FATIGUE OF AIRFRAME STRUCTURES

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DER - Structures

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HISTORICAL BACKGROUND

- 1847 - August Wholer, director of German Imperial Railways, conducted experiments related to fatigue failures in railroad axles.
- 1903 - First flight of Wright Flyer postponed due to fatigue failure of hollow propeller shaft.
- 1920 - A.A. Griffith of the Royal Aircraft Establishment publish the first theory on fracture mechanics.
- 1942-45 - Loss of over 200 Liberty transport ships during WWII due to fatigue failures in hull welding.
- 1954 - Catastrophic in-flight fatigue failure of first jet commercial transport.
- 1969 - USAF bomber suffered catastrophic failure due to manufacturing induced flaw.

HISTORICAL BACKGROUND

- The DeHavilland Comet was the first operational jet commercial transport in the world.

- Comet design suffered from poorly designed structural details and high operating stresses which resulted in the catastrophic failure of the fuselage on two occasions.

- Industry allarmed to dire consequences of metal fatigue in airframes. All future airframes were subjected to thorough structural fatigue testing. In addition, structures were analyzed for fatigue using stress-life methods and by accounting for stress concentrations (Kt).

HISTORICAL BACKGROUND

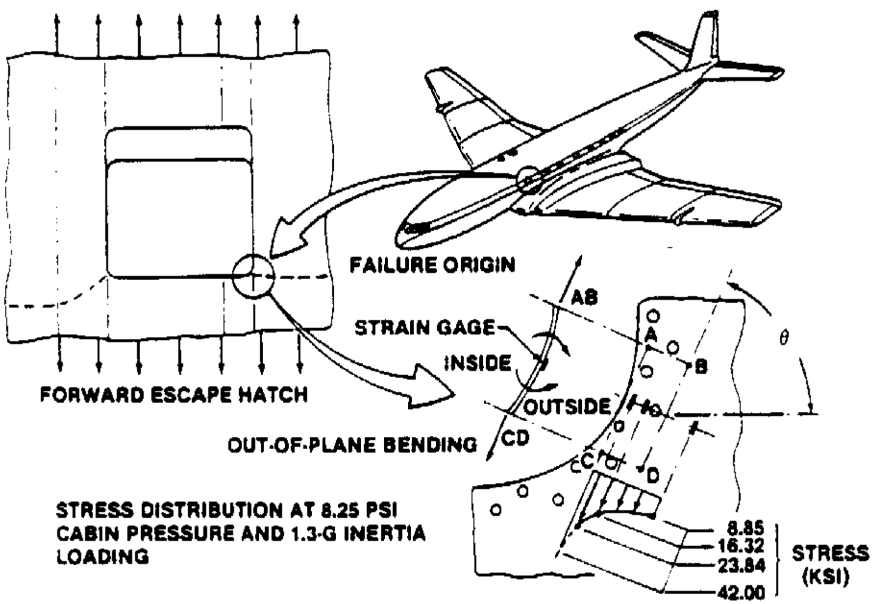


FIGURE 2. PROBABLE FAILURE ORIGIN — COMET I YOKE UNCLE TEST AIRCRAFT

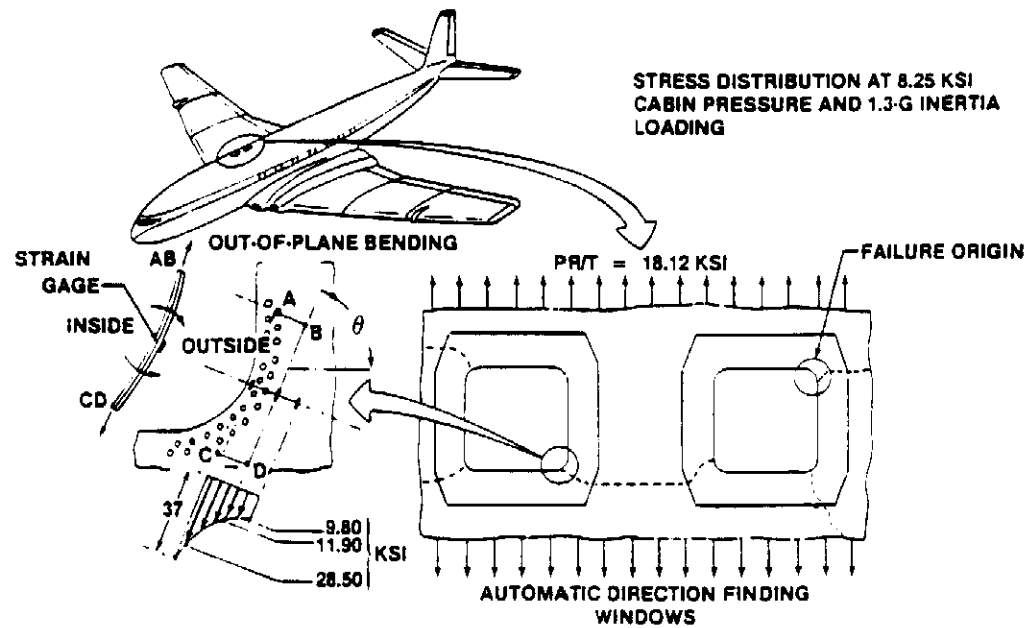
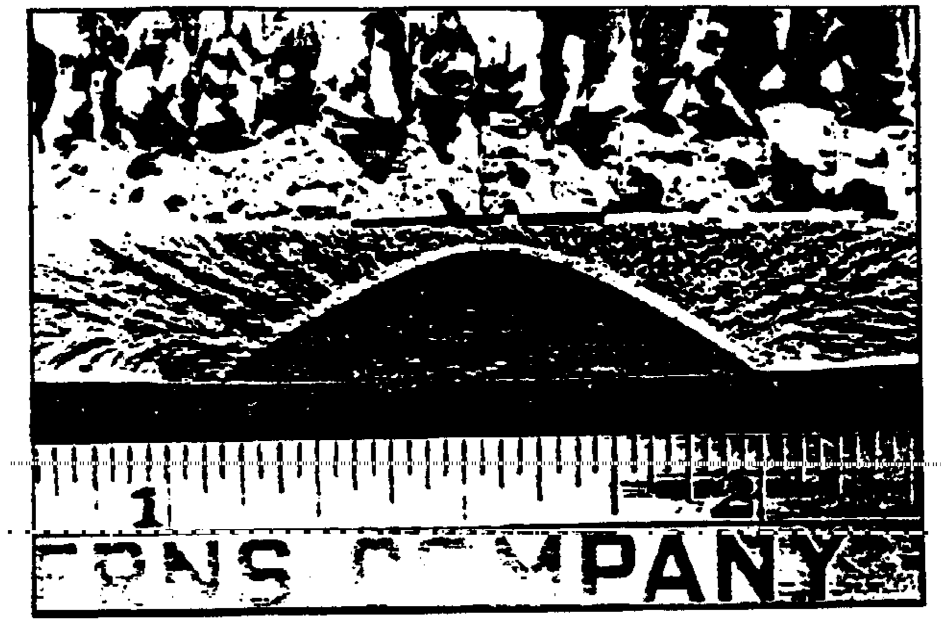


FIGURE 3. PROBABLE FAILURE ORIGIN — COMET I YOKE PETER

- Comet failures attributed to unexpected high corner stresses near window cutouts. Out-of-plane bending was not accounted for in initial analysis. This was later verified by testing.

HISTORICAL BACKGROUND

- Catastrophic failure of F111 occurred due to a manufacturing flaw in the lower wing cover root. Flaw was not detectable due to honeycomb panelling on outer surface.



- The results of this investigation led the USAF to completely change their fatigue design criteria to one which incorporated fracture mechanics and the damage tolerance philosophy.

HISTORICAL BACKGROUND

- The primary objective of the damage tolerance philosophy can be stated as:

  "A fracture mechanics evaluation of the structure under typical load spectra must show that catastrophic failure due to fatigue and accidental damage will be avoided throughout the operational life of the aircraft."

- This approach is accomplished by performing crackgrowth analyses and establishing periodic structural inspections based on the time to reach the critical crack length.

- The damage tolerance approach was adopted by both the USAF and the FAA because it is the most reliable method by which inspections can be performed.

HISTORICAL BACKGROUND

- In 1988, a 737 in Hawaii experienced catastrophic failure of the forward fuselage upper cabin.

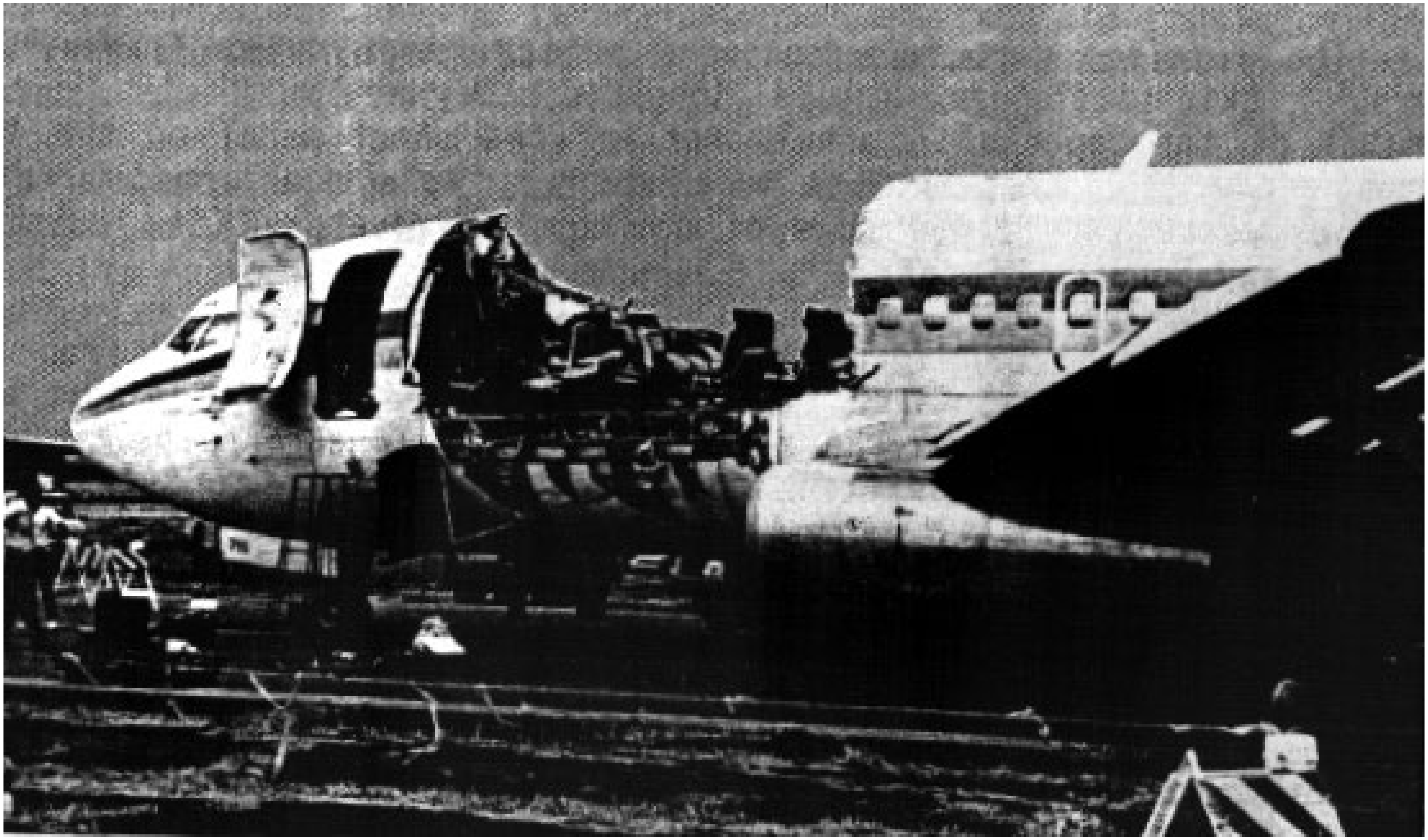
- Failure of the upper fuselage skin of the Aloha 737 was precipitated by disbonding of the longitudinal skin splices and multi-site fatigue cracks.

- The Aloha 737 had over 60000 flight cycles which was well above the original design goal of the aircraft.

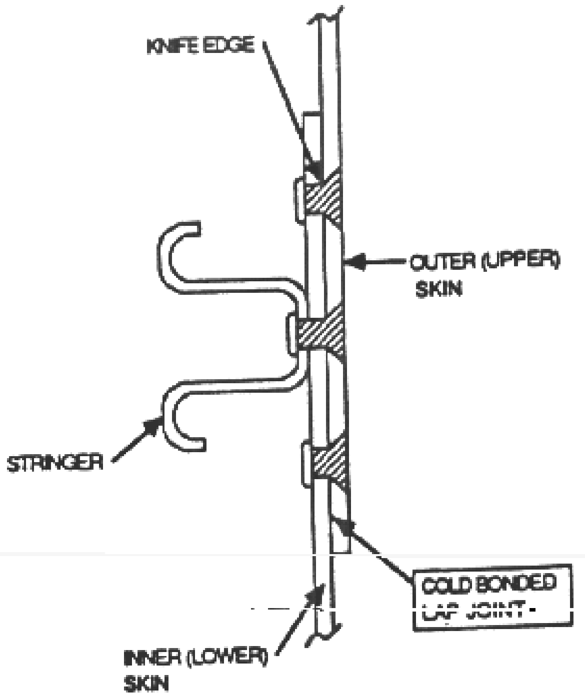
- The large number of multi-site fatigue cracks in the skin precluded the arresting of the cracks by adjacent frame members.

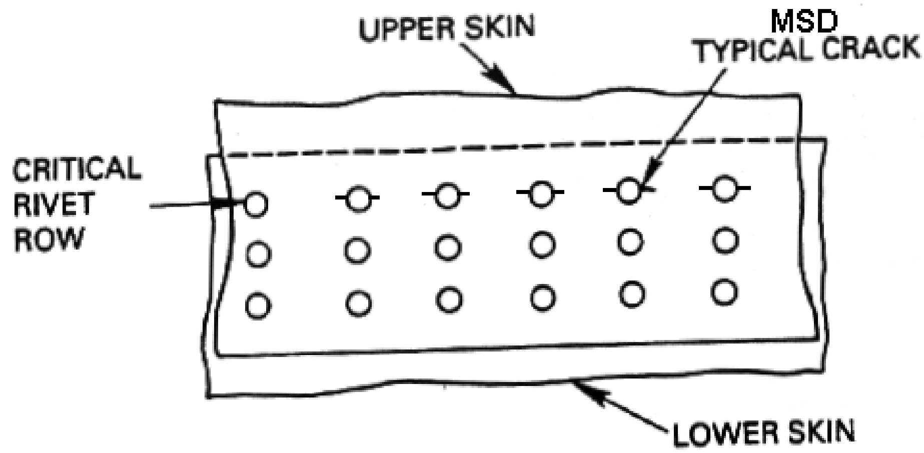
- As a result of this disaster, the issue of aging aircraft operation and the effect of wide spread fatigue damage have gained a great deal of attention by the entire industry.

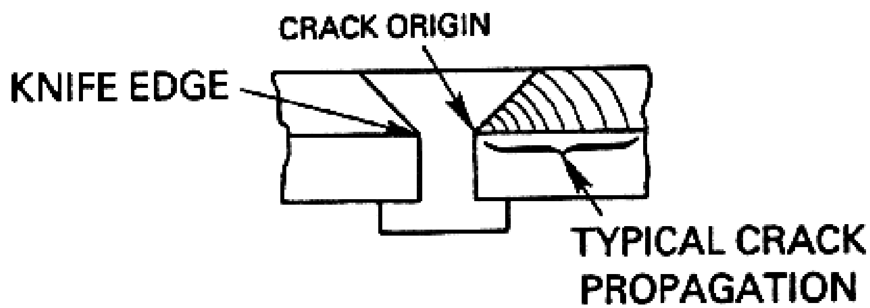
HISTORICAL BACKGROUND



HISTORICAL BACKGROUND

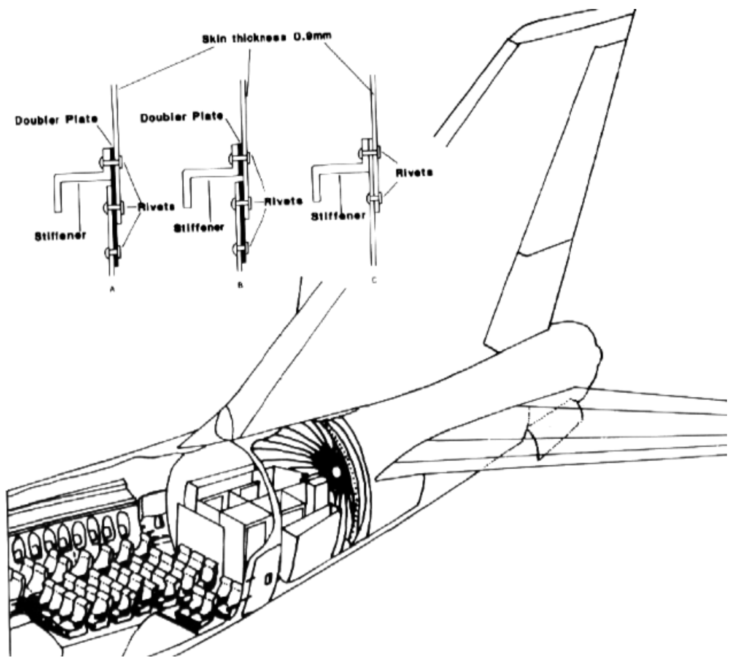






HISTORICAL BACKGROUND

- August 1985, Japanese Airlines Flight 123, a 747 registered as JA8119, flying from Tokyo to Osaka crashed into the side of Mount Osutaka killing 520 passengers.
- Cause of crash due to loss of cabin pressure and flight controls following separation of aft fuselage and empennage. Cause of separation attributed to fatigue failure of aft pressure bulkhead. Fatigue failure caused by incorrectly installed repair.



FATIGUE ANALYSIS METHODS

- Stress life methods have been employed by the aircraft industry for decades in determining the basic fatigue life of structural components.

- S-N Curves have been developed for numerous materials and stress concentrations identified for many structural configurations.

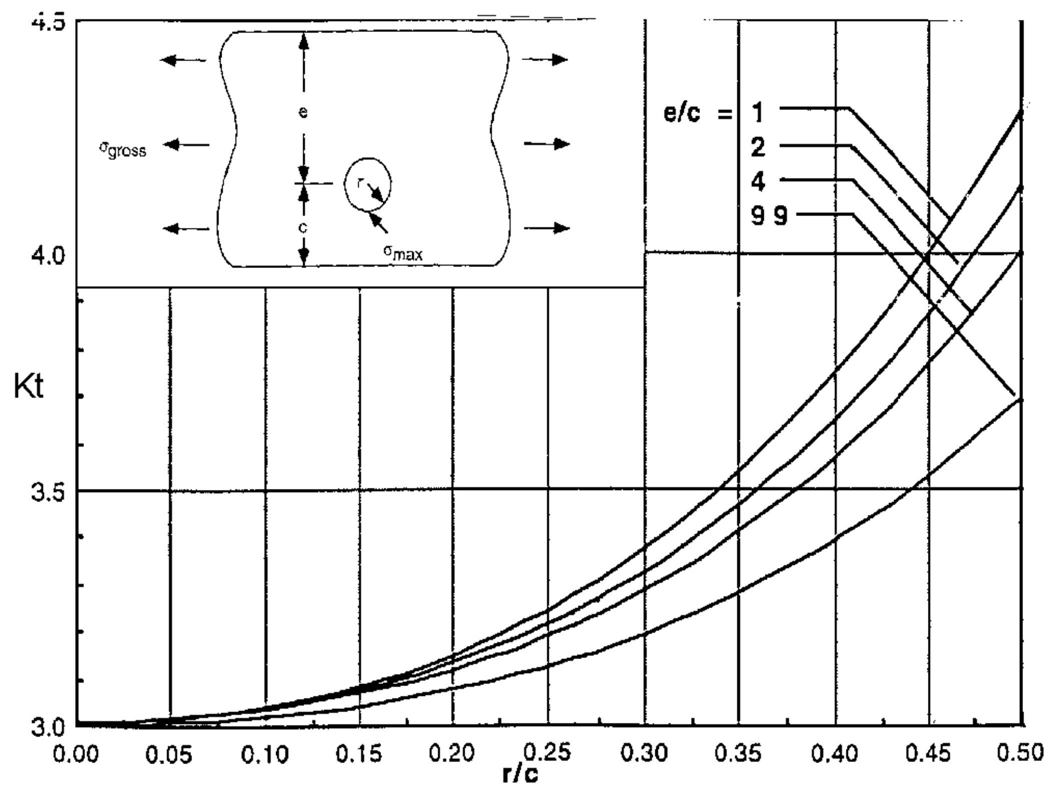
- Typically the basic Palgrin-Miner's damage accumulation method is employed to determine fatigue life from these curves.

- The limitation of this method is that it can only determine the time to total failure. This results in life limited components. This method does not account for accidental damage either.

FATIGUE ANALYSIS METHODS

- Fatigue analysis example:

Stress concentration effects can include: holes, pin loading, countersinks, fillets, ect.
Determine Kt for an open hole with r/c=.45 and e/c=1:   Kt=4.0

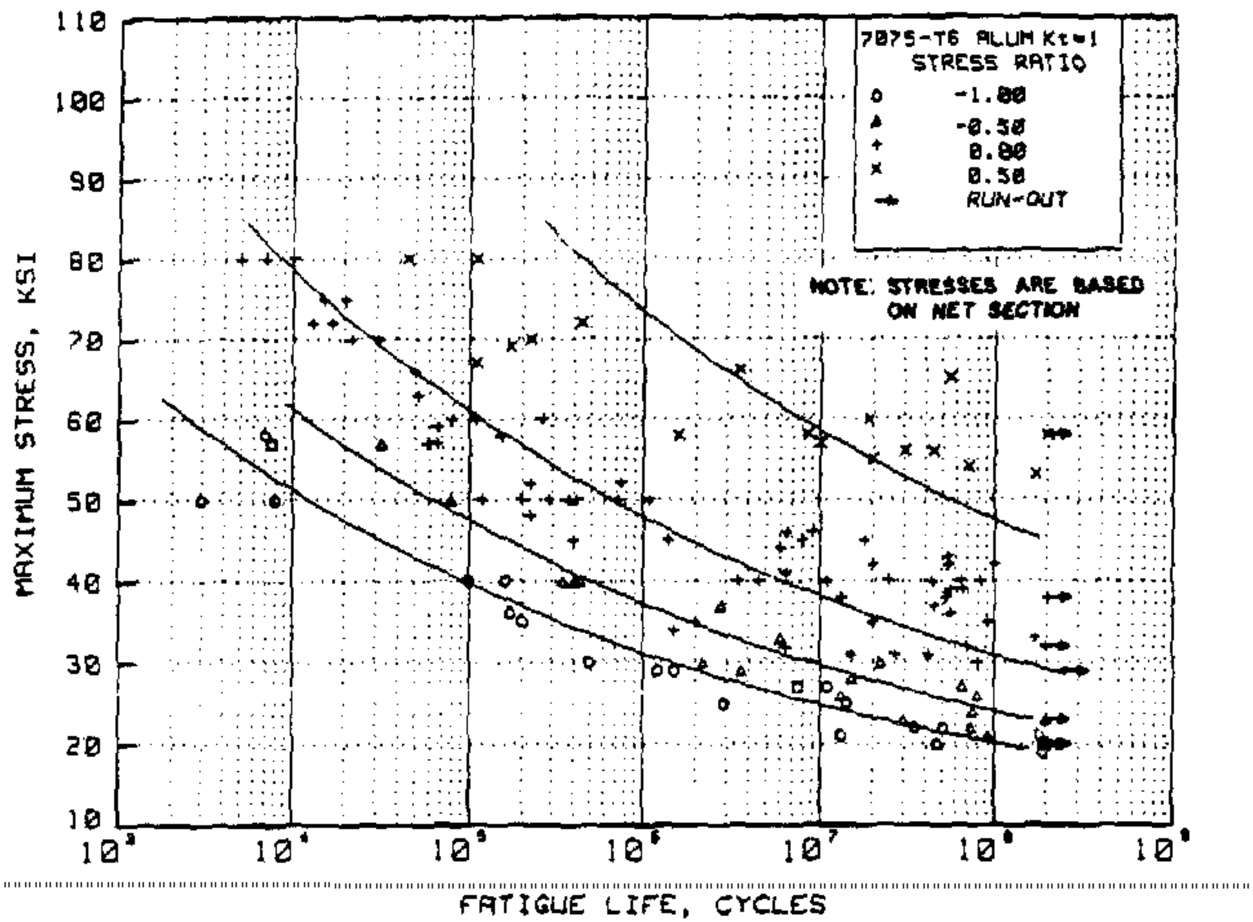


FATIGUE ANALYSIS METHODS

Determine number of life cycles for 7075-T6 sheet:

applied stress = 10 ksi, R ratio = -1.0

Kt x 10 ksi = 40 ksi, Life = 100000 cycles



DAMAGE TOLERANCE METHODS

- The damage tolerance method is founded on the use of fracture mechanics to predict crack propagation. However, there are many other inputs to be considered. The following are the key elements to the the damage tolerance method:

1. Stress Intensity
2. Material Crack Growth Rates
3. Fracture Toughness
4. Repeated Loads
5. Internal Loads and Stress
6. Spectrum Development
7. Crack Growth Analysis
8. Residual Strength Analysis
9. Test Correlation
10. Inspection Criteria and Techniques

STRESS INTENSITY

- The basic equation employed in fracture mechanics is:

$$\mathbf{K} = \sigma\, \beta\, \sqrt{\pi \mathbf{c}} \qquad \text{Stress Intensity}$$

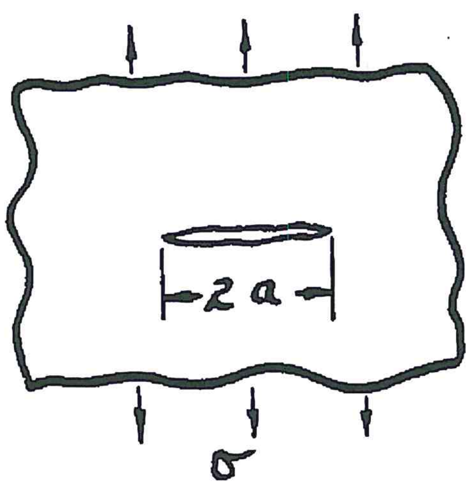
σ = stress β = geometric correction $\mathbf{c}$ = crack length

- K is a measure of the intensity of stress at the crack tip. It is derived from the theory of elasticity.
- K varies with structural geometry, stress level and crack length. Numerous solutions exist for various geometries.
- The change in stress intensity is used to determine the crack growth rate propagation with respect to applied cyclic loading.

STRESS INTENSITY

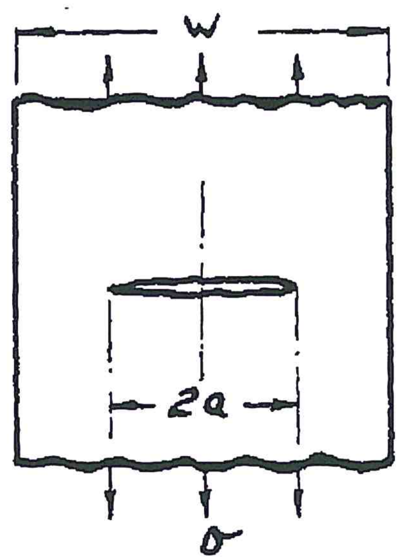
- The effects of structural geometry are typically categorized into various Beta factors. The simplest solution and also the baseline for most others developed is a center thru crack in an infinite plate:

Thru-crack, infinite width



Beta = 1.0

Thru-crack, finite width

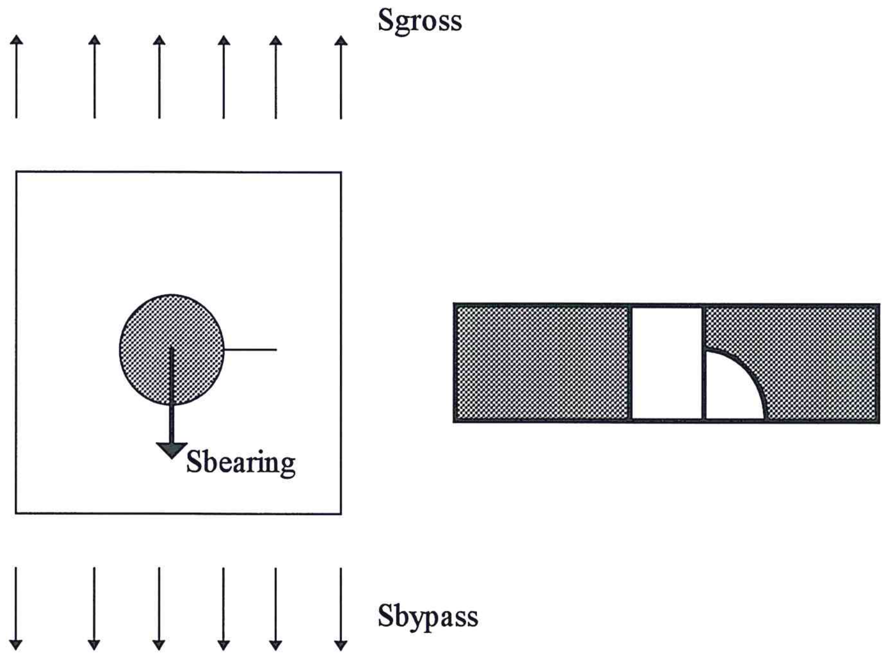


$$\text{Beta} \approx \left[\sec \frac{\pi a}{W}\right]^{1/2}$$

Beta varies due to edge effects

STRESS INTENSITY

- Most typical geometric correction factors for aircraft structures include the following:



- Betas are derived for various different configurations using several methods:

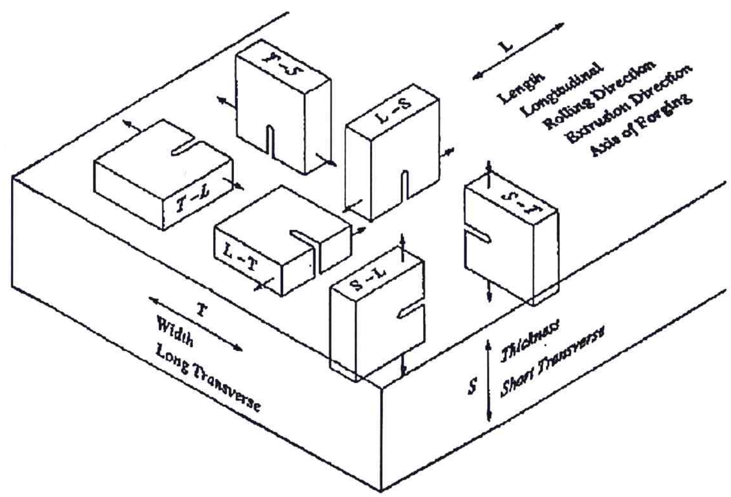
  1. Superposition of basic handbook solutions
  2. Compounding of basic handbook solutions
  3. Finite Element or Boundary Integral Analysis

MATERIAL CRACK GROWTH RATES

- Crack growth rate is a measure of the rate of change in crack length with respect to applied cycles. Typical use of rate data is either in the form of an equation fit or tabular data. The following is an example of a basic slope equation fit called the Paris Law:

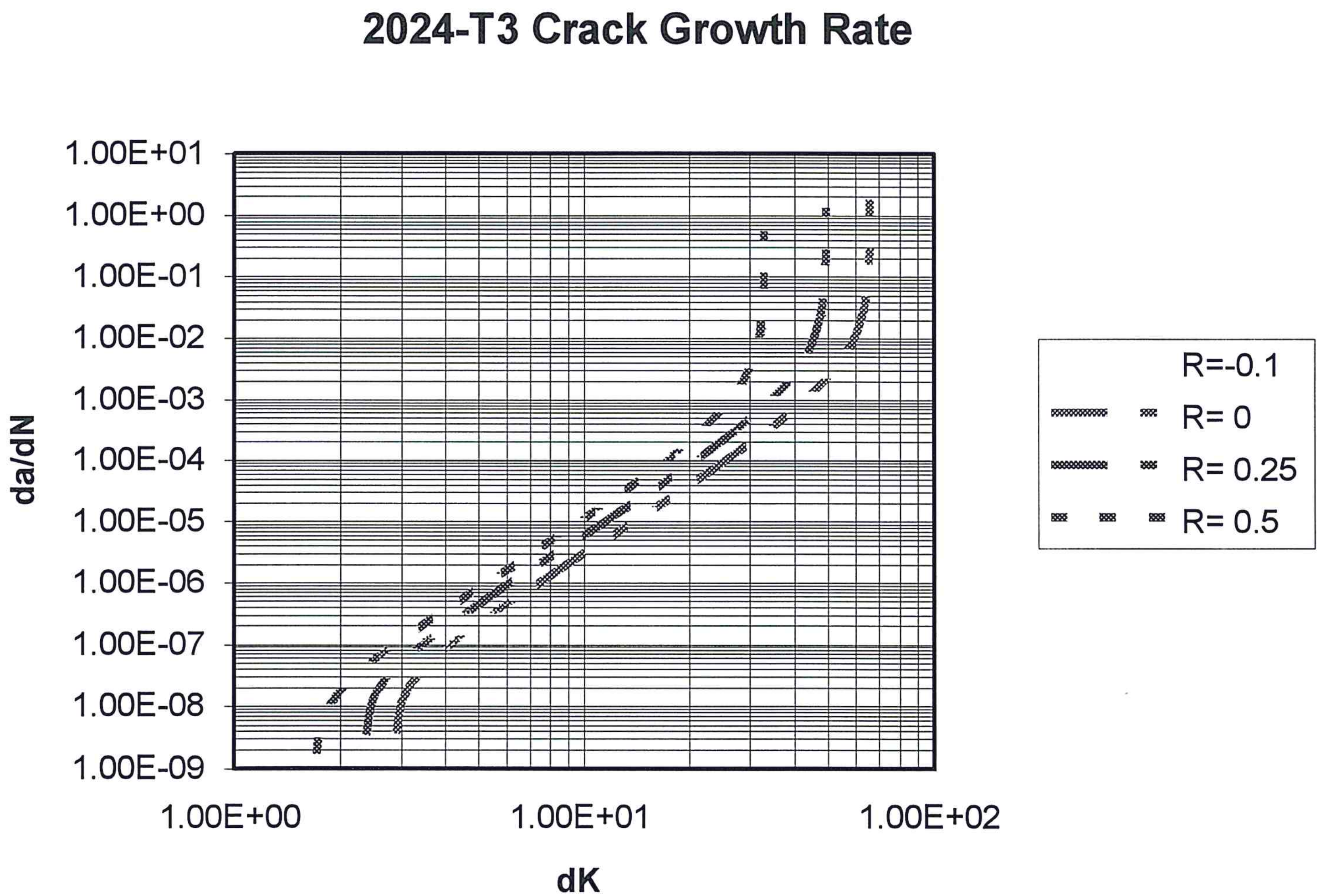
$$\mathbf{dc/dN = C\Delta K^{n}} \qquad \text{Crack Growth Rate}$$

- Growth rates are dependant upon:
    1. Environment
    2. Grain Orientation



MATERIAL CRACK GROWTH RATES

- Crack growth curves are developed for various R ratios.

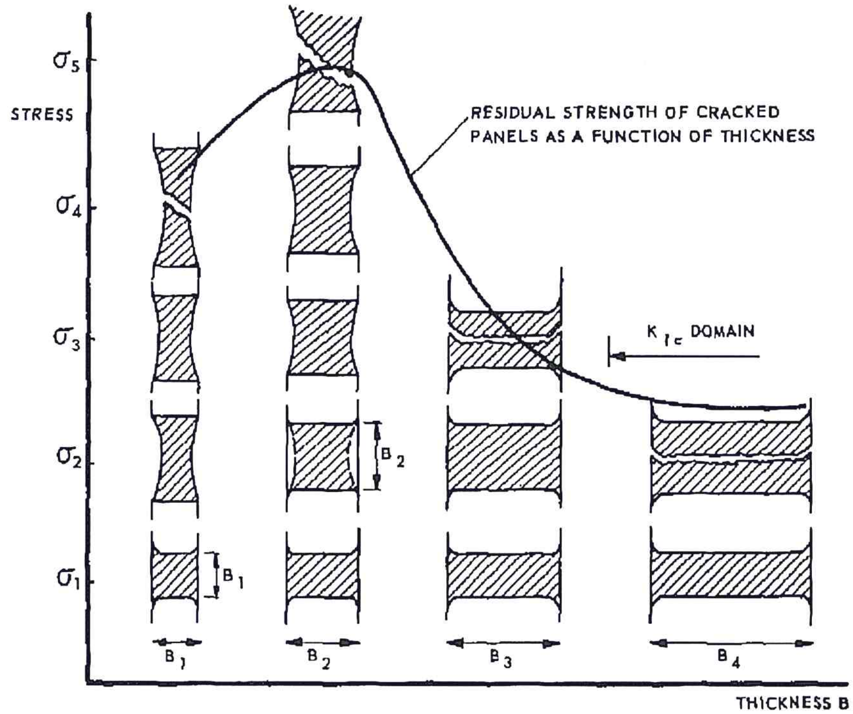


FRACTURE TOUGHNESS

- Fracture toughness is a material allowable which characterizes the point at which fast fracture occurs in a given material.
- There are two modes of toughness: plane strain and plane stress.

K_{IC}= plane strain

K_C= plane stress



REPEATED LOADS

- A key essential to any damage tolerance assessment is the establishment of a loading spectrum. For aircraft, repeated load spectra are developed from the following data:

1. Mission Profile and Usage
2. Loading Environments
3. Loading Histories
4. Dynamic Effects

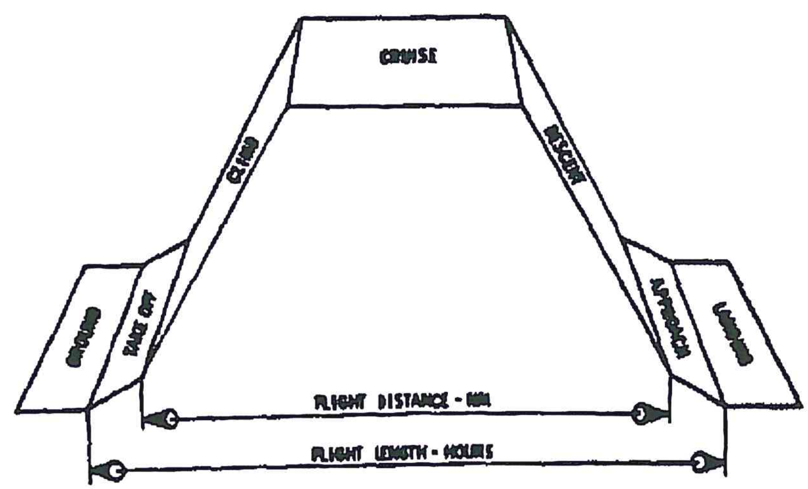
REPEATED LOADS

- Typical mission profiles include all pertinent flight segments and flight parameters. This data is used in developing the required loading conditions. A standard transport might have five different basic flight types such:

1. Short   3. Medium   5. Ferry

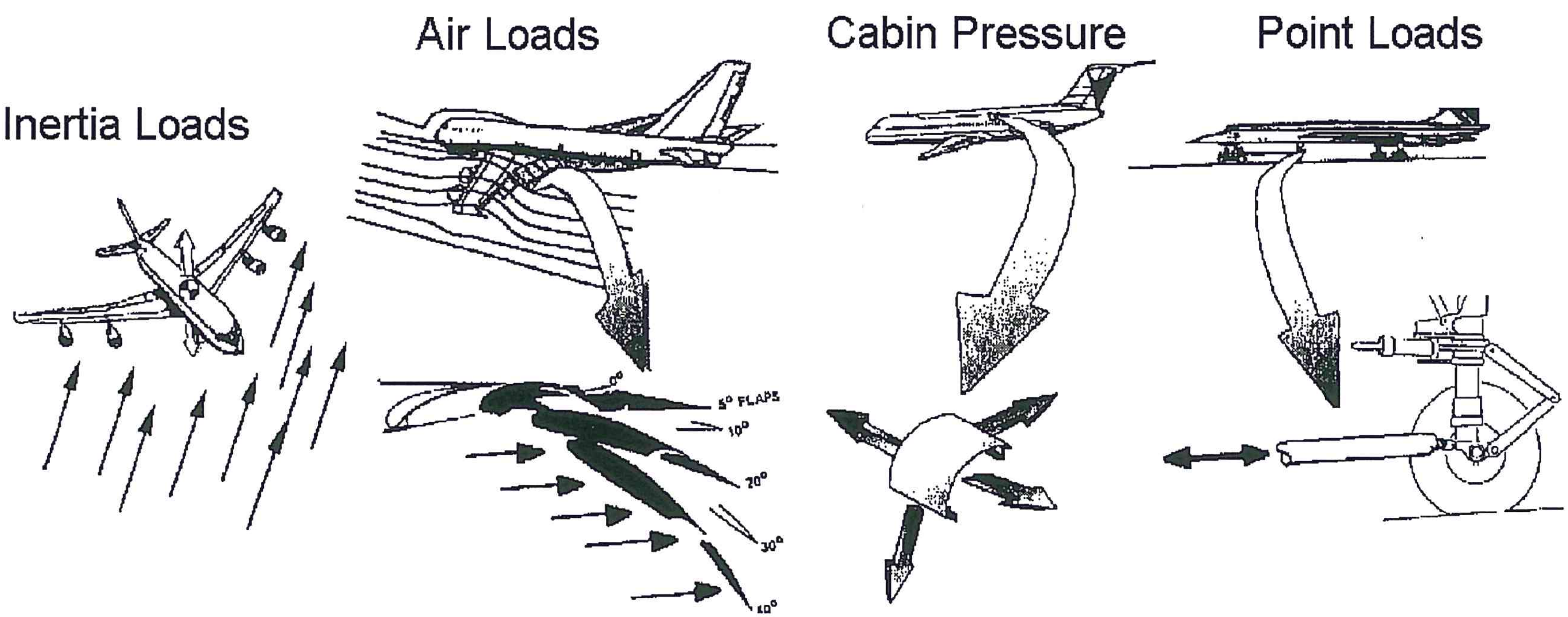
2. Long   4. Training

SEGMENT	NUM	CONDITIONS	ALT	INT P	SPEED	FLAP	WEIGHT	THRUST	CG	DIST	MIN
	1	UNLOADED	0	0.00		0	180.9	0			
	2	CONT CHECK					336.0				
GROUND	3	TOW					336.0				
	4	PIVOT					336.0				
	5	GROUND TURN					336.0		19.3		
	6	BRAKED ROLL					336.0				
	7	TAXI OUT				0	336.0	0			
	8	TAKEOFF ROLL			142	14	330.6	11250	19.9		
TAKEOFF	9	ROTATION			150		330.6	11250	19.9		
	10	LIFTOFF	0		150		330.6	11250	19.9		
	11	GEAR RETRACT	1000	0.00	196		330.6	11250	19.9		
	12	DEPART(F DN)	1000	0.50	216	14	312.0	10950	19.5		
	13	FLAP RETRACT	1000	0.50		14 TO 0	312.0	10950	19.5		
CLIMB	14	INIT CLIMB	3000	1.45	250	0	312.0	10460	19.5	35	8
	15	FINAL CLIMB	17000	7.10	294	0	312.0	7430	19.5	157	32
CRUISE	16	CRUISE	34000	8.60	237	0	302.3	4010	20.5	299	76
DESCENT	17	INIT DESCENT	19000	7.60	252	0	231.5	1000	28.3	43	10
	18	FINAL DESCENT	3000	1.45	258	0	231.0	2000	28.2	13	3
	19	FLAP EXTEND	1000	0.50		0 TO 50	231.0		28.2		
APPROACH	20	APPR (F DN)	1000	0.50	199	50	231.0	8754	28.2		
	21	ROLL MANEUVER	1000	0.50	199	50	231.0	8754	28.2		
	22	YAW MANEUVER	1000	0.50	199	50	231.0	8754	28.2		
	23	GEAR EXTEND	1000	0.50	199	50		8754	28.2		
LANDING	24	FLARE	0		141	50		0	28.2		
	25	TOUCH DOWN	0		141	50		0	28.2		
	26	LANDING ROLLOUT	0		97	50	230.0	-6730	28.0		



REPEATED LOADS

- All load sources contributing to metal fatigue must be accounted for. The repeated loads imposed on an airframe can be generally described by 4 types:



REPEATED LOADS

- Load histories are the cornerstone for generating fatigue spectra. Histories are a statistical database of loads generated over time.

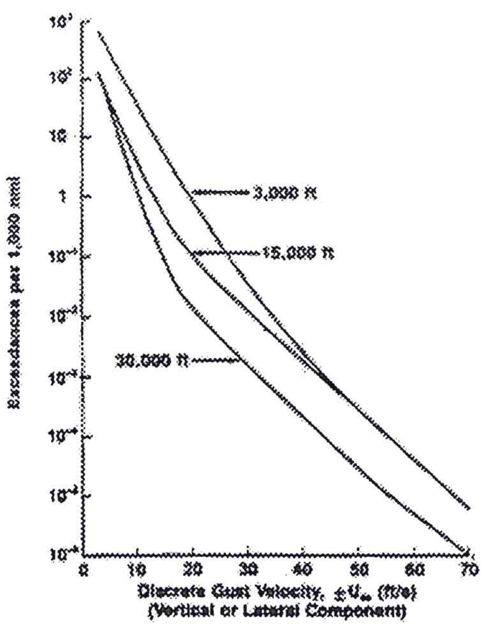
- Typical load histories are generated through flight testing. They generally include the number of cycles observed of a particular load level. Some examples are:

  1. Vertical Gust Velocity (Ude) versus Cycles by altitude
  2. Vertical Acceleration (Nz) versus Cycles for maneuvers
  3. Vertical Acceleration (Nz) versus Cycles for taxi
  4. Control Surface Hinge Load versus Cycles

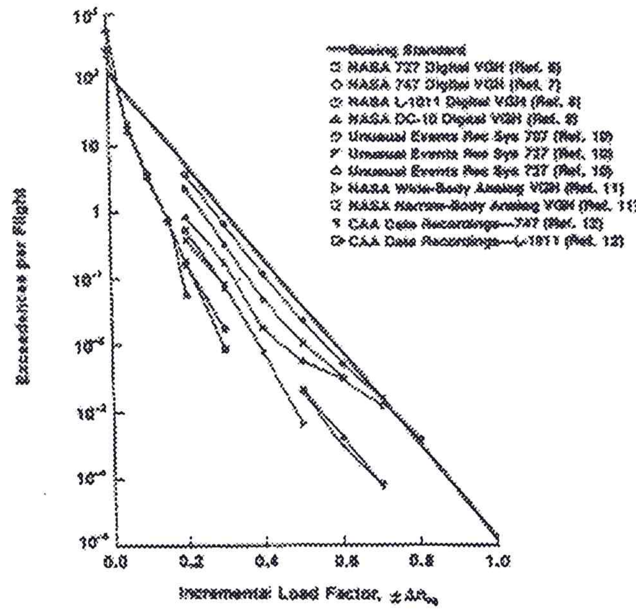
REPEATED LOADS

- Examples of typical load histories are:

Gust



Maneuver



Comparison of Recorded Flight Maneuver Spectra

REPEATED LOADS

- In addition to load sources, dynamic effects on the airframe must be accounted for. Typically these affects are usually related to the following conditions:

    1. Gust Turbulence
    2. Two Point Landing
    3. Taxi

- Dynamic responses to these loadings are determined for affected portions of the airframe typically through finite element analysis.

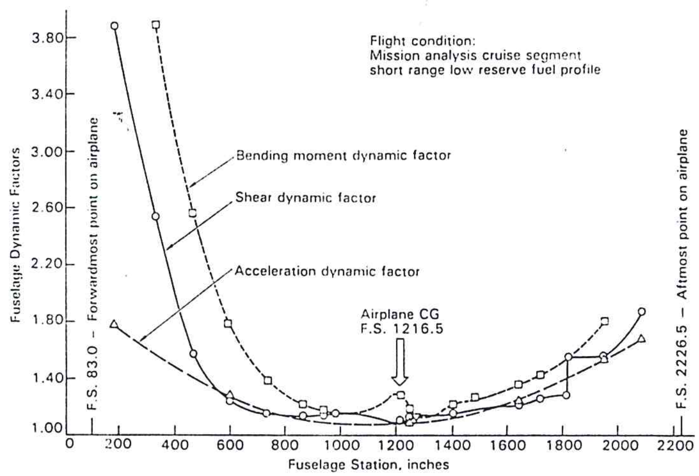


Fig. 8.10b Sample dynamic factors for gust loads: L-1011 fuselage.

INTERNAL LOADS and STRESS

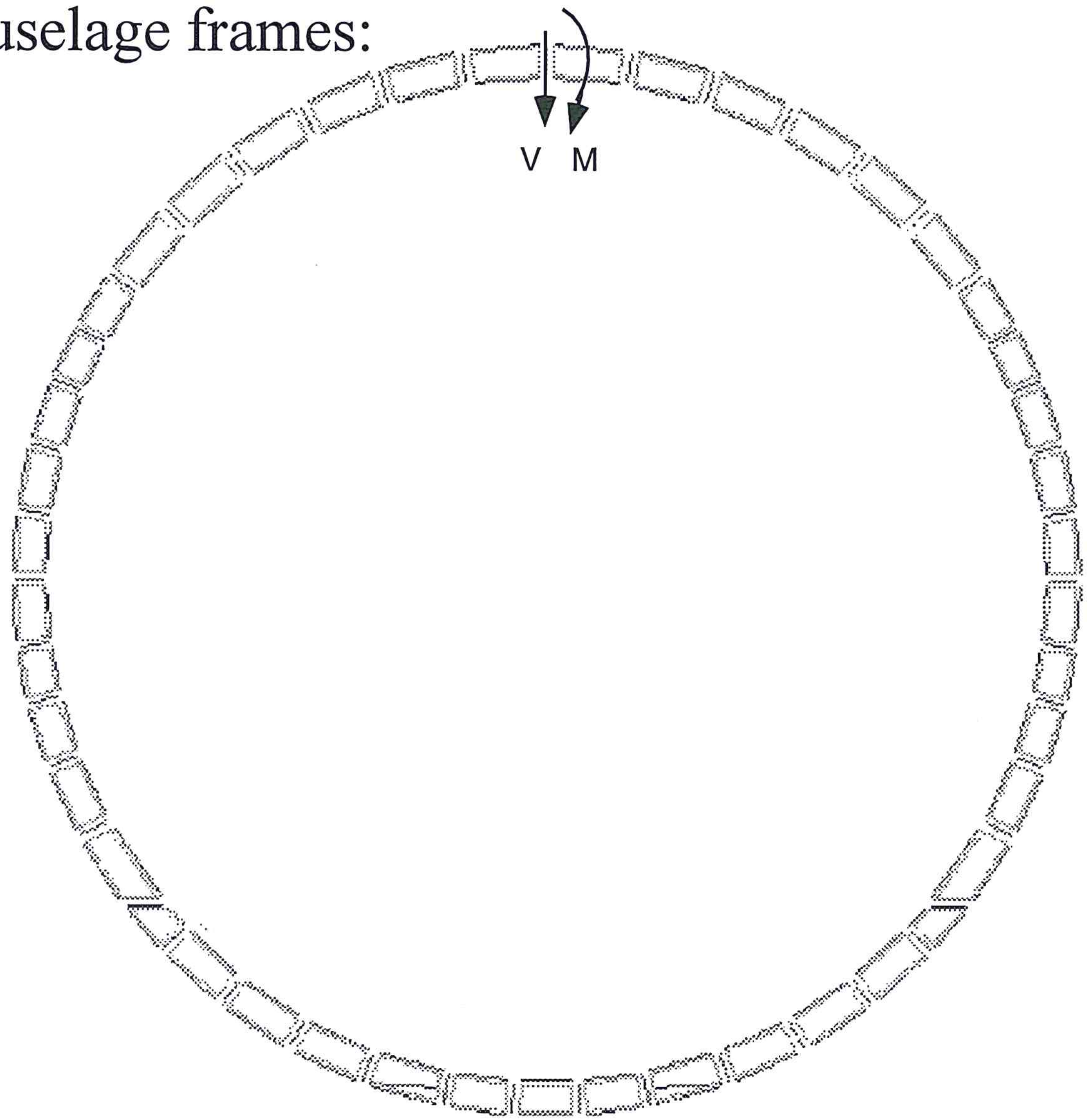
- In order to perform an analysis of a detailed structural component, loads internal to the airframe must be obtained for all damaging load sources.

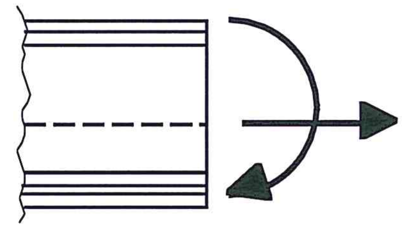
- This data is typically obtained from coarse grid finite element models of the overall airframe. External loads for all fatigue conditions are applied to the FEM.

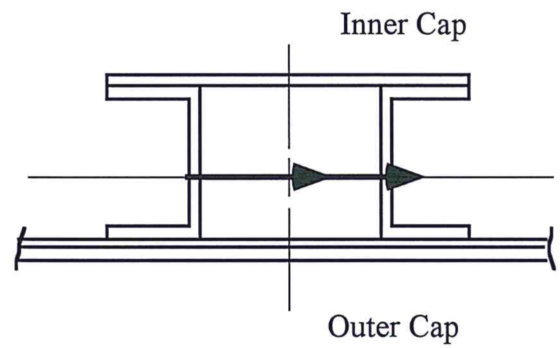
1. **Aerodymic Loads and Pressures - applied as panel loads**
2. **Inertia Loading - applied as distributed masses with load factors**
3. **Landing Loads - applied as discrete forces to gear**

INTERNAL LOADS and STRESS

From results of the coarse grid model, internal loads are assembled for parts of the structure being analyzed. For example, inner and outer cap loads from the FEM are used to develop shear and moment curves for fuseláge frames:







$$f_{ten} = Mc/I + P/A$$

SPECTRUM DEVELOPMENT

- The stress history of a structure is essential in determining its fatigue life. Stress spectra are typically generated from internal loads obtained by FEM.
- Stress analyses are perfomed for each fatigue load condition. Once these stresses are determined, a history is generated by randomly selecting alternating load levels from load exceedance tables. For example:

Assume that a stringer has the following loading:

$$\text{Stress}_{1\text{gCruise}} = 10 \text{ ksi}$$
$$\text{Stress}_{1\text{fpsGust}} = 2 \text{ ksi}$$

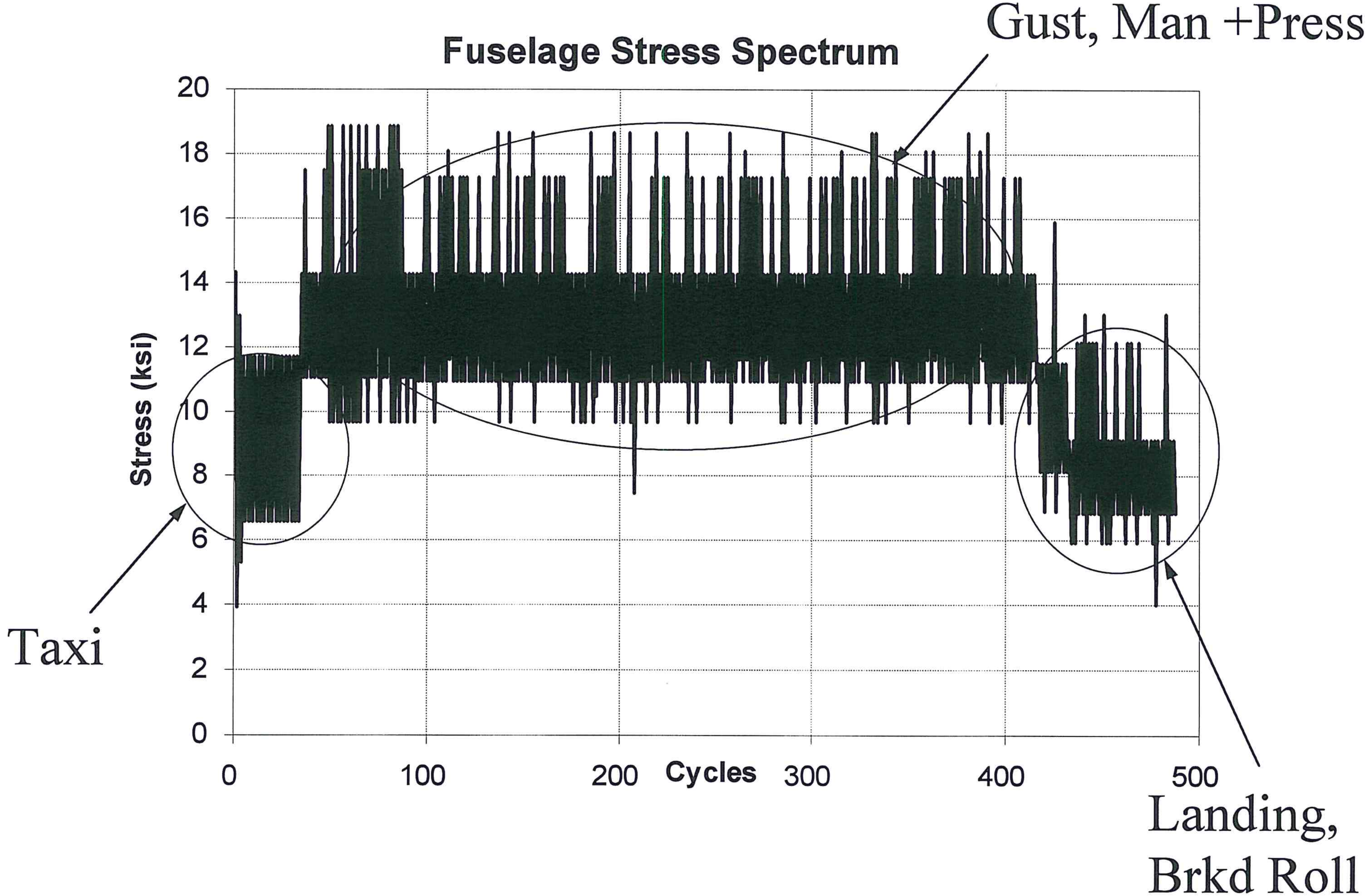
By randomly selecting from a gust exceedance table:

Gust - Ude = 5 fps

Max stress = 10 + 2 x 5 = 20 ksi

Min stress = 10 - 2 x 5 = 0 ksi

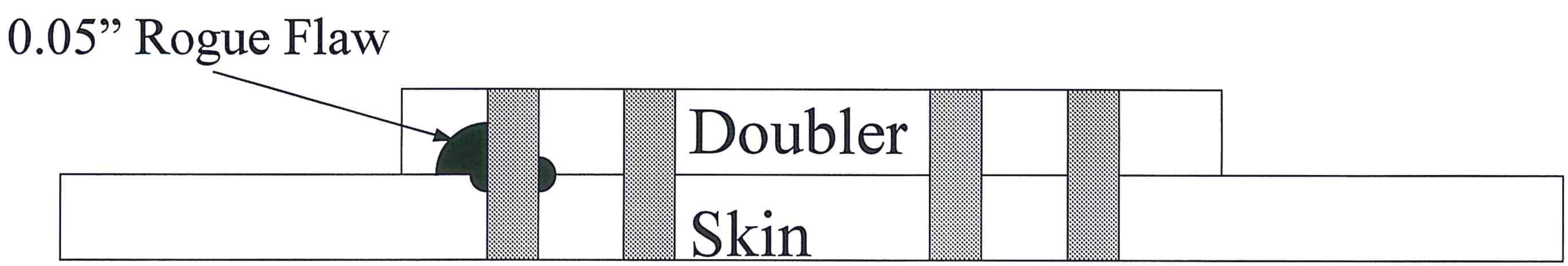
SPECTRUM DEVELOPMENT



CRACK GROWTH ANALYSIS

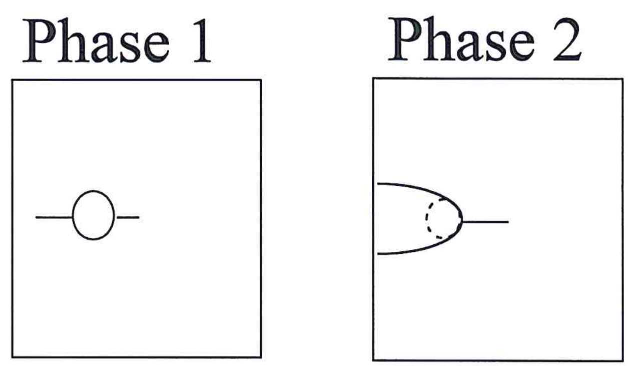
- A crack growth analysis is performed by calculating the total cumulative damage growth rate for each applied fatigue stress from an initial flaw.
- In order to determine the total fatigue life for a structure, both an initial flaw due to manufacturing and a quality flaw must be accounted for. Therefore, crack growths are performed in different phases beginning with the 0.05 inch flaw and then all other continuing damage sites. Standard industry assumptions for initial flaw sizes are:

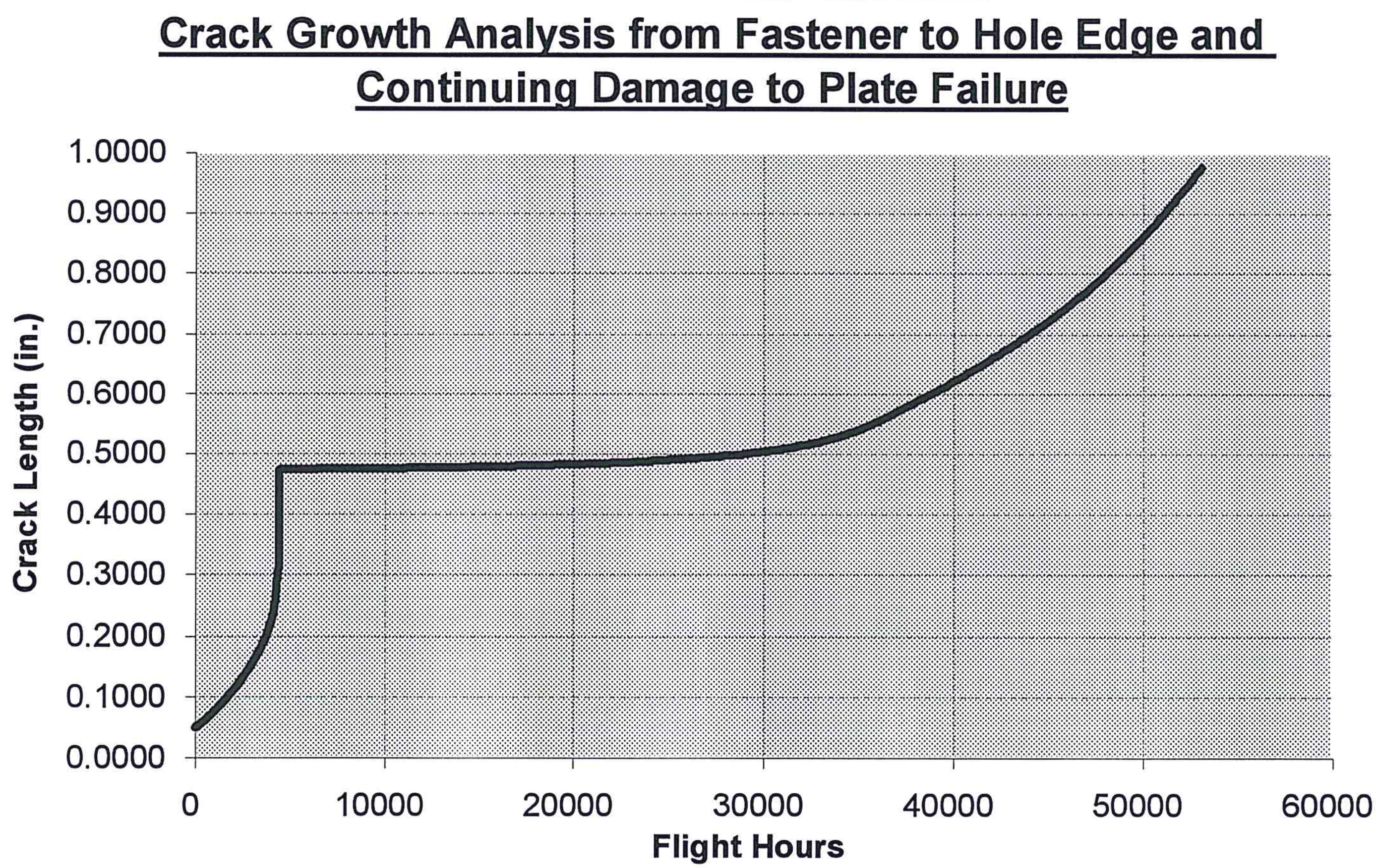
    1. Initial rogue flaw: 0.05” Corner flaw at fastener holes
    2. Continuing damage: 0.005” Corner flaw at fastener holes



CRACK GROWTH ANALYSIS

Example:





RESIDUAL STRENGTH ANALYSIS

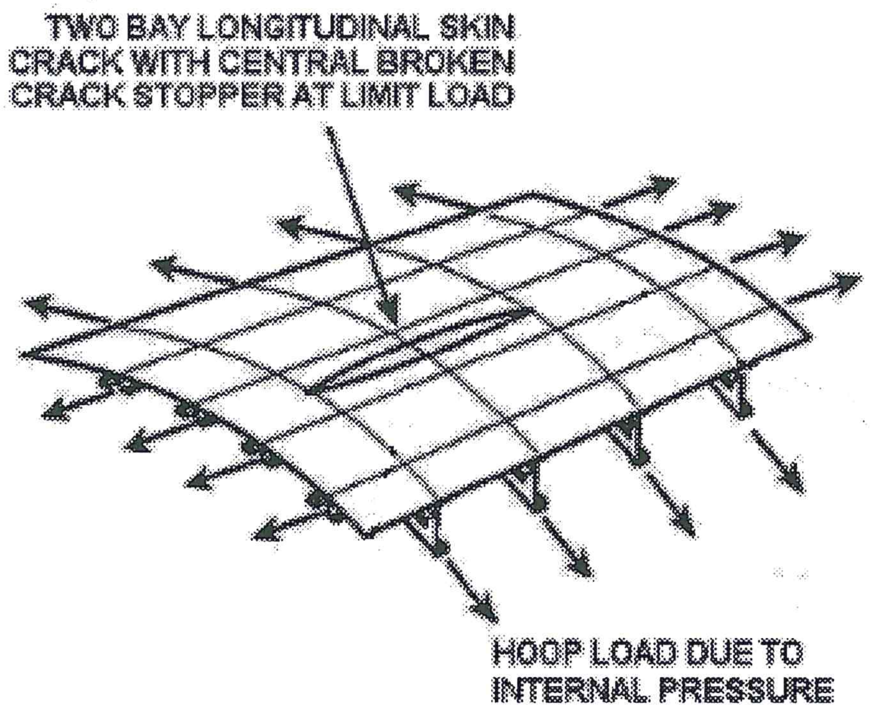
- In addition to predicting the structural fatigue life, the remaining static strength following cracking is determined. This analysis is performed by calculating the stress at each crack length in terms of the allowable fracture toughness:

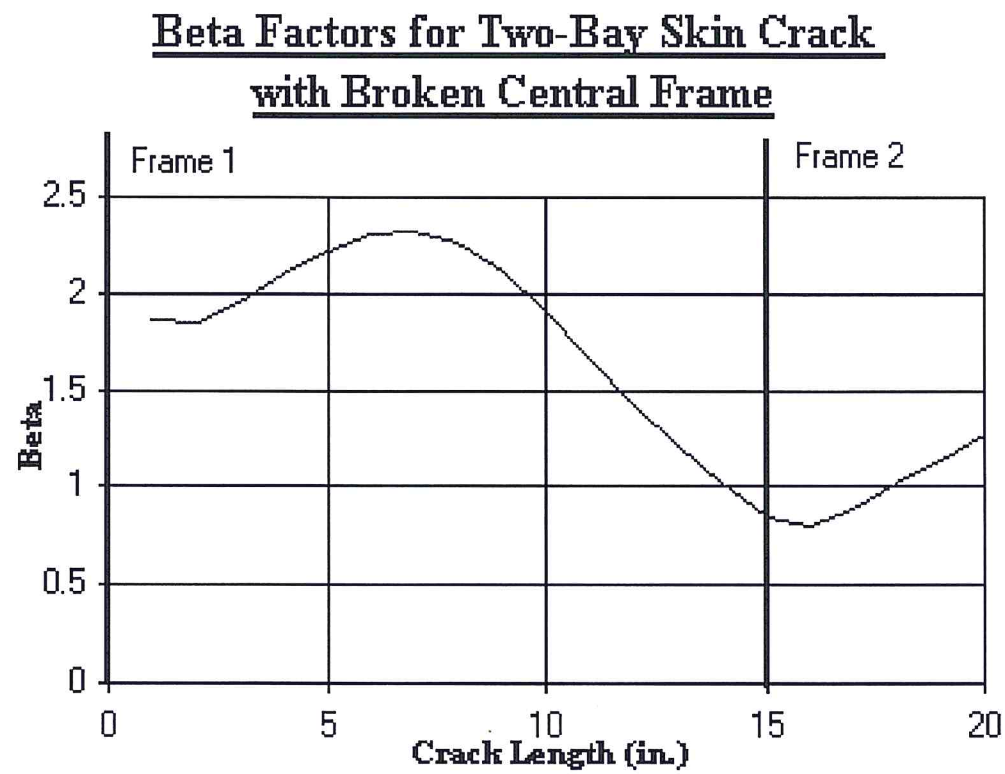
$$\mathbf{K_C / \beta \sqrt{\pi c} = \sigma_{residual}}$$

- Typical residual strength analysis are performed for stiffened panel structures. The resulting residual strength is used to determined if large obvious damage can be contained by adjacent stiffeners.

RESIDUAL STRENGTH ANALYSIS

- A typical example is a longitudinal skin crack with a broken frame in a fuselage panel. The first item needed is a beta solution:

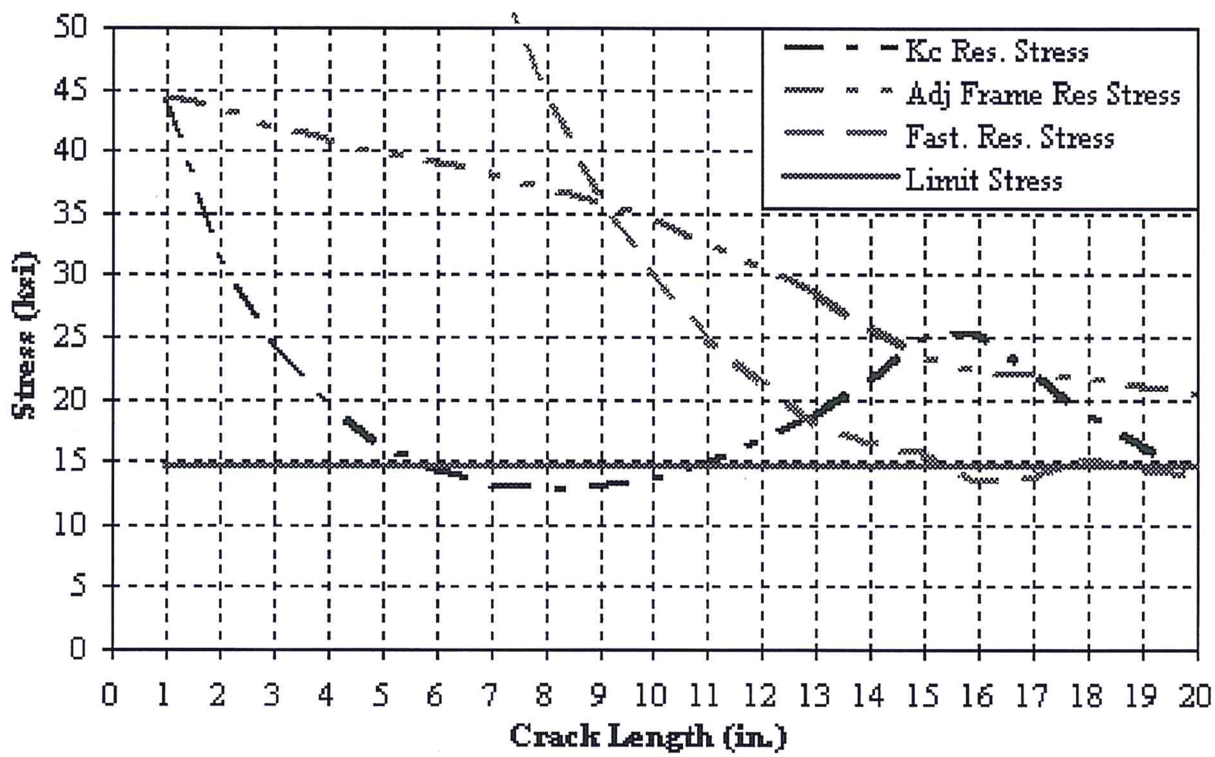




RESIDUAL STRENGTH ANALYSIS

- In addition to calculating the skin residual strength, the remaining strength in the adjacent frame is calculated by reducing its ultimate tension allowable Ftu by the increase in stress due to the failed central frame and skin crack. This results in the following:

Residual Strength Analysis for Longitudinal Skin Crack with a Failed Frame



TEST CORRELATION

- In order to validate the analytical crack growth predictions, testing is required. This can be in the form of coupons, components or even full scale fatigue tests.
- The main reason for testing is to ensure that all variables have been accounted for and to determine if analytical assumptions are conservative. Some of the typical variables which can be tough to accurately account for are:

1. Load interaction - retardation effects
2. Load distributions around cutouts
3. Fastener loads at spliced joints
4. Residual stresses imposed during assembly
5. Localized stress risers imposed during manufacturing

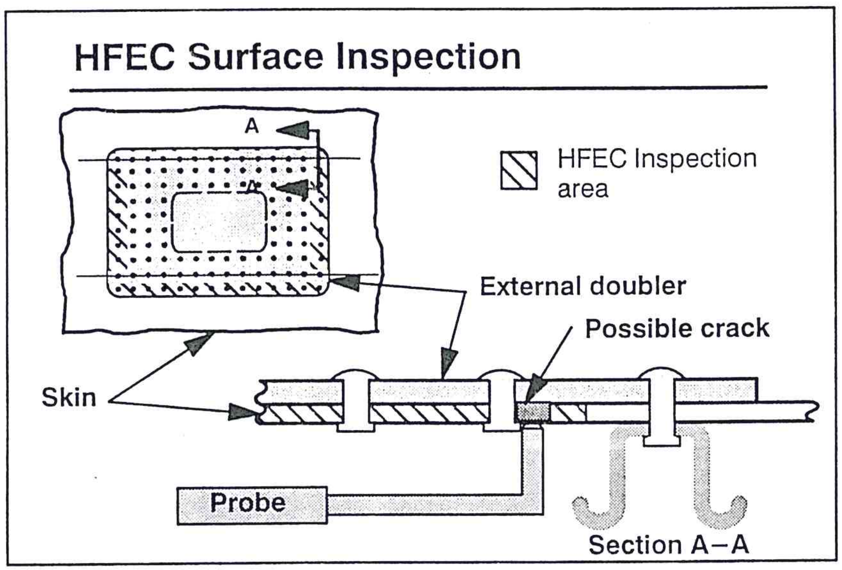
INSPECTION CRITERIA AND TECHNIQUES

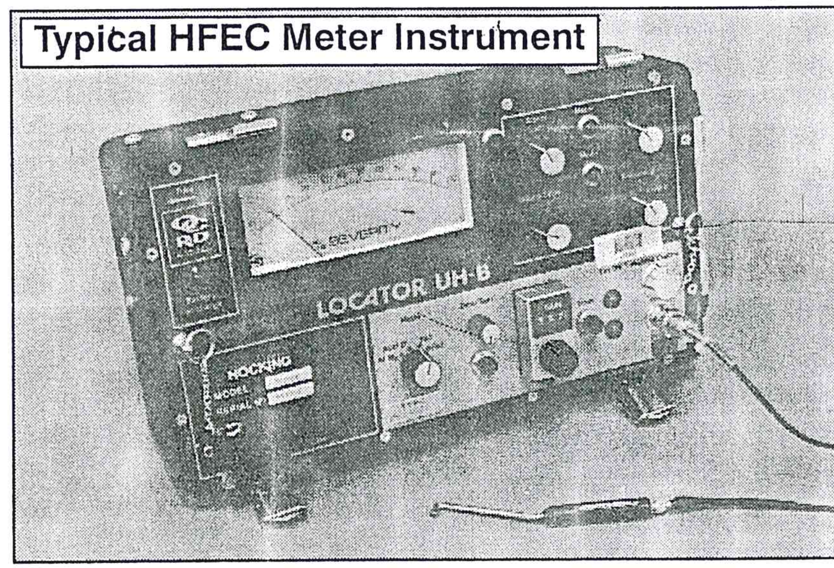
- The end product of all damage tolerance efforts is the establishment of inspection requirements and procedures.
- All inspections are categorized into two basic types:
    1. Visual
    2. NDE (Non-Destructive-Evaluation)
- NDE type inspections are performed for details which require the detection of small cracks with relatively high probabilities of detection. They are performed by FAA qualified inspectors. Some of the more commonly employed methods are:

    1. X-Ray
    2. Dye Penetrant
    3. Ultrasonic (UT)
    4. High Frequency Eddy Current (HFEC)
    5. Magneto Optic Imaging (MOI)

INSPECTION CRITERIA AND TECHNIQUES

- Most of today's current NDT inspections are performed by Eddy Current methods. A typical setup for detecting a skin crack with HFEC would be:





CONCLUDING REMARKS

- Understanding of the mechanisms of airframe fatigue have advanced significantly.
- Modern fracture mechanics has enabled the engineering community to reliably predict cracks due to metal fatigue.
- Damage tolerance has allowed the aircraft industry to provide higher levels of safety through inspections.
- New challenges are always testing the technical abilities of the industry such as Multisite Damage and Widespread Fatigue Damage. Advance methods will therefore be a necessity to address future fatigue related problems.