AOP MECHANICAL ASSEMBLY FATIGUE PLAN

PREPARED BY	ORGANIZATION	DATE
Edwin Penniman	AOP	04/09/2015

APPROVALS	ORGANIZATION	APPROVAL DATE
Hanne Buur	PSE	07/10/2015
Michael Rodriguez	RAMS	07/13/2015

RELEASED BY	ORGANIZATION	RELEASE DATE
Jennifer DeNicholas	CM	07/14/2015

See configuration management system for approval history.

The National Ecological Observatory Network is a project solely funded by the National Science Foundation and managed under cooperative agreement by NEON, Inc. Any opinions, findings, and conclusions or recommendations expressed in this material are those of the author(s) and do not necessarily reflect the views of the National Science Foundation.

Change Record

REVISION	DATE	ECO #	DESCRIPTION OF CHANGE
A	07/14/2015	ECO-02882	Initial Release

TABLE OF CONTENTS

LIST OF TABLES AND FIGURES

1 DESCRIPTION

1.1 Purpose

The purpose of this document is to establish a plan for assessing stress loads on components and assemblies due to mechanical fatigue and to develop the plan for monitoring and inspecting components and assemblies to ensure that failure of mechanical components does not occur due to fatigue. This plan has been developed by the NEON Airborne Observation Platform (AOP) Team to address fatigue issues associated with AOP Payloads and ground support equipment. The major objectives of this plan are:

- To derive and define a plan which is designed to minimize the probably of structural failure due to cyclic loading, including vibration
- To define what needs to be on a proof loading or non-destructive testing schedule
- To define how assemblies are to be proof loaded and inspected
- To define when assemblies need to be inspected and proof loaded
- To define who does the inspection and proof loading

1.2 Scope

This fatigue plan applies to all mechanical assemblies with a single point of failure. Even though fatigue usually occurs to a high number of loading cycles and low stress, this plan also applies to assemblies with high stresses and/or assemblies with a low number of loading cycles.

The 3 risks to the longevity of any mechanical design are:

- Corrosion
- Wear
- Fatigue

This document only addresses fatigue.

2 RELATED DOCUMENTS AND ACRONYMS

2.1 Applicable Documents

Applicable documents contain information that shall be applied in the current document. Examples are higher level requirements documents, standards, rules and regulations.

AD [01]	NEON.DOC.015015 AOP-1 Payload Integration Mount Design Report
AD [02]	
AD [03]	
AD [04]	

2.2 Reference Documents

Reference documents contain information complementing, explaining, detailing, or otherwise supporting the information included in the current document.

RD [01]	NEON.DOC.000008	NEON Acronym List
RD [02]	NEON.DOC.000243	NEON Glossary of Terms
RD [03]		
RD [04]		

2.3 External References

External references contain information pertinent to this document, but are not NEON configuration-controlled. Examples include manuals, brochures, technical notes, and external websites.

ER [01]	Material and Design Optimization for an Aluminum Bike Frame, Worcester Polytechnic Institute Thesis, Dweyer, Shaw, Tombarelli, April 26th 2012
ER [02]	Elements of Metallurgy and Engineering Alloys, ASM International #05224G, 2008
ER [03]	DEPARTMENT OF DEFENCE DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION AERONAUTICAL RESEARCH LABORATORY MELBOURNE, VICTORIA Technical Report 15 HELICOPTER STRUCTURES - A REVIEW OF LOADS, FATIGUE DESIGN TECHNIQUES AND USAGE MONITORING
ER[04]	Gigacycle Fatigue Behavior of High Strength Aluminum Alloys QY Wang,*, T Lib, XG Zeng, Procedia Engineering 2 (2010) 65–70
ER[05]	ANALYSIS OF METHODS FOR DETERMINING HIGH CYCLE FATIGUE STRENGTH OF A MATERIAL WITH INVESTIGATION OF Ti-6Al-4V GIGACYCLE FATIGUE BEHAVIOR DISSERTATION Randall D. Pollak, Major, USAF AFIT/DS/ENY/06-07 DEPARTMENT OF THE AIR FORCE AIR UNIVERSITY AIR FORCE INSTITUTE OF TECHNOLOGY Wright-Patterson Air Force Base, Ohio APPROVED FOR PUBLIC RELEASE; DISTRIBUTION UNLIMITED
ER[06]	Elements of Metallurgy and Engineering Alloys (#05224G), American Society for Metals
ER[07]	http://www.efunda.com/formulae/solid_mechanics/fatigue/fatigue_factor.cfm
ER[08]	Shot Peening A P P L I C A T I O N S NINTH EDITION METAL IMPROVEMENT COMPANY

	A S u b s i d i a r y o f C u r t i s s - W r i g h t C o r p o r a t i o n
ER[09]	IMPROVING THE FATIGUE RESPONSE OF AEROSPACE STRUCTURAL JOINTS Cindie Giummarra and Harry R. Zonker Alcoa Inc., Alcoa Technical Center, Pittsburgh, Pennsylvania, USA
ER[10]	Rooke, D.P. and Cartwright, D.J. (1976). Compendium of stress intensity factors. HMSO Ministry of Defence. Procurement Executive
ER[11]	Estimating Fatigue Curves With the Random Fatigue-Limit Model, Pascual and Meeker, 1999
ER[12]	Fatigue Data Book: Light Structural Alloys, ASM International
ER[13]	Charles Annis, PE, Statistical Engineering, http://www.statisticalengineering.com/random_fatigue_limit.htm
ER[14]	MIL-HDBK-5
ER[15]	http://en.wikipedia.org/wiki/Almen_strip
ER[16]	Summary of Stress Intensity Factors, Alan Liu, Rockwell International

2.4 Applicable Requirements

DOORS Requirement ID	Requirement Title

2.5 Acronyms

AOP	Airborne Observation Platform	WLL	Working Load Limit
SCC	Stress Corrosion Cracking	RFL	Random Fatigue Limit

3 INTRODUCTION

Fatigue is the weakening of a material caused by periodic loads. It is estimated that 90% of structural failures are due to fatigue (ER[02]). Fatigue is cumulative, meaning that the damage adds up over time, eventually leading to failure, except in cases with certain materials when stress is below a fatigue limit (also known as the endurance limit). The existence is still a subject of a debate; one argument is that everything will crack, given enough time and another argument is that unless the stress is high enough to alter molecular bonds, then the structure will last indefinitely. One example would be naturally occurring crystals, which are subjected to stresses from Brownian motion, which is at a very high frequency, yet the crystals do not crack millions of years after formation. Research acknowledges that some materials such as steel and titanium have an endurance limit where “Below a certain stress level, the steel alloy will never fail due to cyclic loading alone.” [ER02]. There is no theorized fatigue limit for other materials such as aluminum because new, longer fatigue tests expose cracking at a higher number of cycles. Presently, there are no tests which go beyond 10^10 cycles. The variability in the test results leads to further uncertainty on whether or not a fatigue limit exists for some, if not all, materials.

Around 1900, the Wohler curve, also known as the S-N curve was introduced. An S-N curve is shown for steel and aluminum inFigure 1; it shows the maximum allowable stress (or, stress amplitude as shown in

Figure 1), S, for a material subjected to N cycles. Materials scientists theorize that steel has a fatigue limit, but not aluminum.As you can see from Figure 1, the S-N curve for aluminum never levels off. This means that it is not possible to design an aluminum structure with the comfort of it never cracking.

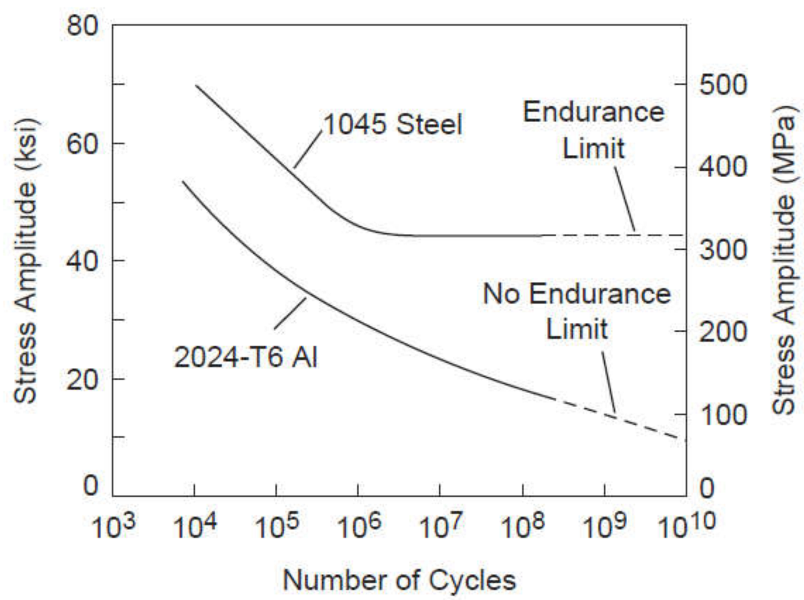

Figure 1 - S-N Curve examples for steel and aluminum ER[06]

Even though aluminum will fatigue, given enough cycles and stress, it is still used for bicycle frames and rock climbing carabiners. Over time, the industry has responded to bicycle frame cracking by modifying the frame design to lower stress to a level which equates to 490,000 cycles [ER1 pg. 80]. For carabiners, the solution is to require the user to replace the carabiner after only 1 fall. This is due to the carabiner having a single point of catastrophic failure. In helicopter blade design, the retirement lifetime of the blade is calculated using the Safe Life methodology as shown in Figure 2

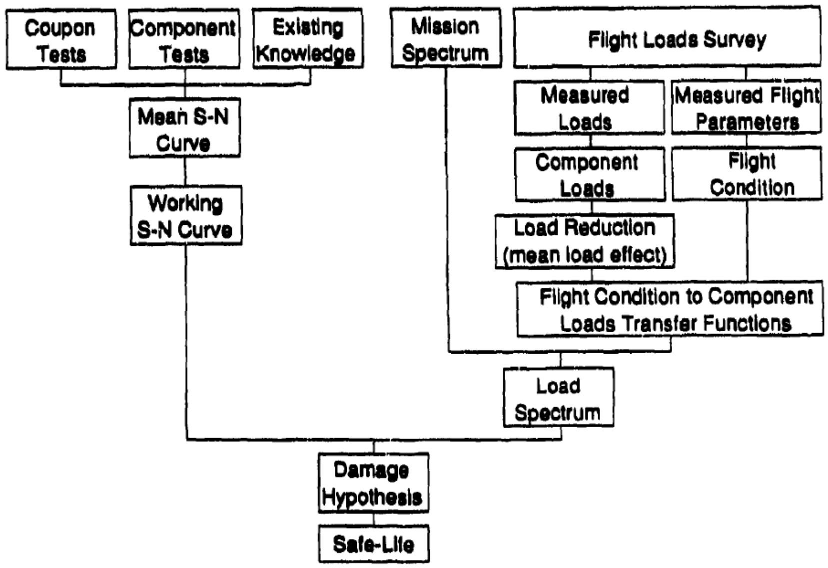

Figure 2 - Safe Life Design Methodology [ER3 pg. 29]

4 FACTORS AFFECTING FATIGUE RESISTANCE

4.1 MICROPOROSITY

Microporosity is controlled for standard billets and stock, but it can become an issue for custom castings.

4.2 ENVIROMENT

It is usually assumed that the fatigue would take place in air, but it makes a great difference, in the lifetime of the structure, if it is immersed in water. For corrosive environments, refer to Secion 4.7 on stress corrosion cracking.

4.3 SURFACE

4.3.1 FINISH

Based on the theory of crack growth rates in ER[06], the crack growth rate is dependent on the initial crack depth. In general, cracks usually initiate at the surface (ER[06]), so one would expect there to be a correlation between roughness and fatigue life span. A small study on the effect of surface roughness is shown in Figure 3; unfortunately, the study didn't state the exact roughness metric such as RMS, peak-to-valley, etc... This study was based on a stress of 665MPa and SAE 3130 steel.

Finish	Median Fatigue Life (kilo-cycles)	Surface Roughness (µm)
Lathe-formed	24	2.67
Partial hand-polished	91	0.15
Hand-polished	137	0.13
Superfinished	212	0.18
Ground	217	0.18
Ground and polished	234	0.05

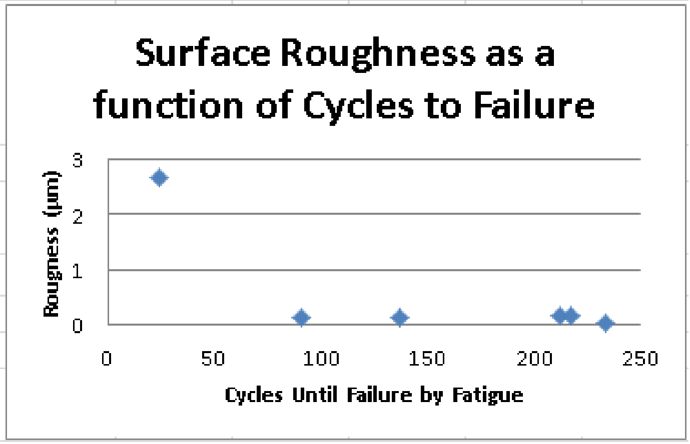

Figure 3 - Fatigue Lifespan as a function of Surface Roughness [ER07]

As you can see, there is a bulk dependence on surface roughness, but there are other factors influencing the fatigue lifetime.

4.3.2 SHOT PEENING

Shot peening creates compressive stresses on the surface of a part by shooting a spray of hard, round particles at it. Shot peening originated from blacksmiths extending the life of leaf springs by hitting the entire concave surface with a ball-peen hammer. Sometimes, shot peening has been reported to increase the fatigue life of a metal part by up to 10x (ER[07]).

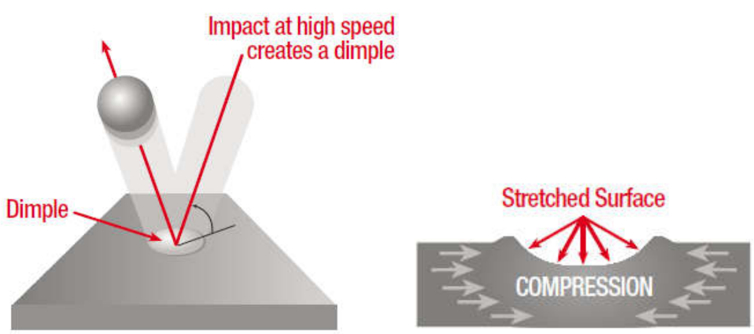

Figure 4 - Shot Peening Illustrations of Mechanics and Stress Formation

The benefits of shot peening are dependent on the material. The parameters of the peening process are also important; these include:

- Air pressure
- Shot material
- Shot diameter
- Shot mass flow
- Nozzle diameter

Below is an example of the improvement in fatigue life of CK45 steel due to shot peening.

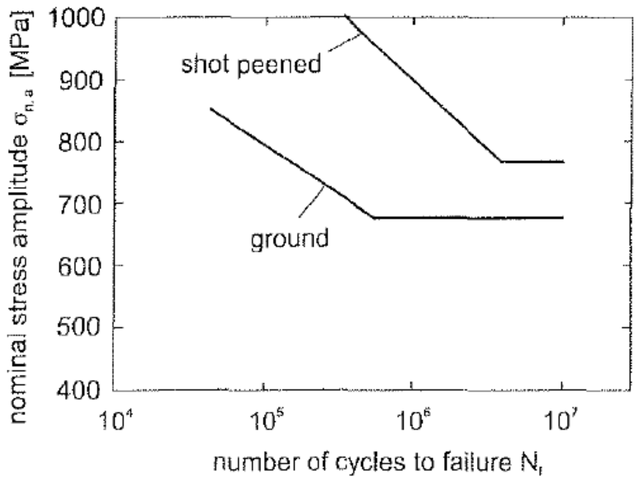

Figure 5 - CK45 Steel Shot Peening vs. Untreated S-N Curves

In conclusion, shot peening is a valuable resource for the improvement of fatigue life. To correctly estimate the fatigue life of a particular part design, one will need to identify an S-N curve of the same alloy and to implement quality controls, such as an Almen Strip (ER [15]), to ensure that adequate surface compressive stresses are developed during the peening process.

4.3.3 ANODIZATION

It has been shown on 7085 aluminum that anodization does not reduce the fatigue life (ER[09]). ER[10] reports a 6X reduction in the fatigue life of a 7010 aluminum part, due to anodization.

In conclusion, it is not possible to make a blanket statement about anodization. This is still a subject of research, so an S-N curve for the exact alloy should be referenced for proper fatigue life estimation.

4.4 TEMPERATURE

There are studies, showing a reduction in fatigue life at high temperatures. This doesn't apply to the AOP, since we do not have any high temperature assemblies.

4.5 GEOMETRY

Maximizing internal radii of machined or cast parts is important because it decreases stress concentrations and extends the fatigue life. Usually the stresses are estimated through finite element model, so the stresses due to internal radii are already taken into account.. The finite element model generates stresses which are supplied to the fatigue analysis.

4.6 INCLUSIONS

Inclusion is any localized defect in the casting or treatment of the material. Inclusions serve as crack nucleation sites which contribute to the formation of cracks. Inclusions are not by design and they could happen in any assembly. Inclusions are mitigated by the early proof loading, testing, and inspection regimens outlined in Table 1.

4.7 STRESS CORROSION CRACKING (SCC)

Pure corrosion would be outside the scope of this document but SCC is different. It is caused by a combination of chemical environment, tensile stress and alloy as shown in Figure 6.

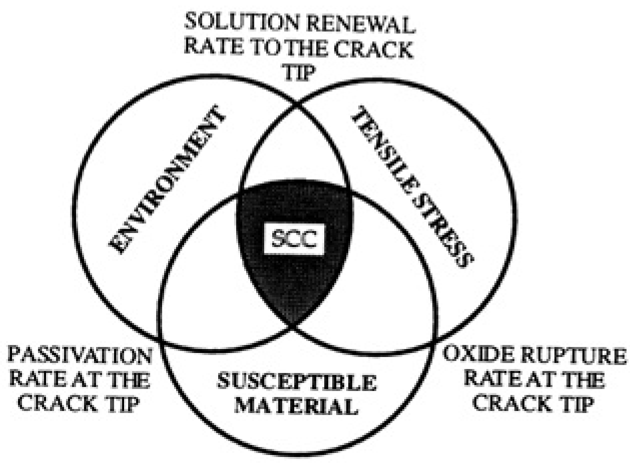

Figure 6 - Factors Required for SCC

In conclusion SCC is something to be aware of if an assembly is continually exposed to a chemical environment, especially salts. It is important that a compatible alloy is selected early in the design cycle. In all of the AOP mechanical systems, there are currently no situations where chemical combinations capable of causing SCC are present.

5 INTERPRETING S-N CURVES

Most S-N curves have few data points relative to the number required for any kind of statistical certainty. Many researchers mentioned in ER[11] have noted how inconsistent the fatigue life is from sample to sample.

5.1 Random Fatigue Model

The random fatigue model takes into account the sample-sample variability in fatigue life. This model is endorsed by ER[05] and ER[13]. In conclusion, the Random Fatigue Model is the most useful and representative interpretation of the S-N curve. Also, the Random Fatigue Model explains the variability in high-cycle test results which have been inhibiting researchers from implementing a useful model for high-cycle fatigue.

The random fatigue model is comprised of 2 parts: the S-N curve shown in Figure 1and the error equation ε pictured in Figure 7. Equation 1 has been adapted from ER[11], as explained in Appendix 11.1, for use with any unit system and expressed in terms of stress as a function of number of cycles. Here is a description of the parameters:

γ_0 (units: pressure): At cycle n=1, this represents how much higher the stress is than the fatigue limit. The value is always a positive scalar.

γ_1 (unitless): This parameter associated with slope and slope change. The value is always a negative scalar.

γ_2 (units: pressure): This is the Fatigue limit. The value is usually a positive scalar, yet there is no reason why it can't be negative. This would mean that the material effectively has no fatigue limit (because it would need to remain under compression to avoid cracking).

Equation 1 – S-N curve portion of the Random Fatigue model (ER[11]). Here stress is denoted by σ(n) consistent with the notation of ER [11]. The symbol S is used to denote stress throughout this document

$$\sigma(n) := \gamma_2 + \gamma_0 \cdot n^{\gamma_1}$$, inversely: $$n(\sigma) = \left(\frac{\sigma - \gamma 2}{\gamma 0}\right)^{\frac{1}{\gamma 1}}$$

(1)

This equation has good agreement for ultra-high cycle fatigue (ER[13]), but not low cycle fatigue. Also, keep in mind that there is also variability in the fatigue limit from sample to sample (ER[13]). This could be due to factors affecting fatigue, listed in Section 4 or it could be the nature of crack initiation which originates from persistent slip bands according to ER[02]. Because the persistent slip bands can form an intrusion or extrusion, the process of crack initiation is a random walk; therefore there is variability in the cycles and stress required for crack initiation.

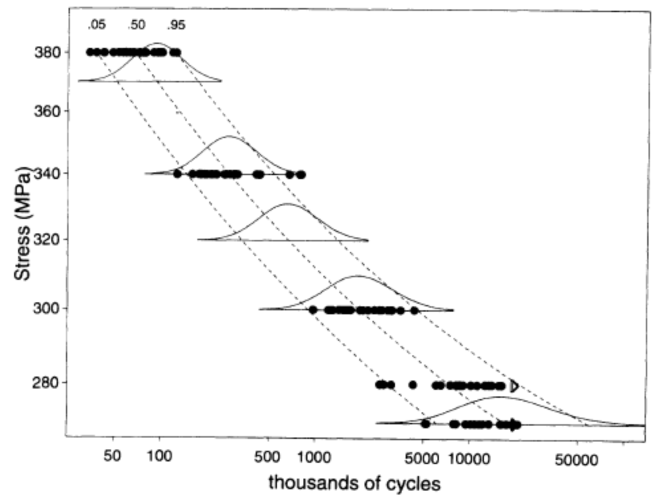

Figure 7 – Different profiles of the error equation along the S-N curve for laminate panel (ER[11])

As you can see, to a small extent, in Figure 7, the error equation becomes wider for a lower stress level. This is especially true for other materials like Ti6Al4V pictured in Figure 8.

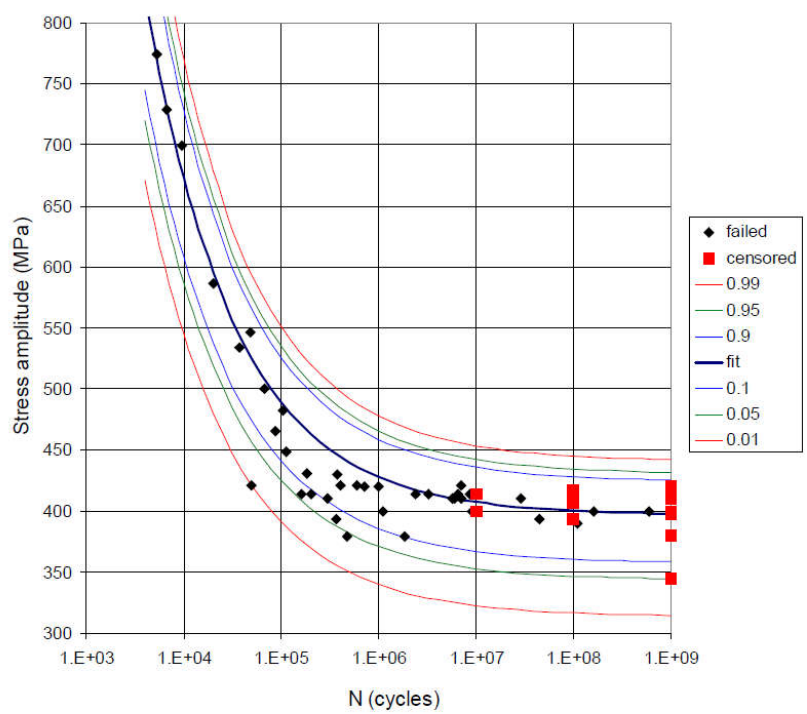

Figure 8 - S-N curve for Ti6AlV4 using the Random Fatigue model (ER[05])

6 FATIGUE PLAN

A lifetime fatigue maintenance regimen was developed to mitigate the following risks:

- Material inclusions and workmanship

- Incomplete, and marginally representative, S-N curves
- Fatigue failure which is up to 2σ away from the mean predicted N (on the S-N curve)
- Weld fatigue limit reduction

Examples of factors, which are not assumed, and must be separately worried about, are:

- Design and analysis mistakes
- Corrosion
- Abuse and wear of the assembly
- Fabrication of and/or modification of the assembly to be different than the intended design

6.1 Developing Lifetime Fatigue Plans for Components and Assemblies

The first step in developing a lifetime fatigue maintenance regimen for each component or assembly is to look up an S-N (Wohler) curve for the material being considered. Then, from a finite element analysis, identify the maximum stress to which the component of assembly will be subjected and look up the maximum number of cycles, N. Table 1 can then be used to determine the appropriate mitigation strategy for the component or assembly under consideration.

Table 1 provides criteria for all components and assemblies. To use this table, first determine the anticipated maximum stress in the assembly (1st column) and the amount of structural redundancy and impact (Top row), then determine the required fatigue maintenance regimen from the matrix in Table 1.

Table 1 –Fatigue Maintenance Chart

<table>
<tr><td></td><td></td><td colspan="3">REDUNDANCY AND IMPACT</td></tr>
<tr><td></td><td></td><td>1. Single failure can cause injury or >$100k in damage</td><td>2. Only multiple failures can cause injury or >$100k in damage</td><td>3. Injury or <=$100k damage is not possible by structural failure</td></tr>
<tr><td rowspan="3">STRESS LEVEL</td><td>A. Stress is within S-N curve</td><td rowspan="3">CATEGORY 4
Period: Time it takes for a crack to grow from 1% to 2% of the structural member thickness.
Service: Proof load to 2x the working load limit (WLL) and inspect for cracks.
Replace: N/2 or cracked</td><td>CATEGORY 3
Period: N/10
Service: Inspect for cracks
Replace: N/2 or cracked</td><td rowspan="2"></td></tr>
<tr><td>B. Stress is too low for S-N curve</td><td>CATEGORY 2
Period: N/10
Service: Inspect for cracks
Replace: If cracked</td></tr>
<tr><td>C. A fatigue limit exists for the material and the stress level is below it</td><td colspan="2">CATEGORY 1
Period: none
Service: none
Replace: if no longer functional</td></tr>
<tr><td></td><td colspan="4">Period: time interval between inspections</td></tr>
<tr><td></td><td colspan="4">Service: activity to be conducted to monitor condition</td></tr>
<tr><td></td><td colspan="4">Replace: time interval or condition when component shall be replaced</td></tr>
</table>

7 FATIGUE LIMIT ESTIMATION PROCESS

The procedure for estimating the fatigue limit for a component or assembly is shown in Figure 9. First establish the working loads on the part and then determine the maximum tensile stress using a Finite Element Model (FEM). This process is documented for the major assemblies making up AOP Payload I in AD [01]. In addition, a similar procedure has been conducted and documented in AD [01] for major components such as the Payload Integration Mount (PIM) lifting ring. Finally, the loading frequency is determined from the operational environment to which the part is subjected to, and N is calculated.

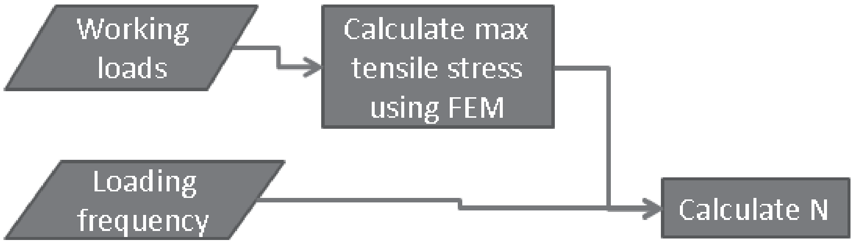

Figure 9: Fatigue Limit Estimation Limit Process

8 CRACK GROWTH RATE ESTIMATION PROCESS

The procedure for estimating the growth rate of a crack is shown in Figure 10. First, establish the working loads on the part and then determine the maximum tensile stress using a Finite Element Model (FEM). Next, most identify likely area for cracking – these are typically notched areas, joints, or internal radii. Determine the stress intensity factor equation corresponding to the crack area geometry. From a study on the same material, determine the crack growth rate (i.e., crack growth distance per cycle). It is important that treatments used in the study correspond to those treatments used on the material being evaluated since these can affect the strength of the material and crack propagation. Determine the loading frequency and then calculate the crack growth rate.

In extreme situations (i.e., high loads, potential for human injury), estimates of crack propagation should be supported with testing to improve confidence in the analysis.

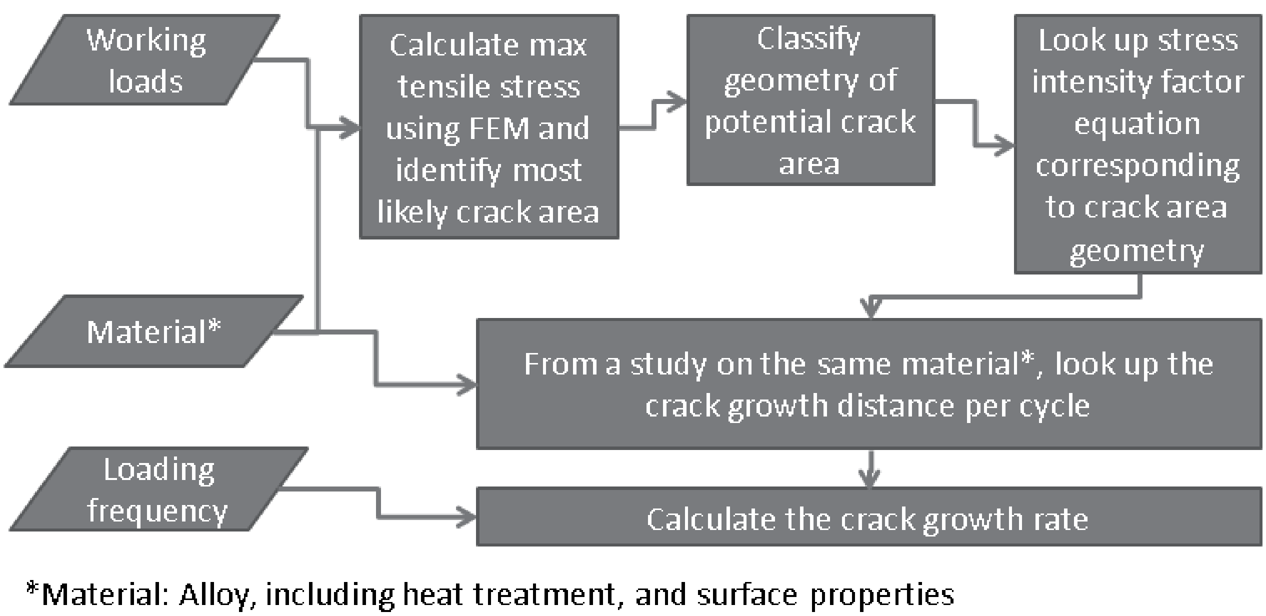

Figure 10: Crack Growth Rate Estimation Process

9 EXAMPLE

As an example of applying this process, we can consider the titanium lifting ring. The lifting ring is the interface between the PIM and the lifting fixture. It comes into play when the Payload is lifted into and out of the aircraft with a forklift during integration and de-integration. The mechanical failure of a lifting ring could potentially result in serious damage to the payload and/or aircraft, and could cause injury to personnel. Therefore, this failure mechanism falls under Category 4 in terms of redundancy and impact in Table 1. Now we need to do the crack growth rate estimation process. Here are the input parameters:

- Working load: 1.3kN
- Material: Ti6Al4V, sanded
- Loading frequency: 20/year

The max tensile stress is conservatively estimated from Figure 11, since the Von Mises >= Tensile stress.

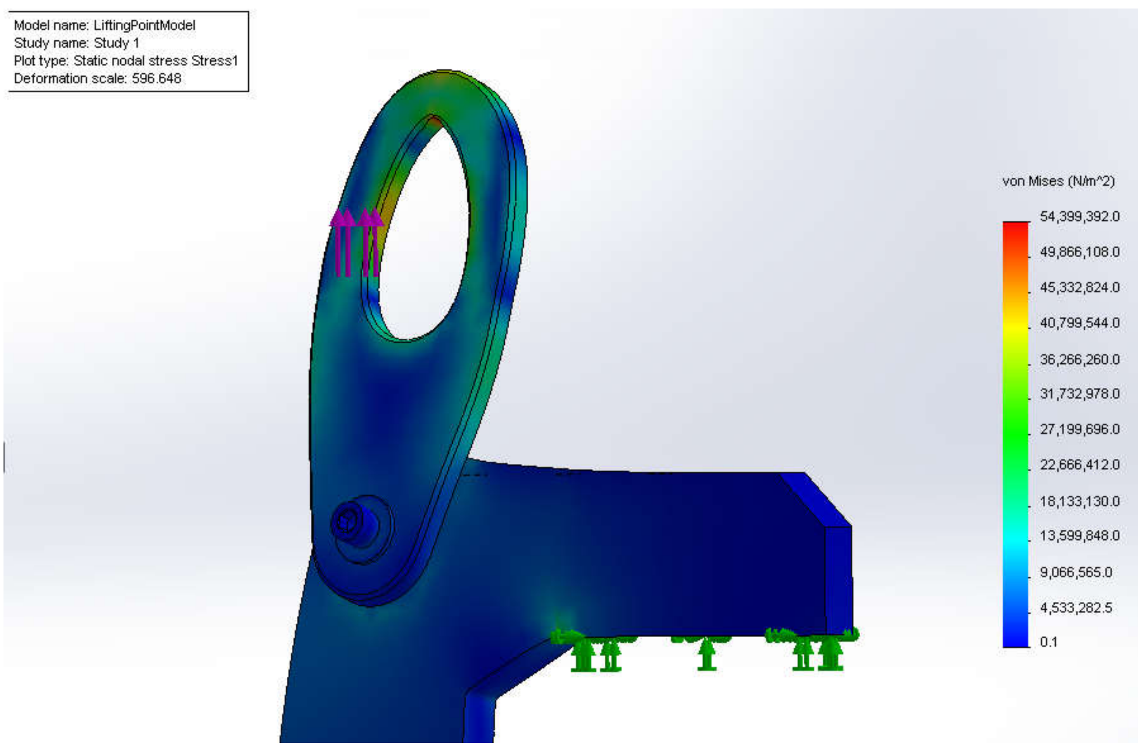

Figure 11 - Ti6Al4V lifting ring loaded with a WLL of 1.3kN

Here is the calculation for the inspection/testing period:

CRACK GROWTH ESTIMATE

max tensile stress on ring $\sigma := 597\mathrm{MPa}$

member thickness $d1 := 12.7\mathrm{mm}$

hypothetical crack size $a := d1 \cdot .01 = 0.127\,\mathrm{mm}$

assuming worst case, with crack at edge, and geometry of sheet under tension, the stress intensity factor, from eq. 7b in ER[16] is

$K := \sigma \cdot \sqrt{\pi \cdot a} = 11.925\,\mathrm{MPa} \cdot \sqrt{\mathrm{m}}$

$K = 10.852\,\mathrm{ksi} \cdot \sqrt{\mathrm{in}}$

the crack growth, for each cycle, shown to the right (ER[14]) is

$dadn := .00006\mathrm{in} = 1.524 \times 10^{-3}\,\mathrm{mm}$

loading frequency $fl := \frac{20}{\mathrm{yr}}$

crack growth rate $dadt := dadn \cdot fl = 0.03\,\frac{\mathrm{mm}}{\mathrm{yr}}$

Maintenance period $\Delta Tinsp := \frac{a}{dadt} = 4.167\,\mathrm{yr}$

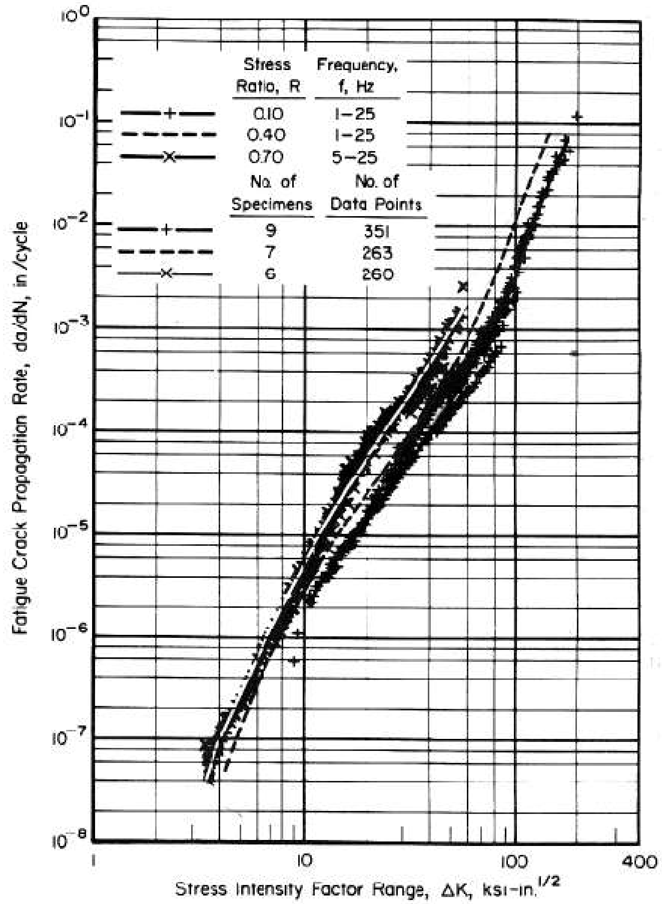

Figure 5.4.1.1.9. Fatigue-crack-propagation data for 0.250-inch-thick Ti-6Al-4V mill-annealed titanium alloy plate with buckling restraint. [Reference 5.4.1.1.9.]

USEFUL LIFE ESTIMATE

equation 1

$n(S, \gamma 0, \gamma 1, \gamma 2) := \left(\frac{S - \gamma 2}{\gamma 0}\right)^{\frac{1}{\gamma 1}}$

$Nc := n(597\mathrm{MPa}, 19020\mathrm{MPa}, -.367, 354.386\mathrm{MPa}) = 1.451 \times 10^{5}$ number of cycles until cracking is 98% probable

MAINTENANCE PRESCRIPTION

working load limit $WLL := 1.3\mathrm{kN}$

Every $\Delta Tinsp = 4.167\,\mathrm{yr}$ or less, proof load to $2 \cdot WLL = 2.6\mathrm{kN}$ and inspect for cracks

After $\frac{Nc}{2 \cdot fl} = 3.627 \times 10^{3}\,\mathrm{yr}$, replace the assembly

10 GUIDELINES FOR SELECTED MATERIALS

The materials evaluated in this section are the primary materials used on the AOP-1 Payload Integration Mount (PIM) and on ground equipment supporting AOP flight operations and laboratory activities.

10.1 6061-T6 Aluminum

6061-T6 is very common, weldable, heat treatable, machinable, and has decent corrosion resistance. The fatigue data from ER[12] were fitted with Equation 1 and are shown in Figure 12, represented by curve S. The fit was shown to have a roughly Gaussian residual, shown in Figure 13. The following curves S84 and S98 were fitted to stay 1σ and 2σ away from S, using localized standard deviations. The coefficients for equations S, S84, and S98 are listed in Table 2.

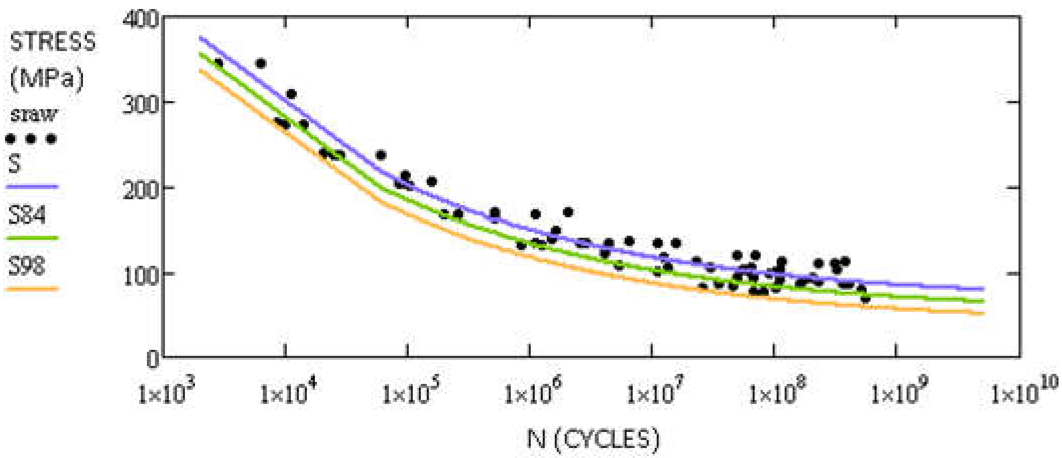

Figure 12 – 6061-T6 S-N data from a rotating beam test (ER[12]) fitted with the RFL Equation 1

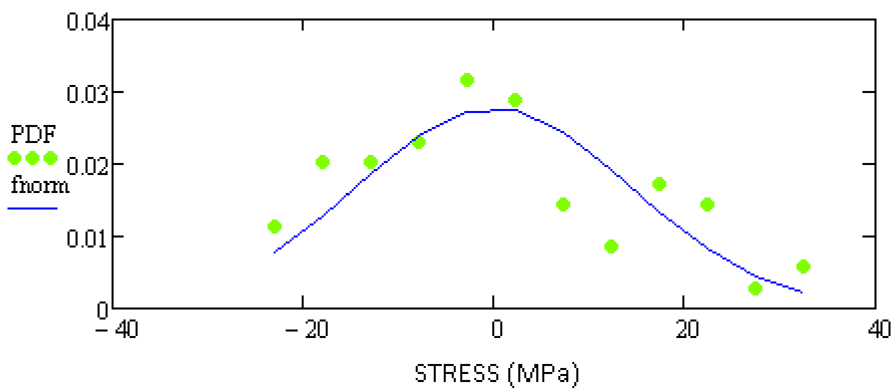

Figure 13 - Probability density of the residual left by fitting Equation 1 to the 6061-T6 S-N data

Table 2 - 6061-T6 fatigue equations indicating max allowable stress as a function of number of cycles (n)

$$S(n) = \gamma_2 + \gamma_0 n^{\gamma_1}$$

Equation	γ0 (Mpa)	γ0 (KSI)	γ1	γ2 (fatigue limit) (Mpa)	γ2 (fatigue limit) (KSI)
S (best-fit)	1547	224.37	-0.211	64.744	9.390
S84 (84% confidence)	1546	224.23	-0.213	51.558	7.478
S98 (98% confidence)	1548	224.52	-0.216	38.372	5.565

As indicated in section 5, there is variability in the stress required for crack initiation and variability in the fatigue limit. This variability can be taken into account by using equations S84 and S98. Also note that there is a significant degradation in the fatigue limit if the structure is notched with sharp internal radii, such as 25µm (ER[12]). There is a slight chance that research will uncover a new lower fatigue limit, but as long as prudent risk mitigation is taken as described in Table 1, then the risk of changing research findings is also mitigated. Therefore equations S84 and S98 can serve as an estimate for fatigue life.

10.2 7075-T6 Aluminum

7075-T6 is one of the strongest aluminum alloys, yet it is unweldable. It has poor corrosion resistance and it is susceptible to stress corrosion cracking. The fatigue data from ER[12] were fitted with Equation 1 and are shown in Figure 14, represented by curve S. The fit was shown to have a roughly Gaussian residual, shown in Figure 15. The following curves S84 and S98 were fitted to stay 1σ and 2σ away from S, using localized standard deviations. The coefficients for equations S, S84, and S98 are listed in Table 3.

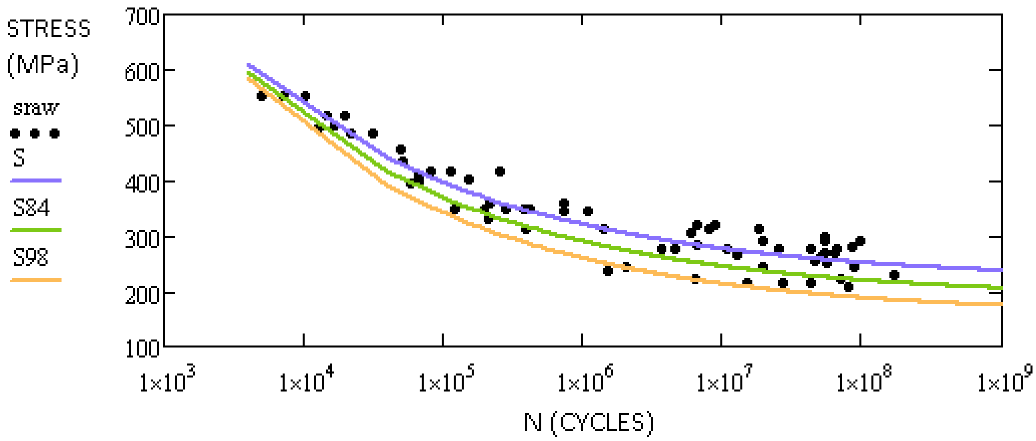

Figure 14 – 7075-T6 S-N data from ER[14], fitted with Equation 1

Test Conditions:

Surface Condition: Unspecified
Loading – Axial
Frequency - 30 Hz
Temperature – RT
Environment – Air

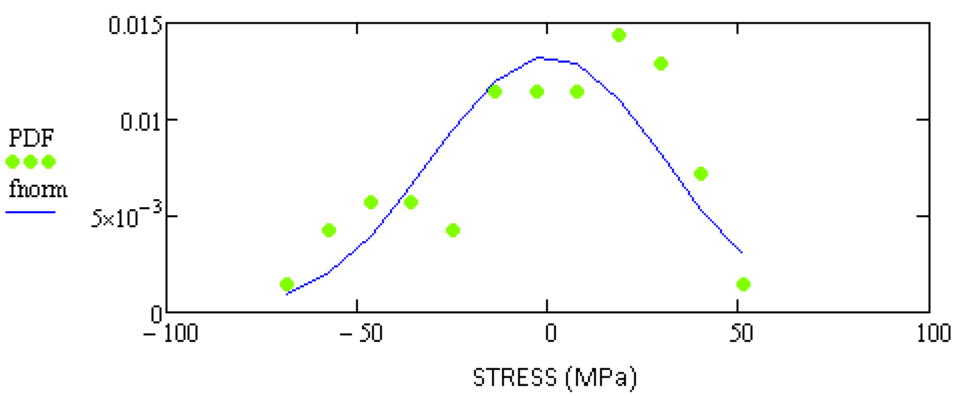

Figure 15 - Probability density of the residual from fitting the 7075-T6 data with Eq. 1

Table 3 - 7075-T6 fatigue equations indicating max allowable stress as a function of number of cycles (n)

$$S(n) := \gamma_2 + \gamma_0 \cdot n^{\gamma_1}$$

Equation	γ0 (Mpa)	γ0 (KSI)	γ1	γ2 (fatigue limit) (Mpa)	γ2 (fatigue limit) (KSI)
S (best-fit)	2967	430.33	-0.244	217.669	31.570
S84 (84% confidence)	3336	483.85	-0.253	187.314	27.168
S98 (98% confidence)	3736	541.86	-0.261	156.958	22.765

As indicated in section 5, there is variability in the stress required for crack initiation and variability in the fatigue limit. This variability can be taken into account by using equations S84 and S98. Also note that there is a significant degradation in the fatigue limit if the structure is notched with sharp internal radii, such as 25µm (ER[12]). There is a slight chance that research will uncover a new lower fatigue limit, but as long as prudent risk mitigation is taken as described in Table 1, then the risk of changing research findings is also mitigated. Therefore equations S84 and S98 can serve as an estimate for fatigue life. Corrosion should also be factored in to the lifetime of the structure.

10.3 AZ31 Magnesium

Magnesium is the lightest commonly available structural alloy. AZ31 is weldable, but it can easily be corroded. It has about 2/3 the density of aluminum. The yield strength of AZ31 is less than 6061-T6, but the fatigue strength is estimated to be higher for AZ31. Magnesium also has the highest damping capacity of all of the structural alloys. Magnesium is the easiest to machine of all structural alloys.

The fatigue data from ER[12] were fitted with Equation 1 and are shown in Figure 16, represented by curve S. The fit was shown to have a roughly Gaussian residual, shown in Figure 15. The following curves S84 and S98 were fitted to stay 1σ and 2σ away from S, using localized standard deviations. The coefficients for equations S, S84, and S98 are listed in Table 3.

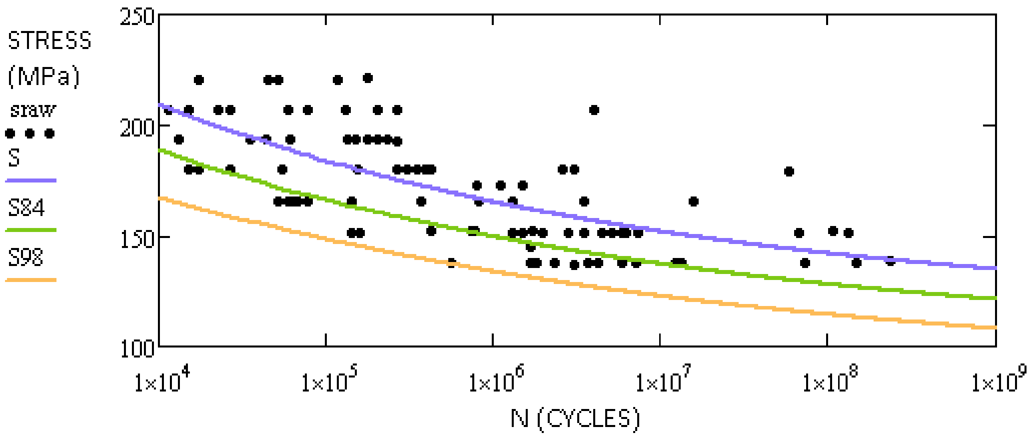

Figure 16 – S-N test results of AZ31B-F samples fit with Equation 1 (ER[14])

Test Conditions:

Surface Condition: Polished sequentially with # 320 aluminum oxide cloth, No. 0, 00, and 000 emery paper and finally # 600 aluminum oxide powder in water
Loading - Axial
Frequency - 1500 cpm
Temperature - RT
Environment – Air

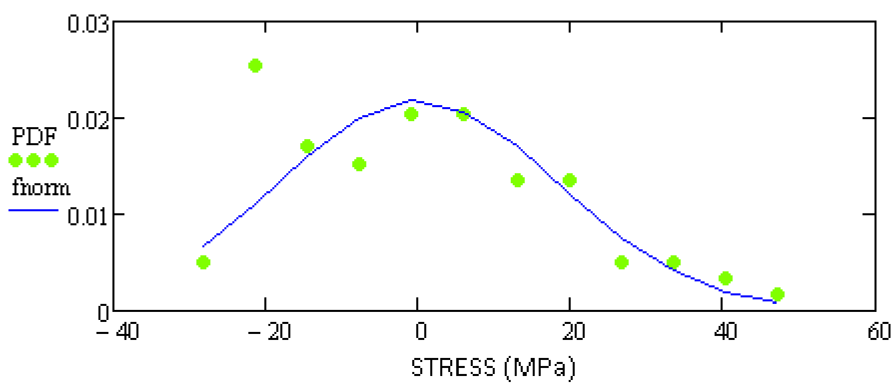

Figure 17 - Probability density of the residual from fitting the AZ31 data with Eq. 1

Table 4 - AZ31B fatigue equations indicating max allowable stress as a function of number of cycles (n)

$$S(n) := \gamma_2 + \gamma_0 \cdot n^{\gamma_1}$$

Equation	γ0 (Mpa)	γ0 (KSI)	γ1	γ2 (fatigue limit) (Mpa)	γ2 (fatigue limit) (KSI)
S (best-fit)	340	49.30	-0.14	116.226	16.857
S84 (84% confidence)	288	41.82	-0.131	101.963	14.788
S98 (98% confidence)	241	34.96	-0.120	87.7	12.720

Figure 16 represents the estimated mean fatigue strength. There will always be variation from this. To achieve these results it is important to maximize internal radii and prevent corrosion.

10.4 Ti6Al4V Titanium

This is the most commonly used titanium alloy. It is very strong and very corrosion resistant, even at high temperatures. It is weldable, but requires a completely oxygen free environment. It has a high fatigue limit and there is a lot of research and testing on it, since it is used throughout the aerospace industry. Titanium is time consuming to machine, requiring low speeds and rigid setups. Paradoxically, titanium is less abrasion resistant than mild steel.

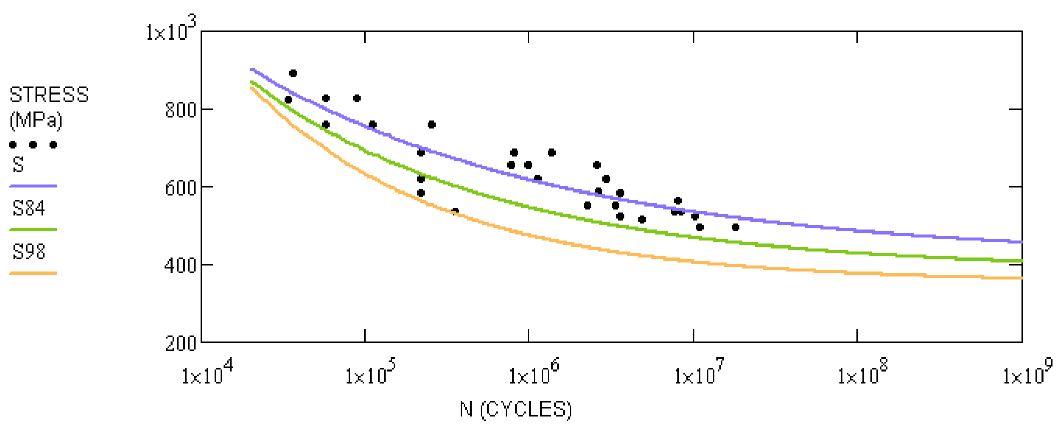

Figure 18 - Ti6Al4V S-N test results from ER[14] using Equation 1

Test Conditions

Surface - RMS 1.6µm
Loading — Axial
Frequency — 1800 cpm
Temperature — RT
Environment — Air

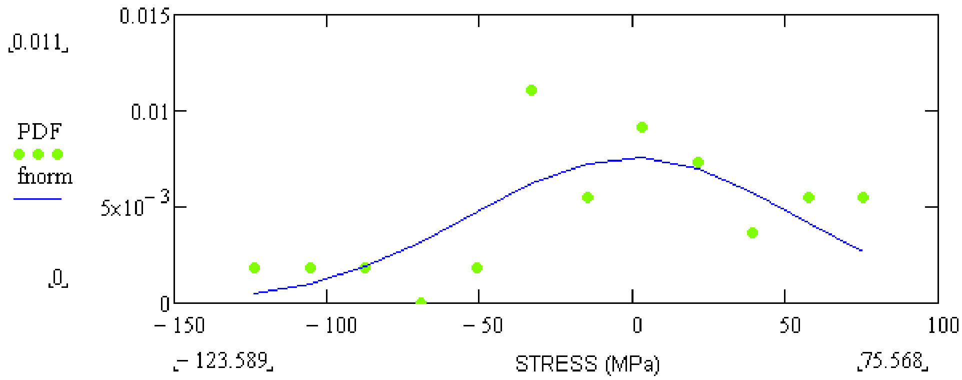

Figure 19 - Probability density of the residual from fitting the Ti6Al4V data with Eq. 1

Table 5 - Ti6AlV4 fatigue equations indicating max allowable stress as a function of number of cycles (n)

$S(n) := \gamma_2 + \gamma_0 \cdot n^{\gamma_1}$ Equation	γ0 (Mpa)	γ0 (KSI)	γ1	γ2 (fatigue limit) (Mpa)	γ2 (fatigue limit) (KSI)
S (best-fit)	4587	665.29	-0.226	413.554	59.981
S84 (84% confidence)	7900	1145.80	-0.281	383.97	55.690
S98 (98% confidence)	19020	2758.62	-0.367	354.386	51.399

As indicated in section 5, there is variability in the stress required for crack initiation and variability in the fatigue limit. This variability can be taken into account by using equations S84 and S98. Also note that there is a significant degradation in the fatigue limit if the structure is notched with sharp internal radii, such as 25µm (ER[12]). There is a slight chance that research will uncover a new lower fatigue limit, but as long as prudent risk mitigation is taken as described in Table 1, then the risk of changing research findings is also mitigated. Therefore equations S84 and S98 can serve as an estimate for fatigue life. Corrosion should also be factored in to the lifetime of the structure.

11 APPENDIX

11.1 Converting the Random Fatigue Limit equation

Original equation presented by Pascual and Meeker (ER [11])

Let Y be the fatigue life and s the stress level. We model Y as

$$\log(Y) = \beta_0 + \beta_1 \log(s - \gamma) + \varepsilon, \qquad s > \gamma,$$

where β_0 and β_1 are fatigue curve coefficients, γ is the fatigue limit of the specimen, ε is the error term, and log denotes natural logarithm.

The goal is to solve for s and eliminate the unit problem caused by having s and γ inside of a natural log.

$$\beta_1 \quad Ln(s \quad \gamma) = Ln(Y) \quad \beta_0$$

$$Ln(s \quad \gamma) = \frac{Ln(Y) \quad \beta_0}{\beta_1}$$

$$s \quad \gamma = e^{\frac{Ln(Y)-\beta_0}{\beta_1}}$$

$$s = \gamma + \left[e^{Ln(Y)-\beta_0}\right]^{\frac{1}{\beta_1}} = \gamma + \left[Y \quad e^{-\beta_0}\right]^{\frac{1}{\beta_1}} = \boldsymbol{\gamma_2 + \gamma_0 \quad n^{\gamma_1}}$$

With the following substitutions

$$\gamma_0 = \left[e^{-\beta_0}\right]^{\frac{1}{\beta_1}}$$

$$\gamma_1 = \frac{1}{\beta_1}$$

$$\gamma_2 = \gamma$$