8929-AN-03-S
PROCEEDINGS

SESSION A

DAMAGE TOLERANCE IN HELICOPTERS - SITUATION TODAY

University of Cranfield, UK.

N68171-00-M-5512

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SESSION A

THE MANUFACTURERS PERSPECTIVE

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FLAW TOLERANCE / DAMAGE TOLERANCE

AGUSTA EXPERIENCE

ON

METAL PARTS

Ugo Mariani

Chief Fatigue Office

Past applications of Damage Tolerant approaches were related to support service bulletins.

Typically inspections were related to:

- easily detectable cracks, like visual detection of trailing edge cracks in rotor blades during routine inspection

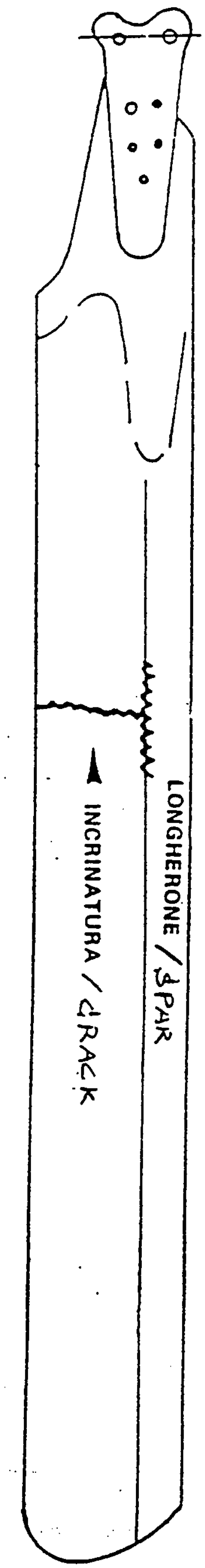
or

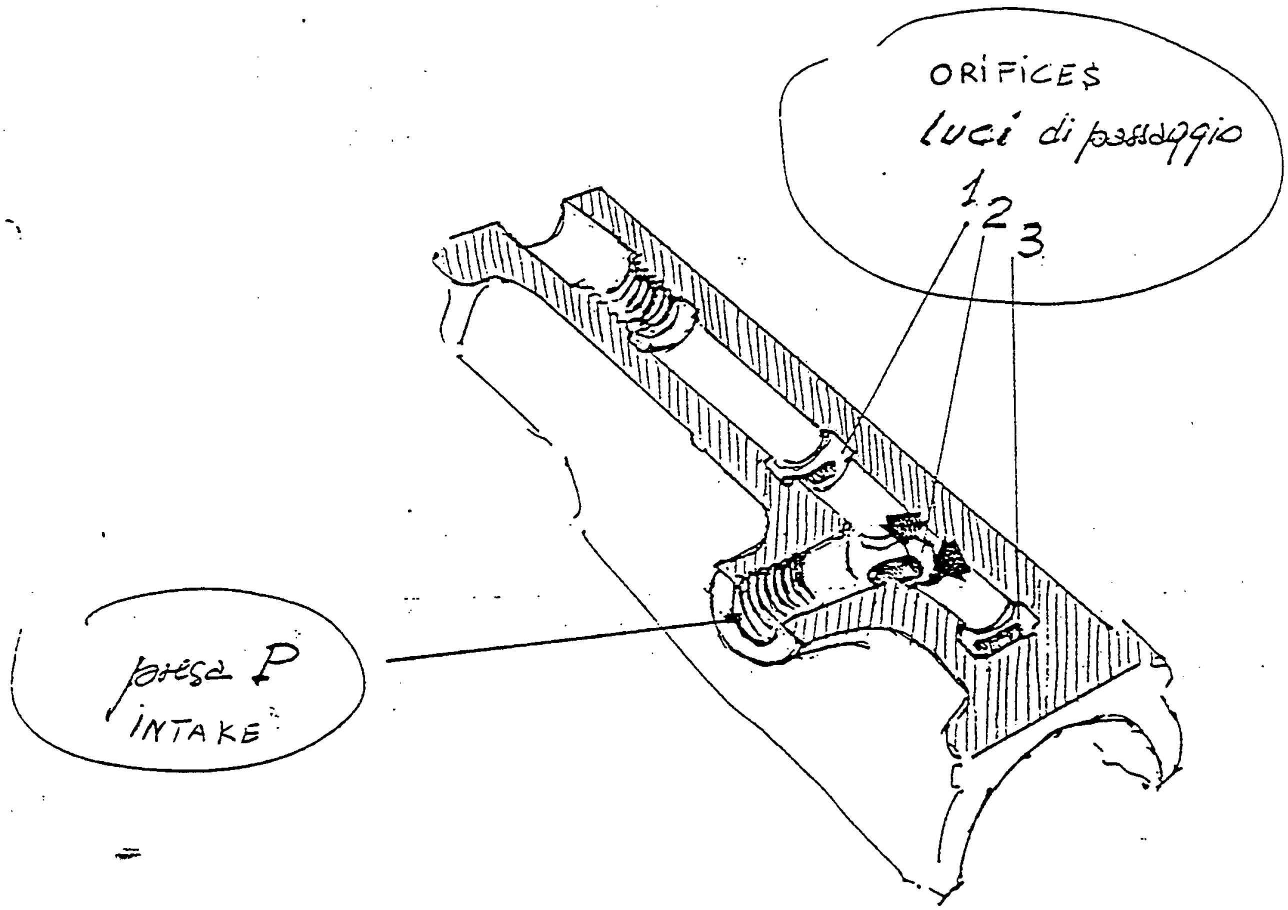
- self evident partial failures, like cracks in servoactuators, self evident due to hydraulic fluid leakage

More complex inspection methods were used for specific critical sections, focusing detection methods in a very limited area of the part.

In these cases frequent inspections could be accepted due to the 'short' time required.

This cannot be the general case for rotor or mechanical parts of a new helicopter.


LONGHERONE / SPAR
INCRINATURA / CRACK



Sez A - A

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FLAW TOLERANCE - ENHANCED SAFE LIFE

Some experience was gained in EH101 development program and research activities.

The approach is now applied to NH90 and AB139.

A proper validation of the type of flaws and the method used to test flawed elements is considered essential.
An example is provided detailing the evidences to validate scratches in aluminum.

A survey of service data and test evidences provided information on a severe manufacturing scratch, showing that plastic deformation and notch radius at crack tip are the relevant parameters for the crack initiation.
The crack growing path of a component failed in service showed negligible effects of the notch were plastic deformation had occurred, although the crack was growing very close to the scratch.

Milling an 'U' shaped gauge in SCT specimens tested at CAL made a very sharp scratch. Considering the scratch like a crack, ΔK was computed for each coupon. Depending on the applied stress, specimens failed if the ΔK was higher than the threshold value, and they gave run out if lower.

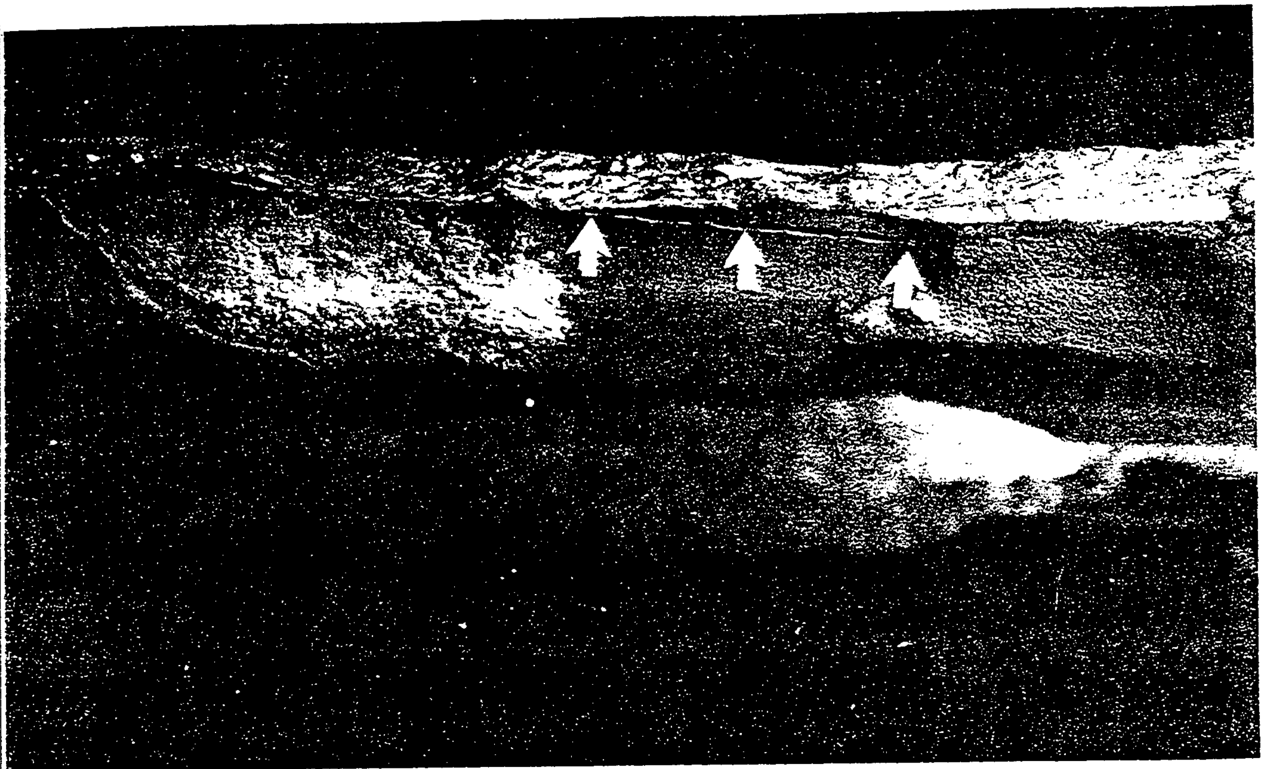
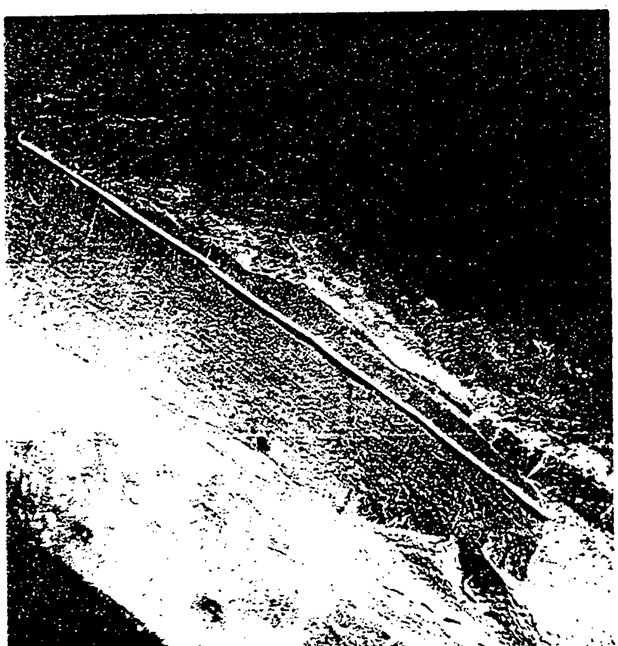
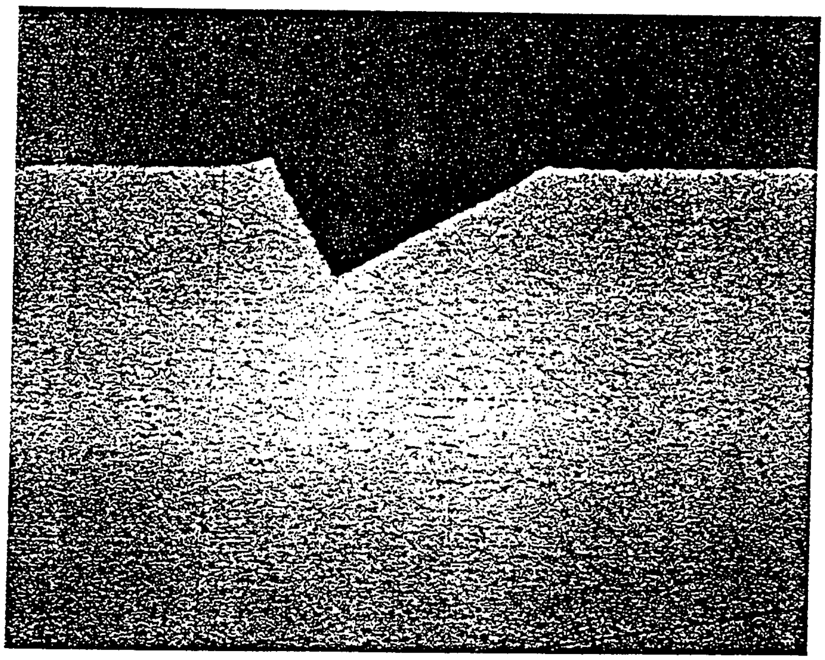


FOTO N. 7



x 9 FOTO N. 8

x 180 FOTO N. 9



- LA PROFONDITĂ DELL'INTAGLIO Ĕ DI mm 0,15 ~.
NON SI NOTANO INCRINATURE.
L'ANGOLO DELL'INTAGLIO Ē DI ~ 90°

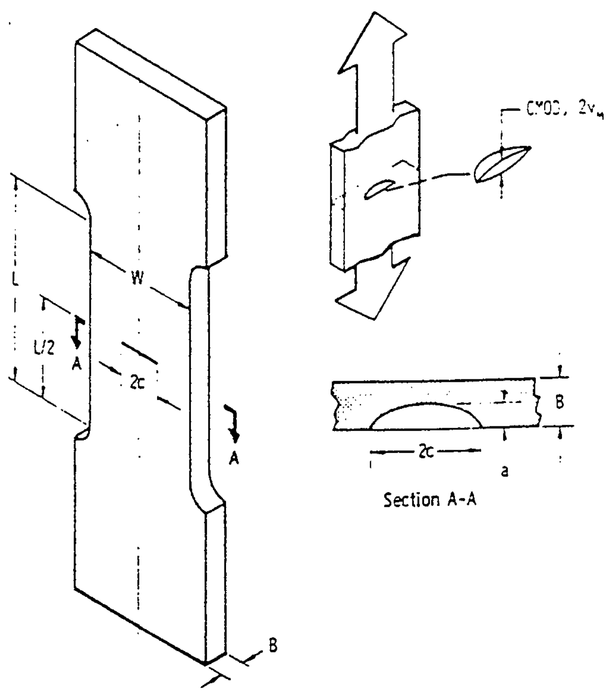
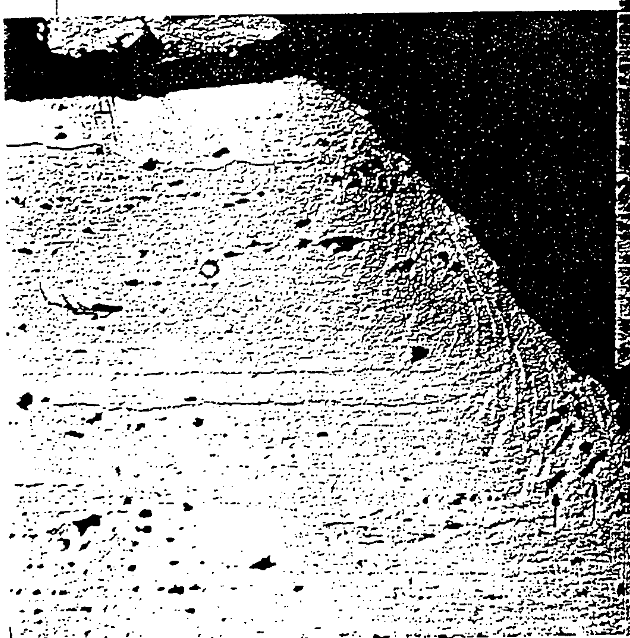
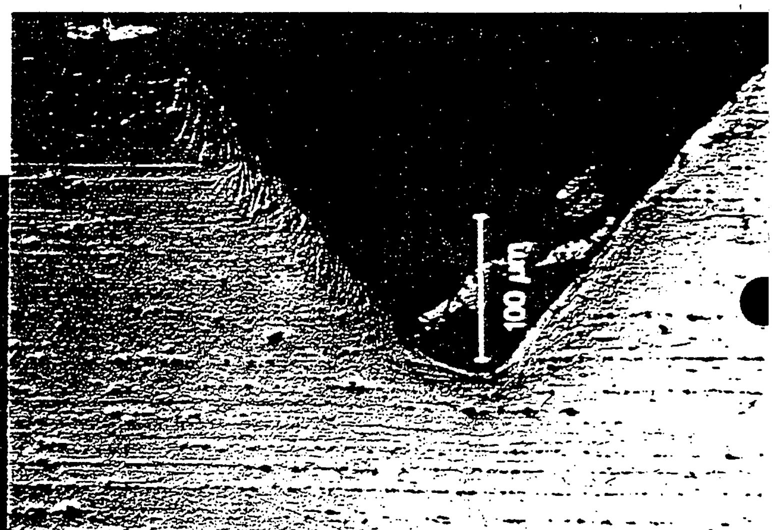


FIG. 1 Typical Surface-Crack Specimen (Grip Details Omitted) and Nomenclature

Foto n°8   200X





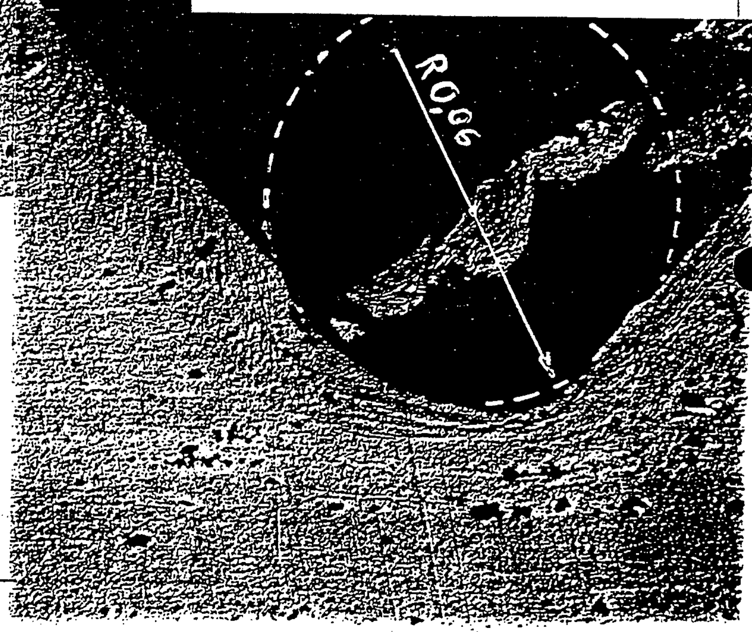


Foto n°9   500X

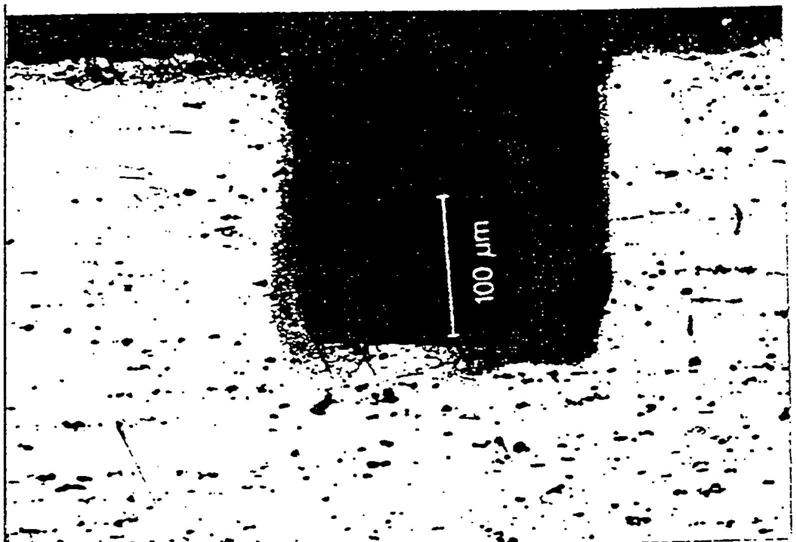
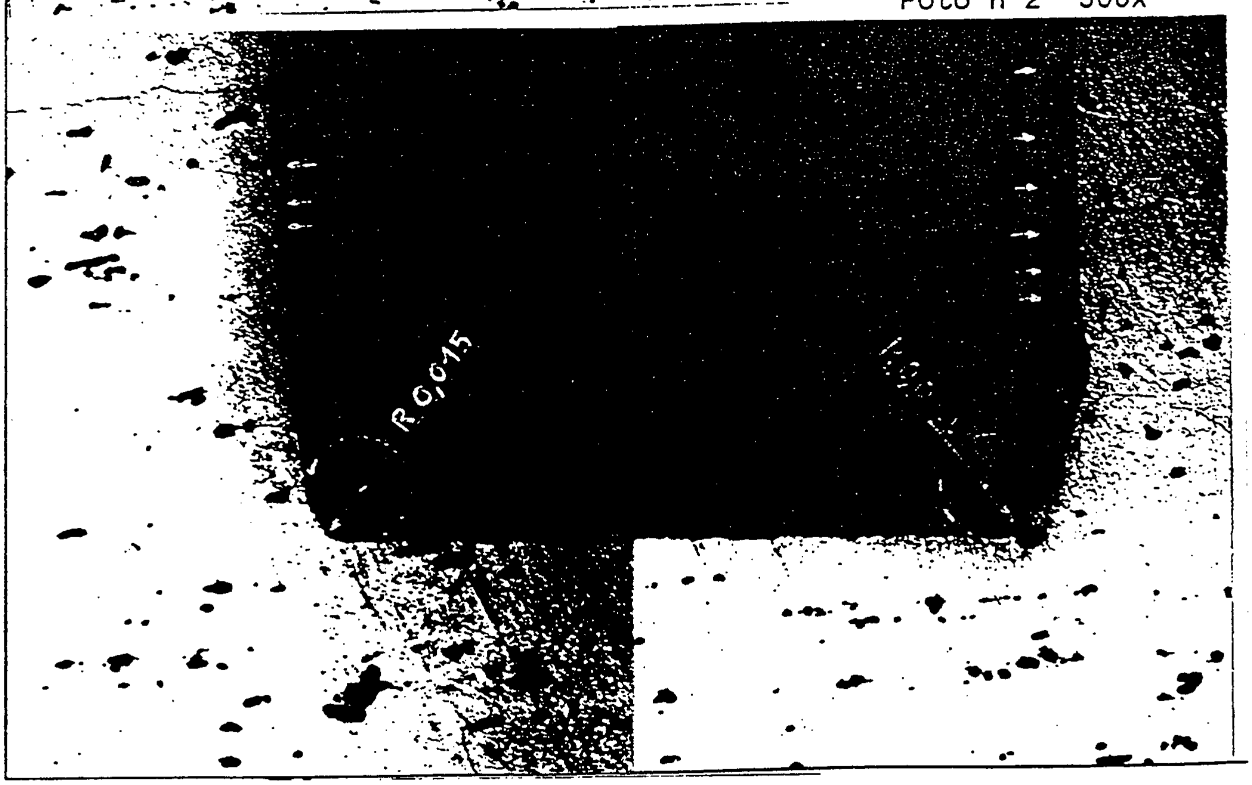
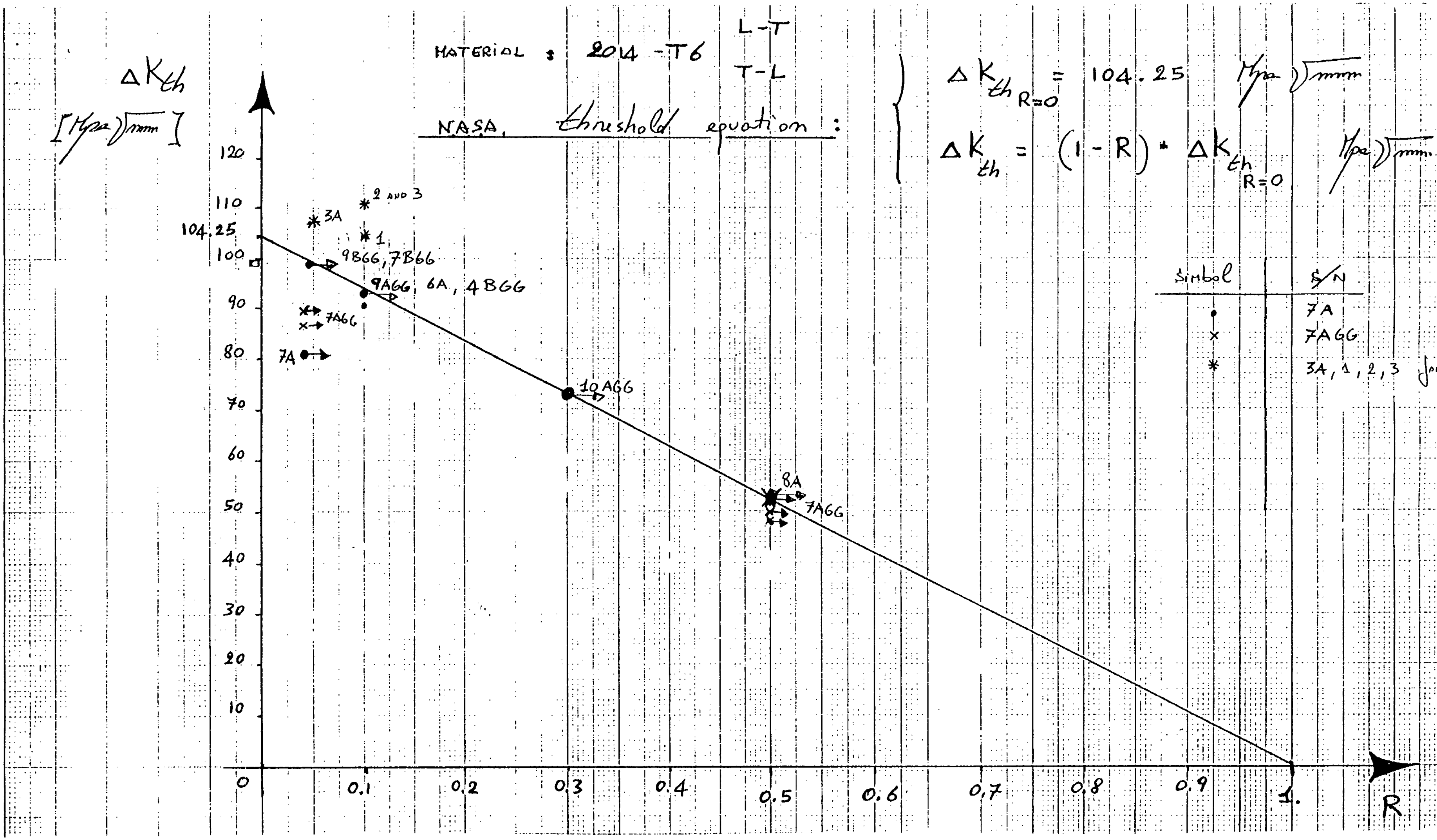


Foto n°1
200X

Foto n°2   500X




MATERIAL : 2014 -T6
L-T
T-L
NASA, threshold equation :
$\Delta K_{th_{R=0}} = 104.25$ Mpa $\sqrt{mm}$
$\Delta K_{th} = (1-R) * \Delta K_{th_{R=0}}$ Mpa $\sqrt{mm}$
ΔK_{th}
[Mpa $\sqrt{mm}$]
120
110
104.25
100
90
80
70
60
50
40
30
20
10
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.
R
3A
2 AND 3
1
9BGG, 7BGG
9AGG, 6A, 4BGG
7AGG
7A
10AGG
8A
7AGG
simbol
S/N
7A
7AGG
3A, 1, 2, 3 failed

Based on these evidences a proper tool could be selected to flaw a Main Gearbox Case, made by aluminum casting, to carry out a flaw tolerance validation.
This is a 'worst case' of scratch, severe but still realistic, since it has:

- 'V' shape according to experience
- minimize the plastic deformation at the tip, shown by etched metallurgical section
- 0.5 mm depth, highest value consistent with experience

A block-loading test was carried out representative of 600 h, factored for scatter, considering 0.5 mm deep scratches as Clearly Detectable Flaws.
Failure occurred during the second test phase that was carried out to improve the result.

The flaw tolerant evaluation, based on 'time to crack initiation', will be used to support inspection intervals for accidental damage.
Assuming this as the retirement life of all the parts would be and unnecessary penalty since the flaws addressed are visually detectable during a Detailed Inspection.

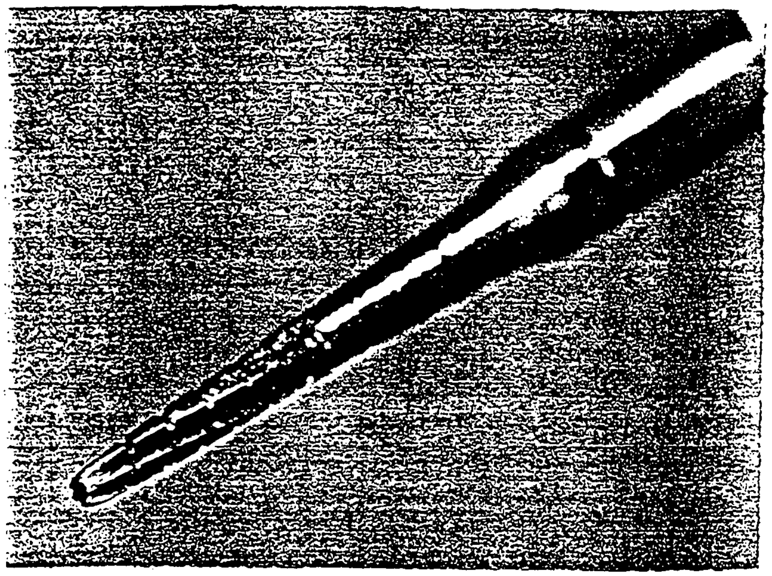


Foto n°18 10X

Fresetta conica a tagliente,

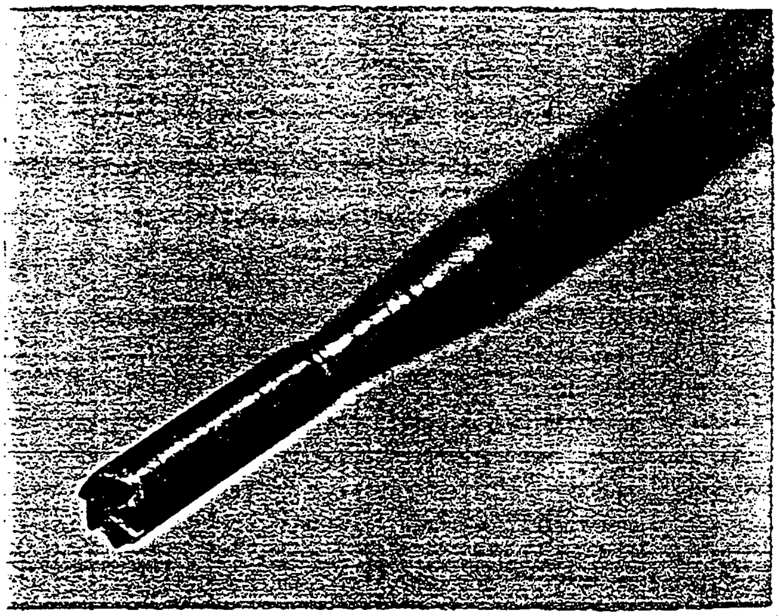


Foto n°19 10X

Fresetta cilindrica con tagliente
sulla testa.

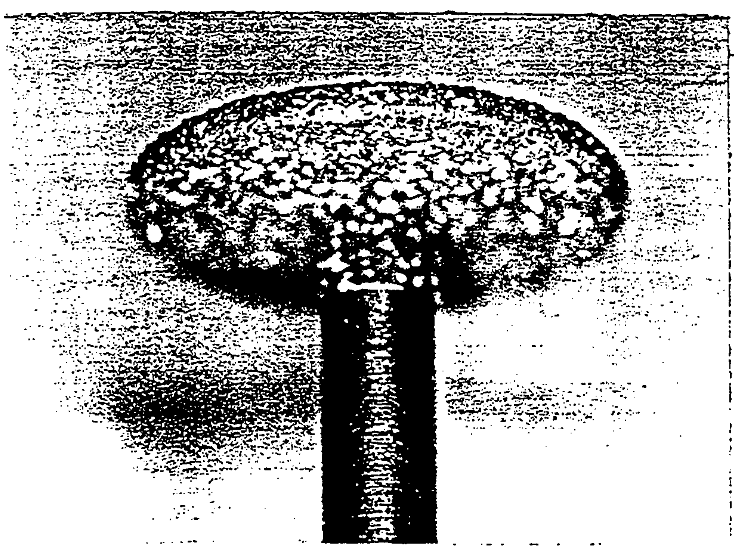
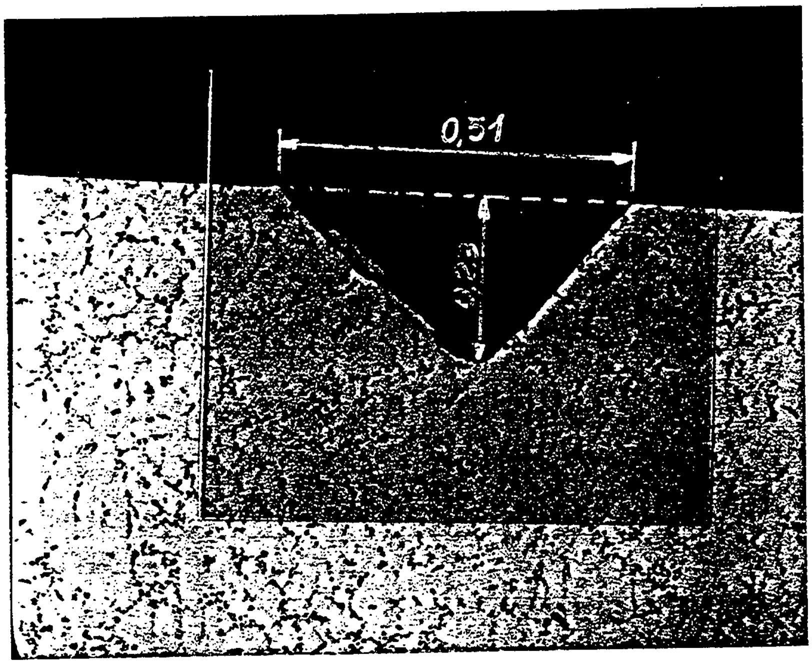
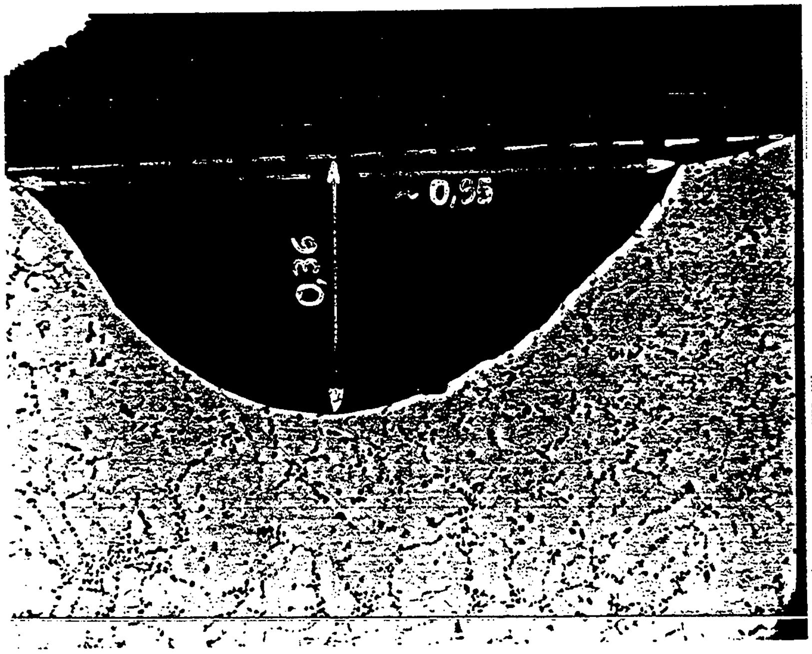


Foto n°20 10X

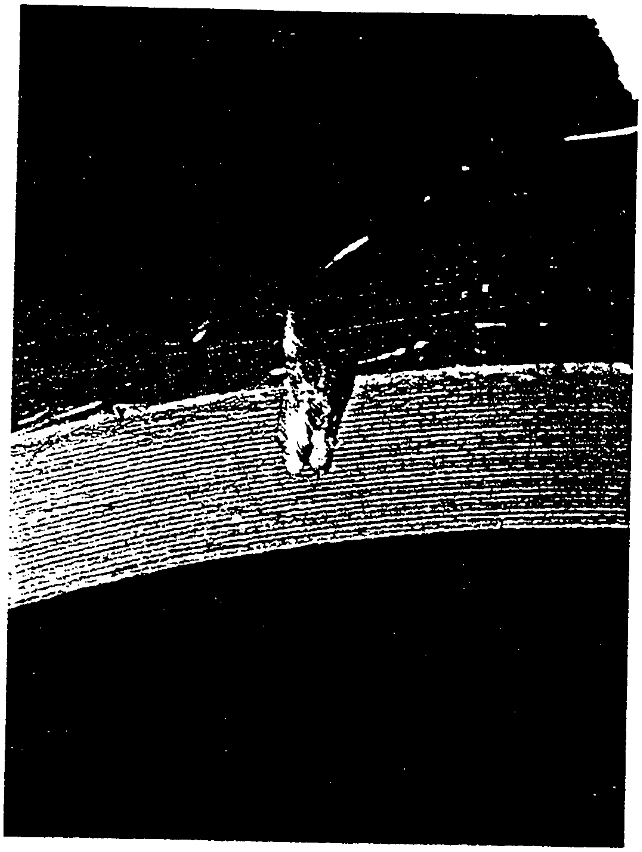
Fresetta, o meglio mola a disco
diamantato,



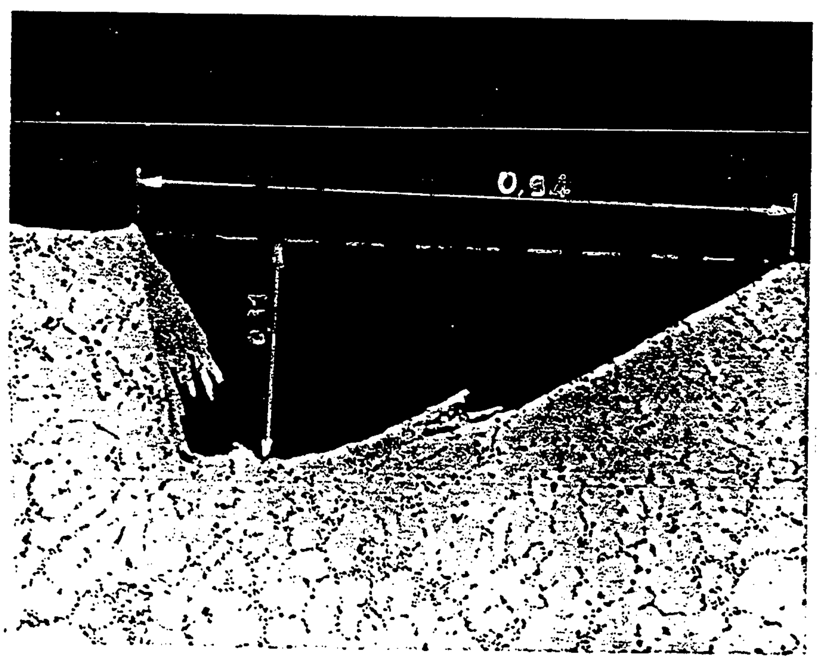
100 x



100 x

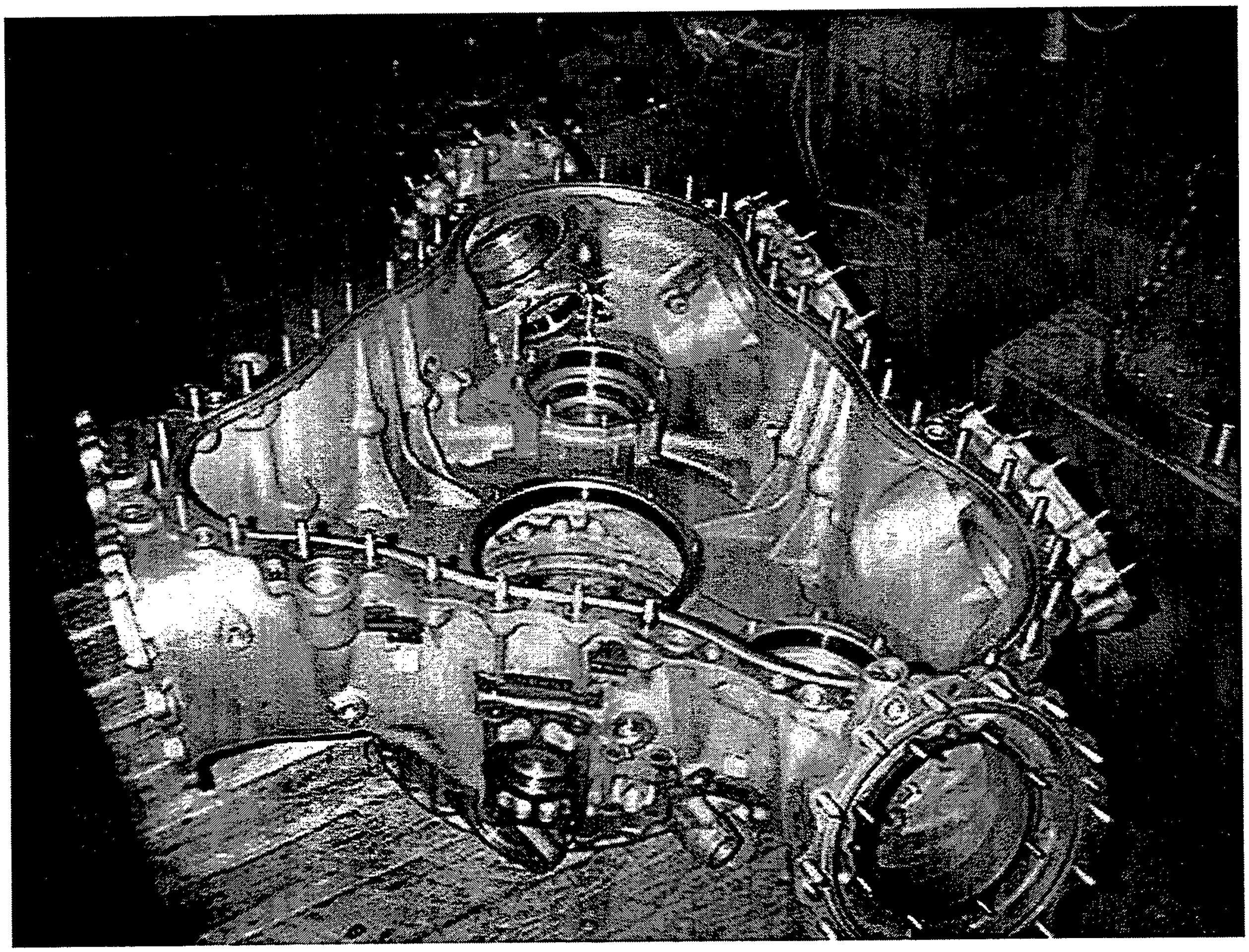


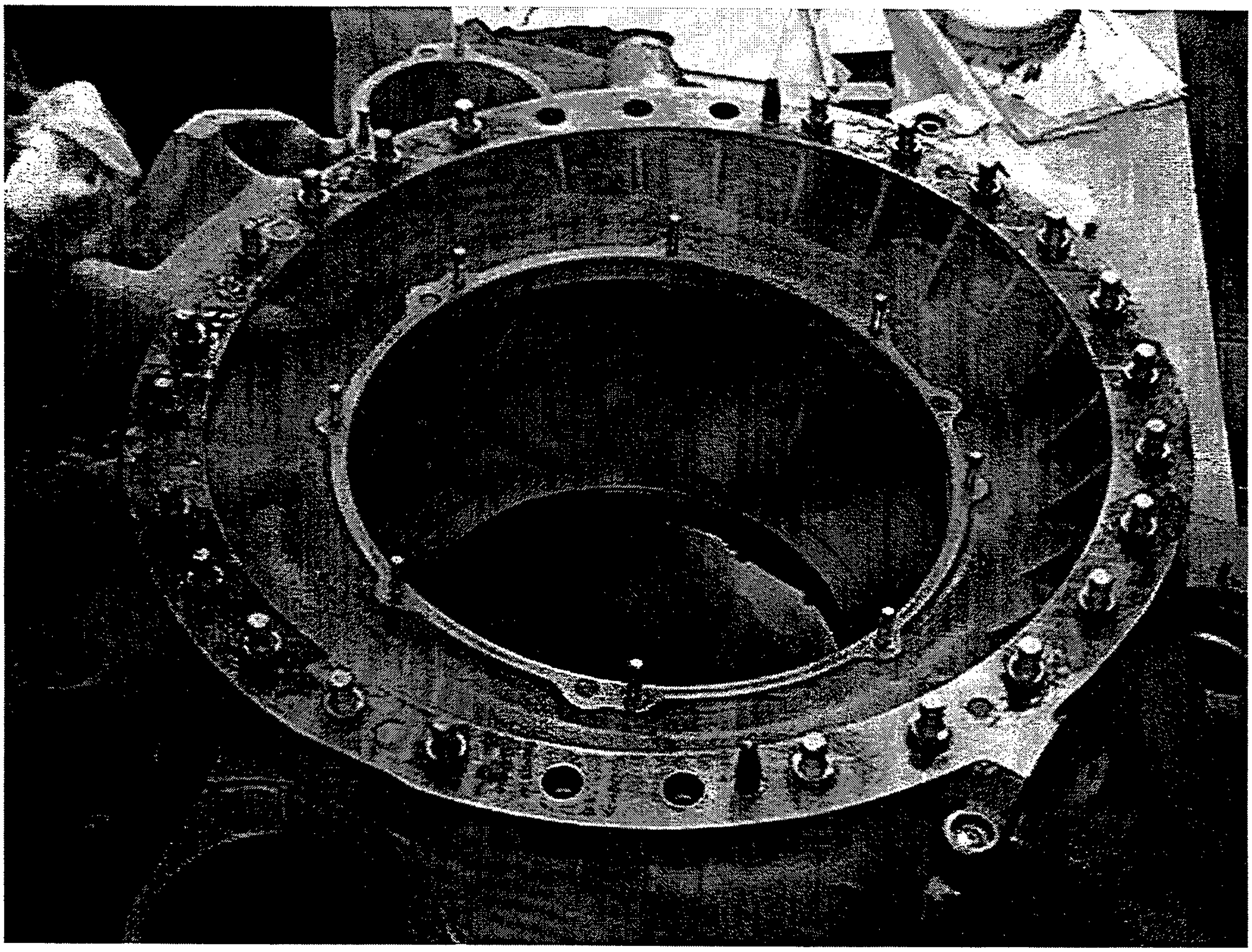
10 x



100 x

S/N 3





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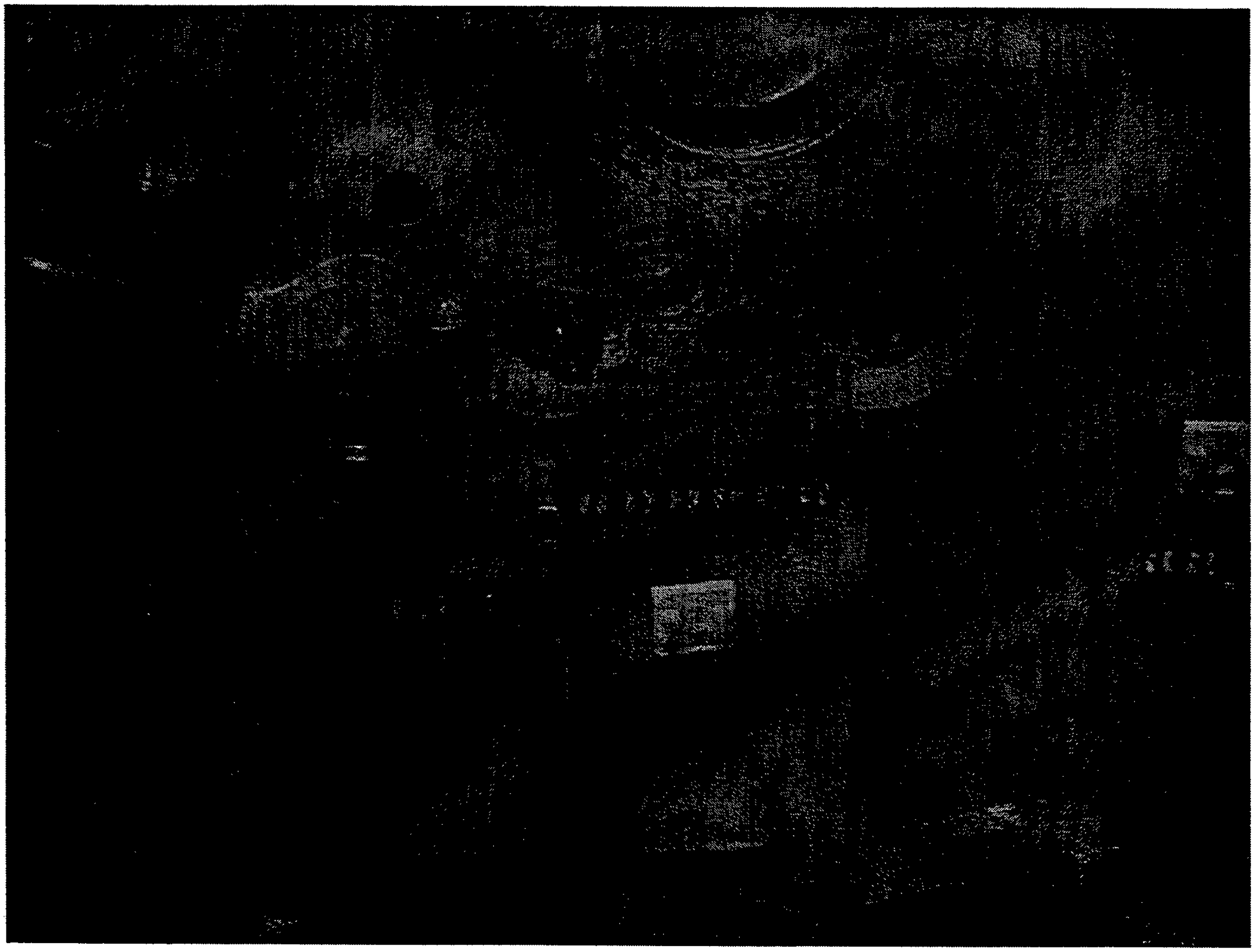

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DAMAGE TOLERANCE – FRACTURE MECHANICS

The experience on SCT specimens at CAL confirmed that for single loading path elements, inspection intervals for cracks are very short if the crack is growing in stabilized flight conditions, were helicopter loading spectra are close to a CAL

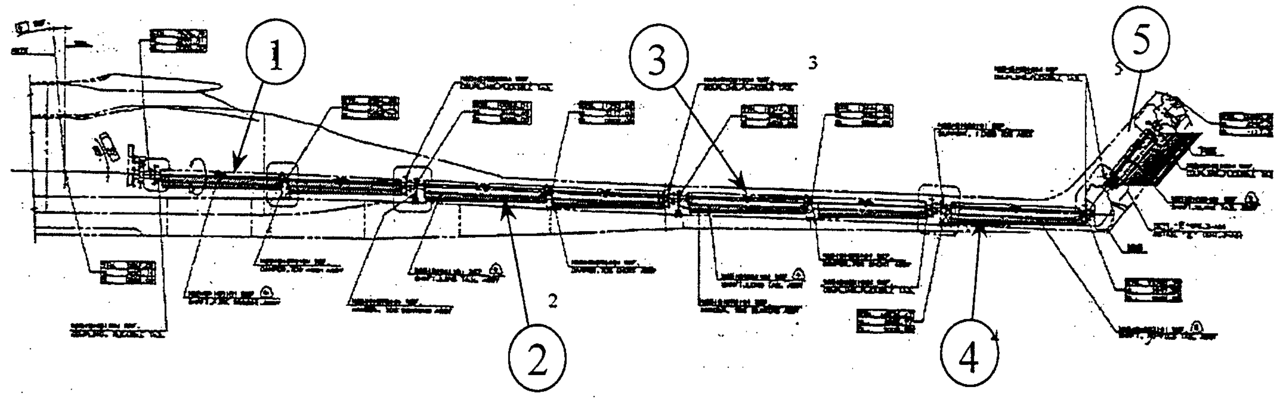
In practice the requirement in this case is 'No Growh', with possibility of slow crack growth limited to start-stop or GAG cycles only.

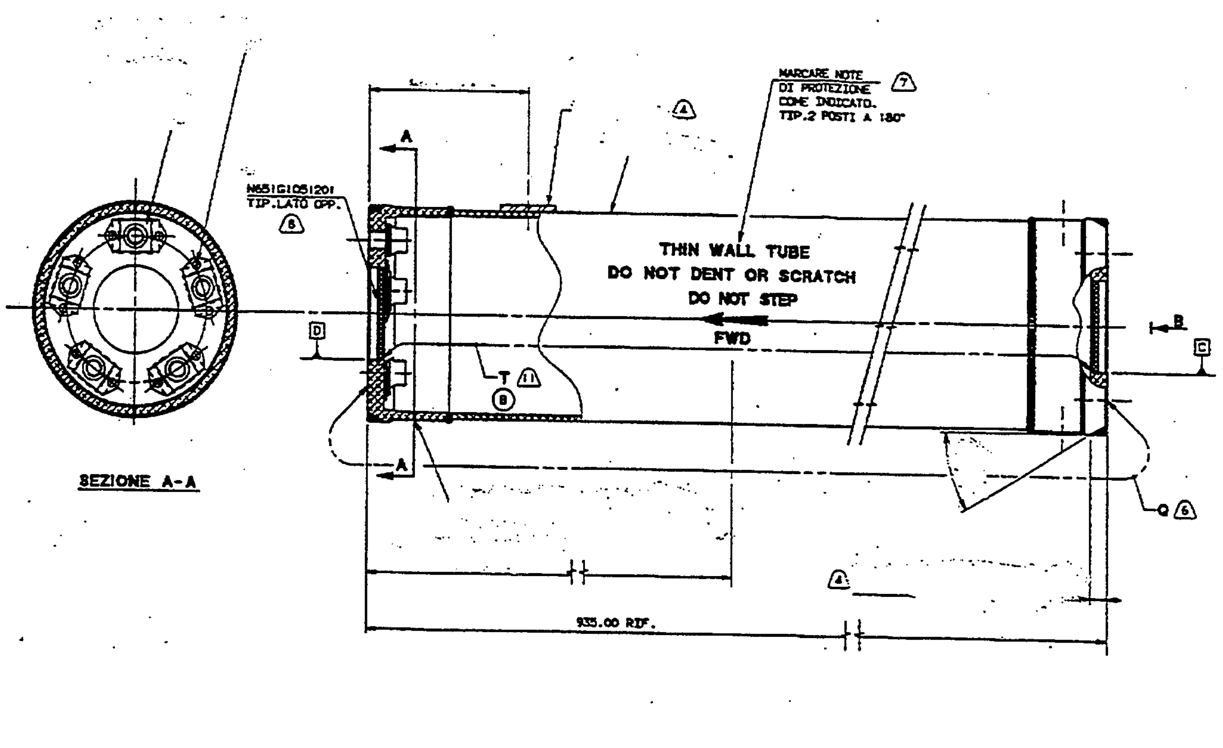
Crack sizes taken into account are based on MIL Spec:

0.38 mm radius	mechanical parts Machined parts of the fuselage
1.27 mm radius	panel & stringers of the fuselage

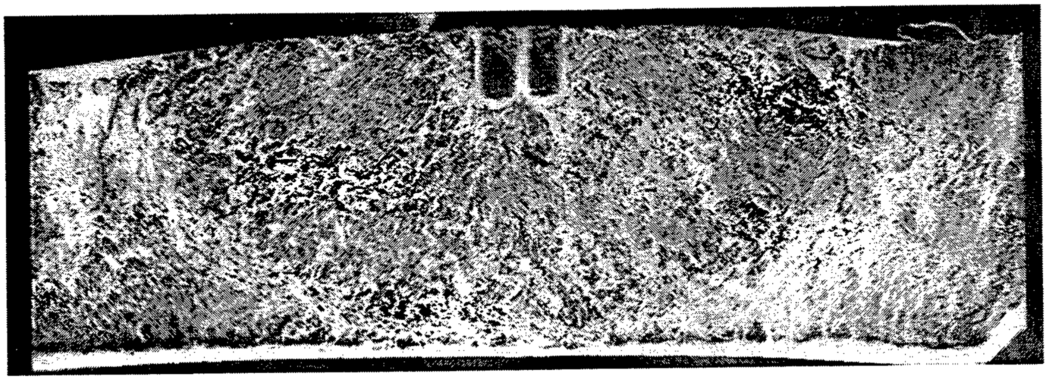
Two cases were carried out to address applicability:

- Tail Rotor Drive-line shafts as 'no growth or benign propagation'
- Inspections of panel / stringers of the naval rear fuselage of EH101 (500-1000 h)

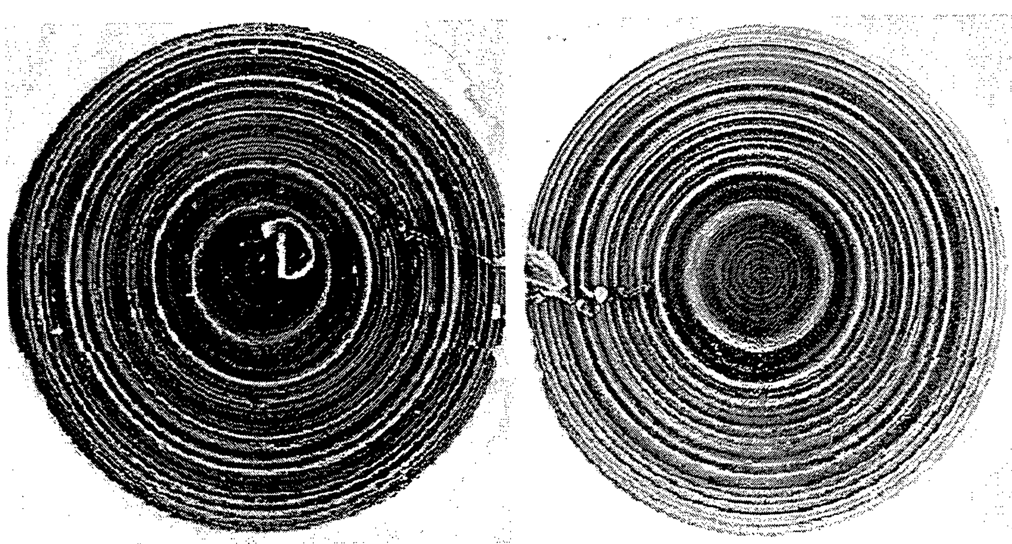




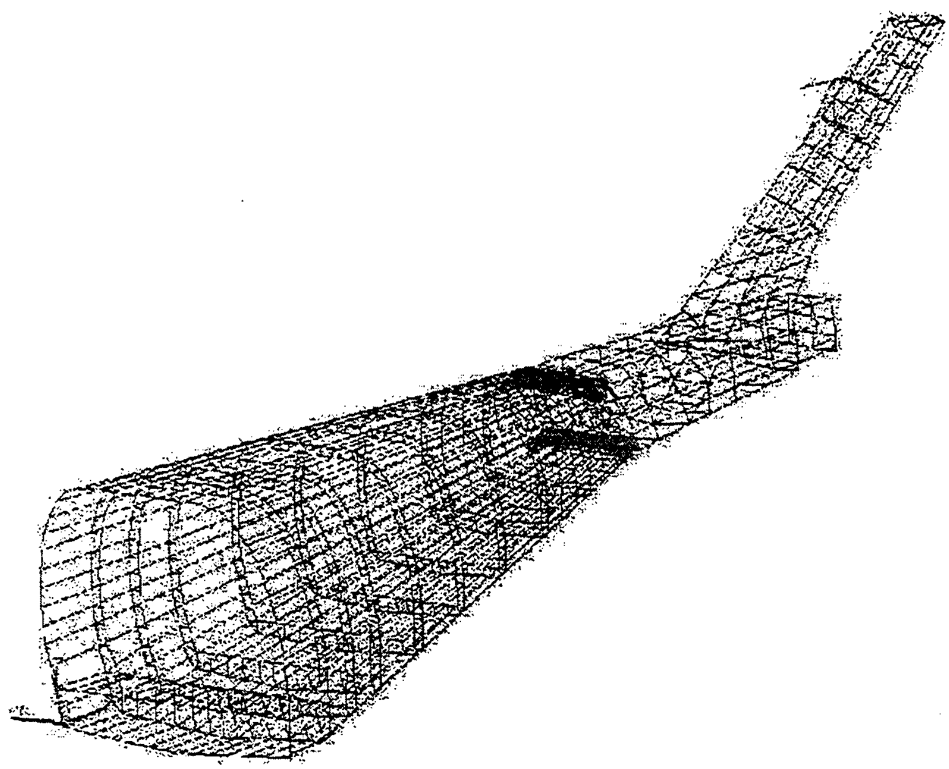
TR Transmission Shaft



Micro-notched specimens for fatigue tests - details of the failure section.



'No crack growth' test data at SEM : a) 'no growth' of the surface crack b) 'no growth' of the crack at the tip of the notch.



US HELICOPTER INDUSTRY
DAMAGE TOLERANCE ANALYSIS AND DESIGN

WORKSHOP ON DAMAGE TOLERANCE IN HELICOPTERS

Cranfield University
April 4-5, 2000

Presented by: George Schneider - Sikorsky Aircraft

Co-author: John Wang - Sikorsky Aircraft

Technical tasks in this document include tasks supported with shared funding by the U.S rotorcraft industry and the Government under RITA/NASA cooperative agreement NCCW-0076, Advanced Technology, August,1995

US HELICOPTER INDUSTRY
DAMAGE TOLERANCE ANALYSIS AND DESIGN
(METALLIC STRUCTURE)

- H-53 Helicopter Sikorsky / Airforce Damage Tolerance Project - 1980's

- Damage Tolerance and Flaw Tolerance Applied to Current Helicopters
  - Sikorsky
    - Damage Tolerance Inspection Intervals - Current Helicopters
    - Flaw Tolerance Inspection Intervals - Current Helicopters
    - S-92 Rotor - FAR29.571 Certification, Amend. 28. October 1989
  - Bell
    - Theoretical Redesign for Damage Tolerance - Current Helicopters
    - Bell 430 Rotor Damage Tolerance Certification

- RITA Damage Tolerance Project
  - Bell crack growth testing, crack growth code evlauations, NDI studies
  - Sikorsky crack growth models and equivalent initial flaw (crack) sizes.
  - FAA/RITA Interaction

- Conclusions and Key Issues

AIR FORCE H-53 DAMAGE TOLERANCE ASSESSMENT

STRUCTURE ANALYZED

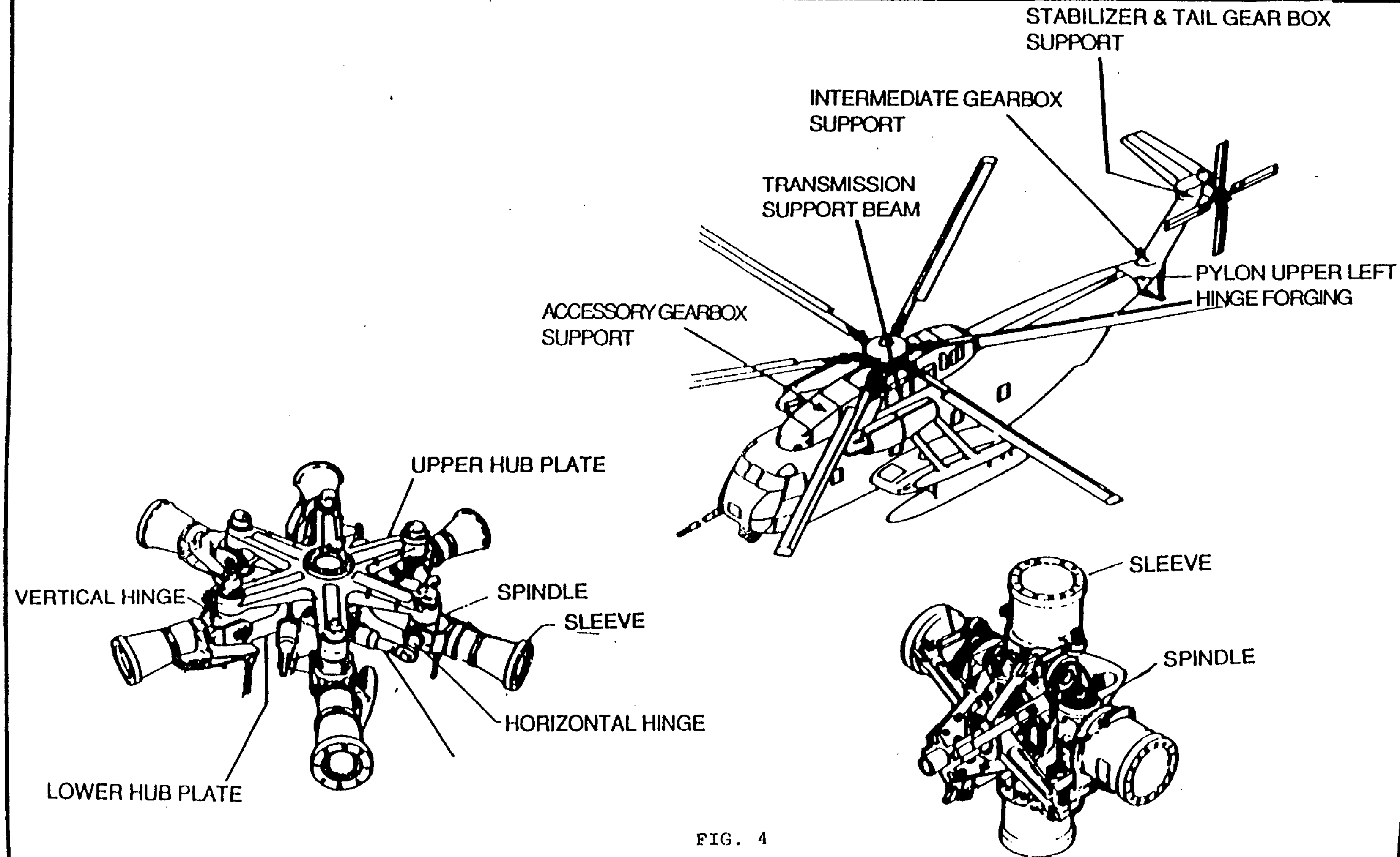


FIG. 4

SOME POTENTIAL MAIN ROTOR CRACK LOCATIONS

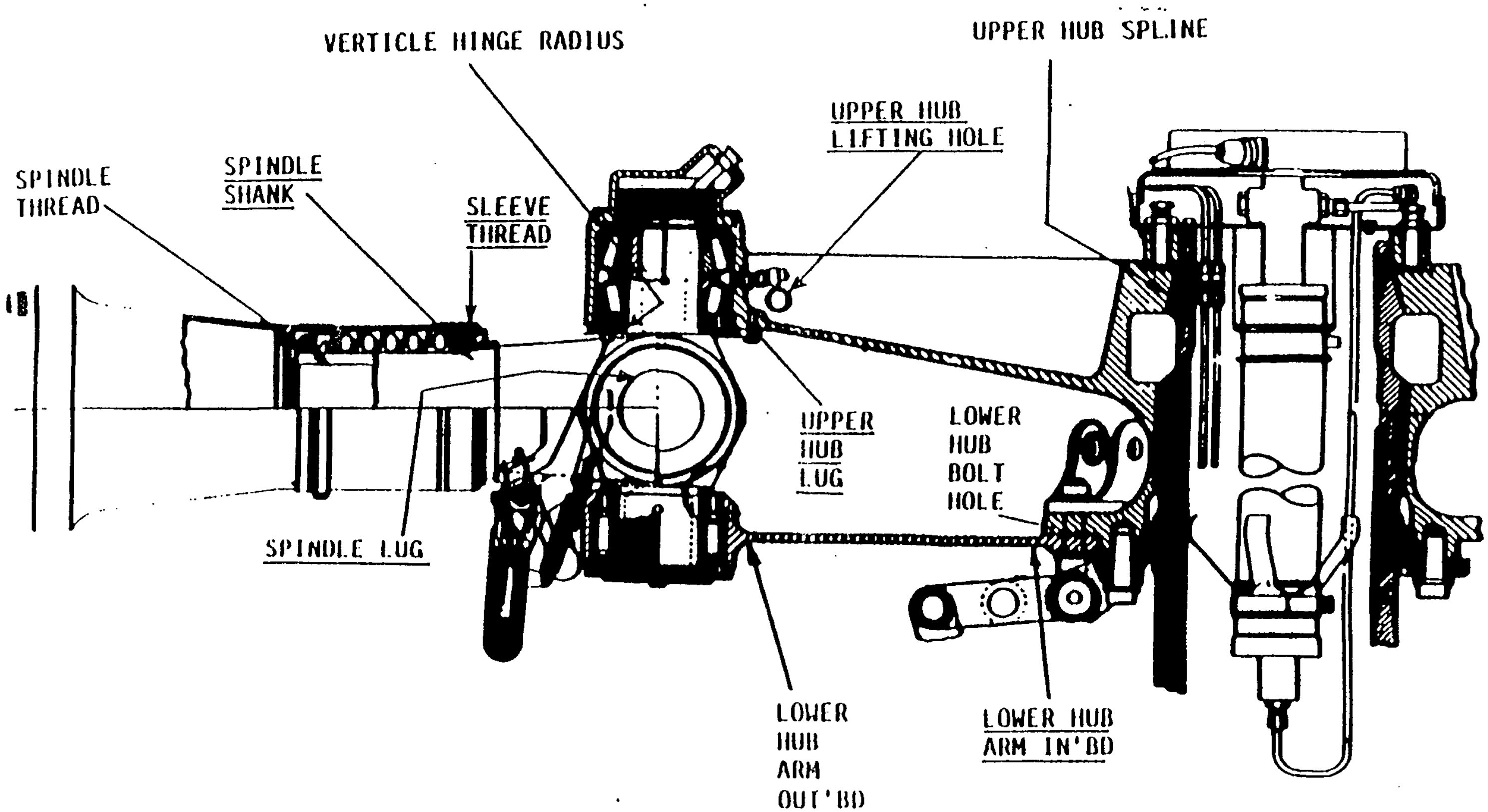


FIG. 5

DAMAGE TOLERANCE ASSESSMENT PERFORMED FOR UNDERLINED LOCATIONS

MAIN ROTOR CRACK GROWTH RESULTS

Phase Inspection is 100 Flt. Hours (Typical)
Time Between Overhaul (TBO) is 1000 Flt. Hours (Typical)

Structure - Crack Location	Material	High Cycle Threshold Crack Depth (inches)	Crack Propagation Time* (Flight Hours) From Initial Crack Depths of:		
			0.005"	0.010"	0.030"
H-53					
Upper Hub Plate	Ti-6Al-4V Alpha-Beta				
- Lifting Hole		0.003	380	50	16
- Lug		0.003+	High	High	5000+
Lower Hub Plate - Inboard Arm	Ti-6Al-4V Alpha-Beta	0.008	High	558	61
Horizontal Hinge - Damper Radius	4340 Steel	0.003	212	14	--
Spindle	Ti-6Al-4V Alpha-Beta				
- Lug		0.022	--	>2000	270
- Shank Radius		0.012	--	>2000	80
Sleeve - Threads	Ti-6Al-4V Alpha-Beta	0.002	150	42	10
Blade Spar	Al 6061-T6	0.030	--	>2000	271

FIG. 6

TAIL ROTOR CRACK GROWTH RESULTS

Phase Inspection is 100 Flt. Hours (Typical)
Time Between Overhaul (TBO) is 1000 Flt. Hours (Typical)

Structure - Crack Location	Material	High Cycle Threshold Crack Depth (inches)	Crack Propagation Time* (Flight Hours) From Initial Crack Depths of:		
			0.005"	0.010"	0.030"
H-53					
Spindle	Ti-Al-4V				
- Shank Radius		0.023	433	56	10
- Threads		0.007	10	7	--
Sleeve	Ti-6Al-4V				
- Threads		0.002	--	>2000	475

FIG. 7

AIRFRAME CRACK GROWTH RESULTS

Phase Inspection is 100 Flt. Hours (Typical)
Time Between Overhaul (TBO) is 1000 Flt. Hours (Typical)

Structure - Crack Location	Material	High Cycle Threshold Crack Depth (inches)	Crack Propagation Time* (Flight Hours) From Initial Crack Depths of: 0.005"	0.010"	0.030"
H-53					
Upper Left Pylon Fold Hinge Ftg. - Rivet Hole	Al7075-T73	0.005	410	170	45
Acc. Gear Box Support Fitting - Bolt Hole	Al7075-T73	0.002	50	29	7
Stablizer Support Fitting - Hole	Al7075-T73	0.004	40	21	6
Transmission Support Frame - Holes	Al7075-T73	0.050	--	--	>2000
		0.100	--	--	>2000
		1.000	--	--	>2000

FIG. 8

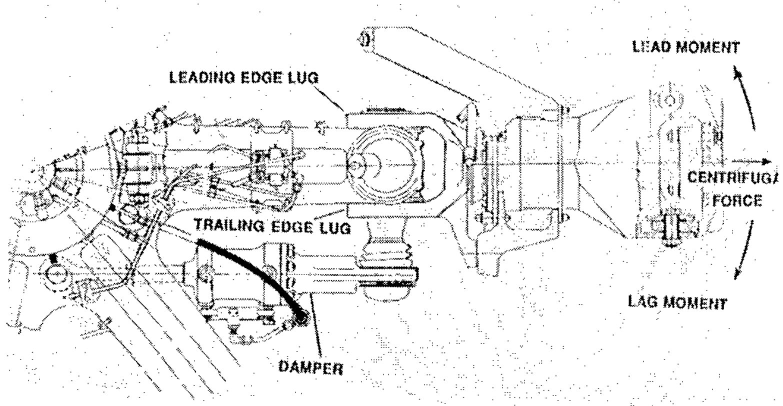
SIKORSKY- DAMAGE TOLERANCE INSPECTION INTERVALS

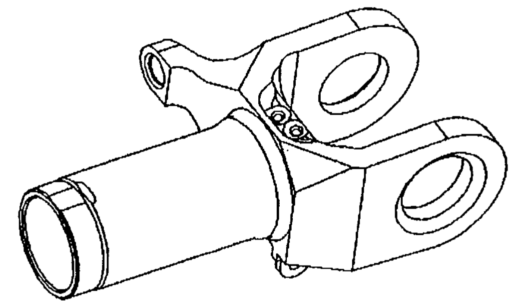
Examples:

Part	Material	Inspection Method	Detectable Crack Size (inches)	Inspection Interval (Flt Hours)
• S-61 Spindle Shank	4340 steel	UT	0.020	50
• H-53E Spindle Lug	Ti-6Al-4V	UT	0.080	50
• H-60 Hub Damper Hole	Ti-6al-4V	FP, EC	0.080	200

H-53E SPINDLE LUG
CRACK GROWTH INSPECTION INTERVAL

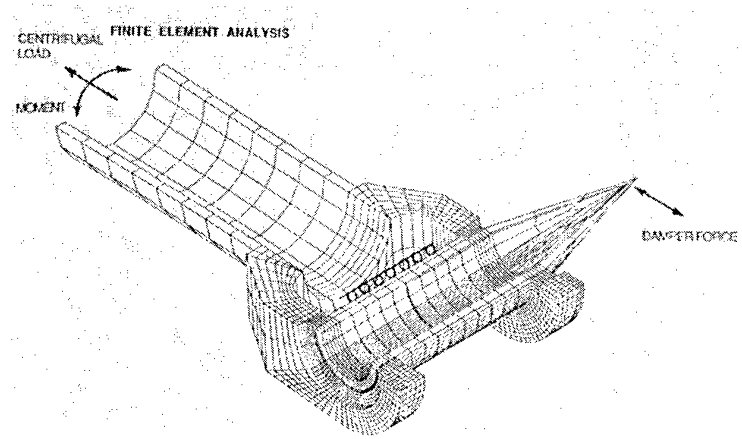
Lug crack under bonded liner resulting from fretting





6AL-4V Titanium Spindle α–β Forged, β STOA

Finite Element Model

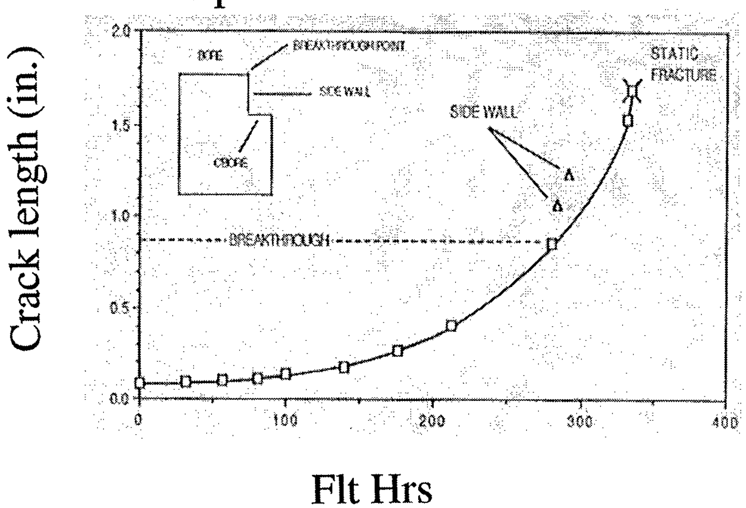


H-53E SPINDLE LUG
CRACK GROWTH INSPECTION INTERVAL

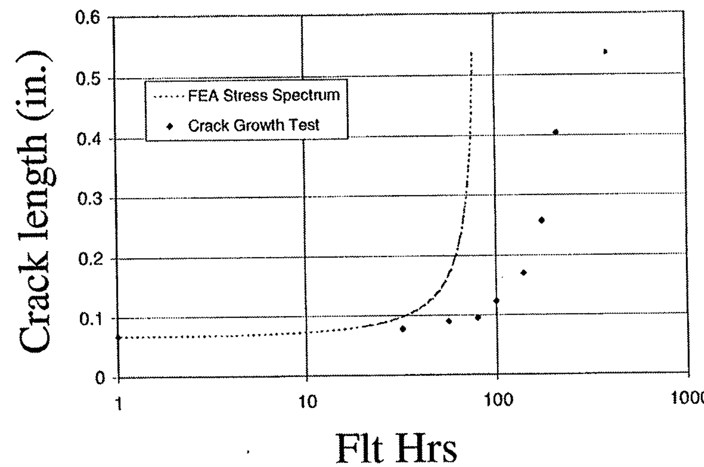
Lug crack under bonded liner resulting from fretting

Analysis: NASGRO strip yield corner crack model
Finite element analysis stress distribution

Spectrum Test Results



Spectrum Analysis Results



Ultrasonic inspection for 0.08 inch crack
Inspection interval 50 flight hours

SIKORSKY - FLAW TOLERANCE INSPECTION INTERVALS

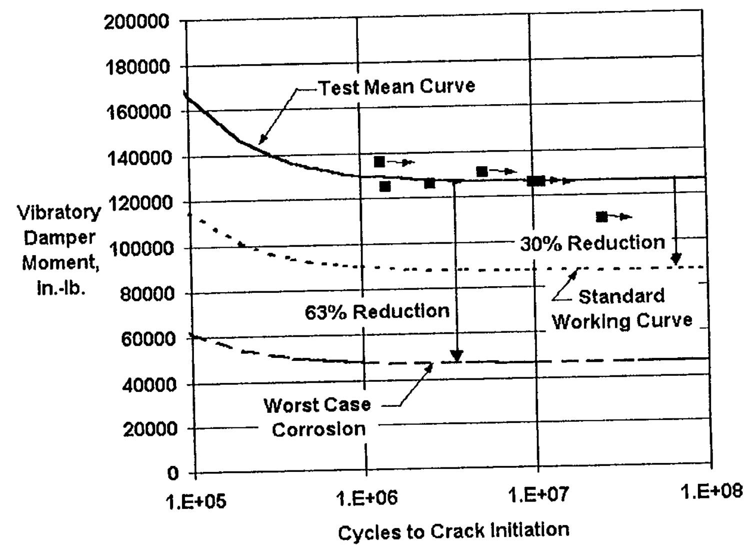
Examples:

Part	Material	Service Degradation	Inspection Interval (Flt Hrs)
• CH-53A/D Horiz. Hinge Pin	4340 steel	corrosion	1500 (1200)
• S-76 TR Pitch Horn	Aluminum	corrosion	12000 *

* Replacement time set to this interval

CH-53 A/D HORIZONTAL HINGE PIN
FLAW TOLERANT APPROACH - INSPECTION INTERVAL

- Subject to in service corrosion
- Coupon testing: “worst case” 63 % reduction
- Reduction applied for full scale test data
- Miner’s Analysis => 1500 flt hours
- Inspection Interval 1200 flt hours
- Replacement Time 8300 Flt Hours



SIKORSKY S-92 - FAR 29.571 ROTOR CERTIFICATION

METAL ROTOR COMPONENTS
FLAW TOLERANT SAFE-LIFE EVALUATION

SIKORSKY S-92 - FAR 29.571 ROTOR CERTIFICATION

METAL ROTOR COMPONENTS
FLAW TOLERANT SAFE-LIFE EVALUATION

FAA Requirements:

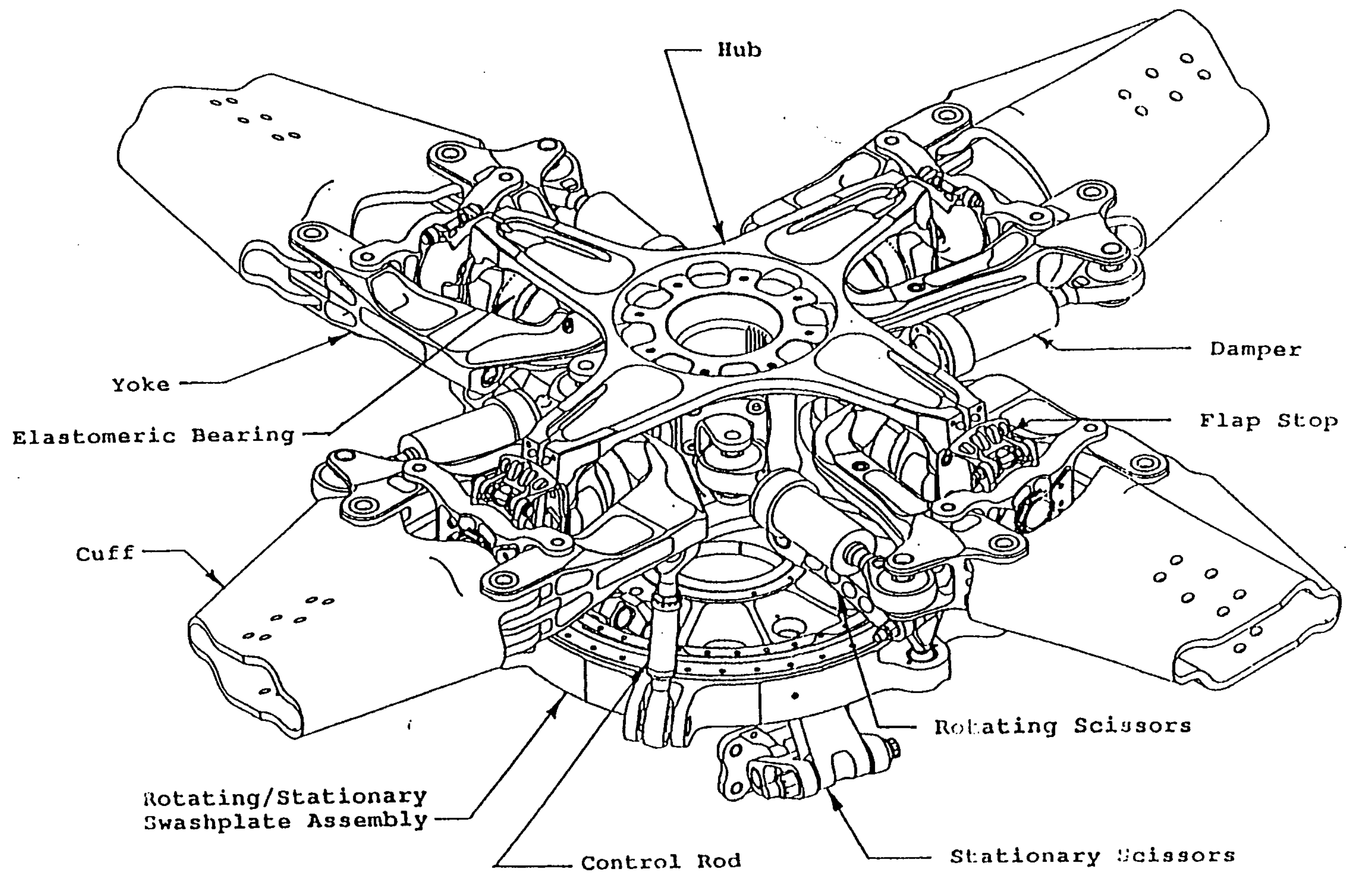
- Replacement Time: based on barely visible flaws
- Inspection Interval: based on severe flaws

Flaw types and sizes based on SAC review of field damage reports

Coupon tests to evaluate effects of flaws

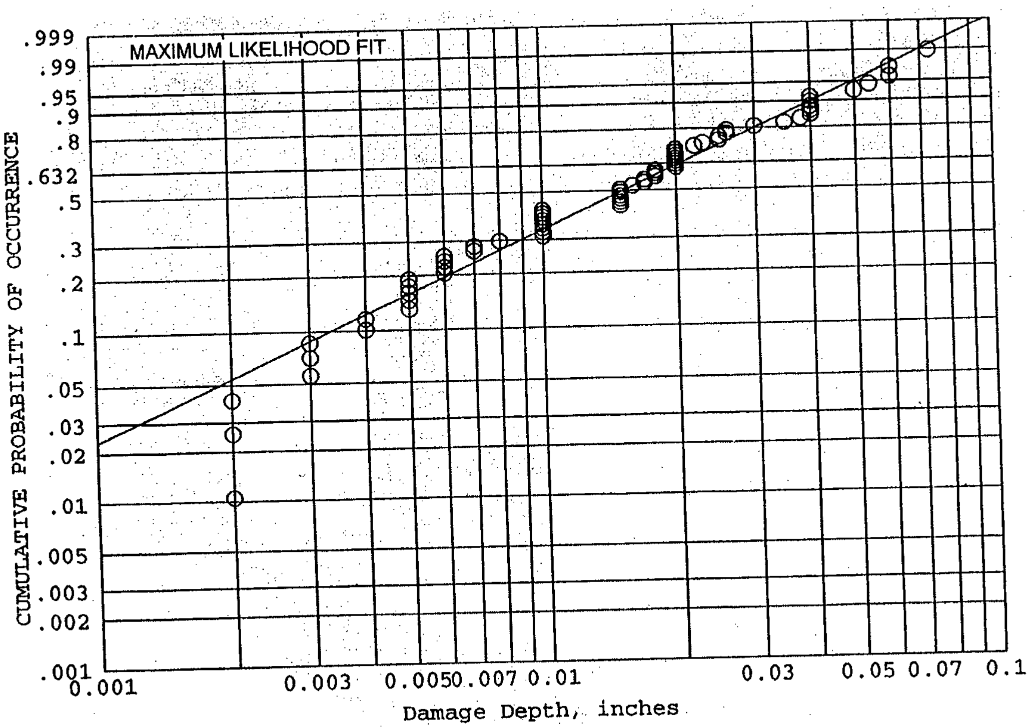
Full scale tests performed with barely visible flaws
Miner's analysis used for determining replacement times

SIKORSKY- S-92 ROTOR HEAD



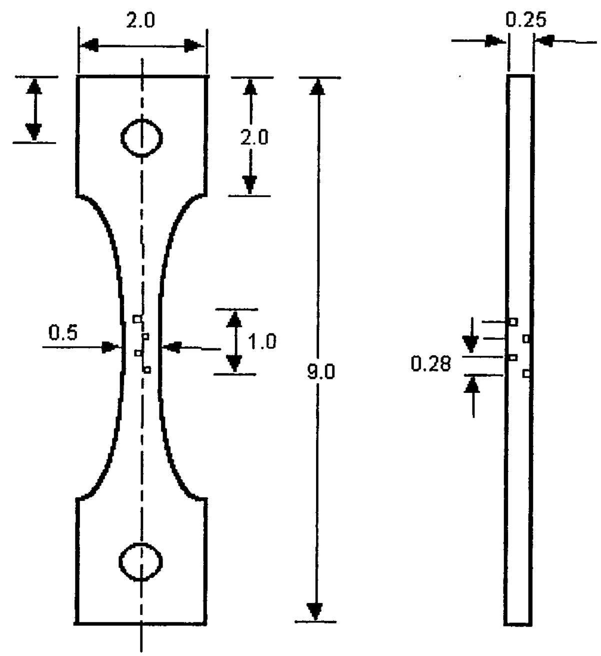
SIKORSKY HISTORY OF SERVICE-RELATED DAMAGE

66 S-76 Aluminum and Titanium Rotor Head Parts
53 Parts Damage > 0.005 inch depth
8000 S-76 Rotor Head Parts in Service



FLAWED SPECIMEN TESTING

Al7075-T73, Tested at 0 Steady Stress

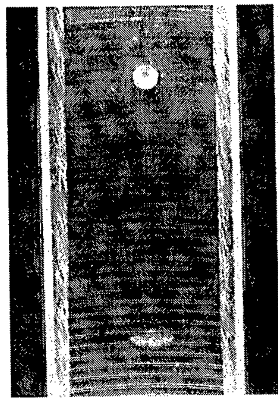


□ Defect introduced into test specimen Gage Section (not to scale)

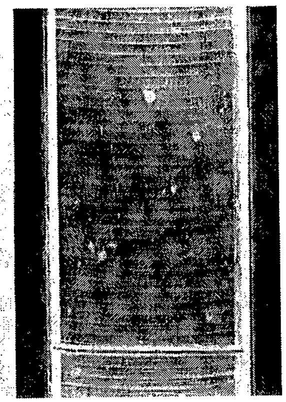
All Dimensions in Inches

Surface	Flaw Type	Depth Range
Highly Polished	N/A	N/A
Polished	N/A	N/A
As-Machined	**N/A**	**0.00012**
Lightly Polished	N/A	0.00005
0.005"Deep Natural Flaws	Corrosion Pit Sharp Scratch Gouge Sharp Indent Dull Indent	.0050-.0068
0.005"Deep Artificial Flaws	EDM Notch Milled Notch ECM Pit	.0040-.0054 .0045-.0054 .0040-.0064
0.040"Deep Natural Flaws	Corrosion Pit Sharp Scratch Gouge Sharp Indent Dull Indent	.034-.045
0.040"Deep Artificial Flaws	EDM Notch Milled Notch ECM Pit	0.040 Nom .039-.043 .038-.042
0.075" Deep Natural Flaws	Gouges	0.071-0.080

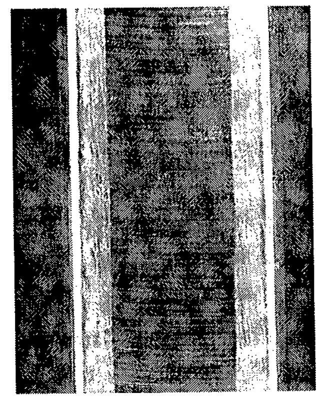
TYPICAL 0.005 INCH DEEP FLAWS



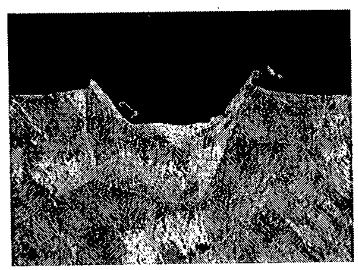
Dull dent and gouge
in Ti-6Al-4V



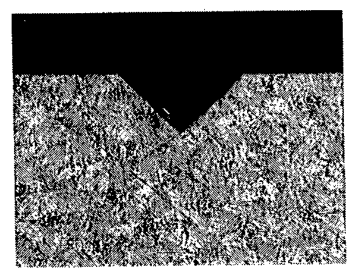
Sharp dent and scratch
in Ti-6Al-4V



Corrosion Pit
in 7075

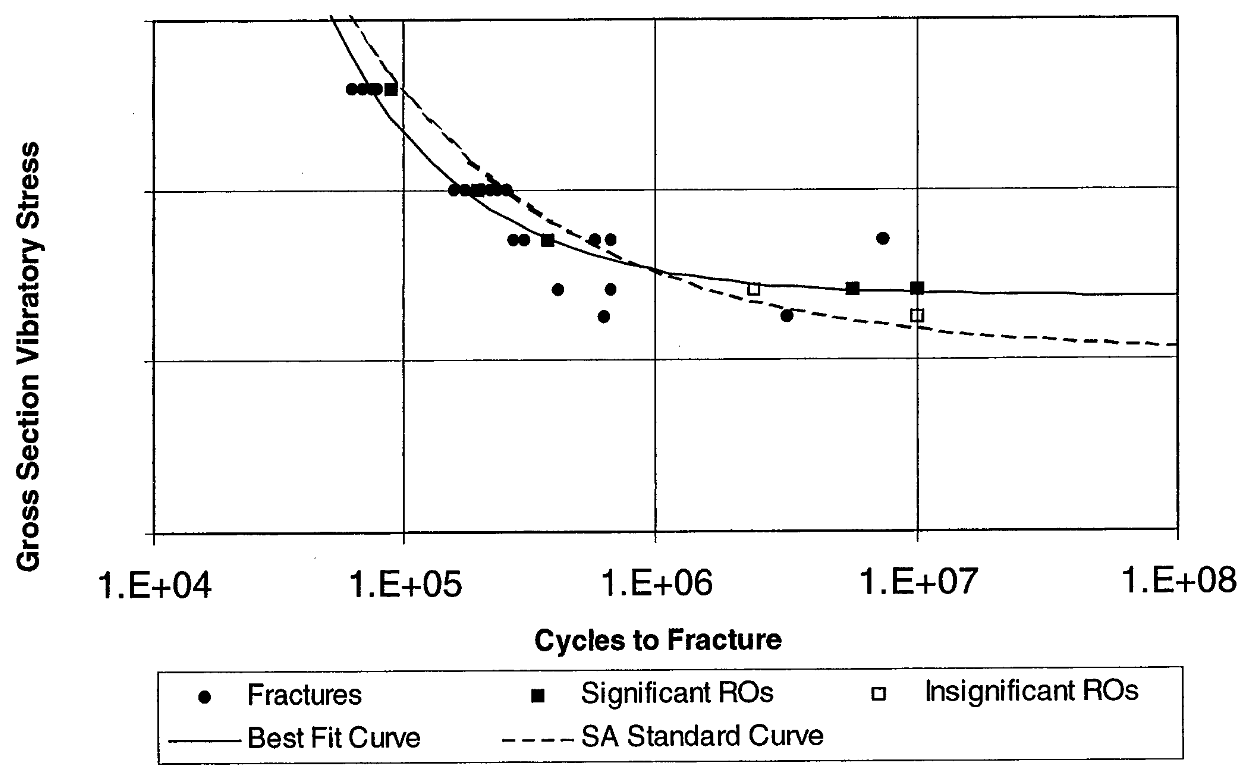


Scratch
in Ti-6Al-4V

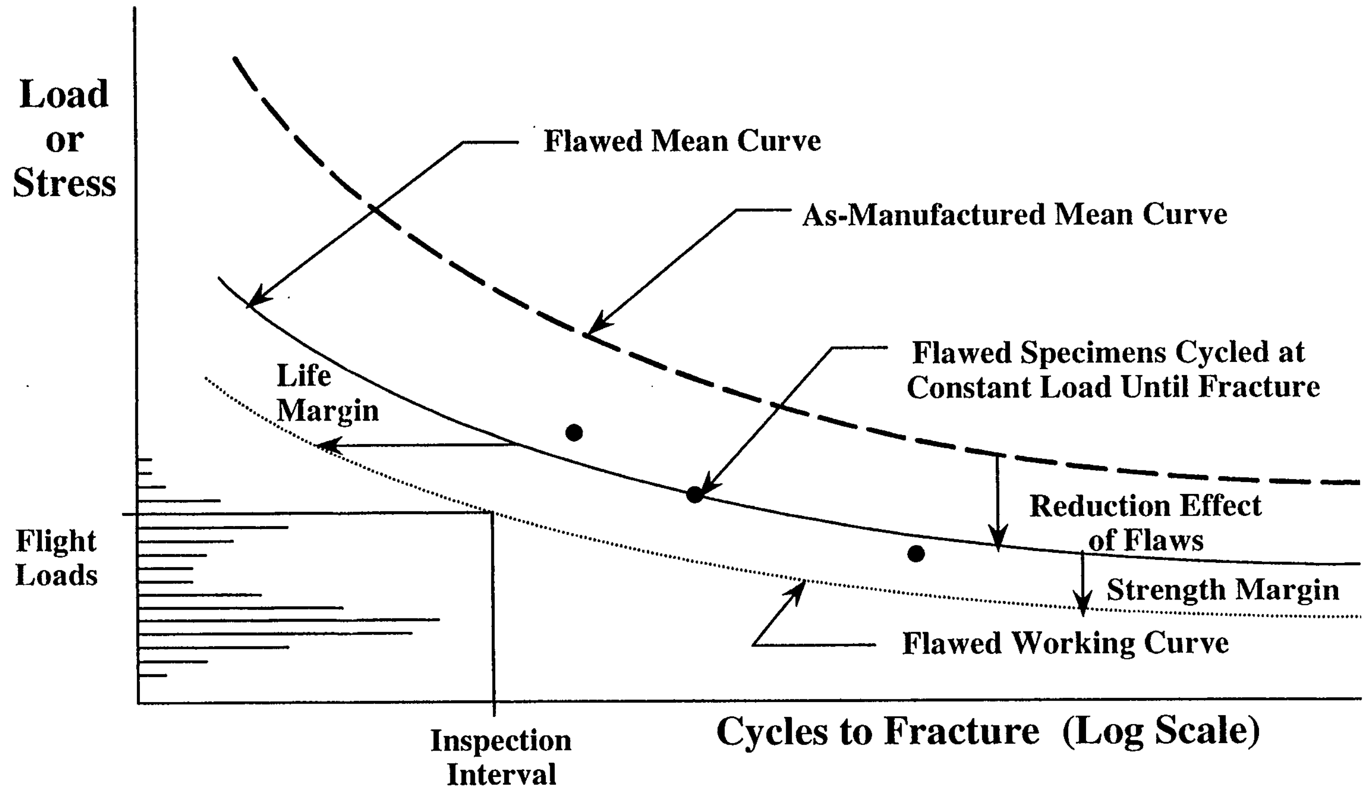


Sharp milled notch
in Ti-6Al-4V

AL7075-T73 -0.005 INCH CORROSION



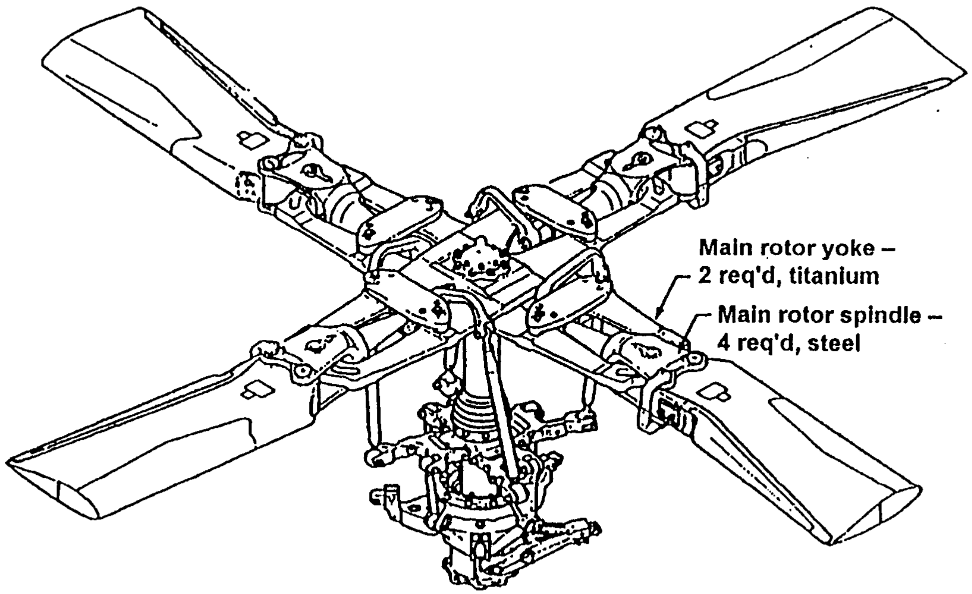
FULL SCALE FLAW TOLERANCE TEST

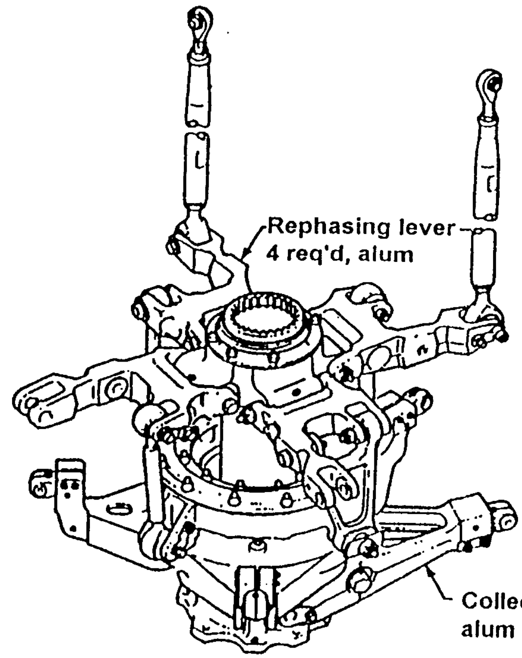


BELL - THEORETICAL REDESIGN FOR DAMAGE TOL.

FAA Investigation - Four selected rotor parts

Presented at 55th Annual AHS Forum, May 1999
James Michael, Bill Dickson, Tom Campbel





BELL - THEORETICAL REDESIGN FOR DAMAGE TOL.

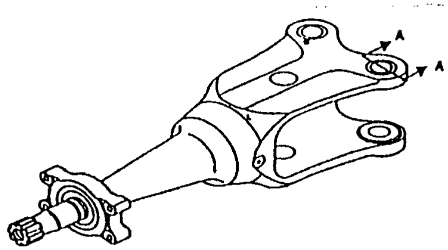
FAA Investigation - Four selected rotor parts

Presented at 55th Annual AHS Forum, May 1999
James Michael, Bill Dickson, Tom Campbell

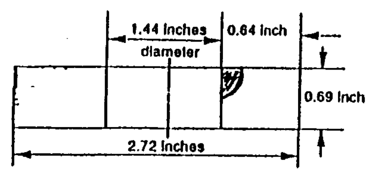
Approach:	No Growth	
	Slow Growth = Twice replacement time	
	Initial Crack Size - 0.015 inch deep (0.381 mm) semicircular	
Four Parts:	Main Rotor Yoke	- Ti-6Al-4V beta stoa
	Main rotor Spindle	- 15-5 Stainless steel
	Main rotor Rephase Lever	- Al7075-T73
	Collective Lever	- Al7075-T73

BELL - THEORETICAL REDESIGN FOR DAMAGE TOL.
MAIN ROTOR SPINDLE

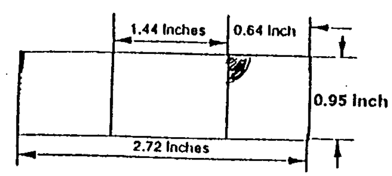
15-5 Stainless Steel
$\Delta K_{th0} = 5.0$ ksi(in)$^{1/2}$
Safe Life Repl. Time = 10000 flt hrs
Crack Growth Time = 143 flt hrs
Original Thickness = 0.69 in
Revised Thickness = 0.95 in. (38 %) (or 1.14 inches)
Crack Growth Time = 20,000 flt hrs (no crack growth)
Weight increase (2 lugs) = 1.13 lbs. (4.6%) per aircraft



Main Rotor Spindle



Section A-A



Redesign - Section A-A

BELL - THEORETICAL REDESIGN FOR DAMAGE TOL. SUMMARY

Helicopter PSE	Baseline safe-life retirement time	Baseline damage tolerant crack growth life (Ref. 2)	Calculated damage tolerant crack growth life[a]	Calculated damage tolerant inspection interval [a]	Material	Weight increase over baseline assembly (%)
Main rotor yoke	5000 hrs	20 hrs	N/A[b]	N/A [b]	6AL-4V titanium with BSHTOA	N/A [b]
Main rotor spindle	10,000 hrs	143 hrs	> 20,000 hrs	10,000 hrs	15-5 stainless steel	4.6%
Collective lever	10,000 hrs	13 hrs	No crack growth	No inspection required	7075-T73 aluminum	22%
Rephase lever	5000 hrs	78 hrs	No crack growth	No inspection required	7075-T73 aluminum	15%

[a] Crack growth life based on limited analytical study results for theoretical designs of each PSE.

[b] Composite material might be an alternative for the main rotor yoke to potentially meet damage tolerance requirements with a 5,000-hour inspection interval.

BELL - 430 ROTOR CERTIFICATION

BELL - 430 ROTOR CERTIFICATION

Transport Canada required certification to FAR29.571 Amendment 28

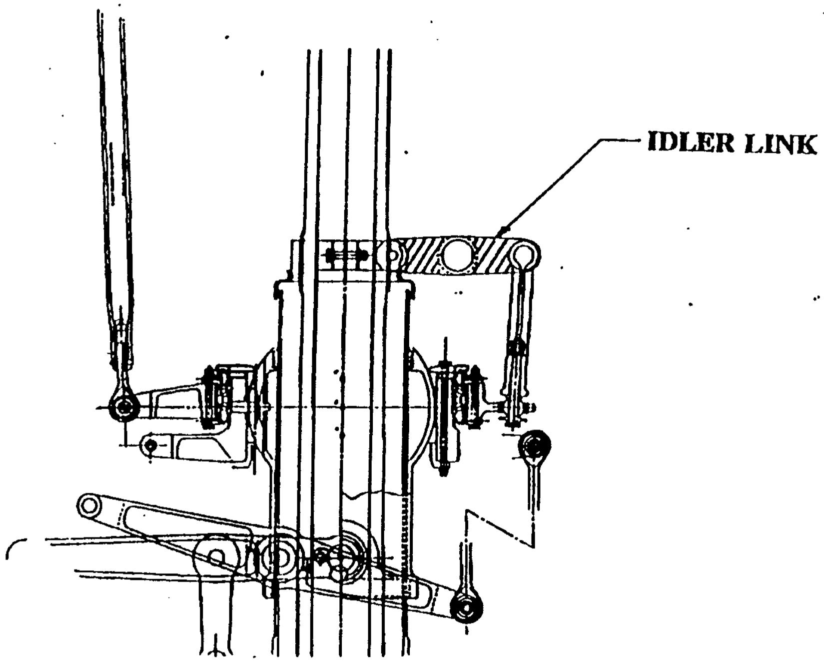
Bell Criteria - No growth or two life times from a 0.015 inch deep crack
- Used similar design criteria for more than 10 years

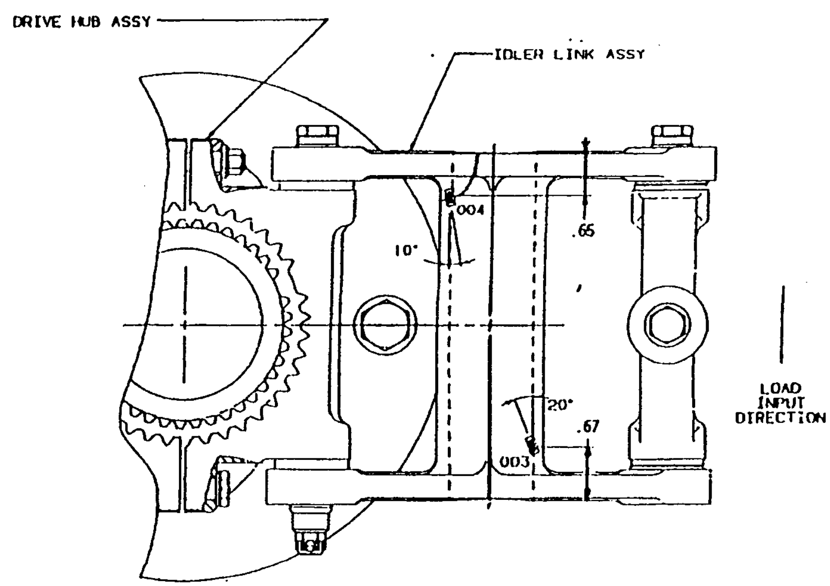
Full scale S-N testing with strain surveys to validate analytical models

Crack growth analysis: 0.015 inch deep crack will not grow, or two lifetimes from an 0.015 inch deep crack

BELL - 430 ROTOR CERTIFICATION

SWASHPLATE DRIVE ASSEMBLY IDLER LINK





BELL - 430 ROTOR CERTIFICATION

SWASHPLATE DRIVE ASSEMBLY IDLER LINK

- Strain measured at crack origin location
- Measured strain above crack growth theshold.
- Part redesigned to reduce strain by a minimum of 15 %.
- Subsequent test of new design - Local strain below crack growth threshold

Rotorcraft Industries Association (RITA)

Damage Tolerance For Helicopter Structure

Jim Cronkhite
Bell Helicopter Textron Inc.

George Schneider
Sikorsky Aircraft

Technical tasks in this document include tasks supported with shared funding by the U.S rotorcraft industry and the Government under RITA/NASA cooperative agreement NCCW-0076, Advanced Technology, August,1995

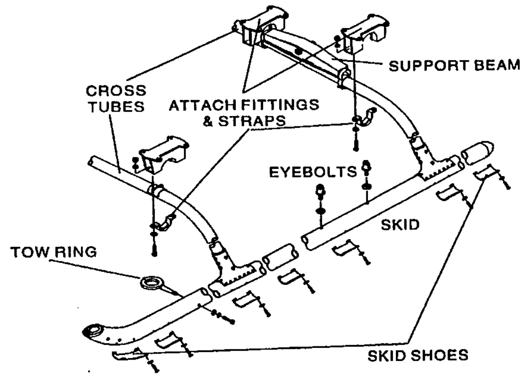
Bell Helicopter Textron Inc.

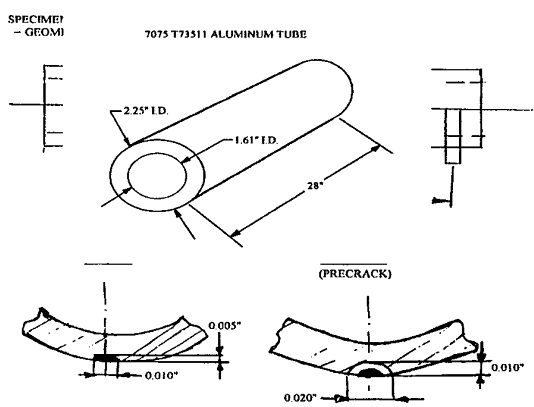
Global Project Objective: Improve safety and reduce design and O&S costs through improvements in damage tolerance methods and tools.

- Software Validation (CRACKS98, NASGRO 3.0, etc.)
- Crack Growth Rate Data (C(T) and K_b - bar specimens)
- Component Crack Growth Tests
  - Landing Gear Cross Tube
  - Idler Link
- NDE
  - Multilayer Aluminum Sheet Corrosion
  - Evaluate Operator Field Techniques
- Structural Integrity Methodology

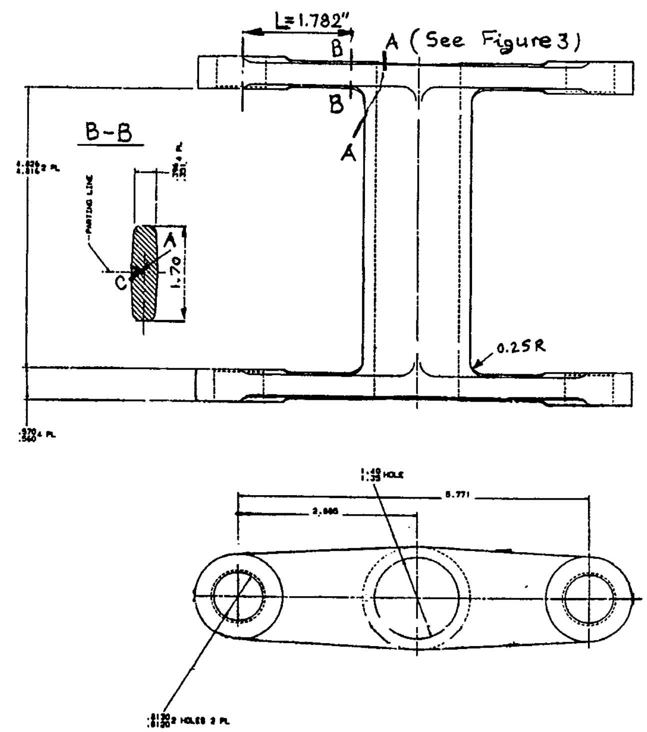
COMPONENT CRACK GROWTH TESTING

M407 Skid Gear Attachment





M430 Idler Link Testing

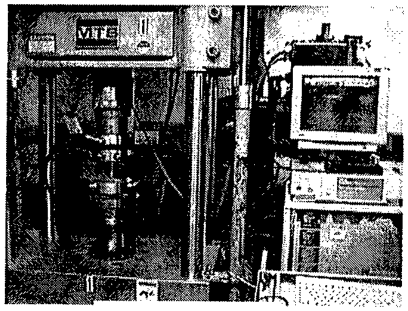


General Objective: Develop technical foundation for the transition from safe-life to damage tolerant design, certification, and management of helicopter structure.

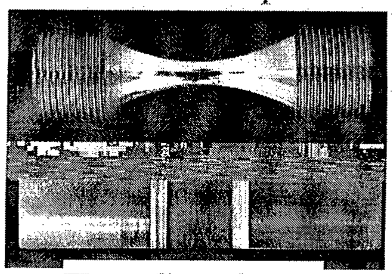
- FASTRAN, NASGRO 3.0, Crack Closure Evaluation
- Polished Specimen Fatigue and Crack Growth Tests
- NASGRO Strip Yield (Closure) Model Developed From Polished Specimen Data
- Machined and Flawed Specimen Fatigue Tests
- Crack Growth Analysis With Residual Stress - Shot Peening
- Corrosion / Crack Growth Testing
- Spectrum Load Polished, Machined, and Flawed Specimen Fatigue Tests
- Equivalent Initial Flaw Sizes (EIFS) - Best Fit and Distributions
- Full Scale Part Fatigue and Crack Growth Evaluation
- Final Report and Damage Tolerance Procedures Manual

Damage Tolerance For Helicopter Structures

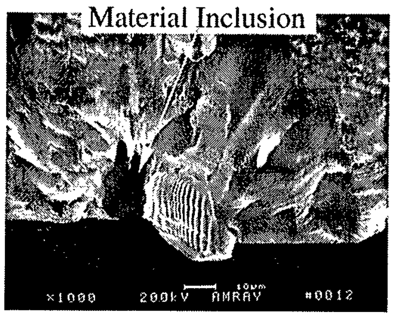
Material Fatigue & Crack Growth Tests



Test Setup

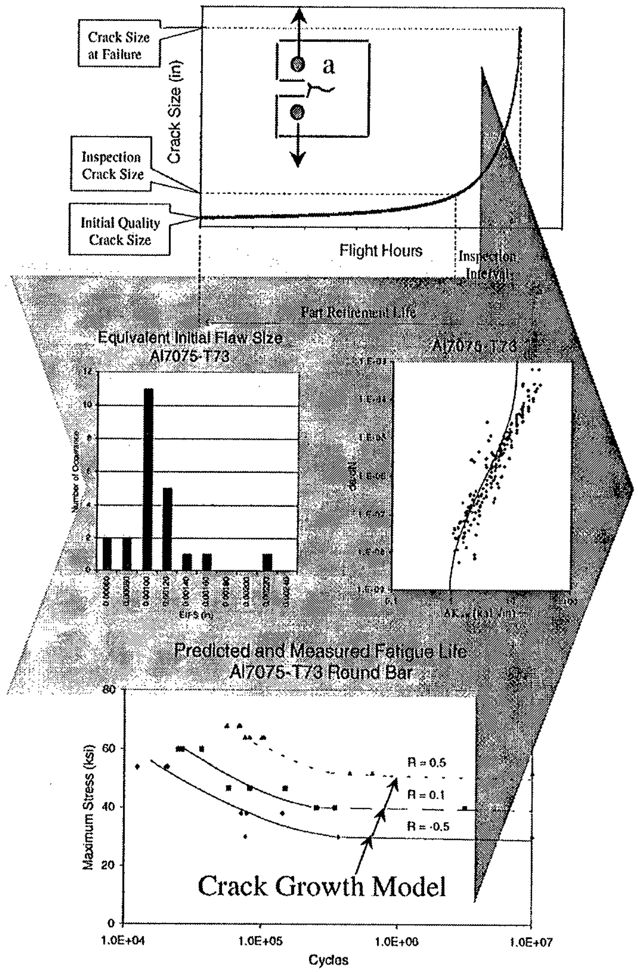


Test Specimen



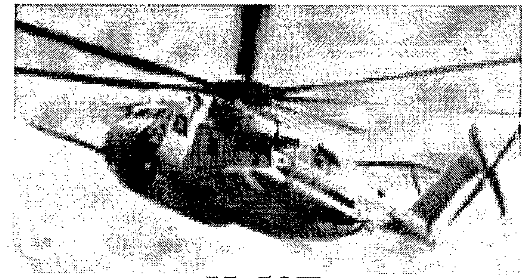
Microstructure Study

Damage Tolerance Crack Growth Model

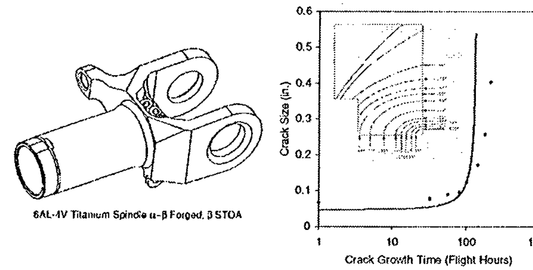


Prediction vs. Measurement

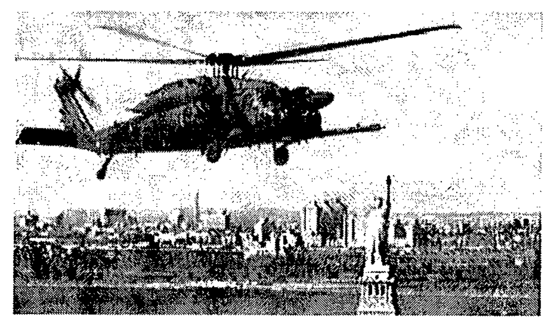
Technology Application



H-53E



Component DT Evaluation



H-60 Black Hawk

CONCLUSIONS

- Damage tolerance requires stress reduction over safe-life.
- Bell theoretical study - 5% to 22 % weight increase (maybe less with optimization)
  - one component switch to composites recommended
- Sikorsky H-53 study - rotor head redesign required for DT

SOME ISSUES

- Crack Growth Rate and Threshold
  - better data base
  - sensitivity of design
  - standardization
- Threshold - problems with load shedding tests and current threshold data
- Life Enhancement -analysis methods needed
  - reluctance to shot peen and cold work due to cost
- NDE
  - 0.010 to 0.030 inch cracks for dynamic systems
  - economics and practicality
  - service inspection - cracks or flaws
- Certification testing requirements ?

SESSION B

DAMAGE TOLERANCE IN DYNAMIC COMPONENTS

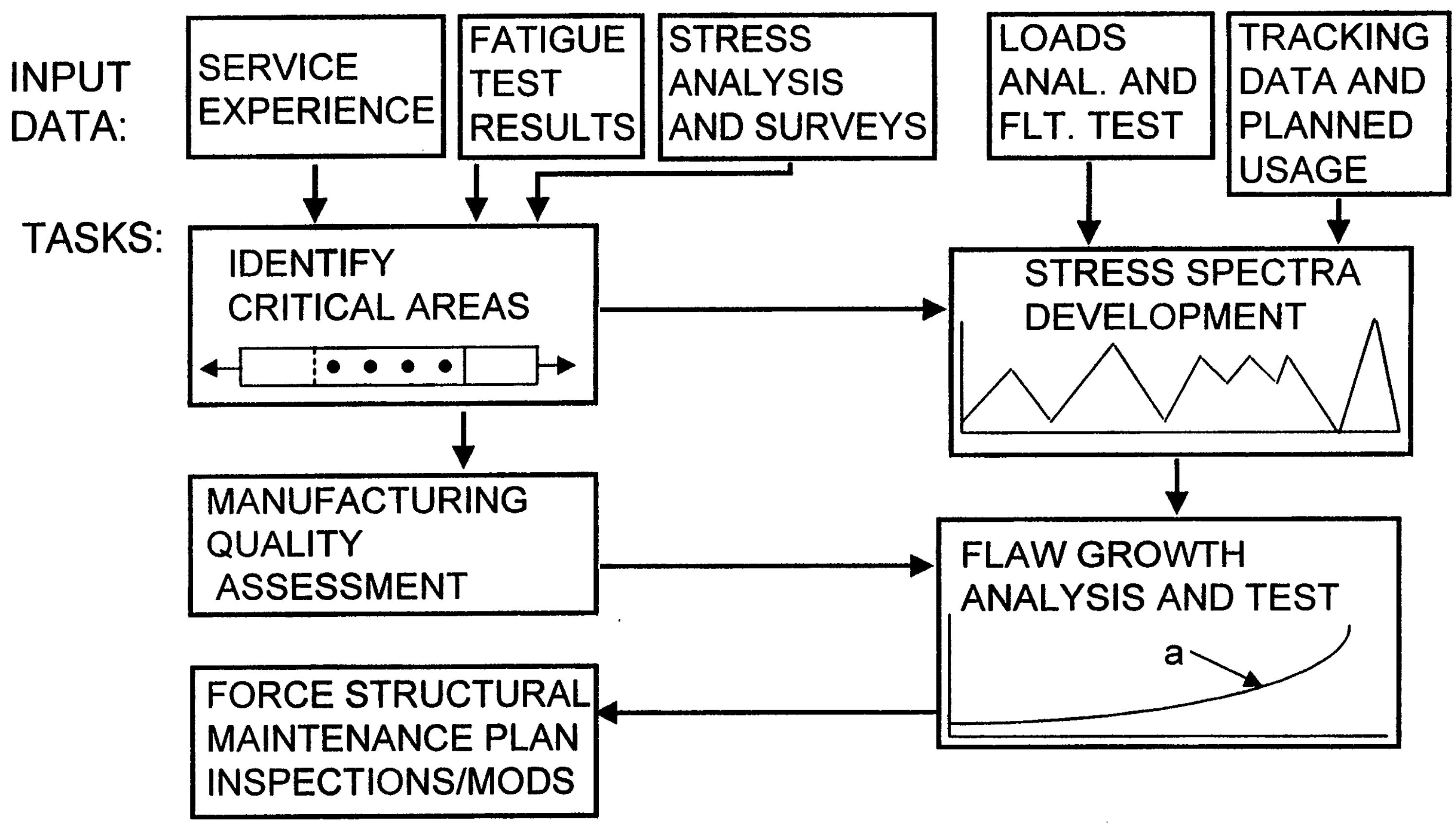
U.S. AIR FORCE VIEWS ON DAMAGE TOLERANCE FOR HELICOPTERS

John W. Lincoln
Engineering Directorate
Aeronautical Systems Center
Wright-Patterson Air Force Base, Ohio

BACKGROUND

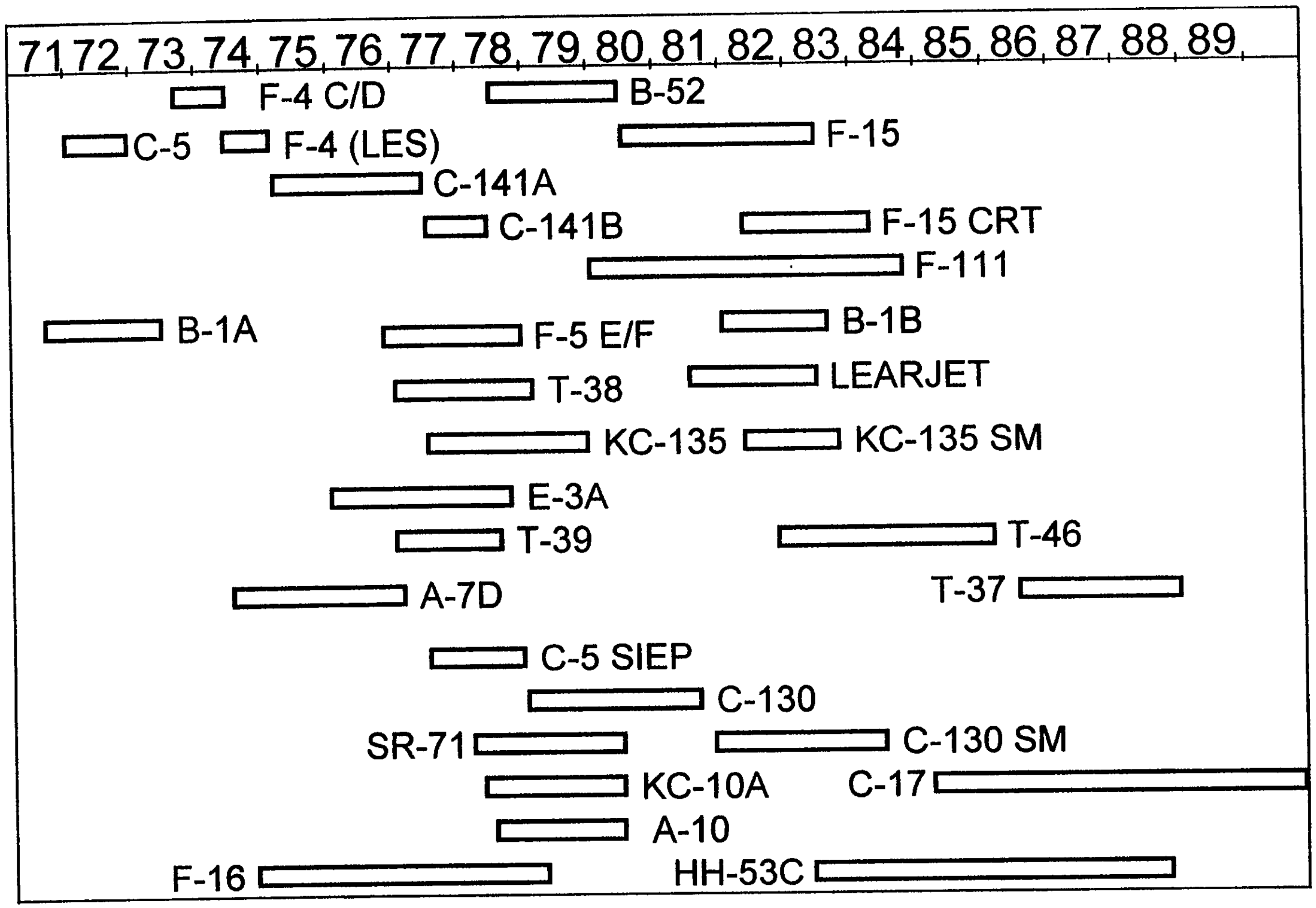
- Success with damage tolerance assessments of fixed-wing aircraft and engines motivated the USAF to perform a damage tolerance assessment of a helicopter
- The USAF initiated a damage tolerance assessment on the Sikorsky HH-53 helicopter starting in 1983

DAMAGE TOLERANCE APPROACH



OUTPUT: INSPECTION AND MOD REQUIREMENTS BY TAIL NUMBER

DAMAGE TOLERANCE EXPERIENCE - AIRCRAFT

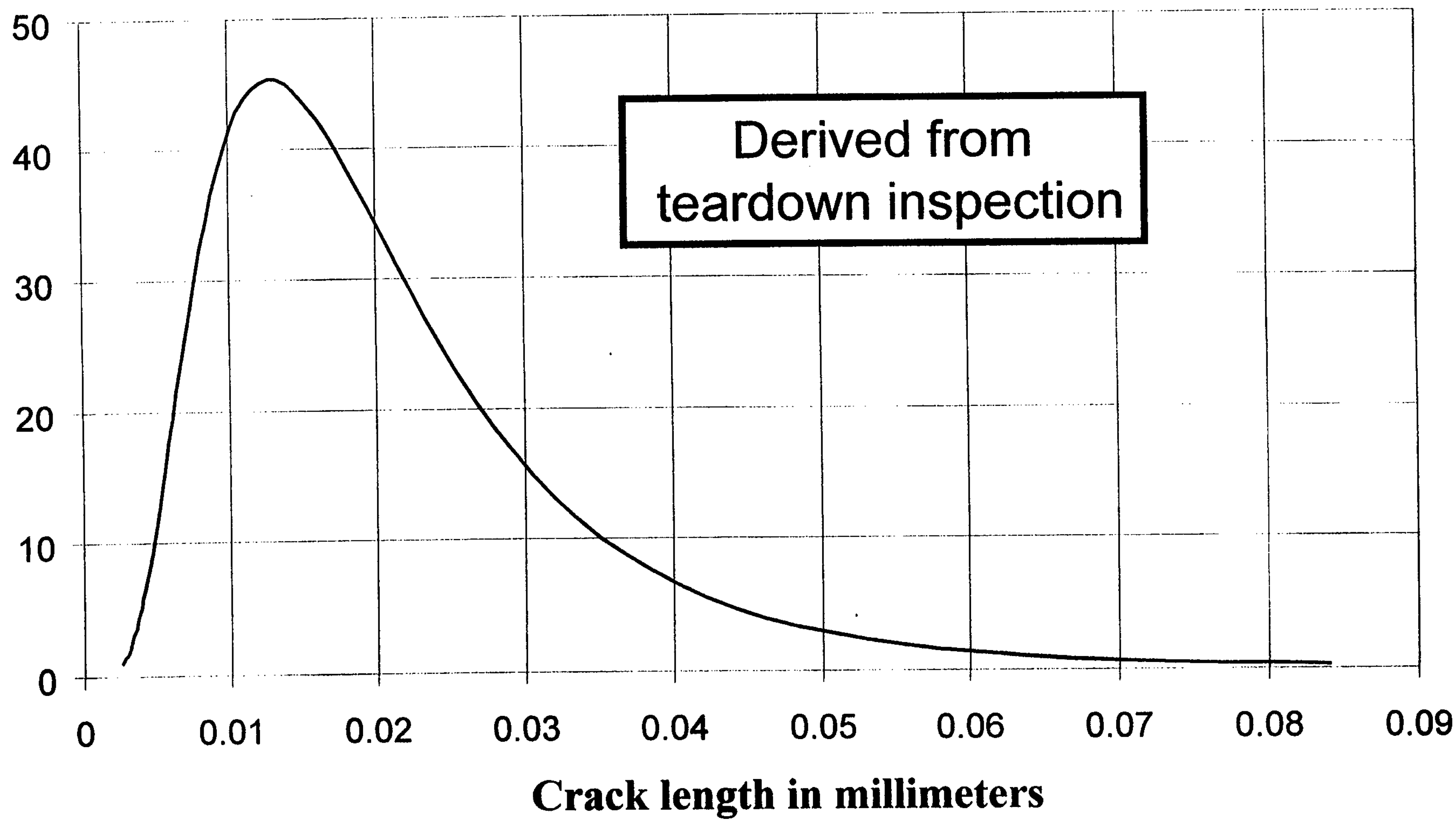


MAJOR FINDING

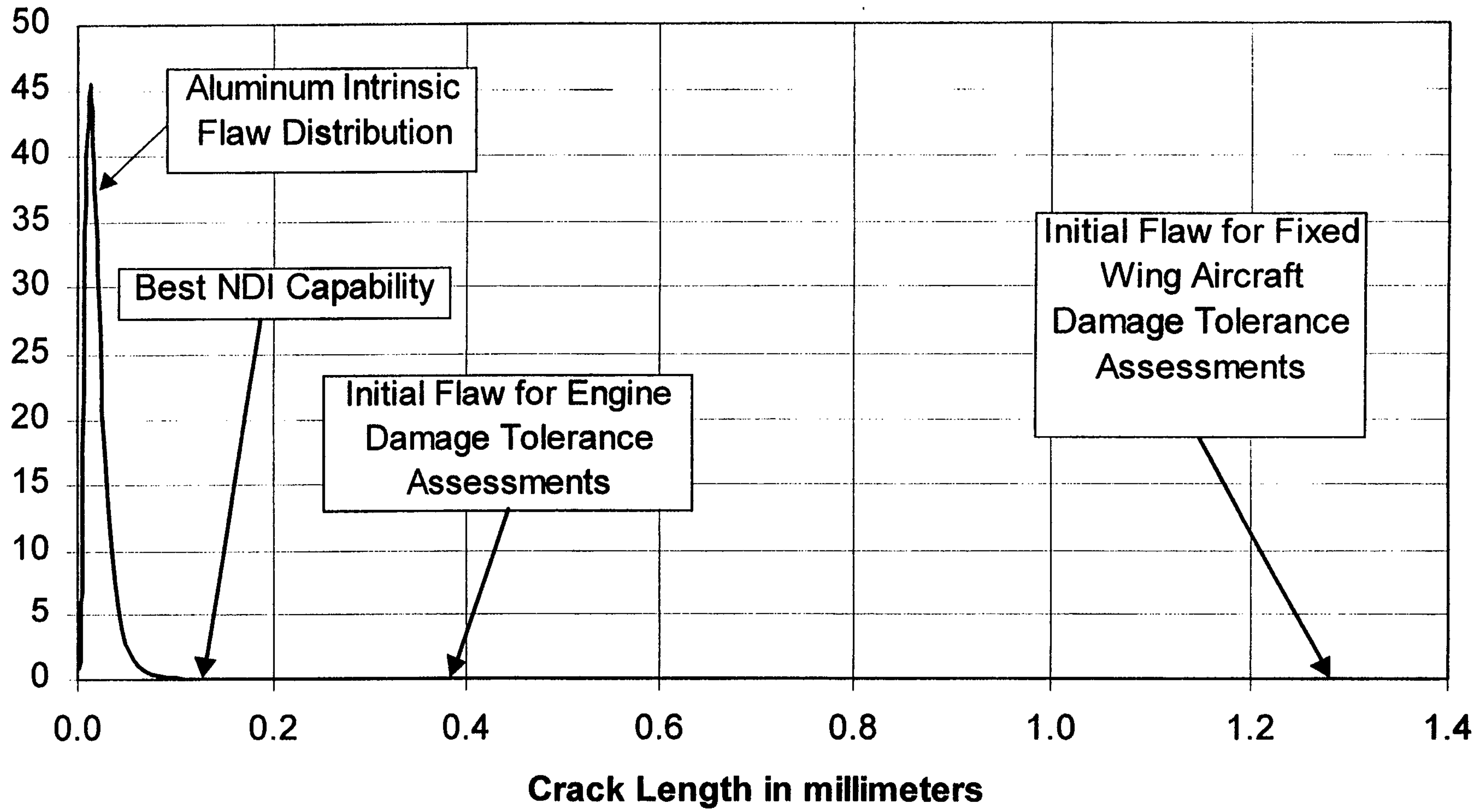
- Original design of the HH-53 based on conventional safe life practice
- Defect distribution for original design representative of intrinsic flaws found in the materials tested

Nondestructive inspection capability inconsistent with defect distribution used in the initial design

INTRINSIC FLAW DISTRIBUTION FOR ALUMINUM



INITIAL STRUCTURAL FLAWS



OTHER FINDINGS FROM THE HH-53 DAMAGE TOLERANCE ASSESSMENT

- Required stress analysis compatible with current technology
- Available stress spectra conservative
- Many critical locations found
  - Although high cycle stresses caused many areas to be not inspectable many areas found to be compatible with inspection capability
- Confidence low in analyses involving crack growth threshold

OBSERVATIONS

- Many parts in existing helicopters are not manageable by damage tolerance methods
- Existing helicopters should be assessed by damage tolerance approach to determine flaw sensitivity of components
- Engine damage tolerance initial flaw sizes are suitable for helicopter design

OBSERVATIONS (CON'T)

- Inclusion of short crack effect is critical for determination of crack growth
  - technology now available to do this
- Effect of stress range truncation should be integral part of damage tolerance assessment
- Shot peening influential in extending lives of component
  - inspection program may still not be viable

OBSERVATIONS (CON'T)

Flaw tolerant design

- database insufficient to determine determine design flaw size
- difficult for certification authorities to determine if the flaws are representative of flaws expected in service operations
- as with conventional safe life design the flaw tolerant design is based on linear cumulative damage

OBSERVATIONS (CON'T)

Damage tolerant design

- for some components requires reduction in stress that would be adequate for conventional safe life design
- provides for a more balanced design and consequently weight penalty should be small
- fracture mechanics technology available for analyses and tests

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DSTO

Crack Growth in a Tail Rotor Output Shaft

Simon Maan
Aerostructures Technologies Pty Ltd

Domenico Lombardo
DSTO

Workshop on Damage Tolerance in Helicopters
Cranfield, U.K., 4 - 5 April 2000

OUTLINE

- Introduction
- Static Load Analysis
- Crack Growth Analysis
- Conclusions
- Future Work

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INTRODUCTION

History

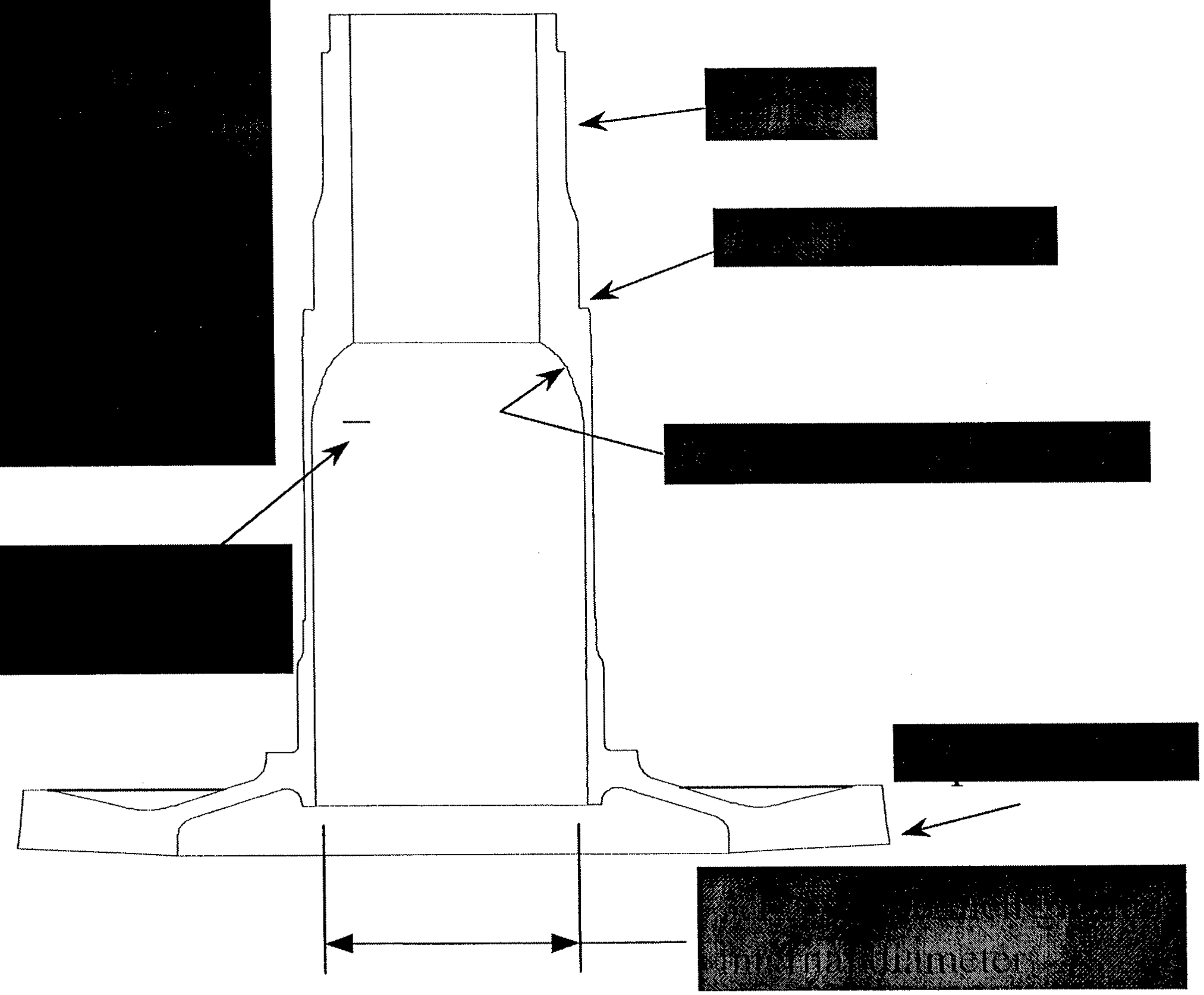
- On June 9, 1993 a US Army EH-60 Black Hawk crashed due to failure of its TR output shaft
- Prior to this, a Chinese S-70C Black Hawk crashed for the same reason
- The Australian Defence Force, supported by DSTO, took action in 1993/94 to ensure the integrity of the shafts in the Australian Black Hawk and Seahawk fleet

Failure Details

- Both failures due to a crack initiating at an internal radius
- For EH-60, crack initiation was thought to be due to incorrect machining
- EH-60 shaft: 580 flight hours at failure (5400 hour retirement life)
- S-70C shaft: 350 flight hours at failure
- This raised questions about the shaft's tolerance to manufacturing flaws

Shaft geometry

[redacted]



Shaft Details

- Two versions of shaft existed: 70358-06620-101 ("Dash 101") and 70358-06620-102 ("Dash 102")
  - "Dash 101" not shot-peened
  - "Dash 102" Shot-peened
- Both failed shafts were "Dash 101"

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STATIC LOAD ANALYSIS

Objective

- Initial Objective:
  - Determine crack growth rates in the shaft
- Longer-term objective
  - Determine time between earliest possible warning if health monitoring had been present and final failure of shaft
  - Determine if health monitoring of the shaft is feasible

Aerostructures Role

- Aerostructures Technologies Pty Ltd was tasked with:
  - Creating a Finite Element Model (FEM) of the tail rotor output shaft,
  - Performing a crack growth investigation
  - Validating FEM results by experiment

Modelling of the Shaft

- Modelling of the Shaft
  - Four 3D NASTRAN models created using brick elements (Hex 20)
    - » Three models based on geometry of three US Army shafts currently with DSTO
    - » One model based on Sikorsky detail drawings
  - Loads and boundary conditions based on Sikorsky fatigue test of shaft
  - Model results correlated against fatigue test results

Results of the Investigation

- Stress concentration factors for all four FEMs varied between 1.42 and 1.52
- Peak stresses < 40% of Yield
- Stresses around one inch radius are sensitive to actual geometry of the radius
- Conclusion:
  - Shaft failure in only a few hundred hours difficult to explain if shaft manufactured as per design and no pre-existing flaw is present

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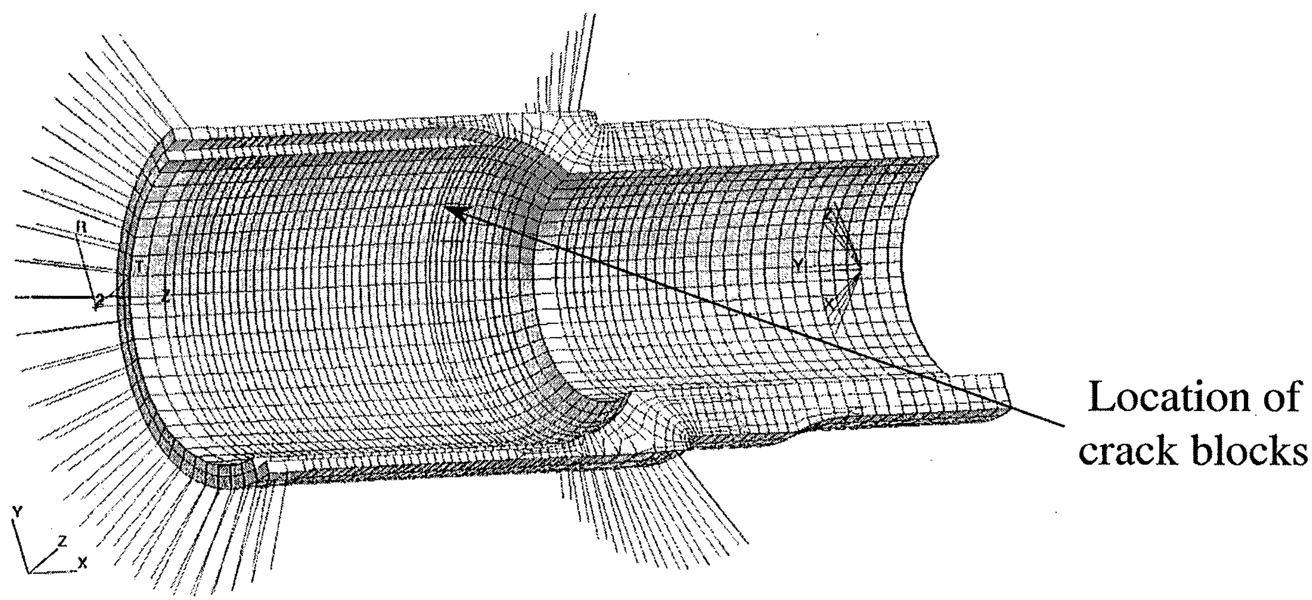
CRACK GROWTH ANALYSIS

Crack Growth Analysis

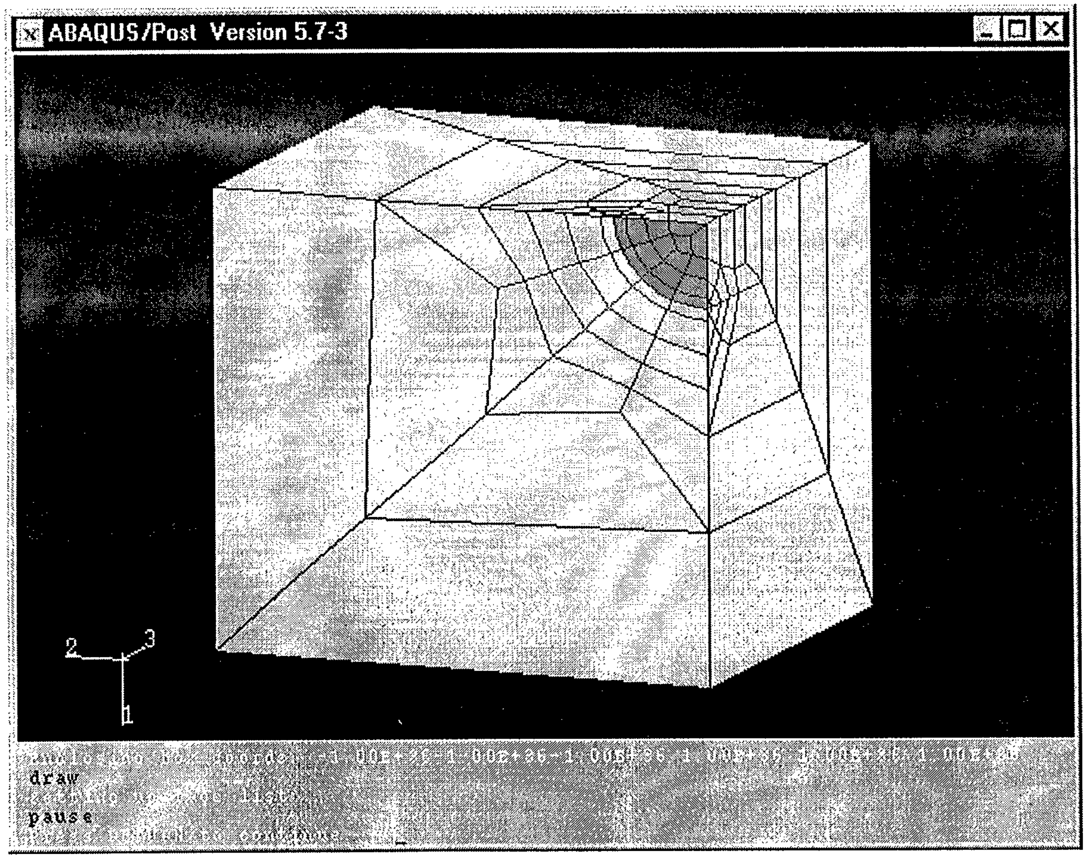
- NASTRAN FEM converted to ABAQUS FE Solver format for use with ZENCRACK
- ZENCRACK allows a 3D crack to be placed into a 3D FE model.
- ZENCRACK also allows the crack to grow in 3 dimensions based upon max stress orientation
- Modified US Army UH-60A/L design usage spectrum used

ZENCRACK Model

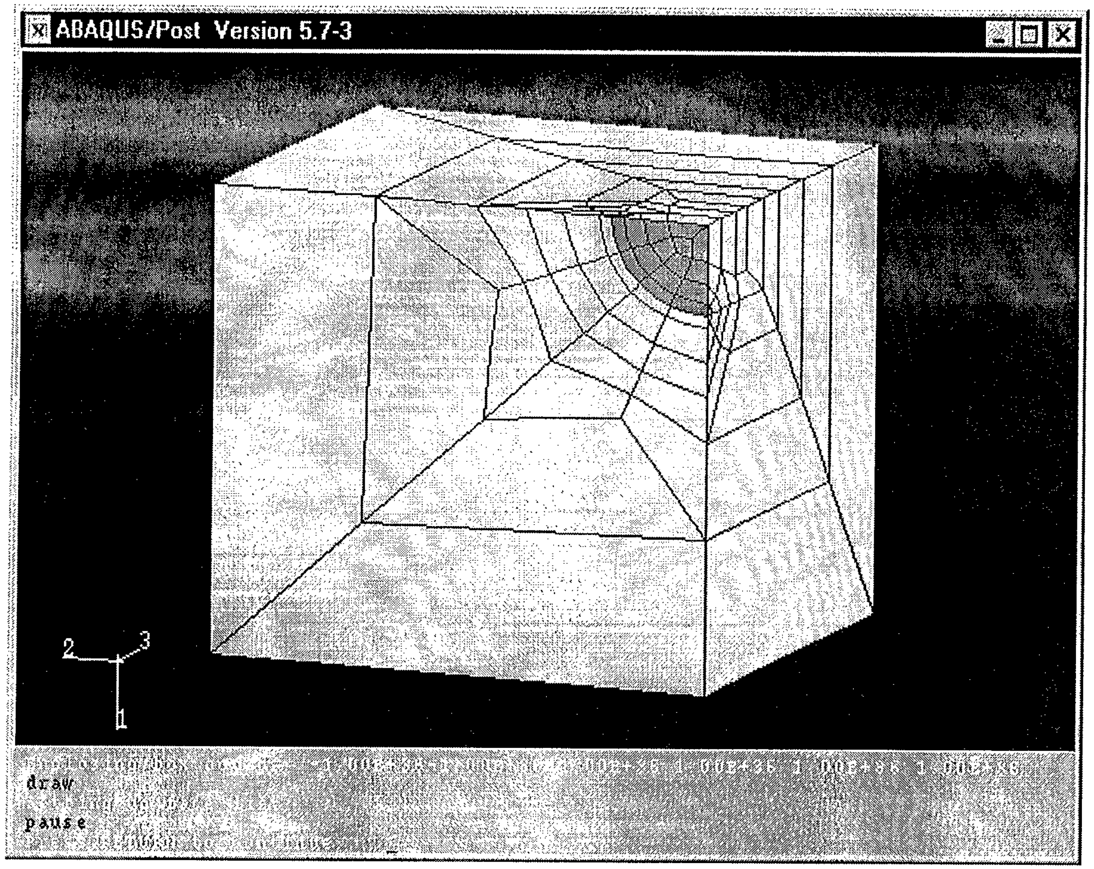
- At the location shown eight crack blocks were inserted into the mesh to model the defect's initial size



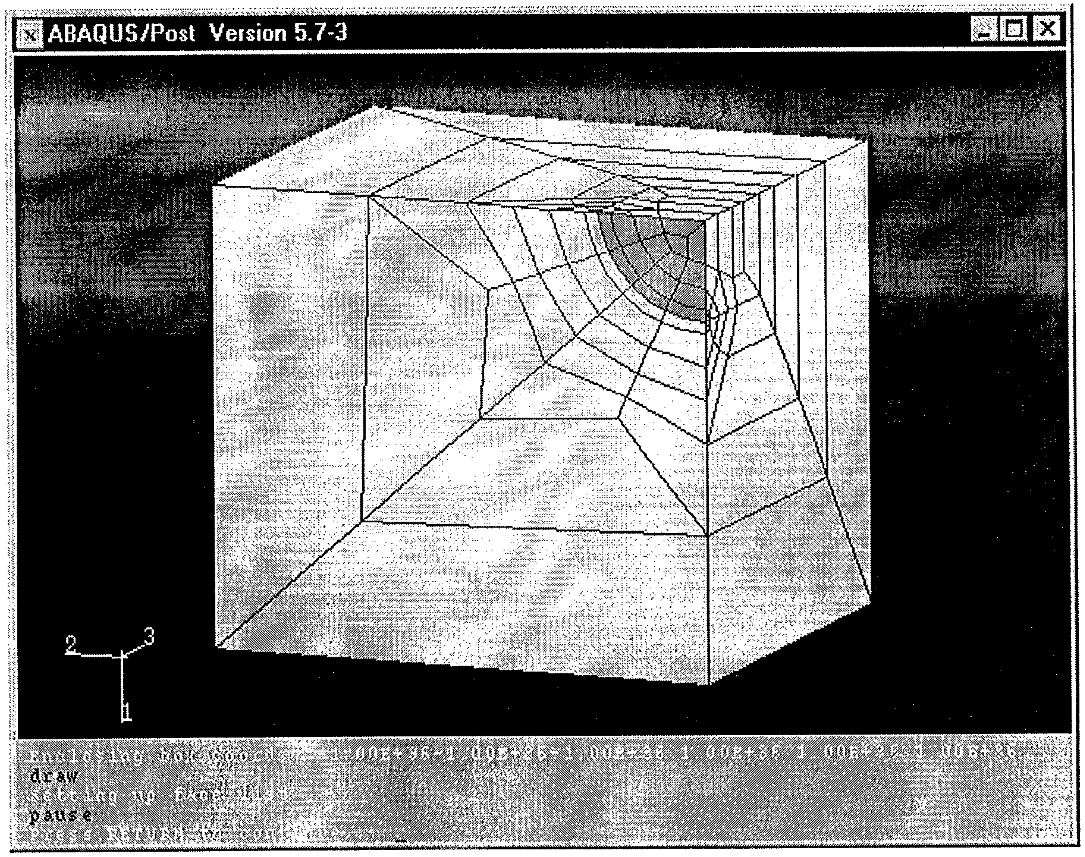
Crack growth



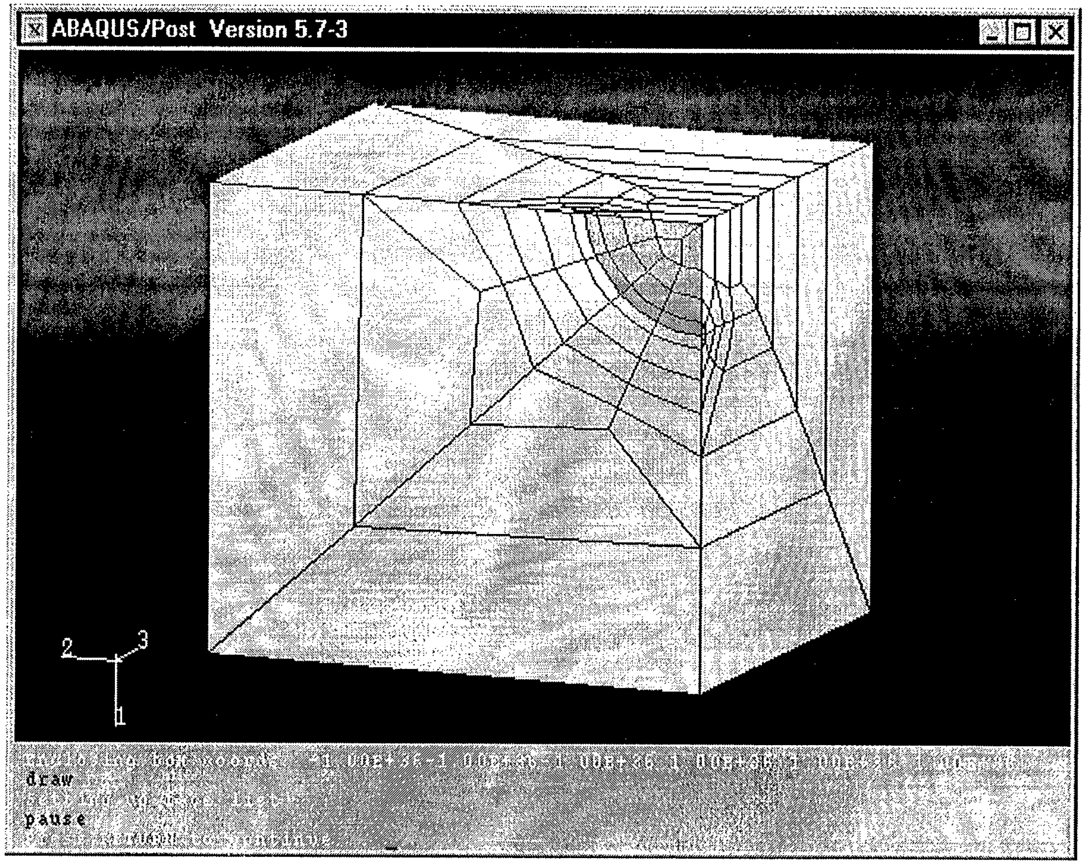
Crack growth



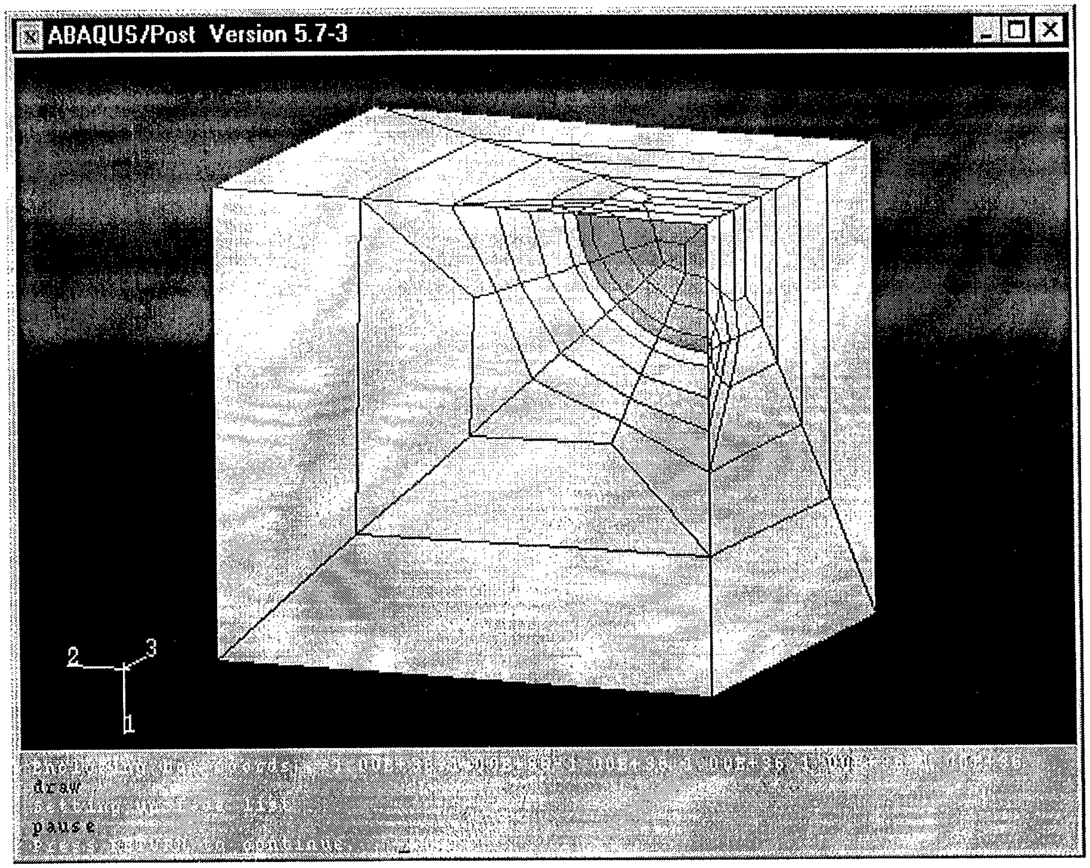
Crack growth



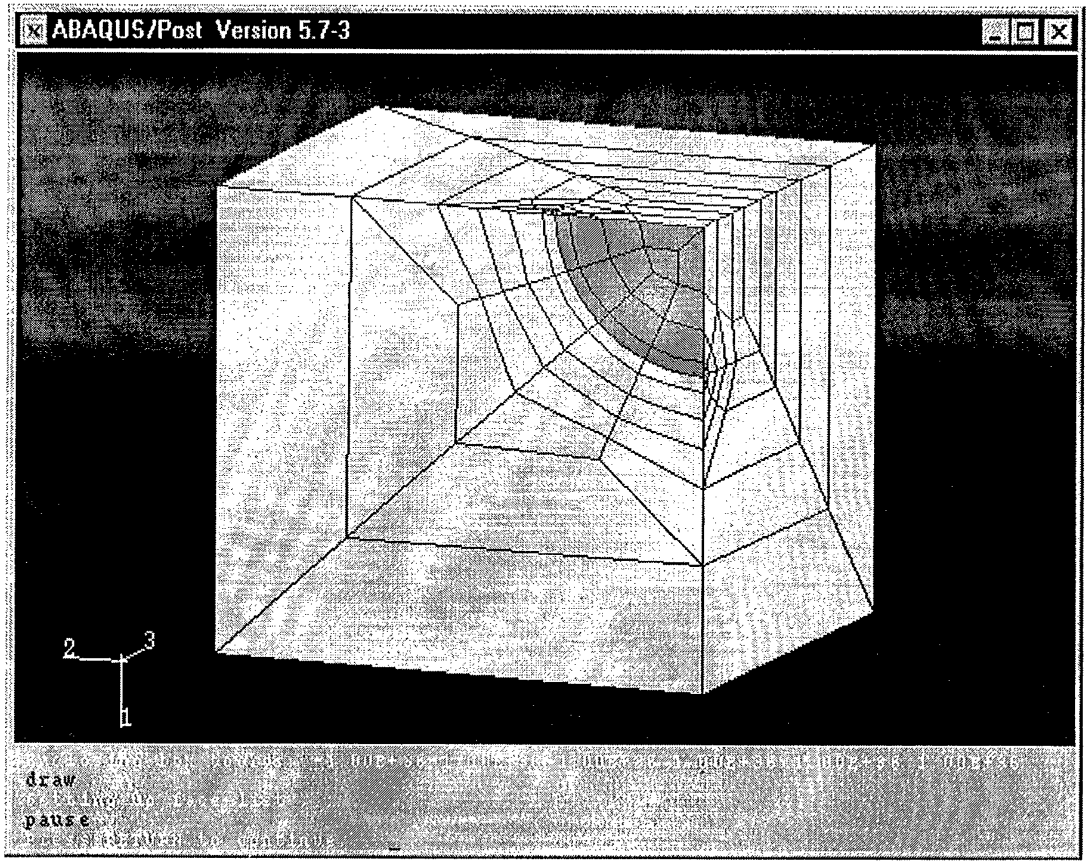
Crack growth



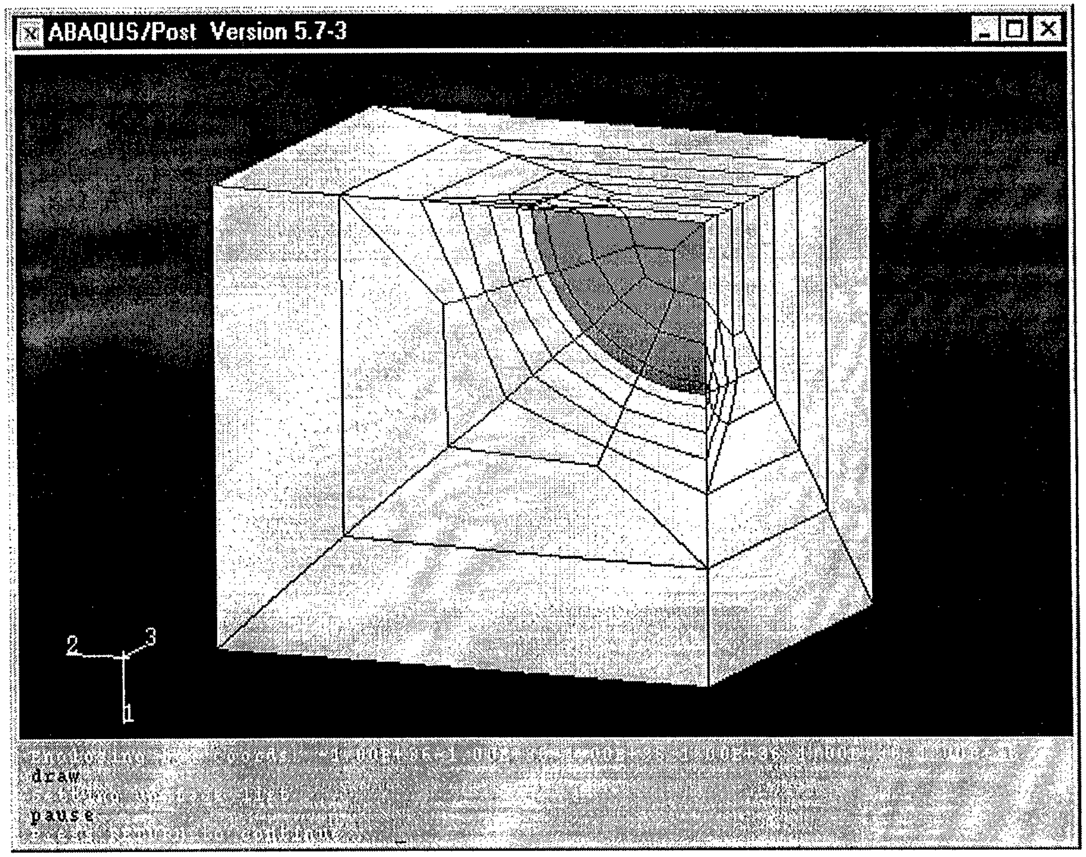
Crack growth



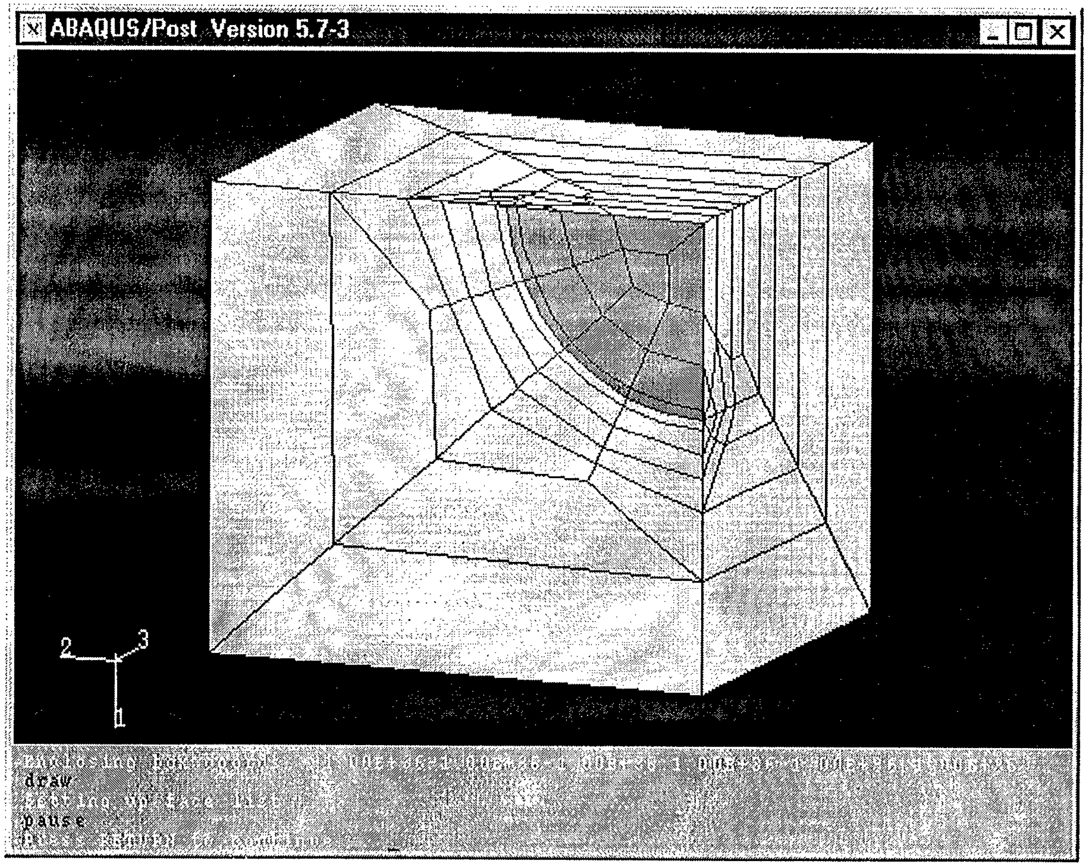
Crack growth



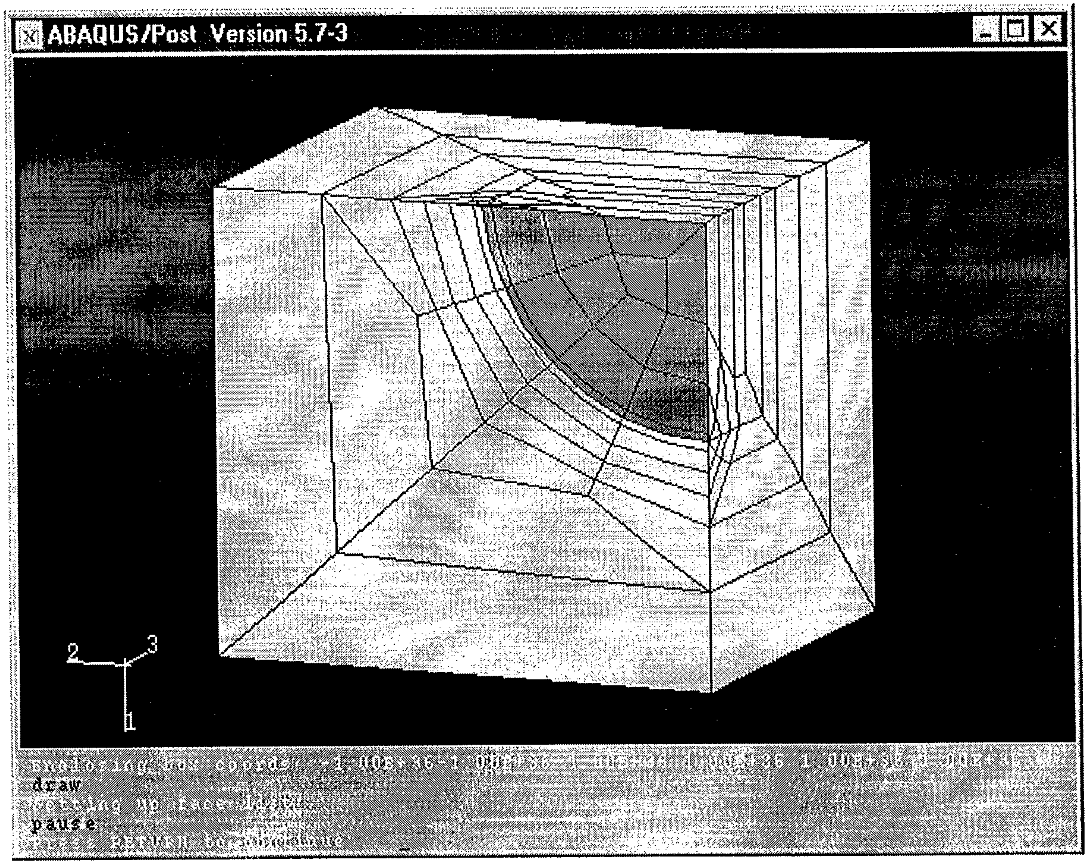
Crack growth



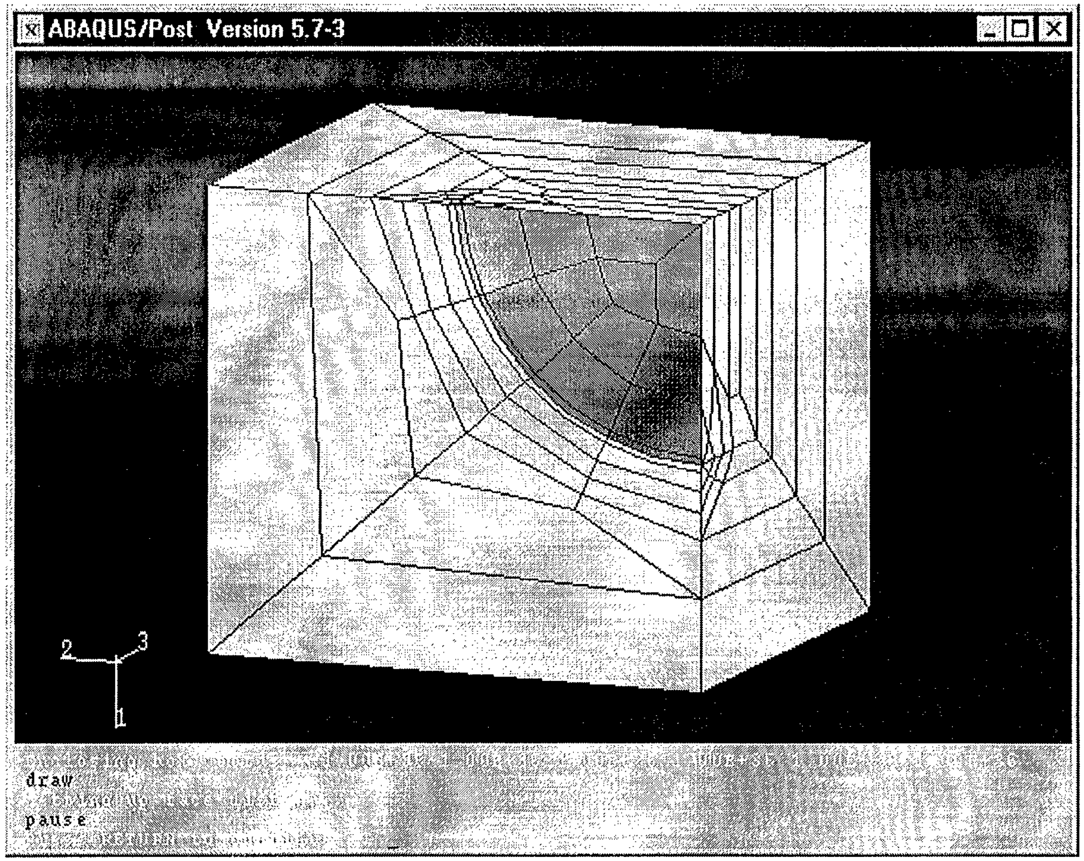
Crack growth



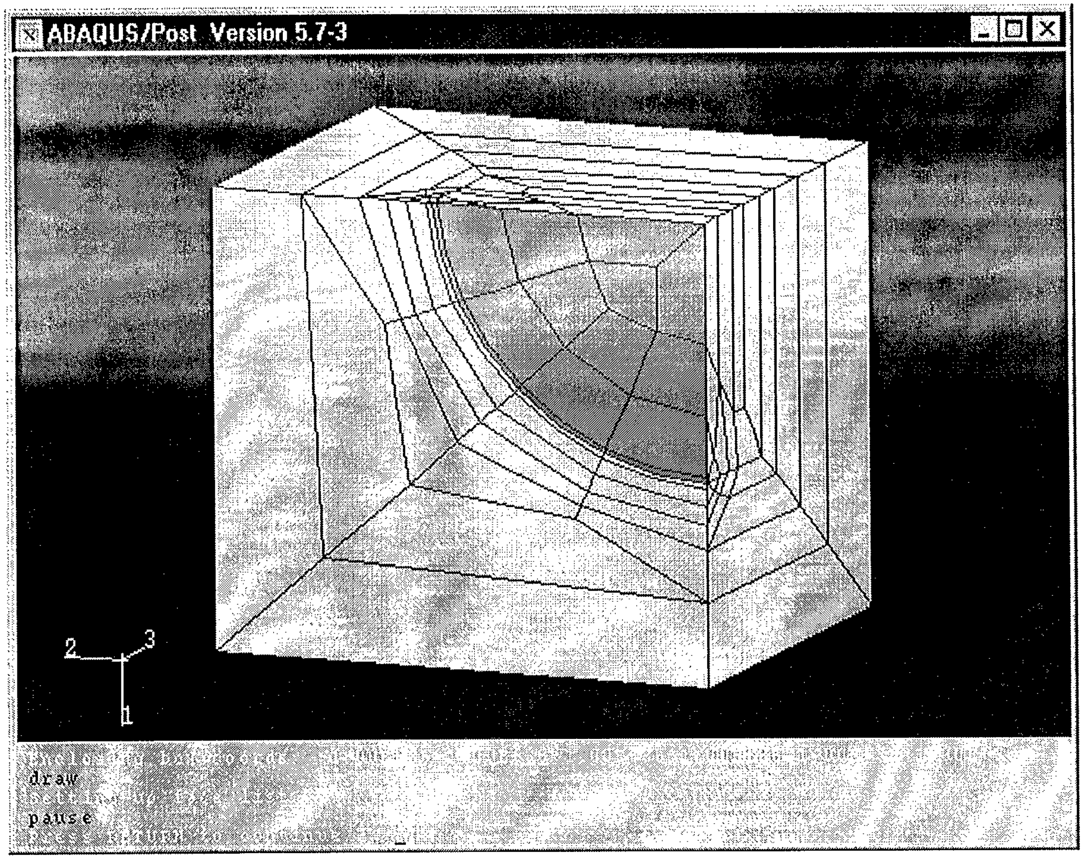
Crack growth



Crack growth



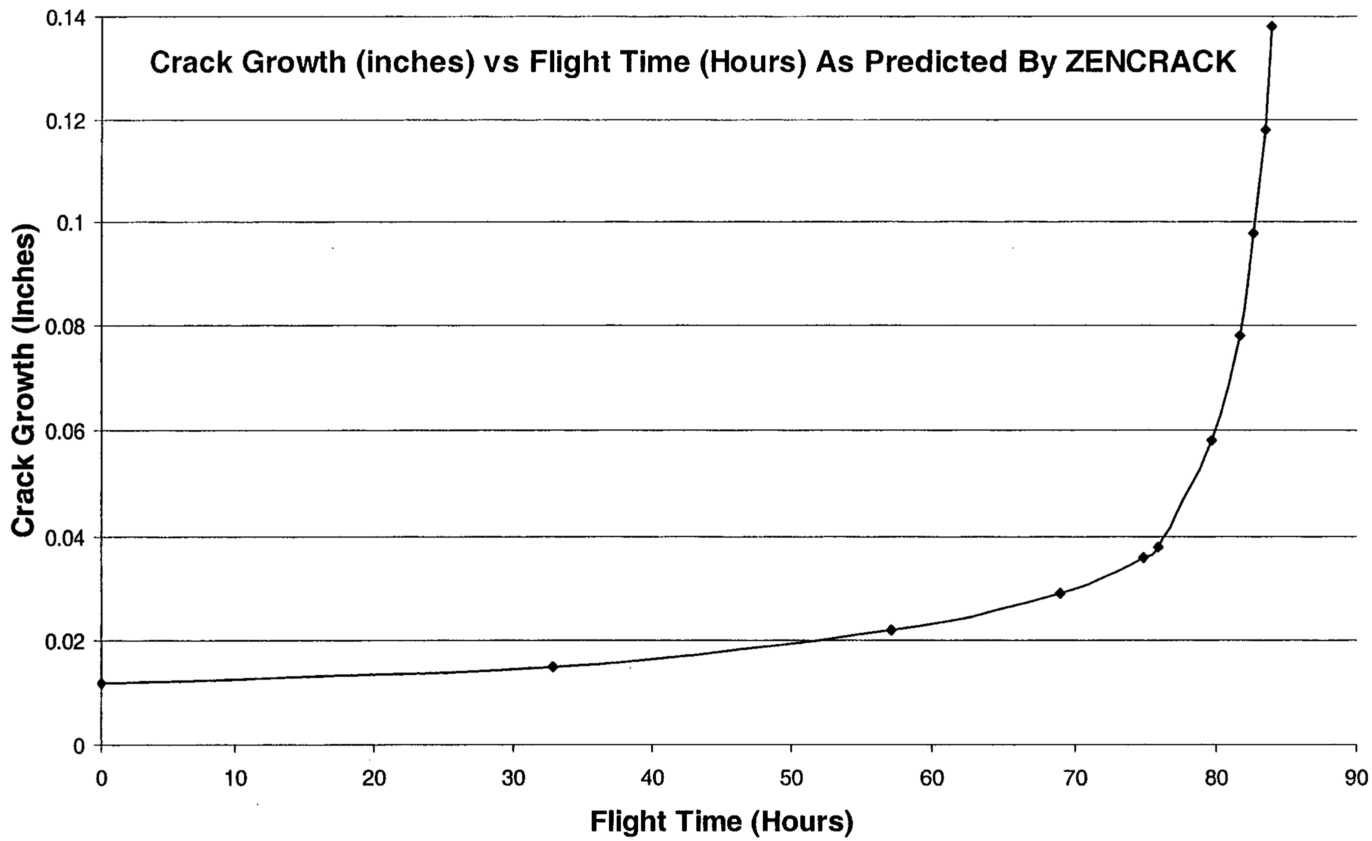
Crack growth



Replay

ZENCRACK Crack Growth Curve

- The smallest crack that could be modelled measured 0.012 x 0.395 inches (depth x width)



ZENCRACK Crack Growth Results

- Time to failure predicted to be $\approx$ 100 to 150 hours
- Compare with 580 hours (EH-60) and 350 hours (S-70C)
- Hence, presence of flaw severely reduces life
- Failure attributed to flaw due to either machining procedure or intergranular corrosion

ZENCRACK Limitations

- Problems with applying the usage spectrum
- Only a part of the crack growth history could be modelled
- Therefore couldn't completely quantify results

AFGROW Crack Growth Analysis

- AFGROW was used to provide a comparison with the ZENCRACK results
  - Semi-circular element used
  - Applied max loads from the four NASTRAN FEMs to the element boundary
  - Applied an unclipped load spectrum
  - 3D model, but 2D crack growth

AFGROW Crack Growth Results

- Similar results to Zencrack
- 67 hours to grow the crack from 0.03 inches x 0.19 inches to a crack that penetrated the outer wall of the shaft (0.233 inches)
- AFGROW would not initiate a crack < 0.03 inches

FEM Experimental Validation

- Correlation so far based on Sikorsky fatigue test results
- However:
  - Insufficient strain data exist for proper correlation
  - Correlation performed at locations of applied load (possible local stress effects)

Static Strain Survey

- NASTRAN FEM results will be verified by experimental testing
- Static strain survey of the shaft to be done
- Shaft has strain gauges at locations near to the crack initiation site
- Results pending

CONCLUSIONS

Conclusions

- Tail rotor output shaft is sensitive to flaws near the internal one inch radius
- Large life reduction results from presence of flaw
- Shot-peening beneficial

FUTURE WORK

Future Work

- Ideally, perform crack growth testing on shafts to validate Zencrack and AFGROW results
- Determine when a health monitoring system might have given warning of impending failure
- Determine feasibility of health monitoring of shaft

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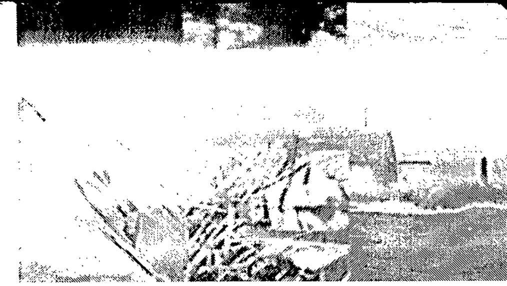
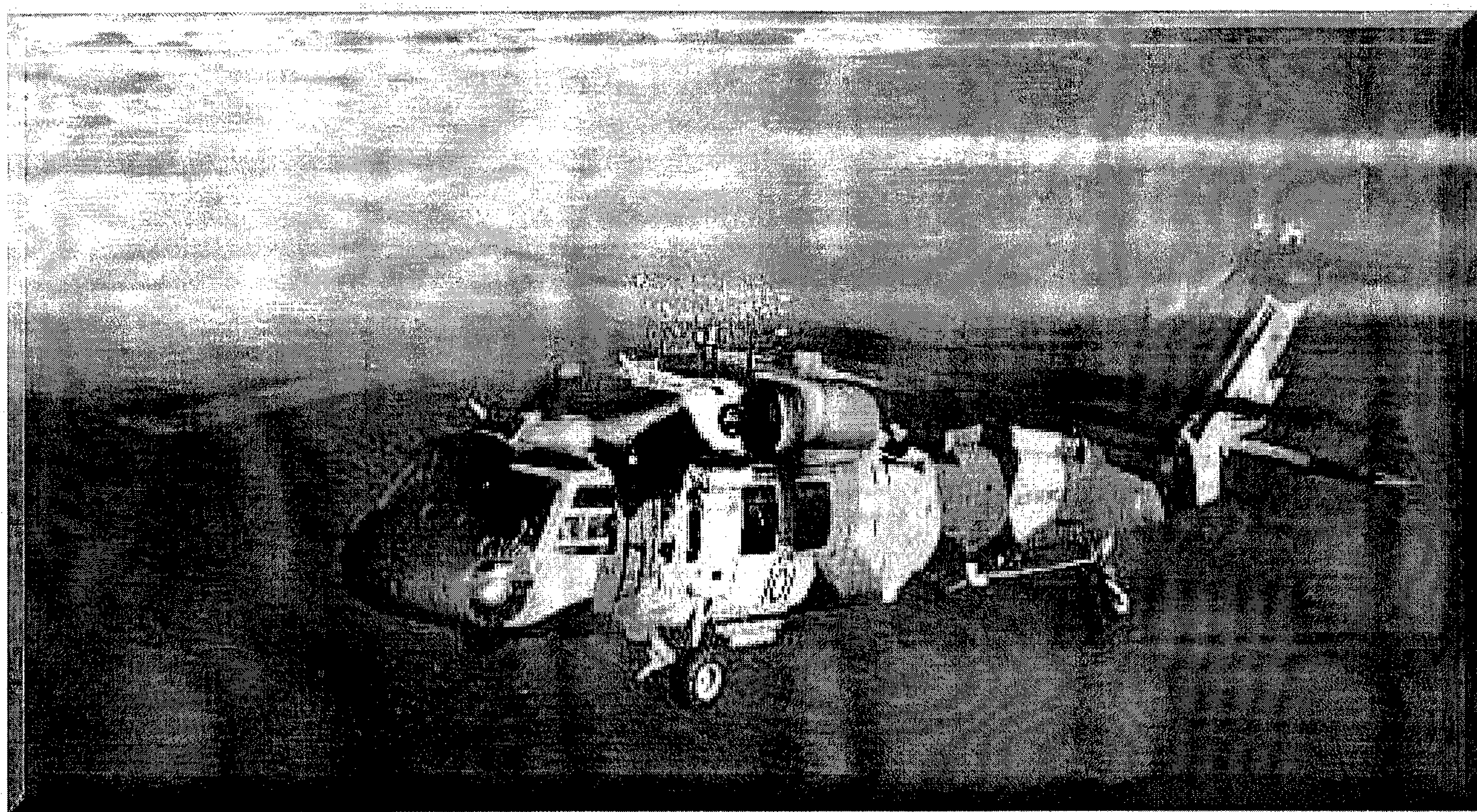


SUMMARY

QUESTIONS?



Summary

- TR Output Shaft failures in Black Hawks
  - Failures at low number of flight hours (<< retirement life)
- Static FEM analysis
  - Showed stresses insufficient to cause such early failures
- Crack growth analysis
  - Flaws in shaft internal radius significantly reduce life

Thank you for your time

PRESENTATION TO

WORKSHP on DAMAGE TOLERANCE in HELICOPTERS

4 - 5 APRIL 2000

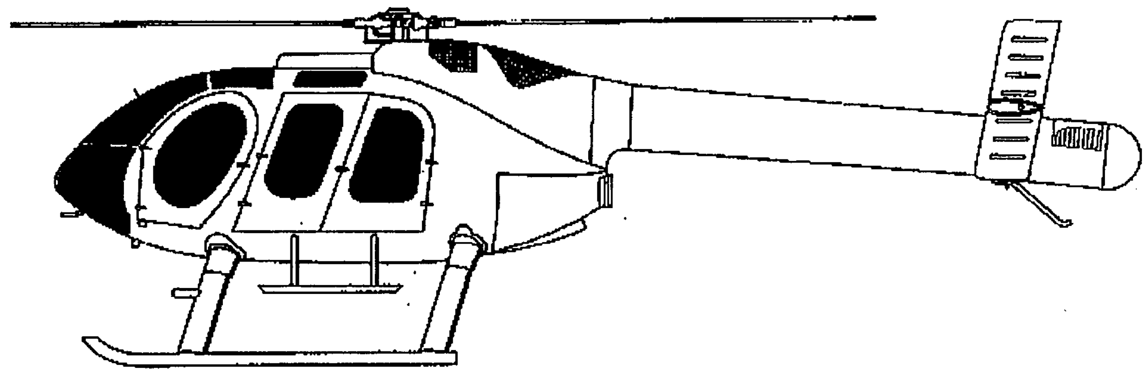
APPLICATION OF DT TO HELICOPTER CERTIFICATION

OVERVIEW

- 600N CERTIFICATION
- FATIGUE & DT METHODOLOGY
- CONTROL SYSTEM COMPONENTS
- INITIAL FATIGUE EVALUATION
- DAMAGE TOLERANCE EVALUATION
- FATIGUE TESTING
- CONCLUSIONS

COMPOSITE
COMPONENTS
- Fan Strap
- Vertical Stab
- Thruster

METAL-COMPOSITE
- Fan Blade
- Tailboom
- Horizontal Stab



MD 600N

CERTIFICATION BASIS

	Cert Basis	Fatigue (Safe-Life)	Damage Tolerance
600N	FAR 27	All	Composites / M/R Blade / Controls

BASIC METHODOLOGY -- LOADS
(Fatigue and Damage Tolerance)

- AC No. 20-95 (Modified for specific aircraft)
- Flight Strain Survey of all specific maneuvers
  (3 Altitudes and 4 Gross weights -- Flown aggressively)
- Cycle Count all data
- Multi-mission spectra
  Worst case Training and Corporate
  External Load Use (Logging)
- Use MS to define Stress Spectrum for Control System Components

BASIC FATIGUE METHODOLOGY -- METALS

- AC No. 20-95
- Reduction Factor taken on S/N curve strength
  - CAM 6 - Appendix A
- Reduction Factor taken on Mean Test Derived S/N curve
  - AFS 120-73-2
  - AFML-TR-69-65, "Reliability Analysis Approach to Fatigue Life Variability of Aircraft Structures"
- Fatigue tests either spectrum or constant amplitude (spectrum preferred)
- Minor's cumulative damage

DAMAGE TOLERANCE CERTIFICATION METHODS FOR METALS

- NASA/FLAGRO* program and database used for all calculations
- Crack growth substantiated inspection intervals for M/R Blade
- Crack growth analysis used on control system components for initial certification

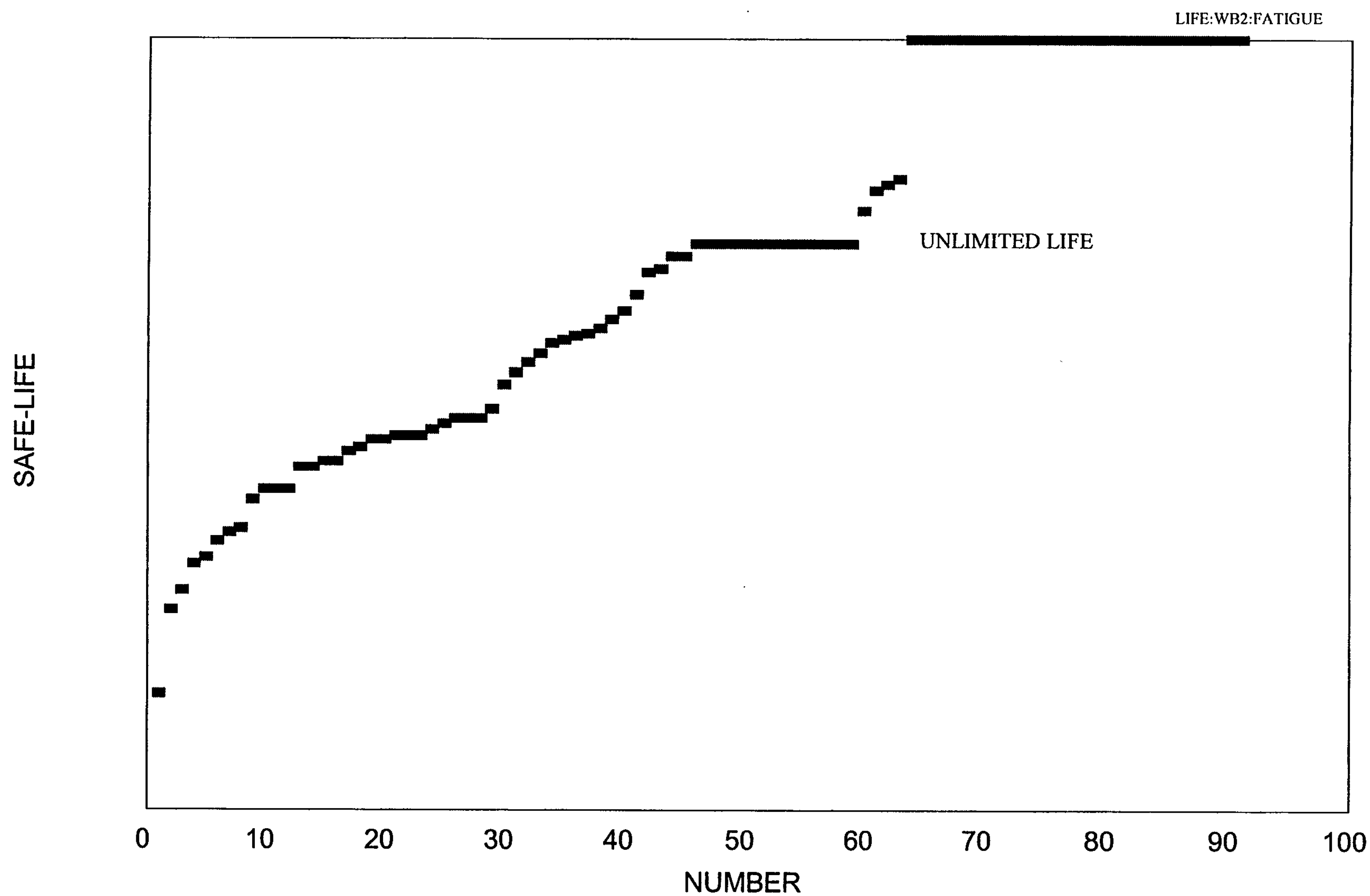
* NASA Lyndon B. Johnson Space Center Houston, Texas

INITIAL FATIGUE EVALUATION

- Ground Rules Were To Use Existing Components
- Very Few Fatigue Tests Conducted On Existing Components
- Conservative Factors Required for Analysis Only Certification

INCREASED CONTROL SYSTEM LOADS

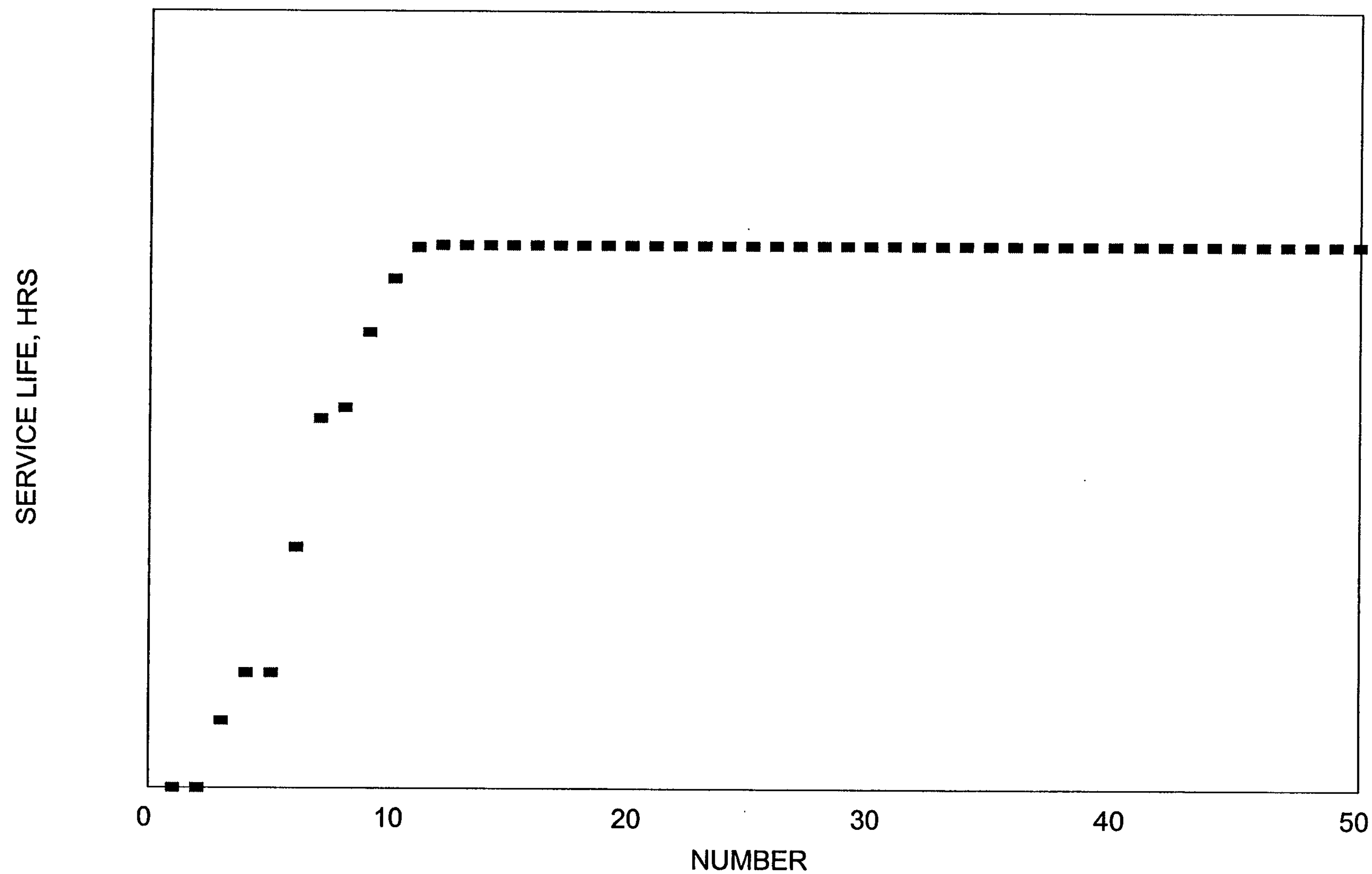
- Six Main Rotor Blades Instead of Five
- Revised Swashplate & Pitch Housing Geometry
- Increased Weight
- Increased Power

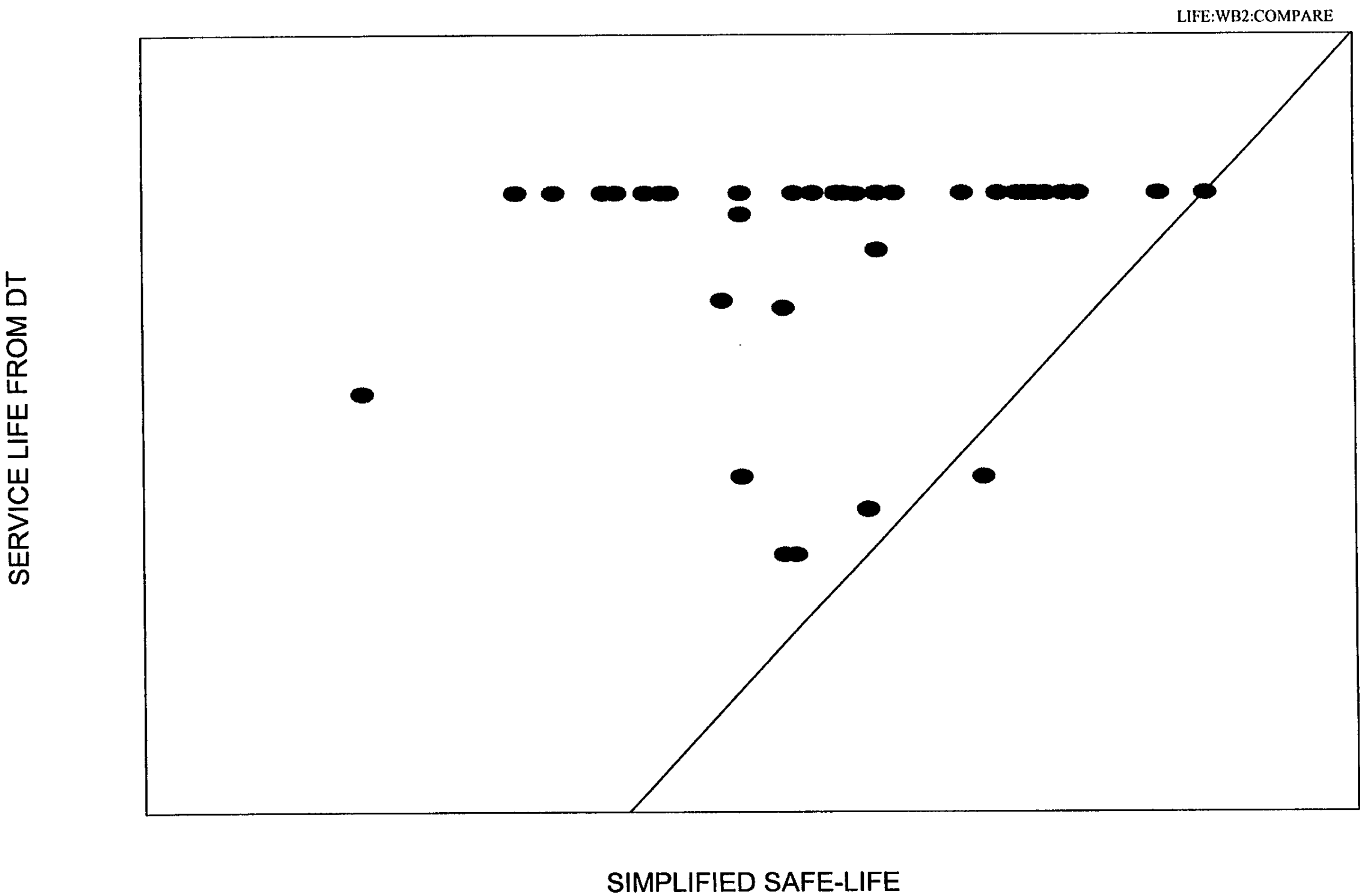

LIFE:WB2:FATIGUE
UNLIMITED LIFE
SAFE-LIFE
0
10
20
30
40
50
60
70
80
90
100
NUMBER

DAMAGE TOLERANT EVALUATION

- Select Equivalent Initial Flaw and Size
  (a = .02 to .05 inch Deep Corner or Surface Flaw)
- Use Same Fatigue Spectrum
- Account for Load Transfer with Bearing Stress
- Apply Factor of Four (4.0) to Calculated Life

LIFE:WB2:DT




LIFE:WB2:COMPARE
SERVICE LIFE FROM DT
SIMPLIFIED SAFE-LIFE

DAMAGE TOLERANT RESULTS

- Reduced Number of Life Limited Components from 43 to 9
- Service Life Ratio (DT / SL) ranged from 700 to 0.67
- Identified 8 Components for Redesign
- Reduced the Number of Fatigue Tests to 6

FATIGUE TEST RESULTS

- Fatigue Testing Showed 7 of the 9 Components to have Unlimited Life Using Safe-Life Methodology
- Fatigue Testing of the Lowest Life Component Confirmed Low Life (Service Life Recalculated Using Safe-Life Methods)
- Other Low Life Component was not Tested - Redesigned

CONCLUSIONS

- Damage Tolerance Successfully Established Service Lives on 600N
- For "Analysis Only", DT Lives are More Accurate than Safe-Life Lives
- DT Lives are Conservative Compared to Final Safe-Lives in Majority of Cases

Galaxy Scientific
Innovative Technical Solutions

Research Issues in the Application of Damage Tolerance Concepts to Rotorcraft

Presented by

Mel F. Kanninen
Galaxy Scientific Corp.

Based on Work in Conjunction with

Charles Harrison, FAA Rotorcraft Directorate
Dy Le, FAA Technical Center
Sharon Miles, FAA Rotorcraft Directorate

Presented at

Workshop on Damage Tolerance of Helicopters
Cranfield University, UK
4-5 April 2000

Current and Anticipated FAA Regulations for Normal and Transport Category Rotorcraft

Acceptable fatigue tolerance evaluation procedures (alone or in combination) currently in CFR Parts 27.571 and 29.571:

- **Damage Tolerance**
- **Flaw Tolerant Safe-Life**
- **Fail-Safe**
- **Safe-Life**

Based on TOGAA recommendations, goal is to allow only damage tolerance and safe life determinations starting in year 2003

Basic Elements of the

Damage Tolerance Methodology for Addressing Crack Growth Failures

- Reliable and precise NDI Procedures that are applicable to all PSE's and all possible crack sizes, shapes and orientations

- Reasonable estimations of the expected service Loading Spectra, and a reliable truncation (simplification) procedure

- Material Fatigue Crack Growth Characterization and other material properties associated with the anticipated modes of cracking

- Robust (efficient, flexible & accurate) Fracture Mechanics Analysis procedures for crack growth and residual strength determinations

Technical Challenges to General Use of DT for New and Existing Rotorcraft

- Complex fuselage and drive train geometries makes the use of conventional finite element models very cumbersome and/or inaccurate

- Rapid accumulation of load cycles requires consideration of very small crack sizes that can be well outside the range of validity of LEFM

- Surface treatments that retard fatigue by creating compressive residual stress fields (e.g., shot peening) require inelastic fracture mechanics

- Mission load profiles that reflect the true severity of actual operating conditions (i.e. combined HCG and LCF load spectra)

- Blunt initial damage (e.g., tool marks, corrosion pits, mechanical impact) for which safe life is nonconservative but LEFM is overconservative

- Composite materials require consideration of failure modes and NDI techniques beyond those of LEFM developed for metallics

Summary and Conclusions

1. Fracture mechanics based damage tolerance methodology currently provides the only practical and scientifically sound approach to establishing in-service inspection intervals for rotorcraft.

2. A new rule embodying damage tolerance as a complement to safe life, with the goal of eliminating flaw tolerance safe life, will be implemented by the FAA by the year 2003.

3. Because of the more demanding conditions facing fracture mechanics analyses in rotorcraft, there is a need to generalize the currently available LEFM-based DT methodologies

4. In collaboration with RITA through the NRTC, with technology transfer from transport aircraft and engines, the FAA is performing research to enable DT to be more widely used for rotorcraft

Additional Research Issues that if Addressed would Enhance the Use of DT for Rotorcraft Applications

- Incubation/formation of small cracks
- crack growth in mixed mode conditions
- crack initiation from initially blunt damage
- crack growth in residual stress fields
- combined corrosion and crack growth
- probabilistic methods to treat uncertainties
- EIFS determinations to link safe life and DT

SESSION C

USAGE, HEALTH MONITORING & INSPECTION

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DERA

Helicopter Fatigue Life Monitoring in the 21st Century

Workshop on Helicopter Damage Tolerance,
4-5 April 2000, Cranfield, UK

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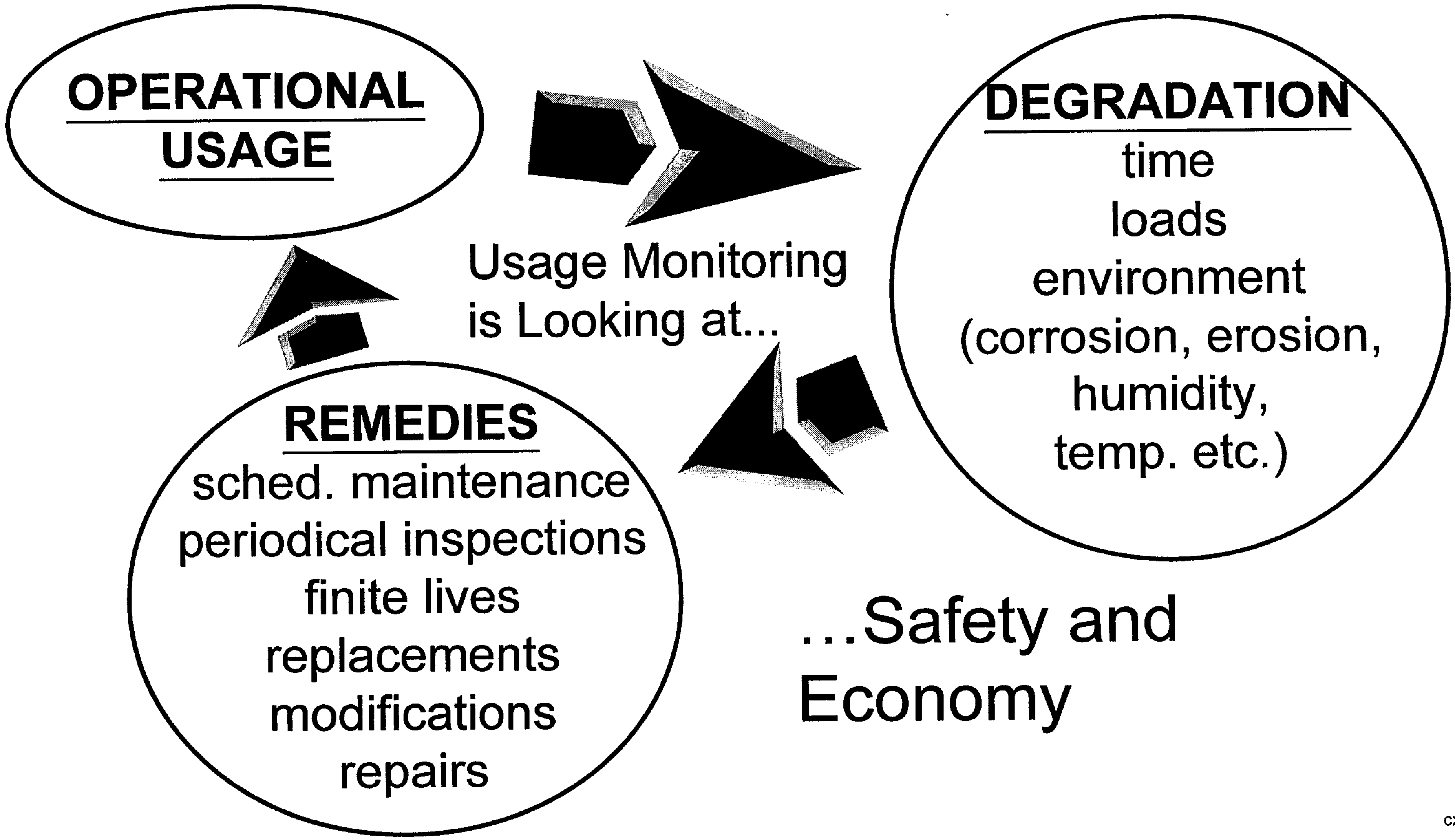
Life Cycle


OPERATIONAL USAGE
DEGRADATION
time
loads
environment
(corrosion, erosion,
humidity,
temp. etc.)
REMEDIES
sched. maintenance
periodical inspections
finite lives
replacements
modifications
repairs
Usage Monitoring
is Looking at...
...Safety and
Economy

Fatigue Life

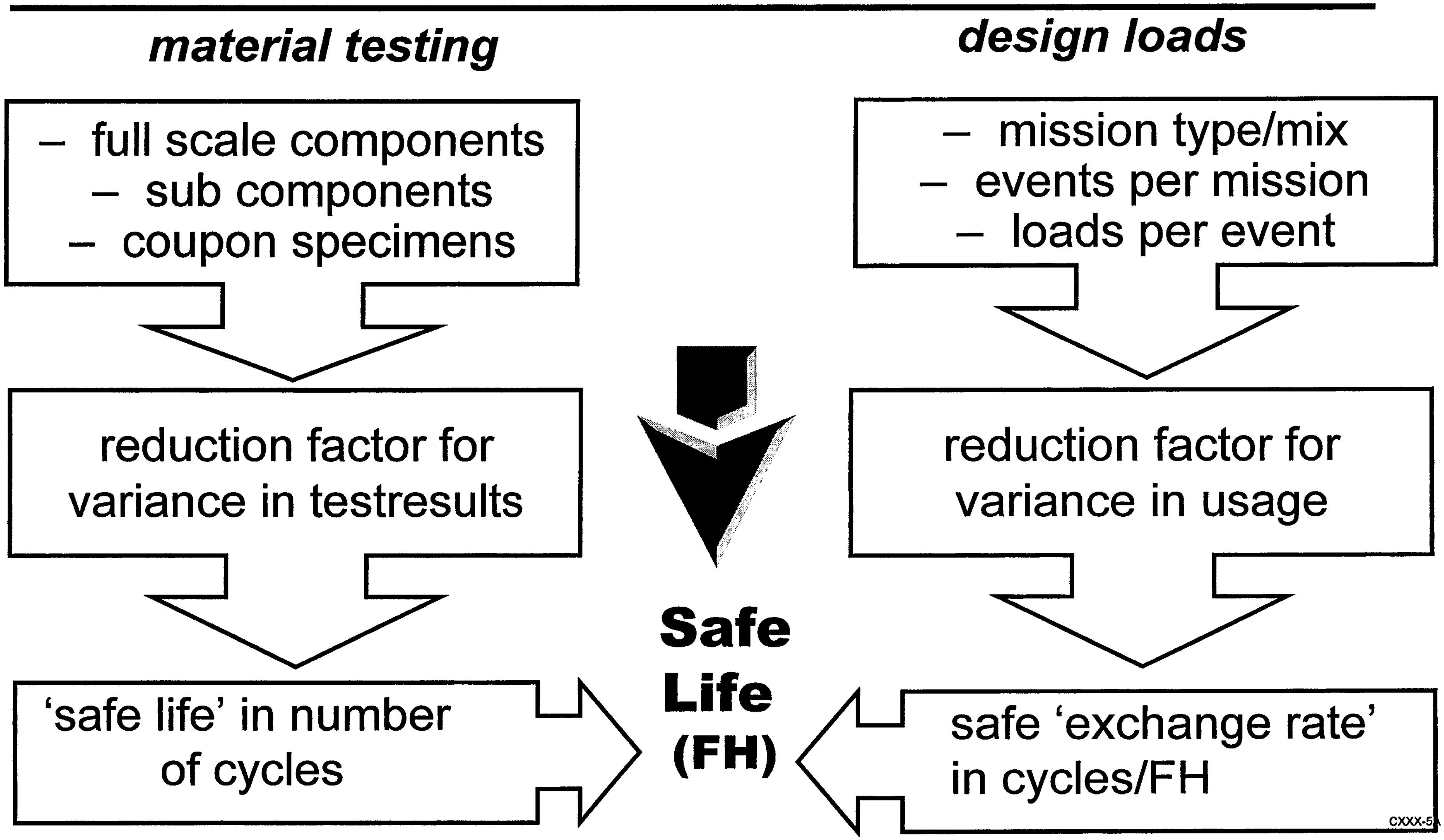
- **On-Condition Items**
  - ‘Damage Tolerant’ design
  - periodical inspections and repairs (if necessary)
  - non-finite fatigue life
  - finite economical life
  - inspection intervals based on operational usage estimate (done by manufacturer)
- **Lifed Items**
  - finite ‘Safe Life’
  - no in-service inspections
  - based on statistics (99%/95%-rule)
  - finite life based on operational usage estimate (done by manufacturer)

Fatigue Life

Typical for helicopter components

- **<u>Lifed Items</u>**
  - * finite 'Safe Life'
  - * no in-service inspections
  - * based on statistics (99%/95%-rule)
  - * finite life based on operational usage estimate (done by manufacturer)

Fatigue Life calculation



Fatigue Life calculation

material testing

- full scale components
  - sub components
  - coupon specimens

reduction factor for variance in testresults

'safe life' in number of cycles

design loads

- mission type/mix
- events per mission
- loads per event

reduction factor for variance in usage

safe 'exchange rate' in cycles/FH

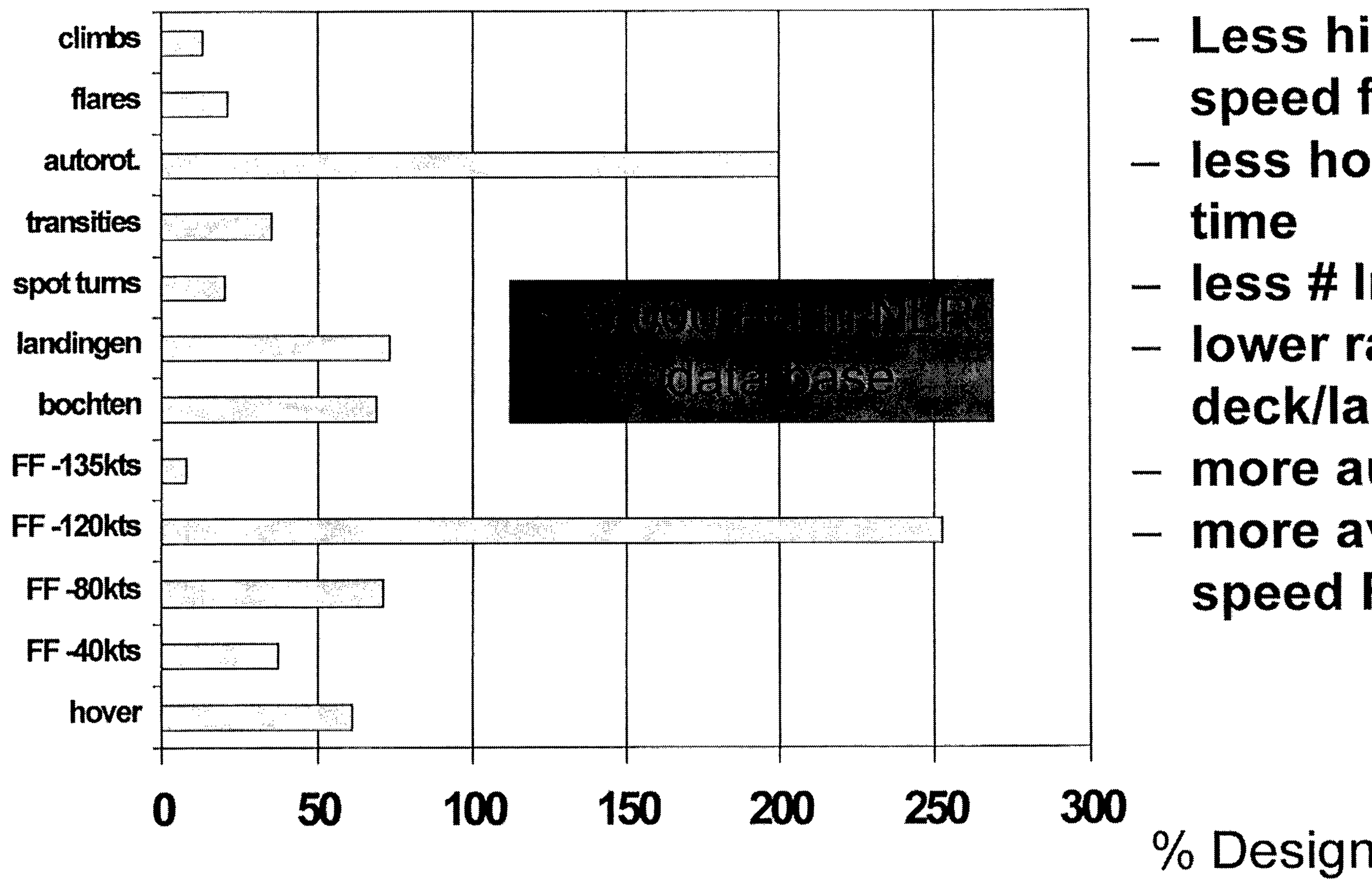
Safe Life (FH)

CXXX-6

RNLN Fatigue Life Monitoring...

- **Lynx Usage Monitoring**
- **Lynx Gem Cyclic Life Control**
  * Rolls Royce Gem engine
- **Lynx AIDA**
  * multichannel retrofit system

Lynx Usage Monitoring



- Less high-speed flight
- less hover time
- less # lndngs
- lower ratio deck/land
- more autorot.
- more avg speed FF

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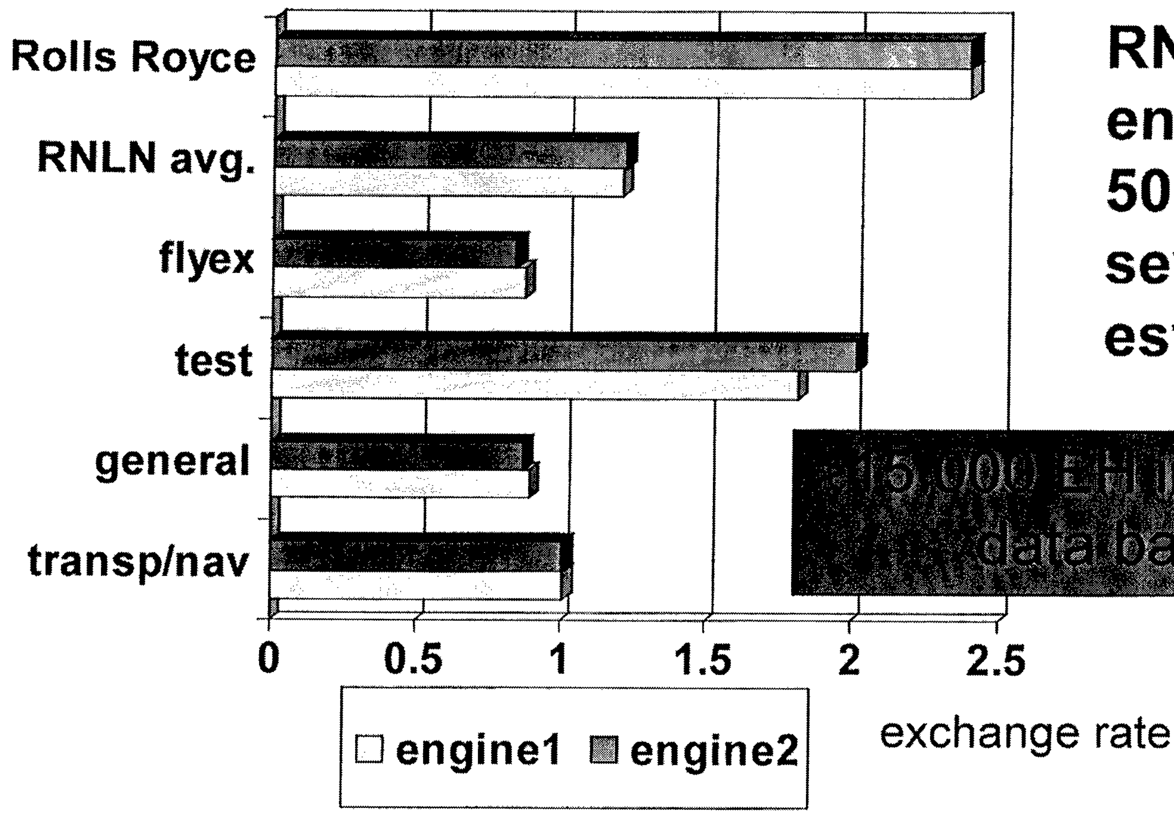
Gem Cyclic Life Control



RNLN Lynx engine usage is 50% less severe than estimated

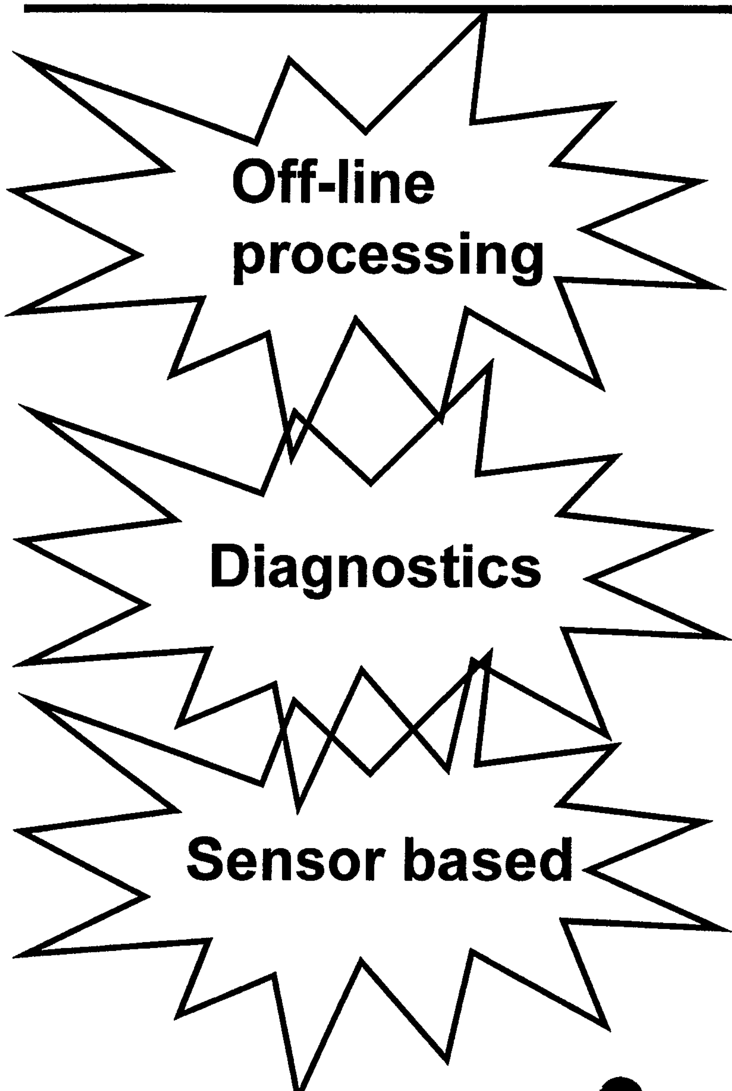
15,000 EH in NLR data base

Lynx AIDA

- **Nr monitoring**
  - audio/visual warnings
  - signal: 1 x RPM
- **Gem engine Cyclic Life Control**
  - Individual Aircraft Tracking
  - Rolls-Royce Service Bulletin
  - signals: 4 x RPM ($Nh_{1,2}$ + $Nf_{1,2}$)
- **Usage Monitoring**
  - CLUMS data-base
  - lifting frame, sponson
  - signals:
    - 2 discretes (WoW, Radalt)
    - 15 analogue (alt, speed, φ,Nr, Nh, Nf, Tq, Sponson strains, spares)

Nationaal Lucht- en Ruimtevaartlaboratorium
National Aerospace Laboratory NLR

Fatigue Life Monitoring - Now -



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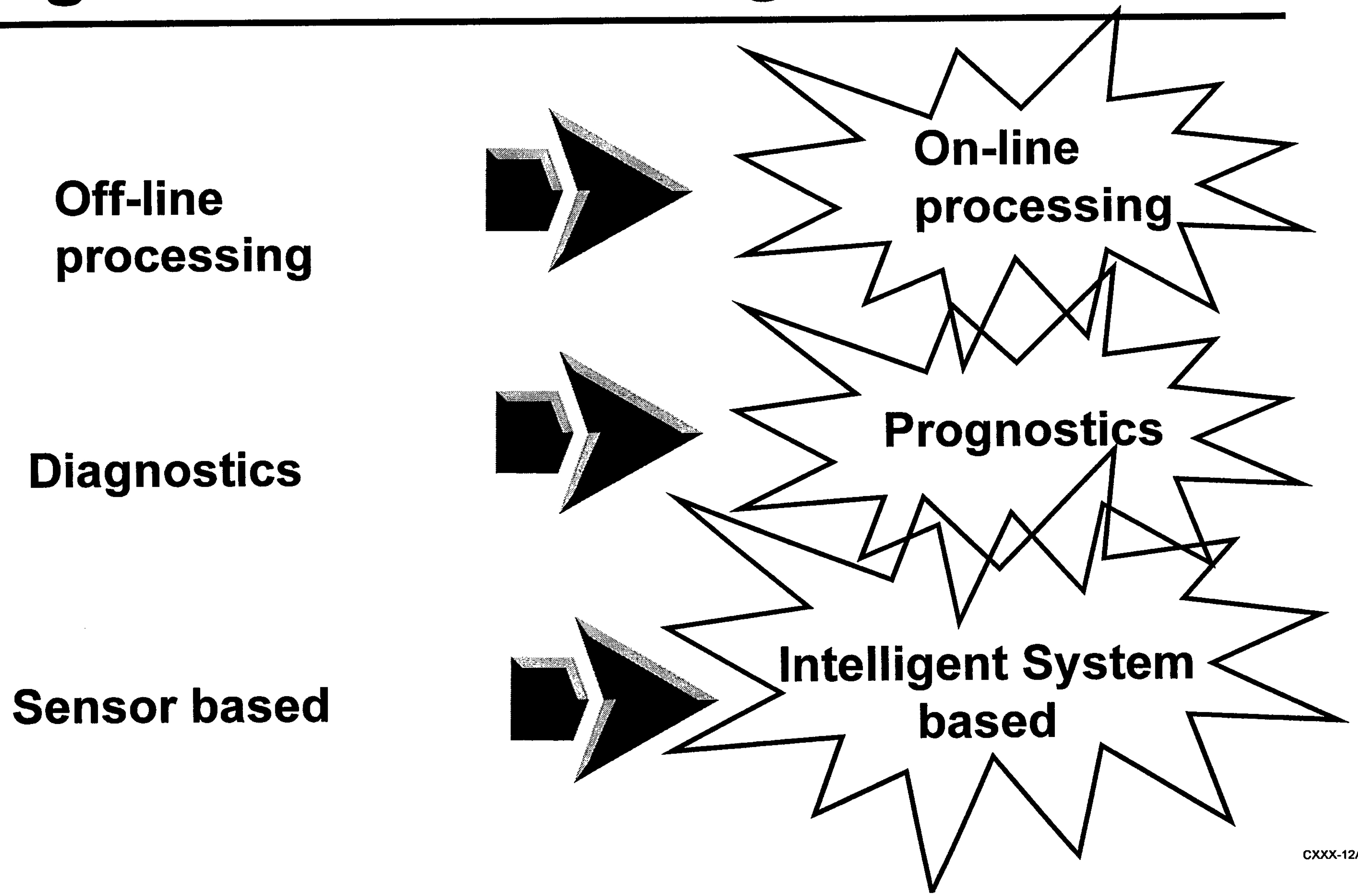
Fatigue Life Monitoring -Transition-



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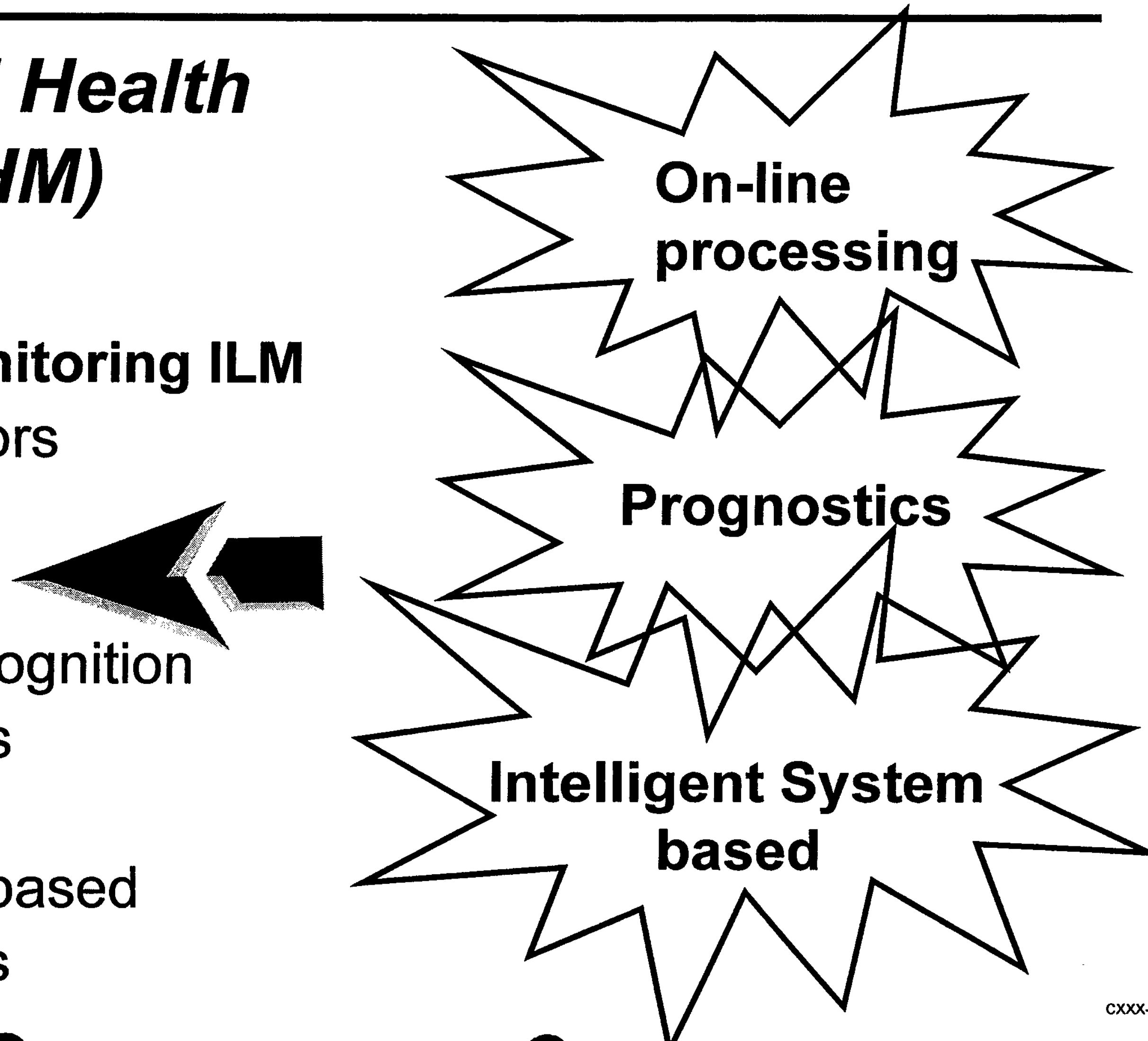
Fatigue Life Monitoring - Future -

Prognostics and Health Management (PHM) systems

- **Intelligent Load Monitoring ILM**
  - virtual strain sensors
  - neural networks
- **Data Mining**
  - autom. regime recognition
  - engine prognostics
- **Reasoners**
  - model, rule, case based
  - genetic algorithms



ILM example - Now -

Off-line Assessment of:

- **Landing Type**
  - deck/land
  - harpoon Y/N
- **Landing Regime**
  - running
  - MPOG
  - T.O. and Landing Weight statistics
- **Limit Load Exceed.**
- **Vibration Level**
- **Damage accumulation**

ILM example - Future -

PHM will enable:

- Elimination of strain sensors
- Strains derived from configuration, flight and environmental data
- On-line calculation of accumulated damage
- Prognostics for fleet and mission planning

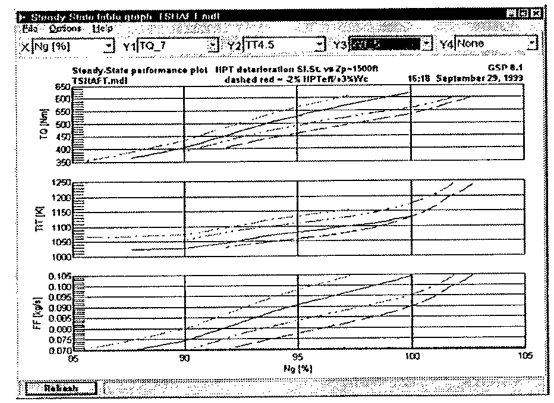
Engine prognostics

Problem: HPT Deterioration

Sea-Level efficiency - 2% with massflow +3%

Deterioration effect is hidden due to flight condition changes (causing variation in performance data)

GSP Turboshaft model



Data Mining finds Known AND Unknown relations

Prediction of Deterioration Effects

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Finally

- Better understand 20th century technology as a basis for future developments
- 21st century goal: Transition of emerging fixed-wing PHM technologies to the rotary-wing environment
- R&D Structural Integrity $$ will shift from the traditional

Mechanical and Materials Engineering

to

IT, Quality Assurance and Certification areas


Non Destructive
Design, Manufacturing
Requirements
Flight Stephen

"Human history becomes more and more a race between education and catastrophe"

H.G. Wells

The Outline of History

"Despite the acclaimed solid education of engineers, it is my experience in teaching fracture mechanics...to practicing engineers, that most have only a vague idea of such subjects as plastic deformation and design; to many Mohr's circle is an enigma; at most one in a class knows the stress concentration factor of a circular hole; fewer even remember yield criteria and their significance"

David Broek

The Practical Use of Fracture Mechanics

Scope

- Non-Destructive Testing
  - Design
  - Manufacturing
  - In Service
- Technology capabilities
- New Technology
- Reliability of Inspection
- Case Studies
  - Sea King
  - Puma

NDT Methods

- Visual Aids
  - Magnifying Glass
  - Endoscopy (Rigid, flexible, CCD imaging)
- Dye Penetrants
  - Colour Contrast & Fluorescent
- Magnetic Particle
  - Colour Contrast & Fluorescent
  - Magnetic Field, Current AC/DC
- Eddy Current
  - Single/Dual/Multiple Frequency
  - Rotary (bolt hole)
  - Transient
  - AC/PD
- Ultrasonics
  - Compression, Shear, Lamb wave
  - Spectroscopy
  - Air Coupled
  - Low frequency techniques (Resonance, tap testing)
- Acoustic Emmission
- Radiography
  - X - γ Ray
  - Filmless (Digital imaging)
    - Phosphor screen
    - Amorphous Selenium
- Optical Techniques
  - D-Sight
  - Edge of Light

NDT Requirements

- Design
  - Determine SSIs for design requirements (Fail Safe, Safe Life or Damage Tolerance)
  - Stress analysis (operational, overload,….)
  - Life component based on Fracture Mechanics, expected operational loads & environmental factors
  - Determine appropriate NDT technique to find defects or faults in structure

NDT Requirements

- Manufacturing
  - Process/Quality control to filter components exceeding acceptable defect criteria
  - Inspection environment enables high quality processes.
    - Automated: PFD lines, Ultrasonic & Eddy Current C-scanning.
    - Scope to develop & implement new technologies
  - Negative aspect: NDT is carried out pre-assembly (potential for induced defects)

NDT Requirements

- In-Service
  - Operator Needs:
    - Cheap/Cost effective
    - High reliability (no false calls)
    - Quick
    - Low frequency of inspection
  - Design for Inspection
    - Accessibility
    - Material selection & manufacturing processes compatible with in-field inspection capabilities
    - Increase Inspection reliability
    - Reduces time & maintenance cost

New Technology

- Aim of implementing new technology is to improve inspection reliability and efficiency, and reduce costs through automation
  - Pulsed Thermography
  - Laser Shearography
  - Field capable C-scan systems
  - Digital X-ray Techniques
- New analyses can lead to unexpected results
  - Complex image interpretation (automated or manual) can lead to increases in false calls
  - New Technology has to provide significant improvement in capability over current practices
    - Simplicity
    - Data interpretation
    - FAA & USAF development programmes are sustaining impetus on technology developments from the private sector

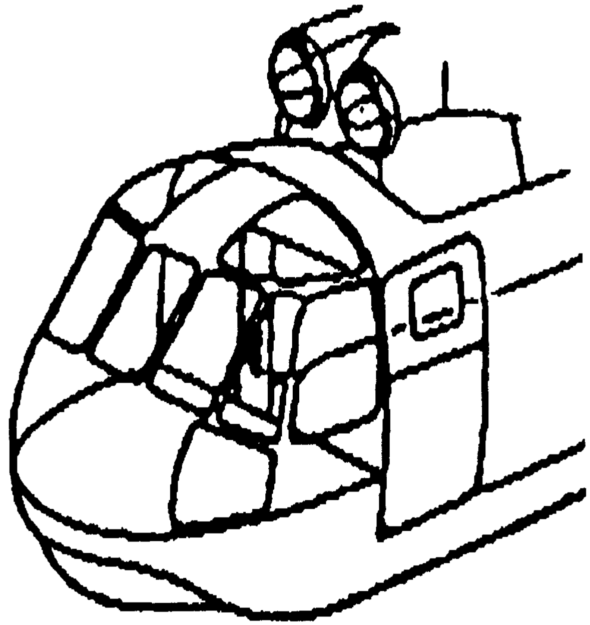
Reliability of Inspection

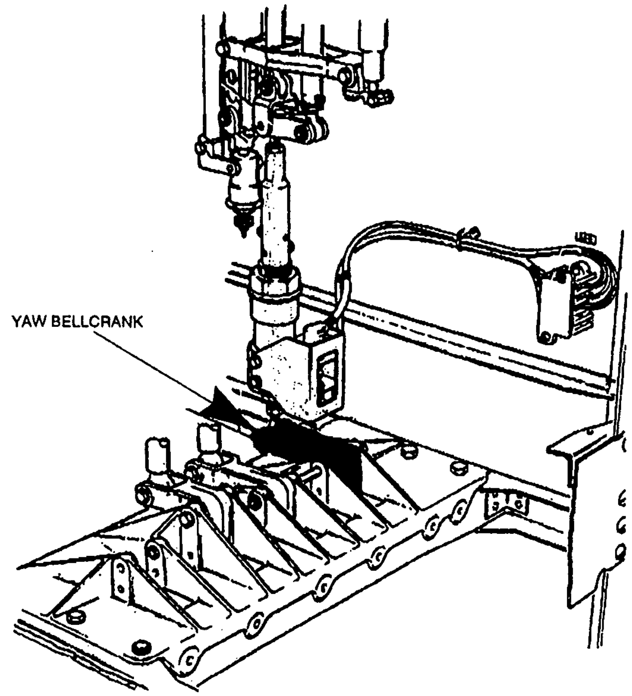
- $R = f$(technique, method, human factor)
- Inspection points must be derived to optimise the selection of technique
- Matching inspections with appropriate level of maintenance
  - Incremental increase in risk!
- Experience derives the most effective technique, and associated limitations
- Ultimately, no system can ever be 100% perfect

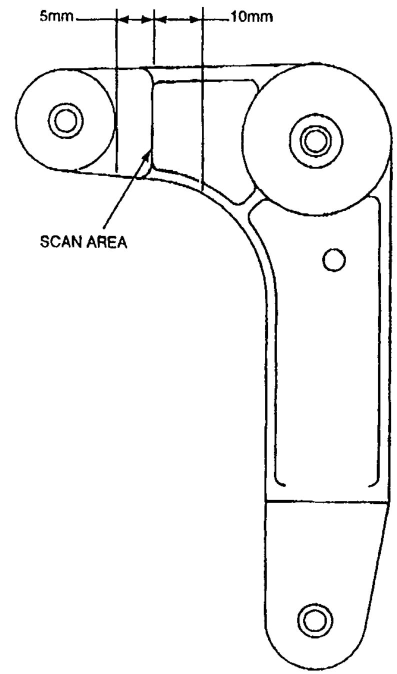
NDT Challenges

- What if the initial flaw size is below the threshold of detection?
  - Durability approach, where no inspections can or will be done.
    - Retire the component after H/2 hrs
    - Expensive
  - Proof Testing
    - Apply every Safe operation period (a_{proof} to a_p)
    - Safe operation period can be short leading to high maintenance penalty
  - Stripping
    - Machining away a surface layer equivalent to a_p
    - Applicable to fillet radii or fastener holes that require oversizing

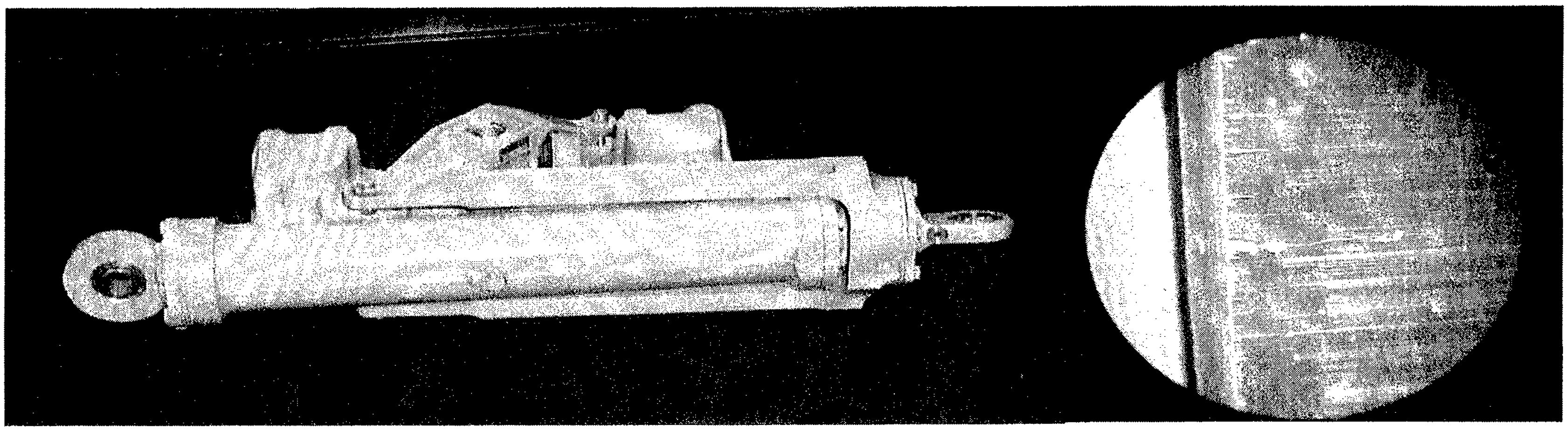
Case Study 1: Seaking




YAW BELLCRANK


5mm
10mm
SCAN AREA

Case Study 2:
Puma MRH Servo



Summary

- NDT is a limiting factor in Damage Tolerant Design
- Pragmatic use of inspection capabilities can facilitate effective fatigue life management
- New technologies must be thoroughly evaluated before implementation either for Quality Control or in-service inspections
- Other inspection/maintenance methods may be required to validate structural integrity of rotorcraft if Damage Tolerance is rigorously applied

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Usage monitoring and materials factors in achievement of damage tolerance

Phil Irving, R A Hudson
Cranfield University

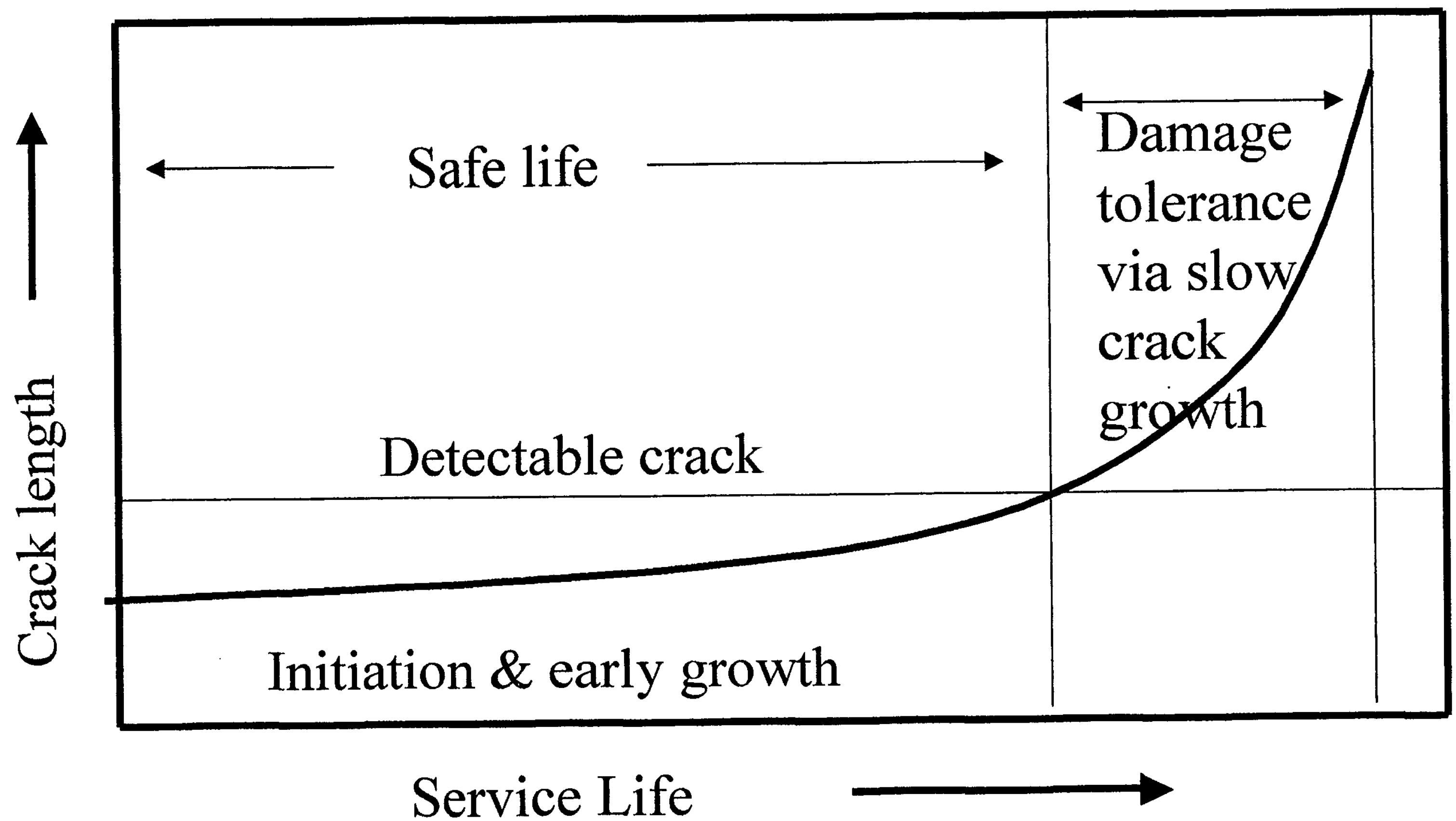
Damage Tolerance discussion workshop

4-5 April 2000

Objectives

- Determine benefits of Usage monitoring in achievement of damage tolerance in helicopter components
- Establish how much difference material choice makes to the damage tolerant situation

Damage Tolerance- schematic



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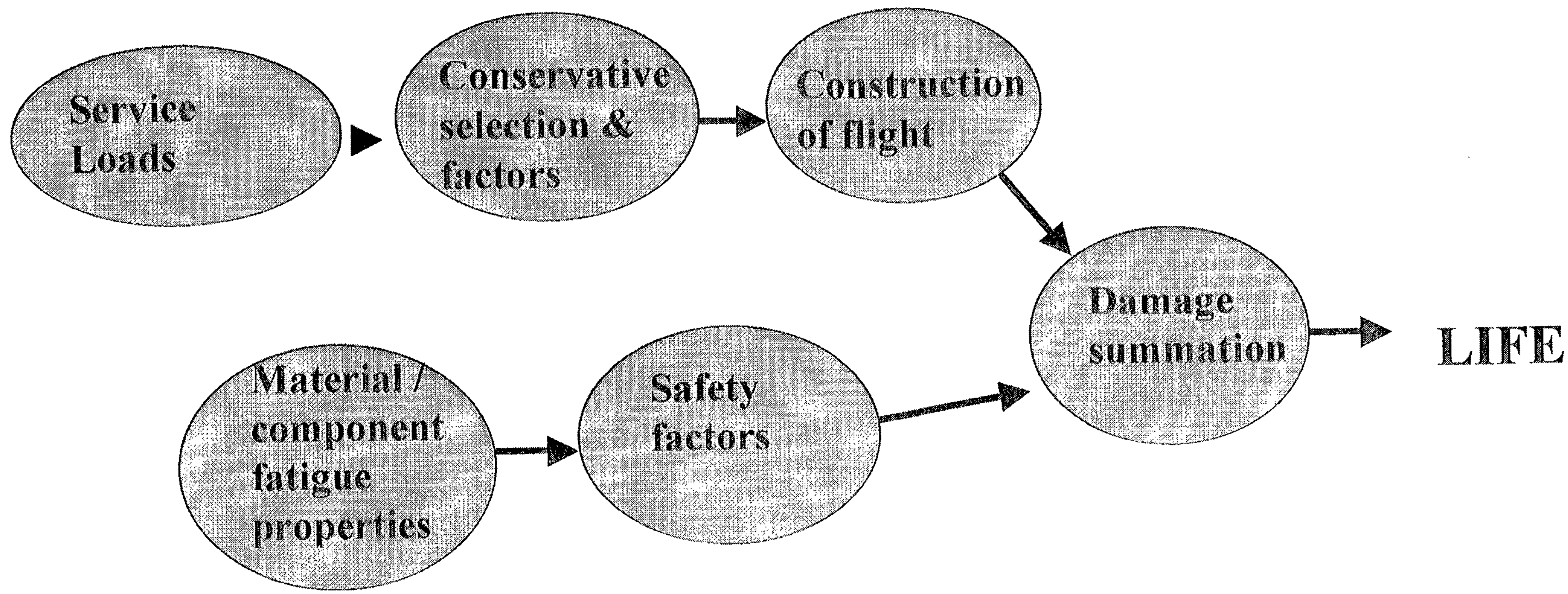
Safe life calculation- schematic



Work- outline

- 10 hours of service load data containing all manoeuvres in design spectrum for Lynx; select dogbone rotor component for study
- Extract load histories for up to 20 examples of each manoeuvre in design spectrum
- Calculate damage for each manoeuvre using component fatigue data supplied by manufacturer
- Develop worst case design spectrum for deterministic life
- Compare worst case fixed lives with distributions of lives developed using Monte Carlo simulations

Work outline - continued

- Repeat for crack growth lives from 1.27 mm defect
- Repeat for alternative materials- steels, aluminium alloys

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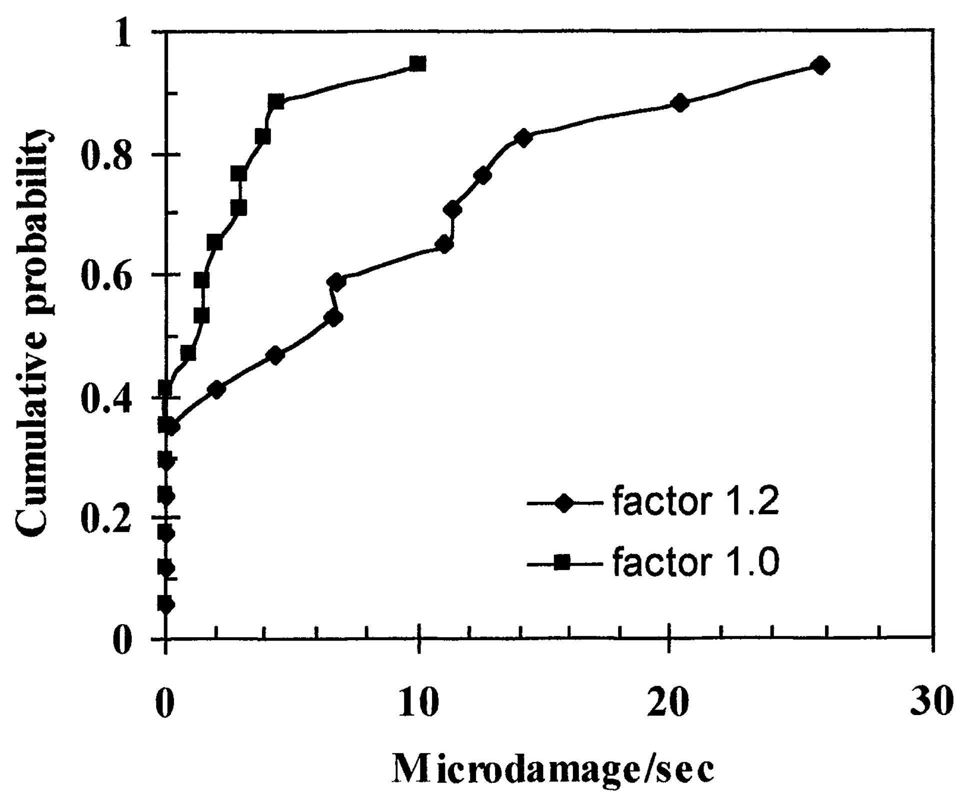
Variability in manoeuvre damage-control reversal



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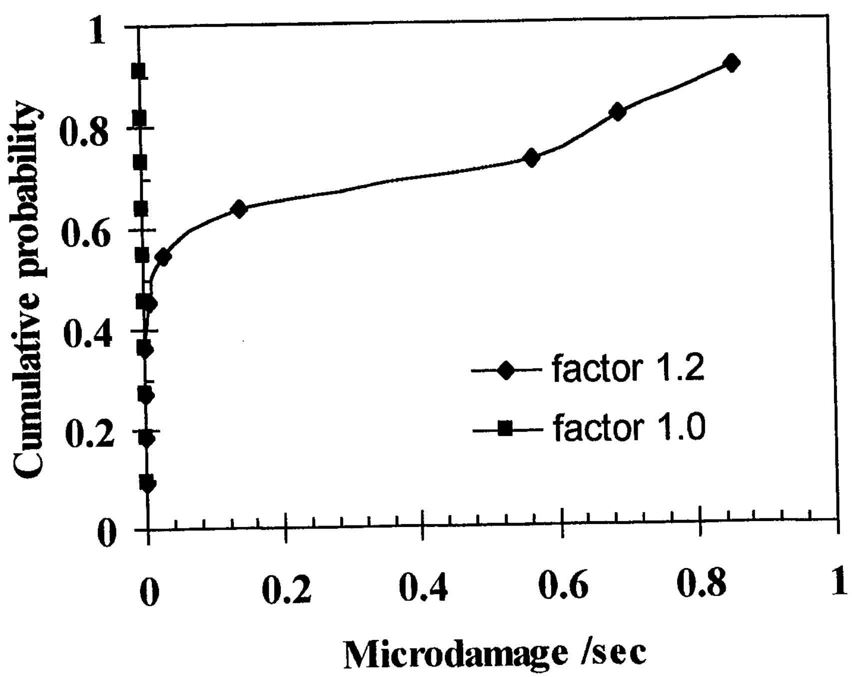
Variability in manoeuvre damage-Hover



Safe life & crack growth comparison- flight hours

Fatigue design approach	Worst case	Simple worst case	50% probability	Simple 50%
Safe life	2,020	273	27,778	2,817
Crack growth	290	73	636	183

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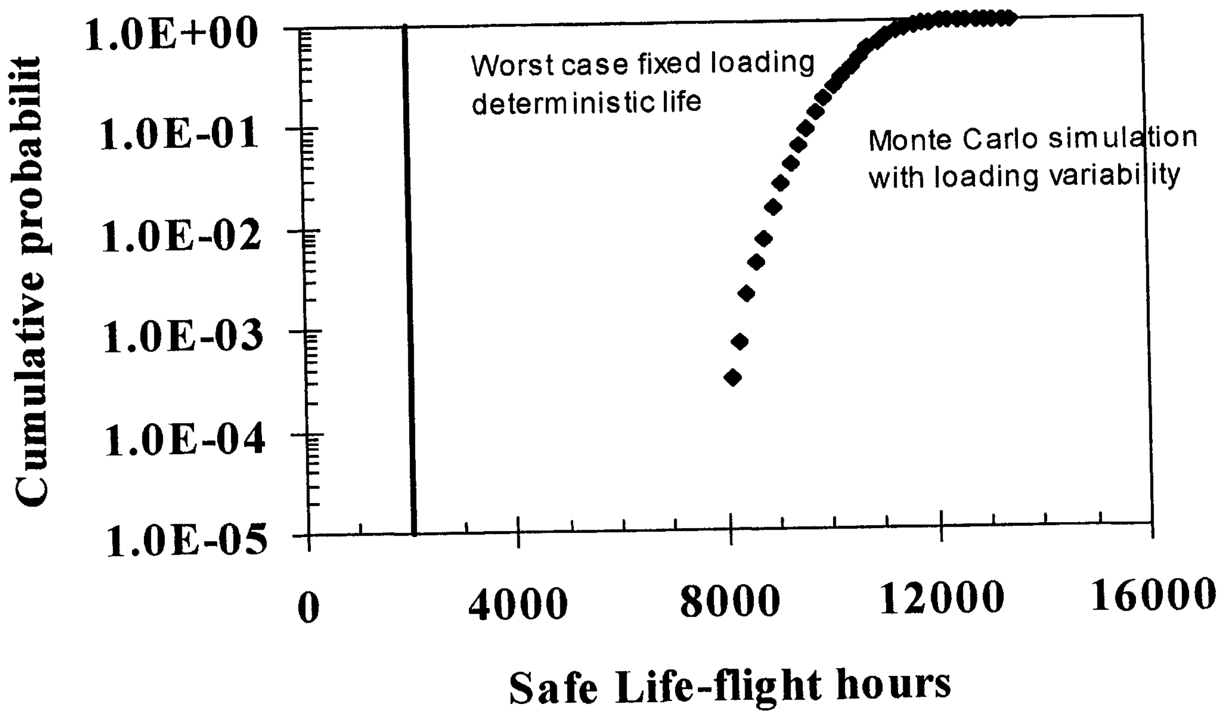
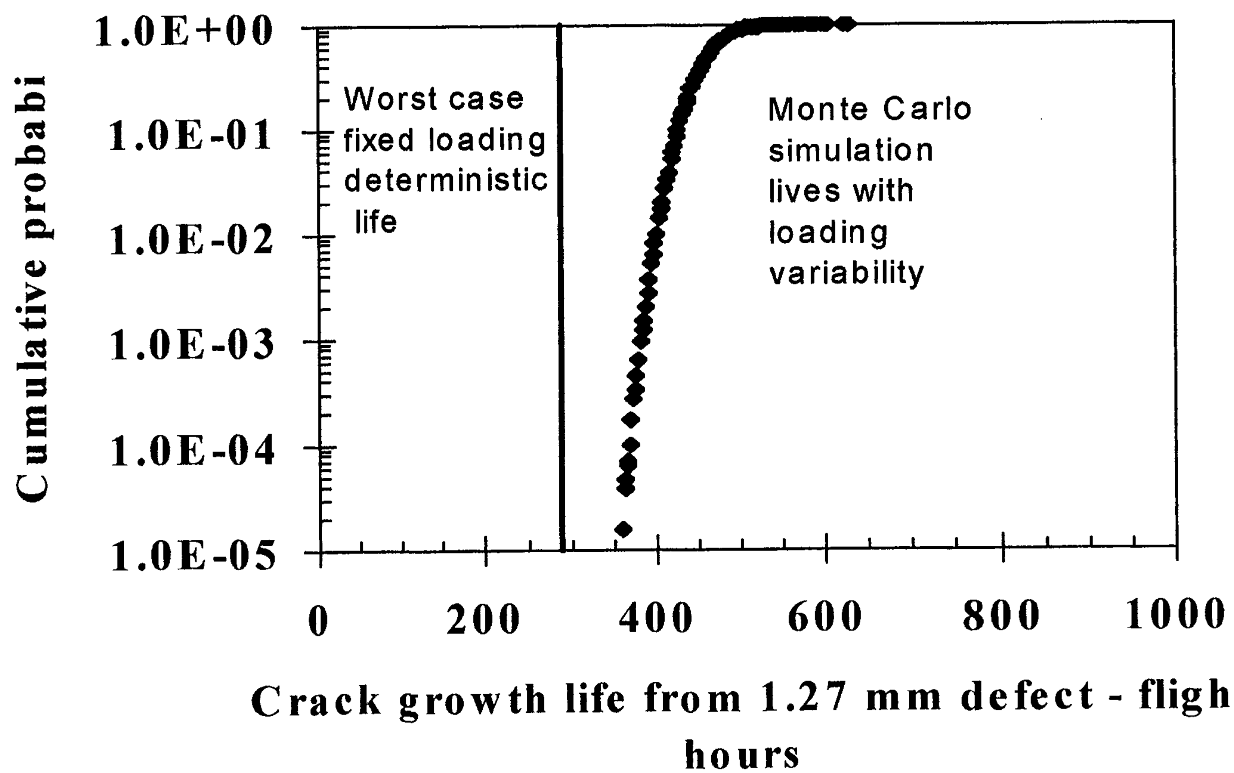
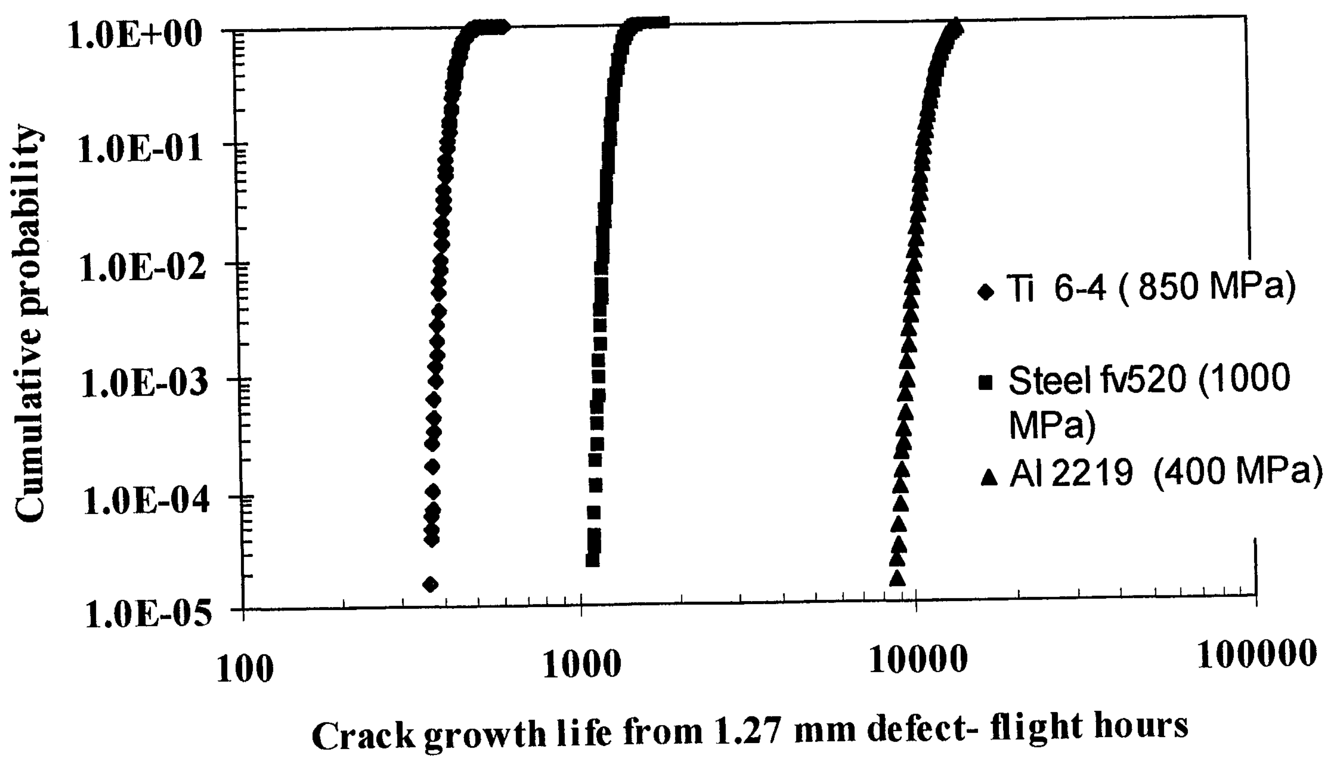
Monte Carlo simulations- safe life-titanium



Monte Carlo simulations- crack growth- titanium



Monte Carlo simulations- crack growth - effect of material- stresses normalised for strength



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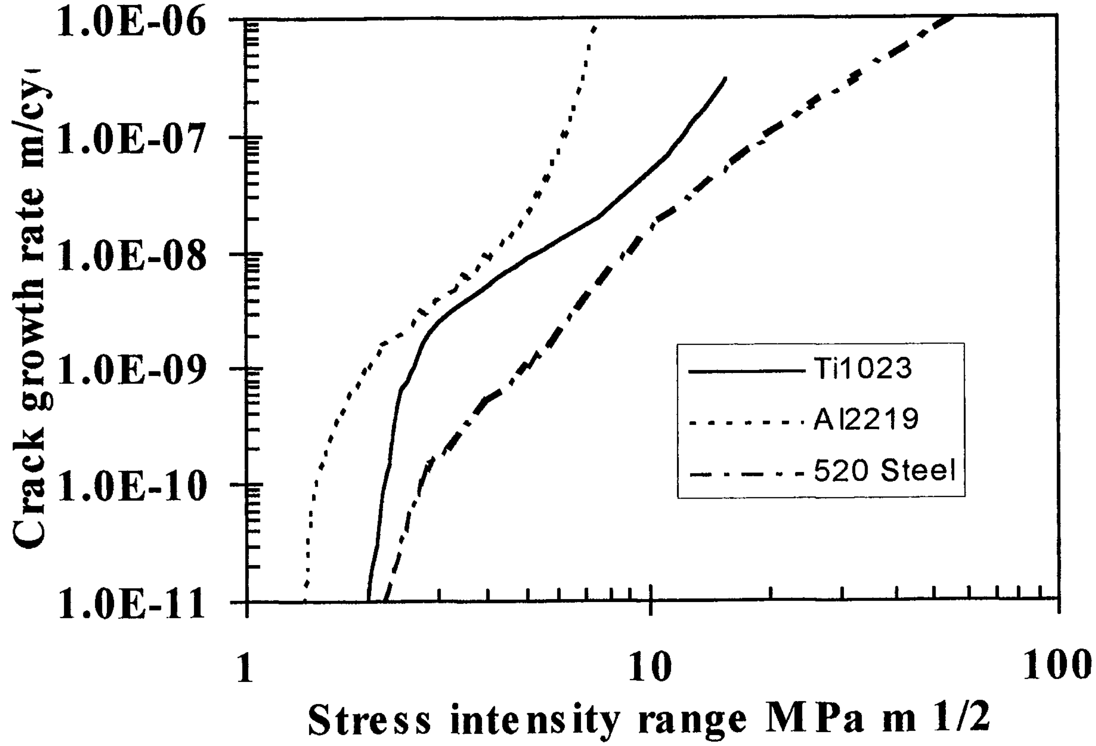
Crack growth data



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Conclusions

- Usage monitoring could improve damage tolerance capability in helicopter rotor components by elimination of uncertainties in service loading;
- Inspections when calculated crack lengths indicate requirement
- Lives to grow cracks from 1.25 mm to 18 mm increased from 290 hours for worst case loading to a most probable life of 500 hours
- Both steel and aluminium alloys show greatly increased crack growth lives whn loads normalised for strength

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Acknowledgements

Thanks are due to many colleagues at Cranfield, CAA and GKN WHL for helpful discussions.

CAA is thanked for continuing support for the CAA chair.

SESSION D

AIRFRAME STATIC COMPONENTS

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DERA

FAILURE PREDICTION OF COMPOSITE FUSELAGE STRUCTURES

Joyanto K. Sen and Jian Li

The Boeing Company

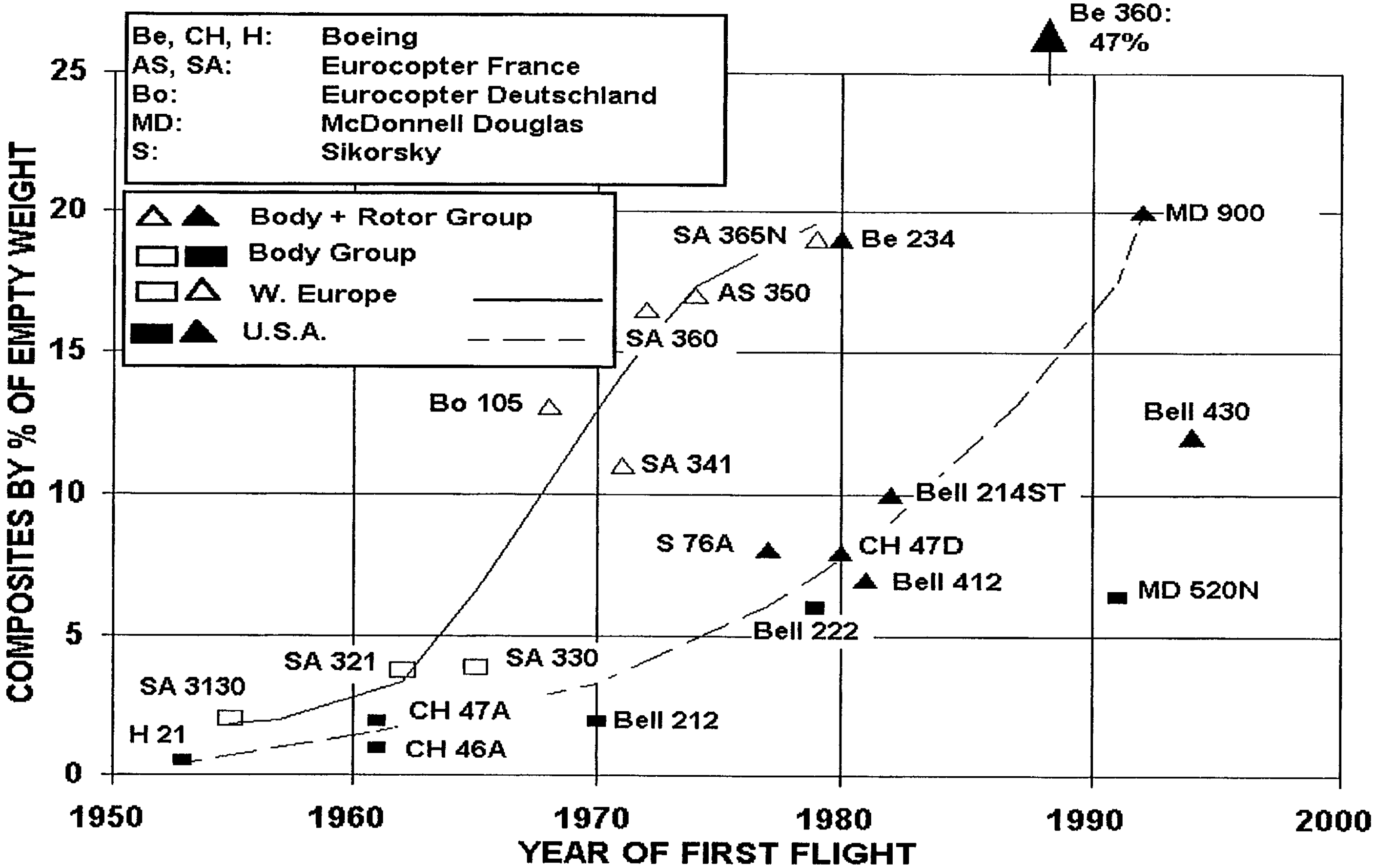
Mesa, Arizona, USA

Workshop on Damage Tolerance in Helicopters

Cranfield University, UK

4-5 April 2000

INCREASING USE OF COMPOSITES IN HELICOPTERS



DESIGNING COMPOSITE STRUCTURES

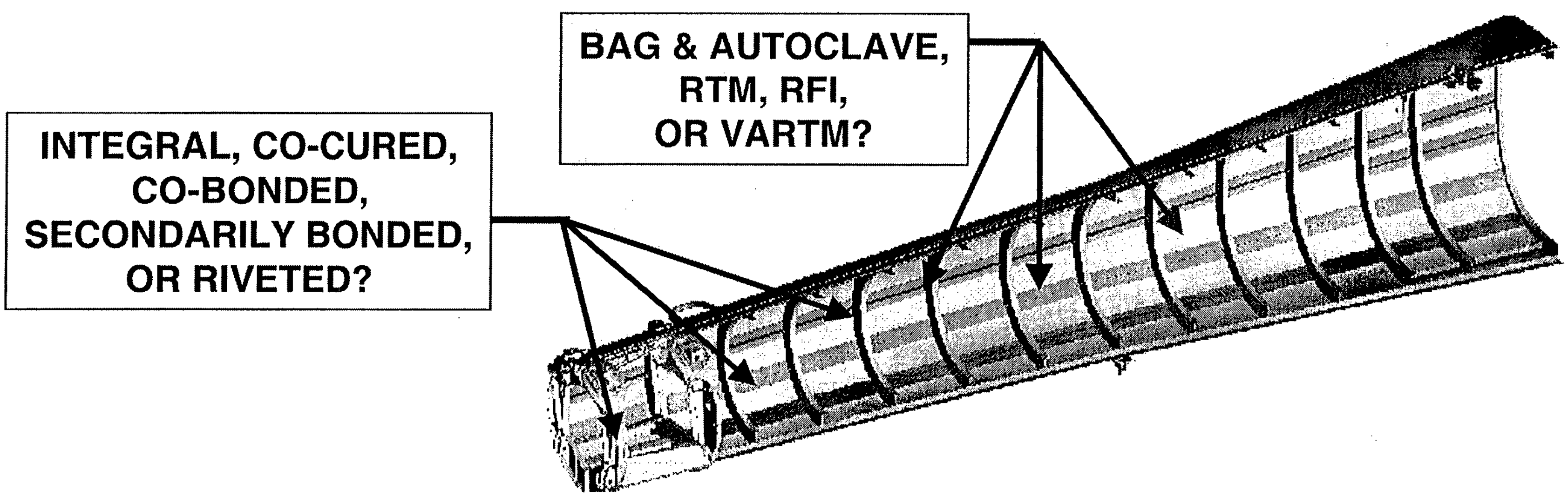
BOEING

- EXHIBITS MULTIPLE FAILURE MODES
  - COMPOSITES ARE MADE OF MULTIPLE MATERIALS
  - MECHANICAL BEHAVIOR CAN BE TAILORED
- DESIGN IS VALIDATED BASED ON
  - MATERIAL CHARACTERIZATION
  - STRUCTURAL ANALYSIS
  - VERIFICATION THROUGH TEST
- LIGHTWEIGHT, AFFORDABLE DESIGNS DEPEND ON SEVERAL DESIGN AND MANUFACTURING ITERATIONS

TYPICAL STRUCTURES REQUIRE INNOVATIONS FOR LIGHTWEIGHT, AFFORDABLE DESIGNS

BOEING

- HAND LAID-UP PREPREG, OR DRY FIBER WITH LIQUID MOLDING?
- UNI-DIRECTIONAL TOWS, OR CLOTH, OR OTHER FABRIC FORMS?
- STITCHED, OR OTHER METHODS OF ENHANCED SURVIVABILITY AND IMPROVED HANDLING CHARACTERISTICS
- AUTOCLAVE-CURED, OR THERMOFORMED, OR SELF-CONTAINED TOOL
- THERMOSET, OR THERMOPLASTIC?



COMPLEXITIES OF COMPOSITE DESIGNS

BOEING

- DESIGN ITERATIONS INFLUENCE MATERIAL BEHAVIOR
  - MATERIAL & CORE FORMS
  - PLY ORIENTATIONS
  - RESIN SYSTEMS
  - STACK-UP SEQUENCE

- MANUFACTURING PROCESS INFLUENCES VALIDITY AND NUMBER OF ITERATIONS OF ANALYSES
  - RESIN POCKETS
  - FIBER VOLUME
  - CO-BONDING & CO-CURING
  - PLY WAVINESS
  - MATERIAL FORMABILITY
  - LAMINATE QUALITY
  - MISPLACED PLY TERMINATION & PARTS LOCATIONS

REQUIREMENTS OF INDUSTRY

BOEING

- REDUCE MATERIAL CHARACTERIZATION CYCLE TIME
  - STATIC
  - FATIGUE
- EASY ITERATION FOR MATERIAL CHOICES AND INSERTION
  - RAPID DEVELOPMENT OF MATERIAL PROPERTIES
- REDUCE DESIGN CYCLE
  - SIMPLIFIED AND RAPID ANALYSES FOR STRUCTURAL INTEGRITY: FAILURE MODES AND LOCATIONS, AND MANUFACTURING DEFECTS

APPROACH TO COMPOSITE ANALYSIS?

BOEING

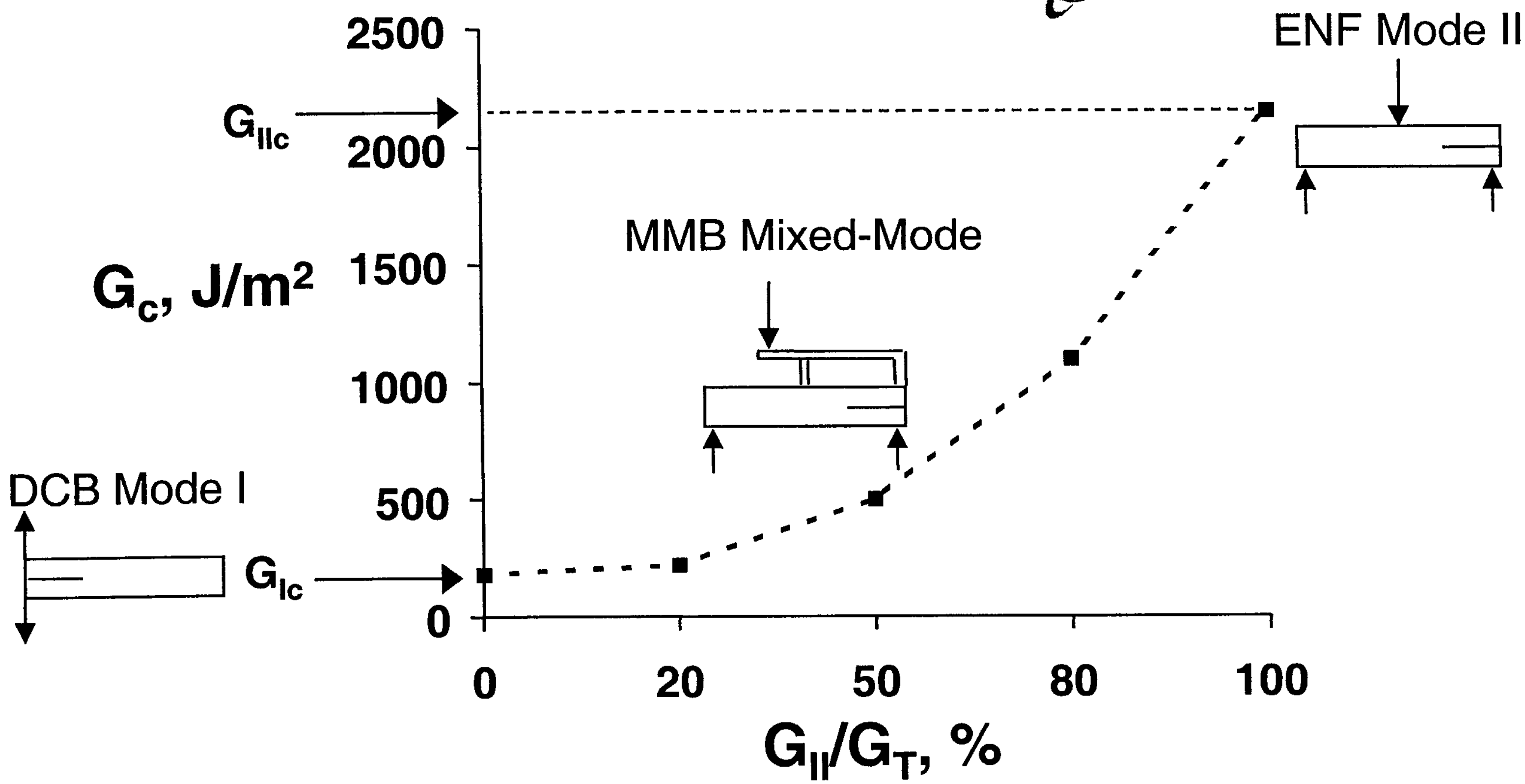
- STRUCTURAL ANALYSIS BASED ON FAILURE MODE(S)
  - REDUCES MATERIAL CHARACTERIZATION CYCLE TIME
  - PREDICTED STRENGTH VALIDATED FOR CRITICAL LOCATION
  - IDENTIFIES INSPECTION TIMES BASED ON FAILURE-CRITICAL LOCATIONS
- PARAMETRIC MODELING
  - PARAMETRIC CHANGES OF THE STRUCTURE ARE RAPIDLY MODELED TO SIMULATE MANUFACTURING DEFECTS FOR STRUCTURAL ANALYSIS
  - REDUCES ANALYSIS CYCLE TIME

FRACTURE MECHANICS APPROACH TO FAILURE PREDICTION METHODOLOGY

BOEING®

- ASSUME CRACK AND DELAMINATION
  - DELAMINATION IS PREDOMINANT FAILURE MODE IN COMPOSITE STRUCTURES
- COUPON TESTS TO DETERMINE MIXED-MODE MATERIAL FRACTURE TOUGHNESS (G_C)
- FINITE ELEMENT ANALYSIS TO CALCULATE TOTAL STRAIN ENERGY RELEASE RATE (G_T) AND MIXED MODE RATIO (G_{II}/G_T)
  - VIRTUAL CRACK CLOSURE TECHNIQUE

MIXED-MODE DELAMINATION CRITERION FOR A GIVEN MATERIAL SYSTEM



$$G_c = M_0 + M_1\,(G_{II}/G_T) + M_2\,(G_{II}/G_T)^2 + M_3\,(G_{II}/G_T)^3$$

FAILURE LOAD PREDICTION

BOEING

- FAILURE OCCURS WHEN

$$[G_T(G_{II}/G_T)]_{FE\ ANALYSIS} = [G_C(G_{II}/G_T)]_{TEST}$$

- FROM G_{II}/G_T DETERMINE MIXED-MODE DELAMINATION FRACTURE TOUGHNESS, G_C,

- FROM FE ANALYSIS CALCULATE CONSTANT OF PROPORTIONALITY, λ, FOR LOAD, P, IN PULL-OFF TESTS

$$G_T = \lambda P^2$$

- THE CRITICAL LOAD, P_C, AT DELAMINATION IS

$$P_C = \sqrt{(G_C/\lambda)}$$

SUMMARY

BOEING

- FRACTURE TOUGHNESS APPROACH IS PROMISING
  - REDUCES ANALYTICAL CYCLE TIME
  - REDUCES DESIGN DEVELOPMENT TIME
  - REDUCES COST
- FRACTURE TOUGHNESS APPROACH IDENTIFIES CRITICAL LOADS AND LOCATIONS, AND RAPIDLY ANALYZES SEVERAL MANUFACTURING DEFECTS
- FURTHER VALIDATED DEVELOPMENT OF FRACTURE TOUGHNESS APPROACH FOR WIDER STRUCTURAL APPLICATION IS NECESSARY

LESSONS LEARNT FROM RECENT ATTEMPTS AT APPLYING DAMAGE TOLERANCE TO HELICOPTER AIRFRAMES

M. L. OVERD
HEAD OF STRUCTURES AND MATERIALS

D. MATTHEW
SENIOR STRESS ENGINEER

GKN WESTLAND HELICOPTERS
UK

GKN Westland Helicopters

PRESENTATION OUTLINE

- **Introduction to the EH101**
- **Origin of damage tolerance requirement**
- **Overview of damage tolerance evaluation**
- **Test Programme**
- **Crack growth analysis & rotorcraft specific problems**
- **Results of the analysis**
- **Reasons for the results and results in perspective**
- **Alternative approaches**
- **Conclusions**

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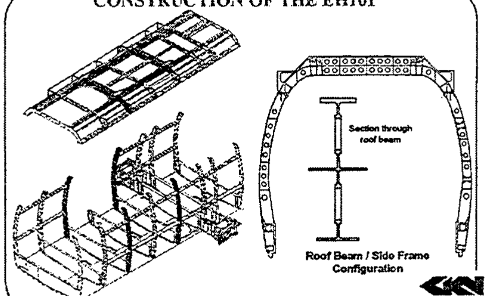
CONSTRUCTION OF THE EH101



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SUBSTANTIATION OF THE EH101

- EH101 Airframe **SAFE LIFE** demonstrated by a full-scale factored load fatigue test.
- Failure modes observed during this test were eliminated from the production standard by design changes. A stand-alone fatigue test of a production standard lift frame, and fine mesh finite element modelling, demonstrated the effectiveness of the changes.
- Safe Lives in excess of 10000 hours have been demonstrated for civil and military variants.
- Damage tolerance evaluation in accordance with JAR 29.571 still required by CAA/RAI.

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DAMAGE TOLERANCE ASSESSMENT OF THE EH101 MAIN LOAD PATH

Programme of testing initiated :

- material data
- structural elements
- full-scale components
- full-scale airframe test.

Objectives :

gather basic material data - develop and prove analytical models - generate compliance functions - demonstrate crack trajectories and failure modes.

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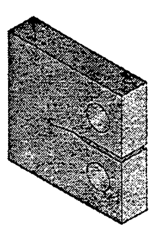
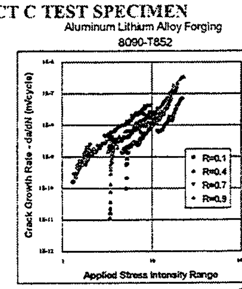
COMPACT C TEST SPECIMEN





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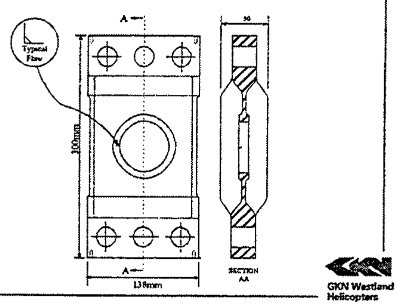
STRUCTURAL ELEMENT TEST SPECIMEN



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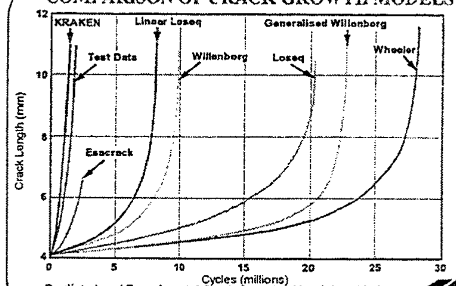
COMPARISON OF CRACK GROWTH MODELS



Predicted and Experimental Crack Growth - Aluminium-Lithium
Surface Crack Tension Specimen - Complex Variable Amplitude Load

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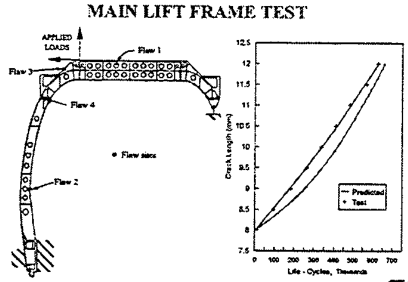
MAIN LIFT FRAME TEST



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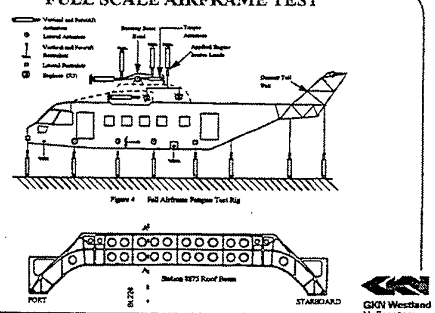
FULL SCALE AIRFRAME TEST



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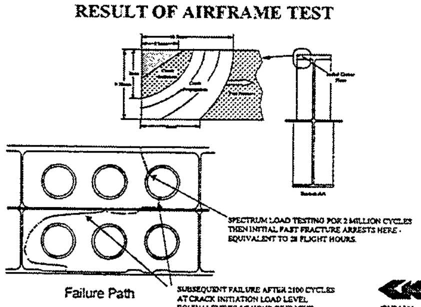
RESULT OF AIRFRAME TEST



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LESSONS LEARNT FROM TEST PROGRAMME

Cracks are very difficult to initiate - even from 4mm saw cuts, scalpel-sharpened and etched.

Critical crack lengths are short and no significant load redistribution occurs before fast fracture. Primary load paths are likely to fail completely.

Accuracy of crack growth predictions depend on accuracy of compliance functions.

Most analytical models are non-conservative under complex helicopter spectra loads (even linear summation models).

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DAMAGE TOLERANCE ASSESSMENT OF THE EH101 MAIN LOAD PATH

Analysis Programme:
building on test results, analyse many locations on the main load path under representative spectrum loads.

Objectives:
calculate primary load path inspection intervals under "fail safe design considering flaw growth" approach.

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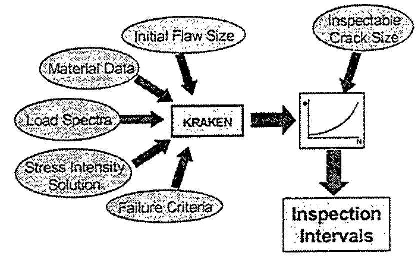
ANALYSIS METHOD



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ANALYSIS PROGRAMME: PROBLEMS WITH HELICOPTERS

- **Large numbers of load cycles at high R-ratio where the threshold value of stress intensity is very low.**
- **Crack growth models all developed for fixed wing use - only validated for low R-ratios and fixed wing load interactions. Many linear models give non-conservative predictions with helicopter spectra.**
- **Short crack behaviour is very important but little understood.**
- **Short critical crack lengths due to high stress compact section designs.**
- **Highly versatile usage means load spectra vary.**

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CRACK GROWTH ANALYSIS

- **FLAWS ASSUMED AT POINT OF HIGHEST STRESS IN FLANGES, LIGHTENING HOLES, BOLT HOLES AND CUT-OUTS. LOCATIONS SELECTED USING A VERY DETAILED FINE MESH FEM.**
- **INITIAL FLAWS ASSUMED TO BE 1.3 MM CORNER CRACKS.**
- **LOAD SURVEY DATA AT ADJACENT STRAIN GAUGES CONVERTED TO LOAD SPECTRA AT THE FLAW.**
- **FAILURE CRITERIA WAS EITHER FAST FRACTURE OR NET SECTION YIELD - BOTH AT LIMIT LOAD.**

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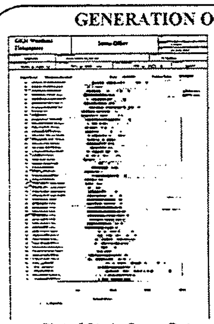
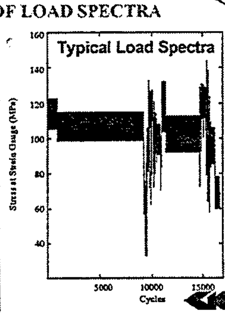
GENERATION OF LOAD SPECTRA



Plot of Strain Gauge Data



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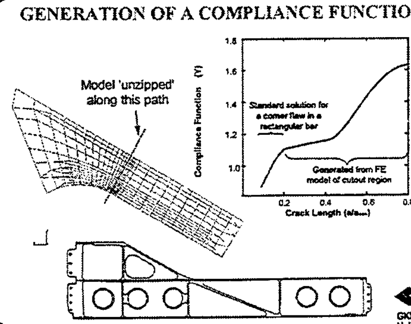
GENERATION OF A COMPLIANCE FUNCTION



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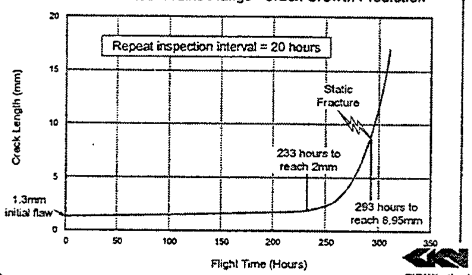
TYPICAL RESULT



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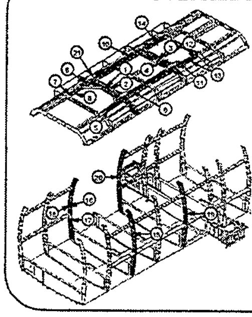
OVERALL RESULTS



Location	Inspection Interval		Life to Failure
	Threshold	Repeat	
1	91	20	89
2	8	11	59
3	459	334	2980
4	77	20	293
5	Greater than starboard		
6	16	4	63
7	181	341	1569
8	62	17	242
9	26	13	264
	Greater than starboard		
	Greater than port		
10	6	3	45
11	1.8	2	13
12	5	10	49
13	27	4	61
14	Greater than port		
15	Greater than Stn 6875 stbd. side frame		
16	427	397	3073
17	14	6	61
18	3	5	27
19	Greater than Stn 6875 stbd. side frame		
20	Greater than Stn 6875 stbd. side frame		
21	130	963	3296

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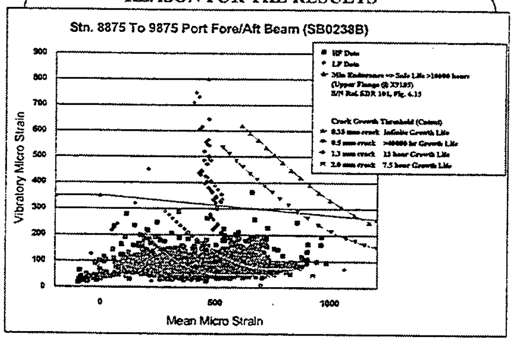
REASON FOR THE RESULTS



RESULTS IN PERSPECTIVE

- **Airframes do not crack regularly in service and result in catastrophic failure.**
- **These results are NOT related to Al-lith which, in fact, has superior da/dN curves to 7010 in the working range.**
- **Manufacture and husbandry are of a high standard.**
- **Flaws and damage are NOT cracks - cracks proved difficult to initiate on test from severe, etched saw cut flaws.**
- **For analysis the flaws are assumed at the points of highest stress - this is unlikely.**
- **Limit load events are very rare.**

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ALTERNATIVE APPROACH - SECONDARY LOAD PATH

POOR RESULTS ABOVE REFER TO DETECTING CRACKS IN THE PRIMARY LOAD PATH BEFORE IT FAILS.

ALTERNATIVE APPROACH
- ASSUME PRIMARY LOAD PATH FAILS
- SET VISUAL INSPECTIONS BASED ON GROWTH FROM SMALLER FLAWS IN THE SECONDARY LOAD PATH.

THIS APPROACH ALSO DOES NOT WORK
- TEST SHOWS COMPLETE LIFT FRAME FAILURE OCCURS
- LOADS GO UP BY 2.5 TIMES IN THE SECONDARY LOAD PATH
- GIVES RF < 1.0 AT LIMIT LOAD
- RESIDUAL STRENGTH TYPICALLY 75% LIMIT LOAD

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HOW COULD THE APPROACH BE MADE TO WORK?

CHANGE	EFFECT
REDUCE THE RESIDUAL STRENGTH REQUIREMENT TO LESS THAN LIMIT LOAD BASED ON LOW PROBABILITY OF OCCURRENCE	MIGHT MAKE SECONDARY LOAD PATH APPROACH VIABLE
IMPROVE NDT CAPABILITY	SMALLER INITIAL FLAWS BELOW THRESHOLD
REAL-TIME MONITORING	REDUCED INSPECTION COST BUT HIGH SYSTEM COST, COMPLEXITY, AND RISK OF FALSE ALARMS
ONLY CONSIDER LOCATIONS AT RISK OF DAMAGE	REDUCED INSPECTION REQUIREMENT BUT MAY MISS SOMETHING

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THE EFFECT OF MANDATORY CRACK GROWTH

- **Higher weight**
- **Designed from the start with higher redundancy**
- **Use of composite materials for primary structure**

New aircraft qualified to these requirements will be penalised compared with older aircraft which may even lack a thorough Safe Life evaluation.

To improve safety the weight would be better used in the dynamic components or in HUMS systems.

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ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

"The capability of flawed structure as shown by tests or analysis based on tests to sustain, without measurable flaw growth, the spectrum of operating loads expected during the service life of the rotorcraft or during an established replacement time."

Coupon fatigue tests ⇒ Knockdown factor for:
- Corrosion
- Impact
- Scores

⇒ Flaw Tolerant (Enhanced) Safe Lives

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ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

Initial Flaw Requirement

- **Largest realistic flaw that would not be picked up by routine inspections.**
- **Ideally needs to be based on in-service experience**
- **Difficult for new aircraft with new materials and design concepts**

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ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

Coupon Test Programme

Damage	Application Method	Levels	Number of Specimens
Datum	No damage	-	10
Score	Machining tool dragged across the specimen.	0.125mm deep	10
		0.25mm deep	10
Sharp Impact	Pyramidal impact was applied perpendicular to the coupon	25J	11
		12.5J	5
Oblique Impact	10mm wide screwdriver impact at 45° to test section	6J	10
Corrosion	Specimen placed in salt environment for 2 days. Damage site pre-drilled to represent a corrosion pit.	0.4mm deep	10
		0.2mm deep	10
		~~No pre-drill~~	3

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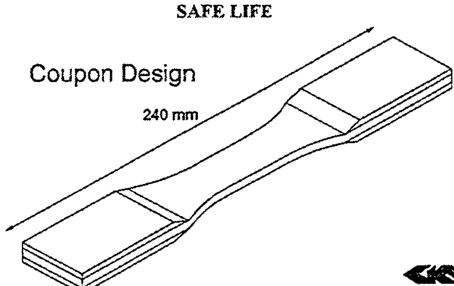
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

Coupon Design



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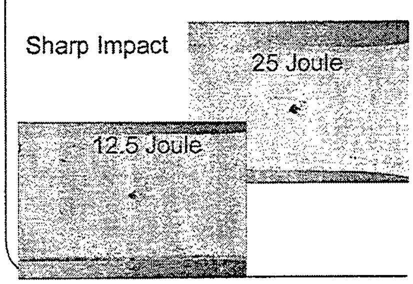
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE

Sharp Impact



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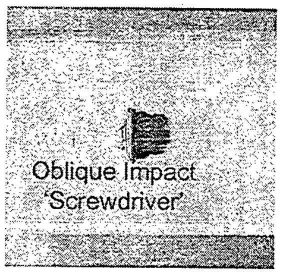
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE



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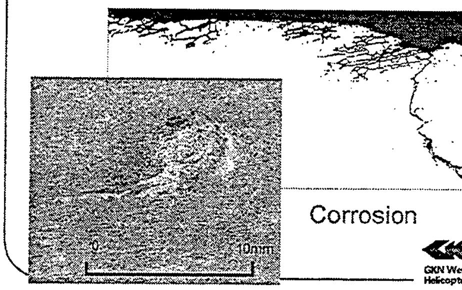
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE



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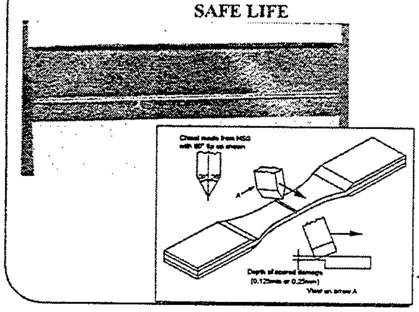

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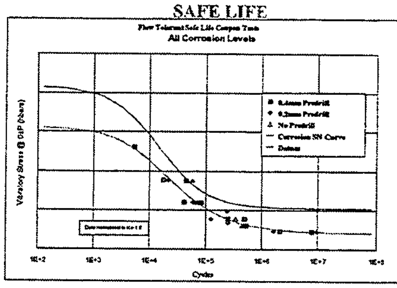
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE



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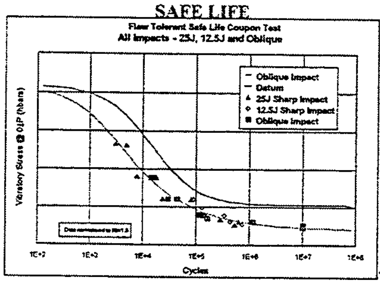
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE



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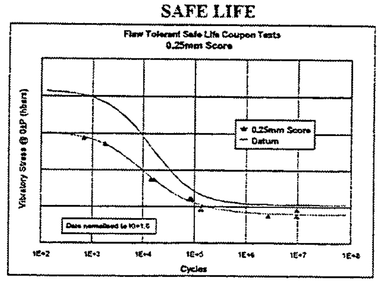
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE



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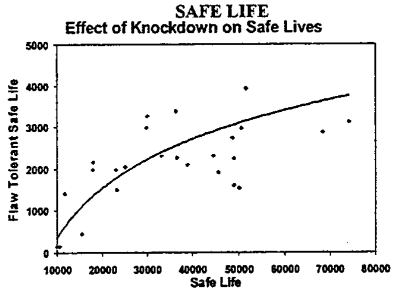
ALTERNATIVE APPROACH: "ENHANCED" SAFE LIFE



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CONCLUSIONS

- This case study illustrates the difference between the Safe Life approach and crack-growth-based damage tolerance.
- GWHL research has shown that many crack growth models are inaccurate and non-conservative under helicopter loads.
- Near-threshold and short crack behaviour are essential elements of accurate predictions.

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CONCLUSIONS - continued

- Our comprehensive test and analysis programme has shown that crack-growth damage tolerance will not work on conventional rotorcraft structures even if they have good, rigorously demonstrated, Safe Lives.
- This is because of their design and high R-ratio (low threshold) + high number of cycles load spectra.
- Service experience shows that more onerous design requirements or inspection regimes are not warranted.

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CONCLUSIONS - continued

- The Enhanced Safe Life approach represents the best approach to Damage Tolerance for helicopters provided that the ESL is used as an inspection period.

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Crack Growth Rates under Vibratory Loading for Helicopter Damage Tolerance

Richard Buller
Senior Stress Engineer
ENGAGE

Engage is a GKN company

Cranfield UNIVERSITY

ENGAGE

Overview

- Damage tolerance in helicopters under FAR 29.571. Differences with fixed wing methods.
- Vibratory loads - static and dynamic components.
- Helicopter component load spectrum development.
- Study of vibratory load crack growth using omission level technique.
- Discussion and conclusions.

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Helicopter damage tolerant design

Differences between application of fixed wing and helicopter crack growth technology:

- Highly variable loading experienced by components due to wide variety of manoeuvre loads and flight sorties.
- High accumulation of vibratory load cycles originating from main rotor revolution.
- High yield strength materials selection based on fatigue initiation properties.

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The application of the fail-safe design considering flaw growth procedure to the damage tolerance analysis of helicopter components can not simply be translated from the well established methods of damage tolerance design used in fixed wing aircraft since the 1970's. For instance most helicopter structures have one main load path and so there is little opportunity of a redundant, fail-safe design. There are several important differences between the fixed wing and helicopters which impact on process of design against fatigue.

Crack growth under vibratory loading

- Source of vibratory loads mostly from main rotor at *1R* for rotor components and *nR* (blade passing frequency) for fuselage.
- Vibratory loads are superimposed on manoeuvre loads so are typically high mean stress (R<0.7).
- Vibratory loads can be removed for testing or analysis.
- Experimental omission level technique developed to study crack growth mechanisms under vibratory loads.

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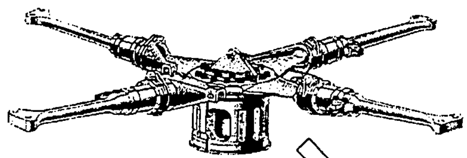
The vibratory cycles typically make-up 90% of the fatigue loads in a helicopter loading spectrum. Therefore, knowledge of the behaviour of these cycles is essential in providing an accurate means to determine inspection periods and to ensure that the helicopter service life is achieved without premature failure due to fatigue.

The understanding of the effect of vibratory load cycles on the crack growth rates in helicopter components under spectrum loading is two-fold: (1) Large reductions in laboratory and full-scale fatigue testing time can be achieved by omitting low amplitude, vibratory cycles below a certain stress cycle range from the loading spectrum using the threshold stress intensity property. (2) The point at which these cycles start to cause significant fatigue crack damage indicate the ability to assign reasonable inspection periods for a component.

However, to do these correctly the crack growth rates of the vibratory load cycles need to be established through consideration of load interaction effects, crack lengths and applied stress levels.

Rotorhead stress sequence development



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Lynx main rotor

- Stress sequence for Lynx main rotorhead based on methodology of Felix/Helix. Strain gauge data taken from flight test data.
- Represents 140 sorties and 190.5 flying hours.
- Contains 1,900,000 *1R* vibratory load cycles.

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The representative stress sequence used for the testing of omission level effects was developed for a helicopter rotorhead component using the well established methods of Helix and Felix. These sequences consist of 140 sorties representing 190.5 hours of flight. Each sortie in the sequence represents either training, transport, anti-submarine warfare (ASW) or search and rescue (SAR) flight. These sorties are in turn described by a defined sequence of twenty two manoeuvre types and are given in three flight lengths of 0.75, 2.25 and 3.75 hours.

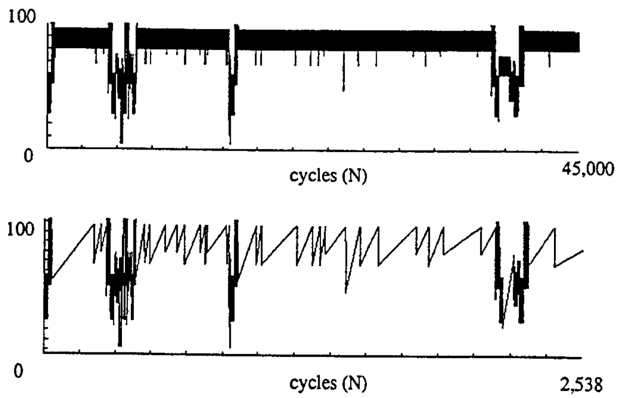
The Felix stress sequence is based on a fixed rotor helicopter so was chosen to be the basis for the development of a stress sequence at a location on the fixed main rotor component studied here. The critical location on the rotorhead component was strain gauged and data were recorded during several test flights to provide data for the stress sequence generation. The strain gauge data were then processed and assembled in accordance with the procedures used in Felix. By using the Felix procedure the small, IR vibratory loads which the component experiences due to rotor rotation are well defined and can be used for testing on laboratory specimens.

One drawback of the Felix procedure is that the load data are strictly defined into integer levels of four between 0 and 100 so that the sequence not as 'continuous' as desired.

Progressive omission level technique

Original sequence sample

Omission of range 16% cycles



- Retains sequencing of major load cycles.

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A novel method was devised to determine the fatigue damage contribution of the different cycle ranges by progressively omitting small range cycles of increasing range from the rotorhead loading sequence. The method considers four omission levels at 16, 20, 24 and 32% cycle ranges. The 'peak-valley filtering' method was used to omit the cycles at each step where trough-peak pairs are retained or omitted according to the desired load range. It maintains the correct ordering of the sequence as is shown in Figure.

The method assumes that the difference in crack growth rates between the two omission level tests will be due to the fatigue crack damage of the omitted cycles. Therefore, the crack growth damage rates of various cycle ranges can be separated out and used for analysis.

Summary of testing

- Four omission level tests each on titanium Ti-10V-2Fe-3Al and aluminium 7010 T73651 compact tension specimens (17.5mm).

Omission range level	Rainflow cycle count	Reduction in length from original (%)
16%	1989925	-
20%	113063	94.3
24%	110907	94.4
32%	51404	97.4

- Crack length monitored with automated direct current electro-potential method.
- Instron digitally controlled machine for spectrum loading.

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Specimens

Ti1023 was used because this is the material from which typical main rotorhead components are manufactured from. The lower strength 7010 was used to compare the effect of yield strength differences on load interaction effects between the two alloys. 7010 is an alloy typically used in the manufacture of helicopter fuselage structures. The CT specimens were t=17.5 mm thick giving predominantly plane strain stress states throughout all the tests.

Crack Length Measurement

A custom designed direct current potential drop (DCPD) crack length measurement system was designed to provide measurement of fatigue crack growth rates for all the fatigue tests.

Loading

Loading was performed under load control, at 10Hz and under ambient laboratory conditions. The tests were carried out in accordance with the procedures defined in ASTM E 647-93 "Standard Test Method for Measurement of Fatigue Crack Growth Rates".

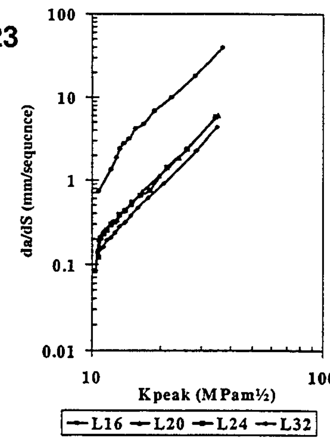
Four tests on each of the Ti-1023 and 7010 CT specimens were conducted at each of the omission levels defined in the table.

Loads are defined in terms of the peak or maximum load experienced by the sequence which is level 100 in the Felix definition. Hence the peak stress and stress intensity factor are referenced to this level and are termed S_{peak} and K_{peak} respectively

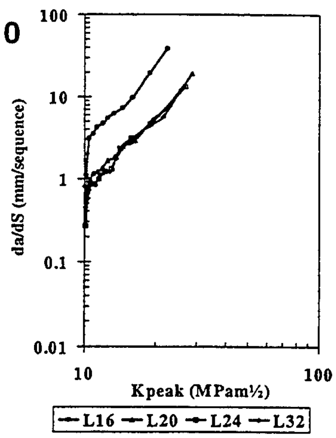
Omission level crack growth rates

- Removal of vibratory cycles (16%) gives increase in crack growth lives of 3.6 for 7010 and 5.4 for Ti1023.

Ti1023



7010



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For both materials the effect of progressively omitting small range cycles is to increase the flights to failure for the specimens. Removal of the level 16 cycles from the full sequence test gave an increase in life of 3.6 and 5.2 times for 7010 and Ti1023 respectively. In each material the number of flights to failure were almost identical for the 20% and 24% omission tests as expected because there are a relatively small number of 24% cycles (2156) and these caused little fatigue damage.

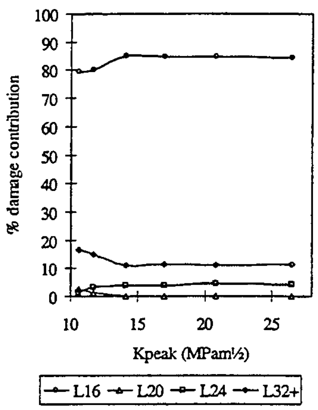
The increase in specimen life due to omission of small cycles has been observed by others and is due to a removal of damaging load cycles. In this case the dominant omission level effect is due to removal of the vibratory load cycles which is in part due to the large proportion of small cycles in the sequence.

Figures show the crack growth rates per sequence for each material against the peak stress intensity factor, K_{peak}. The effect of omitting the 16% vibratory cycles is clearly shown for each material as the growth per sequence is reduced significantly for all the 20%, 24% and 32% sequences indicating the progressive omission of cycles from each were causing fatigue crack damage in the previous test.

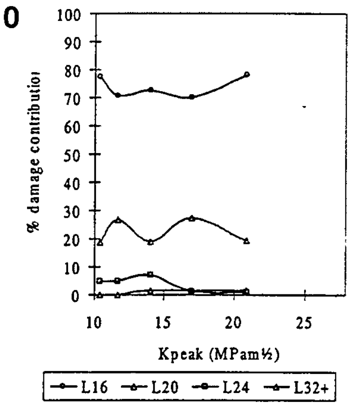
Crack growth damage contributions

- Difference in crack growth rates between omission level test due to crack length damage of omitted vibratory cycles.
- Correlated with SEM fractographic examination of fracture surfaces.

Ti1023



7010



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It may seem obvious that the removal of 94% of the cycles from the sequence would result in an increase in life to failure. However, most of the vibratory cycles operate in the near-threshold regime so would have near-zero growth rates and may not always contribute significantly to fatigue crack damage in a strictly linear manner due to the slope of the log da/dN versus log ΔK curve.

The average crack growth rate per loading sequence is calculated at different levels of K_{peak} and from these the crack growth increment per omission cycle range can be calculated: The crack length damage contribution for each cycle range during the full sequence test (all cycles included) is the sum of all the individual cycle ranges based on the assumption that each cycle range causes the same amount of crack growth increment in each omission level test. That is, the differences between the omission level tests are be due to the omitted cycles.

In both materials the 16% range cycles cause a large proportion of the damage over the entire test crack lengths. On average, the damage for the 16% cycles is about 85% for Ti1023 and 75% for 7010. The next largest contributor to crack growth damage are the 32% range and above cycles. The damage contributions are typically 15% for Ti-1023 and 20% for 7010. In both materials the 20% cycles give negligible contributions to crack damage which is expected because they only constitute 0.1% of the loading sequence

Crack growth damage due to vibratory loads

- Modelling of omission level tests using a crack growth model without load interaction effects (ESACRACK/NASGRO) gives non-conservative results.

Material	Omission	a_i	a_f	Test N_T	Model N_M	N_T/N_M
Ti1023	16	17.43	49.8	1145	3629	0.32
	20	17.43	49.2	6341	6107	1.04
	24	17.43	49.5	6440	6129	1.05
	32	17.43	48.6	8612	8496	1.01

- Vibratory load cycles grew faster than expected under sequence loading conditions when compared with constant amplitude loading data.

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It is significant that, even though the 16% range cycles constitute 94% of the total number of cycles in the sequence, the contribution to crack growth damage by the vibratory cycles is higher than that which a typical model would predict.

The 16% cycles had *greater* fatigue crack growth rates under helicopter sequence test loading than under equivalent CAL conditions. This is a important observation because most fatigue crack models would not predict that these smaller cycles grow faster under CVAL as they are usually 'tuned' for retardation. Hence prediction of growth rates under helicopter CVAL could give misleading predictions.

The Esacrack analysis (Table) indicates the non-conservatism of the modelling of the Ti-1023 full sequence test by a factor of three.

Consequences for helicopter damage tolerance

Accelerated growth rates of high R ratio vibratory load cycles will impact on helicopter damage tolerance design for the following:

- Design of fatigue loading sequences for analysis and testing.
- Selection of materials and component stress levels.

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Simplification of a helicopter loading sequence for testing and analysis is best achieved by omission of the small vibratory load cycles which gives a significant reduction in sequence lengths. However, the main drawback demonstrated here is that these small cycles contribute to a significant amount of crack growth damage so modelling can be misleading. A criteria for 'safe' omission of small load cycles could be based upon the mechanism of accelerated crack growth of these cycles. This would be done by determining when a cycle range can be omitted from a sequence such that its omission will not result in changes in specimen life - best performed by testing.

The damage tolerance capability of a helicopter structure can be demonstrated by examining crack growth rates over a range of flaw sizes for an arbitrary edge cracked component under realistic service stresses.

Omission techniques for helicopter spectra

- Observations indicate that vibratory load cycles have accelerated growth rates under spectrum loading.
- Increased ΔK_{eff} (K_{max}-K_{cl} or K_{max}-K_{PR}) compared to constant amplitude loading.
- Due to frequent underloads in sequence (GAG and manoeuvre) and absence of tensile overloads.

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These observations indicate that the small range cycles have accelerated growth rates under CVAL compared with the equivalent cycles under CAL. An explanation of this observation can be based on transient residual stress fields ahead of the crack tip give an increased ΔK_{eff} when compared with constant amplitude loading is proposed.

The key to the accelerated growth rates is the 'form' of the loading sequence. Typically most cycles have similar K_{max} values, there are no tensile overloads and periodic underloading occurs due to the GAG cycles and low R ratios manoeuvres.

Underload tests have shown that the underload can eliminate or reduce the effect of overloads during high R ratio loading and can cause acceleration during constant amplitude loading. The effect of intermittent underloads causing acceleration of high R ratio CAL crack growth rates has been observed by others. However, the GAG underloads occur infrequently so it is most likely that underloading due to the occasional low R ratio manoeuvre is responsible for increasing the ΔK_{eff} of the high R ratio, vibratory cycles.

Omission techniques for helicopter spectra ...

- 'Safe' omission criteria based on:

$$(K_{max}-K_{PR}) < K_{th}$$

considering cycle range, mean and threshold.

- Careful determination and modelling of near-threshold region over variety of R ratios and considering load-interaction effects.

- 'Safe' omission for crack growth much less than for initiation based methods

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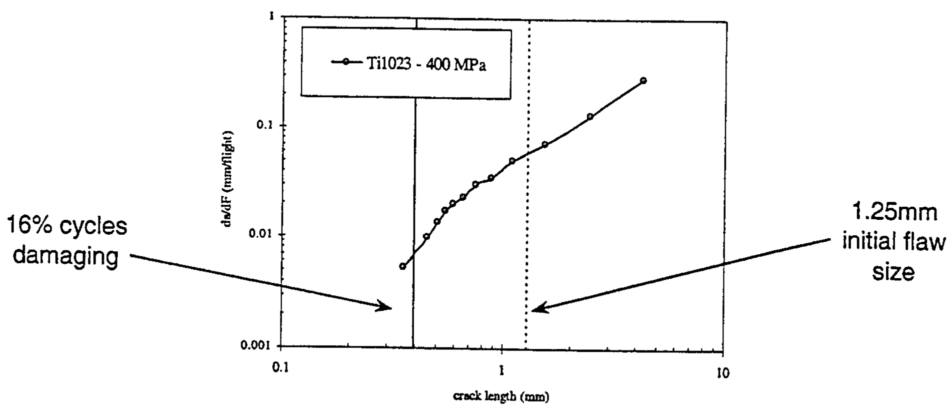
The vibratory loads and the effect of these on fatigue crack growth using the fatigue loading sequence provides a useful means of understanding the mechanisms under which cracks grow in helicopter components. As the concurrent conditions of plane strain, near threshold and high R ratio affecting crack growth are unique to helicopters the mechanisms of growth under such conditions can determined by analysing the crack growth behaviour of vibratory loads through omission. By being able to accurately predict the behaviour of vibratory load cycles, which dominate crack growth damage, a model with general applicability to helicopter damage tolerance design can be developed.

Material and component stress levels

- Damage tolerance capability demonstrated by considering spectrum crack growth rates for arbitrary Ti1023 rotor component, S_{peak}=400MPa.



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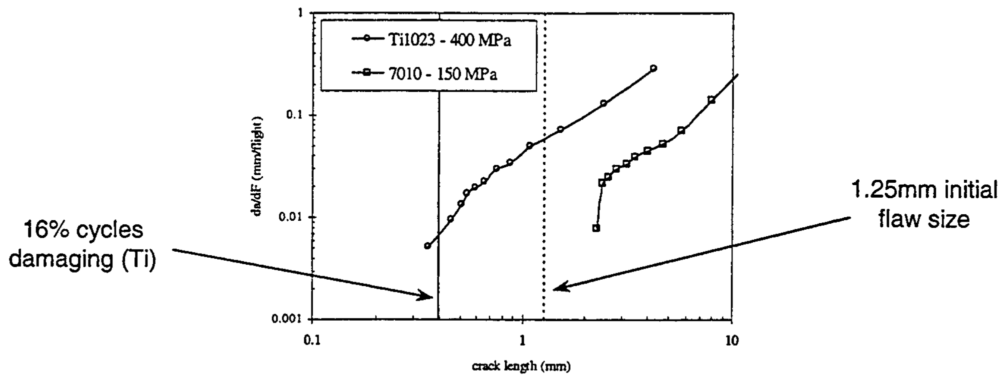
This example component is 'subjected' to the full test sequence with an operating peak tensile stress of 400 MPa (approximately one third of the σ_{UTS} for Ti1023) by converting the crack growth rate per sequence curve into crack lengths that will produce the same K_{peak}'s assuming similitude exists. The experimental data can be applied to range of crack lengths from 0.35 to 4 mm.

Figure indicates the crack length above which the vibratory load cycles will start to cause damage. At 0.39 mm this is will below the typical initial starting defect size of 1.25mm used in the aerospace industry. Improving the damage tolerance capability of a helicopter structure would require the vibratory load cycles to be maintained below their fatigue threshold. Elimination of crack growth due to vibratory load cycles would require a shift of the curve in Figure to the left and/or reducing the size of the initial defect. This could also mean a reduction in component peak stress level which usually translates into an increase in weight

Material and component stress levels...

- Comparison between 'typical' rotorhead (Ti1023) and fuselage (7010) components with S_{peak}=150MPa for 7010.



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Design strength and crack growth rate differences between 7010 and Ti1023 are also indicated in Figure. A typical maximum operating stress of 150 MPa was used for 7010.

This emphasises the differences between different parts of the helicopter structure because some materials will be more sensitive to crack growth under vibratory load cycles. This is mainly because of the lower operating stress that a safe life design gives for aluminium alloys and not necessarily due to material fatigue crack growth properties. Note that a fuselage component would be subjected to the 4R vibratory loads.

Titanium or other high strength metals may have better fatigue crack growth properties but are subject to higher operating stresses. This indicates that aluminium fuselage structures would be amenable to damage tolerance design because vibratory load cycles will not grow from typical initial flaw sizes (1.25 mm).

A reasonable inspection regime could be applied because crack growth will occur under manoeuvre load cycles in a similar manner to fixed wing transport aircraft.

Summary

- Progressive omission level technique used to investigate vibratory load cycle crack growth damage.
- Vibratory load cycles typically 94% of spectrum loads and contribute to 80% of the crack growth damage.
- Standard constant amplitude loading linear summation models are non-conservative by factor of three. ie. Accelerated growth under vibratory loading.
- Damage tolerant analysis of high strength rotor (dynamic) components difficult to achieve with current safe life design stresses.
- Damage tolerant analysis of aluminium fuselage (static) components can be achieved with current safe life design stresses.

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Small vibratory load cycles had the most significant effect on the overall growth rate of the variable amplitude loading sequence, causing up to 80% of the fatigue crack damage which was supported by fractographic examination. Comparison with a constant amplitude loading model revealed that these small cycle were growing at an accelerated rate when applied in conjunction with periodic underloads due to low R ratio underloading.

Significant crack growth rates due to vibratory load cycles under helicopter loading conditions mean that the damage tolerant design of high strength (Ti1023) helicopter components will not be achieved using current stress levels given by a safe life analysis. A reduction in stress levels and initial flaw size is required but with increases in component weight and inspection length capability.

Lower strength aluminium alloy (7010) components are less sensitive to vibratory load cycles because of reduced operating stresses given by a safe life analysis. This indicates that aluminium fuselage structures would be amenable to damage tolerant design because vibratory load cycles will not grow from typical initial flaw sizes (1.25 mm). A reasonable inspection regime could be applied.

SESSION E

CRACK GROWTH PREDICTION METHODS FOR HELICOPTER LOAD SPECTRA AND MATERIALS

CIVIL AVIATION AUTHORITY

Cranfield UNIVERSITY

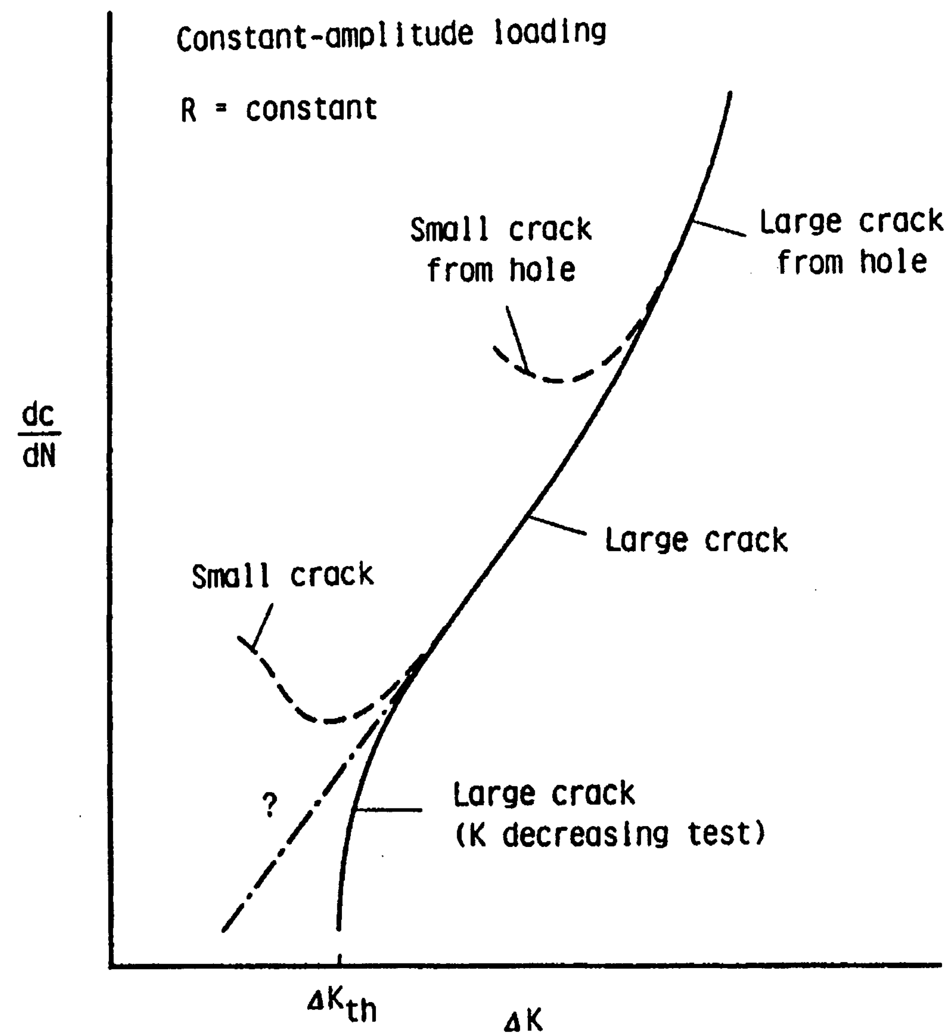
DERA

THE LARGE-CRACK ANOMALY IN THE THRESHOLD REGIME

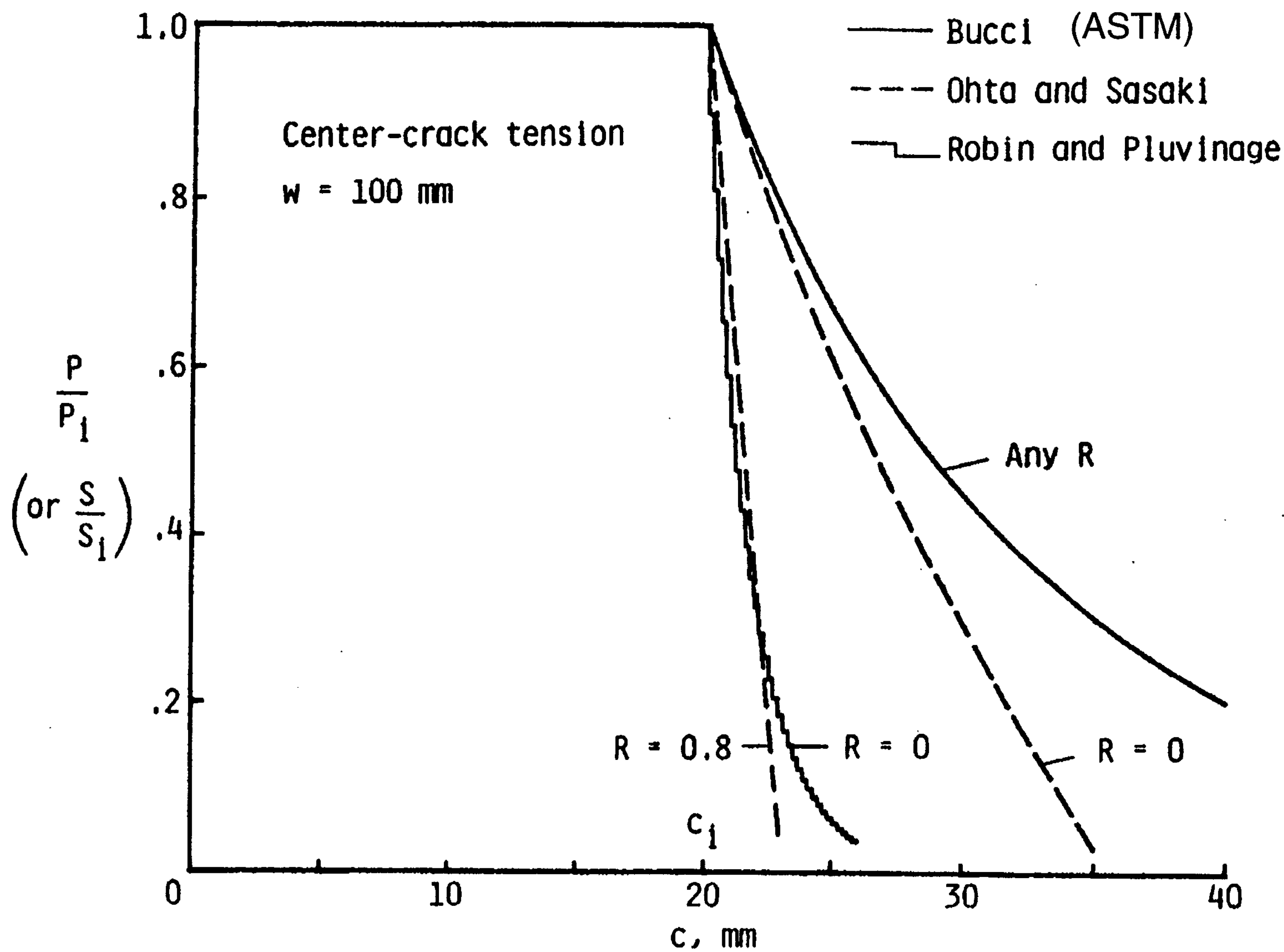
J. C. Newman, Jr.
NASA Langley Research Center
Hampton, Virginia USA

Workshop on Damage Tolerance in Helicopters
Cranfield University
Cranfield, United Kingdom
April 4-5, 2000

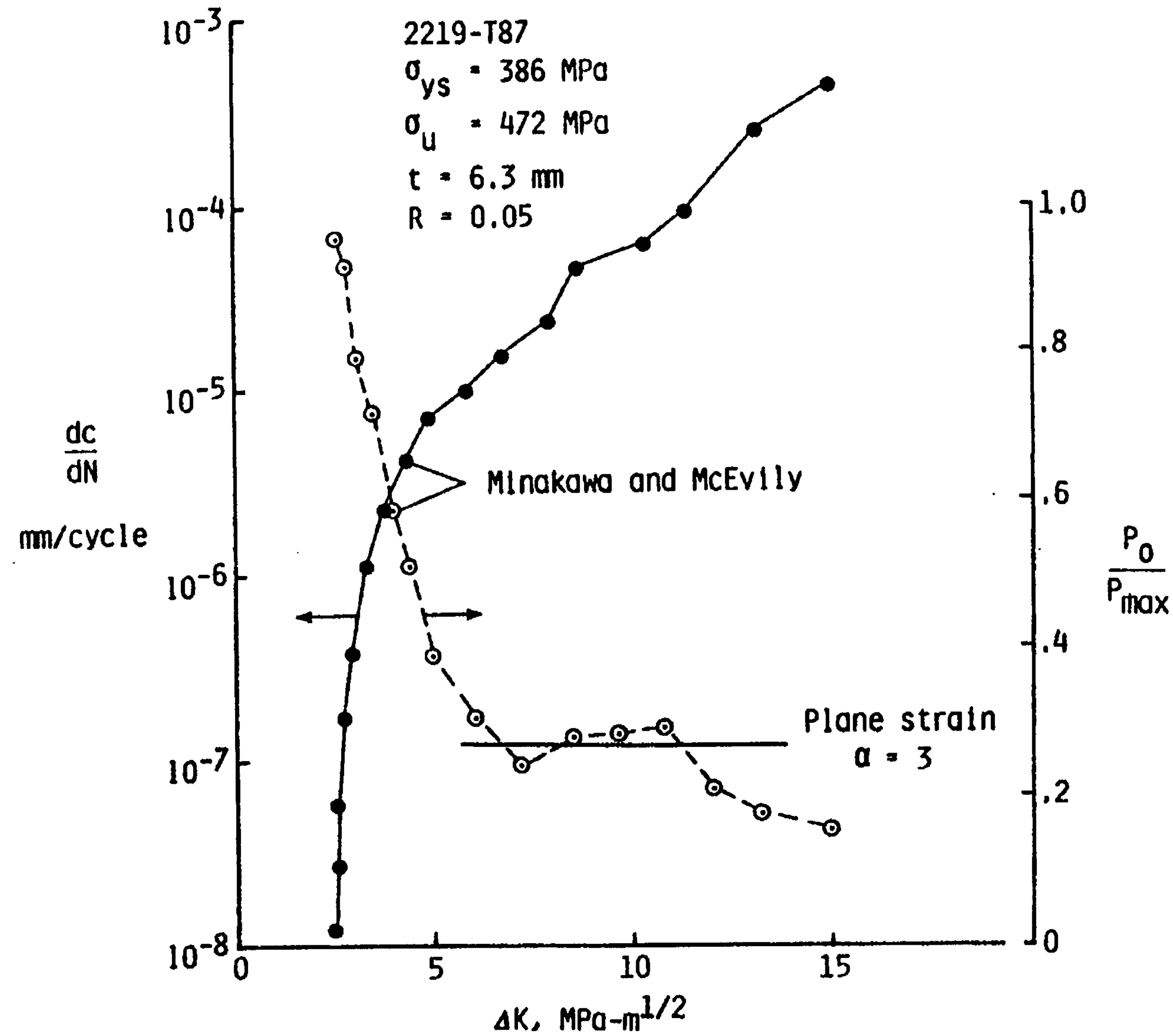
TYPICAL LARGE- AND SMALL-CRACK DATA



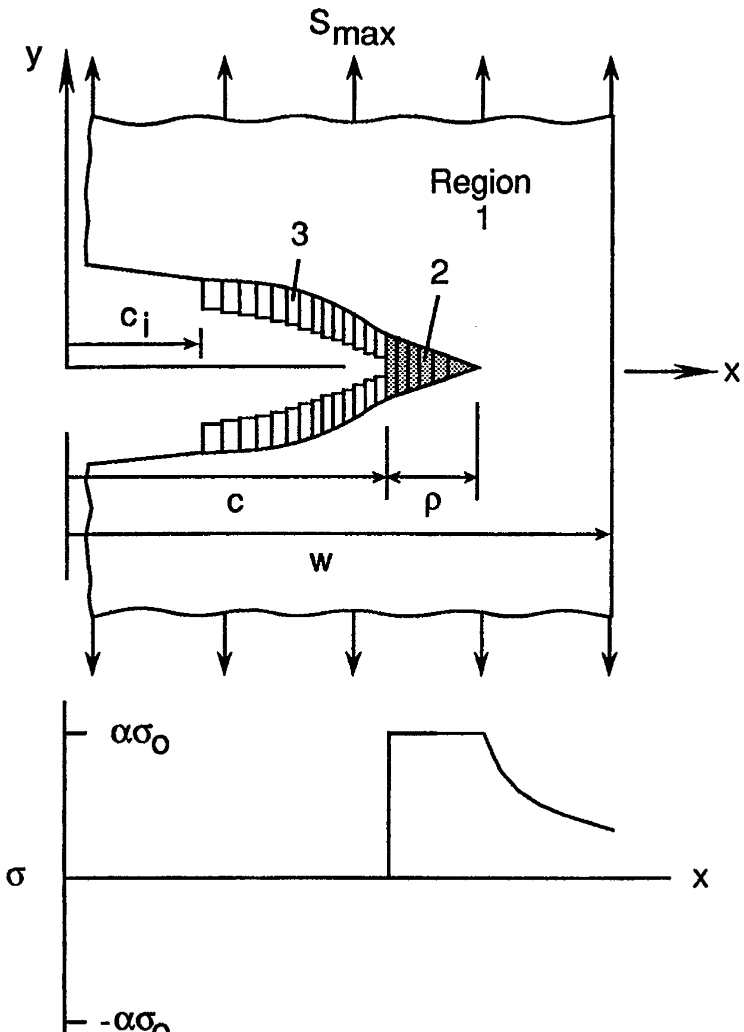
TYPICAL LOAD-REDUCTION PROCEDURES



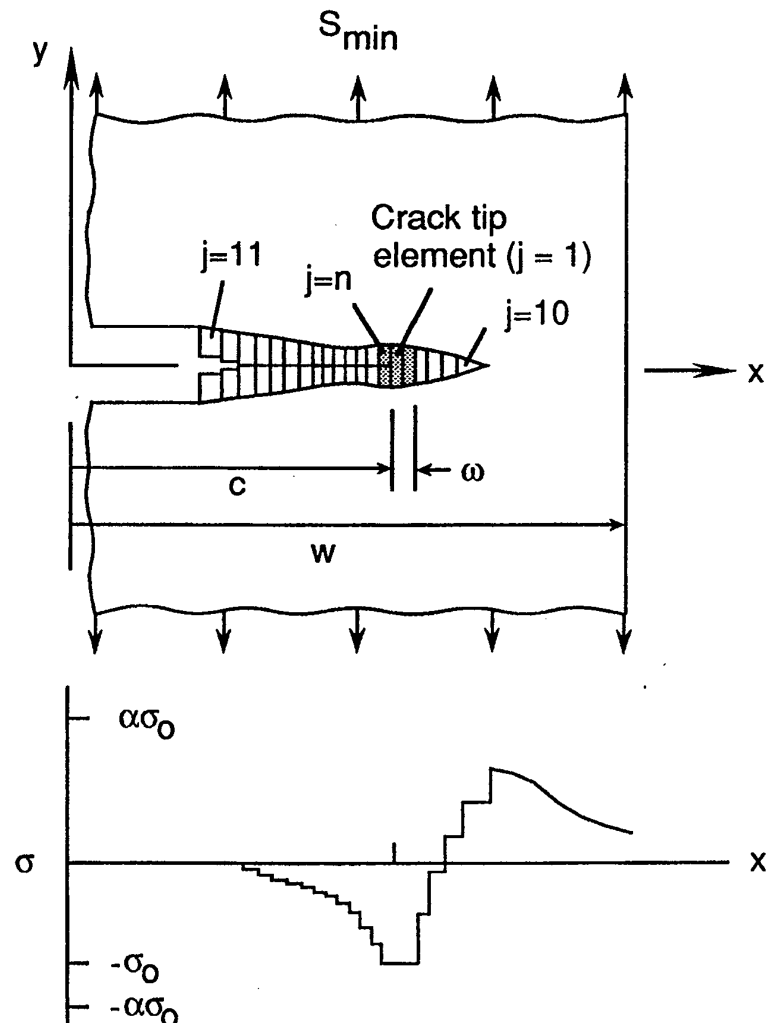
MEASURED CRACK-GROWTH RATES AND CRACK OPENING LOAD LEVELS



FASTRAN -- CRACK CLOSURE MODEL

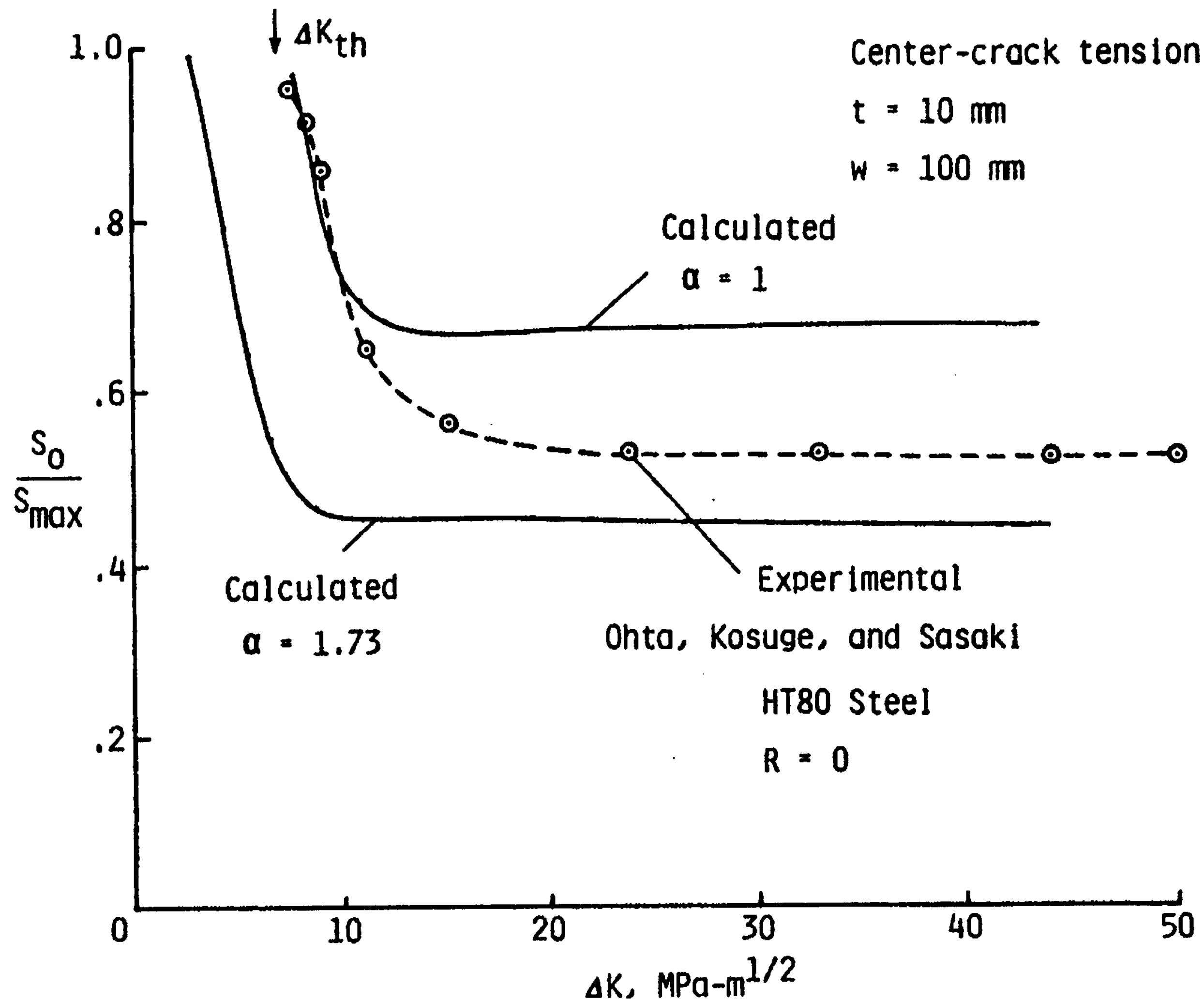


(a) Maximum stress.

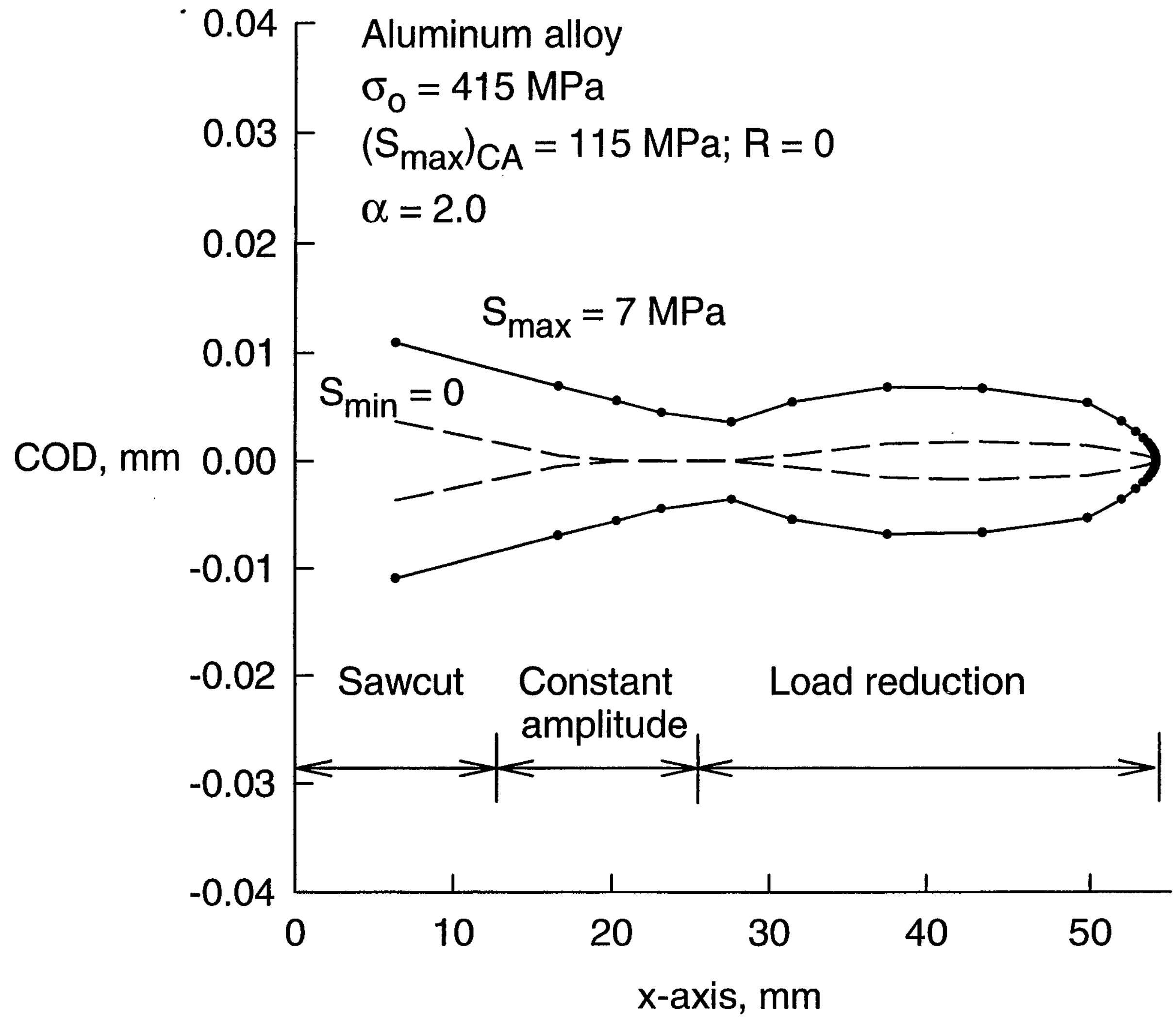


(b) Minimum stress.

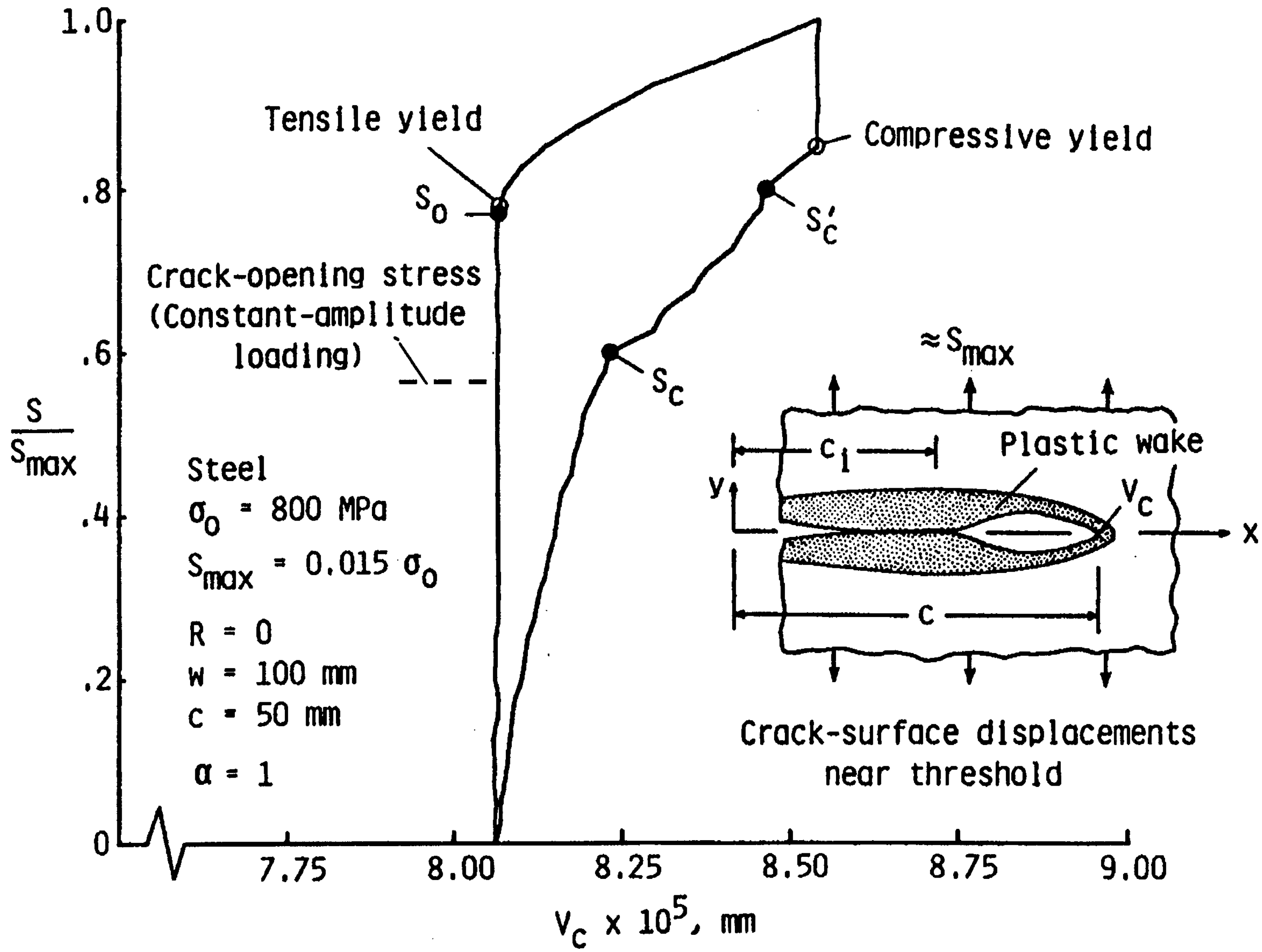
COMPARISON OF MEASURED AND CALCULATED CRACK OPENING LEVELS DURING LOAD REDUCTION



CRACK-SURFACE DISPLACEMENTS AFTER LOAD-REDUCTION TEST SIMULATION

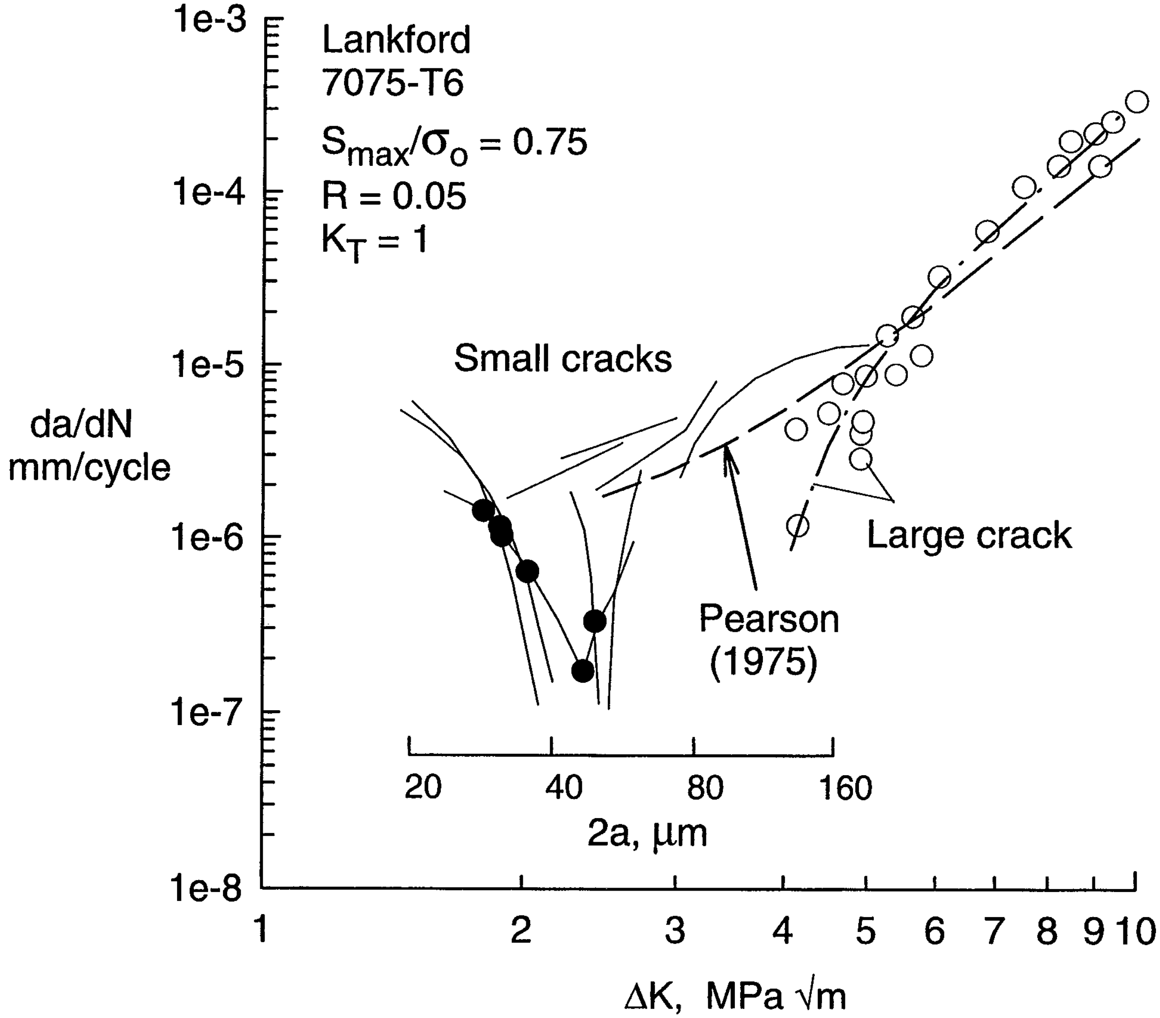


CALCULATED CRACK-TIP DEFORMATIONS AFTER LOAD-REDUCTION PROCEDURE

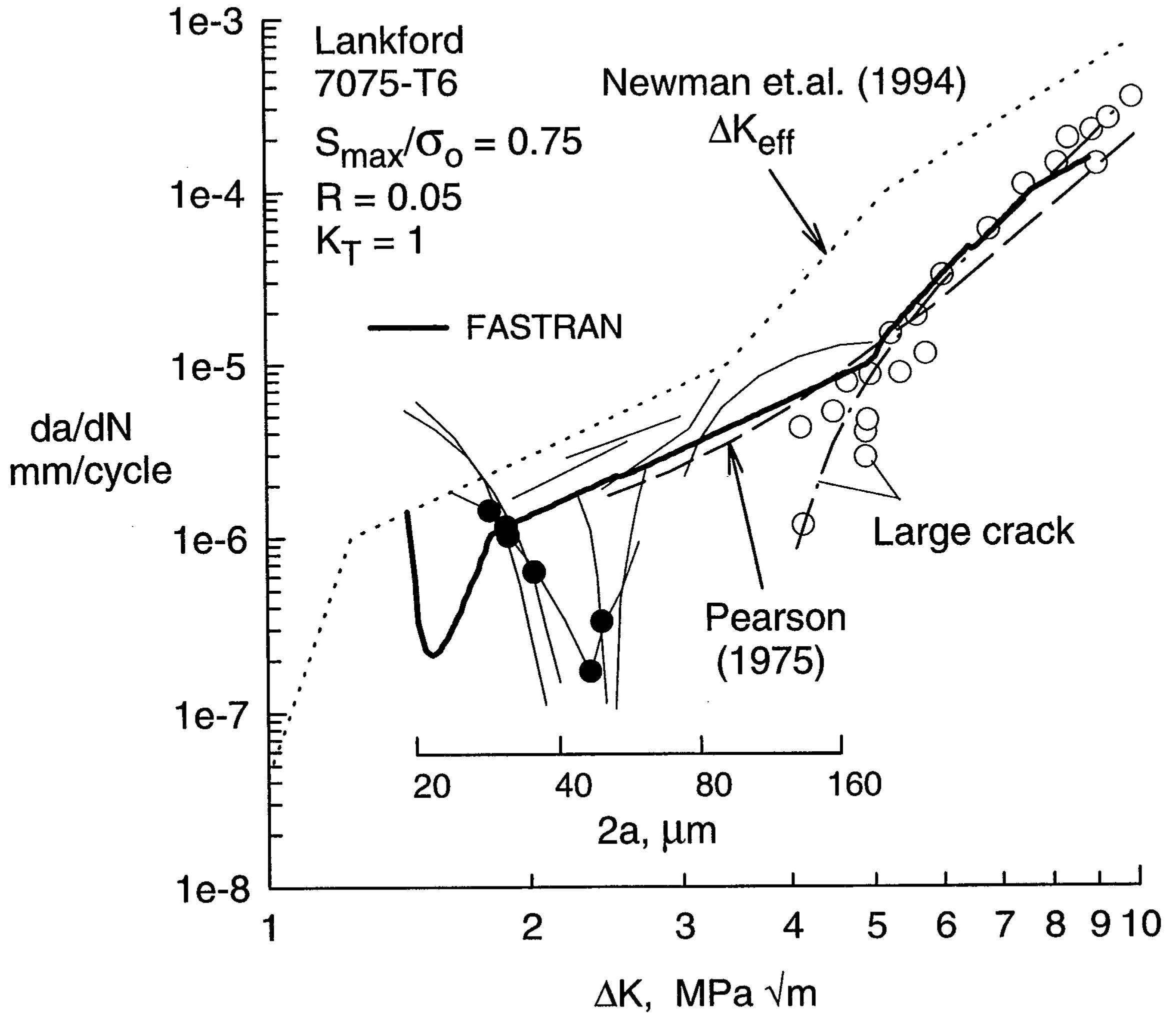


LARGE- AND SMALL-CRACK DATA ON 7075-T6

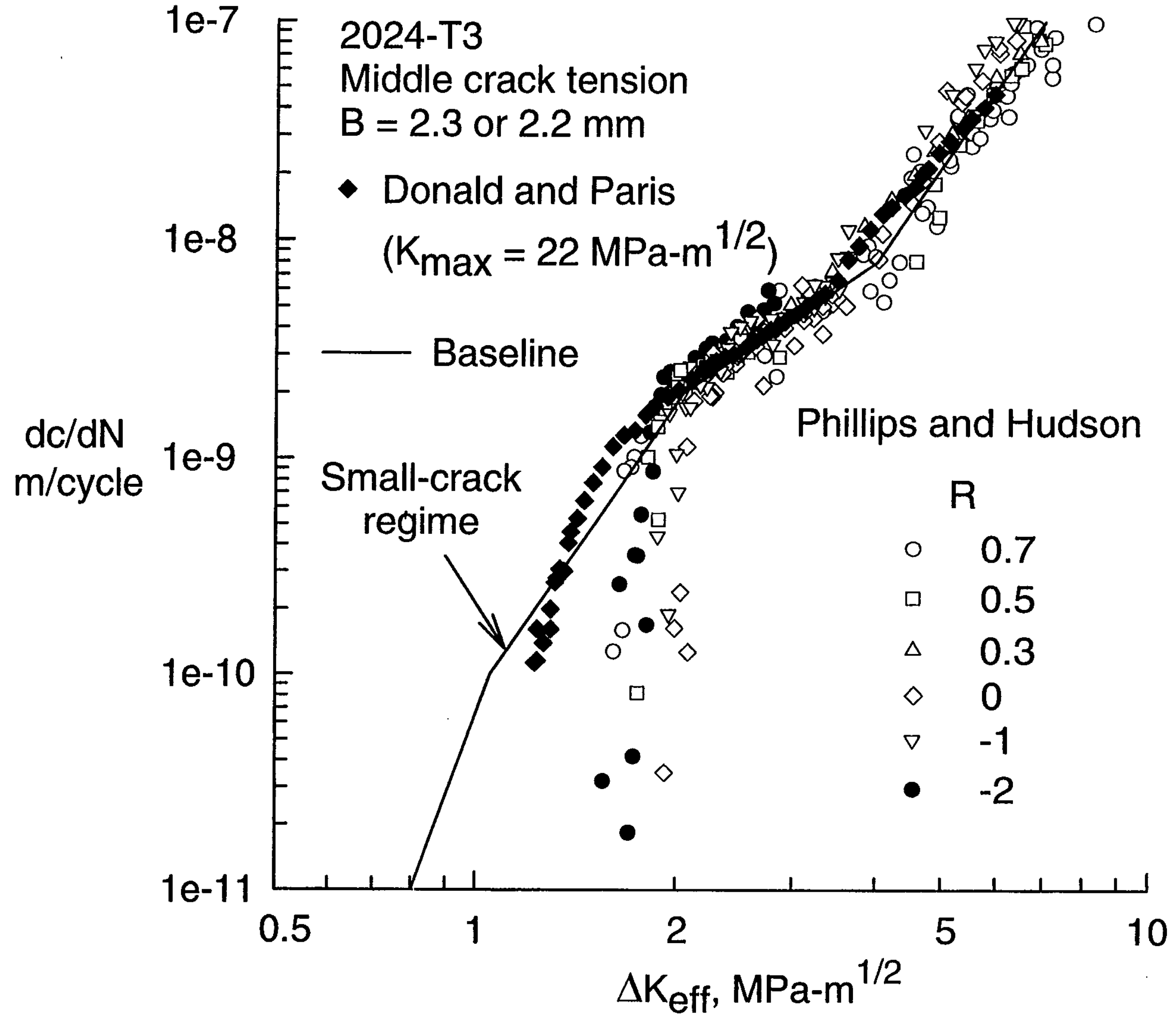
Pearson 1975; Lankford 1983



COMPARISON OF LARGE- AND SMALL-CRACK DATA AND CRACK-CLOSURE MODEL PREDICTION



COMPARISON OF LOAD-REDUCTION, K_{max}-EQUAL-CONSTANT, AND SMALL-CRACK RESULTS



CONCLUDING REMARKS

- Load-reduction test procedures on large cracks cause "remote" closure that elevates the elastic stress-intensity factor range threshold (ΔK_{th}).
- Small-crack effects (lack of closure) occur in the micro-crack length range (< 100 μm or 0.004 inches) for a wide variety of materials.
- Small-crack data for cracks greater than about 100 μm (0.004 inches) in length is "steady-state" crack-growth behavior for a wide variety of materials.
- Small-crack data is "real" and large-crack data in the threshold regime is the "anomaly".

Workshop on Damage Tolerance in Helicopters
Cranfield, UK, 4-5 April 2000.

THE BASICS OF A MODEL FOR FATIGUE CRACK GROWTH IN HELICOPTER COMPONENTS

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Helicopters have traditionally been designed using the "safe life" design philosophy. For about 10 years, the "damage tolerant" design philosophy, successfully applied to fixed wing aircraft design and maintenance procedures [1,2], is either discussed or applied to rotorcraft structures [3,4]. A flaw tolerant design philosophy requires the calculation of crack growth due to service loading with sufficient accuracy. Moreover, the crack growth phase has to be long enough to realise an inspection procedure on a cost effective basis. The main requirements to a crack growth model are robustness (simplicity, parameters are easy to obtain, etc.) and a most accurate mirroring of the physics of crack growth. A fair amount of research has been done in the field of fatigue crack growth and a number of crack growth prediction models exist which are based on different theoretical considerations. Despite the variety of crack growth models, a unified approach to fatigue crack growth is not yet found. The existing models involve various fitting parameters that adjust the result of the calculation to experimental crack growth data from the sequences they should predict. As long as no reliable crack growth code exists, a model that uses some "fitting parameters" is always superior to a complicated model in solving practical problems on a daily basis. It is no surprise that, e.g., the models by Wheeler [5] and Willenborg [6] are so widely established and used. Acknowledging that prediction models involving fitting factors have their place in quickly solving practical problems, we should try to understand the physics of fatigue crack growth. A physically correct model has to predict fatigue crack growth without extensive fitting, thus providing the basis for safer damage tolerant design and management of engineering structures.

This presentation tries to contribute in this effort and puts a new crack growth prediction model for metals into the discussion. Experimental evidence obtained on the aluminium alloy Al 7475-T7351 are generalised to result in a new fatigue crack growth prediction model for variable amplitude loading. The difference to other models is that the central parameter of this model is the crack propagation stress intensity factor, K_{PR}. This is the stress intensity factor at which the crack starts to propagate during the loading part of a cycle. The general description of the relatively simple loading sequences in terms of K_{PR} leads to a scheme that is denoted as the "Fatigue Crack Growth Map." This map defines all possible crack growth conditions and is the heart of the crack growth prediction method. The Fatigue Crack Growth Map is generated for different alloys and temperatures. It is shown that any arbitrary loading spectrum consists only of three different types of load cycles. The model determines K_{PR} cycle by cycle throughout a loading spectrum, which determines the driving force, ΔK_{eff}, for the following loading cycle. The model requires the knowledge of the yield strength and the intrinsic threshold value, ΔK_T, for the material, as well as five different functions that also depend on the material. Phenomenologically, the model accounts for residual compressive stresses ahead of the crack front, the *intrinsic* response of the material, crack closure, which is an *extrinsic* effect, and the *intrinsic* threshold value ΔK_T. A few variable amplitude loading sequences are presented to compare predictions of K_{PR} with experiments. The results are in encouraging agreement.

In the following the viewgraphs of this presentation are listed. The interested reader will find a more detailed description of the model in [7,8]. Also References [9] and [10] will be needed, as well as [11] for more evidence on the Fatigue Crack Growth Map.

1. *Proceedings of the first joint DoD/FAA/NASA Conference on Aging Aircraft*. 1997, Ogden, Utah.
2. Various Authors, Aging of U.S. Air Force Aircraft. *Final Report of the Committee on Aging U.S. Air Force Aircraft, National Materials Advisory Board, Commission on Engineering and Technical Systems, National Research Council*, Publication NMAB-488-2, National Academic Press, Washington, D.C., USA., 1997.
3. An Assessment of Fatigue Damage and Crack Growth Prediction Techniques, *AGARD Report 797*, 1994.
4. Application of Damage Tolerance Principles for Improved Airworthiness of Rotorcraft, *RTO-MP-24*, Feb. 2000.
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6. J. Willenborg, R.M. Engle, H.A. Wood (1971) A crack growth retardation model using an effective stress concept. *AFFDL-TM-71-1-FBR*, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, USA.
7. M. Lang, A Model for Fatigue Crack Growth, Part I: Phenomenology, Accepted fo Publication in *Fatigue and Fracture of Engineering Materials & Structures*, appears in 2000.
8. M. Lang, A Model for Fatigue Crack Growth, Part II: Modeling, Accepted for Publication in *Fatigue and Fracture of Engineering Materials & Structures*, appears in 2000.
9. M. Lang, Description of load interaction effects by the ΔK_{eff}-concept. *Advances in Fatigue Crack Closure Measurement and Analysis, ASTM STP 1343*, American Society for Testing and Materials, 207-223, 1999.
10. M. Lang and X. Huang, The influence of compressive loads on fatigue crack propagation. *Fatigue and Fracture of Engineering Materials & Structures*, Vol. 21, No. 1, 1998, 65-83.
11. M. Lang, J.M. Larsen, Fatigue crack propagation and load interaction effects in a titanium alloy. *Fatigue and Fracture: 30th Volume, ASTM STP 1360,* American Society for Testing and Materials, 201-213, 2000.

The Basics of a Model for Fatigue Crack Growth in Helicopter Components

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Workshop on Damage Tolerance in Helicopters, Cranfield University,
Cranfield, Bedfordshire MK 430AL, England, on -4-5 April 2000.

IABG

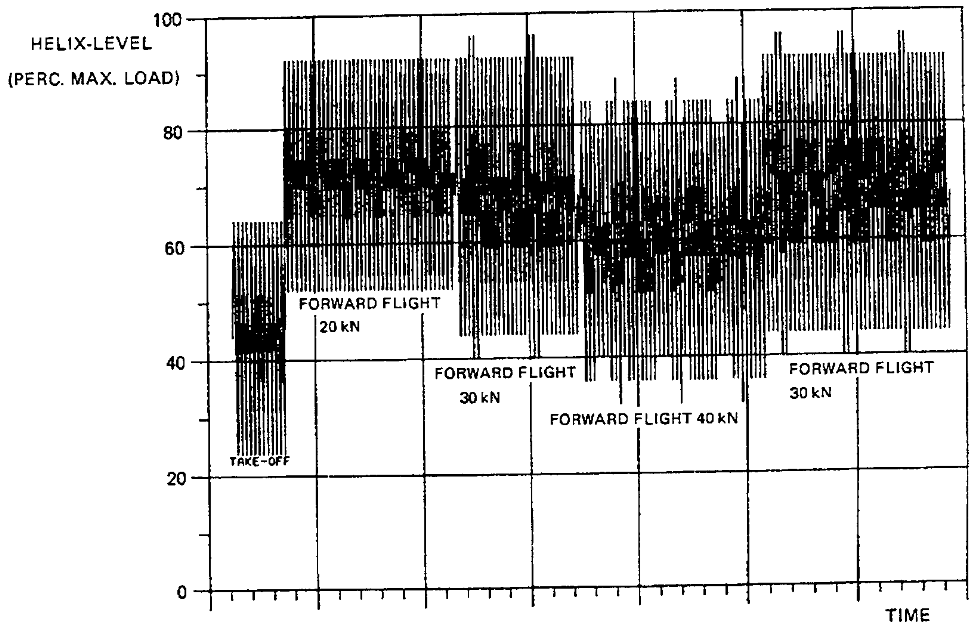
Damage Tolerant Design & Maintenance

- Critical Location
- Crack Shape, K-Solutions
- Size of Initial Failure and Distribution
- Loading Spectrum
- **Fatigue Crack Growth Prediction**
- Confidence
- Corrosion, Temperature, Humidity
- Maintenance / Inspection Procedure (NDT)

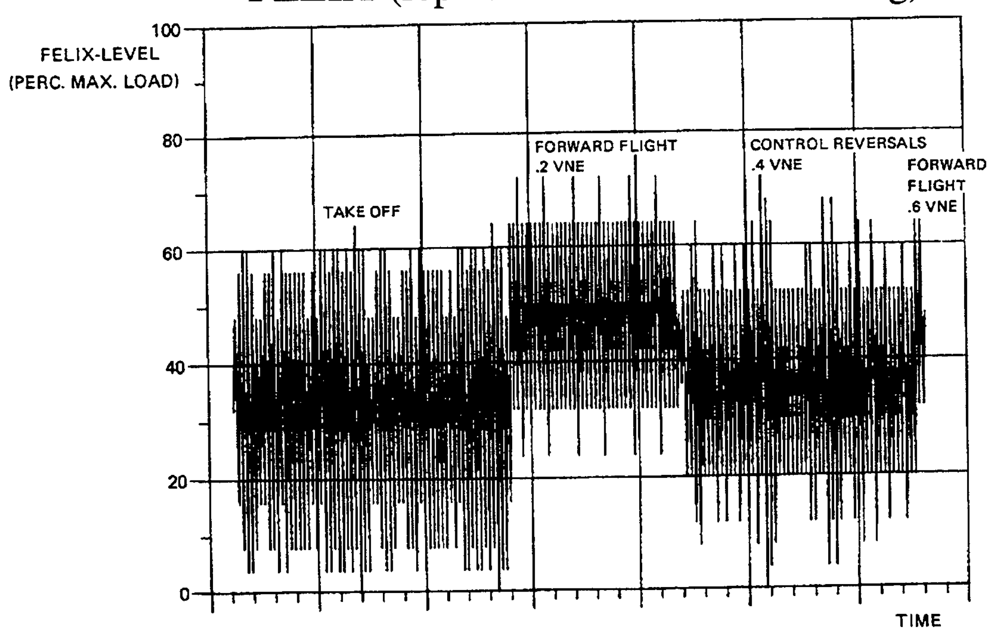
Inspection Procedures (Intervals and NDE Tools)

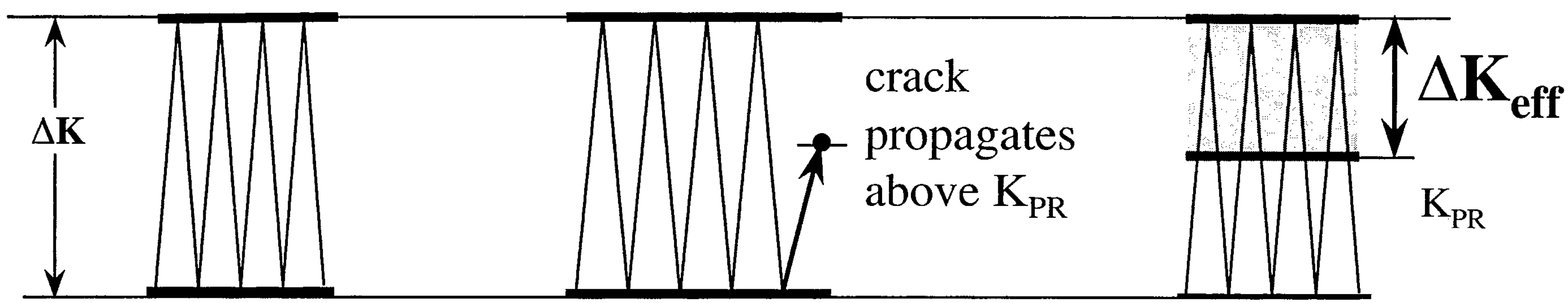
Probability of Failure,

HELIX (represents first 90 sec. Training)

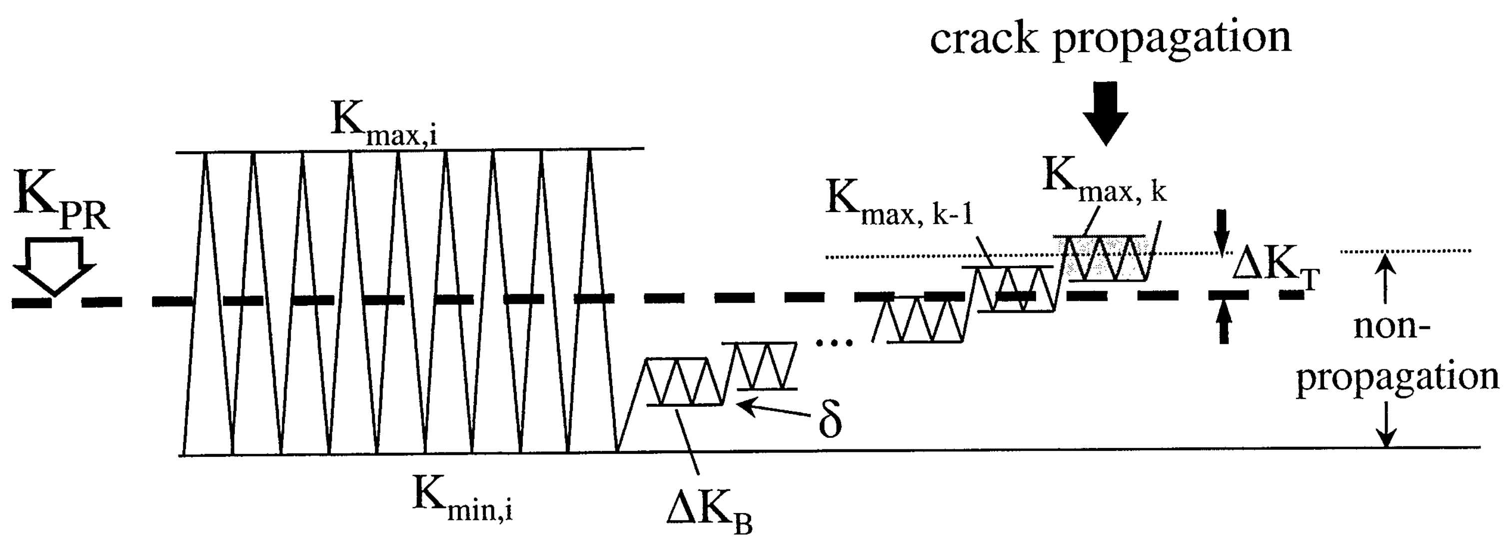


FELIX (represents first 90 sec. Training)

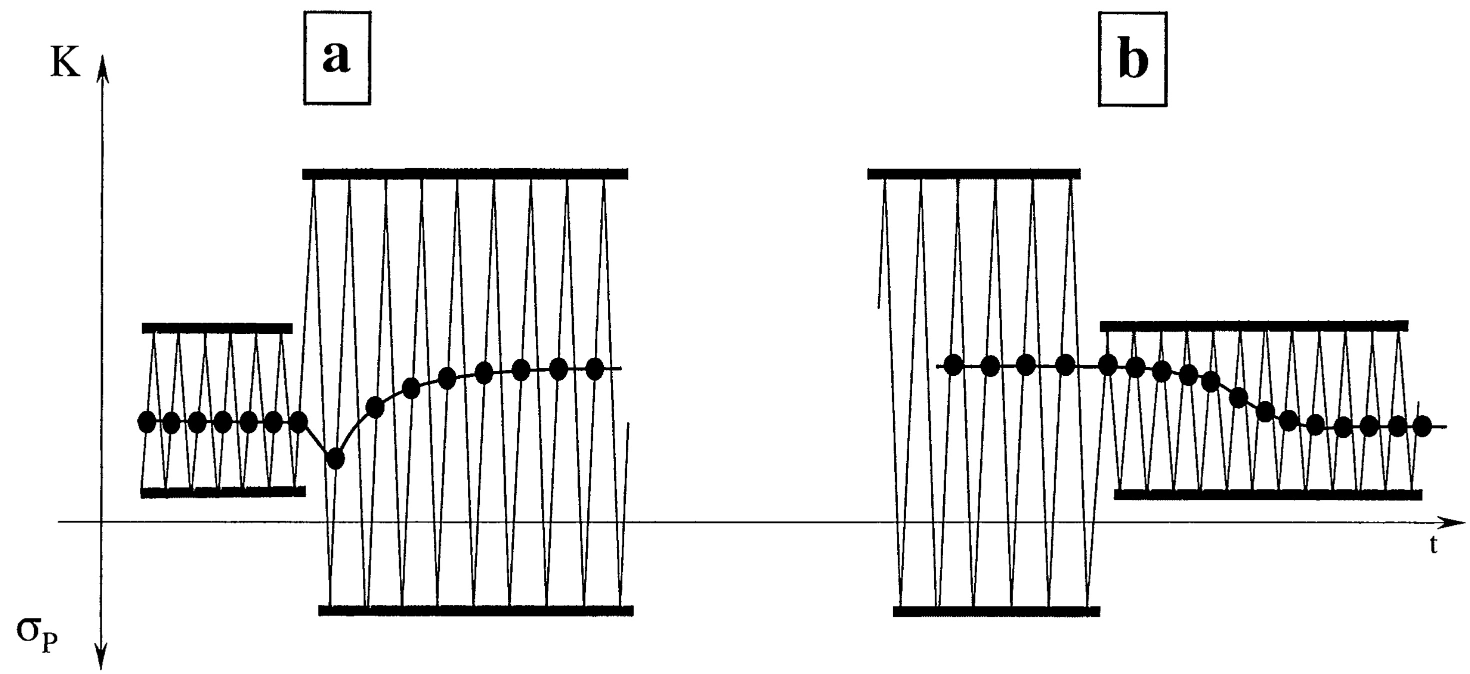



ΔK
crack
propagates
above K_{PR}
ΔK_{eff}
K_{PR}

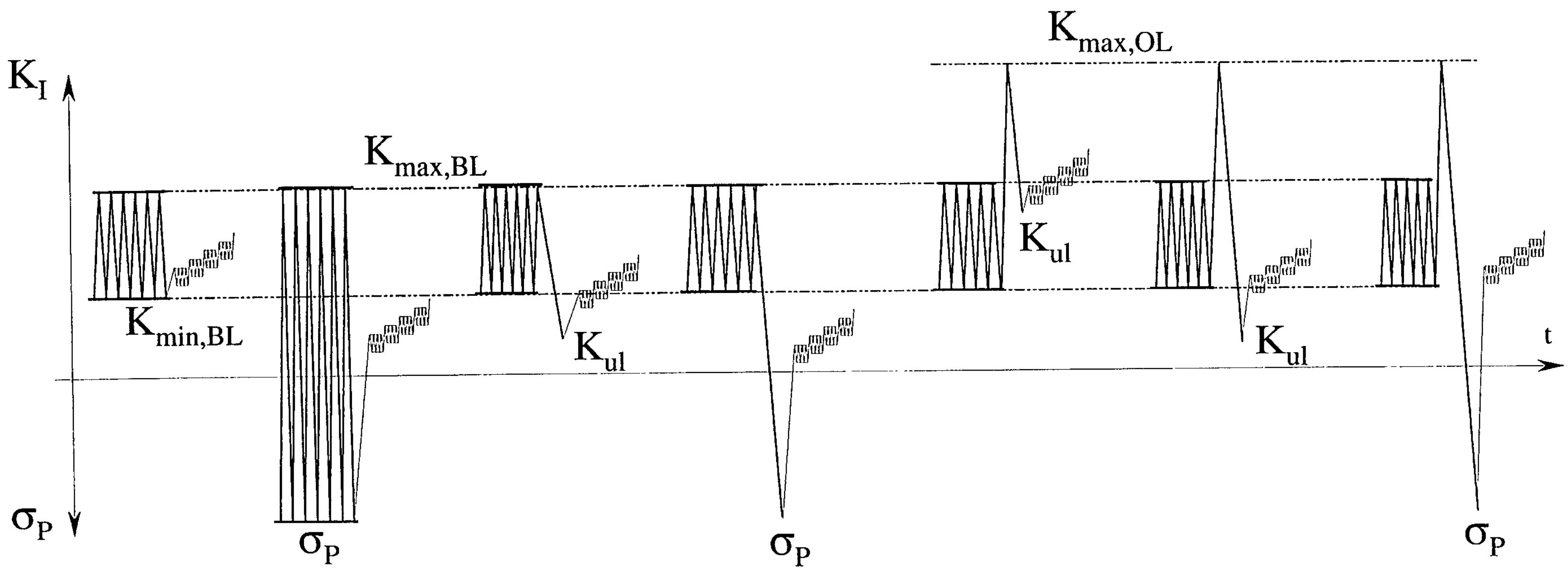
Crack Propagation Load Measurement Method (CPLM)



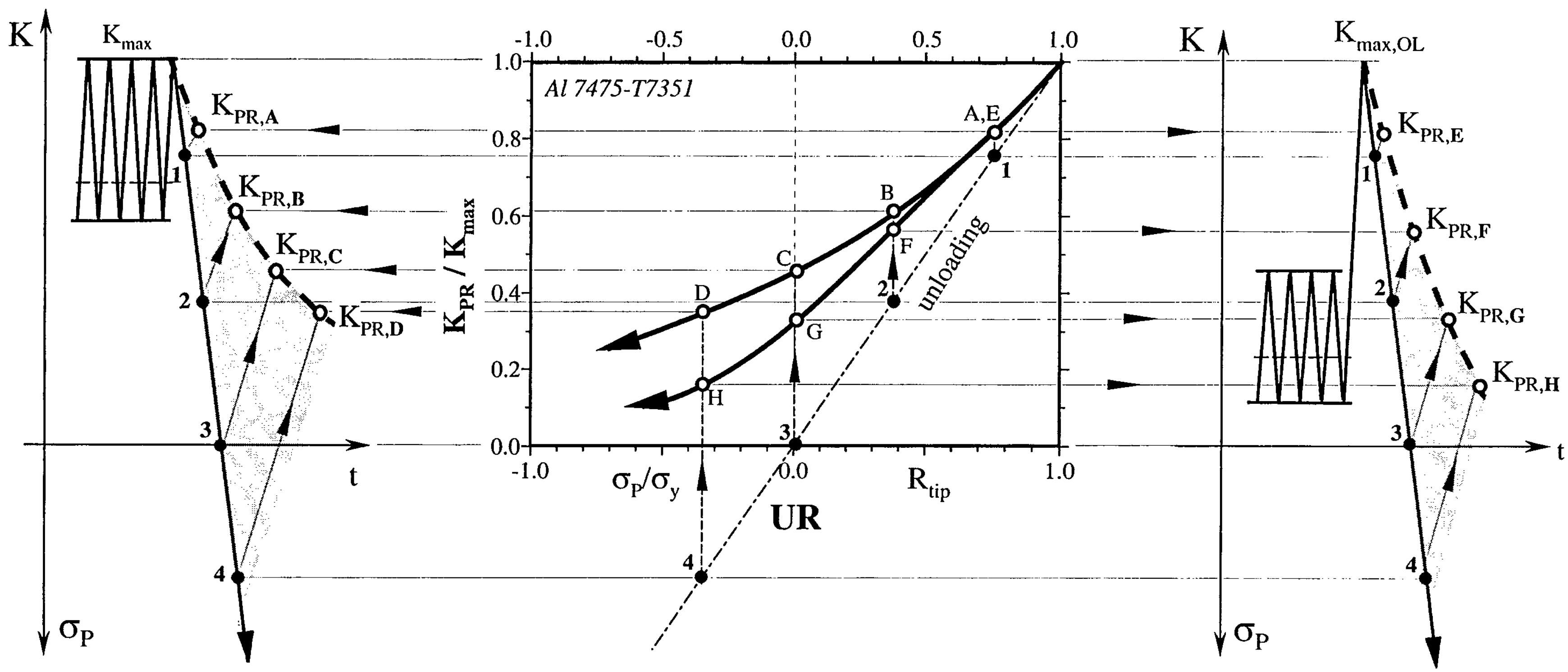
$$K_{PR} = \frac{\left(K_{\max,k} + K_{\max,k-1}\right)}{2} - \Delta K_T$$



a
b
K
σ_P
t



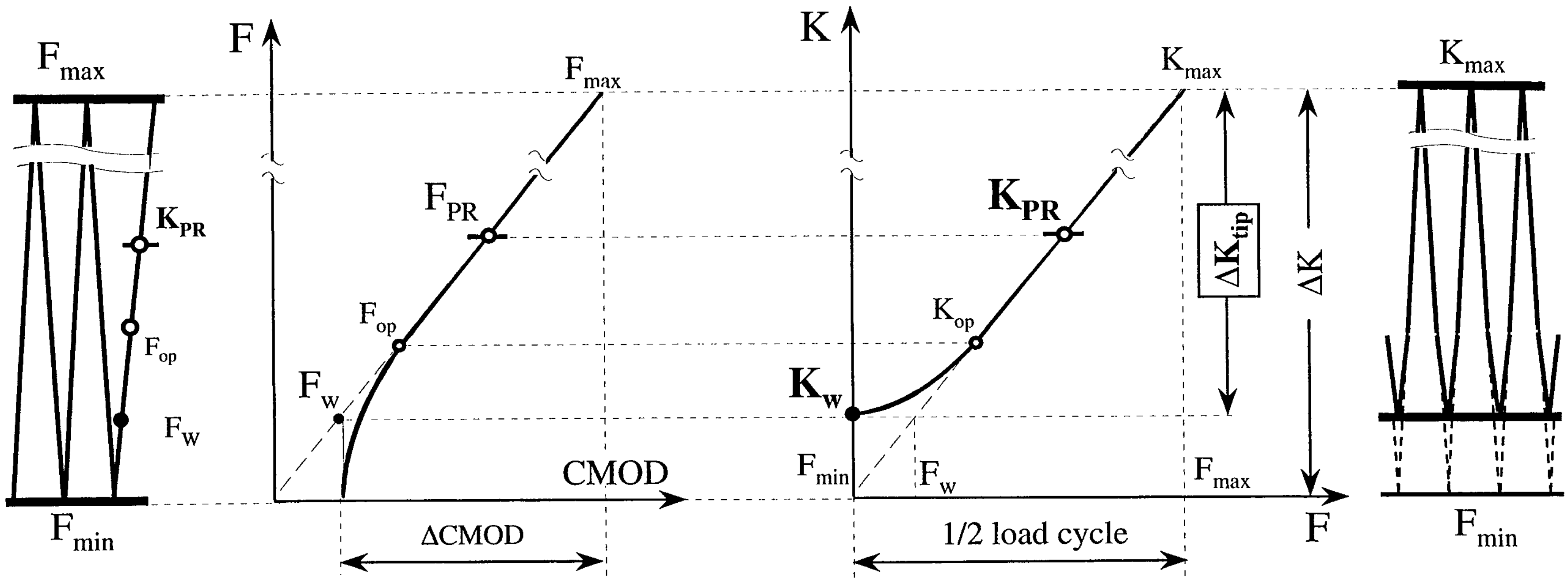
K_I
$K_{max,OL}$
$K_{max,BL}$
$K_{min,BL}$
K_{ul}
K_{ul}
K_{ul}
t
σ_P
σ_P
σ_P
σ_P



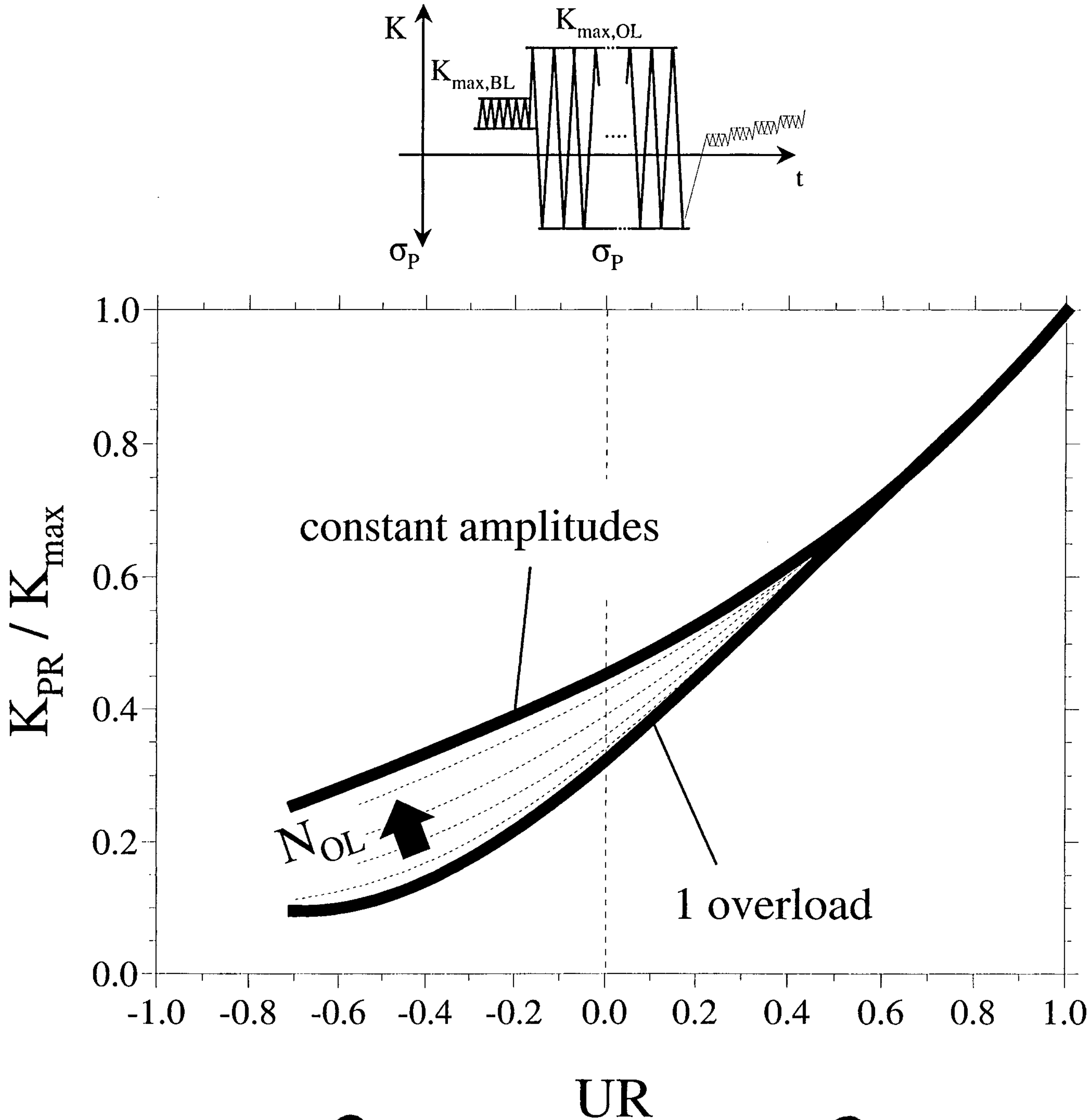
$$K_{PR,C} = \left(0.453 + 0.34 \cdot UR + 0.134 \cdot UR^2 + 0.07 \cdot UR^3\right) \cdot K_{\max}$$

$$K_{PR,1OL} = \left(0.322 + 0.57 \cdot UR + 0.23 \cdot UR^2 - 0.145 \cdot UR^3\right) \cdot K_{\max,OL}$$

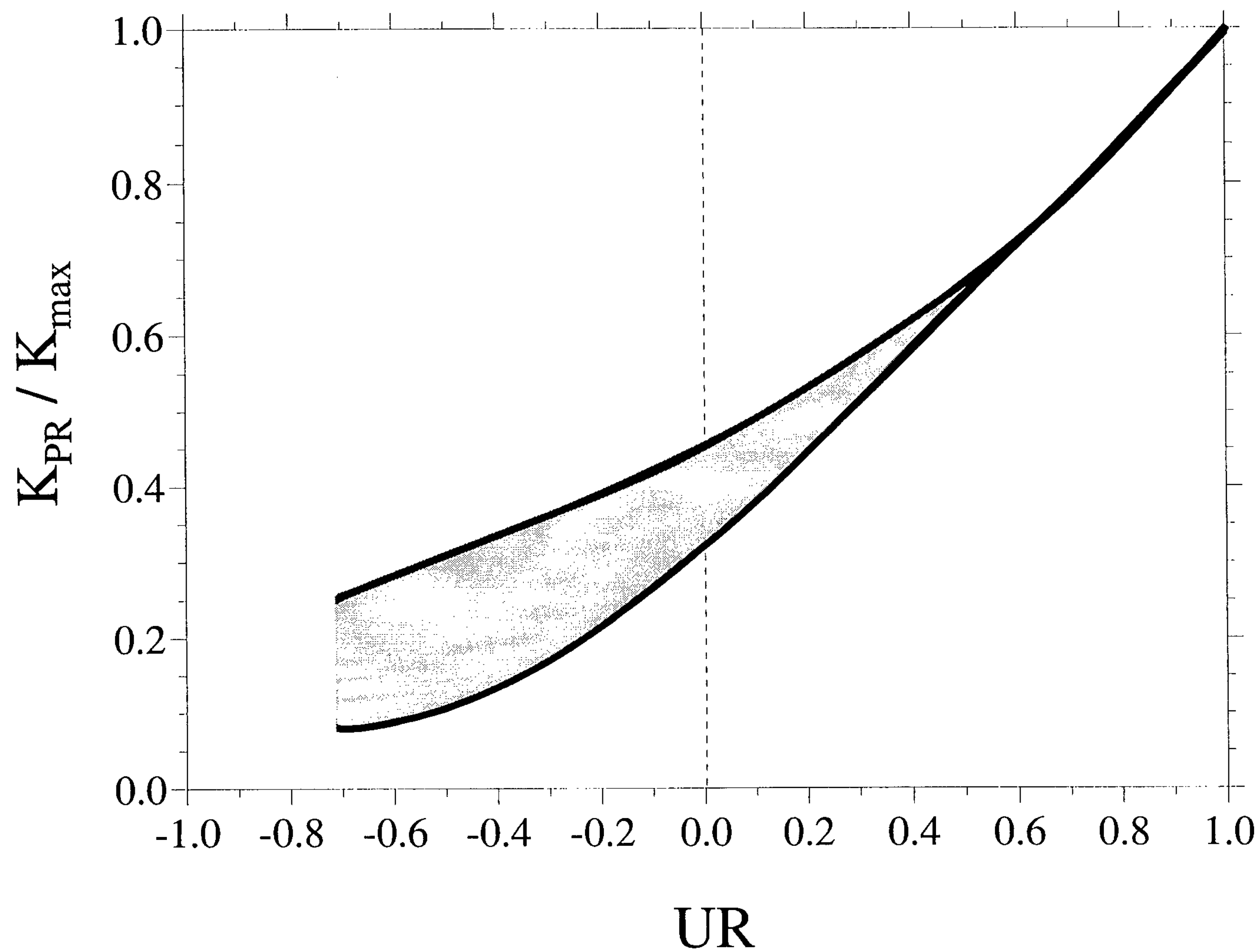
$$UR = R_{tip} = \frac{K_w}{K_{\max}} \qquad UR = \frac{\sigma_P}{\sigma_y}$$



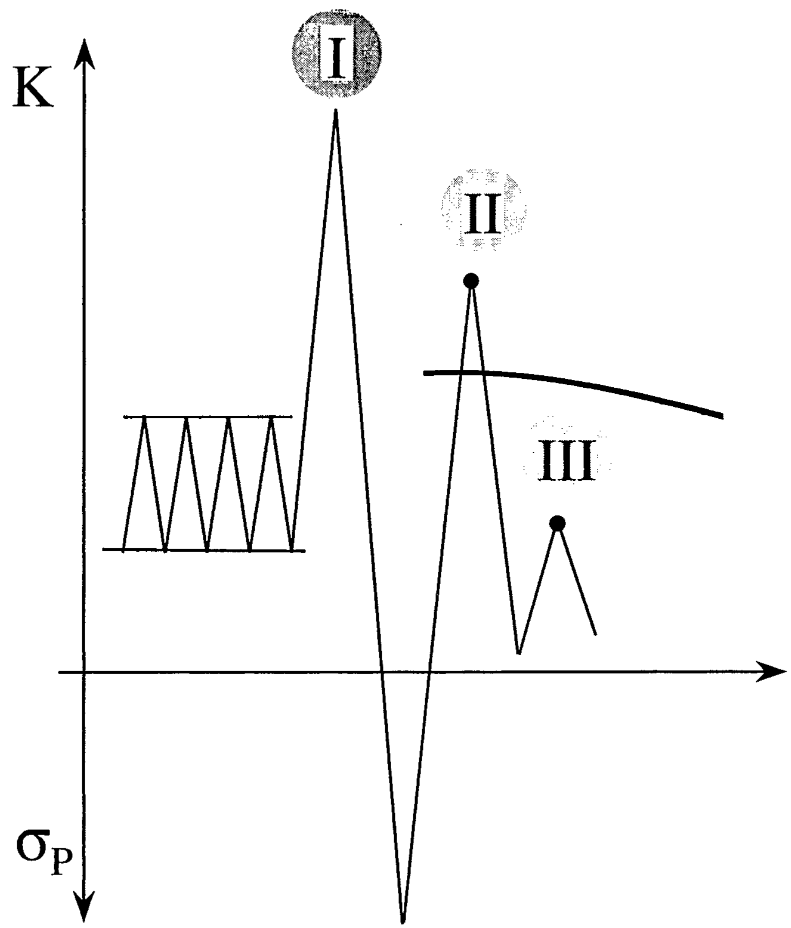
$$\Delta K_{tip} = K_{\max} - K_W$$


K
K max,BL
K max,OL
t
σP
σP
constant amplitudes
NOL
1 overload
KPR / Kmax
UR
1.0
0.8
0.6
0.4
0.2
0.0
-1.0
-0.8
-0.6
-0.4
-0.2
0.0
0.2
0.4
0.6
0.8
1.0

Decisive Crack Propagation Conditions



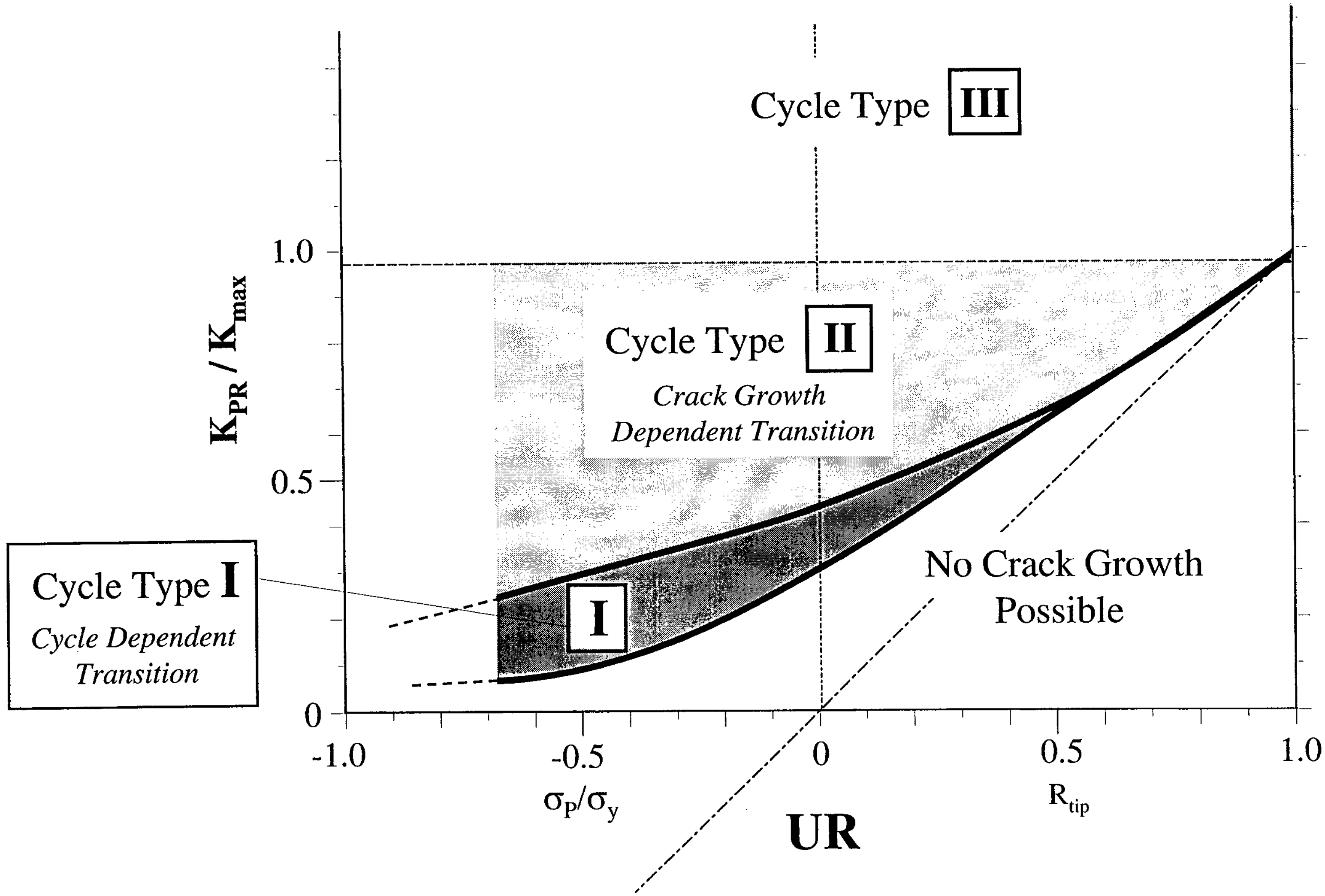
The Three Possible Types of Cycles



I $K_{PR,OL} \le \left(K_{PR} / K_{\max}\right)_I \le K_{PR,C}$

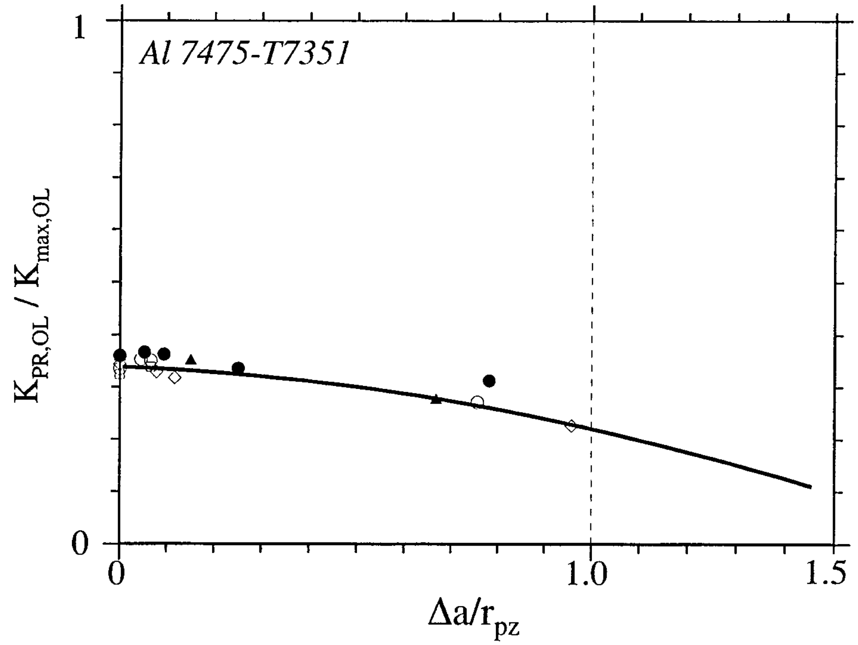
II $K_{PR,C} \le \left(K_{PR} / K_{\max}\right)_{II} \le 1$

III $\left(K_{PR} / K_{\max}\right)_{III} > 1$



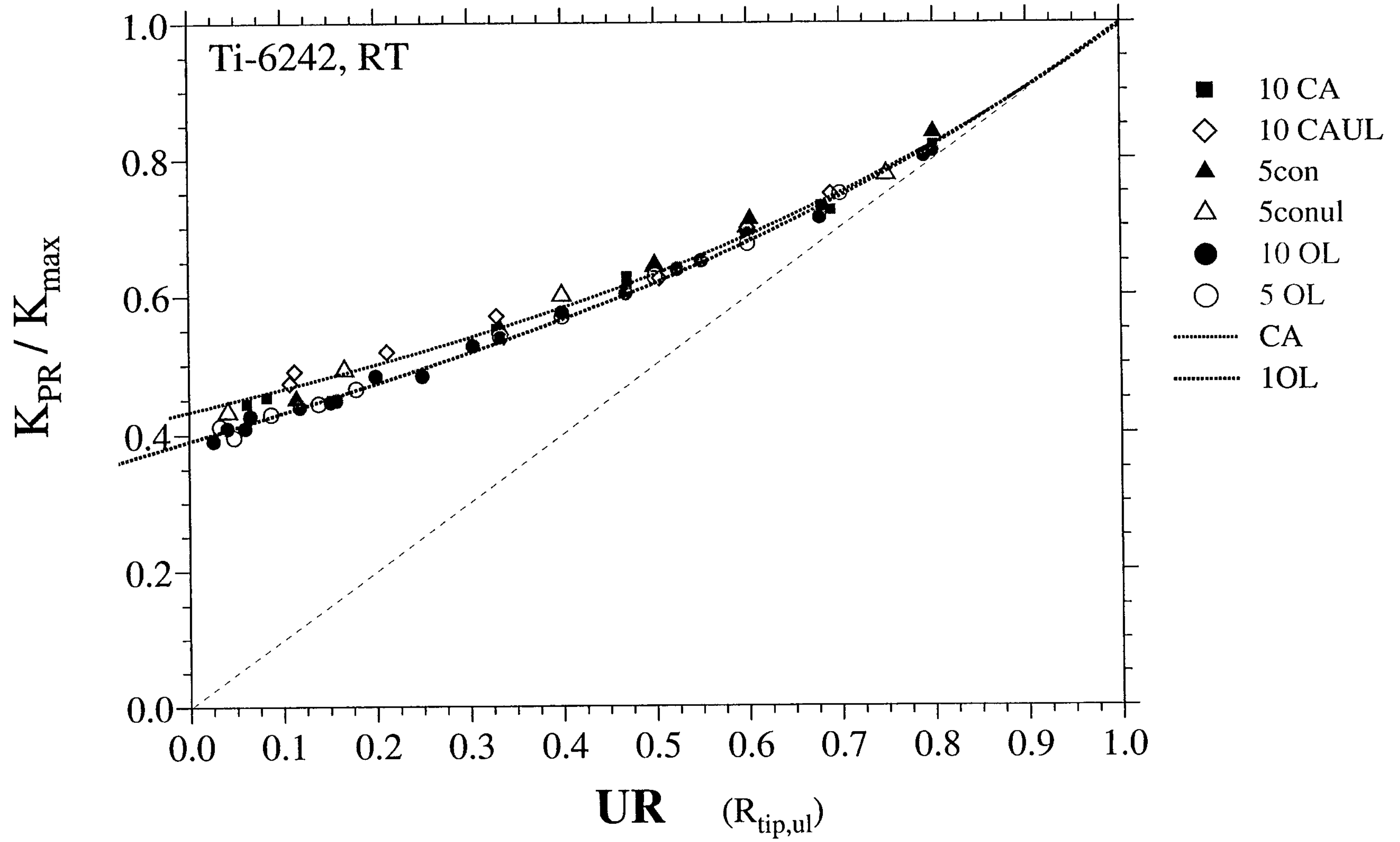
Cycle Type III
Cycle Type II
Crack Growth Dependent Transition
Cycle Type I
Cycle Dependent Transition
I
No Crack Growth Possible
K_{PR} / K_{max}
1.0
0.5
0
-1.0
-0.5
0
0.5
1.0
σ_P/σ_y
R_{tip}
UR

Decline Curve of K_{PR}



$$K_{PR,\,decl.} = \left[\frac{K_{PR,I}}{K_{\max,I}} - \left(l \cdot \left(\frac{\Delta a}{r_{pz}} \right) + m \cdot \left(\frac{\Delta a}{r_{pz}} \right)^2 \right) \right] \cdot K_{\max,I}$$

$$r_{pz} = \frac{2}{3\pi} \cdot \left(\frac{K_{\max} - K_w}{\sigma_y} \right)^2$$



$$K_{PR,C} = \left(0.433 + 0.311 \cdot UR + 0.09 \cdot UR^2 + 0.16 \cdot UR^3\right) \cdot K_{max}$$

[0≤UR≥1]

$$K_{PR,1OL} = \left(0.391 + 0.393 \cdot UR + 0.03 \cdot UR^2 + 0.184 \cdot UR^3\right) \cdot K_{max,OL}$$

$$\underline{\Delta K_{eff} = K_{max} - K_{PR} - \Delta K_T}$$

1 2 3

$$\boxed{\Delta K_{eff} = \Delta K_{tip} - \Delta K_{RCS} - \Delta K_T}$$

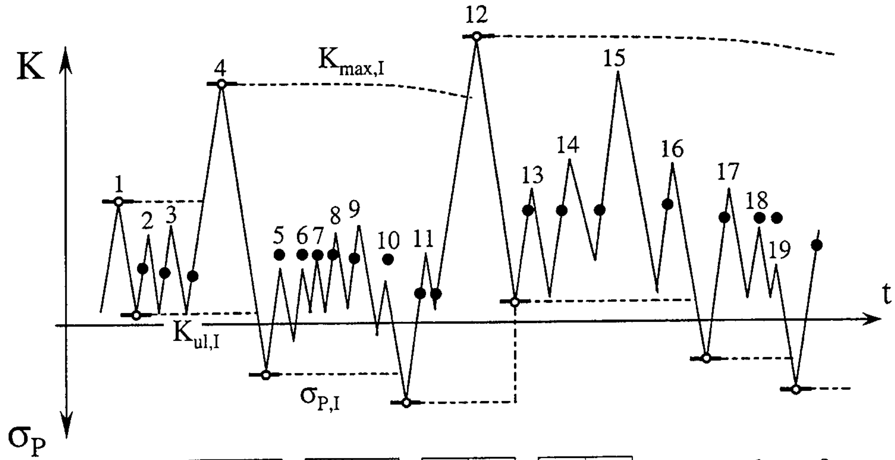
1. **ΔK_{tip}** - Crack closure (crack face contact behind the crack tip) modifies the loading conditions by reducing ΔK to ΔK_{tip}. $\longrightarrow$ $\boxed{\Delta K_{tip} = K_{max} - K_w}$

2. **ΔK_{RCS}** - Local crack tip residual compressive stresses, accounted for by ΔK_{RCS} contribute to the reduction from ΔK to ΔK_{eff}. $\longrightarrow$

(tension)
$$\Delta K_{RCS} = K_{PR} - K_{min} = K_{PR} - K_w$$

(compression)
$$\Delta K_{RCS} = K_{PR}$$

3. **ΔK_T** - The material threshold value, ΔK_T. $\longrightarrow$ $\boxed{\Delta K_T}$



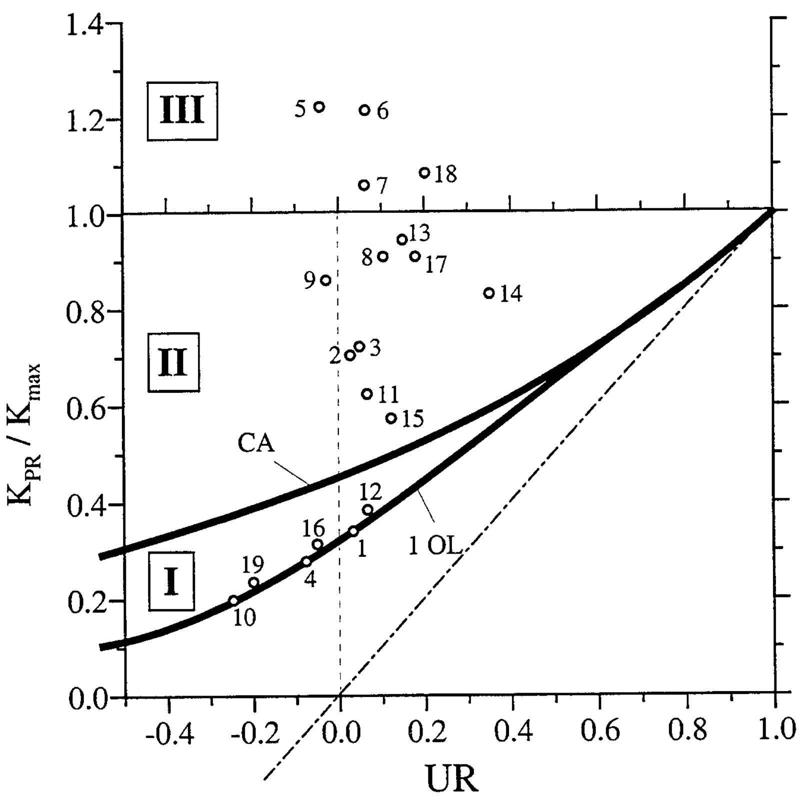
n	T
1	I
2	II
3	II
4	I
5	III

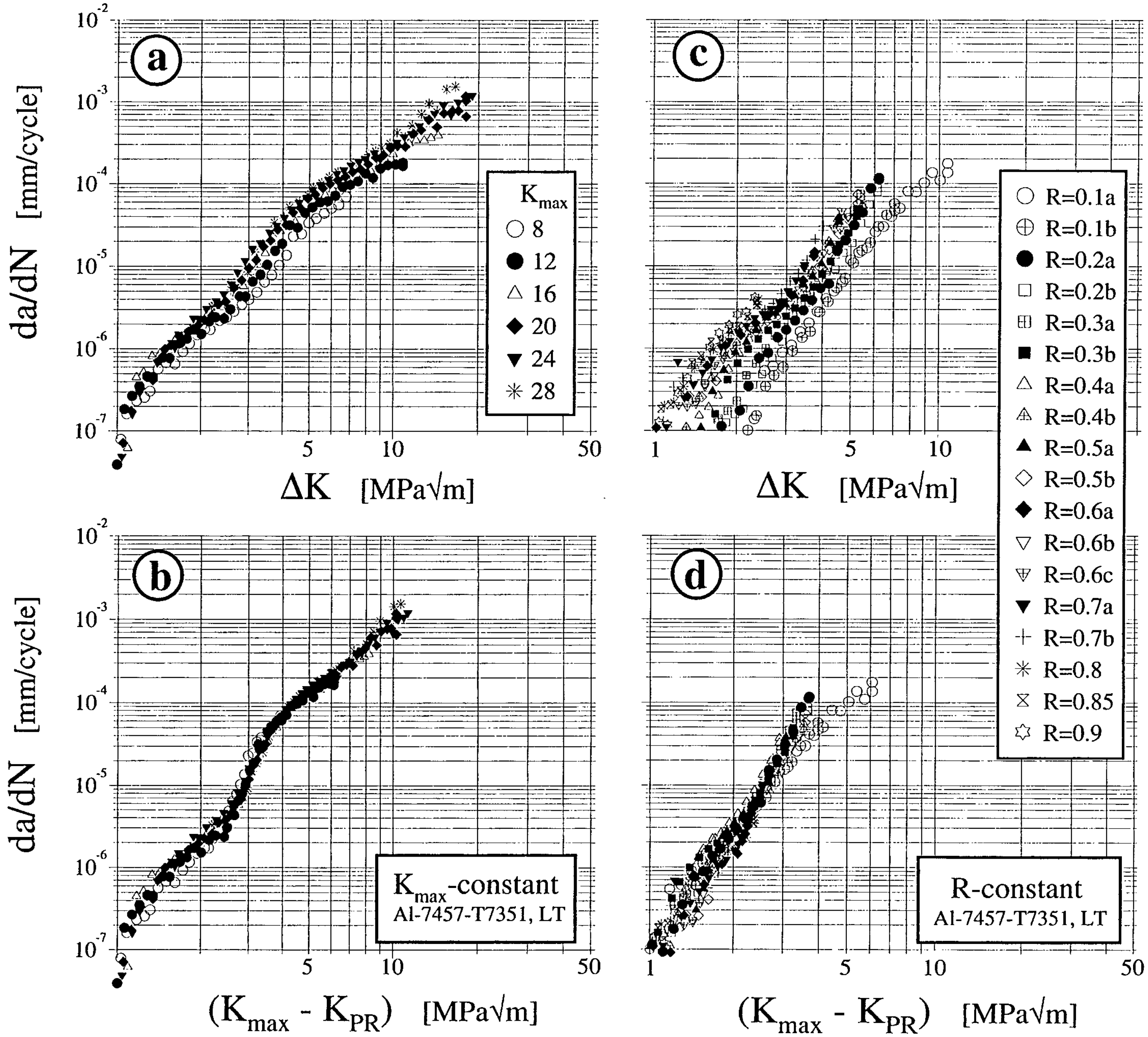
n	T
6	III
7	III
8	II
9	II
10	I

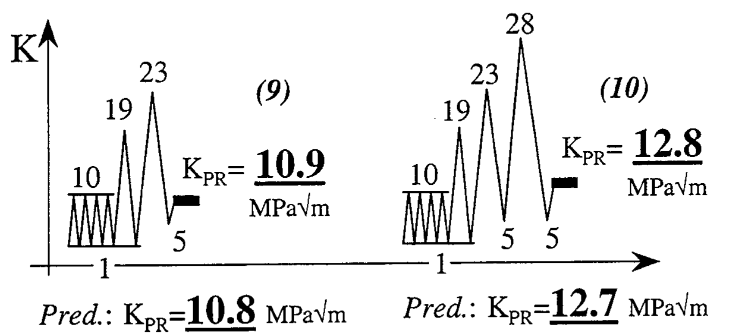
n	T
11	II
12	I
13	II
14	II
15	II

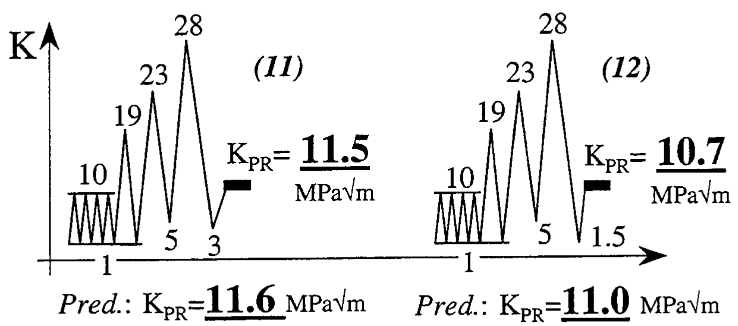
n	T
16	I
17	II
18	III
19	I

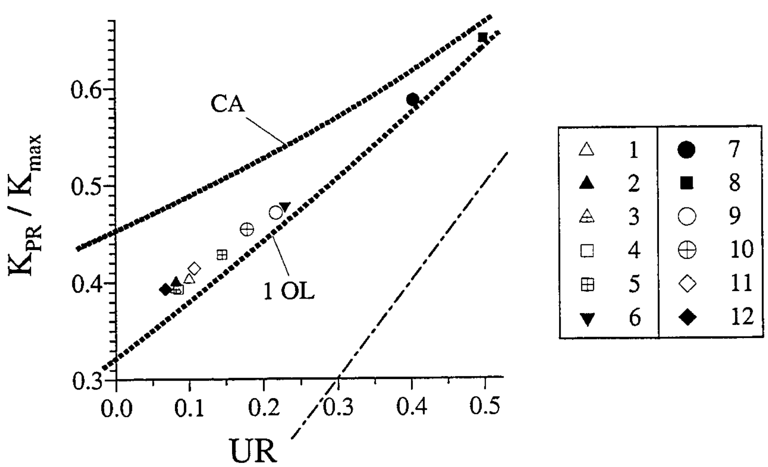
n = number of cycle in above sketch
T = Type of cycle

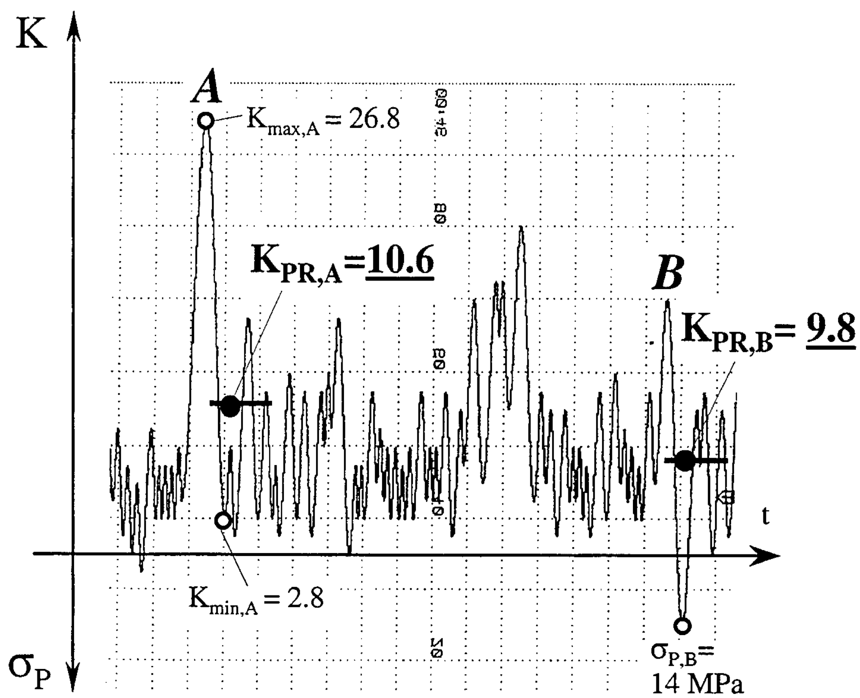



a
b
c
d
da/dN [mm/cycle]
ΔK [MPa√m]
(Kmax - KPR) [MPa√m]
Kmax
8
12
16
20
24
28
R=0.1a
R=0.1b
R=0.2a
R=0.2b
R=0.3a
R=0.3b
R=0.4a
R=0.4b
R=0.5a
R=0.5b
R=0.6a
R=0.6b
R=0.6c
R=0.7a
R=0.7b
R=0.8
R=0.85
R=0.9
Kmax-constant
Al-7457-T7351, LT
R-constant
Al-7457-T7351, LT


K
10
19
23
(9)
K_{PR}= 10.9
MPa√m
5
1
Pred.: K_{PR}=10.8 MPa√m
10
19
23
28
(10)
K_{PR}= 12.8
MPa√m
5
5
1
Pred.: K_{PR}=12.7 MPa√m


K
10
19
23
28
(11)
K_{PR}= 11.5
MPa√m
5
3
1
Pred.: K_{PR}=11.6 MPa√m
10
19
23
28
(12)
K_{PR}= 10.7
MPa√m
5
1.5
1
Pred.: K_{PR}=11.0 MPa√m


K_{PR} / K_{max}
0.3
0.4
0.5
0.6
0.0
0.1
0.2
0.3
0.4
0.5
UR
CA
1 OL
1
2
3
4
5
6
7
8
9
10
11
12



Calculation of $K_{PR,A}$	***Calculation of $K_{PR,B}$***
$K_{max,OL}$= 26.8 MPa√m, K_w=2.8MPa√m, $UR = R_{tip,ul}$ = 0.1045, N_{OL}=1.72, $K_{PR}/K_{max,OL}$ = 0.403, $K_{PR,A}$ = **<u>10.8</u>** MPa√m	UR = -0.03, $K_{max,OL}$= 26.8 MPa√m, N_{OL}=1.72 (from cycle A) $K_{PR}/K_{max,OL}$ = 0.331 $K_{PR,B}$ = **<u>8.9</u>** MPa√m

Summary

- A **new crack growth prediction model** was presented
- Criterion is crack propagation , $\mathbf{K_{PR}}$
- The model accounts for **crack closure**, **residual compressive stresses**, intrinsic **threshold value** ΔK_T.
- Two type of transitions: **Cycle dependent** and **crack growth dependent transition**
- Only **three types of cycles** exist
- Necessary Input/Knowledge:
  1. Loading spectrum; K and σ_P.
  2. The correction of the loading conditions due to crack closure. (In many cases negligible).
  3. The knowledge of the **Fatigue Crack Growth Map** and the methodology.
  4. The definition of the driving force for fatigue crack growth given by

$$\Delta K_{eff} = K_{\max} - K_{PR} - \Delta K_T$$

5. Material dependent parameters and functions which are:

5.1. yield strength, σ_y,

5.2. the intrinsic threshold value, ΔK_T,

5.3. the two "master curves" for CA and 1OL

$$K_{PR} = f(UR) \cdot K_{\max}$$

5.4. the transitional function of K_{PR} between the two "master curves,"(**crack growth dependent transition**)

5.5 the decline function of K_{PR} (**cycle dependent transition**)

5.6 the function

$$da/dN = f\left(\Delta K_{eff}\right)$$

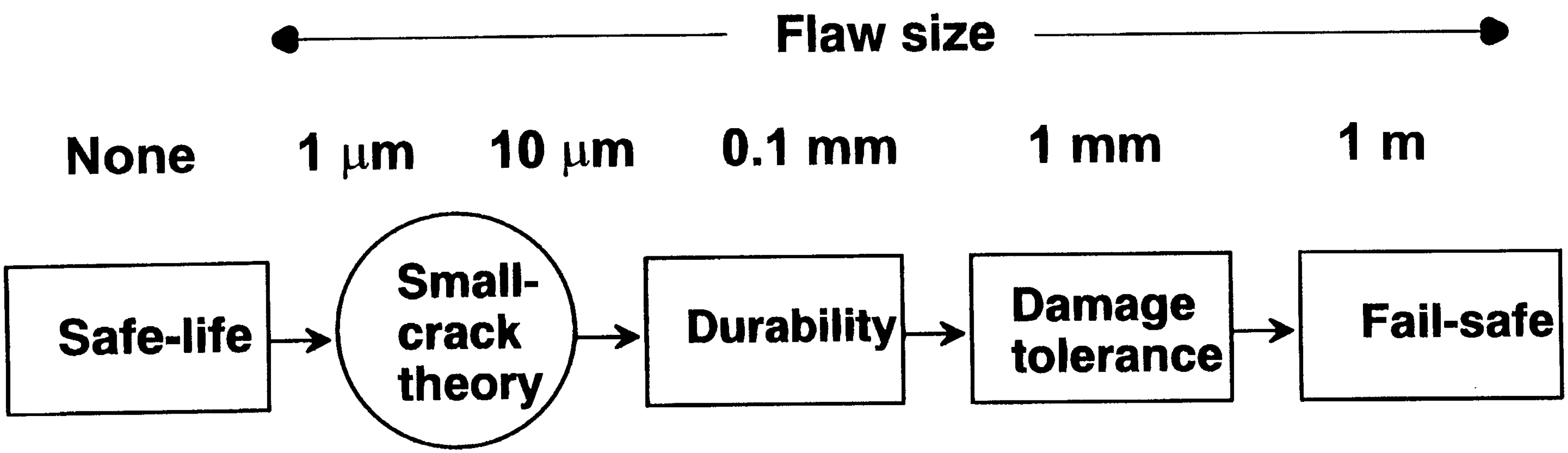
to calculate the crack growth increment *da* per cycle.

FATIGUE LIFE ANALYSIS OF 4340 STEEL WITH A MACHINE-LIKE SCRATCH (Using Small-Crack Theory)

R.A. Everett, Jr.
U.S. Army Vehicle Technology Directorate
NASA Langley Research Center
Hampton, Virginia USA

Workshop on Damage Tolerance in Helicopters
Cranfield University
Cranfield, United Kingdom
4-5, April 2000

DESIGN CONCEPT USING SMALL-CRACK THEORY



- Hidden or uninspectable structure
- Manufacturing- and service-induced damage
- EIFS to fit S-N or ε - N behavior

- Economic-life extension

- Inspectable flaw size

- Redundant structure

FAR 29.571

(b) F*atigue tolerance evaluation(including tolerance to flaws)*

(1) *Flaw tolerant safe-life evaluation*. It must be shown that the structure, with flaws present, is able to withstand repeated loads of variable magnitude <u>without detectable flaw growth</u> for the following time intervals ---

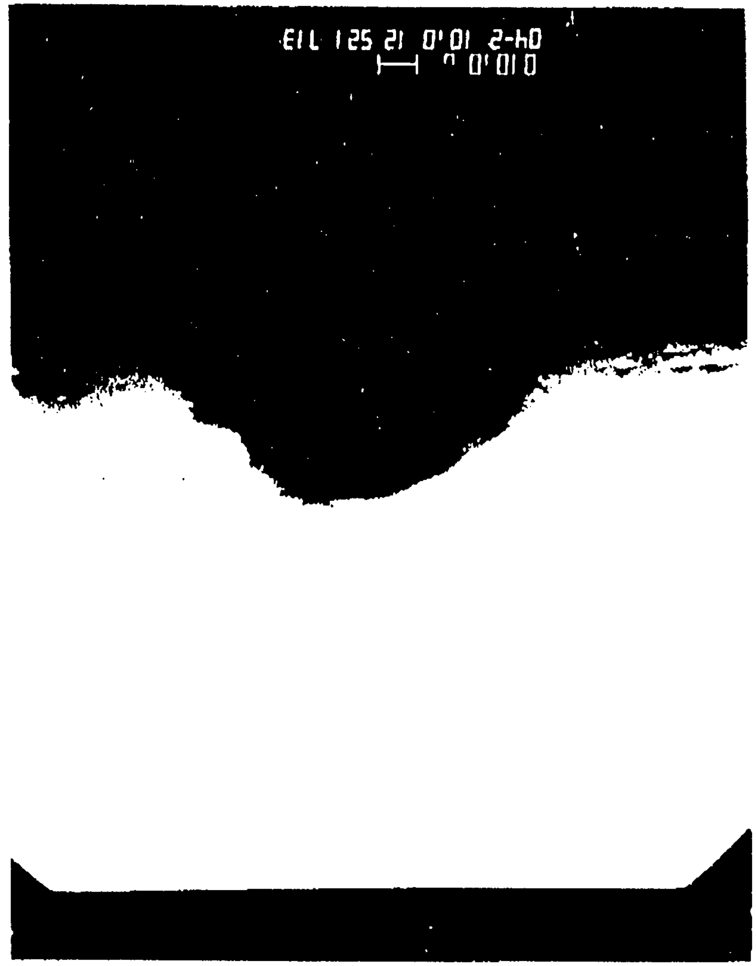
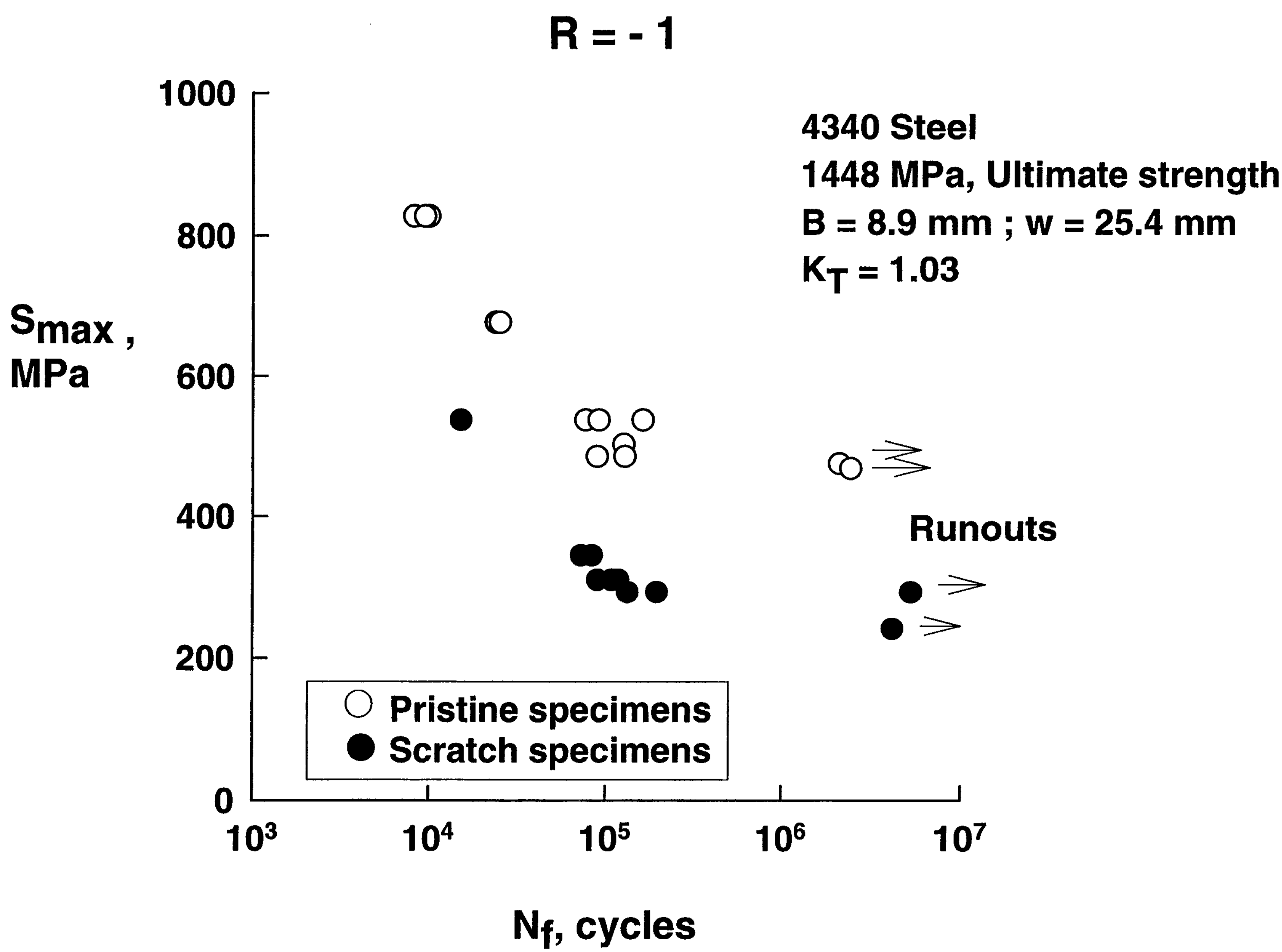
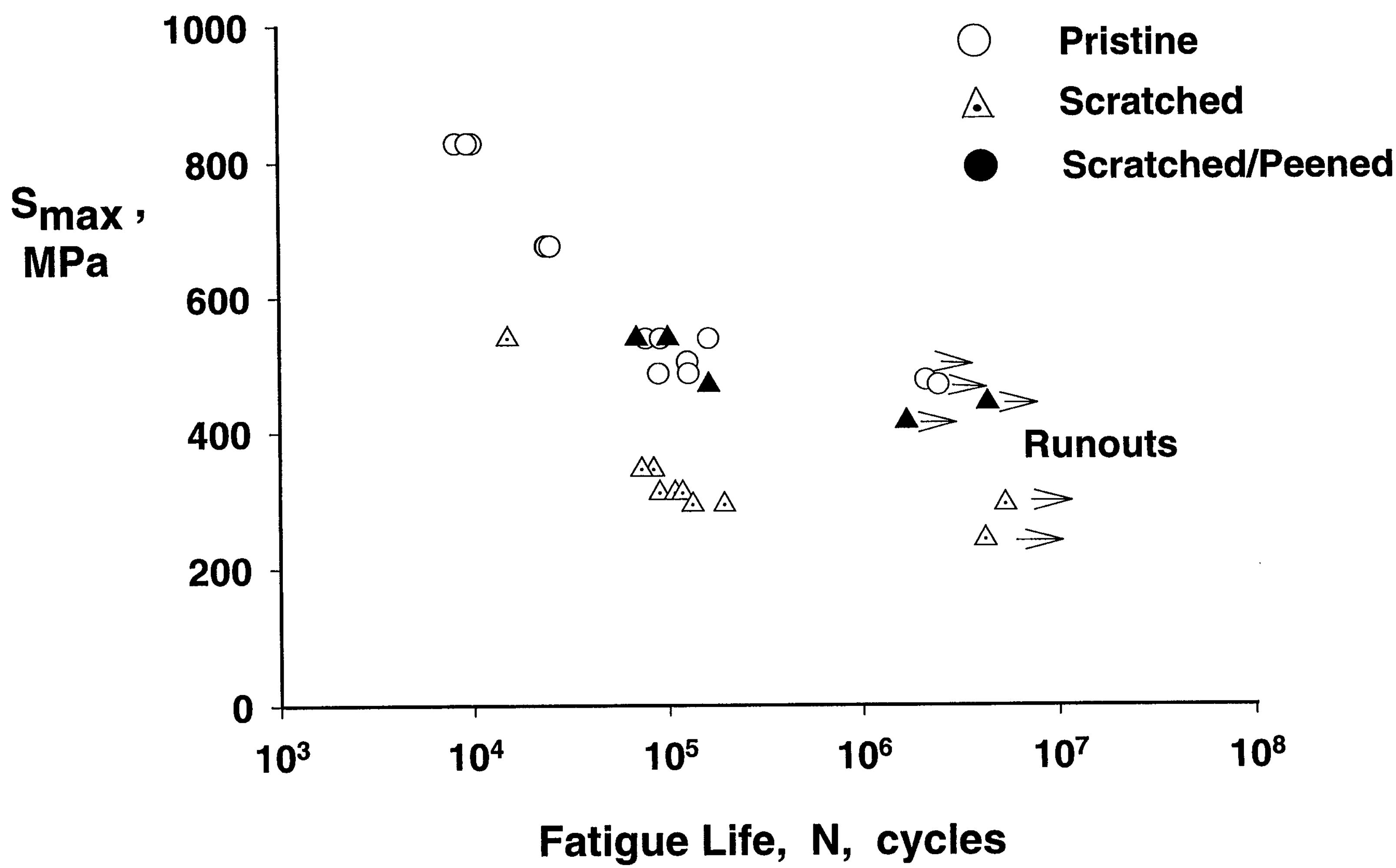


Fig. 6. Cross section of scratch with two small cracks (400X)

FATIGUE LIFE OF PRISTINE AND SCRATCH SPECIMENS (SCRATCH DEPTH = 0.05 mm)

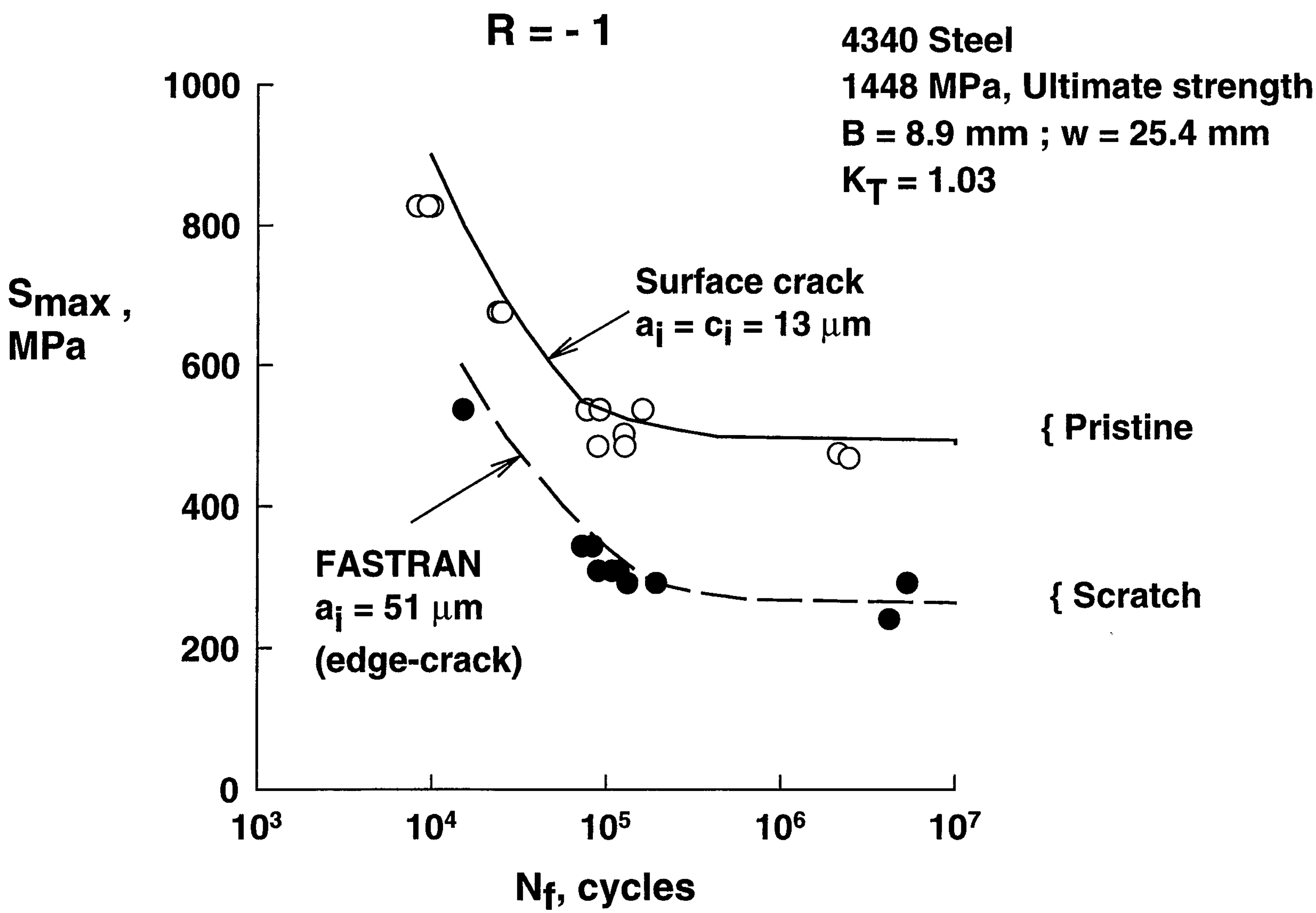


FATIGUE LIFE OF PRISTINE AND SCRATCH/PEENED SPECIMENS

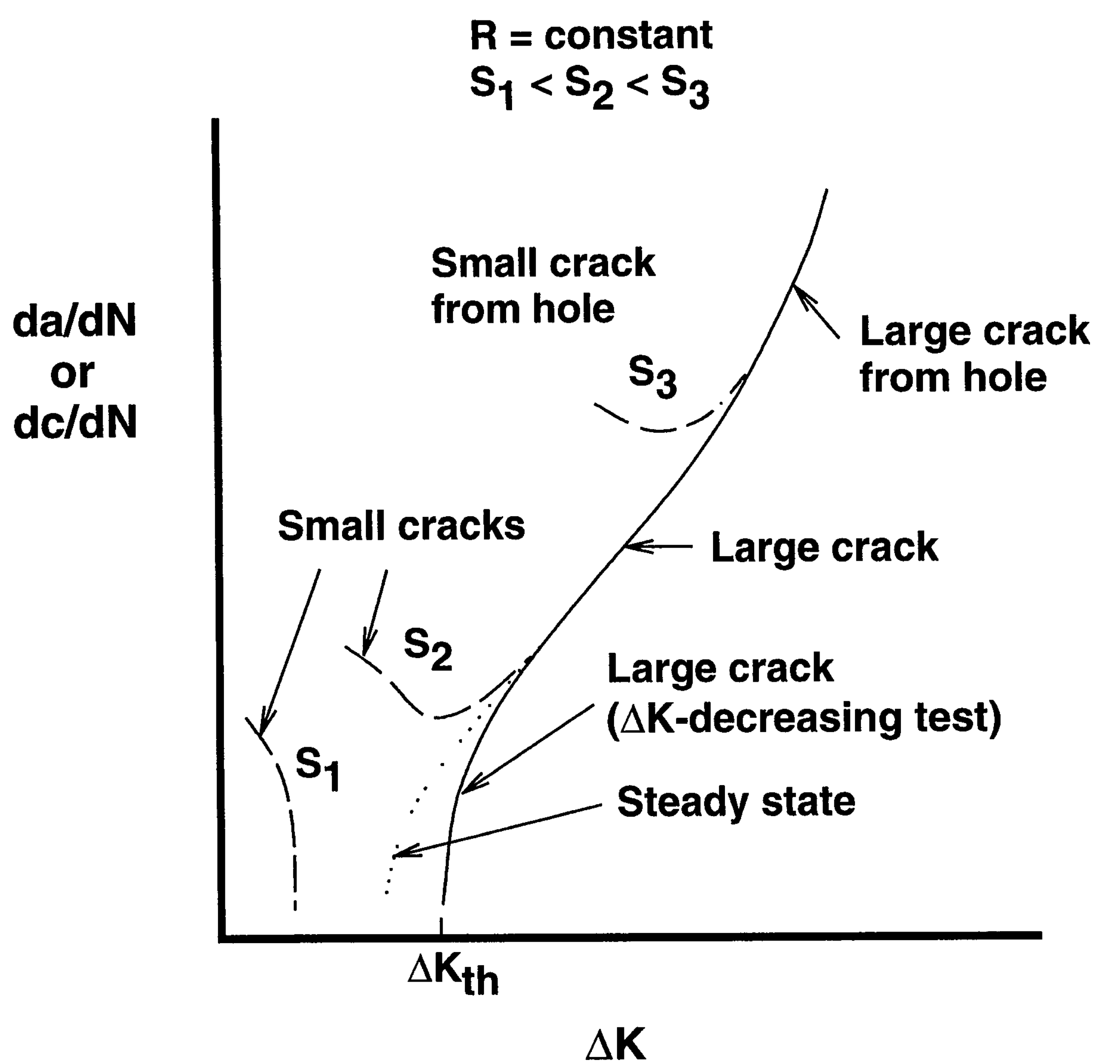


MEASURED AND PREDICTED FATIGUE LIVES FOR THE PRISTINE AND SCRATCHED SPECIMENS

(Scratch Depth = 0.05 mm)

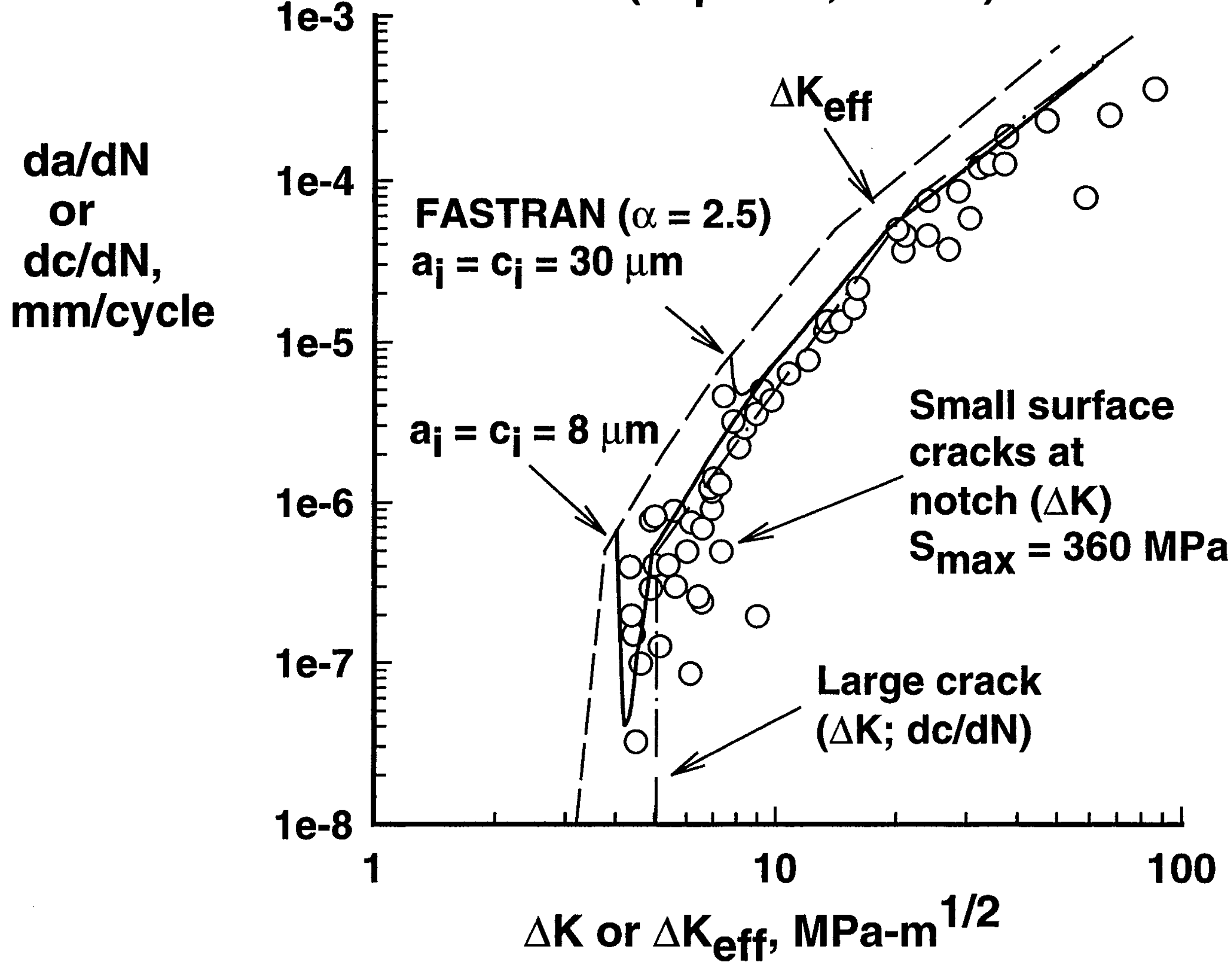


THE "SMALL-CRACK" EFFECT



MEASURED AND PREDICTED SMALL CORNER CRACK GROWTH AT A NOTCH IN 4340 STEEL

($K_T = 3.3$, $R = 0$)



AGARD–ARMY SUPPLEMENTAL SHORT CRACK PROGRAM ON EDGE-NOTCHED 4340 ALLOY STEEL

CRACK GROWTH MEASUREMENT ON NOTCH SURFACE FROM REPLICAS

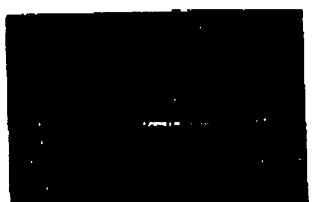
60,000 cycles

L = .04mm

130,000 cycles

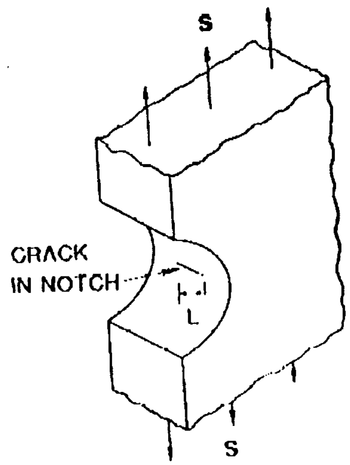
L = .15mm

160,000 cycles



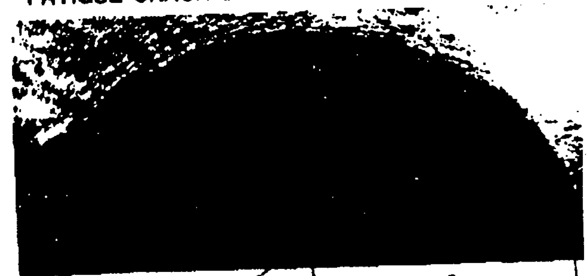
L = 2.8mm

FATIGUE SPECIMEN



FRACTURE SURFACE OF SPECIMEN

FATIGUE CRACK SHAPE AFTER 160,000 CYCLES

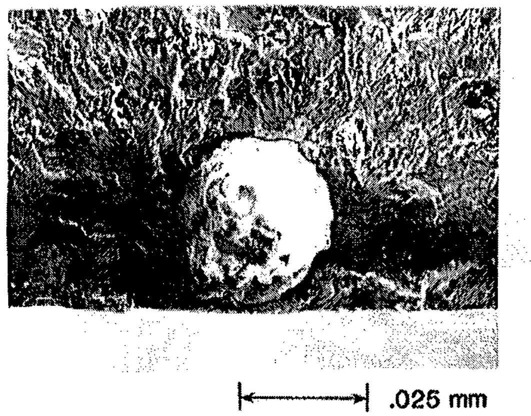


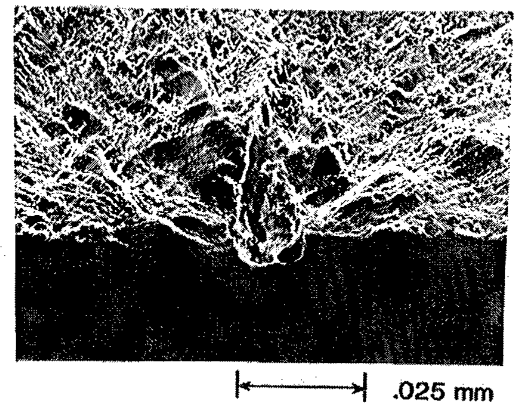
L = 2.8mm

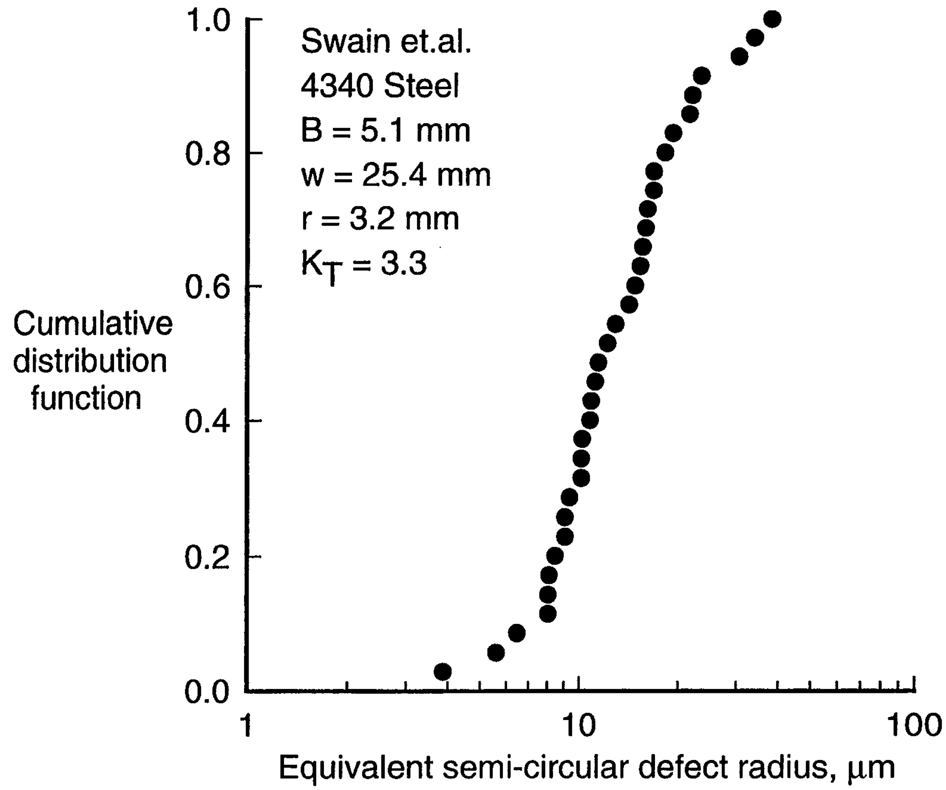
CRACK INITIATION SIT

.016mm

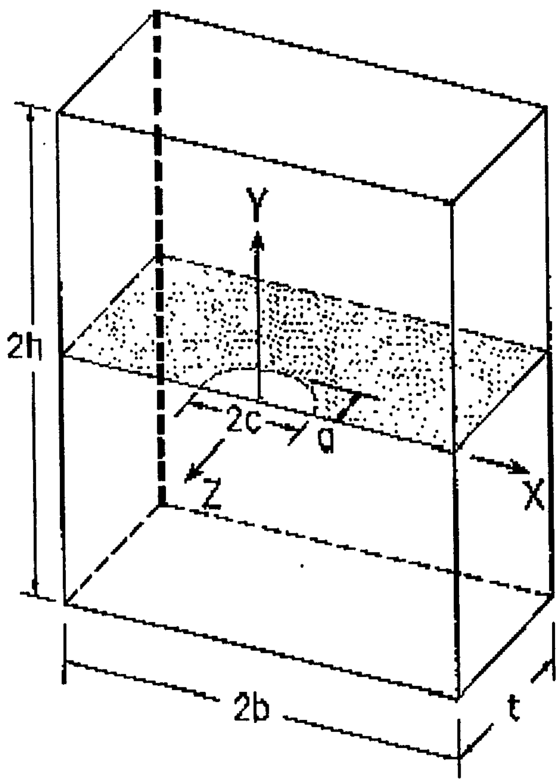
Cumulative Distribution Function for Initiation Sites in 4340 Steel



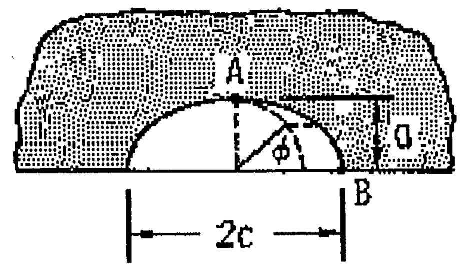




Analytical & Weight Function Methods

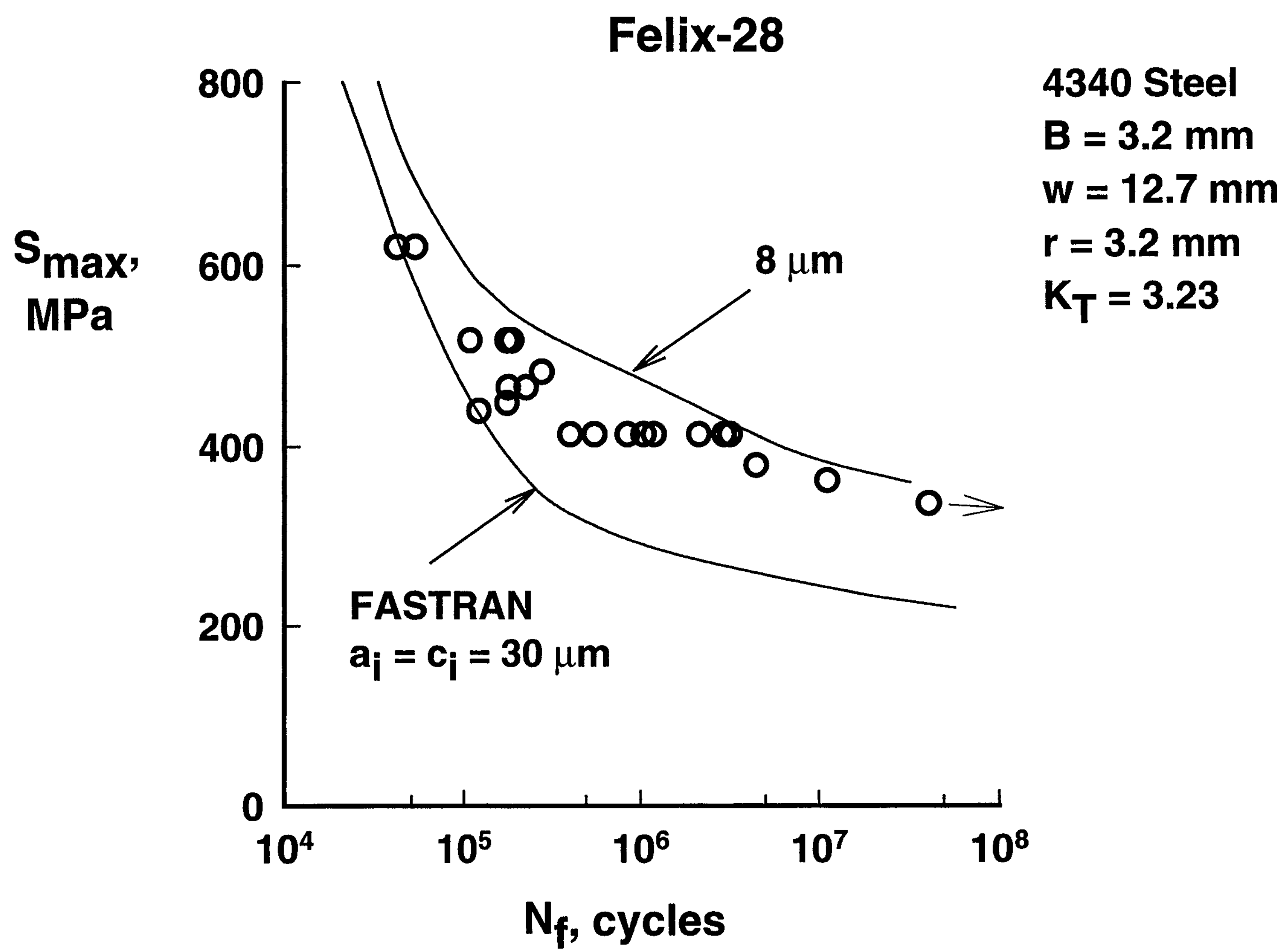


$$K = 1.12\sigma\sqrt{\pi a / Q}$$



$$da/dN = C\,(\Delta K)^n$$

MEASURED AND PREDICTED FATIGUE LIVES UNDER SPECTRUM LOADING



CONCLUSIONS

- A Machining-Like Scratch Reduced the Fatigue Strength of a High Strength By 40 %.

- Shot Peening Restored the Fatigue Strength.

- A Crack Growth Total Fatigue Life Model Predicted the S /N Behavior (Pristine and Scratched).

- A New Emerging Crack Growth Model for Predicting S / N Lives.

SESSION F

DISCUSSION WORKSHOP: REGULATION ASPECTS

CIVIL AVIATION AUTHORITY

Cranfield UNIVERSITY

DERA

JAA Rulemaking and Advisory Material Development Initiatives, Harmonisation with FAA and their status

Workshop at Cranfield University - 4th ,5th April 2000

Bruno Moitre - ENAC , Aircraft Certification Division

JAA Rulemaking and Advisory Material Development Initiatives, Harmonisation with FAA and their status

- Introduction
- Regulatory initiatives in an FAA-JAA harmonised perspective
- The JAA Helicopter Airworthiness Study Group (HASG) and the Joint Harmonisation Working Group (JHWG)
- The Working Groups sponsored and their T.O.R.'s
- Concluding remarks

Introduction

- Airworthiness design standards define minimum design requirements for ensuring a safety level congruent with the state of the art of the technology developments.
- Service experience, unairworthy conditions occurred should also be taken into account
- Emphasis put in the recent past on fatigue requirements and the associated need for continually review them by regulatory agencies are related to both events

Introduction (cont .)

- The Technical Oversight Group for Ageing Aircraft (TOGAA) established in U.S.A. with their recommendations to the Rotorcraft Working Group and the Amendment 28 to FAR Part 29 are examples of how the said process develops under the said circumstances.

Regulatory initiatives in an FAA-JAA harmonised perspective

- Nowaday a key word has entered into play within the regulatory civil aeronautical context :

“ Harmonisation”

- In such context Harmonisation has a meaning well beyond his literal one since the FAA and JAA committed in 1992 to develop a structure and formal procedure for their joint harmonisation program.
- The idea of harmonisation does not differ from the concept of defining a recognised and universally accepted standard however : ....

Regulatory initiatives in an FAA-JAA harmonised perspective (cont.)

Why Harmonisation ?

- The global market perspective
- The liberalisation of product's exchange
- The limit imposed to the barriers for air traffic growth
- Fair competition between U.S. and European Industry, Operators

What's Harmonisation?

- A fundamental process for the achievement of :
  - Common safety standards
  - Common interpretations of the requirements/means of compliance
- A challenge for the aviation community to achieve such goal while maintaining and possibly improving aviation safety.

Regulatory initiatives in an FAA-JAA harmonised perspective (cont.)

New regulations imply, in an *harmonised context* :

- A cost on the aviation community
- Costs have to be compensated on the ground of the mutual confidence and recognition of the Certificating Authorities
- Essential contributors for :
  - Certification cost reductions
  - Avoiding duplication of work
  - More efficient use of man power resources

The Helicopter Airworthiness Study Group and the Joint Harmonisation Working Group

The JAA Helicopter Airworthiness Study Group is the Authorities' group established by the JAA to

- develop harmonised proposals for JAR-27 and JAR-29 (small and large rotorcraft respectively)
- support the maintenance of JAR-27, JAR-29

The Helicopter Airworthiness Study Group and the Joint Harmonisation Working Group (cont.)

The Joint Harmonisation Working Group (JHWG) is the joint Authorities' and Industries' group (U.S. and European) set up to cooperate for the development of harmonised airworthiness requirements for small and large rotorcraft .

The JHWG is composed of :

- the HASG members
- an FAA representative
- U.S. and European Industry representatives

The Helicopter Airworthiness Study Group and the Joint Harmonisation Working Group (cont.)

The JHWG is supported by ad hoc working groups composed of experts selected from Authorities and Industry when specific regulatory and interpretative issues need to be addressed in light of harmonisation.

Rulemaking initiatives agreed within the JHWG are approved by the Harmonisation Management Team to which proposals are addressed together with the T.O.R.'s envisaged.

The Working Groups sponsored by the JHWG and their T.O.R.'s

Rulemaking

W.G. C5 ← Metals → W.G. C8

W.G. C6 ← Composites → W.G. C7

Advisory Material (existing req.'s)

The Working Groups sponsored by the JHWG and their T.O.R.'s (cont.)

Four Working Groups for Fatigue of Rotorcraft Structures have been agreed to be formed

- W. G. C5 - D.T. and fatigue evaluation of metallic rotorcraft structure (rulemaking)
- W.G. C8 - Fatigue evaluation of metallic rotorcraft structure (A.C. updating)
- W.G. C6 - D.T. and fatigue evaluation of composite rotorcraft structure (rulemaking)
- W.G. C7 - Fatigue evaluation of composite rotorcraft structure (A.C. updating)

T.O.R.'s for W. G. C5

BACKGROUND

- The Technical Oversight Group for Aging Aircraft (TOGAA) has recommended that current safe-life methods be complemented by D.T. assessment methods and that flaw tolerance safe life be removed from the regulations.
- The Industry (AECMA and AIA) in the White Paper, recommends retention of Flaw Tolerance as a Damage Tolerance option, and that the overall fatigue methodology be improved and updated.
- A Road Map is developed in the White Paper
- There is agreement that use of D.T.for part 27 will be optional

T.O.R.'s for W. G. C5 (cont.)

SPECIFIC TASKS

- Evaluate the Industry White Paper, the recommendations contained in the TOGAA letters to the FAA, and the on-going activities and results of rotorcraft D.T.R&D.
- Identify the information needed to conclude rulemaking and define acceptable M.O.C.
- Recommend appropriate changes to FAR/JAR 29 regarding D.T. and fatigue evaluation of metallic structure, and to FAR/JAR 27 (D.T. as an option)

T.O.R.'s for W. G. C5 (cont.)

SPECIFIC TASKS (cont.)

- Any recommended change should be practical/appropriate to the unique characteristics of rotorcraft.
- Where feasible and appropriate, provide consistency with FAR/JAR 23/25
- Prepare related AC material for both FAR/JAR 27 and 29.
- If requested, recommend disposition of comments received in response to the NPRM or NPA.

T.O.R.'s for W. G. C5 (cont.)

SCOPE/EXPECTATION

- The project is to be a harmonized JAR/FAR 27/29 ARAC program.
- The ARAC working group will present its final document to ARAC by April 2002
- Four meetings are anticipated, beginning April 2000
- A progress report will be provided at each JHWG meeting
- The ARAC recommendation will be provided to the FAA and the JAA by September 2002.

T.O.R.'s for W. G. C5 (cont.)

CAVEAT

- A separate rotorcraft ARAC working group will be developing a rulemaking proposal for composite structure.
- Although this tasking for metallic structure does not depend on the completion of the composite structure project, the two working groups should communicate to avoid possibly conflicting recommendations to amend the same regulatory sections

ENAC

T.O.R.'s for W. G. C6

BACKGROUND

- Current FAR/JAR 27/29 regulations do not provide adequate certification standards for composite materials.
- Past certifications based on advisory material and a broad interpretation of the fatigue substantiation.
- Some AA's have issued Special Conditions because the advisory material is not supported by an adequate airworthiness standard.

T.O.R.'s for W. G. C6 (cont.)

SPECIFIC TASKS

- Revise current FAR/JAR 27 and 29 to add regulations for composite structure
- Consider creating a new FAR/JAR 27/29.573.
- Evaluate and revise, as appropriate, the regulations-advisory material to achieve the goal of improved tolerance to flaws and defects in composite structure with methodology and procedures which are practical and appropriate to rotorcraft.
- Consistency with FAR/JAR 23/25 where feasible-appropriate

T.O.R.'s for W. G. C6 (cont.)

SCOPE/EXPECTATIONS

- The project is to be a harmonized JAR/FAR 27/29 ARAC program.
- The ARAC working group will present its final document to ARAC by June 2002.
- Four working group meetings are anticipated, beginning June 2000.
- A progress report will be provided at each JHWG meeting.

T.O.R.'s for W. G. C6 (cont.)

CAVEAT

- A separate rotorcraft harmonisation W.G. will be developing a D.T. rulemaking proposal for metallic structure.

- Although this tasking for composite structure does not depend on the completion of the metallic structure project, the two W.G.'s should communicate to avoid possibly conflicting recommendations to amend the same regulatory sections.

Concluding remarks (cont.)

On the reason for the 4 distinct W.G.'s sponsored

- Existing FAR/JAR 27,29 requirements for fatigue evaluation of rotorcraft structures are in the TC basis of a number of projects
- FAA Rotorcraft Directorate has mandate for revising and maintaining AC 27-1 and AC 29-2
- This task is promoted by the FAA Rot.Dir. through a two years revision cycle which forms also part of the JHWG routine work for the seeking of harmonisation
- WG's for AC revision are composed by the same people involved in the WG's for rulemaking thus allowing consistency, synergistic efforts, cost saving,

Concluding remarks (cont.)

On the Road Map proposed by Industry

- R & D effort should parallel and support but not drive the rulemaking effort,to enhance the advisory material and expand the practical application of a very general proposed regulation.
- The AA's do not consider appropriate to tie the rulemaking effort to the accomplishment of any specific R&D plan
- The FAA is committed in their Aging Aircraft Plan to publish the Part 29 DT final rule in fiscal year 2003

Concluding remarks (cont.)

On the Harmonisation aspects

- W.G.'s for Rulemaking are supported by the FAA-JAA HMT in accordance with the outcoming of the 16th Annual JAA/FAA Conference and the " Better Plan for Harmonisation "
- FAA,JAA,U.S. and European Industry agree to coordinate their effort in a suitable timeframe to accomodate TOGAA recommendations without disrupting harmonisation
- JHWG will monitor progress of different W.G.'s
- Progress from R&D supportive of the W.G.'s ; coordination is a key for success.

SESSION G

RAPPORTEURS CONCLUSIONS & SUMMARY

CIVIL AVIATION AUTHORITY

Cranfield UNIVERSITY

DERA