EASA
European Aviation Safety Agency

Damage Tolerance for Antenna installations

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Introduction

- The intent of this presentation is to discuss recurring issues for antenna DT certification like:
- Accepted methods and tools
- Conservatism of assumptions and simplifications discussion, considering data available and its reliability.
- Installation and inspection instructions
- DOA and CVE qualification

Contents

Criticality of damage tolerance

- An antenna installation increases baseline fatigue stresses on the fuselage skin. Therefore an assessment is necessary to determine that the installation won’t lead to catastrophic failure due to fatigue.

- The baseline aeroplane design may be intrinsically robust in damage tolerance, however this can’t be assured for all aeroplanes, all fuselage areas, and all antenna installation designs.

What we want to prevent

- Crack extending catastrophically, for structural or physiological reasons (decompression)

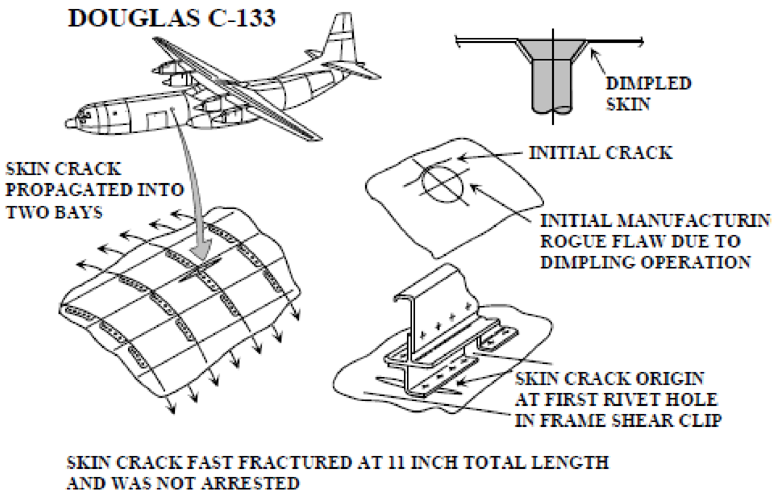


Figure B8: Details of the Cracking that Led to Failure of C-133 Transport Aircraft

Crack on unmodified skin (old design).

From "Threats to aircraft structural safety", Aeronautical Systems Center

Current regulation

- CS-25.571 Amendment 15:
  - (a) *General.* An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage, will be avoided throughout the operational life of the aeroplane.
  - …the identification of principal structural elements and detail design points, the failure of which could cause catastrophic failure of the aeroplane
  - …inspections or other procedures must be established as necessary to prevent catastrophic failure, and must be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness required by CS 25.1529.

Current regulation

- Part 26 (ageing aircraft) compliance for FAA validations.
- 23.571, for metallic pressurized fuselage, allows choosing between fatigue, fail-safe or damage tolerance analysis (see 23.573(b)) .
- DT required in 23.573(a) for composite structure and 23.574 for metallic structure, commuter category, unless impractical

Current regulation: Establishing thresholds

- The structure can be considered as safe-life in case it is not prone to accidental or corrosion damage and either the calculated fatigue threshold is more than 1.25 DSG, (to allow for life extensions), or a replacement time can be established.
- AMC to 25.571 2.1.1 (b) manufacturing errors and AD
- For FAA validations 25.571 and AC 25.571-1d paragraph 6.j. has to be considered: Only SL if the structure is easily inspectable and fail-safe, and the resulting damage initiation is beyond the expected service life.
- This will be harmonised in CS-25.571

Current regulation. Establishing thresholds

- For fatigue analysis, the same as for crack growth analysis, the effect of loaded fasteners should be considered (ref. FAA-AIR-90-01 Repairs to Damage Tolerant airplanes, conservative)

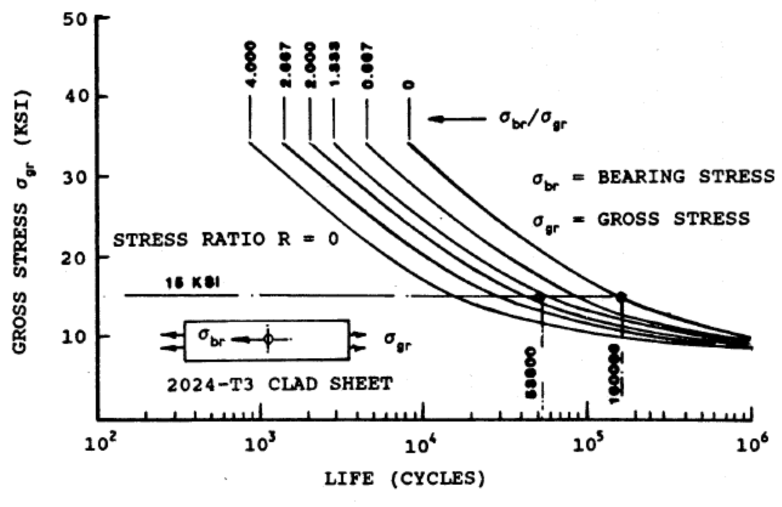
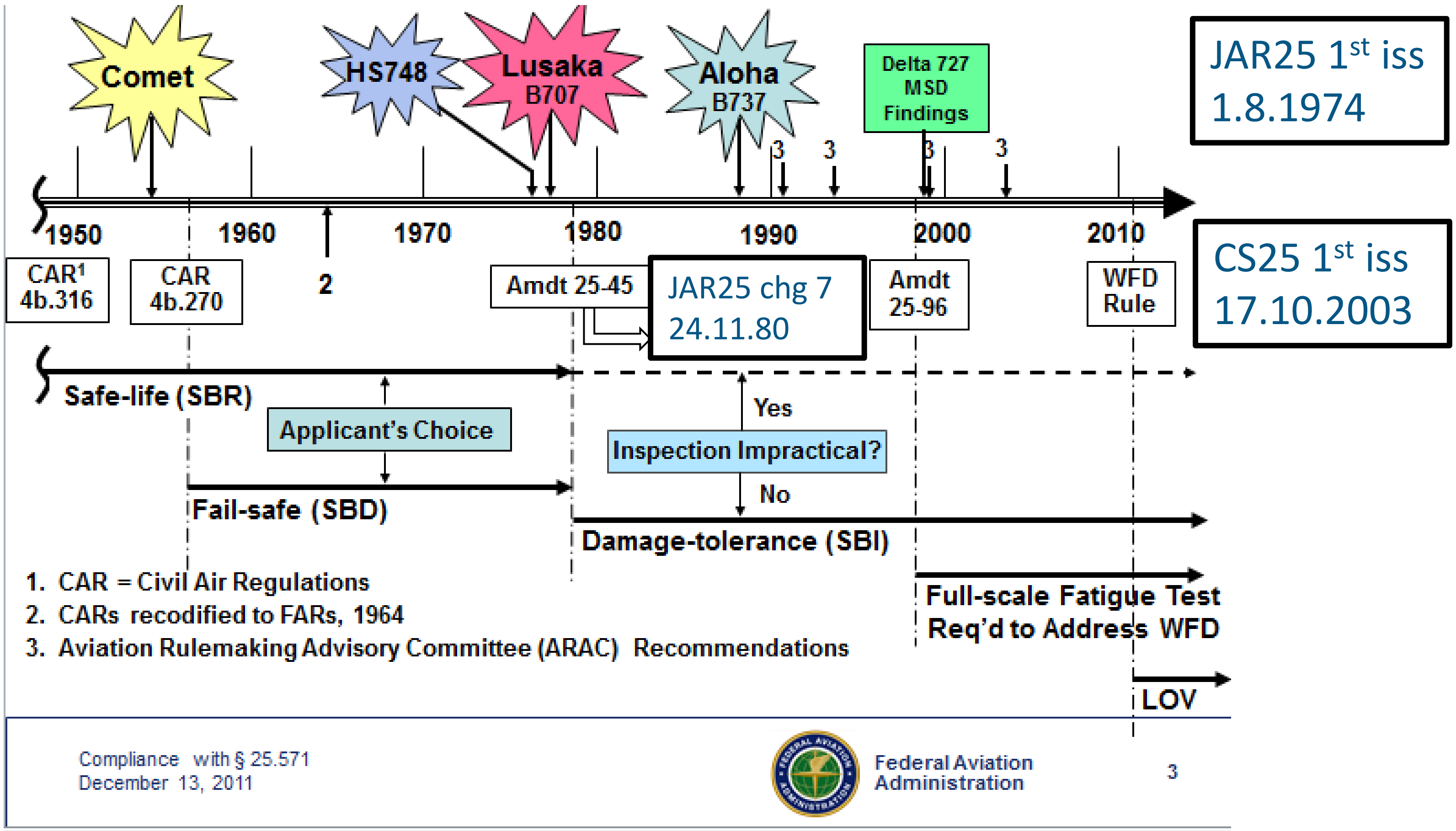


Fig. 2. Open Hole Sn Data 2024-T3

Regulation history. CS-25/JAR and FAR



Cert basis main milestones and other considerations for DT and inspections

- Amdt 45 (10.01.78; JAR Chg. 7 Eff. 24.11.80)
  - Inspections and limits must be included in the Maintenance Manual
- Amdt 54 (14.10.80; JAR Chg 10 Eff. 19.12.83)
  - inspections must be included in the **ALS of the ICA** required by Sec. 25.1529.
- Amdt 96 vs. part 26
- Another aspect to take into account is whether the baseline ALS already contains DT based inspections.

References used in the presentation

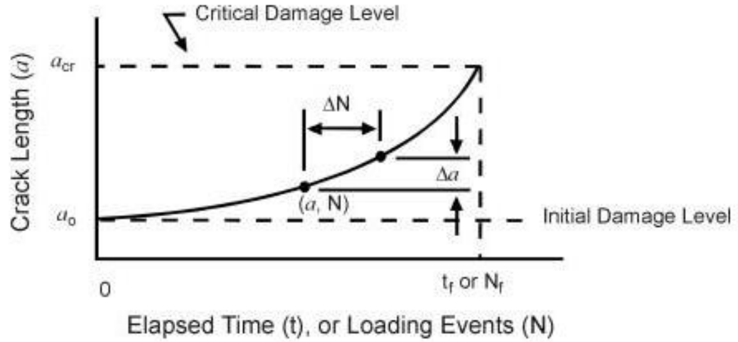
- Chicago ACO paper "*Damage Tolerance Analysis for antenna Installations on Pressurized Transport Airplanes*" R.Eastin (presentation) or J. McGarvey (paper)
- T. Swift FAA-AIR-90-01 " *Repairs to Damage Tolerant Aircraft*"
- Safarian "*DTA guidelines for antenna installations*"

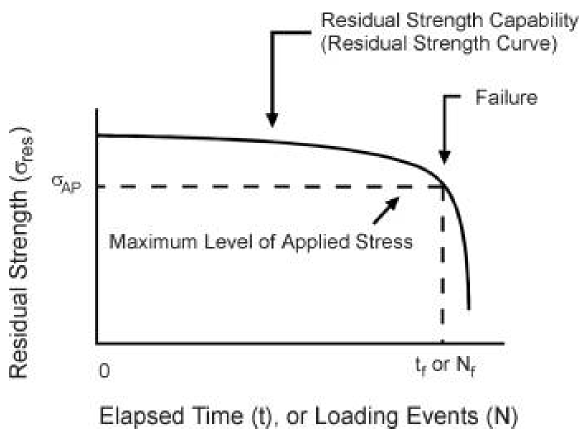
Figures used, and other useful references

- NASGRO *Reference Manual*
- M. Niu “*Airframe Structural Design*”
- AFGROW DTD Handbook online
- HSB *Handbuch Struktur Berechnung* ch. 60000
- FAA *DT Handbook* DOT/FAA/CT-93/69
- ESDU Series

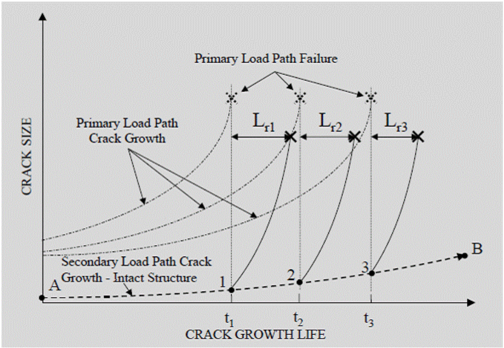
Crack growth principles

- Threshold: Initial to critical with a SF
- Interval: Detectable to critical with a SF





For small antennas, the reinforced fuselage works as Multiple Load Path



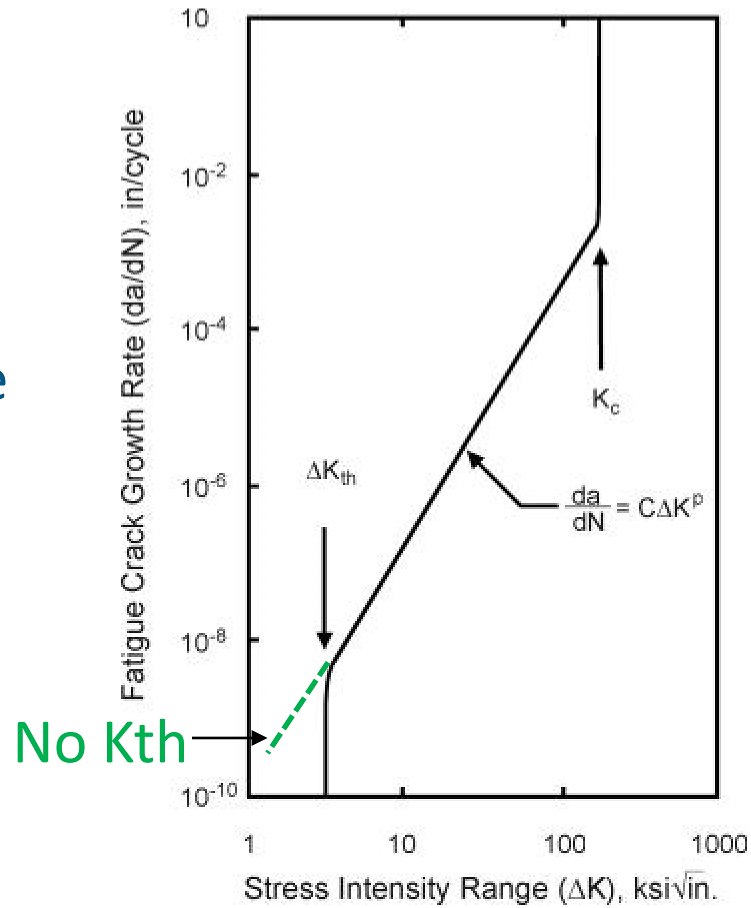
Crack growth principles: Linear elastic fracture mechanics with plastic corrections

The stress intensity factors for each geometry can be described using the general form:

$$K = \sigma\beta\sqrt{\pi a}$$

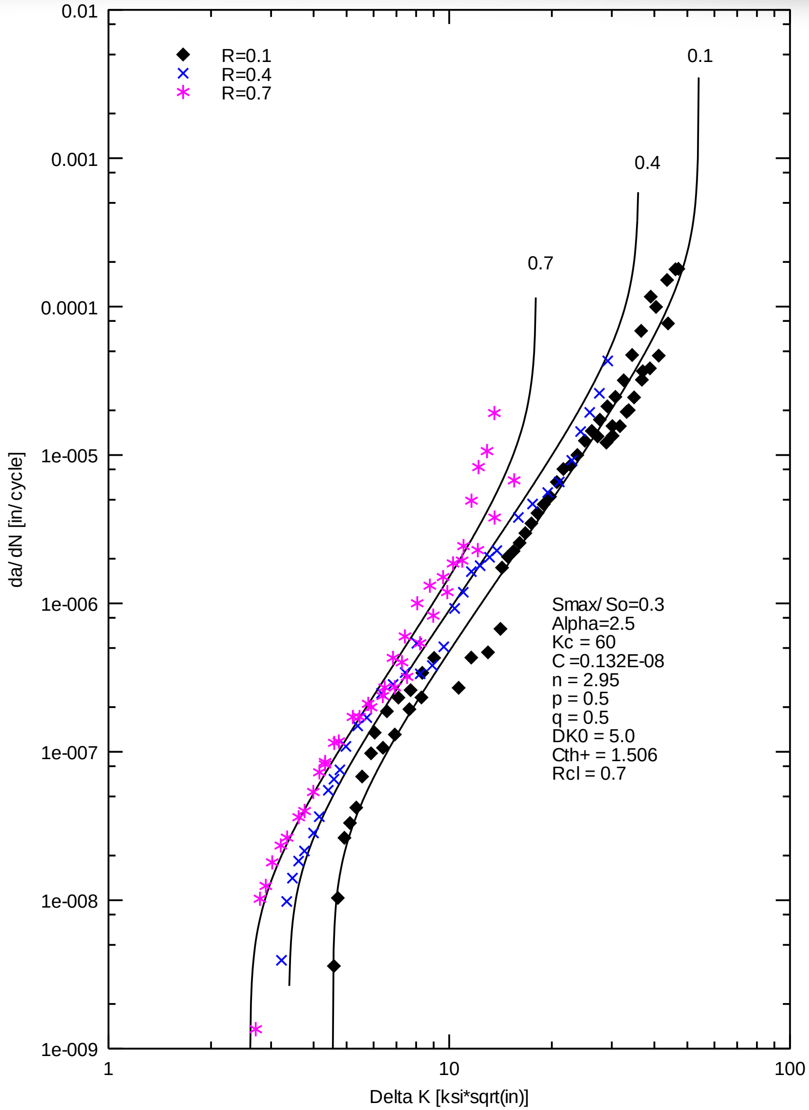
$$\frac{da}{dN} = C(\Delta K)^p$$

- P~3: Stress, β, high influence
- Conservative approach: no Kth to avoid issues like small crack (e.g. Forman model)
- Failure: Fracture toughness or net section yield.



Crack growth principles: Linear elastic fracture mechanics with plastic corrections

- R: Ratio of min to max stress
- Closure effects
- E.g, Walker model, NASGRO equation.



From NASGRO Reference manual

Antenna structure

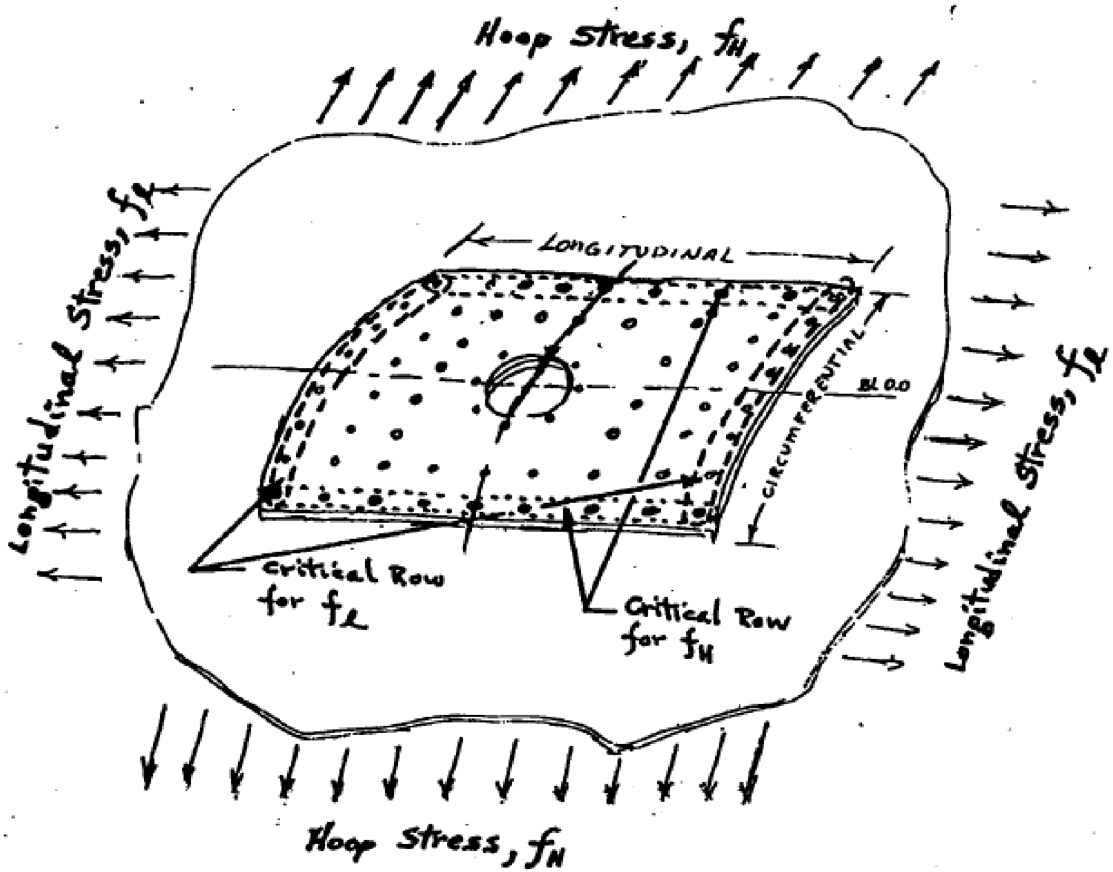


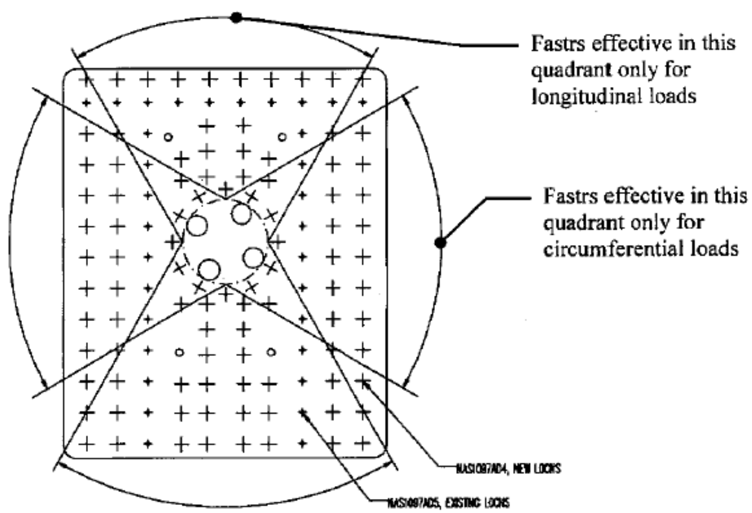
Figure 4 Critical Fastener Rows

Design choices

- Doubler on stiffeners or thickened skin: reduces disturbance to skin, however stiffeners/ frame may need checking, including increased stiffness consideration like e.g. use of intercostals
- Important aspects:
  - Doubler internal or external (detectability). Impact on inspection program.
  - Doubler/skin thickness ratio (static strength vs. stiffness change/fastener load)
  - Fastener pitch, edge distance, distance to radius, to minimise stress concentrations. Refer to SRM for good design practices.

Don’t forget static strength checks

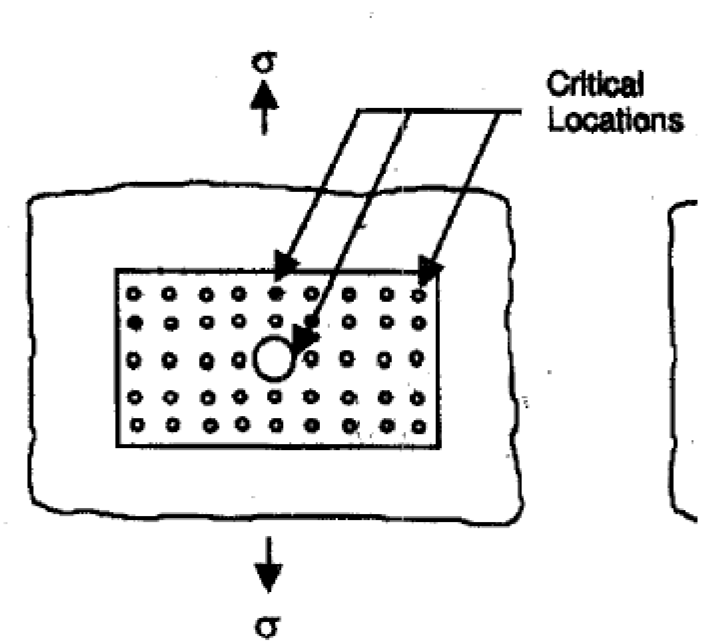
- Check area restitution and aerodynamic loads
- Check fasteners for shear
- Check bearing strength in skin and doubler



- Plastic redistributions: If enough fasteners / doubler one gauge thicker than basic fuselage skin
- Static strength isn’t usually the design driver

F&DT issues

- New fatigue critical details
  - Antenna or doubler attachments (fasteners) => filled loaded holes
  - Cable penetrations => open holes. Large hole has large influence on adjacent small hole. All elements, including all doublers, need evaluation



Ref. Peterson, ESDU, Niu:

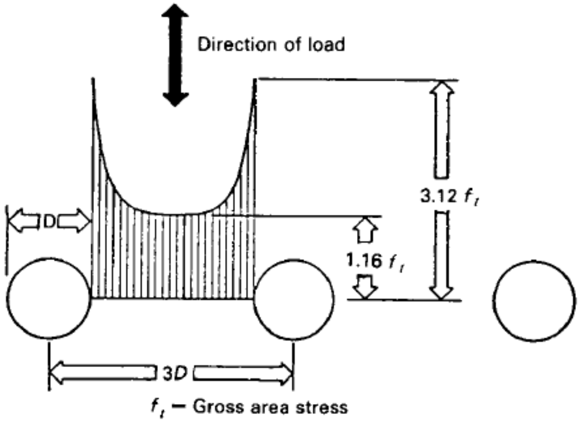
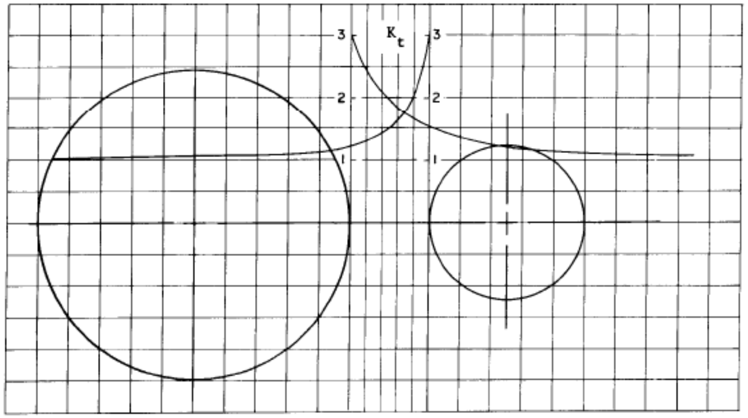


Fig. 7.7.6 Stress distribution between two holes at 3D-spacing.

- From Niu



- From ESDU 85045

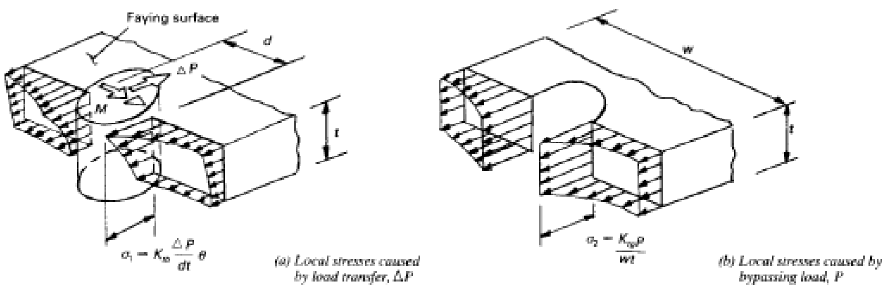
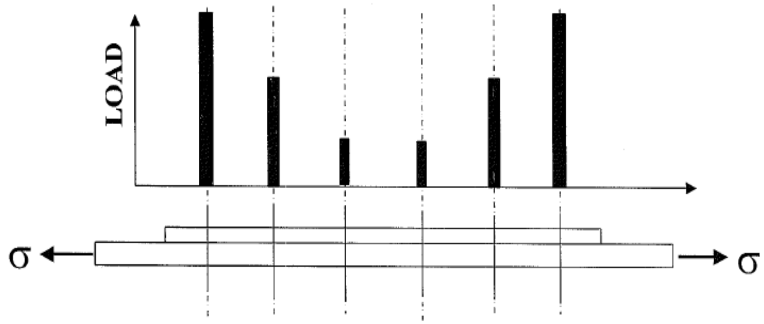


Fig. 7.7.24 Local peak stresses caused by load transfer and bypass load.

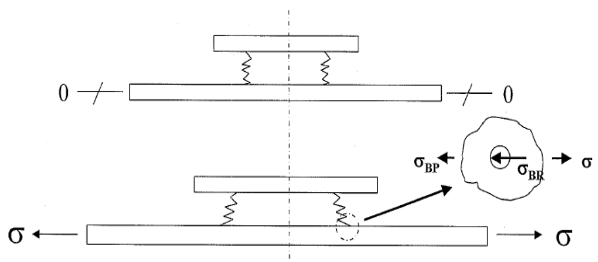
SRM repairs where doubler doesn't have a hole lead to smaller stress concentrations at the skin hole

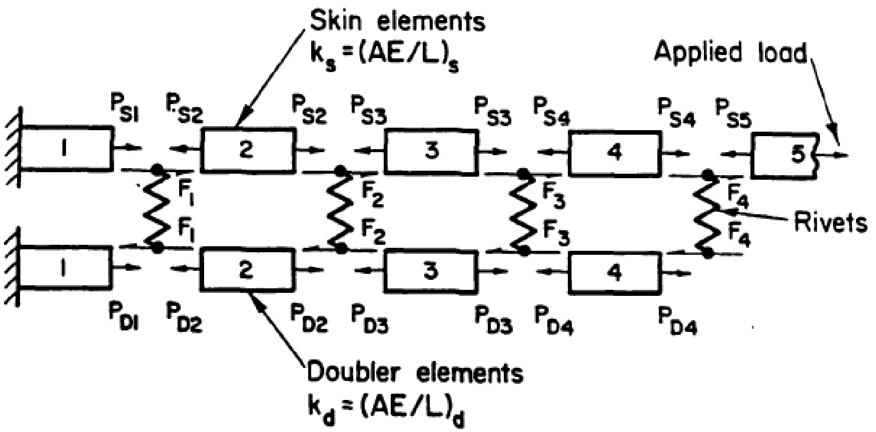
Fastener load

- First Row of Fasteners in Multi Row Design Will Have Most Critical Combination of Bearing and Bypass Stress



- Skin Holes Used for Doubler Attachment Will Get Induced Fastener Bearing Loading in Addition to Basic Stress





(After Swift)

FIGURE 4-1. SWIFT CDA MODEL

Giovanni from the ENAC will discuss conservative simplified approach for fastener load distribution

Crack growth analysis

- Example of crack propagation steps for a certain initial scenario scenario

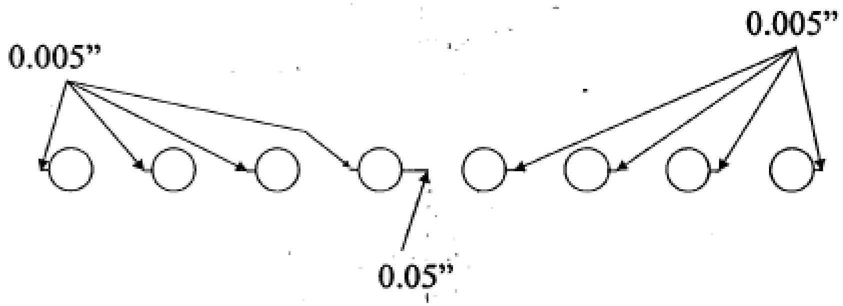


Figure 10 Initial Flaw Assumptions

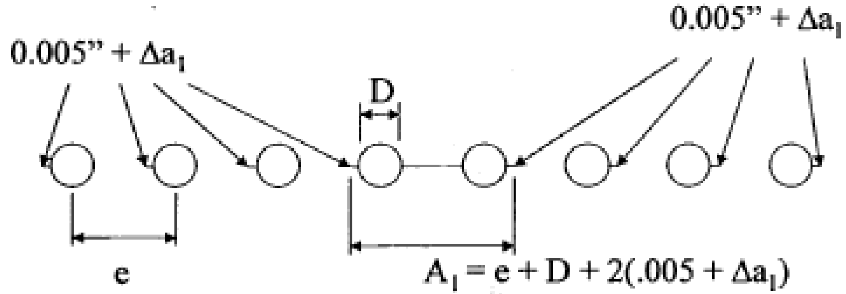


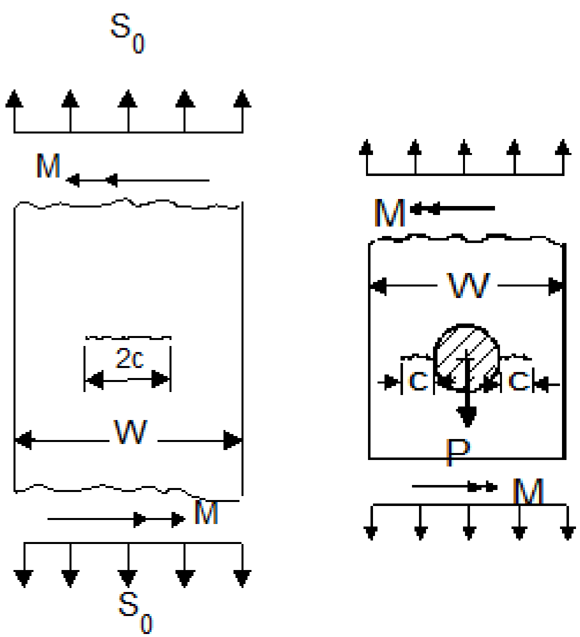
Figure 11 End of First Stage of Continuing Damage

Typical simplifications

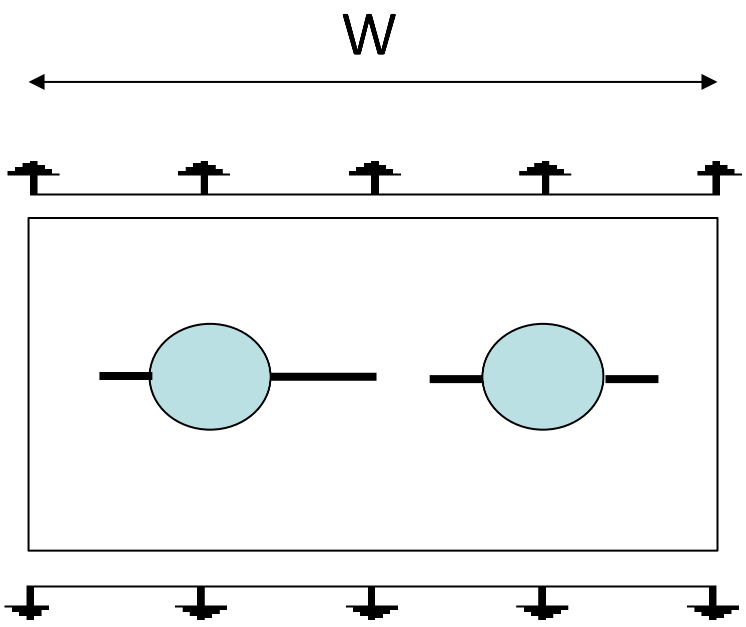
- In reality the skin is supported by stringers and frames.
- As the crack grows, the stress intensity increases if the tip close to a broken stiffener, but it decreases as it approaches an intact stiffener. However the load on the stiffener increases to a point where it could potentially fail.

Typical simplifications

- Therefore a common approach is to assume the skin is a strip of finite width



Simplest configuration



Two holes model

Typical simplifications

- It is a common conservative approach to assume the width of the strip as the distance to the next hole or fastener
- Otherwise all elements contributing to stress should be considered
- Additional analysis can be complex and doesn't usually bring much benefit in terms of extended inspections, because growth is much slower for small cracks.

Complex approach

- See backup slides for complex approach
- The following aspects need to be considered
  - Hole and crack interaction
  - Unequal load distribution among fasteners
  - Design details like fastener head, surface finish influence crack propagation
  - Bending due to doubler eccentricity (can be alleviated by good clamp-up, avoid blind rivets)
  - Fuselage bulging factor increases intensity for large cracks
  - Stiffener/frame overload

DTA analysis strategies

(1) Use of OEM data
(SRM data, and/or similarity with previous installation)

(2) FAA Chicago ACO method (by Bob Eastin)
(generic guidance)

(3) RAPID(C) (stress distribution + fatigue spectrum + crack growth)
(.) = for transport a/c ; (C) = for commuter a/c
Rapid runs under Windows 98, and support from the FAA is limited to commuter and smaller aeroplanes. It's use has been accepted in the past. Use to be evaluated on a case-by-case basis. (Ref. backup slides)

(4) FEM (stress distribution) + (simplified) fatigue spectrum + (e.g.) NASGRO / AFGROW (crack growth)

Note: Methods (2) and (3) use similar concepts

Potential issues with DTA strategies

(1) Use of OEM data :
(a) Data needs to be current and up-to-date
(b) Similarity needs to be substantiated
(c) Differences between prior installation/repair and (new) antenna installations need to be addressed (e.g. hole in doubler for connector non-existing for repair/SRM)

(2) FAA Chicago ACO:
(a) Typically leads to longitudinal stress higher than circumferential stress (contrary to expectation and RAPID(C))
(b) Seems to be conservative approach, but should not to applied to other installations (e.g. freighter conversion)
(c) Antenna hole not discussed

(4) NASGRO / AFGROW:
(a) Normal modelling concerns (program(s), mesh, elements, boundary conditions, loads, etc.)
(b) Choice of (far field, detail) stress from FEM to DTA needs to be understood
(c) Choice of options (crack growth model, etc.) needs to be understood

Analysis: Initial crack scenarios

- Rogue, primary crack: 0.05 “ corner
- Initial secondary crack: the USAF who first created the 0.05 primary and 0.005” corner secondary crack scenario have since 2008 required that a 0.01” corner flaw (plus damage growth until element failure) is used for continuing damage scenarios. (Ref. USAF Structures Bulletin No. EN-SB-08-002)
- Accepted scenarios:
  - A 0.05” rogue crack on critical hole location and 0.005” cracks on both sides of every hole
  - A 0.05” crack and 0.01” corner crack on one side of each other hole (or 0.005” through crack)

Analysis: Fuselage loads and spectrum

- Antenna installations typically located on the fuselage skin, bounded by frames and longerons, away from discontinuities like doors and windows
- For most of the fuselage the stress state is mainly biaxial loading (circumferential and longitudinal) due to pressure plus vertical inertia fuselage bending (longitudinal) only. Other loading could be reasonably neglected. Locations of the fuselage where primary loading has other components, should be avoided.

<u>Total stress:</u>

$f_{hoop} = \frac{\Delta p \cdot R}{t}$ [Pa] , circumferential stress

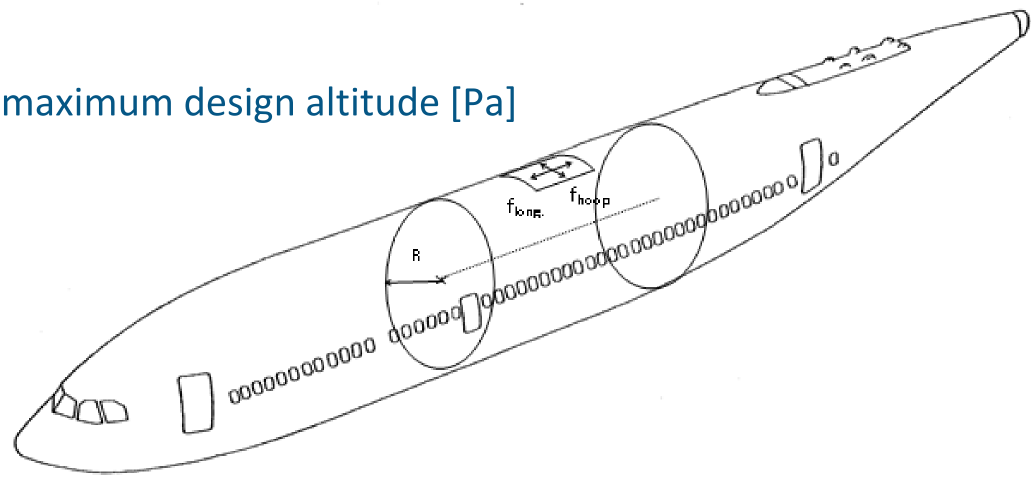
$f_{longitudinal} = \frac{\Delta p \cdot R}{2 \cdot t}$ [Pa] + bending stress , longitudinal stress

The longitudinal bending stress needs to be considered also for non-pressurized aircraft F&DT analysis

Δp = normal operating pressure differential at maximum design altitude [Pa]

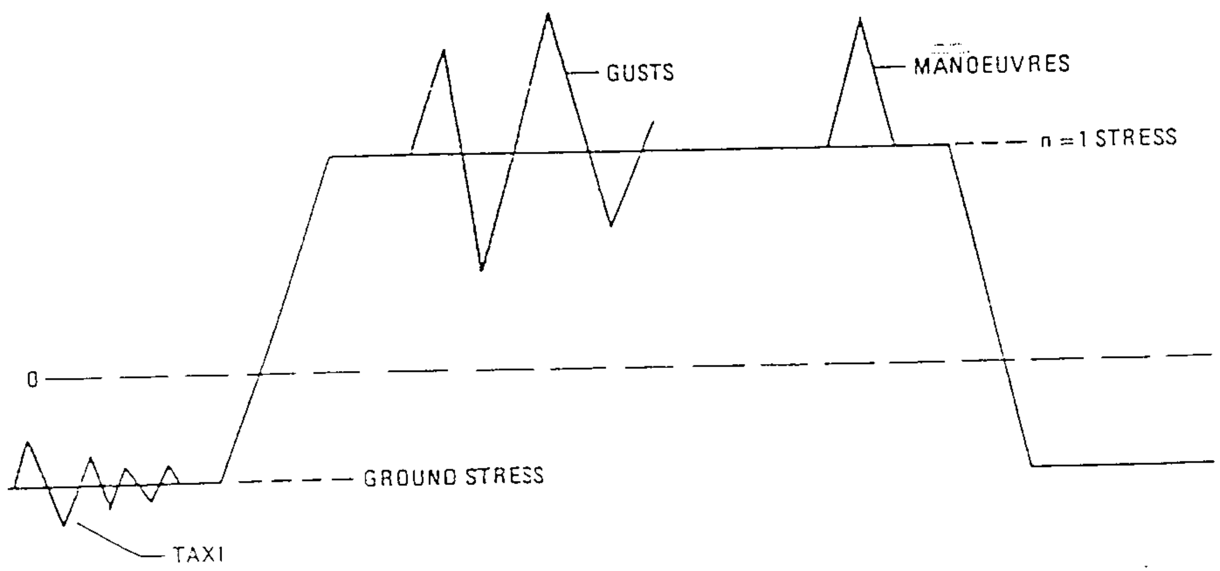
R = fuselage radius [m]

t = wall thickness [m]



Analysis: Spectrum I

- Typical fatigue profile



- Normally post-processed for simplified crack growth analysis

Analysis: Spectrum II

- Either a full rational analysis, or a simplified approach with inherent conservatism
- If an equivalent constant amplitude stress is calculated, take into account:
  - Rainflow count, if not full cycles.
  - Miner’s rule or fracture mechanics for equivalent constant amplitude stress.
  - Sequencing (Random and semi-random), truncation and retardation (often conservatively omitted).

Analysis: Spectrum III

- See Spectrum discussion at:
  - DOT-FAA-CT-93-79 Effects of Repair on Structural Integrity
  - DOT-VNTSC-FAA-91-16 Generation of spectra and Stress histories for F&DT analysis of fuselage repairs

- They include stress derivation from fuselage bending as a beam with some simplifications

Chicago ACO constant amp. Spectrum I

Longitudinal Gross Fatigue Stress:

The longitudinal skin stress typically results from

- The vertical fuselage inertia bending loads, assumed to vary along the x-axis of the A/C, and
- The differential pressure loads

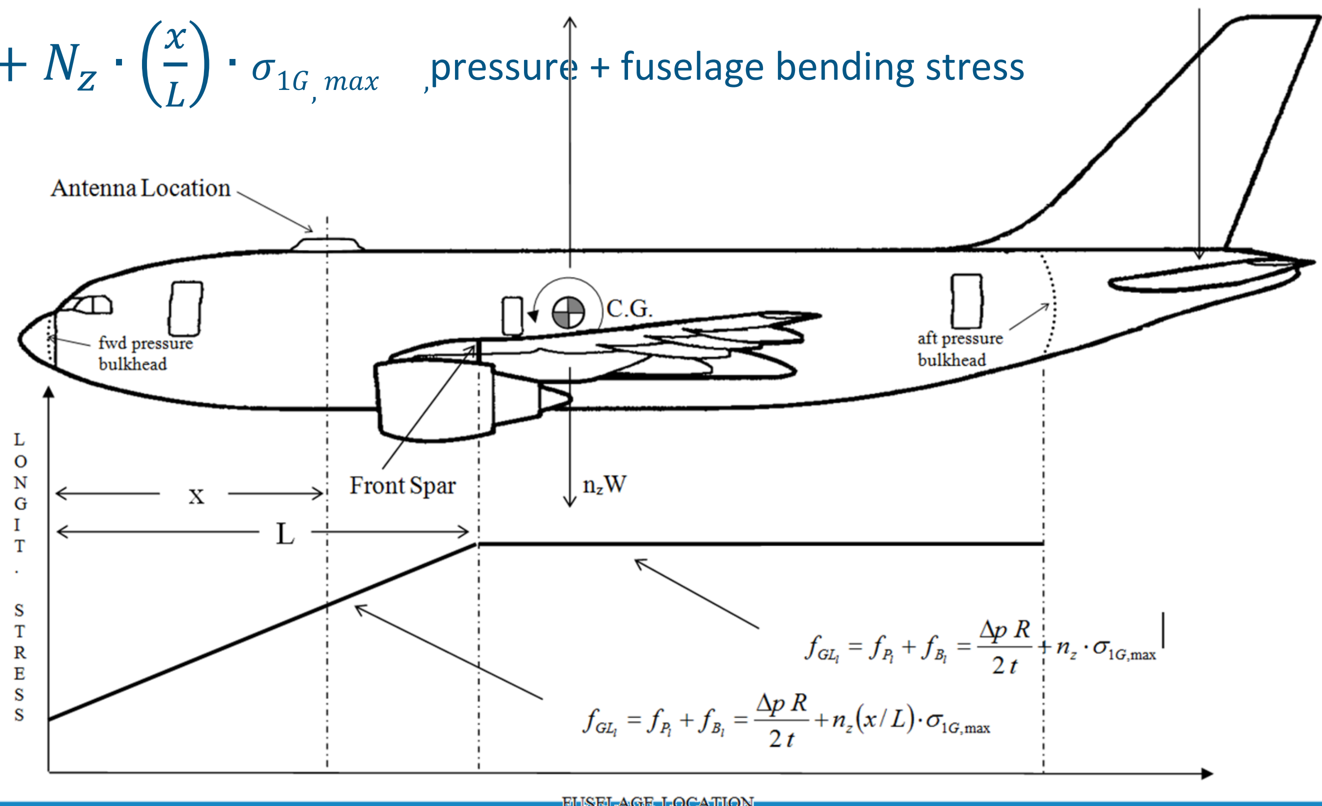
$$f_{GL_l} = f_{P_l} + f_{B_l} = \frac{\Delta p \cdot R}{2 \cdot t} + N_z \cdot \left(\frac{x}{L}\right) \cdot \sigma_{1G,\ max}$$, pressure + fuselage bending stress

Nz=Vertical load factor

f_{GL_l} = gross loading in longitudinal direction

f_{P_l} = pressure loading in longitudinal direction

f_{B_l} = bending in longitudinal direction



Chicago ACO constant amp. Spectrum II

In case $\sigma_{1G,max}$ from the TCH is not available, it can be estimated conservatively assuming that the skin has been sized with MS=0 for stress resulting from ultimate manoeuvre, combined with pressure.

$$f_{GL_l} = F_{tuB} = 1.5 \cdot \left(\frac{\Delta p \cdot R}{2 \cdot t} + N_z \cdot \sigma_{1G,max} \right) \quad [Pa]$$

$$\sigma_{1G,max} = \frac{\left(\frac{F_{tuB}}{1.5} - \frac{\Delta P \cdot R}{2 \cdot t} \right)}{N_z} \quad [Pa]$$

Δp = normal operating pressure differential at maximum design altitude, [Pa]

F_{tuB} = "B" basis ultimate tension allowable (ref. MMPDS-01), [Pa]

N_z = max. allowable positive limit g-load factor taken from the flight envelope (Ref. 25.337(b))

CRACK GROWTH STRESSES

- Longitudinal cracks,

  $f_{H,min} = 0$

  $f_{H,max} = \Delta PR/t$

- Circumferential cracks,

  $f_{l,min} = 0$

  $f_{l,max} = \Delta PR/2t + 1.3\sigma_{1G,max}$ (aft of front spar)

  $f_{l,max} = \Delta PR/2t + 1.3(L/S)\sigma_{1G,max}$ (forward of front spar)

ΔP = normal operating pressure at maximum design altitude plus .5psi (aerosuction)

RESIDUAL STRENGTH STRESSES

- Longitudinal cracks,

  $f_{H,RES} = 1.15\ (\Delta pR/t + 0.5R/t)$

- Circumferential cracks,

  $f_{l,RES} = \Delta PR/2t + N_z\sigma_{1G,MAX}$ (aft of front spar)

  $f_{l,RES} = \Delta PR/2t + N_z(L/S)\sigma_{1G,MAX}$ (forward of front spar)

Δp = normal operating pressure at maximum design altitude

ΔP = normal operating pressure at maximum design altitude plus 0.5psi for aerosuction

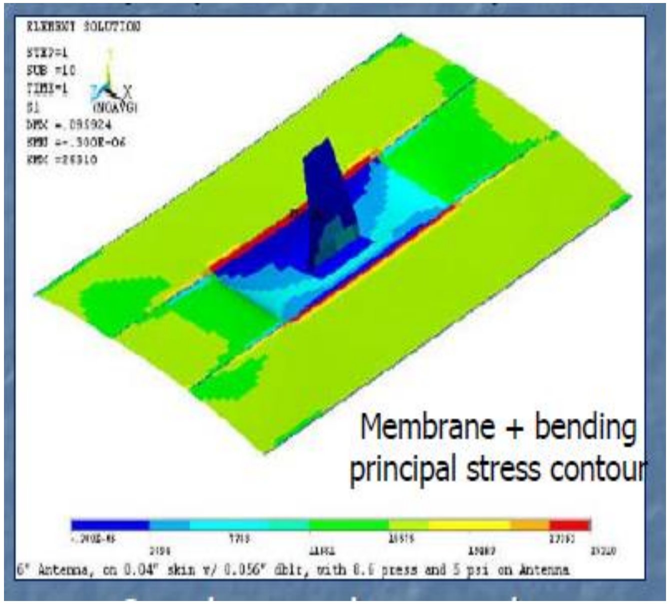
N_z = maximum design limit load factor (at least 2.5 but not greater that 3.8)

Constant amplitude spectrum discussions

- 1.3 inertial fatigue factor:
  - There are some discussions whether a 1.5 factor would be a better conservative approach
  - 1.3 Currently considered as acceptable within the Chicago paper fatigue stress calculation approach. Ultimate load factor is taken from manoeuvre, not gust.

FEM

- The FEM should include a representative section of the structure, at least including adjacent stiffeners and frames



- Local stresses and fastener loads can be obtained from it
- Sensitive to representation in model of boundary conditions and stiffness

Analysis: Inspection safety factor discussion

- Depending on uncertainty, criticality, and general conservatism
- A factor of 2 is usually accepted for threshold and interval determination and interval of multiple load path structures.
- A factor of 3 or more is recommended for interval inspection for single load path (Ref. Swift) or equivalent, such as large antenna.
- For fatigue based demonstration, a factor of 5 is used by TCH when there is no specific test data available. For STCH, less information is available, therefore a factor of 8 is recommended.

Analysis: Sources of material properties

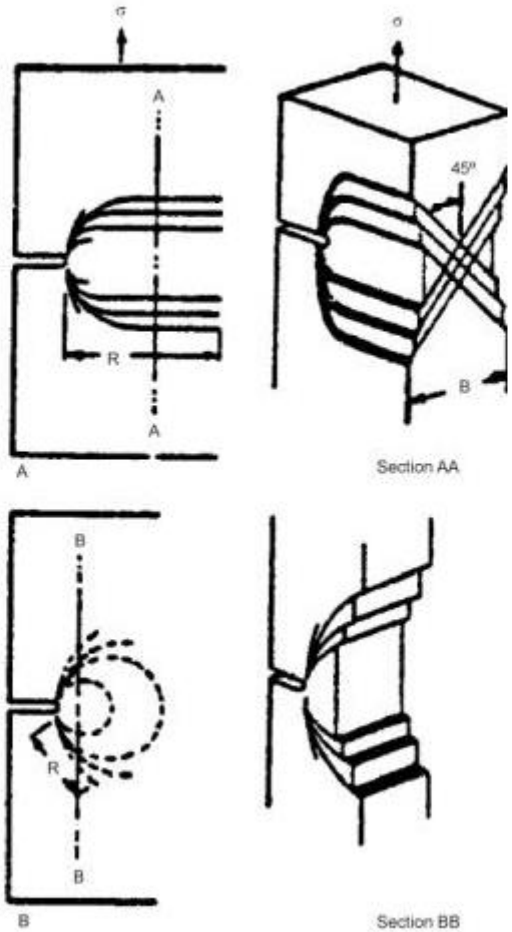
- Material property accepted references:
  - Ar-mmpds
  - Walker coefficients from Chicago Paper
  - ESDU
  - Handbuch Struktur Berechnung Ch. 60000
  - ASM handbook
  - NASGRO/AFGROW database
- In principle not right to extrapolate properties to different materials, tempers or orientations.

Analysis: References for beta

- Handbuch Struktur Berechnung Ch. 60000
- ESDU Intensity Factors
- NASGRO Manual
- Swift papers/courses

Stress State

- Plane stress (thin)/plane strain (thick). The stress state affects the critical stress intensity.
- While there is a constant K1c for plane strain, for thin sheet and ductile materials there is stable crack growth beyond K1c. Kc for unstable crack growth depends on thickness, initial crack size and geometry.
- Internally calculated by AFGROW
- R-curves for an elaborate approach



Analysis: Detectable crack

- The detectable crack assumed for the inspection interval has to be consistent with the access and inspection technique
- The inspection instruction should detail all necessary access (e.g. remove lining, antenna, etc.) and inspection instructions to ensure this
- Consider part of the crack hidden by a doubler, antenna, fastener head, etc.

- [NAVAIR technical manual]
- [NDT Resource Center]

TABLE 2. Detectable Crack Sizes Associated with Inspection Techniques (Reference [4])

Method	Description	Detectable Crack Length (inch)
Visual	Unpainted Surface*: 3 to 5x Magnification	1.0 or Hole-to-Edge
	Painted Surface	None
Penetrant	Unpainted Surface: 3 to 5x Magnification Without Magnification	 0.125 0.250
	Painted Surface	None
Magnetic Particle	Unpainted Surface: 3 to 5x Magnification Without Magnification	 0.0625 0.125
	Painted Surface: Without Magnification	 0.250
X-RAY Radiography	Uncovered length of crack in aluminum (not covered by a steel member)	0.75 or Hole-to-Hole or Hole-to-Edge
Ultrasonic Shear-Wave (Angle Beam)	Crack at fastener hole using mini probe (0.25 x 0.25 inch element) at 5 to 10 Mhz	0.125 Long x .0625 Deep
	Crack in Clevis or Lug	0.125 Long x 0.0625 Deep
Ultrasonic Longitudinal Wave (Straight Beam)	Bolts	¼ to 1/3 Diameter
	Crack at Fastener Hole	0.125
Bolt Hole Eddy Current (Faster Removed)	Edge Corner Crack	0.030 x 0.030
	Inside Diameter Surface	0.060 Long x .030 Deep
Eddy Current Surface Probe	Crack at Fastener	0.0625 Uncovered Length
	Crack Away Fastner	0.125

High Altitude considerations

- EASA current policy is to assume loss of antenna
- For some models, adoption of FAA Special Condition in the EASA TCDS
  - 1.67 factor instead of 1.33 for static stress 25.365 (d)
  - Maximum allowed opening size in the TCDS to ensure physiologically safe pressure histories:
    - Limits size of antenna hole: it is assumed that the antenna can be lost.
    - Additional criteria for critical crack : Depending on crack length and skin/doubler stiffness, bulging will create an opening.
  - Typically requiring a SF of 4 for inspection interval calculation.

Corrosion protection and installation issues

- Sealant application vs. drainage and ventilation
  - The doubler, protective coatings, sealing etc should be based on OEM design principles relevant to the fuselage skin or else justified separately;
- Installation instructions for fatigue critical quality:
  - Fit. Deburring to remove stress concentrations
  - Paint/protection removal (avoid scratches) and reapplication (ensure right material and process).
- Re-use of holes: if oversized, initial damage can be considered as 0 for fatigue life calculations.
- Anticipate and prevent contact with adjacent structures after structure deformation. Applicability: Set minimum distance to adjacent repairs/mods, to avoid stress interaction.

DOA capabilities: CVE and documentation

- For installations modifying PSEs, substantiation data should be prepared, checked (by a Compliance Verification Engineer (CVE)) and approved by persons with sufficient capability, knowledge and experience regarding F&DT.
- This can be shown through a history of substantiations agreed by EASA/NAA.
- The data supporting F&DT compliance verification should contain sufficient descriptive and substantiating data.
- Reference to the FAA’s Advisory Circular (AC) 43.13-2A is not generally sufficient by itself.

DOA: Organisational issues

- The Design Assurance System, should appropriately cover also the structures area in avionics changes/modifications
- Particularly change classification and assessing the impact on compliance with 25.571
- Updating, adapting or initiating all appropriate design procedures in the handbook/manual to appropriately include the structural aspects of the design (for instance in preparation of certification plan, showing of compliance, producing of manuals and instructions, control of subcontractors, etc. etc.)
- Availability of appropriate design tools, methods and guidance
- Availability of CVE’s capable to also cover the structures area
- (Continuation) Training for CVE’s and general awareness training for all other staff
- …

Conclusions

- Either simplified conservative analysis, or all detrimental factors and uncertainties should be considered.
- Adequate installation and maintenance instructions
- Adequate DOA qualification and organisation

- Questions?

EASA

European Aviation Safety Agency

End slide

Your safety is our mission.

An agency of the European Union

Backup slide : Complex analysis

- Crack/hole interaction

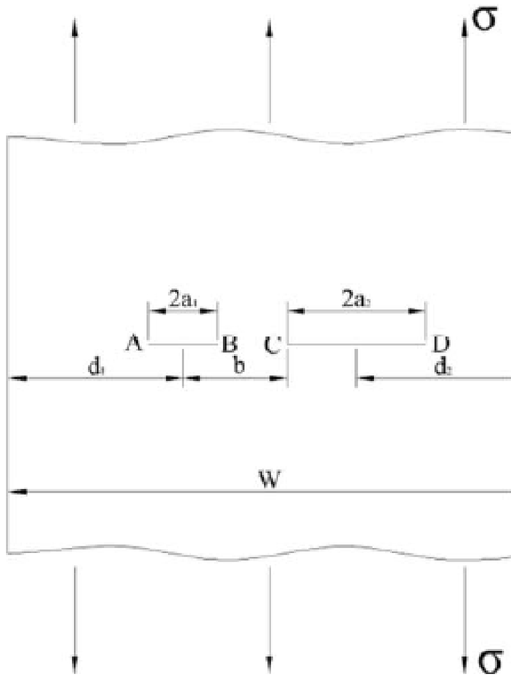


Fig. 2 –The finite width plate with two collinear cracks.

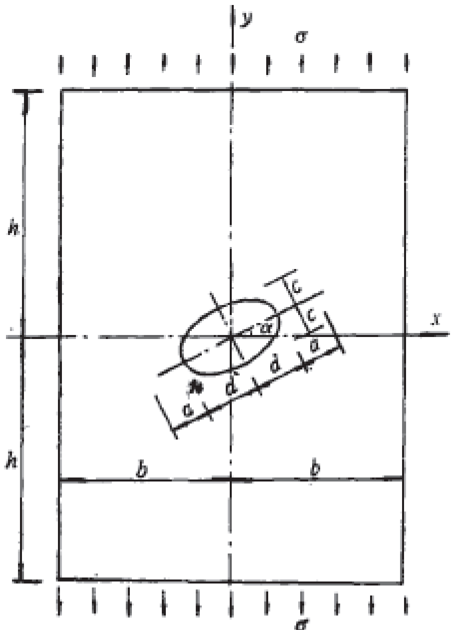
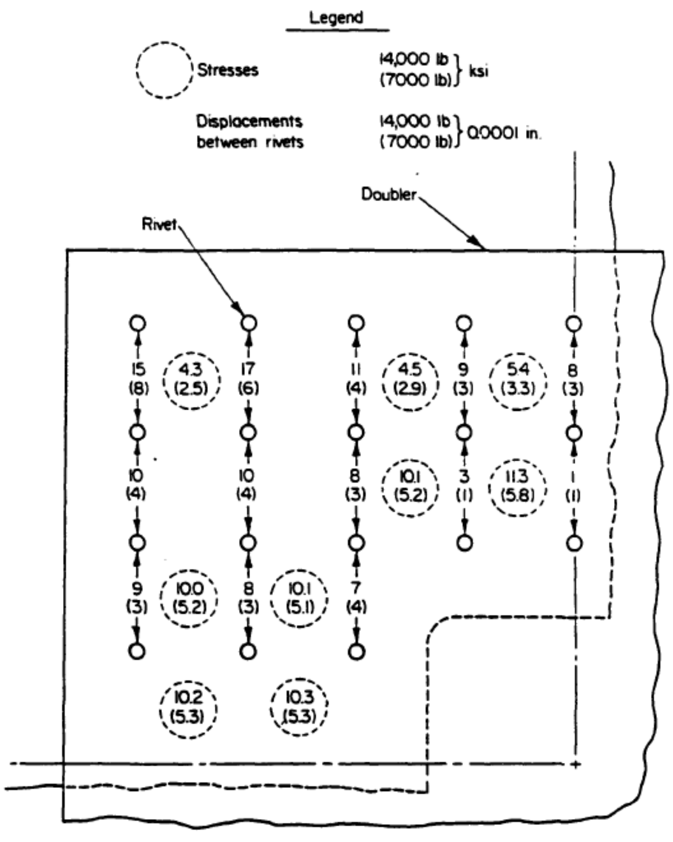


Fig. 6 Two slant cracks emanating from an elliptical hole in a finite plate

Backup slide : Complex analysis



0.039 Panel with 0.039 patch
Specimen No. 25 (with cutout)

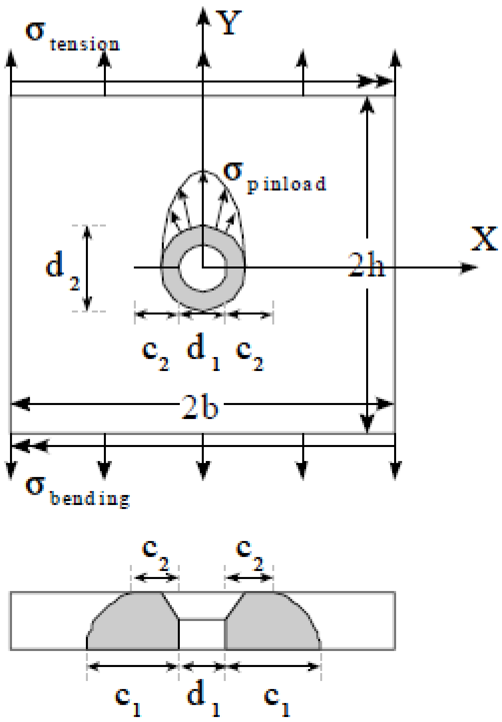
FIGURE 4-11. DOUBLER STRESSES AND RIVET DISPLACEMENTS

4-18

- Load distribution among fasteners
- From DOT-FAA-CT-93-79

Backup slide : Complex analysis

- Design details influence: fastener type, fit, surface treatment.

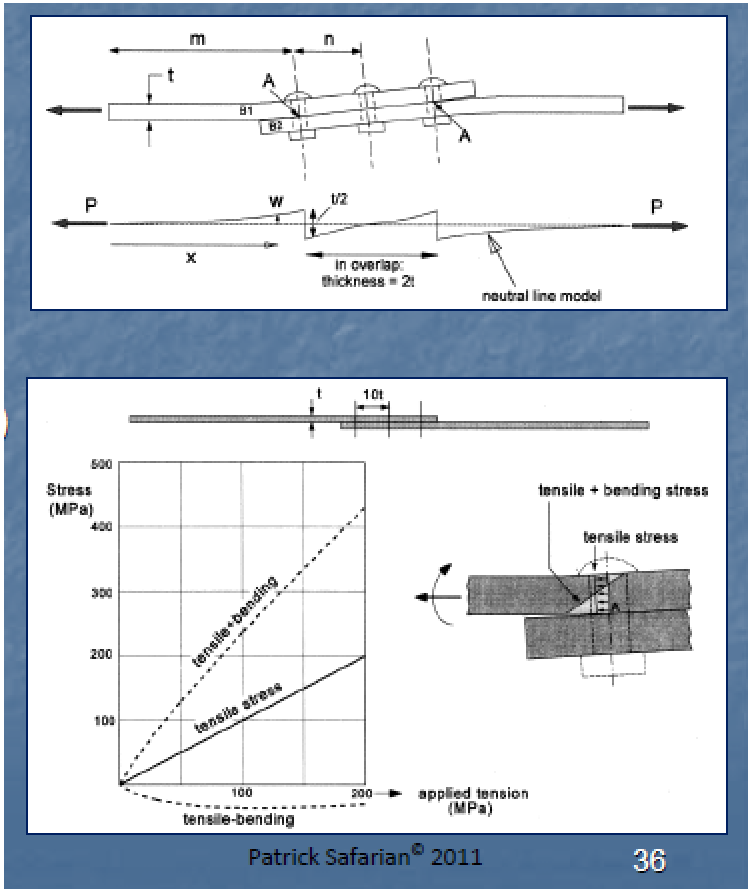


Backup slide : Complex analysis

- Bulging factor: Stress intensity is increased by fuselage bulging. Factor could be of the order of up to 2.5 for large cracks (Ref. Swift paper, Bigelow /Bakuckas paper, HSB)

Backup slide : Complex analysis

- Secondary bending. Safarian has pointed out that eccentricity for hoop stress can have an effect on stress.
- It can be noted that this is a splice, not a doubler, but the eccentricity effect would be similar.



Backup slide: RAPID Issues

(3) RAPID(C) (Ref. Manual):

(a) Limitations in program (only certain size and shape of antenna installations) need to be understood

(b) Only Rapid C is supported by the FAA Small Airplane Directorate office, for commuter aeroplanes only. Otherwise unconservative results could be obtained.

(c) Similarity between actual installation and RAPID (stylized) model chosen needs to be substantiated

(d) Default values (e.g. wide-body fatigue spectrum fatigue spectrum) need to be understood

(e) Typically the cable penetration hole is the critical location, contrary to other investigations, where fastener holes seem to be more critical

(f) Equivalent stress option leads to less than 1 delta-p (circumferential) per flight

(g) Use of Kc rather than KIc

(h) Use of factor 1.1 for residual strength where CS-25 would require 1.15

(i) 70% payload factor: The equivalent stress calculated is usually conservative, but if this stress is derived independently, assumptions need justification

(j) Similarly the far field ultimate load alleviation for structure with holes would not be applicable to pocketed fuselage skin

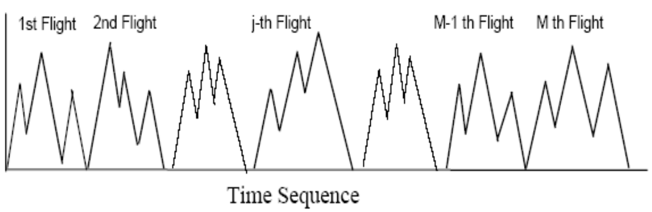
Backup slide: RAPID

- RAPID(C)
- Equivalent stress
- Flight by flight

Circumferential

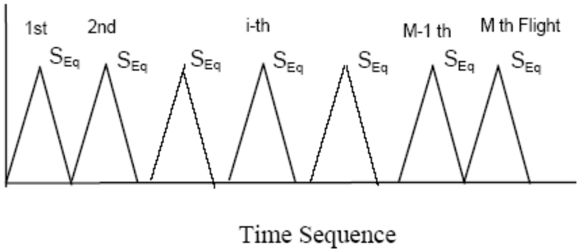
$$\frac{pr}{t} = \sigma_p$$

Longitudinal



$$\sigma_{1g,max,100\%\ Payload} = \frac{C\left(F_{tu}/1.50\right) - (p+1.1)\ r/2t}{2.5}$$

C = material allowable reduction factor (0.88) for assembled structure
p = pressure differential
r = fuselage radius at the location
t = skin thickness at the location
F_{tu} = ultimate tensile strength of the skin material (psi)
$\sigma_{1g,max,\ 100\%\ Payload}$ = maximum longitudinal design stress



$$\sigma_{1g} = \sigma_{1g,\ max,\ 70\%\ Payload}\left(\frac{L}{S}\right)$$