出國報告（出國類別：實習）

參加美國波音公司
飛機結構修理訓練課程：
「**Metallic Repair Structural Repair for Engineers - Part III**」
出國報告書

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參加美國波音公司飛機結構修理訓練課程:「Metallic Repair Structural Repair for Engineers - Part III」出國報告書

目　錄

壹、目的

本局經常審查航空公司依法呈報之飛機結構大修理技術資料，故本次參加美國波音公司飛機結構修理訓練課程：「Metallic Repair Structural Repair for Engineers - Part III」，此次訓練課程，係三個系列課程中的 Part III 課程，Part I 為金屬件修理訓練，為多數航空公司參加之基礎課程，Part II 為承受壓力負載結構之修理訓練，而本次參加之 Part III 課程，為介紹飛機結構安全重要之疲勞分析(Fatigue Analysis)及容損分析(Damage Tolerance Analysis)訓練課程。

課程主要分為三部份，第一部份，說明飛機結構疲勞分析方法；第二部份，進行蒙皮與補片鉚接結構之負載分析，利用分析公式，及有限元素方法(Finite Element Analysis)計算負載分佈；第三部份，說明結構容損分析方法，計算裂紋成長率(Crack Growth Rate)及結構剩餘強度分析(Residual Strength Analysis)，以訂定檢查起始點(Inspection Threshold)及檢查時距(Inspection Interval)。課程講授觀念與實際分析計算並重，可了解飛機結構最重要兩種設計基礎：疲勞及容損設計，以及了解實際應用方法，可增進本局對大修理資料審查之技術，且對老舊飛機持續安全之法規符合性審查有所助益。

貳、行程經過摘要

一、本次行程安排如下：

11 月	11 月	11 月
15 日	16-20 日	21-22 日
台北－美國西雅圖	美國西雅圖	美國西雅圖－台北

二、課程安排：

2009/Nov/16
- Course/Student Introduction
- Fatigue Concept
- Joint Modeling

2009/Nov/17
- Stress Severity Factor
- 737 Fuselage Skin Exterior Repair
- 777 Fuselage Skin Flush Repair

2009/Nov/18
- 747-400 Extrusion Repair
- Effective Stress and After Press Loads

2009/Nov/19
- Effective Stress and After Press Repair
- Damage Tolerance Concepts
- Damage Tolerance Elements
- Residual Strength Analysis

2009/Nov/20
- Crack Growth Rate Analysis
- SSID Example
- Repairs
- Review
- Final Examination

參、訓練內容摘要

本章節主要在整理出此次結構修理訓練課程中相關重點內容。

一、飛機結構疲勞簡介

(一) 疲勞定義：

ASTM 對疲勞的定義為：”The process of progressive localized permanent structural change occurring in a material subjected to conditions that produce fluctuating stresses and strains at some point or points and that may culminate in cracks or complete fracture after a sufficient number of fluctuations”。

其重要特性為：

1. 結構所受的為重複性外力。
2. 會對材料造成局部應力及形變。
3. 對材料產生的形變永久性的。
4. 損壞過程是漸進的。
5. 最後會對結構造成裂紋或完全的破壞。

因此裂紋產生時間(Crack Initiation Life)即為評估結構的疲勞特性的重要指標，亦影響結構之疲勞壽命(Fatigue Life)。

(二) 波音飛機結構疲勞設計理念：

如圖1所示，波音公司飛機設計使用壽命(Design Service Objective, DSO)為 20 年，且預期在飛機使用 15 年，也就是到達飛機 75%

DSO 時，疲勞效應才會對飛機結構強度之衰減產生顯著影響，且隨著飛機使用時間越久，疲勞對飛機結構的影響性也越來越大。

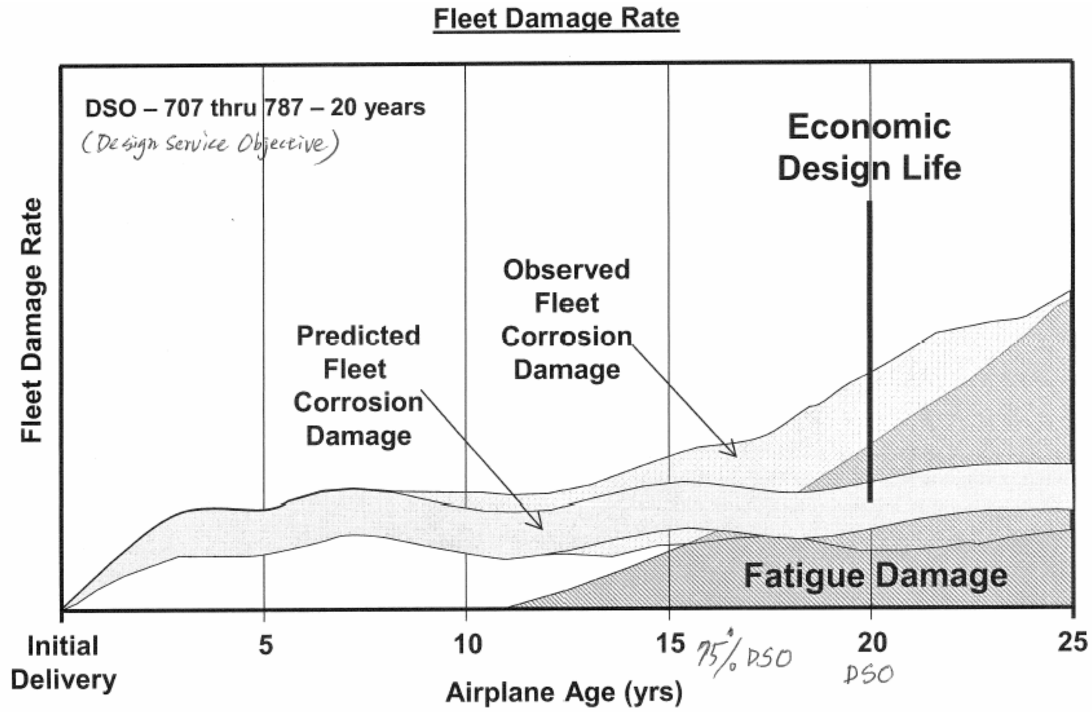


圖 1 飛機結構損壞因素分析

(三) 飛機結構疲勞的種類：

飛機結構疲勞的種類可分為下列幾類：

1. 正常性疲勞(Normal Fatigue)：

   是一般設計可預期產生的疲勞現象，可利用疲勞分析(Fatigue Analysis)及疲勞試驗(Fatigue Test)定義相關安全係數(如：Scatter Factor)。

2. 不正常性疲勞(Anomalous Fatigue)：

   是設計沒有預期到而產生的疲勞現象，其發生是一般由於材料瑕疵、製造瑕疵或維修不當所造成。

3. 不預期的正常性疲勞(Unexpected Normal Fatigue)：

超出設計所預期產生的疲勞現象，其發生是由於設計不良或使用不當所造成，例如：結構受力超出預期(Overloading)、錯誤的蒙皮設計厚度等。其改正須由各機隊使用情形回報至原廠後，以採取進一步的改正措施，例如：技術通報(Service Bulletin, SB)或適航指令(Airworthiness Directive, AD)之發布。

4. 多重疲勞損傷(Widespread Fatigue Damage, WFD)：

指的是飛機結構組件多處產生疲勞裂紋，使得該結構組件剩餘強度無法承受設計負載。又可分成結構多重損傷(Multiple Site Damage, MSD)及多重結構組件損傷(Multiple Element Damage, MED)兩種。

(1) 結構多重損傷(Multiple Site Damage, MSD)：

指的是在同一組件中，有多處產生疲勞損傷。例如：在同一張蒙皮的不同鉚釘孔位產生疲勞裂紋。通常該組件中具有重覆性的應力集中因素者(Stress Riser)，常發生此類之疲勞損傷，所謂的 Stress Riser 一般而言指的是鉚釘孔，因其周圍會有應力集中的現象，在重複性受到張力下，會導致裂紋開始由鉚釘孔產生後，延受力垂直方向開始成長，最終使得多處裂紋結合成一較大的損傷而破壞結構。

(2) 多重結構組件損傷(Multiple Element Damage, MED)：

指的是多處相同受力模式的結構組件同時產生疲勞損傷，該類結構通常是具有 Redundant 設計理念，例如：飛機機翼下蒙皮多處下蒙皮板片受到彎矩(Bending Moment)而產生疲勞裂紋。

二、飛機疲勞結構設計

(一) 設計使用壽命(Design Service Objective, DSO)：

波音公司利用設計使用壽命(Design Service Objective, DSO)，做為機體結構設計疲勞壽命之用語，一般設定為 20 年，其目的是希望當飛機使用時間達 20 年時，有 99% (信心水準 Confidence Level)的結構件不會產生目視可檢查到的裂紋，在使用時間達 30 年時，有 95%的結構件不會產生目視可檢查到的裂紋。至於 DSO 長短的選定，則取決於維修成本(檢查及修理成本)及燃油成本(較佳的 DSO 有較強但較重之機體)間之最佳化考量。

(二) 飛機結構負載種類：

飛機結構負載種類分為下列幾種：

1. 操作負載(Operating Loads)：

   此為每次飛航過程所會承受之負載，一般垂直 Load Factor 約為 1.4g。進行疲勞分析時，通常並非以飛機從起飛至降落之間(亦即 Ground-Air-Ground/GAG Flight Cycle)的各個階段所可能承受之最大或最小負載做為分析輸入，而是以平均應力(Mean Stress)做為結構疲勞分析的負載來源，以評估該處結構可能累積的疲勞損傷(Cumulative Damage)。圖 2 所示為飛機機翼下蒙皮受力之應力譜(Stress Spectrum)，以平均應力表示各飛航階段之受力情形。

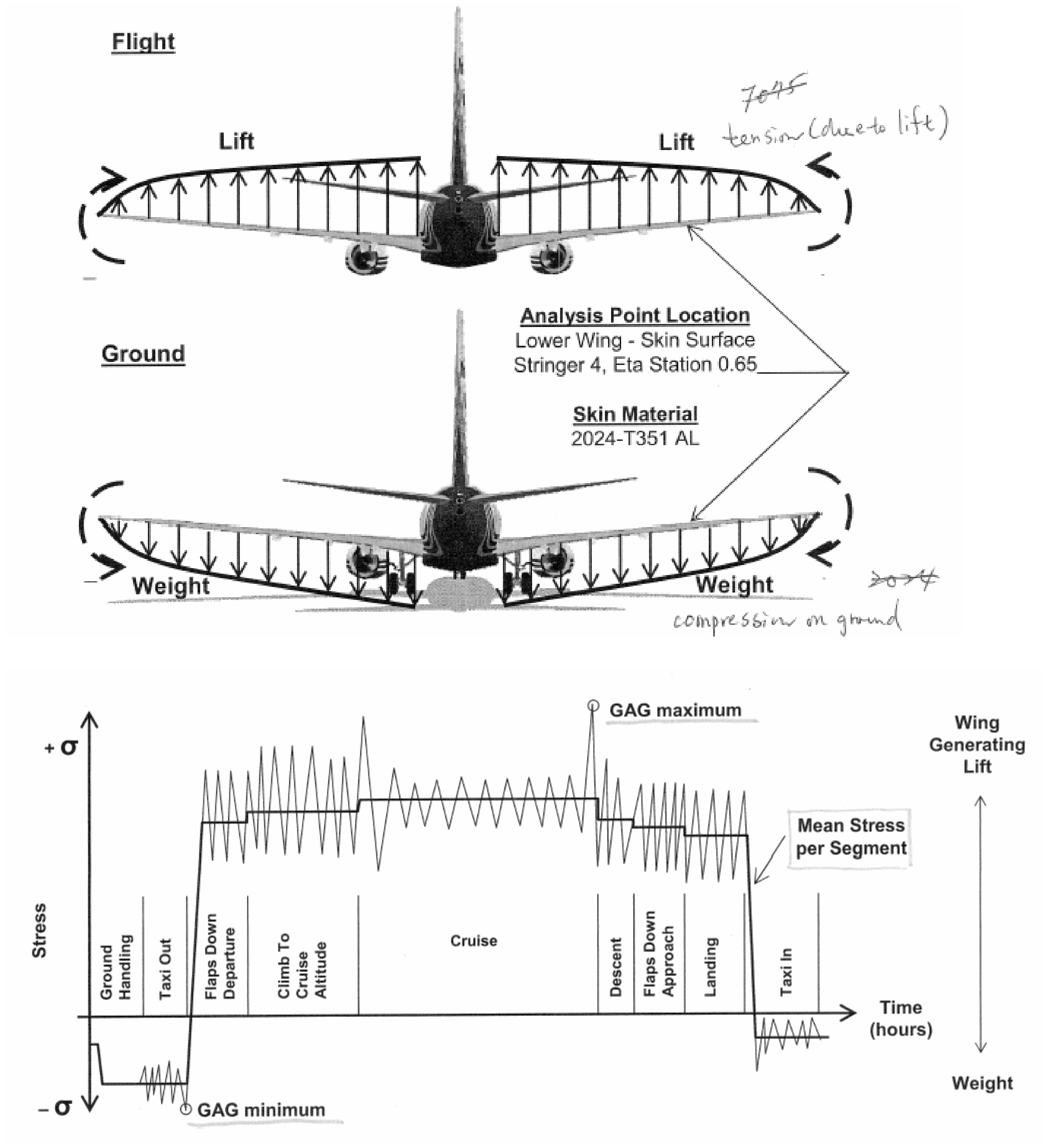


圖 2 飛機機翼下蒙皮受力之應力譜(Stress Spectrum)

2. 限制負載(Limit Loads)：

依 FAR 25.301 定義為飛機使用期間，預期可能承受之最大負載。一般垂直操控(Vertical Maneuver) Load Factor 約為 2.5g。

3. 極限負載(Ultimate Loads)

依 FAR 25.305 定義為限制負載的 1.5 倍，在限制負載不足以定義結構所需承受之負荷時，即需要測試條件須提昇至極限負載。一般垂直操控(Vertical Maneuver) Ultimate Load Factor 約為 3.75g (=2.5gx1.5)。

結構修理則需將結構回復至可承受限制負載及極限負載之能力，而進行結構疲勞分析或容損分析時，則須同時考慮飛機之操作負載。因此建立整個飛航過程中的機體可能遭受的負載譜(Load Spectrum)，為後續結構疲勞分析及容損分析重要的前置工作。

(三) 疲勞試驗(Fatigue Test)：

1. 飛機結構疲勞試驗，從前述之各項負載數值，做為測試輸入，利用固定大小之往覆式負載，測得應力(Stress, S)及至結構損壞之循環週期(Cycles to Failure, N)圖表，簡稱 S-N Curve，其目的是用以比較在相同的受力環境下，不同材料的疲勞特性，其另外的一項特性為，當應力低於某個值時，該材料可謂不會損壞，該應力值為「疲勞限度(Endurance Limit)」，如圖 3 所示。一般鋼材有此疲勞限度，而鋁及鈦金屬則無此疲勞限度。

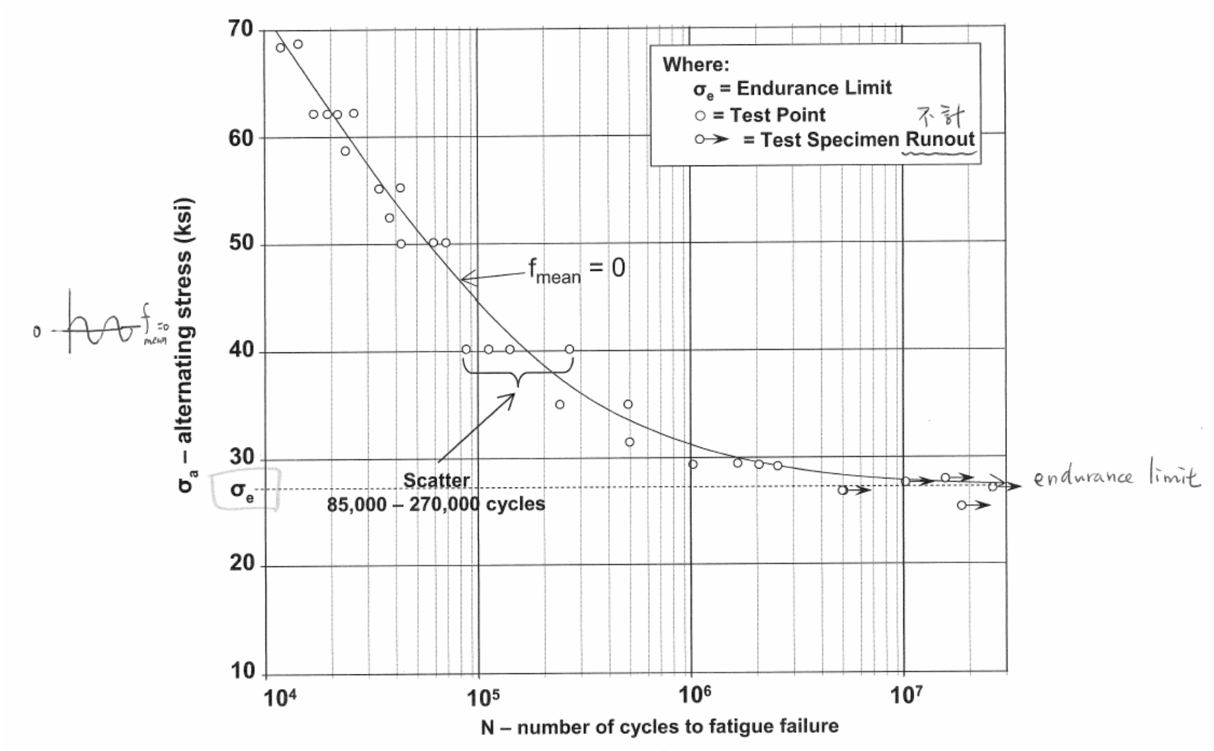


圖 3 S-N 曲線

2. 累積疲勞損傷(Cumulative Damage)：然而飛機結構受力並非固定大小，因此結構的疲勞損傷是在不同的應力下慢慢累積而成，而要評估此疲勞損傷的大小，則需要利用 Miner's Rule：

Miner's Method

$$\boxed{D = \sum_{i=1}^{J} \frac{n_i}{N_i}}$$

Where:

D = Fatigue damage. If D = 1.0 a fatigue crack has formed and grown to a failure level.

n_i = Applied number of cycles at the i^{th} specific stress level.

N_i = Allowable number of cycles to failure at the i^{th} specific stress level.

j = Total number of stress levels.

亦即利用不同應力下之損壞循環週期(Cycles)為基礎，代入飛機結構在不同應力程度下，所受到外力(飛行負載、地面負載等)的循環週期，估算出該結構開始產生疲勞破壞之 Cycles，

可做為持續適航維護檢查、結構壽限/更換時間點評定之用。

(四) 飛機結構之防疲勞設計(Design for Fatigue)：

1. 良好的結構細部設計：避免結構產生應力集中，例如避免：Notch、Short Edge Margin、Close Adjacent Holes、Perpendicular to Load Path 或 Rough Surface Finishes 等。
2. 利用珠擊(Shot Peening)或冷作加工(Cold Work)：對於 2024 鋁合金，可使用珠擊，而對於 7075 鋁合金，則可使用冷作加工，使其材料表面產生 Compressive Layer，降低 Peak Stress 值，增加承受拉力負載之能力，可有效提昇結構疲勞壽命。

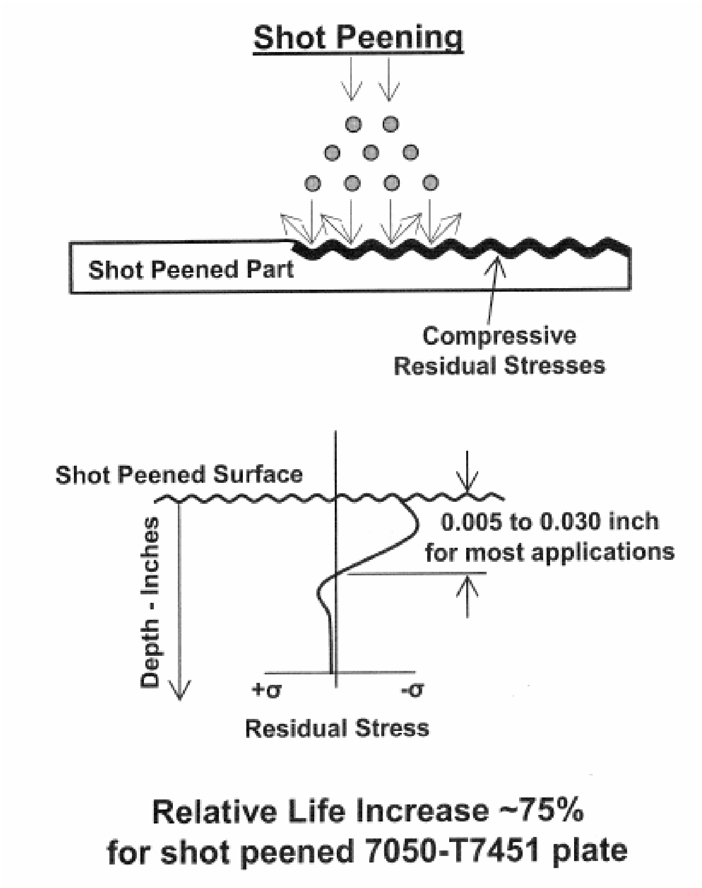


圖 4 Shot Peening 可降低 Surface Peak Stress

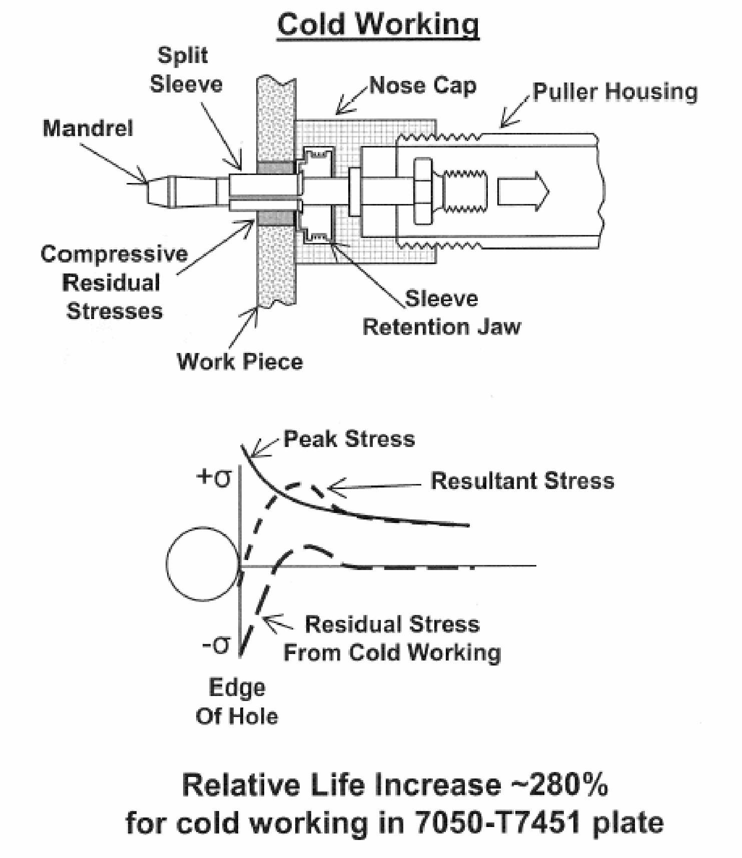


圖 5 Cold Work 可降低孔位附近結構之 Peak Stress

3. 使用 Fatigue Resistance Fastener (例如：Rivet, Hex-drive Bolt, Lockbolt 等)，並以緊配(Interference Fit)的方式安裝，此方法亦即使材料表面產生 Compressive Layer，提昇結構疲勞壽命。

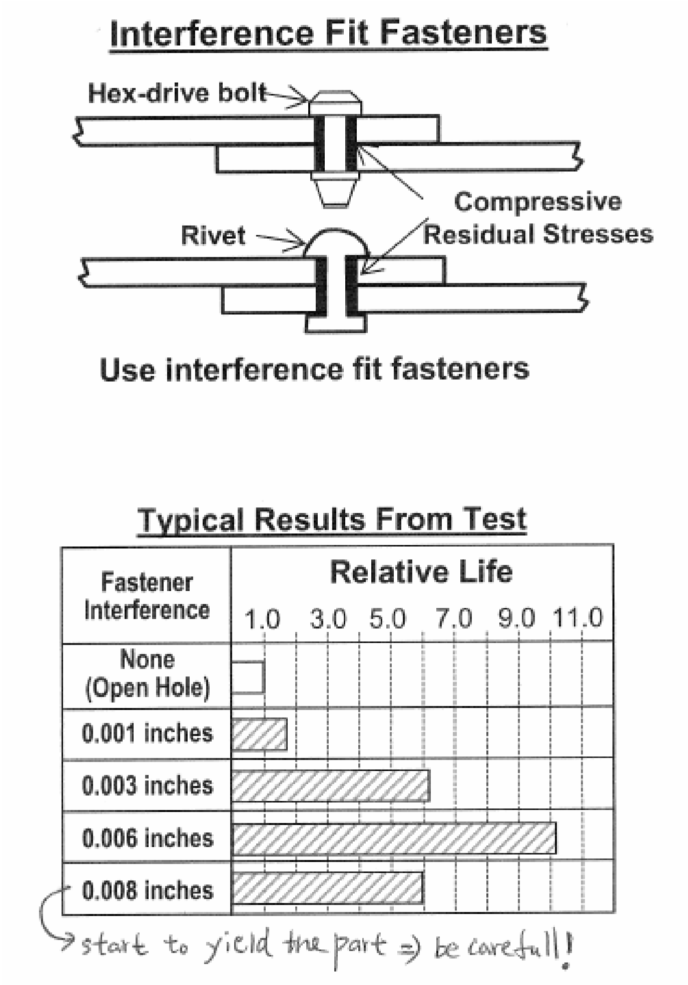


圖 6 扣件緊配安裝可產生孔位 Compressive Layer

(五) 飛機結構疲勞裂紋之修理：

1. 當飛機結構產生疲勞裂紋時，必須進行修理。因為飛機結構必須隨時在可承受適航標準規定之 Limit Load 及 Ultimate Load 的狀態(如：FAR 25.301, 25.305)。
2. 止裂孔(Stop Drill)議題研討：當將結構裂紋以打止裂孔(Stop Drill)方式進行處理時，此種作法僅能降低該處之應力集中因子(Stress Concentration Factor) Kt，由於可受負載之截面積在扣除裂紋長度後會相對減少，但在所受外力仍不變下，剩餘結構承受應力因而增加，而形成 Overload 情況，造成結構之

Safety Margin 減少，若不進行永久性修理，終究仍會因為裂紋持續成長而產生結構破壞，因此打止裂孔只能當做暫時性修理(Temporary Repair)。最佳的修理方式是將受損結構進行換新，但通常此較花成本，另外的選擇則是將該處進行打磨(Trim Out)後加上加強補片(Splice)的方式進行修理。

三、飛機結構修理設計及分析

(一) 使用結構修理手冊(Structure Repair Manual, SRM)進行結構修理設計及分析：

1. SRM 係飛機原製造廠工程師依據飛機疲勞負載數據(Fatigue Load Data)，以發展出符合 Durability 及容損分析原則之修理方法。SRM 中所列之修理方法係一般性修理原則，由於航空公司工程師並無從得知相關飛機結構負載數據，當飛機細部結構些微差異、進手空間限制及庫房可供使用鉚釘型號之限制等因素，無法完全依照原製造廠之 SRM Repair 進行修理時，最佳方法即是利用原製造廠 SRM Repair 做為飛機結構修理設計基礎，提出修理方法建議(Repair Proposal)，分析修理方法建議是否在結構應力之安全裕度範圍內。然後再比較 SRM Repair 與所提修理方法建議之間疲勞壽命的差異，當所提修理方法建議之疲勞壽命應高於 SRM Repair 時，該之修理方法建議即為可行。
2. 結構修理疲勞分析(Fatigue Analysis)方法：在進行結構修理疲勞分析時，其中一項困難點在於決定板片及鉚釘間之負載分佈(Load Distribution)，板片及鉚釘間之負載並非平均分佈，因此須得知板片及鉚釘間之分配負載值，以計算評估所提出之

修理建議產生之結構應力是否小於會產生疲勞破壞之應力。

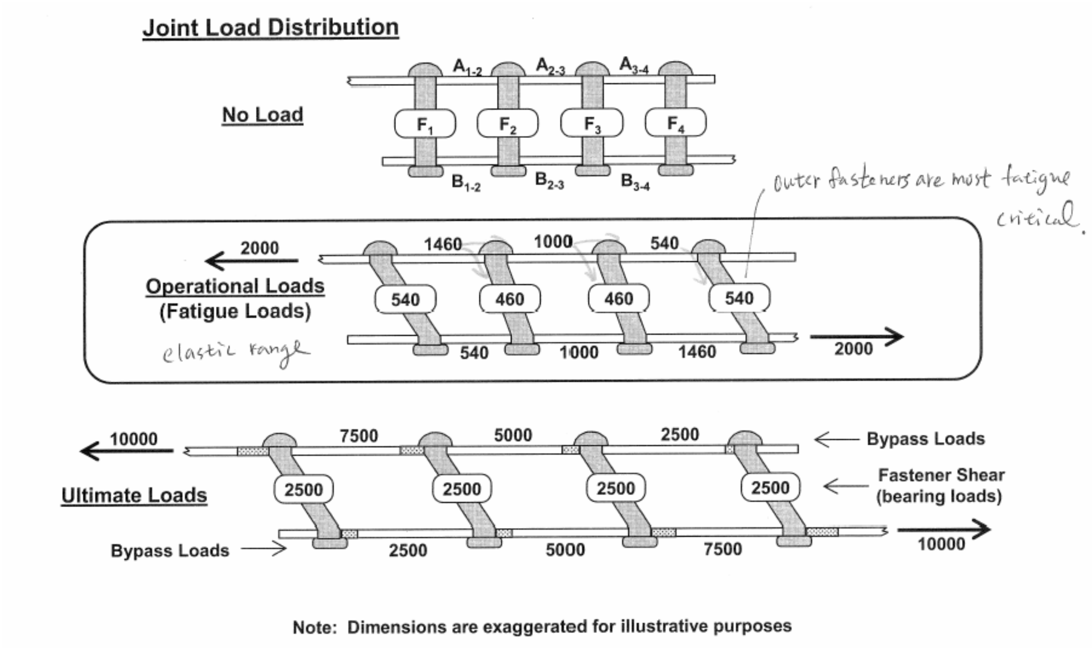


圖 7 板片及鉚釘間之負載分佈示意圖

(二) 計算板片/鉚釘安裝結構之負載分佈：

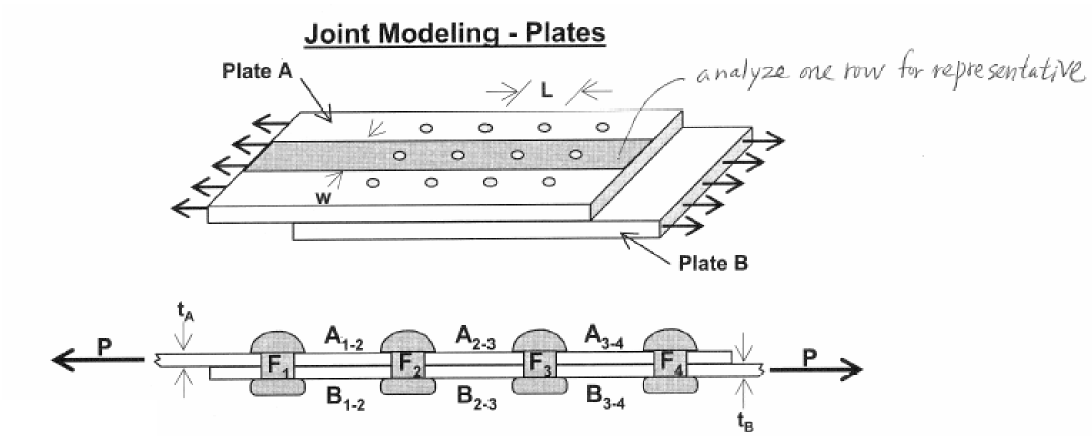
1. 建立板片/鉚釘結合處之分析模型(Joint Modeling)，以進行結構修理負載分析，是一簡化可行的分析方法，其原理係將板片及鉚釘視為一彈簧體(Spring)，而計算方法即在決定板片及鉚釘在分析模型中的 Spring Constant。
2. 板片 Spring Constant 計算方式：

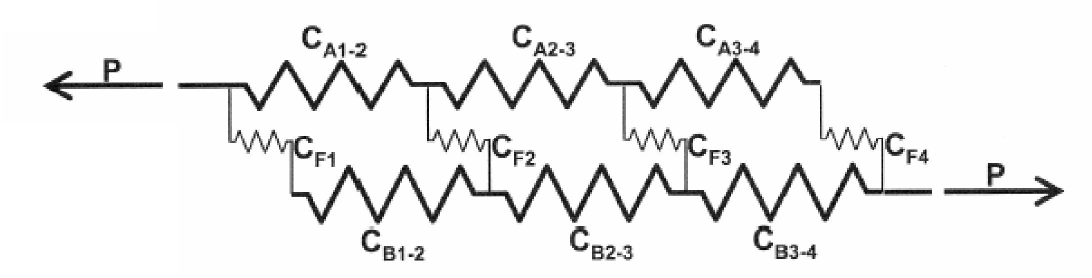
   板片 Spring Constant 計算公式如下：

$$C_{plate} = L / (AE)$$

where:

C_{plate} = plate spring constant (in./lb)
L = length (in.), fastener spacing in direction of load
A = cross-sectional area of plate between fasteners (in.^{2}) = w x t
E = Modulus of elasticity of the plate material (psi)
w = width (in.), fastener spacing perpendicular to load
t = plate thickness (in.)





3. 鉚釘 Spring Constant 計算方式：

計算鉚釘 Spring Constant 的經驗公式，通常得自於許多的測試資料，所需之數據資料包括：鉚釘直徑、鉚釘材質、鉚釘長度、鉚釘剪力強度、板片剪力強度、板片厚度、受到彎矩及剪力下之鉚釘變形量，以及鉚釘與板片之安裝構型等。常

使用的公式有兩種，一種出自於 FAA-AIR-90-01，另一種則出自於波音公司規範 D6-29942。

(1) FAA-AIR-90-01 計算方式：

FAA-AIR-90-01 “Repair to Damage Tolerant Aircraft”報告由 Mr. Tom Swift 於 1990 年發表，並由 FAA 列入正式文件，計算公式如下：

FAA-AIR-90-01

Repair to Damage Tolerant Aircraft

T. Swift, FAA (1990)

Fastener spring constant

$$C_F = [A + B\cdot(D / t_d + D / t_s)] / (E\cdot D)$$

where:

C_F = fastener spring constant (in/lb)

A = 5.0 for aluminum fasteners

= 1.666 for steel fasteners

B = 0.80 for aluminum fasteners

= 0.86 for steel fasteners

D = fastener diameter (in.)

E = modulus of elasticity of plates (psi) (only one modulus, so, this equation applies to only same doubler, skin, fastener material)

t_d = doubler thickness (in.) upper plate

t_s = skin thickness (in.) lower plate

Deflection Equation:

$$\delta_{fast} = P_F \cdot C_F$$

where

P_f = fastener shear load (lb)

此經驗公式是從測試數據進行歸納得出，但有其使用限制，亦即鉚釘材料須為鋼材或鋁材，且搭配安裝的板片須與鉚釘具有相同的材質。

(2) 波音公司規範 D6-29942 計算方式：

波音公司規範 D6-29942 ”Stress Severity Factors for Axially Loaded Mechanically Fastened Joints”於 1969 年提出，可適用各種材料，例如：Titanium 或 Monel 合金亦可，且鉚釘搭配安裝的板片材料也可以是不同材質，計算公式如下：

D6-29942

Stress Severity Factors for Axially Loaded Mechanically Fastened Joints
The Boeing Company (1969)

Fastener Spring Constant

$$C_F = \frac{4(t_i + t_j)}{9G_bA_b} + \frac{t_i^3 + 5t_i^2t_j + 5t_it_j^2 + t_j^3}{40E_{bb}I_b} + \frac{1}{t_i}\left(\frac{1}{E_{bb}} + \frac{1}{E_{ibr}}\right) + \frac{1}{t_j}\left(\frac{1}{E_{bb}} + \frac{1}{E_{jbr}}\right)$$

where:
- C_F = fastener spring constant (in/lb)
- t_i = thickness of plate i (in.) upper plate
- t_j = thickness of plate j (in.) lower plate
- G_b = shear modulus of the bolt material (psi)
- A_b = cross-sectional area of the bolt (in.^{2})
- E_{bb} = modulus of elasticity of the bolt material (psi)
- I_b = moment of inertia of the bolt cross-section (in.^{4}) $= \pi D^4 / 64$
- E_{ibr} = modulus of elasticity of plate i (psi)
- E_{jbr} = modulus of elasticity of plate j (psi)

Deflection Equation:

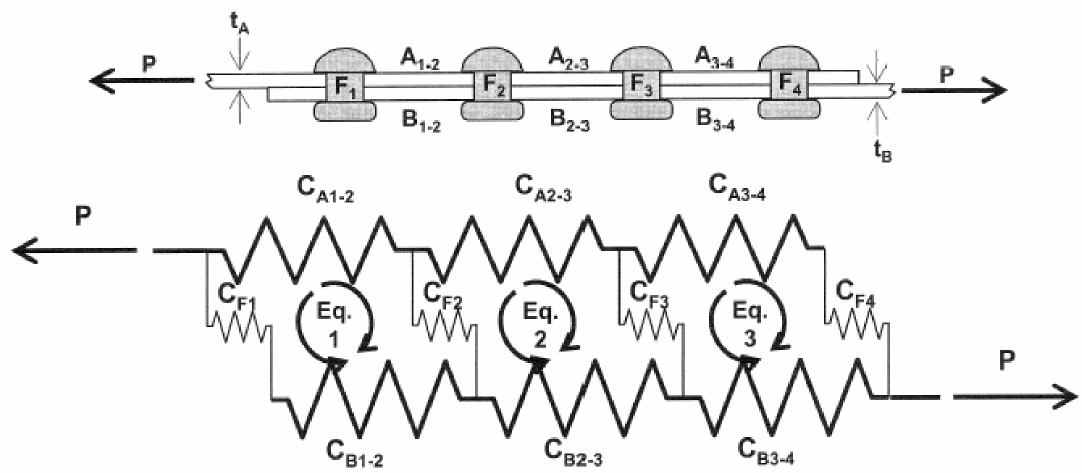
$$\delta_{fast} = P_F \cdot C_F$$

where
- P_F = fastener shear load (lb)

4. 鉚釘負載分配(Load Distribution)計算方法：一般有兩種方法計算上述之鉚釘 Spring Constant，其中一種為 Matrix Algebra，另一種為有限元素方法(Finite Element Analysis, FEA)。

(1) Matrix Algebra：首先先列出變形量方程式(Deflection Equations)，此方程式利用 Displacement Compatibility 及 Equilibrium 原理列出相關方程式，所謂 Displacement Compatibility 為相互之位移量是相同的觀念，Equilibrium 原理指得的是力學平衡的觀念，說明如下：

Joint Modeling – Deflection Equations



Equation 1

$$C_{A1\text{-}2}\cdot(P-P_{F1})+C_{F2}\cdot P_{F2}=C_{F1}\cdot P_{F1}+C_{B1\text{-}2}\cdot P_{F1}\Rightarrow$$

deflection 1
||
deflection 2

Equation 2

$$C_{A2\text{-}3}\cdot(P-P_{F1}-P_{F2})+C_{F3}\cdot P_{F3}=C_{F2}\cdot P_{F2}+C_{B2\text{-}3}\cdot(P_{F1}+P_{F2})$$

Equation 3

$$C_{A3\text{-}4}\cdot(P-P_{F1}-P_{F2}-P_{F3})+C_{F4}\cdot P_{F4}=C_{F3}\cdot P_{F3}+C_{B3\text{-}4}\cdot(P_{F1}+P_{F2}+P_{F3})$$

Equation 4

$$P_{F1}+P_{F2}+P_{F3}+P_{F4}=P$$

依據所列出的 Deflection Equations 再利用行列式運算，計算出各鉚釘所受負載，如下圖列示：

4 Equations – 4 Unknowns

Equation 1

$$C_{A1-2} \cdot (P - P_{F1}) + C_{F2} \cdot P_{F2} = C_{F1} \cdot P_{F1} + C_{B1-2} \cdot P_{F1}$$
$$C_{F1} \cdot P_{F1} + C_{A1-2} \cdot P_{F1} - C_{F2} \cdot P_{F2} + C_{B1-2} P_{F1} = C_{A1-2} \cdot P$$
$$\mathbf{(C_{A1-2} + C_{B1-2} + C_{F1}) \cdot P_{F1} - C_{F2} \cdot P_{F2} = C_{A1-2} \cdot P}$$

Equation 2

$$C_{A2-3} \cdot (P - P_{F1} - P_{F2}) + C_{F3} \cdot P_{F3} = C_{F2} \cdot P_{F2} + C_{B2-3} \cdot (P_{F1} + P_{F2})$$
$$C_{F2} \cdot P_{F2} + C_{A2-3} \cdot P_{F1} + C_{A2-3} \cdot P_{F2} + C_{B2-3} \cdot P_{F1} + C_{B2-3} \cdot P_{F2} - C_{F3} \cdot P_{F3} = C_{A2-3} \cdot P$$
$$\mathbf{(C_{A2-3} + C_{B2-3}) \cdot P_{F1} + (C_{A2-3} + C_{B2-3} + C_{F2}) \cdot P_{F2} - C_{F3} \cdot P_{F3} = C_{A2-3} \cdot P}$$

Equation 3

$$C_{A3-4} \cdot (P - P_{F1} - P_{F2} - P_{F3}) + C_{F4} \cdot P_{F4} = C_{F3} \cdot P_{F3} + C_{B3-4} \cdot (P_{F1} + P_{F2} + P_{F3})$$
$$C_{F3} \cdot P_{F3} + C_{A3-4} \cdot P_{F1} + C_{A3-4} \cdot P_{F2} + C_{A3-4} \cdot P_{F3} + C_{B3-4} \cdot P_{F1} + C_{B3-4} \cdot P_{F2} + C_{B3-4} \cdot P_{F3} - C_{F4} \cdot P_{F4} = C_{A3-4} \cdot P$$
$$\mathbf{(C_{A3-4} + C_{B3-4}) \cdot P_{F1} + (C_{A3-4} + C_{B3-4}) \cdot P_{F2} + (C_{A3-4} + C_{B3-4} + C_{F3}) \cdot P_{F3} - C_{F4} \cdot P_{F4} = C_{A3-4} \cdot P}$$

Equation 4

$$\mathbf{P_{F1} + P_{F2} + P_{F3} + P_{F4} = P}$$

$$[C_{ij}] \cdot [P_{Fi}] = [CP]$$

$$\begin{bmatrix} C_{A1-2} + C_{B1-2} + C_{F1} & -C_{F2} & 0 & 0 \\ C_{A2-3} + C_{B2-3} & C_{A2-3} + C_{B2-3} + C_{F2} & -C_{F3} & 0 \\ C_{A3-4} + C_{B3-4} & C_{A3-4} + C_{B3-4} & C_{A3-4} + C_{B3-4} + C_{F3} & -C_{F4} \\ 1 & 1 & 1 & 1 \end{bmatrix} \cdot \begin{bmatrix} P_{F1} \\ P_{F2} \\ P_{F3} \\ P_{F4} \end{bmatrix} = \begin{bmatrix} C_{A1-2} \cdot P \\ C_{A2-3} \cdot P \\ C_{A3-4} \cdot P \\ P \end{bmatrix}$$

2 x 2 matrix – 2 fastener joint

3 x 3 matrix – 3 fastener joint

上述計算方法可利用現有 Microsoft Excel 軟體寫出巨集進行運算，本次課程時即是利用 Excel 軟體進行相關實例演練，軟體畫面如圖 8 所示。

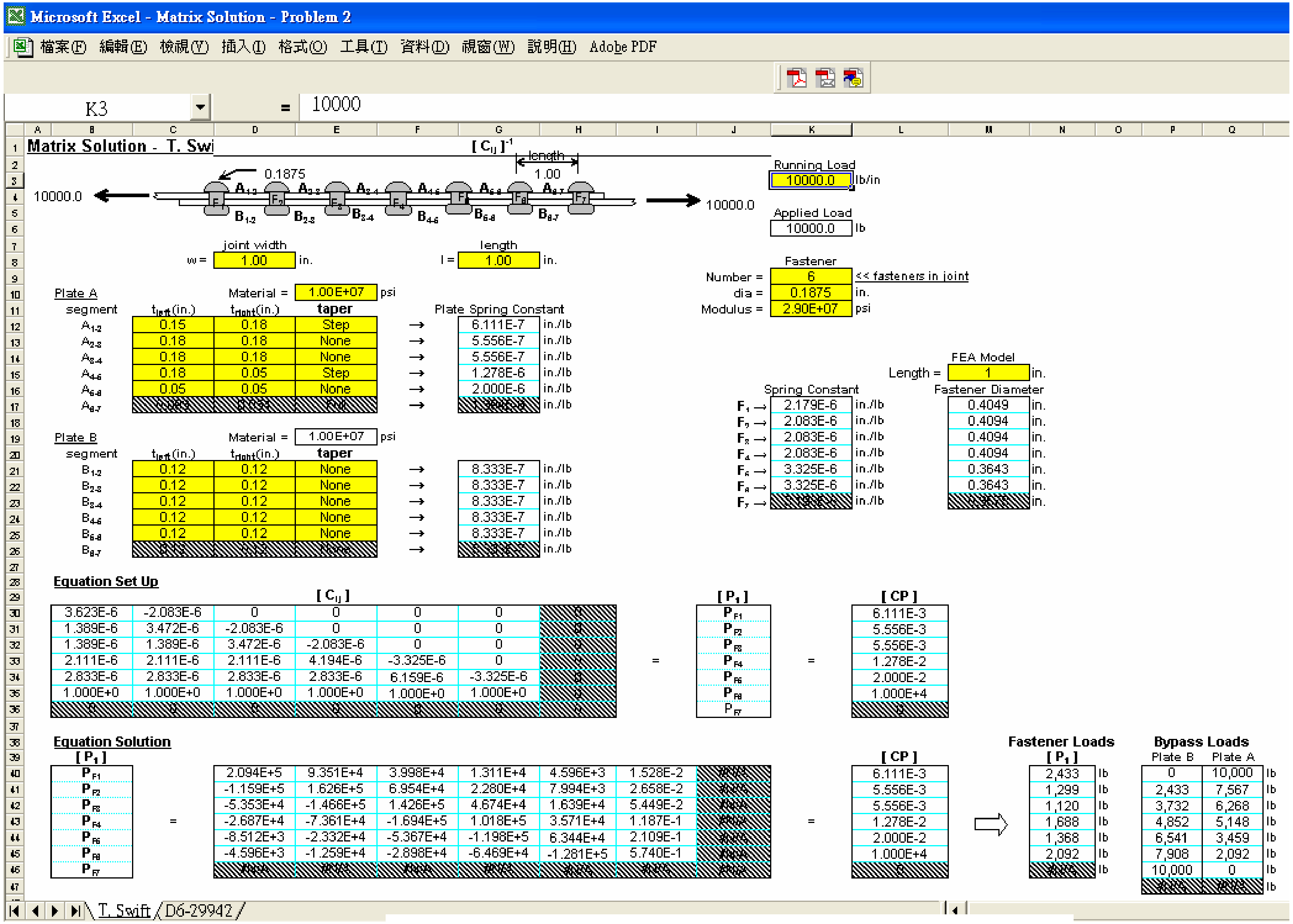


圖 8 利用 Excel Spreadsheet 計算鉚釘及板片負載

(2) 有限元素方法(Finite Element Analysis, FEA)

前述 Matrix Algebra 計算方法可應用於簡單的鉚釘及板片安裝構型，但當安裝構型涉及更多的鉚釘數、Tapered Doubler 或 Double-Shear Joint 構型時，便變的相當費時，因此使用現有之有限元素方法(FEA)軟體，可快速模擬鉚釘及板片安裝構型，建立分析模型(Analysis Model)，經由電腦運算可迅速得到板片及鉚釘間之負載分佈情形。本次課程利用波音公司提供之”IES Visual Analysis” FEA 套裝軟體，建立鉚釘及板片安裝構型之 2-D 模型，進行負載分佈運算。但未來在分析軟體的選用上，須注意該軟體對於待分析結構構型之適用性。

由於鉚釘及板片安裝構型相對單純，因此建立 2-D 模型即可完整模擬其負載分佈。而模擬方法如圖 9 所示，將其中上板片視為力學上之簡易長板樑(Simple Beam)，將每個安裝的鉚釘亦視為具有另一種性質的圓柱樑(Round Beam)，再連結至下板片，此下板片亦視為另一長板樑，由此建立一個組合樑之分析模型。然後利用相互力學作用原理，將上板片一端視為固定點，於下板片一端施力，拉動整個組合樑，由此分析各個樑柱元素(Element)受力情形。

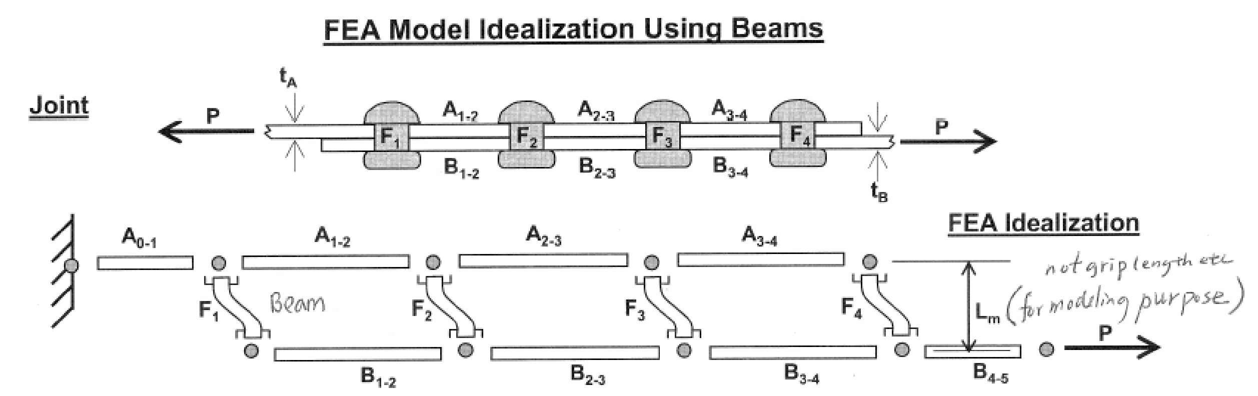


圖 9 鉚釘及板片安裝構型之 FEA 分析基礎示意圖

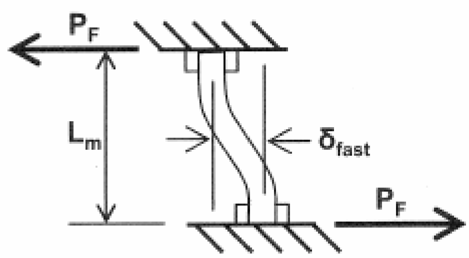
在建立每個樑柱元素時，同時輸入其材質(如：2024, 7075 等)及幾何形狀資料(板片厚度或鉚釘直徑等)。因為將鉚釘視為一圓柱樑，須先計算鉚釘在分析模型中之等效圓柱樑直徑(Equivalent Beam Diameter) D(model)後，再代入 FEA 模型中進行分析，D(model)計算公式如下：

(其中為分析方便起見，常假設 Lm=1)

$$\boxed{D_{model} = \sqrt[4]{\frac{L_m^{\,3} \cdot 64}{12 \cdot E_{fast} \cdot C_F \cdot \pi}}}$$

where:

L_m = distance between plates in FEA model
E_{fast} = fastener modulus of elasticity in model
D_{model} = diameter for a circular beam FEA element
C_F = fastener spring constant from Swift or D6-29942
δ = shear deflection of beam
P_F = fastener load



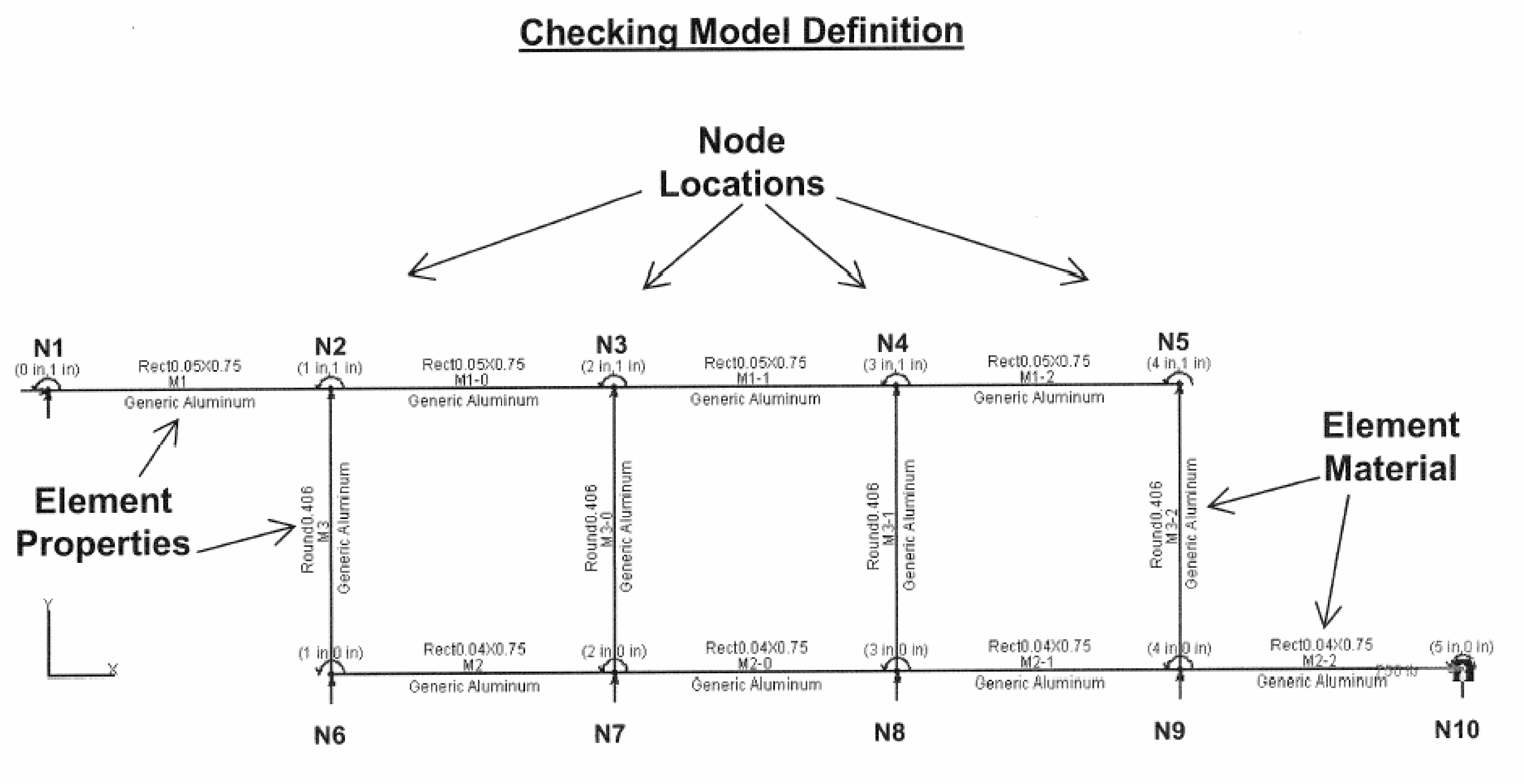
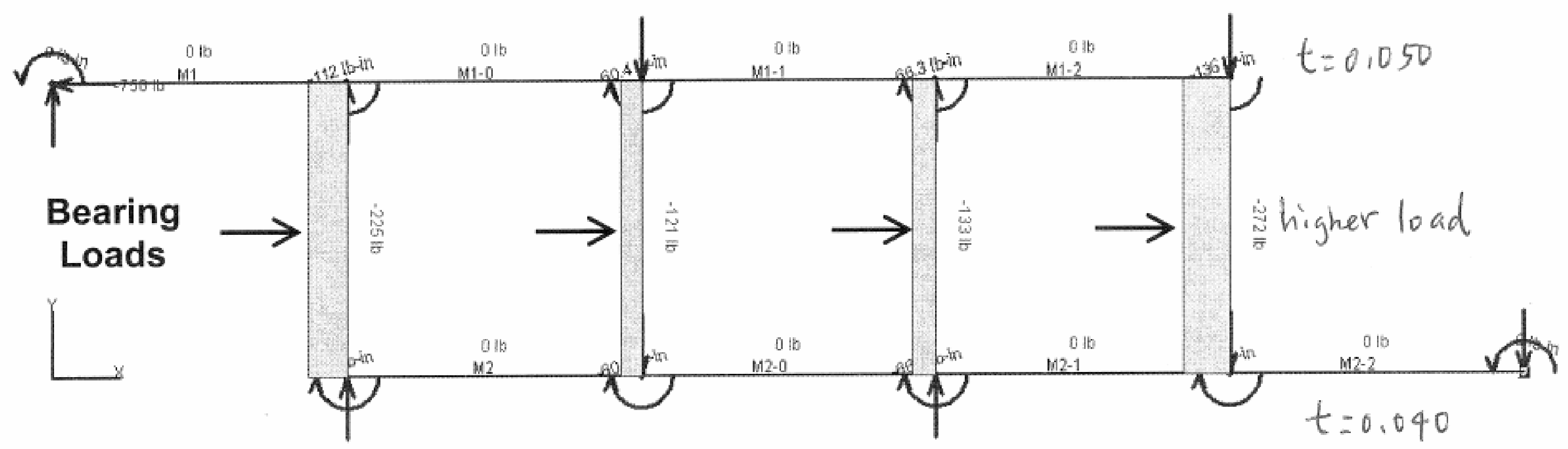


圖 10 鉚釘及板片安裝構型之 FEA Model

Reviewing Model Results – Fastener Bearing Loads



F1 + F2 + F3 + F4 = P

225 lb + 121 lbs + 133 lbs + 272 lbs = 750 lb

圖 11 鉚釘及板片安裝構型之 FEA 分析結果 – 鉚釘負載分佈

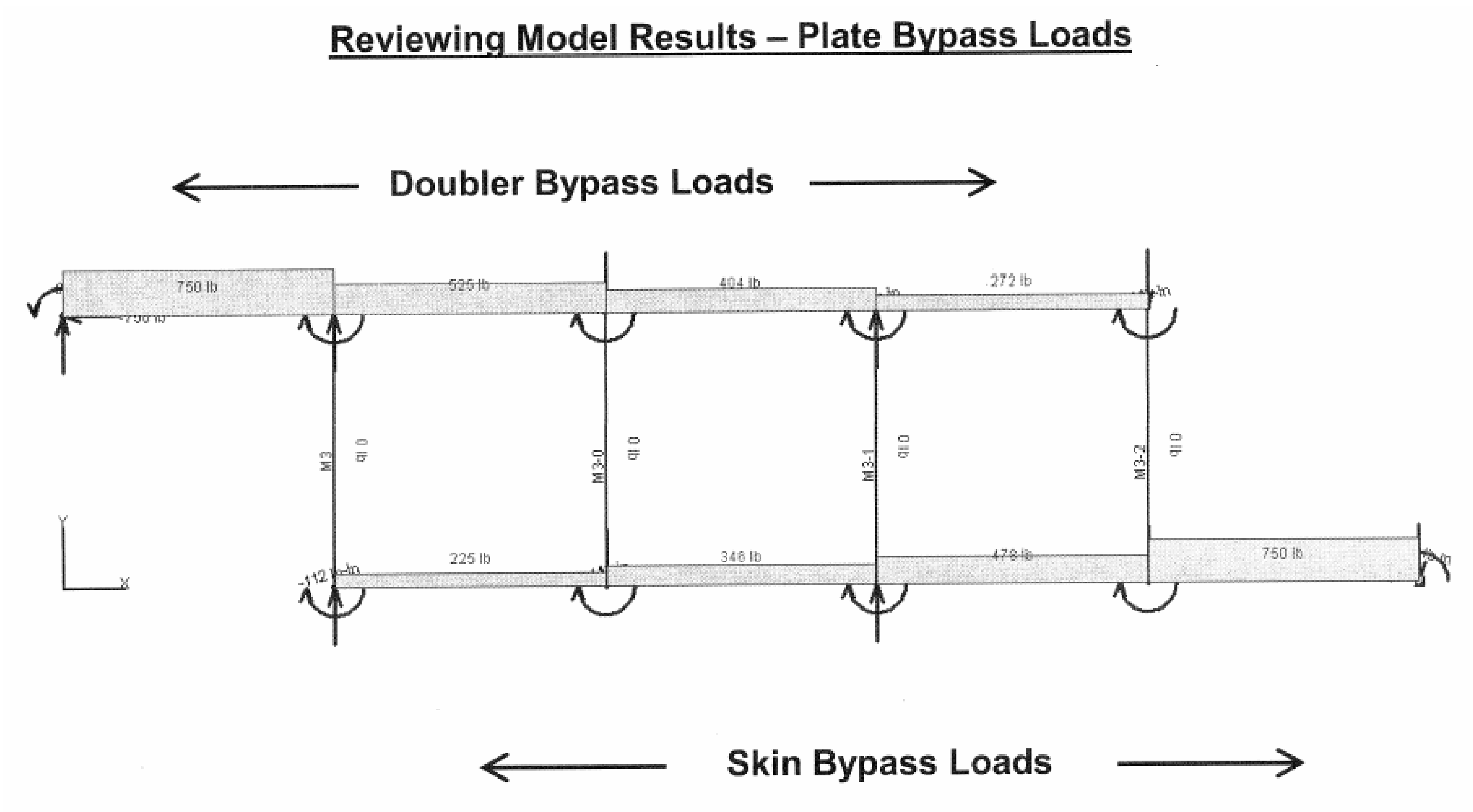


圖 12 鉚釘及板片安裝構型之 FEA 分析結果 – 板片負載分佈

(三) 分析修理方法之 Peak Stress：

1. 如前述第(一)節所述，當航空公司工程師因為飛機細部結構些微差異、進手空間限制及庫房可供使用鉚釘型號之限制等因素，無法完全依照 SRM Repair 進行修理時，可利用原製造廠 SRM Repair 做為飛機結構修理設計基礎，提出建議之修理方法，接下來再進行結構分析，比較所提建議修理方法與 SRM Repair 之 Peak Stress，以評估所提建議修理方法所產生之 Peak Stress 是否小於 SRM Repair Peak Stress。

2. 應力集中因子(Stress Severity Factor, SF)：

   由於結構細部構型變化甚多，因此利用 SF 因子用以評定結構細部構型之應力集中效應，該因子可估算複雜結構中的負載分佈，以評估重要結構項目之疲勞壽命。波音公司規範 D6-29942 中所列 SF 及 Peak Stress 計算公式如下：

Stress Severity Factor Formula

The severity factor (SF) relates the local peak stress (σ_{peak}) caused by fastener bearing load and plate bypass loads to a reference far-field gross stress (σ_{ref}).

$$SF = \frac{\alpha\beta_1 \cdot e_2 \cdot \gamma}{\sigma_{ref}}(\sigma_1 + \sigma_2)$$

$$\boxed{\sigma_{peak} = SF \cdot \sigma_{ref} = \alpha\beta_1 \cdot \gamma \cdot (\sigma_1 + \sigma_2) \cdot e_2}$$

where :

$$\sigma_1 = \frac{K_{tbr} \cdot P_{br} \cdot \theta}{D \cdot t} \quad \{\text{Bearing}\}$$ Fastener

$$\sigma_2 = \frac{K_{tbp} \cdot P_{bp} \cdot \cancel{\theta}}{w \cdot t} \quad \{\text{Bypass}\}$$ Plate (Skin / Doubler)

α = fastener hole condition factor
β_1 = hole filling factor
$\alpha\beta_1$ = 0.62 (Rivets)
= 0.65 (Hex-drive bolts in transition fit holes)

D = fastener diameter
w = width (in.), fastener spacing perpendicular to load
t = plate thickness

σ_{ref} = reference stress in the structure
σ_1 = peak stress caused by fastener bearing load, P_{br}
P_{br} = bearing load (fastener load transfer)
K_{tbr} = bearing stress concentration factor (Figure 1)
θ = bearing distribution factor (Figure 2)

σ_2 = peak stress caused by bypass loads, P_{bp}
P_{bp} = bypass load
K_{tbp} = bypass stress concentration factor (Figure 3)

γ = countersink factor (Figure 4)
e_2 = load transfer factor (Figure 5)

上述公式中以不同的參數表示不同的安裝構型(鉚釘型式、Countersink、Rivet、Hex-Drive Bolt、板片厚度、應力集中因子等)，以求得在不同板片與鉚釘安裝構型下之負載分佈情況，而公式中的參數值可由測試數據所得之不同圖表中查得。

3. 經由 Matrix Algebra 或 FEA 方法所求得鉚釘及板片安裝結構 Element 所受之負載後，可代入上述之 SF 公式，可求得各個鉚釘及板片所受之應力，利用此分析過程分別求得 SRM Repair 及所提修理建議方法之 Peak Stress 後，即可進行比較，若所提修理建議方法之 Peak Stress 小於 SRM Repair Peak Stress，則代表所提修理建議方法不會產生疲勞破壞，亦即此修理建議方法是可接受的。

(四) 分析修理方法之疲勞壽命(Fatigue Life)：

1. Mr. Tom Swift 在 FAA-AIR-90-01 “Repair to Damage Tolerant Aircraft”報告中引用 Mr. Lars Jarfall 提出的「有效應力“Effective Stress”」概念，亦即利用簡化的方法，在僅考慮鉚釘緊配安裝及無偏心情況下，可用以估算鉚釘及板片結合結構之疲勞壽命。Effective Stress σ_{eff} 計算方式如下：

$$\boxed{\sigma_{eff} = [1.3K_{tbr}\ \sigma_{br} + K_{tbp}\ \sigma_{bp}]/K_{tbp}}$$

Where σ_{eff} = effective stress

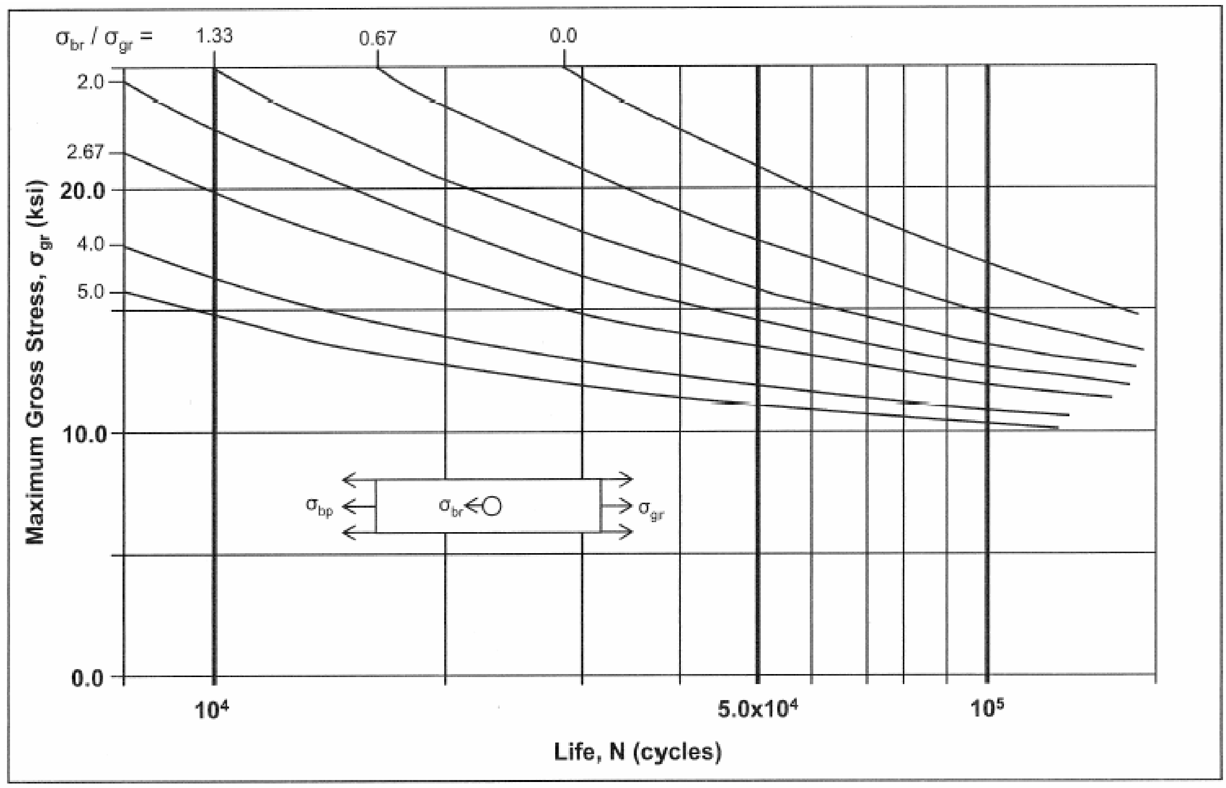
σ_{br} = bearing stress

σ_{bp} = bypass stress

K_{tbr} = stress concentration factor relating peak stress at edge of hole to σ_{br}

K_{tbp} = stress concentration factor relating peak stress at edge of hole to σ_{bp}

估算疲勞壽命的方法，為利用板片 Gross Stress (σ_{gr})與鉚釘之 Bearing Stress (σ_{br})比值，查詢測試數據所得出之圖表，可求得結構產生疲勞損壞之疲勞壽命，圖表如下：



Fatigue S-N Data 2024-T3 Clad Sheet, 1
Open or Loose Fit Holes, Stress Ratio = 0.0

Reference: FAA-AIR-90-01, Figure 13

四、飛機結構容損分析(Damage Tolerance Analysis, DTA)

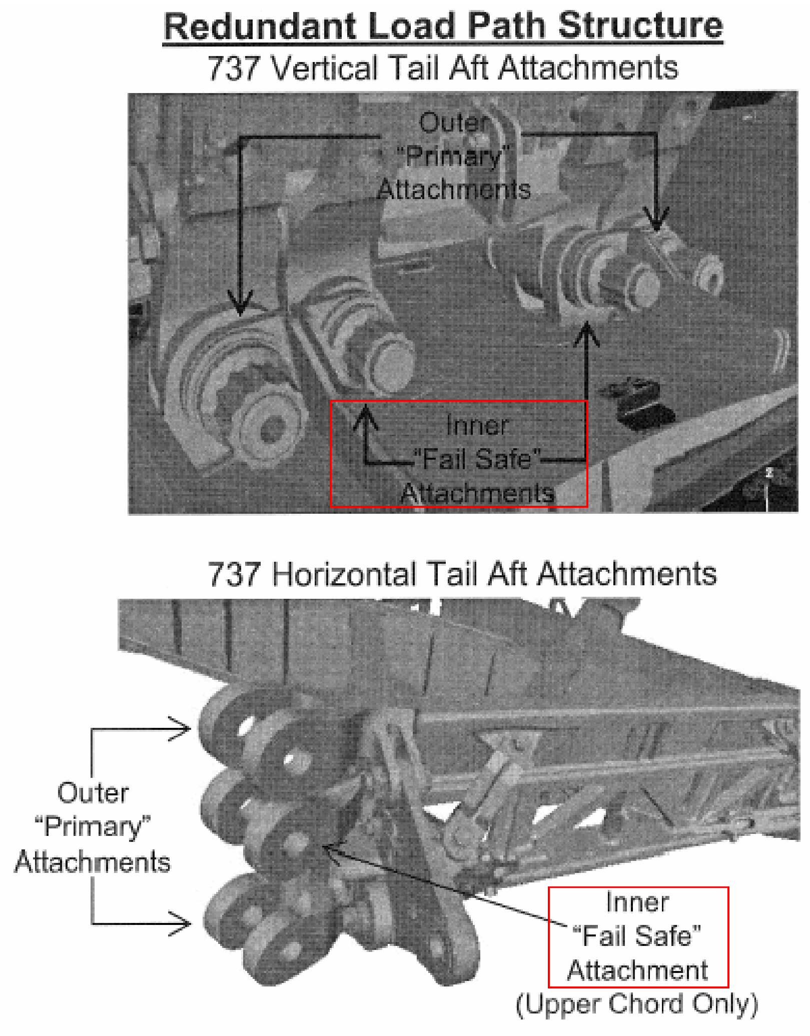
(一) 容損分析法規簡介：

FAA 於 1978 年在 FAR Part 25 之 Amendment 25-45 中，修訂 FAR 25.571 法規，將容損設計原則納入適航標準。過去是以分析飛機結構之疲勞壽命及失效安全(Fail-Safe)為主，但新的 FAR 25.571

法規考量結構在製造過程可能產生微小裂紋等損傷時，隨著飛機使用時間日異增長，含此損傷之結構剩餘強度(Residual Strength)須尚能承受外在負載，以免在下次檢查發現該損傷前，導致該處結構破壞，而危及飛機結構完整性影響飛航安全。因此飛機結構設計方向，自此導向容損設計概念，同時在後續飛機結構修理上，透過對修理構型進行容損分析(DTA)，確認修理方法滿足 FAR 25.571 的要求，以維持飛機之持續適航。

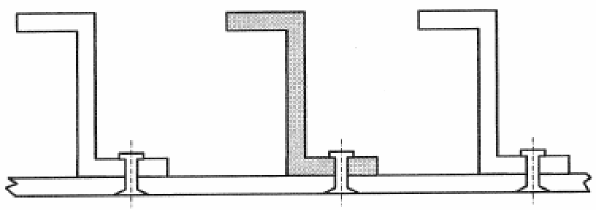
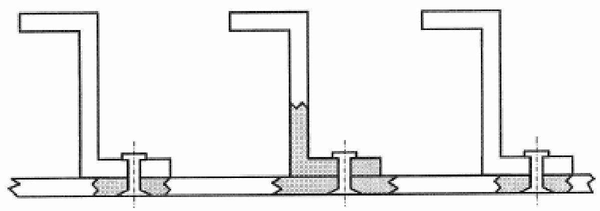
(二) 飛機結構設計原則及相關限制簡介：

1. 安全壽命(Safe Life)：當飛機結構實際產生疲勞的應力低於「疲勞限度(Endurance Limit)」時，基本上這樣的結構是不會損壞的，即所謂 Long Life 的設計，但此設計之機體重量將會太重而不合經濟效益。因此後來發展出 Safe Life 設計理念，透過疲勞試驗及疲勞分析，求出結構在一定反覆式操作負載下，在 Safe Life 使用循環周期內，飛機結構不會產生疲勞破壞。但 Safe Life 設計理念非常保守，並不允許結構初始狀態有裂紋存在，而且需要大量的疲勞試驗數據，因此非常費時及耗費成本。一般而言，飛機鼻輪起落架即是 Safe Life Parts。
2. 失效安全(Fail-Safe)：通常利用多重負載路徑(Multiple Load Path)的結構設計理念，亦即當其中一個結構組件損壞時，其他組件仍然可承受設計負載，使飛機整體結構不致於產生重大破壞，以允許在檢查時距(Inspection Interval)下，檢測出已損壞結構，並進行修理以恢復適航。以下是一些 Fail-Safe 結構設計例子：



3. 容損設計原則：當飛機結構因為疲勞(Fatigue)、銹蝕(Corrosion)、製造瑕疵(Manufacturing Defects)或意外損傷(Accidental Damage)等所產生損傷後，在該受損結構負載承受能力低於 Fail-Safe Load 前，可被檢查出並進行適當的修理，以回復結構強度，此即容損設計原則。整體來說，Fail-Safe 設計為容損設計之基礎，但容損設計增加對結構多重損傷、銹蝕及意外損傷之設計考量，並分析結構受損後的剩餘強度，據此訂出適合的檢查起始點(Inspection Threshold)及檢查時距。容損分析與失效安全之設計原則比較表如圖 13。

Fail Safe / Damage Tolerant Criteria

ANALYSIS	Fail - Safety FAR 25.571 (Pre – 1978)	Damage Tolerance FAR 25.571 (Post – 1978) Amdt. 25-45
Residual Strength	 • Single Element Or Obvious Partial Failure	 • Multiple Active Cracks (more realistic)
Crack Growth	• No Analysis Required	• Extensive Analysis Required
Inspection Program	• Based on service history • FAA Air Carrier Approval	• Related to Structural Damage Characteristics and Past Service History • Initial FAA Engineering and Air Carrier Approval

727, 737 Classic

圖 13 容損分析與失效安全之設計原則比較表

(三) 容損分析(DTA)三大要素：

1. 結構剩餘強度分析(Residual Strength Analysis)：

   當飛機結構產生損傷後，須決定其他結構組件所能承受之最大負載，此即結構剩餘強度(Residual Strength)，而此最大負載須低於 Fail-Safe Loads (大於或等於 Limit Loads)，以使得結構在損壞後，其他結構組件仍能承受 Fail-Safe Loads，並由此計算出 Critical Crack Length，此裂紋長度亦即可承受 Fail-Safe Loads 之最大裂紋。

2. 裂紋成長率分析(Crack Growth Rate Analysis)：

   分析可檢測裂紋(Detectable Crack)成長至 Critical Crack Length 之速率，即為裂紋成長率分析。

3. 檢查需求制訂(Inspection Requirements)：

   從裂紋成長率分析結果所得出 Crack Length v.s. Cycles 曲線，由此曲線可決定後續結構檢查起始點及檢查時距，檢查時距之訂定以各檢查方法可檢測出損傷之機率進行估算，以定義出最適當的檢查時距，例如：以可檢測微小裂紋之 High Frequency Eddy Current (HFEC)為檢查方法時，由於其檢測能力佳，其檢查時距可以拉長。

(四) 疲勞裂紋成長及破壞力學(Fracture Mechanics)：

容損分析在於當結構產生微小損傷後，基於損傷成長模式的了解，透過破壞力學的理論，決定 Critical Crack Length，然後計算在此最差的條件，結構的剩餘強度，同時經由破壞力學的理論決定此裂紋之成長速率，進而得出該何時進行檢查，用什麼檢查方法，才可在裂紋未成長至對飛機結構產生重大損害(Catastrophic

Failure)前，發現裂紋並加以修理，以恢復其承受 Ultimate Load 的能力，因此了解疲勞裂紋成長模式及相關破壞力學原理，亦為進行容損分析前重要的學習課題，因此本訓練課程亦講授相關觀念。

1. 疲勞疲勞裂紋成長模式：

   疲勞裂紋的成長分為下列二個階段：Crack Initiation Phase 及 Crack Growth Phase，如圖 14 所示，其中第一階段為微小裂紋成長至可首次可檢出裂紋(First Detectable Crack)，而第二階段為從 First Detectable Crack 成長至結構破壞失效為止。

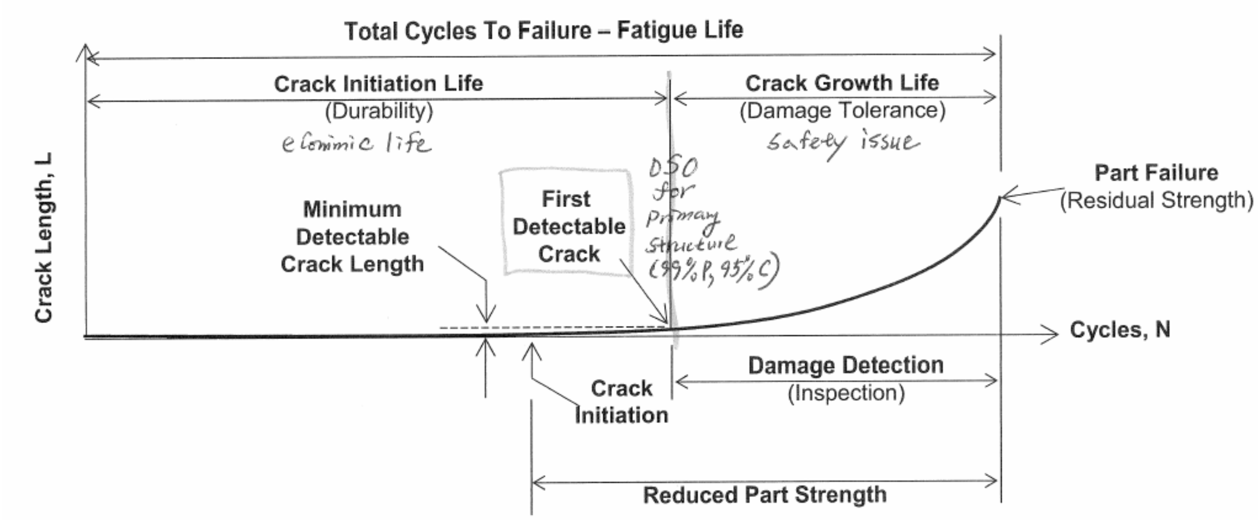


圖 14 裂紋成長階段

而第一階段又可分為下列三個進程：(1) Cyclic Slip (2) Crack Nucleation (Formation or Initiation) (3) Micro Crack Growth，進而產生 First Detectable Crack，如下圖 15 所示。

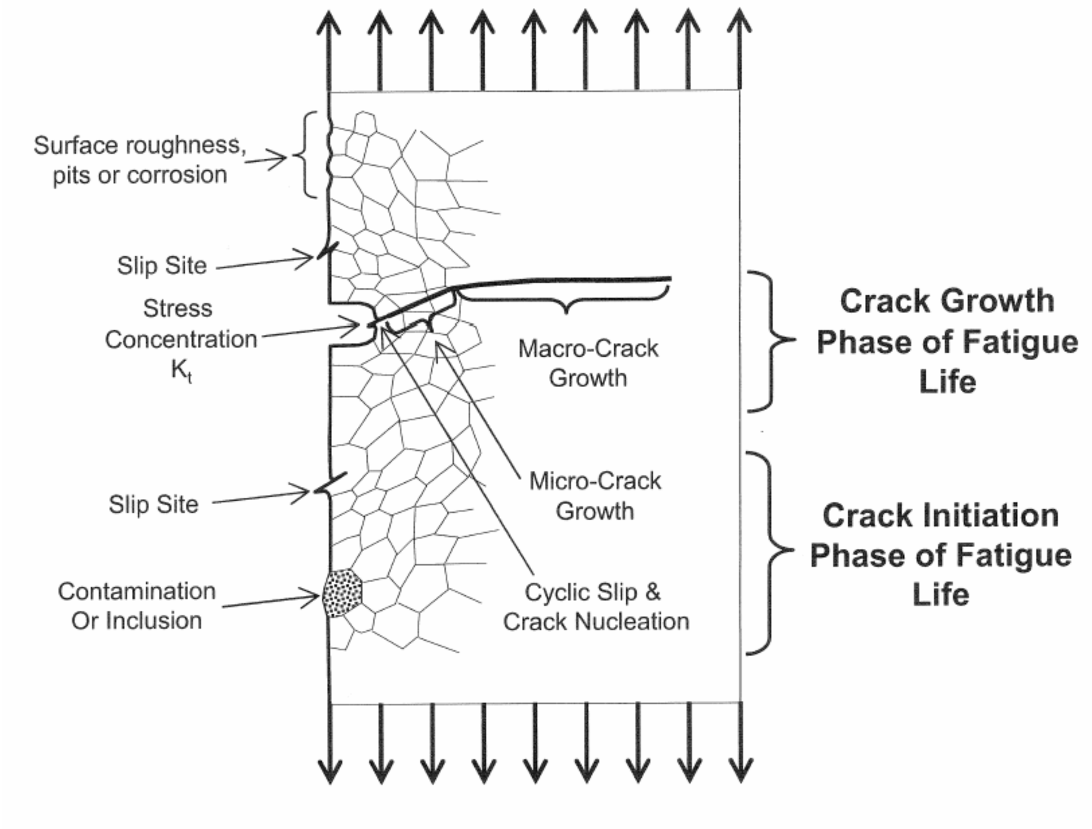


圖 15 疲勞裂紋成長模式

此階段裂紋成長速率將隨著結構表面粒度、環境因素(濕度、鹽份、燃油等)、應力集中因子 Kt，以及所受外力等因素而定，而裂紋將會沿著晶粒中最大剪力平面的方向成長。

此後裂紋成長進入第二階段：Crack Growth Phase，此時由 Micro Crack Growth 轉為 Macro Crack Growth，此階段裂紋成長速率將隨著材料性質，以及所受外力等因素而定，但此階段的裂紋成長速率大過於第一階段，因為 Crack Tip 會產生應力集中，且隨裂紋成長尺寸加大，結構受力截面積隨之遞減，因而局部應力變得更大而加速裂紋成長，直到結構破壞為止。

而 Crack Growth Phase 亦可分為三個區間：(1) Threshold Region (2) Paris Region (3) Stable Tearing Region。如圖 16 所示：

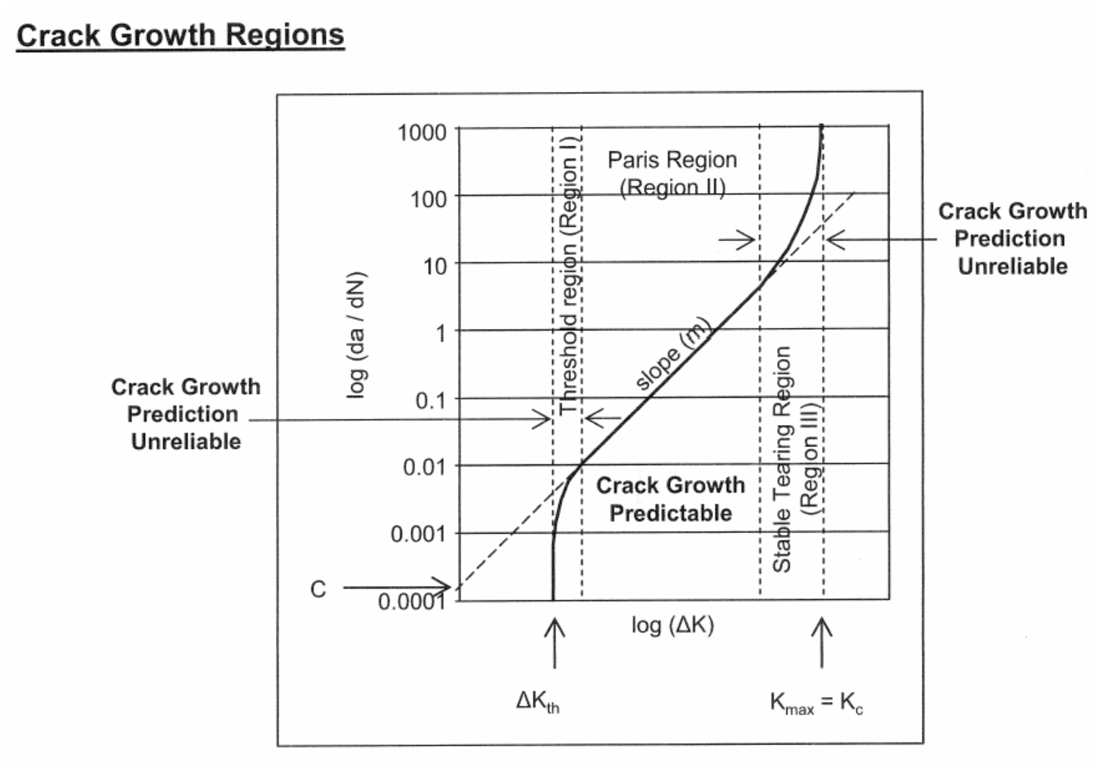


圖 16 裂紋成長第二階段：Crack Growth Phase 之三個區間

其中第一區間 Threshold Region 的裂紋成長速率快，但裂紋尺寸仍小，而第二區間 Paris Region 的裂紋成長趨緩，但其裂紋尺寸較具可檢測性，可使用非破壞檢測方法(Non-Destructive Inspection, NDI)量得裂紋尺寸，且此區間之裂紋成長模式，可用公式加以歸納之，由 Paris, Gomez 及 Anderson 所導出的 Paris Equation 即是其中一例，Paris Equation 公式如下：

$$da / dN = C \cdot \Delta K^m \qquad \text{Paris Equation}$$

where:
- da = change in crack length
- dN = number of cycles to achieve da
- C = a material constant
- ΔK = change in stress intensity for da
- m = slope of the curve of $\log(da/dN)$ versus $\log \Delta K$, a material constant

第三區間 Stable Tearing Region 的裂紋成長速率又加快，其成長速率由破壞力學中的材料破壞韌性(Fracture Toughness)、平均應力(Mean Stress)及環境因素所決定。

一般在容損分析時，常假設初始裂紋長度(Initial Crack Length)為 0.05”，然後開始計算從 Initial Crack Length 成長至 Critical Crack Length 期間的週期循環次數，藉以定義飛機結構檢查時距。依照一般容損設計準則，檢查時距之訂定，須安排在結構最終破壞失效前，至少有 2 次可檢查出裂紋的機會。

2. 破壞力學相關原理：

在進行裂紋成長計算之前，須先了解 Crack Tip Stress 與 Gross Stress 之關連性，依據破壞力學之父 Mr. A. A. Griffith 及 Mr. G. R. Irwin 提出「線彈性破壞力學(Linear Elastic Fracture Mechanics, LEFM)」理論，可利用「應力強度因子(Stress Intensity Factor)」定義裂紋尖端應力，而應力強度因子(Stress Intensity Factor), ”K”可依下列公式求得：

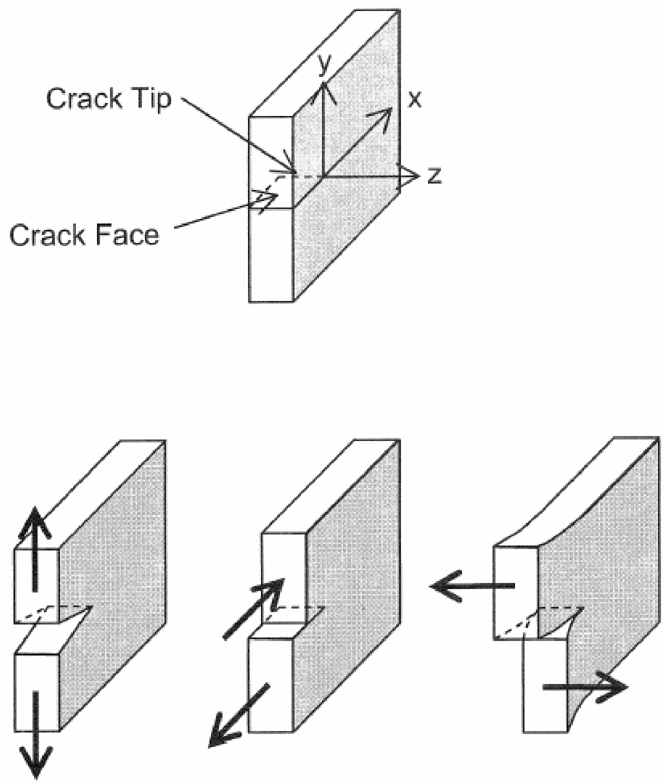
Stress Intensity Factor

$$K = \beta \cdot \sigma \sqrt{\pi \cdot a} \quad (\text{ksi} \cdot \sqrt{\text{in.}})$$

where: β = geometry factor (unit-less)
σ = far field stress (ksi)
a = crack length (in.)

而裂紋成長模式(Crack Extension Modes)共有三種，圖示如下。但飛機結構裂紋的成長模式為第一種模式(Mode I)，亦即裂紋成長方向與受到張力(Tension)方向垂直。

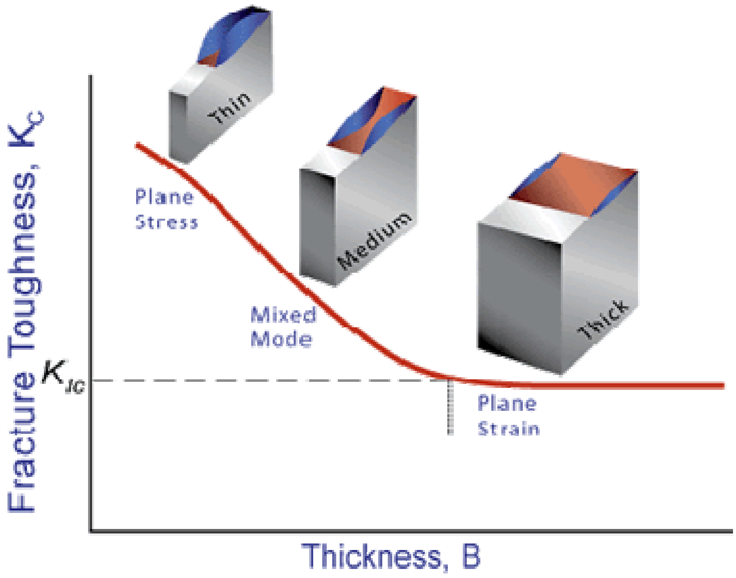
Crack Extension Modes



Mode I (Opening) **Mode II** (Sliding) **Mode III** (Tearing)

fatigue happen most in this mode.

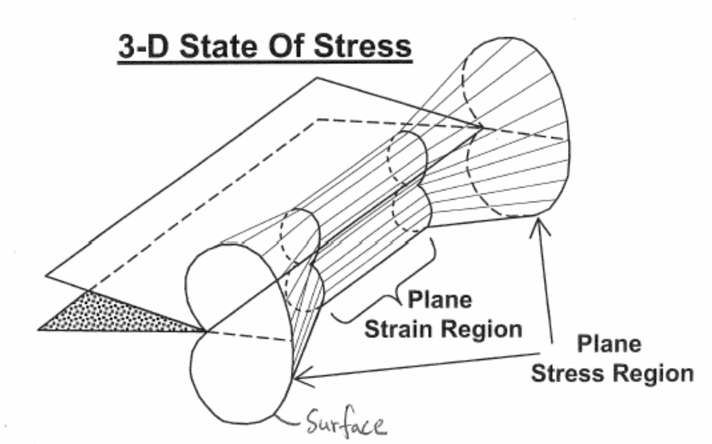
「破壞韌性(Fracture Toughness)」為衡量材料抵抗破壞的能力，為一種材料特性，通常以 Kc 表示之。此因子可用以表示裂紋快速成長時裂紋尖端的應力情形，會受到溫度及外力大小的影響。Fracture Toughness 為材料在某種特定條件下(Plain Stress, Plain Strain 等)，可承受外力不致破壞之最高應力強度因子(Stress Intensity Factor), K，圖示如下。



從上圖可知當材料到達一定厚度值時，Kc 呈一常數值，以 K_{IC} 表示，亦即代表材料在 Plain Strain 的條件下，抵抗材料產生破壞的最大能力，亦即 Stress Intensity Factor, K 之最大值，其中 I 代表在裂紋成長模式 Mode I 下所求得之 Kc。

所謂 Plain Strain (平面-應變)及 Plain Stress (平面-應力)分為代表裂紋在材料內部及材料表面的受力後之應力及應變情形。其中在材料內部，由於受到周圍材料之束縛，裂紋係延一定平面而產生應變，尤其當材料厚度越厚時，材料內部產生此

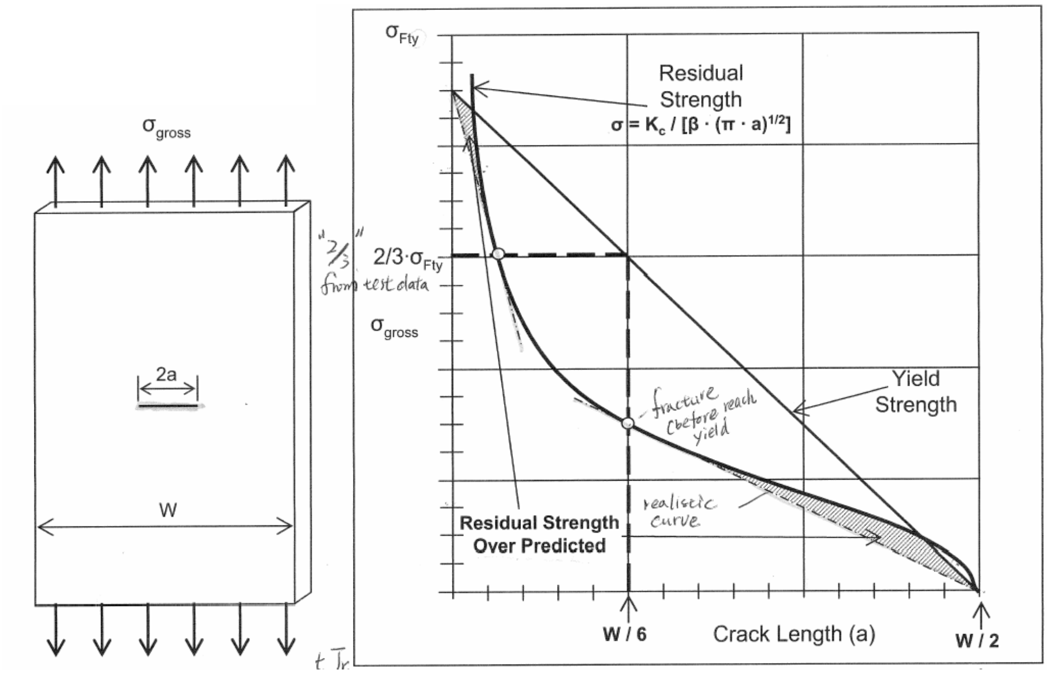
現象越明顯。而在材料表面因為是自由面(Free Surface)，而使應力延一定平面產生，當材料厚度越薄時，材料表面產生此現象越明顯。Plane Stain Region 及 Plane Stress Region 圖示如下：



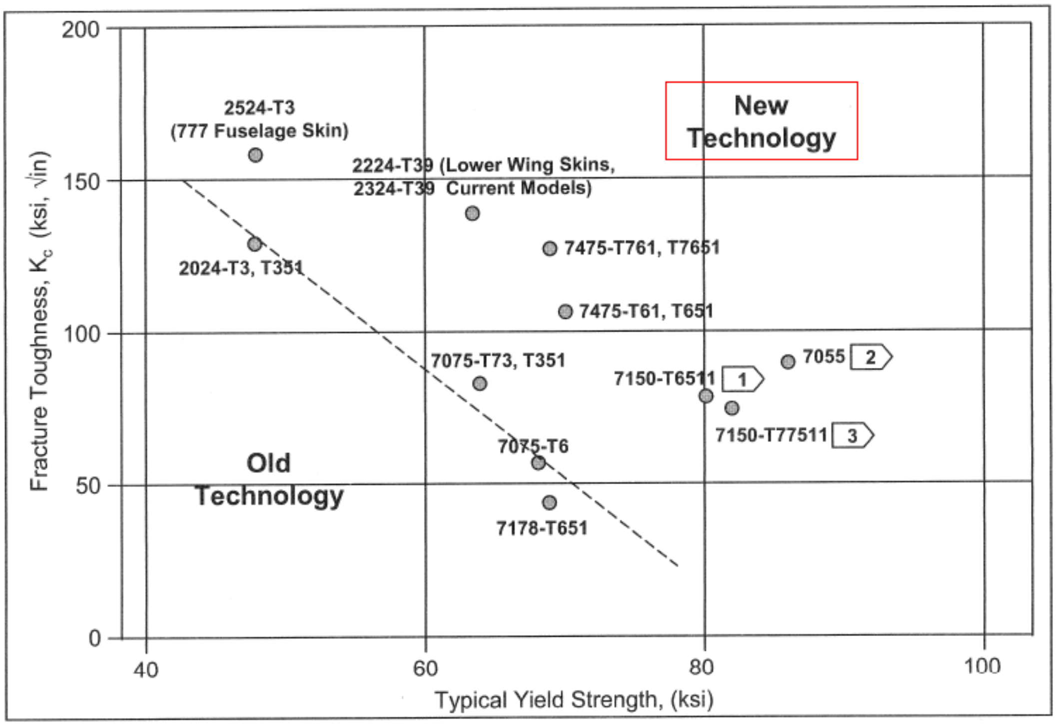
利用 Fracture Toughness Kc 可計算出 Critical Crack Length，整理計算 Stress Intensity Factor, K 之公式，可得到下列公式計算產生 Critical Crack Length 時的 Critical Stress σ：

$$\sigma = K_c / [\beta \cdot (\pi \cdot a)^{1/2}]$$

由上述公式可得出下列圖表，在進行容損分析時可用以決定 Critical Crack Length 及相對應之 Critical Stress σ，此圖表同時評估材料之 Yield Stress 及 Residual Stress，以決定實際的 Critical Stress σ：



此外，Fracture Toughness 與材料 Yield Strength 成反比，例如：7075 鋁合金 Yield Strength 比 2024 鋁合金 Yield Strength 高，則 7075 鋁合金 Fracture Toughness 則較弱，而 2024 鋁合金較具延展性，也具有較高之 Fracture Toughness，因此裂紋成長速率也較 7075 鋁合金慢。目前波音公司已開發新的鋁合金材料，可提昇材料的 Fracture Toughness，如：2524 鋁合金、7150 鋁合金，即分別為 2024 鋁合金及 7075 鋁合金的 Fracture Toughness 提昇版本，除可增強飛機結構容損能力外，亦可減輕飛機結構重量。圖 17 則列出過去及近幾年開發之鋁合金性質圖表，可比較不同鋁合金之 Yield Strength 及 Fracture Toughness 差異。



NOTES: Approximate K_c and F_{ty} properties for comparison purposes only

1 Upper Wing Skins, 737-300 / 400 / 500, 747-400, 757

2 Upper Wing Skins, 737-600 / 700 / 800, 777

3 Body Stringers, 777

圖 17 鋁合金 Yield Strength 及 Fracture Toughness 圖表

3. 裂紋成長計算：

目前有許多學者提出不同的裂紋成長計算公式，說明如下。

(1) Paris Equation：

Paris Equation 如下，其缺點在於由於公式僅考慮應力強化因子之差值ΔK，因此無法區分不同平均應力(Mean Stress)下的裂紋成長。

Paris Crack Growth Equation

– 1961 - P. C. Paris presented the following equation for crack growth rate as a function of ΔK

$$\frac{da}{dN} = C \cdot (\Delta K)^m \Rightarrow \log\left(\frac{da}{dN}\right) = \log(C) + m \cdot \log(\Delta K)$$

(2) Walker Equation：

Walker Equation 如下，此公式導入了應力比值(Stress Ratio) R，此為最小與最大應力比，因此可以處理不同平均應力下的裂紋成長計算，當 R=0 時，此公式即成為 Paris Equation。

Walker Crack Growth Equation

Boeing use modified Walker equation

– **In 1970, K. Walker published a modification to the Paris equation that accounts for increasing crack growth rate with increasing stress ratio, R**

$$\frac{da}{dN} = C\left[\frac{\Delta K}{(1-R)^{(1-n)}}\right]^m \Rightarrow \log\left(\frac{da}{dN}\right) = \log(C) + m \cdot \log(\Delta K) - m(1-n) \cdot \log(1-R)$$

(3) Forman Equation：

Forman Equation 如下，該公式增加對於裂紋成長第二階段之第三區間 Stable Tearing Region 之裂紋成長計算功能。

Forman Crack Growth Equation

– **Forman published a modification to the Paris equation that accounts for different stress ratio, R, and incorporates Region III crack growth behavior**

$$\frac{da}{dN} = \frac{C\Delta K^m}{(1-R) \cdot (K_c - \Delta K)}$$

五、飛機容損結構檢查及修理

(一) 飛機容損結構檢查方法：

一般飛機容損結構檢查方法分為兩類：第一類為目視檢查(Visual Inspection)，第二類為非破壞檢查(Non-Destructive Inspection, NDI)。其中目視檢查又可分為General Visual Inspection及Detailed Visual Inspection 二種方法；而常用之非破壞檢查又可分為 Dye Penetrant Inspection、Magnetic Particle Inspection、Radiography (X-Ray) Inspection、Ultra-Sonic Inspection 及 Eddy Current Inspection 五種方法。

1. 一般目視檢查法(General Visual Inspection)：

   為一般最方便容易的檢查法，通常僅利用一般光線(Daylight, Hangar Light 等)，加上鏡子等簡易工具即可進行，但無法檢測之細小裂紋。

2. 詳細目視檢查(Detailed Visual Inspection)：

   利用鏡子、放大鏡、內視鏡、電腦影像等輔助裝備，在輔助加強光源下，進行結構詳細檢查，並視需要進行檢查前之結構表面清潔，以利後續檢查工作之進行。

3. 液滲檢查(Dye Penetrant Inspection)：

   為低成本、簡易之非破壞檢查法，主要針對非多孔性金屬之表面裂紋之檢查法。而使用之滲透液有 Fluorescent Dye 及 Visual Dye (僅限使用於非機體結構上)，可檢測 0.05”以下細小裂紋。本檢查法缺點是需多道處理程序、結構表面有清潔要求，以及對於端角、多孔性金屬及粗糙表面等之檢查結果並不穩定。

4. 磁檢(Magnetic Particle Inspection)：

   適用於鐵磁性金屬(通常為低碳鋼、可熱處理之不銹鋼等)之非

破壞檢查法。可檢測出表面及次表面之裂紋。本檢查法缺點在於進行重要結構表面檢查前須先去除表面塗層，且無法探知位裂紋深度。

5. 放射線檢查(Radiography (X-Ray) Inspection)：

   經常使用於已知結構裂紋模式及位置之結構檢查，可檢測表面及內部結構損傷，已知結構裂紋模式及位置的來源係得自於疲勞試驗的結果，或是使用中飛機結構裂紋資料。放射線檢查亦可用於檢查嚴重之結構銹蝕。本檢查法缺點在於放射線設備具危險性，以及檢查結果判讀較複雜。

6. 超音波檢查(Ultra-Sonic Inspection)：

   利用超音波穿透(Through Transmission Ultrasonic, TTU)或反射(Pulse Echo)回波，以檢測結構裂紋，適用於大部份飛機結構金屬，為一方便簡易之非破壞檢查法。本檢查法缺點在於待檢測表面須具探頭可及性以進行檢查，以及待檢測之裂紋有方向性要求。

7. 渦電流檢查(Eddy Current Inspection)：

   此方法利用設備產生電流並量測電流變化，進行結構缺點檢查，其原理為利用交流電壓使探頭產生磁場，依法拉第定律，此磁場會產生渦電流，而渦電流則再引發額外磁場，造成磁阻(Impedance)，可經由設備量測後於螢幕顯示檢測結果。渦電流檢查為飛機容損結構裂紋之重要檢查法，依電流頻率可區分為：高頻渦電流檢查(High Frequency Eddy Current, HFEC)、中頻渦電流檢查(Medium Frequency Eddy Current, MFEC)及低頻渦電流檢查(Low Frequency Eddy Current, LFEC)三種。

   (1) 高頻渦電流檢查(HFEC)：電流頻率 50-500 kHz，可檢測

表面缺點，可檢測最小至 0.1"之表面裂紋，亦可以旋轉方式檢查鉚釘孔。

(2) 中頻渦電流檢查(MFEC)：電流頻率 30-50 kHz，可檢測至第一層次表面缺點，可檢測最小至 0.1"之表面裂紋，以在 0.02"深度以內的 0.25"次表面裂紋。

(3) 低頻渦電流檢查(LFEC)：電流頻率小於 30 kHz，可檢測至第二層及第三層次表面缺點，適用於檢查結構內部損傷。對於 Doubler，可檢測出 0.036" ~ 0.220"之裂紋；對於蒙皮，則可檢測出 0.036" ~ 0.100"之裂紋。三種渦電流檢查方法之示意圖如圖 18 所示。

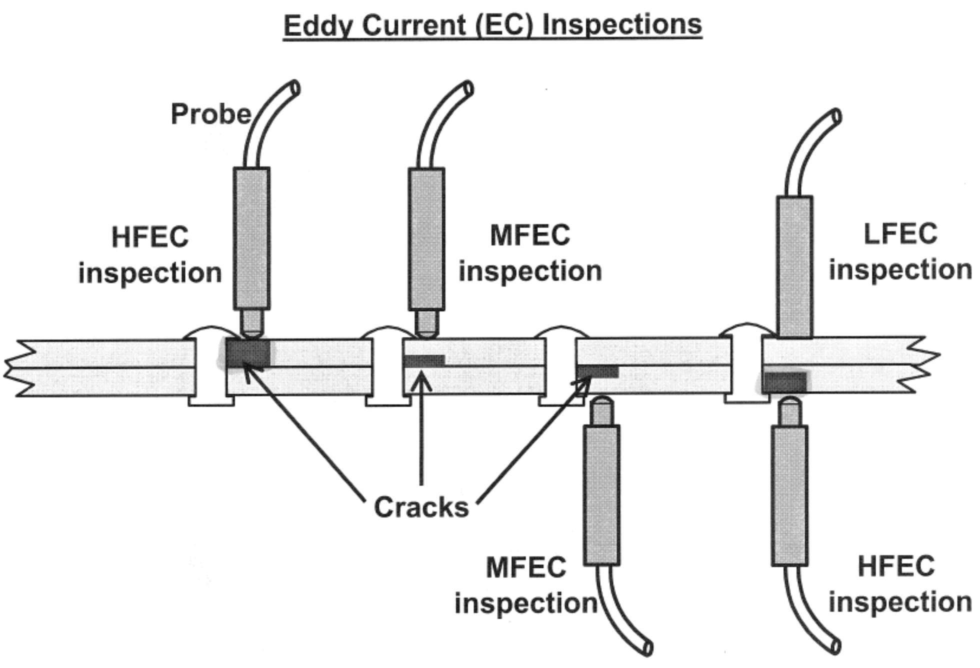


圖 18 三種渦電流檢查法可檢測裂紋位置示意圖

(4) 開放孔位檢查(Open Hole Inspection)：

當修理結構孔位進行「孔位歸零(Zero-Timing Holes)」工作時，必須先進行開放孔位檢查，以確認結構孔位裂紋

情況，所使用之檢查設備為 Bolt Hole HFEC，可檢測之最小孔位裂紋為 0.030”。開放孔位檢查須將所有鉚釘移除，執行 HFEC 前，須先以標準 0.030” Notch 進行校正後。檢查之進行可分為手動檢查或使用 Rotary Scanner，其中 Rotary Scanner 檢查法較快，檢測結果也較穩定可靠，適合執行大量孔位檢查時機。

8. 裂紋檢查時距研討：裂紋檢查時距(Inspection Interval or Damage Detection Period)之訂定，視該檢查方法可檢出最小裂紋尺寸而定，當可檢出最小裂紋尺寸越小，檢查時距則較長，圖 19 則說明可檢出裂紋長度與檢查時距之關係圖，圖 20 列出各種檢查方法可檢出之最小裂紋尺寸。

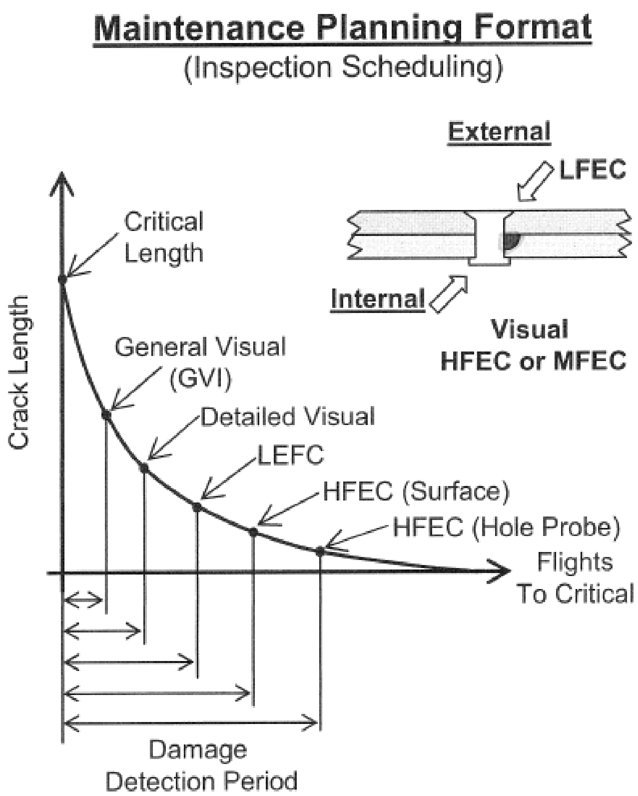


圖 19 可檢查裂紋長度與檢查時距關係圖

Method	Description	Detectable Crack Length (in.)
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
Radiography (X-Ray)	Uncovered length of crack in aluminum (not covered by steel)	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 in.) at 5 to 10 Mhz	0.125 long x 0.0625 Deep
	Crack in Clevis or Lug	0.125 x 0.0625 Deep
Ultrasonic Longitudinal Wave (Straight Beam)	Bolts	1/4 to 1/3 Diameter
	Crack in Fastener Hole	0.125
Bolt Hole Edge Current (Fastener Removed)	Edge Corner Crack	0.030 x 0.030
	Inside Diameter Surface	0.060 long x 0.030 Deep
Eddy Current Surface Probe	Crack at Fastener	0.0625 Uncovered Length
	Crack Away From Fastener	0.125

圖 20 各種檢查方法可檢查之最小裂紋尺寸

9. 結構檢查補充計畫(Supplemental Structural Inspection Program, SSIP)：

SSIP 係針對在 1978 年容損設計適航標準(FAR 25.571 Amendment 25-45)生效前所檢定飛機，所訂定的結構檢查補充計畫，計畫內容相較容損設計適航標準涵蓋範圍為少。

(1) SSIP 計畫內容訂定的假設前題：SSIP 並非涵蓋所有 FAR 25.571 Amendment 25-45 之考量條件，而是有限度地針對結構重要項目(Structurally Significant Items, SSIs)進行監控檢查，SSIP 計畫內容訂定的假設前題說明如下：

- SSIP 僅考慮疲勞裂紋(Fatigue Cracking)。
- SSIP 僅包含結構重要項目(SSIs)，但並非所有 SSI 項目皆須檢查，SSIP 僅將含不確定性疲勞特性之 SSI 項目列入補充檢查對象。一旦波音研究出其疲勞特性後，即會以技術通報(Service Bulletin, SB)內容取代 SSIP 訂定之補充檢查項目。
- 所有受影響飛機在執行完最近一次適航指令(Airworthiness Directive, AD)檢查後之一定時間(Inspection Threshold)起，開始執行 SSIP 所要求之結構補充檢查工作。剛開始 SSID 僅針對沒有執行修理的 SSIs 進行檢查，並不包含修理區域，後來 FAA 發佈 AD 強制要求所有 SSIs 皆須進行結構補充檢查。
- SSIP 並不包含銹蝕損傷議題，此議題由另一份「銹蝕預防與管制計畫(Corrosion Prevention and Control Program, CPCP)」文件予以規範。

(2) SSIP 補充檢查內容之制訂基礎，與容損分析同樣是基於裂紋成長及剩餘強度之理論。

(3) 結構檢查補充文件(Supplemental Structural Inspection Document, SSID)：SSIP 所制訂的檢查時距及檢查要求皆被列入 SSID 文件中，並由航空業界所組成的各機型 Structure Task Group 定期審視修訂，並經由民航主管機關核准(例如：波音所設計飛機之 SSID，由美國聯邦航空總署 FAA 核准)。

(4) SSIP 與容損設計適航標準(FAR 25.571 Amendment 25-45)之比較如下：

DAMAGE TOLERANCE AND SSIP COMPARISON

	Damage Tolerance	SSIP
Airplane Models Affected	757, 767, 777, 737 BBJ & 737-900 (Certified After 1978)	707, 727, 737, & 747 (Certified Before 1978)
Regulation	FAR 25.571, Amendment 45	AC 91-56 and AD's For Each model
Inspection Basis	Crack Growth & Residual Strength	Crack Growth & Residual Strength
Damage Included	Fatigue, Corrosion & Accidental Damage	Fatigue
Airplanes Included	All Airplanes	All Airplanes *

* Per Latest AD's

SSIP DOCUMENTS (SSID)	
707	D6-44860
727	D6-48040-1
737	D6-37089
747	D6-35022
747-SR	D6-35655

(二) 飛機容損結構分類：

波音將飛機容損結構分類，依安全性及經濟性等考量可區分為四類：Category 1, 2, 3, 4，其中 Category 1 為 Secondary Structure，而 Category 2, 3, 4 為 Primary Structure。亦即整體飛機分為 Primary Structure 及 Secondary Structure 兩種。

而 Primary Structure 即為 Structurally Significant Items (SSIs)，在容損設計適航標準生效後，法規又定義了 Principal Structure Element (PSE)，眾多的結構分類原則分述如後。

1. Category 1 Structure：

   為 Secondary Structure，並非主要影響持續飛航或非承受飛機負載之主要結構項目。

2. Category 2 Structure：

   為 Primary Structure/SSIs 中，其結構裂紋可由明顯易見的損傷現象檢測而得，例如：燃油滲漏等。

3. Category 3 Structure：

   為 Primary Structure/SSIs 中，須經由妥善規劃之檢查計畫始可檢出裂紋之結構項目。此類型結構項目，於檢查時常無法進手，因此需特定之檢查計畫，而此檢查計畫是經容損分析中裂紋成長分析、結構剩餘強度分析及考量檢查需求所制訂而得。

4. Category 4 Structure：

   為 Primary Structure/SSIs 中，無法經由妥善規劃之檢查計畫檢出裂紋，或是其裂紋成長速率過快之結構項目。此類型結構項目，即是所謂年限管制件(Life-Limited Parts, LLP)，而其設計是採用疲勞試驗所訂出的安全壽命(Safe Life)為其設計原則。

5. 主要結構項目 Principal Structure Element (PSE)：

   所有 Primary Structure 又可分為 PSE 及 non-PSE 項目。有關 PSE 定義列於 AC 25.571-1C (目前最新之版別) 中，摘錄如下：

” PSE, is an element that <u>contributes significantly</u> to the carrying of <u>flight, ground, or pressurization loads</u>, and whose integrity is essential in <u>maintaining the overall structural integrity</u> of the airplane.”

而 PSE 包含了大部份的 Category 3 Structure、Category 4 Structure 的 Landing Gear，以及一些 Category 2 Structure。在 SRM 中會列出 PSE 結構項目。圖 21 總結上述飛機結構之分類項目。圖 22 簡要說明飛機容損結構分類及維護檢查需求制訂之流程。

Structural Category			Safety Analysis Requirements		Maintenance Considerations		
			Required Design Attributes	Technology Control Method	Primary Purpose	Requirements	Planning Basis
Other Structure	Damage Tolerant Design	1 Secondary Structure	Design for loss of component or safe separation	• Continued safe flight	Economics	Scheduled maintenance tasks for detection and repair or prevention of damage	Previous experience when similar to existing structure. Manufacturer's recommendations when new material and/or concept
Structurally Significant Items (Primary Structure)	Damage Tolerant Design	2 Damage Obvious or Malfunction Evident	Adequate residual strength with extensive damage-obvious during walkaround or indicated by malfunction	• Residual strength			
Structurally Significant Items (Primary Structure)	Damage Tolerant Design	3 Damage Detection by Planned Inspection	Inspection program matched to structural characteristics	• Residual strength • Crack growth • Inspection program	Safety	Adequate inspections for timely detection and repair or prevention of damage	Manufacturer controlled rating systems • AD[1] • ED[1] • Fatigue damage[2]
Structurally Significant Items (Primary Structure)	Safe Life	4 Safe Life Design	Design for conservative fatigue life	• Fatigue analysis verified by test	Safety	Detection and repair or prevention of accidental damage and corrosion	Manufacturer controlled rating system

1. Applicable through operational life for Accidental Damage (AD) and environmental deterioration (ED).
2. Applicable after aircraft reaches threshold for detectable size fatigue damage.

圖 21 飛機結構之分類項目

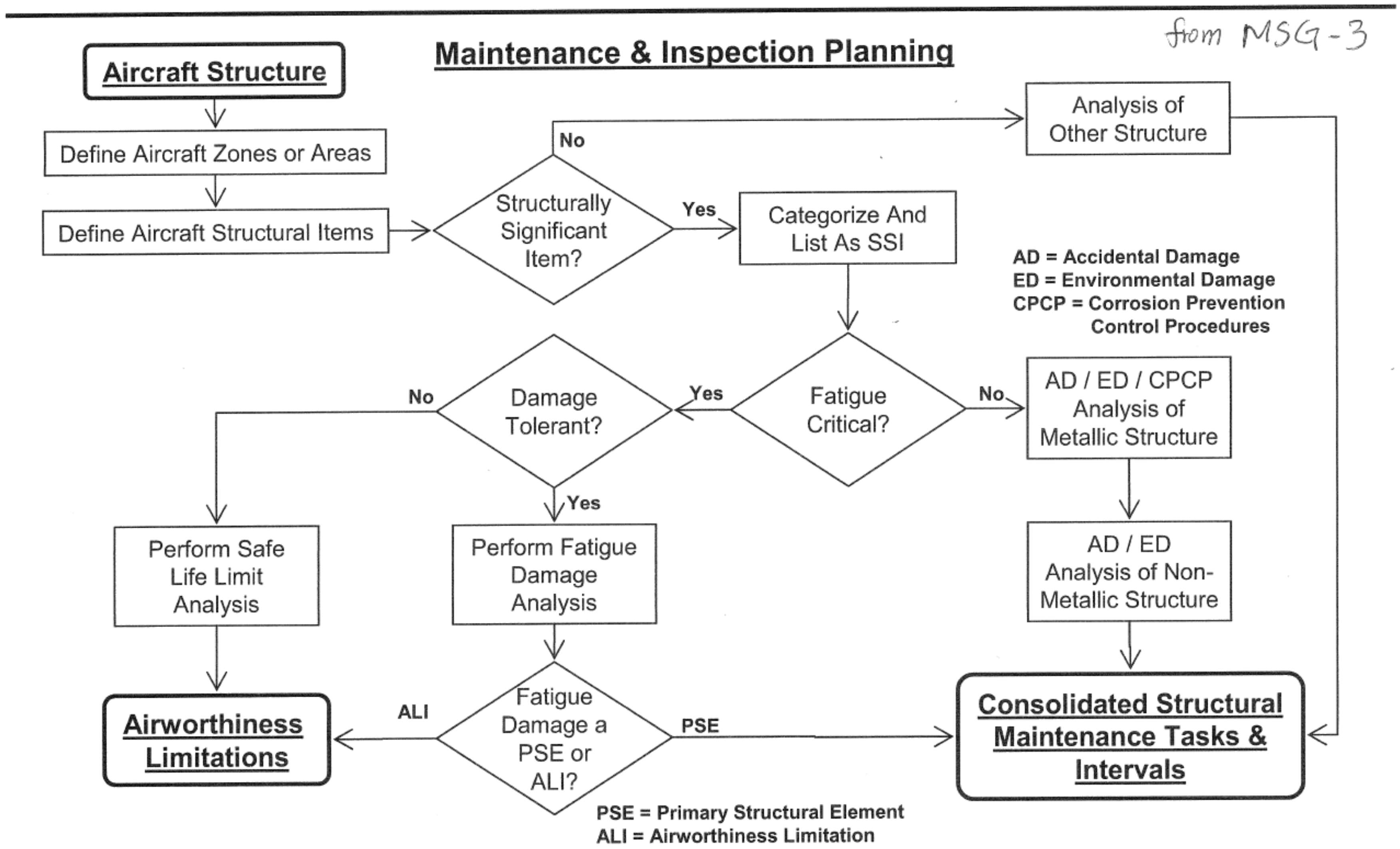


圖 22 飛機容損結構分類及維護檢查需求制訂流程圖

(三) 容損分級(Damage Tolerance Rating, DTR)說明：

對於飛機結構中，無法在裂紋達到 Critical Size 前，以檢查方法檢測出此裂紋時，一般利用容損分級(DTR)將 SSI 結構項目中，依產生疲勞損傷之可能性，以及對此損傷之檢出率(Probability of Detection, POD)加以區分，以制訂出適當之飛機結構檢查計畫。以下說明結構損傷檢出率(POD)制訂基礎、DTR 來源及在訂定檢查計畫上之應用。

1. 結構損傷檢出率(Probability of Detection, POD)：

由於不確定飛機結構損傷會在何次的檢查工作中被發現，因此須考慮所使用檢查方法對於結構損傷之檢出率。結構損傷之檢出率是統計檢查工作人員實際檢查發現結果所訂出，由參與測試人員實際發現結構損傷的比率，定義結構損傷檢出率。

圖 22 列出波音公司針對不同檢查方法對於結構裂紋之檢出率圖表，同樣的裂紋長度，當所使用的檢查方法所能檢出最小尺寸越小時，則檢查工作人員利用該檢查方法，以檢測出該裂紋之機率則越高。

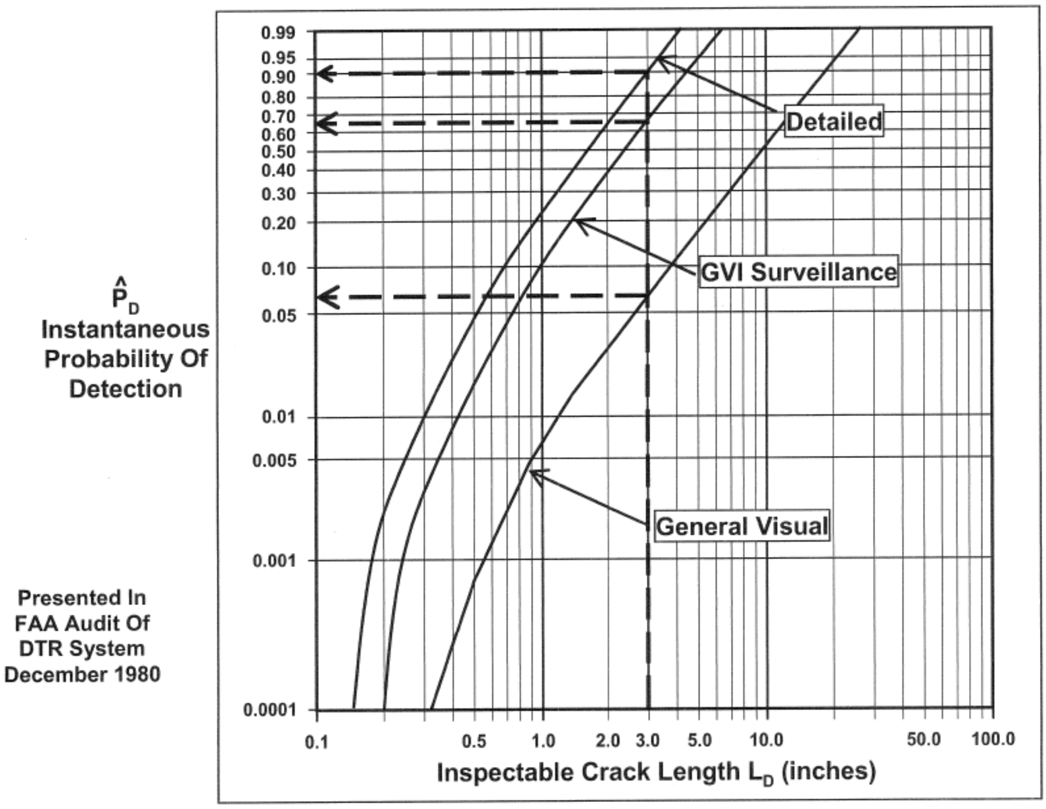


圖 22 不同裂紋長度及檢查方法之檢出率(POD)

2. 容損分級(Damage Tolerance Rating, DTR)：

DTR 是依據結構損傷檢出率(POD)所訂出。結構損傷檢出率(POD) P_D與 DTR 之關係，定義如下：

$$P_D = 1 - 1 / 2^{DTR}$$

由上述公式可換算出 DTR 計算公式如下：

$$DTR = \mathrm{Log}\,(1 - P_D) / \mathrm{Log}\,(0.5)$$

當 DTR=1 時，代表 50% (0.5)的結構損傷檢出率(POD)；

當 DTR=4.32 時，代表 95% (0.95)的結構損傷檢出率(POD)；

圖 23 所列為波音對於不同之飛機結構項目，定義所需之 DTR

(Required DTR)，而所謂的 Basic DTR，指的是在正常的結構檢查下之結構損傷檢出程度，而 Incremental DTR，則是當執行重要檢查有所困難，或是 Fail-Safe Load 與 Operation Load 之 Safety Margin 較少時，所需加強檢查的程度值。

Required DTR For Various Structures

Structure			Required DTR		
			Basic	Incremental	Total
Wings And Nacelles	Externally Visible Areas		4	0	4
	Areas Not Externally Visible		4	2	6
	Primary Flap Structure		4	4	8
Empennage	Primary Structure		4	2	6
Fuselage	Contribution of Cabin Differential Pressure To Total Fail-Safe Stress	< 50%	4	2	6
		> 50%	4	6	10

95%P

NOTES:
- Reflects previous service cracking history
- Base DTR reflects detection during structural inspections
- An incremental DTR is required if casual inspection is difficult, or if fail-safe load is close to operational loads

圖 23 飛機結構檢查 Required DTR 數值

為方便飛機結構檢查補充計畫之制訂，可利用 DTR Form (如圖 24 所示)，將航空公司機隊之目前檢查計畫，與 SSID 所規定的檢查要求相互比較，以調整訂定符合法規要求之飛機結構檢查補充計畫。經查詢圖表得出加總後之 DTR 低於 Required DTR 時，代表航空公司須針對該結構項目，擬定額外檢查工作，以確保飛機結構完整性。

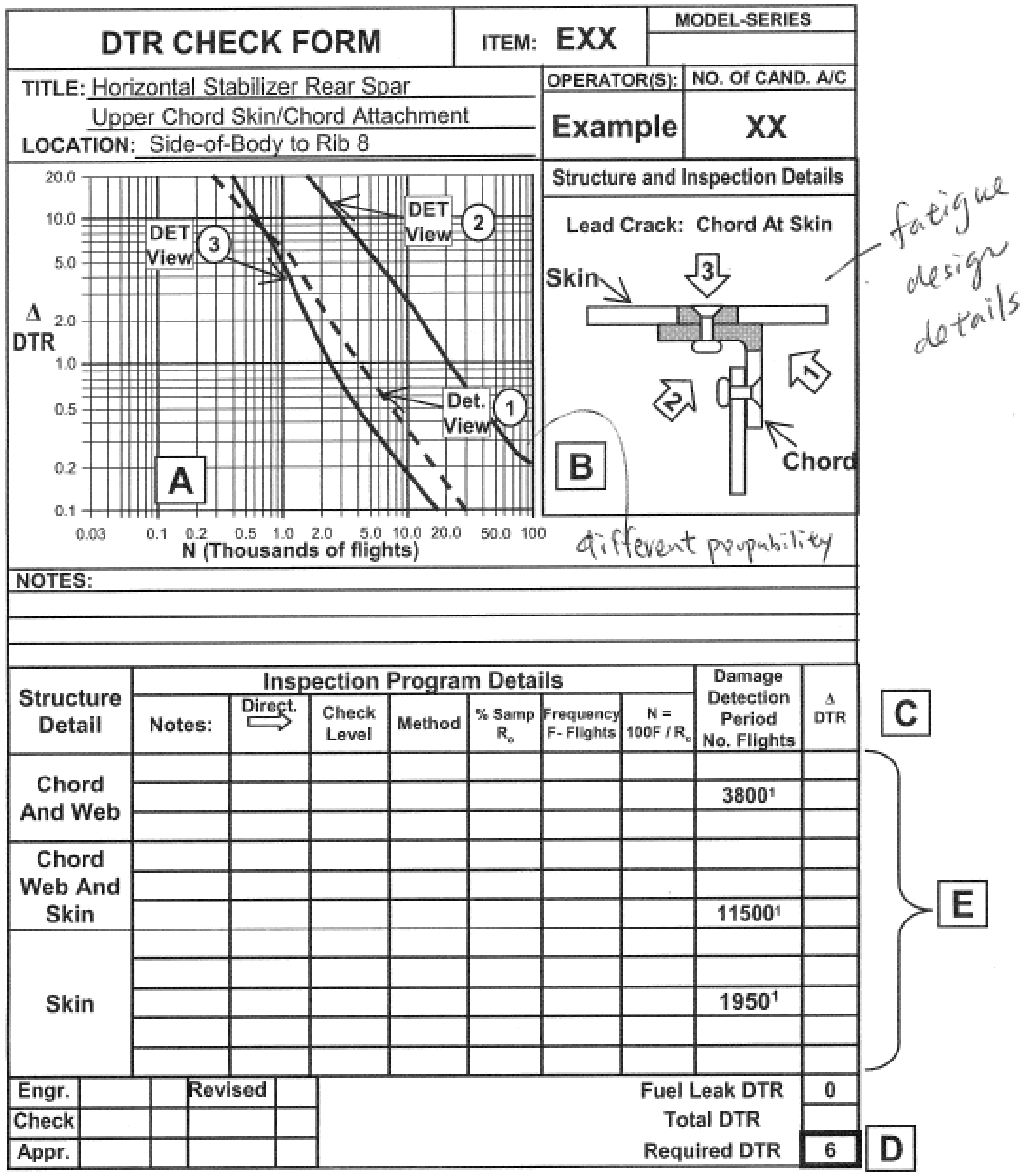


圖 24 DTR 範例

3. DTR 制訂來源：DTR Form 中的曲線係依據裂紋成長分析結果所得曲線而訂出。此外，波音公司尚會依據依據現行機隊結構損傷檢查經驗數據，調整訂定不同結構項目的 Required DTR。

換言之，DTR 之決定係來自於結構剩餘強度之分析及計算、

裂紋成長分析所得出之 Critical Damage，以及結構基礎檢查計畫內容適用性及完整性評估結果。至於 Critical Damage 之計算，除結構剩餘強度分析外，尚須考量結構項目所受之限制負載情況，加上材料性質而訂定。而以此 DTR 分析所訂出之飛機結構檢查計畫，即可符合結構補充檢查(Supplemental Inspections)要求。

(四) 容損結構修理設計原則：

飛機容損結構設計原則有二項：第一項為 Static Strength Repair，第二項為 Damage Tolerant Repair。分述如下：

1. 符合結構靜力強度要求之修理設計(Static Strength Repair)：

   此項設計原則在於使修理後之飛機結構可承受所要求的限制負載及極限負載，可利用 SRM 之修理原則設計所需之結構修理方法。

2. 符合容損要求之修理設計(Damage Tolerant Repair)：

   符合容損要求之修理設計，基本上必須考量容損分析(DTA)三大要素：剩餘強度分析(Residual Strength Analysis)、裂紋成長率分析(Crack Growth Rate Analysis)及檢查需求制訂(Inspection Requirements)。此外，另須考量下列因素：

   - 修理構型，例如：平補片、外補片或內補片。
   - 鉚接安裝構型設計。
   - 孔位及鉚釘排數及鉚釘型式。
   - 修理件(如：Doubler)材質及其厚度。
   - 檢查位置之進手性及檢查方法之選定。

3. Zero Timing Hole：

意指在於將受損傷之結構孔位予以擴孔(Oversizing)以消除損傷，使其檢查起始點可以延後，但須注意原有結構所要求之最小鉚釘間距(Riveting Spacing)及最小邊距(Edge Distance)要求仍需滿足不可違背。有關 Zero Timing Hole 一般程序如下：

- 先以 HFEC 檢查方法檢查結構孔位之損傷情況。
- 若發現裂紋，則以每次擴孔 1/64”之漸進方式，直至結構孔位檢查已無損傷為止。
- 在已無損傷之結構擴孔孔位，再予以擴孔 1/16”。

4. 避免產生孔位尖銳邊角(Knife Edge)：

須注意於製作安裝沉頭鉚釘之沉頭孔(Countersink)時，當沉頭孔深度大於蒙皮厚度 2/3 時，則會使蒙皮沉頭孔形尖銳邊角，將對鉚釘產生一集中應力，使得鉚釘提早損壞，而影響結構完整性。

5. 評估使用之補片厚度(Doubler Thickness)：

雖然越厚的補片可使修理後的結構回復其靜力強度，但也將改變該處結構之負載分佈(Load Distribution)，進而影響修理後結構之疲勞壽命及容損特性。同時較厚之補片亦會降低檢查方法(如：LHEC)之檢出率，因此可能需要縮短檢查時距。

6. 依據 SRM 進行容損修理設計：

目前波音機型中，757/767/777 SRM 中所有修理方法皆符合結構容損設計要求。而 727/737/747 SRM 並非所有修理方法皆符合結構容損設計要求，目前波音已進行 SRM 更新作業，包含重新設計或加入容損分析後所得之檢查要求。因此 SRM 中如有適用的修理方法時，應依據 SRM 以進行容損修理設計。

(五) 高齡飛機結構修理評估計畫(Repair Assessment Plan, RAP)：

為維持高齡飛機結構之持續適航，Air Transportation Association of America (ATA)及 Aerospace Industries Association (AIA)於 1988 年 6 月組成了一個 Airworthiness Assurance Working Group (AAWG)工作團隊，此工作團隊由航空產品製造廠、航空公司、民航主管機關及航空組織所組成，AAWG 下亦成立一個結構工作小組(Structures Task Group, STG)，其成立目的在處理有關高齡飛機之相關議題並研擬相關方案，其一即為「結構修理評估計畫(Repair Assessment Plan, RAP)」

RAP 計畫內容係規畫飛機加壓艙結構(Pressurized Vessel Structure)之補充檢查，以確保高齡飛機持續適航。AAWG 建立了 RAP 之架構，包含三階段評估流程(Three-Stage Process)，以及結構修理評估分類(Repair Assessment Categories)，再由 STG 針對各高齡機型發展出個別之 Repair Assessment Guidelines (RAG)，說明相關工作內容。

1. 三階段評估流程(Three-Stage Process)：

   RAP 三階段評估流程說明如下。

   (1) Stage 1：建立修理評估計畫。航空公司須於一定時間內提出結構評估計畫，並經由其民航主管機關核准。

   (2) Stage 2：檢視修理並將其分類。(Category A, B, C)

   (3) Stage 3：決定適當的檢查計畫。RAG 文件提供各常見修理類型之檢查起始點及檢查時距之訂定依據。

2. 結構修理評估分類(Repair Assessment Categories)：

   經評估現有結構修理項目後，可將修理項目分為四類，說明

如下：

(1) Category A Repair：為永久性修理(Permanent Repair)，且無需進行結構補充檢查，亦即一般標準結構檢查方法即已足夠。例如：小尺寸平補片修理(Flush Repair)等。

(2) Category B Repair：為永久性修理，並有結構補充檢查需求。例如：外補片修理、大尺寸平補片修理等。

(3) Category C Repair：為時限修理(Time-Limited Repair)或稱為暫時性修理(Temporary Repair)，由於其疲勞壽命低或無法確定其疲勞壽命等因素，故須在一定時間內更換為永久性修理。例如：盲鉚釘修理等。

(4) Not Structurally Acceptable Repair：代表既有之修理無法滿足飛機結構 Static Strength 之要求,須在下次飛航前更換為符合規定之修理。

(六) 高齡飛機相關法案：

FAA 為維持高齡飛機之安全性，持續制訂相關法案，說明如下：

1. Widespread Fatigue Damage (WFD)法案：

   由於高齡飛機之持續使用，終究會到達一個時間點，使整體飛機結構性能衰減至發生危害之風險提昇的程度，這個時間點定義為 Limit of Validity (LOV)，也就是過了 LOV 後，飛機即不可再使用，目前此法案尚在立法階段(NPRM, Notice of Rule Making)中。

2. Aging Aircraft Safety Rule (AASR)法案：AASR 法案已於 2005 年 3 月正式生效。本法案包含兩個部份：第一部份為高齡飛機結構檢查及維護紀錄檢視；第二部份針對整個飛機結構包含修理、改裝在內項目，建立容損為基礎之結構補充檢查。

3. FAR Part 26：包含電子線束系統安全(Subpart B)、燃油箱安全(Subpart D)及高齡飛機修理及改裝之容損分析(Subpart E)。其中自 FAR Part 26 Subpart E 生效以後，所有疲勞關鍵基礎結構(Fatigue Critical Baseline Structure, FCBS)之修理項目必須進行容損分析(DTA)。

肆、心得與建議

一、訓練心得：

此次至飛機製造原廠 – 波音公司訓練中心參加負有盛名之結構修理訓練課程，實是難得機會。綜觀其上課教材內容深入淺出，而且波音飛機結構設計，在民航製造業自有其第一把交椅之優勢，因此總有第一手設計資料，故於闡述相關疲勞原理、破壞力學及容損分析概念，兼有廣度及深度，可於此短期五天課程中，讓上課學員系統性地了解全貌，相當難能可貴。

而授課講師 Mr. John U. Gokcen 本身最早於土耳其航空工作至 Line Maintenance 經理，後至美國深造，亦曾至加拿大工作，後來任職波音公司後，曾參多項波音機型結構設計工作，具有豐富之實際結構設計及分析經驗。故能帶領學員由淺入深、由繁入簡地領略飛機結構設計及修理的兩大重要領域：疲勞分析及容損分析。

而課程中亦安排多項疲勞分析、有限元素分析、容損分析之計算演練，更加深相關公式及理論之應用與了解。此外，課程第二天亦安排至西雅圖附近之 Fatigue Technology Inc. (FTI)參訪，這家公司為波音重要外包廠商之一，他們亦承接許多軍機及工業界之業務，透過實地了解疲勞試驗過程及金屬孔位冷作加工原理與工作程序，可以應證上課所學。

另外，學員有來自德航技術工程部門、加拿大民航局、加拿大航空、波音 737 Service Engineer 等不同單位結構工程師，以其自身經驗，在上課發問、相互學習，對於提昇與加強學習效果甚有助益。

整體訓練課終了，我們亦對波音提出此課程之建議，建議其增加容損分析之課程講授時間，並增加 FAR Part 26 之課程內容，此部份亦得到講師正面回應，詳如以下 e-mail 內容。

"Gokcen, John U" <john.u.gokcen@boeing.com>
11/25/2009 00:43

收件人 "'ericchen@mail.caa.gov.tw'" <ericchen@mail.caa.gov.tw>, "alanchen@mail.caa.gov.tw" <alanchen@mail.caa.gov.tw>
副本抄送 "Post, Anthony" <anthony.post@boeing.com>, "Davis, John R" <John.R.Davis2@boeing.com>, "Schroeder, Richard D" <richard.d.schroeder@boeing.com>
副本密送
主旨 [This message is clean of malware] RE: RE_Receipt for 472 Class payment to Taiwanese CAA participants

Dear Eric and Alan,

I am very glad to hear that you already got the receipts from Mike and you enjoyed the class and found it very useful. Your comments, inputs regarding to extension of the course to two weeks and coverage of more part 26 requirements and damage tolerance examples will be evaluated.

Thanks and look forward to seeing you both in the future classes,

Best,
John

二、建議事項：

建議未來可派員多參與原廠結構訓練課程，以學習完整技術資料內容，有助於本局執行國內飛機結構大修理之審查時,能與國際間相關法規要求有一致性及符合性。

三、補充文件：

此次上課，由於德航技術工程部門的同學亦有改裝飛機天線的經驗，因此講師特別提供一份 FAA Chicago ACO (驗證辦公室)撰寫的一份報告「Damage Tolerance Analysis for Antenna Installations on Pressurized Transport Airplanes」給所有學員，該報告以加壓艙改裝天線為例，說明容損分析之考量要點，是一份很好的參考文件，特附於本報告之後供未來應用參考。

伍、附件

附件：「Damage Tolerance Analysis for Antenna Installations on Pressurized Transport Airplanes」

DAMAGE TOLERANCE ANALYSIS

FOR

ANTENNA INSTALLATIONS

ON

PRESSURIZED TRANSPORT

AIRPLANES

Chicago Aircraft Certification Office

DAMAGE TOLERANCE ANALYSIS FOR ANTENNA INSTALLATIONS-GENERAL OUTLINE

1.0 WHAT IS DAMAGE TOLERANCE ANALYSIS – A DEFINITION

Damage Tolerance Analysis is the application of Fracture Mechanics to assess how a structure, assumed to be cracked, will respond to loads (cyclic and static) over time. Specifically, a Damage Tolerance Analysis assesses: (1) How quickly a crack(s) will grow over time; and (2) How the strength of the structure is affected by crack size and shape.

2.0 WHEN IS DAMAGE TOLERANCE ANALYSIS REQUIRED?

2.1 Damage Tolerance was adopted as a rule for large transport category airplanes in October 1978 by Amendment 45 to FAR 25.571. Any airplane with Amendment 45 as part of its certification basis must have damage tolerance evaluations performed on any modifications (e.g. alterations, repairs, and STC's).

2.2 Several older transport category airplanes certified prior to the adoption of Amendment 45 to FAR 25.571 have undergone damage tolerance evaluations by their manufacturers. These evaluations resulted in the development of Supplemental Inspection Documents (SIDs) that where subsequently mandated by Airworthiness Directives (ADs). These airplanes are: Airbus A300; BAC1-11; Boeing 707/720, 727, 737,747; Douglas DC-8, DC-9/MD-80, DC-10; Fokker F-28; and Lockheed L-1011. Consequently, even though these aircraft were certified prior to Amendment 45, they were brought up to that level by virtue of mandating the SIDs and consequently must have damage tolerance evaluations performed on any modifications.

It should also be noted that the original intent was that repairs to all aircraft models having damage tolerance based SIDs should also be subject to damage tolerance evaluations and appropriate inspections established. However, this intent was not always met. As a result a new rule was issued on April 18, 2000 entitled "Repair Assessment for Pressurized Fuselages". This rule requires that a comprehensive damage tolerance repair assessment be completed for fuselage pressure boundary repairs to the 11 aging aircraft models. The required inspections, modifications, and corrective actions are to be accomplished in accordance with an airplane's mcdel specific repair assessment guidelines and incorporated into a certificate holders FAA approved maintenance program.

2.3 Damage Tolerance was established as a requirement for commuter category airplanes in February 1996 by the adoption of FAR 23.574. Any airplane having FAR 23.574 as part of its certification basis must have damage tolerance evaluations performed on all modifications.

2.4 Airplanes that operate at high altitude (41,000 feet and above) generally have a special condition requiring a damage tolerance evaluation as part of its certification basis. The purpose of this requirement is to assure that the critical crack length, and resultant leakage rate precludes rapid decompression. Certificated airplanes in this category include: Cessna 525A, 560 and 650; Challenger (Bombardier) 600 model series; Falcon 50; Hawker 800; Israeli Aircraft Industries 1125; Lear 35A, 45, 55 and 60; the Sino Swearingen model SJ30-2; and the Raytheon (Beech) 400A.

3.0 HOW ARE THE ANALYSIS RESULTS USED

3.1 FAR 25.571 requires the establishment of inspections or other procedures, as necessary, to prevent catastrophic failure. These requirements must be consistent with the damage tolerance characteristics of the structure as determined by analysis and test.

3.2 FAR 21.50 and FAR 25.1529 require Instructions for Continued Airworthiness and Manufacturer's Maintenance Manuals having Airworthiness Limitations sections be delivered with each aircraft. The AD mandated SIDs bring the above listed older airplanes up to this same requirement level. Appendix H to FAR 25, section H25.4 requires a separate Airworthiness Limitations section to be approved by the certificating Aircraft Certification Office (ACO). This section must set forth each mandatory replacement time, structural inspection interval, and related structural inspection procedure approved under FAR 25.571.

3.3 Because of the aforementioned requirements, when a damage tolerance evaluation is required for an aircraft, the results of the damage tolerance analysis for all antenna installations will be used to develop a Maintenance Manual Supplement specifying the necessary inspection locations, methods and frequencies. Each Maintenance Manual Supplement must be approved by the certificating ACO, and cited in the limitations section of the STC approving the installation (e.g. the approved antenna modification must be maintained in accordance with the FAA-approved Maintenance Manual Supplement.)

4.0 ANTENNA INSTALLATION CHARACTERISTICS

A typical antenna installation with a single external doubler is illustrated below. Installations are typically mid-bay between frames and longerons. The antenna straddles the connector hole, and is easily removed to facilitate inspection by being mechanically fastened to the fuselage skin via screws into the nutplates. In turn the nutplates are riveted to the doubler and fuselage skin.

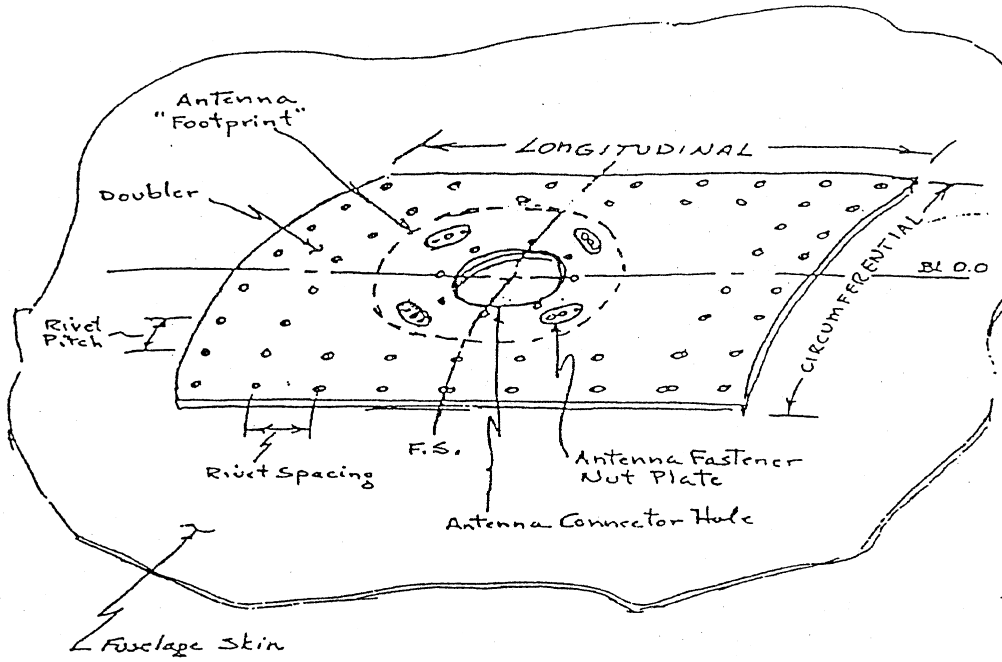


Figure 1 Typical Antenna Installation

4.1 QUALIFYING ASSUMPTIONS

4.1.1 INSTALLATION CHARACTERISTICS

The scope of this document is limited to antenna installations that: (1) Do not involve cutting any major load paths such as stringers or frames; and (2) Are located in the fuselage shell away from discontinuities such as doors, windows, wing-fuselage intersection, etc. where significant load redistribution takes place. Typically, the antenna installations addressed herein are located midbay between frames and stringers.

4.1.2 GENERAL APPROACH

The guidance presented herein is generally consistent with the approach used in the RAPID programs which may be downloaded from the Internet web site given in Reference [1]. In some cases this document is more conservative than the RAPID programs since the approach presented herein is less rigorous.

4.1.3 LOADING

A conservative once-per-flight constant amplitude Ground-Air-Ground (GAG) loading cycle is suggested in lieu of a more precise, complex flight-by-flight loading spectrum. Reference [2] states that for circumferential cracking the rule of thumb is to use a stress equivalent to stress due to pressure (e.g. $\Delta pR/2t$) plus inertia stress due to 1.5G. Experience indicates that this should be conservative for large transport aircraft and a lessor inertia component might be acceptable (e.g. 1.3G).

5.0 GROSS AREA STRESSES

The design stresses used by the airplane manufacturer are generally unavailable. However, a conservative estimate of gross area stresses can be made as discussed below. The fuselage will be treated as a pressurized cylinder and the beneficial effects of longeron and frame areas ignored. Basic cylindrical pressurized shell equations are used.

5.1 HOOP STRESS

The stress in the hoop direction is assumed to be due to pressure only and is given by,

$$f_H = \Delta pR/t$$

Where,

Δp = shell differential pressure, psi
t = shell thickness, in
R = shell radius, in

5.2 LONGITUDINAL STRESS

The stress in the longitudinal direction is assumed to be due to pressure and vertical fuselage bending due to inertia (i.e. vertical acceleration) and is given by,

$$f_l = \Delta pR/2t + n_z\sigma_{1G}$$

Where,

Δp = shell differential pressure, psi
t = shell thickness, in.
R = shell radius, in.
n_z = vertical load factor
σ_{1G} = 1G stress at antenna location

The longitudinal stress is assumed to vary along the length of the fuselage as shown below. Any variation around the circumference is conservatively neglected.

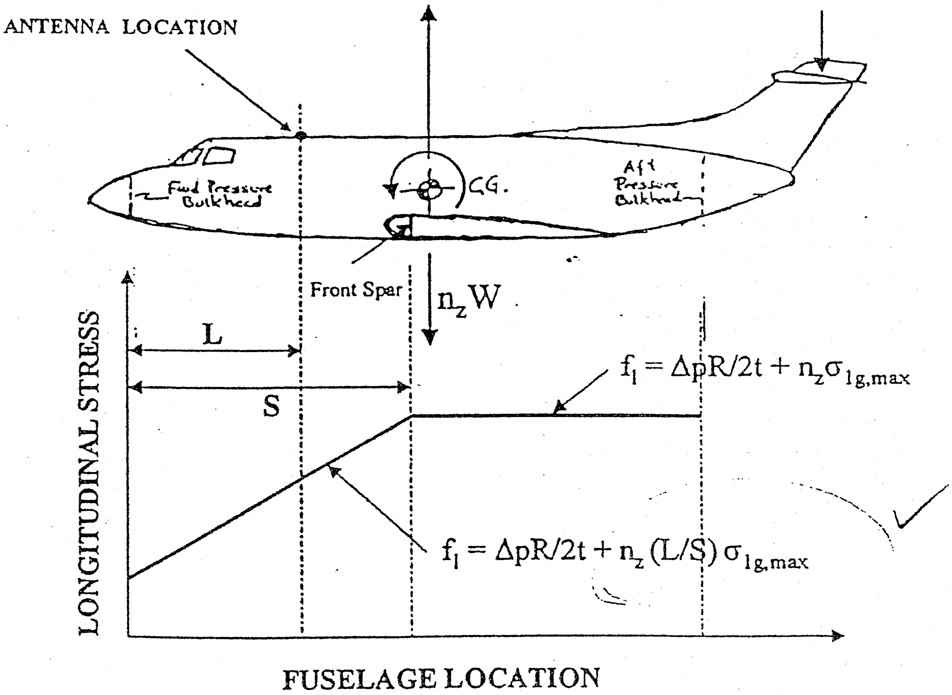


Figure 2 Assumed Longitudinal Stress Distribution

5.3 ESTIMATE OF MAXIMUM 1G STRESS

The maximum 1g stress is assumed to occur at the top of the fuselage over the wing where vertical fuselage bending is the greatest. In order to insure conservatism in estimating it any static strength "hole out" factor as well as additions to normal operating pressure (e.g. aerosuction, pressure regulator tolerance, etc.) are ignored in the following calculation. It is assumed that the airplane manufacturer designed to zero margin at ultimate. The total ultimate design stress is composed of a pressure and inertia component and is given by,

$$\text{Total Ultimate Stress} = 1.5(\Delta PR/2t + n_z \sigma_{1G,max})$$

Where,

ΔP = normal operating pressure at maximum design altitude, psi
R = fuselage radius, in.
t = fuselage skin thickness, in.
n_z = maximum limit design load factor
$\sigma_{1G,max}$ = maximum 1G longitudinal stress in the fuselage due to bending
1.5 = Safety factor, ref. FAR 25.303

(NOTE: n_z should be conservatively assumed to be 2.5 unless it can be substantiated that the airplane manufacturer used a higher value (reference FAR 25.337(b)).)

For a zero margin design the total ultimate stress would be just equal to the ultimate allowable used for the material (reference MIL-HDBK-5, F_{tu}, B value). Therefore $\sigma_{1G,max}$ is given by,

$$\sigma_{1G,max} = (F_{tu}/1.5 - \Delta PR/2t)/n_z$$

May be Lesser of

5.4 CRACK GROWTH

Stresses to be used to compute crack growth should represent typical or nominal loading. Adverse pressure regulator tolerance as well as relief valve setting and tolerance are not included. Although many cycles actually occur during any given flight, a once per flight stress cycle may be used to estimate the crack growth behavior. This is schematically illustrated below.

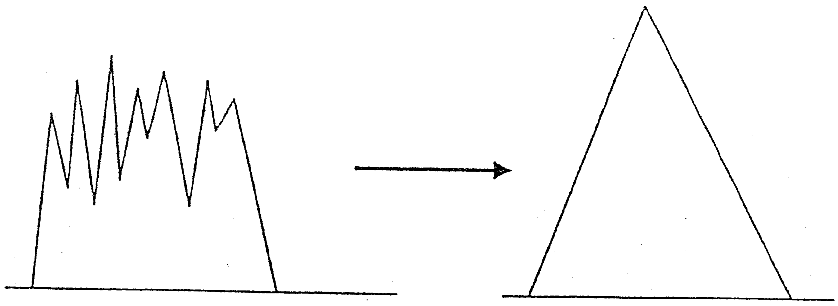


Figure 3 Once Per Flight Equivalent Cycle

The equivalent once per flight stress cycle to be assumed for the location being analyzed will depend on whether a longitudinal or circumferential crack is being evaluated.

5.4.1 LONGITUDINAL CRACKS

For longitudinal cracks it is assumed that inertia effects are negligible and that total stress is in the hoop direction. Hoop stress is given by,

$$f_H = \Delta PR/t$$

where ΔP is the normal operating pressure at maximum design altitude plus 0.5 psi to approximate a once per flight maximum aerodynamic suction loading.

5.4.2 CIRCUMFERENTIAL CRACKS

For circumferential cracks both pressure and inertia loading is accounted for and the longitudinal stress is given by the equations in section 5.2 dependent on whether the location being analyzed is forward or aft of the front spar. ΔP is the same as for the hoop stress as given in section 5.4.1. The load factor, n_Z, taken to be 1.3G, represents the equivalent once per flight load factor to be used in the crack growth assessment. As mentioned in Section 4.1.3 a value of 1.5 has been accepted in the past for large aircraft however lesser values may be acceptable. The Chicago Aircraft Certification Office recommends the use of 1.3.

5.5 RESIDUAL STRENGTH

The residual strength requirements are given in FAR 25.571(b)(5)(i)&(ii). (NOTE: The load requirements of FAR 25.365(d) defines 1.33xΔP as the <u>limit static strength</u> loading and should not be confused with the 25.571 requirements.) Residual strength condition(s) (i) includes normal operating differential pressure plus aerodynamic pressure combined with the limit flight loads given in 25.571 (b)(1) through(4). Residual strength condition (ii) includes the maximum value of normal operating differential pressure factored by 1.1 to account for relief valve setting and tolerance plus aerodynamic suction at 1G.

5.5.1 LONGITUDINAL CRACKS

For longitudinal cracks, condition (ii) prevails since inertia effects are assumed to be negligible and the required residual strength hoop stress is given by,

$$f_{H,RES} = 1.1\Delta PR/t + 0.5R/t$$

where ΔP is the normal operating pressure at maximum design altitude and 0.5 psi is the assumed aerodynamic suction pressure.

5.5.2 CIRCUMFERENTIAL CRACKS

For circumferential cracks, condition (i) prevails since inertia effects are significant. The required residual strength longitudinal stress is calculated from the equations given in section 5.2 depending on whether the location is forward or aft of the front spar. That is,

$$f_{L,RES} = \frac{\Delta PR}{2t} + n_z \left(\frac{L}{S}\right) \sigma_{1g\,max} \quad \text{(Forward of front spar)}$$

$$f_{L,RES} = \frac{\Delta PR}{2t} + \sigma_{1g\,max} \quad \text{(Aft of front spar)}$$

n_z

where,

Δp = normal operating pressure at maximum design altitude + 0.5 psi for Aerosuction.

n_Z = maximum design limit load factor (at least 2.5 but not greater than 3.8)

6.0 FASTENER LOADS

Typical antenna installations employ a reinforcing doubler fastened to the fuselage skin with rivets. When the airplane is loaded (stress is applied) the doubler and skin will attempt to strain together, and some amount of load will transfer out of the skin and into the doubler. The load is transferred by rivet shear and bearing. The highest (critical) rivet load associated with the transfer load is at the first fastener row as illustrated in the figure below. Cracking in the skin is most likely at the first fastener due to the high bearing load in combination with basic gross stress.

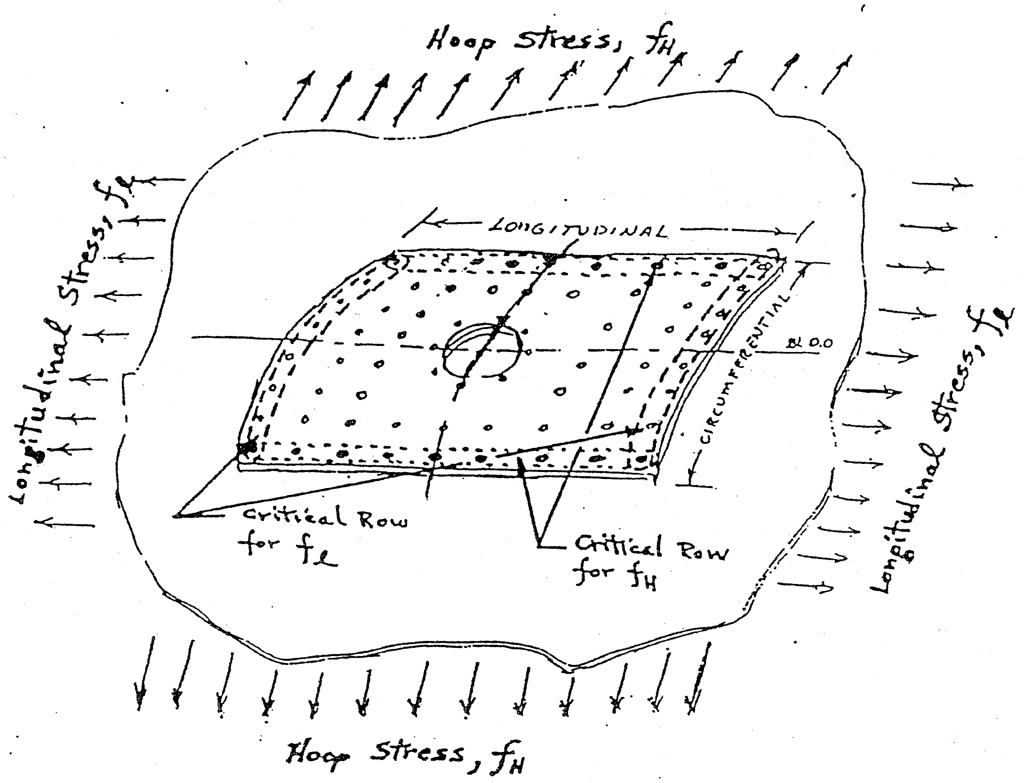


Figure 4 Critical Fastener Rows

Fastener loads may be approximated by a number of different methods. A common approach is to use a strip analogy and perform a displacement compatibility analysis as described in section 6.1 below. Finite element methods may also be used on a 2D or 3D basis to more rigorously determine loads.

6.1 DISPLACEMENT COMPATIBILITY APPROACH

To understand this method refer to the Figure 5 where the idealized strip (1 rivet spacing in width) is loaded with skin hoop stress, f_H. A similar approach is used to determine rivet loads associated with longitudinal skin stress.

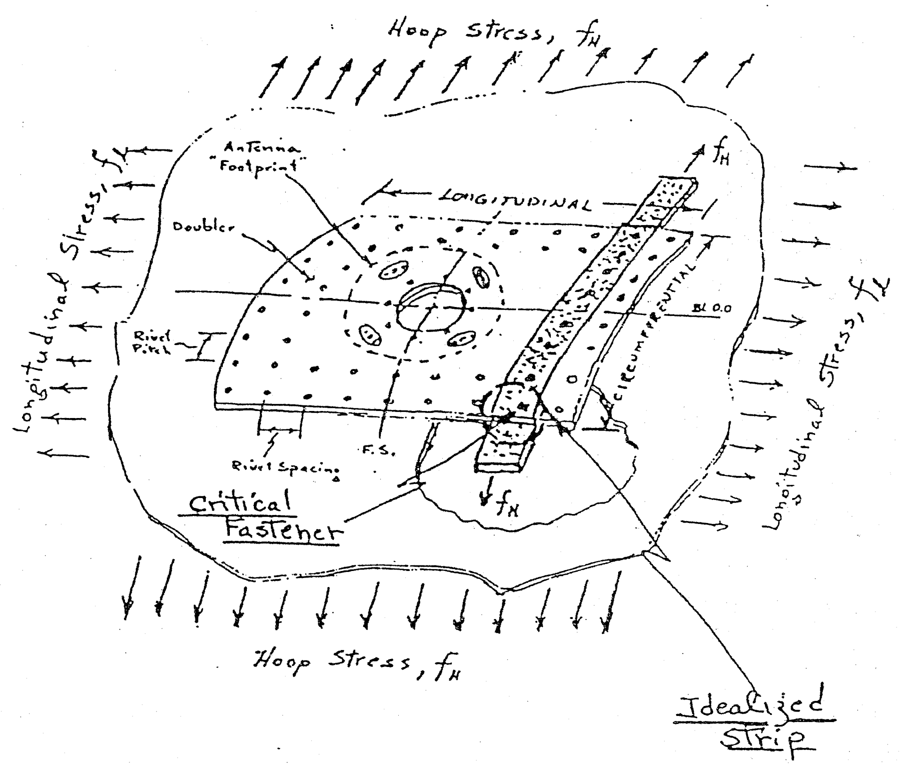


Figure 5 Strip Analogy

A displacement compatibility analysis will yield a fastener load distribution as depicted below.

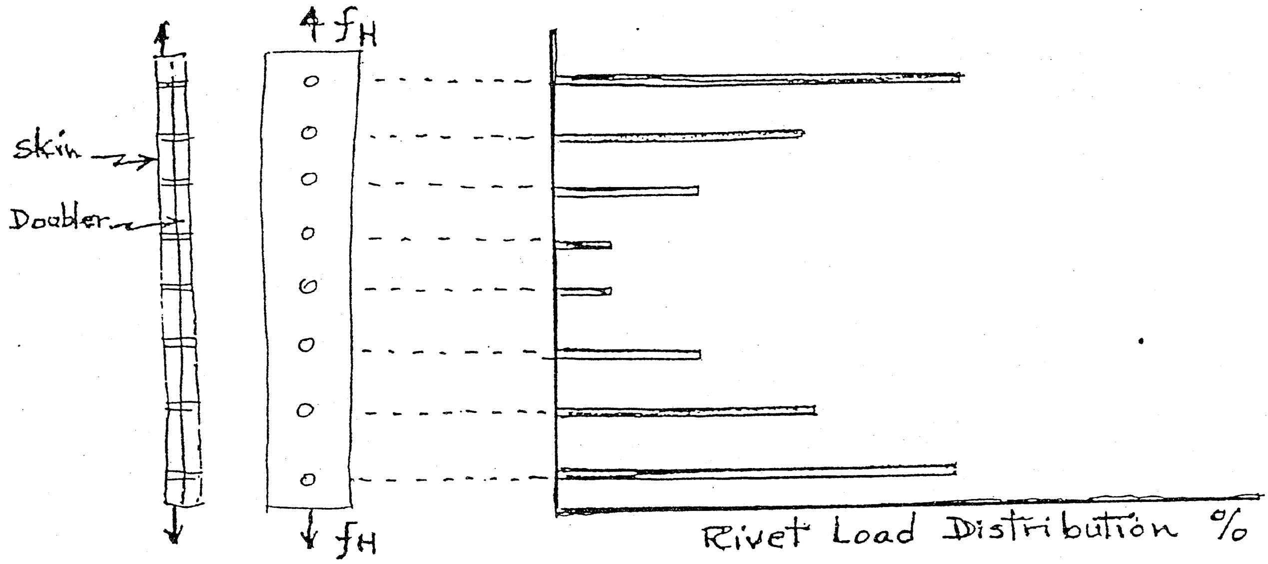


Figure 6 Rivet Load Distribution

7.0 CRACK GROWTH ASSESSMENT

Crack growth is typically determined by numerically integrating the growth rate, da/dN, from some initial crack size to some final crack size. Computer programs are available to do this but it may also be performed by hand using a spread sheet. Crack growth rate is directly proportional to the stress intensity range, ΔK, and the general relationship is illustrated below.

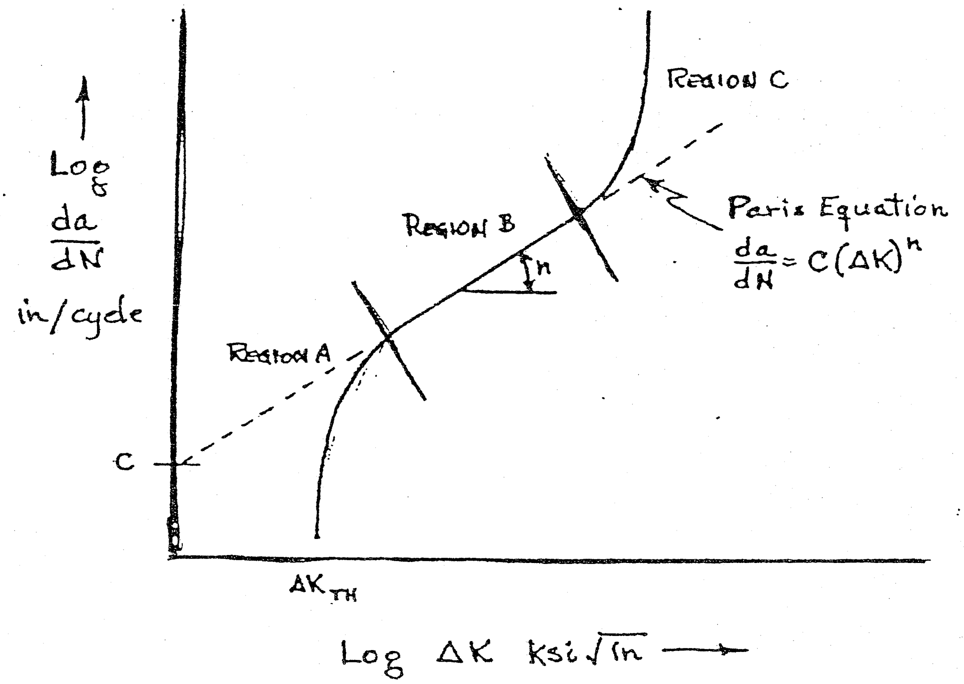


Figure 6 Crack Growth Rate as A Function of ΔK

Notes: Region A – Growth rate decreases asymptotically with decreasing ΔK. Below a threshold value of ΔK (i.e. ΔK_{TH}) there is no growth.

Region B – Growth rate and ΔK follow a Log-Log linear relationship and can be reasonably approximated using the Paris Equation where;

n = Slope of line

C = Intercept of da/dN axis

Region C – Growth rate increases asymptotically with increasing ΔK.

Crack growth rate libraries are included with many of the crack growth programs that are available. A number of different approaches are used to represent the data. The simplest mathematical representation is the Paris equation as shown in Figure 7 however it does not include any stress ratio effects. Walker generalized the Paris equation to make da/dN also a function of stress ratio and it takes the form given below.

$$da/dN = C[(1.0-R)^q K_{max}]^p$$

The RAPID computer programs use the Walker equation. C, q and p values for a number of commonly used aluminum alloys are included in Table 1 for reference.

TABLE 1. Walker Equation* Coefficients and Exponents for Room Temperature, Laboratory Air Ambient Conditions

ALLOY	FORM	DIRECTION	C	q	p
2014-T6	Sheet	L-T	9.66482×10^{-10}	0.57937	3.78906
2024-T3 & -T42	Sheet	L-T	6.76125×10^{-10}	0.64647	3.71980
2024-T3 & T42	Sheet	T-L	9.01566×10^{-10}	0.62910	3.68842
2024-T351/T3511	Plate/Extrusion	L-T	8.86005×10^{-10}	0.67178	3.71010
7050-T7452	Forging	L-T & T-L	1.08344×10^{-9}	0.68746	3.72313
7050-T74511 & T76511	Extrusion	L-T	1.98718×10^{-9}	0.76890	3.60885
7050-T7651 & T7451	Plate	L-T & T-L	1.32927×10^{-9}	0.57452	3.55242
7075-T6	Sheet	L-T	1.11737×10^{-9}	0.60750	3.79719
7475-T7351 & T7651	Plate	L-T	1.05576×10^{-9}	0.60418	3.54815
7475-T761	Sheet	L-T	1.11412×10^{-9}	0.66473	3.74701

* $da/dN = C[(1.0 - R)^q K_{max}]^p$ where, da/dN = crack growth rate, in./cycle
K_{max} = maximum (i.e. peak) stress intensity, ksi(in)$^{1/2}$
R = stress ratio, K_{min}/K_{max}

7.1 STRESS INTENSITY FACTOR RANGE

The stress intensity factor range, ΔK, is the difference between the maximum and minimum applied stress intensity,

$$\Delta K = K_{max} - K_{min}$$

For antenna installation assessment the loading cycle is from zero load to the once per flight equivalent load. Therefore K_{min} is zero and ΔK is equal to K_{max}. The cyclic history of applied stress and stress intensity are illustrated below.

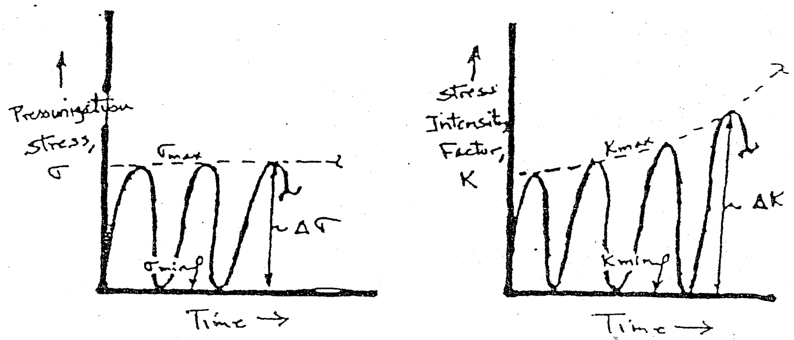


Figure 7 Stress and Stress Intensity Time History

Stress intensity is given by,

$$K = \sigma(\pi a)^{1/2}\beta$$

Where,

σ = reference stress or load (typically far field gross Stress)
a = crack dimension (typically crack length)
β = geometry factor (typically a function of at least 'a')

(Note: Units for stress intensity factor, K, are typically psi(in)$^{1/2}$, or ksi(in)$^{1/2}$)

8.0 CRITICAL LOCATIONS

Several detail locations should be evaluated to determine the damage tolerance characteristics. As a minimum longitudinal and circumferential cracking at the outer row of fastener holes and at the antenna connector hole should be considered. Consistent with this, some candidate cracking scenarios are illustrated below. Other scenarios should be considered if it is suspected that those described herein do not sufficiently bound the details involved.

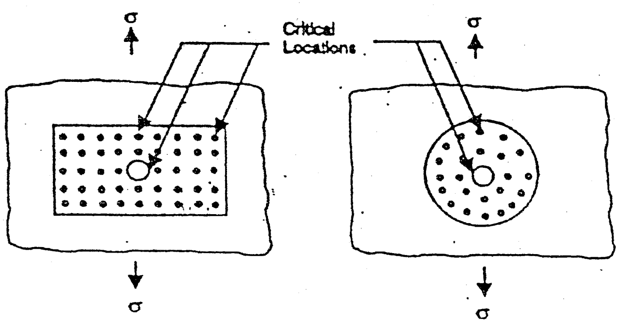


Figure 8 Candidate Critical Locations

9.0 INITIAL FLAW SIZES AND CONTINUING GROWTH

To perform a Damage Tolerance Analysis (DTA) of the antenna modification, assumptions of initial flaw sizes, and subsequent growth are necessary. To achieve a uniform approach to DTA for antenna installations, the following initial flaw size and subsequent growth scenarios are assumed.

9.1 INITIAL FLAW ASSUMPTIONS

At any analysis location, a primary 0.050" through the thickness crack is assumed to exist in the skin on the most critical side of the hole from a damage tolerance perspective. Simultaneously, a secondary 0.005" through the thickness crack is assumed to exist in the opposite side of the hole having the primary crack, and in each adjacent hole on the remote side on the primary 0.050" crack as depicted below:

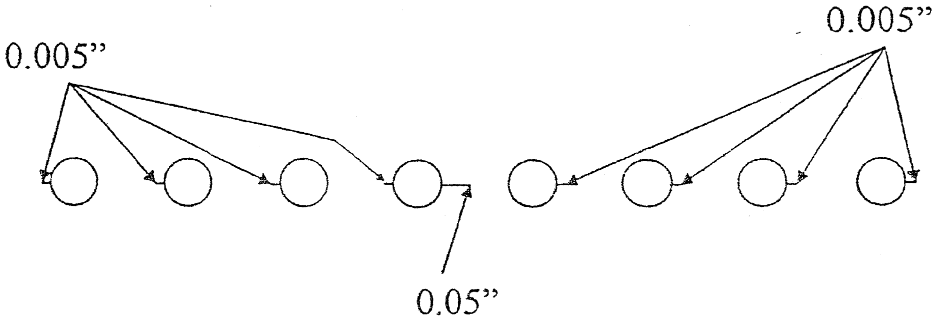


Figure 9 Initial Flaw Assumptions

9.2 SUBSEQUENT GROWTH ASSUMPTIONS

All cracks, both primary and secondary, are assumed to grow concurrently, but independently. The interaction between cracks is ignored. Growth of the primary 0.050" crack is evaluated until it breaks through into an adjacent hole. At this juncture, all the secondary 0.005" cracks are assumed to have simultaneously grown an amount, Δa_1, and the total crack length would have become the center-to-center distance between the adjacent holes plus a hole diameter plus (.005" + Δa_1) beyond each hole. The end of the first stage of growth is depicted below.

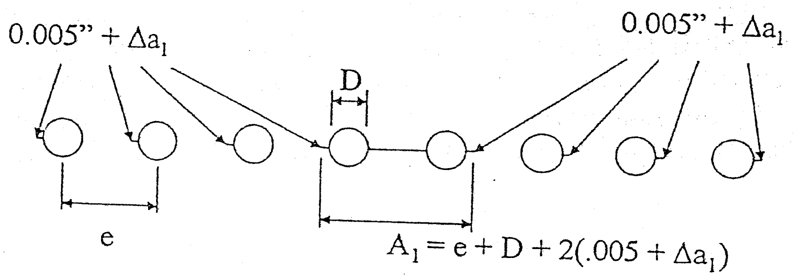


Figure 10 End of First Stage of Continuing Damage

Determination of the secondary 0.005" crack incremental growth, Δa_1, should be with the same cyclic load as was used for the growth of the primary 0.05" crack, and for the number of cycles necessary for it to grow to the adjacent hole.

This same process continues in successive growth. Specifically, the primary crack of length "A_1" in the first stage, will grow to the next adjacent holes. During this growth period, all the secondary cracks of length 0.005"+ Δa_1 are assumed to have grown an additional amount, Δa_2, and the total primary crack length at the end of the second stage of continuing damage would be equal to three hole pitches plus a hole diameter plus (0.005"+ $\Delta a_1 + \Delta a_2$) beyond each hole. The end of the second stage of continuing damage is depicted below.

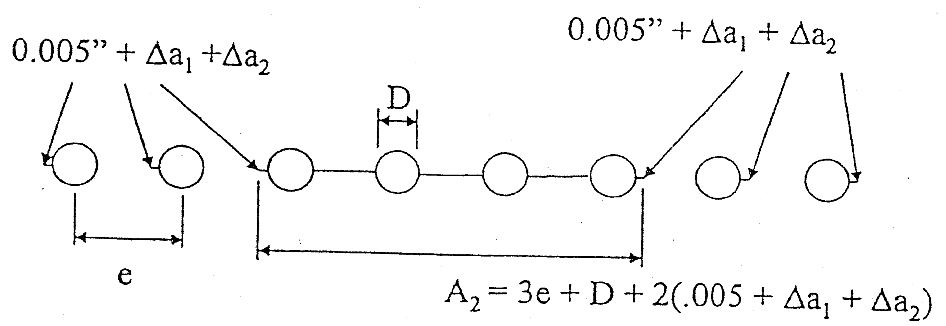


Figure 11 End of Second Stage of Continuing Damage

Determination of Δa_2 would be the same process as used to establish Δa_1 except that the number of growth cycles would be those necessary for the primary crack at the end of the first stage to grow out to the next adjacent holes. This same scenario would be followed as long as continuing damage is to be calculated.

10.0 STRESS INTENSITY FACTOR SOLUTIONS/CONSIDERATIONS

The actual problem of cracking along a row of loaded holes or cracking from the antenna connector hole as described about is very complex. However there are reasonable ways to approximate these problems. Both situations are discussed below along with some possible simplifying assumptions.

10.1 LOADED FASTENER HOLE

The initial problem is illustrated below. This is the case of a loaded hole with bearing and bypass stress. Solutions for this case are readily available in the literature and are typically part of the stress intensity library in available crack growth codes. These are typically based on superposition of a through stress case and a loaded hole case.

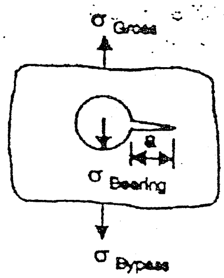


Figure 12 Single Crack From Hole with Bearing and Bypass Stress

If continuing crack growth is to be evaluated after growth of the primary crack terminates, the scheme discussed in section 9.0 is followed. In order to approximate this a through crack may be considered with total length as shown below. Continuing growth would be approximated with a jump discontinuity equal to two hole diameters plus .005" and whatever Δa had occurred up to that point in time beyond each hole. This is illustrated below.

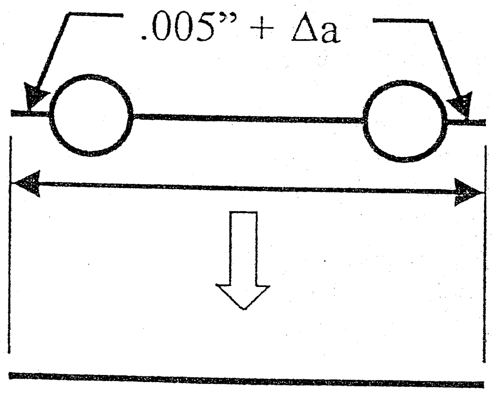


Figure 13 Through Crack Approximation

10.2 ANTENNA CONNECTOR HOLE

The initial problem is illustrated below. This is the case of a single crack out of an open hole. In reality the reinforcing doubler has a beneficial effect in lowering the local stresses around the connector hole. However unless a finite element analysis has been performed to accurately quantify the effect it may be conservatively assumed that it is ineffective and that the hole is subject to the full value of gross hoop and longitudinal stress. Continuing crack growth after the primary crack grows into the adjacent fastener hole (if any) would proceed in a similar fashion as described above in section 10.1.

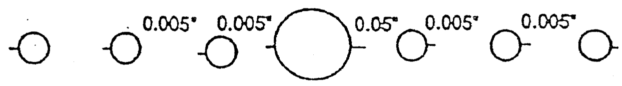


Figure 14 Antenna Connector Hole Initial Condition

11.0 CRITICAL CRACK SIZE

The critical crack size would be the crack size that would just be stable with the required residual strength stresses of Section 5.5 applied. This size sets an upper bound on the crack growth life that can be used for credit when determining inspection threshold and/or intervals. The critical crack size can be estimated using linear elastic fracture mechanics (LEFM) principles. The expression for stress intensity is used to solve for critical crack size, a_{crit}, as a function of critical stress intensity, K_C, and applied stress, σ.

$$K_C = \sigma(\pi a)^{1/2}\beta$$

Therefore,

$$a_{crit} = (1/\pi)(K_C/\sigma\ \beta)^2$$

It is typically convenient to generate a plot of σ versus a_{crit} and then enter the curve at $\sigma = f_{RES}$ (the required residual stress) to determine the critical crack size for the location being addressed. This is depicted in Figure 16.

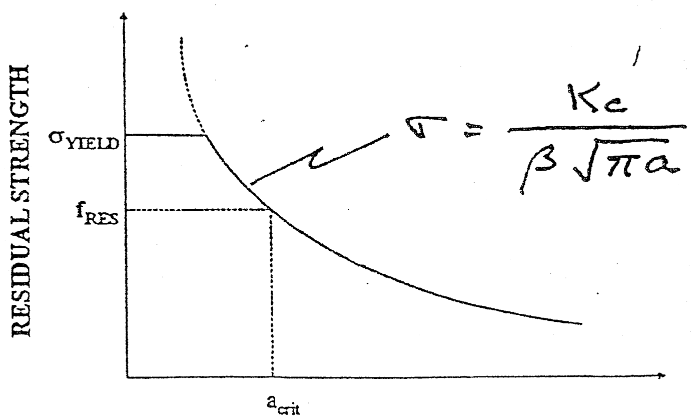


Figure 15 Residual Strength Curve

(NOTE: The curve is truncated at the gross stress that would result in net section yield. Under no circumstances should a residual strength larger than this ever be used.)

12.0 LIFE MANAGEMENT – INSPECTION REQUIREMENTS

Damage Tolerance Analysis is a life management philosophy wherin an undetectable crack is assumed to be present, and is evaluated to assess: (1) If and how quickly it will grow over time; (2) Establish a detectable crack length a_d, and a critical crack length, a_{crit}, to maintain adequate residual strength under limit loading condition; and (3) Impose inspection requirements to allow its discovery and corrective rework prior to it causing a flight safety problem.

The results of a damage tolerance evaluation can be used to establish inspection requirements that will mitigate a failure due to fatigue. Given the damage tolerance characteristics of the structure in question and a detectable crack size, one is ready to determine when the inspection should start (i.e. threshold) and how often it should be performed (i.e. interval). Inspection requirements based on the guidance provided herein are only sufficient by themselves during the period of time when normal fatigue degradation (as opposed to "rogue" or anomalous fatigue) is not expected to occur. That is, the resulting inspections are not meant to protect a structure that has exhausted its normal fatigue life, especially if the structure is susceptible to the MSD/MED threat, which can lead to widespread fatigue cracking. Recommendations for managing MSD/MED have been proposed by the AAWG in Reference [3] and this same group are currently drafting a proposed rule and advisory material relative to WFD under the ARAC process.

NOTE:
MSD = Multi Site Damage
MED = Multi Element Damage
AAWG = Aging Aircraft Working Group
WFD = Widespread Fatigue Damage
ARAC = Aviation Rulemaking Advisory Committee

12.1 INSPECTION THRESHOLD

The inspection threshold or first inspection is the time in cycles or flight hours that inspections should begin. The inspection threshold, N_1, is established as the lesser of:

N_d = cycles to detectable crack length, a_d (see Section 12.3)

$\frac{1}{2} N_{cr}$ = ½ of cycles to critical crack length, a_{crit}

12.2 INSPECTION INTERVAL

The inspection interval is established to assure that there are at least two inspections before a critical crack develops. The interval for repetitive inspections, N_R, is:

$$N_R = (N_{cr}-N_1)/2$$

A graphic description of determining threshold and interval from the damage tolerance characteristics is presented in Figure 17.

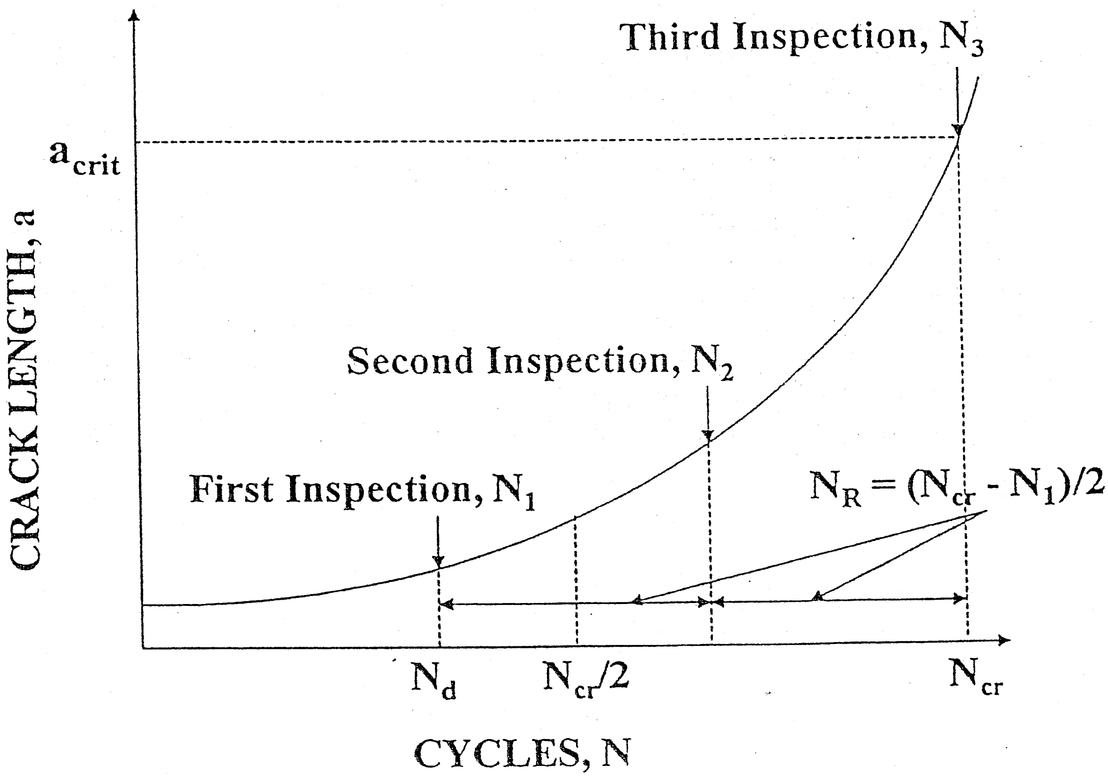


Figure 17 Inspection Requirements

12.3 DETECTABLE CRACK LENGTH

The Non-Destructive Inspection (NDI) procedures employed determine the detectable flaw size. Industry and Agency consensus has been to establish the detectable crack length as that which can be found with a 90% Probability of Detection (POD). Common NDI methods include Visual (3 to 5x magnifying glass), Penetrant (dye or fluorescent), Magnetic Particle (for ferromagnetic material), X-Ray (radiographic), Ultrasonic, and Eddy Current (high and low frequency).

Table 2 presents reasonable estimates of detectable crack length with a 90% POD. Of these, for antenna installations, the surface probe Eddy Current method using graduated templates is preferred.

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

* Only primer is allowed on unpainted surfaces.

13 REFERENCES

1. RAPID website – *http://aar400.tc.faa.gov/rapid;* For more information contact Dr. Xiaogong Lee at 609-485-6967 or xiaogong.lee@faa.gov.
2. Los Angeles Aircraft Certification Office Policy Memo, "INFORMATION: ANM-120L Damage Tolerance Philosophy, dated May 3, 1995.
3. Airworthiness Assurance Working Group for the Aviation Rulemaking Advisory Committee – Transport and Engine Issues, "Recommendations for Regulatory Action to Prevent Widespread Fatigue damage in the Commercial Airplane Fleet", March 11, 1999.
4. D. Hagemaier, Maintenance Engineering Plan, May 1988.