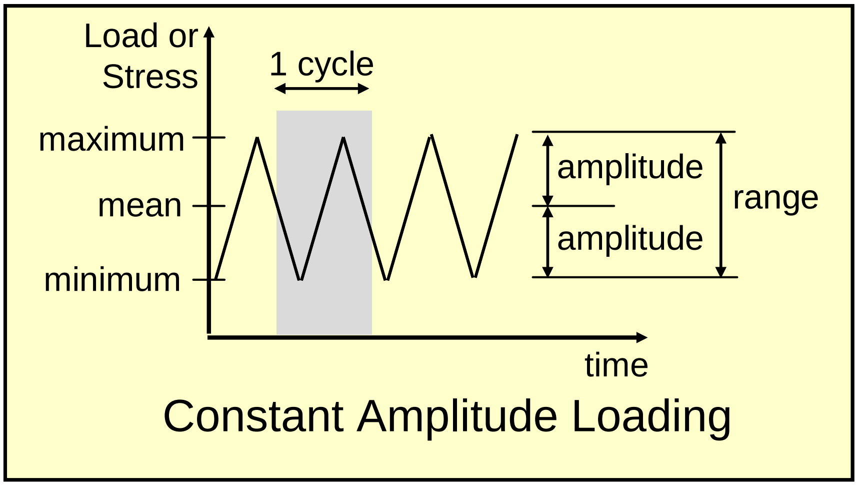

Figure 14.5-2 Fatigue Stress Cycle Nomenclature for Constant Amplitude Cycling

Aircraft load spectra sequences are typically random. Figure 14.5-3 shows an example of a variable random amplitude loading and the terminology used to describe it. A reversal refers to either a peak or valley. A positive range is the magnitude of a load increase from a valley to a peak and a negative range is the magnitude of the load decrease from a peak to a valley.

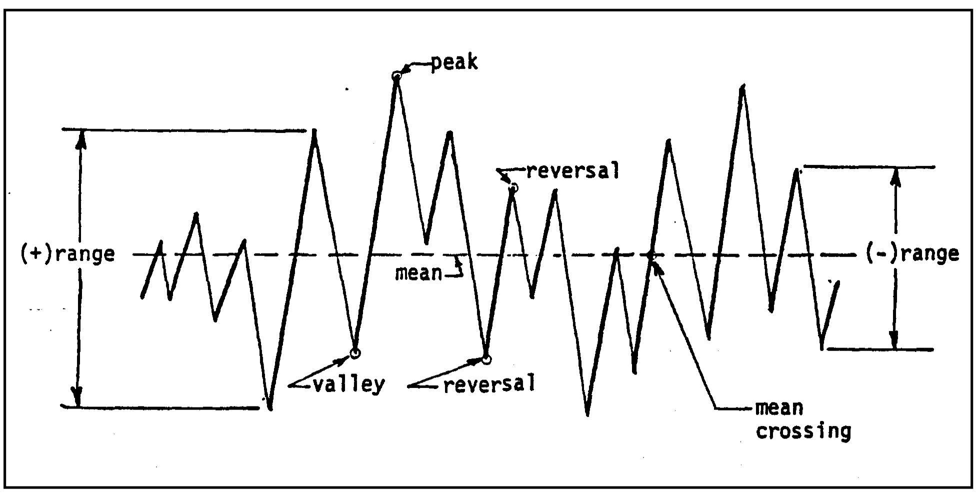

Figure 14.5-3 Fatigue Stress Cycle Nomenclature for Variable Amplitude Random Cycling

For the analysis spectrum, blocks of cycles which represent a random flight-by-flight sequence of some number of flights which are comprised of some large number of flight hours, *e.g.*, 500 to 800 hours. It's random because the order of the missions is random and within any flight segment, the load magnitudes are drawn randomly from the exceedance curve corresponding to that segment type. It is called flight-by-flight because the segments and their duration follow the flight mission profiles. These block can then be repeated to form one lifetime of hours. This is shown in Figure 14.5-4.

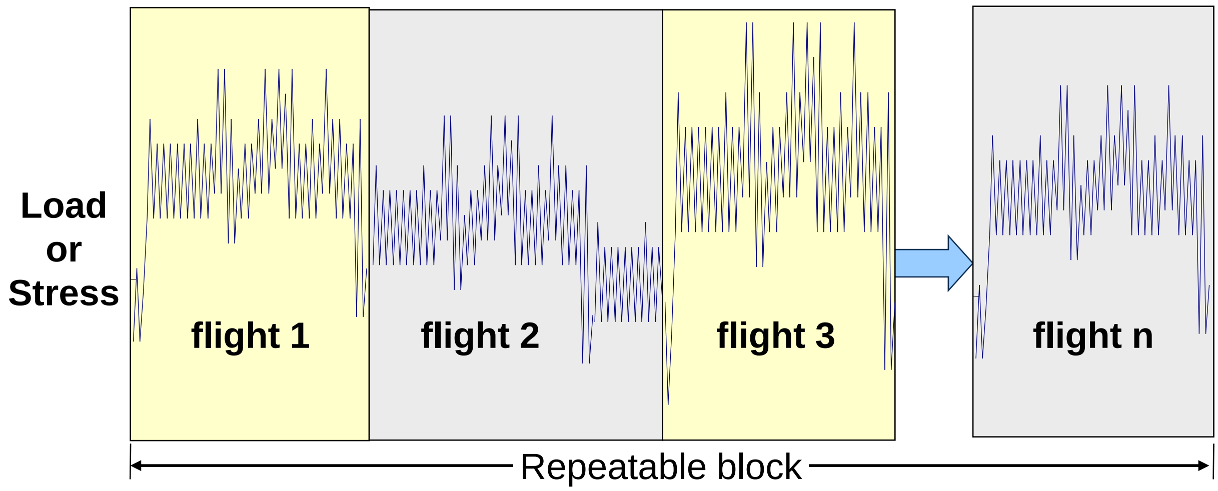

Figure 14.5-4 Flight-by-Flight Random Repeatable Spectrum Block representing an 800 hour, 1/10th Lifetime Repeatable Sequence

Also of interest are the ground-air-ground (GAG) cycles which are the largest cycle that occurs once in a flight. It represents the lowest valley to the highest peak. For transport aircraft they are large contributors to the fatigue of the airframe. Additionally, testing has shown that the GAG cycle causes faster crack growth when the cycle is partially or fully reversed and the valley is in compression.

Load or stress time histories used for durability or damage tolerance analysis are referred to as fatigue spectra. The fatigue spectra can be purely analytically based as is typically the case for new aircraft design. They can also be based, later in the aircraft life cycle, on data obtained from the Loads and Environmental Spectrum Survey (L/ESS) of fielded aircraft. For the analytically based spectra three input components are:

- Mission profiles which depict how the aircraft will be flown. For a given aircraft platform 10-20 missions are defined based on the type of aircraft. Also defined is the percent of time that will be spent on each mission for the life of the airframe. Both of these sets of information are defined in the Structural Design Criteria (SDC) document for the specific aircraft program, derived from a governing specification like JSSG-2006 (Reference 14.9)
- Loads criteria which depict the severity as a function of usage. This is also typically from the SDC.
- Internal loads or stresses from the FEA representing the aircraft structure and stiffness.

While fighter aircraft are maneuver critical because of the high g-levels and amount of time spent in maneuvering the aircraft, transport aircraft (and larger bombers) are typically gust critical. Most of the cycles they see are generated from gust response or continuous turbulence. However, mission profiles are still used as a part of the spectrum definition for these aircraft.

Maneuver loads are treated as quasi-static meaning they are a result of the external forces on the aircraft and not the internal response to those loads. Gust loads are dynamic loads meaning they include the effects of the response of the aircraft's structure to the loads input. Gust spectra development is the statistical determination of the cyclic loading that the aircraft structure will experience under a given usage scenario within the operating environment. A statistical determination is required because the turbulence is defined in terms based on mathematical probabilities and correlated to both test and in-service measured data.

Once the spectra are generated, DADT engineers may make adjustments to the spectra for a number of reasons. To ensure the spectrum is representative of all aircraft in the fleet, they might employ cyclic clipping. To reduce the number of cycles they may perform cyclic truncation.

Cycle clipping or spectrum clipping is reducing the magnitude of the rare high-tensile peaks. Compressive peaks are not clipped. This does not reduce the number of cycles as the cycle with the peak is still present but its magnitude has been reduced. When these high tensile peaks occur they create favorable compressive residual stresses that prolong life. If they are infrequent (low probability of occurrence) these beneficial peaks may not occur on a given aircraft; hence, no prolonged life for some aircraft in the fleet. An analysis without clipping would tend to unconservatively over-estimate the actual lives of the individual aircraft that don't experience these peaks.

Cycle truncation is the removal of cycles which do not cause significant damage and are therefore not necessary. Truncation is done to save cost, time, and data storage requirements. Stress range ($\Delta S = S_{MAX} - S_{MIN}$) is the major driver in fatigue; therefore stress range truncation is the most basic method of truncation. Truncation, as defined here, includes an initial step to delete all intermediate events in a stress sequence. An intermediate event is neither a peak nor a valley. This is illustrated in Figure 14.5-5.

Cycle truncation can be important when planning and performing fatigue testing. Because of the number of cycles involved, testing can take a long time to complete. An analytical truncation study is recommended prior to a spectrum fatigue or spectrum crack growth test. By analysis, a truncation level that will eliminate lots of cycles and thus greatly reduce test time, but not significantly increase the spectrum life in flight hours can be determined. In addition, a coupon test program may be used to confirm analytical results.

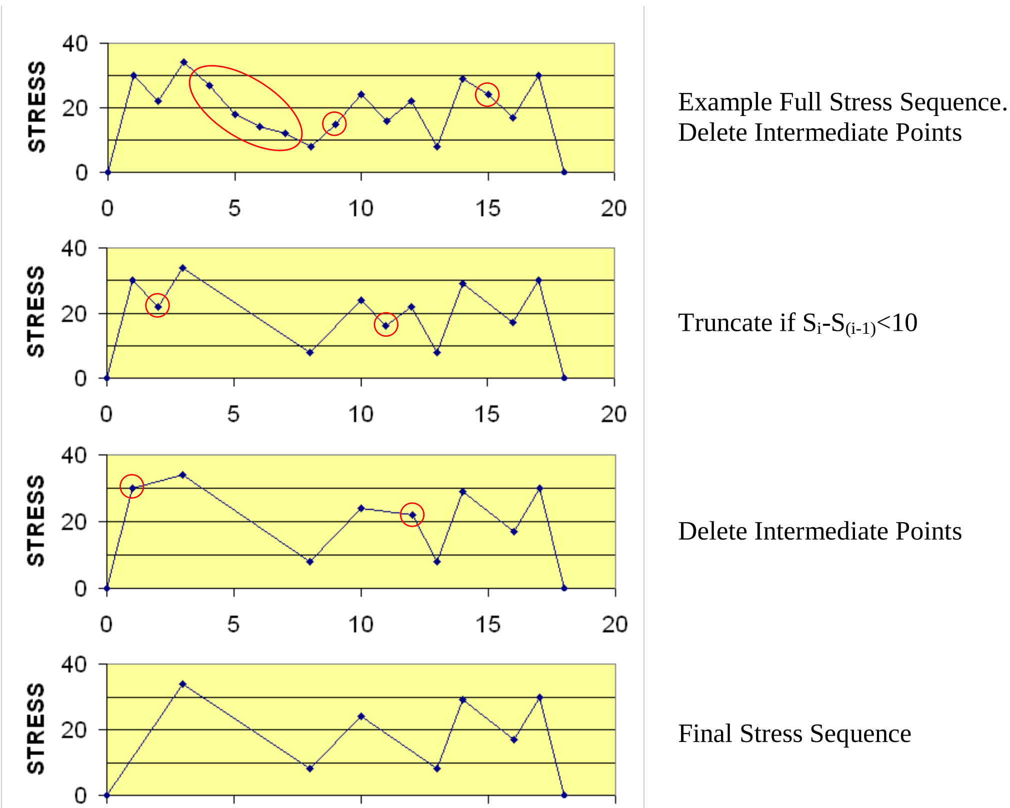

Figure 14.5-5 Example of Cycle Truncation Steps

14.6 Material Properties for Fatigue

Fatigue is the process of progressive localized permanent structural change in a material in the presence of fluctuating stresses. It culminates in cracking of the part or, in the extreme, total failure. The fatigue property of a material is often characterized through the use of an S-N curve. The curve usually represents the mean of fatigue test data, sometimes with slight conservative bias. An example of an S-N curve, taken from Reference 14.5, is shown in Figure 14.6-1.

Sometimes alternating stress is plotted instead of maximum stress in the S-N curve. Note that the number of cycles are plotted on a log scale. The geometric mean[1] life is used in place of the arithmetic mean to be consistent with the use of log scale. Note that the legend indicates what stress concentration value was used in the testing, *i.e.*, k_t=1.0. Some curves will have multiple k_t values plotted and curves are not always available for the desired k_t.

In addition to the curve, the raw data is also shown. This is an indication of the fatigue scatter or the variation in fatigue test data lives with respect to the S-N curve Also note symbols with arrows to the right near the right edge of the plot. These are called run-out values and occur when fatigue tests that are discontinued at a high number of cycles before cracking occurs in the relevant test section.

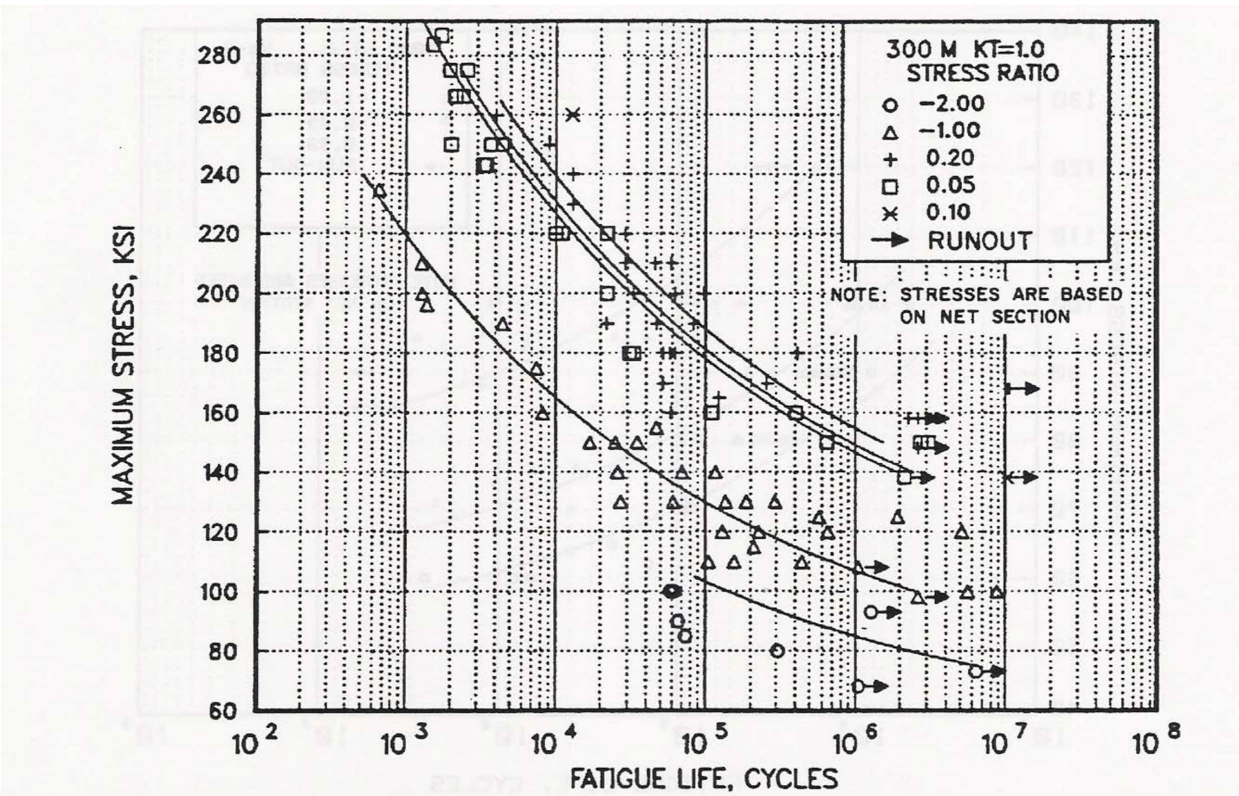

Figure 14.6-1 Example S-N Curve for 300M Steel (Reference 14.5)

The endurance limit of a material is the largest constant amplitude stress below which fatigue life is, for engineering purposes, unlimited. Some materials have clear endurance limits but others, like aluminum, do not. In that case, if a runout value exceeds the range of cycle of interest it is sometimes of sufficient duration to be used as a pseudo-endurance limit.

Unlike static properties which are provided as statistically based 90th or 95th percentile values, fatigue data is presented and used as mean values. Part of the role of the scatter factor (Reference Section 14.4-1) is to account for the variation in test results.

[1] To calculate the geometric mean life of, say, 5 test lives subjected to identical fatigue stresses, multiply the 5 lives together and take the 5th root.

A material property used in damage tolerance analysis is the fatigue crack growth rate, da/dN. Fatigue cracks grow at speeds that are inherent to the material and as a function of the stress intensity factor. The crack growth rate is given by the change of crack length, a, with respect to a number of applied cycles, N and is typically plotted as a function of ΔK or the change in stress intensity factor. Crack growth rate is not constant but changes as a function of crack length, stress level and environment.

The stress intensity factor, K, represents the intensity of stresses and deformations near the crack tip. Its value changes as the crack grows. There are several alternative means of obtaining a representation of K:

- From equations, graphs or tables in standard handbooks or other literature sources
- Handbooks often present K equations in terms of β factors which account for crack length, structural detail and type of loading
- By mathematically combining those handbook K solutions using superposition or approximate techniques
- By advanced techniques such as FEM analysis
- By selection from a list of internal K solutions available in a crack growth program or other fracture mechanics software

Section 17.5.3 defines the test specimens typically used for the determination of material properties for use in durability and damage tolerance analysis. Figure 14.6-2 shows a sample crack growth curve for 1.5 in thick Aluminum 7574-T7351 plate in two different environments.

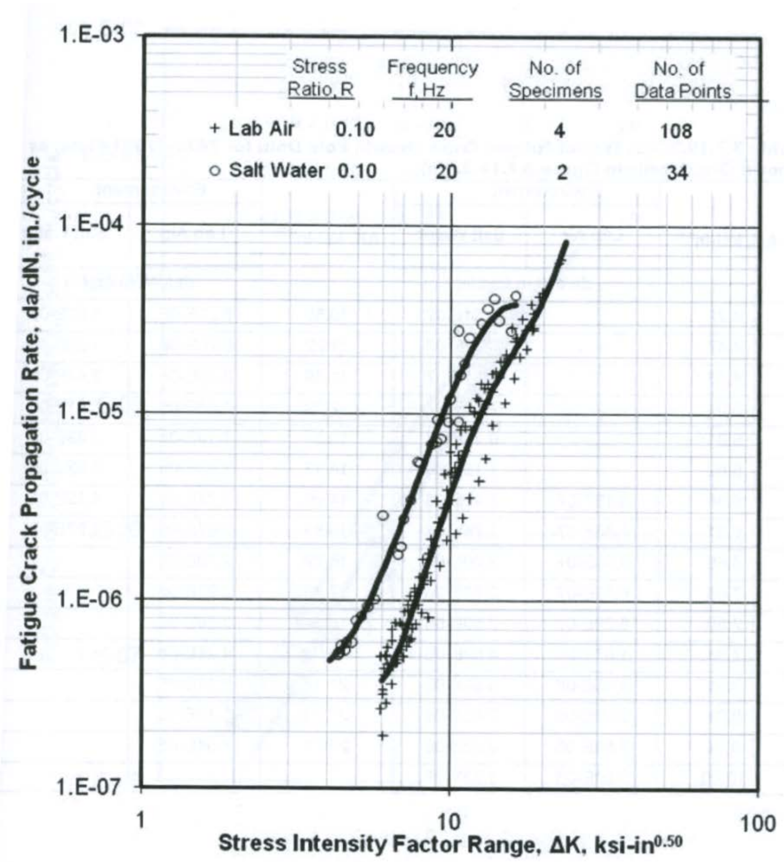

Figure 14.6-2 Stress Intensity Factor versus Crack Growth Rate for 1.5 in thick AL 7475-T7351 Plate (Reference 14.5)

Both of the material properties discussed above are a function of material and temper, specific product form, and grain orientation and care must be used in selecting the appropriate curve.

14.7 Durability Analysis

Durability analysis addresses the economic life of the aircraft. It is concerned with the initiation and growth of cracks. There are two philosophies for calculating the analytical durability life.

- Assume an initial flaw, representing poor initial quality, and grow it to a critical crack length defined as functional impairment. The assumed flaw, in this case, is typically 0.01 inch and the analysis method is described in Section 14.8 as damage tolerance analysis.
- Predict the mean crack initiation life and allow the crack to grow to some small length, typically 0.01 inch. This is commonly referred to as a fatigue analysis method. There are two methods for predicting fatigue life: Stress-Life and Strain-Life.

Stress-Life Analysis: Parts with stresses less than yield exhibit plastic deformation and have relatively long lives. This falls into the realm of high cycle fatigue with the number of cycles greater than 10^7. Transport aircraft see many, many more occurrences of lower elastic stresses in their spectrum due to the gust loading and thus typically use stress life. That isn't to say there aren't some areas of a transport that see high stresses at low frequency.

Stress Life analysis is based on Miner's Linear Cumulative Fatigue Damage Rule which proposes that the total life of a part may be estimated by adding up the percentage of life consumed by each cycle for a given stress level. The stress level includes any stress concentration factors or stress severity factors.

The resulting damage D_i at each stress level, is expressed by the following equation and the total damage, D, is the sum of the damage for each individual stress level.

$$D_i = \frac{n_i}{N_i}$$

$$D = \sum_{i=1}^{Number\ of\ Stress\ Levels} D_i$$

Equation 14.7-1

n_i = the number of applied cycles in spectrum at a stress level
N_i = the number of allowable cycles at same stress level

Failure is expected if D is equal to or greater than 1. The analytical fatigue life is defined as the inverse of D. This process is illustrated in Figure 14.7-1. This methodology is codified in the Legacy Transport aircraft fatigue programs.

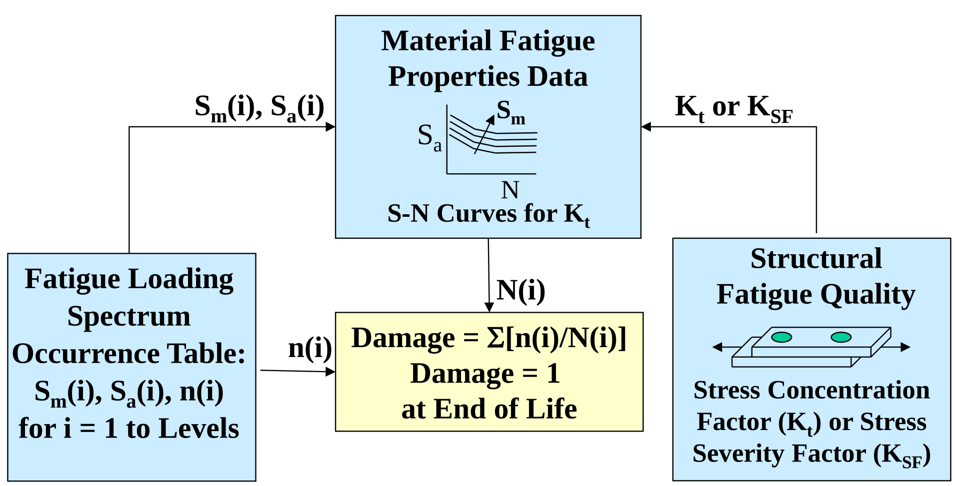

Figure 14.7-1 Analysis Flow for Stress Life Analysis

Strain-Life Analysis: Parts with stresses greater than yield do see plastic deformation and as result have shortened lives. This is low cycle fatigue with the number of cycles ranging from one hundred to 10 million. Fighter airframes see higher stresses in their spectrum due to maneuver loads, often with plastic deformation, and thus typically use strain life.

Figure 14.7-2 is an overview of the strain-life fatigue analysis method. As with the stress-life method data is inputs for the loading spectrum, the material, and the structural location. But for the spectrum and for the material, the data needed for this method are different. For the stress life approach only the number of occurrences are used but, in addition, for the strain-life analysis method the loading sequence is also required.

In the strain-life method, damage is accumulated based on calculated peak and valley occurrences of the local elastic-plastic stress and strain at the edge of the notch. So two types of material properties are required:

- Strain-Life Curve: Fatigue data plot of life as a function of the local cyclic fatigue strain
- Cyclic Stress-Strain Curve: Elastic-plastic stress-strain relationship developed specifically from cyclic (rather than static) testing

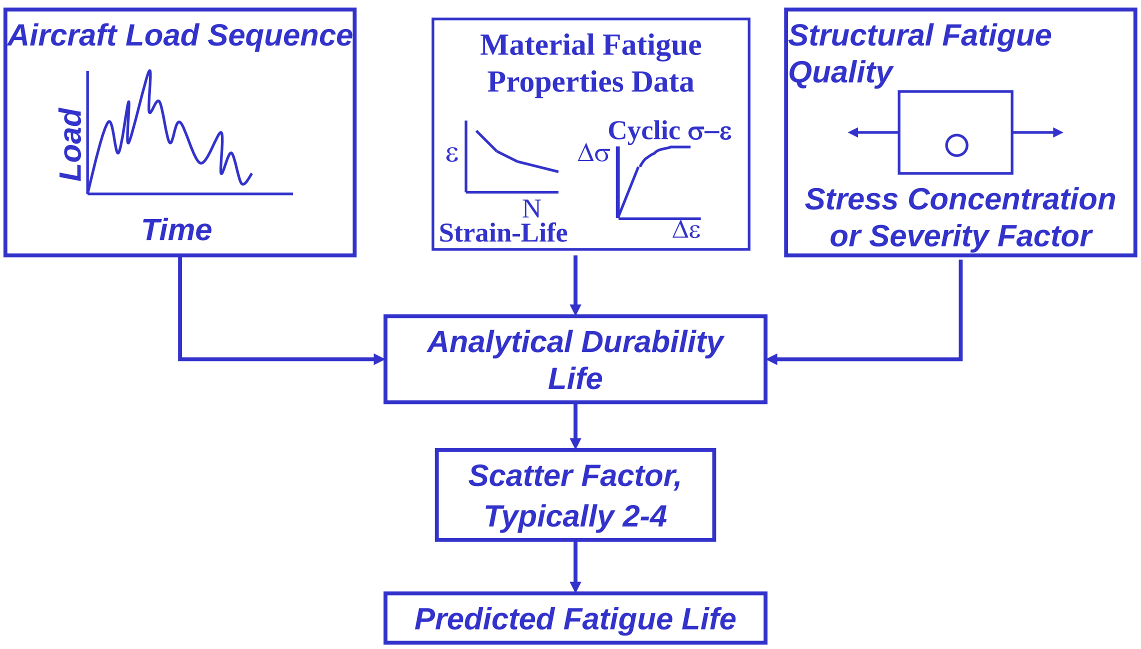

Figure 14.7-2 Analysis Flow for Strain-Life Analysis

There are two primary strain-life computer programs at Lockheed Martin:

- LOOPIN: F-22, F-35, F-16
- FAMS: P-3

While they both use the same basic theory there are some key differences between them.

14.7.1 Stress Concentration Factors

A hole or other geometric irregularity in a uniform tensile stress field causes the stress to magnify or concentrate near the hole or irregularity. In general the location of the largest magnified stress is at the edge of the irregularity. Assuming linear elastic stresses the k_t or stress concentration factor can be defined as

$$k_t = \frac{\sigma_{max}}{\sigma}$$ **Equation 14.7-2**

where
σ_{max} is the maximum stress (psi)
σ is the farfield stress (psi)

A stress concentration factor, k_t, is an indicator of how much the stress peaks at the edge of a hole, radius, defect, *etc.*, and is used to determine when cracking will start. A high k_t in the structural part is an indicator of a likely fatigue problem. The concept of a k_t is shown in Figure 14.7-1 for a hole in a plate. Note the stress at the edge of the hole under a uniaxial loading is three times the farfield stress. Tabulated values for stress concentrations are available through a number of references such as Reference 14.10.

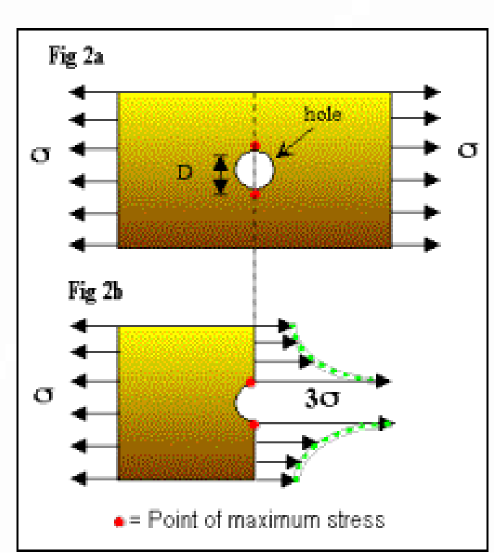

Figure 14.7-3 Illustration of a Stress Concentration and the Peaking of Stresses at a Hole in a Plate

14.7.2 Stress Severity Factor

Stress Severity Factors, K_{SF}, are equivalent stress concentration factors at bolted joint holes. They are the ratio of the total peaked stress at the hole to the nominal reference stress. The total peaked stress is comprised of the peak bearing stress plus the peak bypass stress plus additional factors for hole condition, hole fill, and tilt. Both the peak bearing and peak bypass stresses peak at the hole centerline. The loading is illustrated in Figure 14.7.2. And the equation for K_{SF} is given by Equation 14.7-3

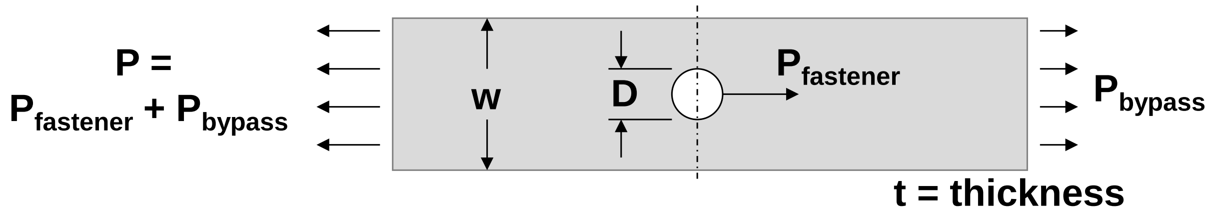

Figure 14.7-4 Loading and Geometry in a Bolted Joint

The peak bearing stress is the load transferred by the fastener divided by the product of the thickness of the material and the diameter of the fastener.

The peak bypass stress is the load passing the hole to be transferred by fasteners downstream in the joint, divided by the product of the width and thickness of the material.

The nominal reference stress is the farfield stress.

$$\sigma_{brg} = \frac{P_{fastener}}{tD}$$

$$\sigma_{bypass} = \frac{P_{bypass}}{wt} \qquad \textbf{Equation 14.7-3}$$

$$\sigma_{ref} = \frac{P}{wt}$$

$$K_{SF} = {\alpha\beta\left(K_{tb}\theta\sigma_{brg} + K_{tg}\sigma_{bypass}\right)}\Big/{\sigma_{ref}}$$

where
P is the total load (lbs)
$P_{fastener}$ is the fastener load (lbs)
P_{bypass} is the bypass load (lbs)
t is the plate thickness (in)
D is the fastener diameter (in)
w is the plate width (in)
K_{tb} is the bearing stress concentration factor
K_{tg} is the bypass stress concentration factor
α is the hole condition factor, typically 0.7 to 1.0
β is the hole filling factor, typically 1.0 for open holes, 0.75-0.90 for threaded bolts and 0.50 for Taper-Loks
θ is the fastener tilt factor which is a function of t/D and whether the joint is single or double shear

Depending on the loading and geometry, there may be additional factors included in the calculation of K_{SF}.

14.8 Damage Tolerant Structure and Damage Tolerance Analysis

There are two classes of damage tolerant structure:
- Slow Crack Growth Structure
- Fail Safe Structure

The residual strength of a structure is remaining strength of the structure after degradation due to cracking or failure of an element. The residual strength load level has been defined as the “maximum of limit load or the maximum load in 20 lifetimes.” The current definition per Reference 14.8 is the maximum load in 10 million flights.

Slow Crack Growth Structure is structure where flaws or defects are not allowed to attain critical length required for unstable rapid crack growth. This is depicted in Figure 14.8-1 for non-inspectable structure and in Figure 14.8-2 for inspectable structure for the initial inspection increment.

In the slow crack growth scenario, an initial defect that remains undetected, shall not grow to critical crack size for the designated residual strength load in less than the required period of unrepaired service usage. Safety is assured through slow crack growth for periods of usage depending on the degree of inspectability. For structure which cannot be inspected, the structure is designed with no inspection required.

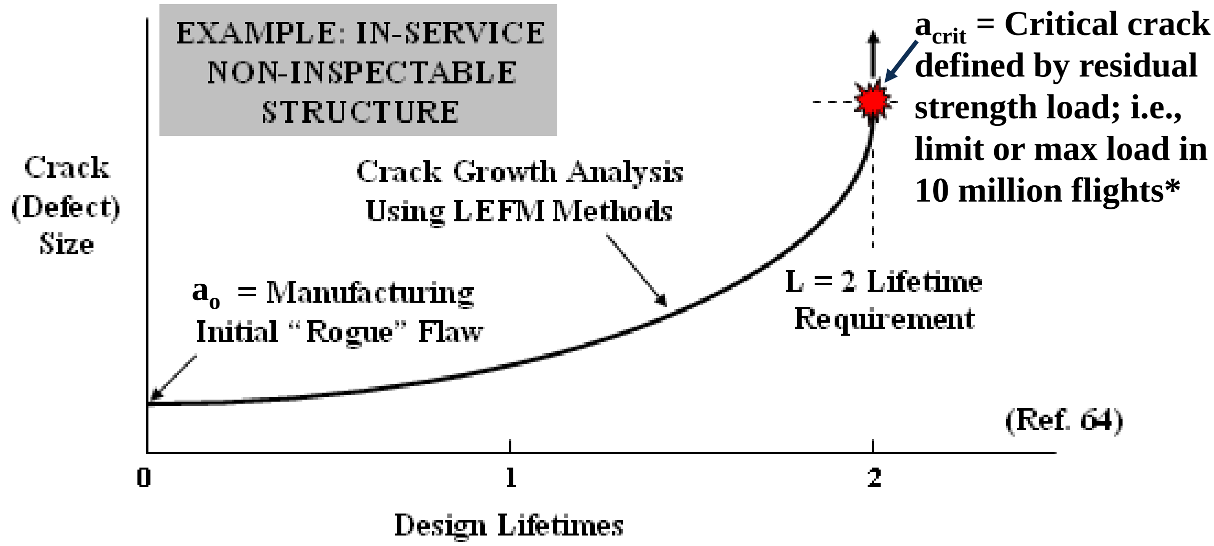

Figure 14.8-1 Depiction of Slow Crack Growth Non-Inspectable Structure

For slow crack growth inspectable structure, in-service inspections are planned and the DADT analyst must specify those inspection intervals. The initial interval is typically different than subsequent intervals. The initial interval is based on one-half the time it takes to analytically grow the crack from the assumed initial flaw size to critical crack length.

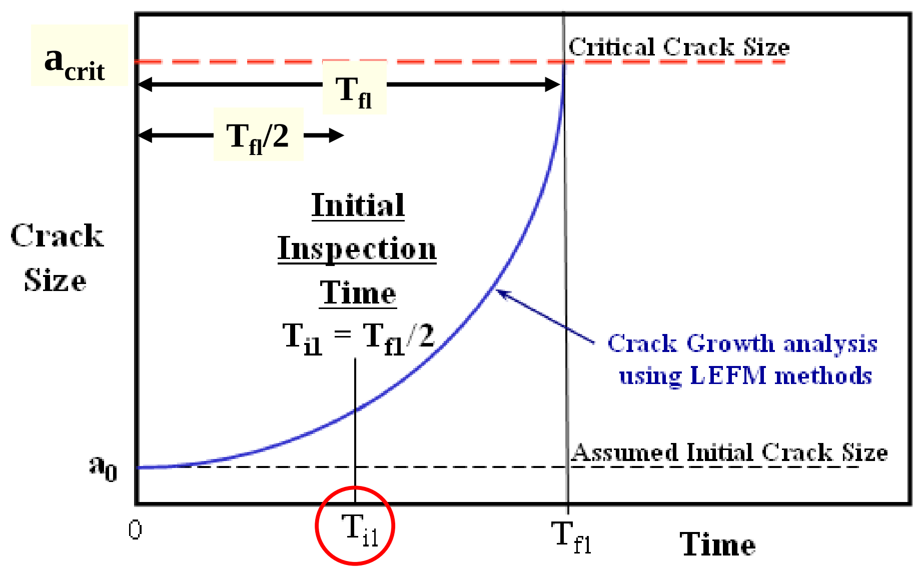

Figure 14.8-2 Depiction of Slow Crack Growth Inspectable Structure – Initial Inspection Increment

The initial defect is presumed to be an undetected manufacturing flaw. This flaw is called a **rogue flaw** and it is assumed to be in the worst location and is statistically rare in occurrence. Initial flaw sizes for the rogue flaw are a part of the aircraft's criteria. Some standard sizes of flaws are depicted in Figure 14.8-3.

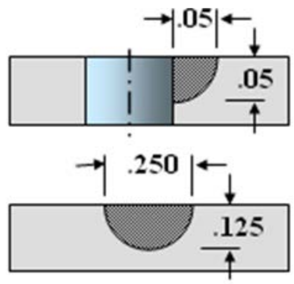

Figure 14.8-3 Standard at Hole (Top) and Remote (Bottom) Assumed Rogue Flaw Sizes

The standard flaw sizes are specified in Reference 14-9 and represents a flaw that is both highly improbably but also what could reasonably expect to be found using standard inspection techniques. With a successful NDI demonstrations, smaller initial flaw size standards can be assumed, with customer concurrence.

Critical crack length is material and part-geometry dependent. It is called a_{crit} or sometimes c_{crit}. The required period of unrepaired service usage, at the beginning of an aircraft program is typically the life of the aircraft as the contract often requires this for new aircraft. Later in the program or for a Service Life Extension Program this might be defined as something different based on service experience and inspections may be more palatable to the customer.

Under the slow-crack growth scenario, the sub-critical damage cannot degrade the strength of the structure below the specified residual strength level.

Fail-Safe Structure is designed and fabricated such that unstable rapid propagation of cracks will be stopped within a continuous area of structure prior to complete failure. Safety is then assured through slow crack growth in the remaining structure and detection of damage and repair on subsequent inspections. If the airframe should exhibit widespread fatigue damage the structure is no longer failsafe.

Fail-safety, in principle, is a very good approach; however, in practice it can be difficult to achieve and demonstrate through test. The basic premise is that the primary structural element may fail at any time. There are no crack growth requirements for the primary element. The remaining adjacent element has assumed initial damage and when the primary structure fails, the adjacent element must be able to sustain the specified residual strength load without failure with any assumed initial damage plus crack growth up to the point of primary element failure. It must be able to carry this load and maintain structural integrity until the failed condition is found.

Another key element of fail safety is the ability to inspect the primary element. The primary element failure must be either readily apparent from in-flight or post-flight observations or in an obvious safety-scheduled special visual inspection. Or it must be discoverable due to a malfunction of other systems which would alert the flight personnel or ground crew that a primary element failed.

Damage tolerance analysis, at a very high level, involves assuming a rogue flaw is present in the worst location on a given part and based on that crack length, determining the stress intensity factor, ΔK. This value for ΔK is used along with the material crack growth curve to determine the change in crack length, Δa for one cycle. This can be added to the existing crack length to determine the new length and a cycle is added to the sum of cycles. If this length exceeds the critical crack length as determine by a residual strength analysis, then the analytical life is the number of cycles that have been summed up. If not, the process is repeated. For a spectrum loading, the process is more complicated as the stress spectrum of the individual element is used. This is illustrated in Figure 14.8-4.

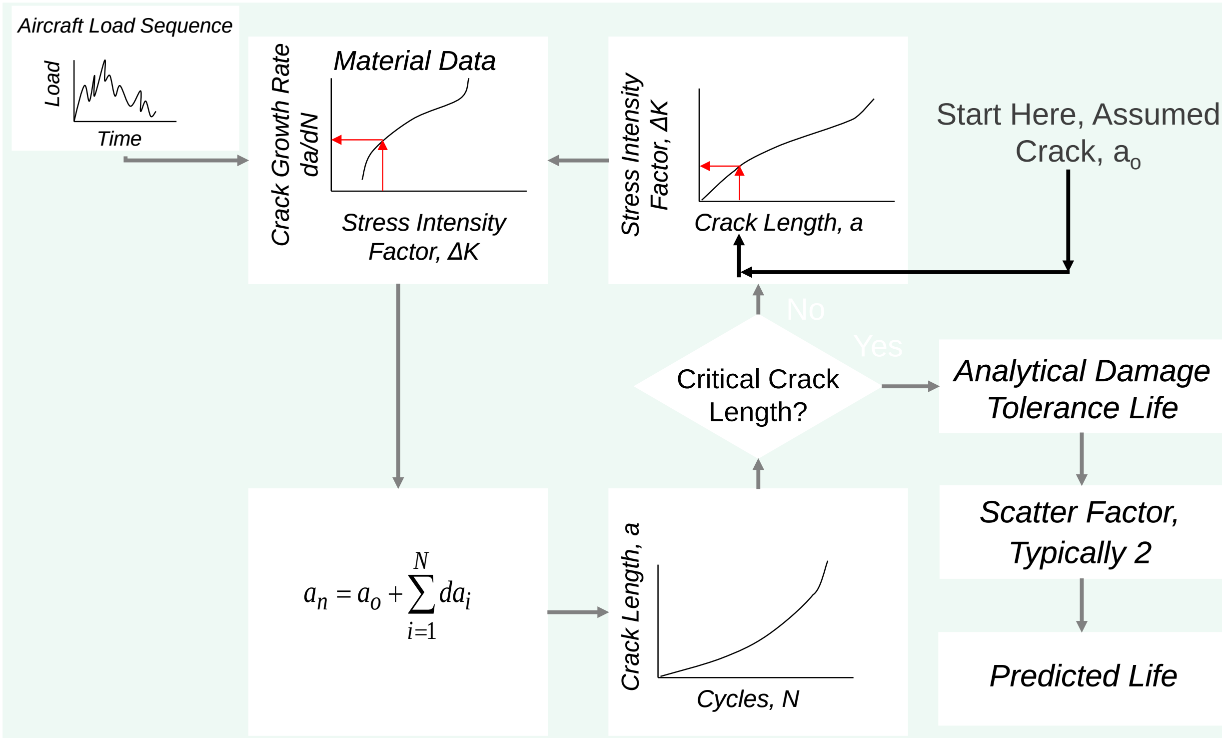

Figure 14.8-4 Damage Tolerance Analysis Flow

Not all structural parts require a damage tolerance analysis. Considerations for selecting locations where analysis should be performed include all safety-critical structure, *i.e.*, structure classified as fracture critical, and on durability critical structure where there are high cyclic tensile stresses or high stress concentration values. Areas where there are questionable design details such as large changes in section or multiple stress concentrations may also be analyzed, along with cases where there is a material review issue that might increase a stress concentration, lead to a bad design detail or result in tensile residual stresses.

Once the durability testing is complete, areas that exhibited cracking in test may be additional candidates for damage tolerance analysis. If there is reported cracking in service, those areas might also be subjected to a damage tolerance analysis.

14.9 Widespread Fatigue Damage

Widespread fatigue damage (WFD) is the simultaneous presence of cracks at multiple structural locations that are of sufficient size and density such that the structure will no longer meet the residual strength requirements. While it was known before the Aloha Airlines Flight 243 incident in April 1988, after that time the Federal Aviation Administration became proactive in studying and regulating for WFD. Design features should provide provisions to limit the probability of concurrent multiple damage, particularly after long service, which could conceivably contribute to a common fracture path. Examples of such multiple damage are:

- A number of small cracks which might coalesce to form a single long crack (Multiple Site Damage)
- Failures, or partial failures, in adjacent areas, due to the redistribution of loading following a failure of a single element; and
- Simultaneous failure, or partial failure, of multiple load path discrete elements, working at similar stress levels (Multiple Element Damage)

Multiple site damage (MSD) is a source of widespread fatigue damage characterized by the simultaneous presence of fatigue cracks in the same structural elements.

Multiple element damage (MED) is a source of widespread fatigue damage characterized by the simultaneous presence of fatigue cracks in similar adjacent structural elements. Key to this being a threat to the aircraft is that the adjacent structural elements are working at approximately the same stress levels.

Each FAA certified aircraft program must now establish a limit of validity for that model line. The **Limit of Validity** (LOV) of the engineering data that supports the maintenance program is the period of time (in flight cycles, flight hours, or both), up to which it has been demonstrated by test evidence, analysis and, if available, service experience and teardown inspection results of high-time airplanes that widespread fatigue damage is unlikely to occur in the airplane. This is the number of cycles, given no other data, when the airplane faces retirement.

14.10 High Cycle Fatigue

The S-N curve shown in Section 14.6-1 has a maximum number of cycles of approximately 10^7. This is considered to be in the low cycle fatigue realm and most structure on aircraft fall into this category. However, high cycle fatigue can occur in aircraft due to aero-acoustic loading or in response to dynamic loads, such as would be found in electronic equipment or its mounting. For this type of fatigue the number of cycles may be 10^{11} or more, of varying magnitudes. Similar analysis methods are used to analyze high cycle fatigue in these applications as what was explained under Stress-Life analysis in Section 14.7. Extended S-N curves may be required for the material properties. An example of such a curve is shown in Figure 14.10-1

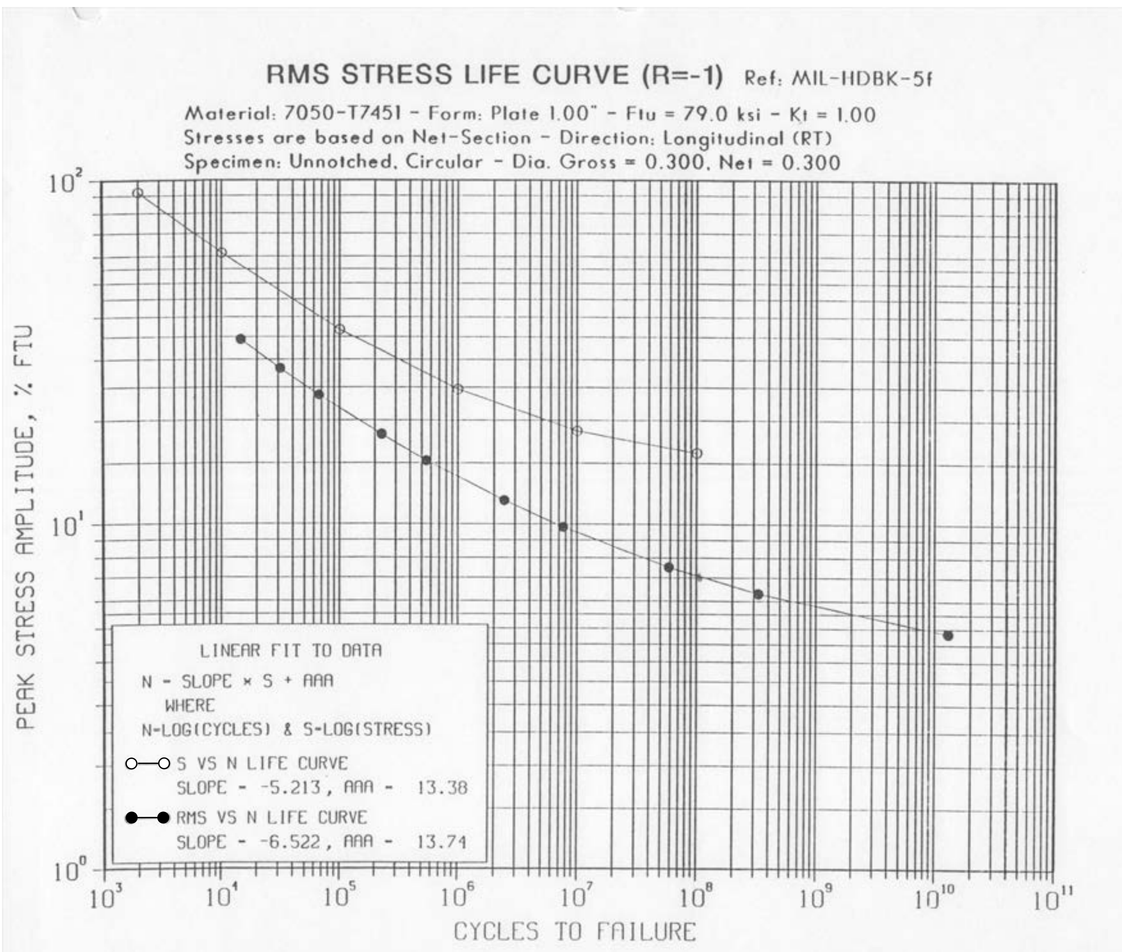

Figure 14.10-1 Sample High Cycle Fatigue Material Property Curve for 7050-T7451 Plate, 1 in thick

Reference 14-11 addresses high cycle fatigue calculations for electronic equipment. The approach recommended is Miner's Linear Cumulative Fatigue Damage Rule that proposes that the total life of a part may be estimated by adding up the percentage of life consumed by each cycle for a given stress level.

The resulting damage, D_i, at each stress level is given by the number of applied cycles at that stress level divide by the number of cycles for that stress level from the material property curve. The total damage is the summation of the individual D_i values. This was shown in Equation 14.7-1. Failure of the part is predicted if D is equal to or greater than one. With the addition of a scatter factor, the practical limit for the summed ratio is one divided by the scatter factor. For example if the scatter factor were 4, then the summation of D_i has to be less than or equal to 0.25 to satisfy the requirements. An example calculation is shown in Figure 14.10-2. In this example the allowable value for N was taken from a material curve generated from test data for the specific material/joint design.

	Tension	Shear						
Dynamic Design Load (Applied)	334	334	lbs					
Applied G's	4.2	3	1.8					
% Dynamic Design Load	No Cycles, Vertical	No Cycles, Transverse	No Cycles, Longitudinal	Load (lbs)	n =max of (Vertical or Transverse Cycles)	Allowable N From test data (based on stress level)	$D_i = n/N$	
10	107000	132000	84800	33	132000	30892789	0.00427	
20	29200	25500	24200	67	29200	9543644	0.00306	
30	19200	5330	20800	100	19200	2948298	0.00651	
40	4540	2850	14700	134	4540	910811	0.00498	
50	2180	2050	4800	167	2180	281375	0.00775	
60	1180	1230	2140	200	1230	86925	0.01415	
70	580	600	1080	234	600	26853	0.02234	
80	100	200	430	267	200	8296	0.02411	
90	100	100	350	301	100	2563	0.03902	
100	100	50	150	334	100	792	0.12631	
						Summation	**0.25**	**D**
						D≤0.25: Acceptable life with a scatter factor of 4		

Figure 14.10-2 Example Miner's Rule Calculation for Equipment Mounting based on Dynamic Loading

14.11 Best Design Practices For Long Life Aircraft

Designing an aircraft for long life is all about paying attention to detail. It starts with material selection and choosing a material that performs well under cyclic loading, has stable crack growth characteristics and that isn't susceptible to the many types of corrosion including exfoliation and stress corrosion cracking. Refer to Section 3.4 for a discussion of corrosion. Part of the requirements of the ASIP Master Plan per Reference 14.7 is to develop an approved material list and a corrosion control plan. So for the individual analyst or designer, usually there is program guidance available on the preferred materials within the approved materials for use in durability or fracture critical parts.

Utilizing good design practices discussed in other sections of this manual and PM4007 the Design Manual are also important:

- Make sure there are well thought-out load paths
- No unintentional eccentricities

- Minimize intentional eccentricities or support them
- All changes in load path direction (kicks) should be supported
- Avoid abrupt changes in cross-section which will naturally concentrate the load
- Avoid abrupt growths or add-ons to structure to avoid rapid changes in stiffness or load concentration
- Avoid long unsupported spans
- For brackets and support structure, avoid diving boards (long flexible cantilevered beams). Even with minimal applied load, high cycle fatigue due to vibration loading could cause failure.

Manufacturing and Finishing Details

- Surface Finish should be a minimum of RA 125 or better. (Better is a smaller number)
- Beware of cold forming as residual stresses can be locked into the part reducing life
- Do not pull up gaps which induce residual stresses but rather shim to within 0.005 inch
- Avoid hard plating such as sulfuric acid anodize. These coatings can crack and propagate into the material
- Beware of dissimilar materials and use barrier materials and sealants to prevent galvanic corrosion.

Design details should be carefully examined

- No square holes
- Make large holes oval (Reference Figure 18.4.22 and Figure 18.4.22-1)
- Avoid sharp notches: make radii as large as possible
- Do not have multiple adjacent radii as the k_t's will multiply
- Use generous radii at corners, transitions, steps, *etc.*
- Break all sharp edges with a chamfer. Feather edge can become damaged and cracked, propagating into the structure
- Do not use impression stamping

Joints and Holes

- Do not use sloppy-fit fasteners in structural holes. Anything greater than 0.004 diametrical clearance is sloppy and results in the potential for fastener fatigue, head failures and fretting fatigue in the hole and in the faye surface.
- No knife edge countersinks, *i.e*,. countersinking beyond 80% of material depth, as this has been a source of cracking in older aircraft.
- Do not use counterbores or spotfaces as the corner radii, by design are sharp notches. Instead use radius block to preclude interference between fasteners and adjacent part radii.
- Be sure structure is destacked and deburred where practically possible. See Section 18.4.12 for more discussion. If it cannot be, because of assembly sequence, areas must be specifically noted on the drawing and additional factors will need to be used in the durability analysis.
- Do not use large non-structural shims, where the thickness is greater than 0.15D to 0.25D, because it can cause greater working of the joint resulting in fretting, bolt bending and bolt fatigue.
- Wet install fasteners
- Use fay surface sealant to reduce fretting
- Be sure bolts are torqued to an appropriate level for the type of joint. Refer to Section 5.2.3.1.3.
    - For shear joints, bolt torque reduces joint motion and fretting fatigue and prevents nut back off and loosening of the joint
    - For tension joints, the preload minimizes the cyclic tension load on the bolts and bolt fatigue.

Many of these items can occur in parts designed in CATIA 3D models without anyone realizing it. It is crucial that a first article inspection of every machined part, irrespective of material product form, occur with a joint team of engineers from design, stress, DADT and Materials and Process organization so that bad details can be caught and fixed before production.