University of Bristol
Department of Aerospace Engineering

Structures & Materials
StM3A

FATIGUE

Test 787 airframe surrounded by steel framework (Boeing)

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Course objective

At the end of this course you should be able to:

- understand the fatigue behaviour of metals
- understand the idealisation of fatigue loading histories.
- understand the issues and manipulation of fatigue data
- understand the fatigue damage accumulation process
- carry out fatigue life calculations for metal aircraft structures under service loading.
- identify the critical fatigue design details of aircraft

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Contents

Fatigue of Metals

- Behaviour
- Analysis
- Design
- Examples

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References

Sandor "Fundamentals of Cyclic Stress and Strain"

ESDU "Engineering Science Data Units"

Niu "Airframe Structural Design"

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Fatigue Behaviour of Metals

Contents

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Definition

The phenomenon:

Repeated cyclic loads - *below elastic limit*

↓

Accumulation of fatigue damage - *microscopic plastic deformations at localised zones*

↓

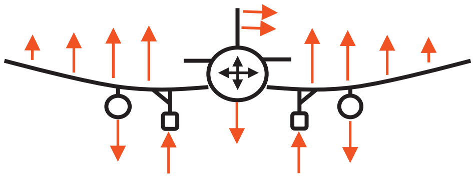
Crack formation & growth

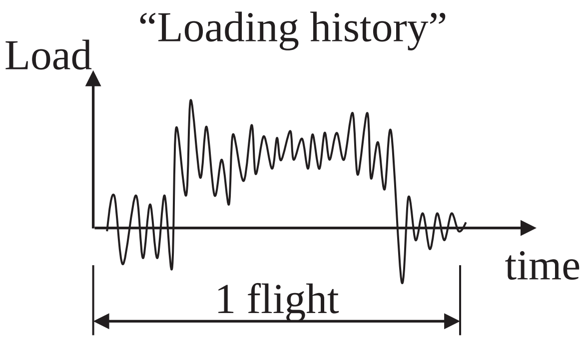
↓

Catastrophic failure!

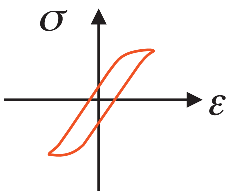
The Problem:

Complex service loading





Complex material response



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Instantaneous and catastrophic failure can occur under moderate levels of r**epeated cyclic loading** which has already been resisted many times. The reason for such a failure is due to the **accumulation of microscopic damage** at a detail or flaw which represents a discontinuity, eventually leading to cracking which causes progressive loss of strength and final failure by fracture.

The microscopic fatigue damage can be traced to **plastic deformations**. This localised plastic deformation can occur at localised zones even though the applied global stress is well below the material elastic limit. Stress concentrations at details or flaws can produce **dislocations** which increase and move under the action of high stresses. Eventually the density of the dislocations increases to such an extent that they impede one-another, i.e. a local work-hardening. The only way remaining to continue to absorb energy at the stress concentration is then to crack and form new surfaces.

The problem for the fatigue design of aircraft structures is to determine the **service loading** imposed on the material and the **material response** to this loading, particularly at discontinuities. However, We are faced with complex cyclic loading histories and each material having its own complex personality for response.

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The fatigue process

Initiation:
plastic strain accumulation
↓
Microscopic crack

Fatigue Analysis

>75% life

Propagation:
crack growth
↓
Critical crack
Failure

Fracture Analysis

Fatigue analysis is used for **"Safe life"** design
assuming no initial macroscopic flaws
e.g. safety critical - undercarriage

Fracture Analysis is used for **"Fail safe"** design
assuming an initial macroscopic flaw
e.g. more easily inspected components

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Classical fatigue analysis was initially developed to predict the **"crack free initiation life"** of metals. For high quality well-finished metals (with no initial macroscopic cracks) the crack free initiation phase can account for over three quarters of the total fatigue life. Because of the relatively large proportion of life accounted by the initiation phase and because of the difficulty in determining when the macroscopic crack stage starts, fatigue analysis is often carried out based on **life to final failure**.

Aircraft fatigue design usually involves estimates of life using **fatigue analysis** - assuming no initial macroscopic cracks and **fracture mechanics analysis** - assuming an initial finite crack length.

Fatigue analysis is used to provide **Safe life** designs for structural components which are difficult to inspect for cracks, such as undercarriage components made from high strength alloys which exhibit a short critical crack length or components which are not readily accessible. Here, a safe life is estimated - after which the component is removed from service – despite "no apparent fatigue damage". This method of analysis is also used as an"overall"estimate of fatigue life for general aircraft structure to assess the likely "crack free life".

Fracture mechanics can provide **Fail Safe** designs which nowadays predominate for all general aircraft structural components. Here the component is designed so that a certain amount of fatigue damage can be sustained and confidently detected before reaching a critical level.

Usually both fatigue and fracture analyses are carried out to characterise an aircraft structural life. This course looks in detail at **classical fatigue analysis** as opposed to fracture mechanics.

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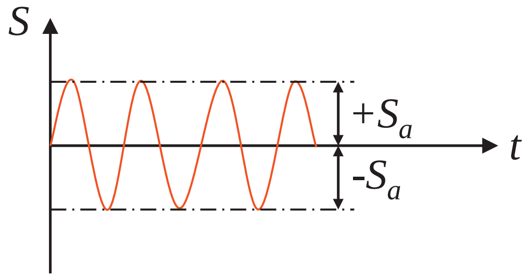
Material Response

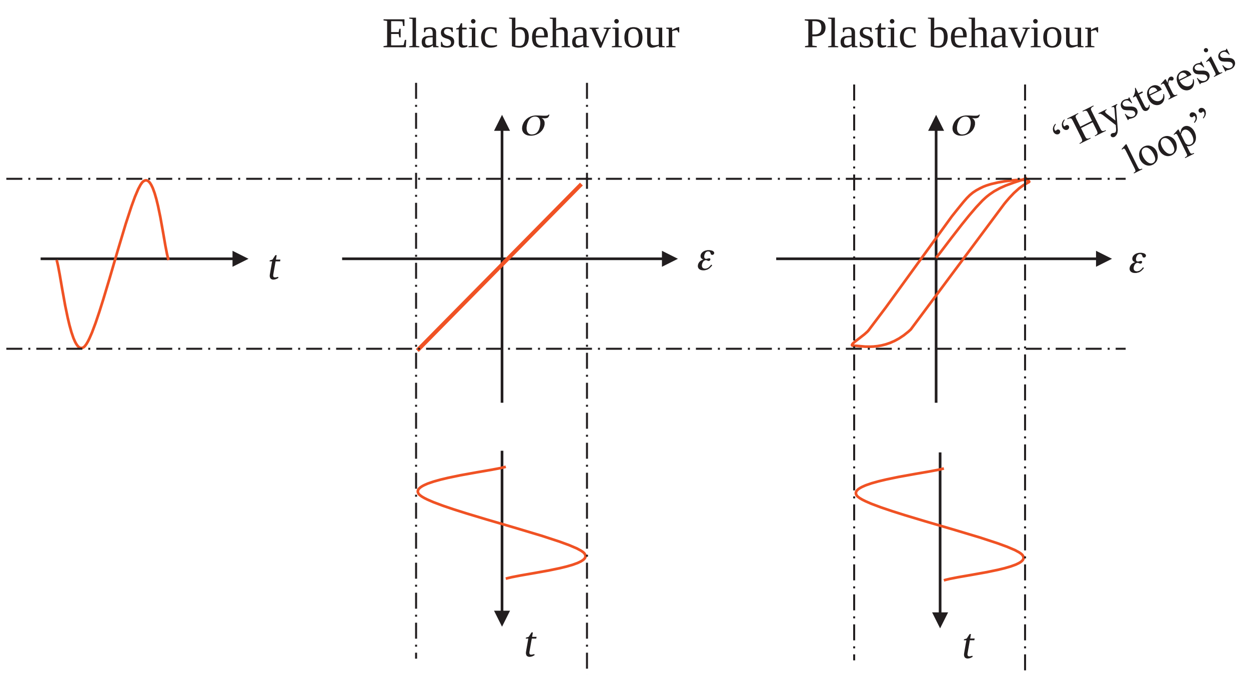
Cyclic Stress and Material Response

Zero mean (fully reversed) constant amplitude cyclic stress



S = stress or strain, σ,ε
S_a = alternating σ,ε
ΔS = Range $2S_a$

Stress-strain "hysterisis"



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First consider simple constant-amplitude fully-reversed (i.e. zero mean stress) cyclic loading.

Cyclic stress-strain plots, in the form of **hysterisis** curves, are useful to understand material response under cyclic loading. The area within the hysterisis loops effectively represent the “work done” or “energy absorbed” by the material on each complete load cycle and this can be associated with the accumulated fatigue damage.

The cycle dependent material response will depend upon the controlling action, e.g. **stress** or **strain control.** Cyclic hardening or softening may occur during repeated loading resulting in changing hysteresis characteristics. Cycle dependent material response is **exponential** and usually most change in response occurs by the "**fatigue half-life"** at a given stress level - as indicated in the following diagrams of cyclic hardening and softening.

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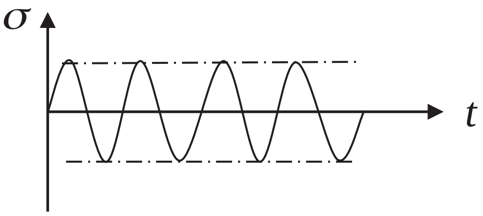
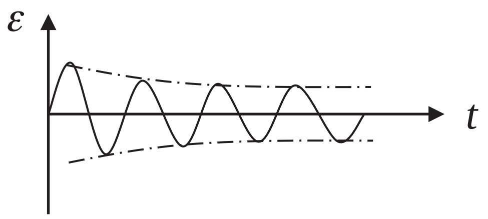
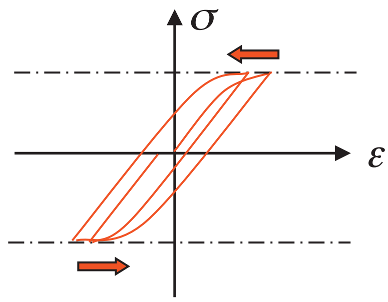
Cyclic hardening and cyclic softening

Stress control

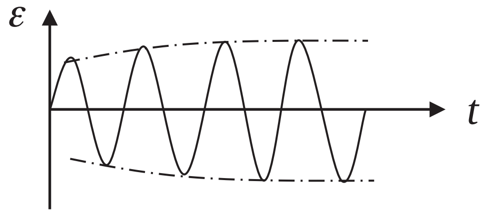


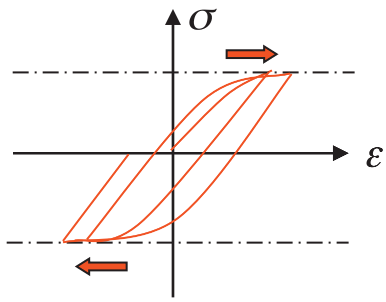
Cyclic hardening





Cyclic softening





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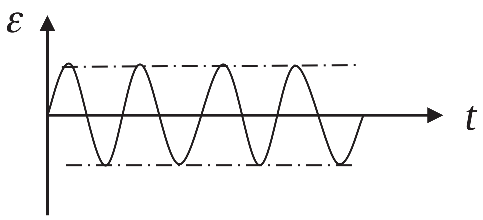
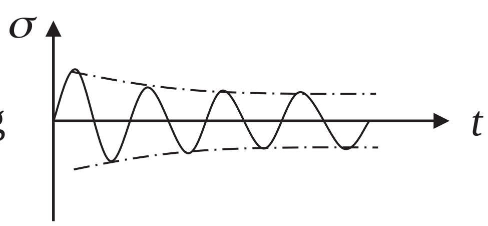
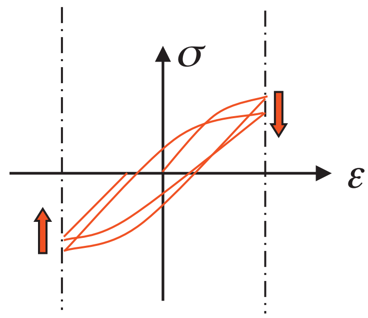
Cyclic hardening and cyclic softening

Strain control

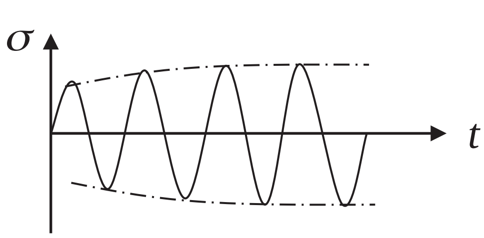


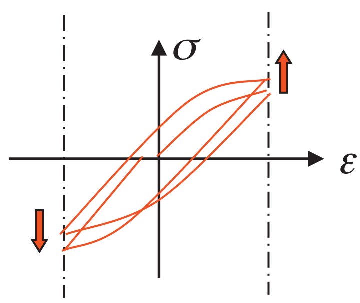
Cyclic softening





Cyclic hardening





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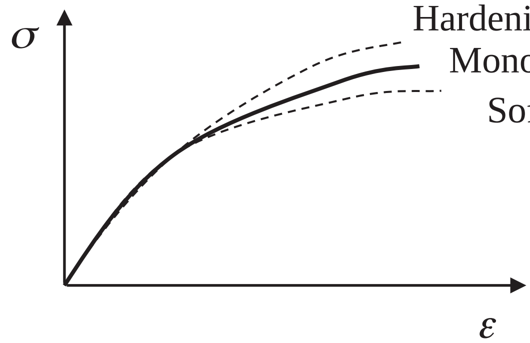
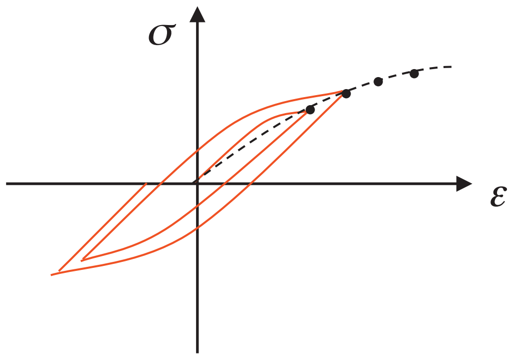
Cyclic dependent changes in properties

The fatigued stress-strain response
may not be the same as the unfatigued response

E.g.:



"Multiple step method" for cyclic stress-strain curve



- Generate curves from hysteresis peaks

Material flow dependent properties are most affected

E.g.: σ_y , (σ_{ult})

"Fatigue half life properties" are used to indicate change in response (σ-ε response after cycling to ~ ½ fatigue life)

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Some metal properties may change significantly under repeated cyclic loading. Consequently, engineers must use **initial static stress strain** curves with some caution. Cyclic change in stress strain response can lead to higher stresses or strains than estimated from data - depending on whether under stress or strain control.

A quick method to determine the fatigued stress-strain response curve of a material is given by the **"Multiple Step Method"**. Usually a strain controlled cyclic test is used where the strain amplitude is increased in steps after sufficient cycles have been performed to create a stable hysteresis loop at that strain level. Successive stress-strain hysteresis loops are plotted on one diagram and the tips are joined to form the cyclic stress strain curve.

Material flow properties such as yield and ultimate strength are most significantly affected by cycling. Fracture properties are less affected and elastic modulus remains relatively unchanged.

It is impractical to characterise the whole range of cyclic stress strain responses that might exist in a fatigue life and instead a **fatigue half-life** response may be used to give an indication of the expected change. I.e. the stress-strain response of a material after cycling to approximately half of it's fatigue life.

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Typical cycle dependent changes in properties:

Material flow	✓
Fracture	(✓)
Elastic Modulus	✗

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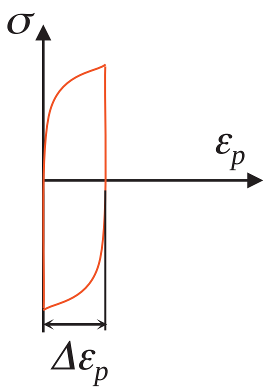
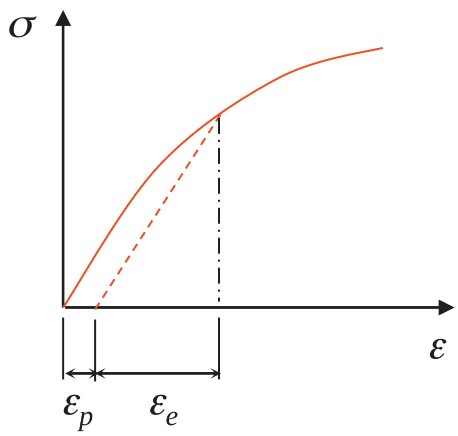
Energy considerations in fatigue

Initiation stage plastic deformation involves energy dissipation

Consider energy per cycle:





Total plastic strain energy: $W_p = \Delta W.N_f$ *"Fatigue toughness"*

W_p (hi cycle fatigue) > W_p (lo cycle fatigue) > W_f (static)

Note:

fatigue

- Plastic strain energy : is responsible for crack initiation
- Elastic strain energy : is responsible for crack propagation

fracture

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Energy is dissipated in fatigue because of plastic deformation. The energy per cycle, W, is measured by the area of the hysteresis loop. The total energy expended during life is the summation of all the loop areas.

Where W varies only slightly from cycle to cycle the total plastic strain energy, $W_{p,}$ is given by summing the individual cycle hysteresis energy increments over the fatigue life and this value is effectively the "fatigue toughness".   (This is <u>not</u> the same thing as fracture toughness).

From an energy point of view, fatigue is not a bad problem. Even though fatigue is considered as a serious weakening process causing fractures at stresses below the ultimate or even the yield strengths, typically more than 100 times the monotonic fracture energy may be required to cause fatigue failure in 10^6 cycles.

Hysteresis energy is a useful basis for fatigue failure. Essentially **plastic strain energy** accounts for damage accumulation before crack propagation and **elastic strain energy** accounts for fatigue cycle crack growth up to separation. The stress increase leading to final fracture is due to increase in stress intensity at the crack tip and gradual reduction in the load bearing area caused by fatigue crack growth.

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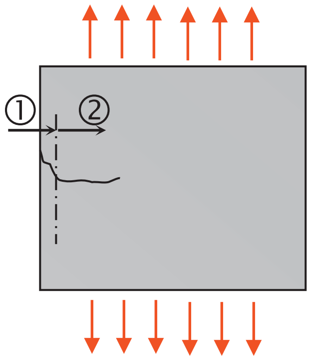
Damage Phases

“phases of damage accumulation”:

Initiation and stage 1 growth



Under tension or compression

Dependent on **shear-slip deformation**

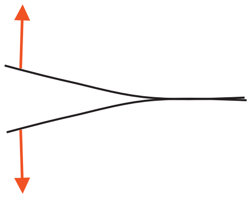
In direction of **max shear stress planes**

Dependent on properties at surface

Slow growth over a few grains

Lasts for most of fatigue life

Propagation and Stage 2 growth



Predominantly under tension

Dependent on tensile stress concentration at crack tip

Growth on planes of maximum tensile stress (over large areas)

Occurs at end of fatigue life

- *up to catastrophic failure at critical crack length*

Characterised by striation “shell” markings

- *making up majority of fracture surface*

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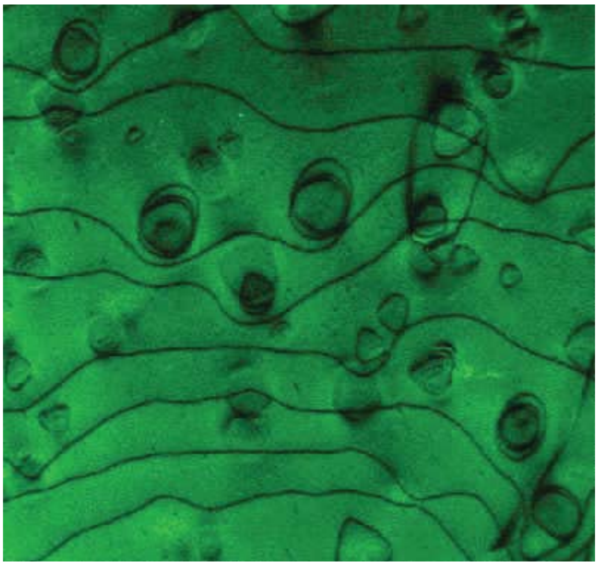
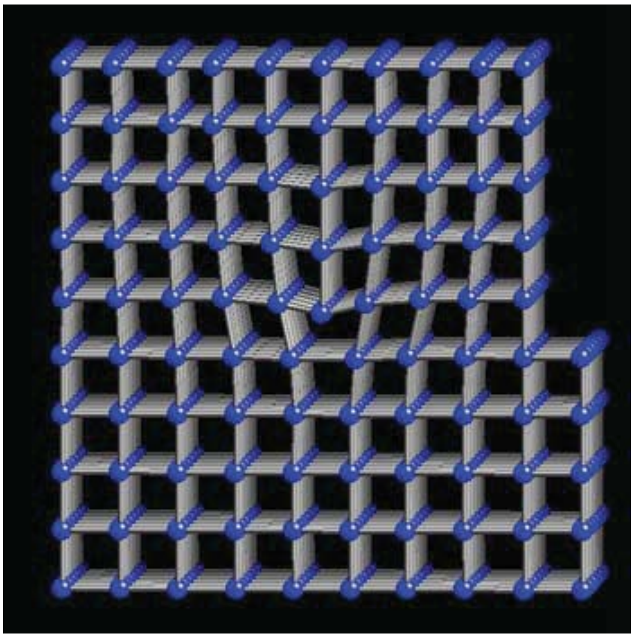
DISLOCATIONS



•WHAT ARE THEY?

•Defects that cause lattice distortion centred around a line.

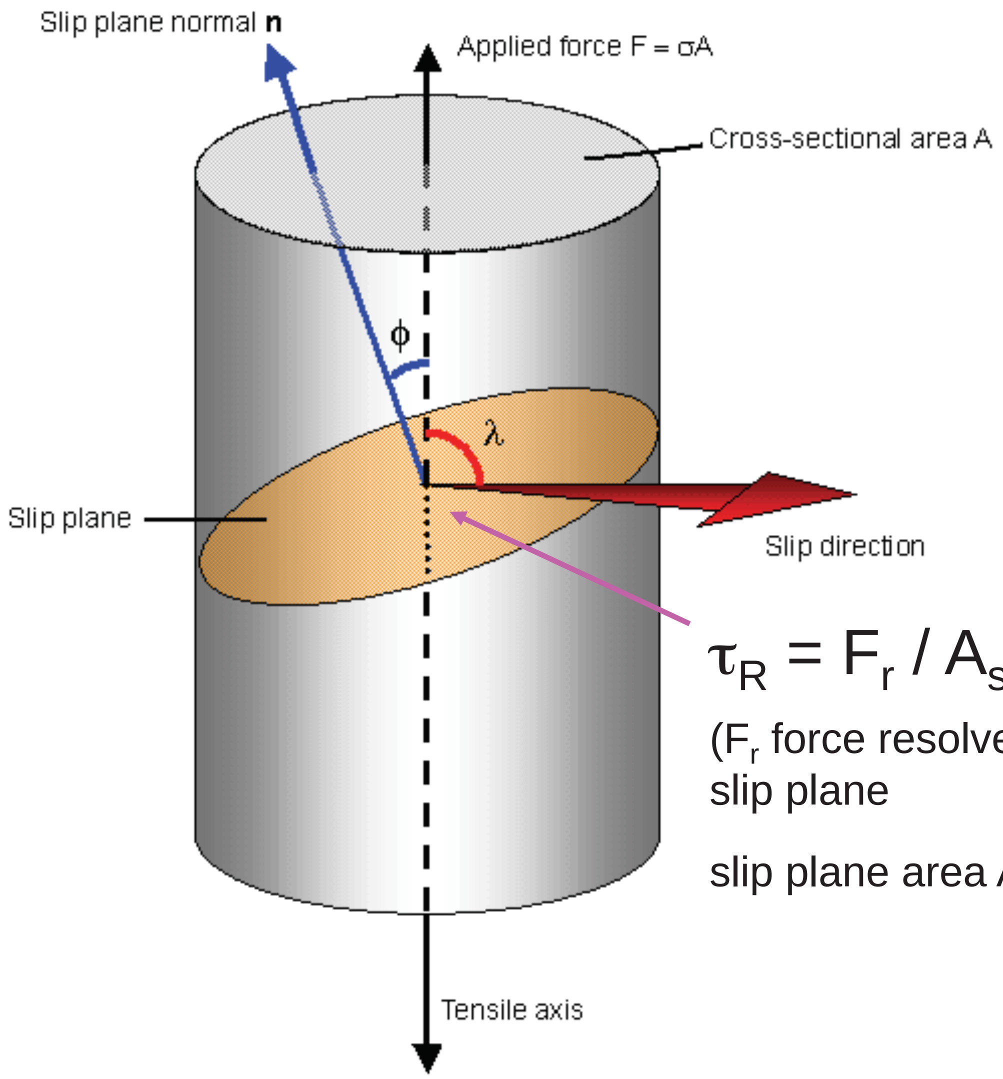
•Extra half plane of atoms in lattice



[p74-p78 Callister, Materials Science & Engineering – An Introduction TA403CAL]

Critical Resolved Shear Stress, (τ_{CRSS})

•A resolved shear stress τ_R may be produced on a slip system, causing the dislocation to move on the slip plane in the slip direction



$\tau_R = F_r / A_{sp}$

(F_r force resolved on slip plane

slip plane area A_{sp})

•When $\sigma \rightarrow \tau_R \geq \tau_{CRSS}$ then SLIP occurs

[p159-p163 Callister, Materials Science & Engineering – An Introduction TA403CAL]

FACTORS AFFECTING SLIP

- PERFECT, SINGLE CRYSTALS
- (N.B. REAL MATERIALS NOT SO!)

Factor	FCC	BCC	HCP
τ_{CRSS} (MPa)	**0.35-0.70**	**35-70**	**0.35-0.70**
No. of slip systems?	**12**	**48**	**3**[a]
Cross-slip?	**Can occur**	**Can occur**	**Cannot occur**[a]
Summary of properties	**Ductile**	**Strong**	**Relatively brittle**

[a]By alloying or heating to elevated temperatures, additional slip systems are active in HCP metals, permitting cross-slip to occur and thereby improving ductility.

•Why plastic deformation (slip) in metals?
•Why low for FCC and high for BCC?

- Slip planes & directions →

SLIP DIRECTIONS AND PLANES IN METALS

Crystal structure	Slip plane	Slip direction	Slip systems
BCC	**{110}** **{112}** **{123}**	**<111>**	**48**
FCC	**{111}**	**<110>**	**12**
HCP	**(0001)** **$\{11\bar{2}0\}$*** **$\{10\bar{1}0\}$*** **$\{10\bar{1}1\}$***	**<100>** **<110>**	**3**

• *These planes are active in some metals and alloys at elevated temperatures

READ; Chapter 9: The New Science of Strong Materials - JE Gordon

[p159 Callister, Materials Science & Engineering – An Introduction TA403CAL]

stage 1 is dominated by initiation and crack growth oriented at 45° to the direct principal stress axis, i.e. on planes of high shear stress which promote slip processes and plastic deformation. Note shear stresses will be created by compressive as well as tensile loading so that plastic deformation and stage 1 cracks can grow equally well in tension or compression during initiation. Growth usually depends on the properties of the surface and the immediate sub-surface. Stage 1 cracks extend at very low rates and only for a short distance, typically over a few grains and can last for most of the fatigue life.

In **stage 2** cracks grow on planes of high tensile stress essentially perpendicular to the principal stress axis and is responsible for most of the fracture of the material. Stage 2 cracks tend to grow (propagate) only under tension. In compression, crack faces may close and temporarily reduce the stress concentration and perhaps even cold weld the flaw. Stage 2 cracks are characterised by striations or "**shell markings"** on the crack surface which can be traced back to the initiation point. The crack grows through the material cross-section up to a critical length before catastrophic propagation through the remainder of the section.

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Striations “Shell markings”

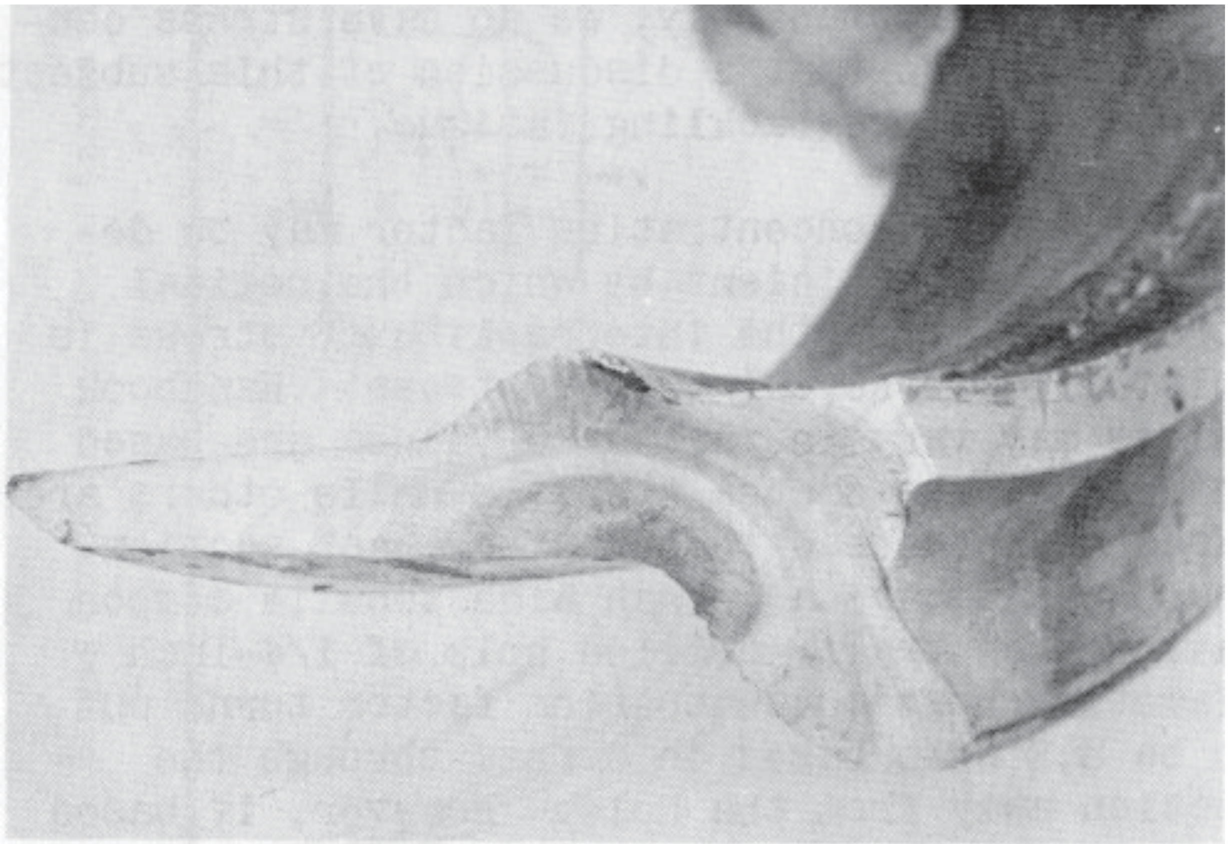

Fig. C13.1 Airplane wheel rim showing striations at fatigue fracture.

B787 Fatigue Testing

B787 Wing Testing 2

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Definition

The phenomenon:

Repeated cyclic loads — *- below elastic limit*

↓

Accumulation of fatigue damage — *- microscopic plastic deformations at localised zones*

↓

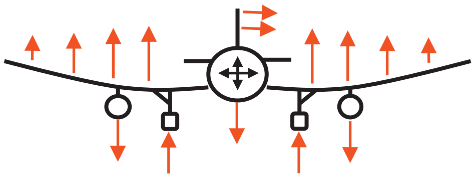
Crack formation & growth

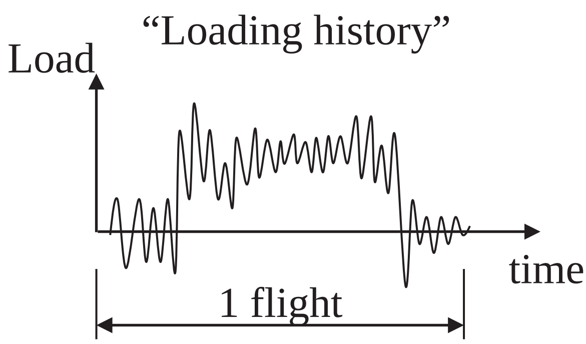
↓

Catastrophic failure!

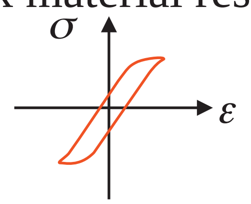
The Problem:

Complex service loading





Complex material response



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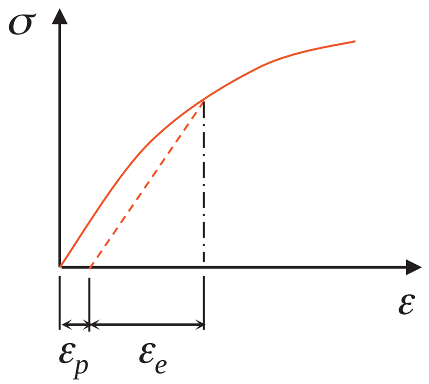
Fatigue Life and Base Data

There are 2 Regimes of Fatigue Life:

Low cycle fatigue < 50000 cycles *(High stress)*

High cycle fatigue > 50000 cycles *(Low stress)*



<u>Low Cycle Fatigue</u>

Consider elastic and plastic strains:

ε_p = *total strain less elastic strain*

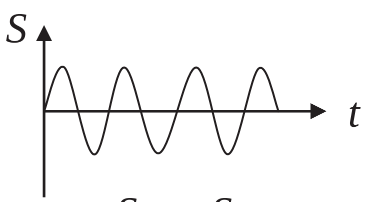
$= \varepsilon_{tot} - \varepsilon_e$

"S-N" Diagrams

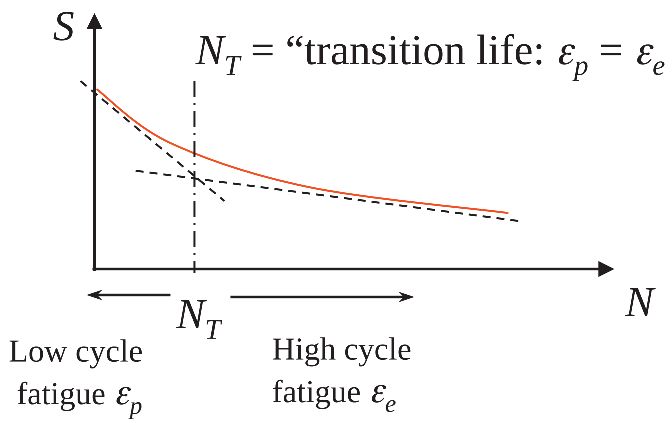
based on constant amplitude stress or strain

e.g. for **<u>zero</u>** mean cyclic stress:

For:



$S = \pm S_a$



Plastic strain ε_p dominates at high stress/strain, *i.e. short lives*

Elastic ε_e strain dominates at low stress/strain levels: *i.e. long lives*

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There are two regimes of fatigue life analysis depending on the level of cyclic loading and numbers of cycles to failure, commonly referred to as **Low cycle fatigue** (i.e. @ high cyclic stress) and **High cycle fatigue** (i.e. @ low cyclic stress).

Low cycle fatigue tests (i.e. @ high stress-strain levels) are usually carried out under strain control to avoid "runaway" instabilities common under high levels of cyclic stress .

Analysis of the low-cycle fatigue strain-life curve usually involves an assumption for the superposition of **elastic and plastic strain components** and an attempt to relate fatigue fracture to the initial quasi-static monotonic fracture behaviour.

The monotonic test effectively corresponds to a half a fatigue cycle and extrapolation back to the monotonic value assumes a relation between fatigue & monotonic strengths or strains, however this is not always the case!

The straight lines representing the elastic and plastic behaviour on the strain life diagram are described by power functions and are often of universal significance in fatigue. **Manson-Coffin** relations are available to estimate the plastic line.

Considering the total strain-life curve as the summation of the elastic strain life curve and the plastic strain-life curve, then at short lives (i.e. high stresses and strains) the plastic strain contributes more to the total strain than the elastic strain. The reverse is true for long lives, i.e. low strains and stresses. At the **"Transition Life"** the components of elastic and plastic strain are equal and this point marks the region where each strain component dominates.

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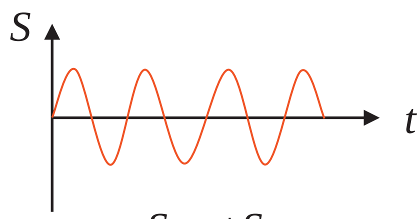
High-Cycle Fatigue

"S-N" Diagrams

based on constant amplitude stress

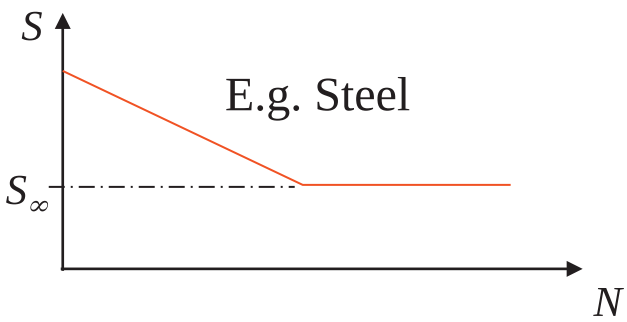
e.g. **zero** mean cyclic stress

For:



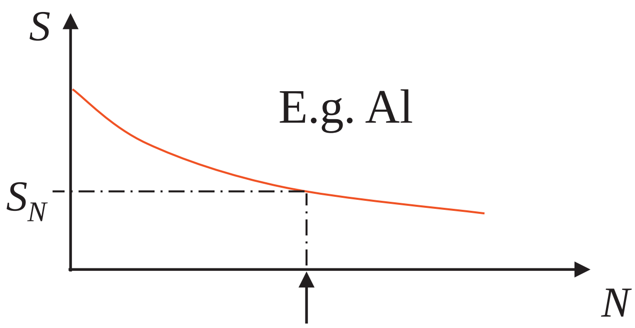
$S = \pm S_a$

Endurance or fatigue **limit**



[Note: in most cases of aircraft structural design the value of the fatigue or endurance limit is too low to be useful as a design limit]

Endurance or fatigue **strength**



S_∞ , S_N used as a measure of material fatigue resistance

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High cycle fatigue has received the most attention and is probably best understood in classical fatigue analysis. Furthermore, high cycle fatigue design is more relevant to aircraft structures and it is this regime which is covered in this course.

High cycle fatigue material performance is usually characterised by stress-life **"S-N" diagrams** obtained for constant amplitude tests - e.g. fully reversed (zero mean) cyclic stresses**.** Cyclic stress for a given life or life for a given cyclic stress can be obtained from S-N plots.

Note, the "**fatigue or endurance limit"**, S_∞ , is the reversed cyclic stress that can be sustained for effectively infinite fatigue life (i.e., so long that it can not practically be measured). Some steels exhibit an effective endurance limit. *Note however, in most cases of aircraft structural design the value of the fatigue or endurance limit is too low to be useful as a design limit.*

The "**fatigue or endurance strength"**, $\mathbf{S_N}$, is the reversed cyclic stress that can be sustained for a given life of N cycles, typically 10^6 cycles. This value is often quoted as a measure of a material's resistance to fatigue where there is no obvious fatigue limit.

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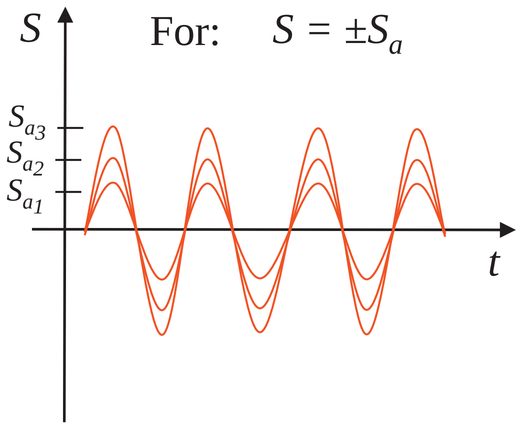
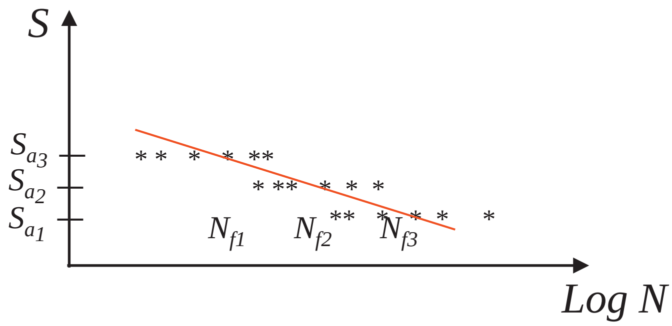
Fatigue S-N test data

Obtained from lab specimen tests

Significant scatter in lives is common *(increasing with N_f)*

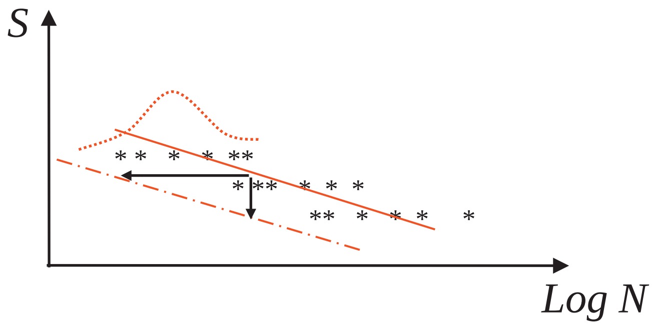




E.g. typically factors of 3 in fatigue life @ 10^6 cycles!

"Safe Design Curves"

Stress factors:
$S_{Nmean} - kS_{std\ dev}$



Life factors: N_{mean}/k

Factor k depends on No. of tests carried out

Typically k = 3 or 4 (ESDU Fatigue Vol. 1)

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S-N curves are usually derived from constant amplitude fully reversed cyclic **stress fatigue tests** on standard quality lab specimens. Despite the use of apparently identical specimens there is usually significant **scatter of fatigue lives** which tends to increase with increasing life. Life scatter factors of up to 3 are typical in the 10^6 cycle range.

Because of the exponential nature of fatigue it is usual to consider the log No. of cycles.

To account for scatter the S-N curves are factored by life or stress factors to produce **"safe" design curves.** The value of factor to be used is calculated according to the number of test specimens making up each sample at a given stress level and the resulting scatter. E.g. For a sample size of 6 specimens, life and stress factors of 3 or 4 are typical**.** Standard guidelines can be found in the ESDU Fatigue data sheets volume 1.

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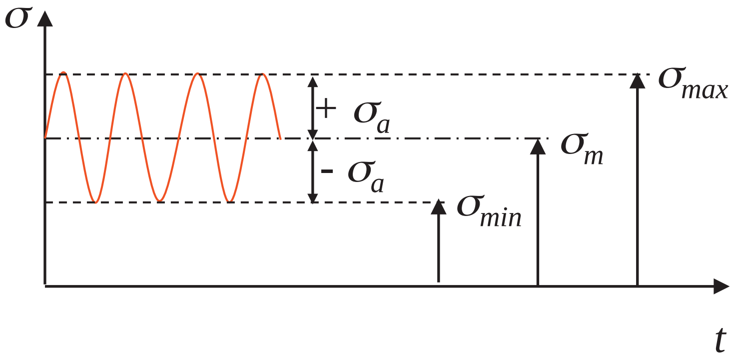
Stressing Effect

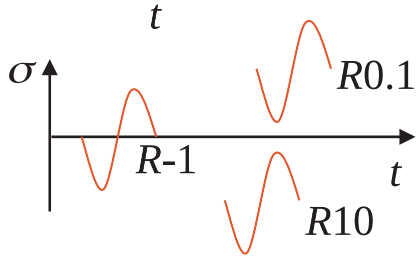
Non-Zero Mean Cyclic Stresses

In most cyclic loading histories: alternating stresses are superimposed upon non-zero mean stresses



$$\sigma_{max} = \sigma_m + \sigma_a$$
$$\sigma_m = (\sigma_{max} + \sigma_{min})/2$$
$$\sigma_a = \pm(\sigma_{max} - \sigma_{min})/2$$
$$\sigma_{min} = \sigma_m - \sigma_a$$

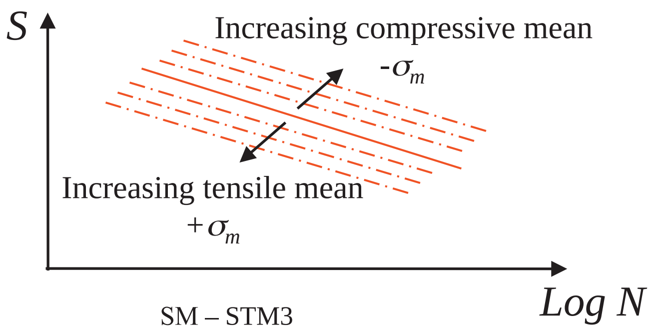
R-Ratio $R = \dfrac{\sigma_{min}}{\sigma_{max}}$ E.g.



Mean stress effect on fatigue life:

Tensile mean stress → reduction in life

Compressive mean stress → increase in life



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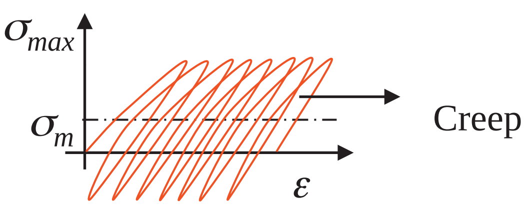
Although many S-N curves are based on load cycles with zero mean, most load cycle histories include significant mean loads. In the early days, fatigue test equipment limitations meant that only reversed cyclic loading was readily available, e.g. rotating bending machines. Nowadays programmable servo-hydraulic test equipment allow variable amplitude fatigue testing -including changing mean loads as well as changing amplitudes.

Load cycles including non-zero means are conveniently described by “R-ratio” values. I.e. the ratio of minimum to maximum cyclic stress.

Mean stresses can significantly alter the resistance to cyclic loads. Tensile mean stresses tend to reduce fatigue life and compressive mean stresses tend to increase fatigue life. The relative effects of tensile and compressive mean stresses can be understood when considering a crack within the material. I.e. tension opening and compression closing cracks.

Mean stresses can in fact result in creep under load control. If the material has different tensile and compressive responses then creep can even occur under fully reversed cyclic stresses.



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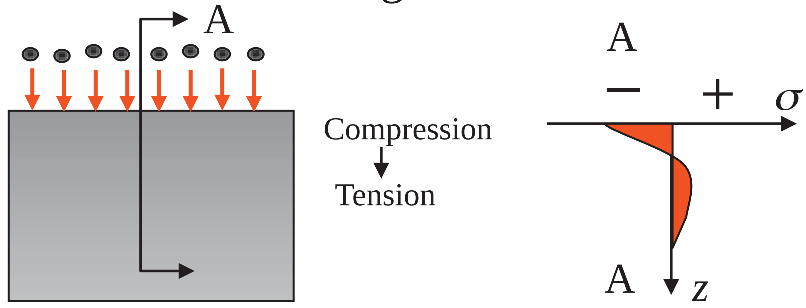
Residual Stresses

Residual stresses are effectively **<u>local</u>** internal mean stresses *which will be additional to externally applied stresses*

Can be beneficial if in correct sense. E.g.:

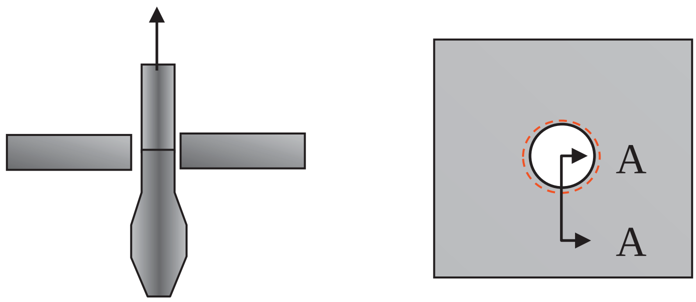
Peening

e.g. x10 Fatigue life
for mild steel

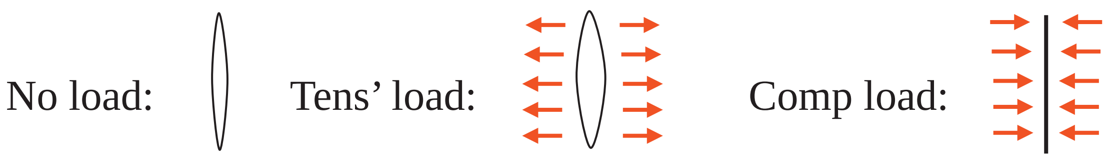


Cold worked holes

After drilling and reaming, oversized tapered mandrel pulled through the hole.
Result is a compressed material zone around hole which will delay onset of cracking.

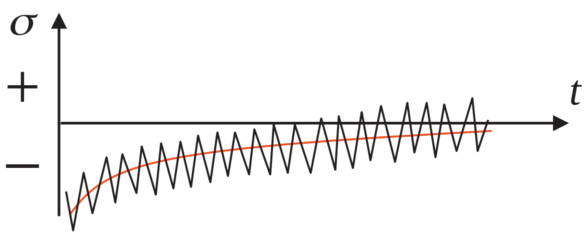


Mean stress effect on cracks



Relaxation!

Residual σ can change exponentially
with load cycling & heating – benefits not reliable.
If crack already in equilibrium tensile zone
then fatigue life reduced.
These methods **should be used with caution**.



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Residual stresses exist in most materials and act effectively as local mean stresses upon which the external mean and cyclic stresses are superimposed.

Residual stresses must be balanced within the material, i.e., a tensile residual stress will be balanced by a compressive residual stress and vice versa.

Acting in the desired sense, residual stresses can be beneficial. For example **Peening** a metal surface (i.e. striking the surface with lead or steel shot) is a common method of creating a beneficial compressive residual stress at the surface which will delay the onset of cracking. Using this method fatigue life can be increased significantly, e.g. by a factor of 10 for mild steel.

Another example of the use of favourable compressive residual stresses is the **cold-working** of drilled holes. In this process after drilling and reaming an oversized tapered mandrel is pulled through the hole. The result is a compressed material zone around the hole which will delay the onset of cracking.

However, residual stresses can change exponentially with load cycling and other actions such as heating and so these benefits may not necessarily be relied upon. Also if a crack already exists which extends into the equilibrium tensile zone then the fatigue life may be reduced. Consequently these methods **should be used with caution.**

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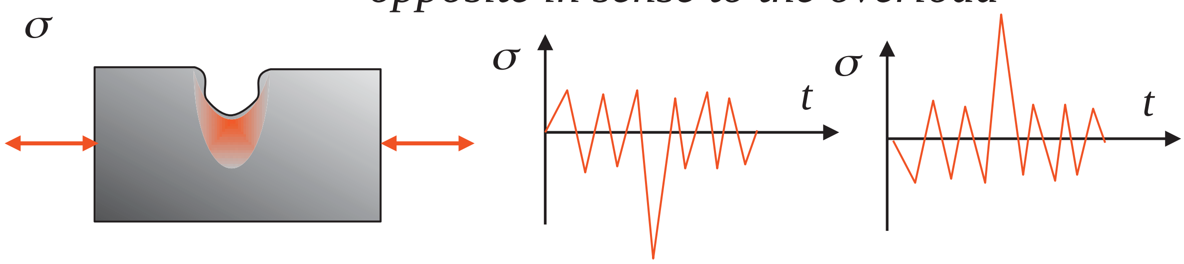
Overloads

[Abrupt transfer from low to high σ or single high loads "over-loads" can produce residual stresses at notches which can play an important role in fatigue.]

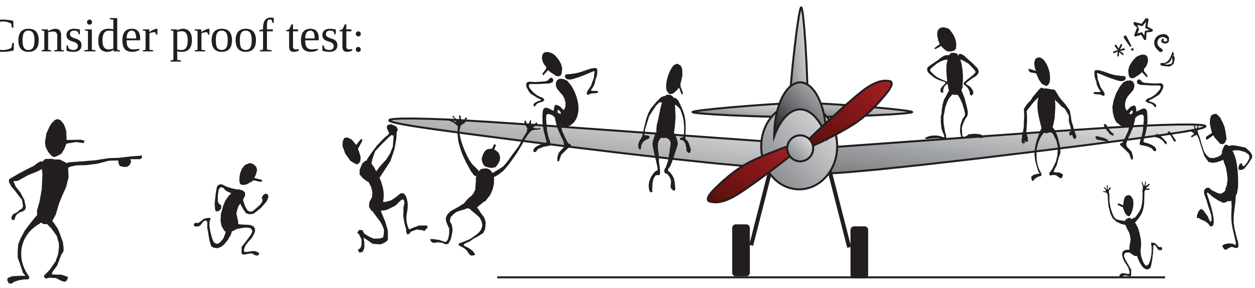
Overloads create residual stresses @ notches

opposite in sense to the overload

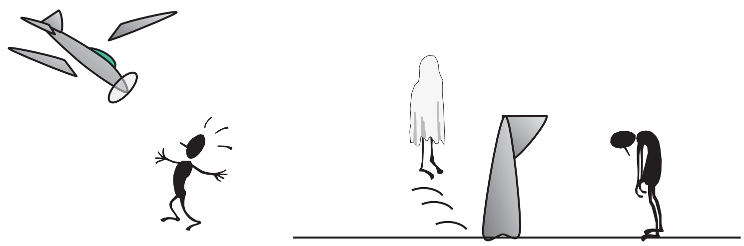


E.g. a tensile overload produces a compressive residual stress & vice versa

Consider proof test:



Sense of overloads and resulting residual stresses should be considered

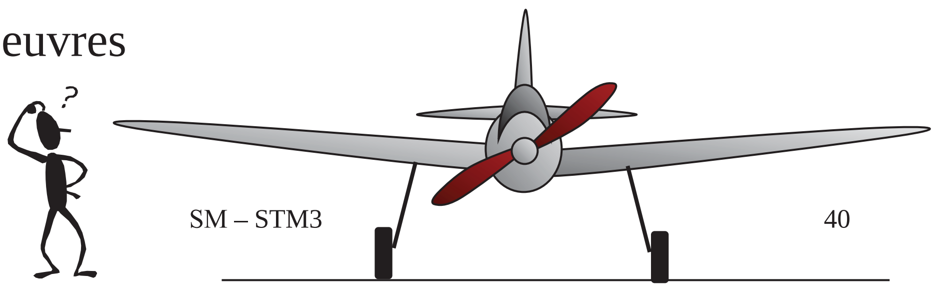


E.g.s of overloads:

Freak gusts

Emergency manoeuvres

Heavy landings



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Abrupt transfer from low to high stress or single high loads "over-loads" can produce residual stresses at notches which can play an important role in fatigue.

Notched members overloaded in tension and then cycled under small fully reversed loads can give increased fatigue life and when overloaded in compression give decreased fatigue life! This is the opposite of the general trend for mean stress effects discussed earlier, the reason is that overloads tend to produce residual stresses of opposite sign in stress concentration regions.

It is therefore important to consider the generation of residual stresses, at least for sign and location for further analysis!

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Move over!

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Notches

Notch fatigue strength reduction factor:

$$K_f = \frac{\text{fatigue strength without notch}}{\text{fatigue strength with notch}}$$

Sometimes used to estimate material notched fatigue performance from plain material S-N data

But! K_f = f(size, shape, applied stress, fatigue life range)
i.e. **NOT** a material property
∴ Use with care!

[Is NOT a material value and depends on size, shape, applied stress and fatigue life range AND the approach can NOT account for the effect of mean and residual stresses]

Fatigue notch sensitivity parameter:

$$q = \frac{K_f - 1}{K_t - 1}$$

Note, full fatigue notch sensitivity when	Kf = Kt, q = 1
.. .. insensitivity when	Kf = 1, q = 0

Where K_t = static notch stress concentration factor

$$= \frac{\text{static strength without notch}}{\text{static strength with notch}}$$

This is indicative rather than exact for fatigue – use with caution!

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Attempts have been made to estimate the fatigue performance of notched material from plain material making use of a "Notch fatigue strength reduction factor", K_f. Note, K_f is not a material value and depends on the size, shape, applied stress and fatigue life range and the approach can not account for the effect of mean and residual stresses. Therefore, care is needed for correct application.

A "Fatigue notch sensitivity parameter", q is also defined based on the fatigue reduction factor and static monotonic stress concentration factor, K_t.

Note, full fatigue notch sensitivity when	$K_f = K_t$, $q = 1$
.. .. insensitivity when	$K_f = 1$, $q = 0$

There are simple expressions available for these concentration factors based on elastic behaviour, however, the true stress concentrations including the stress reduction effects of plastic deformation should be used. I.e. plastic yielding can occur at the roots of notches and other sources of stress concentrations such as cracks, which can be beneficial in reducing the stress concentration.

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Summary

(i) Fatigue is a plastic process.

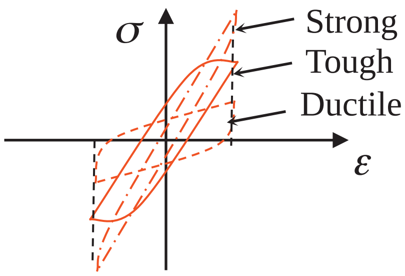
Cyclic plasticity leads to local exhaustion of ductility

(ii) Fatigue is a localised problem.

(iii) Use ‘safe life’ fatigue design where crack detection is difficult

(iv) Different metals have different hysteresis response.

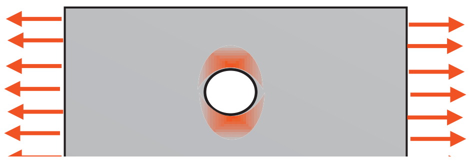
e.g. 7075 upper wing skins vs. 2024 lower wing skins



(v) Stress-strain hysterisis response can change with cycling.

(vi) Metals provide global stiffness and strength

and local ductility!



Local ductility

@ stress conc

(

Dangerous to take anything for granted in fatigue without having experimental evidence!

A320 Extended life program

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Airbus begins tests to extend service life of A320 family

+1 Tweet

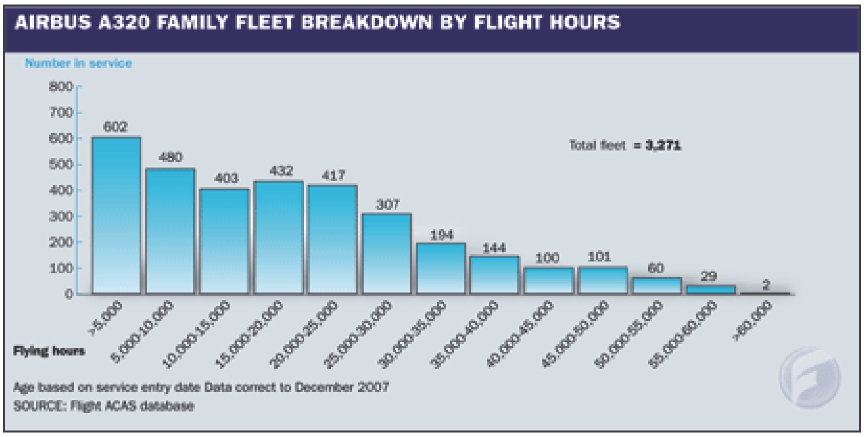
By: MAX KINGSLEY-JONES LONDON
Source: FLIGHT INTERNATIONAL

01:38 22 Jan 2008

Airbus is about to begin fatigue tests of full-scale A320 sections as it undertakes a major programme to extend the life of twinjet that could ultimately result in the flight-hour (FH) limit being tripled to 180,000h.

The programme, Extended Service Goal (ESG), was launched a year ago as the high-time A320 was approaching the 60,000h goal originally set for the twinjet when it entered service in 1988 with Air France. With 3,300 A318, A319, A320 and A321 aircraft in service, and 2,500 on order, Airbus is moving to ensure that the fleet can remain operational well into the second half of the century.

"When we launched the A320 we planned for a 25-year design service goal based on an assumption that each cycle would be 1.25h," says Antoine Vieillard, vice-president A320 family customer services. "Today the actual average flight duration for the fleet is 1.82h - we are doing many more flight hours than we had originally expected."



60,000 hour threshold

The A320 flight-hours fleet leader passed the 60,000h threshold in October 2007 and "six or seven aircraft have now exceeded 60,000h", says Vieillard. The fleet leader is currently at just over 40,000 cycles and is set to reach 48,000 in early 2011. "The hourly DSG threshold has been reached first as the aircraft is being worked on longer routes than we estimated," says Stuart Mann, director of product marketing, A320 family.

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(i) Cyclic plasticity is responsible for the exhaustion of ductility in small localised regions of metal. When the available ductility is used up in a region then the material can only continue to absorb energy by creating a new surface area, i.e. a crack. So that even ductile materials fracture in a brittle manner.

Fatigue damage accumulation is effectively a plastic deformation process and a crack is the final manifestation of the cumulative fatigue damage process.

Fracture is an elastic process where an existing crack grows to critical length causing catastrophic failure.

(ii) Fatigue is a local problem so look for local stress concentrations! Increasing the cross-sectional area of a structure will not necessarily increase the fatigue life. Under strain control larger cross-sections may result in higher bending stresses and make the situation worse!

(iii) An easily detectable fatigue crack often appears only in the last 10% of fatigue life! Therefore early detection of cracks can be difficult and expensive.

(iv) Different material cyclic behaviours can be illustrated by stress strain hysteresis plots.

(v) The cyclic stress strain response of a material can be altered with cyclic loading.

(vi) In metals it is possible to provide global strength to transmit loads without excessive plastic deformation and local ductility to provide plastic flow and prevent crack initiation and subsequent propagation at notches.

(vii) **It is dangerous to take anything for granted in fatigue without having experimental evidence!** A range of tests from coupons to complete airframe structures must be tested under representative fatigue loading to confirm fatigue performance.

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Fatigue Analysis of Metals

Life Prediction under aircraft service loading

Contents

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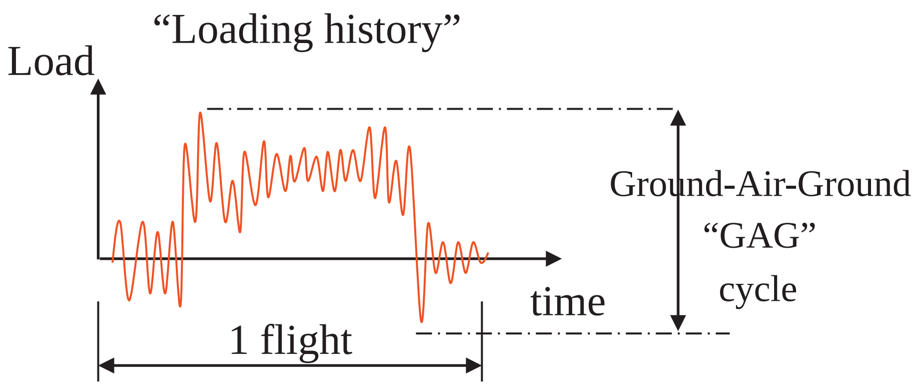
Load Cycle History Idealisation

Flight load cycles: *Gusts & manoeuvre*

Ground Load cycles: *Taxi, take-off, landing*



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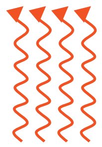
Thermal cycles 

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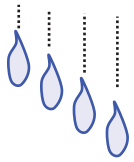
Moisture cycles 

Definition of individual "damage events" = Fatigue problem

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Aircraft service fatigue loading is made up of complex sequences of flight and ground loads. Flight loads originate from manoeuvres and gusts and Ground loads originate from taxi, take-off and landing. The resulting spectrum of aircraft service fatigue loading consists of a mix of deterministic and random cycles which must be idealised for cumulative fatigue analysis.

As well as mechanical load cycling the structure may also be subjected to significant thermal cycling and moisture cycling and it may also be necessary to account for these effects in fatigue analysis.

Implicit in the idealisation of a loading history is the definition of “damage events” making loading history idealisation a fatigue problem rather than just a statistical problem. Each cyclic event must defined and accounted for in terms of damage for subsequent cumulative fatigue analysis.

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Aircraft load history monitoring & idealisation

Monitoring of A/C variable amplitude load histories (since 1950's)

for selected a/c types

↓

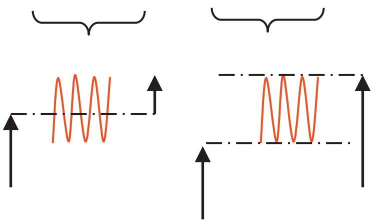
Cycle idealisation

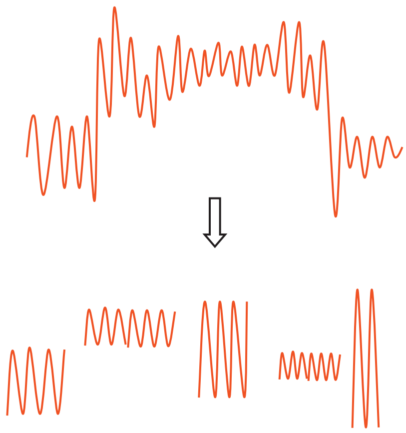
using "cycle counting methods"

↓

Sets of constant amplitude cycles

defined as σ_m, σ_a, or σ_{pk}, σ_{tr}





N.B. attempt to reduce complex VA load sequences to sets of simpler CA cycles described by peak and trough values or mean and alternating values.

Cycle counting methods

Depend on assumptions about fatigue damage accumulation:

- Cycle not time dependent
- Independent of loading wave-form
- Independent of load sequence

The definition of CA cycles from VA cycles is ambiguous and arbitrary leading to many different counting methods

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Since the 1950's selected aircraft load histories for particular aircraft types have been monitored and the results compiled to provide generalised data over hundreds of hours of flight time. The degree of information retrieved has been limited by the monitoring equipment capabilities and cumulative damage theories available at the time. Typical information from early monitoring programmes only provided values of major load peaks.

The idealisation of complex service loading histories has traditionally been carried out by "Cycle Counting Methods". Counting methods attempt to reduce complex variable amplitude load sequences to sets of constant amplitude cycles described by peak and trough values or mean and alternating values. (It is important to realise that all counting methods assume that fatigue is cycle rather than time dependent and independent of loading wave form and sequence).

Counting methods can vary from simple arbitrary peak or range counts to counts based on the theoretical assumptions of the fatigue process and the specific method used can have a significant effect on the fatigue assessment. The selection of the most appropriate counting method is essentially a fatigue problem and not a statistical problem. However, incomplete knowledge of the fatigue process means that the majority of the counting methods have no real theoretical background and are essentially arbitrary. The ambiguity of defining discrete constant amplitude cycles from variable amplitude cycles has led to the development of various counting methods.

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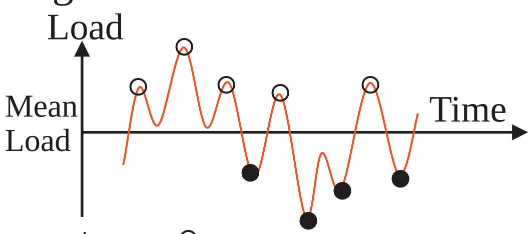
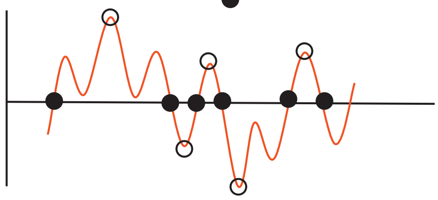
Different counting methods:

E.g.:



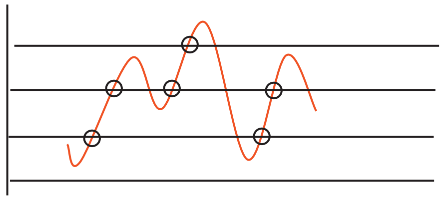
Peak

- ○ maxima counted
- ● minima counted



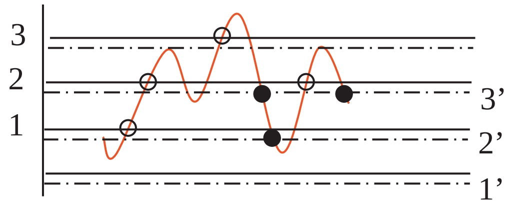
Peak between mean crossing

- ● mean crossing
- ○ peaks counted



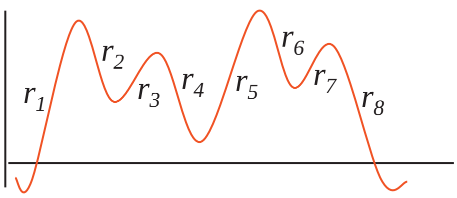
Level crossing

- ○ Level crossings counted



Restricted level crossing

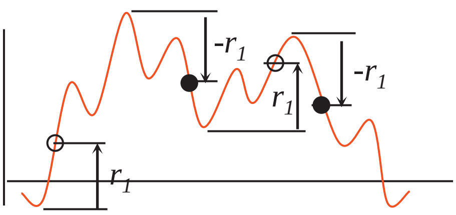
- ○ 1st counting condition, levels 1,2,3
- ● 2nd counting condition, levels 1',2',3'



Range

+ve ranges: r_1, r_3, r_5, r_7

-ve ranges: r_2, r_4, r_6, r_8



Range - pair exceedance

- ○ 1st counting condition, r > +r1
- ● 2nd counting condition, |-r | > | r |

essentially arbitrary

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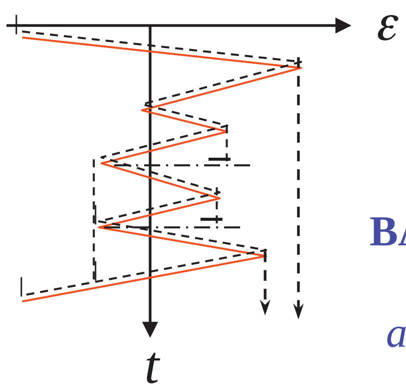
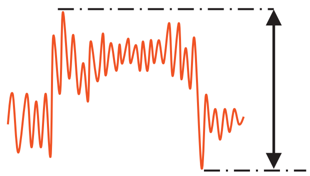
The Range-Pair-Range “Rainflow” counting method

Theoretical basis: (http://en.wikipedia.org/wiki/Rainflow-counting_algorithm)

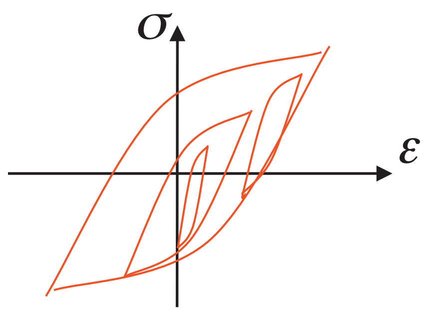
Cycles defined by stress-strain hysteresis loops:

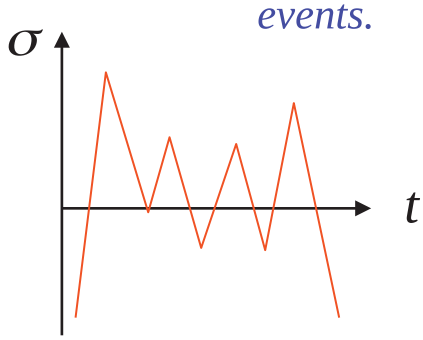




“GAG” cycle

BASIC ASSUMPTION: *Pairs of load reversals which correspond to closed hysteresis loops assuming that these represent discrete damage events.*





Each closed loop count has a mean and range value associated with it so that CA fatigue data can be defined from VA loading.

"The damage caused by a large hysteresis event is not affected by interruption to complete a small hysteresis event"

$\Rightarrow$ *"Material strain memory effect"*

Assuming stable cyclic stress-strain behaviour, regarded as best assessment of cyclic damage events for fatigue life estimation of metals using cumulative damage theories

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The only method with a firm theoretical base is the “Range-Pair-Range” technique, known as the “Rainflow” counting method.

The Rainflow method selects pairs of load reversals which correspond to closed hysteresis loops assuming that these represent discrete damage events. A basic assumption is that the damage caused by a large event is not affected by its interruption to complete a small stress-strain loop, the damage of the interruption is simply added to the total as an event in its own right. This assumes a “material memory effect” which results in continuous hysteresis curves for interrupted events.

Assuming a stable cyclic stress-strain behaviour exists, the Rainflow method can be regarded as the best assessment of cyclic damage events for fatigue life estimation of metals using cumulative damage theories.

Each closed loop count has a mean and range value associated with it so that constant amplitude fatigue data can be defined from variable amplitude loading.

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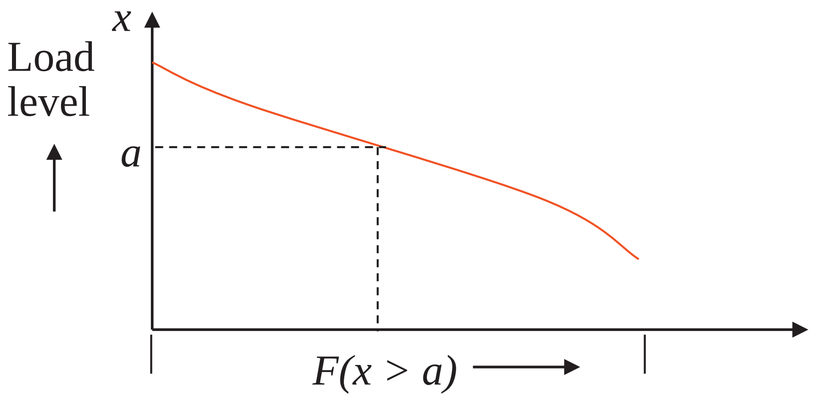
Load spectrum "exceedence curves"

"Load spectrum" = range of CA counted cycles represented by "exceedence curves" (derived from counting method e.g. 'Rainflow' analysis):



1	Eg. 5000	10^6	Cumulative (total) occurrences
10^{-6}	Eg. $5x10^{-3}$	1	Cumulative frequency of occurrence *(Frequency of exceedings)*

E.g. 5000 cycles in 10^6 as least as large as *a* etc.

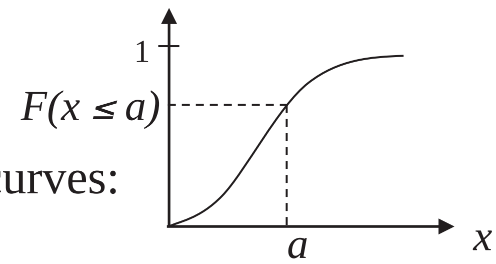
Convenient for input to simple *cumulative damage algorithms* and to perform fatigue tests simulating the service loading environment.

Tests can be carried out by repeating the basic spectrum "block" of cycles until failure.

Note:

Exceedence curves

≡ cumulative distribution function curves:



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The range of constant amplitude cycles counted from monitored load histories is referred to as a “Loading Spectrum”. The up and down load peak occurrences obtained from the counting methods are usually presented as cumulative load distribution curves showing the number of times a given or higher load occurs. The information is usually illustrated graphically and the number of occurrences or cumulative frequency are plotted as the x-axis and load level as the y-axis. The plot is often referred to as an “Exceedence Curve” and is a convenient form of the usual cumulative distribution plot.

Spectrum exceedence curves provide a convenient form for input to simple cumulative damage algorithms and to perform fatigue tests simulating the service loading environment. Tests can be carried out by repeating the basic spectrum "block" of cycles until failure. Cycles within the block can be re arranged according to step approximations of the spectrum or may be "Pseudo randomised" into flight by flight type sequences.

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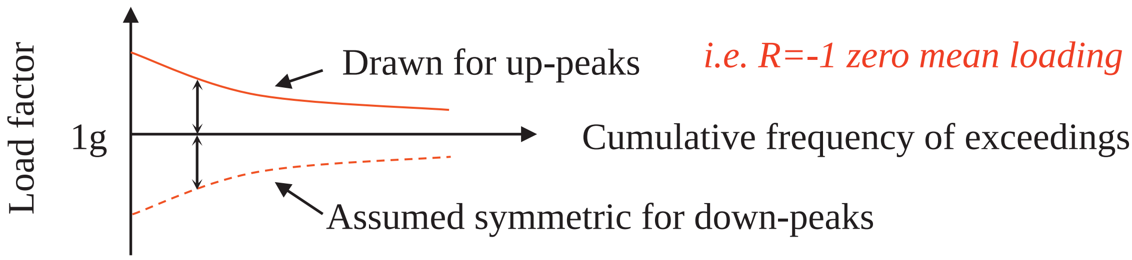
Uses of spectrum exceedence curves

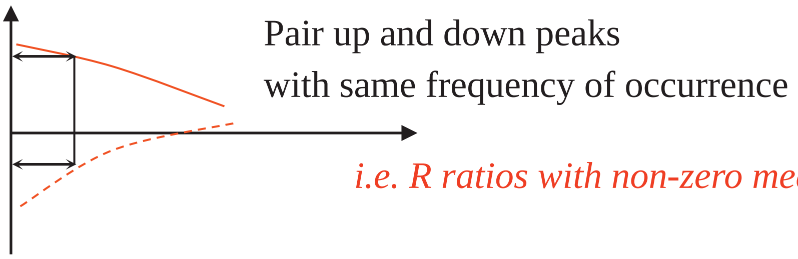
Combination of up and down peaks

Symmetric spectrum *Gust loading



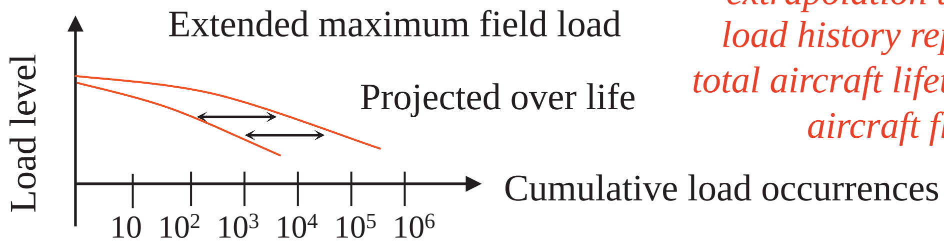
i.e. R=-1 zero mean loading

Unsymmetric spectrum *Manoeuvre loading



i.e. R ratios with non-zero mean

Spectrum extrapolation



extrapolation to produce a load history representing a total aircraft lifetime or even aircraft fleet lifetime.

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Exceedence curve plots provide a useful form for spectrum presentation and analysis. Some of the uses are outlined below:

Combination of Up and Down Peaks:

For relatively **symmetric** load peak distribution functions, typical of **gust loading**, the cumulative spectrum exceedence curves are usually drawn for the up-peaks only and the whole spectrum is then assumed to be symmetrical about the mean load.

For distinctly **unsymmetric** load peak distribution functions, typical of **manoeuvre loading**, the up and down peak values are either paired by the counting method, or by the exceedence curve pairing up and down peaks which have the same frequency of occurrence. Although, there is no evidence to justify this, the procedure is thought to be conservative in maximising possible load cycle amplitudes.

Spectrum Extrapolation: The distribution shape obtained from a limited sample of load history data can be extrapolated to produce a load history representing a total aircraft lifetime or even aircraft fleet lifetime.

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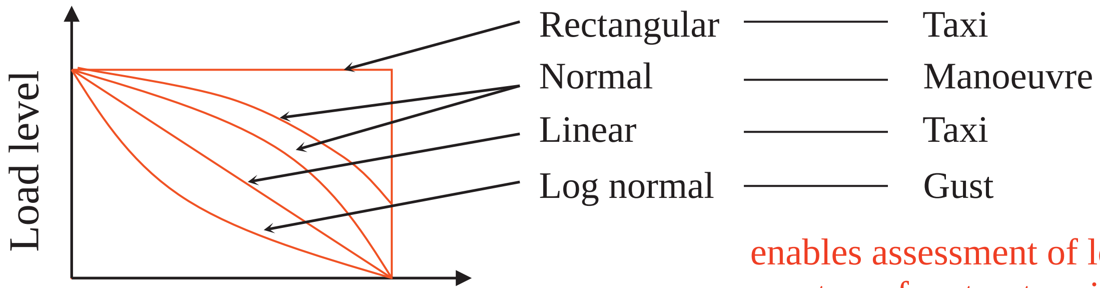
Load spectra forms

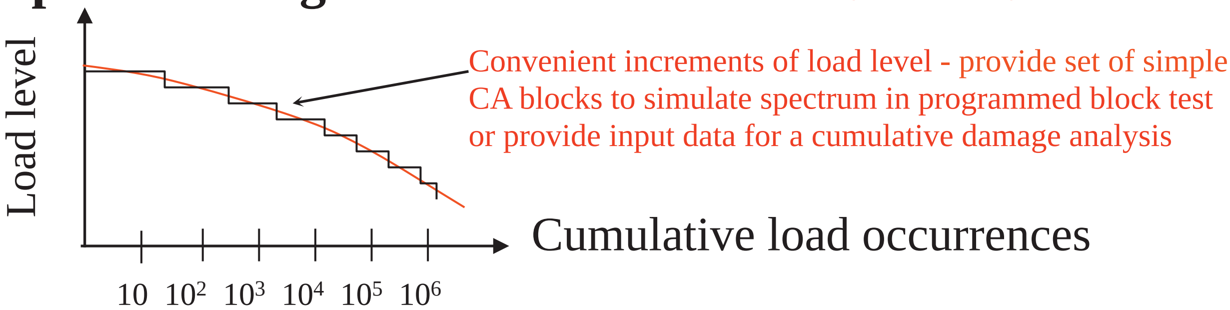
Load spectrum distribution forms



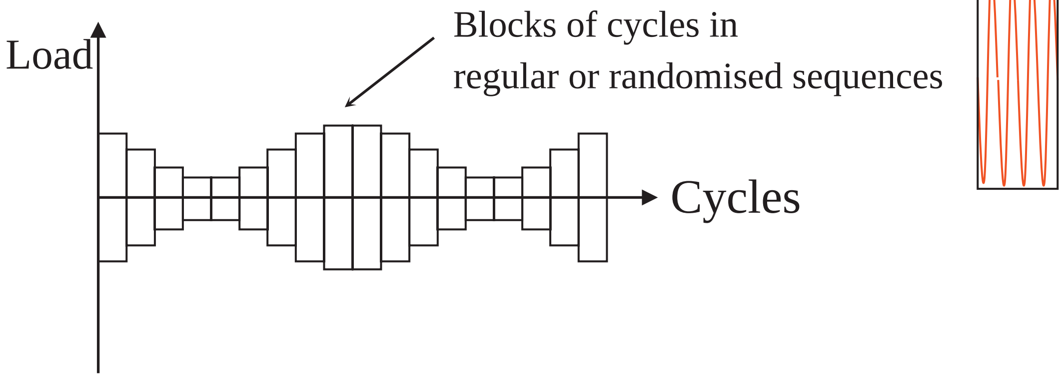
F(x>a) Cumulative load occurrences

enables assessment of load spectrum for structure in design stage & estimation of fatigue life before entering service or before availability of prototype.

Step-load histogram form



Program block spectrum simulation



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Load spectrum distribution forms: Exceedence curves offer a convenient means of comparing different load spectra distributions, allowing results from different spectrum tests to be correlated. The availability of recognised load spectrum distributions enables assessment of the load spectrum of a particular structure in the design stage and estimation of the fatigue life before the structure enters service or even before the availability of a prototype.

Step-load Histogram Approximation: Exceedence curves can be stepped in load level increments to provide a set of simple constant amplitude blocks to simulate the spectrum in a **programmed block test** or to provide input data for a cumulative damage analysis.

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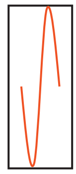
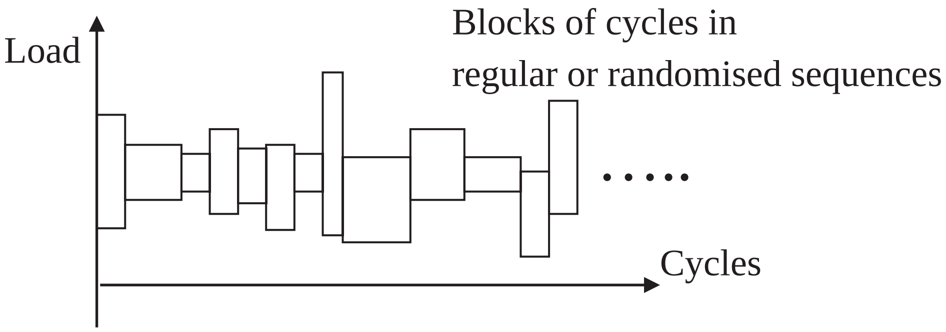
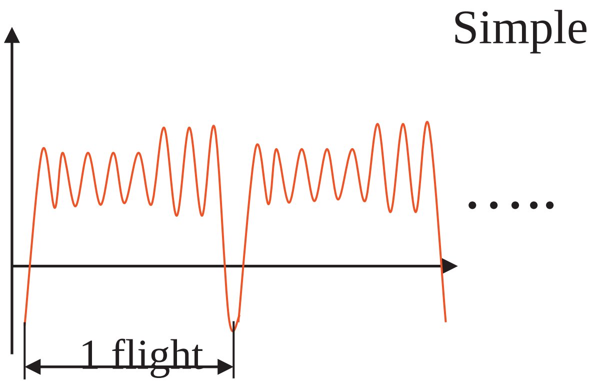
Resequenced test spectra

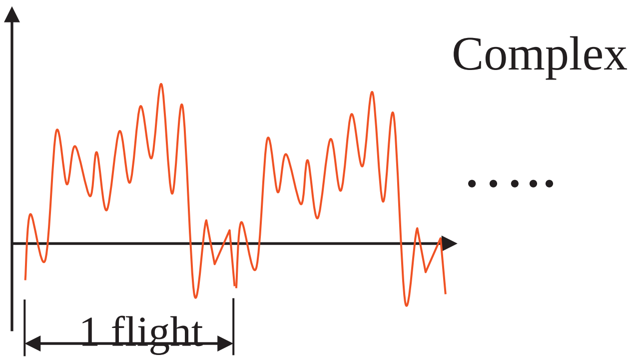


"Programmed Block"



"Flight by Flight"





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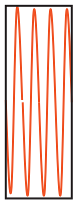
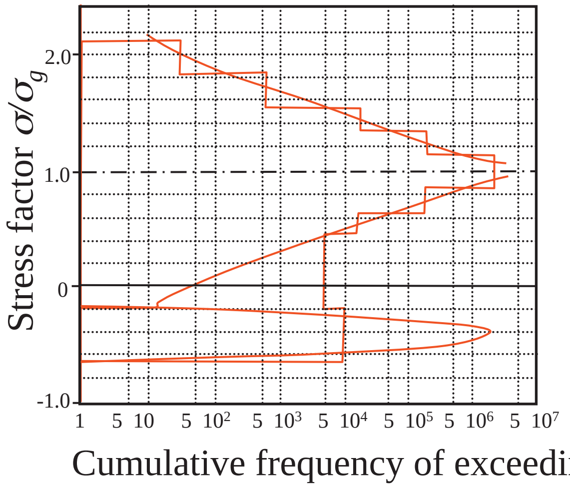
Standard Load Spectra

- Ensure conformity of load spectrum tests at different establishments

Transport aircraft spectra e.g. "TWIST”

Transport WIng Standard Test

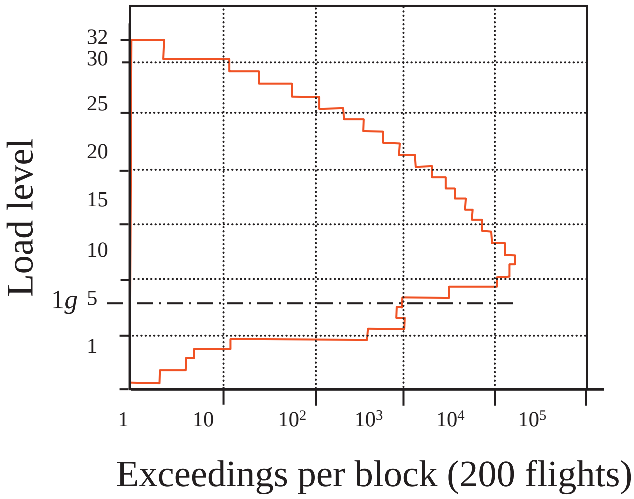




- <u>Gust dominated</u>
- Log-normal
- Symmetric
- Separate taxi

Fighter aircraft spectra: e.g. "FALSTAFF"

Fighter Aircraft Loading STAndard For Fatigue



- <u>Manoeuvre dominated</u>
- Normal/ Log-normal
- Unsymmetric
- Taxi included

Helicopter rotor load spectra are “HELIX” and “FELIX”.

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Investigations have been carried out to provide standardised spectra for specific types of aircraft and associated loading environments to ensure conformity of load spectrum tests at different establishments.

The type of flight envelope, level of aerodynamic wing loading and relative magnitude of gust and manoeuvre response result in particular load spectrum trends for different aircraft types. **Transport aircraft type** flight load sequences are described to a large extent by gust loading and as such tend to be of log-normal distribution and symmetric with respect to up and down peaks. **Fighter Aircraft type** flight load sequences are described entirely as manoeuvre loading and tend to be of normal distribution for up peaks and log-normal for down peaks and non-symmetric. Ground load sequences are seldom described in detail.

Examples of standardised fixed wing aircraft load spectra are “TWIST” (Transport Wing Standard Test) and “FALSTAFF” (Fighter Aircraft Loading Standard for Fatigue). Examples of helicopter rotor load spectra are “HELIX” and “FELIX”.

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Assumptions and simplifications of standard load spectra

Data base: - accelerations of a/c c.g.

- a/c type
- flight samples

Falstaff based on Al alloy fighter with primary interest in wing root. Data taken from 5 European air forces operating 4 different fighter a/c

Spectrum block size: - limited flights per block

Falstaff block size = 200 flights = ave annual European fighter utilisation

Omissions of low loads: - range filtering

Falstaff uses range filter value = 10% highest stress reached in 200 flight block, giving an ave. 80 cycles/flight

Truncations of high loads: - based on metal response

High loads - plastic zone at crack tips, reduce stress concn., increase fatigue life.
Will not often occur in service, unwise to include in test spectra.
Highest value in Falstaff taken as the 1 in 100 flight value.

Sequence effect: - original info lost

- arbitrary regeneration
- block by block

or flight by flight

Falstaff - 200 flight blocks, incl. T/O & landing, represents 3 mission severities, i.e. Air-to-Ground Combat, Air-to-Air Combat & Navigation

Wave-form shape and rate: - original info lost

- arbitrary sine wave
- convenient frequency

Simple cumulative damage theories - no account of wave-form shape or load rate

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The creation of a standardised load spectrum involves many assumptions and simplifications, these are listed below *(with examples for the Falstaff spectrum):*

Data base: Load factor histories are usually obtained from centre of gravity measurements from a number of different aircraft representing a particular aircraft type, e.g. Transport, or Fighter. *(Falstaff is based on aluminium alloy fighter aircraft structure with primary interest in the wing root region. The data was taken from 5 European airforces operating four different fighter aircraft.)*

Size of spectrum block: The overall number of flights in a spectrum definition must be limited to a convenient size and yet represent the expected aircraft loading. The spectrum block is repeated identically in test and analysis. *(Falstaff block size is 200 flights representing average annual European fighter aircraft utilisation.)*

Omission of low loads, i.e. range filtering: To limit the number of load cycles represented only load ranges above a given size are included, assuming smaller load ranges are not damaging. *(Falstaff uses a range filter value of 10% of the highest stress level reached in the 200 flight block, giving an average of 80 cycles per flight.)*

Truncation of high loads: Spectrum high load truncation is arbitrarily applied based on metal response. For metals, low occurrences of high loads

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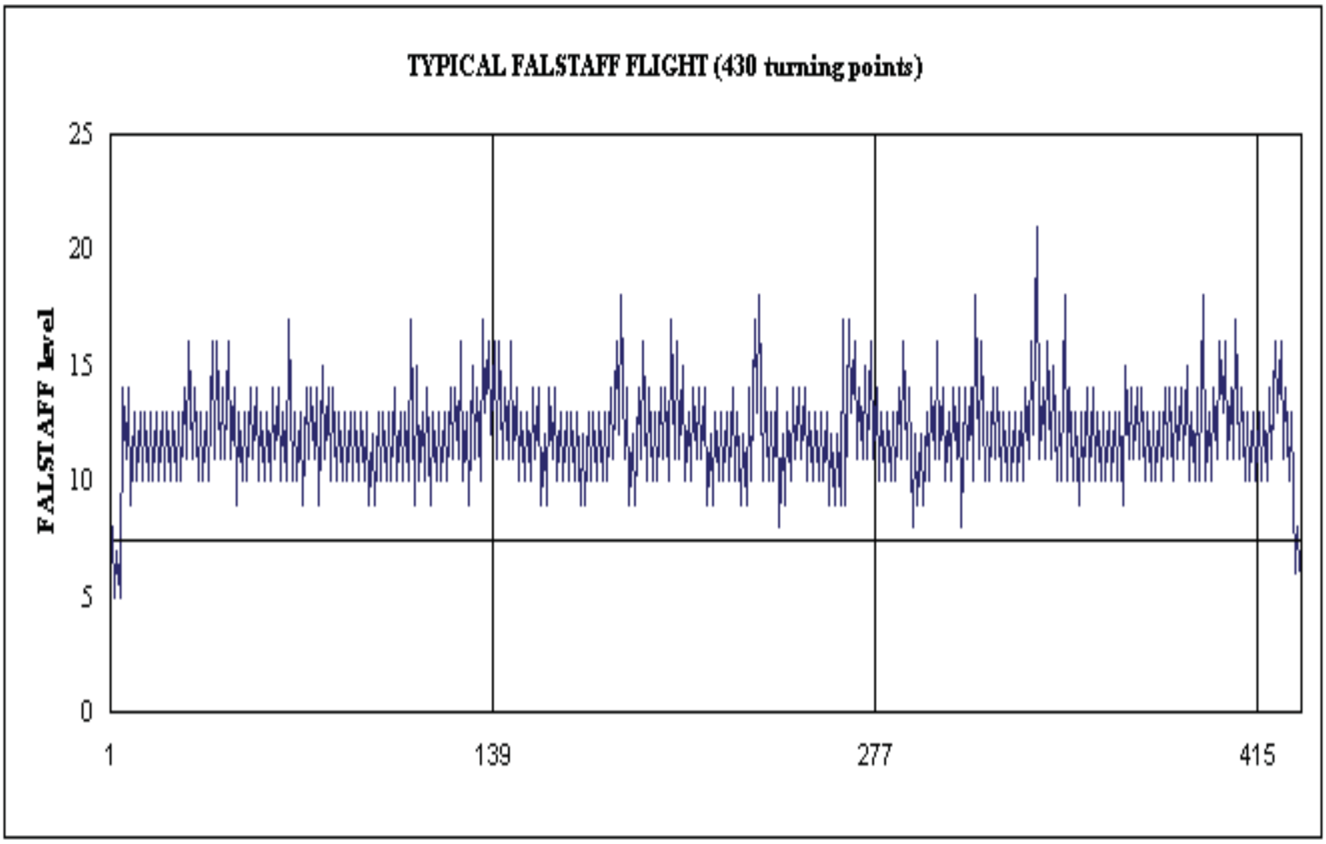
The Falstaff loading spectrum

Part of the Falstaff sequence



(The Falstaff spectrum block consists of 200 flight blocks of loading, each with take-off and landing sequences. These flight blocks have been generated to represent three different training mission severities, i.e. Air-to-Ground Combat, Air-to-Air Combat and Navigation missions.)

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can be beneficial by creating plastic zones at crack tips, reducing crack tip stress concentration and increasing fatigue life. We can not be confident that these high load occurrences will occur at convenient regular intervals in service and so it is unwise to include them in fatigue test spectra applied to metals. *(The highest value in the Falstaff spectrum is taken as the once-per-hundred flight value.)*

Sequence effect: Most of the original load sequence information is lost and an idealised sequence of cyclic loads is arbitrarily regenerated in block form or pseudo-random form to represent individual flight blocks within the main spectrum block of loading. Note, simple cumulative damage theories take no account of load sequence. *(The Falstaff spectrum block consists of 200 flight blocks of loading, each with take-off and landing sequences. These flight blocks have been generated to represent three different training mission severities, i.e. Air-to-Ground Combat, Air-to-Air Combat and Navigation missions.)*

Loading wave-form shape and rate effect: Use of counting methods to generate equivalent constant amplitude data loses all information of load wave-form shape and rate. Usually arbitrary sine-wave loading and convenient load frequencies are used. Note, simple cumulative damage theories take no account of wave-form shape or load rate.

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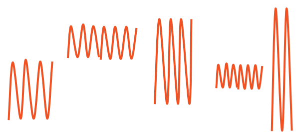
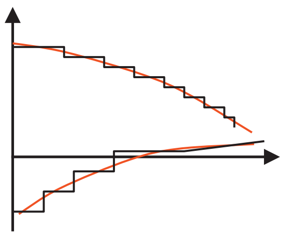
Summary of load history idealisation

Service loading history

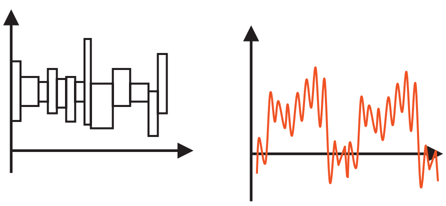
↓ *counting method*



Cumulative occurrence spectrum

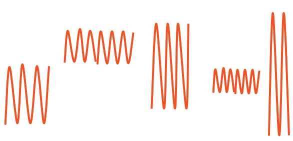


↓ *arbitrary sequence regeneration*



Block or flight by flight test spectrum

↓ *counting method*



Cumulative damage analysis (CDA)

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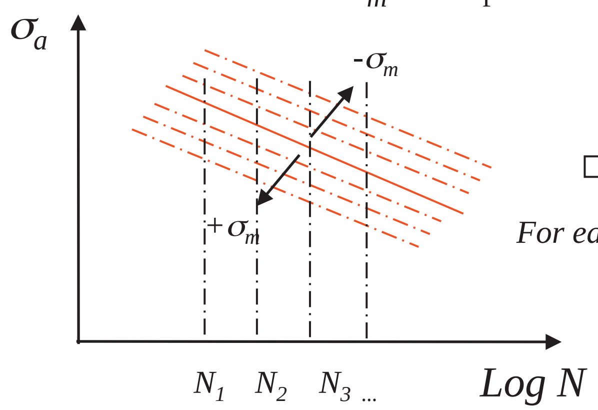
Data Manipulation

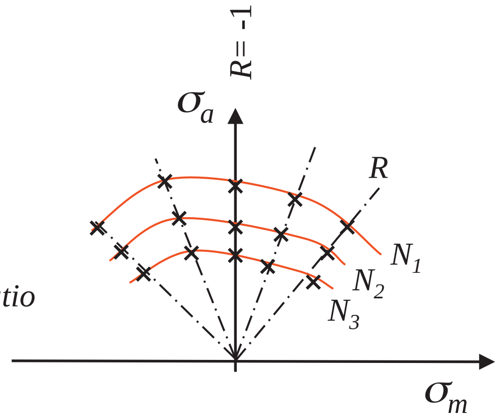
S-N fatigue data limited, typically 3 or 4 R-ratios & 3 or 4 severities. For CDA, manipulate (interpolate & extrapolate) data to any σ_m / σ_a defined in loading spectrum

Empirical Constant Life Diagrams

Effect of σ_m on N_f



For each R ratio

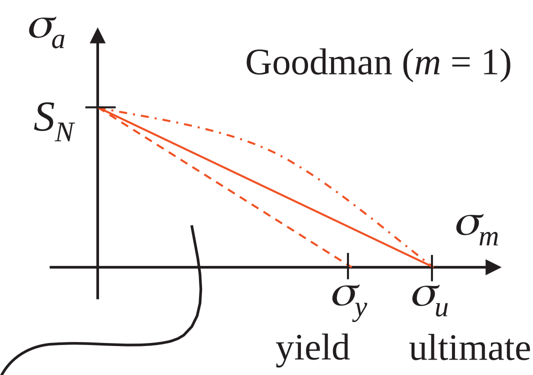


CLD determined from S-N curves at different constant σ_m, σ_a or R-ratios.

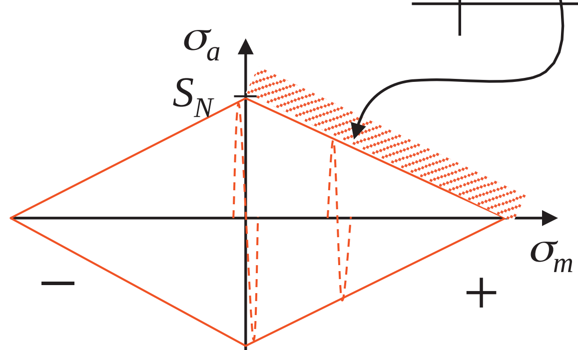
Constant life models:

$$\sigma_a = S_N\,[1 - (\sigma_m / \sigma_u)^m]$$

$$S_N = \sigma_a / [1 - (\sigma_m / \sigma_u)^m]$$



Alternatively, use empirical constant life models e.g. Goodman or Soderberg, etc. which relate to R=-1 S-N data



CLD extrapolate limited R-ratio / severity S-N data over complete load spectrum

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S-N fatigue data is usually limited to a relatively small base consisting of typically three or four R-ratios (i.e. combinations of mean and alternating stress) and three or four severities. For cumulative damage analysis it is necessary to manipulate (interpolate and extrapolate) from this data to any mean / alternating stress defined in the loading spectrum.

Usually empirical "**Constant Life Diagrams**" are used which consist of curves defining the combinations of mean and alternating stresses which cause failure at a particular life. I.e. at a particular number of cycles to failure.

These curves can be determined experimentally from S-N curves at different constant σ_m, σ_a or R-ratio values. Alternatively empirical constant life models can be used, such as Goodman or Soderberg, etc.. Essentially constant life diagrams provide a means of extrapolating limited R-ratio / severity S-N data over a complete load spectrum.

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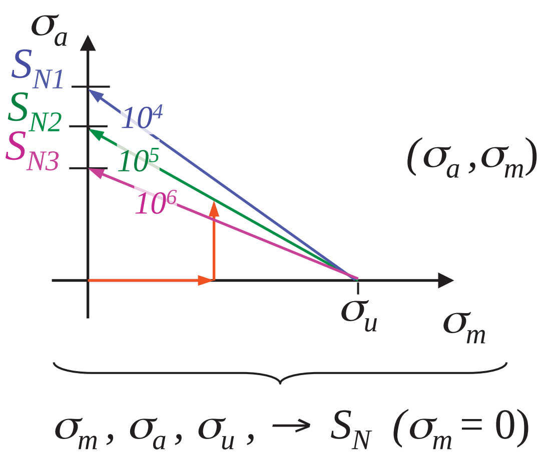
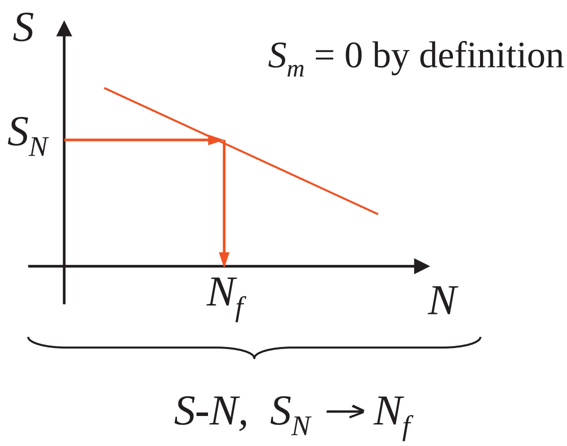
Use of Goodman Constant Life Diagrams

Calculate **Life;**

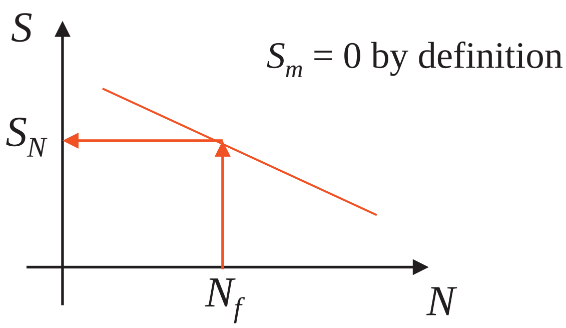
E.g. N_f **for given** σ_m, σ_a, σ_u, *S-N*

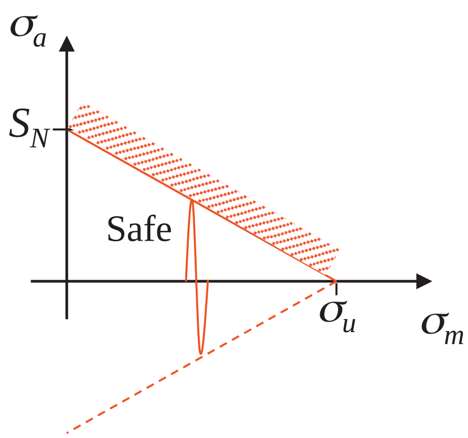




Calculate **allowable combinations of mean and alternating stress**

E.g. σ_m, σ_a **for given** N_f, σ_u, *S-N*





Note: Goodman assumes damage from load cycle with non-zero mean equivalent to damage from R=-1 load cycle which requires the same number of cycles to failure!

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By manipulation of load cycle data Goodman diagrams allow simple calculation of either:

life - for a given cyclic stress, S-N data and static strength

Or

allowable combinations of mean and alternating stress - for a given life and S-N data and static strength.

Note, the application of Goodman assumes that the damage caused by a load cycle with non-zero mean is **equivalent** in form and effect to the **damage** caused by a fully reversed load cycle with zero mean which requires the same number of cycles to failure!

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Cumulative Damage Analysis

Cumulative Damage Rule

Palmgren-Miner (1930's)

Linear Cumulative Damage (LCD) Rule

$\mathbf{D_i \propto n_i/N_i \quad @ \; S_i}$ "fatigue cycle fraction rule"

Damage $\propto$ n/N where n = no. applied cycles at the equivalent endurance strength, i.e. R=-1, and N = no. constant amplitude cycles to cause failure at that endurance stress.

For spectrum loading:

$\mathbf{\Sigma \; n_i/N_i = 1 \quad @ \text{ failure}}$

i.e.:

$$\mathbf{n_1/N_1 + n_2/N_2 + n_3/N_3 + \ldots\ldots\ldots = 1 \quad @ \text{ failure}}$$

$$@ \quad \underbrace{\mathbf{S_1 \qquad S_2 \qquad S_3}}_{(\sigma_m = 0)}$$

Comparison of LCD with measured spectrum fatigue lives often shows failure at n/N $\neq$ 1. Indicates favourable or unfavourable interaction effects and with experience summation value for failure can be empirically adjusted.

Assumptions of LCD rule:

Damage accumulated is:

- independent of current damage state

  i.e. given cycle of loading assumed to have same damage effect throughout fatigue life.

- independent of loading sequence

Also, load history idealisation assumptions and data manipulation (Goodman assumptions)

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Palmgren and Miner independently produced a simple **Linear Cumulative Damage (LCD) Rule** to calculate the proportion of fatigue life consumed for a loading history idealised as sets of constant amplitude cycles and material with fatigue performance defined by S-N curves.

The **damage is assumed proportional to n/N** where n is the number of applied cycles at the equivalent endurance strength, i.e. with zero mean, and N is the number of constant amplitude cycles to cause failure at that endurance stress.

For a spectrum of loading, idealised as sets of constant amplitude cycles, the damage ratios of each set of cycles is summed and failure may be assumed to occur when the sum adds up to 1.

Comparison of LCD results with measured spectrum fatigue lives often shows that failure occurs at summations higher or lower than 1. This indicates **favourable or unfavourable interaction effects** respectively and with experience the summation value for failure can be empirically adjusted.

Note that the summation is usually carried out for fully reversed cycles so that **equivalent zero mean cyclic stresses** must be deduced by Goodman for any spectrum cycles with non zero means.

The linear cumulative damage rule implies the following assumptions:

- The damage accumulated is assumed to be **independent of the current damage state**. I.e., a given cycle of loading is assumed to have the same damage effect throughout the fatigue life.
- Damage accumulation is assumed to be **independent of the order of loading**.

And of course any assumptions related to definition of input data, (e.g. Goodman) and load history idealisation (e.g. standardised load spectra) will be inherent in the analysis.

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Summary of classical fatigue analysis

Complex v/a load cycle history

↓

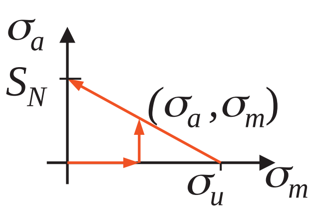
Sets of c/a load cycles

↓

Analysis

n_i, (σ_m, σ_a)

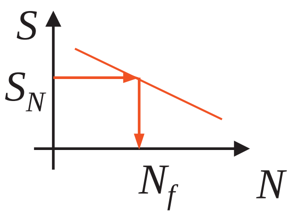
Goodman



↓

S-*N*

Programmed blocks



↓

N_{fi}

Equivalent zero mean cycles

↓

LCD

test

$D = \sum n_i/N_{fi} = 1$ @ failure

↓

Predicted Life

Test

Spectrum simulation

e.g. blocked

or

Flight by flight

↓

Fatigue

↓

Test life

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Data Manipulation Summary

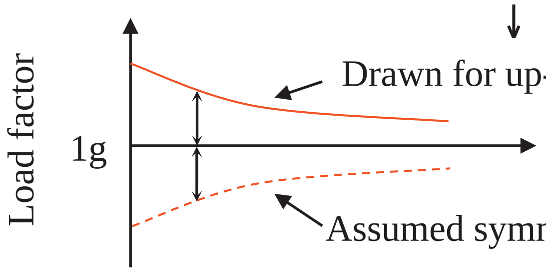
Breakdown of loading from Load Spectra

Complex v/a load cycle history

↓

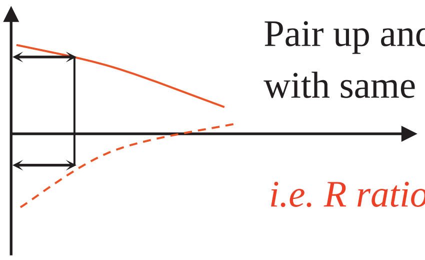
Sets of c/a load cycles

↓



i.e. R=-1 zero mean loading

Cumulative frequency of exceedings



Pair up and down peaks
with same frequency of occurrence

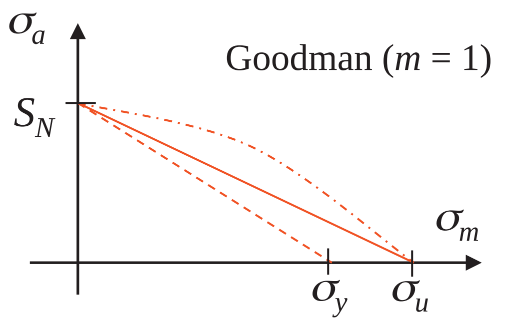
i.e. R ratios with non-zero mean

Use constant life model e.g. Goodman, to convert ALL σ_m and σ_a derived from load spectrum above to equivalent S_N at R=-1

$$\sigma_a = S_N[1 - (\sigma_m / \sigma_u)^m]$$

$$S_N = \sigma_a / [1 - (\sigma_m / \sigma_u)^m]$$

⇨



Apply Linear Cumulative Damage (LCD) Rule:

$$\Sigma \;\; \mathbf{n_i/N_i = 1 \;\; @ \; failure}$$

i.e.: $\mathbf{n_1/N_1 + n_2/N_2 + n_3/N_3 + \ldots\ldots = 1}$ **@ failure** (where σ_m = 0, i.e. R=-1)

@ $\quad S_1 \qquad S_2 \qquad S_3$

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Fatigue Design of Metal Aircraft Structures

Contents

- Design philosophies
- Materials
- Manufacture and Assembly
- Joint configurations
- Details
- Stress levels
- Environment
- Testing

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Design Philosophies

Safe Life

Considered in this course

Use **Safe Life design** to demonstrate resistance to fatigue damage initiation (to macroscopic crack stage)

Provide **Safe Life** interval equal to projected lifetime of aircraft.

i.e. practical crack-free service life: "Safe life interval"

Fail Safe

Use **Fail Safe design** to demonstrate resistance to fatigue crack growth (to critical length stage)

Provide **Fail Safe** interval to allow certainty of inspections to detect crack before propagation to critical length.

i.e. Slow/restricted crack growth
for maximised inspection intervals: "Fail safe interval"

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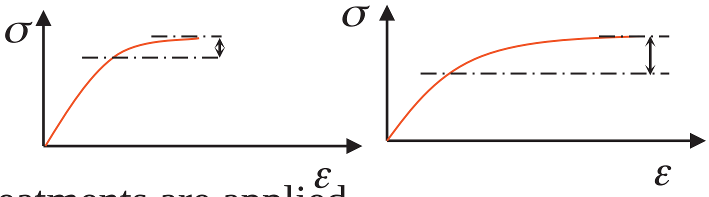
Materials

•Choose material for good fatigue resistance to crack initiation and good resistance to crack propagation

E.g. 2024-T3 (clad & anodised for erosion protection) for fuselage & lower wing skin (predominantly tensile loading

•Avoid Aluminium alloys with high yield to ultimate strength ratios to ensure large critical crack lengths (*plastic safety margin!)*

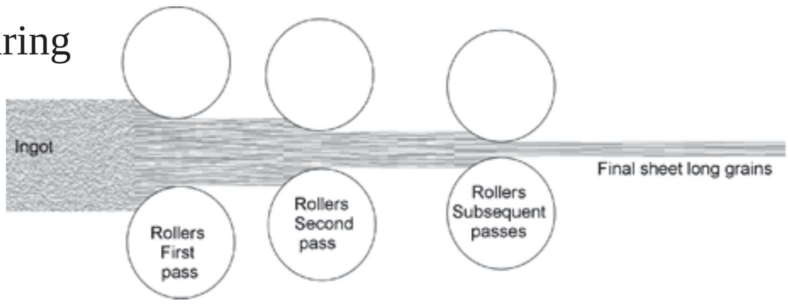


•Ensure proper heat treatments are applied

E.g. T6 c/w T3

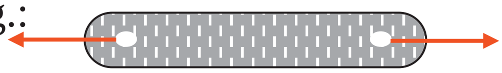
•Check appropriate material substitutions are carried out

•Account for manufacturing method, e.g. forging, extrusion, sheet etc.,

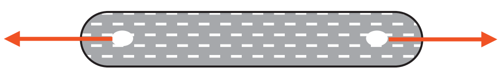


E.g. Check appropriate grain directions are used - sheet

E.g.:



Load normal to grain direction ×



Load parallel to grain direction ✓

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Manufacture & Assembly

Ensure proper cleaning of surface treatments

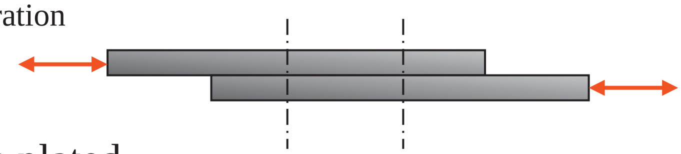
e.g. complete removal of etchants

Use protective finishes

e.g. corrosion protection, fretting protection

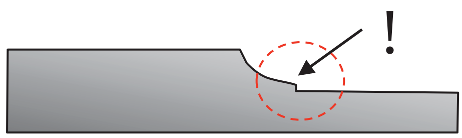
Minimise wear of fretting surfaces due to relative sliding

by protective finishes, lubrication, or separation



Shot peen surfaces to be chrome plated

to reduce crack sensitivity from hardened surface

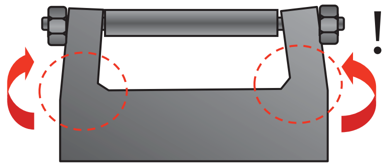


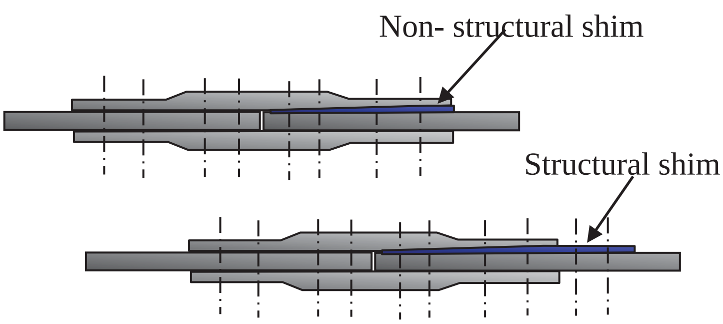
Check mechanical surface finish, e.g. surface roughness,

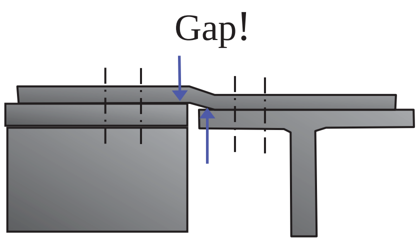
wrt process, e.g. chemical milling, plating, rolling, shot peening

Check fit-up and use of shims

use shims to avoid excessive clamping







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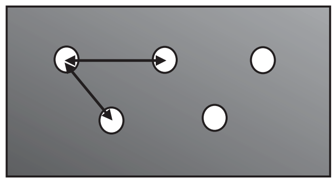
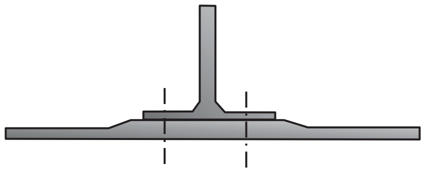
Joint configurations

Use adequate hole/edge spacings
to reduce stress conc'n interactions

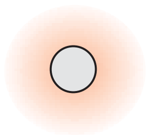


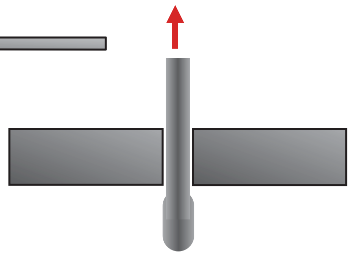
Limit countersink depth
increase edge distance to allow for csks

Use thickening/ pad-up
for reduced stress levels



Use cold worked holes
for beneficial compressive hole edge stress



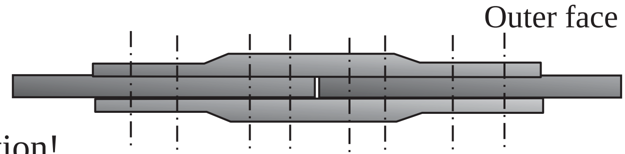


Ensure correct fastener selection and installation

Consider fastener flexibility

Use interference fit
for significant fatigue life improvement

Avoid blind areas
preferential failure on outer layers – inspection!



Avoid eccentricity

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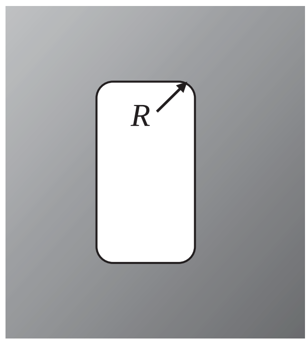
Details

Identify critical areas

Attachment points, joints geometric details

Use large cut-out radii

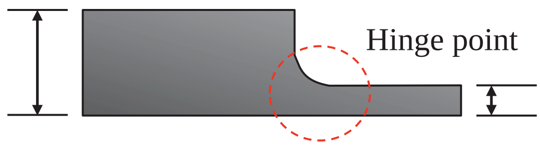
to reduce stress concentrations



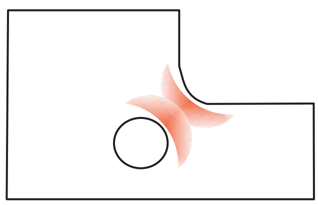
Use large corner/ bend radii

to avoid cracking along folds, e.g. min bend radius = 2t

Avoid eccentricities and sudden discontinuities of sections



Avoid interacting stress concentrations



Use "fatigue quality index"

- to allow for manufacturing damage, holes, fastener installation, repairs & scratches.
- a proprietarty index that captures relevant previous experience etc. for all variables in design

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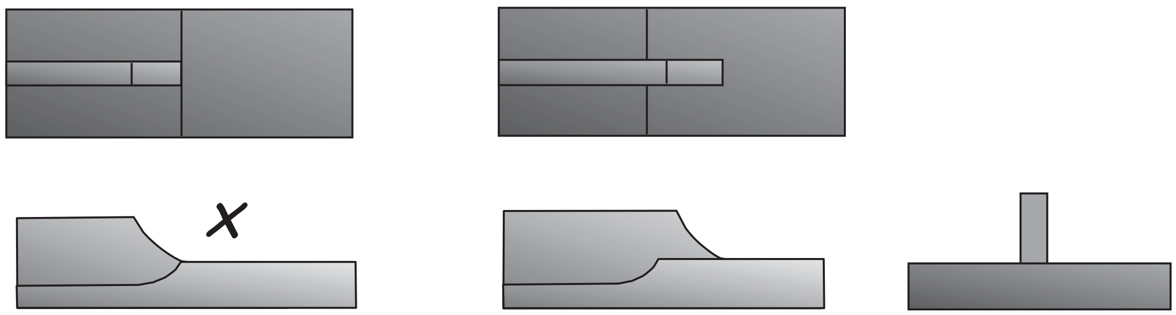
Details

Avoid change in sections at same location e.g. stringer run-out



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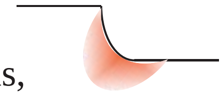
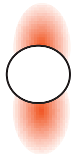
Stress levels

- Use detailed level of stress analysis

  detailed enough to identify stress gradients and concentrations,

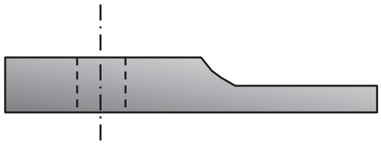
  e.g. @ holes, cut-outs, notches, etc.





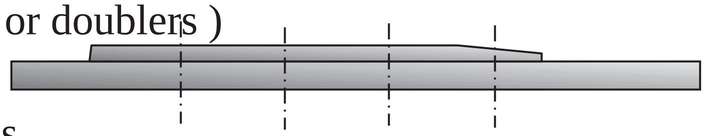
- Limit maximum design ultimate allowable values

  e.g. 310-380 N/mm^2 for Aluminium transport a/c structures



- Use local reinforcements and "beef-up"

  to reduce stresses (taper straps or doublers)



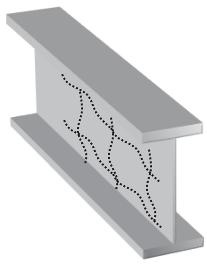
- Reduce initial buckling of webs

  e.g. use design for ‘no buckling’ up to 1.0g steady flight for civil a/c

  up to 1.5g steady flight military a/c

  use design ratio of $\tau_{crit}/\tau_{ult} > 20\%$

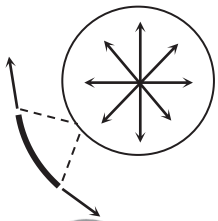
  use limit minimum t/b panel sizing for lightly loaded webs (avoid in-plane shear stress)

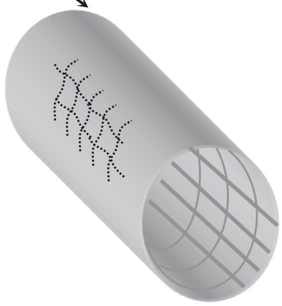


- Limit fuselage gross area hoop tensile stress

  to allow for stored energy of pressurised fuselage and possible catastrophic failure (balloon burst – fast fracture!)

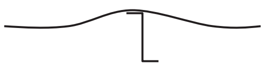
  e.g. limit stress to 80-100 N/mm^2 at design operating pressure level





- Design to allow for fuselage panel quilting or pillowing (unpressurised state)

  by considering resulting bending stresses across stiffener rivet lines



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Stress levels

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Environment

Determine actual aircraft experience
"Operational Loads Measurement" OLM current trend

Increase structural damping of panels subjected to acoustic excitation to prevent sonic fatigue.

PA-30 Twin Comanche Tail Flutter Test

Protect against corrosive conditions

How to break a Glider Wing

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Testing

Carry out development tests

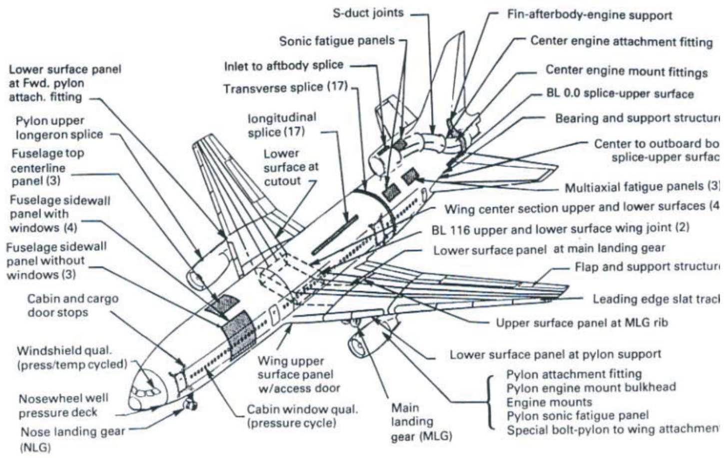
to assess fatigue initiation performance and generate crack growth data

Determine fatigue variability (scatter)

as a product derived from consideration of test sample size, environmental effects...

Perform a/c component tests

full scale flight by flight tests to locate critical areas



Components structural fatigue tests — L-1011.
(Courtesy of Lockheed Aeronautical Systems Co.)

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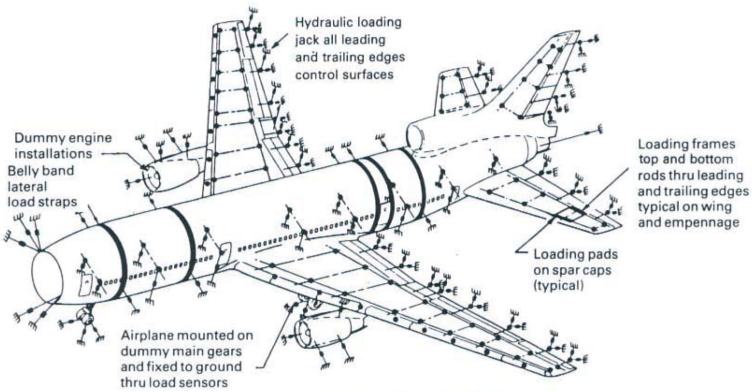

(a) Illustration of fatigue test set-up

Complete airframe fatigue tests (indoor) — L-1011.

B787 Fatigue Testing

B787 Wing Testing 2

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