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PROCEEDINGS OF THE 1988 STRUCTURAL INTEGRITY PROGRAM CONFERENCE

EDITORS:

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Wright-Patterson AFB OH

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May 1989

29 November - 1 December 1988

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Hilton Palacio Del Rio Hotel
San Antonio, Texas

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MATERIALS LABORATORY
WRIGHT RESEARCH & DEVELOPMENT CENTER
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AFB OHIO 45433-6533

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This technical report has been reviewed and is approved for publication.

THOMAS D. COOPER, Chief
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Systems Support Division
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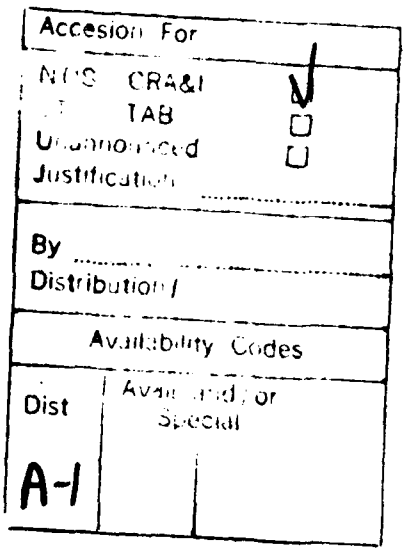
FOREWORD

This report was compiled by the Materials Integrity Branch, Systems Support Division, Materials Laboratory, Wright Research & Development Center, Wright-Patterson AFB, Ohio. It was initiated under Task 24180704 "Corrosion Control & Failure Analysis" with Thomas D. Cooper as the Project Engineer.

This technical report was submitted by the editors.

The purpose of this 1988 Conference was to bring together technical personnel in DOD and the aerospace industry who are involved in the various technologies required to insure the structural integrity of aircraft gas turbine engines, airframes and other mechanical systems. It provided a forum to exchange ideas and share new information relating to the critical aspects of durability and damage tolerance technology for aircraft systems. The Conference was sponsored by the Aeronautical Systems Division Deputy for Engineering and Materials Laboratory of the Wright Research & Development Center, Wright-Patterson AFB, Ohio. It was hosted by the Air Force Logistics Command's San Antonio Air Logistics Center.



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1988 USAF STRUCTURAL INTEGRITY PROGRAM

29 November - 1 December 1988

Hilton Palacio Del Rio Hotel
San Antonio, Texas

SPONSORED BY:

ASD/Deputy for Engineering

WRDC/Materials Laboratory

HOSTED BY:

San Antonio Air Logistics Center
Material Management Directorate

Fighter/Tactical/Trainer
Systems Program Management
Division (SA-ALC/MMS)

AGENDA

TUESDAY, 29 NOVEMBER 1988

SESSION I. OVERVIEWS
Chairman - T. Cooper, WRDC/MLSA

0830 - 0900 The Integrity Programs
J. Halpin, ASD/EN

0900 - 1000 The Integrity Process as Applied to the Advanced Tactical Fighter Engine (AFTE)
J. Ogg, D. Irwin, Lt J. Satterfield, ASD/YFE
P. Domas - General Electric Co.
J. Estes - Pratt & Whitney

1000 - 1030 REFRESHMENT BREAK

1030 - 1100 The MECSIP Approach for Assessing In-Service Mechanical Systems
H. Wood, ASD/ENF

1100 - 1130 WRDC Aircraft Mechanical Systems Integrity Program
K. Schwartz, WRDC/FIEMB

1130 - 1200 The Development of Improved NDE/I to Meet Structural Integrity Requirements
D. Forney Jr, WRDC/MLLP

1200 - 1330 LUNCH AND PRESENTATION
Long Life Assurance Issues for the Space Station
T. Cruse, Southwest Research Institute

SESSION II. STRUCTURAL ANALYSIS, FASTENERS AND TESTING
Chairman - R. Bader, WRDC/FIB

1330 - 1400 Control of Error in Local Stress Analysis with PROBE, a New P-Version FEA Program
B. Taylor, Noetic Technologies

1400 - 1430 Predicting Fatigue Crack Growth Under Combined Tension and Out of Plane Bending in Transitional Thickness Plates
Flt Lt K. Walker, WR-ALC/MMEMD
D. Showers, WR-ALC/MMEMD
W. Ayer, WR-ALC/MMEMD
R. Bell, Lockheed Aeronautical Systems Co.

1430 - 1500 Inspection of Fatigue Critical Fastener Holes Using Capacitance Measurement Systems Equipment
D. Hammond Jr, WR-ALC/MMSHR

1500 - 1530 REFRESHMENT BREAK

1530 - 1600 An Update on the Study of the Quality of Screw Threaded Products
C. L. Petrin Jr., ASD/ENFS

1600 - 1630 CF116 Full-Scale Durability & Damage Tolerance Test
E. Van Blaeren - Canadair Inc.

1630 - 1700 F-16 Full-Scale Airframe Durability Test
C. Babish, ASD/YPE

1730 RECEPTION

WEDNESDAY, 30 NOVEMBER 1988

Session III - ASIP/METHODS
Chairman - H. Wood, ASD/ENF

0800 - 0830 T37B Damage Tolerance Analysis Results
C. Massey, SA-ALC/MMSA

0830 - 0900 The C-17 Aircraft Structural Integrity Overview
T. Bivins and R. Eastin, Douglas Aircraft

0900 - 0930 The X-30 Structural Integrity Program
M. Snead, AFSC/NAE

0930 - 1000 Structural Risk Analysis in Aging Aircraft Fleets
A. Berens, The University of Dayton

1000 - 1030 REFRESHMENT BREAK

1030 - 1100 Probabilistic Methods for Inspection, Planning & Life Extension of an Aging Aircraft Fleet
R. Skjong, Veritas Research Oslo, Norway

1100 - 1130 Structural Risk Analysis - Capability to Conservatively Forecast Initial Crack Instability
H. Horsburgh, Lockheed Aeronautical Systems Co.
O. Greenhaw, WR-ALC/MMSFRA

1130 - 1200 Risk Assessment of the F-16 Wing
D. Cornog, ASD/YPE
J. Lincoln, ASD/ENFS

1200 - 1330 LUNCH & PRESENTATION
The Materials Revolution
V. Russo, Director, WRDC Materials Laboratory

SESSION IV - METALLIC/COMPOSITE STRUCTURES
Chairman - Lt Col J. Kolek, WRDC/MLS

1330 - 1400 ARALL Laminates - Scale-Up from R&D to Flying Articles
R. Bucci, L. Mueller, R. Bentley & M. Gregory, Alcoa Laboratories

1400 - 1430 ARALL Laminates on the Douglas C-17 Transport Aircraft
W. Leodolter & R. Pettit, Douglas Aircraft
M. Gregory, Alcoa Laboratories

1430 - 1500 C-130 Composite Structural Repair Development
J. Grosko, Lockheed Aeronautical Systems Co.

1500 - 1530 REFRESHMENT BREAK

1530 - 1600 Low Velocity Impact Damage Detector Development
S. Huang & A. Scotese, Naval Air Development Center

1600 - 1630 Recent Advances in the Fatigue Enhancement Technology
E. Easterbrook, M. Landy & L. Reid, Fatigue Technology Inc.

1630 - 1700 Life Cycle Cost Reductions Realized from Implementation of Damping
M. Drake, University of Dayton

THURSDAY, 1 DECEMBER 1988

SESSION V - AVIP/ENSIP
Chairman, C. Petrin Jr., ASD/ENFS

0800 - 0830 Initiating & Integrating AVIP with ENSIP
L. Allen, General Electric

0830 - 0900 The Application of ENSIP to the F101-GE-100 Engines, Practical Perspectives & Future Needs
R. Craman, General Electric

0900 - 0930 Structural Integrity Program NDE Application and Development
W. Herron, General Electric

0930 - 1000 Engine Monitoring System for On Condition Maintenance of the T800-LHT-800 Turboshaft Engine
D. Metz & B. Aldieri, Garrett Engine Div of Allied Signal Aerospace Corp.

1000 - 1030 REFRESHMENT BREAK

1030 - 1100 F-109-GA-100 Engine Accelerated Mission (AMT) Testing & Condition Monitoring
H. Maertins, Garrett Engine Div of Allied Signal Aerospace Corp.

1100 - 1130 Engine Mishap Data Analysis
W. Cowie, ASD/YZEF
T. Stein, USAF Reserve

1130 - 1200 Automation of Oil Analysis Laboratories
L. Miller, A. Souza, M. Anderson & J. Frenster, Systems Control Technology

1200 - 1330 LUNCH AND PRESENTATION
Certification Program for ASTRA 1125 Aircraft as a Result of Damage Tolerance Requirements
A. Davidsonn, Israeli Aircraft Industries

SESSION VI - ENSIP/FORCE MANAGEMENT
Chairman: J. Turner, SA-ALC/MMSA

1330 - 1400 T-38 Severe Usage Spectra Development/Update
J. P. Kaplan, Willis, Kaplan & Associates, Inc.
R. E. Welch, Chapetta, Welch & Associates Ltd.
T. A. Knott, Willis, Kaplan & Associates, Inc.
M. Reinke, SA-ALC/MMS
Lt S. Arnold, SA-ALC/MMS
C. Massey, SA-ALC/MMS

1400 - 1430 C-141 Force Management Technology
J. Cochran, Lockheed Aeronautical Systems Co.
T. Christian, WR-ALC/MMSFR
D. Hammond, WR-ALC/MMSHR

1430 - 1500 Navy's Structural On-line Fatigue Information System
V. Elchuri, Aerostructures Inc.
J. Braun, Systems & Electronics Inc.
B. Sturgis, Naval Air Systems Command

1500 - 1530 REFRESHMENT BREAK

1530 - 1600 Multichannel On-Board Fatigue Monitoring
H. Wilms, Spectralab Kilchberg, Switzerland

1600 - 1630 ASIP Concepts for the Standard Flight Data Recorder
T. Conquest, F. Saggio & K. Todoroff, SLI Avionics Corp.

1630 - 1700 F-15 Aircraft Information Retrieval System (AIRS)
M. Sullentrup, McDonnell Douglas
B. Trussell, WR-ALC/MMFRB

1700 ADJOURN

INTRODUCTION

This report contains the proceedings of the 1988 USAF Structural Integrity Program Conference held at the Hilton Palacio Del Rio Hotel in San Antonio, Texas from 29 November to 1 December 1988. The conference, which was sponsored by the ASD Deputy for Engineering and the WRDC Materials Laboratory, was hosted by the San Antonio Air Logistics Center Material Management Directorate Fighter/Tactical/Management Division (SA-ALC/MMS).

This conference, which in recent years has been held annually, has evolved from government only participation to a conference where government and industry personnel meet to exchange views on how to improve the structural integrity of military weapon systems. It is also evolving from a conference where the emphasis was on issues derived from the Aircraft Structural Integrity Program (ASIP) to a conference where the Engine Structural Integrity Program (ENSIP) and the Mechanical Subsystems Integrity Program (MECSIP) are a vital integral part of the program. It is anticipated that these three integrity programs will be the basis for the agenda for future meetings.

The sponsors are indebted to their hosts for their support of the conference. The sponsors are also indebted to the speakers for their contributions. In particular, thanks are due to Dr. John Halpin from ASD/EN for his remarks in the lead paper on "The Integrity Programs." Thanks are also due to Dr. Thomas Cruse from Southwest Research Institute, Dr. Vince Russo, Director of the WRDC Materials Laboratory and to Mr. Ami Davidsohn from the Israel Aircraft Industries for their luncheon presentations. But, most of all thanks to all of the attendees for making the conference a success. Much of the success of the conference is due to the efforts of Jill Jennewine and her staff from the Universal Technology Corporation. Their cooperation is appreciated.

John W. Lincoln
ASD/ENFS

Thomas D. Cooper
WRDC/MLSA

PROPULSION AND POWER SYSTEM INTEGRITY:

THE INTEGRITY OF PROPULSION STRUCTURES, CONTROLS, ACCESSORIES, AND EXTERNALS; THE PROPULSION AND POWER SYSTEM INTEGRITY PROGRAM, AND ITS APPLICATION TO THE ADVANCED TACTICAL FIGHTER ENGINES.

PRESENTER: DAVID IRWIN
ASD/YFEP
WPAFB OH

OVERVIEW

- THE INTEGRITY PROCESS
- ATFE INTEGRITY
- HISTORICAL PERSPECTIVE ON ENGINE CONTROLS, ACCESSORIES, AND EXTERNALS (CA&E) DEVELOPMENT
- CA & E INTEGRITY
- ATFE INTEGRITY PROGRAM IMPLEMENTATION
- SUMMARY

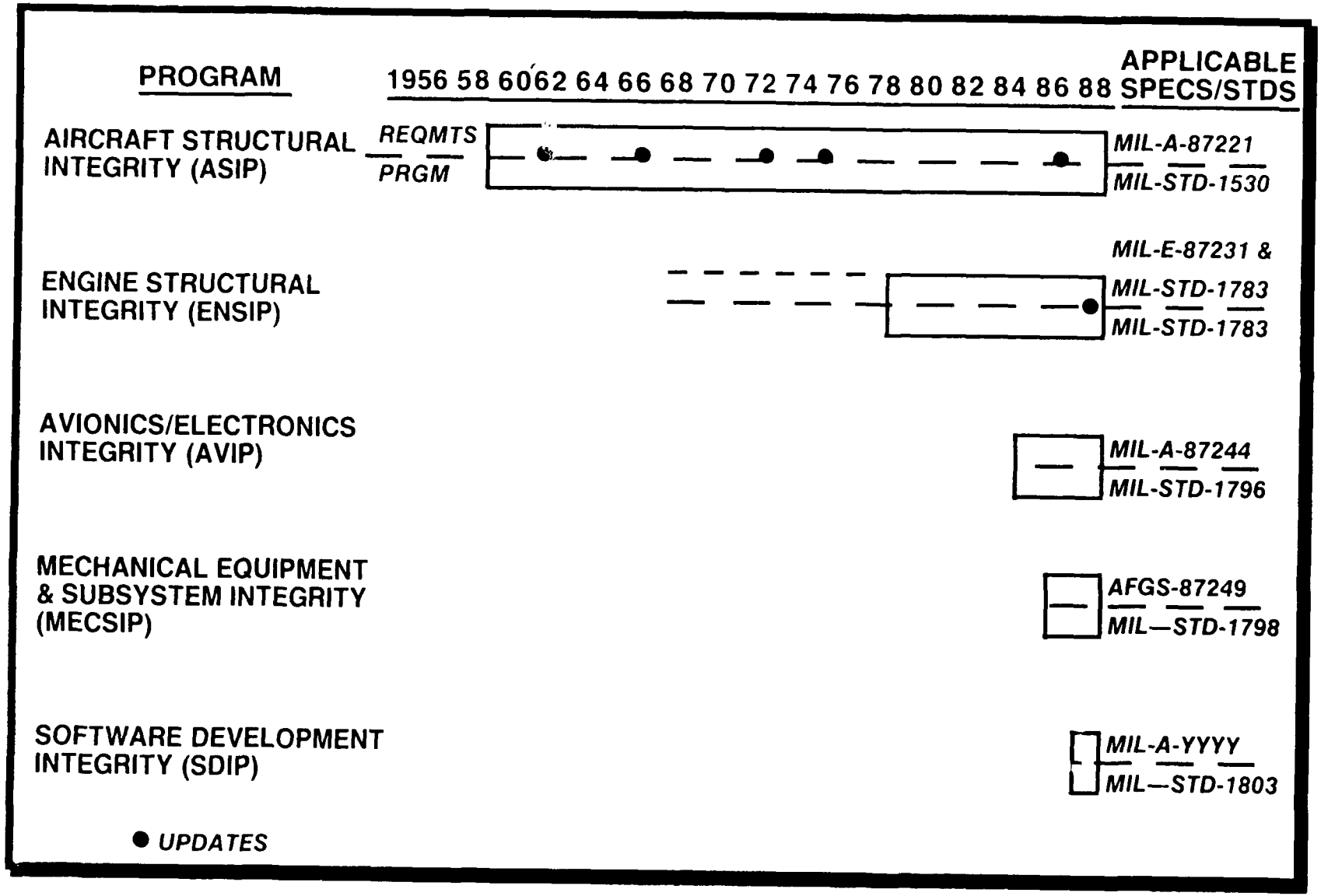
THE INTEGRITY PROCESS

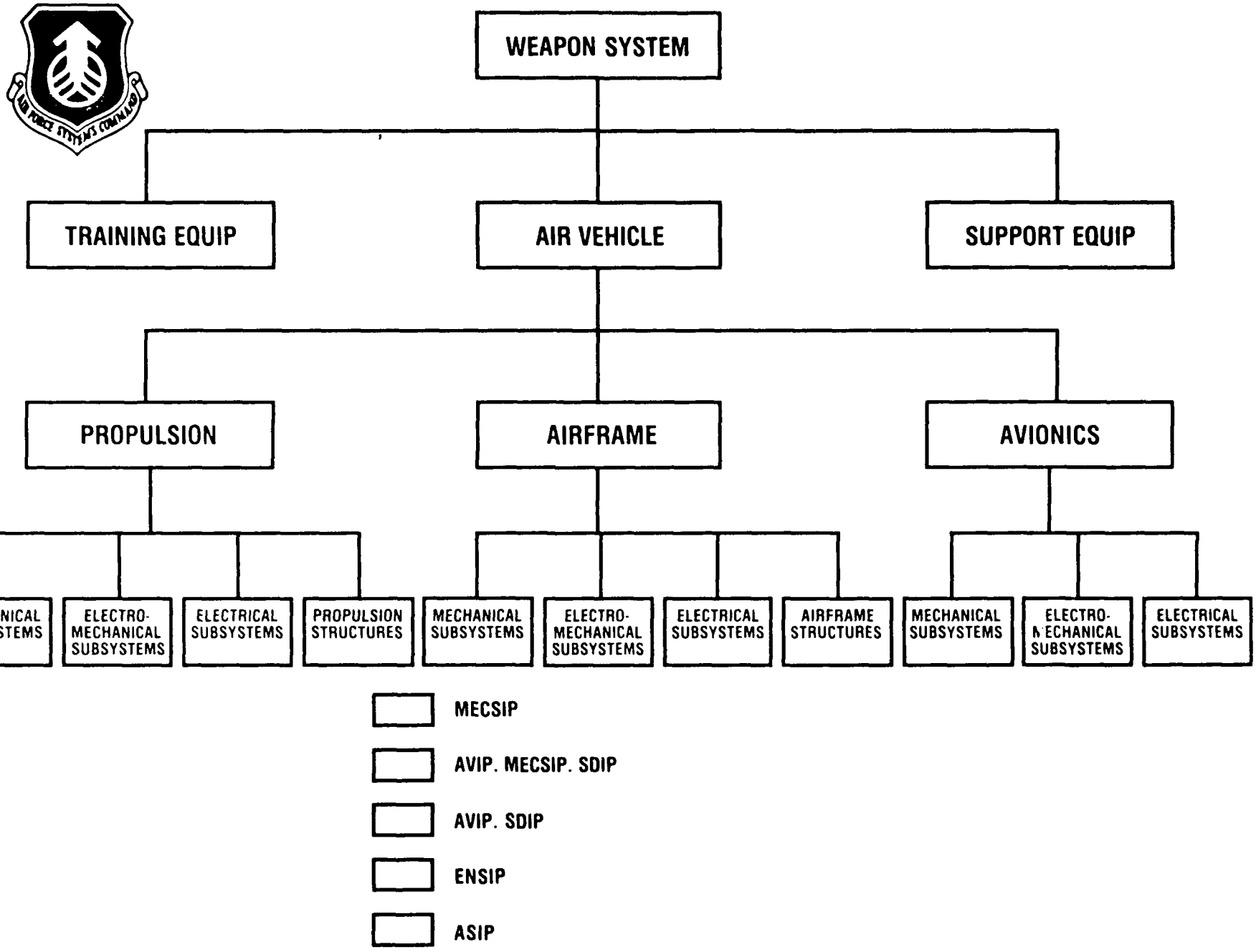
- AN ORGANIZED AND DISCIPLINED APPROACH TO THE DESIGN, DEVELOPMENT, QUALIFICATION, PRODUCTION, AND LIFE MANAGEMENT OF A PRODUCT WITH THE GOAL OF ENSURING:
  - SAFETY
  - MISSION CAPABILITY
  - RELIABILITY
  - MAINTAINABILITY
  - REDUCED COST OF OWNERSHIP (LCC)

KEY ELEMENTS OF THE INTEGRITY PROCESS

- THOROUGH UNDERSTANDING OF THE OPERATIONAL USAGE ENVIRONMENT
- REALISTIC REQUIREMENTS DERIVED FROM INTENDED APPLICATION
- IDENTIFICATION OF CRITICAL COMPONENTS/FEATURES
- THOROUGH CHARACTERIZATION OF DESIGN VIA:
  - ANALYSIS
  - INSTRUMENTED TESTING
  - MATERIAL AND PROCESS CHARACTERIZATION
  - DURABILITY AND RESIDUAL LIFE TESTING
- ESTABLISHMENT OF DESIGN CRITERIA BASED UPON APPLICATION
- DEFINITIZATION & IMPLEMENTATION OF MANUFACTURING AND DEPOT/FIELD MAINTENANCE PROCESS CONTROLS AND QUALITY ASSURANCE REQUIREMENTS
- LIFE MANAGEMENT OF CRITICAL HARDWARE
- CLOSED LOOP ON FEEDBACK OF DEVIATIONS

INTEGRITY PROCESS EVOLUTION & DISSEMINATION




WEAPON SYSTEM
TRAINING EQUIP
AIR VEHICLE
SUPPORT EQUIP
PROPULSION
AIRFRAME
AVIONICS
MECHANICAL SUBSYSTEMS
ELECTRO-MECHANICAL SUBSYSTEMS
ELECTRICAL SUBSYSTEMS
PROPULSION STRUCTURES
MECHANICAL SUBSYSTEMS
ELECTRO-MECHANICAL SUBSYSTEMS
ELECTRICAL SUBSYSTEMS
AIRFRAME STRUCTURES
MECHANICAL SUBSYSTEMS
ELECTRO-MECHANICAL SUBSYSTEMS
ELECTRICAL SUBSYSTEMS
MECSIP
AVIP. MECSIP. SDIP
AVIP. SDIP
ENSIP
ASIP

ATFE PPSIP REQUIREMENTS

INTEGRATION OF ENSIP, AVIP, AND MECSIP REQUIREMENTS AS TAILORED TO THE ATF ENGINES.

I USAGE AND ENVIRONMENTAL DEFINITIONS

II MATERIALS AND PROCESS CHARACTERIZATION

III PARTS CLASSIFICATION

IV ENVIRONMENTAL REQUIREMENTS

V STRENGTH REQUIREMENTS

VI DURABILITY REQUIREMENTS

VII DAMAGE TOLERANCE REQUIREMENTS

VIII MISCELLANEOUS REQUIREMENTS

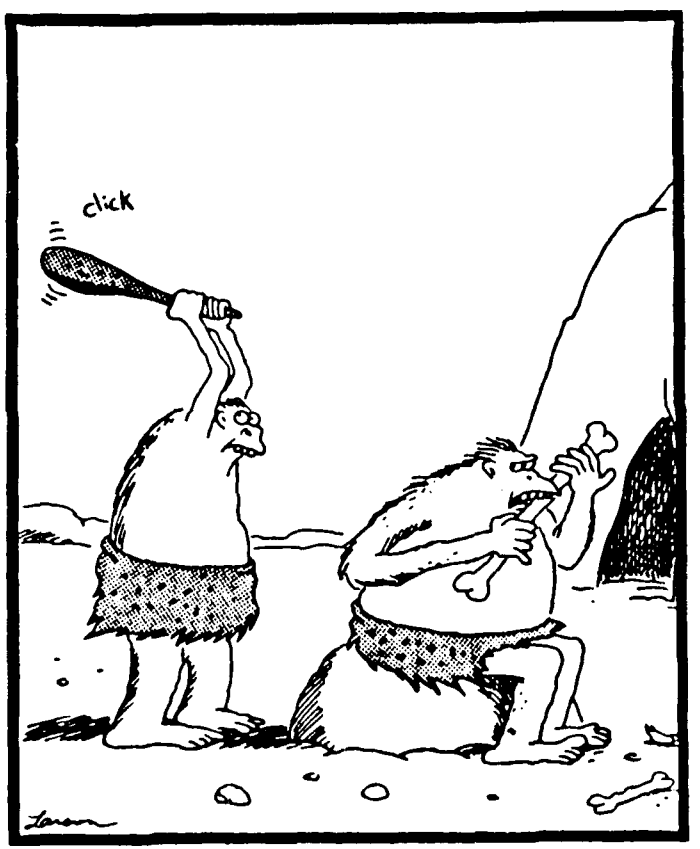
- CREEP
- OVERSPEED, OVERTEMP
- VIBRATION REQUIREMENTS
- SURGE AND STALL
- FOD/DOD
- INSPECTABILITY

IX INTEGRITY MANAGEMENT REQUIREMENTS

ADDITIONAL CA & E UNIQUE INTEGRITY REQUIREMENTS

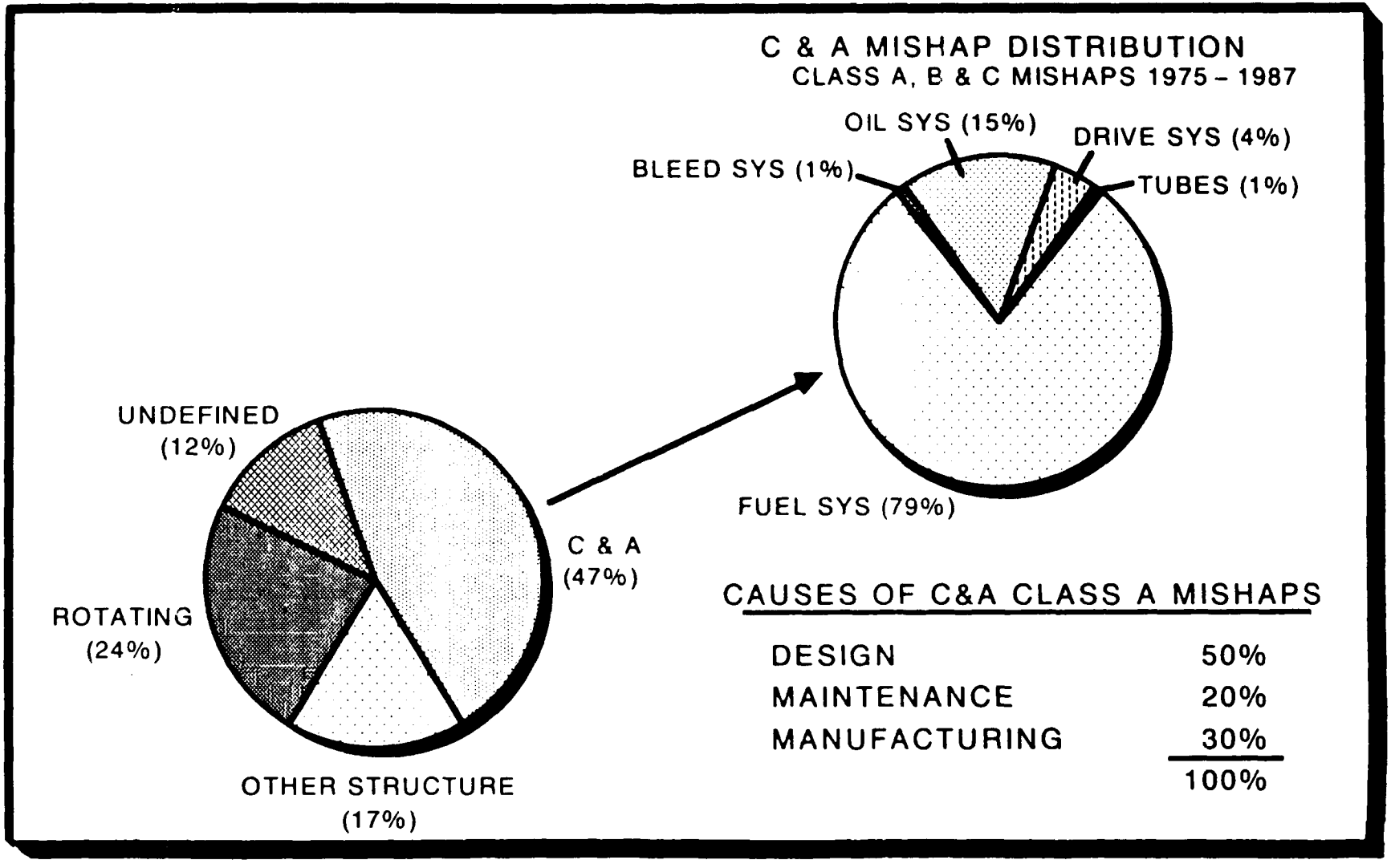
- SENSITIVITY OF PARTS, ASSEMBLIES, AND EQUIPMENT TO ELECTROSTATIC DISCHARGE
- INTERNAL PRESSURES
- CONTAINMENT
- DIELECTRIC MATERIAL MARGIN
- CURRENT CARRYING CAPACITY
- ELECTRICAL/ELECTRONIC DESIGN (DERATED) STRESS LEVELS
- MODIFICATION/REWORK OF PRINTED WIRING AND SURFACE MOUNTED ASSEMBLIES
- THERMAL CYCLES ASSOCIATED WITH UNIT REPAIRS
- FAIL SAFE EVIDENT/LEAK BEFORE BURST
- EQUIVALENT LIFE & MARGIN TESTING

HISTORICAL PERSPECTIVE



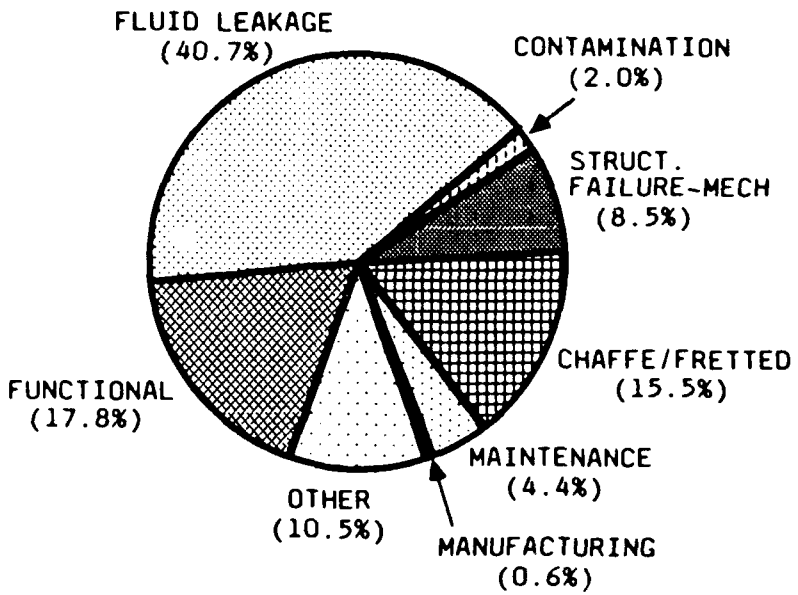
At a critical moment, Zak's club jams.

ENGINE MISHAP EXPERIENCE
CLASS A, B, & C MISHAPS

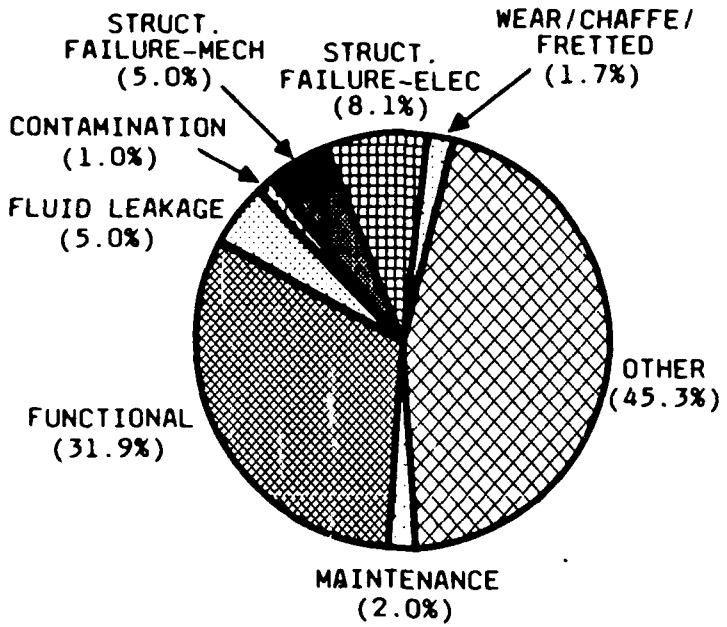


C&A MALFUNCTIONS

MECHANICAL COMPONENTS



ELECTRICAL - MECHANICAL COMPONENTS



ECONOMIC SIGNIFICANCE

USAF

- LOSS OF AIRCRAFT
- LOSS OF MISSION
- AIRCRAFT DOWNTIME
- SPARE PARTS (OVERHAUL) REPLACEMENT
- SHIPPING
- SUPPORT EQUIPMENT
- PERSONNEL MANHOURS
  - MAINTENANCE
  - LOGISTICS
  - ENGINEERING
  - DATA TRACKING
- PAPERWORK

ENGINE CONTRACTOR

- WARRANTY CLAIMS
- PRODUCTION DOWNTIME
- SHIPPING
- TEST FACILITIES
- PERSONNEL MANHOURS
  - FIELD REPS
  - LOGISTICS
  - ENGINEERING
  - DATA TRACKING
- PAPERWORK
- IMAGE

COMPONENT VENDOR

- WARRANTY CLAIMS
- PRODUCTION DOWNTIME
- SHIPPING
- TEST FACILITIES
- PERSONNEL MANHOURS
  - TECHNICIANS
  - LOGISTICS
  - ENGINEERING
  - DATA TRACKING
- PAPERWORK
- IMAGE

PAST "DESIGN", DEVELOPMENT & QUALIFICATION

- COMPONENT BENCH TESTS

  IDENTICAL DURATION FOR ALL APPLICATIONS; NOT MISSION NOR COMPONENT DESIGN SPECIFIC

  - SIMULATED OPERATIONAL TESTS (NO DIFFERENTIATION BETWEEN MECHANICAL & ELECTRICAL COMPONENTS)

    100 HRS/600 CYCLES AT HIGH TEMP

    300 HRS/1800 CYCLES AT ROOM TEMP WITH CONTAMINATED FUEL

    20 HRS/10 CYCLES AT LOW TEMP

  - ENVIRONMENTAL TESTS (ELECTRICAL COMPONENTS ONLY)

HUMIDITY	SHOCK
FUNGUS	VIBRATION
ACCELERATION	EMI/LIGHTNING

  - SPECIAL TESTS

    IGNITION SYTEM FOULING — FIRE TESTS

    LUBE OIL TANK PROOF PRESSURE/10,000 PRESSURE CYCLES — FUEL FILTER CAPACITY

    ALTERNATOR OVERSPEED/ELECTRICAL OVERLOAD

- ENGINE AMT TESTS
  - INSIGNIFICANT LOADING ON MOST C&A COMPONENTS
  - VERIFIES PERFORMANCE/OPERABILITY
- NO DESIGN ANALYSIS REQUIRED

OVERALL PHILOSOPHY WAS TO PASS QUALIFICATION TESTS (COOKBOOK STYLE)
NOT REQUIRED TO UNDERSTAND DESIGN USAGE/LIMITATIONS

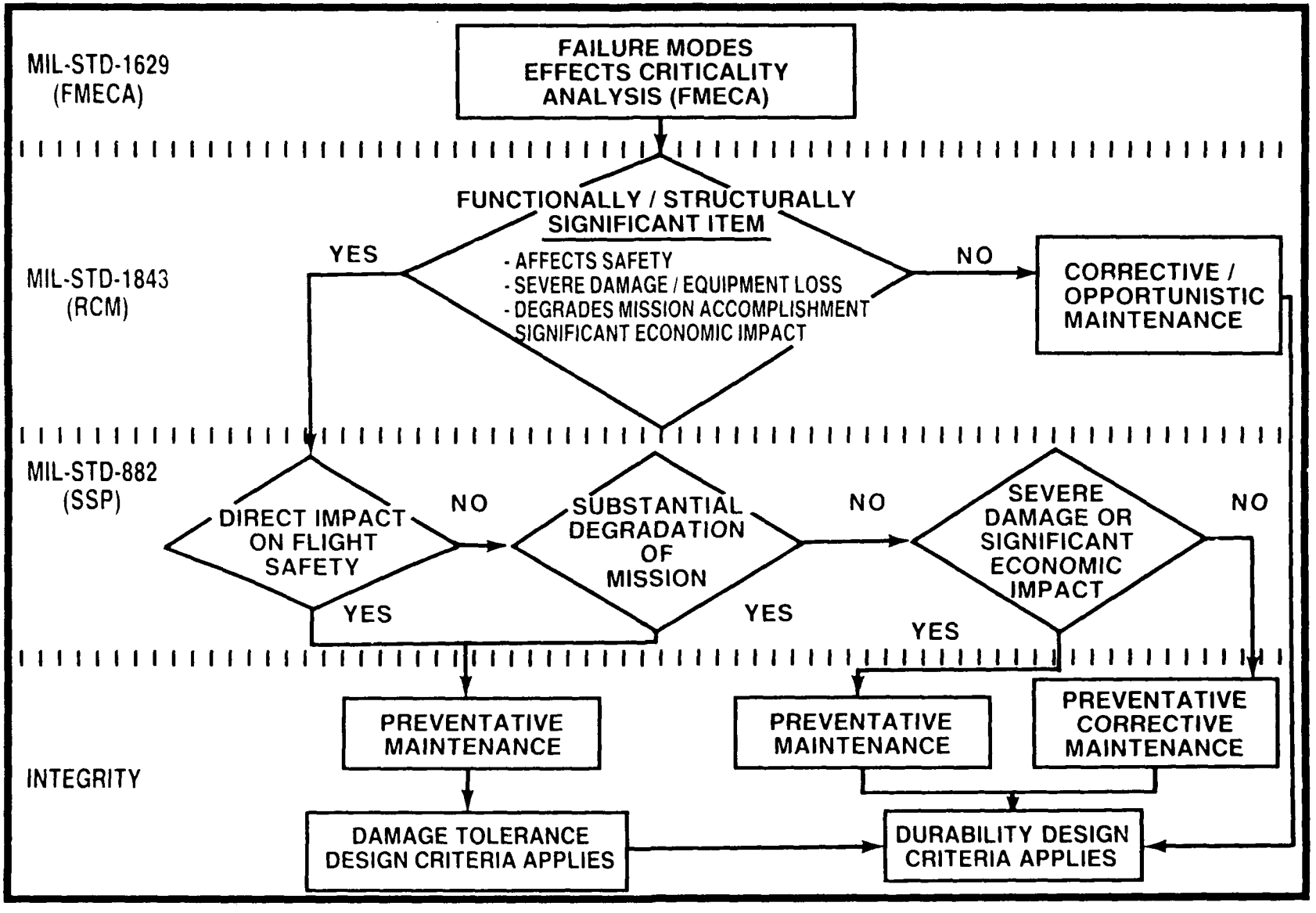
DEFICIENCIES WITH TRADITIONAL APPROCH

- COMPONENTS "SCALED" TO MEET INDIVIDUAL PERFORMANCE/ FUNCTIONAL REQUIREMENTS
- COMPONENTS NOT "DESIGNED" FOR SYSTEMS USAGE/ ENVIRONMENTS AS APPLIED TO DURABILITY, RELIABILITY, MAINTAINABILITY
- COMPONENTS WERE "DESIGNED" TO PASS QUALIFICATION TESTS
- LITTLE UPFRONT ANALYSIS CONDUCTED TO VERIFY COMPONENT'S ABILITY TO MEET REQUIREMENTS
- QUALIFICATION TESTS NOT NECESSARILY REPRESENTATIVE OF COMPONENT USAGE - LIMITED IN EVALUATING COMPONENT & USAGE VARIABILITIES
- FMECA AS A CONTRACTUAL REQUIREMENT RATHER THAN A DESIGN/DEVELOPMENT TOOL
- DESIGN CAPABILITIES/LIMITATIONS NOT UNDERSTOOD PRIOR TO PRODUCTION
- REQUIRED ONLY QUALIFICATION TEST COMPLETION, DID NOT INCLUDE LOGICAL, ORGANIZED ENGINEERING DESIGN EFFORT

DERIVATION OF ENGINE CONTROL, ACCESSORY, & EXTERNAL INTEGRITY REQUIREMENTS

- PRIOR SPECIFICATION REQUIREMENTS
  - FUNCTIONAL OPERATION
  - PERFORMANCE
- OPERATIONAL EXPERIENCE ON PAST/EXISTING SYSTEMS
  - DURABILITY
  - RELIABILITY/MAINTAINABILITY
  - SAFETY
  - OPERABILITY
- ONGOING INSTITUTIONALIZATION OF INTEGRITY DISCIPLINE FOR MECHANICAL & ELECTRICAL SUBSYSTEMS/COMPONENTS
  - FORMAL STRUCTURED APPROACH TO DESIGN & DEVELOPMENT
  - DAMAGE TOLERANCE
  - DURABILITY DEMONSTRATION
- RECOGNIZED NEED FOR FACTORING ECONOMIC & MISSION EFFECTIVENESS ELEMENTS INTO DESIGN CRITERIA
  - FMECA/RCM/ILS
- TECHNOLOGICAL ADVANCEMENTS
  - INSTRUMENTATION
  - ANALYTICAL TOOLS
- LESSONS LEARNED
  - TEST TECHNIQUES
  - ESTABLISHMENT OF REALISTIC REQUIREMENTS
  - DEFINITIZATION OF REQUIREMENTS
  - MAXIMUM USE OF NUCLEUS OF EXPERIENCE

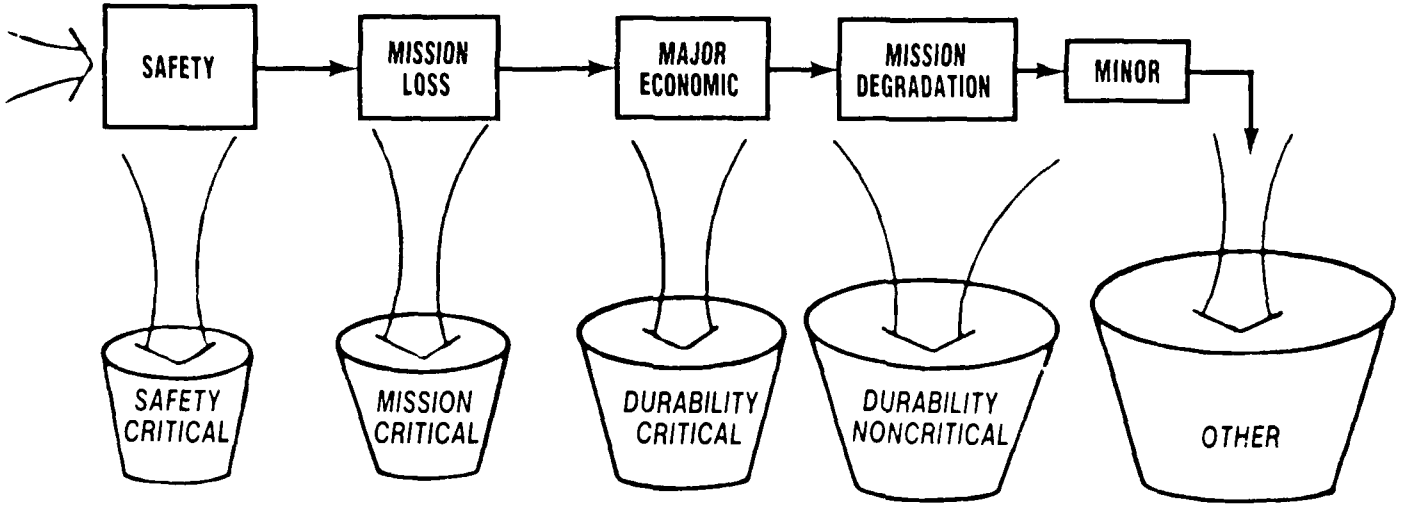
CA & E DESIGN METHODOLOGY



COMPONENT CLASSIFICATION DECISION LOGIC

INPUT

- **EXPERIENCE**
- **FMECA**
- **ENGINEERING ANALYSES**

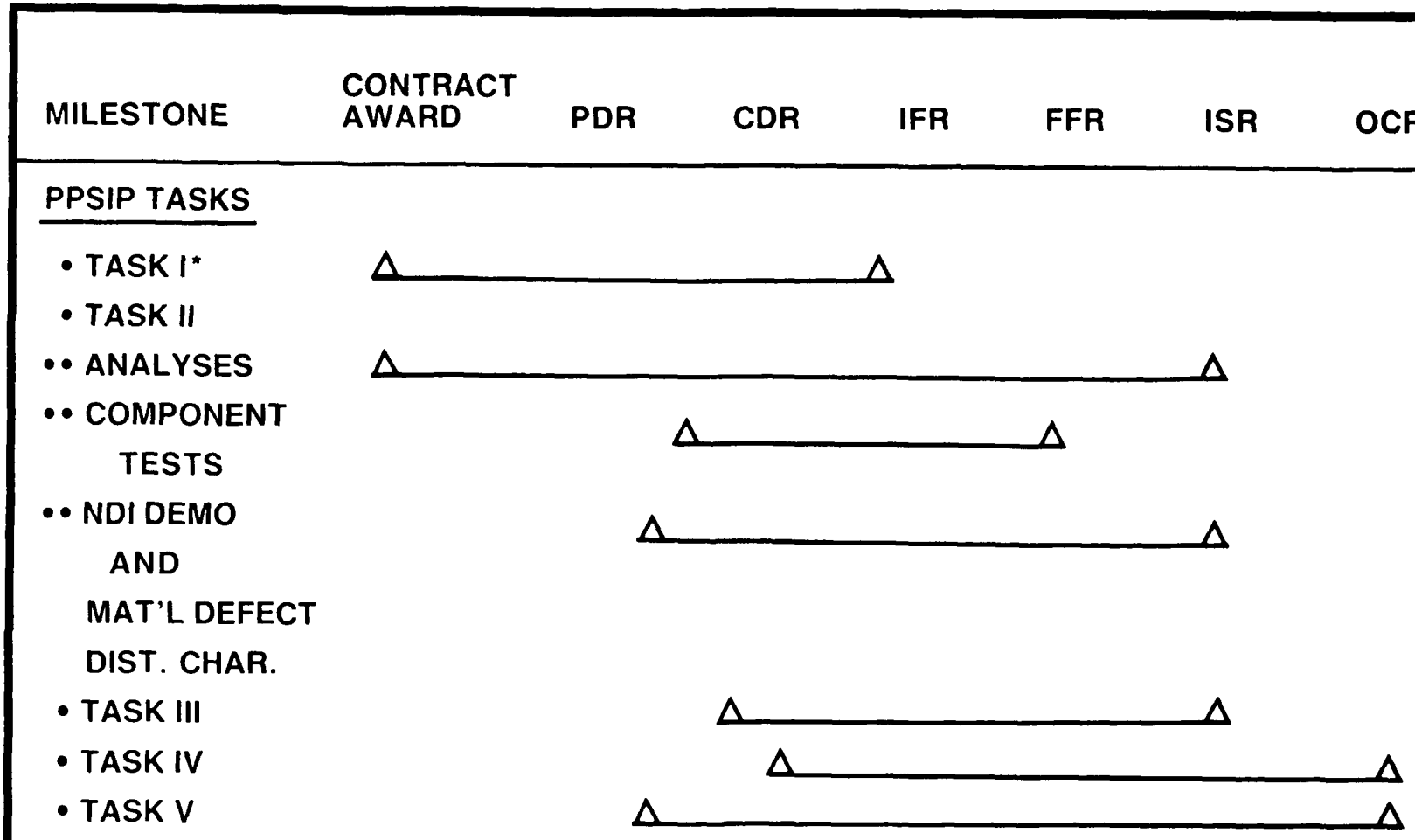


CLASSIFICATION CATEGORIES	CANDIDATE SPECIAL PROVISIONS*						
	EXPANDED DESIGN CRITERIA	TIGHTENED PROCESS CONTROL	ENHANCED QUALITY CONTROL	ACCEPTANCE TESTING	RESTRICTED REPROCUREMENT	PARTS TRACKING	PREVENTATIVE MAINTENANCE
SAFETY CRITICAL	●	●	●	●	●	●	●
MISSION CRITICAL	●	●	●	●	●	●	●
DURABILITY CRITICAL		●	●	●	●	●	●
DURABILITY NONCRITICAL				●			●
OTHER							

* THE EXTENT TO WHICH THE SPECIAL PROVISIONS ARE APPLIED IS COMPONENT SPECIFIC

RELATIC
DISK A8

RELATIONSHIP OF PPSIP TASKS TO ATFE MILESTONES



*SOME ITEMS SUCH AS THE DADT CONTROL PLAN WILL NEED UPDATING AS THE PROGRAM PROGRESSES

ATFE RFP

REQIREMENTS (SRD)	GUIDANCE (REFERENCE)	INTEGRITY PROCESS (REFERENCE)
MIL-E-87231 (SECTION 3) MILITARY SPECIFICATION FOR AIR BREATHING TURBINE ENGINE	MIL-E-87231 HANDBOOK	
STRUCTURAL INTEGRITY REQUIREMENTS (MIL-STD-1783 SECTION 4) ENSIP	MIL-STD-1783 HANDBOOK	MIL-STD-1783 SECTION 4 & HANDBOOK
MECHANICAL CA&E INTEGRITY REQUIRMENTS (TAILORED MECSIP)	AFGS-87247	MIL-STD-1798
ELECTRICAL CA&E INTEGRITY REQUIREMENTS (TAILORED AVIP)	MIL-A-87244 HANDBOOK	MIL-STD-1796

SUMMARY

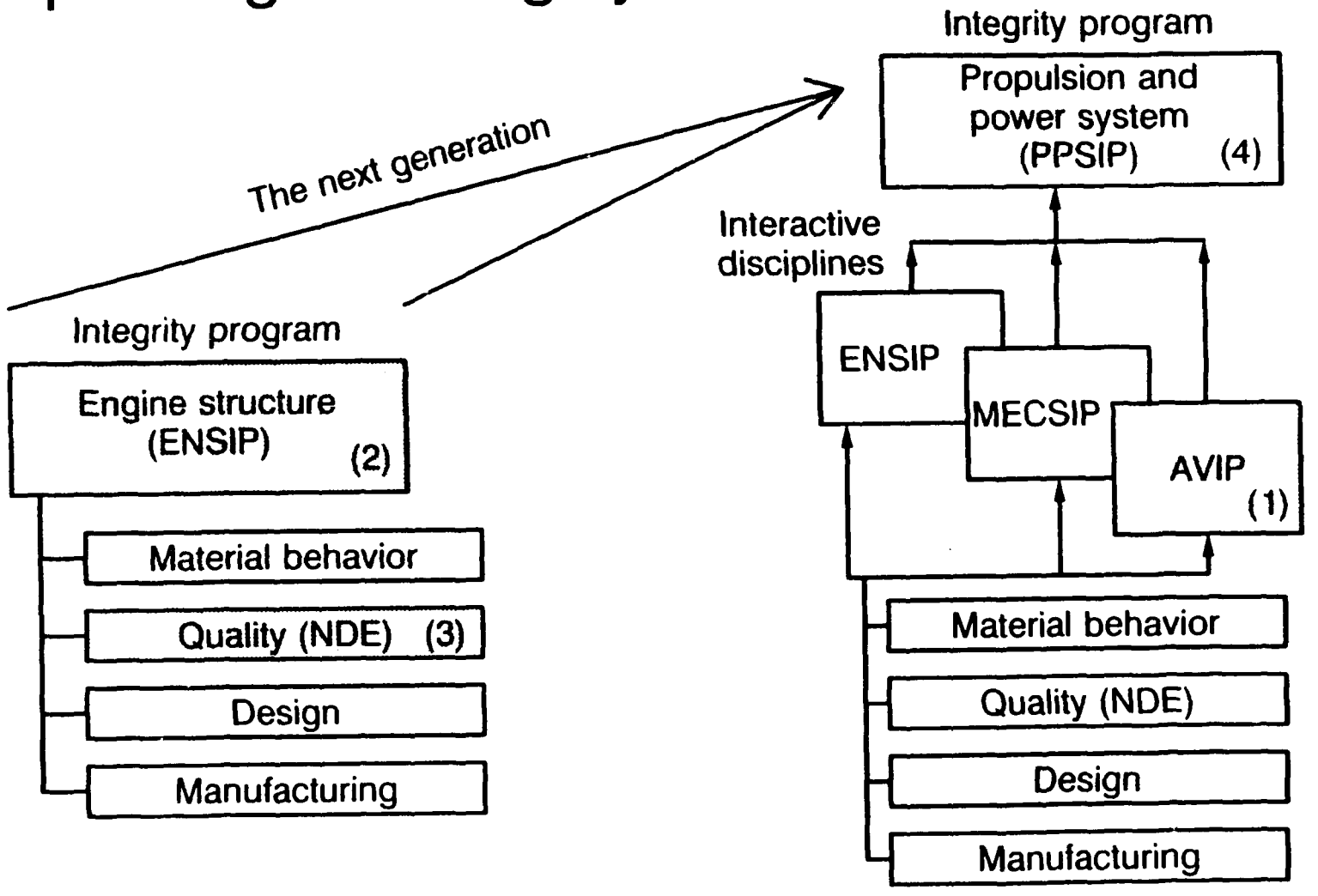
- PPSIP REPRESENTS AN INTEGRATION AND APPLICATION OF THE INTEGRITY PROGRAMS TO THE DESIGN, DEVELOPMENT, QUALIFICATION, AND LIFE MANGEMENT OF THE ATFE PROPULSION SYSTEM
- THE INTEGRITY PROCESS IS A VIABLE MEANS FOR IMPROVING THE RELIABILITY OF ENGINE CA & E THUS REDUCING LCC AND INCREASING WEAPON SYSTEM READINESS
  - PAST EXPERIENCE SUPPORTS THE NEED OF ADOPTING SUCH AN APPROACH TO THE DESIGN, DEVELOPMENT, MANUFACTURE, AND LIFE MANAGEMENT OF CA & E
  - CA & E INTEGRITY REQUIREMENTS HAVE INCORPORATED LESSONS LEARNED FROM PAST PROGRAMS WHILE INTRODUCING A DESIGN PERSPECTIVE THAT EMBODIES MISSION EFFECTIVENESS & ECONOMICS ALONG WITH SAFETY
  - EXPANSION OF THE PRODUCTION ACCEPTANCE REQUIREMENTS FOR CA & E TO INCLUDE ESS, ENHANCED INSPECTIONS AND PRODUCTION TEARDOWN EVALUATION IS FUNDAMENTAL TO CAPITALIZING ON THE INHERENT DESIGN CAPABILITY
- IMPLEMENTING PPSIP WILL ENHANCE RELIABILITY THROUGH IMPROVED DURABILITY AND QUALITY WHILE ALSO SERVING TO STRENGTHEN INDUSTRY'S KNOWLEDGE OF THE CAPABILITIES AND LIMITATIONS OF THEIR DESIGNS
- ALTHOUGH ADDING TO THE UPFRONT COSTS, THE INSTITUTION OF PPSIP WILL RESULT IN A POSIVITVE RETURN-ON-INVESTMENT IN TOTAL LIFE CYCLE COST
- THE RELIABILITY AND MAINTAINABILITY ENHANCEMENTS STEMMING FROM THE PURSUIT OF AN INTEGRITY PROCESS FOR ENGINE CA & E IS IN KEEPING WITH THE R&M 2000 GOALS & INITIATIVES

The Integrity Process as Applied to the Advanced Tactical Fighter Engine (ATFE)

1988 USAF Structural Integrity Conference
November 29 - December 1, 1988
San Antonio, Texas

Dr. P.A. Domas
GE Aircraft Engines
Cincinnati, Ohio

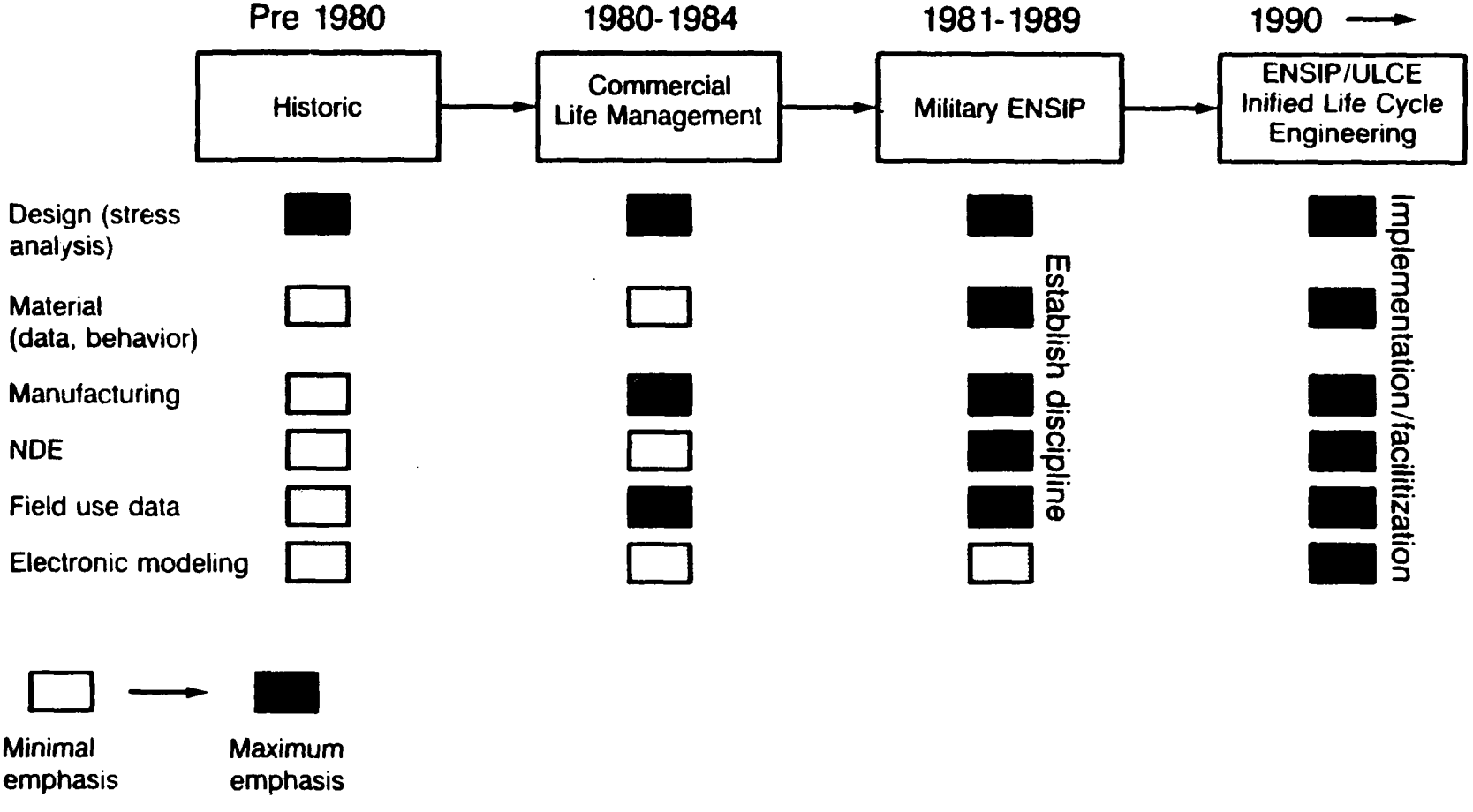
Expanding the Integrity Influence



Agenda

- Engine Structural Integrity Programs (ENSIP) - successful step toward unified (integrated) engineering for aircraft engines
- The expanded integrity challenge - beyond "structure"
- Meeting the challenge through commitment, innovation, and cooperation
- Propulsion and Power System Integrity (PPSIP) implementation for the ATFE

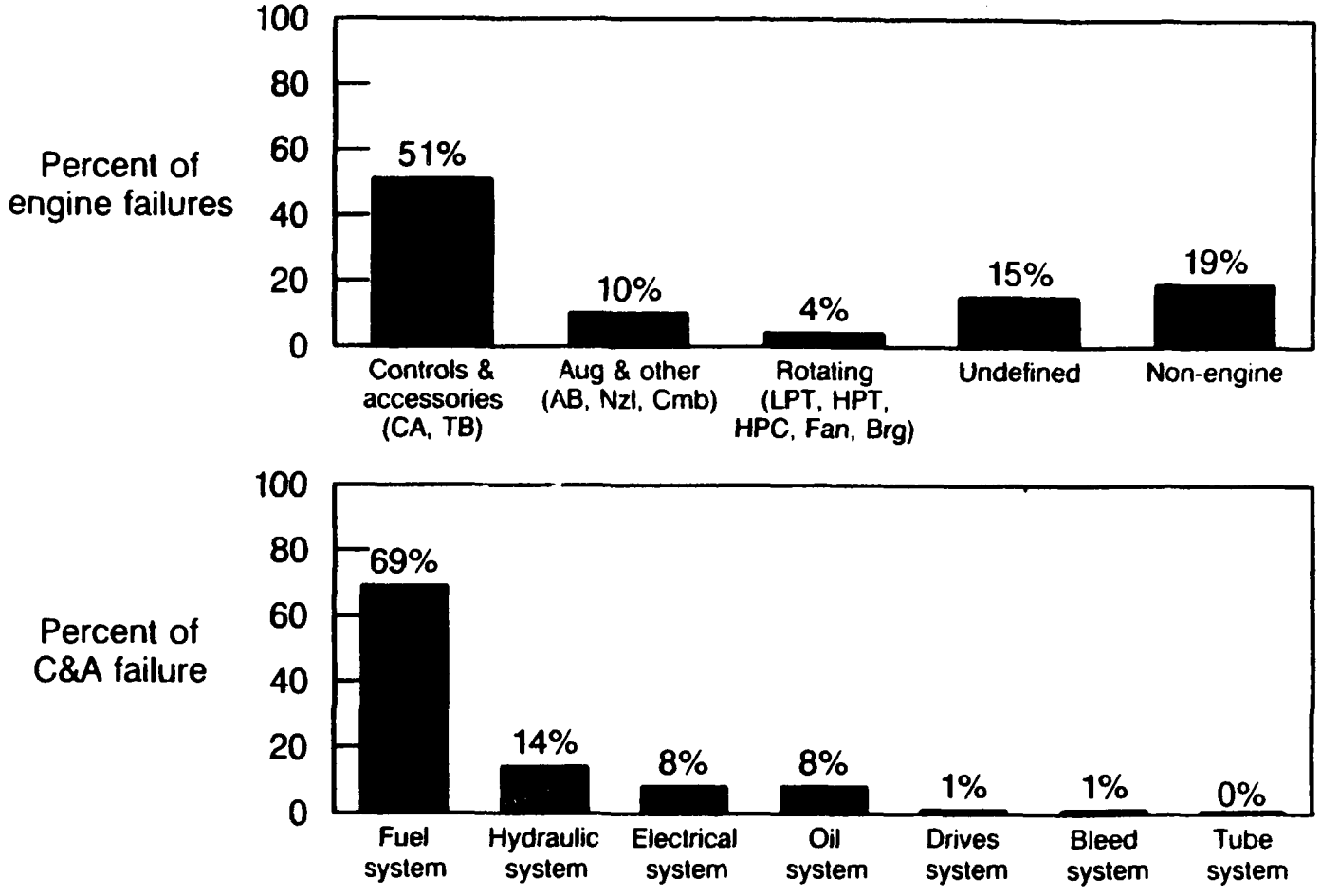
The Quest for Integration



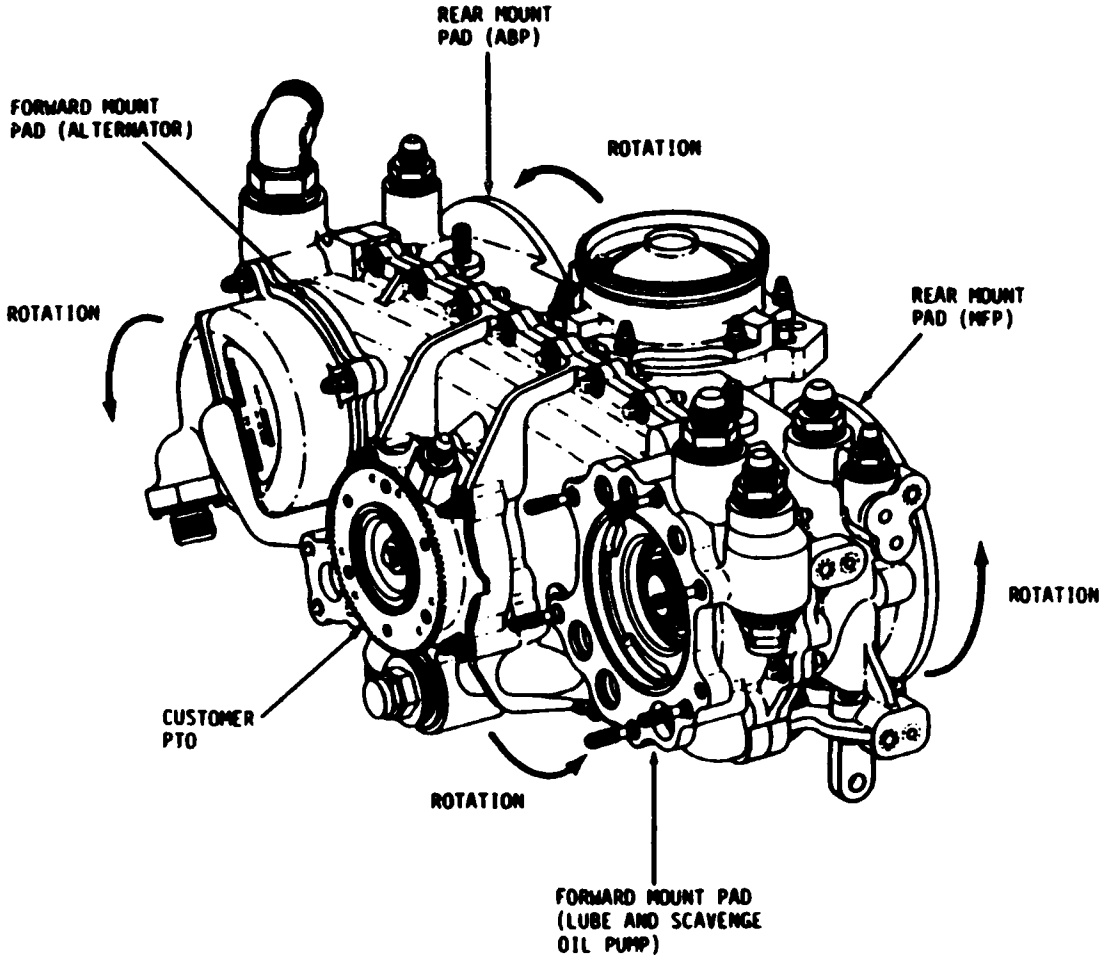
The Expanded Integrity Challenge

- Field data indicate needed control, accessory, and external component improvement
- Increased complexity
  - Geometry
  - Failure modes
- Increased quantity of parts addressed by integrity programs

In-Flight Engine Malfunctions GE-F404 Field Service Data

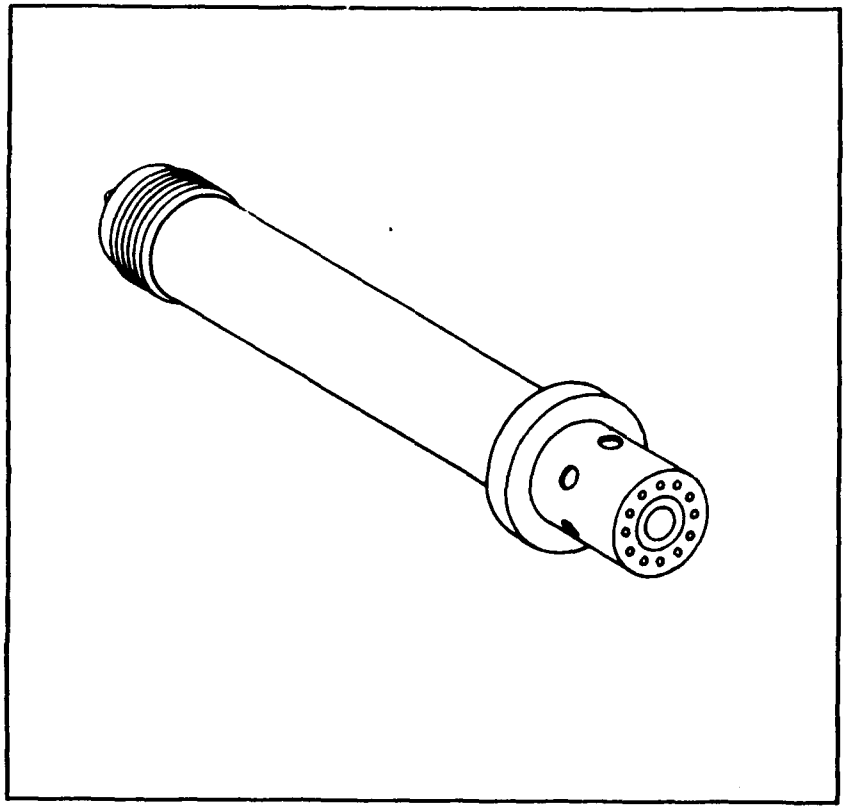


Typical Accessory Gearbox Example of Potential CA&E Complexity

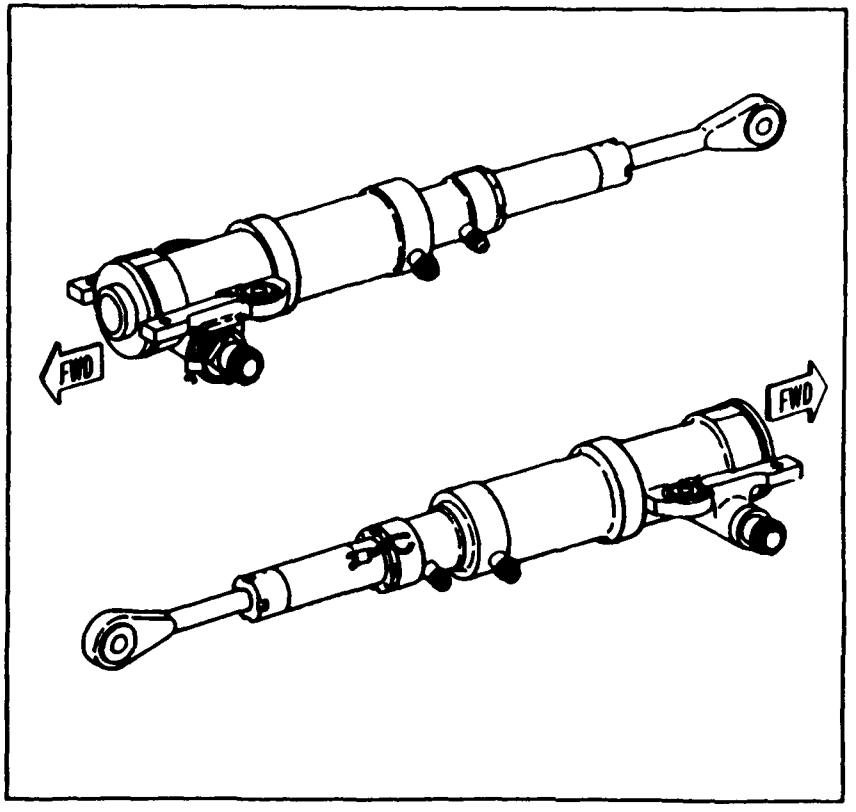


Other Typical CA&E Component Examples

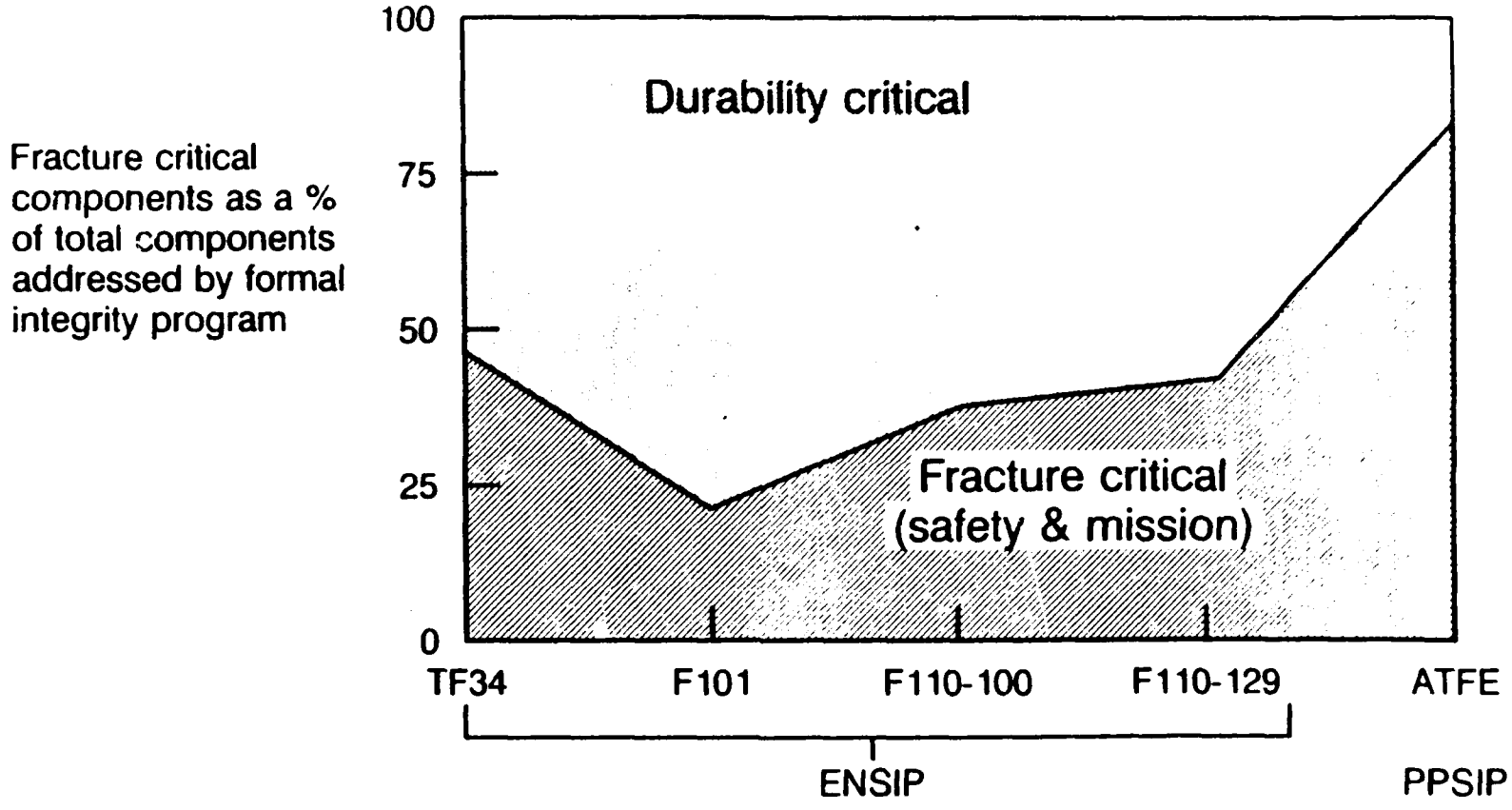
Igniter



Actuator



Increased Integrity Implementation Challenge



- Increased quantity of fracture critical parts
- Higher technology - more sophistication
- Greater structural/functional interdependence

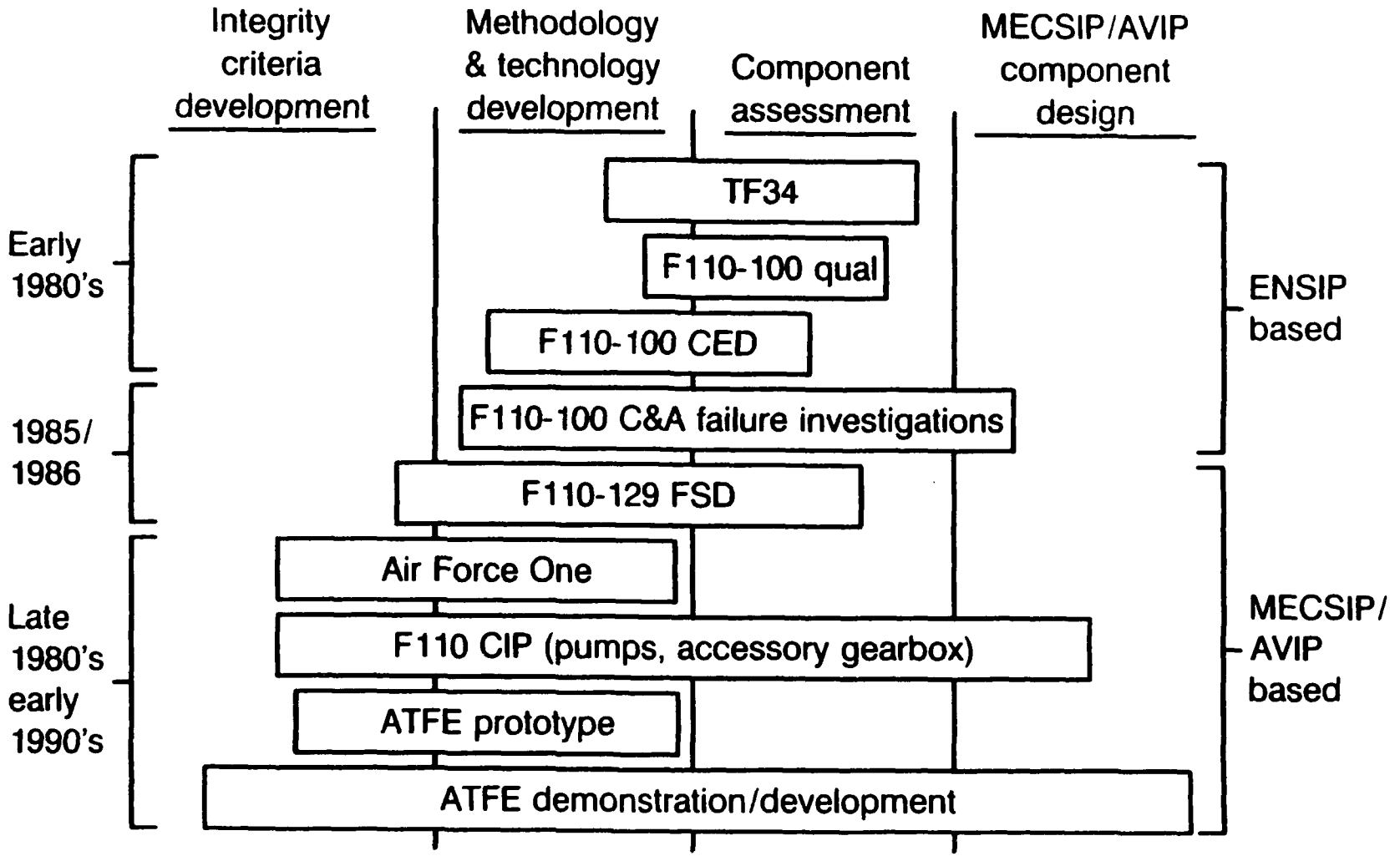
Typical CA&E Failure Modes

- Low and high cycle fatigue
- Fracture
- Yield
- Clogging/fouling
- Wear
- Leakage
- False sensor indications
- Short or open circuits
- Fluid starvation (oil/fuel/hydraulic)

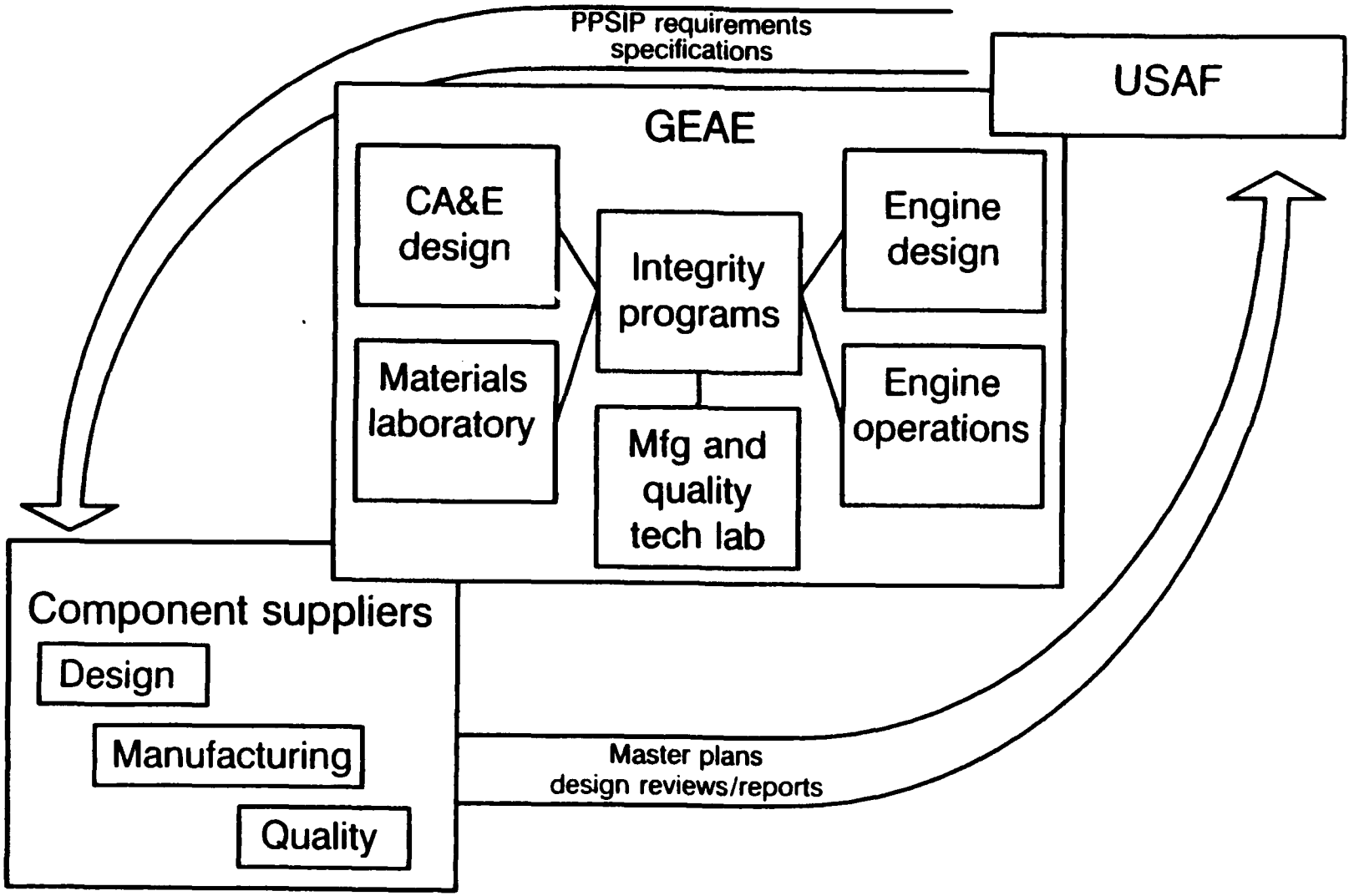
Meeting the Challenge

- ENSIP experience - PPSIP commitment to CA&E integrity
- Integrated customer - GEAE - supplier organization structure
- Building from established disciplines

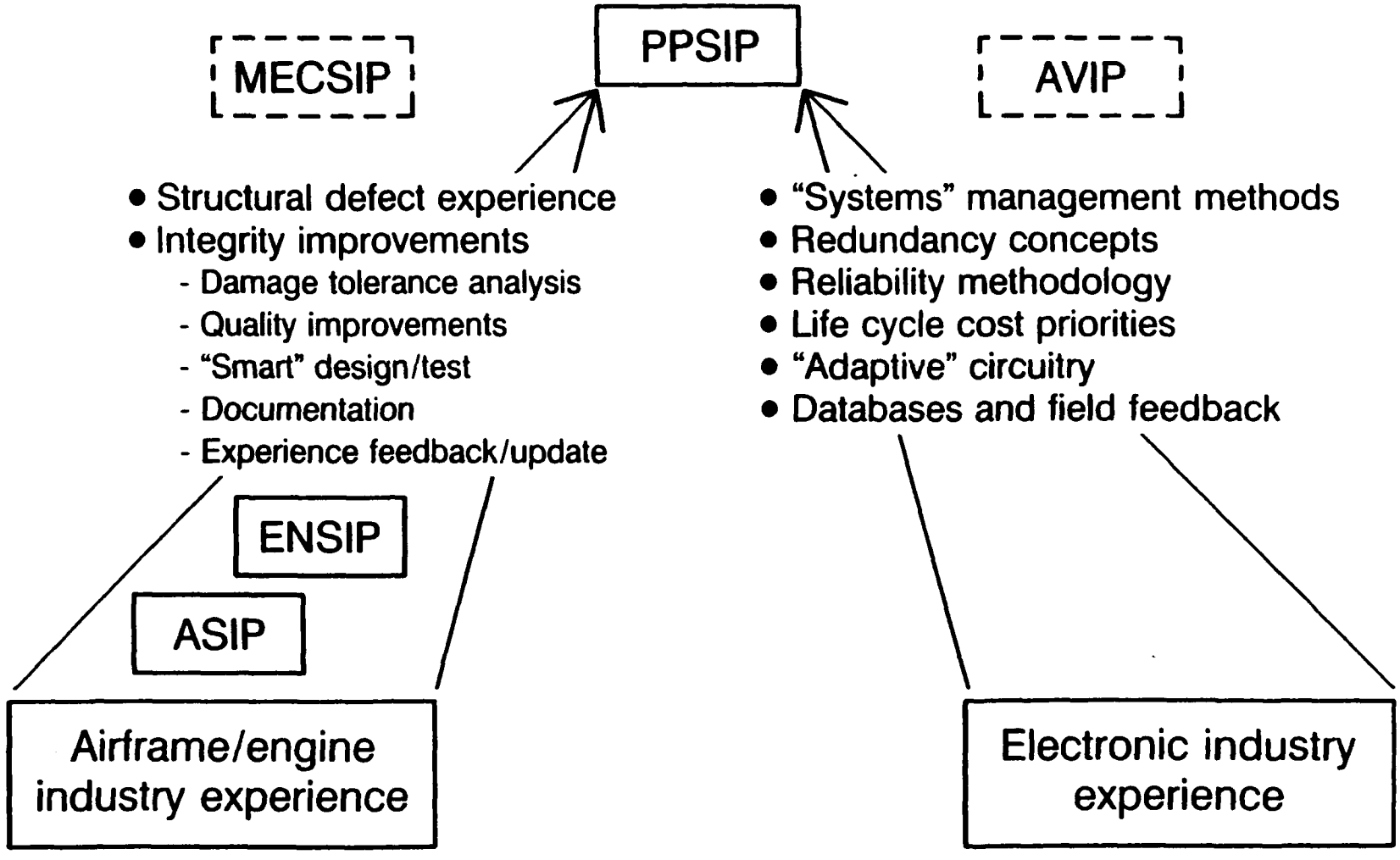
GEAE CA&E Integrity Programs



Integrated Organization Structure



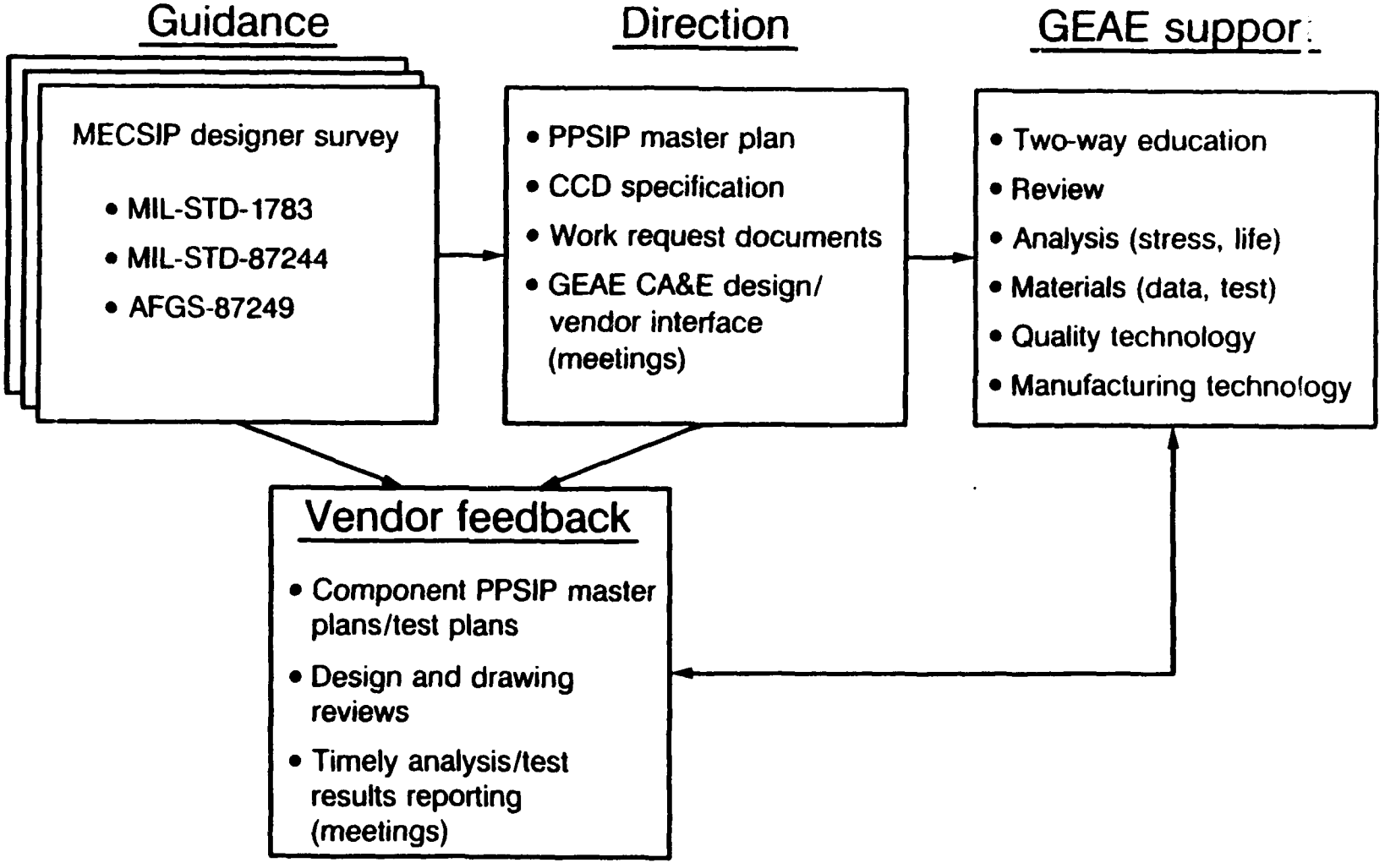
PPSIP Will Build from Two Experience Bases



PPSIP Implementation for the ATFE

- Integrated contract proposal, PID specification, and PPSIP master plan
- Generation and execution of Vendor Involvement Plan (VIP)
- Balanced enhancements to CA&E analysis and test procedures

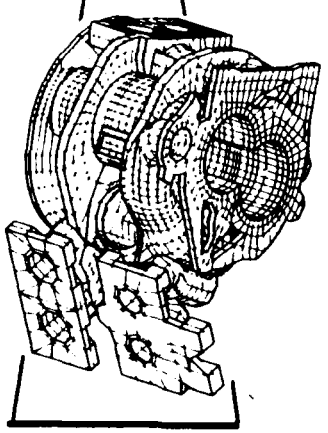
VIP (Vendor Involvement Plan)



Prudent Analysis Balanced By Prudent Testing

Elastic stress analysis
- Handbook formulas
- Design safety factors
- Material factors
- Etc.

Elastic/Plastic FEM



Predicted/documented
- Strength
- Wear
- Fatigue
- Damage tolerance

Well characterized usage environment
- Air, fluid
- Temperature
- Loads
- Actual load spectra

Disciplined state-of-art quality control
- In-process
- NDE
- "Green runs"
- Proof tests

"Smarter" standard tests
- Development
- Design assurance (DAT)
- Qualification
- Engine system
- Performance/acceptance (P/V)

Augmented and/or replaced with integrity focused tests
- Endurance
  - Piece part
  - Subcomponent
  - Component
- "AMT"
  - Mission simulation
  - Field run parts
- Damage tolerance proof

Verified/documented
- Strength
- Wear
- Fatigue
- Damage tolerance

Encompass the Old - Expand and Improve Where Needed

- Enhanced analysis (thermal, stress, life) where appropriate
- Engine mission driven strength and endurance testing (exercise components under expected usage environment)
- Development experience (success/failure) tracking (database)
- More detailed and interactive design reviews
- Improved quality control process (e.g. SPC, environmental stress screening)

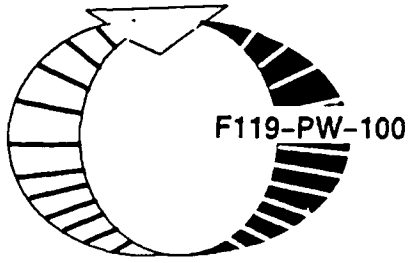
Summary

- Engine field experience shows a clear need for propulsion and power system integrity program
- ENSIP experience and technology advancements provide a firm base for expanding integrity programs to address all engine subsystems and components
- The quantity, diversity and deceptive complexity of the CA&E components presents a significant integrity challenge to component suppliers and engine contractors
  - Educating suppliers and flow-down of requirements
  - Cultural change to accept new and more disciplined approach
  - Building supplier technology base in support of aircraft/engine industry

Summary (Continued)

- GE Aircraft Engines is working with:
  - Internal design, laboratory, and support functions
  - Component suppliers
  - Weapon system contractors, and
  - The USAF

To implement a comprehensive yet cost effective integrity program for the ATFE



The Integrity Process As Applied To The F119-PW-100 Advanced Tactical Fighter Engine

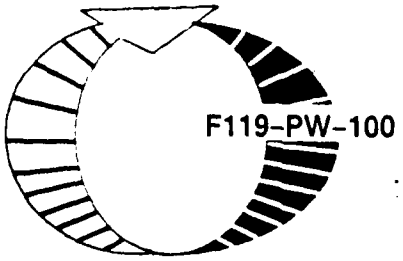
Presented By

Jerald D. Estes

Pratt & Whitney

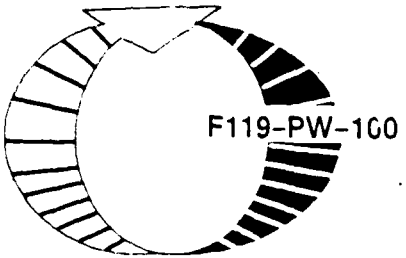
Government Engine Business

November 29, 1988



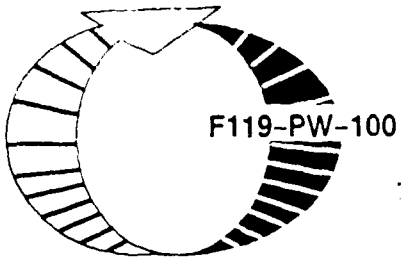
Integrity Program Experience at Pratt & Whitney

- PW5000 first military engine fully designed to meet current ENSIP requirements
- Broad background developed in:
  - Structural mechanisms
  - Materials
  - Inspection requirements
  - Fracture mechanics



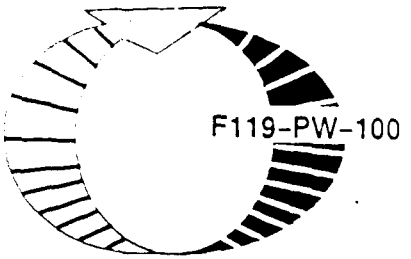
Integrity Program Experience at Pratt & Whitney

- Limited application to CA&E–designated safety critical components
  - Control castings
  - Prime reliable pumps
- Recent emphasis placed on quality
  - Total quality management (TQM)
  - Environmental stress screening
  - Nondestructive inspection
- Product Integrity organization
  - Independent audit
  - Engineering/material quality



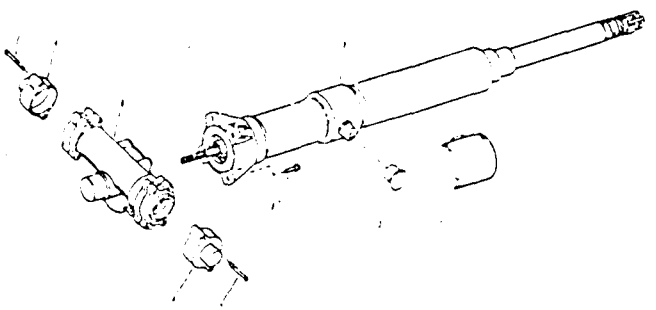
Total Quality Management - A Supplier Program To Continuously Improve The Manufacturing Process Through Process Control

- All Processes must be –
  - Documented
  - Followed
  - Reviewed
  - Continuously Improved
- Result –
  - All processes are stable, predictable & capable
    - Statistical Process Control
  - Measures of merit continue to improve
    - Reduce scrap, rework & repair rates
    - Decrease throughput time
    - Decrease in M.D. and field rejections
    - Decrease in product cost



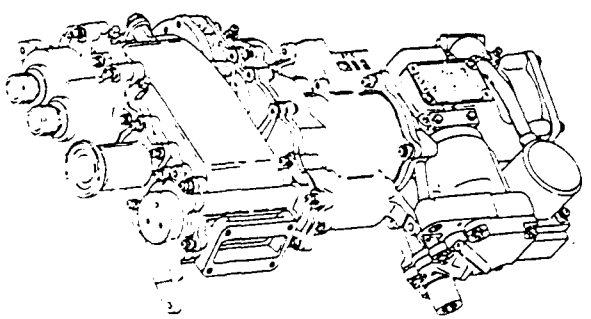
Component Problems in the Field Encompass Both Design and Quality Related Categories

F100 Exhaust Nozzle Actuator

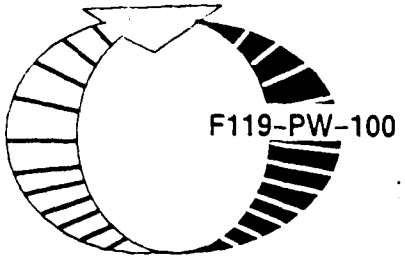


- Cross shaft bevel gear failure
- Broken quill shafts

F100 Exhaust Nozzle Control

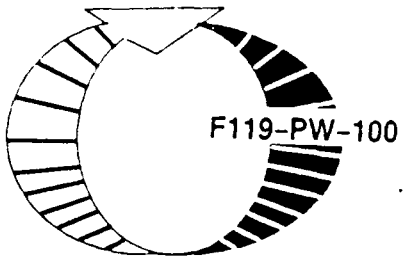


- Fuel leakage
- Defective o' rings
- Bearing failures



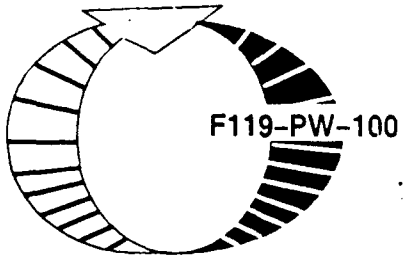
Past Component Specifications Did Not Emphasize Durability / Reliability

- Limited emphasis on actual environments/usage
- Designs structured to pass bench qualification tests (BQT)
- Reliance on engine accelerated mission test (AMT) to verify many design requirements
- Extended endurance testing (reliability growth testing–CERT) applied only to critical, control electronics



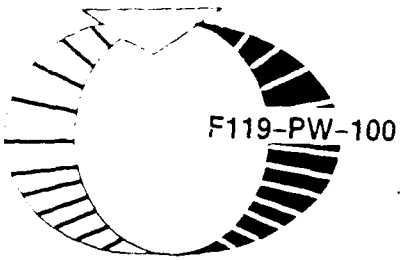
PPSIP (MECSIP / AVIP) Presents New Challenges

- Characterize the engine/component operational environment
- Component Classifications
  - Fracture (mission & safety) critical
  - Durability critical
  - Durability non–critical
- Up front detailed design analyses
  - Stress
  - Durability
  - Damage tolerance
- Redefinition of qualification testing
  - Environmental
  - Durability
  - Design verification
- Defining the "SMART" thing to do



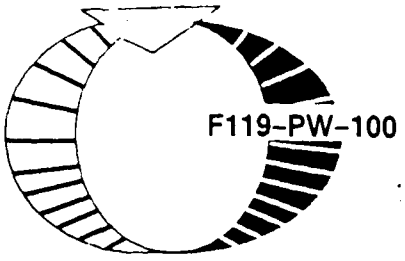
PPSIP (MECSIP / AVIP) Presents New Challenges

- Applying the requirements by component
- Better execution and documentation of analyses
  - Stress/strength
  - Vibration
  - Acoustic
  - Critical Speeds
  - Damage tolerance (mission & safety critical components)
  - Durability
  - Thermals
  - Duty cycles/loads



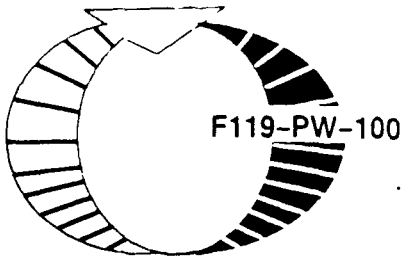
Component Design / Development Tasks Must Change To Meet The Integrity Program Requirements

- Instrument components to define the actual engine environment
- Characterize material behavior
- Expand mission/life analysis models to define component duty cycles
- Define application of PPSIP requirements to electromechanical interfaces on mechanical components
- Apply economic analysis to define non–durability critical parts
- Redefine qualification test (full life equivalency test)
- Define for producibility to reduce "Quality" related failures
- Reflect logistic environment in up–front analyses
    - storage
    - transportation
    - repair level / field support



Bringing The Component Supplier Up-To-Speed

- Developing the required design tools
- Doing the "smart" analyses / tests
- Iterate during the design process to define "margin" and "design drivers"
- Enhanced flowdown / documentation
    - PDR / DDR requirements
    - Comprehensive design reports



Formulating A Comprehensive, Yet Cost Effective Program

- Define best mix of analyses and testing
- Establish facilities for component full life testing
- Demonstrate full life at service introduction
- Goal: Elimination of "Component Improvement Programs"
    - Use CIP's only to enhance the product, not fix problems
- Bring the suppliers on–board
    - Make best use of supplier expertise
    - Change the supplier mindset
    - Provide the necessary design tools

MECHANICAL EQUIPMENT AND SUBSYSTEMS INTEGRITY PROGRAM

MECSIP

PRODUCT INTEGRITY PROGRAM

PRODUCT	PROGRAM	DOCUMENTATION	OPR
• AIRFRAME STRUCTURES • STRUCTURAL COMPONENTS	ASIP	MIL-STD-1530	ASD/ENF
• ENGINE STRUCTURES	ENSIP	MIL-STD-1783	ASD/ENF
• AVIONICS SYTEMS • ELECTRICAL COMPONENTS	AVIP	MIL-STD-1796	ASD/ENA
• MECHANICAL SYSTEMS • MECHANICAL COMPONENTS	MECSIP	MIL-STD-1798	ASD/ENF
• SOFTWARE	SDIP	MIL-STD-1803	ASD/EN

AFGS-87249

AIR FORCE GUIDE SPECIFICATION - "MECHANICAL EQUIPMENT AND SUBSYSTEMS REQUIREMENTS FOR THE INNTEGRITY OF"

(15 MARCH 1988)

- ESTABLISHES REQUIREMENTS AND VERIFICATIONS RELATED TO
  - SERVICE LIFE
  - OPERATIONAL USAGE
  - ENVIRONMENT
  - MATERIAL CHARACTERISTICS
  - DURABILITY
  - DAMAGE TOLERANCE
  - STRENGTH
  - MAINTAINABILITY
  - SUPPORTABILITY

MIL-STD-1798 (USAF)

MILITARY STANDARD "MECHANICAL EQUIPMENT AND SUBSYSTEMS INTEGRITY PROGRAM"

(20 JUNE 1988)

"SETS FORTH PROGRAMMATIC TASKS FOR THE DEVELOPMENT, ACQUISITION, MAINTENANCE, MODIFICATION, AND OPERATION OF MECHANICAL EQUIPMENT AND MECHANICAL ELEMENTS OF AIRBORNE, SUPPORT AND TRAINING SUBSYSTEMS . . ."

MIL-STD-1798
MECSIP SUPPLEMENTAL INFORMATION

- SUMMARIZE MECSIP TASKS IN RELATION TO SYSTEMS ACQUISITION, DEPLOYMENT AND OPERATION
- CROSS REFERENCES SPECIFIC TASKS WITH SPECIFIC VERIFICATIONS REQUIREMENTS OF AFGS-87249
- LISTS DATA REQUIREMENTS

THE MECSIP APPROACH FOR ASSESSING IN SERVICE MECHANICAL SYSTEMS

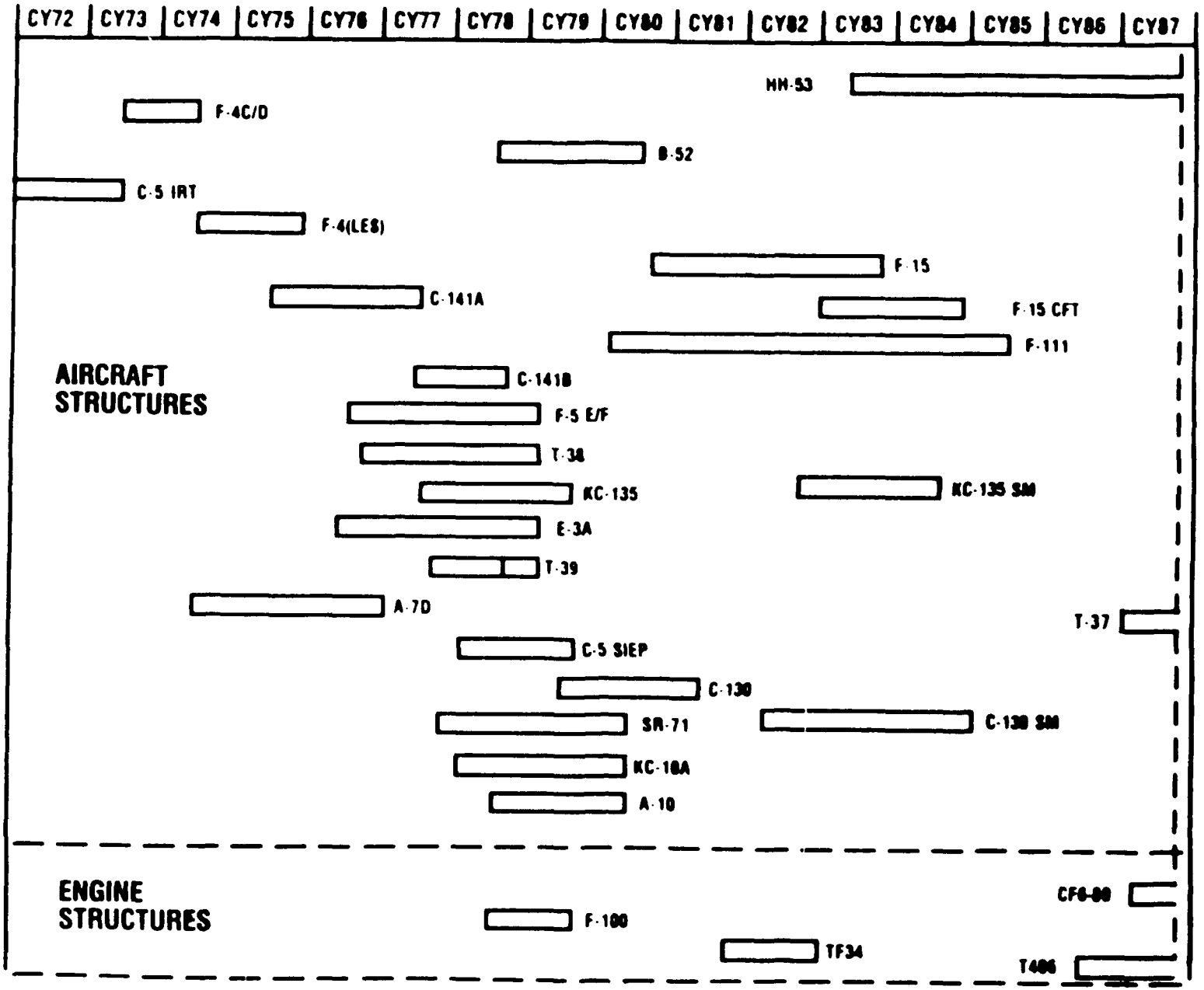
HOWARD A. WOOD
ASD/ENF

ASD 98 2733

PURPOSE OF ASSESSMENTS

- EVALUATE SELECTED FUNCTIONAL SUBSYSTEMS USING ESTABLISHED INTEGRITY PROCESS AND CRITERIA
- DEFINE:
  - REQUIRED TASKS AND OPTIONS TO PROTECT SAFETY AND MAINTAIN MISSION READINESS
  - MODIFICATION/IMPROVEMENT OPTIONS FOR LOW LIFE COMPONENTS/ELEMENTS WITH KNOWN OR PROJECTED HIGH MAINTENANCE COST
- DEVELOP STRATEGY AND PLAN FOR LOST EFFECTIVE IMPLEMENTATION OF FINDINGS

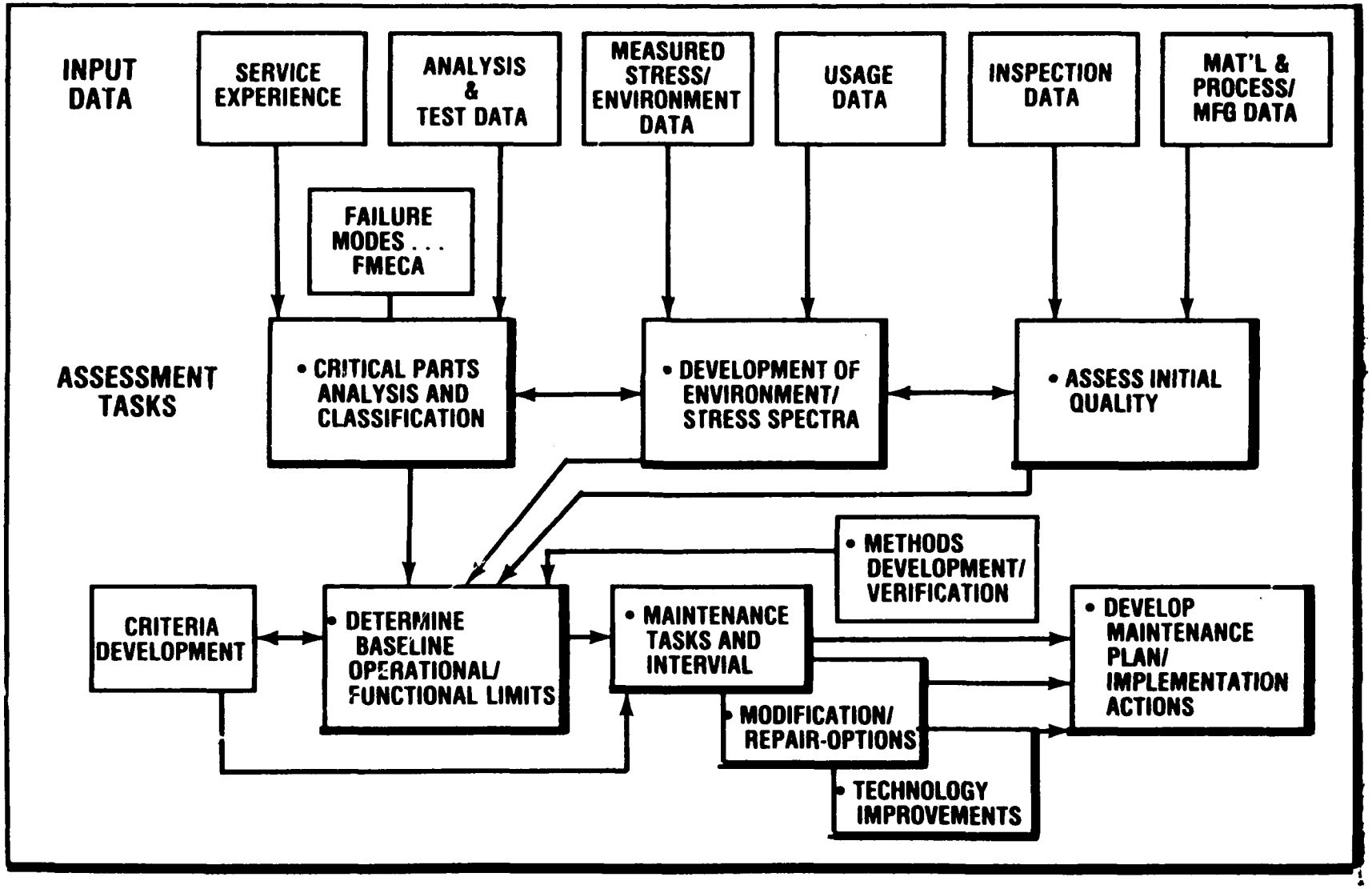
ASSESSMENT EXPERIENCES



ASSESSMENT FOCUS

- SYSTEMS CURRENTLY IN USE
- SYSTEMS UNDERGOING MODIFICATION
- NEW OR SIGNIFICANT CHANGES IN USAGE
- SYSTEMS DEVELOPED BY/FOR OTHERS (EG. FAA, NAVY)

TECHNICAL APPROACH



CRITICAL PARTS ANALYSIS & CLASSIFICATION

- **OBJECTIVE**
  - SELECT CANDIDATE COMPONENTS
  - CLASSIFY COMPONENTS (EG. SAFETY, MISSION CRITICAL, ETC.)
  - FOCUS FOLLOW ON EFFORT
- **BASIS/APPROACH**
  - REVIEW FIELD EXPERIENCES
  - REVIEW DEVELOPMENT DATA BASE/QUALIFICATION TESTS/ ANALYSES/MFG & QUALITY EXPERIENCES
  - ESTABLISH CLASSIFICATION CRITERIA
  - CONDUCT FAILURE MODES ANALYSIS
  - ASSEMBLE CANDIDATE LIST & PRIORITIZE

DEVELOPMENT OF STRESS/ENVIRONMENT SPECTRA

- **OBJECTIVE**
  - DEVELOP COMPONENT "USAGE CYCLES" AND STRESS/ENVIRONMENT SPECTRA TO SUPPORT LIFE ANALYSES
- **BASIS/APPROACH**
  - ASSEMBLE OPERATIONAL USAGE/ENVIRONMENT DATA AS AVAILABLE (EG. MEASURED STRESSES, PRESSURES, THERMAL CYCLES, EVENTS, ETC.)
  - CONDUCT NEW ANALYSES AS APPROPRIATE
  - EXAMINE VARIABILITY IN USAGE-GROUP POPULATIONS
  - DEVELOP SPECTRUM/DUTY CYCLE FOR EACH COMPONENT/CRITICAL PART/CRITICAL AREA

ASSESS INITIAL QUALITY

- OBJECTIVE
  - ESTABLISH INITIAL BASIS (STARTING POINT) FOR DETERMINING BASELINE OPERATIONAL LIMITS
- BASIS/APPROACH
  - SHOULD REFLECT INITIAL MFG CONDITION OF COMPONENTS
  - INITIAL FLAW SIZE CONCEPT UTILIZED FOR STRUCTURAL ASSESSMENTS OF AIRFRAME AND ENGINE STRUCTURES
  - FLAW SIZE APPROACH APPLICABLE TO METALLIC COMPONENTS/ELEMENTS
  - OTHER CONCEPTS MAY BE REQUIRED
    - REFLECT MATERIALS/MFG CONCEPTS
    - CAN BE RELATED TO ANALYSIS/EMPIRICAL METHODS

DETERMINE BASELINE OPERATIONAL LIMITS

- **OBJECTIVE**
  - ESTABLISH FUNCTIONAL/LIFE LIMITS FOR CRITICAL COMPONENTS
- **BASIS/APPROACH**
  - PREDICT OPERATIONAL LIVES IN TERMS OF BASELINE USAGE FOR CRITICAL/FUNCTIONAL MODES (EG. LEAKAGE, WEAR STRUCTURAL CRACKING, STRUCTURAL STRENGTH, ETC.)
  - DEVELOP AND VERIFY LIFE PREDICTION METHODOLOGY
  - DEVELOP EMPIRICAL DATA BASE AS APPROPRIATE
  - DETERMINE SENSITIVITIES TO VARIATIONS IN BASELINE USAGE

DETERMINE MAINTENANCE TASKS & INTERVALS

- **PURPOSE**
  - IDENTIFY PREVENTIVE MAINTENANCE TASKS/OPTIONS AND INTERVALS FOR CRITICAL COMPONENTS AND CRITICAL FUNCTIONAL MODES
- **BASIS/APPROACH**
  - EMPHASIS ON SAFETY/MISSION CRITICAL COMPONENTS
  - ESTABLISH CRITERIA
    - DEVELOP INTERVALS IN TERMS OF BASELINE USAGE AND OPERATIONAL LIMITS

DETERMINE MODIFICATION/REPAIR OPTIONS/TECHNOLOGY IMPROVEMENTS

- **OBJECTIVE**
  - IDENTIFY OPTIONS FOR LOW LIFE COMPONENTS OR COMPONENTS WITH KNOWN OR PROJECTED HIGH MAINTENANCE COST
- **BASIS/APPROACH**
  - ESTABLISH OPTIONS TO EXTEND LIVES THROUGH MOD/REPAIR
  - IDENTIFY AND QUANTIFY OPTIONS, POST MOD MAINTENANCE TASKS AND INTERVALS
  - EVALUATE COST/BENEFITS

DEVELOP MAINTENANCE PLAN/IMPLEMENTATION ACTIONS

- **OBJECTVE**
  - IDENTIFY PREFERRED MAINTENANCE ACTIONS - PLAN IMPLEMENTATION STRATEGY
- **BASIS/APPROACH**
  - USE ASSESSMENT RESULTS TO DEVELOP PREFERRED ACTIONS
  - DEVELOP MOST COST EFFECTIVE APPROACH FOR IMPLEMENTATION (EG. HOW, WHAT, WHERE, WHEN, HOW OFTEN)
  - INTEGRATE INTO OPERATIONAL PLANNING (EG. FACILITIES, MANPOWER, ETC.)
  - IDENTIFY OPR'S, FUNDING REQUIREMENTS AND SCHEDULING

RELATIONSHIP TO RCM PROCESS

- RCM PROCESS ~ IDENTIFY MAINTENANCE TASKS TO REALIZE INHERENT RELIABILITY OF EQUIPMENT

- INSPECTIONS/MAINTENANCE TASKS IN RCM UTILIZE ASIP AND ENSIP STRUCTURAL DATA (EG. FORCE STRUCTURAL MAINTENANCE PLAN)

- INSPECTIONS/MAINTENANCE TASKS FOR SAFETY AND MISSION CRITICAL COMPONENTS AND EQUIPMENT SHOULD BE BASED ON THE MECSIP AND AVIP DATA BASE

ADDITIONAL CONSIDERATIONS

- IMPLEMENTATION MAY REQUIRE TRACKING OF USAGE FOR SOME INDIVIDUAL COMPONENTS TO ALLOW ADJUSTMENT AND SCHEDULING OF REQUIRED MAINTENANCE TASKS - FOCUS SHOULD BE ON SAFETY AND MISSION CRITICAL COMPONENTS

SUMMARY

- ASSESSMENT APPROACH AIMED AT EVALUATING "IN-SERVICE" SUBSYSTEMS AND EQUIPMENT TO AIR FORCE INTEGRITY PHILOSOPHY/CRITERIA
- AIRCRAFT AND ENGINE STRUCTURAL ASSESSMENT EXPERIENCE IS EXTENSIVE - PAYOFF AND BENEFITS CLEARLY DEMONSTRATED
- LIMITED EXPERIENCES FOR OTHER TYPES OF EQUIPMENT
- CHALLENGE - MAKE IT WORK EFFECTIVELY

THE DEVELOPMENT OF IMPROVED NDI/E TO MEET STRUCTURAL INTEGRITY REQUIREMENTS

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WRDC/Materials Laboratory

The availability and application of accurate, reliable nondestructive inspection and evaluation (NDI/E) methods and procedures are recognized as a cornerstone of a successful structural integrity program. As summarized in Figure 1, MIL-A-83444 establishes requirements and guidelines for linking certain critical design factors to our demonstrated ability to detect consistently small flaws in both manufacturing and in-service operational settings. Without such demonstrated NDI/E capabilities, the existence of larger threshold flaws must be assumed initially and a more conservative (less efficient) damage tolerant design must be adopted.

AIRPLANE DAMAGE TOLERANCE REQUIREMENTS MIL-A-83444

- ASSUMES FLAWS EXIST IN CRITICAL COMPONENTS
- QUANTITATIVELY DESCRIBES THEIR EFFECT
- REQUIRES GUARANTEE OF LARGEST SIZE FLAW THAT CAN GO UNDETECTED IN MANUFACTURE
- REQUIRES DEMONSTRATION OF MANUFACTURING/ FIELD INSPECTION CAPABILITIES

FIGURE 1

DEFICIENT NDI/E CAPABILITIES

With the change some years ago to a damage tolerance design and associated force life management philosophy for US Air Force structural systems came a much

greater recognition of the vital (and frequently controlling) role of NDI/E throughout the system life cycle (Figure 2). However, most of the deployed

FIGURE 2

IMPACT OF NDI/E ON SYSTEMS PERFORMANCE

- IN DESIGN OF CRITICAL COMPONENTS
  - CONFIGURATION, MATERIALS SELECTION LINKED TO NDI/E CAPABILITIES
- IN MANUFACTURING
  - NDI/E A KEY ELEMENT OF PRODUCT QUALITY ASSURANCE
    - PROCESS CONTROL
    - INSPECTION AGAINST SPECIFICATIONS
    - MANUFACTURING DEFECTS
- IN SERVICE
  - MONITORING EXPECTED DAMAGE / DETERIORATION
  - AVOIDANCE OF SUDDEN FAILURES FROM UNKNOWN FLAWS
  - APPLICATION OF RETIREMENT-FOR-CAUSE

FIGURE 3

MIL-A-83444,
"AIRPLANE DAMAGE TOLERANCE REQUIREMENTS"
ASSUMED IN-SERVICE NDI CAPABILITIES

CONDITION	GEOMETRY	FLAW SIZE
REMOVED FROM AIRCRAFT	ALL	SAME AS PRODUCTION NDI
ON AIRCRAFT ACCESSIBLE	HOLE	0.250 UNCOVERED LENGTH
	OTHER	0.500 x 0.250
ON AIRCRAFT UNACCESSIBLE	ALL	UNINSPECTABLE

methods and procedures at the time fell short of requirements and many still do today. Figure 3 outlines flaw sizes to be assumed present by the designer in a manufactured airframe component in the absence of a demonstrated capability to detect smaller sizes with 90% probability at a confidence limit of 95%. Figure 4

outlines criteria for in-service NDI/F. Figure 5 presents a similar outline of design flaws adopted within the Engine Structure Integrity Program (ENSIP). The perception of many was that these threshold requirements were "not only meetable but exceedable". However, as shown in subsequent major undertakings by the Air Force Logistics Command to measure in-service inspection capabilities (1-2), the required detection performance requirements could not, for the most part, be met.

FIGURE 4
MIL-A-83444,
"AIRPLANE DAMAGE TOLERANCE REQUIREMENTS"
ASSUMED PRODUCTION NDI CAPABILITIES

SLOW CRACK GROWTH			FAIL-SAFE		
GEOMETRIC FEATURE	MATERIAL THICKNESS	FLAW SIZE,TYPE	GEOMETRIC FEATURE	MATERIAL THICKNESS	FLAW SIZE,TYPE
HOLE	≤ Ø.050"	0.050" A	HOLE	≤ 0.020"	0.020" A
HOLE	> 0.050"	0.050" B	HOLE	> 0.020"	0.020" B
OTHER	≤ 0.125"	0.250" A	OTHER	≤ 0.050"	0.100" A
OTHER	> 0.125"	0.125 x 0.250" C	OTHER	> 0.050"	0.050 x 0.100" C

FLAW TYPES

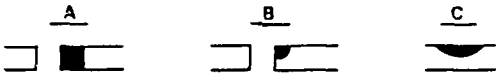


Figure 5

- SUGGESTED DESIGN FLAW:
  - 0.030" LENGTH, OR 0.015 x .015 CORNER
- SUGGESTED PRODUCTION NDI CAPABILITY

METHOD(S)	FLAW TYPE	SIZE	COMMENT
FPI, MPI	SURFACE	0.030" LENGTH	NO ASPECT RATIO DEFINED
FPI	SURFACE	0.200" x 0.010"	WELDMENT, AS WELDED
ET, UT	SURFACE	0.015" LENGTH	
UT	IMBEDDED	0.002 in³	
RT	IMBEDDED	SPHERE, 20% WELD THICKNESS	

- DEMONSTRATION REQUIRED

MIL-STD 1783 ENGINE
STRUCTURAL INTEGRITY PROGRAM (ENSIP)

Figure 6 shows results of depot and field level NDI of service-damaged airframe components. Figure 7 illustrates depot inspection results on flawed engine component specimens. There are some straight forward reasons why an NDI/E

FIGURE 6

RELIABILITY OF AIRFRAME INSPECTION METHODS
(AF "HAVE CRACKS-WILL TRAVEL" PROGRAM - 1978)

METHOD / APPLICATION	DETECTABLE FLAW LENGTH-MEAN CURVE (INCHES) 75% PROB. OF DETECTION
• EDDY CURRENT / RADIAL LENGTH AROUND COUNTERSUNK FASTENER HEAD	0.380
• ULTRASONIC SHEAR / " HAND SCAN	0.880
• ULTRASONIC SHEAR / " SEMI-AUTO	0.070
• RADIOGRAPHIC / "	0.750
• PENETRANT / RADIAL LENGTH-EDGE CRACKS IN WING RISER	0.120
• ULTRASONIC / RADIAL LENGTH-EDGE CRACKS IN WING RISER	0.300
• EDDY CURRENT / AXIAL LENGTH IN BOLT HOLES HAND ROTATIONAL SCAN	
SAMPLE E	> 1.0
SAMPLE F	>> 1.0
• EDDY CURRENT / AXIAL LENGTH IN BOLT HOLES AUTOMATIC SCAN	
SAMPLE E	0.080
SAMPLE F	0.220

FIGURE 7

ENGINE NDI RELIABILITY STUDY MEASURED CAPABILITIES

INSPECTION TECHNIQUE	BEST INDIVIDUAL FLAW SIZE/ POD	AVERAGE DEPOT FLAW SIZE/ POD
MAGNETIC PARTICLE	.250/65%	.300/60%
PENETRANT - AUTOMATED LINE	.175/85%	.230/75%
PENETRANT - HAND PROCESS	.125/90%	.240/85%
PENETRANT - ELECTROSTATIC	.175/90%	.220/90%
PENETRANT - BATCH PROCESS	.125/90%	.190/75%
EDDY CURRENT - STATIC	.150/30%	.200/30%
EDDY CURRENT - DYNAMIC	.030/98%	.090/80%
ULTRASONIC SURFACE WAVE	.180/90%	.375/80%

(PROBABILITY OF DETECTION (POD) FIGURES ARE GIVEN FOR DATA SAMPLING AT A 95% CONFIDENCE LEVEL.)

capability shortfall in meeting many requirements has existed and continues to exist and several key factors are shown in Figure 8. Inherent in all of these is

FIGURE 8

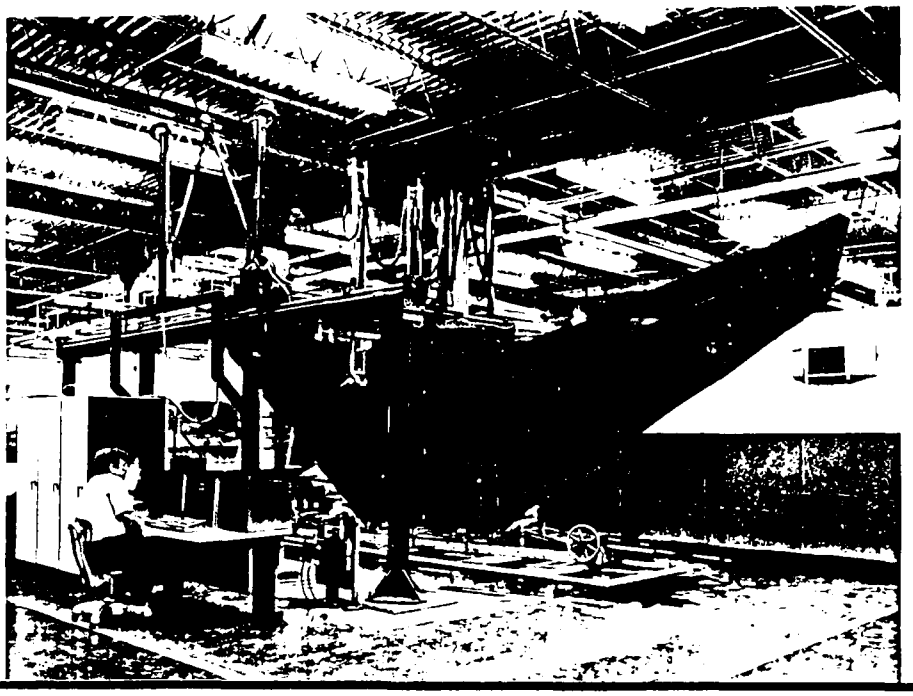
LIMITATIONS IN NDE CAPABILITIES

- METHODS / EQUIPMENT / PROCEDURES MAY BE INADEQUATE FOR JOB
  - REQUIREMENTS MAY EXCEED STATE-OF-THE-ART
  - APPROACH MAY NOT BE OPTIMUM
  - INSPECTION ENVIRONMENT MAY BE LIMITING
- DEFECT CONDITION MAY BE INHERENTLY DIFFICULT TO FIND / IMAGE
- SOME CANDIDATE APPROACHES MAY BE TOO COMPLEX, COSTLY TO USE

the fact that the NDI/E technology area has only relatively recently been a focus of significant research and development emphasis and much work remains to be accomplished to deal with methods, equipment and procedure inadequacies to to the job. In some important ways, advancements in manufacturing NDI/E have been much easier to bring about than in the case of in-service (depot, field) operations. Figure 9 exemplifies how computer-automated ultrasonic scanning and data processing methods have been coupled with precision part fixturing to produce highly reliable inspection procedures - in this instance for a graphite-epoxy composite wing cover. In contrast, many in-service NDI/E tasks such as illustrated in Figure 10 are, by necessity or because of facility/procedural limitations, manual labor intensive and more variable in nature. We have also experienced cases where some flaws or flaw locations are nearly inaccessible due either to geometry complexities or an inability to secure an image of or signal from the damage. Inner structure flaws and hidden corrosion are examples. Finally, some new flaw scanning methods have been proposed, but because they depend on complicated or expensive procedures, appear to have doubtful utility for most inspection tasks. Vibration-induced thermal methods, optical holography and acoustic emission are examples.

COMPUTER AUTOMATED ULTRASONIC INSPECTION OF GRAPHITE-EPOXY WING COVER

FIGURE 9



TYPICAL DEPOT NDI/E PROCEDURES

FIGURE 10

AIR FORCE NDI/E R&D RELATED TO ASIP/ENSIP

The Air Force Systems Command conducts our ongoing R&D program to help in the development of new and improved NDI/E technology to enhance the inspection capabilities used as part of the ASIP/ENSIP program. This R&D initiative, in a broader sense, is also required as a supporting effort by Air Force Regulation 66-38 "Nondestructive Inspection Program" which governs the role and performance of NDI/E in the design and logistics support of weapon systems throughout their life cycle. Figure 11 gives an overall objective statement for the R&D efforts.

Figure 11

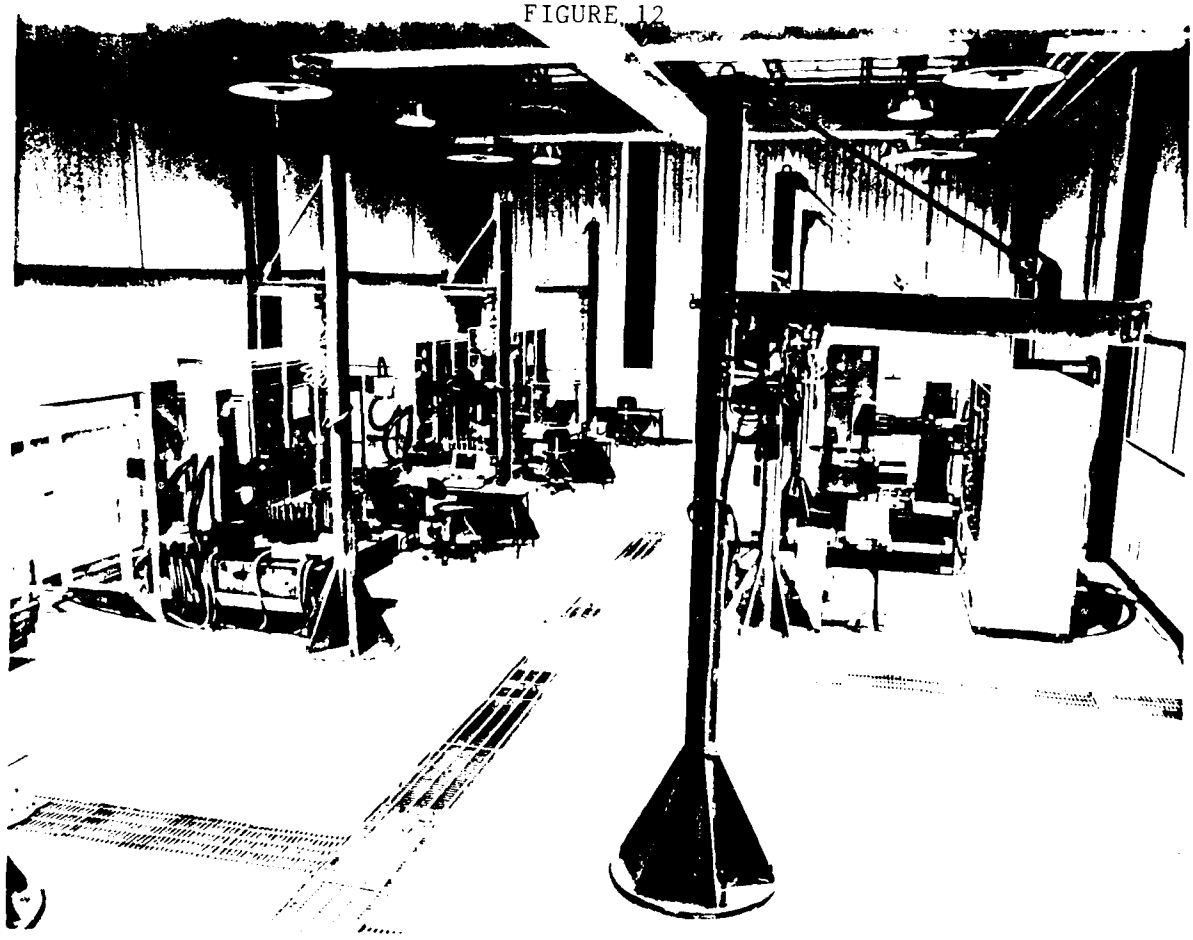
NDE DEVELOPMENT DIRECTIONS

- REFINEMENTS IN
  - METHODS
  - PROCEDURES
  - EQUIPMENT
- COMPUTER TECHNOLOGY INTEGRATION
- FLAW / FEATURE IMAGING
  - INCLUDING IMAGE ENHANCEMENT TECHNOLOGY
- EVOLUTION / INTRODUCTION OF NEW METHODS
- MODELING TO IMPROVE INHERENT INSPECTABILITY
  - IMPROVEMENT OF POD / CL
- EMPHASIS ON
  - INCREASED SENSITIVITY / CONSISTENCY (RELIABILITY)
  - COST EFFECTIVENESS / OPTIMUM SIMPLIFICATION
  - KEEPING PACE WITH ADVANCED MATERIALS / STRUCTURES

Successful Technology Development and Transfer Examples

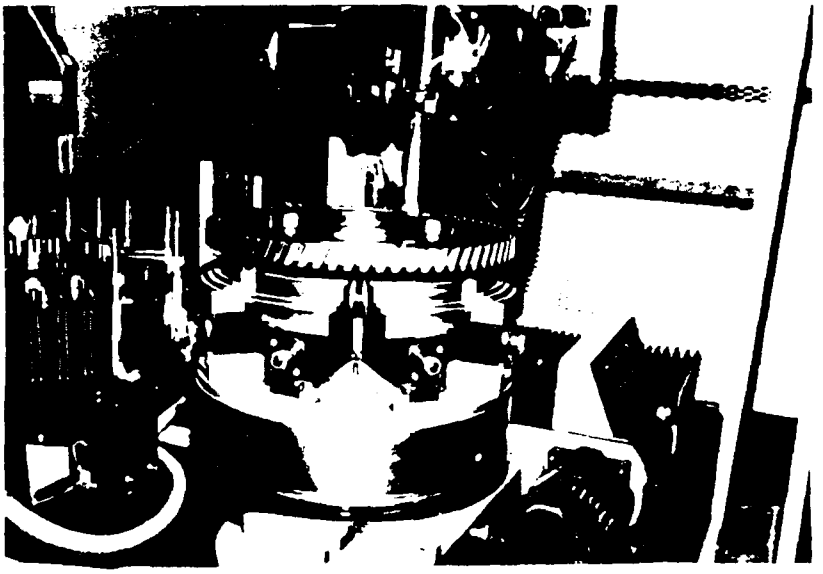
Significant NDI/E improvements have been implemented recently to increase flaw detection capabilities in several key weapon system life management applications. One is the new turbine engine disk inspection system at Kelly Air Force Base, Texas, developed to support periodic inspection and life extension of the F100 turbine engine through the process termed "retirement for cause" (e.g., retirement upon detection of a safety limit crack versus retirement at a prescribed safe usage life without regard to the presence or absence of a crack). Shown in Figure 12 is a state-of-the-art, computer based, semi-automated inspection system consisting of eddy current stations (module) on the left and an ultrasonic "squinter" station on the right (floor space for a fourth eddy current

FIGURE 12



module and second ultrasonic module are seen in the foreground)(3). Seen in Figure 13 is a close-up of an eddy current module. The seven axis probe manipulation robot is shown automatically lowering a rotating probe into a turbine

FIGURE 13



disk bolt hole as part of the programmed scan plan. As outlined in Figure 14, this "RFC NDE System" has performed a significant workload since implementation, exceeding system consistency reliability goals. A similar system is scheduled for installation soon at a second engine maintenance depot.

In another development effort, a unique X-ray computed tomography (CT) system has been developed with AF Materials Laboratory sponsorship by General Electric Engine Business Group to inspect turbine engine blades with internal cooling

FIGURE 14

RFC EXPERIENCE - F-100 ENGINE

- OCTOBER 1986 THRU JULY 1988
- 4553 PARTS INSPECTED
- 1275 PARTS INITIALLY ACCEPTED
- 2449 PARTS ACCEPTED AFTER RE-EVALUATION
- 829 PARTS SENT TO MRB
  - VERIFIED CRACKS REJECTED (6 TOTAL)
  - UNVERIFIED INDICATION RELEASED FOR SERVICE, INSPECTION INTERVAL ONE-HALF (1800 CYCLES) THAT OF RFC ACCEPTED PARTS (3600 CYCLES)

geometries (4). This CT system is able to reconstruct cross section image slices, as shown in Figure 15, perform computerized measurement and comparison of internal wall thickness with specified values and automatically accept or reject blades accordingly. Not only does this new capability replace certain slow, manual, subjective, qualitative-only and less accurate inspection operations, but it also produces for the first time quantitative measurements. A factory-installed system for production NDI/E is pictured in Figure 16.

FIGURE 15

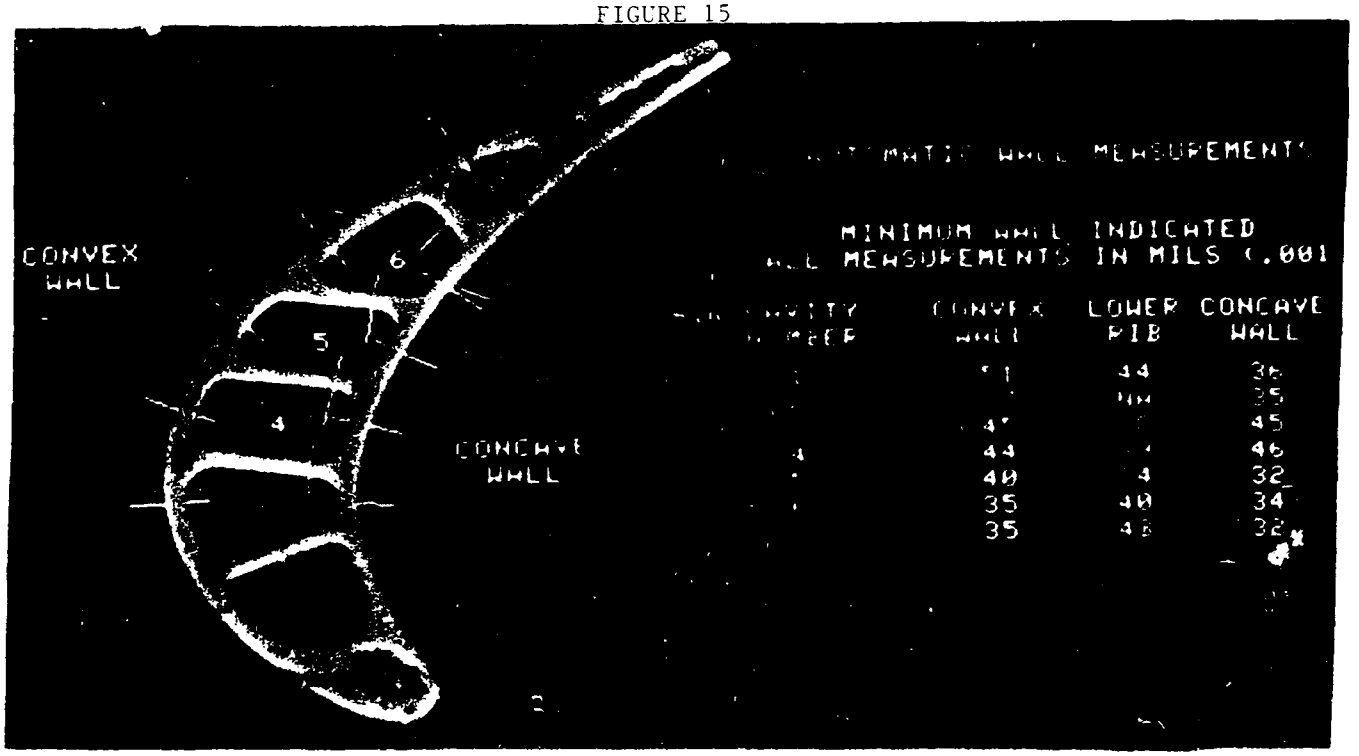
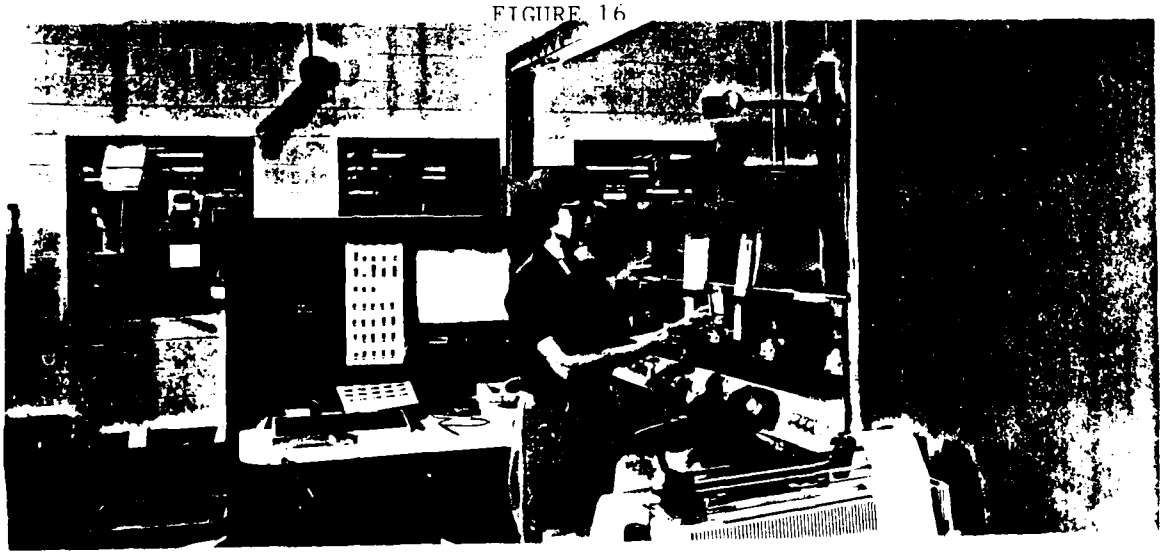


FIGURE 16



In response to a critical airframe wing splice joint fastener hole inspection requirement, the AF Materials Laboratory directed the development of a microprocessor-based ultrasonic scanning system to seek radial fatigue cracks as small as 0.030 in. without requiring expensive and damage-risking fastener removal (5). The resulting system, which is now produced commercially, is shown in a depot setting in Figure 17. A companion system has been designed and partially field-evaluated for detection of hole cracks in a second layer of a fastened stack-up, again without fastener removal (a problem which is predicted to occur in the near future in at least one currently operational fighter aircraft system).

FIGURE 17

Emerging NDI/E Technology

A host of advanced technology methods and techniques emerging are expected to provide important solutions or aids to current integrity monitoring requirements and needs. A few are reviewed here.

One of the more important development thrusts in the R&D program addresses improved methods for composite component inspection. Figure 18 illustrates a typical in-service inspection procedure in which the inspectors perform manual ultrasonic scanning and depend on instantaneous detection information captured from a CRT display. To bring about the needed improvement, the R&D effort

FIGURE 18

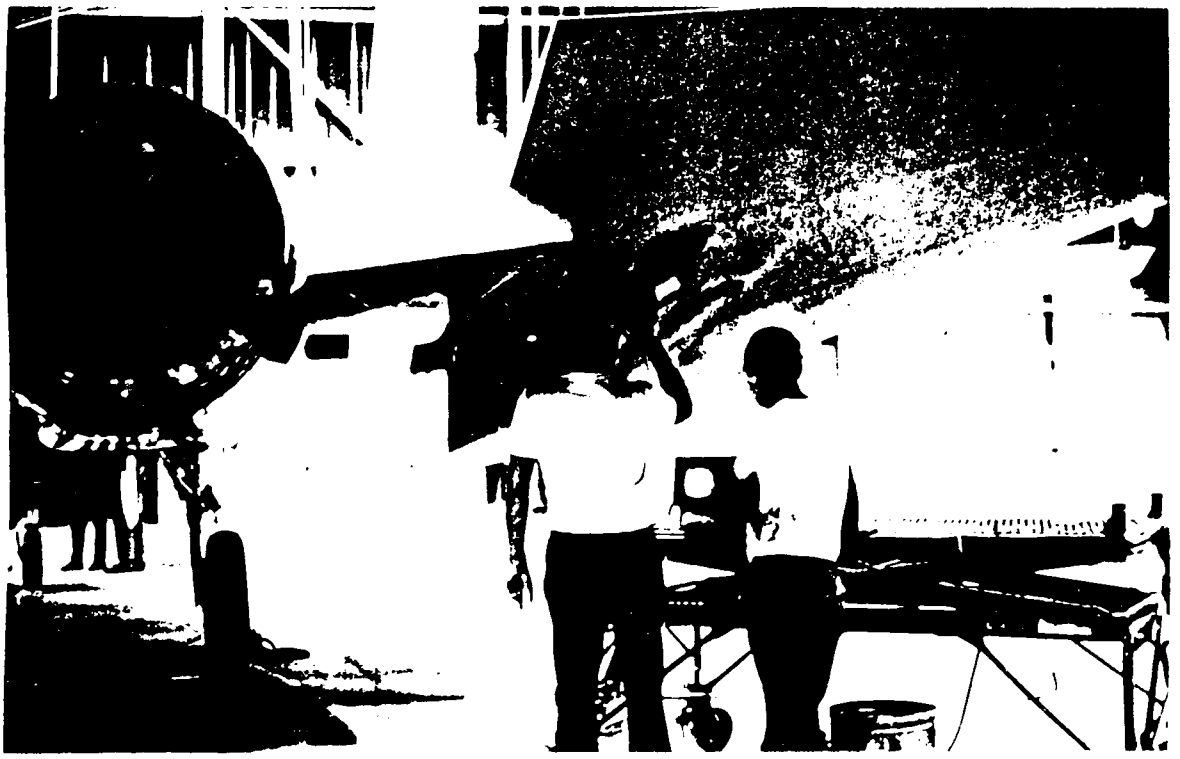


FIGURE 19

COMPOSITES NDI/E THRUST

- DEPOT-LEVEL MANUAL SCANNING WHICH PRODUCES HARDCOPY IMAGES
- RAPID LARGE AREA SCAN CAPABILITY
- RAPID IMPACT DETECTION METHODS

underway (Figure 19) is aimed at 1) an articulatible instrument that gives a monitor-viewable and hardcopy C-scan record of the type now available in the manufacturing factory, 2) new methods to rapidly scan large areas, and 3) a method to locate rapidly at an operational site any possible impact damage (e.g., FOD) which may require subsequent detailed inspection. A new ultrasonic system prototype has been built to satisfy the first requirement (6). The system, named Automated Realtime Inspection System (ARIS) for composites is pictured in Figure 20 being employed to scan the graphite-epoxy skinned horizontal tail of an F-16 fighter. The system consists of a hand scanned transducer whose housing contains

FIGURE 20

a sonic emitter that transmits its location in x-y space to a sound bar fixed to the component surface with suction cups. Two microphones in the bar (one at each end) receives the signals which are used to record the transducer position by a

triangulation calculation. By combining this information with the ultrasonic measurements, a C-scan image is constructed as displayed on the monitor at the bottom right. Shown in this view is the system operating in the pulse-echo mode. The rest of the system components seen in the background have been designed and packaged specifically to make the system portable as five pieces of commercial airline baggage.

In order to perform through-transmission mode NDI/E, the APIS is provided with an easy-to-handle yoke which aligns the transmitting and receiving transducers, as seen in Figure 21. The system, which is capable of scanning a 24 in. x 24 in. area without repositioning the sound bar, and whose sensitivity provides a defect detection level (e.g., unbond or delamination) of 0.25 square inch, is currently undergoing extensive field trials at operational sites and on operational

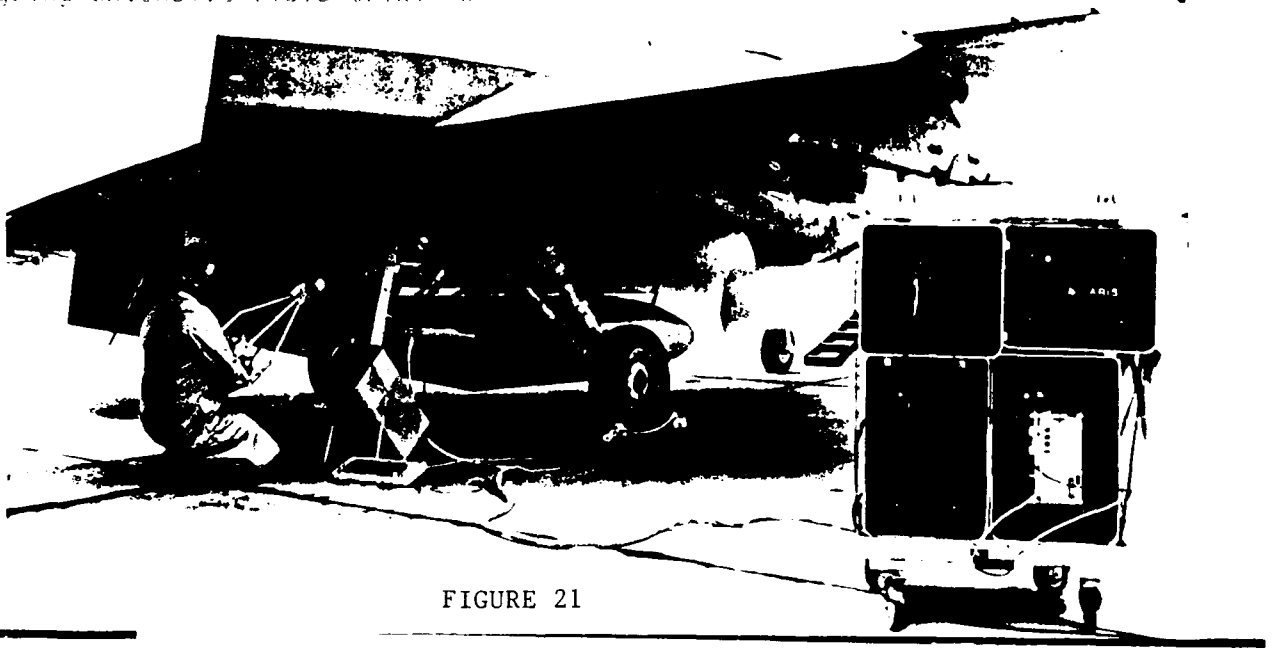


FIGURE 21

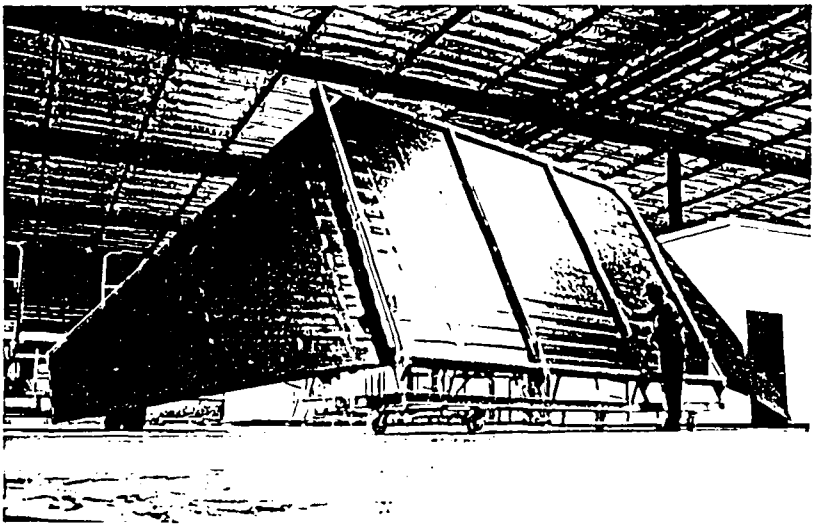


FIGURE 22

aircraft. Following these trials and after any modification needs are identified and resolved, the system should be ready for implementation by the AF.

The second major composites NDI/E development effort listed in Figure 19 relates to the projected need for rapid (economical) scanning of large composite structural areas anticipated in some future aircraft systems. Figure 22 illustrates a large graphite thermoplastic composite prototype wing cover that typifies such structures. While an advanced NDI/E system of the type shown in Figure 9 may be suitable during initial manufacturing, an in-service NDI/E procedure will require an approach designed for scanning assembled and installed structure and adaptable to the more restrictive depot setting. The sketch in Figure 23 emulates such a concept on which some development work has been conducted. In this example, an inspector moves a sensor head along a large surface area to acquire ultrasonic signal information to process a C-scan map as

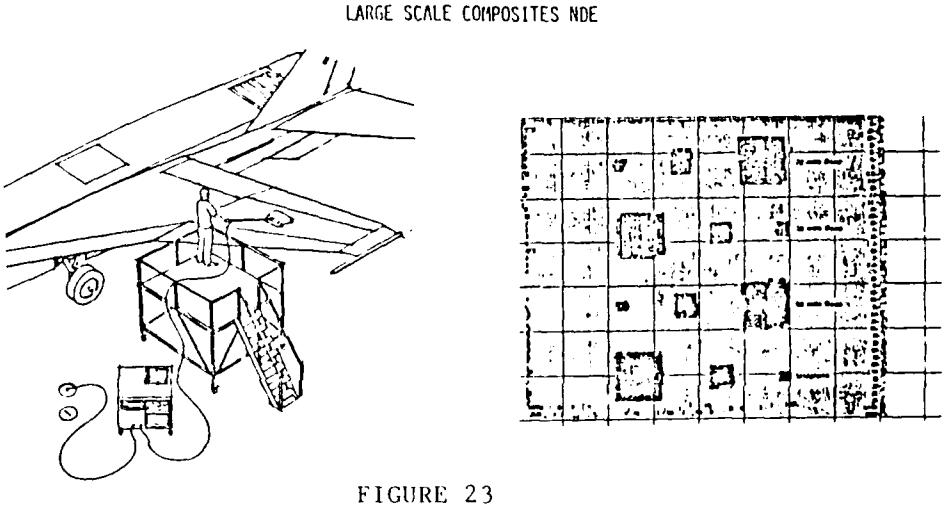
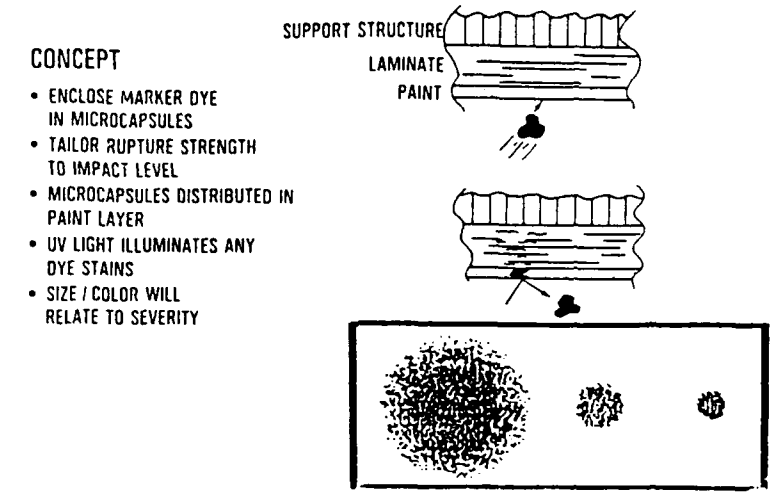


FIGURE 23

can be obtained now with the ARIS system previously described. A prototype of such a sensor head has been developed by McDonnell Aircraft Company with the aid of WRDC/ML funding (7). Other concepts are being considered also, such as laser-generated ultrasonic scanning (in which a laser beam might sweep rapidly across a surface to initiate an ultrasonic wave, then sense the response from a flaw) as well as some type of advanced wide-area thermographic technique.

Finally, the possibility of significant interlaminate impact damage (e.g., FOD) sometime during the service life of a composite component can pose a challenge to the inspector. Although robust damage detection equipment items (e.g., ARIS) should come on line in the near future, they are most effective for interrogating specific (and local) areas in a reasonable time. However, a procedure to identify possible damage sites quickly in an operational setting will be needed in order to screen out rapidly all undamaged components from those that require a more detailed (and more time-consuming) subsequent scanning. One screening concept that has been studied by WRDC/ML and further developed under contract by Southwest Research Institute is illustrated in Figure 24.

FIGURE 24
METHOD TO INDICATE IMPACT EVENTS
IN COMPOSITES



EXPERIMENTAL RESULTS OF IMPACT TESTS ON COMPOSITE SPECIMENS COVERED WITH A PAINT CONTAINING DYE-FILLED MICROCAPSULES

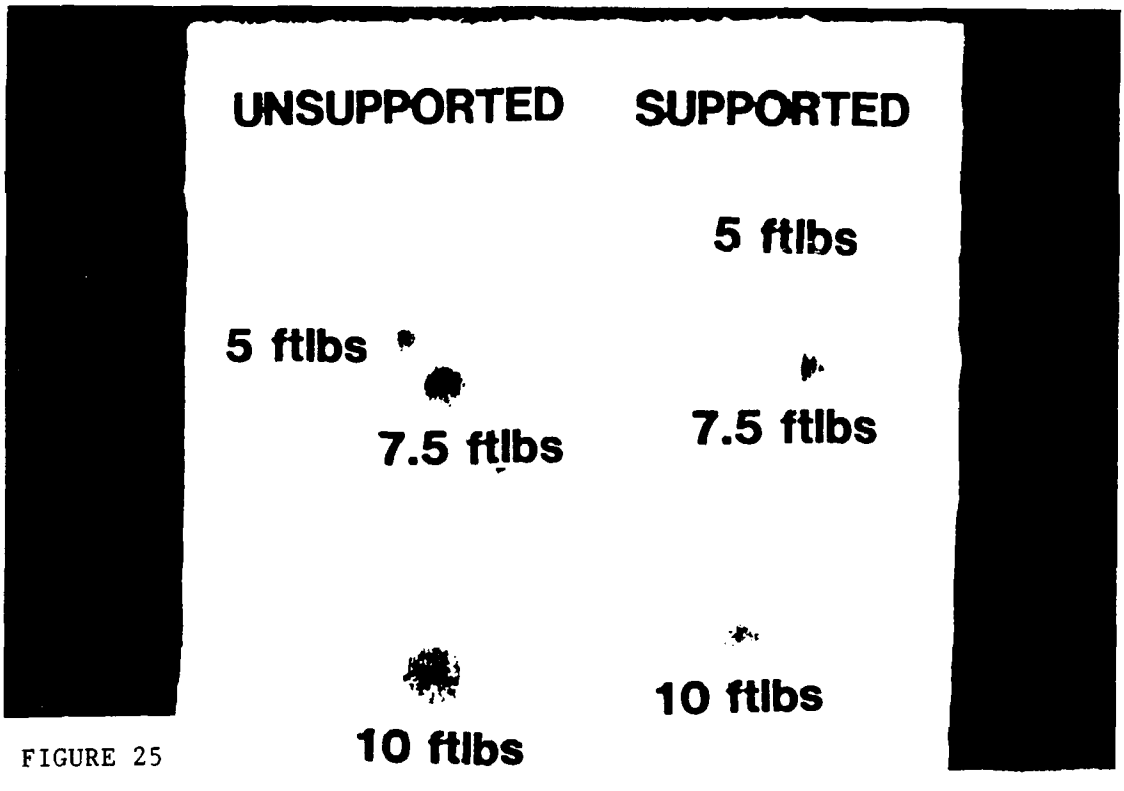


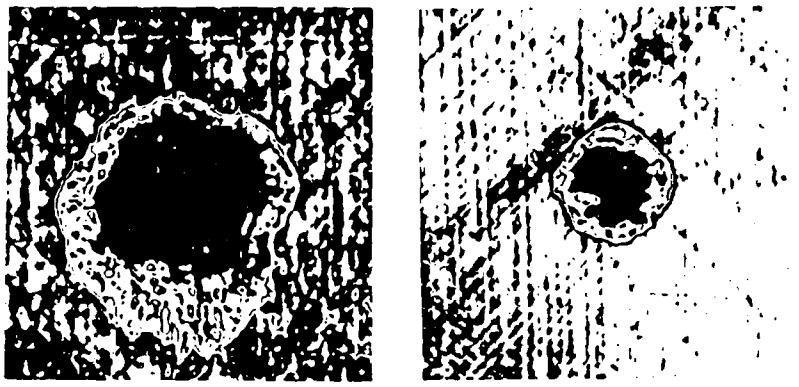
FIGURE 25

Microencapsulation technology in place today for other applications has been shown to provide a feasible approach to the idea prescribed here (8). Figure 25 presents experimental results using a loaded paint system placed on an organic-matrix composite specimen. Impacts of specific energy levels on both unsupported and supported (typical interior stiffener) areas selectively ruptured the dye-containing microcapsules resulting in the "graded" impact indications shown. The next effort planned is an advanced development program to scale up, demonstrate, and validate such a system under operational conditions.

The capability to produce an ultrasonic C-scan map of a delamination damage envelope in composite materials, such as shown in Figure 26, is well established for through-transmission (T-T) immersion systems. Such a map indicates the interruption by the delamination (or defect) of signal passage through the thickness to a receiving transducer located on the back side. In a related way, the pulse-echo (P-E) mode C-scan indicates signals reflected back to the transmitting transducer from an impeding defect in the signal path. More recently, equipment and procedures have been developed to use time-of-flight measurements from various depths in the P-E mode to gain an idea of the through-the-thickness distribution of delamination damage. The ARIS system described earlier and a couple of commercially available instruments have such a capability. However, since these are only close estimates of the defect location,

FIGURE 26

CONVENTIONAL ULTRASONIC C-SCAN RESULTS
- IMPACT DAMAGE IN COMPOSITES



GRAPHITE/EPOXY **GRAPHITE/PEEK**

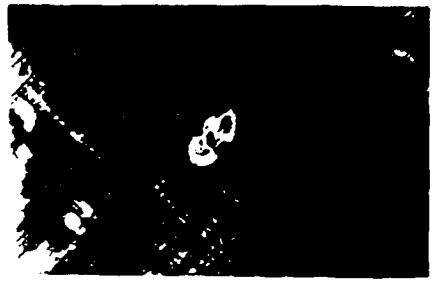
a degree of imprecision is characteristic. WRDC/ML researchers have now developed a new method by which they can produce precise maps of the delamination segments that exist at specific laminate interfaces anywhere through a thickness that exceeds 100 plies (9). The resulting image quality is demonstrated in Figure 27 which compares the precisely located and mapped ultrasonic images of delam segments with the interfacial delamination scars revealed by post-test deplying. The follow-on development strategy includes the incorporation of this new methodology into systems such as ARIS.

Conventional X-ray radiography has been an important traditional NDI/E method of value for the Air Force. It will remain so. It is the first line of inspection for numerous flaw conditions. There are a number of new developments and opportunities that will undoubtedly increase capabilities dramatically and significantly expand its application. A few are covered here.

Among the disadvantages of conventional film radiography is the dependency on the film itself, handling inconveniences, interpretation inaccuracies and limitations, and materials costs (primarily film). In contrast to conventional setups, as illustrated in Figure 28 left, real-time filmless imaging systems (which are already commercially available), seen in Figure 28 top right, offer some important advantages, such as image viewing in virtual real time, ability to

FIGURE 27

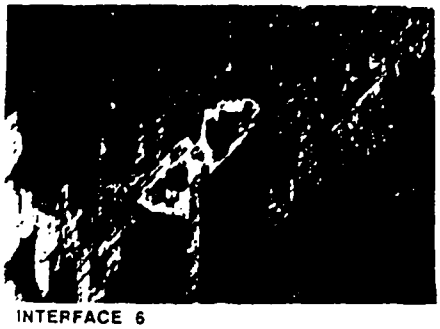
IMAGING OF IMPACT DAMAGE IN COMPOSITES



INTERFACE 3

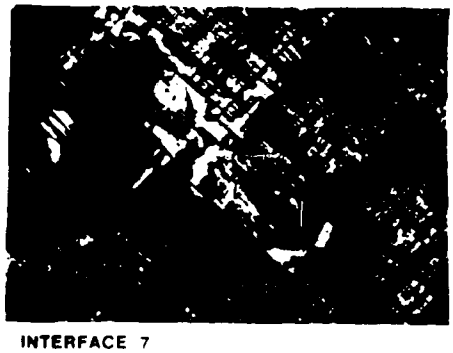
25 mm

INTERFACE 4,5



INTERFACE 6

25 mm



INTERFACE 7

DESTRUCTIVE VERIFICATION

ULTRASONIC IMAGES

change scan speed and reverse direction, and digital memory archiving. Still, image size and resolution do not match that of film. The long range Air Force plan calls for development of a portable real time filmless system, to be utilized as depicted in Figure 28 lower right. To achieve portability will require a move away from a real time radiography (RTR) system based on the currently used optical imaging subsystem (Figure 29a) to a non-optical method of acquiring, storing and readout to a computer of image information using solid state components. Such a system is described in Figure 29b. Improvements in X-ray detection and image storage devices are being made by the technical community almost daily and this will lead shortly to a technology integrating and RTR system prototype development effort.

FIGURE 28

HIGH RESOLUTION FILMLESS RADIOGRAPHY

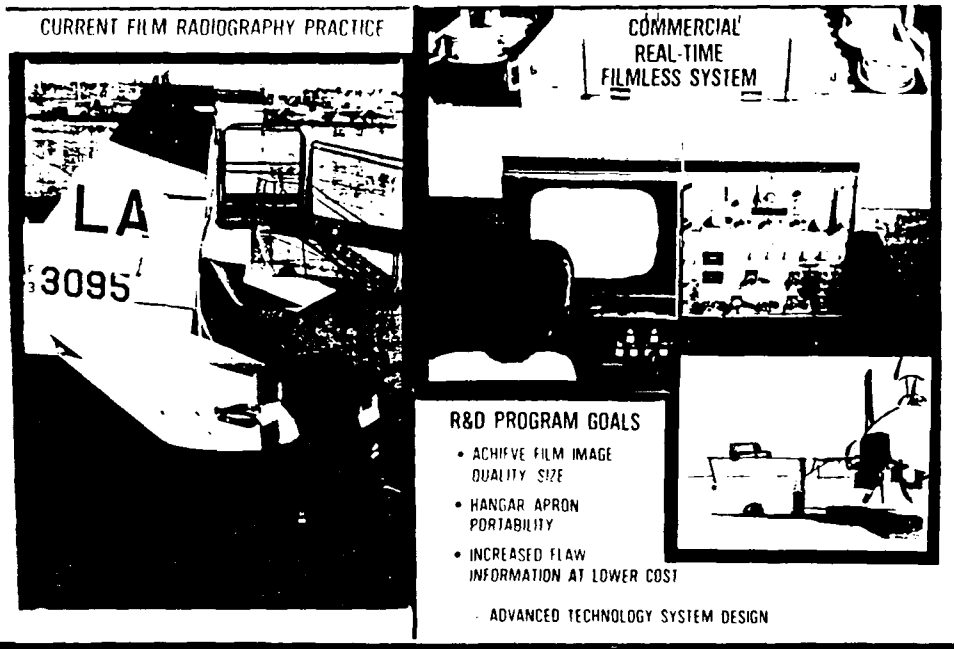


FIGURE 29a

FILMLESS RADIOGRAPHY SYSTEM

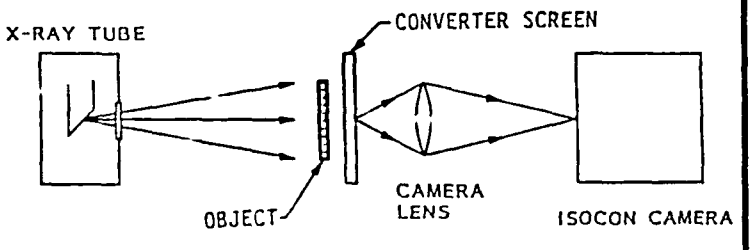
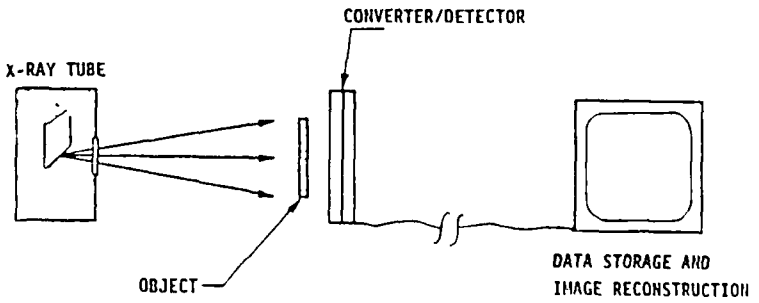


FIGURE 29b

NON-OPTICAL FILMLESS RADIOGRAPHY SYSTEM CONCEPT

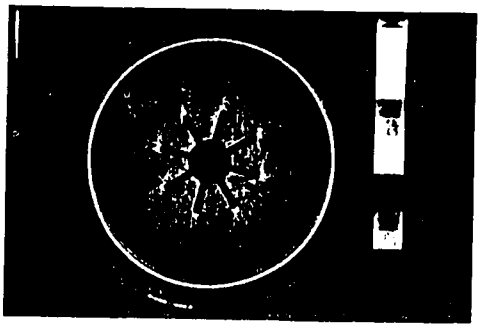


The substantial development work on X-ray computed tomography (CT) led by WRDC/ML over the past 10 years has demonstrated the enormous potential payoff of this process with its ability to produce high resolution defect images not available through other means. In contrast to conventional radiography which produces a flat projection of an object perpendicular to the X-ray beam direction, CT measures the intensity of X-ray transmission through the object along many ray paths in a fanlike pattern lying in the plane to be imaged. After this procedure has been repeated at many angles in the plane, the point-by-point X-ray attenuation in the image plane is calculated and an image "slice" is produced. The WRDC/ML program has produced several CT system prototypes of various sizes to demonstrate and refine the technology One of these, known as AFACTS II, is pictured in Figure 30 left (10). Here, the object to be imaged, using the 15 MeV X-ray source (on the right tower) and solid state photon detector bank (left tower) is an 8 foot diameter second stage solid rocket motor standing on the system's turntable. A full-diameter image is shown on the right (unreproduced image contains substantial detail). Another machine produced and installed recently at an Air Force maintenance depot is shown in Figure 31. This system, an

FIGURE 30

COMPUTED TOMOGRAPHY

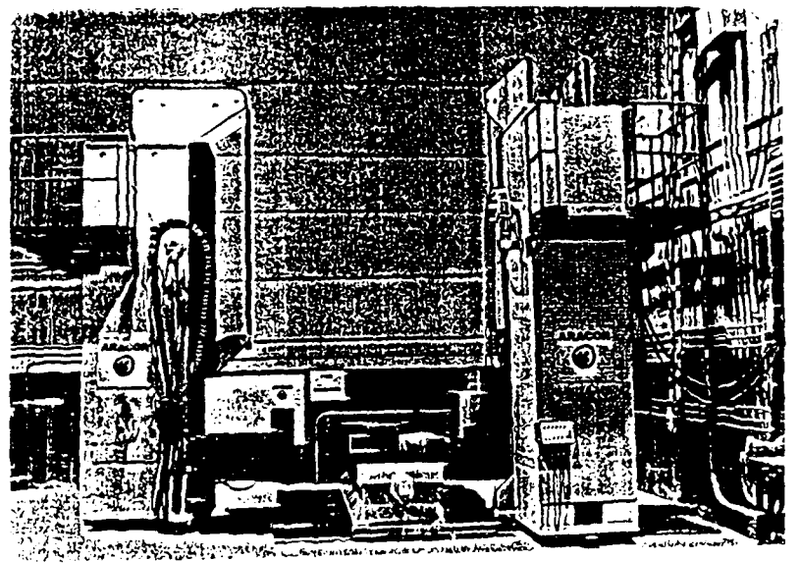
AIR FORCE ADVANCED SYSTEM II (AFACTS)



PEACEKEEPER 2ND STAGE MOTOR COMPUTED TOMOGRAPHY CROSS SECTION IMAGE

improved version of earlier Air Force machine designs, includes a 2 MeV X-ray source with a maximum inspection object size of 1 meter in diameter. A high

FIGURE 31



quality image is of a seven inch diameter igniter is pictured in Figure 32. A 2 millimeter wide unbond is detected at the 8 o'clock position, a flaw which was very difficult to detect by conventional radiographic means. The application of this machine to specific maintenance NDI/E functions is under study currently. In other work, a major advanced development program is underway.(11) to investigate possible aeronautical equipment inspection applications and the associated economics and payoff for CT (most earlier development efforts have focused on missile component NDI/E needs). Some exploratory scans are shown here. Figure 33 shows a CT image of full depth honeycomb helicoptor blade with a cast aluminum leading edge and glass phenylic skins (leading edge casting porosity was found, as

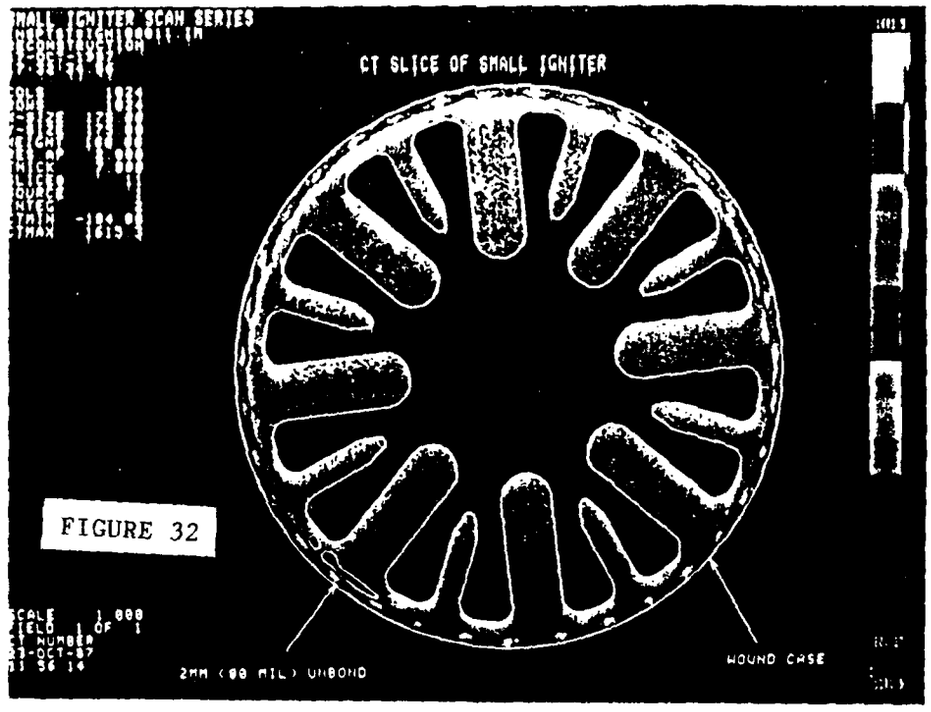
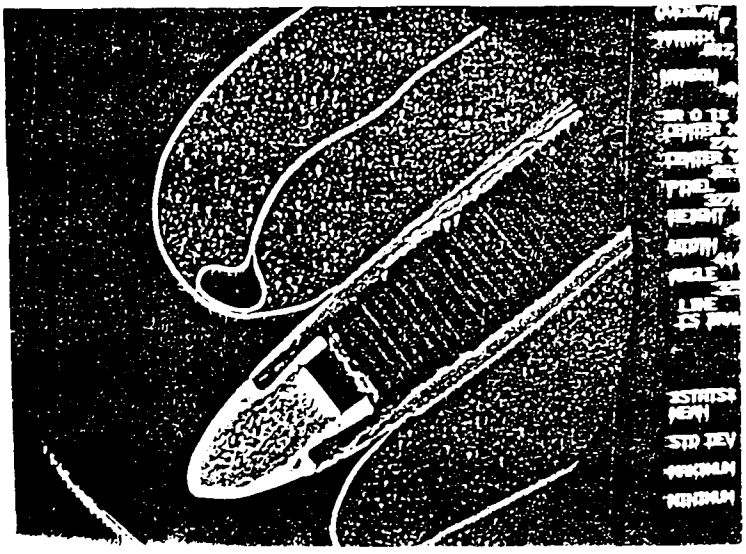


FIGURE 32

FIGURE 33



was unbonds and porosity in the skin material). Figure 34 demonstrates imaging capability with a six-inch diameter silicon carbide bladed turbine disk. A low density region is revealed from the six to nine o'clock position. In a final example, seen in Figure 35, successive CT slices through surface mounted electronic components on a circuit board are displayed. Once the tremendous potential of this new inspection modality is better understood and harnessed, broadened applications in support of ASIP/ENSIP are expected.

FIGURE 34

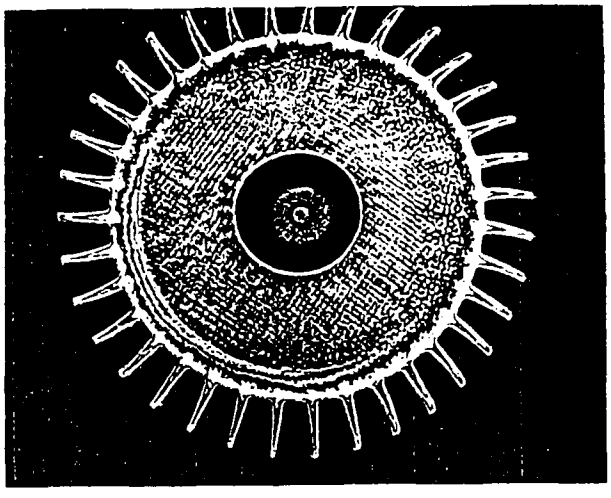
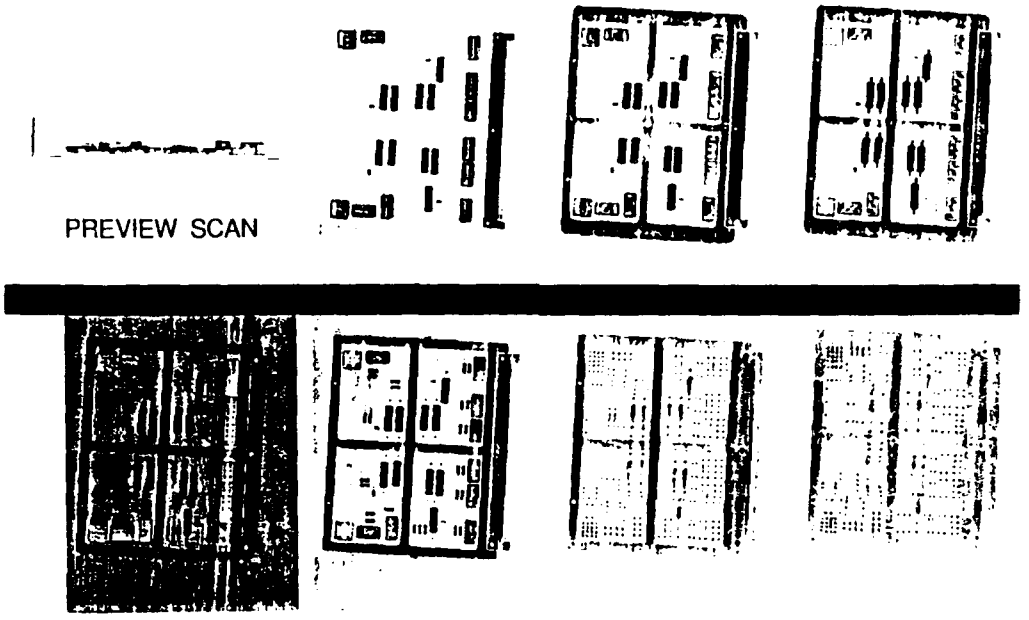


FIGURE 35

CT OF ELECTRONIC CIRCUIT BOARD

MULTIPLE SCANS THROUGH MOUNTED COMPONENTS

(ML LAM/DE)



Exploration of New Methodologies

Many new methods and concepts are being studied and developed by the NDI/E scientific community and some of them should provide improved capabilities to support IP efforts in the future. A few examples within the WRDC/ML program are given here.

Thermal Wave Imaging. With this method, variations in heat conduction across the surface of a part due to near surface discontinuities can be sensed. Heat is periodically deposited, such as with a repeatedly interrupted laser beam, at a point on the part surface and then the consequent temperature change at that point is measured. As with ultrasonic imaging both the amplitude and phase of the resulting thermal wave are recorded with an IR camera (in this case, "phase" refers to the time delay between switching the laser off and the maximum of the subsequent temperature rise). With this method, a digital "thermal wave" image (TWI) is obtained revealing near surface (1-2 mm) discontinuities with considerable resolution (12). Figure 36 shows a TWI of a fatigue crack in the surface roughly half way down a fastener hole. In another example shown in Figure 37, a defective electrical interconnect is scanned. Here a copper conductor is deposited onto a polymeric substrate in a way that an unbond is present under the neck. Conventional thermography failed to reveal the unbond, as shown in (a), but with the TWI technique was revealed robustly, as shown in (b).

FIGURE 36

FIGURE 37a

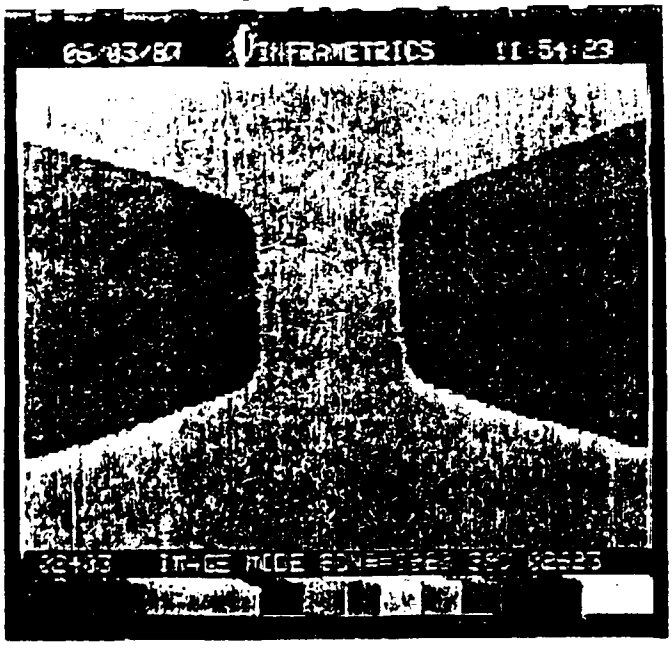
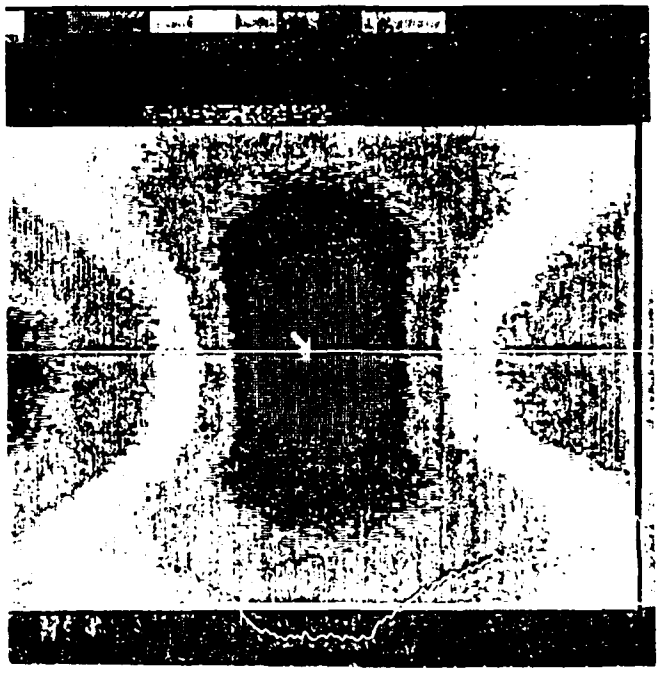


FIGURE 37b



Leaky Lamb Wave Ultrasonics. A number of researchers have experimented with ultrasonic Lamb wave excitation as a means of revealing subtle defects within certain types of materials, for example composites. With this method, an ultrasonic surface wave can be made to excite one or more plate modes within a laminate. Any discontinuities present within the excited area will change the response predictably and sufficiently to be recorded [13]. The response changes can be detected by characterizing the portion of the surface wave energy that radiates or "leaks" off the material surface. Figure 38 presents images of several types of subtle defects contained in a graphite/epoxy test piece using Lamb wave excitation.

Additional work by D. Chimenti of WRDC/ML has demonstrated our ability to plot variations. Additional studies have demonstrated the potential of Lamb wave methods to measure some materials properties nondestructively. Chimenti has succeeded in plotting the elastic property map shown in Figure 39 for a ceramic composite specimen using methods developed in his laboratory. Additional work should expand this capability further.

DEFECT IMAGING IN BIAXIAL COMPOSITES USING LAMB WAVES - 24 PLY GRAPHITE/EPOXY

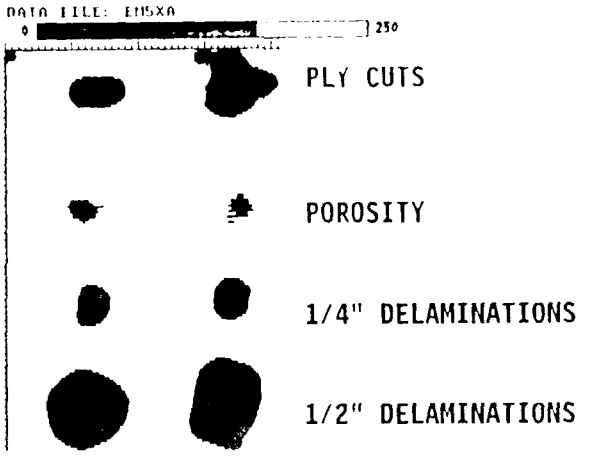
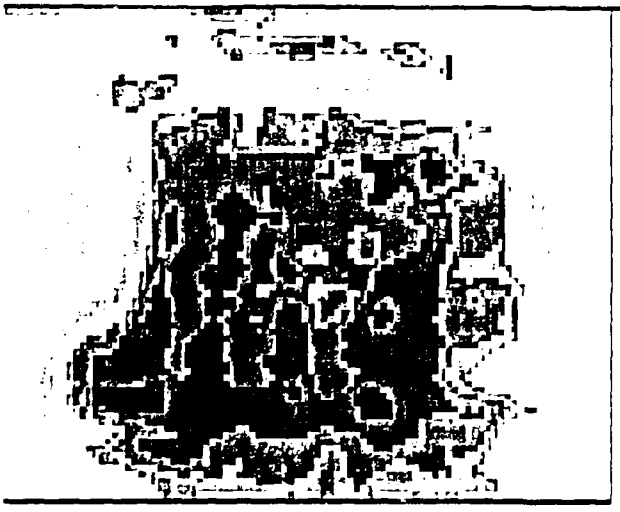


FIGURE 38

HIGH RESOLUTION ULTRASONIC SCAN OF CERAMIC COMPOSITES



- NEW SCAN CONCEPT UNDER STUDY
  - LEAKY LAMB WAVE EXCITATION
- UNIQUE IMAGING OF MECHANICAL PROPERTIES POSSIBLE
  - ELASTIC MODULUS VARIATIONS SHOWN HERE
  - TO 1 PERCENT RESOLUTION
- DEVELOPED METHOD COULD BE SCALED UP
- KEY APPLICATIONS ENVISIONED
  - CRITICAL FIBER/MATRIX BONDING CHARACTERIZATION
  - DATA TO ASSIST NEW MATERIALS DEVELOPMENT

FIGURE 39

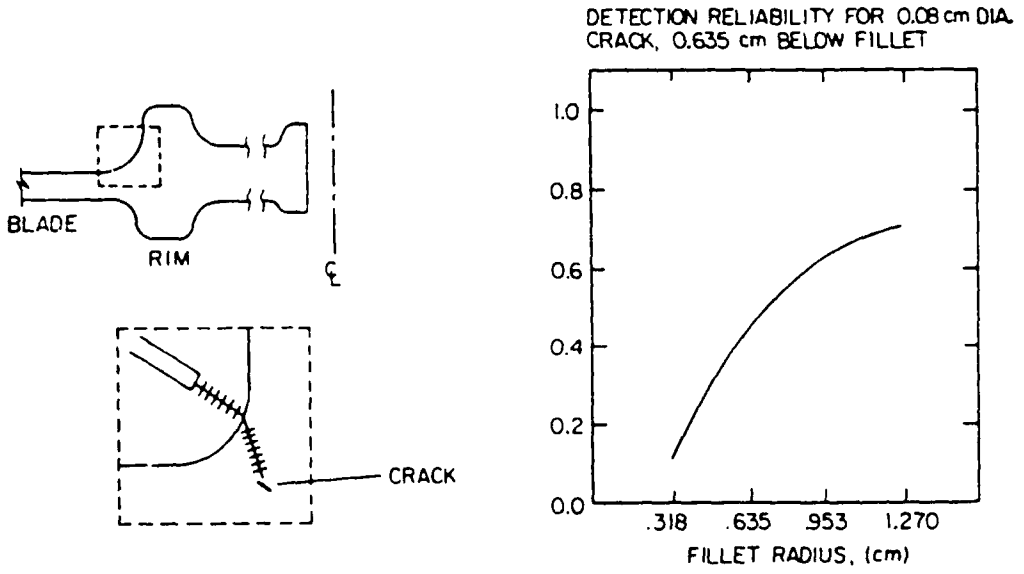
Flaw Image Enhancement Methods. The development of new and more revealing flaw imaging methodologies of the type described in this paper for applications to NDI/E has taken on growing importance over the last few years. Not only is more information about detected flaws revealed through their images, but the reliability of the detection itself may be increased as well. As a next step, it has been shown that the use of some of the powerful digital image enhancement methods that have been developed and used for other applications can reveal flaw evidence that may not be seen within a raw image. As an example, optimal enhancement techniques were employed by The Analytical Sciences Corporation to analyze a conventional radiograph of a turbine engine blade in the vicinity of cooling holes, shown in Figure 40a, to highlight the presence of an anomaly in the blade. The enhanced image is seen in Figure 40b. Although a crack is not visible, the concentration of fringes clearly indicates what was identified as a crack in a subsequent microscopic examination. The WRDC/ML development program is funding work to use enhancement techniques for ultrasonic, radiographic (especially CT) thermal wave imaging methods.

FIGURE 40a

FIGURE 40b

NDI/F Reliability Modeling. Considerable research and development is underway by WRDC/ML as well as several other groups to create the methodologies to model the inspection process analytically, or with some empiricism, as a step toward maximizing flaw detection reliability. Typically, the selection of NDI/E procedures for given requirements is experientially-based and established only after a design is completed or well scoped out. Probability of Detection (POD) models, once available, should allow the designer to call upon such analyses at the CAD terminal to predict NDI/E reliabilities for candidate design configuration iterations to identify optimum combinations. Such models are being developed for ultrasonic, eddy current and radiographic techniques. Figure 41 shows experimental results from validation tests of an ultrasonic model developed with WRDC/ML support by T. Gray and R.B. Thompson (14). Their model predicts the characteristics of ultrasound scattered from planar, crack-like defects as affected by simple geometries, as illustrated in Figure 41. Follow on refinement and validation efforts for several initial models are beginning now in order to produce accurate, flexible and robust simulations representing realistic use environments.

FIGURE 41



SUMMARY

Much of the NDI/E research and development work underway currently relates to, and indeed is motivated by, structural integrity requirements and issues. The WRDC/ML program is, of course, no exception. The emphasis is on (a) increasing the consistency/reliability of flaw detection (first and foremost), (b) detecting smaller flaw conditions, (c) establishing detection capabilities in cases where no adequate capability exists presently and (d) increasing, or at least maintaining, cost-affordability. Overlaying all of this, naturally, is the frequently difficult task of transitioning new technology to the intrinsically more restrictive applications settings, such as depot and field level NDI/E organizations, where a critical part of monitoring successful performance against ASIP/ENSIP requirements resides. A new NDI/E capability development can, in no way, be considered successfully completed until the critical application steps have been taken, including acceptance and incorporation by the user. While much work remains to be done just to keep pace with continually expanding IP requirements, the NDI/E technology field continues to grow richer in ideas and opportunities.

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Control of Error in Local Stress Analysis

with PROBE,

a New p-Version FEA Program

Presented by

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Noetic Technologies
St. Louis, MO, USA

at the

1988 USAF Structural Integrity
Program Conference

November 29, 1988

San Antonio, Texas

Control of Error in Local Stress Analysis with PROBE, a New p-Version FEA Program

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Noetic Technologies, St. Louis, MO

The Goals of Mechanical Analysis

There are two major classes of mechanical analysis routinely performed with the help of finite element methods, structural and stress, or, as some prefer, global and local.

The goal of structural or global analysis is an understanding of the general behavior of an assembled object, how internal loads are distributed and what displacements may be expected under various applied external loads. The primary emphasis is typically placed on approximating the stiffness of each structural member. Exact geometry is not usually required, and significant leeway may be allowed in the analytical idealization of the object without invalidating the general structural conclusions. Familiar element types such as beams, rods, shear panels, plates, and shells are employed effectively in this type of analysis. In general they perform rather well in the hands of a seasoned analyst, and yield results that are acceptable by most engineering standards. The FEA models of autos and aircraft often seen in publication are examples of models prepared for global analysis.

The goal of stress or local analysis is, by contrast, an understanding of the specific response of a detailed portion of an object. Here idealization must be far less crude, because the desired output is usually the location and magnitude of the largest stresses. Accuracy in stress demands much more precision of the model than does accuracy in displacements or forces, because stresses are based on the derivatives of the displacements. And geometric features that result in high stress gradients play an important role in overall solution quality.

With only structural information from FEA, that is, displacements and internal loads, an object can be manufactured to satisfy requirements for structural integrity and safety; but without detailed stress information, no judgment can be made about the real mechanical and economic effectiveness of the design. As a consequence, significant effort is expended following a 'structural' analysis in the performance of detailed stress analysis, often including cumbersome hand calculations, handbook rules of thumb or simply gross approximations derived from the output of a 'structural' model. In spite of the high value of good local stress data, the methods commonly used to obtain it have generally remained relatively crude.

Solution Quality

Engineering managers throughout industry routinely confirm that they have confidence in traditional FEA programs to predict the internal load distribution within reasonable engineering tolerances. They complain, however, that these programs are generally less reliable with regard to local stress distribution.

Furthermore, they have no explicit assurance with regard to the quality of the analysis other than the experience of the individual analyst, and, occasionally, a handbook of results from similar, but seldom identical, models, sometimes correlated with physical test data.

The quality of the approximation of the real mechanical response of an object to applied loads provided by finite element analysis is controlled by four factors: the theory selected for the idealization, the finite element mesh, the polynomial degree of the shape functions employed in the elements, and the mapping of the functions onto the geometry of the object.

Traditional FEA codes

Selection of common beam or shell elements, for instance, limits the analysis to the confines of beam and and shell theories, subsets of the full theory of elasticity. Accurate local stress distributions at shell intersections, cutouts, and other similar locations cannot be expected from such an idealization.

The finite element mesh selected has several important characteristics, its regional density, the grading, aspect ratios and skewness of elements, and so on. The relative diameter of an element is often referred to as 'h'; hence, the h-version name is often applied to traditional codes, because the analyst's choice of mesh tends to dominate the numerical quality of the result.

The polynomial degree of the shape functions, referred to as 'p', is not variable to any great extent in traditional FE codes. These shape functions are the basic mathematical building blocks used by the program to describe the displacement of the object as loads are applied. Four noded quadrilateral linear displacement constant strain elements are of the first polynomial degree; similarly, eight noded quadrilaterals are of the second degree. Beyond this sort of distinction, there is no recognition by traditional codes that 'p' has significant influence over the solution.

All mathematical degrees of freedom are located at the nodes of traditional elements; and mapping is restricted to formulations involving the nodal locations. Nothing is known about regions between nodes except by interpolating assumption. This is why very high mesh density is required of traditional codes to make even a crude analysis of stresses around the perimeter of a hole.

An analyst using a traditional FEA program will choose an idealization (theory) and a discretization (mesh) that will, in his opinion, produce answers very near to the exact (but unknown) solution.

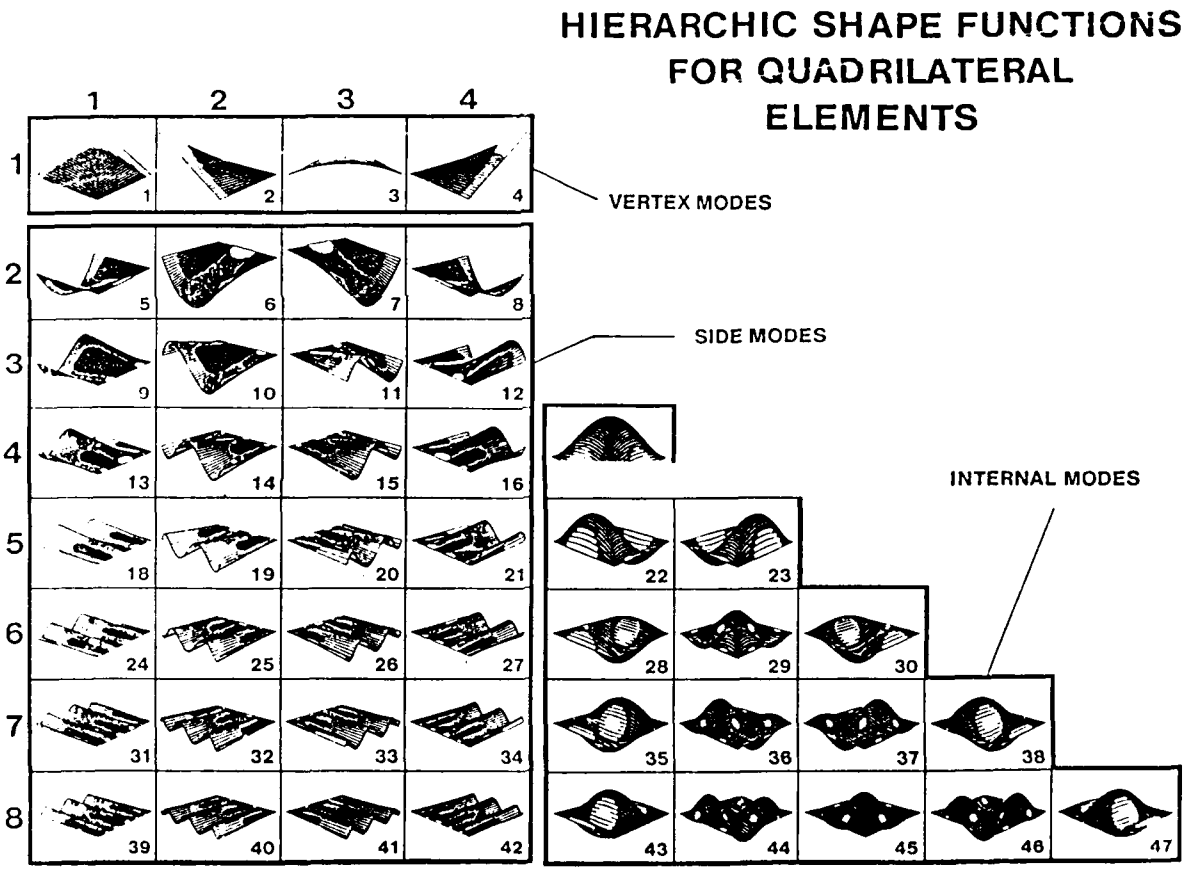
Once, however, the analyst has, by experience, selected the element types and mesh design, the predictive quality of the model has been fixed; and the relative error of approximation is essentially unknown. Control of solution quality is typically a priori. There is no direct feedback from the program.

Only one course of action is open to the analyst desiring some confirmation of the validity of his model. That is to perform a series of analyses, an orderly sequence of mesh refinements, increasing the degrees of freedom, and, presumably, the accuracy of the result. In practice, this is seldom done, mainly because of the time and effort required to refine the mesh, re-submit the analysis, and compare the

results in a rigorous fashion.

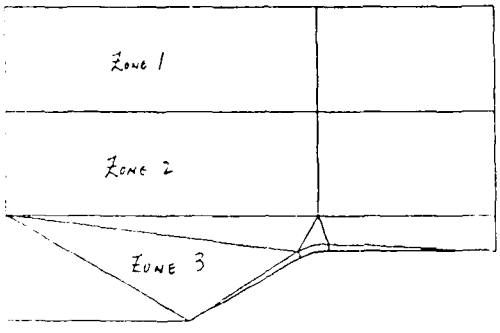
The p-version

Increasing the polynomial degree of the shape functions beyond first or second order, commonly called the p-version of FEA, yields immense dividends in local analysis. First, the mathematical robustness of the higher order elements facilitates much greater freedom in discretization. There is far less sensitivity to regional density, grading factors, element shapes, and so on. Second, the polynomial degree may be controlled by the analyst, within the specific scope of the actual computer program. And third, the mapping is not dependent solely upon the location of the element nodes; and more precise geometric descriptions may be used. In fact, the factors combine to provide stress solutions that are very nearly continuous over the domain; that is, results may be validly and explicitly extracted at any point within the model, not just at nodal points. A set of shape functions for polynomial degree 1 through 8, is shown below:

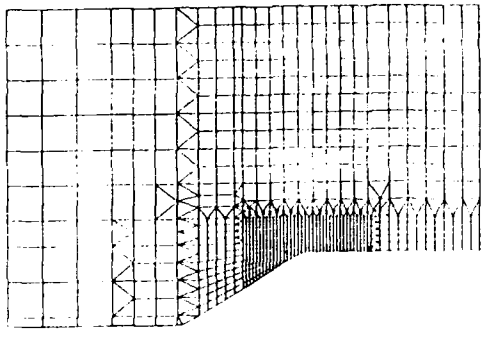


With this continuous representation of the displacement field (and all derivatives) the p-version enjoys significant inherent advantages over h-dominated codes for local stress analysis. It is interesting to note that the h-version is actually a subset of the p-version, with fixed values of p. Traditional h-version codes typically utilize linear (p=1) or quadratic (p=2) shape functions. The p-version continues the sequence of polynomial extension. It is also possible for the analyst to refine the mesh while using a p-version code, and the term 'hp' has made its way into the literature to describe concurrent manipulation of both 'h' and 'p'.

Here are some valid p-version meshes (compared to their h-version counterparts) employed for a range of interesting problems.

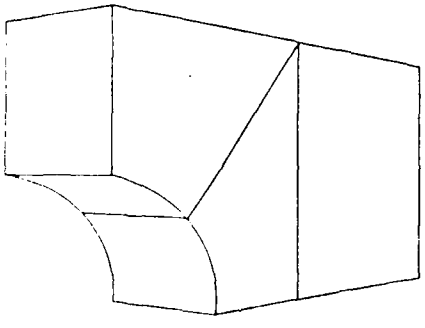


p-version mesh
(12 elements)

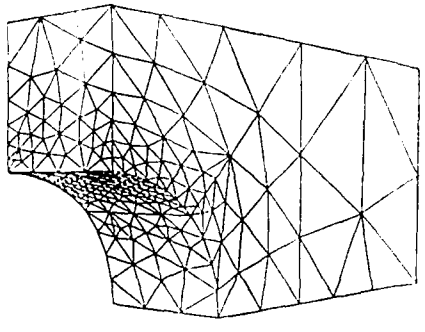


Traditional h-version mesh
(1000 elements)

Composite Wing Tab

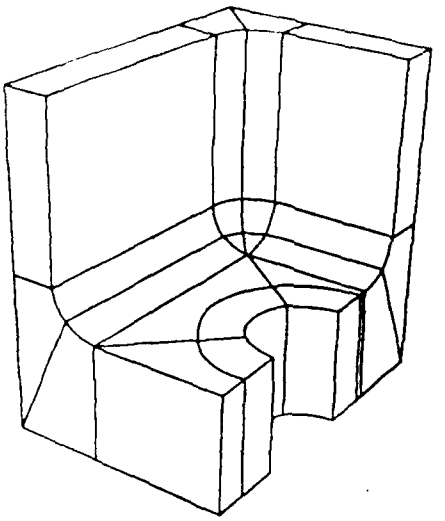


p-version mesh
(3 elements)

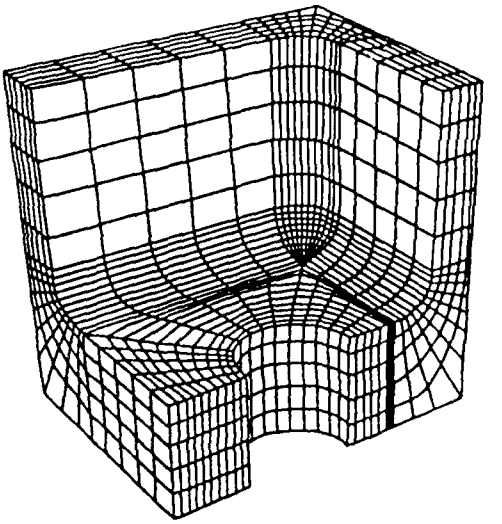


Traditional h-version mesh
(1894 elements)

Simple tension strip

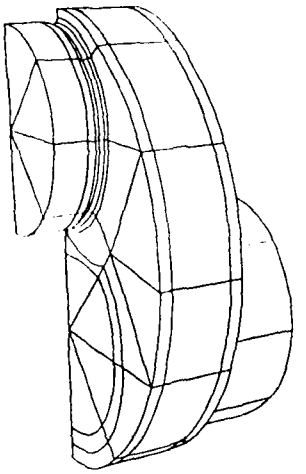


p-version mesh
(18 elements)

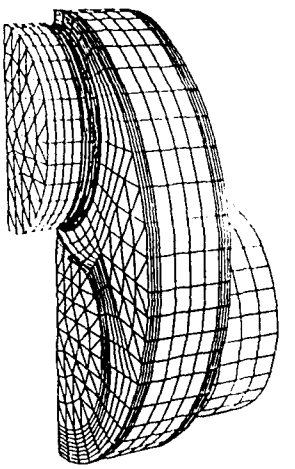


Traditional h-version mesh
(4572 elements)

Aircraft splicing fixture



p-version mesh
(107 elements)



Traditional h-version mesh
(2300 elements)

Automotive crankshaft

A relatively obvious characteristic of p-version elements apparent from these meshes is the mathematical "robustness". Aspect ratios, distortion, and grading factors cause almost no limitations for the p-version, as opposed to the rigorous demands for traditional meshes. Internal angles of 160 degrees, aspect ratios of 100:1, and exponential grading factors are all valid with the p-version of FEA.

Quality Assurance with the p-Version

All this would be reason enough to employ a p-version formulation for local stress analysis, but the most significant advantage of the p-formulation is that it facilitates direct and understandable feedback regarding the quality of the result. Enabling the elements to have a variable polynomial degree allows convenient and economical performance of multiple analyses on a single mesh. With properly selected hierarchical shape functions, results from p=1 through p=n may be computed and compared in order to evaluate the convergence of important variables, namely strain energy, forces, and stresses. Each analysis is self-evaluating, a posteriori. The 'art' of FEA becomes a science.

Specifically, the conveniently available feedback includes:

1. The estimated relative error in the energy norm, which serves as an indicator of the overall quality of the approximation.

2. Energy balance tests whereby the equilibrium of all or any part of the model may be checked by explicit integration of stresses or flux.

3. Convergence of results with specific engineering significance, such as the magnitude of the maximum principal stress.

If the model is not adequately performing its intended task, these tests first reveal that fact, and then provide the analyst with specific instructions as to how to modify the model to achieve a good result. More on that later.

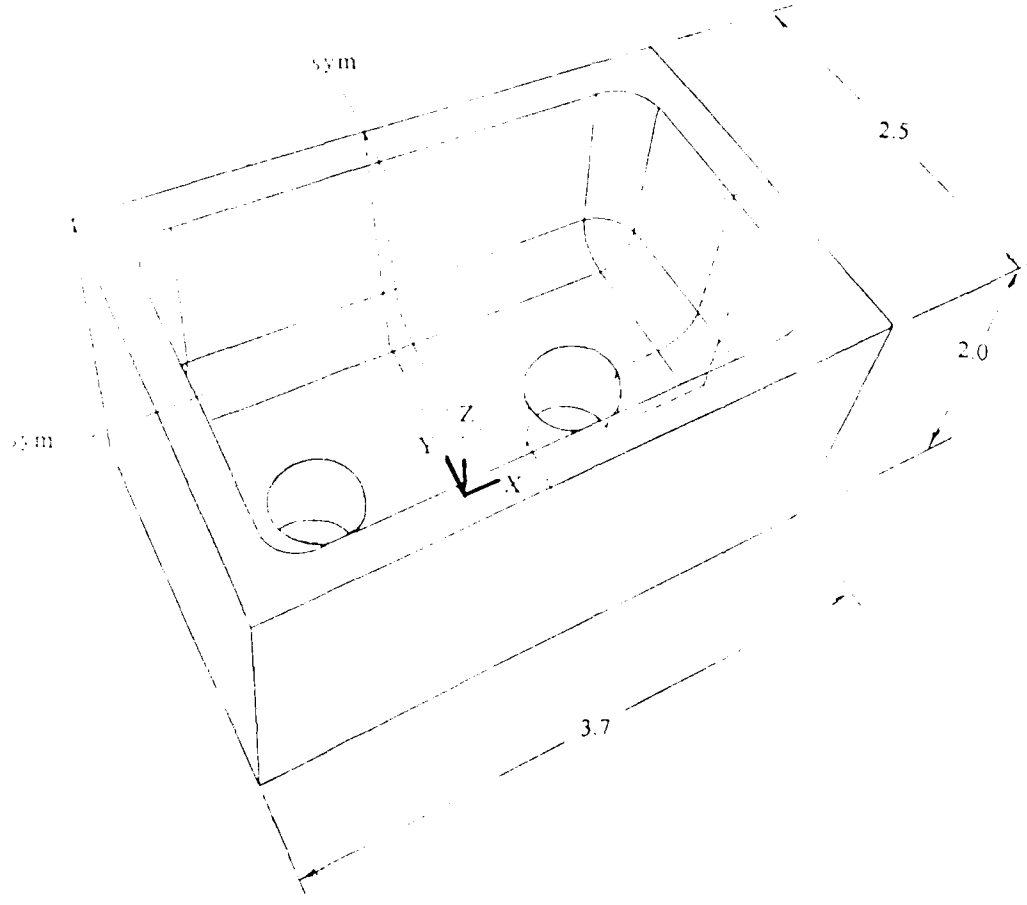
Improved Productivity

Simpler meshes, straightforward idealizations, explicit quality assurance, continuous solutions and other, more subtle, advantages of the p-formulation combine to yield significant gains in engineering productivity for local analyses. Specific gains are somewhat problem dependent, but they are clearly measureable due to their magnitude.

An Example

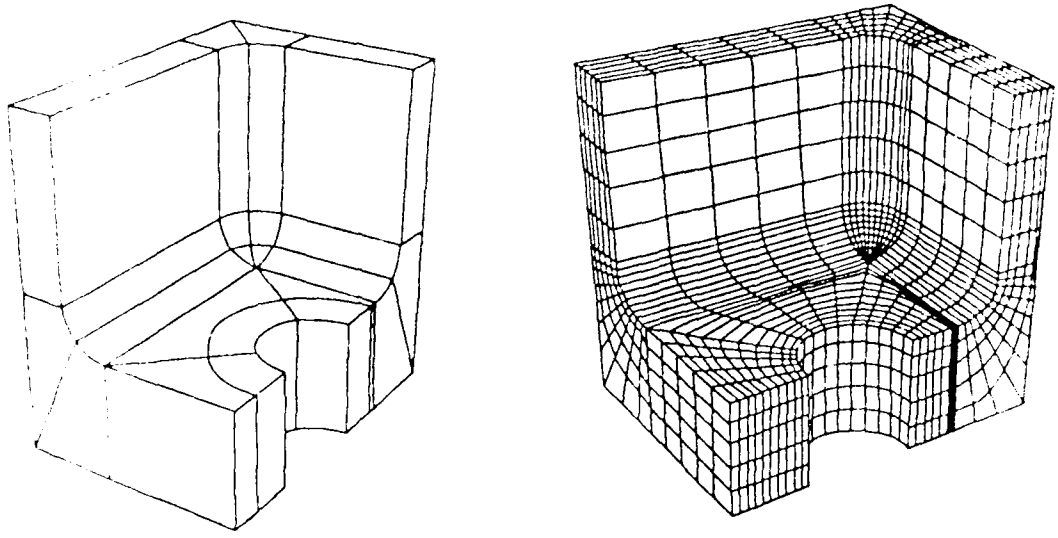
A relatively new code, PROBE, incorporates all the advantages inherent in the p-version of FEA. It has met with good commercial acceptance and provides both 2D planar and 3D solid capabilities. Excerpts from a 3D solid analysis are presented below to illustrate the unique capabilities of the p-formulation.

The object of study was a splicing fixture commonly known as a 'bathtub'. A pull-through force at the bolt holes was modeled as a compressive normal pressure acting on the washer contact area surrounding each bolt, and the normal displacement was fixed on another surface.



Splicing Fixture

Symmetry allows the analysis to be reduced to 1/4 of the total splice, which can be modeled with as few as 18 PROBE elements. Two typical traditional h-version models prepared for the same analysis contained 2,567 and 4,752 elements, respectively. The latter case is shown below as a comparison to the p-version model.



p-version mesh (18 elements)

Traditional h-version mesh (4572 elements)

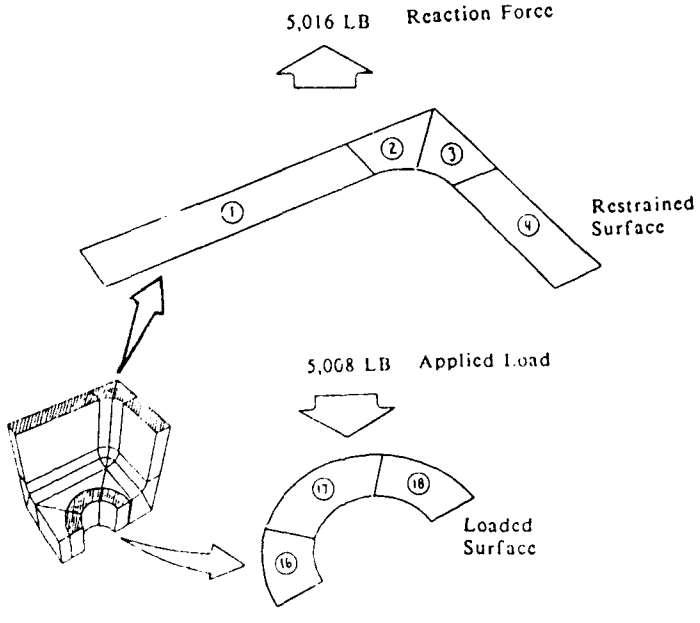
Comparative Finite Element Models

Automatic computation of results from p=1 through p=6, with global degrees of freedom ranging from 124 to 3,074, yielded convergence of the energy norm to within 3.3%.

Polynomial Degree, p	DOF	Total Energy	Extrapolated Energy	Relative Error
1	124	7.72	-	54.8 %
2	420	10.30	-	25.9
3	728	10.80	11.54	15.0
4	1261	10.95	11.02	8.9
5	2025	11.01	11.06	5.3
6	3074	11.03	11.04	3.3

Relative Error in the Energy Norm

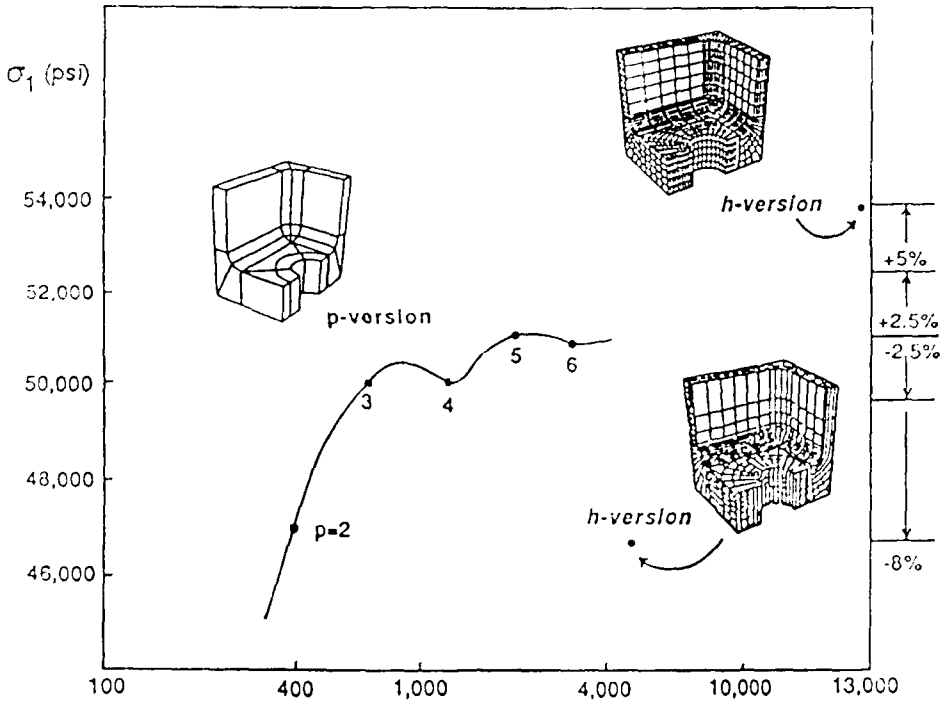
By explicit integration around the boundaries (no averaging or smearing) the overall forces were shown to be in equilibrium within 1%, 5,008 pounds applied, 5,016 pounds of reaction. Again by explicit integration, individual elements were examined for net force as a percent of total load; and face-to-face interelement action/reaction was checked. These data also indicated acceptable numerical behavior of the model.



Convergence of Equilibrium

Finally, the maximum value of the first principal stress was examined from p=1 through p=6 and exhibited strong convergence.

Two traditional models, independently prepared and analyzed at an industrial site, with roughly 5,000 and 13,000 degrees of freedom, yielded respective answers about 8% below and 5% above the converged PROBE result. No convergence conclusion could be drawn from such widely divergent examples.

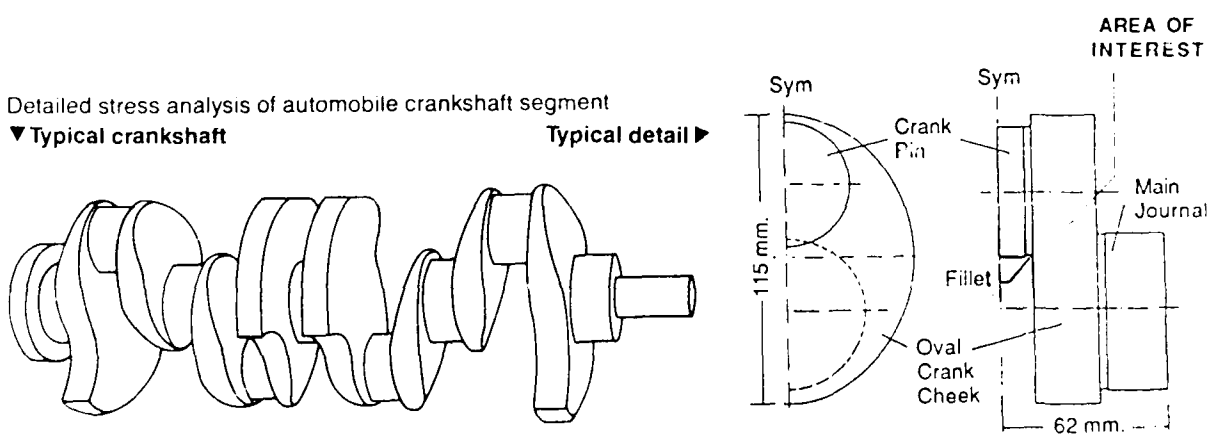


Convergence of Maximum Stress Sigma 1

Direct and unambiguous tests such as these are not possible without a hierarchy of solutions known as an 'extension'. Performing an extension with traditional codes requires creation of multiple models of increasing mesh density (since 'p' may not be increased), with the obvious corollary of multiple computer runs and examination of multiple detailed batch results.

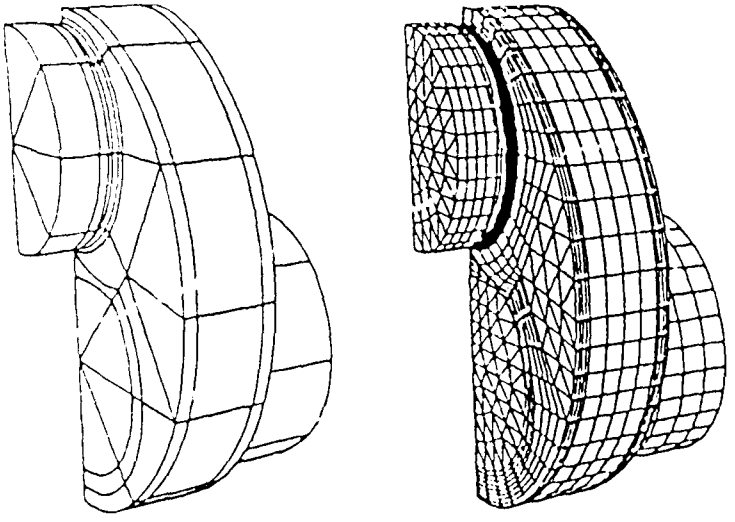
It should be pointed out that this example is a relatively smooth problem. In more demanding instances involving high gradients or true singularities, the p-formulation advantages are even more apparent. In the absence of explicit quality assurance capabilities, explained only briefly here, immense amounts of analytical effort could (and often do) result in misleading predictions, and thus contribute to questionable design decisions.

A more challenging problem involved the stress distribution near the joint of the main journal and crank pin in an automobile crankshaft (shown below).



Crankshaft geometry

The p-version model involved 107 elements. A traditional model with similar mathematical power would require about 2300 elements!

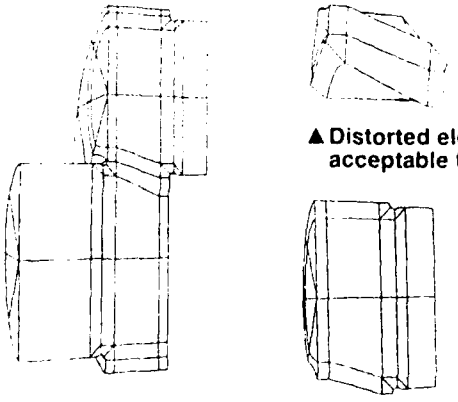


p-version mesh (107 elements)

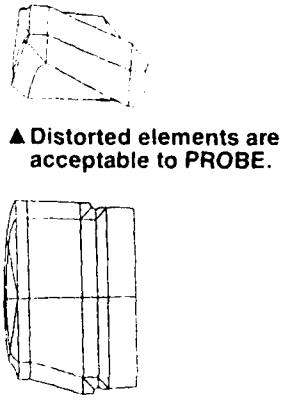
Traditional h-version mesh (2300 elements)

Comparative Meshes

The inner details of the p-version mesh (shown below) clearly expose the robustness of the p-version elements.



▲ High aspect ratios are acceptable to PROBE.



▲ Distorted elements are acceptable to PROBE.

▲ High grading factors are acceptable to PROBE.

The quality control features, highlighting the convergence of energy, equilibrium and stress results, is illustrated below:

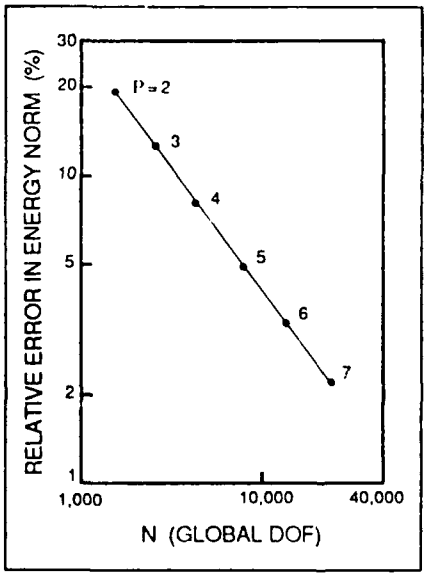
Convergence of Energy

The multiple hierarchic solutions provide explicit convergence analysis for every model.

Results are typically sufficiently accurate when the energy error estimator is in the 1-3% range (2 significant digits). Here the analysis was stopped at p=7, with the error in energy norm at 2.61%.

p	Global DOF	Total Energy	Extrapolated Energy	Convergence Rate	Percent Rel. Error Est'd.
1	433	0.432173D+02	0.000000D+00	0.00	66.50
2	1607	0.746549D+02	0.000000D+00	1.90	19.10
3	2944	0.761487D+02	0.767187D+02	1.24	13.12
4	5354	0.769611D+02	0.779649D+02	1.57	8.21
5	8975	0.772571D+02	0.775021D+02	1.62	5.40
6	14128	0.773784D+02	0.774964D+02	1.70	3.67
7	21134	0.774302D+02	0.774830D+02	1.70	2.61

▲Convergence of global energy—tabular format

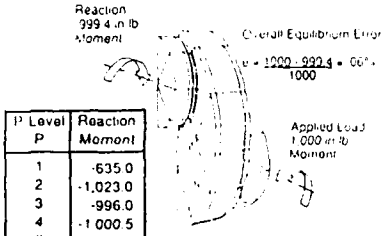


PROBE's built-in 'convergence calculator' extrapolates true energy to infinite degrees of freedom, computes the rate of convergence, and provides the root-mean-square error.

PROBE automatically produces results for all available solutions. Each p-level has its own DOF/energy pairing.

◀Convergence of global energy—graphic format

Convergence of equilibrium



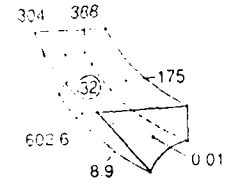
P Level P	Reaction Moment
1	-635.0
2	-1,023.0
3	-996.0
4	-1 000.5
5	-999.3
6	-999.1
7	-999.4
∞	-1000

▲Convergence of global equilibrium

P-Level P	Element Imbalance ΔFY
1	422.0
2	106.0
3	134.0
4	96.0
5	77.0
6	38.0
7	17.5
∞	0

Elemental Equilibrium Error

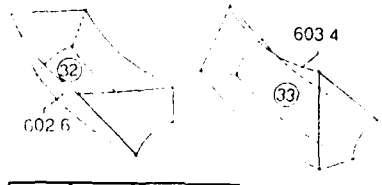
$e = \frac{17.5}{603} = 2.9\%$



▲Convergence of elemental equilibrium

Action/Reaction Error

$e = \frac{603.4 - 602.6}{603} = .13\%$



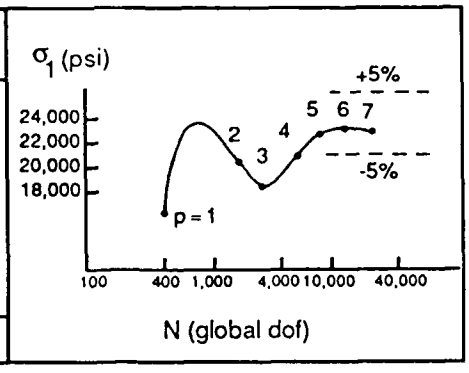
P-Level P	Force FY Ele 32	Force FY Ele 33	Net ΔFY
1	-517.6	573.0	55.4
2	-620.4	597.3	28.1
3	-609.3	638.9	29.6
4	-609.0	602.6	6.7
5	-609.7	607.8	8.1
6	-605.4	602.6	2.8
7	-602.6	603.4	0.7
∞	-603	603	0

▲Convergence of elemental action/ reaction equilibrium

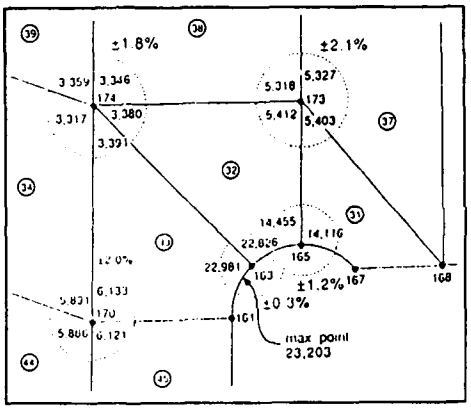
Forces are computed directly from the stress tensor. They are not "equivalent" forces.

Convergence of stress results

Polynomial degree, p	(σ_1) max
1	17,000
2	21,500
3	19,700
4	21,400
5	22,900
6	23,100
7	23,000
8	not needed
∞	23,XXX



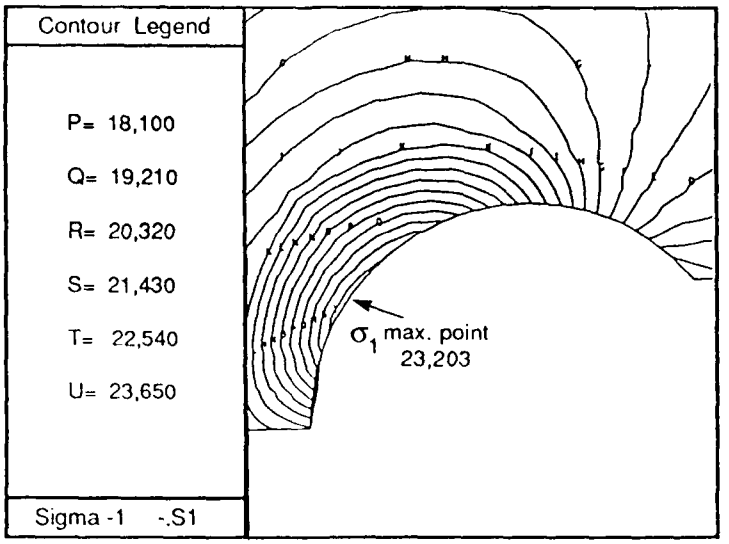
▲ Convergence of stresses
Convergence of maximum stress with increasing polynomial degree illustrates the integrity of the local data in the critical region, with variation of less than 1%.



Continuity of stresses ▶
Continuity of the stresses computed from each element at a specific point further demonstrates the integrity of the finite element result. Again, variation is less than 1%.

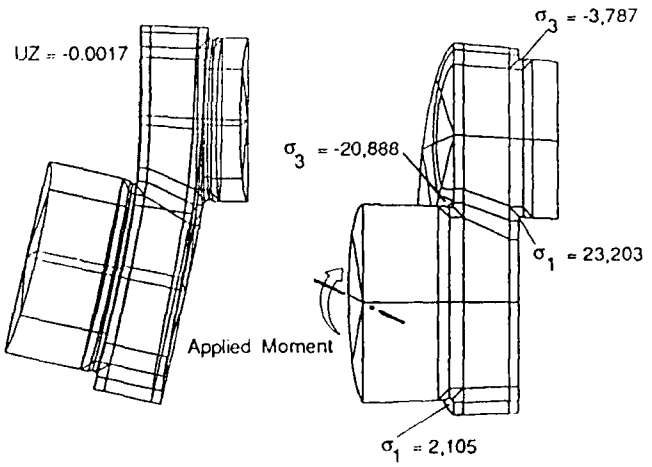
Engineering data, of known high quality, is shown below:

First, the contour and deformed plots:

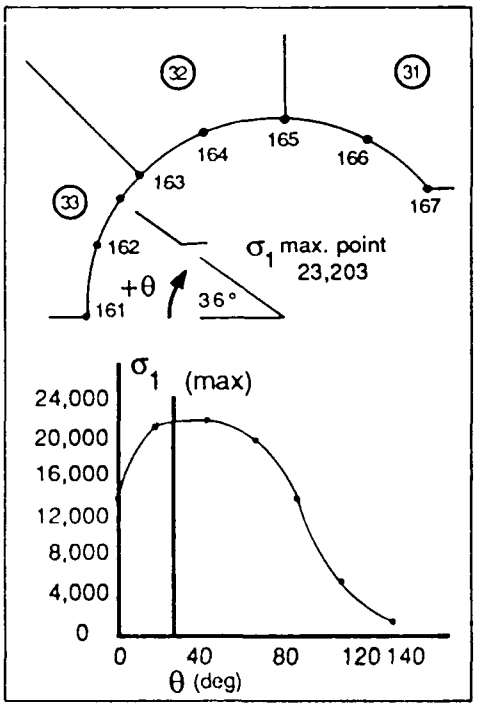


▲Contour plots
Standard contour plots are easily created through interfaces with the popular graphics packages such as Patran®, Supertab®, or MOVIE.BYU.

Deformed shapes▶
A typical plot of the deformed shape shows the effect of the moment loading and illustrates the weakest cross-section through the throat.

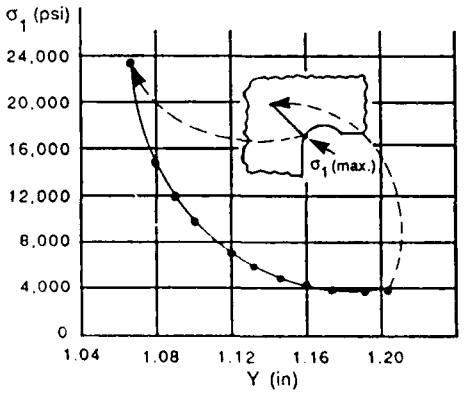


Second, the maximum stress location and relevant stress profiles, computed from the continuous shape functions:



▲Maximum stress

PROBE solutions are continuous over the domain. This allows easy extraction of maximum stress points or creation of stress profiles in critical regions as part of interactive post-processing.



▲Stress profiles

Conclusions

Traditional methods and computer programs for global structural analysis are performing well against the goal of predicting general distribution of internal loads and displacements in complex structures. The p-version of FEA, however, provides a more appropriate set of capabilities for detailed analysis of local phenomena; and new codes, such as PROBE, should be considered as important complementary tools for the well-equipped designer/analyst.

Acknowledgements

We wish to thank Cray Research for their technical assistance and for the use of the Cray-XMP in Mendota Heights, MN. Both example problems were solved on that machine.

PROBE[tm] is a product of Noetic Technologies Corporation, St. Louis, Missouri, USA, is represented also by Altair Engineering, Inc., Troy, MI, and in Europe by TEDAS GmbH.

Additional information may be obtained by contacting any of the following organizations.

Noetic Technologies	USA	(314) 961-6960
Altair Engineering	USA	(313) 680-1670
TEDAS, Marburg	FRG	(06421) 26077
TEDAS, Coventry	UK	(0203) 692252
TEDAS, LeChesnay	FR	(1) 39.63.32.32

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PREDICTING FATIGUE CRACK GROWTH UNDER COMBINED TENSION AND OUT OF PLANE BENDING IN TRANSITIONAL THICKNESS PLATES

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1.0 INTRODUCTION

1.1 Analysis of fatigue cracking problems in the lower surface wing panels of C-141 aircraft (References A and B) has shown that the local bending loads combine with tension loads to propagate the cracks. Material test data and stress intensity solutions ,however, are usually only available for the uniaxial case thereby making analytical predictions difficult. This paper presents an example of the C-141 lower wing skin in the stomper fitting area in which finite element analysis and fracture mechanics were applied in combination with available 3 dimensional stress intensity solutions for semi elliptical corner flaws subjected to combined bending and tension loads. The skin is machined from a 7075-T6 aluminium alloy extrusion. The 0.185 inch thickness of the skin in this area meant that it fell in the transitional region where neither plane stress nor plane strain conditions dominate.

1.2 Figure 1 shows the area of cracking in the C-141 center wing and details the location of cracks which were discovered during full scale fatigue testing and later in service. There are 11 panels which make up the lower surface of the C-141 center wing. The cracks analyzed here, known as joggle cracks, were found to initiate at the forward edge of the panels immediately outboard of the spanwise splice termination. The spanwise splice terminates due to a bend in the panels which accommodates the chordwise splice between the center wing and the inner wing. Although the predominent loading in the lower surface panels is pure tension, the local geometry in the joggle area (ie. at the bend) is such that local bending stresses are induced. This bending results in variation in the stresses through the thickness of the panels in the joggle area. Higher stresses are induced in the top of the thickness and lower stresses in the bottom of the thickness.

1.3 The higher stresses in the top of the thickness caused the joggle area cracks to start there with a quarter circular shape. Reference C details failure analysis for a 1.47 inch joggle crack and a 0.406 inch long joggle crack. That analysis revealed that as the crack grew, it developed a quarter

elliptical shape with the semi-major axis on the upper surface of the skin panel and eventually penetrated the thickness. Once through the thickness, the crack propagated quite rapidly with a pointed appearance. Figure 2 details a schematic of the crack geometry and highlights our concerns about the joggle cracks which were as follows;

a.Location and direction of cracks: The cracks were running chordwise in the lower surface of the wing indicating that they were mode 1 fatigue cracks.

b.Size of crack jumps: Metallurgical analysis revealed that once the crack became a through flaw, it grew in large overload jumps interspersed with regions of stable fatigue growth.

c.Shape of crack front: After penetrating the thickness, the crack developed a pointed appearance indicating that it was close to critical size.

1.4 Finite element models were created at WR-ALC (Section 2) and Lockheed (Section 4). These models simulated the stomper fitting area and revealed that significant bending stresses were present in the stomper fitting area. This explained, to some extent, the shape of the crack front which showed that it grew faster at the top surface due to higher tensile stresses there. The higher tensile stresses in the top surface of the skin resulted from the local bending. With finite element models to calculate stress values and distribution, a crack growth simulation could be accomplished. Prior to any analysis, flight restrictions were imposed on the aircraft pending fleetwide inspections. This decision was based on the observations commented on in paragraph 1.3.

2.0 WR-ALC FINITE ELEMENT MODEL

2.1 The MSC NASTRAN finite element model (FEM) used in the analysis is shown in Figure 3. The model represented a 19.6 by 19.8 inch section of the center wing panel splice and stomper fitting composed of thin shell quadrilateral elements. End restraints on the model were applied opposite the axial loads to simulate the end fasteners. All remaining fasteners located on the interior were modelled as springs with a stiffness. Springs were used instead of coupled constraints to minimize the stress concentrations which coupled constraints would impose. Restraints were also applied along the free edges of the model along the z axis to simulate the continuation of the panels. The axial loads applied were aquired from Lockheed stress reports on the C-141.

2.2 Figures 4 and 5 show the stress contour results after running the model. The riser visibility was turned off to permit a complete view of the skin stress distribution. As expected a stress concentration area was identified around the stomper fitting area (Figure 4). A plot of the top, middle, and bottom surfaces of the fitting area revealed three different stress distributions (Figure 5). The different distributions were concluded to be due to localized bending which had been previously overlooked. As the panel is loaded in tension, the joggle in the panel attempts to straighten inducing a localized bending moment. Figure 6 is a plot of the top, middle and bottom surface stresses from the joggle area of the FEM with a unit stress applied. The plots show the stresses moving aft from the panel edge toward the riser. Moving toward the riser the bending stresses were reduced because the riser provided additional stiffness and removed the load offset. At the edge of the panel the ratio of bending to tension stress was nearly 1.0, and this ratio went almost to zero at the riser. Based on the results shown in Figure 6, the ratio of bending to tensile stress was assumed to vary linearly from 1.0 at the panel edge to 0.0 at the riser.

2.3 The contour plots were of prime importance in the crack growth analysis. The identification of the localized bending prompted background research into known stress intensity solutions involving tension and bending. The stresses obtained from the FEM were used to compute the critical crack length and service life of the center wing/stomper fitting configuration.

3.0 WR-ALC CRACK GROWTH RUN AND RESULTS

3.1 Reference D is a paper which details the stress intensity solutions for three dimensional bodies subjected to combined tension and bending. Figure 7 details the geometry and loading conditions for which a solution is available. The stress intensity factor varies along the crack front. This is taken into account in the equations by the crack parametric angle, phi. As detailed in Figure 7, the stress intensity depends on the following parameters:

a. St - applied tensile stress

b. Sb - applied bending stress (outer fiber)

c. a/t - crack length (depth direction) to thickness ratio

d. a/c - crack aspect ratio (depth to surface direction)

e. c/b - crack length (surface direction) to width ratio

3.2 Joggle area cracks had been discovered during full scale fatigue testing at Lockheed. Reference E details the examination of such cracks. The appearance and shape of these cracks were identical to those found in service and reported in Reference C. Thus it was possible to determine an equation to represent the shape of the crack, ie. the crack aspect ratio (a/c) as a function of crack length in the surface direction (c). This reduced the stress intensity equations to vary with only one crack dimension. The number of variables was further reduced by assuming that the bending stress was induced by the remote tensile stress, reducing the stress intensity to vary with one stress. Thus, for a given width and parametric angle, the Reference D equations could be used to determine a representative stress intensity factor. A small Fortran program was written for this purpose.

3.3 The stress intensity factors were calculated at the surface (crack parametric angle, phi=0). This produced a set of geometric correction factors (betas). These betas were multiplied by 1.222 to account for the fact that the finite element model produced 44 KSI of tension in the stomper fitting area with the application of 36 KSI of remote tension (44/36=1.222). The crack growth programs apply remote tension. The stress intensity solution therefore accounted for the shape of the crack, the combined tension and bending loads, and the local geometry.

3.4 With the stress intensity solution known, a comparison could be made with the Lockheed test. Marker loads had been applied at various stages of the full scale fatigue test, so several points on the crack growth curve were known. The loading spectrum used in the Lockheed test was found in Reference F, and this was converted to a format suitable for use with the CRACKS84

(Reference G) crack growth computer program. After an initial run on CRACKS84, the model was fine tuned by making successive runs in which the betas were multiplied by a scaling factor. This scaling procedure was continued until a reasonable match was obtained resulting in a scaling factor of 0.625. Figure 8 shows the known crack growth curve for the Specimen A test compared with the prediction using the adjusted betas. Figure 9 is a plot of the adjusted betas versus crack length.

3.5 The adjusted betas were then used with the Lockheed Runstream program (Reference H) to predict the crack growth behaviour under actual C-141B usage. In addition to the SLA2B baseline, a new spectrum was created based on the reported usage for 1987. Figure 10 shows a plot of the predicted crack growth from a 0.05 inch initial flaw to critical. The critical crack length (computed at 100% design limit stress using a fracture toughness value of 66.0 ksi inch) was 0.976 inch. The 66.0 ksi inch value for fracture toughness was determined from the plot shown in Figure 11. The safety limit was found to be 9560 hours and 12265 hours for the 1987 usage and the SLA2B baseline respectively. Inspection interval here is defined as the time for the crack to grow from an assumed NDI detectable size (0.10 inch) to critical, divided by two. This would result in inspection intervals of 2200 hours and 2800 hours for the 1987 usage and SLA2B baseline respectively. Figure 10 shows growth up to a critical crack length corresponding to 80% of design limit stress. This was in order to determine the safety afforded by flight restrictions imposed to limit the loads to 80% of design pending inspections to determine the extent of cracking.

4.0 LOCKHEED FINITE ELEMENT MODEL AND RESULTS

4.1 A finite element model was also produced at Lockheed. The grid is detailed at Figure 12. It yielded similar results to the WR-ALC model. Figure 13 Details the stress contour plot revealing a high stress of 80 KSI in the stomper fitting area, thus highlighting the significant bending stress.

5.0 LOCKHEED CRACK GROWTH RUN AND RESULTS

5.1 Figure 14 details the joggle area crack fracture surface from the full scale fatigue test specimens. The crack growth curve could be re-constructed for this crack using the known marker load positions. This re-constructed curve is shown in Figure 15. With the specimen loads known, a set of geometric correction factors (betas) were obtained and scaled or back tracked to obtain a crack growth curve which matched the test. Figure 16 shows a plot of the "back tracked" betas. With these betas, a crack growth run was made with the Runstream program and the resulting crack growth curve is shown in Figure 17. This result compares well with the WR-ALC result shown in Figure 10. The reason for the WR-ALC curve being slightly more conservative was attributed to less severe loads in the Lockheed data base for the Runstream program.

6.0 DISCUSSION AND CONCLUSION

6.1 The prediction of fatigue crack growth and fracture is a difficult business, which is made even more so in situations where the crack shape and loading are complex, and the thickness falls in the transitional region. The 0.185 inch thickness in the joggle area of the C-141 lower wing skin is in the transitional region. In this case, it was fortunate that a crack growth curve under a given loading was known so that the crack growth model could be calibrated before being run with the latest aircraft loads. The results from this analysis enabled decisions to be made concerning the recurring inspection interval for panels inspected with no cracks found. It also highlighted the importance of the problem which will eventually result in a program to improve the damage tolerance and durability of the C-141 center wing.

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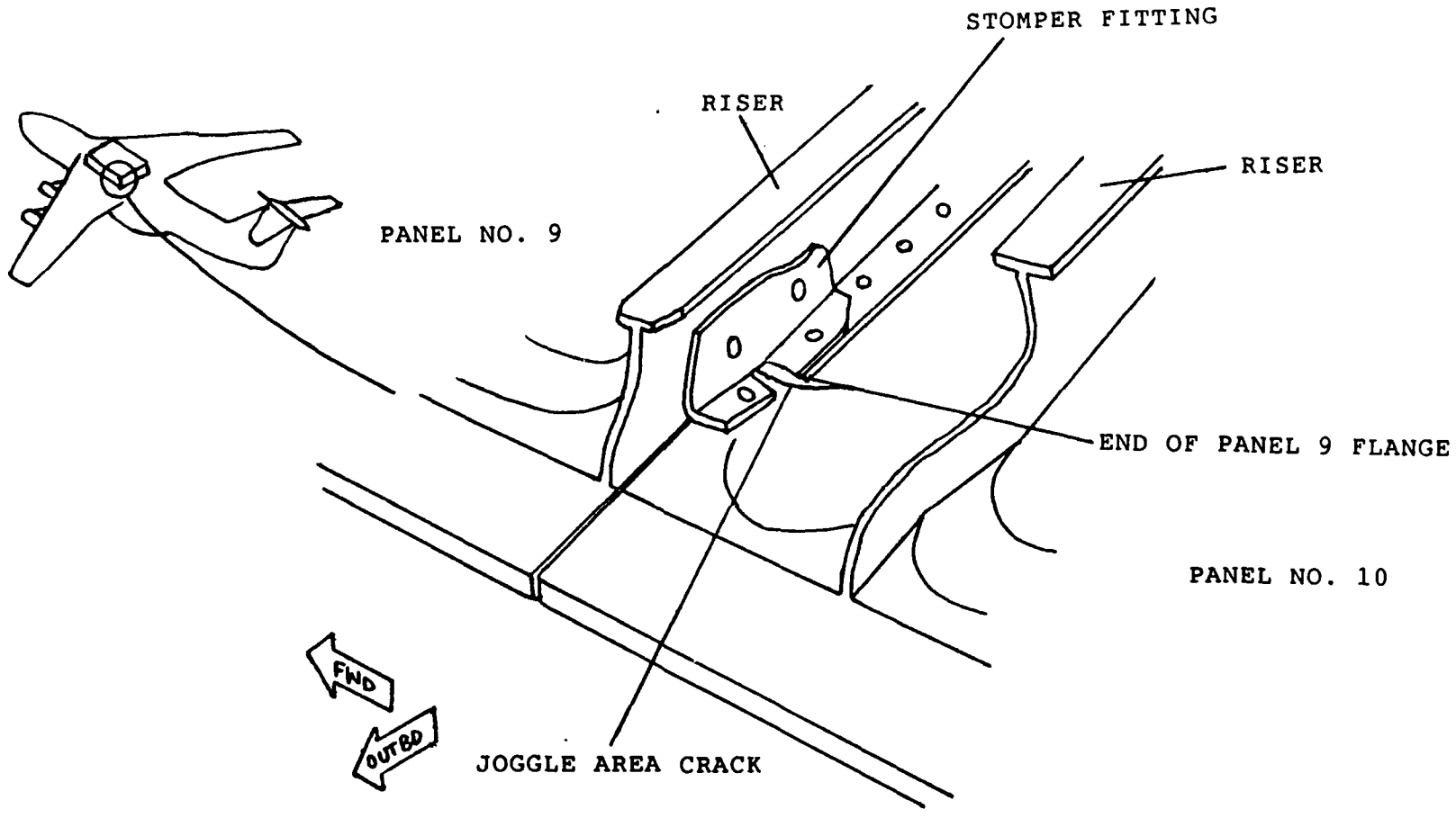
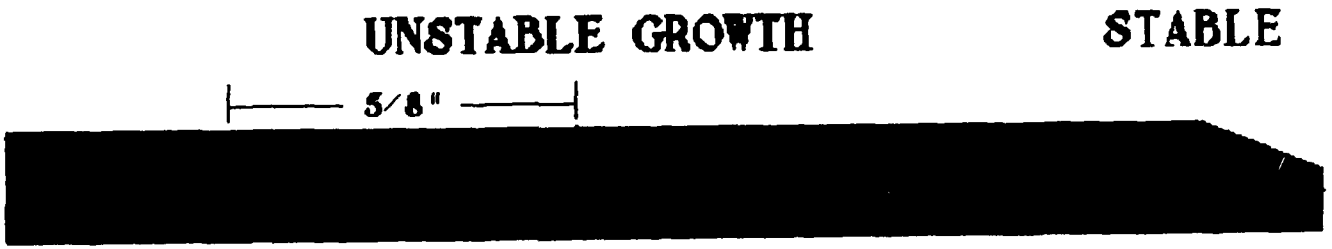


Figure 1 : C-141 Center wing lower panel cracking area

"JOGGLE" CRACK
METALLURGICAL ANALYSIS



CONCERNS:

* LOCATION DIRECTION OF CRACKS
* **SIZE** OF CRACK JUMPS
* **SHAPE** OF CRACK FRONT

Figure 2 : Schematic of crack geometry

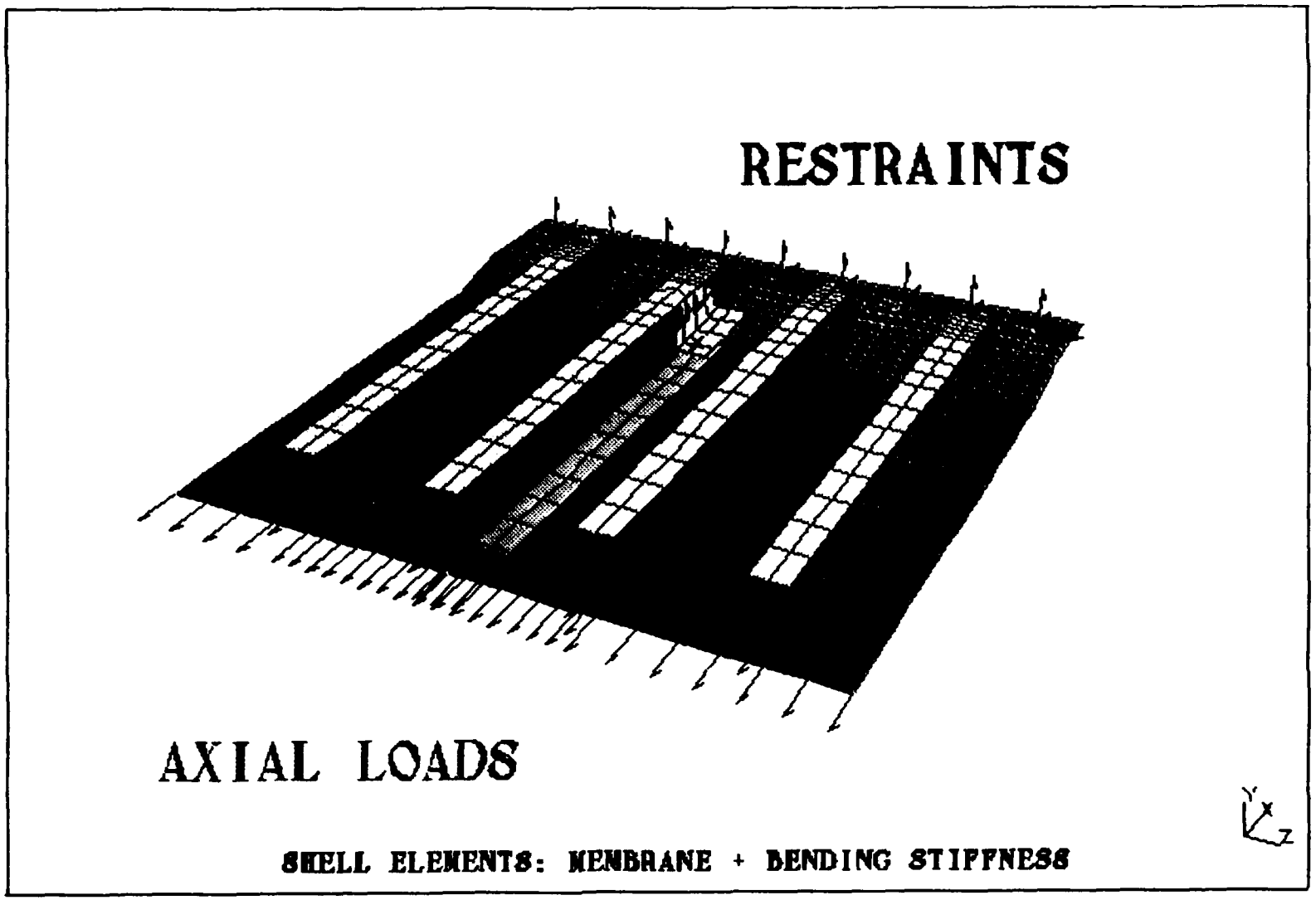


Figure 3: WR-ALC Finite Element Model

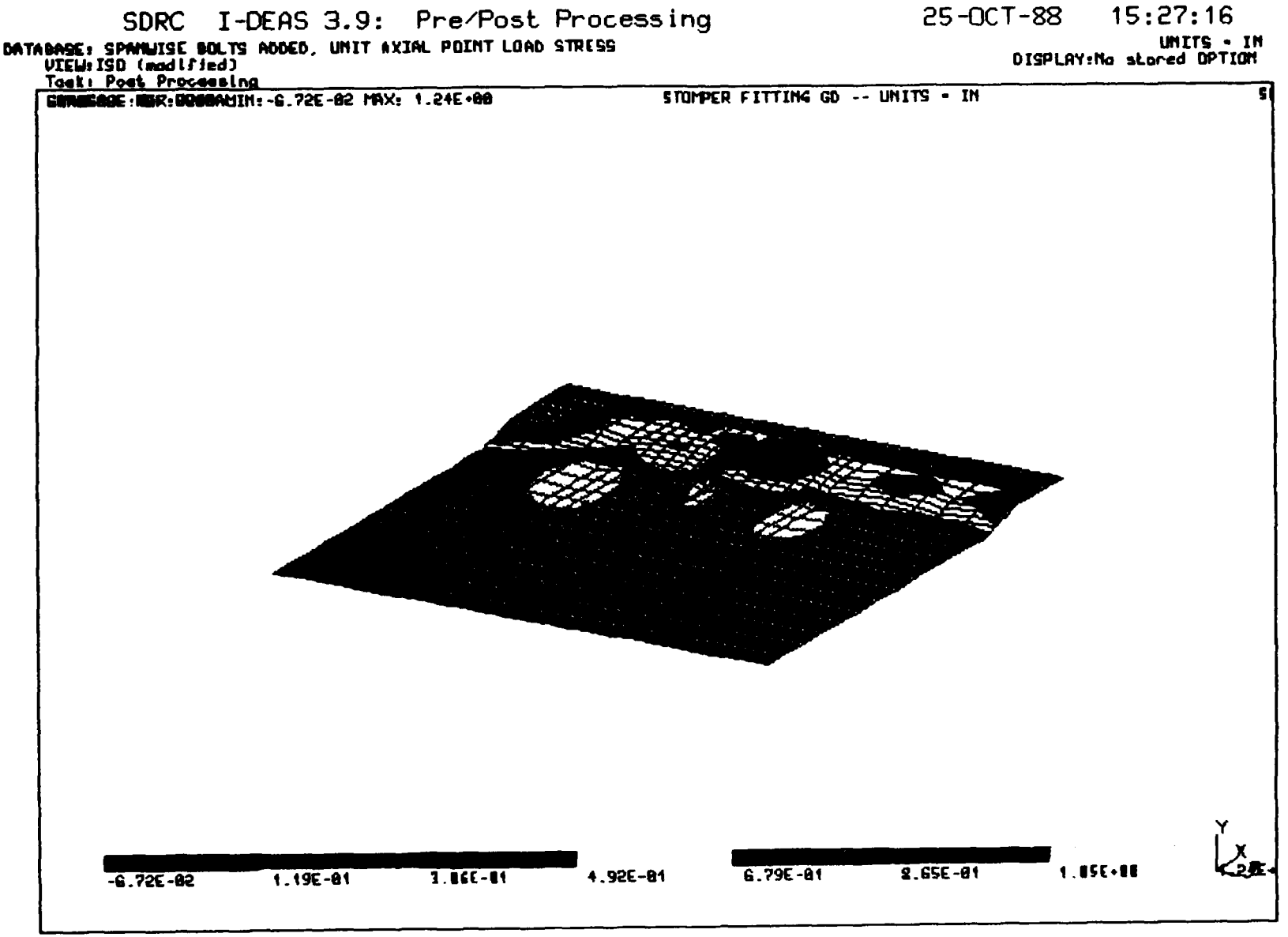


Figure 4: Stress Contour Plot (Skin Elements Only)

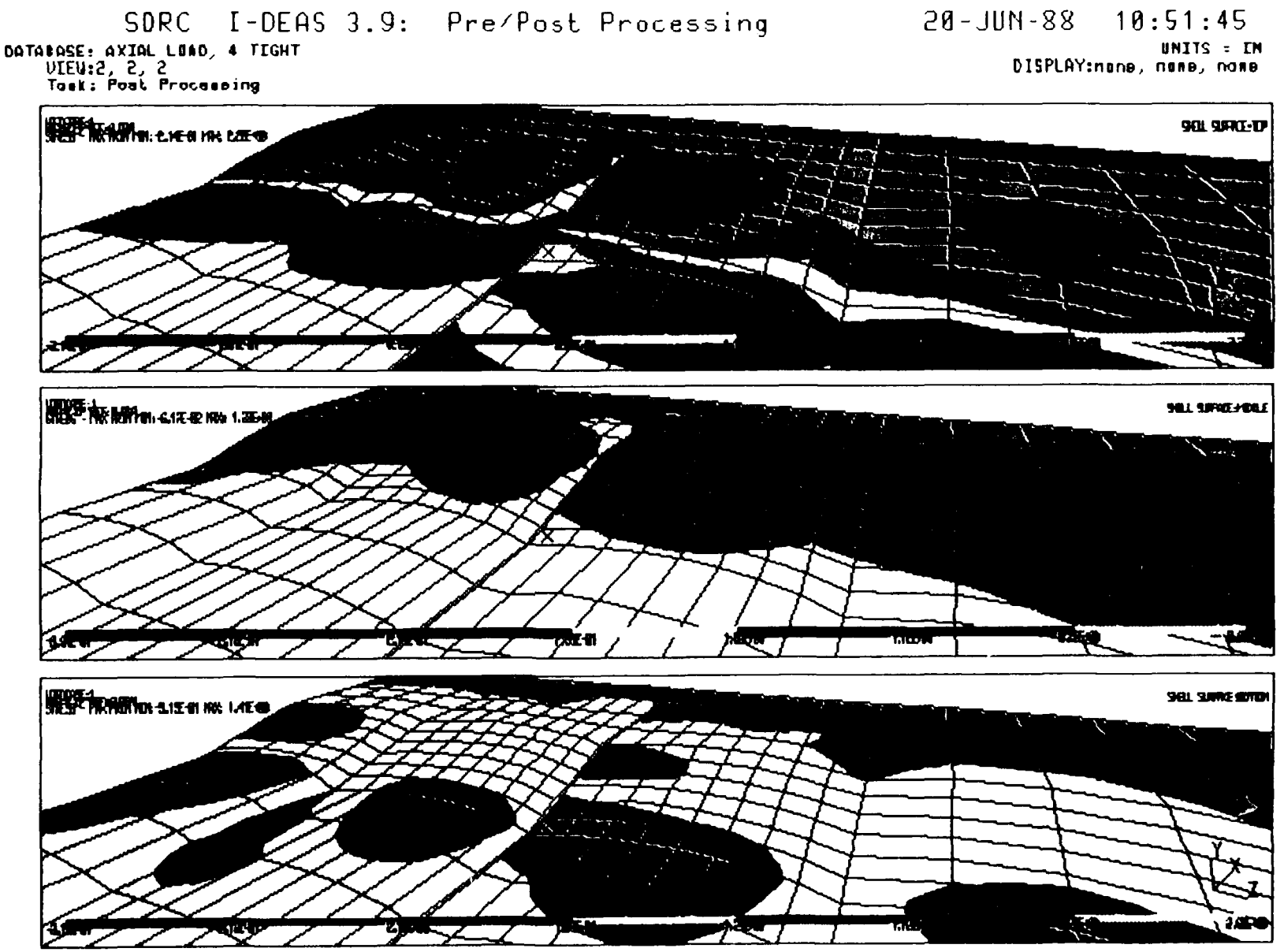


Figure 5: Stress Contour Plots for Top, Middle, and bottom Surface Stresses.

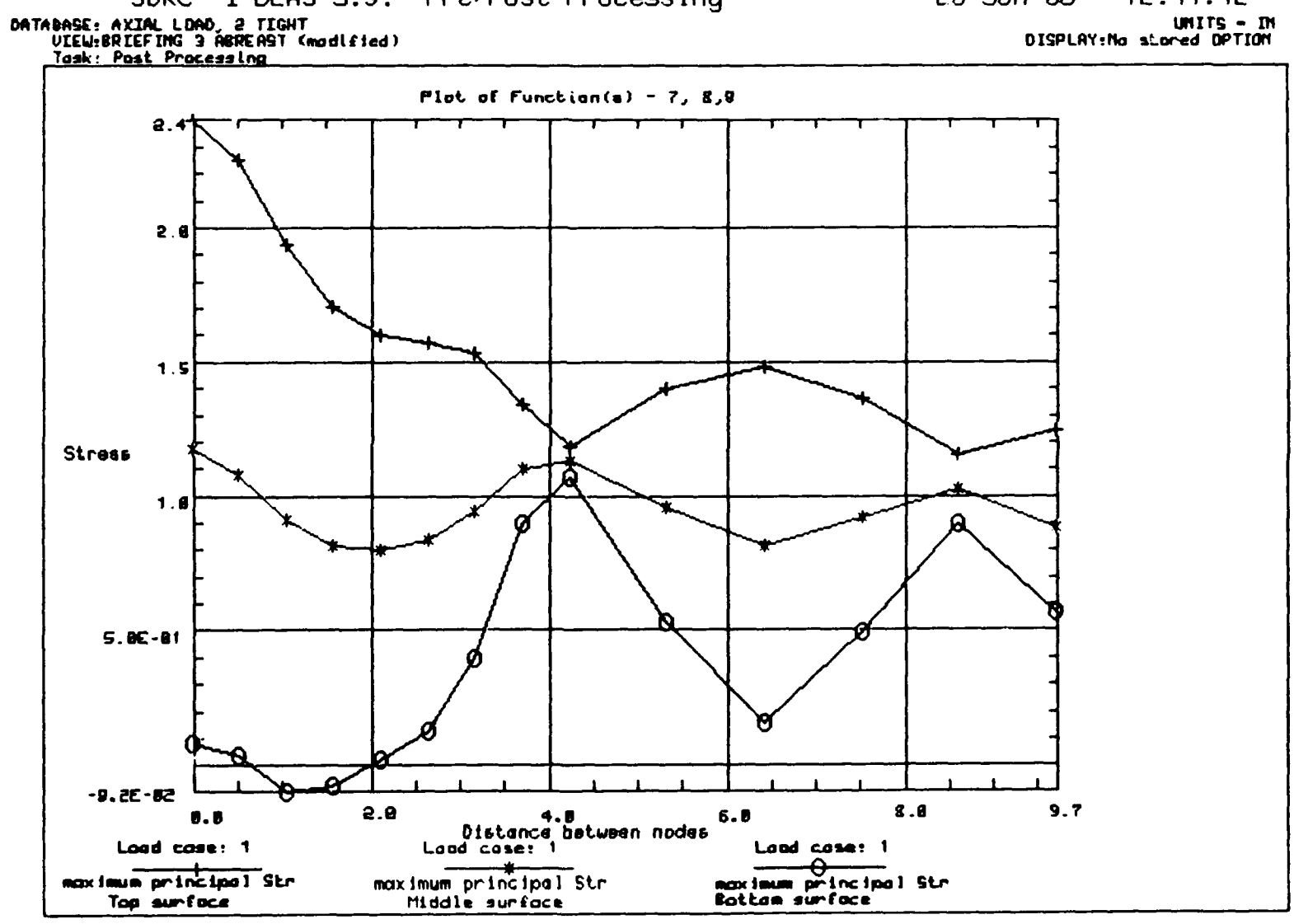
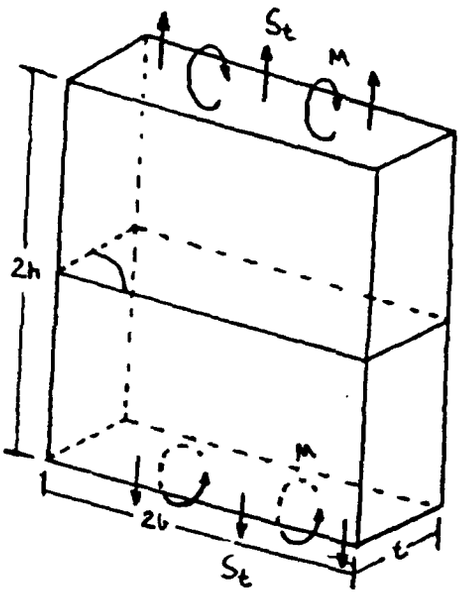
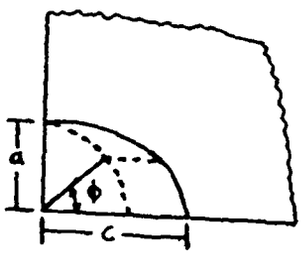


Figure 6: Reduction in bending Stress Near Risers



$$S_b = \frac{3M}{bt^2}$$



K is a function of $S_t, S_b, a/t, a/c, c/b$ and ϕ

Figure 7 : Quarter elliptical corner crack configuration

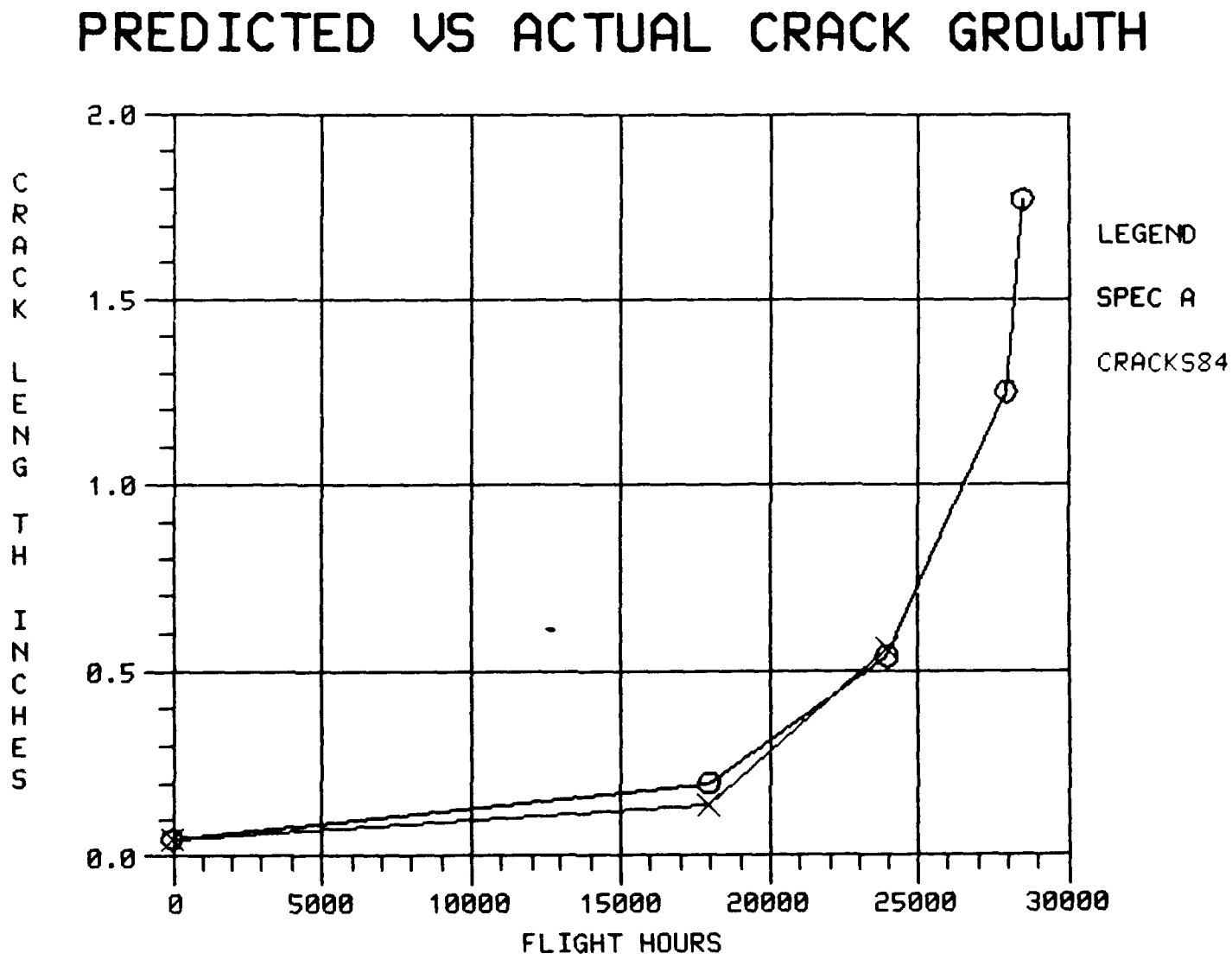


Figure 8: Specimen A Fatigue Test Result Versus Model Prediction

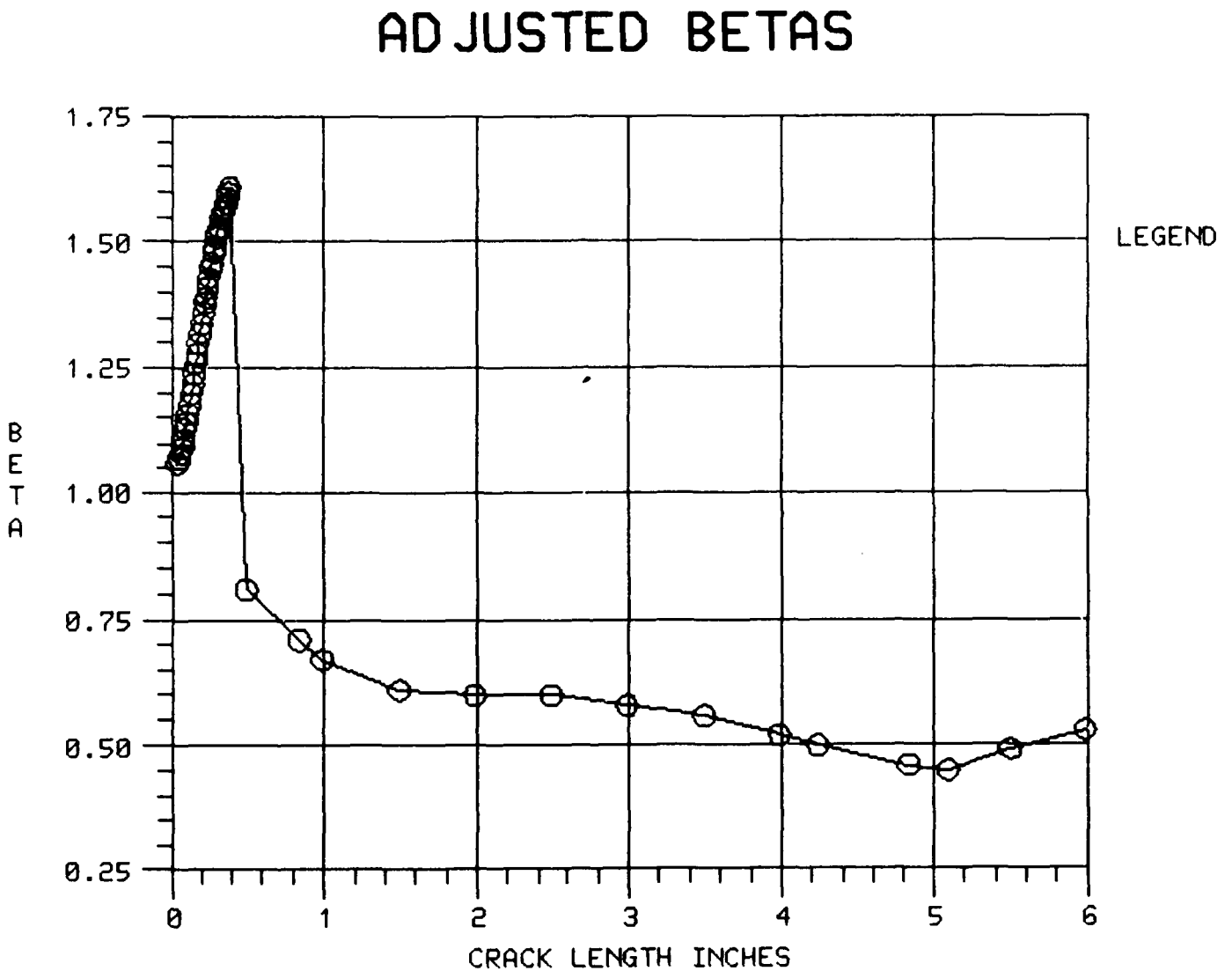


Figure 9: Adjusted Betas Versus Crack Length

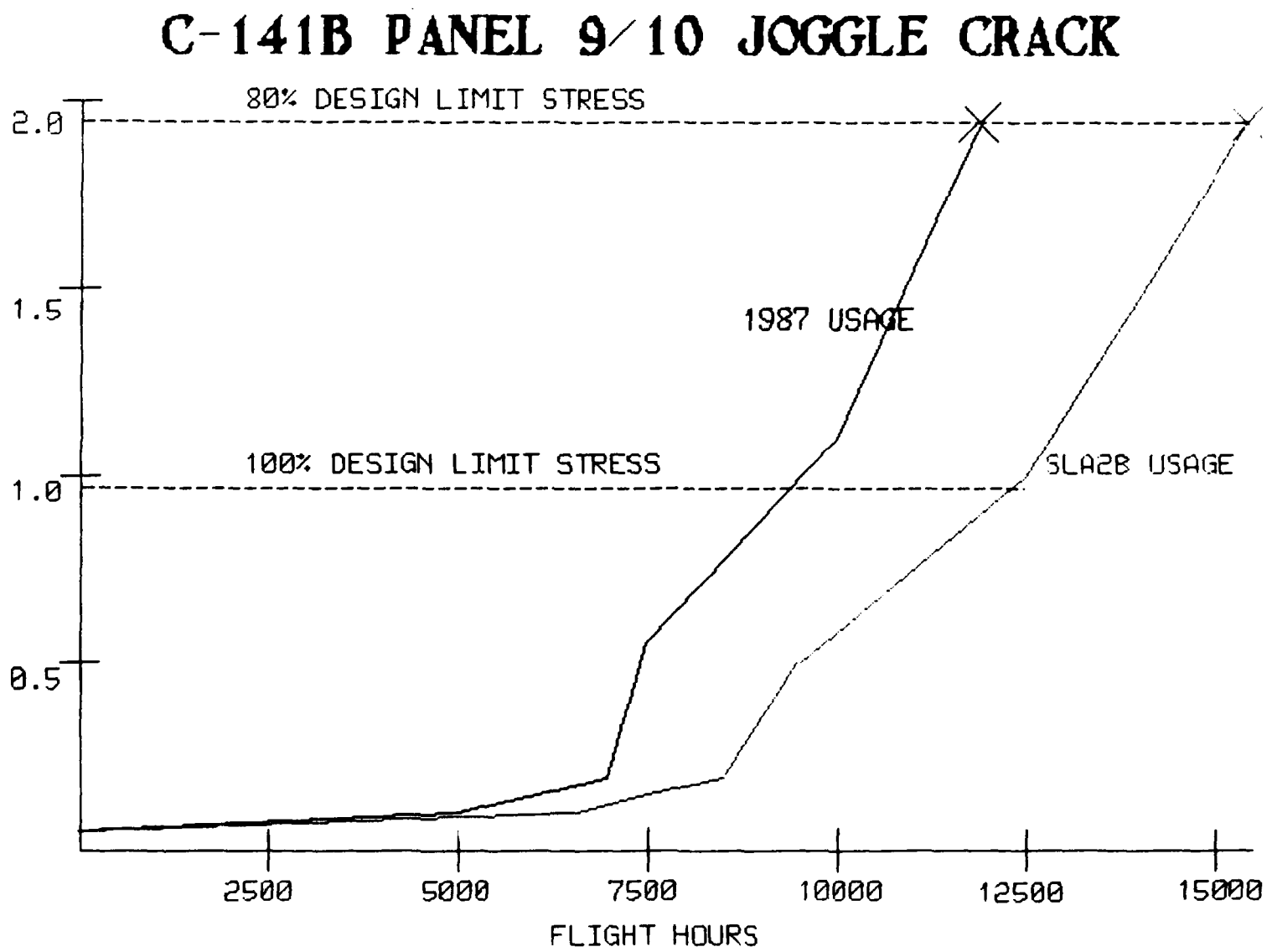


Figure 10: Predicted Crack Growth Under SLA2B Spectrum and 1987 Usage

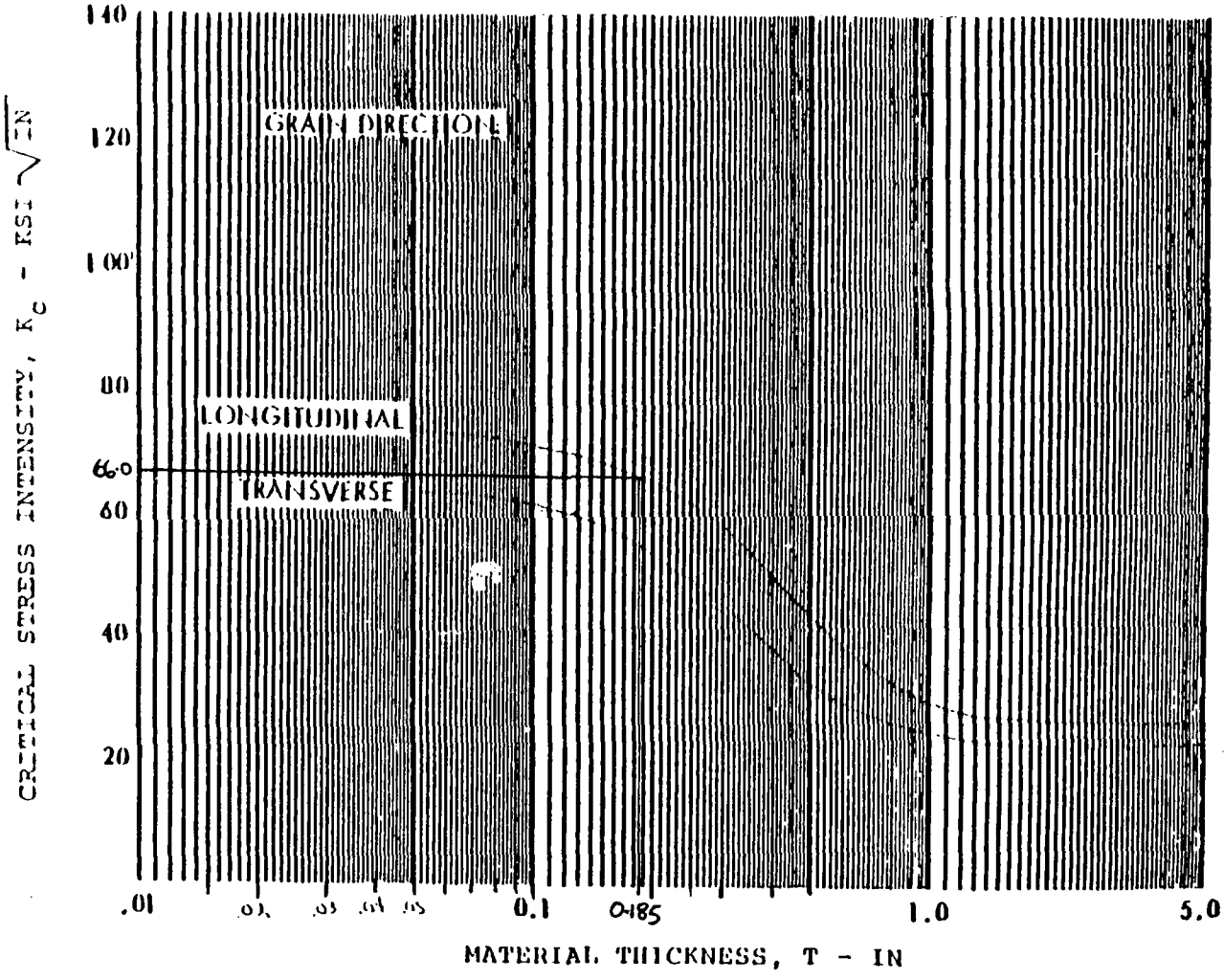


Figure 11 : Critical stress intensity factor versus thickness 7075-T6

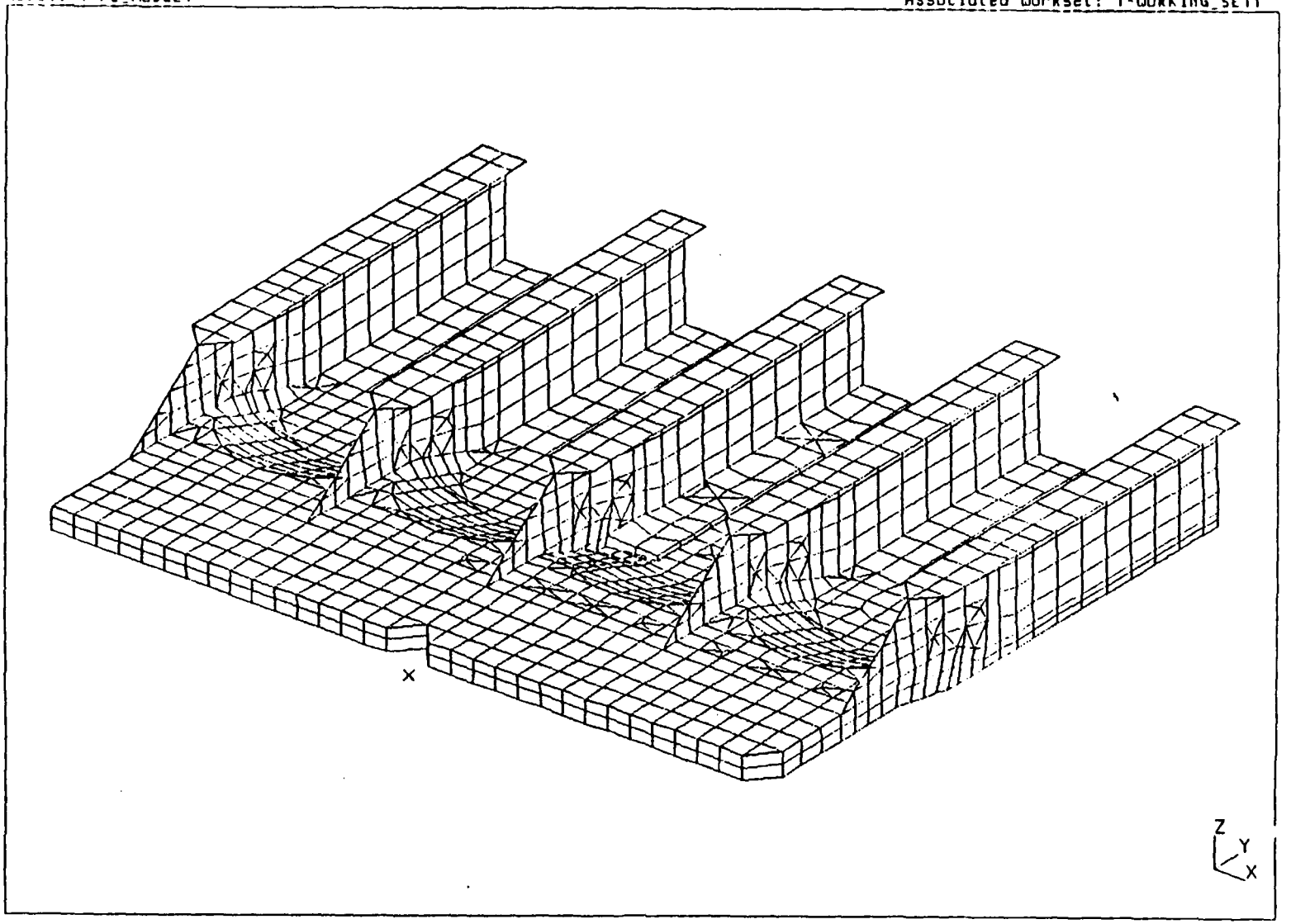


Figure 12 : Lockheed finite element grid

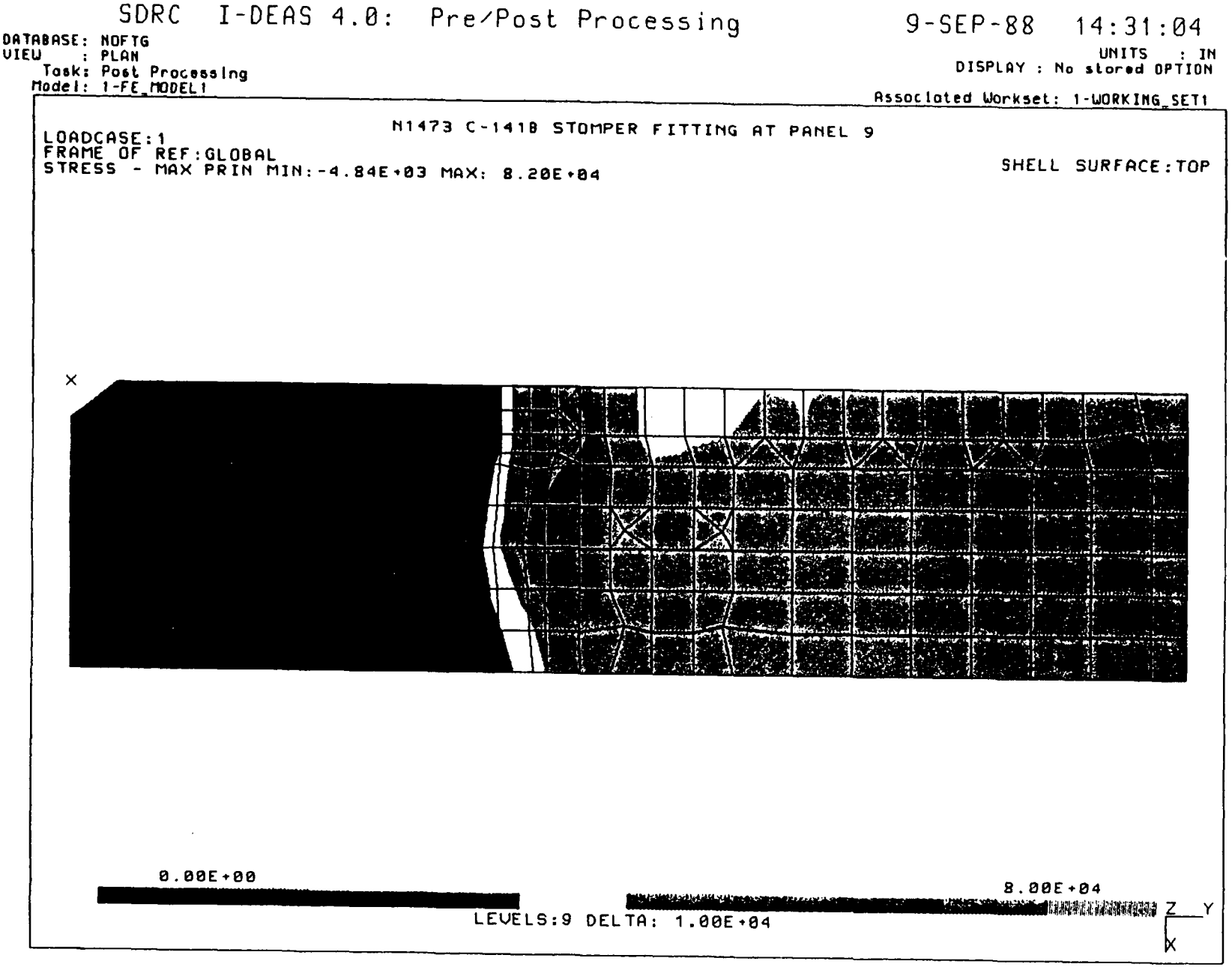
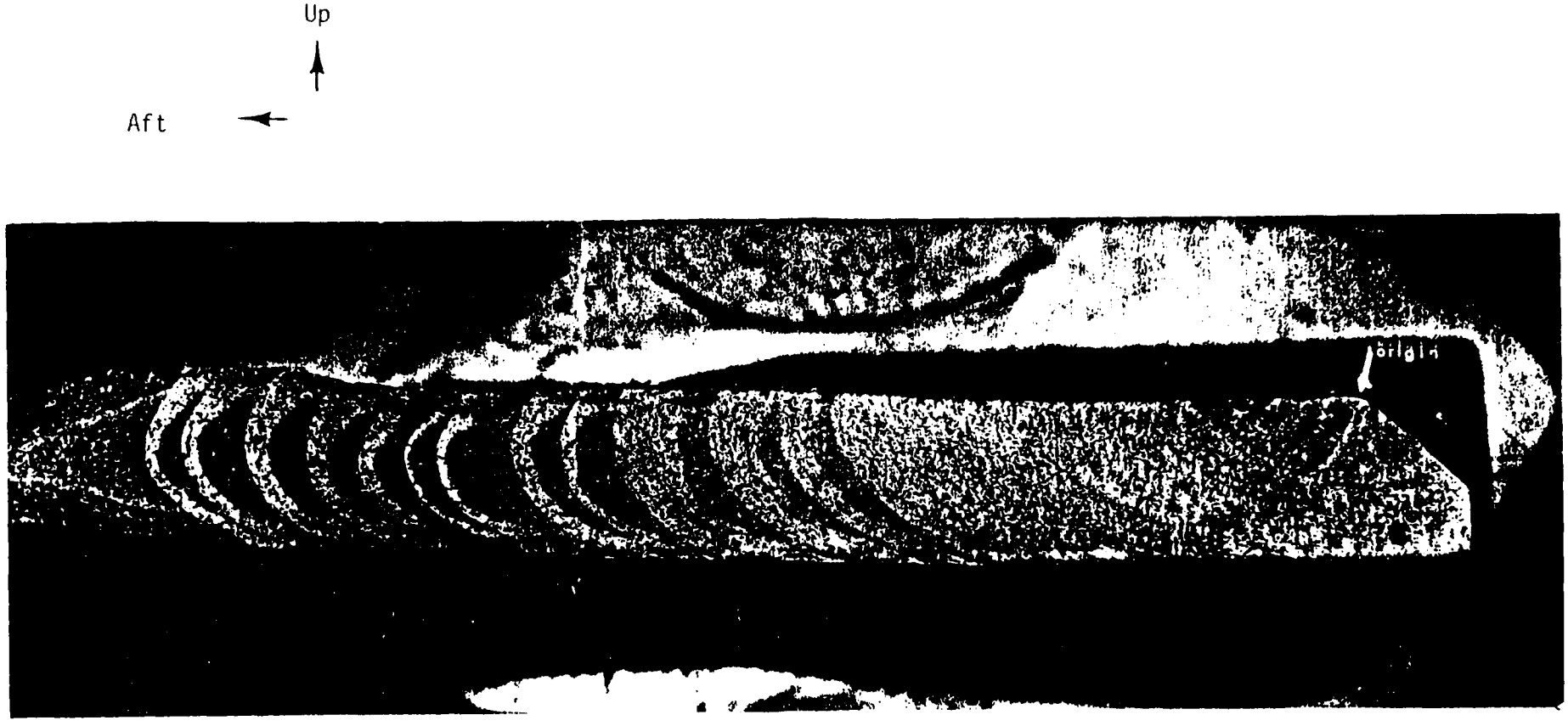


Figure 13 : Stress contour plot detailing bending stress



5.9X.

FIGURE 14. FRACTURE SURFACE OF CRACK FROM THE FORWARD EDGE OF THE SKIN PANEL.

The origin (arrow) was at the internal surface corner of the 45° chamfer. The location of the Pass 36 and 38 markers are as noted.

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CENTER WING JOGGLE AREA STUDY

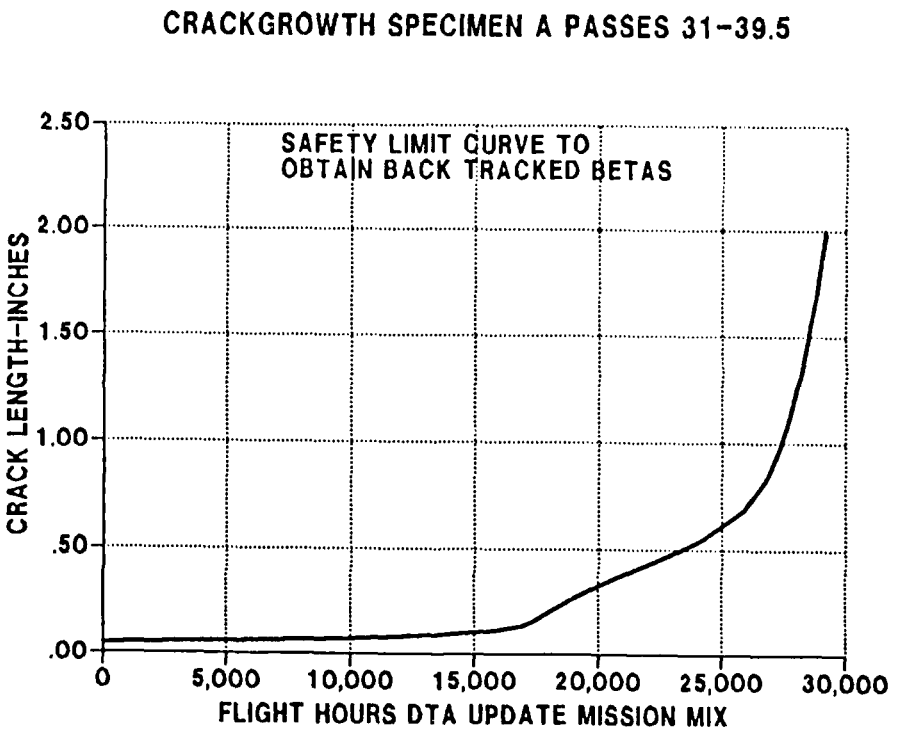


Figure 15 : Re-constructed crack growth curve

CENTER WING JOGGLE AREA STUDY

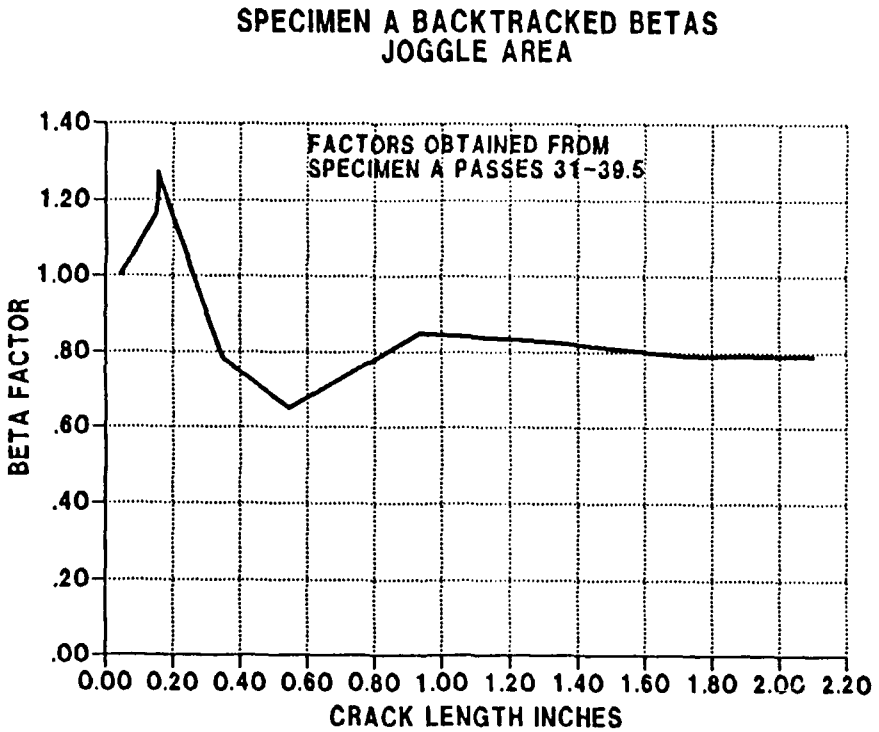


Figure 16 : Back tracked betas

CENTER WING JOGGLE AREA STUDY

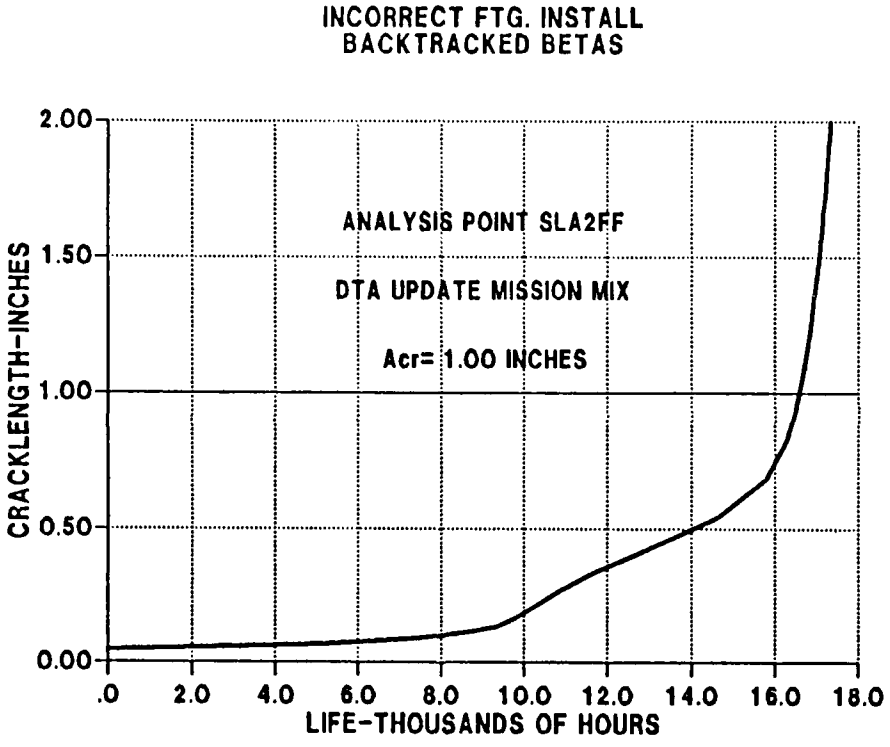


Figure 17 : Crack growth prediction using back tracked betas

CAPACITANCE MEASUREMENT SYSTEM (CMS)

USE AT

WARNER ROBINS AIR LOGISTICS CENTER

PRESENTED BY:
DEVARD O. HAMMOND JR.
C-141 ASIP MGR
WARNER ROBINS ALC

BACKGROUND

- REQUIREMENTS
- HOLE QUALITY
- EFFECTS

REQUIREMENTS

- DADTA
- LIFE EXTENSION OF C-141
- MODIFICATION PROGRAMS
- CENTERWING PROGRAM

HOLE QUALITY

- AUTOMATED TOOLING
- 100% INSPECTION
- EFFECTS
  - INCREASED LIFE
  - MAINTENANCE REQUIREMENTS

CMS DESCRIPTION

- OPERATING PRINCIPLES
- SYSTEM COMPONENTS
- CAPABILITIES

PRINCIPLES OF CMS OPERATION

- ELECTRICAL CAPACITANCE VARIATION
  - PROBE
  - HOLE WALL SURFACE
  - AIR GAP
- BASIC EXAMINATION MODES
  - HOLE SIZE
  - WALL SURFACE CONDITION

BASIC SYSTEM COMPONENTS

- SENSING DEVICE (PROBE)
- ELECTRONIC UNIT
- COMPUTER
- CONTROL UNIT

SYSTEM CAPABILITIES

- QUANTITATIVE DETERMINATIONS
- ACCURACY
- SPEED
- MEASURED PARAMETERS (MULTIPLE)

PARAMETERS

- SIZE
- OVALITY
- BELLMOUTHING
- BARRELING
- ANGULARITY
- FAYING SURFACE GAPS
- HOLE SURFACE CONDITION
  - ROUGHNESS, SCRATCHES, RIFLING TOOL MARKS

CONVENTIONAL INSPECTION

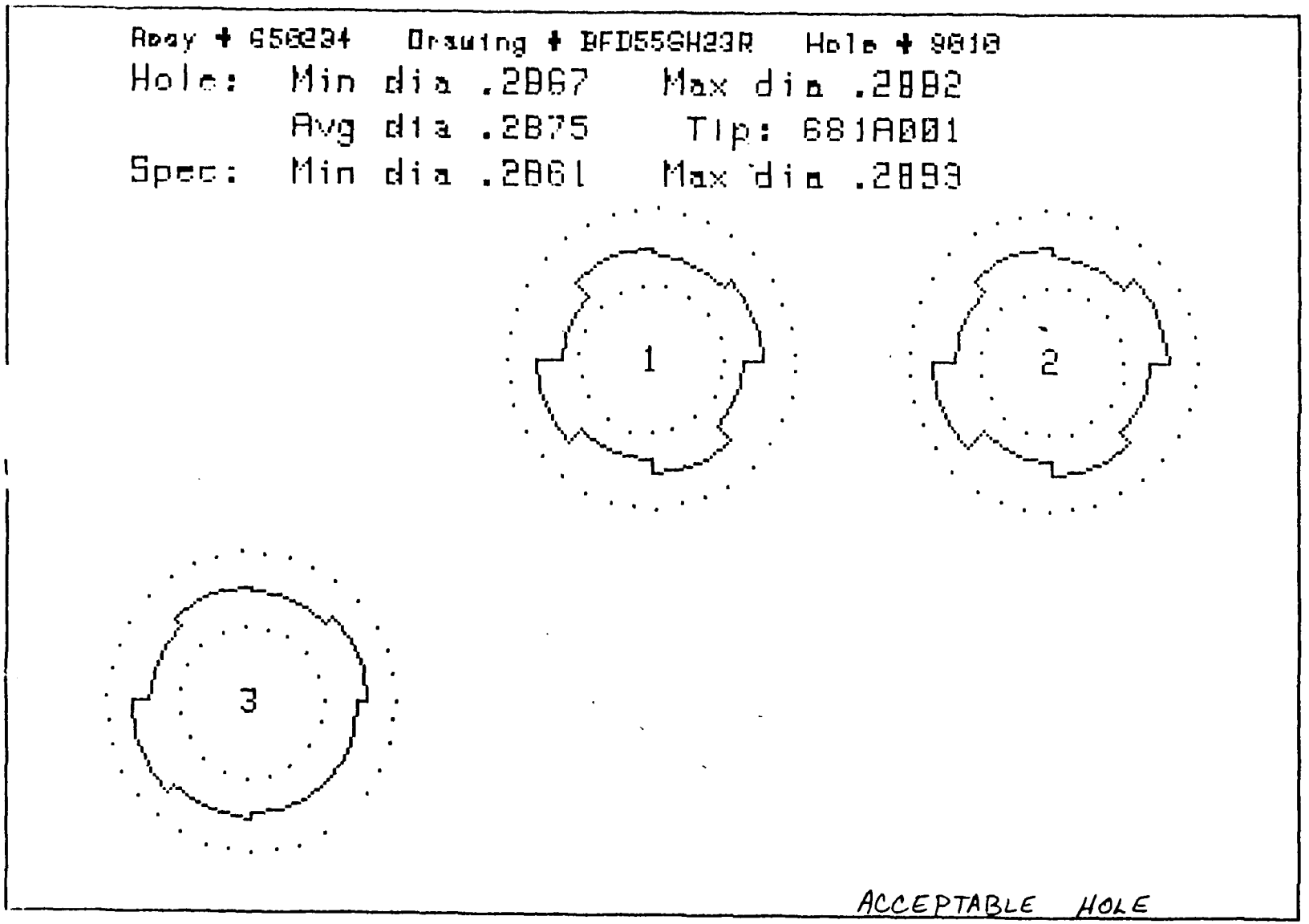
HOLE SIZE	PROTRUSION MEASUREMENT/AIR GUAGE
BELLMOUTHING	AIR GUAGE/DYE CHECK
BARRELING	AIR GUAGE/DYE CHECK
OVALITY	AIR GUAGE/DYE CHECK
TAPER	AIR GUAGE/DYE CHECK
FAY SURFACE GAPS	VISUAL/DYE CHECK
SCRATCHES	VISUAL/DYE CHECK
RIFLING	VISUAL/DYE CHECK
LAP/TEARS	VISUAL/DYE CHECK
PLASTIC DEFORMATION	VISUAL/DYE CHECK
TOOL MARKS	VISUAL/DYE CHECK
CHATTER	VISUAL/DYE CHECK
SURFACE FINISH	PROFILOMETER OR EQUIVALENT
ANGULARITY	PHYSICAL MEASUREMENT

USE AT WRALC

- 1987 START
- 6 UNITS, 12 WORK STATIONS & TOOL CRIB
- PROGRAMS
  - CENTER WING REPAIR
  - FATIGUE LIFE EXTENSION
- NDI SPECIALIST OPERATORS

RESULTS

- METHOD COMPARISON
  - 5 MIN/HOLE/INSPECTION
  - 3 SEC/HOLE/INSPECTION
- OVER 1500 HOLES AT WRALC
- OVER 2000 HOLES IN UPPER SURFACE ASSEMBLY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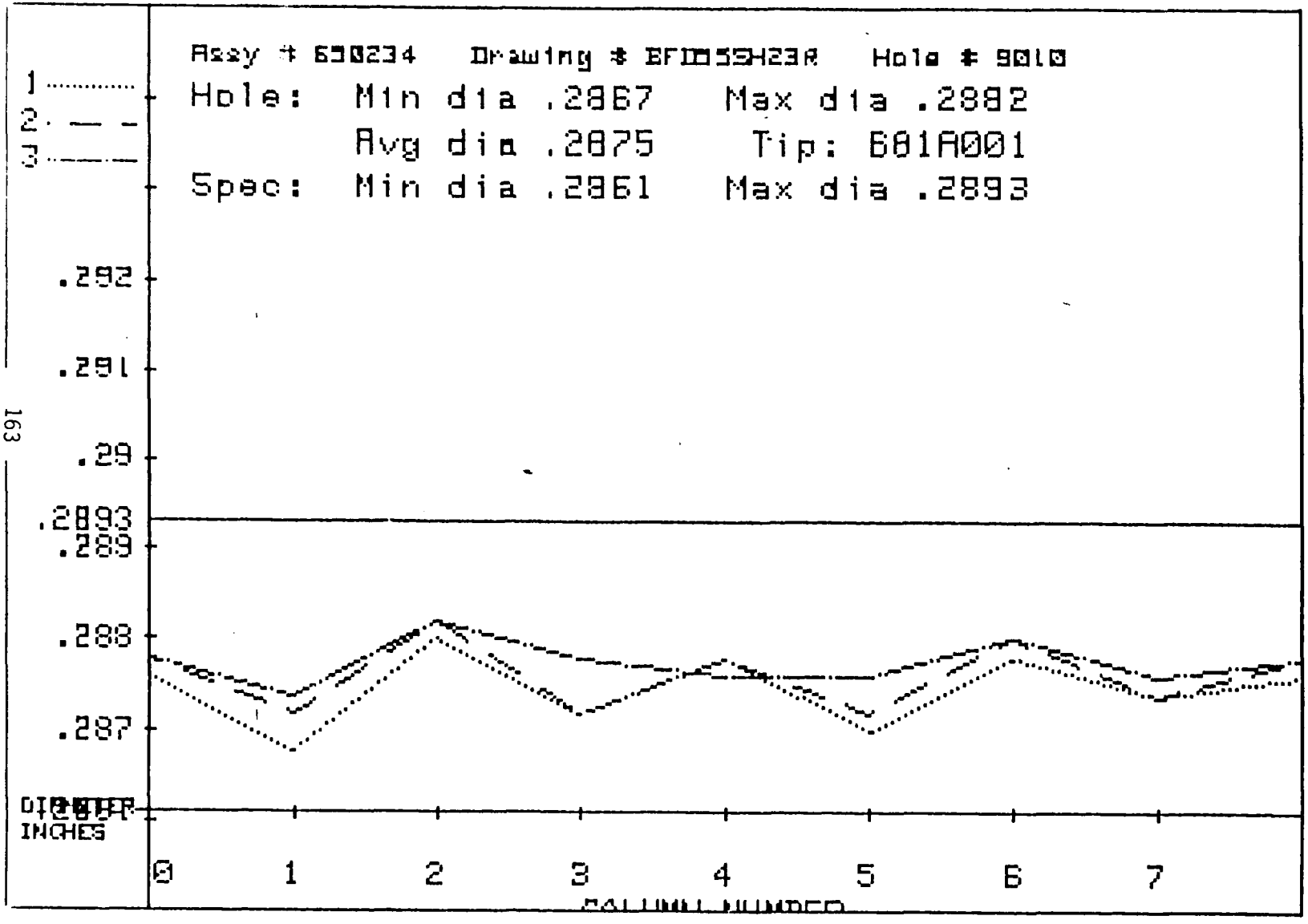
Avg dia .2875 Tip: 681A001
Spec: Min dia .2861 Max dia .2893
1
2
3
ACCEPTABLE HOLE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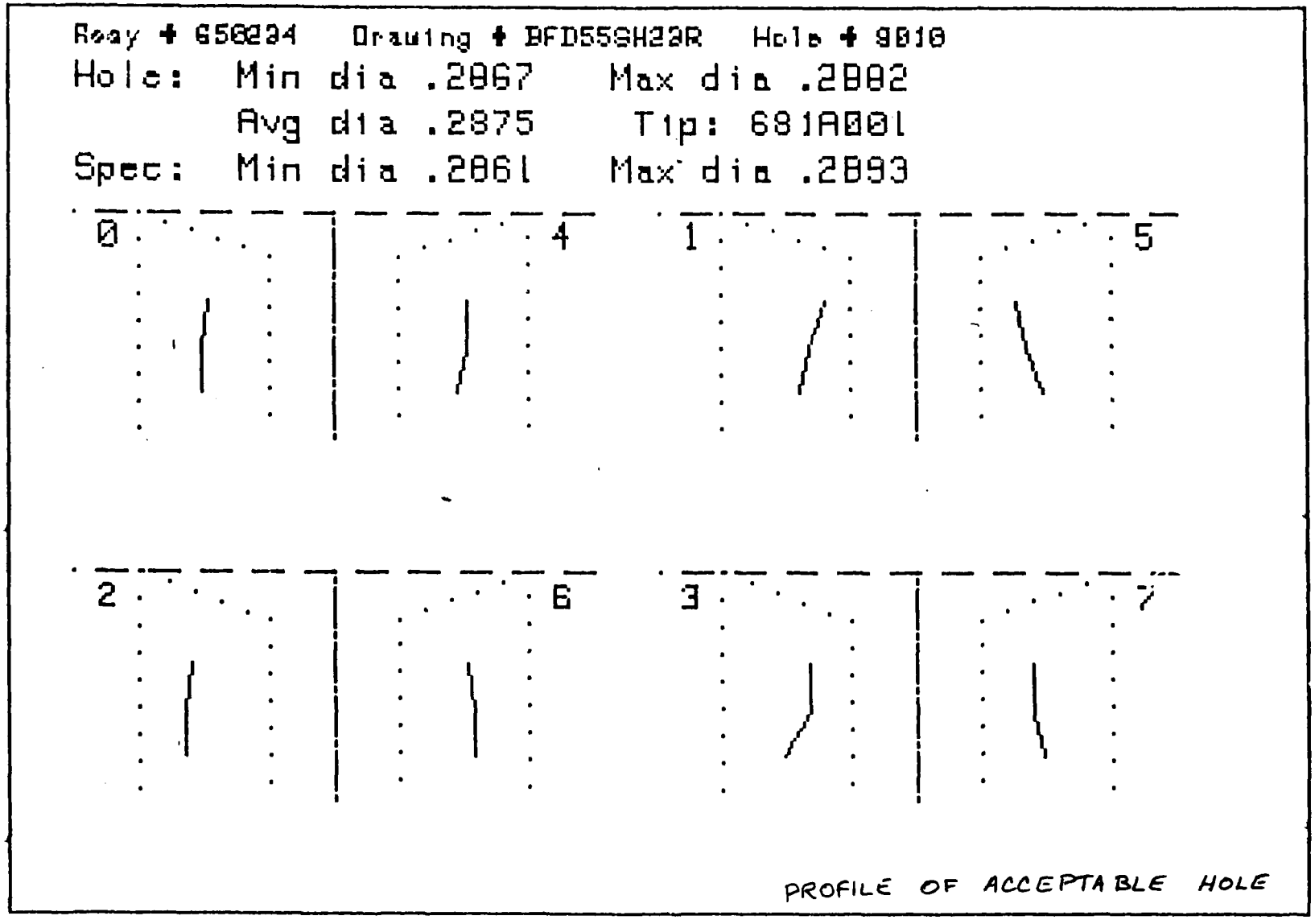
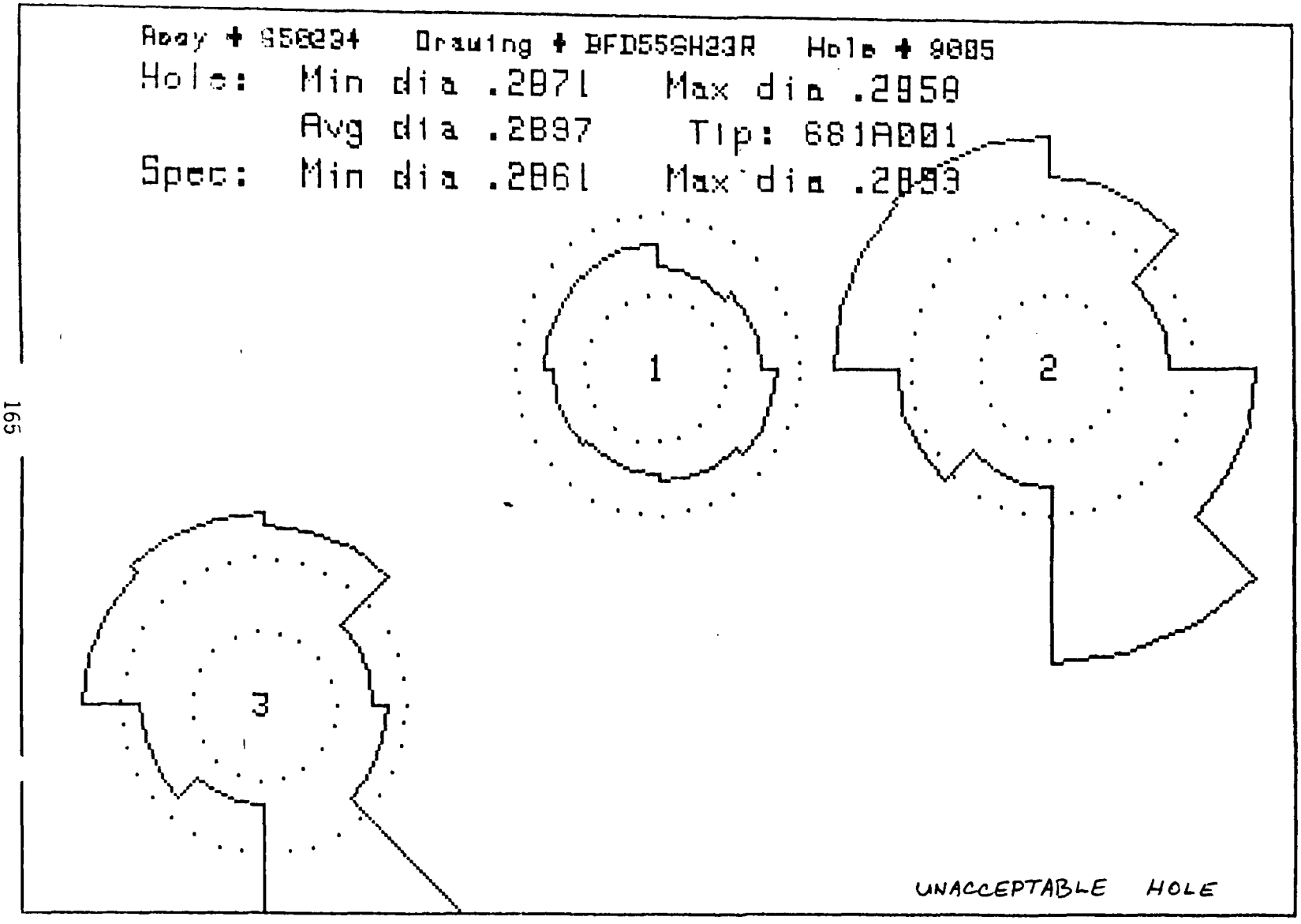
Avg dia .2875 Tip: 681A001
Spec: Min dia .2861 Max dia .2893
1
2
3
.292
.291
.29
.2893
.289
.288
.287
INCHES
0
1
2
3
4
5
6
7



PROFILE OF ACCEPTABLE HOLE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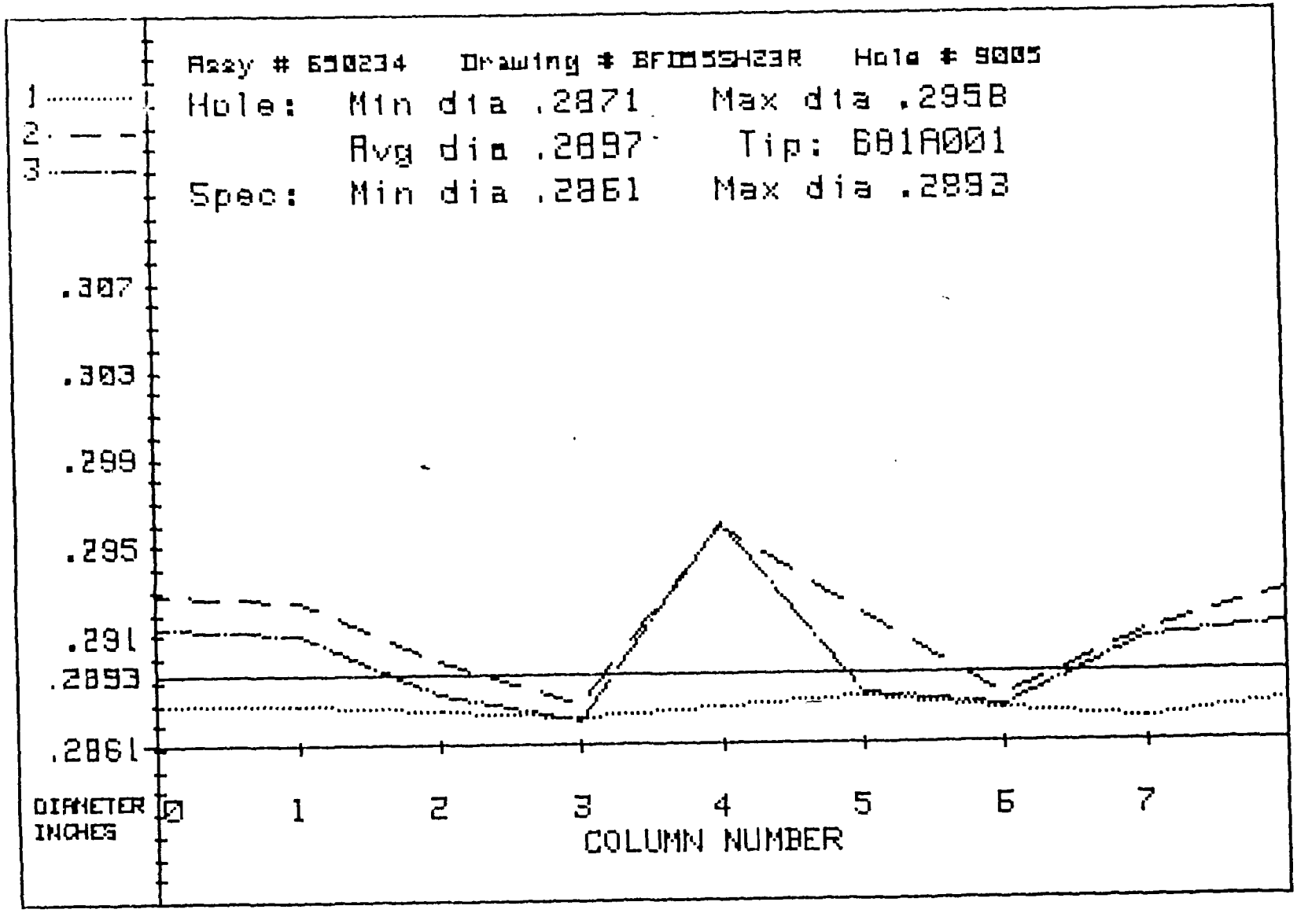
Hole: Min dia .2871 Max dia .2958
Avg dia .2897 Tip: 681A001
Spec: Min dia .2861 Max dia .2893
1
2
3
UNACCEPTABLE HOLE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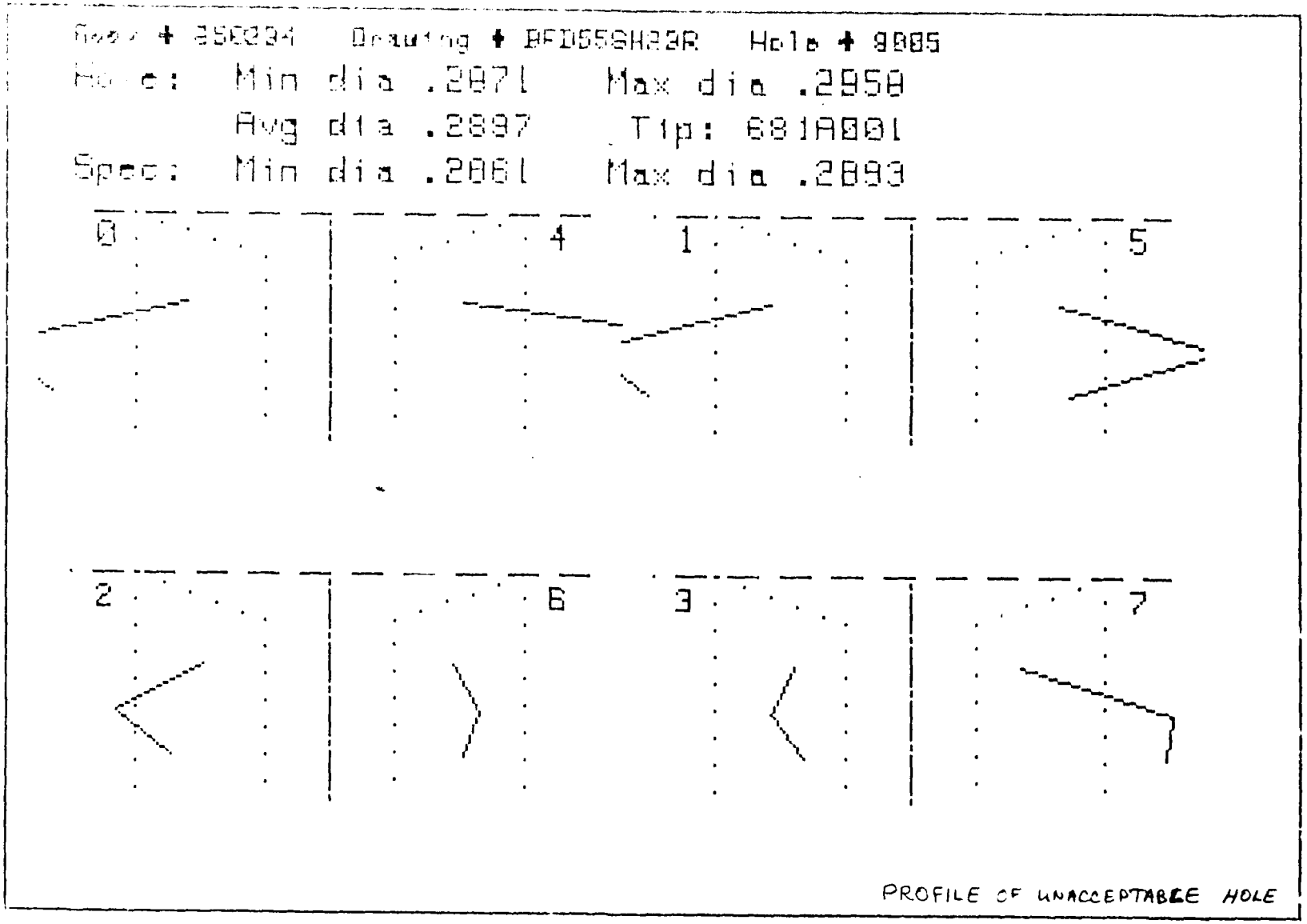
Hole: Min dia .2871 Max dia .2958
Avg dia .2897 Tip: 681A001
Spec: Min dia .2861 Max dia .2893
1
2
3
.307
.303
.299
.295
.291
.2893
.2861
DIAMETER INCHES
0
1
2
3
4
5
6
7
COLUMN NUMBER



PROFILE OF UNACCEPTABLE HOLE

CONCLUSIONS

- ACCURATE, QUANTITATIVE RESULTS
- PERMANENT RECORDS
- LESS INSPECTION TIME
- OBJECTIVE, REPEATABLE

CMS AT WRALC

INTRODUCTION

The Durability And Damage Tolerance Assessment (DADTA) for the C-141 aircraft was accomplished in the mid 1970s. Several areas of the aircraft were identified as requiring modification action to enable the service life of the aircraft to be extended from 30,000 flight hours to 45,000 flight hours. Most of the modification efforts consist of ream up of the fastener holes and installation of new oversize fasteners in fatigue susceptible areas. Efforts in these areas reaffirmed the importance of quality holes in aircraft structure. Methods to ensure this quality included stable mounted, automatic tooling and a rigid 100% quality inspection requirement. With this tight control it was determined that a smaller initial flaw could be assumed when performing DADTA analysis. This equates directly to longer life and fewer subsequent inspections. Subsequent to this type of rework initial inspection is scheduled to occur at the safety limit instead of at one half the safety limit. Recurring inspection intervals are scheduled at eighty per cent of the time to grow a crack from the maximum undetectable flaw size to critical crack length instead of one half the time.

Efforts have continued to better assure hole quality. These

efforts led us to investigate the use of the Capacitance Measurement System in this quest for quality fastener holes. Inspection methods used at Warner Robins Air Logistics Center (WRALC) to inspect for hole quality consisted of blue pin checks for tapered holes, ball gage checks for straight shank holes and visual inspections for surface finish inside the holes. With the advent of the C-141 Center Wing (CW) rework program the requirement to inspect fatigue critical fastener holes using CMS equipment was implemented for the C-141. Future Fatigue Life Extension Modification (FLEM) programs will include this requirement.

SYSTEM DESCRIPTION

The CMS system was developed by the Lockheed Corp. with support from the Air Force Wright Aeronautical Laboratories. It is manufactured and marketed by Measurement Systems Incorporated in Marietta, Georgia.

The CMS is based on the principle that electrical capacitance varies inversely with the distance between capacitor plates. CMS employs this property by locating a number of capacitor plates on a ceramic probe shaped to fit a particular fastener hole. This is then covered by a dielectric material (glass). The opposing capacitor plate is the material forming the walls of the hole to be inspected. What is actually measured is the air gap between the probe and the hole wall.

The system examines the hole in two basic modes, hole size measurement and wall surface condition determination. Hole size

is inspected by measuring the effective distance of each individual capacitor element from the hole wall adjacent to that element. Surface condition is determined by examining variations between capacitor elements. This determination is done by using algorithms in the software. Actual measurements are mathematically analyzed, compared with the algorithms for defect conditions, and the CMS is programmed to accept or reject the hole accordingly, regardless of the orientation of the defect or the probe.

The CMS consists of four major components; a sensing device or probe; an electronic unit which converts the signals into data that can be processed by the computer; an HP 9826 computer; and a hand held control unit which receives the analysis from the computer and tells the operator whether the hole meets the established criteria. All the components have been incorporated into a mobile cart to allow use in the maintenance areas.

The system offers the capability of quantitatively determining whether all the configuration parameters of a fastener hole have been met. With one probe insertion, size, ovality, bellmouthing, barreling, hole angularity, faying surface gaps and the surface condition of the hole (including surface roughness, scratches, rifling, and tool marks) can be determined. These readings are obtained with accuracies of $\pm$.0001 inch and resolution of $\pm$20 microinches.

Actual inspection is performed by inserting the probe into the hole and pushing the test button on the control unit. In less than three seconds the hole will be accepted as signified by a green light, rejected by a red light or a yellow light will

require a retest. Hole numbers can be called up automatically or any hole can be selected by simple input. The readings are stored on diskettes for permanent storage. The results for each hole inspected can be displayed graphically on a built in CRT if desired. Also, paper copies with tabular data and graphical hole representations are obtainable.

USE AT WRALC

WRALC began using the CMS in 1987. The CW rework program replaces the CW upper surface assembly and refurbishes the interior of the CW box. CMS is required for use in the manufacture of the upper surface assemblies which are procured from a contractor. It is also required at WRALC for use on fastener holes during installation. Each CW work station (12) will be equipped with a CMS unit. The CMS equipment is operated by Non Destructive Inspection (NDI) specialists.

The inspection of an individual fastener hole takes approximately 3 seconds. This compares to the blue dye pin check of a tapered fastener hole which takes about 5 minutes per hole, including cleanup of dye from hole. There are 2200 tapered fastener holes in the upper surface assembly. There are over 1500 holes inspected at WRALC during assembly. One can see that man hour savings can accumulate rapidly on the CW program alone.

Conclusions

The CMS is a significant state-of-the-art advance in the inspection of fastener holes. The benefits of its use are numerous. Inspection using this system is a one step operation which eliminates human error and subjective judgment. Accurate, objective, quantitative and repeatable results are readily obtainable. The overall image is a composite of up to 48 separate sections of the hole, therefore the entire hole surface is well covered by the inspection. Permanent records are easily obtained and retained. Quality is assured to a greater degree at less cost than previous methods.

AN UPDATE ON

A REVIEW OF THE QUALITY OF

SCREW THREADED PRODUCTS

MR C PETRIN ASD/ENFS

ASD 88 2446

NARRATIVE

TITLE CHART

LAST YEAR I GAVE YOU A RUNDOWN ON THE RESULTS OF A YEAR LONG STUDY ON THE GOVERNMENT REPROCUREMENT OF SCREW THREADED PRODUCTS. AT THAT TIME RECOMMENDATIONS FOR CHANGE HAD BEEN MADE AND WERE BEING MULLED AROUND IN THE AIR STAFF. TODAY I WOULD LIKE TO BRIEFLY COVER THE EVENTS OF THIS YEAR.

NARRATIVE

WHERE WE WERE A YEAR AGO

AS I SAID, A YEAR AGO WE WERE ANTICIPATING THE RELEASE OF A FINAL POLICY FROM HQ USAF. WE HAD STARTED TO ADDRESS SOME CHANGES TO SPECS AND STANDARDS WHICH THE STUDY HAD SHOWN NEEDED UPDATING REGARDLESS OF THE FINAL POLICY. MOST OF THESE CHANGES WERE BRIEFED LAST YEAR AND I WILL NOT GO OVER THEM AGAIN.

WHERE WE WERE A YEAR AGO

- AF/LE-RD WAS ABOUT TO ISSUE AN UPDATE OF THEIR FEB 87 POLICY
- SPECIFICATIONS FOR CLASS 3 SCREW THREAD FORM WERE UNDER REVISION
- OTHER SPECS AND STANDARDS HAD BEEN IDENTIFED AS REQUIRING REVISION

NARRATIVE

POLICY

SAF/A AF/LE LETTER 8 DEC 87

THE POLICY WAS FINALLY ISSUED ON 8 DECEMBER. IT APPLIED ONLY TO AEROSPACE CLASS 3 FIT SCREW THREAD PRODUCTS.

THE STRONGEST STATEMENT WAS SIMPLY THAT THE AIR FORCE WILL USE SCREW THREADED PRODUCTS THAT CONFORM TO SPECS. THIS IMPLIES OF COURSE THAT WE MUST MAKE ALL NECESSARY EFFORTS TO ENSURE WHAT WE BUY CONFORMS TO THE DESIGN SPECS.

THE POLICY ALSO STATED THAT THE DESIGNERS SHOULD HAVE DESIGN FREEDOM TO MEET THOSE PERFORMANCE CHARACTERISTICS WE WANTED. IT SAID ESSENTIALLY - CHANGE MIL STANDARDS WHICH MANDATE THREAD FORM.

THE THIRD ELEMENT OF THE POLICY WAS A WARNING TO THREADED PRODUCT SELLERS THAT THE AIR FORCE WAS NO LONGER GOING TO BEAR THE COST OF FINDING AND REPLACING NON-CONFORMING PRODUCTS.

POLICY

SAF/AQ AF/LE LETTER 8 DEC 87
SCREW THREADED PRODUCT QUALITY

- FOR AEROSPACE (CLASS 3) SCREW THREADED PRODUCTS
- USE SCREW THREADED PRODUCTS THAT CONFORM TO SPECS
- USE STANDARDS THAT ALLOW DESIGNERS FLEXIBILITY TO ACHIEVE AN OPTIMUM BALANCE OF PERFORMANCE, SAFETY, RELIABILITY AND INTERCHANGEABILITY WITH MINIMUM COST, LOGISTICS INVENTORY AND MAINTENANCE
- PAST HISTORY WILL BE A FACTOR IN FUTURE CONTRACTS TO SCREW THREAD PRODUCT MANUFACTURERS, SUPPLIERS AND DISTRIBUTORS

NARRATIVE

POLICY

THE POLICY ALSO STATED THAT THE INTENDED USE OF THE PRODUCT OUGHT TO ESTABLISH HOW IT WAS DESIGNED AND HOW WE VERIFIED ITS COMPLIANCE WITH THE DESIGN. IT ESTABLISHED DIFFERENT GROUND RULES FOR THREADED PRODUCTS USED IN SAFETY APPLICATIONS. THIS WAS INTENDED TO RAISE THE AWARENESS THAT SOME MUNDANE FASTENERS COULD MEAN LIFE OR DEATH.

THE POLICY ALSO SET ITS OWN EXPIRATION TERMS. IT WAS DEFINITELY DESIPED TO INSTITUTIONALIZE THESE POLICIES.

POLICY

- ORGANIZATIONS RESPONSIBLE FOR TECHNICAL REQUIREMENTS SHALL CLASSIFY SCREW THREADED PRODUCTS IN WEAPON SYSTEMS AND SUPPORT EQUIPMENT ACCORDING TO THE CONSEQUENCES OF FAILURE.
  TWO CLASSIFICATIONS ARE ESTABLISHED:
  SAFETY CRITICAL
  OTHER
- USE FMECA PROCESS FOR SYATEMS IN DEVELOPMENT OR PRODUCTION. USE TAILORED FMECA AND/OR EXPERIENCE ON OTHER SYSTEMS AND SUPPORT EQUIPMENT. MODIFY DRAWINGS TO REFLECT CLASSIFICATION CATEGORY AND UPDATE REPROCUREMENT DATA
- THIS POLICY SHALL REMAIN IN FORCE UNTIL INCORPORATED INTO APPROPRIATE REGS, DIRECTIVES, SPECS, AND STANDARDS

NARRATIVE

POLICY

GUIDELINES FOR ACCEPTANCE

THE BUYING OF SAFETY CRITICAL THREADED PRODUCTS NEEDED THE SAME EMPHASIS AS PARTS GENERALLY RECOGNIZED AS BEING SAFETY CRITICAL. YET IT WAS DESIRED THAT THE GOVERNMENT NOT HAVE TO INVEST MORE MANPOWER IN ACHIEVING THIS. THE POLICY WANTED THE SELLER TO DEVELOP THE PROOF OF COMPLIANCE WITH GOVERNMENT IN A REVIEWER ROLE. WHILE THIS IS SEEMINGLY WHAT WE WERE DOING WE WERE NOT VERIFYING INDEPENDENTLY THE SELLERS DATA. NOW WE MUST VERIFY BY WITNESSING OR CONDUCTING OUR OWN INDEPENDENT TESTS FOR COMPLIANCE. THE POLICY DID ALLOW FOR RELAXATION OF THIS STRINGENT INSPECTION REQUIREMENT ONCE A SUPPLIER HAD SHOWN HE COULD CONSISTENTLY MEET THE REQUIREMENTS FOR HIS PRODUCT.

POLICY

GUIDELINES FOR ACCEPTANCE

- FOR SAFETY CRITICAL PRODUCTS:
  - SELLER SHALL NDE/NDI EACH PART FOR IDENTIFIED CHARACTERISTICS AND PERFORM DESTRUCTIVE TESTS IDENTIFIED ON LOT SAMPLE BASIS
  - BUYER SHALL ASSURE THE SELLER HAS MET INSPECTION REQUIREMENTS:
    - WITNESS SELLER INSPECTIONS
    - CONDUCT RECEIVING INSPECTIONS
  - BUYER SHALL RERVIEW INSPECTION REQUIREMENTS AFTER HISTORY OF PERFORMANCE HAS BEEN ESTABLISHED

NARRATIVE

POLICY

GUIDELINES FOR ACCEPTANCE

THE KEY DIFFERENCE WITH OTHER THREADED PRODUCTS IS THE FREQUENCY OF INSPECTION TECHNIQUES. THE POLICY STILL REQUIRES THE SELLER TO SHOW PROOF OF COMPLIANCE FOR ALL PARTS BUT THE INDEPENDENT CHECKS CAN BE ON A SAMPLING BASIS.

POLICY

GUIDELINES FOR ACCEPTANCE

- FOR OTHER PRODUCTS:
  - SELLER PERFORM INSPECTIONS PER THE PRODUCT SPEC
  - BUYER SHALL ASSURE SELLER HAS COMPLIED WITH INSPECTION REQUIREMENTS AND SHALL CONDUCT SAMPLING INSPECTIONS AT ACCEPTANCE

NARRATIVE

POLICY

ACCEPTANCE INSPECTION METHODS

THE POLICY SPECIFIED HOW THE THREAD FORM IS TO BE VERIFIED FOR SAFETY CRITICAL AND OTHER APPLICATIONS. WHAT IS REALLY SOUGHT AS THE FINAL INSPECTION METHOD IS THE USE OF PROCESS CONTROL TECHNIQUES WHICH WILL ELIMINATE THE POST MANUFACTURING INSPECTIONS AND REDUCE THE COST OF MANUFACTURE. NOTE, HOWEVER, EVIDENCE OF DIMENSIONAL COMPLIANCE IS STILL REQUIRED.

POLICY

ACCEPTANCE INSPECTION METHODS

- FOR ACCEPTANCE OF SAFETY CRITICAL APPLICATION CLASS 3 EXTERNALLY AND INTERNALLY THREADED PRODUCTS USE MIL-S-8879A METHOD C OR FED-STD-H28/20 SYSTEM 23
- FOR ACCEPTANCE OF OTHER APPLICATION CLASS 3 EXTERNALLY AND INTERNALLY THREADED PRODUCTS USE MIL-S-8879A METHOD B OR FED-STD-H28/20 SYSTEM 22
- ALTERNATIVE METHODS MAY BE ADOPTED. PROCESS CONTROL SHOULD BE EMPLOYED. VERIFICATION OF COMPLIANCE WITH SPECIFIED DIMENSIONS AND TOLERANCES MUST BE OBTAINED

DISC (CONT'D)

TWO NEW ACTIVITIES HAVE BEEN STARTED WITH DISC AS THE CENTRAL GOVERNMENT PLAYER. THE ASME HAD BEEN UNABLE TO MAINTAIN FUNDING REQUIRED TO REGISTER HEAD MARKINGS USED BY FASTENER COMPANIES TO IDENTIFY THEIR PRODUCTS. THERE ALSO APPARENTLY HAD BEEN SOME DUPLICATION OF CHOICE OF MARKINGS. DISC AGREED TO TAKE OVER THE COMPILATION AND DISTRIBUTION OF A MANUAL OF HEAD MARKINGS. IN ADDITION TO IDENTIFYING THE ORIGIN OF FASTENERS, HAVING SUCH A REGISTRATION CAN BE USED IN ESTABLISHING CRIMINAL INTENT IN COUNTERFEIT CASES.

A NEWER EFFORT IS THE FORMATION OF A STUDY GROUP WITH INDUSTRY TO ESTABLISH A MEANS FOR SUPPLIERS OF FASTENERS TO BE CERTIFIED FOR MANUFACTURING OR SELLING OF THESE PRODUCTS TO INDUSTRY OR THE GOVERNMENT. THIS MAY SOUND LIKE A QPL APPROACH, BUT THE INTENT IS TO GO MUCH FURTHER IN OVERSIGHT AND PENALTIES FOR FAILURE TO DELIVER PRODUCTS TO SPEC.

NARRATIVE

RESULTS

LET US LOOK AT WHAT HAS HAPPENED AS THIS POLICY HAS BEEN IMPLEMENTED. SINCE THE DEFENSE INDUSTRIAL SUPPLY CENTER BUYS MOST OF THE GOVERNMENT PROCURED THREADED PARTS, MOSTLY FASTENERS, I WILL START WITH THEM. THEY HAVE PRETTY WELL DISCIPLINED THE SYSTEM AND HAVE TAKEN THE LEAD IN TRYING TO HEAD OFF THE INTRODUCTION OF COUNTERFEIT FASTENERS INTO THE AEROSPACE STOCKS. THE ACTIONS SHOWN HERE HAVE GOTTEN THE MESSAGE OUT TO SELLERS THAT THE GOVERNMENT IS CLEANING UP ITS ACT. FOR THOSE WHO DID NOT BELIEVE THEM, DISC HAS GOTTEN WITH THE DEPT OF JUSTICE TO PROSECUTE SEVERAL SUPPLIERS FOR FRAUD. MGEN PIGATY, THE COMMANDER AT DISC, HAS STRESSED COMMUNICATIONS WITH ALL PLAYERS AND HAS SPONSORED TWO MEETINGS FOR ALL THE GOVERNMENT AGENCIES INVOLVED AND A GOVERNMENT/INDUSTRY MEETING TO ALLOW BOTH SIDES TO COMMUNICATE THEIR POSITIONS AND POTENTIAL PROBLEMS.

RESULTS

- DEFENSE INDUSTRIAL SUPPLY CENTER (DISC)
  - REQUESTED ALL SERVICES IDENTIFY SAFETY CRITICAL SCREW THREADED PARTS BY NATIONAL STOCK NUMBER (NSN)
  - FROZE ALL STOCKS ON HAND OF NSN'S. CONDUCTED TESTS PER TECH DATA PACKAGES. NON-COMPLIANCES SENT TO CUSTOMERS FOR WAIVER DETERMINATION
  - IMPOSED RECEIVING INSPECTION REQUIREMENTS FOR NEW STOCK
  - REQUIRED DCAS OBTAIN SELLER CERTIFICATION DATA
  - INITIATED TRACKING OF SELLERS' PERFORMANCE
  - NOTIFIES SERVICE POC'S OF TEST RESULTS
  - HELD GOVT/IND CONFERENCE FOR EMPHASIS AND FEEDBACK

RESULTS

- DISC (CONT)
  - HAS TAKEN OVER CATALOGING AND DISTRIBUTION OF IDENTIFICATION MARKS USED BY FASTENER MANUFACTURERS
  - HAS TAKEN LEAD WITH INDUSTRY TO ESTABLISH CERTIFICATION REQUIREMENTS FOR SUPPLIERS OF FASTENERS

NARRATIVE

RESULTS

AFLC

AFLC'S MAJOR EFFORT HAS BEEN TO IDENTIFY THE SAFETY CRITICAL NSN'S FOR SYSTEMS THEY MANAGE. THIS EFFORT IS ESSENTIALLY COMPLETE. TO DATE ONLY 373 SAFETY CRITICAL NSN'S HAVE BEEN IDENTIFIED.

BECAUSE MANY NSN'S HAVE BEEN IDENTIFIED ON SEVERAL SYSTEMS, THE HQ HAS SET UP A DATA BASE TRACKING SYSTEM TO PERMIT ALL USERS OF THESE NSN'S TO BE NOTIFIED OF THE RESULTS OF INSPECTIONS ON STOCK IN DISC WAREHOUSES AND NEW BUYS. SINCE SOME NSN'S MAY HAVE NON-CONFORMANCES WAIVED BY ONE SYSTEM MANAGER IT IS NECESSARY FOR ALL TO BE AWARE OF NON-CONFORMANCES THAT MAY EXIST IN DISC STOCK.

RESULTS

- AFLC
  - HAS DIRECTED IDENTIFICATION OF SAFETY CRITICAL NSN'S FOR WEAPON SYSTEMS/SUPPORT EQUIPMENT
  - HAS ESTABLISHED A DATA BASE FOR CROSS TRACKING OF COMMON NSN'S AND INSPECTION RESULTS
  - HELD COMMAND CONFERENCE ON IMPLEMENTATION OF POLICY FOR SAFETY CRITICAL PRODUCTS

NARRATIVE

RESULTS

AFSC

AFSC HAS BEEN WORKING IN TWO MAJOR AREAS.

FIRST, IT HAS REQUIRED EACH PRODUCT DIVISION TO COMPLY WITH THE POLICY FOR ALL SYSTEMS IN DEVELOPMENT AND PRODUCTION. PROGRAMS ARE TO IMPLEMENT THE POLICY AT THE NEXT CONTRACTUAL OPPORTUNITY. ALL DRAWINGS THAT HAVE NOT BEEN RELEASED WILL INCORPORATE THE APPLICATION CATEGORY FOR THREADED PARTS. THIS EFFORT IS STILL IN PROCESS AS CONTRACTORS ASSESS THE COST AND BREAK-IN POINTS.

SECONDLY HAS BEEN TO REVISE THE SPECS AND STANDARDS TO IMPLEMENT THE POLICY. NEW SPECS FOR CLASS 3 THREAD FORM HAVE BEEN ISSUED FOR AIR FORCE ONLY USE IN PROCUREMENTS. THEY HAVE BEEN SUBMITTED FOR TRI SERVICE REVIEW AND WILL GO TO INDUSTRY FOR REVIEW EARLY IN 1989.

A REVISION TO MIL-STD-1515 HAS BEEN SUBMITTED FOR TRI SERVICE REVIEW. THIS REVISION WILL ELIMINATE THE LIMITATION ON USE OF ONLY CLASS 3 THREAD FORM IN DESIGN.

RESULTS

- AFSC
  - HELD COMMAND CONFERENCE ON IMPLEMENTATION OF POLICY
  - ISSUED DIRECTION TO PRODUCT DIVISIONS TO COMPLY AND TO SUBMIT IMPLEMENTRATION SCHEDULES
  - ISSUED MIL-S-007742C (USAF) AND MIL-S-008879B (USAF) 29 JULY 88 FOR ALL NEW USAF PROCUREMENTS. SUBMITTED FOR TRI-SERVICE COORDINATION.
  - SUBMITTED REVISION TO MIL-STD-1515 FOR TRI-SERVICE COORDINATION TO ELIMINATE RESTICTION ON SELECTION OF SCREW CLASS OF FIT
  - PRIME CONTRACTORS DETERMINING IMPLEMENTATION FOR SYSTEMS

NARRATIVE

FEEDBACK

THIS CHART SHOWS SOME OF THE RESULTS TO DATE. THE MOST SIGNIFICANT RESULT IS THAT THE SUPPLIERS ARE NO LONGER DUMPING THEIR NON-CONFORMING STOCK ON DISC. SECONDLY, ONLY 628 NSN'S HAVE BEEN IDENTIFIED AS SAFETY CRITICAL BY ALL THREE SERVICES AND NASA. COMPARED TO THE 93000 CLASS 3 NSN'S PROCURED THROUGH DISC, THIS IS A VERY SMALL NUMBER AND THE COST OF VERIFYING COMPLIANCE CAN BE TOLERANTED.

RESULTS

FEEDBACK

- NON-CONFORMANCES AT DISC HAVE DROPPED FROM 40 PERCENT OF BUYS TO 10 PERCENT
- SEVERAL FRAUD CASES HAVE BEEN INITIATED
- SUPPLIERS ARE TAKING THEIR NON-CONFORMING PRODUCTS ELSEWHERE
- THE NUMBER OF SAFETY CRITICAL NSN'S IS CURRENTLY AT 628 WITH ARMY AND AIR FORCE ESENTIALLY COMPLETE

NARRATIVE

SUMMARY

TO WRAP UP. THE POLICY IS RESULTING IN HIGHER QUALITY THREADED PRODUCTS COMING IN AS SPARES AND REPLACEMENTS. THE COST TO ACHIEVE SAFE SYSTEMS IS TOLERABLE. IMPLEMENTATION IS IN ITS FINAL STAGES. THE BIGGEST REMAINING CONCERN IS WHERE REJECTED PARTS ARE GOING. SO FAR, THESE STOCKS ARE NOT BEING DESTROYED. THIS MEANS THERE IS STILL A POTENTIAL FOR BAD PRODUCT TO SHOW UP AND THE NEED FOR INSPECTION CANNOT BE RELAXED.

SUMMARY

- POLICY HAS POSITIVE BENEFITS IN GOVERNMENT REPROCUREMENT OF SCREW THREADED PARTS
- NUMBER OF SAFETY CRITICAL SCREW THREADED PARTS IS VERY SMALL AND COST TO MAINTAIN SAFETY IS TOLERABLE
- COMPLETION OF IMPLEMENTATION SHOULD OCCUR BEFORE NEXT INTEGRITY MEETING
- WATCH OUT FOR SUPPLIERS TRYING TO UNLOAD NON-CONFORMING FASTENERS THEY USED TO SELL TO THE DISC

BOMBARDIER Inc.

CANADAIR, Military Aircraft Division

CF116 FULL SCALE DURABILITY AND DAMAGE TOLERANCE TEST

Eric E. Van Blaeren, Eng., M. Sc.

Manager, Aircraft Engineering

Systems Engineering & Maintenance

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SUMMARY

The Canadian Forces and Canadair Inc. are presently undertaking a full scale durability and damage tolerance test for the CF116 (CF-5) airplane. This paper describes the test planning phase and the test preparation. It is broken down in 5 sections:

1. The background section explains the reasons to undertake such a test. It also compares the durability and damage tolerance analyses and testings performed on similar types of airplane (T-38, F-5). It compares the present usage load spectrum with the spectra used in previous testings.

2. The scope section details the main characteristics of the tests:

   a. Full scale testing of a CF aircraft representative of the current fleet usage.

   b. Inspection of the test article to establish the condition of the test article.

   c. Durability testing to 16000 hours.

   d. Damage tolerance testing to 8000 hours subsequent to durability testing.

   e. The application of a load spectrum representative of current and future CF-5A/D mission profile (419 SQD).

   f. The provision of marker blocks in the load spectrum for fractographic analysis.

   g. The strain gauging of the test article prior to testing to provide a baseline load data.

   h. Teardown of the test article subsequent to testing for a final condition inspection.

3. The purposes section shows the goals of the test:

   a. To ensure a high level of confidence in the structural integrity of the CF-5A/D to operate to 6000 A/F hours. The test will also be used to evaluate the possibility of extending the life of the aircraft beyond 6000 hours, up to 8000 A/F hours.

   b. To ensure that a highly efficient support program is provided for the service life of the airplane.

4. The test plan section describes the test rig facility. The load spectrum derivation, the critical location policy and the inspection policy.

5. The schedule section shows the time frame of the program.

1.0 INTRODUCTION

This paper discusses the requirements of a full scale fatigue test on the CF116 (CF-5) aircraft. It also presents the work done during the planning phase of the test.

The paper is divided in 5 Sections:

- Background
- Goals
- Scope
- Test Plan
- Schedule

1.1 ROLE OF FULL SCALE TESTING IN THE ASIP PROCESS

An Aircraft Structural Integrity Program (ASIP) has been implemented on the CF116 fleet. The general requirements of the ASIP are to:

- Establish, evaluate, and substantiate the structural integrity (airframe strength, rigidity, damage tolerance and durability) of the aircraft.

- Acquire and evaluate operational usage data to provide a continual assessment of the in-service integrity of individual airplanes.

- Provide a basis for determining logistics and fleet planning requirements such as maintenance, inspections, modifications, supplies, phaseout and future roles.

Considerable analytical and usage data have now been collected. The proper method of substantiating this data used in the ASIP process is by full-scale testing.

A full-scale fatigue test would establish and evaluate the structural integrity as described above. The results of such a test would aid in evaluating the in-service integrity of individual airplanes. And most importantly in light of cost and safety, a full-scale test would validate maintenance, inspections, spares, phaseout, and future requirements.

Other methods have been used in the ASIP process such as fatigue and damage tolerance analysis. The results are not entirely consistent and also the methods include a number of conservatisms designed for safety. These theoretical analyses should be substantiated by representative tests.

1.2 F-5/CF-5 MILESTONES

In this section are the relevant dates and milestones of the F-5 family of aircraft. The F-5 family includes the following aircraft:

T-38 First aircraft in the series. Two seat trainer, no guns or wing pylons.

F-5A: Single seat armed version of T-38.

F-5B: Two seat, armed trainer.

F-5E: Updated version of F-5A.

F-5F: Updated version of F-5B.

F-5G or F-20: Latest version of F-5A; single engine.

The CF-5A/D are versions of the F-5A/B. Some of the differences are shown in a subsequent section.

1.2 F-5/CF-5 MILESTONES (CONT'D)

DATE	MILESTONE
1959	T-38 first flight
1960-63	T-38 fatigue test - tested to 16,000 simulated flight hours
1964-66	F-5A fatigue test - tested to 16,000 simulated flight hours
1967-74	CF-5 production at Canadair
1970	F-5A crash due to wing critical radius failure
1973-77	F-5E fatigue test - tested to 24,000 simulated flight hours. Vertical stabilizer failure led to redesign
1975	T-38 found with 1.9" crack in wing critical radius
1980	T-38 found with 1.8" crack in wing critical radius
1980	CF-5 ASIP Master Plan first issue
1980-82	CF-5 Wing coupon testing in the Netherlands
1986	CF-5 Wing Components Redesign
1987	CF-5 Dorsal Longeron Redesign
1988	CF-5 Aft former redesign

2.0 BACKGROUND

2.1 ANALYTICAL LIFE ESTIMATES

The CF5 was designed using the safe life approach. This is a pre-1970 approach where the mean life prediction was based on:

- Unflawed lab specimen data
- Miner's cumulative damage analysis
- A constant scatter factor of 4
- No crack growth analyses requirement

The life estimate was 4000 flying hours.

In July 1977, a service life extension program (SLEP) was initiated. It was based on:

- A new approach: The damage tolerance assessment
- Selection of typical, mild and severe load spectra
- A NASTRAN finite element model of the F-5A/B
- Inspection interval determination

The life estimate obtained during the SLEP was 6000 flying hours providing that a set of structural inspections be performed.

2.2 FATIGUE TESTS

This section describes the full-scale fatigue testing done on aircraft similar to the CF116. The T-38 and F-5A fatigue tests were performed on components such as the wing, forward fuselage, horizontal stabilizer, etc. The F-5E test is the only one carried out on a complete airframe.

The only full-scale fatigue testing on the CF-5 was for the two position nose landing gear on the CF-5A.

T-38 Durability Test (1960-1963)

The T-38 durability tests included full-scale testing on the following components:

Wing and centre fuselage

Control surfaces

Main landing gear

Nose landing gear and forward fuselage

The test was to demonstrate a satisfactory service life of 7,000 hours and 15,000 landings. In order to accomplish this, the target test life was 60,000 simulated flight hours and 75,000 landings. After the completion of 60,000 simulated flight hours and 64,300 landings, the full-scale wing testing was terminated for the following reasons:

1. Fatigue cracks detected at 60,000 hours in the lower wing skin critical radius. (T-38 skin thickness is .42" compared to .52" for the CF-5).

2. A review of T-38 usage indicated that the applied ratio of landings per flight hour (64,300:60,000) was more realistic than the original requirement (75,000:60,000).

For the wing test, all manoeuvre conditions were cycled from a 1 g load level. The gust conditions were cycled from a zero load level. The effect of the omission of negative load cycles is discussed in a subsequent section.

F-5A Fatigue Test (1964-69)

The F-5A wing and centre fuselage were tested to 16,000 simulated flight hours. No major structural failures were reported. As discussed later, the spectrum for the F-5A fatigue test is milder than that of the CF-5A/D operational spectrum.

F-5E Fatigue Test (1973-1975)

The first comprehensive full-scale fatigue test was for the F-5E. A total of 24,000 simulated flight hours were applied to a complete airframe. Although the wing critical radius had been redesigned for the F-5E, the vertical stabilizer critical radius and attachment angles were similar to the CF-5. The cockpit longeron is also similar to the CF-5A. The results, are given in Table 1 below.

TABLE 1 F-5E FATIGUE TEST RESULTS

ITEM	REMARKS	TEST HOURS
Vertical stabilizer forward attachment angle	Left and right cracked section replaced Left and right cracked	1,000 1,417 1,495
Vertical stabilizer attachment angle at 46.5% spar	Left (stop drilled)	1,417
Rear fuselage frames (near v/stab)	Left and right cracked	2,334
Vertical Stabilizer Skins (critical radius)	Left and Right Cracked	2,528
Cockpit Longeron	Left side at F.S. 194 and 257 cracked	8,000

The results of the F-5E fatigue test prompted a flight load survey of the CF-5 vertical stabilizer as well as special inspections. It should be noted that the F-5E full scale fatigue test led to a redesign of the vertical stabilizer side skins and attachment angles.

2.3 STRUCTURAL DIFFERENCES AND SPECTRA COMPARISON

The Canadian Forces CF-5 has structural and spectral differences compared to the aircraft used in the full-scale fatigue tests described in previous sections. These differences are described in this section.

2.3.1 Structural Differences:

The basic dimensions and general arrangements of the relevant F-5 aircraft are given in Figure 1.

A list of the main structural differences between F-5 type aircraft can be found in Table 2.

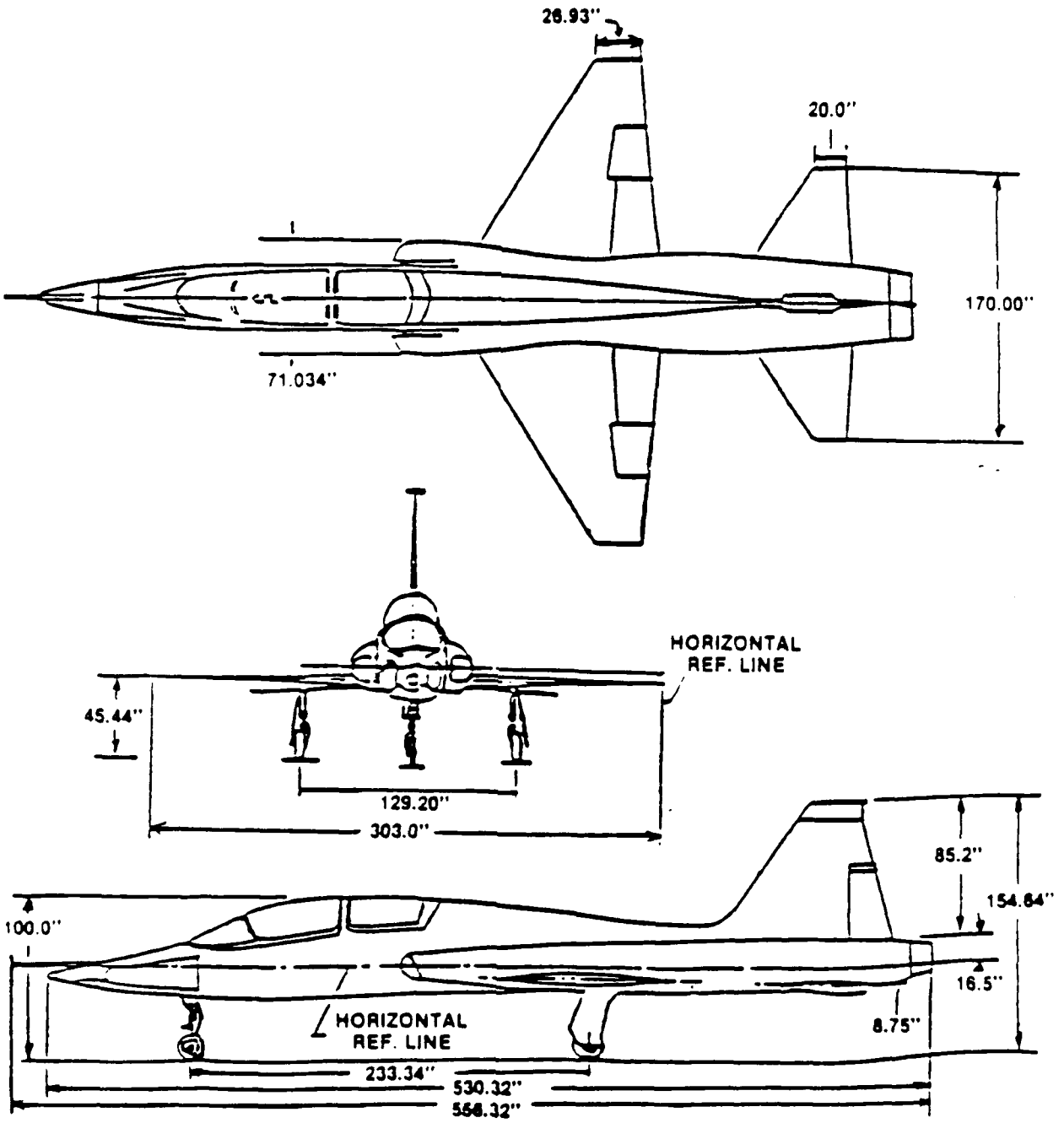


FIGURE 1 a) - T-38 GENERAL ARRANGEMENT

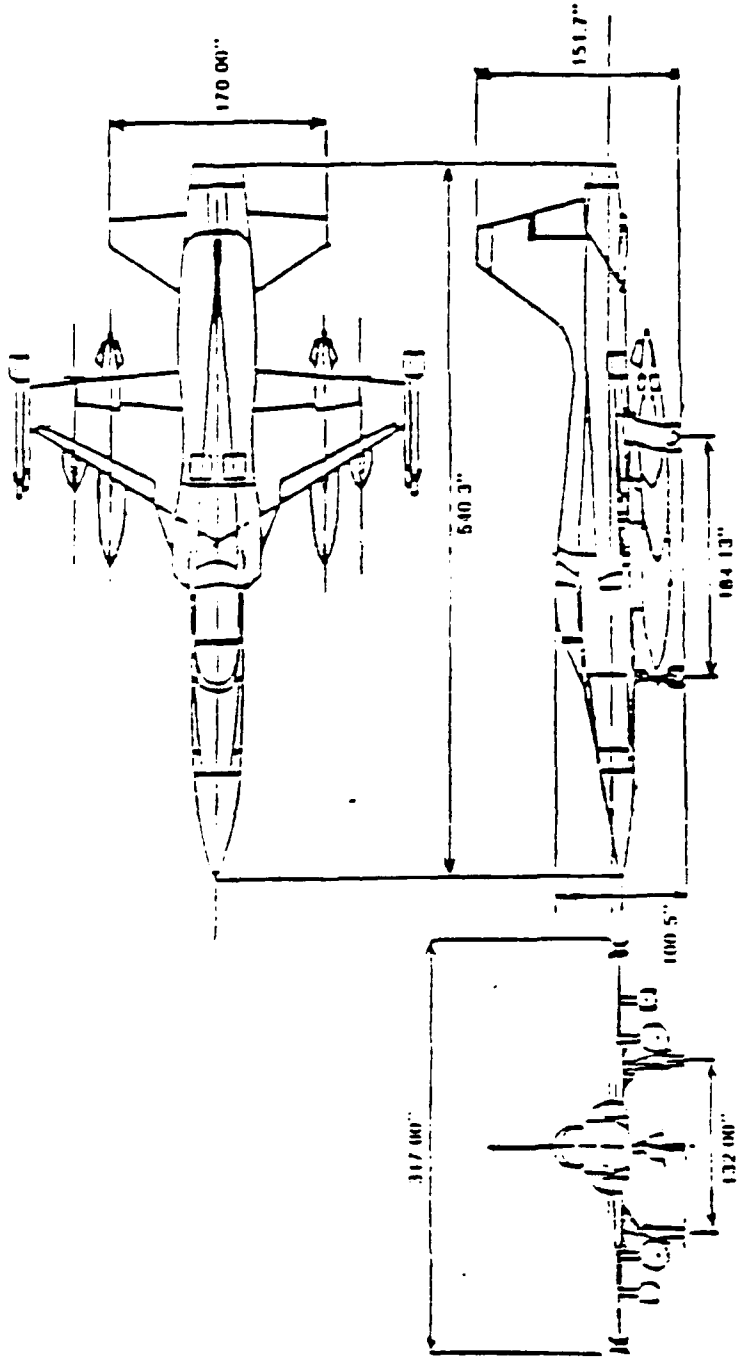


FIGURE 1 b) - F-5A GENERAL ARRANGEMENT

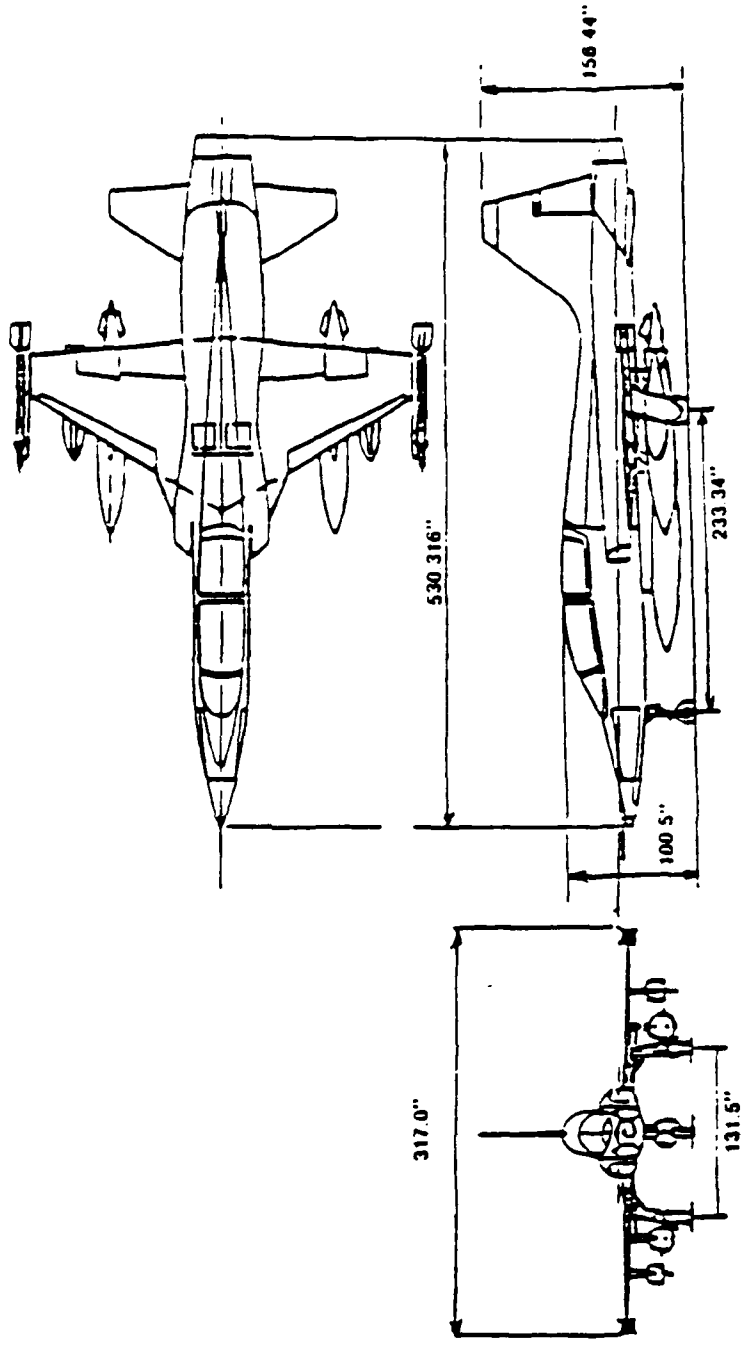


FIGURE 1 c) - F-5B GENERAL ARRANGEMENT

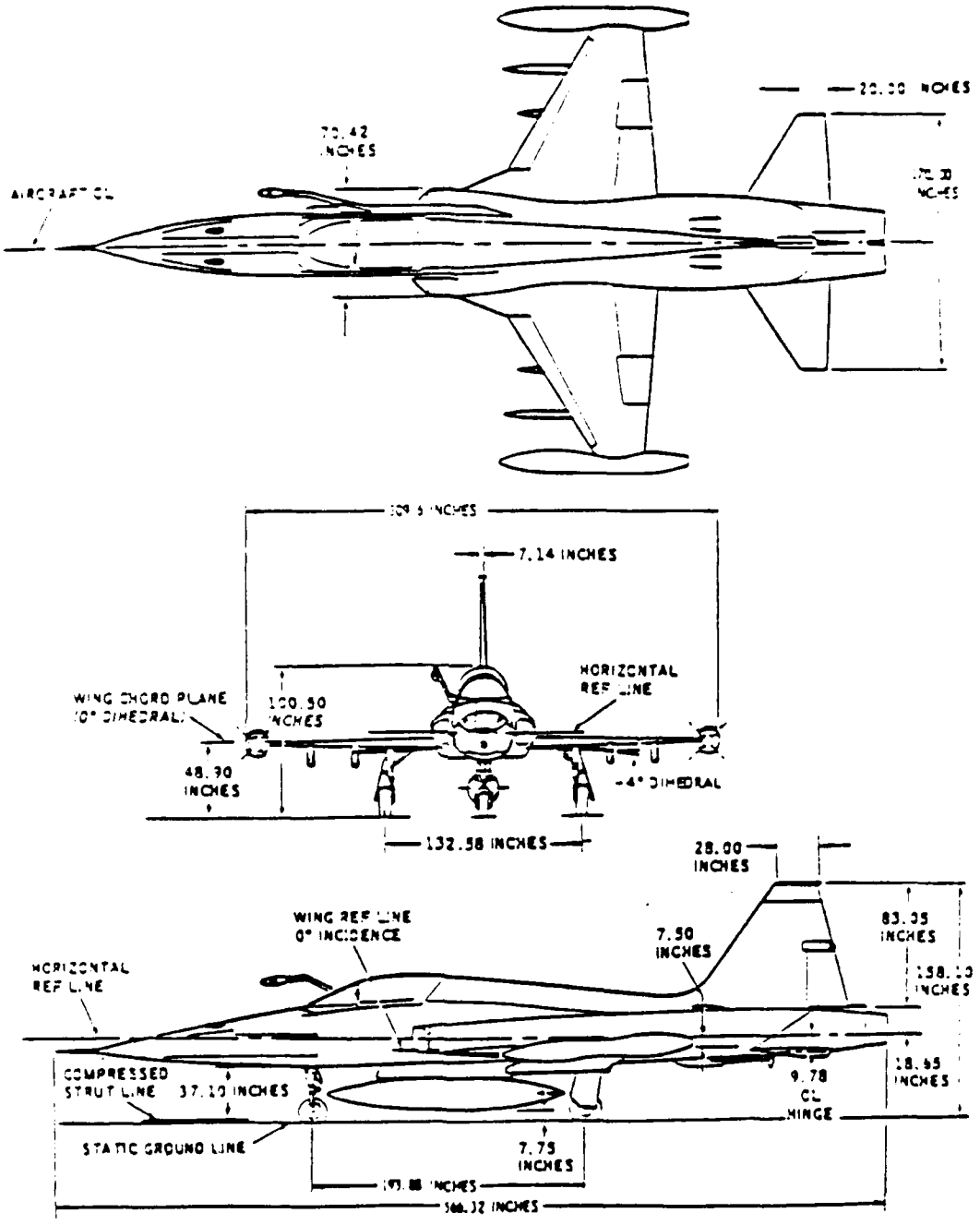


FIGURE 1 d) - CF-5A GENERAL ARRANGEMENT

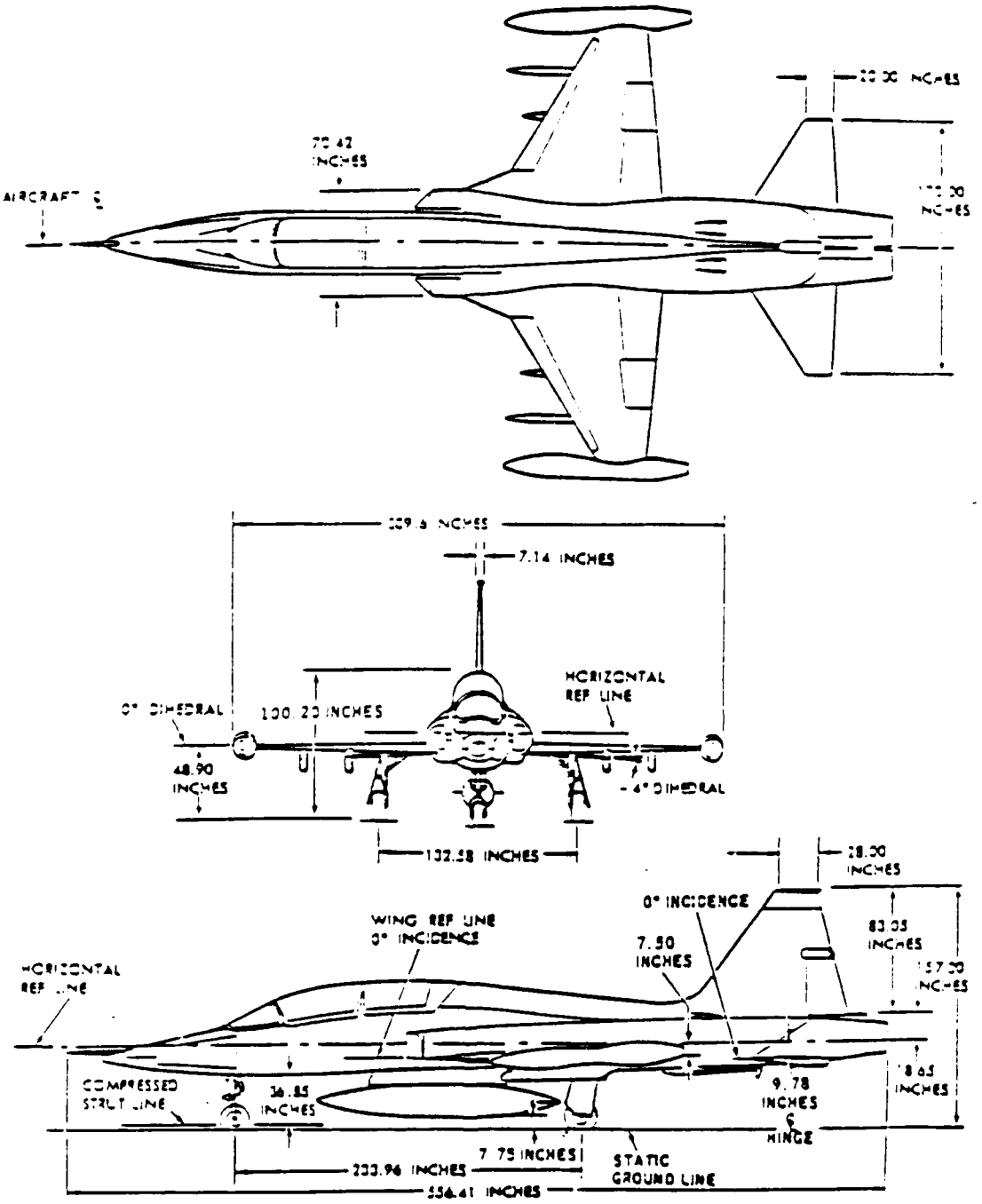


FIGURE 1 e) - CF-5D GENERAL ARRANGEMENT

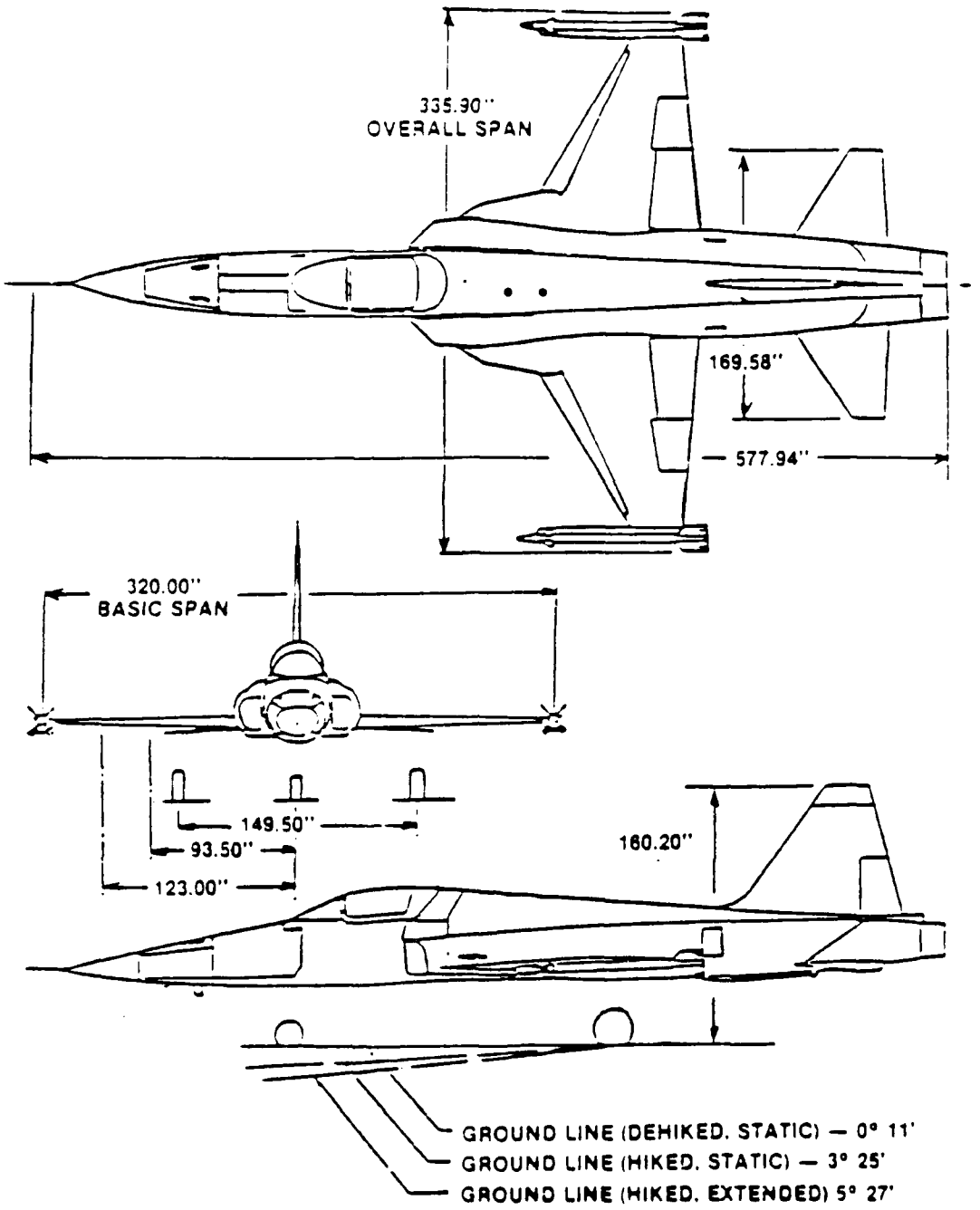


FIGURE 1 f) - F-5E GENERAL ARRANGEMENT

TABLE 2 STRUCTURAL COMPARISONS OF F-5 TYPE AIRCRAFT

Criteria	AIRCRAFT TYPE					
	T-38	F-5A	F-5B	CF-5A	CF-5B	F-5E
Lower Wing Skin Critical Radius Thickness	-	.42"	.42"	.52"	.52"	Redesigned
V/Stab Configuration	similar	similar	similar	similar	similar	Similar until redesigned due to fatigue test failure
Leading Edge Extensions	No	Yes	Yes	Yes	Yes	Yes; larger
Two Position Nose Gear	No	No	No	Yes	No	Yes
Leading Edge Flap	No	Yes	Yes	Yes	yes	Yes; Manoeuvre flap
Wing Span	303"	303"	303"	303"	303"	320"
Overall Length	556.5"	566.3"	556.4"	560.3"	560.1"	573"
Centre and Aft Fuselage	similar	similar	similar	similar Except aft Form. redesign	similar	15" longer 16" wider for new engines
Dorsal Longeron	similar	similar	similar	redesigned	redesigned	Dorsal area redesigned for new engines
Cockpit Longeron FS's 194 and 257	-	similar	-	similar	-	Similar until redesign due to fatigue test

2.3.1 Structural Differences (Cont'd)

Most of the problem areas are similar in the T-38, F-5A/B and CF-5 A/D. However, the F-5E structure is completely different from the CF-5 A/D in four key areas. These are:

a. Wing Critical Radius - This area of the F-5E lower wing skin has been fully redesigned. Figure 2 shows the CF-5 wing skin critical radius and Figure 3 shows the F-5E wing skin critical radius.

b. Vertical Stabilizer - During the F-5E full scale fatigue test the covering skin and attachment angles failed. These were similar to CF-5 configuration. The failures led to a redesigned configuration.

c. Dorsal Longeron - This area was completely redesigned to make room for larger engines (Figure 4).

d. Cockpit Longeron - Redesigned after fatigue testing produced cracks. The F-5E is also a heavier aircraft than the CF-5. This is due to:

- 17" wider wing span
- larger leading edge extensions
- extended fuselage allows 570 lbs. extra fuel
- wider fuselage to accommodate engines
- bigger engines

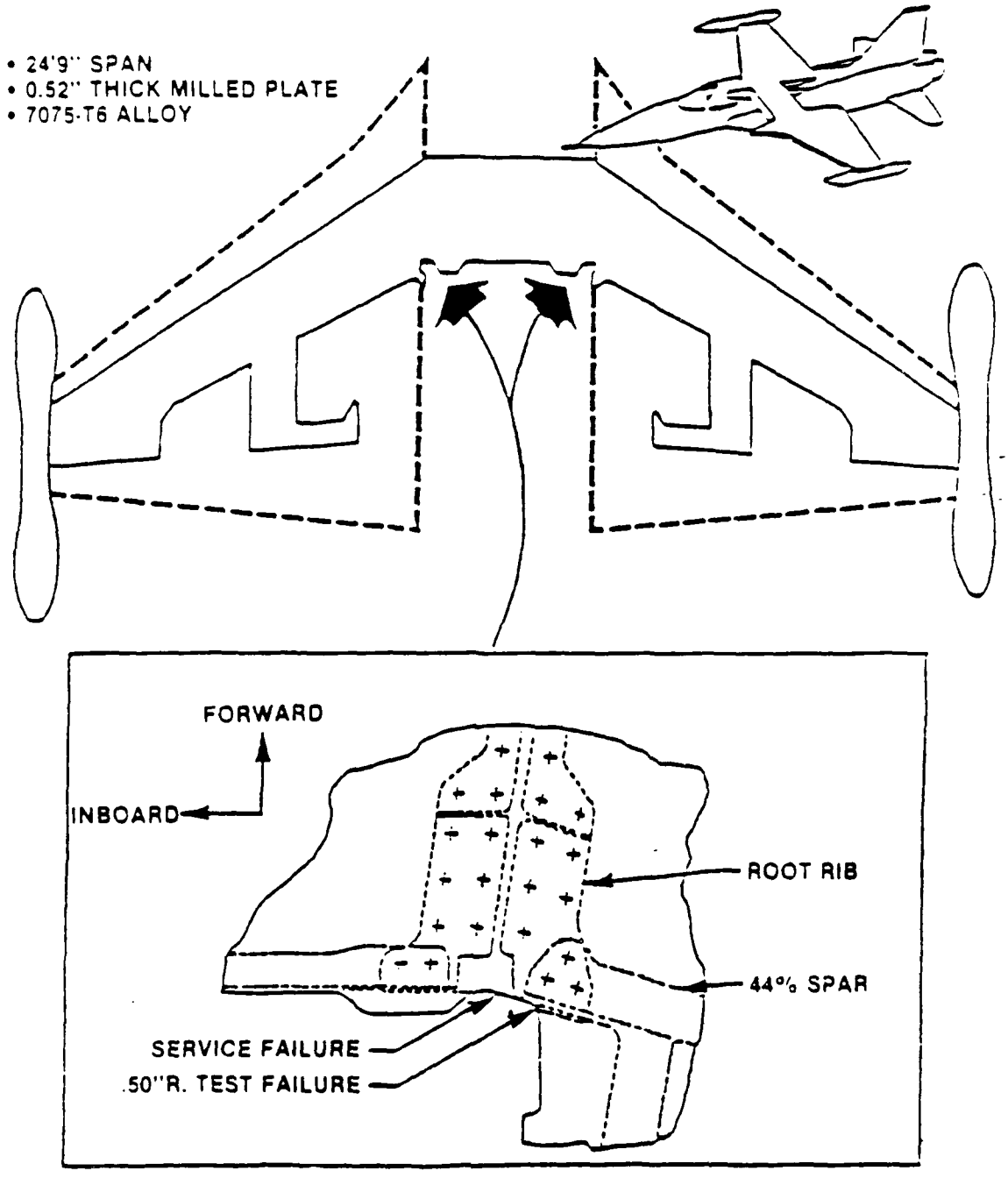


FIGURE 2 - CF-5 LOWER WING SKIN CRITICAL RADIUS

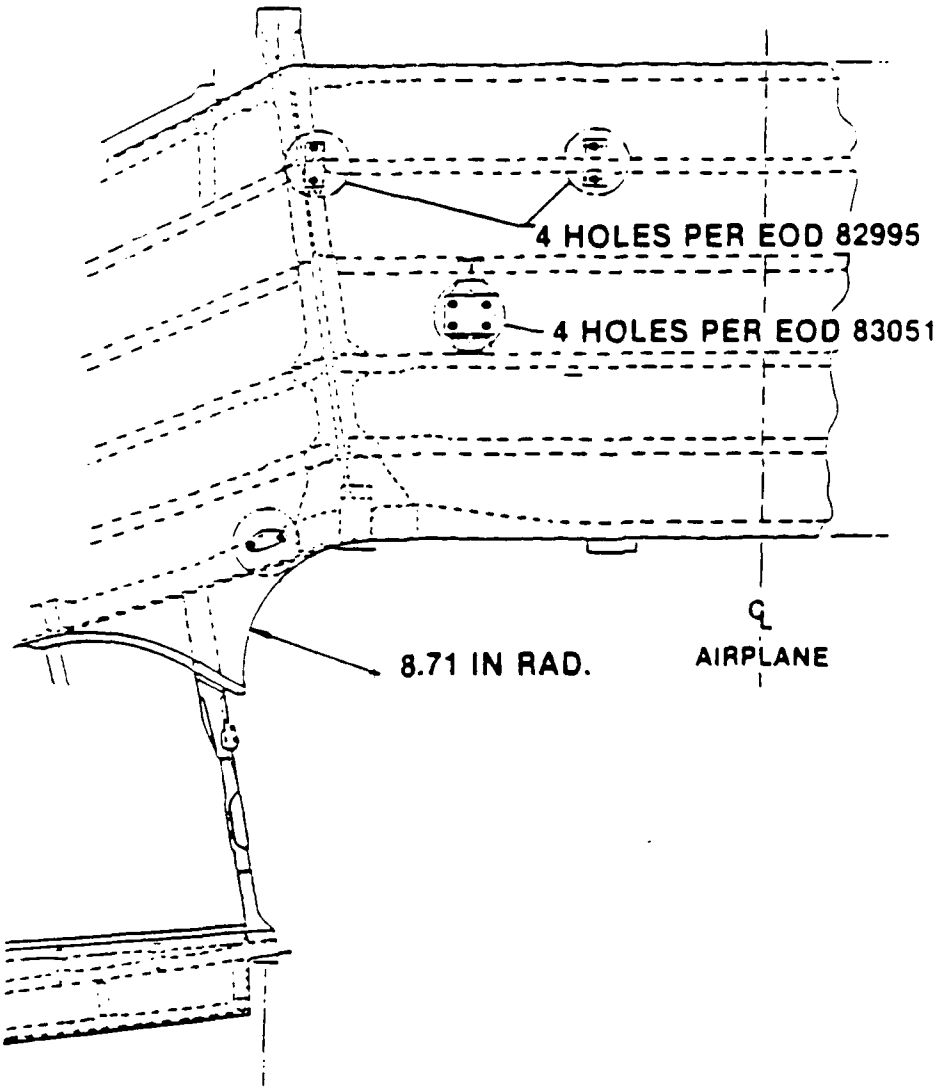


FIGURE 3 - F-5E LOWER WING SKIN CRITICAL RADIUS

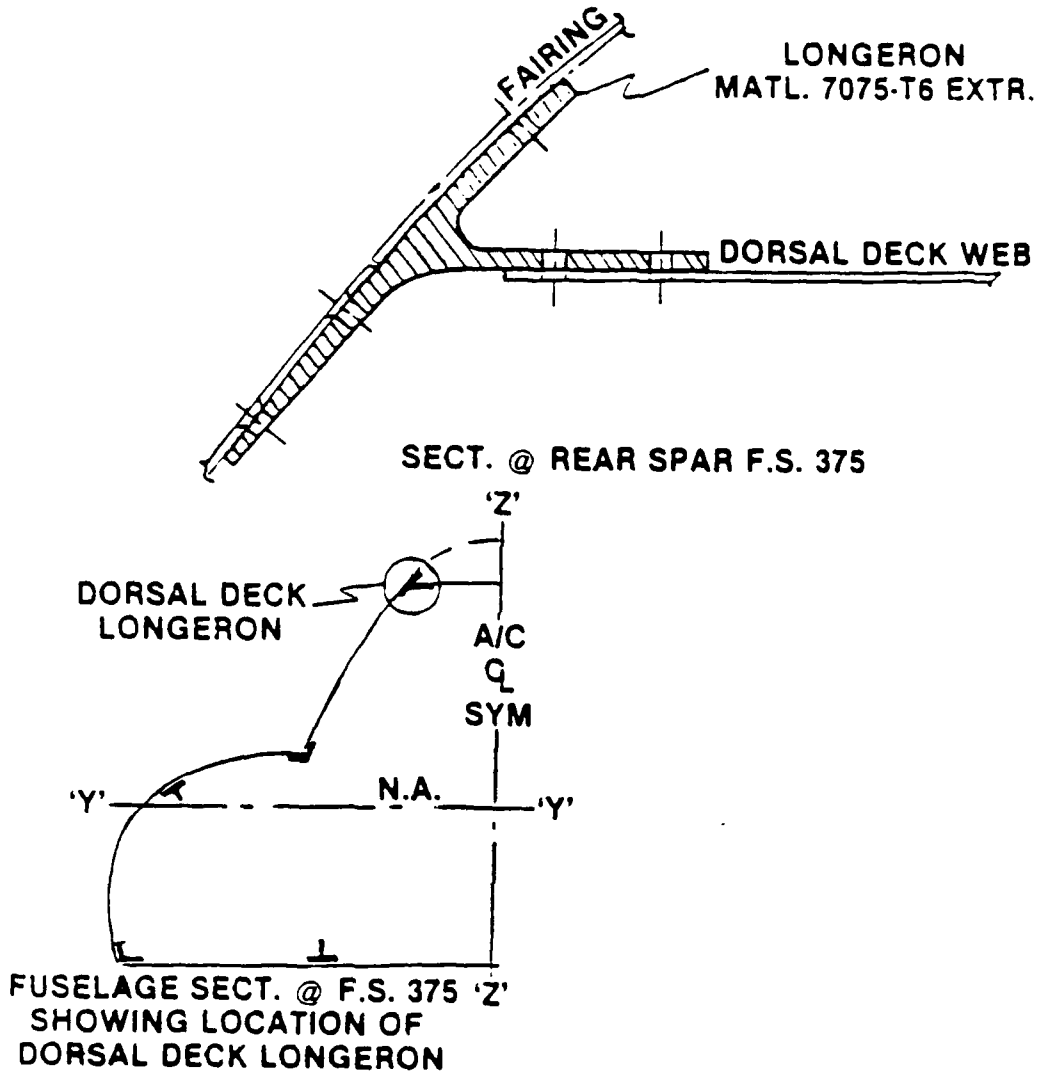


FIGURE 4 - F-5E DORSAL LONGERON

2.3.2 Spectra Comparison:

The current CF-5A vertical accelerometer spectrum is compared with the F-5A fatigue test spectrum in Fig. 5.

The spectra used in the T-38 and F-5A fatigue tests were too mild compared to actual CF-5 usage spectra.

It can be noted that:

- The CF-5 operational spectrum is more severe than the one used for the F-5A test
- The F-5A test did not include a full manoeuvre spectrum, ie., the portion of the spectrum under 1 'g' was not included (thus no overswing was accounted for)

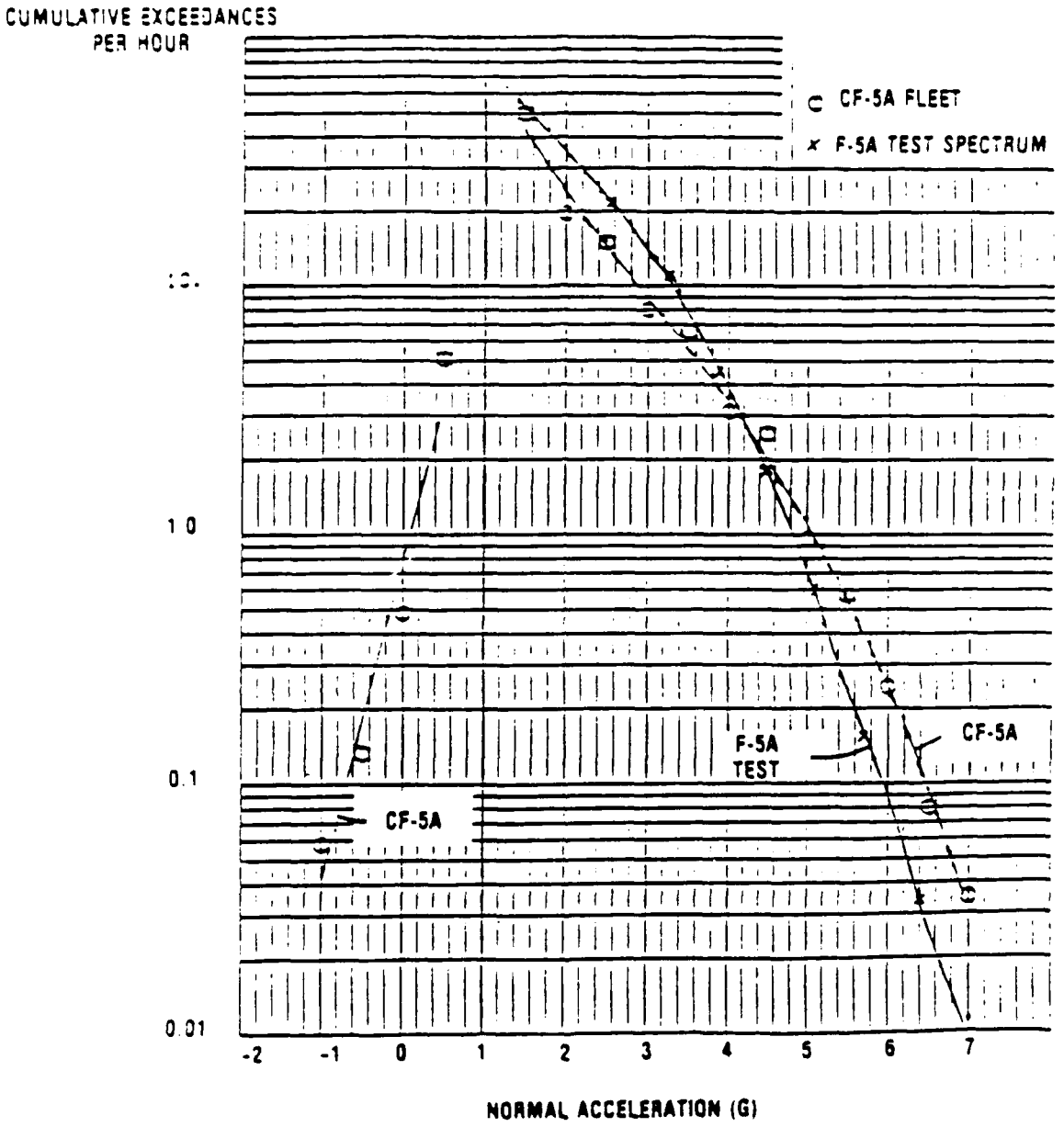


FIGURE 5 - F-5A AND CF-5A SPECTRA

2.3.2 Spectra Comparison (Cont'd)

The difference between calculated fatigue lives for the F-5A and CF-5A spectra of Fig. 5 will be shown in the following analysis. The CF-5 wing critical radius will be used. The S-N curve used is shown on Figure 6.

Note: the F-5A test spectrum does not include the overswing assumed for the CF-5. Therefore the minimum stress for the F-5A spectrum is the stress at 1 'g'.

Stress/g = 4377 psi/g

$$\text{Maximum Stress} = \frac{n_i + n_{i+1}}{2} \times 4377$$

$$\text{Minimum Stress} = \left(1 - \left[\frac{n_i + n_{i+1}}{2}\right] - 1 \times 20\%\right) \times 4377$$

- for CF-5A

Minimum Stress = 4377 psi

- for F-5A test spectrum

The calculation of the damage sum is shown in Tables 3 and 4. Note that the calculations are shown for comparison purposes only.

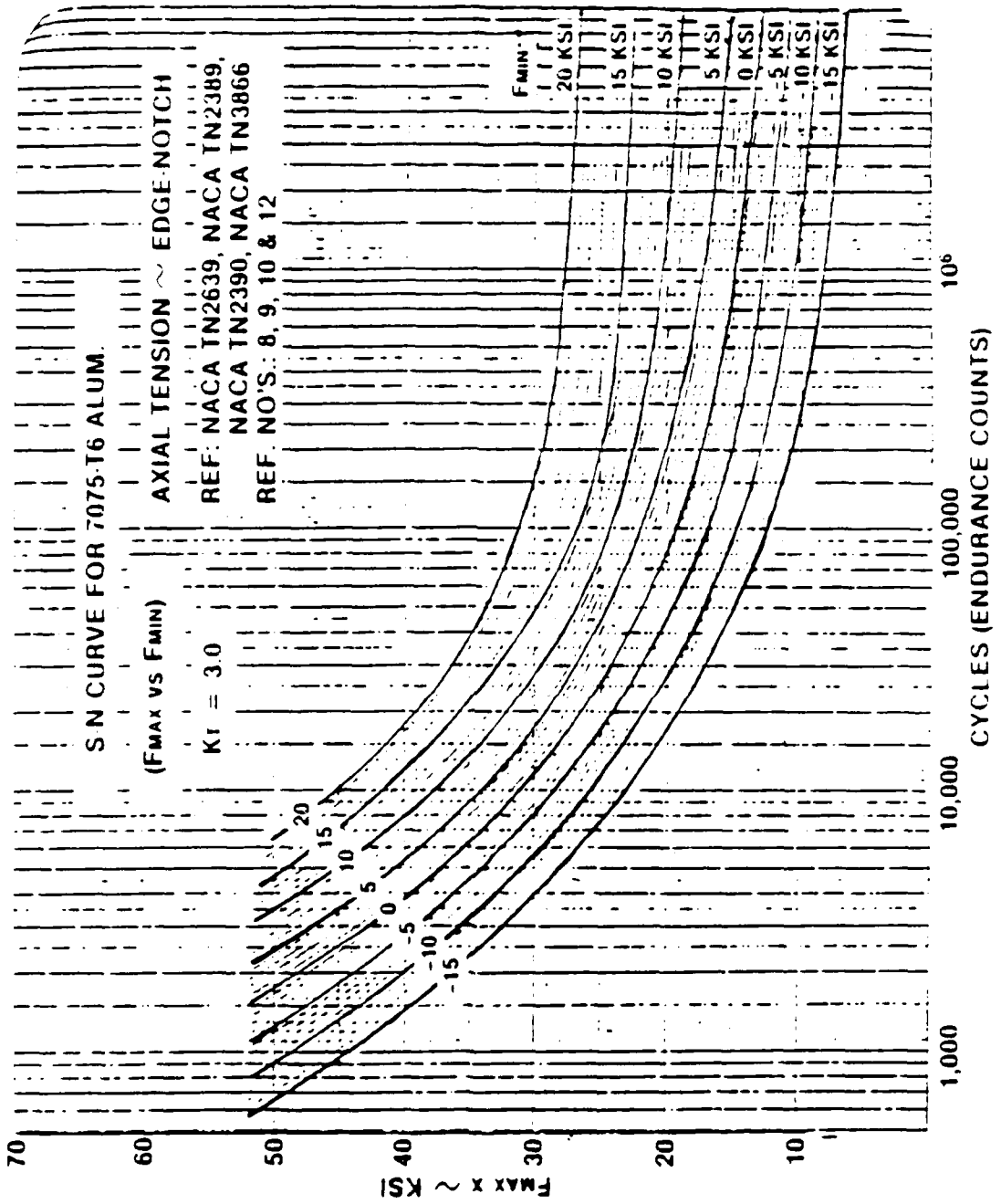


FIGURE 6 - S-N CURVE

TABLE 3 - CALCULATION OF CF-5A DAMAGE SUM FOR WING CRITICAL RADIUS

'g' level	Occurrences per 4000 hours (n)	Maximum Stress (psi)	Minimum Stress (psi)	N Cycles to Failure	Damage n/N
2.5					
	27745	12037	2845	infinite	0
3.0					
	5904	14225	2407	7×10^6	.0008
3.5					
	13683	16414	1970	1.2×10^6	.0114
4.0					
	2260	18602	1532	2.5×10^5	.0090
4.5					
	6294	20791	1094	85000	.0740
5.0					
	1585	22979	657	42000	.0377
5.5					
	1315	25168	219	27000	.0487
6.0					
	651	27356	-219	17000	.0383
6.5					
	182	29545	-657	10200	.0178
7.0					
	92	31733	-1094	9000	.0102
7.5					
	32	33922	-1532	6800	.0047
8.0					
					= .2526

TABLE 4 - CALCULATION OF F-5A TEST SPECTRUM DAMAGE SUM FOR WING CRITICAL RADIUS

'g' level	Occurrences per 4000 hours (in)	Maximum Stress (psi)	Minimum Stress (psi)	N Cycles to Failure	Damage n/N
2.5					
	34000	12037	4377	infinite	0.0
3.0					
	21200	14225	4377	infinite	0.0
3.5					
	15600	16414	4377	2×10^6	.0078
4.0					
	8400	18602	4377	320000	.0263
4.5					
	4000	20791	4377	140000	.0286
5.0					
	1880	22979	4377	70000	.0269
5.5					
	568	25168	4377	41000	.0139
6.0					
	260	27356	4377	26000	.0100
6.5					
	48	29545	4377	18000	.0027
7.0					
	40	31733	4377	13000	.0031
7.5					
8.0					
					= .1196

2.3.2 Spectra Comparison (Cont'd)

From Tables 3 and 4 the damage sums are:

a. CF-5A operational spectrum = .2526

b. F-5A fatigue test spectrum = .1196

Therefore, according to the above analysis, the F-5A test spectrum is 2.1 times milder than the CF-5A spectrum.

2.4 VERTICAL STABILIZER

The F-5E fatigue test spectrum is compared to the CF-5 operational spectrum in Fig 7. The F-5E spectrum is more severe, however, it should be noted that cracks have been found in relatively low-life CF-5's.

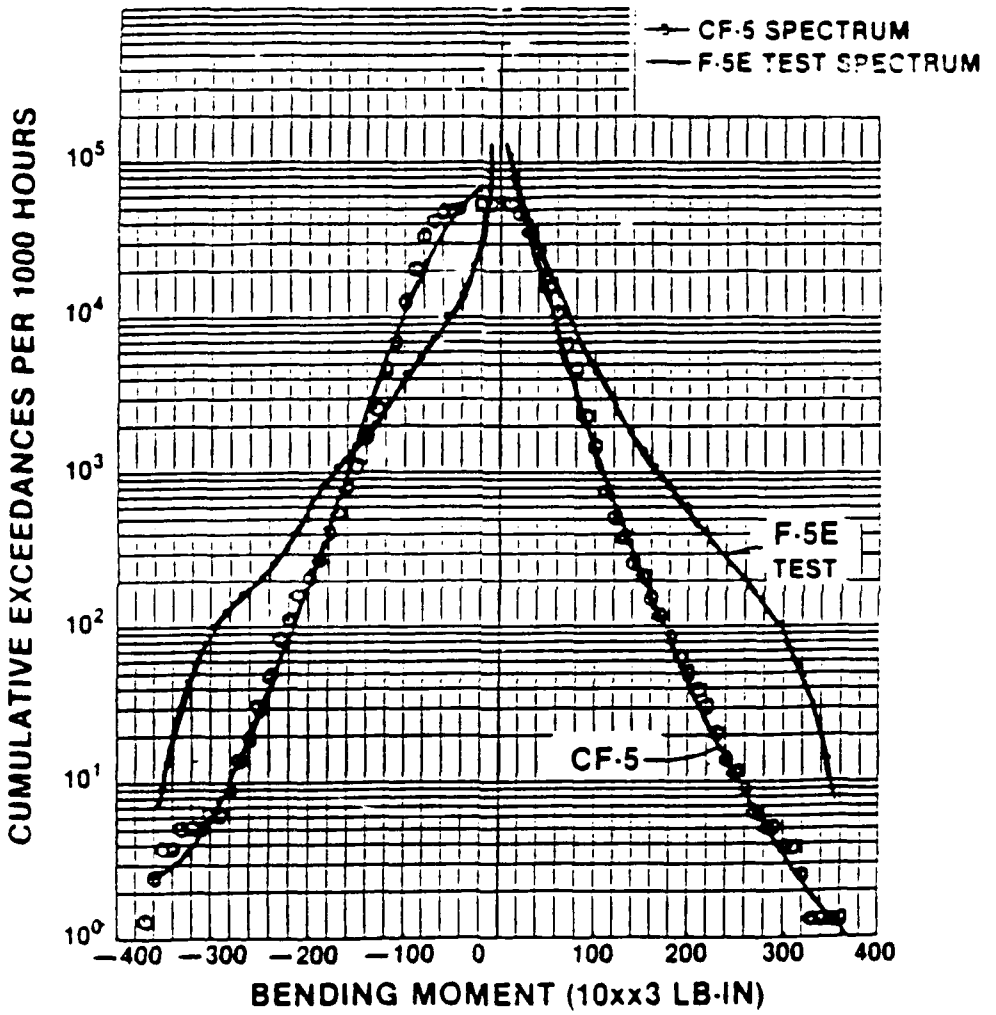


FIGURE 7 - VERTICAL STABILIZER BENDING MOMENT SPECTRA (CF-5 & F-5E)

2.5 CONCLUSIONS

From the previous discussions it can be concluded that:

1. A full-scale fatigue test would play an important role in the ASIP process. It would validate ASIP tasks such as inspection frequencies, spares requirements, phaseout, etc.;

2. The F-5A and T-38 full-scale test spectra were milder than CF-5 measured spectra;

3. There has been no test on a complete thick (.52 in.) wing skin. The other tests on wing skins with critical .5 in. radii similar to the CF-5 have been on thin (.42 in.) ones;

4. The F-5E test was the only test on a complete airframe. The F-5E has significant structural differences compared to the CF-5;

5. A fatigue test would substantiate results of fatigue and damage tolerance analyses;

6. The F-5E vertical stabilizer test spectrum was more severe than for the CF-5. However, CF-5 vertical stabilizer cracks have been found.

3.0 GOALS

The test will be performed for the following purposes:

Increase confidence in the CF116 structural integrity

Optimize the aircraft technical support program by:

a. Substantiating theoretical analyses (DDTA and NASTRAN FEM)

b. Identifying problem areas

c. Validating inspection techniques

d. Establishing spare part requirements

e. Evaluating modifications and repair schemes. For instance, there has been three structural areas of the CF-5 which have been redesigned:

1) Dorsal Longeron: The Canadair redesign consists of removing the upper leg of the existing longeron and installing an external "L" section steel longeron. The longeron is manufactured from 4340 steel heat treated to 160-180 Ksi. Figure 8 locates and shows the redesigned dorsal longeron.

2) Wing: Several wing structural components have been redesigned by using material with improved physical properties (e.g. static strength, durability and damage tolerance characteristics, stress corrosion resistance). As well, design details are considered to prevent fatigue and stress corrosion problems. A summary of the changes is presented in Table 5 and Figure 9.

3) Aft Formers: The modification consists of removing and replacing, by a machined aluminium section, the upper portion of formers between left and right upper longerons and the centre portion above the horizontal reference plane. Also, integrated bathtub type fittings are incorporated on forward and aft faces of redesign former sections. This will allow an improvement of the load path for transmission of vertical stabilizer loads into the fuselage formers. Figure 10 shows the redesigned formers location.

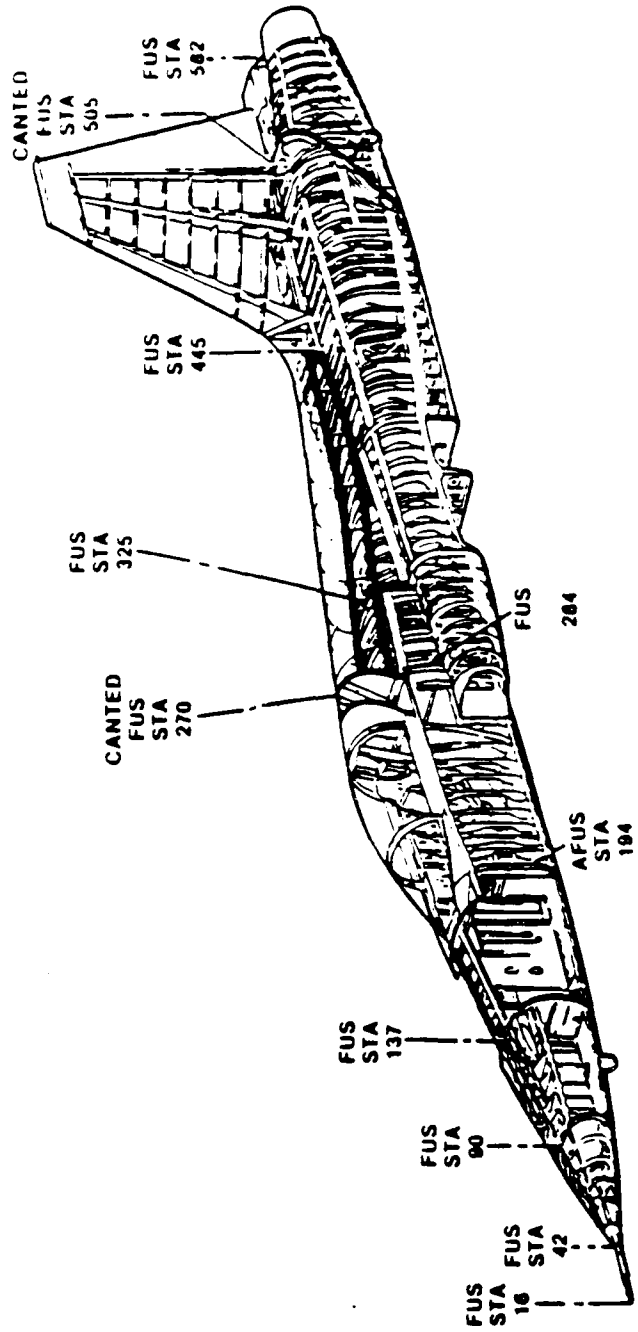


FIGURE 8 a) - DORSAL LONGERON MODIFICATION

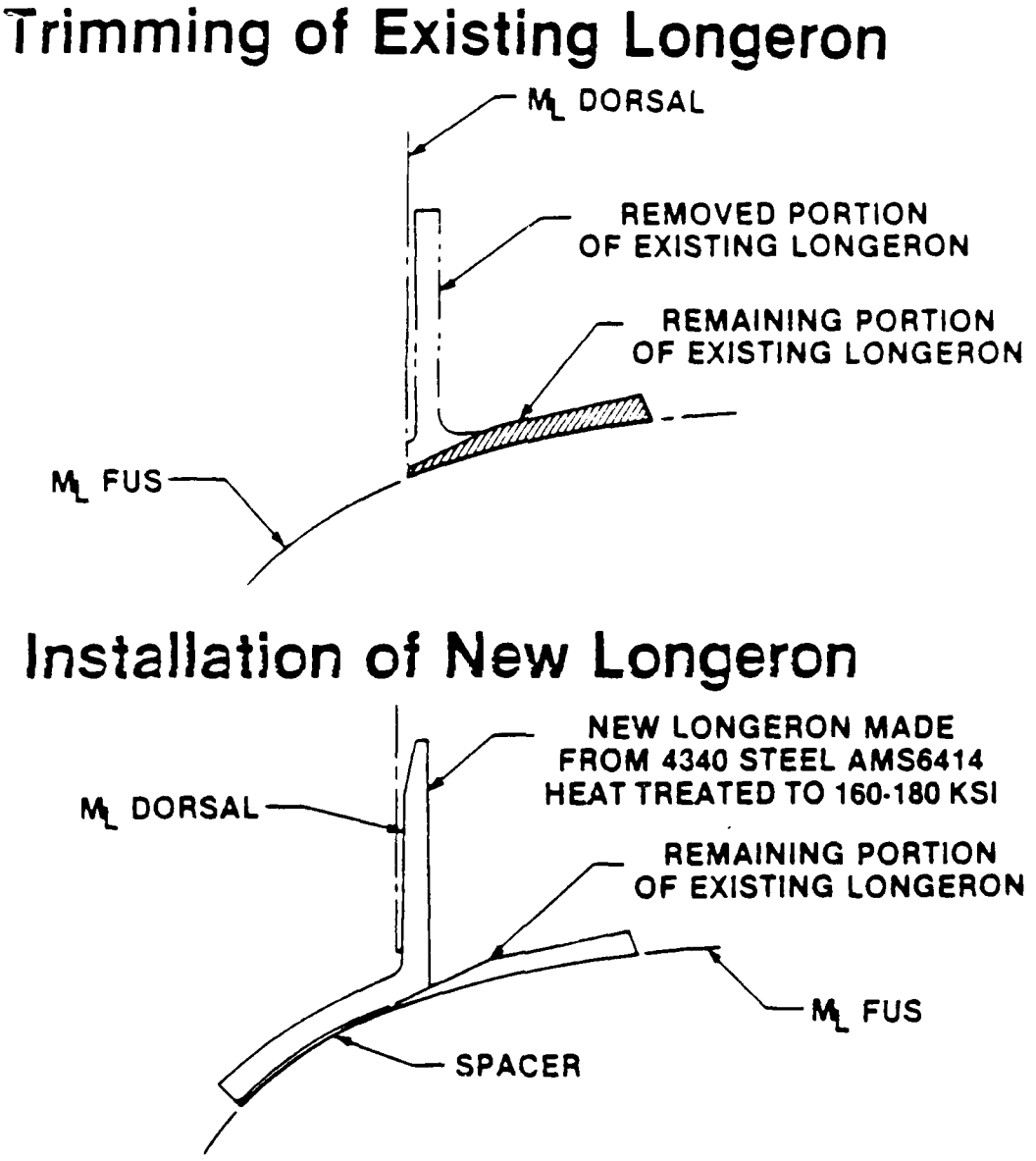


FIGURE 8 b) - DORSAL LONGERON MODIFICATION

TABLE 5 - WING COMPONENT CHANGES

COMPONENT	PART No.	DESCRIPTION OF CHANGES	REDESIGN PART No.
15% Spar-Inboard	6-23441	Grain direction change	219-20005
15% Spar-Outboard	6-23442	Redesign + material change	219-20006
44% Spar	6-23252	Redesign + material change	219-20002
66% Spar	6-23459	Redesign + material change	219-20004
Aileron Actuator Support Rib	8-23410	Material change	--
Main Landing Gear Uplock Rib	6-23225	Redesign + material change	219-20003
Root Rib	8-23404	Cold working + add. of anti-roll pin + redesign + material change	219-20001
Centre Section Spar:			
FS 330.021	8-23421	Cutting + splicing + enlarging fuel line holes	--
FS 336.408	8-23423	Cutting + splicing + material change	--
FS 342.867	8-23425	Cutting + splicing	--
FS 349.400	8-23427	Cutting + splicing	--
FS 361.572	2-23239	Cold Working of holes	--
Lower Skin	8-23444	Material change + shot peening in critical radius area + cold working	--

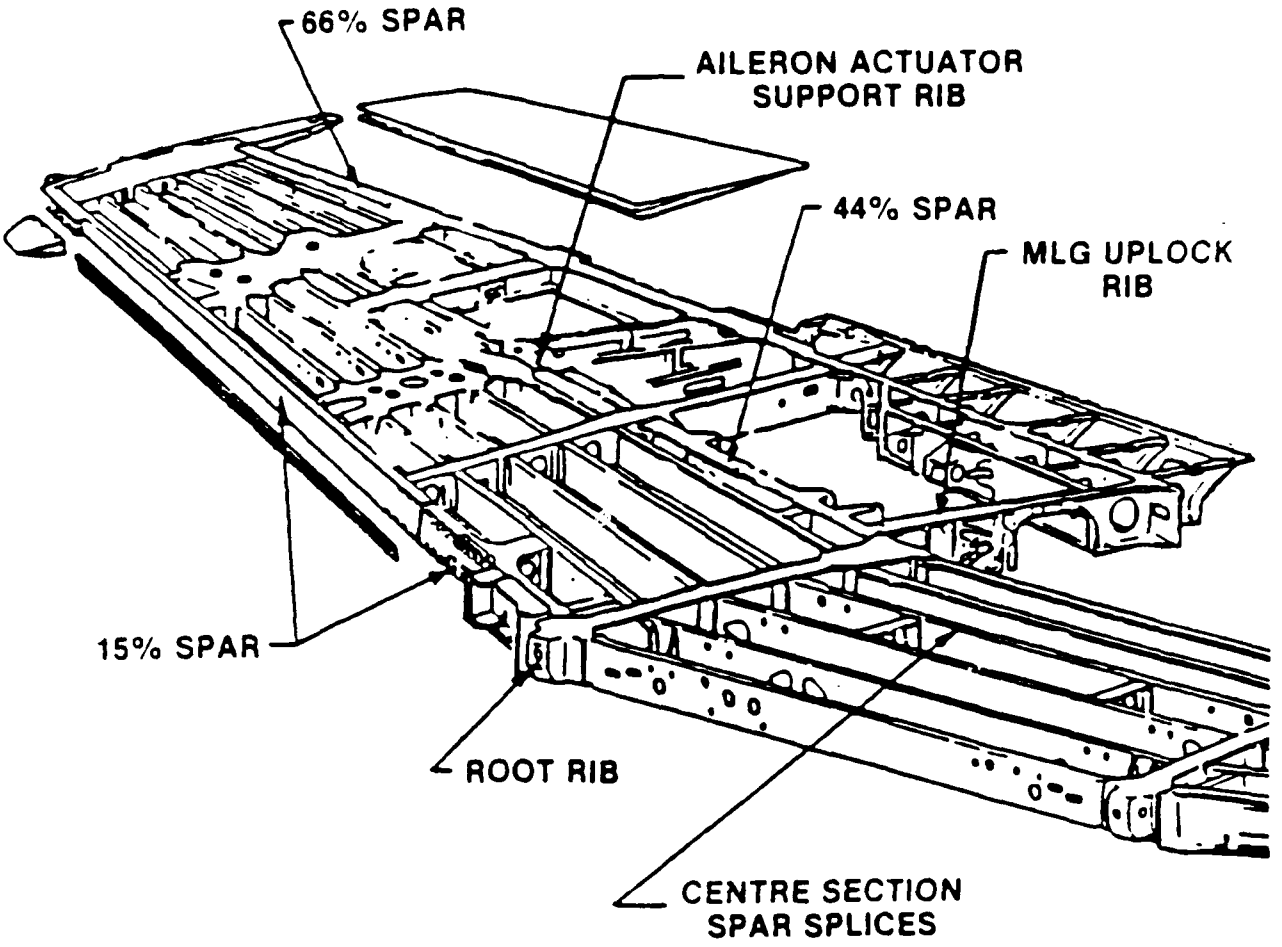


FIGURE 9 - WING COMPONENT CHANGES

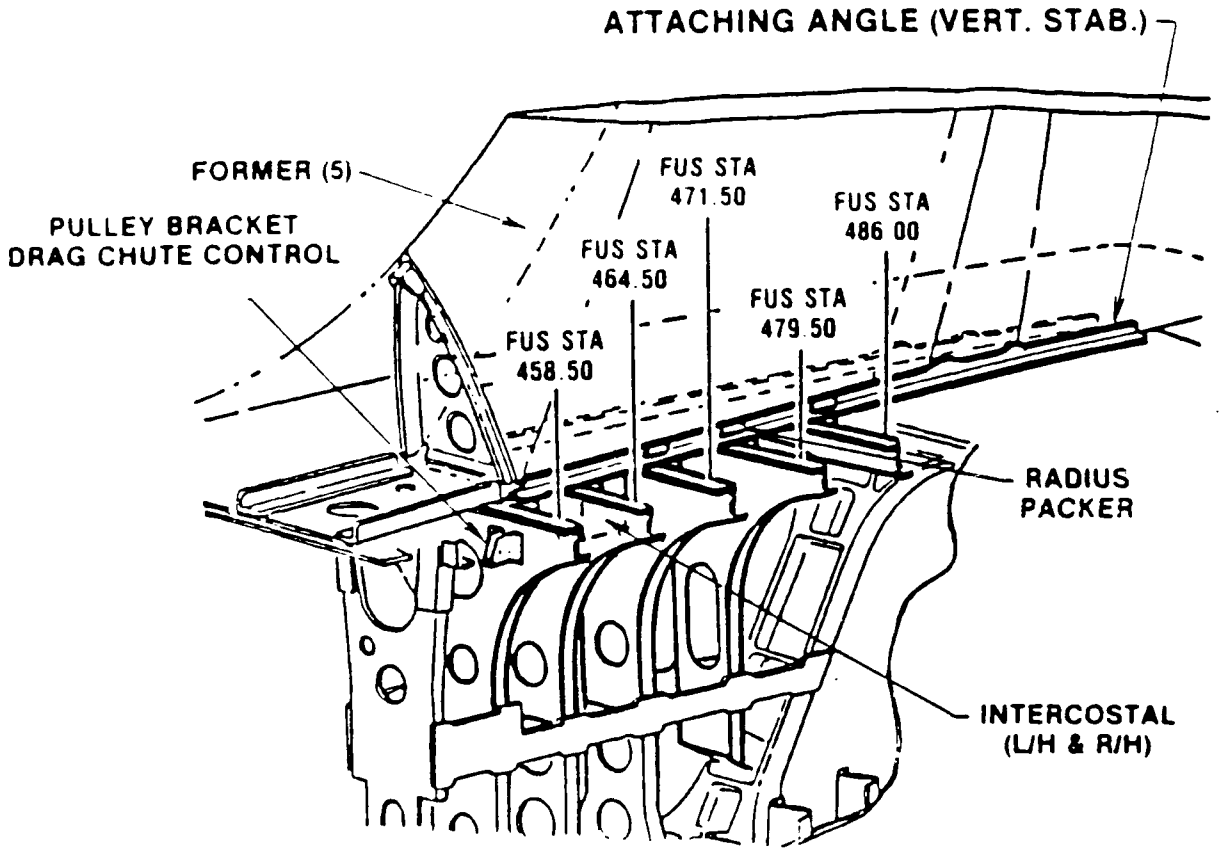


FIGURE 10 - REDESIGNED FORMERS LOCATIONS

4.0 SCOPE OF THE TEST

The scope of the program will be:

1. The full scale testing of a CF aircraft representative of the current fleet both in usage and environment.

2. The inspection of the test article under a modified ASI to establish the condition of the test article.

3. The durability testing of the test article to 16000 hours plus the test article airframe hours. This is required by the military specification MIL-A-008867B "Airplane Strength and rigidity Ground Tests" to take into account the differences between the actual load spectrum in flight and the applied load spectrum.

4. The damage tolerance testing of the test article to 8000 hours subsequent to durability testing.

5. The application of a load spectrum representative of current and future CF116A/D mission profiles. The load spectrum will be based on squadron 419 mission profile: This squadron is a training squadron and the CF116 will be used as a trainer in the future.

6. The strain gauging of the test article prior to testing to provide baseline load data.

7. The teardown of the test article subsequent to testing for a final condition inspection.

8. The provision of marker blocks in the load spectrum for fractographic analysis.

5.0 TEST PLAN

5.1 TEST ARTICLE SELECTION

A decision analysis has been performed for the selection of the test article.

The objectives considered in this analysis were divided in "MUSTS" and "WANTS" and these are shown in Table 6.

The first step of the decision analysis was to eliminate all the aircraft that could not meet the requirements of the objectives classified as "MUSTS".

In a nutshell an aircraft was rejected if it had:

1. Non-Standard items.

2. Non-Standard structural modifications and/or repairs.

3. Overstress cases experienced and/or defects detected.

It was conservative to do so since an overstress could possibly have a retardation effect on the crack propagation. Moreover, it is difficult to measure and compare the impact of different levels of overstress cases.

Finally, the aircraft 116729 has been chosen as test article.

TABLE 6 LIST OF OBJECTIVES FOR CF116 TEST ARTICLE DECISION ANALYSIS

MUSTS

a. Standard configuration

b. Structural modifications representative of 419 Squadron

c. Structural repairs representative

d. Overstress status representative.

WANTS

a. Load spectrum representative of 419 Squadron

b. Ability to interpret results. Maximum findings in durability test

c. Occurrences of natural cracks

d. Minimize impact on CF operations.

5.2 TEST RIG FACILITY

The stand alone test rig will be located in a secure fenced in area adjacent to the CL-600 Challenger fatigue test rig Building 117, Plant 1, Canadair. Most of the same equipment, facilities and accumulated expertise, used during CL-600 Challenger airframe test will be used. The loading arrangement will be similar to that described in Figure 12 to 15. Approximately forty five (45) tension-compression jacks will be used to apply the required loads. The cockpit will be pressurized.

The test article will be tested as a semifree-floating body in a balanced state, with fixed instrumented reaction linkages possibly at the forward and aft fuselage. Each loading condition will then consist of applied jack loads and fixed reactions in a statically determinate relationship. Counterbalancing of the test article will be by a deadweight system to simulate an approximately weightless condition.

When designing the basic test rig, accessibility to the test frame will be implemented as follows:

a. All loading arrangement attached to the test article will use mechanical fasteners and be easily removable.

b. It will be possible to disassemble the wing from the fuselage and remove it from the test rig using a specially designed handling dolly.

c. It will be possible to disassemble the vertical stabilizer from the rear fuselage and remove it from the test rig.

d. It will be possible to work on the upper dorsal longeron area from the outside with the test article 'in situ'.

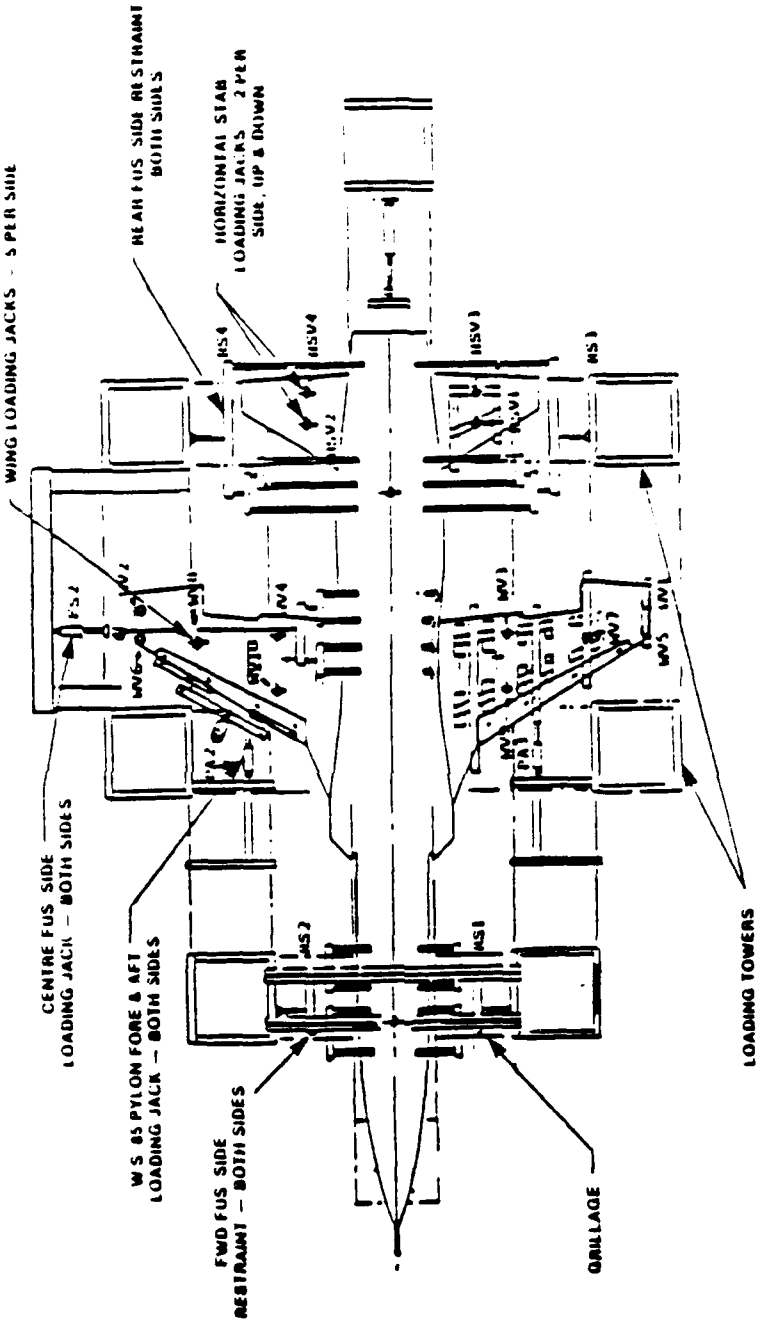


FIGURE 12 LOADING ARRANGEMENT VIEW LOOKING UP

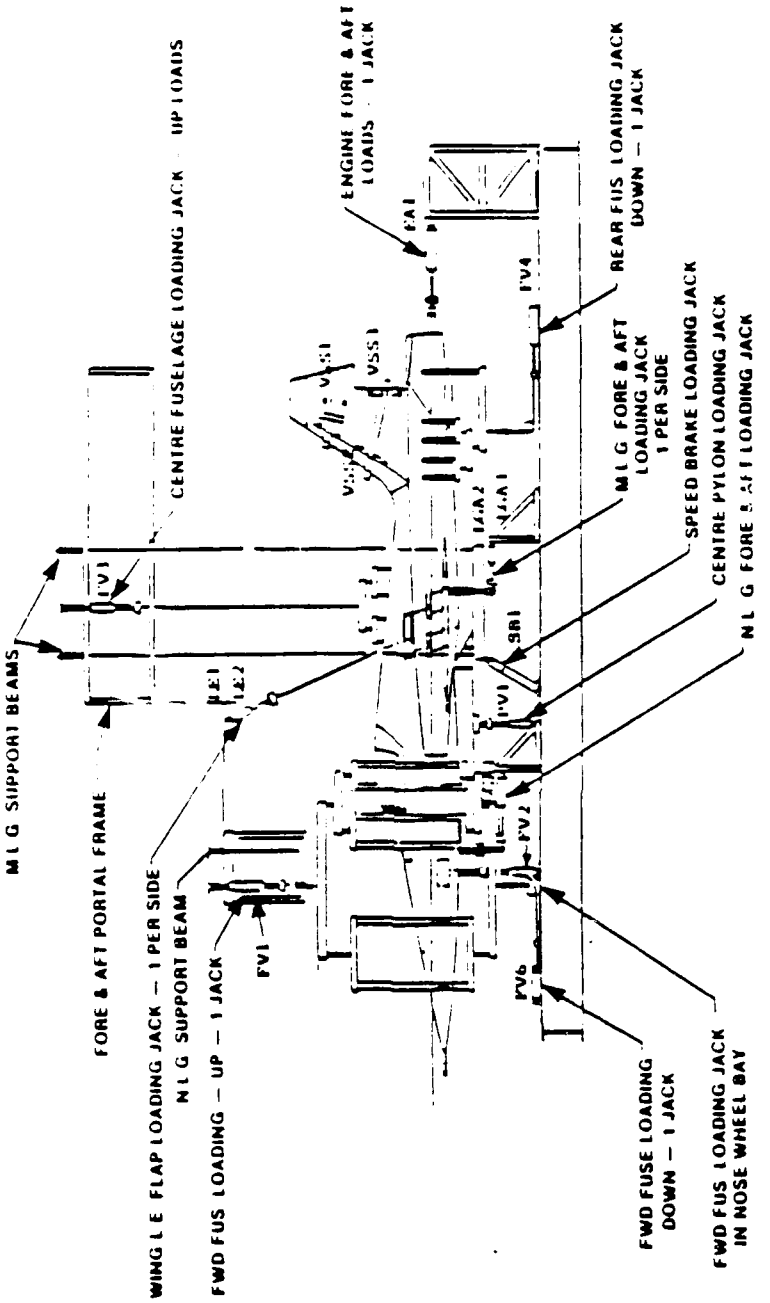


FIGURE 13 LOADING ARRANGEMENT SIDE VIEW

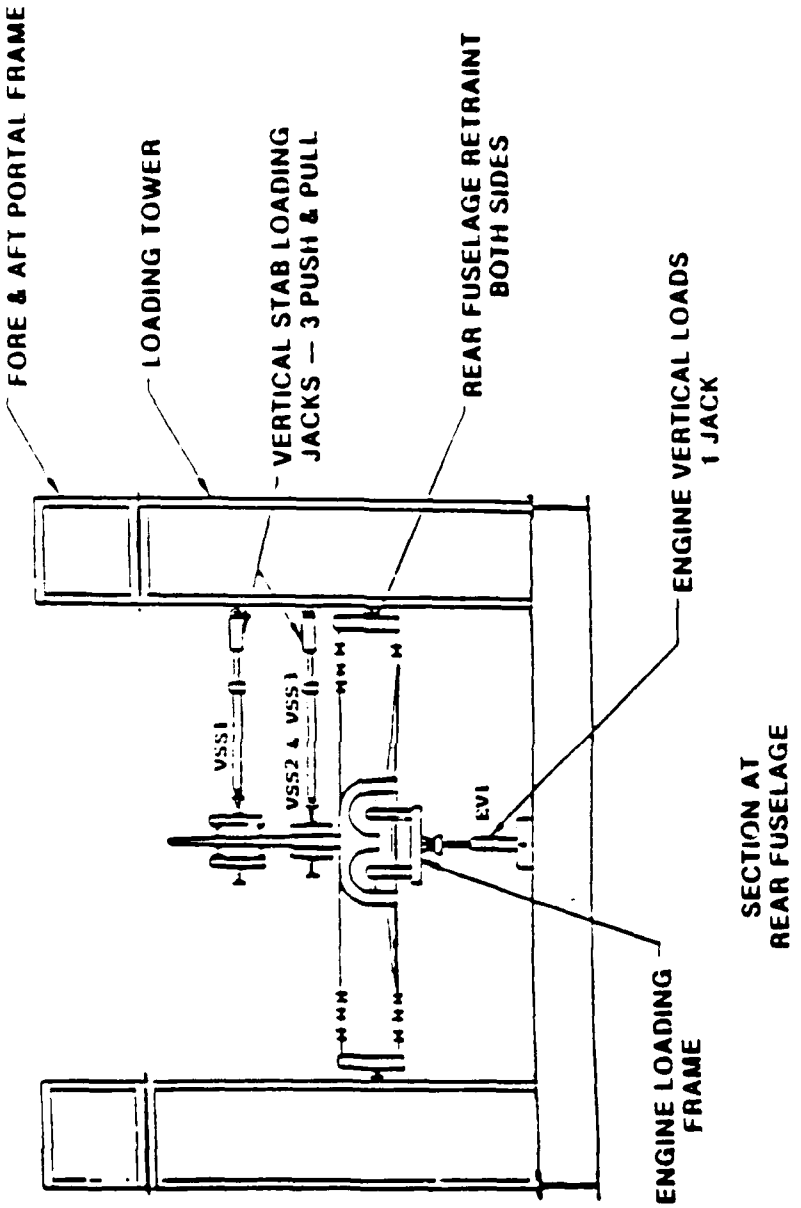


FIGURE 14 LOADING ARRANGEMENT SECTION AT REAR FUSELAGE

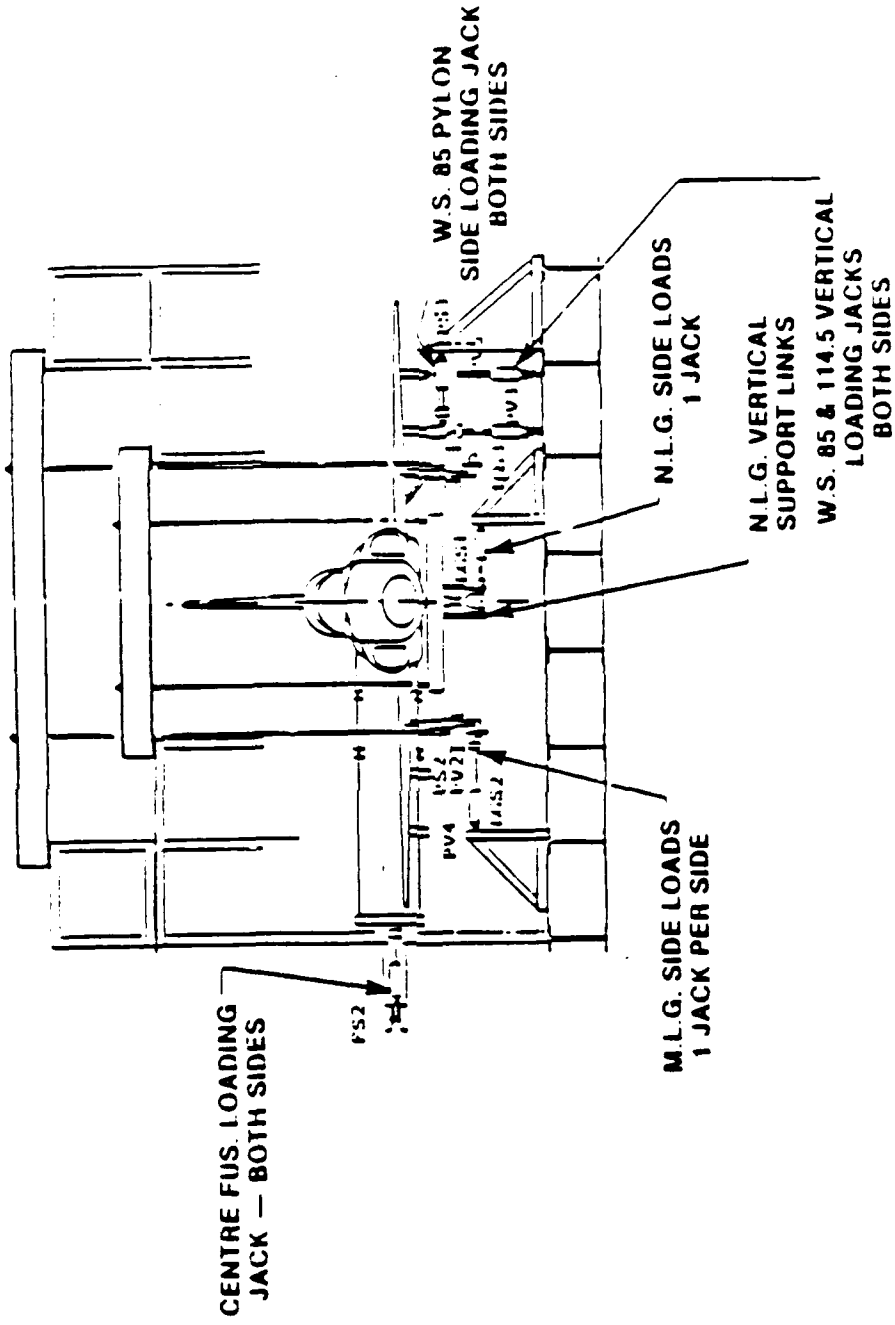


FIGURE 15 LOADING ARRANGEMENT VIEW LOOKING AFT

5.3 LOAD SPECTRUM DERIVATION

Figure 16 shows the steps followed to obtain the untruncated spectrum.

The load spectrum is based on 419 Squadron operations for the CF116A aircraft. It will be modified to balance the loads on the test article and to ensure a conservative approach to damage initiation in critical areas. It will also be truncated and tested on coupons representative of the critical locations. Different stress levels and truncations will be tested. Then marker blocks will be provided to the load spectrum for fractographic analyses.

The derivation of the load spectrum will employ data from a number of studies including:

1. Accelerometer data
2. Vertical Stabilizer MSR data
3. VGH (Velocity, Acceleration (g), Altitude)
4. Wing mechanical strain recorder data
5. Longeron Stress Survey
6. Mission Profile Data
7. Vertical Stabilizer Strain Gauges Data
8. Northrop Spectrum
9. Instrumented Wing Flight Load Survey
10. CF Landing Survey
11. Military Specifications, etc.

5.3.1 Flight Sequence

The test will be carried out on a mission by mission basis.

Identical blocks equivalent to 1,000 flights will be developed and will be representative of the 419 Squadron Mission Profiles.

Each block will be obtained by creating a random sequence of the missions flown by the 419 Squadron CF116A aircraft.

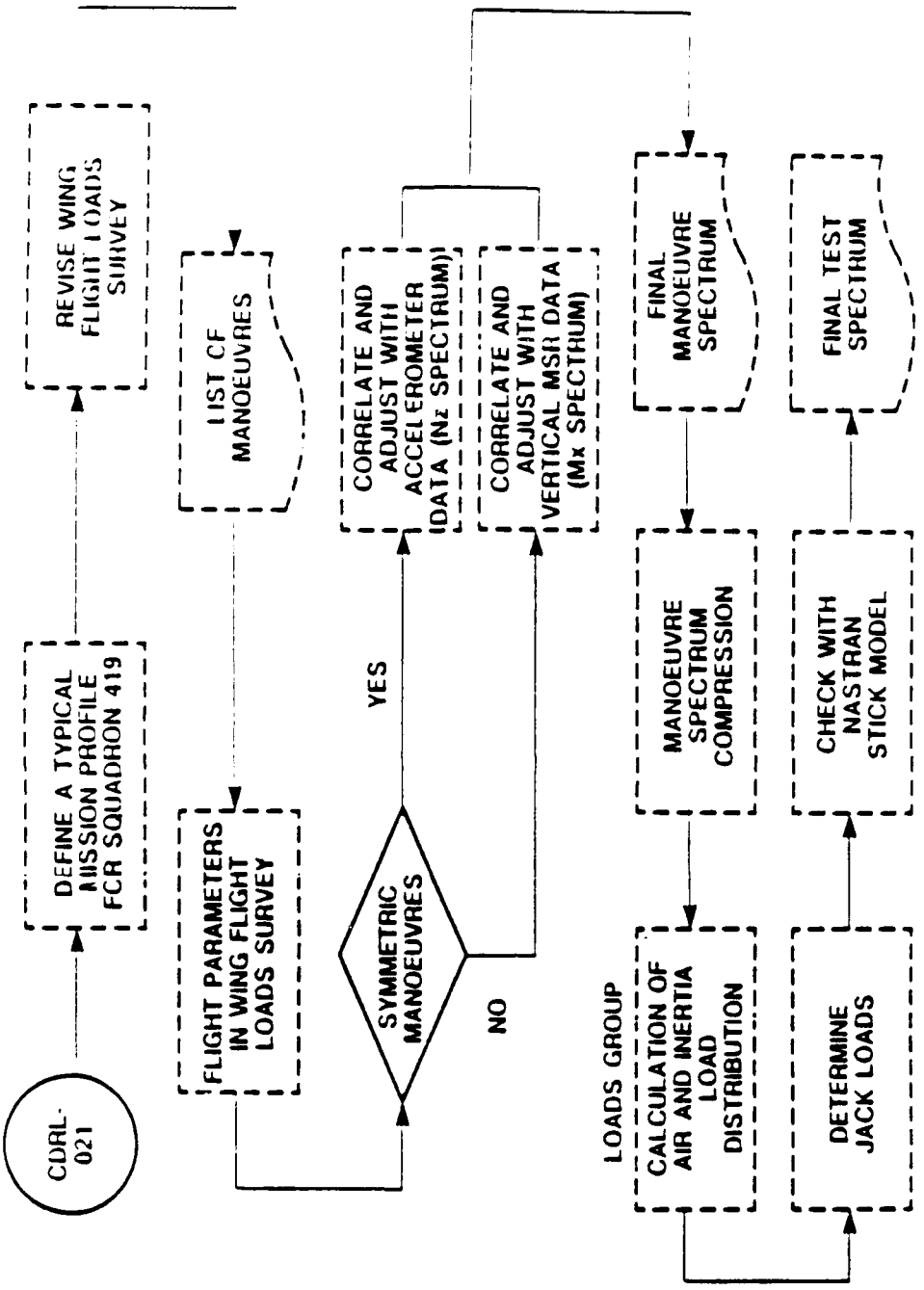


FIGURE 16 SPECTRUM DEVELOPMENT

5.3.1 Flight Sequence (Cont'd)

The types of missions in the flights are:

1. Familiarization/Basic Handling/Clearhood
2. Formation (Day or Night)
3. Instrument Training/Ferry Flights
4. Navigation Training
5. Air to Air Refuelling
6. Air to Air Gunnery
7. Dart Tow
8. Advanced Handling/BFM/ACM
9. Ground Attack on the Manned Range
10. Ground Attack on the Tactical Range
11. Ground Attack with Heavy Ordnance
12. Air Display
13. Test Flight

5.3.2 Loading Occurrences

The sequence of the manoeuvres will be compiled in accordance with the 419 Squadron Mission Profile.

The total occurrences of each "g" level in the loading block will be matched with the accelerometer data.

The pitching and roll rates for the manoeuvre will be obtained from the instrumented wing flight load survey data.

The frequency and intensity of the gusts will be determined from Military Specifications.

The vertical stabilizer bending moment spectrum will be checked against the data coming from the MSR data.

5.3.3 Balanced Test Loads

The manoeuvre spectrum will be compressed to eliminate manoeuvres having the same load distribution as others.

Then the jack loads will be determined to match the air and inertia loads distribution by a optimisation process.

5.3.4 NASTRAN Model

A NASTRAN stick model, see Figure 17, will be used to represent the structural stiffnesses of the airframe in shear, bending and torsional modes. This model will be used for the following purposes:

1. Predict the deflections of the airframe structure at various loading points under the applied loads.

2. Determine the attachment reaction loads which can be used to design the test rig. The validity of the end conditions of the test article can also be checked based on this information.

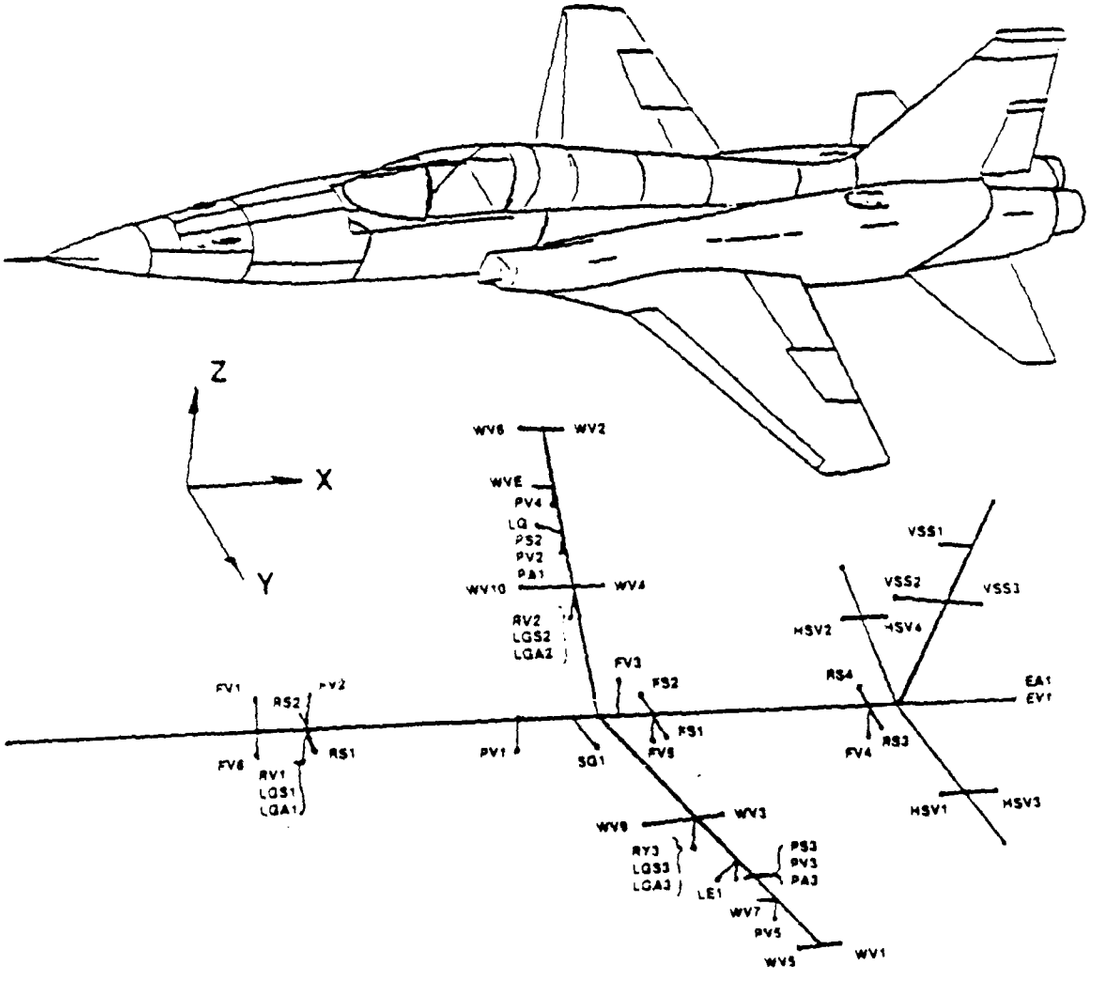


FIGURE 17 CF116 NASTRAN DEFLECTION MODEL

5.4 DURABILITY TESTING

Provisions will be made to apply 16,000 equivalent flight hours (2 lifetimes) of flight and ground loads to the test article. The test loads will be applied in a mission by mission manner.

The application of the test load spectrum is expected to result in the occurrence of natural cracks in the structure of the airframe.

5.5 DAMAGE TOLERANCE TESTING

A further 8,000 equivalent flight hours of damage tolerance testing will be carried out on completion of the durability testing. This will consist of a continuation of the mission by mission load applications in order to measure crack propagation rates, both for naturally occurring cracks and for introduced cracks at critical locations.

To encourage the rapid initiation of cracks, simulated damage will be introduced using 0.008 inch cuts configured with square corners. This technique was developed and successfully used for the FAA Damage Tolerance Certification to FAR25-571 amendment 45 of the CL-600 and CL-601 aircraft.

The simulated damage will be introduced into accessible areas. If and when required, initiated damages will be introduced into inaccessible areas where some parts of the structures will have to be removed. Where the flaw is required in a fastener hole, the fastener will first be removed, the edge of the hole will have the saw cut introduced and the fastener will be reinstalled.

Some information relevant to the use of saw cuts is contained in Canadair report RAT-600-173 "Correlation of Natural/Artificial Cracks in 7475 Material".

5.6 IDENTIFICATION OF CRITICAL ZONES

The test article will be divided in three zones regarding the accuracy of the load spectrum applied during the test.

1. Zone A:

   In this zone, the applied load spectrum must reflect the real life load spectrum with the greatest possible accuracy.

   This zone contains areas:

   a. which experienced cracks in the past.

   b. For which the previous fatigue or damage tolerance analyses have shown life lower than twice the expected life of the aircraft.

2. Zone B:

   In this zone, the applied load spectrum must reflect the real life load spectrum with a good accuracy.

   This zone contains areas which have been considered by the SLEP program.

3. Zone C:

   In this zone, no crack has ever been found and no previous fatigue and damage tolerance has been done.

The locations of the three zones are as follows:

1. Zone A:

   a. Wing from WS 65R to WS 65L, (Figure 18). It contains the following critical areas:

      1) Lower skin critical radius WS 27

      2) Lower skin critical fastener holes

      3) 44% Spar (W.S. 65R to W.S. 65L)

      4) Wing attachment points

      5) Lower skin aft edge in main landing gear strut-well

      6) Lower skin drain holes

7) Centre section spars other than 44%

8) Main landing gear uplock rib at Web cut-out

b. 66.6% spar from W.S. 80R to W.S. 80L (Figure 18).

c. 15% spar from W.S. 105R to W.S. 105L (Figure 18).

d. Main landing gear (Figure 19). It contains the following critical areas:

1) A crack has been found in the side brace trunnion pin.

2) Side brace attachment lug

e. Nose landing gear (Figure 20). It contains the following critical areas:

1) Drag Brace Lug

2) Nose Landing Gear Housing

f. Fuselage from FS 220 to FS 418 (Figure 21). It contains the following critical areas:

1) Upper cockpit longeron FS 220 to FS 284

2) Fuselage upper skin at centre fuel cell floor FS 333 to FS 355

3) Centre fuel cell floor FS 355 to FS 362

4) Dorsal longeron FS 343 to FS 418 and attachment to FS 388.75 bulkhead

g. Canted frame FS 487 (Figure 21). It contains the following critical areas:

1) Upper frame flange

2) Tail cone attach. bolt

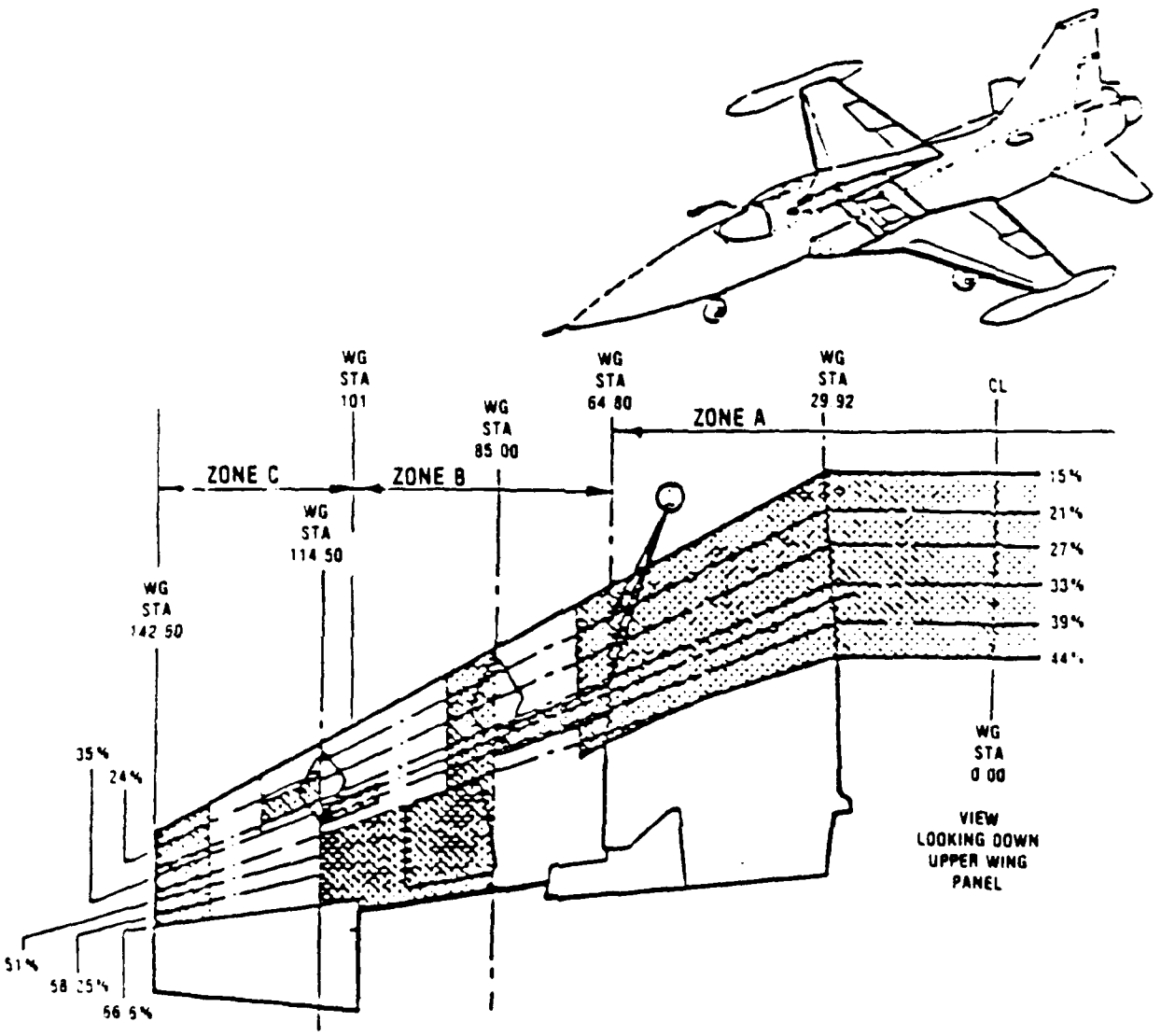


FIGURE 18 WING ZONE PARTITION

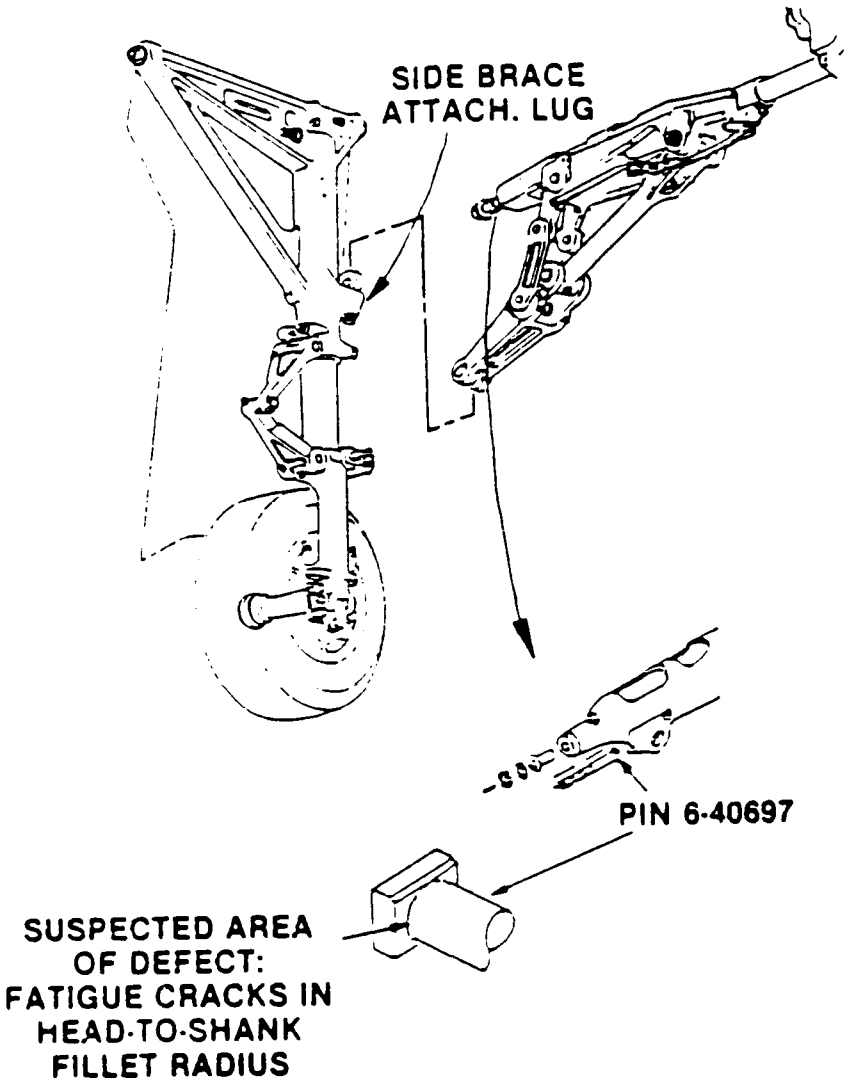


FIGURE 19 MAIN LANDING GEAR

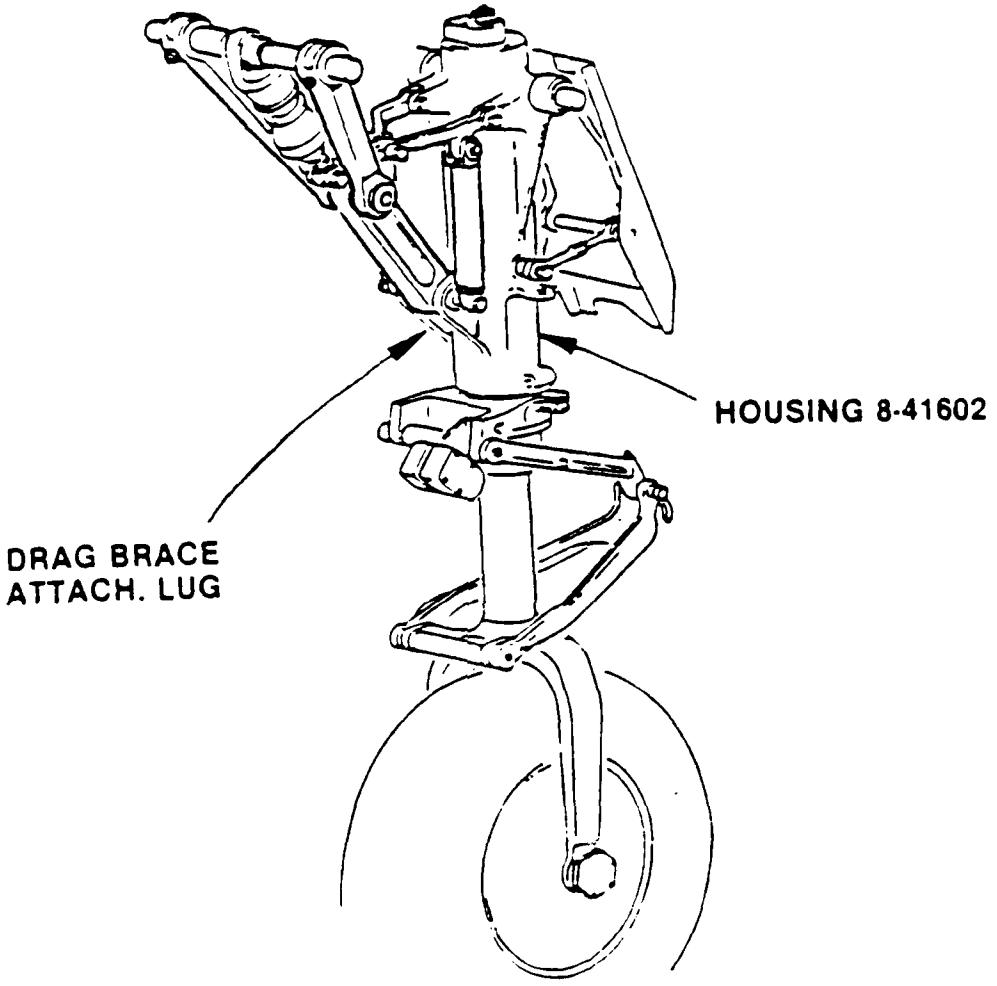


FIGURE 20 NOSE LANDING GEAR

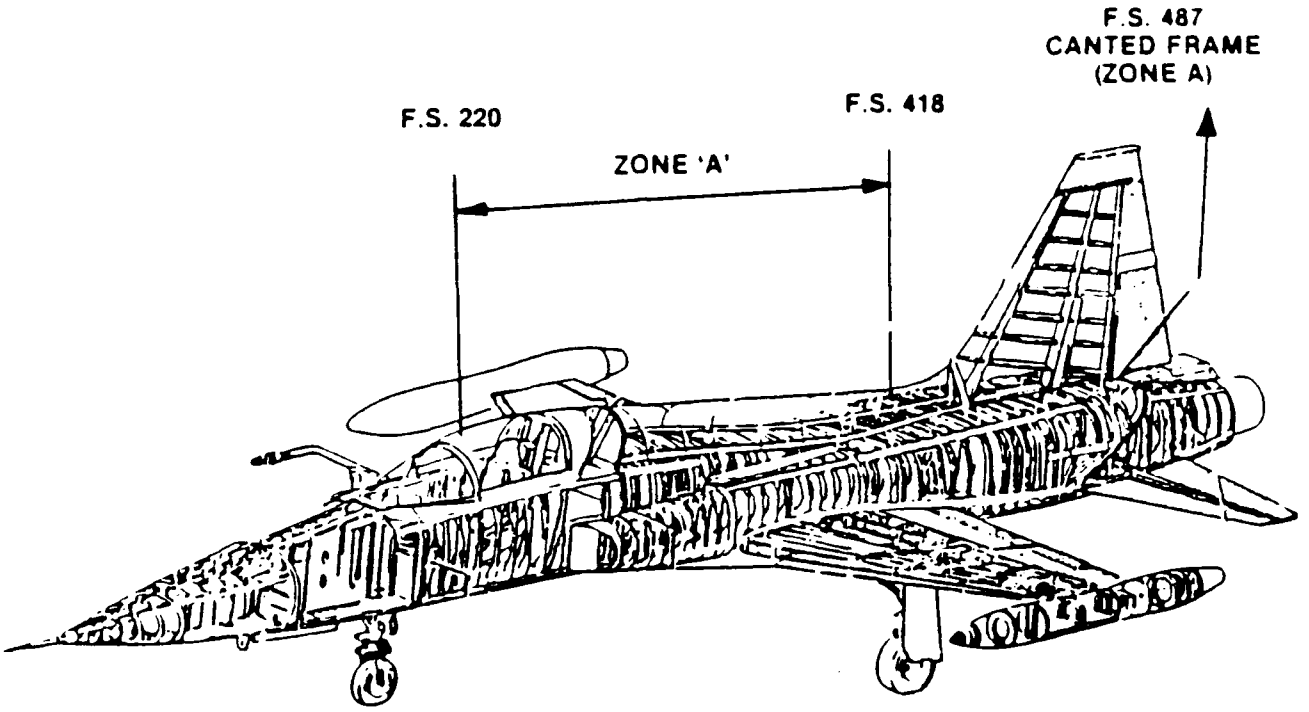


FIGURE 21 FUSELAGE ZONE A

h. Vertical Stabilizer (Figure 22 and 23). It contains the following critical areas:

1) Skin at VSS 17.0 Forward Radius

2) Spar at VSCS 7.0

3) Spar at VSS 55.0

4) Attachment Angle

2. Zone B:

a. Wing from WS 65 to WS 101 (Figure 18). It contains the following critical areas:

1) 44% Spar

2) Spars other than 44%

3) Fasteners holes

4) Drain holes

b. Engine Mounts (Figures 24 to 26)

c. Horizontal stabilizer. It contains the following critical areas:

1) Main spar (Figure 27)

2) Horn fitting and torque tube (Figure 28)

3. Zone C:

a. Zone C. contains all other areas

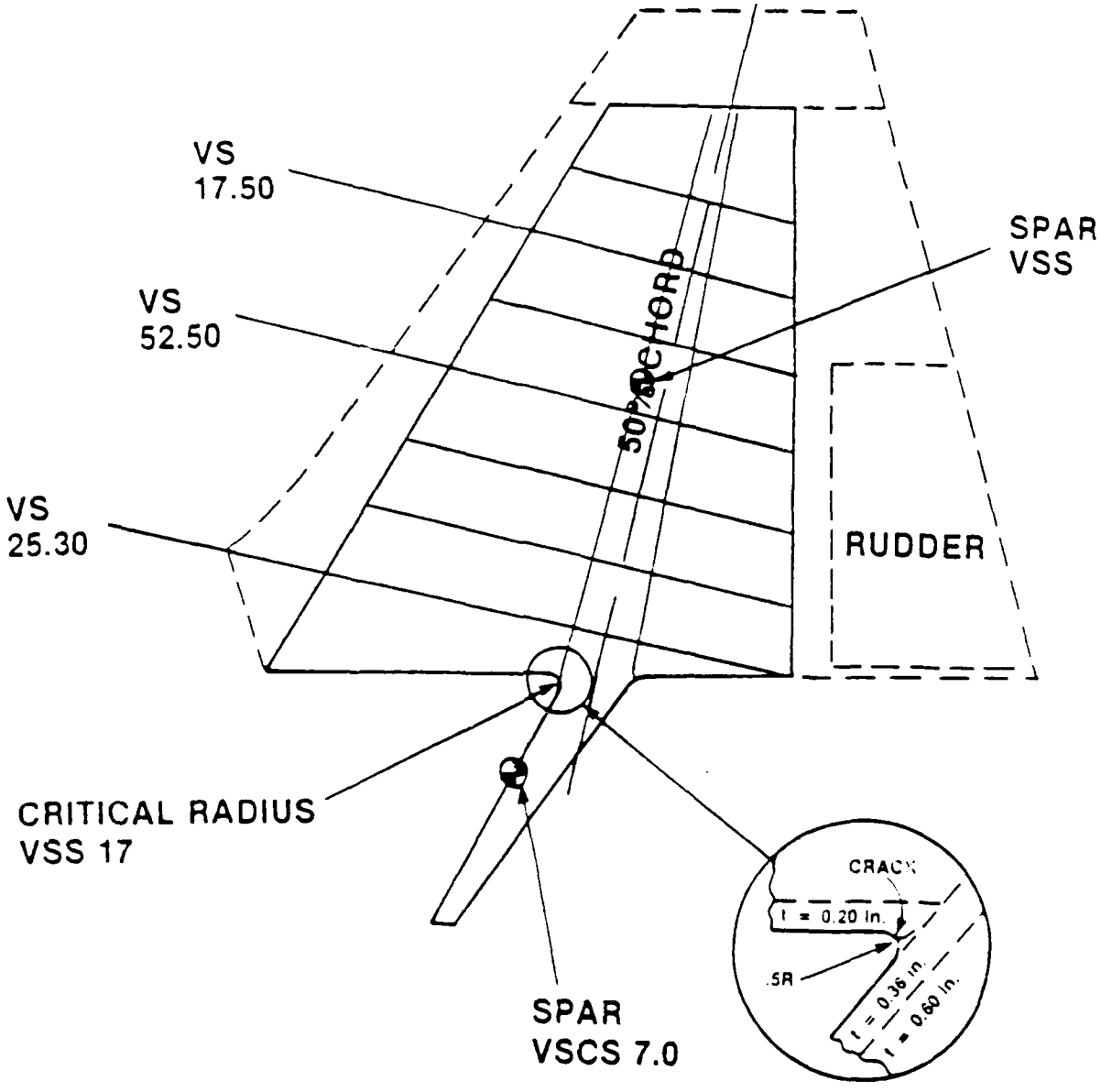


FIGURE 22 VERTICAL STABILIZER

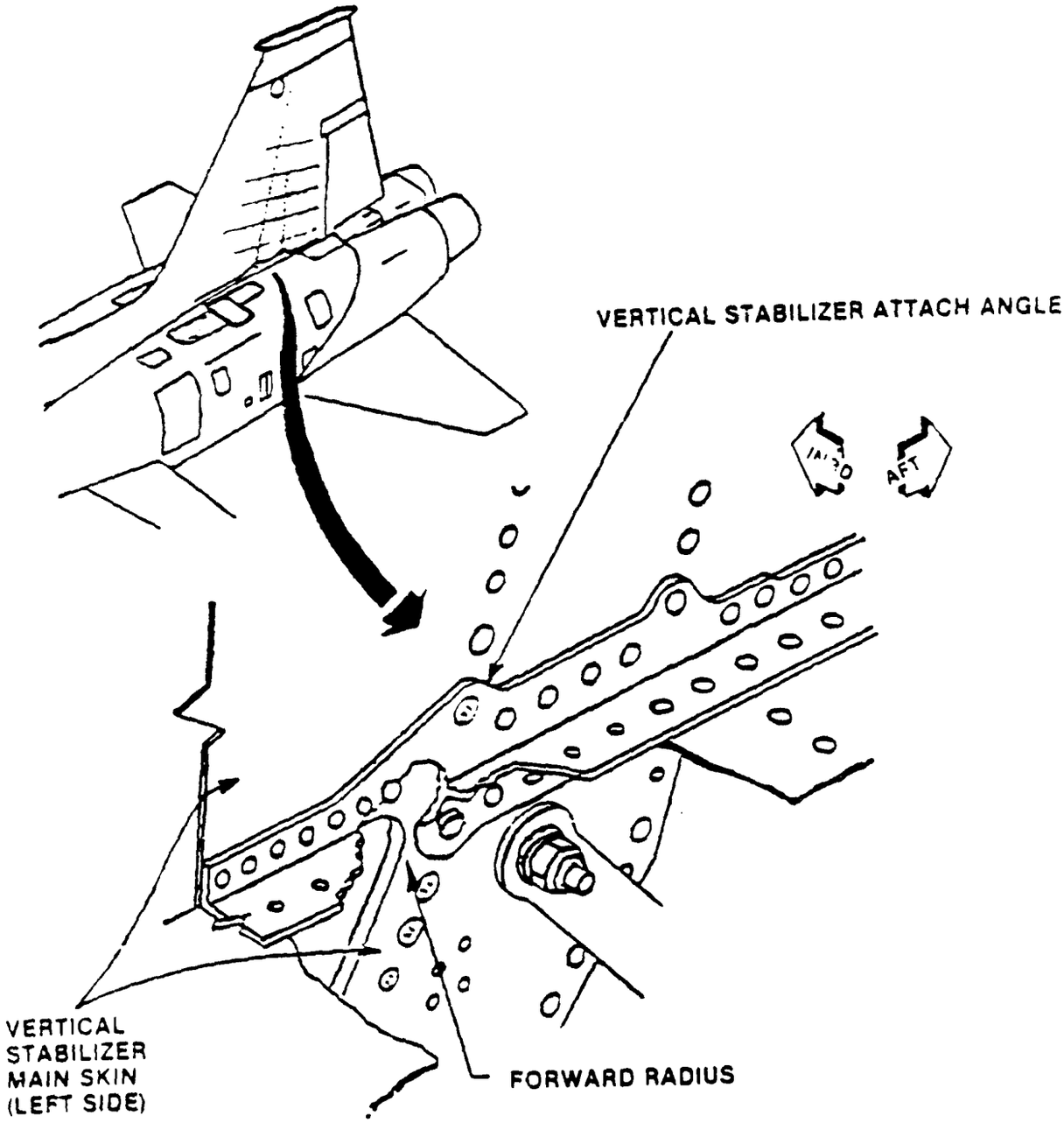
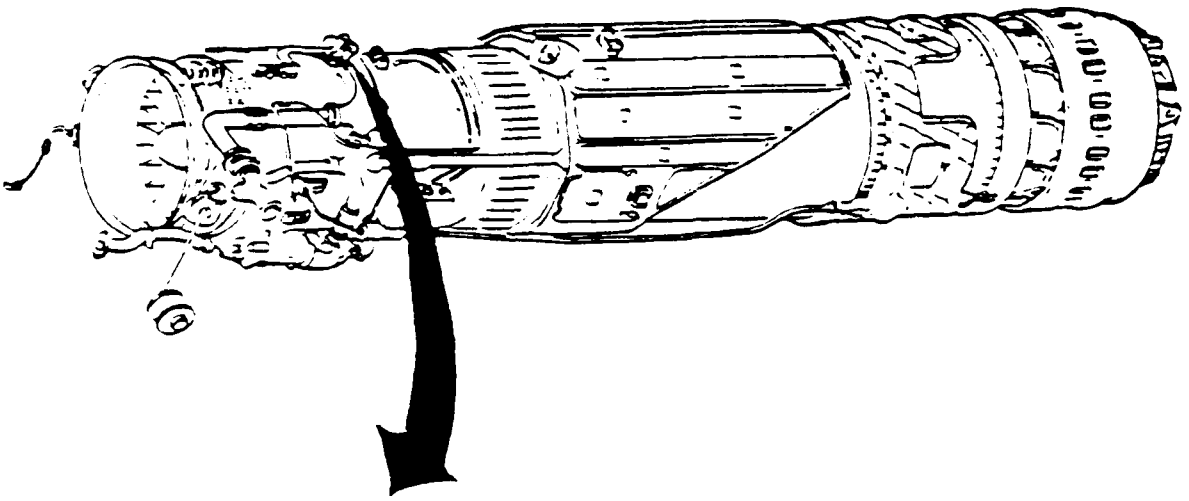
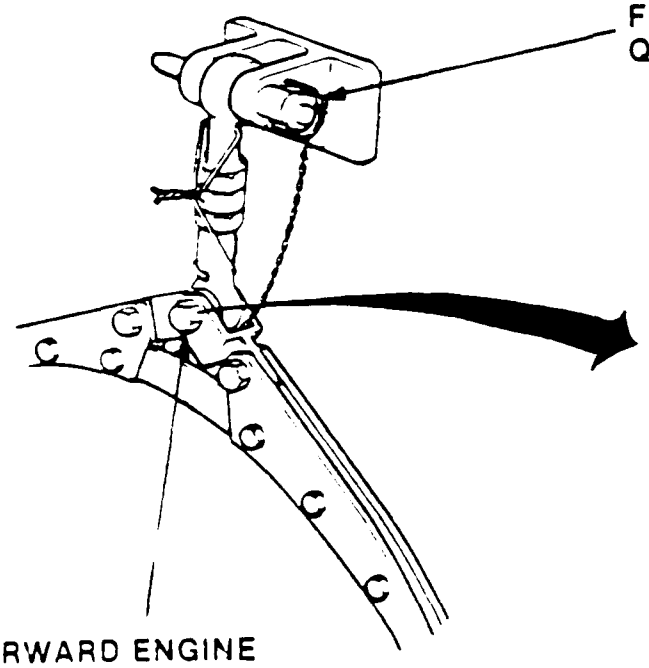
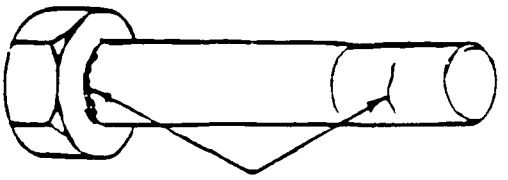


FIGURE 23 VERTICAL STABILIZER







SUSPECTED AREAS OF DEFECT

SUSPECTED DEFECTS:
• TRANSVERSE DISCONTINUITIES

FIGURE 24 FORWARD ENGINE MOUNT LINE BOLT AND QUICK RELEASE PIN

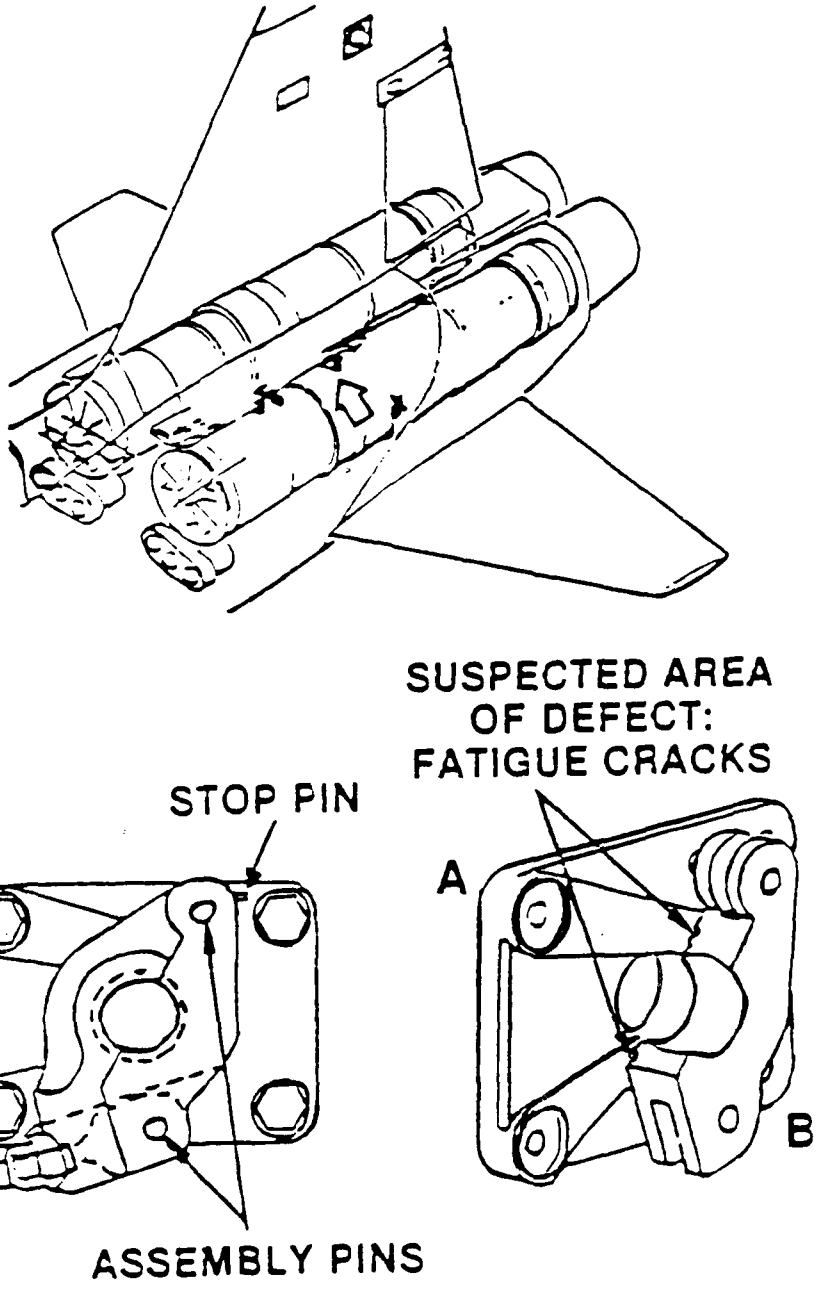


FIGURE 25 MAIN INBOARD ENGINE MOUNT

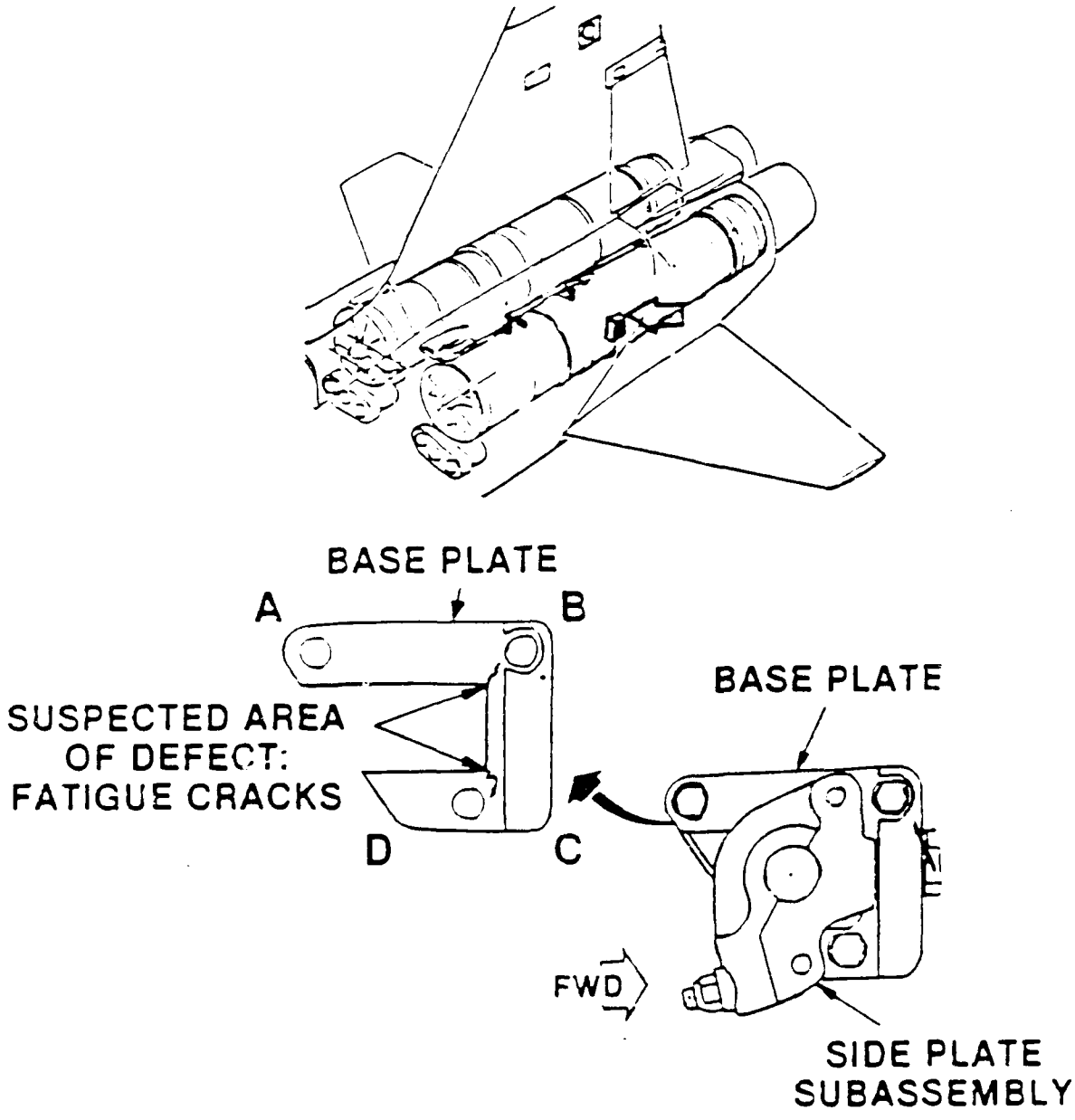


FIGURE 26 MAIN OUTBOARD ENGINE MOUNT

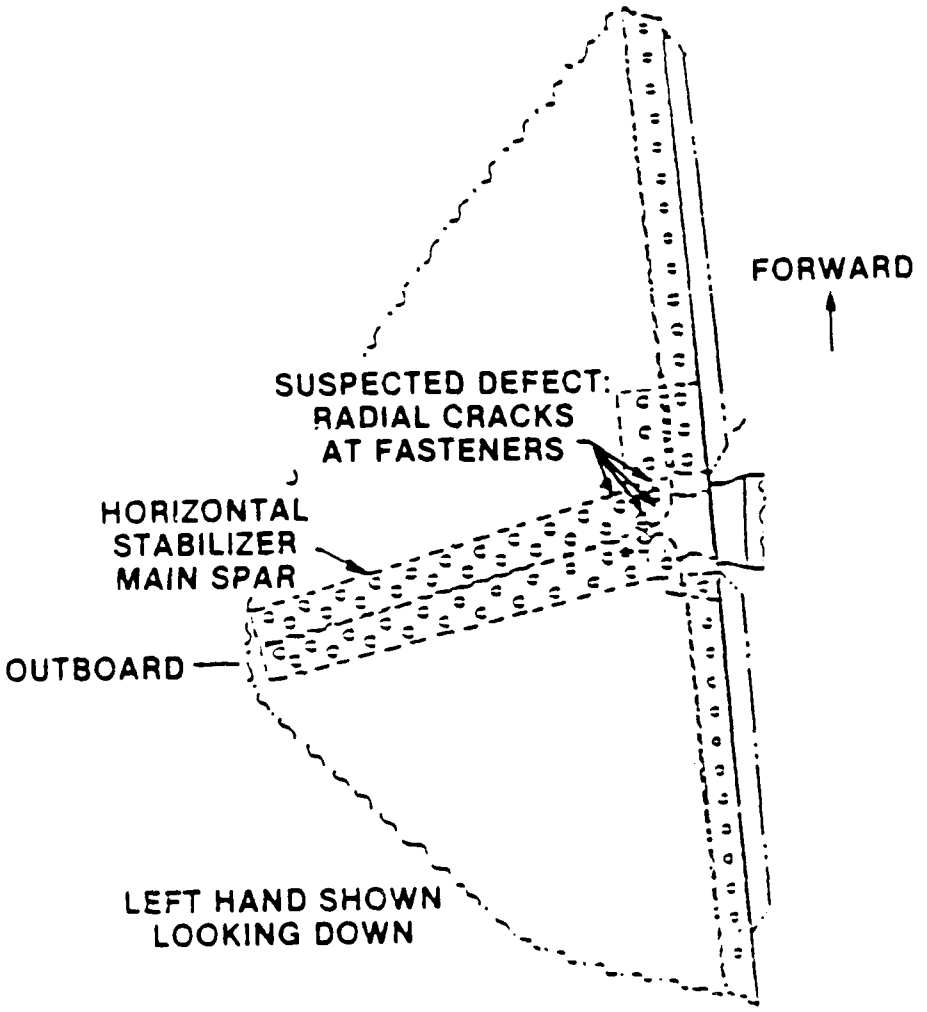


FIGURE 27 HORIZONTAL STABILIZER MAIN SPAR

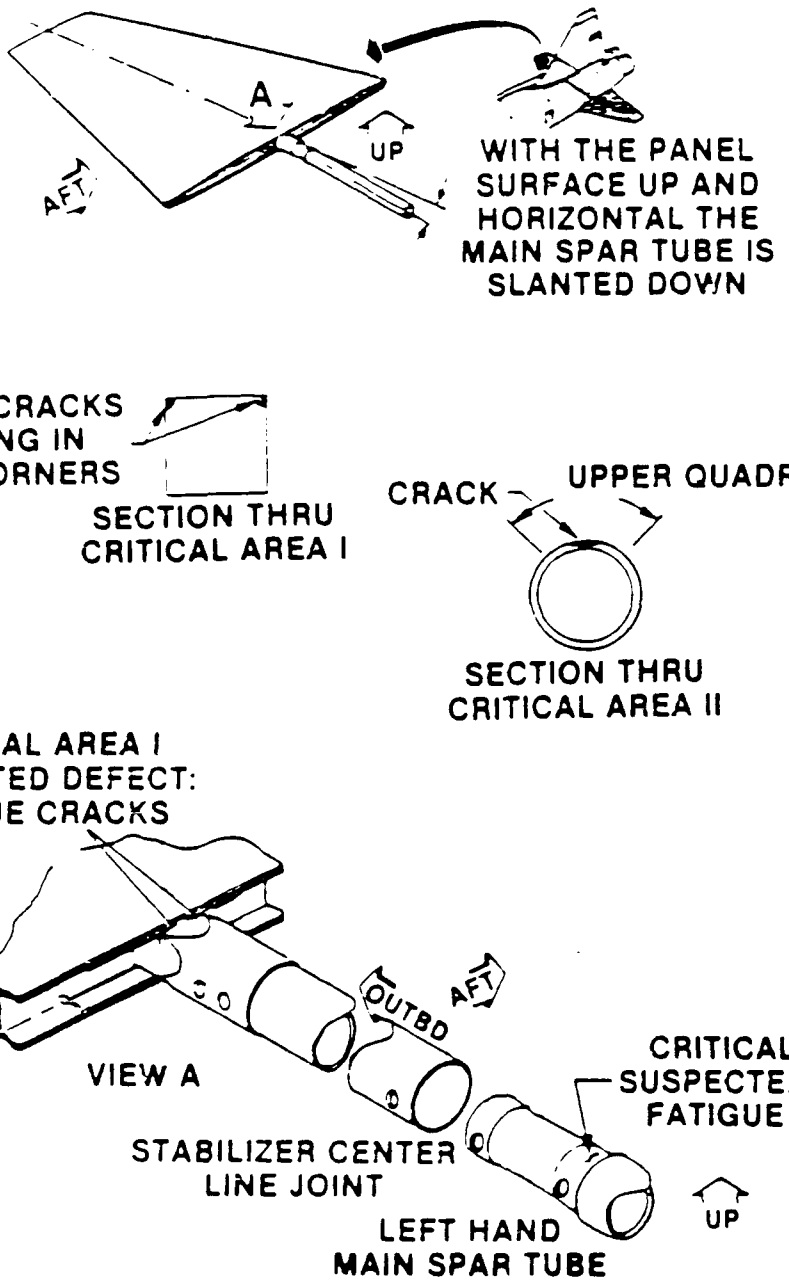


FIGURE 28 HORIZONTAL STABILIZER HORN FITTING AND TORQUE TUBE

5.7 INSPECTION AND DETECTION OF STRUCTURAL DAMAGE

MONITORING OF THE STRUCTURAL CRITICAL AREAS

A total of 120 strain gauges shall be installed on the test article for the purpose of static strain survey, ie.e. measure and record the static strain at selected end points for selected missions. This will allow to substantiate the NASTRAN finite element model.

Up to 20 strain gauges shall be selected in critical locations to monitor the health of the test article during the test. Strain values shall be checked for out of limit condition at each endpoint.

5.8 INSPECTION DURING THE TEST

A continuous surveillance during the test by an experienced test engineer and QC personnel shall be made for the timely detection of structural damage or cracks in the critical areas during load spectrum application.

Throughout testing, a number of down times are planned for the Q.C. inspector(s) to inspect the test article for damage and cracks with varying degree of disassembly. The level of inspection, the areas to be stripped and inspected, and the time intervals, respectively, are described below.

Inspection Levels

Level 4 Inspection

The access panels (where practical) shall be removed. The opened areas and the fuselage (where practical) surface shall be visually inspected while test is operating routinely.

This inspection shall be performed approximately every 500 equivalent flight hours or on every other day basis when the test is running, whatever is most practical.

Level 3 Inspection

The test shall be stopped. The canopy and leading edge flaps shall be removed. The wing shall be removed from test article. A visual inspection of the airframe together with NDT checks, using dye-penetrant, shall be performed paying particular attention to Zone A and items 1 through 9 below.

1. Lower wing skin critical radius
2. Lower wing skin fastener and drain holes
3. Lower wing skin aft. edge in main landing gear strut well
4. Upper cockpit longeron
5. Dorsal longeron (at the locations which are accessible)
6. Vertical stabilizer critical radius
7. Fuselage upper skin at centre fuel cell access door frame
8. Centre fuel cell floor
9. wing attachment points

This inspection shall be performed approximately every 1,000 equivalent flight hours.

Level 2 Inspection

This inspection level shall be identical to Level 3 but shall also include landing gear removal. Particular attention shall be given to Zone B, and item 1 listed below and all items listed in Level 3.

1. Main landing gear side brace trunnion pin.

This level of inspection shall be performed approximately every 2,000 equivalent flight hours.

Level 1 Inspection

This level of inspection shall be identical to Level 2 but shall include aft section and dummy engines removal. A comprehensive visual inspection of the airframe shall be performed.

A nondestructive inspection shall be performed as in Level 2 and Level 3. The following items shall also be nondestructively inspected.

a. Wing internal structure

b. Canted frame upper flange

c. Vertical stabilizer internal structure

d. Aft section attachment bolts

e. Engine Mounts

f. Horizontal stabilizer internal structure

This inspection shall be performed approximately every 8,000 equivalent flight hours.

6.0 MASTER SCHEDULE

This paper shows (Figure 29) the major schedule details to be followed throughout the course of the CF116 Full Scale Durability and Damage Tolerance Test (FSDADTT).

The schedule closely defines all work required for the program. This includes not only all work currently in progress, but also all work already completed at the time of this writing, as well all work to be performed through to the program's termination in 1991.

In addition, the schedule reflects various modifications to be incorporated into the test article including the installation of:

a. the redesigned dorsal longeron

b. five (5) redesigned aft fuselage formers

c. improved landing gear

d. improved and reskinned vertical stabilizer, and

e. improved wing (currently in manufacture at Bristol Aerospace Ltd.)

All modifications are expected to be installed on the test aircraft (CF116729) by the rig start date of February 1989.

The go-ahead date for the program was 27 April 1987. Project completion including the submission of the final report is scheduled for 29 March 1991.

The initial vertical stabilizer (2-31400-519) from CF116729 shall be reskinned prior to commencement of the test. Subsequent vertical stabilizers utilized shall have been modified to drawing K219-31001-1. Vertical stabilizers shall be replaced upon failure and not at specifically scheduled intervals.

Included on the FSDADTT Master Schedule are the following major milestones.

1. Spectrum Development
2. Test Article Preparation
3. Test Commission
4. The End of the First Durability Lifetime
5. The End of the Second Durability Lifetime
6. The End of the Damage Tolerance Lifetime
7. Test Article Strip Down (Decommission)
8. Final Report Submission

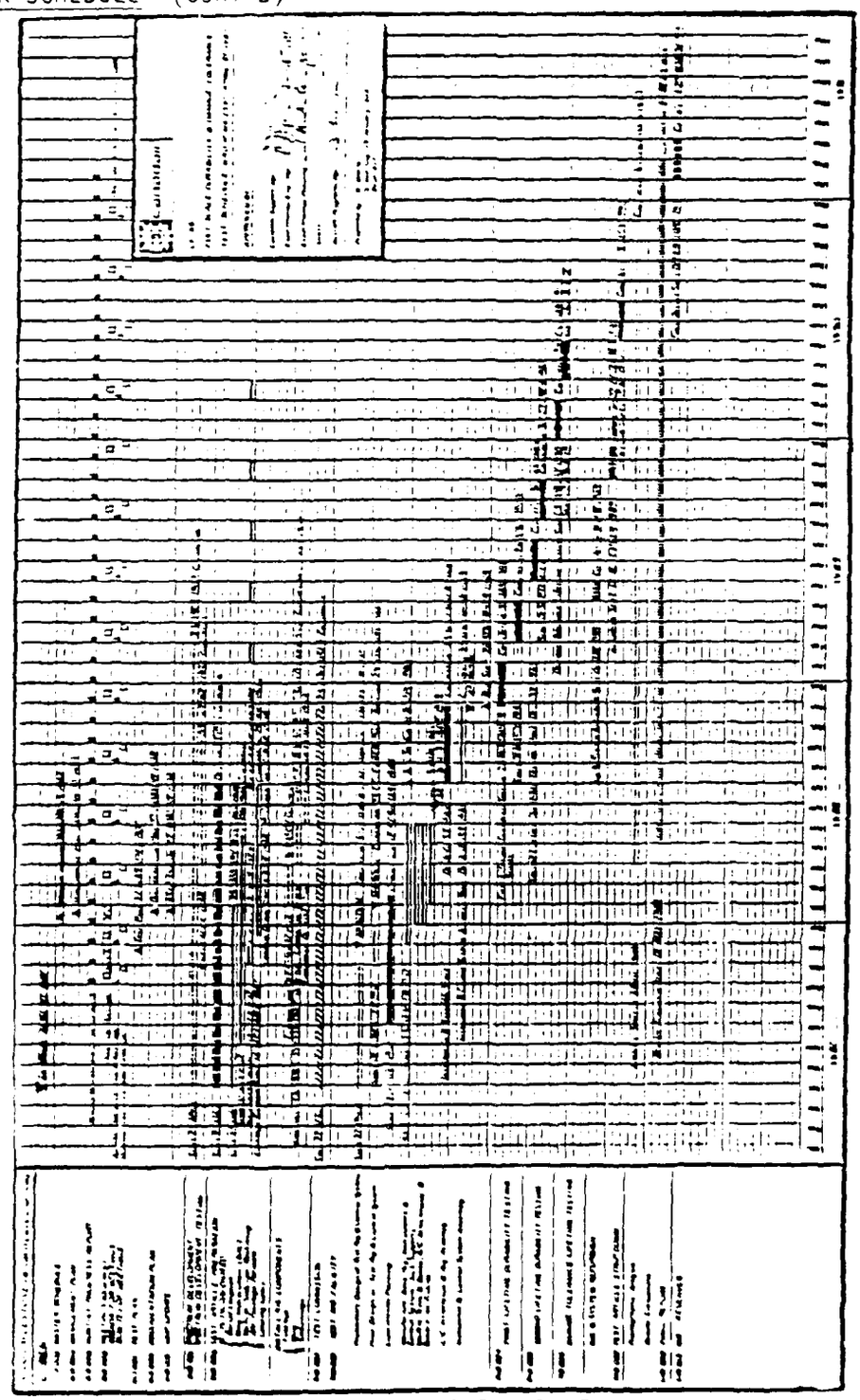


FIGURE 29 MASTER SCHEDULE

F-16C FULL-SCALE AIRFRAME DURABILITY TEST STATUS/RESULTS

MR CHARLES A. BABISH, IV

ASD/YPEF

INTRODUCTION

- RATIONALE FOR TEST
- TEST PROGRAM
- TEST STATUS
- TEST RESULTS TO DATE

RATIONALE FOR TEST

- INCREASE IN BASIC FLIGHT DESIGN GROSS WEIGHT FROM 22500 LBS TO 26910 LBS
- INCREASE IN MAXIMUM TAKEOFF GROSS WEIGHT FROM 33000 LBS TO 37500 LBS
- CHANGE IN OPERATIONAL USAGE FROM

  55% A-A
  20% A-G

  TO

  28% A-A
  57% A-G

- 8000 HOUR SERVICE LIFE BECAME A GOAL FOR THE F-16C/D AIRCRAFT, THUS A GREATER POTENTIAL FOR PROBLEMS EXIST

TEST PROGRAM

- TEST ARTICLE
  - STRUCTURALLY COMPLETE BLOCK 30 AIRFRAME
  - DUMMY LANDING GEARS, HORIZONTAL TAILS, AND ENGINE
- APPLIED LOADS
  - AIR, INERTIA, ENGINE THRUST, GROUND, STORE
  - COCKPIT AND FUEL TANK PRESSURIZATION
  - LEADING EDGE FLAP ACTUATED UNDER LOAD
  - TRAILING EDGE FLAPERON ACTUATED UNDER LOAD
  - RUDDER LOADED BUT NOT ACTUATED

TEST PROGRAM (CONT'D)

- TEST USAGE
  - 2 LIVES OF 8000 HOURS EACH
  - 5840 FLIGHTS PER LIFE
  - TEST SPECTRUM CONSISTS OF A 500 HOUR BLOCK OF RANDOMLY ORDERED FLIGHTS REPEATED AS NECESSARY
  - AVERAGE CYCLING RATE:
    - 30 LOAD PTS/MIN
    - 168 LOAD PTS/FLIGHT HOUR
    - 10 FLIGHT HOURS/CLOCK HOUR

TEST STATUS

- TEST WAS STOPPED AT 3988 HOURS DUE TO A LEADING EDGE FLAP FAILURE IN AUG 88
- THE HALF-LIFE MAJOR INSPECTION WAS CONDUCTED AT THIS TIME
- 4 ADDITIONAL CRACKED AREAS WERE FOUND
- REPAIRS ARE BEING INSTALLED
- ESTIMATED RESTART DATE IS JAN 89

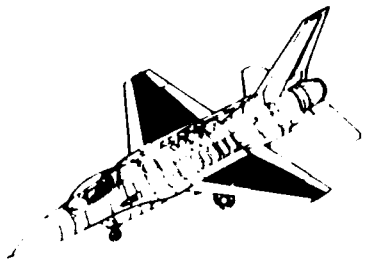
TEST RESULTS TO DATE

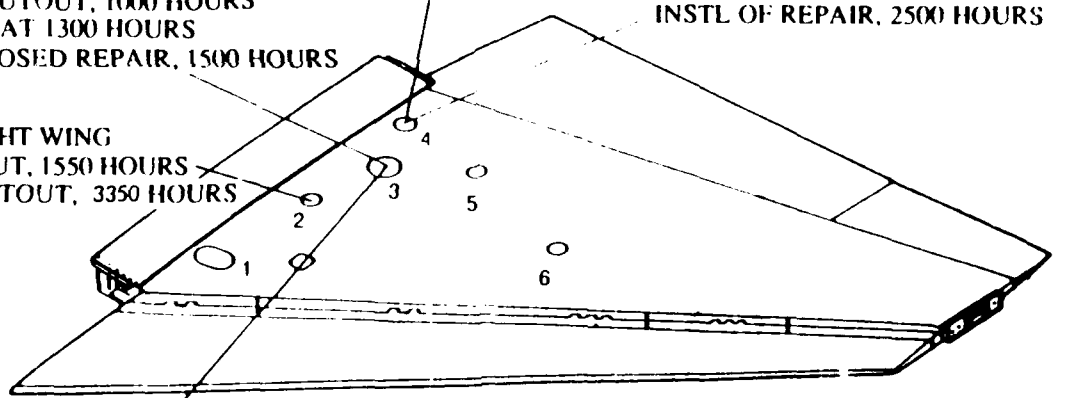
- CRACKS IN UPPER WING SKIN ACCESS HOLES
- LEADING EDGE FLAP FAILURES
- FORWARD FUSELAGE LONGERON FAILURES
- CRACKS IN BULKHEAD AT F.S. 341.8
- CRACKS IN BULKHEAD AT F.S. 357.8
- CRACKS IN BULKHEAD AT F.S. 446.1

UPPER WING SKIN

ACCESS HOLE CRACKS

SUMMARY OF RESULTS





CRACKING, RIGHT WING
TOWARD CUTOUT, 2500 HOURS

CRACKING, LEFT WING
HOLE TOWARD CUTOUT, 600 HOURS
AWAY FROM CUTOUT, 1000 HOURS
STOP-DRILLED AT 1300 HOURS
INSTL OF PROPOSED REPAIR, 1500 HOURS

CRACKING, LEFT WING
HOLE TOWARD CUTOUT, 1000 HOURS
AWAY FROM CUTOUT, 2300 HOURS
INSTL OF REPAIR, 2500 HOURS

CRACKING, RIGHT WING
TOWARD CUTOUT, 1550 HOURS
AWAY FROM CUTOUT, 3350 HOURS

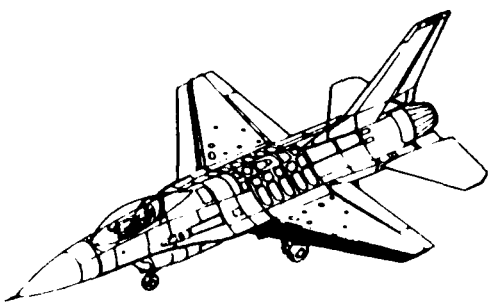
CRACKING, RIGHT WING
HOLE TOWARD CUTOUT, 1150 HOURS
AWAY FROM CUTOUT, 2700 HOURS
INSTL OF REPAIR, 2968 HOURS

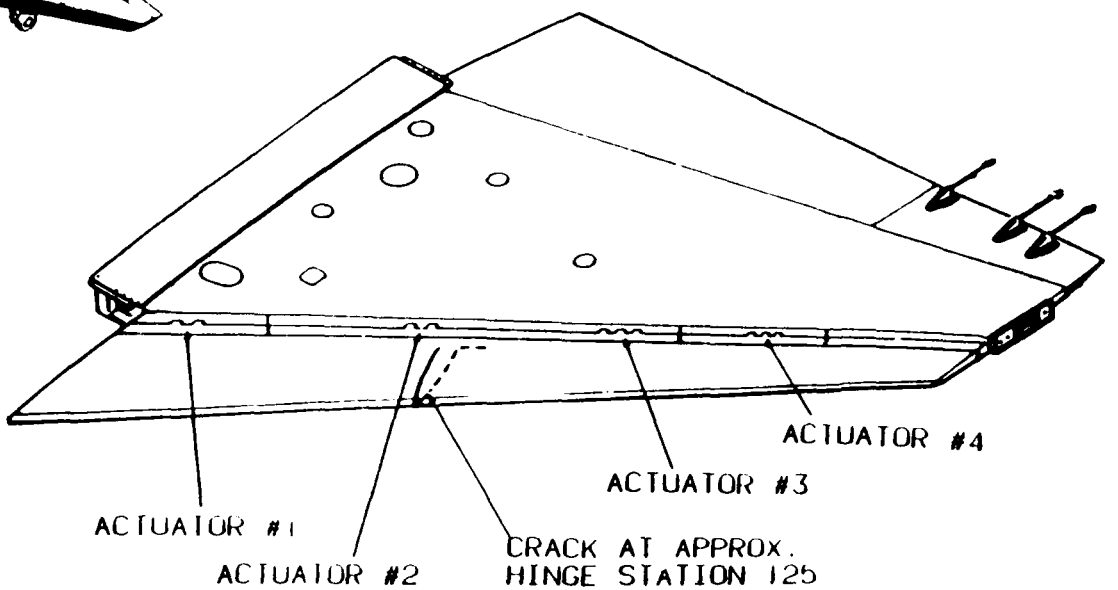
LEADING EDGE FLAP

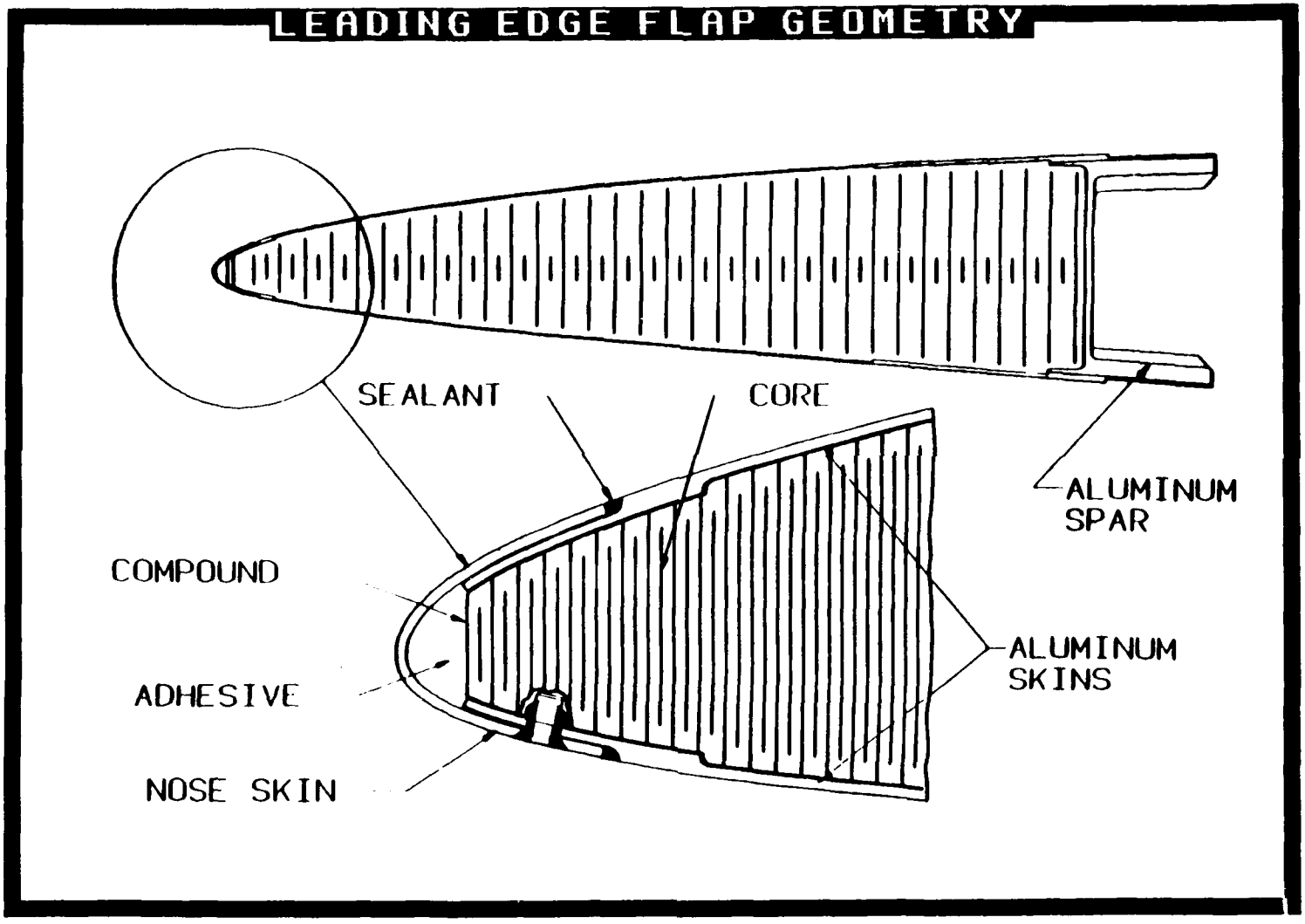
(LEF)

FAILURES

FAILURE LOCATIONS






LEADING EDGE FLAP GEOMETRY
SEALANT
CORE
ALUMINUM SPAR
COMPOUND
ADHESIVE
NOSE SKIN
ALUMINUM SKINS

HISTORY OF LEF FAILURES

DATE	SIDE	FLT HRS	△FLT HRS	ACTION
OCT 87	LEFT	704	704	REPLACED
JAN 88	RIGHT	1200	1200	CRACK CUT-OUT
APR 88	RIGHT	1800	1800	CRACK CUT-OUT
APR 88	LEFT	1848	1144	BOTH REPLACED
JUN 88	LEFT	2968	1120	REPAIRED
JUL 88	LEFT	3024	1176	REPAIRED
AUG 88	LEFT	3988	2140	BOTH REPLACED

FORWARD FUSELAGE

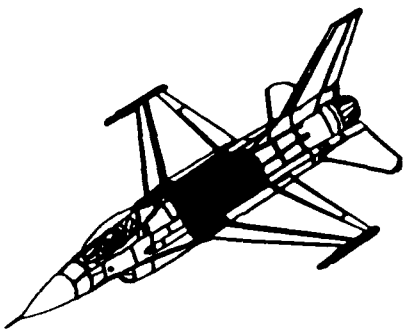
LONGERON FAILURES

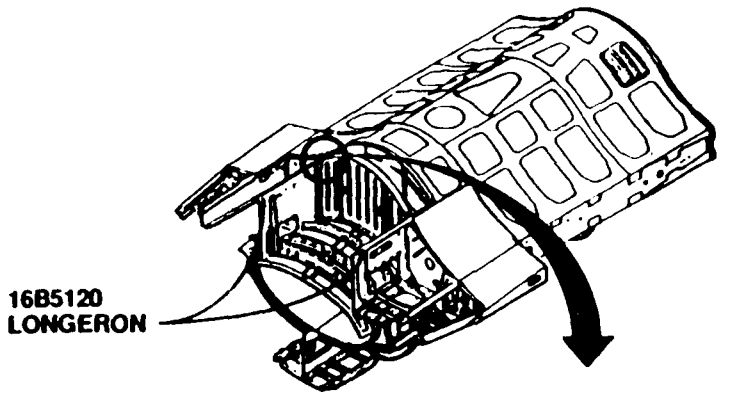
HISTORY

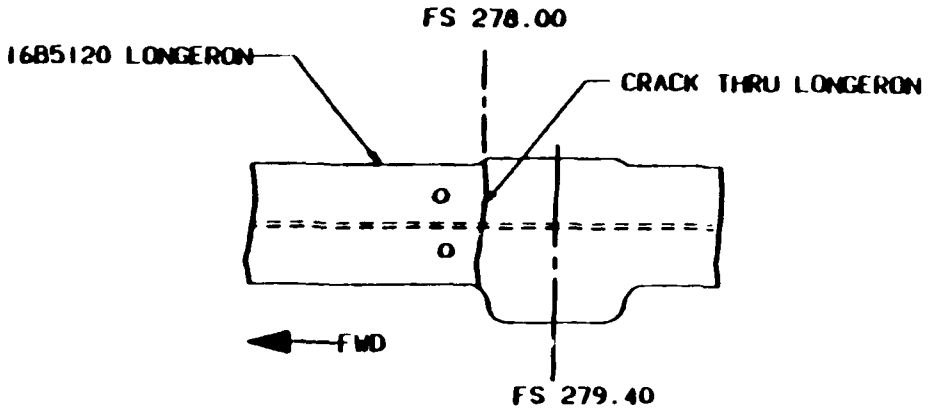
- R/H FAILURE AT F.S. 278 AT 2968 HOURS (JUL 88)
  - REPLACED WITH REDESIGNED PRODUCTION LONGERON

- L/H FAILURE AT F.S. 267.4 AT 3988 HOURS (AUG 88)
  - REDESIGNED LONGERON (DIFFERENT FROM ABOVE) WILL BE INSTALLED

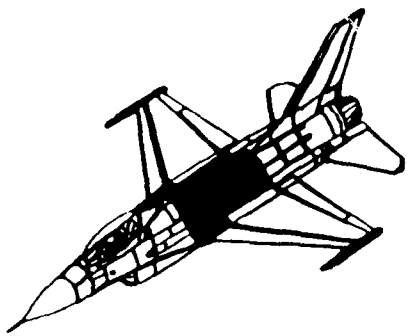
FAILURE LOCATION AT 2968 HOURS

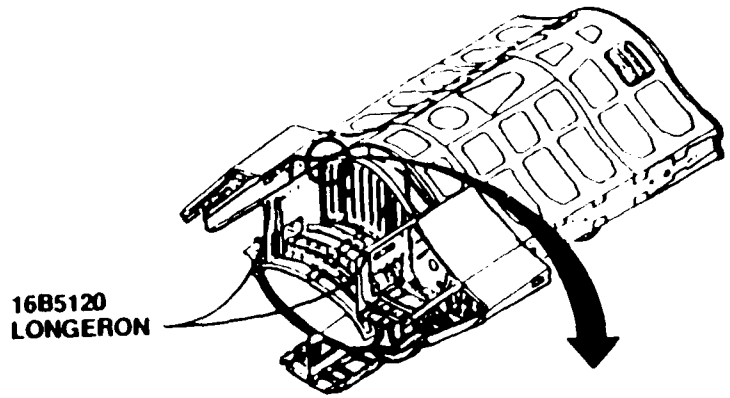


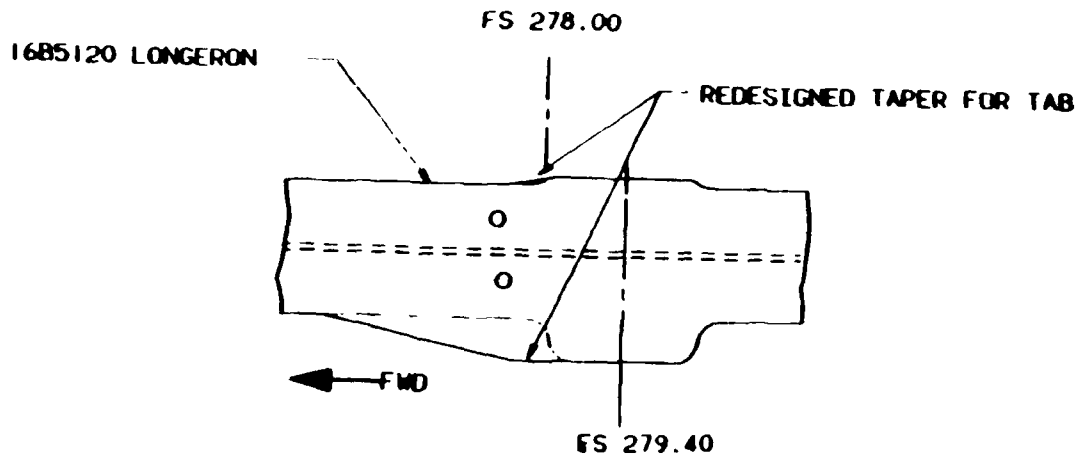




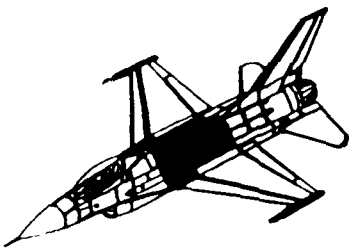
REPAIR AT 2968 HOURS

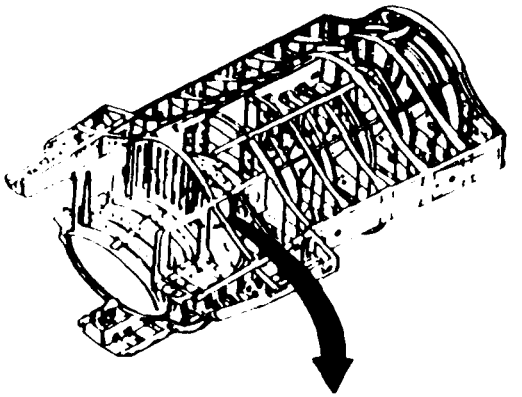


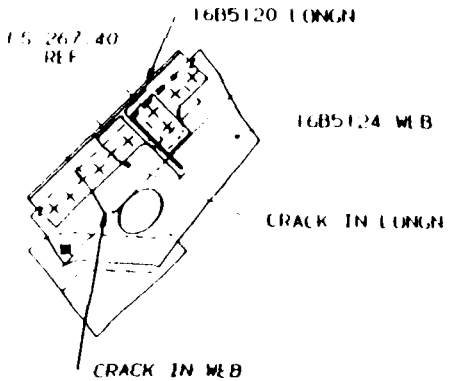




FAILURE LOCATION AT 3988 HOURS

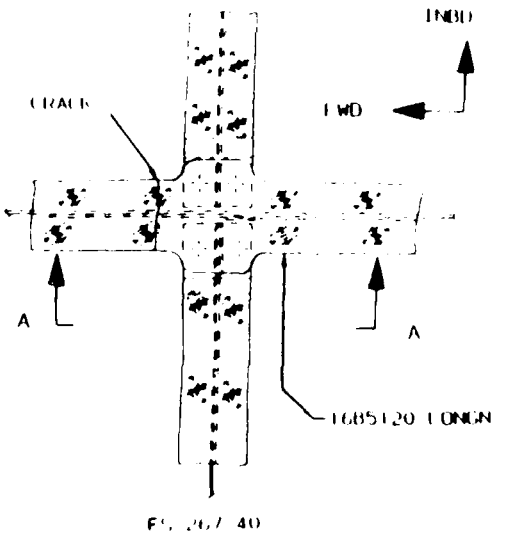




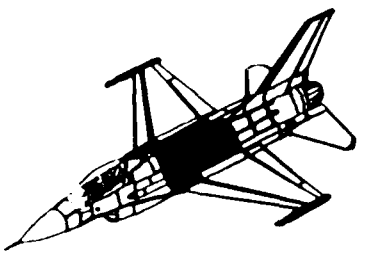


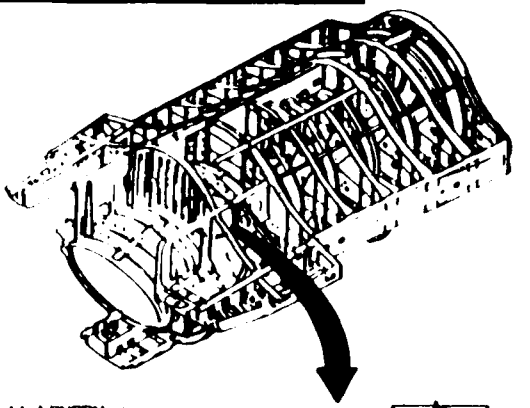
VIEW A A

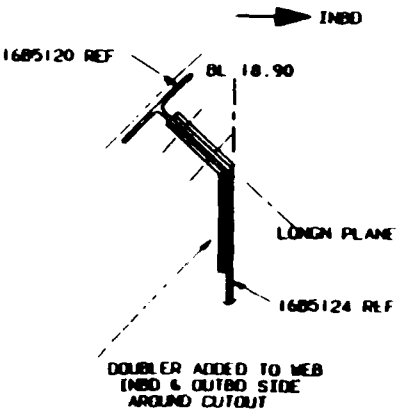
LRG INBD NORMAL TO LONGN PLANE

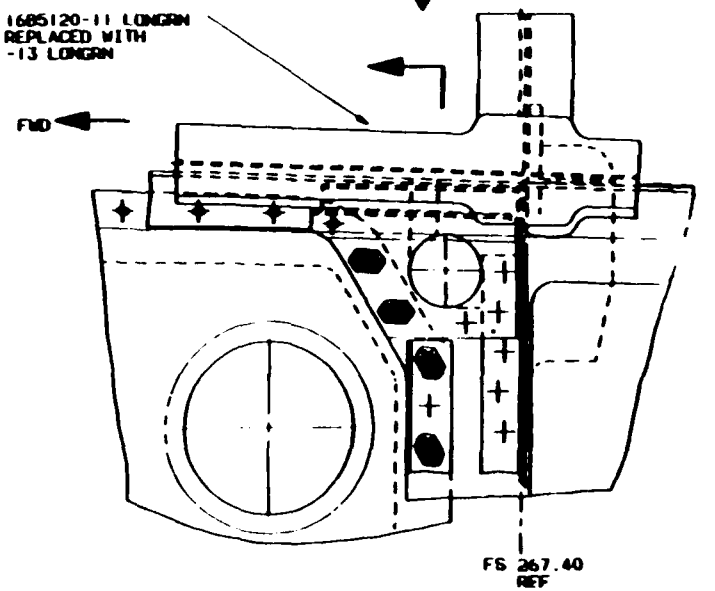


REPAIR AT 3988 HOURS



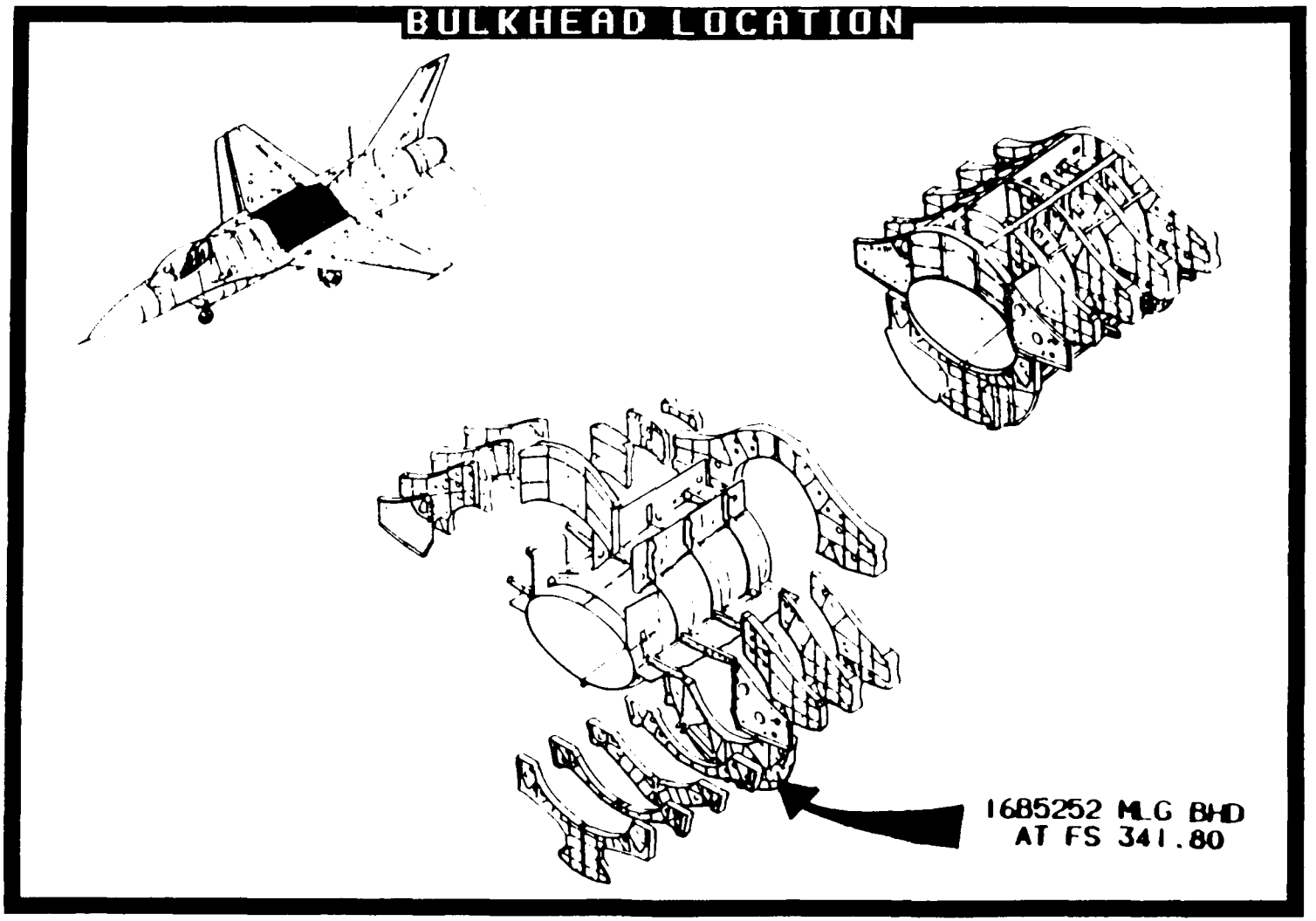




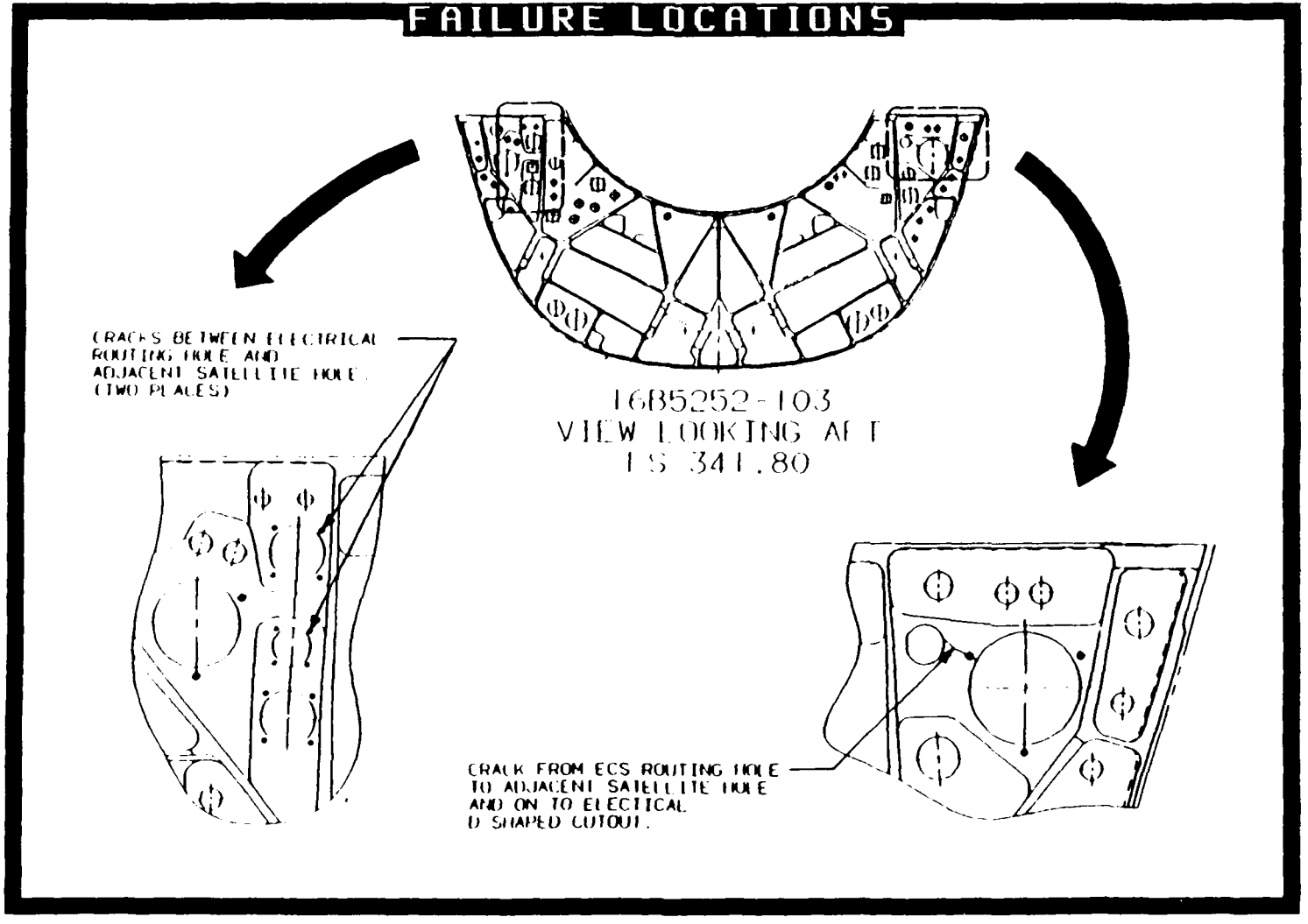


BULKHEAD AT F.S. 341.8

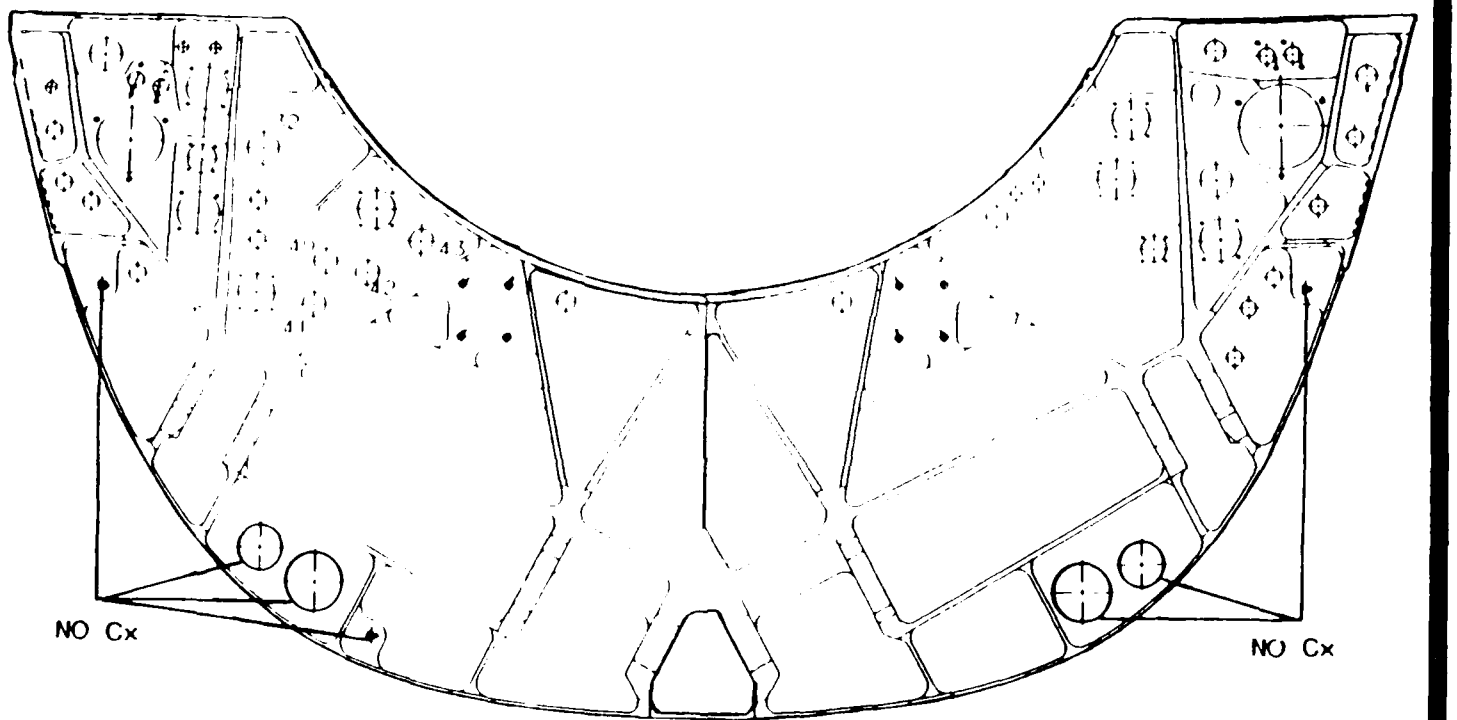
MAIN LANDING GEAR BULKHEAD


BULKHEAD LOCATION
1685252 MLG BHD
AT FS 341.80

FAILURE LOCATIONS

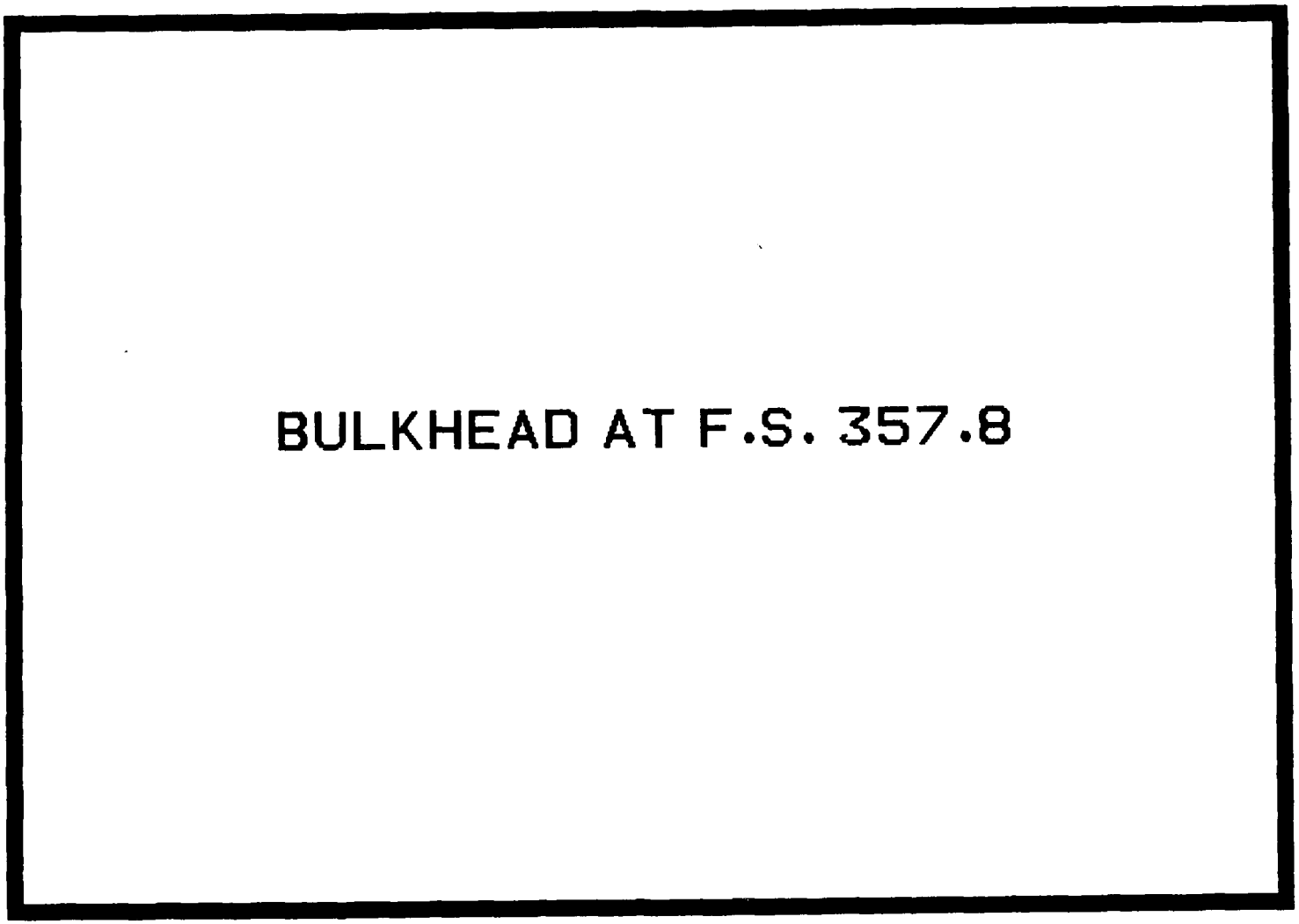


REPAIR CONFIGURATION

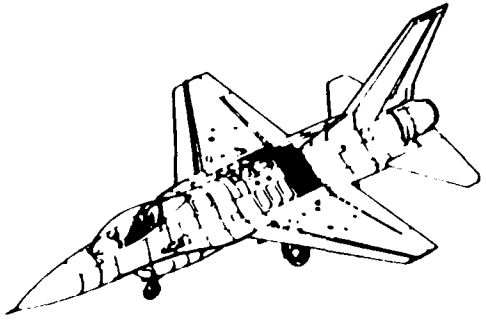


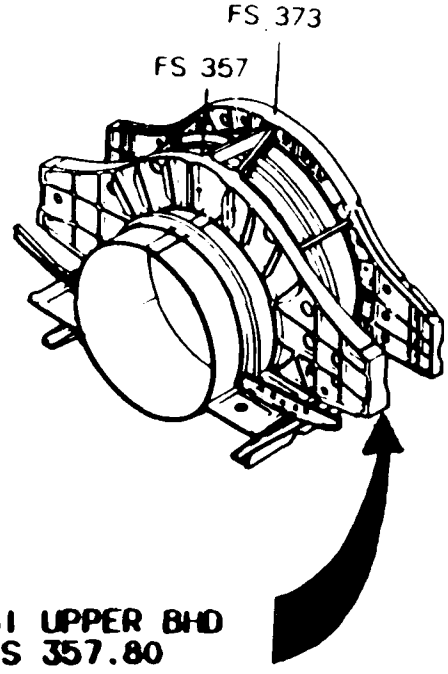
VIEW LKG AFT

ALL HOLES COLDWORKED (Cx) EXCEPT NOTED SEVEN

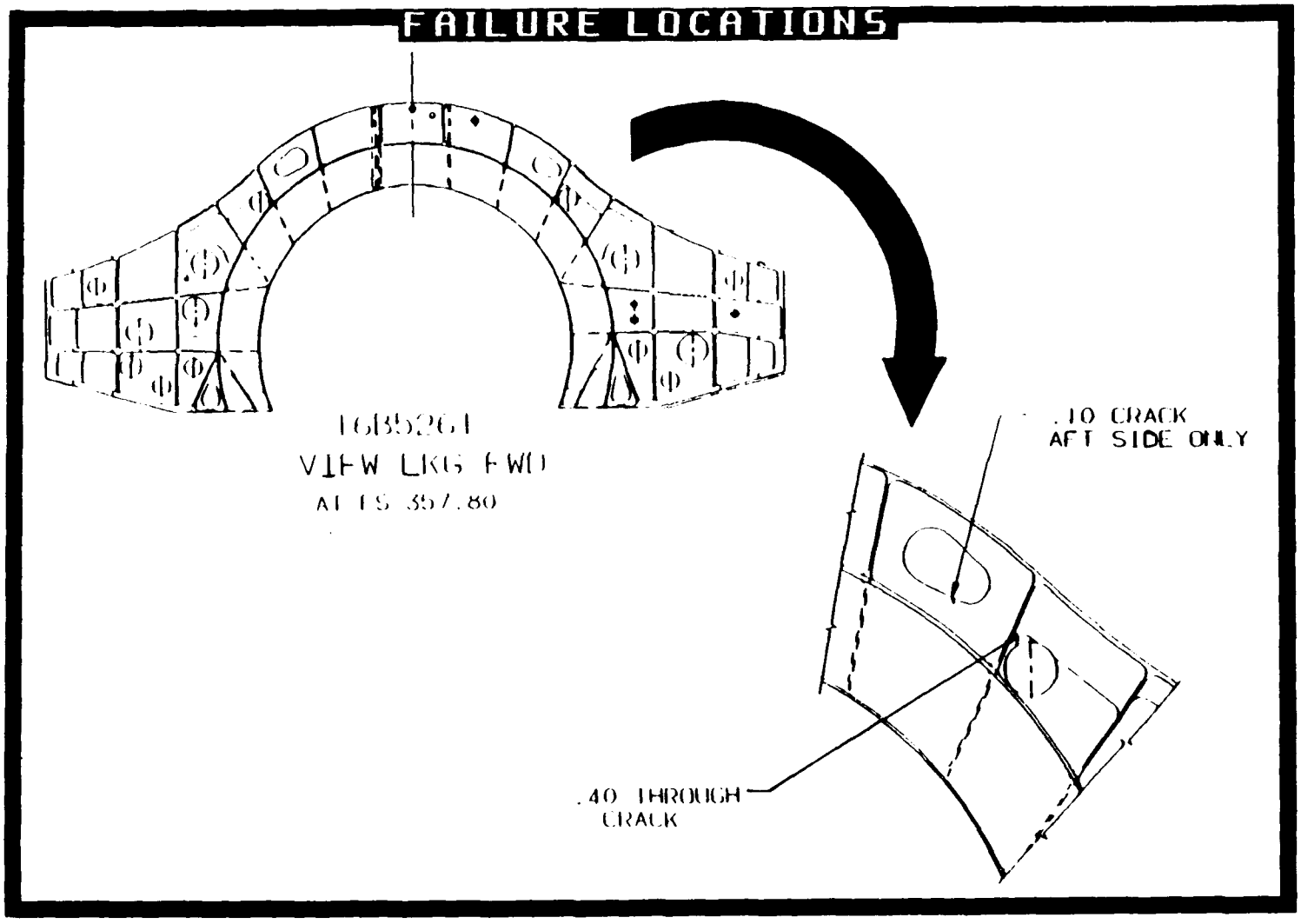

BULKHEAD AT F.S. 357.8

BULKHEAD LOCATION

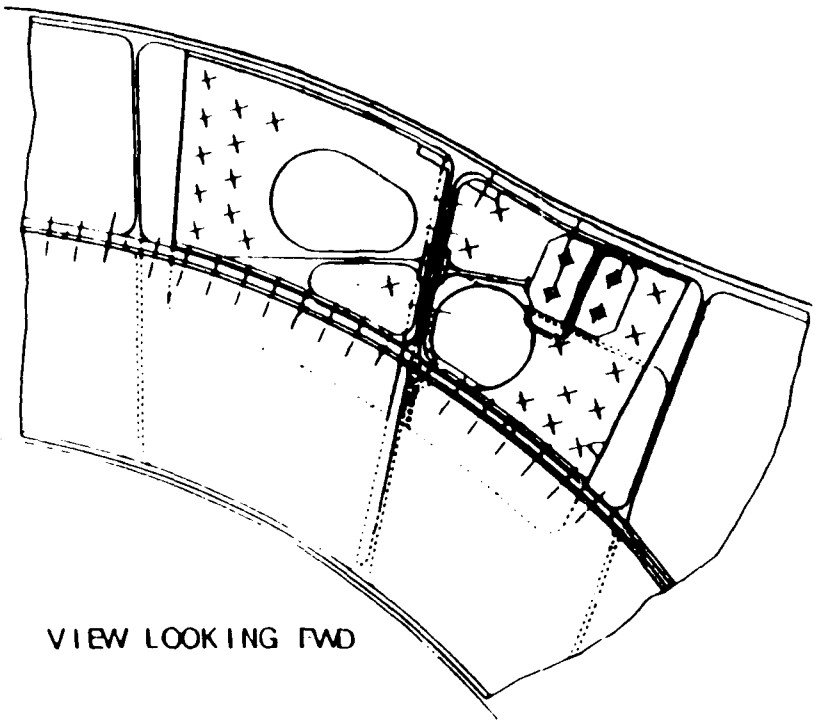




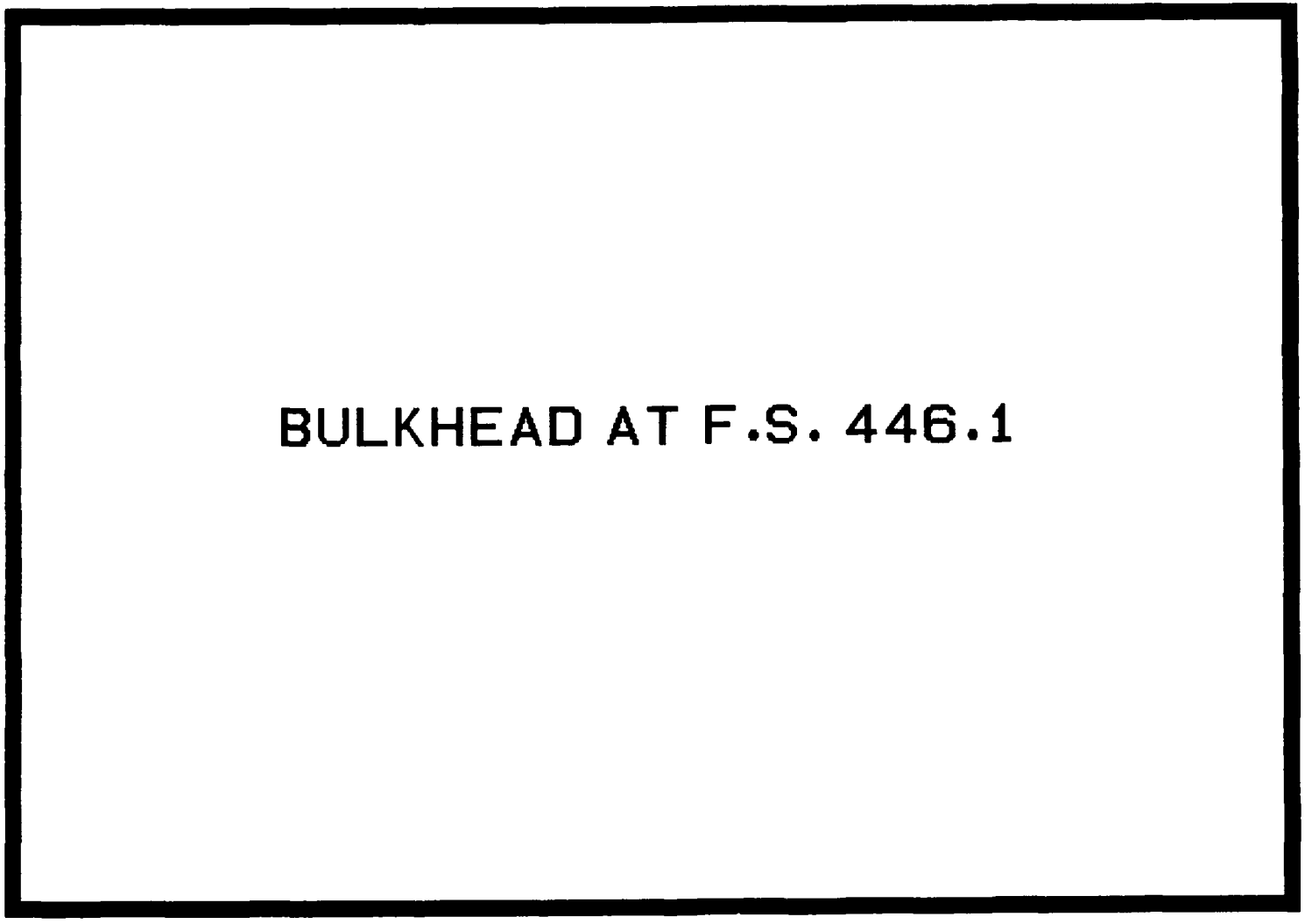
1685261 UPPER BHD
AT FS 357.80


FAILURE LOCATIONS
1G352G1
VIEW LKG FWD
AT FS 357.80
.10 CRACK
AFT SIDE ONLY
.40 THROUGH
CRACK

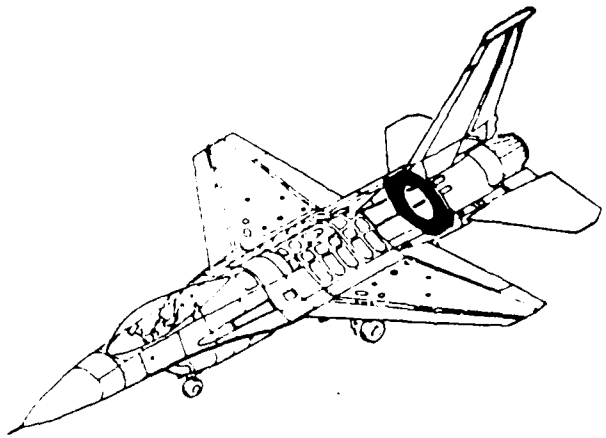
REPAIR CONFIGURATION

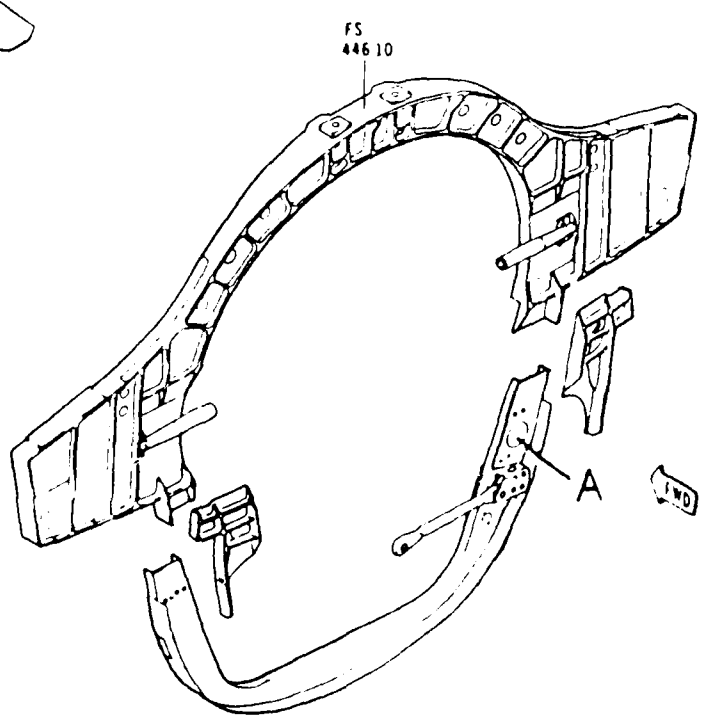


VIEW LOOKING FWD

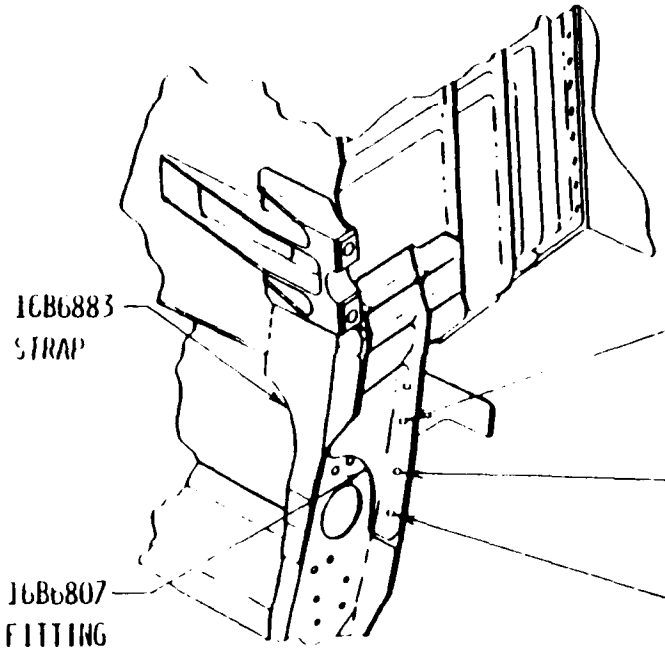

BULKHEAD AT F.S. 446.1

BULKHEAD LOCATION





CRACK LOCATIONS



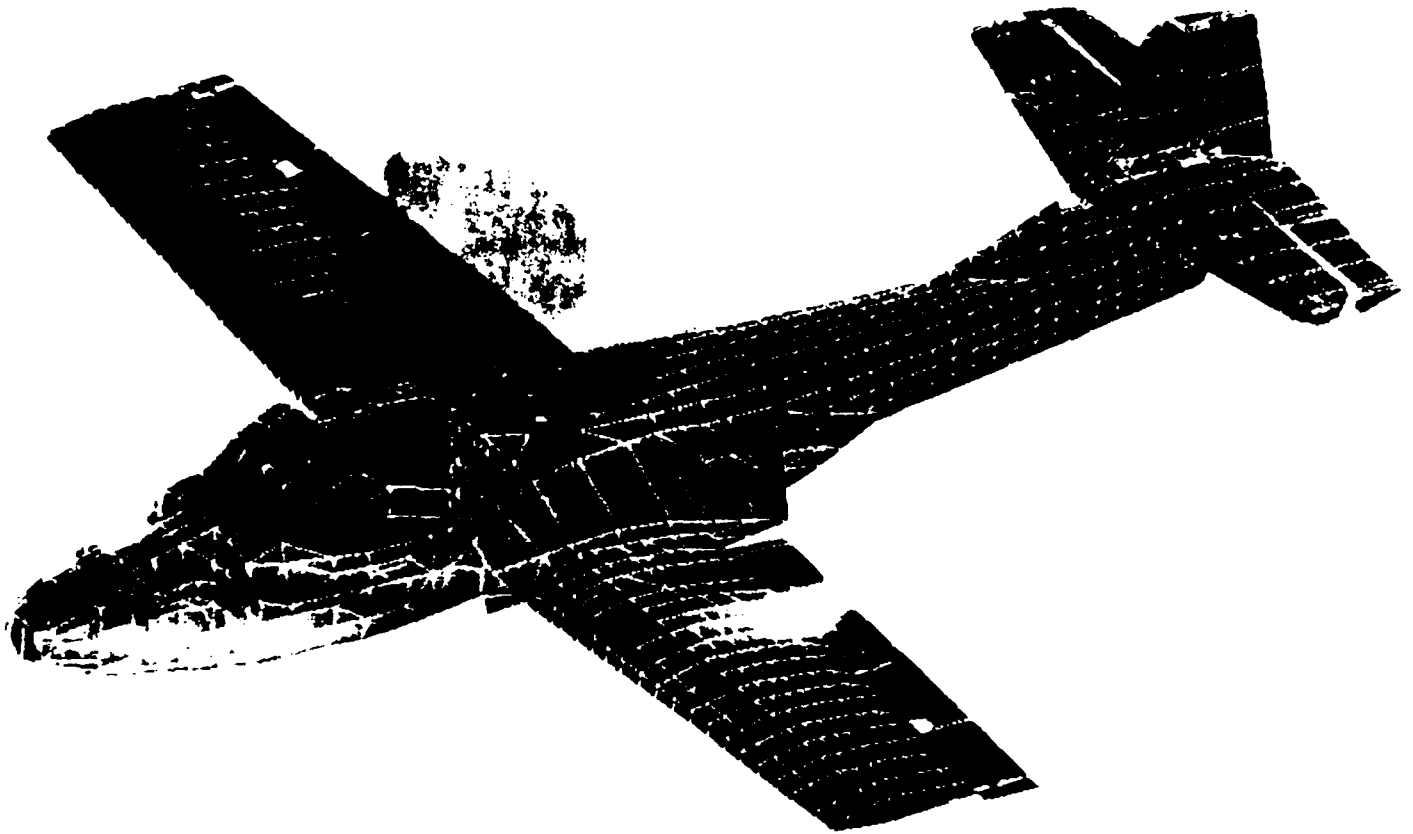
#2 TRIDAIR HOLE: CRACK INDICATION IN LONGERON SPLICE LH & RH SIDES. VISUAL CONFIRMATION OF CRACK LH SIDE ONLY.

#3 TRIDAIR HOLE: CRACK INDICATION BUT NO CONFIRMATION LH SIDE ONLY.

#4 TRIDAIR HOLE: CRACK INDICATION BUT NO CONFIRMATION RH SIDE ONLY.

AFLC

T-37B
DAMAGE TOLERANCE ANALYSIS



OVERVIEW

- DTA PROGRAM
  - — REQUIREMENT
  - — BACKGROUND
  - — PROGRAM
    - -- OBJECTIVE
    - -- DTA
    - -- RESULTS
  - — ACTION
  - — SLEP IMPACT

REQUIREMENT

- AFR 80-13, AIRCRAFT STRUCTURAL INTEGRITY PROGRAM

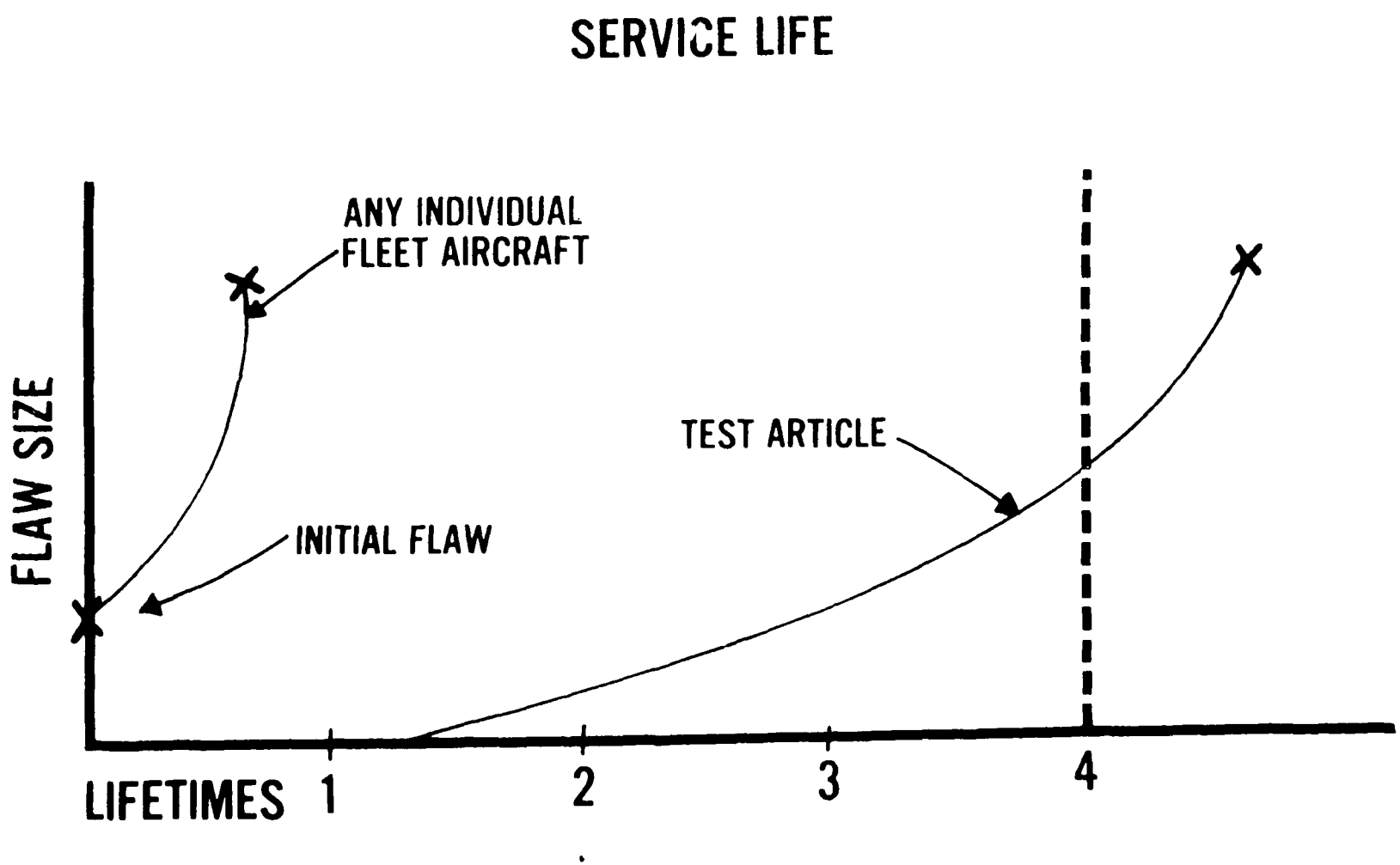
  - ESTABLISH, EVALUATE AND SUBSTANTIATE AIRCRAFT STRUCTURAL INTEGRITY FOR DAMAGE TOLERANCE, DURABILITY, ECONOMIC LIFE ...

  - MIL-A-87221

BACKGROUND

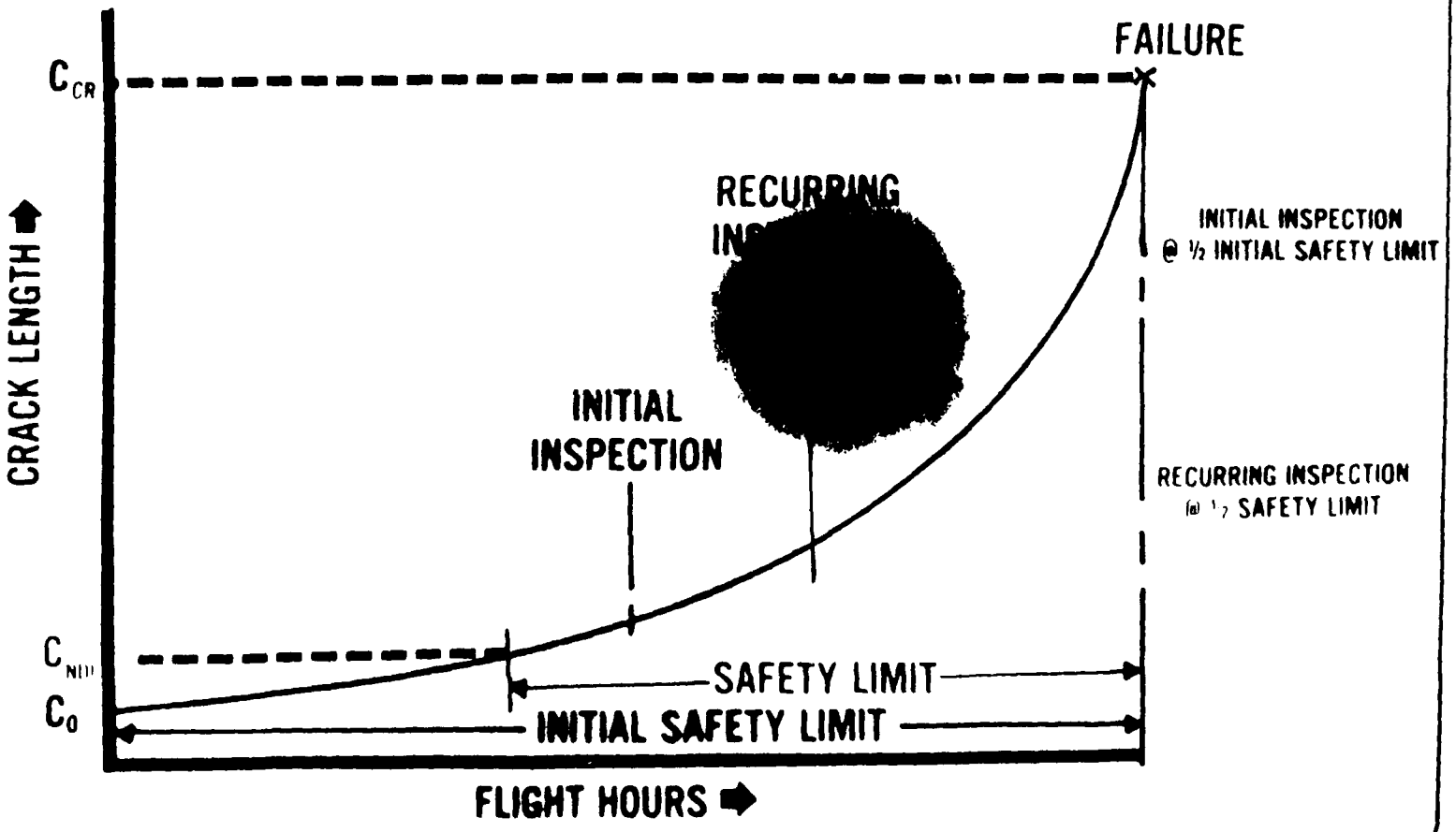
- SERVICE LIFE
- DAMAGE TOLERANCE ANALYSIS
  - SLOW CRACK GROWTH
    - -- SAFETY LIMITS
    - -- INSPECTION INTERVALS

SAFE LIFE METHODOLOGY



6 KBK

DAMAGE TOLERANCE ANALYSIS



BACKGROUND

- DEVELOPED IAW SERVICE LIFE CONCEPT
  - 1950's DESIGN CRITERIA
  - ORIGINAL AIRFRAME LIFE - 8000 HRS
  - TEST AND MODIFICATION
    - -- 15000 HRS
    - -- 18000 HRS

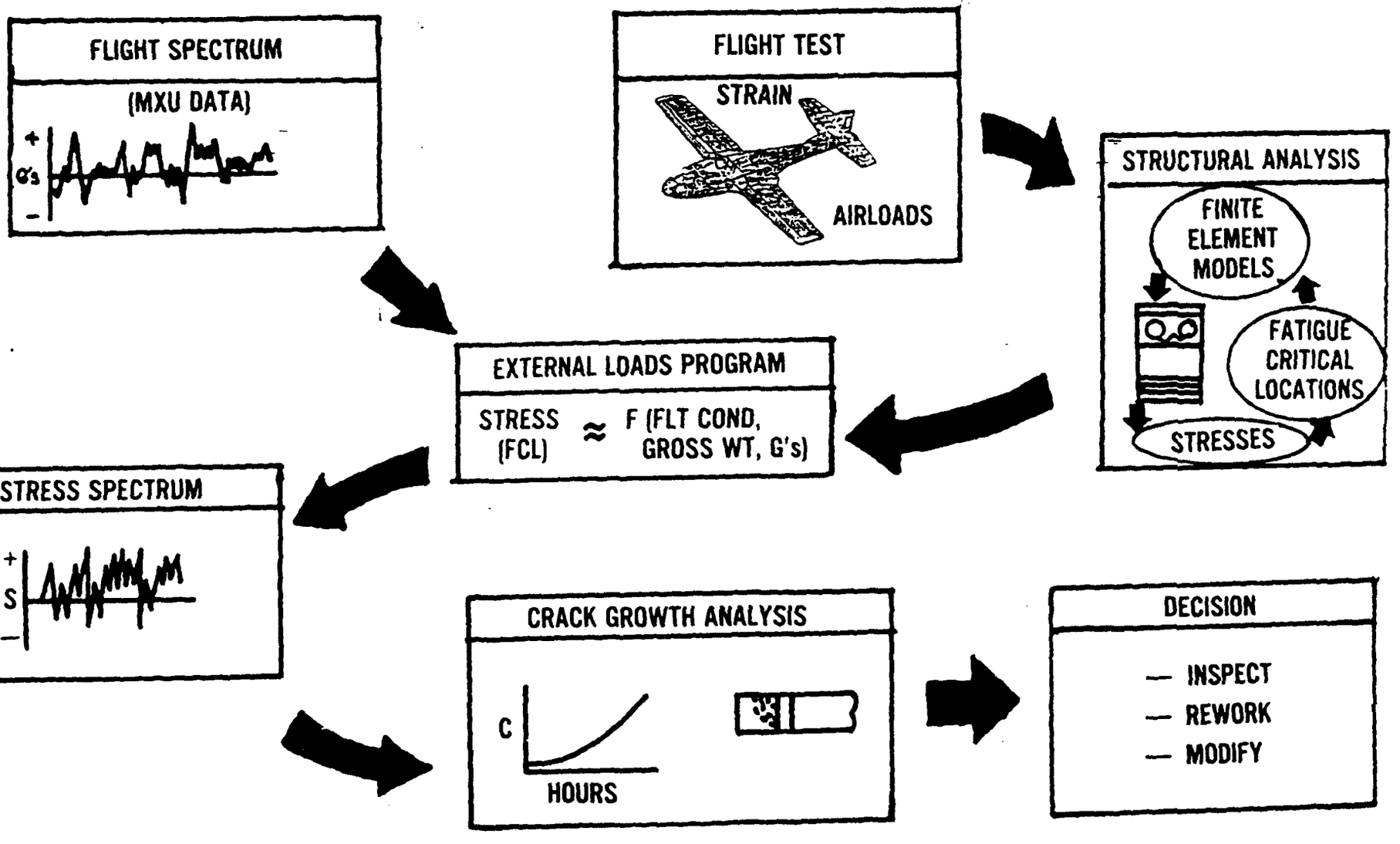
PROGRAM

- T-46 CANCELLATION
  - — SEP 85 TO MAY 86
- CONTRACT
  - — CESSNA
  - — APR 86 TO MAR 88
- FINAL REVIEW

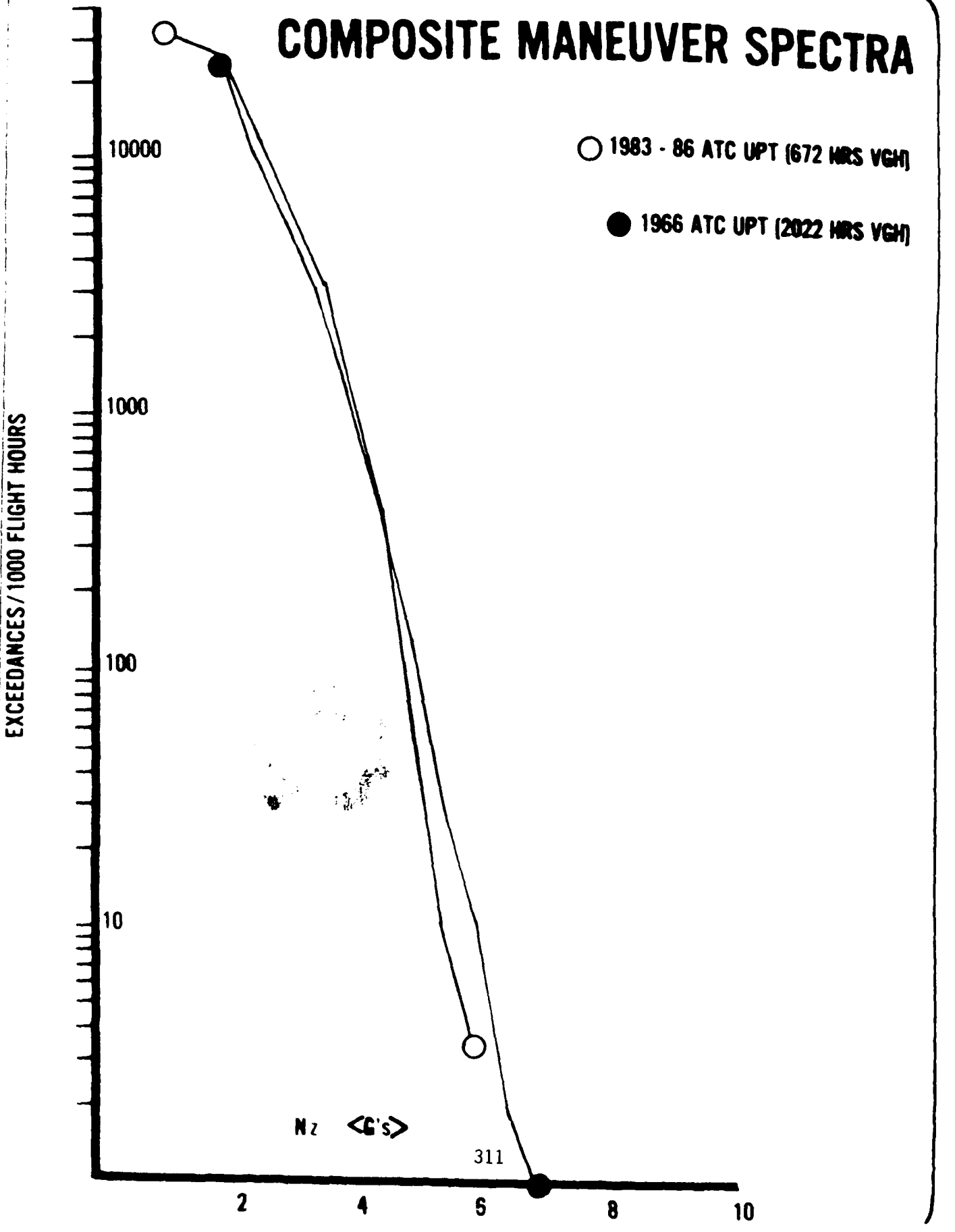
PROGRAM

- OBJECTIVES
  - IDENTIFY FATIGUE CRITICAL LOCATIONS
  - DETERMINE INSPECTION INTERVALS

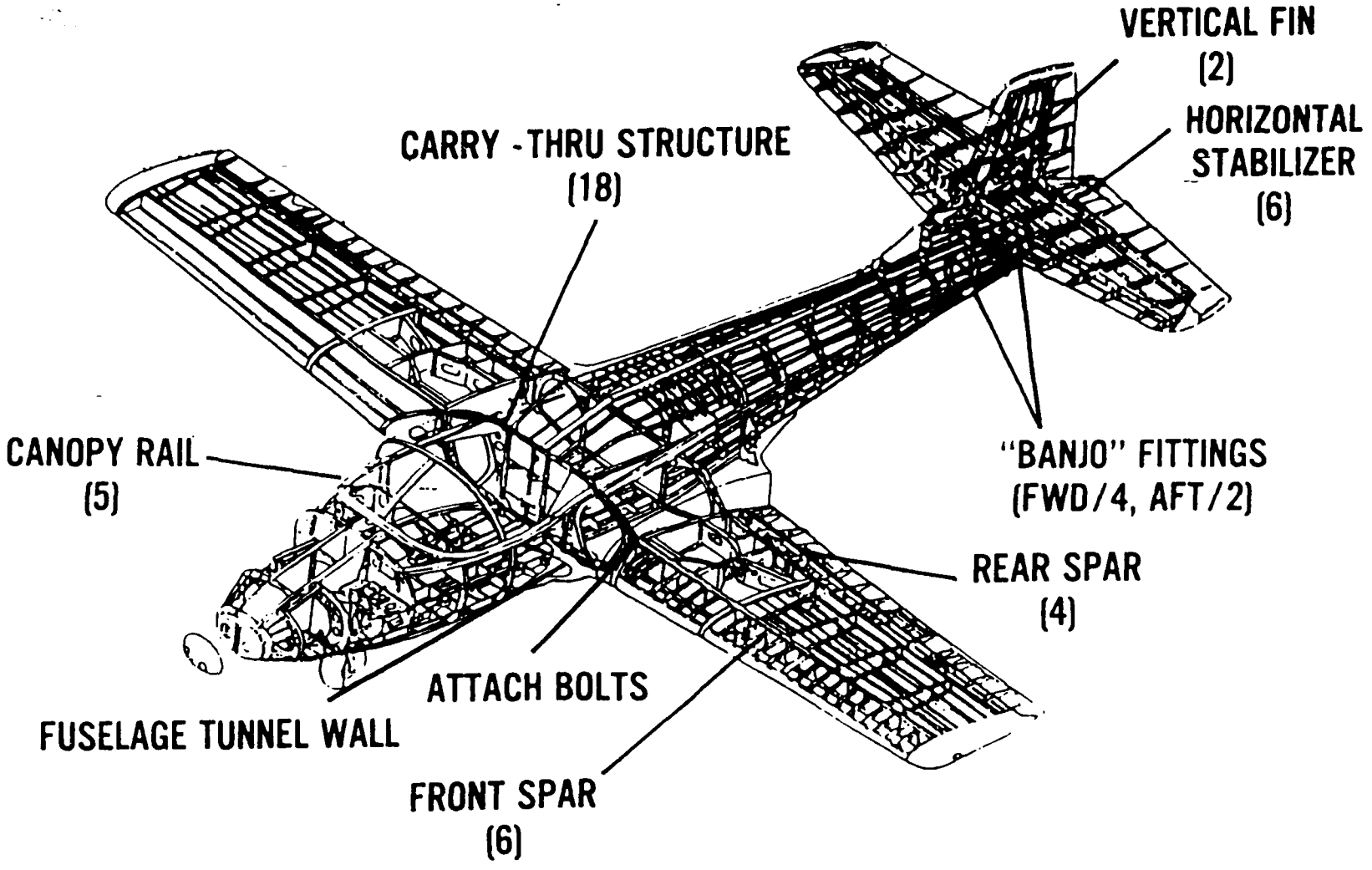
DAMAGE TOLERANCE ANALYSIS



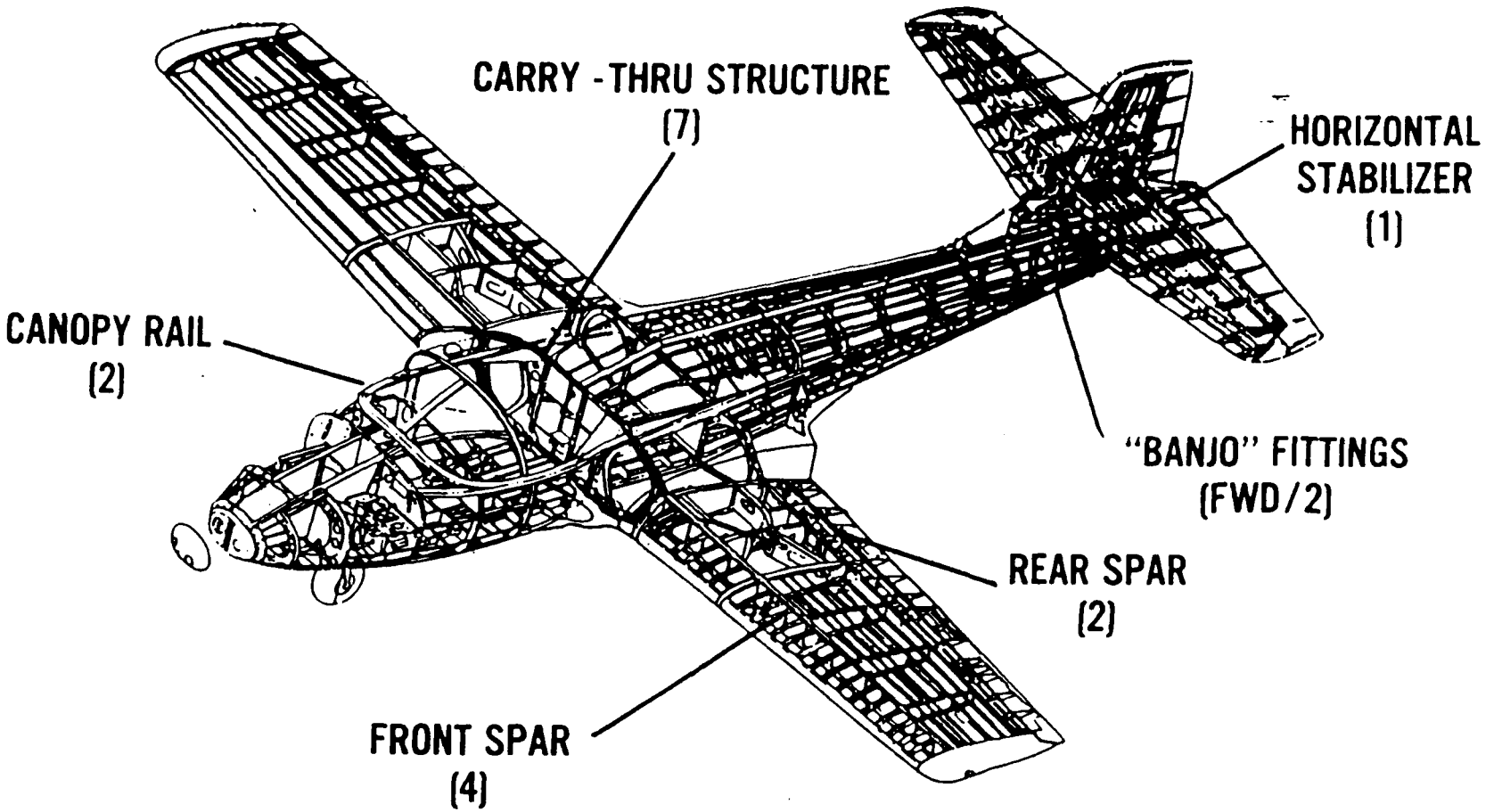
COMPOSITE MANEUVER SPECTRA



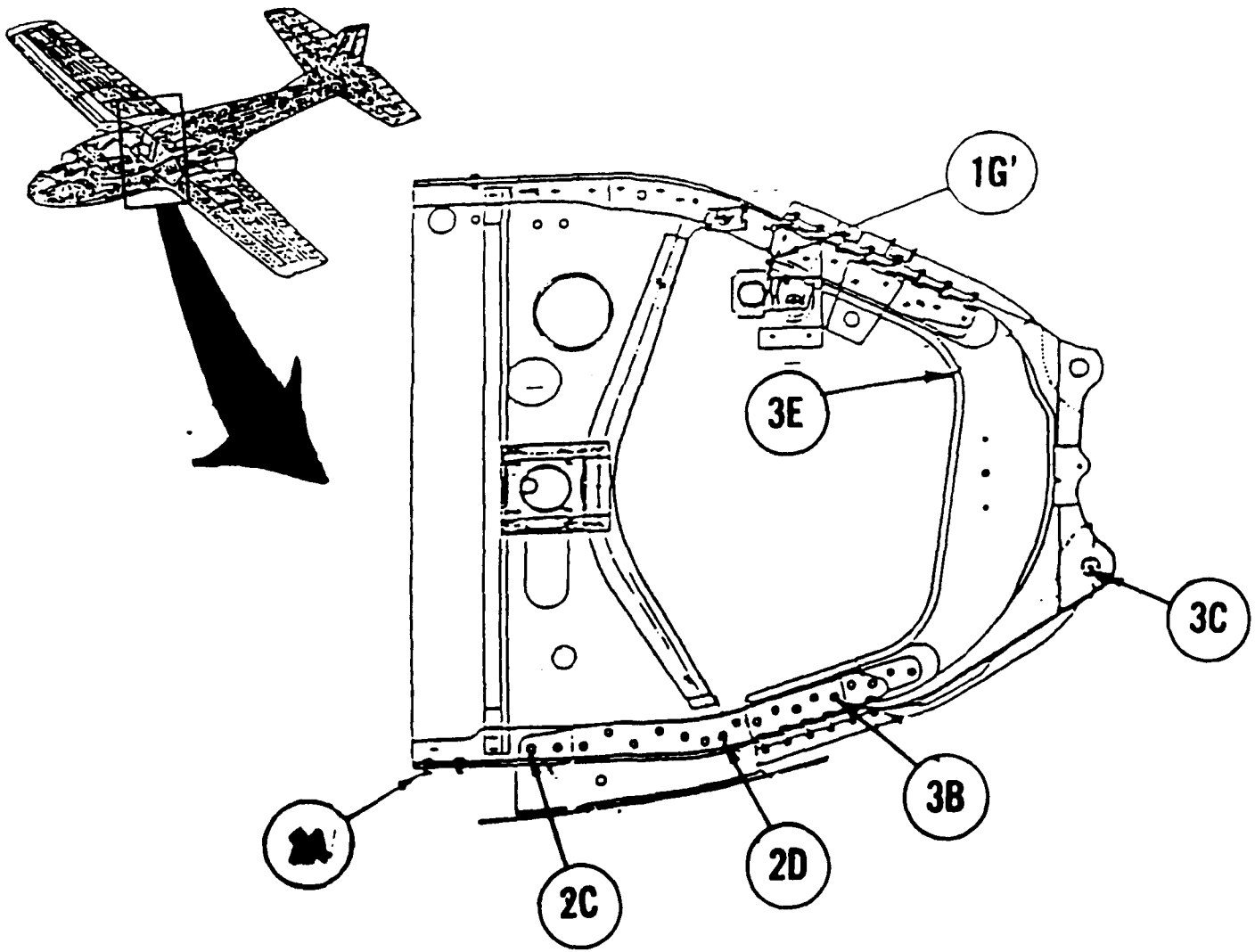
CANDIDATE AREAS



FATIGUE CRITICAL LOCATIONS



CARRY-THRU



RESULTS

- FORWARD CARRY-THRU

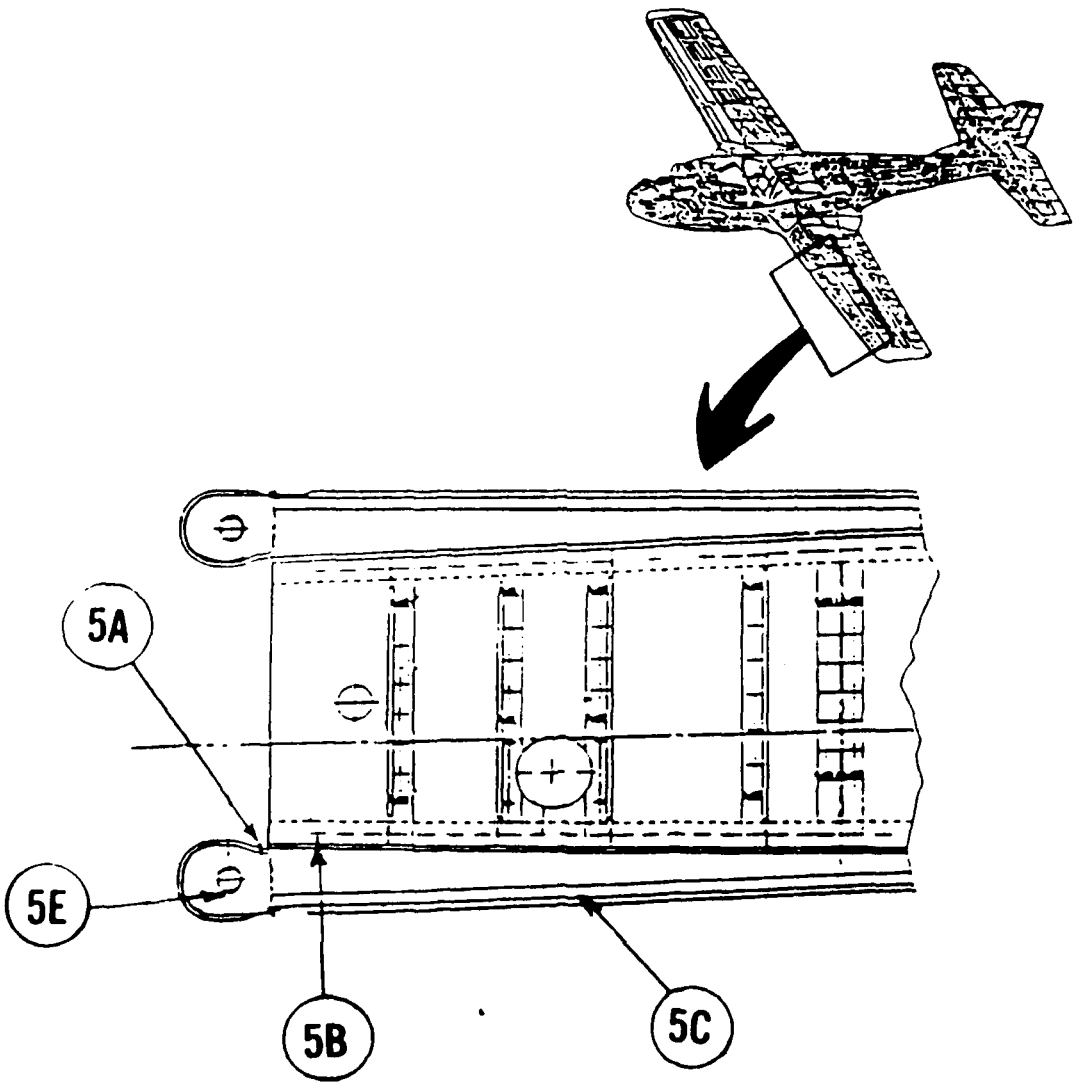
LOC	DESCRIPTION	CRITICAL CRACK LEN (IN)	INSPECT INTERVALS ..... HOURS ..... INITIAL	RECURR
1G'	UPPER CAP	.88	>30,000	>30,000
2A	LOWER CAP LONG ATTACH HOLE	.71	4,911	3,899
2C	LOWER CAP END FASTENER HOLE	1.24	11,218	7,898
2D	LWR CAP BOOMERANG ATTACH HOLE	.19	5,150	208

RESULTS

• FORWARD CARRY-THRU

LOC	DESCRIPTION	CRITICAL CRACK LEN (IN)	INSPECT INTERVALS -----(HOURS)----- INITIAL	RECURR
3B	FORG-LWR CAP	.26	965	703
3C	FORG-LWR LUG HOLE EDGE	.23	835	456
3C'	FORG-LWR LUG HOLE BORE	.15	437	190
*3E	FORG-UP CORNER	1.14	3,804	1,190

WING FRONT SPAR

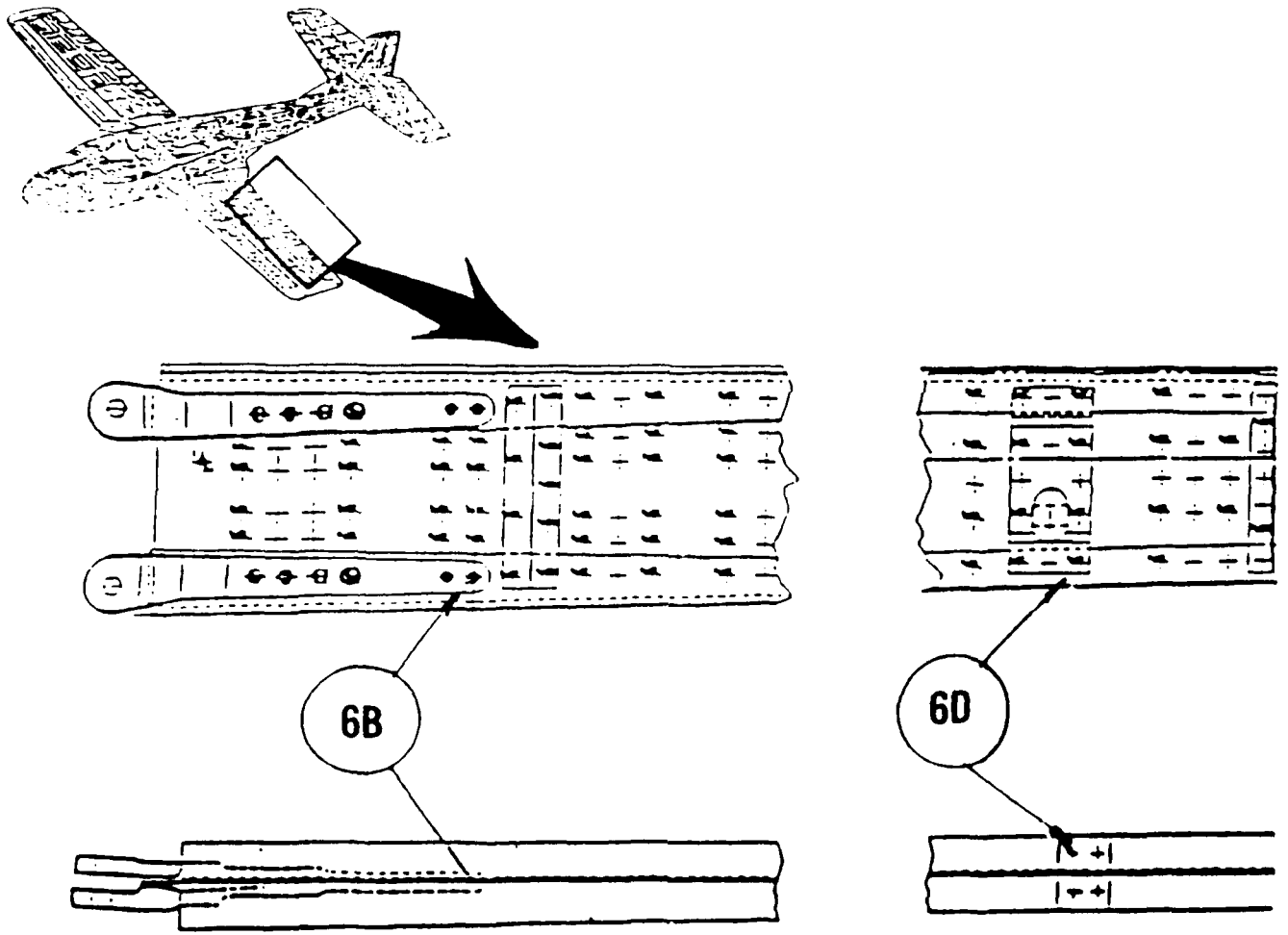


RESULTS

• WING

LOC	DESCRIPTION	CRITICAL CRACK LEN (IN)	INSPECT INTERVALS -----(HOURS)------ INITIAL	RECURR
- FRONT SPAR LOWER CAP				
5A	LUG FILLET	.16	1,076	348
*5B	VERTICAL FLANGE	4.07	16,075	9,467
5C	ATTACH HOLE	1.88	>30,000	29,127
*5E	LUG HOLE EDGE	.34	1,374	899
	(5E W/COLD WORK BENEFIT)		6,000	4,375

WING REAR SPAR

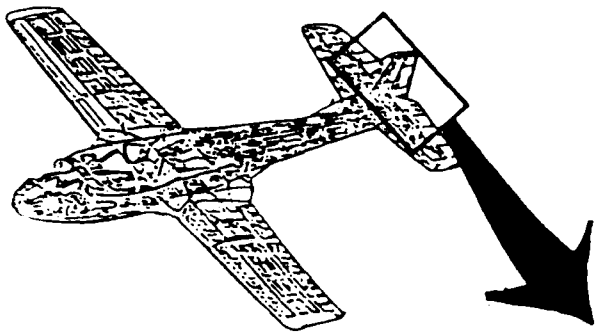


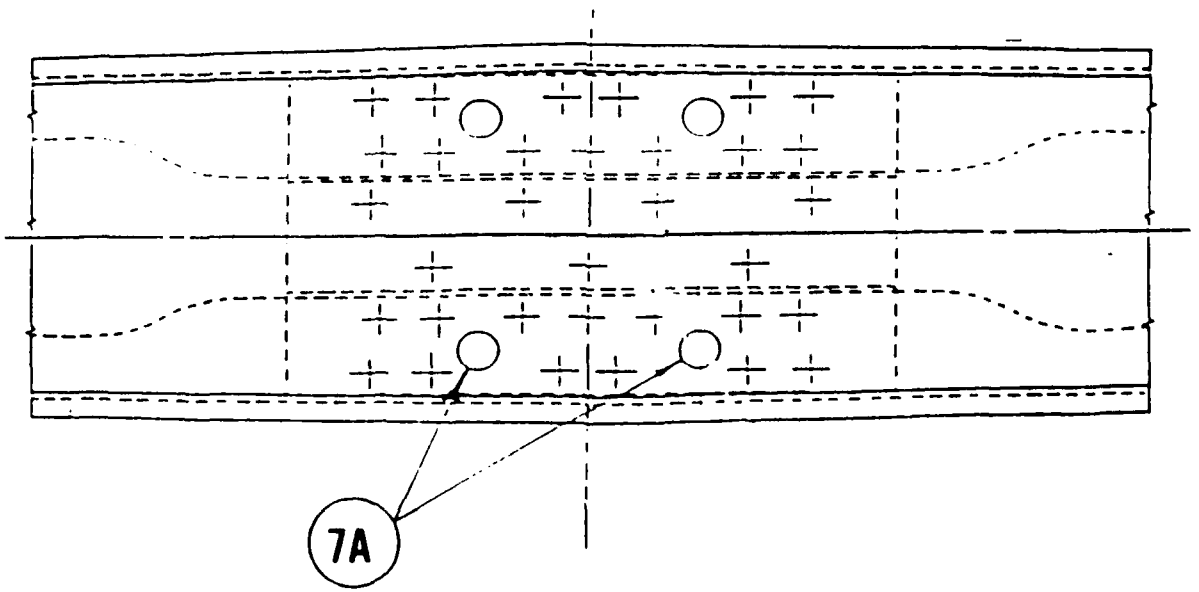
RESULTS

• WING

LOC	DESCRIPTION	CRITICAL CRACK LEN (IN)	INSPECT INTERVALS -----(HOURS)------ INITIAL	RECURR
- REAR SPAR LOWER CAP				
*6B	FITTING END TAPER-LOK HOLES	.53	4,333	2,404
6D	STRAP END FASTENER HOLE	1.01	6,256	3,300

HORIZONTAL STABILIZER

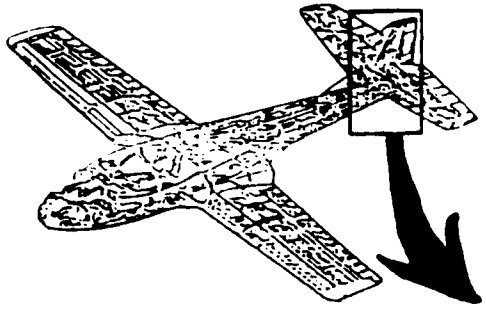


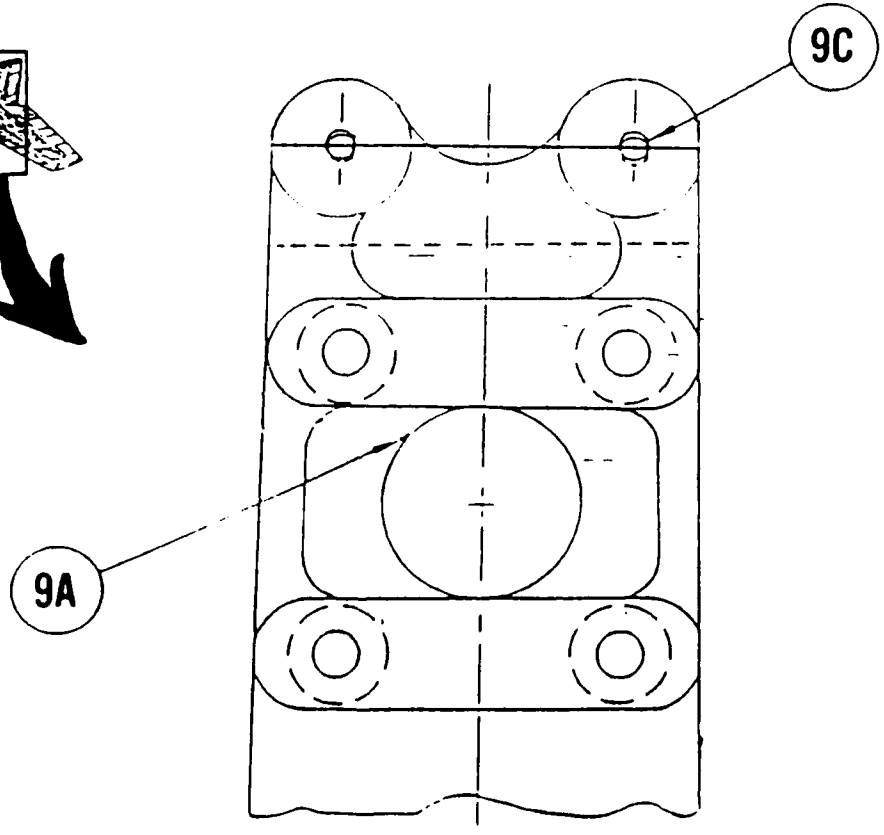


RESULTS

LOC	DESCRIPTION	CRITICAL CRACK LEN (IN)	INSPECT INTERVALS (HOURS) INITIAL	RECURR
- HORIZONTAL STABILIZER				
*7A	FRONT SPAR BANJO ATTACH HOLES	.48	1,087	766
- BANJO FITTING				
*9A	LIGHTENING HOLE	1.03	2,340	1,559
9C	LUG HOLE EDGE	.45	>30,000	>30,000

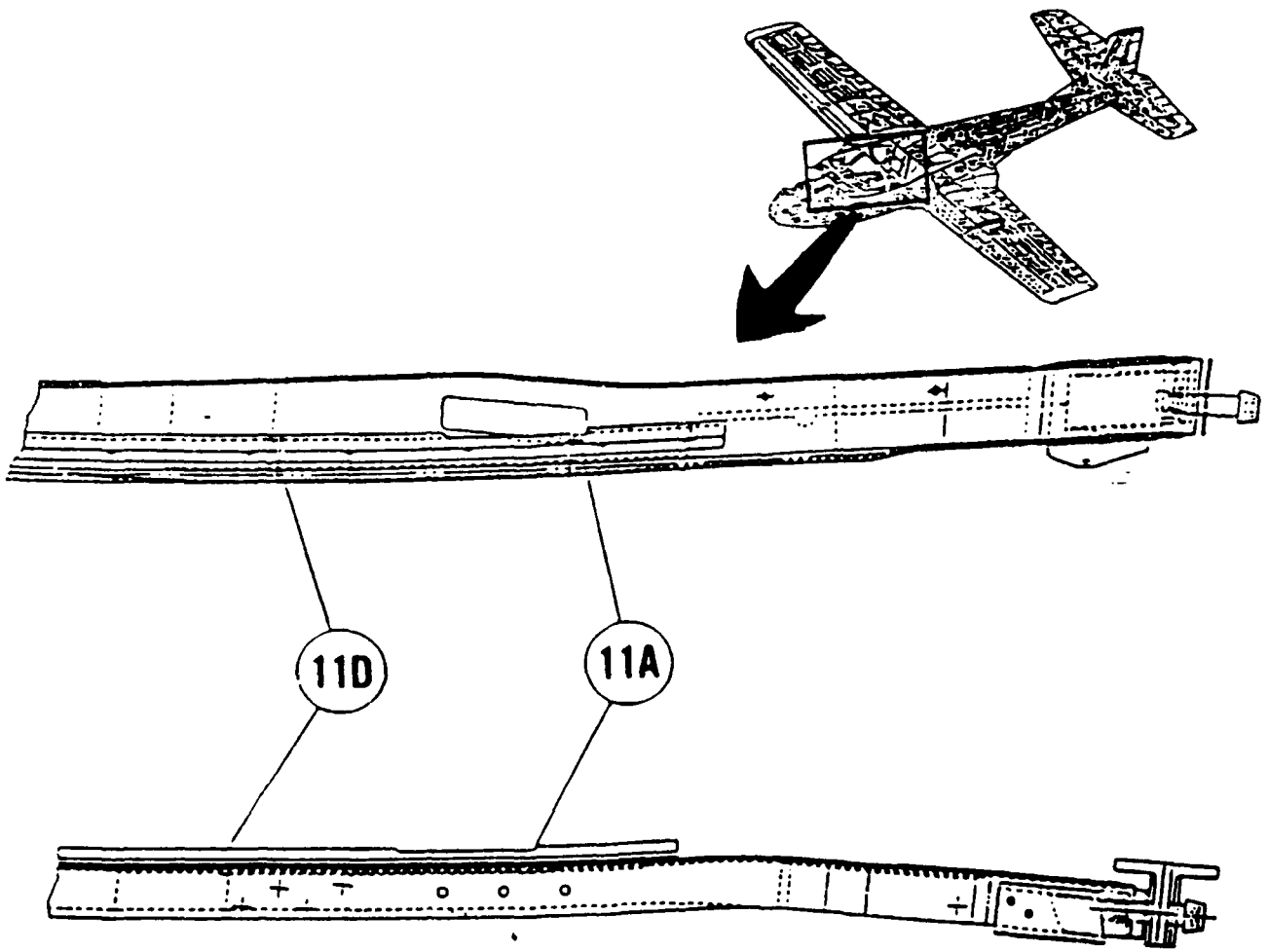
BANJO FITTINGS





VIEW LOOKING FWD

CANOPY RAILS



RESULTS

• CANOPY RAIL

LOC	DESCRIPTION	CRITICAL CRACK LEN (IN)	INSPECT INTERVALS -----(HOURS)------ INITIAL	RECURR
11A	AFT LATCH CUT-OUT	1.04	>30,000	26,352
11D	ATTACH HOLE	1.36	>30,000	15,186

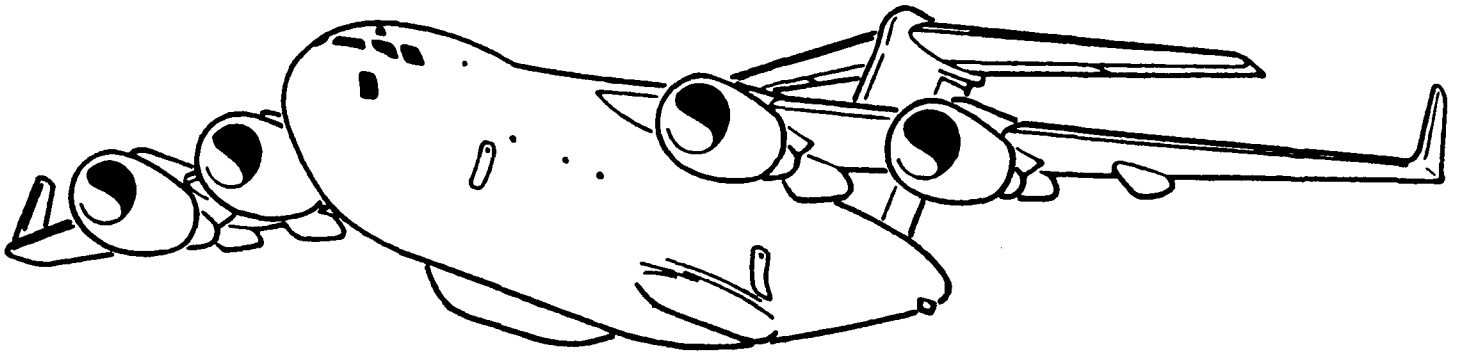
ACTION

- RISK ANALYSIS
  - — ASD (MAY - JULY 88)
  - — AREAS OF CONCERN
    - -- CARRY-THRU (FCL 3B)
    - -- WING FRONT SPAR (FCL 5A)

IMPACT

- SERVICE LIFE EXTENSION PROGRAM
  - — PROGRAM REDUCTION
    - -- COMPONENTS
  - — REVISED CRITERIA
    - -- REPLACE (2 COMP)
    - -- REPLACE-ON-CONDITION (OTHERS)
  - — TENTATIVE SCHEDULE
    - -- FY91

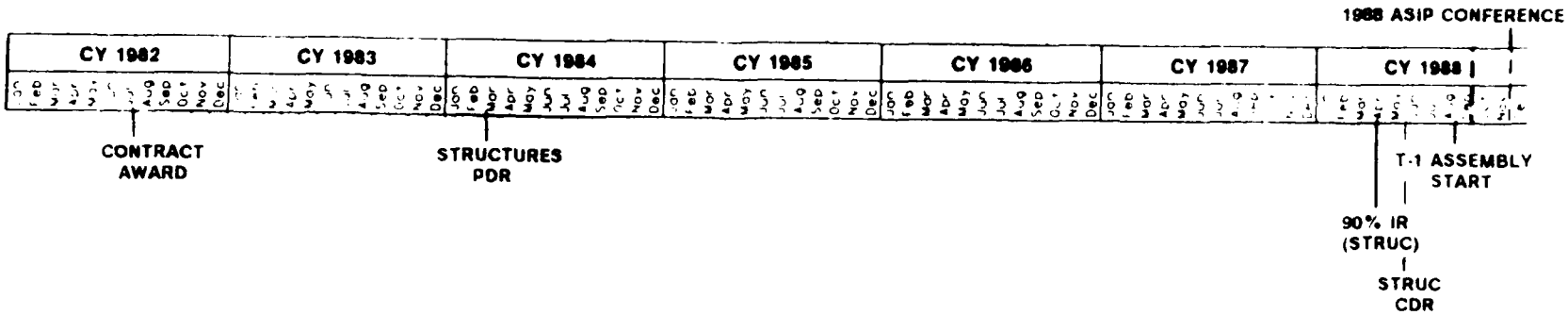
C-17
AIRCRAFT STRUCTURAL INTEGRITY OVERVIEW

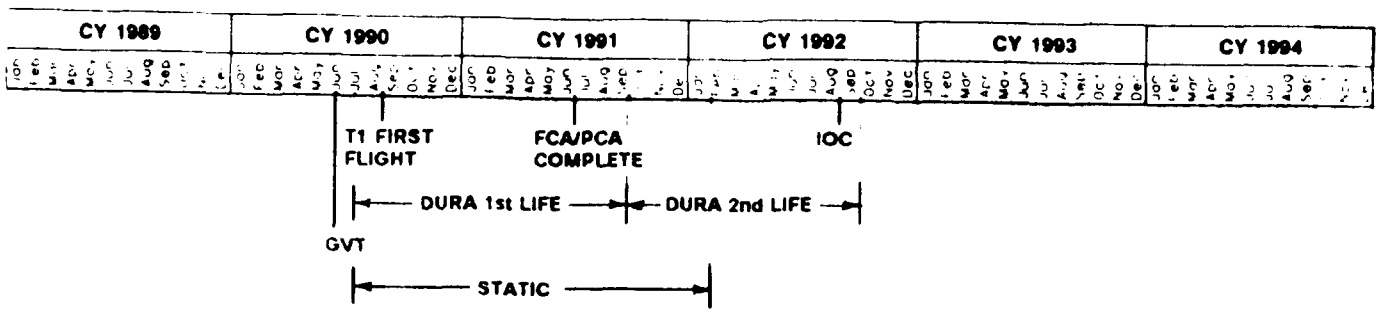


1988 ASIP CONFERENCE
NOV. 29 - DEC. 1
SAN ANTONIO, TEXAS

M. T. Bivins
R. G. Eastin
Douglas Aircraft Co.
Long Beach, Ca.

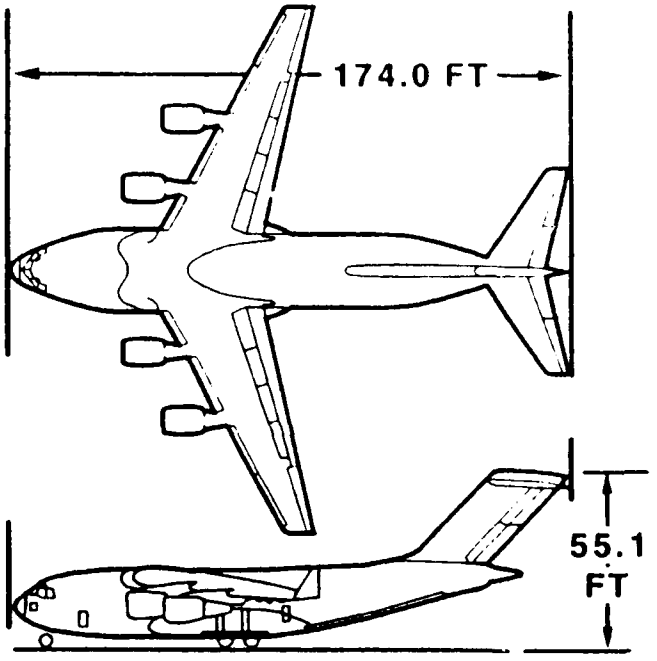
PROGRAM TIMELINE





USAF / MCDONNELL DOUGLAS C-17A

MAX T.O. GROSS WEIGHT	580,000 LB
MAX PAYLOAD	172,200 LB
MAX INTERNAL FUEL	176,000 LB
CRUISE SPEED	MACH 0.77
THRUST PER ENGINE	39,800 LB
AIRDROP	110,000 LB



CAPABILITIES

LONG RANGE	CARGO		LOADING DRIVE ON/OFF	SMALL AUSTERE AIRFIELD	AIRDROP & LAPES	AERIAL REFUELING	COMBAT OFF-LOAD
	OVERSIZE & BULK	OUTSIZE					

C222.01B

GENERAL REQUIREMENTS

- Structural Design Requirements are Derived in Accordance with MIL-STD-1530A and Related Specifications Together with Amplifications and Exceptions Contained in the Air Vehicle Specification
- Missions and Load Factors
  - Maximum Payload Mission: -0.25g, +2.25g
  - Heavy Logistics Mission: -0.25g, +2.25g
  - Intertheater Logistics Mission: -.5g, +2.5g
  - High Performance Logistics Mission: -1.0g, +3.0g
  - Takeoff or Landing Configuration: 0.0g, +2.0g
- Useful Life = 30 Years
- Service Life = 30000 Flight Hours

SIGNIFICANT MIL-SPEC EXCEPTIONS AND/OR AMPLIFICATIONS

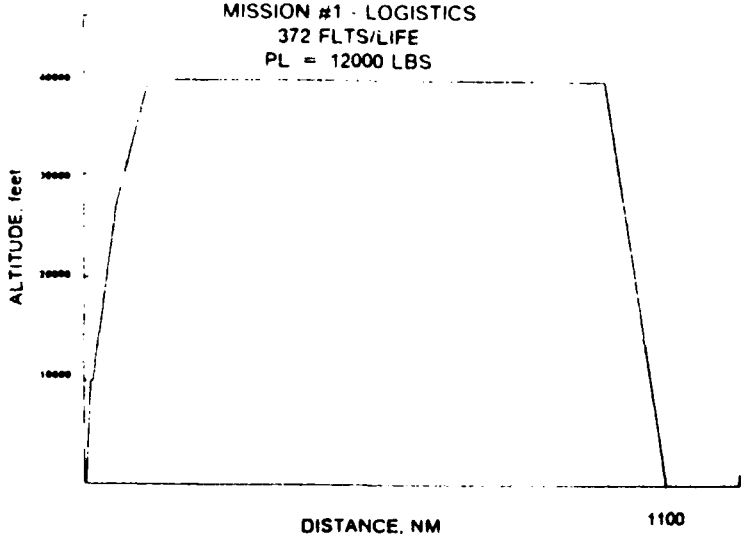
- MIL-STD-1530
  - Average Values of da/dN Data
  - Minimum Values of Fracture Toughness
- MIL-A-8861
  - Modified Turbulence Field Parameters
  - Continuous Turbulence Analysis to Include Response to Asymmetric Turbulence and Account for Time Phasing of Loads
- MIL-A-8862
  - Maximum Design Sink Speed of 15.0 fps
- MIL-A-8866
  - Modified Maneuver Load Factor Spectra Based on C5A, C-141, and C-130 Data
  - Maneuver Load Factor Spectra for Low Level High Speed Cruise Specified
  - No Functional Impairment in Two Lifetimes Based on Growth from an Initial .010″ Flaw
  - Credit for Effects of Fatigue Life Enhancement Allowed but Limited to Effect of Reducing Initial Flaw Size to .005″

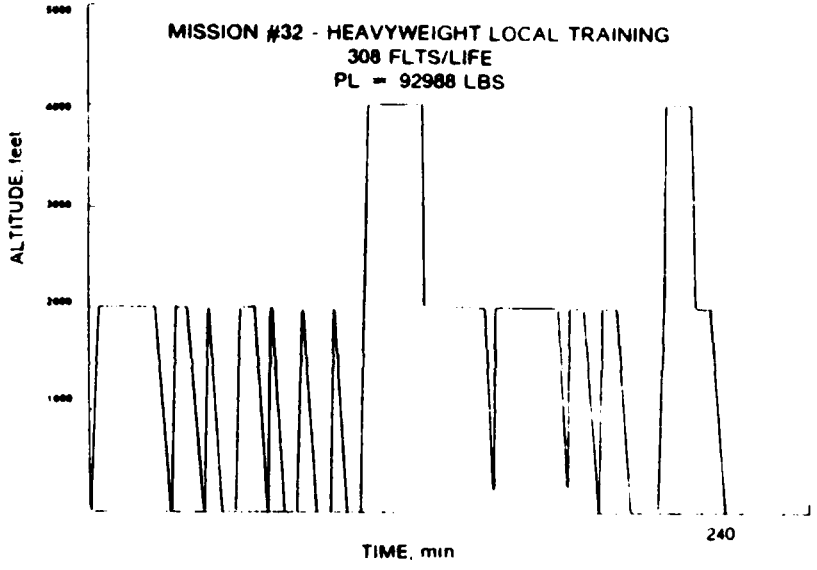
EXCEPTIONS AND/OR AMPLIFICATIONS (cont'd)

- MIL-A-83444
  - Credit for Beneficial Effects of Life Enhancement Techniques Disallowed
  - Pressurized Door Mechanical Systems Designed and Qualified as Failsafe Multiple Load Path Structure
  - Depot or Base Level Inspection Option Deleted for Structure Qualified as Slow Crack Growth
  - Intact Structure Requirements Deleted for Structure Qualified as Failsafe

SPECIFIED SERVICE USAGE

- 35 Unique Peacetime Design Mission Profiles Specified

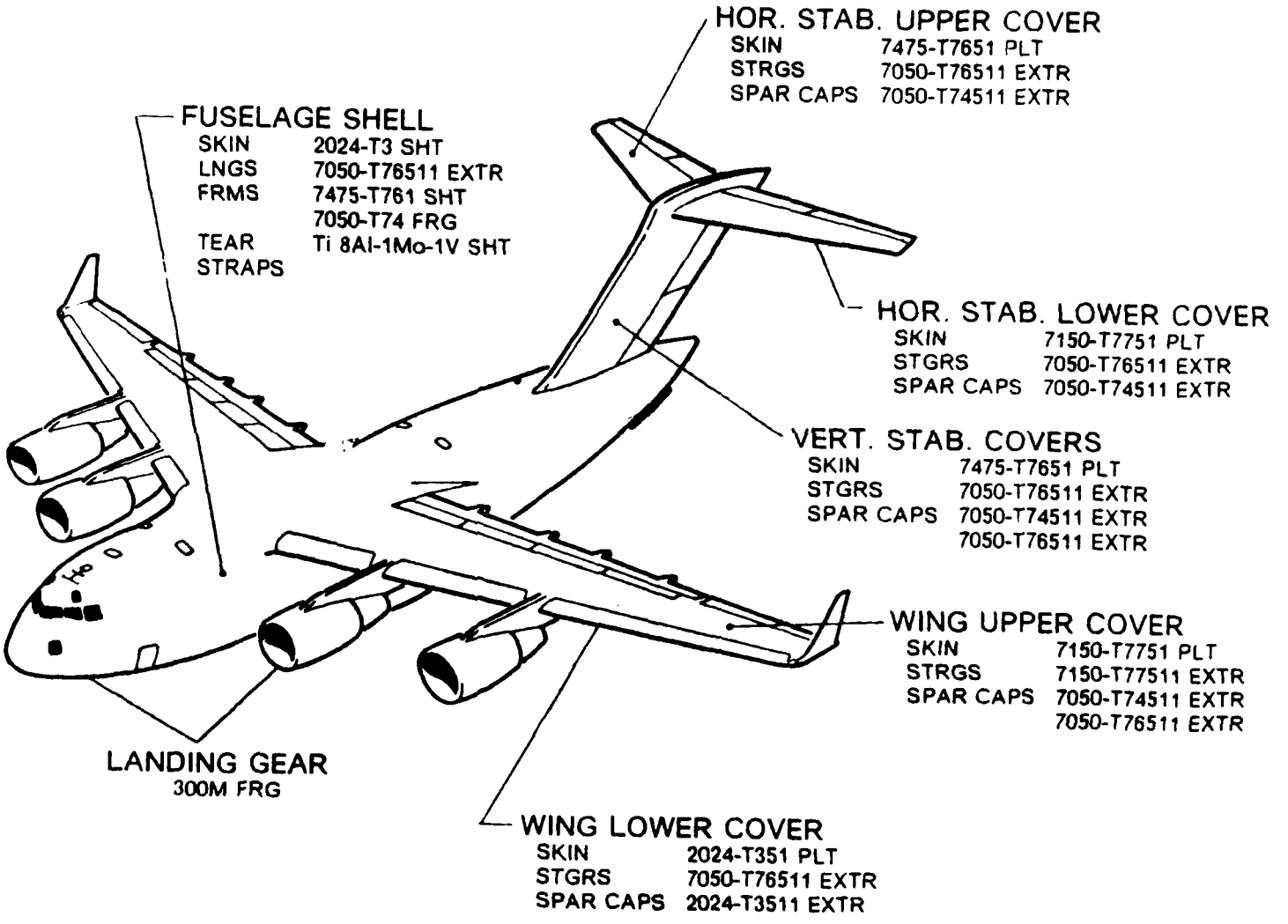




- One Lifetime of Operation Consists of:

  8564 Flights
  30000 Flight Hours
  18950 Landings (Includes Touch and Gos)
  4200 Full Pressure Cycles
  3200 Hours @ $V_E \geq 300$ KEAS & $h \leq 2000$ feet
  6050 Hours @ $V_E < 300$ KEAS & $h \leq 2000$ feet

C-17A PRIMARY STRUCTURE MATERIALS



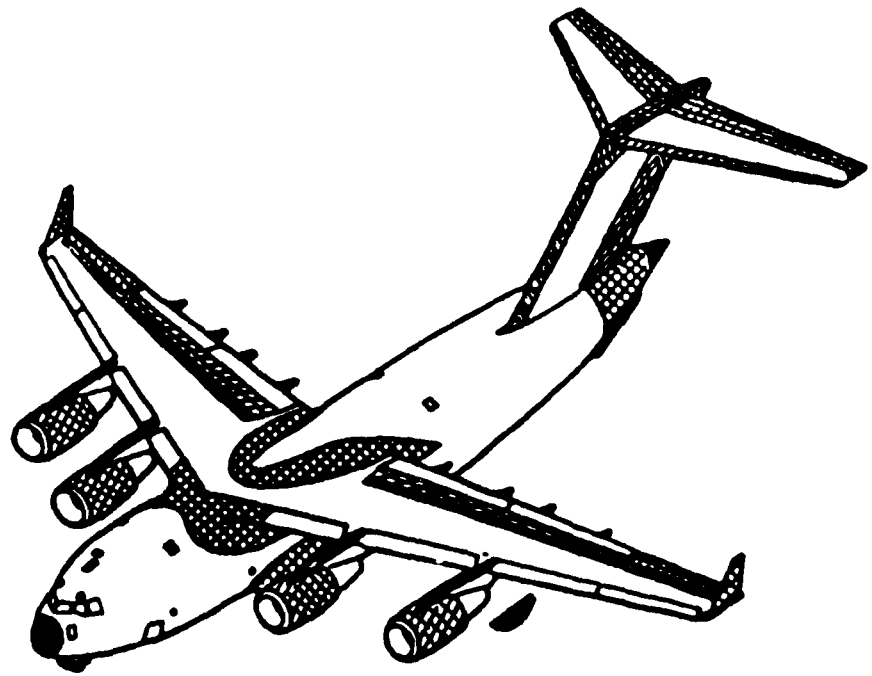
C-17A COMPOSITES APPLICATIONS

- Major Composites Applications

  Empennage L.E. and T.E. and Fairings
  Wing T.E.
  Flap Hinge Fairings
  Strake
  MLG Pod
  Wing/Fuselage Fillet
  Ailerons, Elevators and Rudders
  Flap T.E. and Vane
  Winglet
  Tailcone
  Gear Doors

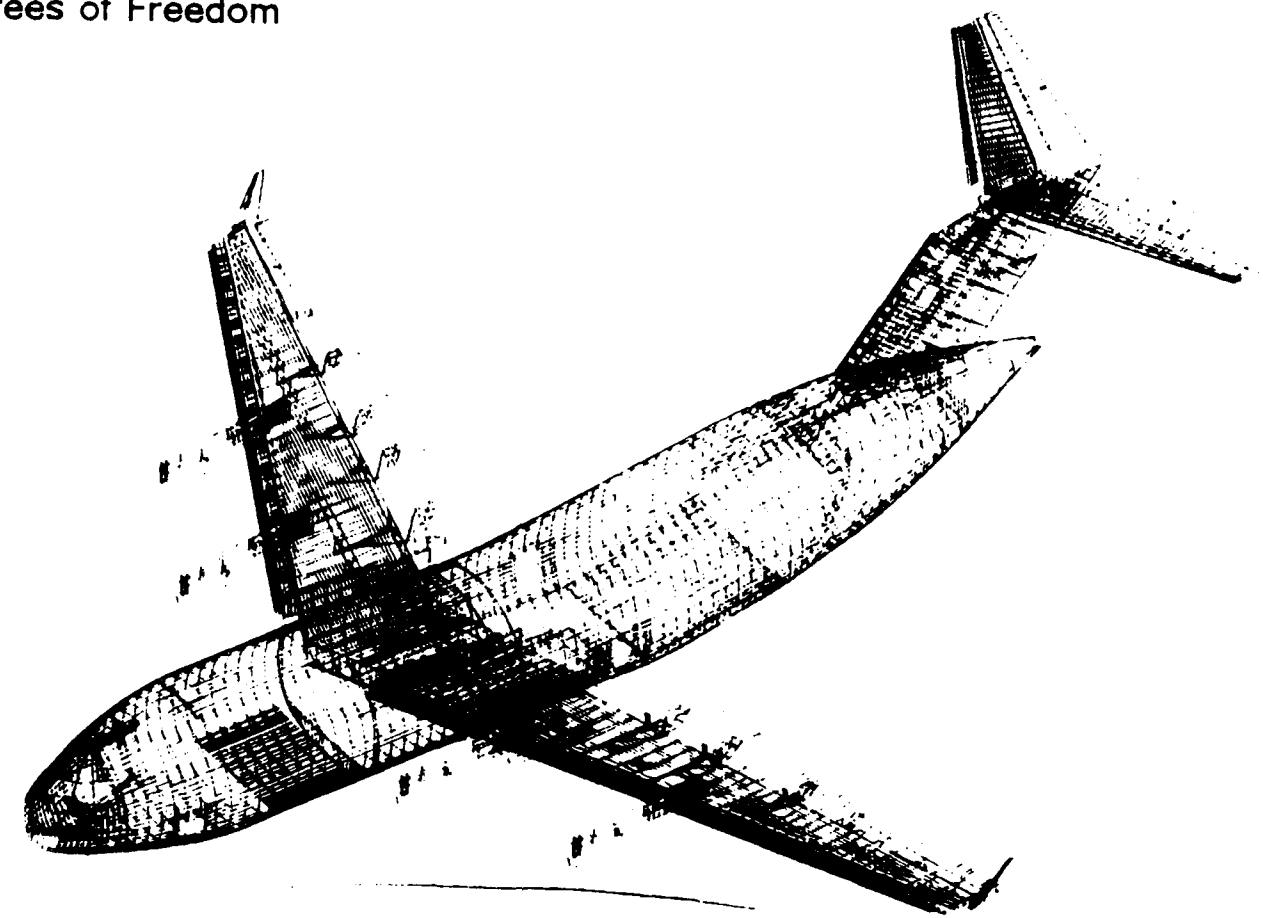
- Significant Development Test Program

  264 Specimens
  Shear Panels
  Compression Panels
  Fitting/Attachment Details



INTERNAL LOADS ANALYSIS

- One for One Modeling of Frames, Longerons, and Stringers
- 398,000 Degrees of Freedom



C-17 FINITE ELEMENT MODEL

MATERIAL PROPERTIES COUPON TEST PROGRAM

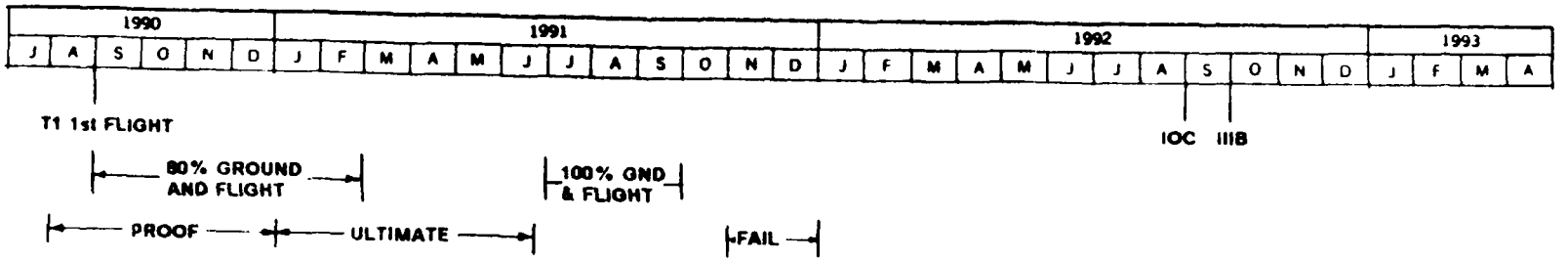
- Comprehensive Coupon Test Program Conducted To:
  - Generate Data Base for New Materials
  - Expand/Augment Data Base for Established Materials
- Tests Included:
  - Static Strength Allowables (F_{tu}, F_{ty}, F_{cy}, F_{su}, *etc.*)
  - Crack Growth (da/dN vs ΔK)
  - Fracture Toughness (K_{Ic})
  - Crack Growth Resistance (K_R)
  - Stress Corrosion (K_{ISCC})
- Materials Tested Included:

2024	7475	Ti 6Al-4V ANN
2090	ARALL 3	Ti 6Al-4V RA
6013	INCO 718	Ti 6Al-2Sn-4Zr-2Mo
7050	PH13-8Mo	Ti 10V-2Fe-3Al
7150	15-5PH	

- 6079 Total Specimens

C-17A AIRFRAME STATIC TEST PROGRAM

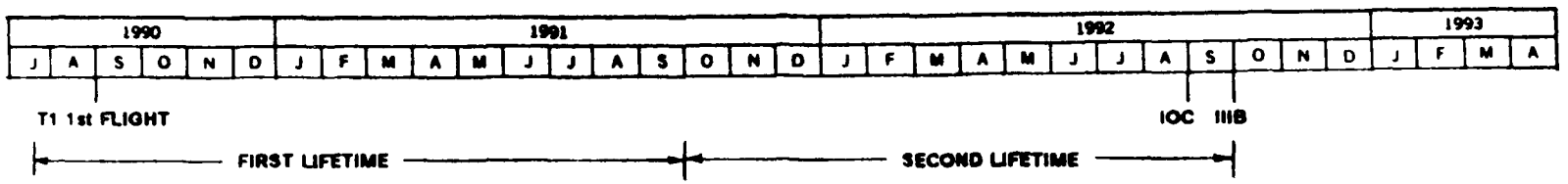
- Schedule



- Extensive Use of Glue-on Tension Pads to Load Structure
- Proof Test Critical Conditions for Major Components
- Conditions Include Cabin Pressure as Applicable
- Condition Sequence and Number of Conditions Currently Being Established
- Wing Failure Test at Conclusion of Static Test Program

FULL SCALE AIRFRAME DURABILITY TEST PROGRAM

- Schedule



- Limited Residual Strength Testing and Teardown Inspection at End
- Random, Cycle-by-Cycle, Flight-by-Flight, 1/10 Lifetime Sequence which Includes all 35 Design Flights
- Two Lifetime Duration (i.e. 60000 Flight Hours)
- Extensive Use of Glue-On Tension Pads to Load Structure
- All Significant Cabin Pressure Cycles Applied
- 1305 Strain Gages Specified to Date

OFF-AIRFRAME COMPONENT STRUCTURAL QUALIFICATION TESTS

- Major Structural Components Tested Separately from the Full Scale Static and Durability Articles Include the Following:

  Landing Gears
  Control Surfaces
  Flaps and Slats
  Horizontal Stabilizer
  Pylons
  Engine Thrust Reversers

- Benefits Include the Following:

  Loading and Fixture Flexibility
  Schedule Flexibility and Compression
  Tailoring of Applied Loads Spectra
  Facilities and Manpower Availability

STRUCTURAL FLIGHT TEST AND LOADS CALIBRATION

- First Aircraft Assembled is Used for Structural Flight Test (Designated as T-1 Aircraft)
- Perform Loads Calibration in Laboratory Prior to First Flight
  - Apply Unit Load Conditions
  - Apply Limit Load Conditions
  - Perform Control Surface Functional Tests
- Perform GVT in Laboratory Prior to First Flight
- Perform Flight and Ground Operations Load Survey to 80% Design Loads
- Perform Flight and Ground Operations Tests to 100% Design Loads after Completion of Ultimate Static Test Program

C-17A IATP and L/ESS OBJECTIVES

- Collect Sufficient Data to Allow Reconstruction of Time History of Missions
- Analyze Collected Data and Report Individual Aircraft Usage and Structural Status (i.e. Control Point Crack Growth) Every 6 Months - First IAT Report 12 Months after First Flight
- Analyze Collected Data and Report on Loads/Environment Fleet Exposure Every 12 Months - First L/ESS Report 18 Months after First Flight

DATA COLLECTION

- Onboard Data Recorder (SFDR) Used to Record:
  - IATP Data for 100% of Operation for 100% of Fleet
  - L/ESS Data for 20% of Operation for 100% of Fleet
- 49 Parameters Recorded - 5 Different Record Types

	IATP	L/ESS	R		IATP	L/ESS	R		IATP	L/ESS	R
Aircraft Serial Number	x	x	D	Pressure Altitude	x	x	T	Nx		x	H
Sequential Flight No.	x	x	D	Radar Altitude	x	x	T	Pitch Rate		x	H
Airframe Time	x	x	D	Indicated Air Speed	x	x	T	Yaw Rate		x	H
Mission Type	x	x	D	Airplane Gross Weight	x	x	T	Roll Rate		x	H
Date of Mission	x	x	D	Total Fuel Weight	x	x	T	Strain #1		x	H
Departure Base	x	x	D	Airplane CG	x	x	T	Strain #2		x	H
Auto Pilot On or Off	x	x	E	EPR Engine #1	x	x	T	Srtain #3		x	H
Thrust Reverser Deployed or Stowed	x	x	E	EPR Engine #2		x	T	Strain #4		x	H
Slats In or Out	x	x	E	EPR Engine #3		x	T	Strain #5		x	H
AR Door Open or Closed	x	x	E	EPR Engine #4		x	T	Strain #6		x	H
Cargo Ramp/Door Open or Closed	x	x	E	Hor. Stab. Position		x	T	Strain #7		x	H
Landing Gears Up or Down	x	x	E	Flap Position		x	T	Elevator Position		x	C
Weight On or Off Wheels	x	x	E	Taxi Speed		x	T	Aileron Position		x	C
Parking Brake On or Off	x	x	E	Speed Brake Position		x	H	Left Spoiler Position		x	C
Flaps In or Out	x	x	E	Nz	x	x	H	Right Spoiler Position		x	C
Speed Brake Deployed or Stowed	x	x	E	Ny		x	H	Rudder Position		x	C
Troop Doors Open or Closed	x	x	E								

D - Data Tag
E - Event
T - Time Track
C - Counted
H - Peak Valley Time History

THE X-30 STRUCTURAL INTEGRITY PROGRAM: THE CHALLENGES AHEAD

John W. Lincoln and James M. Snead***

Some of the major issues are discussed that need to be addressed in the structural integrity program for the X-30 (NASP) vehicle. The primary emphasis in this paper is a method that may prove attractive for qualifying the structure for flight. A probabilistic approach is described that will enable the structural engineer to determine the risk of structural failure from the loading environment imposed on a hypersonic vehicle. The basis for the method is a time domain analysis where the loads from all significant sources are calculated and then a Monte Carlo technique is used to determine their probability distribution function. The probability distribution for strength is then determined based on an examination of the failure modes for each critical location. The combination of these distributions can then be used to determine the probability of failure.

INTRODUCTION

Over the last five years, the Air Force and NASA have conducted a wide range of design studies to identify the best technical course for building the next generation of manned Transatmospheric Vehicles (TAV) and supersonic and hypersonic military and commercial aircraft. Central to these studies was a joint Air Force/NASA technical study to assess the scientific feasibility of building a single-stage-to-orbit aircraft. In the fall of 1985, the Air Force and NASA jointly concluded that it was within the capabilities of this country to develop and demonstrate, through a tightly focused experimental aircraft program, the requisite technologies for a generation of new and highly advanced aircraft.

The National Aero-Space Plane (NASP) program, a joint Air Force, NASA, Navy, Defense Advanced Research Projects Agency (DARPA) and the Strategic Defense Initiative Organization (SDIO) program, was initiated to develop and demonstrate the technologies needed for safe and economical single-stage-to-orbit (SSTO) flight and endoatmospheric hypersonic cruise. To achieve and demonstrate these program goals, the NASP program will design, build, and flight test two experimental aircraft which have been designated the X-30. These two X-30 aircraft will be the

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first of a new generation of transatmospheric vehicles that can efficiently operate in the atmosphere or in space in low Earth orbit. While the X-30's will be experimental aircraft, one of the primary goals of the program will be to develop and demonstrate not only the technologies needed to accomplish SSTO flight and sustained hypersonic cruise capabilities but to also develop and demonstrate those technologies needed to accomplish these flight capabilities in a safe and economical manner. Because of this latter emphasis, the X-30's primary design goals include the following:

- Aircraft-like operations (ground support, maintenance, flight planning, etc.).
- A fully reusable aircraft - no expendable parts other than normal wear and tear (brakes, tires, etc.) and extended time between subsystem replacement or repair.
- Normal aircraft horizontal takeoff and landing.
- Normal aircraft safe flight characteristics - system redundancies, safe abort, safe unpowered flight characteristics, etc.
- Good overall system reliability, safety, and maintainability.

The achievement of these design goals, while at the same time producing an aircraft capable of SSTO flight, is the major challenge in deciding upon the appropriate structural integrity program for the NASP program.

THE NASP PROGRAM

The NASP Program has been organized into three phases (Figure 1). Phase 1 focused on the assessment of the feasibility of the concept of a SSTO vehicle and the technologies needed to build such a vehicle. This phase concluded in the fall of 1985 with the aforementioned decision to proceed with the X-30 program.

Phase 2 of the program began in 1986 and will extend until the fall of 1990. In this phase of the program the propulsion and airframe contractors, working under separate contracts, are defining the propulsion system and airframe concepts. In conjunction with these conceptual design studies, the contractors and the government are conducting highly focused technology maturation programs to ready needed technologies in time for Full Scale Development.

At the end of FY 90, following a formal program technical review and decision to proceed, the program will enter Phase 3 which includes full scale development, production, and the flight test program. As Figure 1 shows, the current program schedule is aiming at a FY 1995 first flight, followed by two years of flight test envelope expansion leading to the first SSTO flight in FY 1996. Phase 2 and Phase 3 will take about ten years. In perspective, this is about the same amount of time as the

Mercury, Gemini, and Apollo space programs of the 1960's took to develop the technology needed to land men on the moon.

STRUCTURAL DESIGN DRIVERS

The central issue in designing a SSTO vehicle is the ability to synthesize a vehicle design that can carry sufficient propellant to accelerate the vehicle from the earth into low Earth orbit (LEO). The total change in velocity required is on the order of 24,500 ft/sec.

Rockets have had this capability for thirty years. But they achieve orbital velocity through the mechanism of staging where all stages but the last stage, which carries the payload, are discarded as the rocket accelerates to orbital velocity. Staging is necessary to compensate for the low propulsion efficiency of these rockets. The problem with staging in this manner is that these expensive stages are only used once and then discarded. This prevents the rockets from achieving the economics of operation that fully reusable vehicles, such as aircraft, routinely achieve.

The goal of having routine and economical access to space drives the design of advanced space transportation systems towards fully reusable systems. The Space Shuttle system was originally intended to be fully reusable but reductions in development funding forced the program into substituting throw-away stages (the external tank) and partially reusable stages (the solid rocket boosters) for a fully reusable first stage. The economical impact of these decisions was not fully appreciated when these design changes were made.

The NASP program has selected a true single-stage-to-orbit vehicle over competing designs of quasi-SSTO and two stage systems. [A quasi-SSTO system is one such as the Boeing Reusable Aerospace Vehicle (RASV) which drops the main landing gear during takeoff and lands on a second, much lighter landing gear. This is a version of staging and was needed because the RASV was rocket powered.] The design of a true SSTO vehicle is governed by a single equation:

$$V_{ORBIT} = - g_o I_E \ln(W_{ORBIT} / W_{TAKEOFF})$$

where:

V_{ORBIT} = Orbital velocity (24,500 ft/sec)

I_E = Trajectory averaged propulsion efficiency (specific impulse, sec)

W_{ORBIT} = Vehicle weight in orbit (empty weight plus payload plus propellant reserves)

$W_{TAKEOFF}$ = Vehicle weight at the mission start

While simple in appearance, this equation, sometimes referred to as the "rocket equation", is deceptive in actual application to a vehicle design. The first item to note is that the required orbital velocity is a constant for a given orbital altitude, orbit inclination, and takeoff latitude. The second item to note is that the ratio of weights is, in the first order, an expression of the structural efficiency of the design. This ratio relates the weight of the structure to the mass (or volume) of the propellant required to be carried. For a typical SSTO vehicle design, this ratio will have values between .25 and .50. This means that every pound of structure must carry one to three pounds of propellant.

The structural difficulty in designing a SSTO vehicle comes from the demands placed on the structures from the remaining term in this equation. This is the propulsion efficiency or, in rocket terms, the specific impulse. (Aircraft designers are usually used to this being expressed as specific fuel consumption.) Our design studies have shown that in order to incorporate the aforementioned program goals into a vehicle of reasonable size, the trajectory averaged propulsion efficiency must be significantly higher than what can be achieved using only rocket propulsion. These studies have indicated that airbreathing propulsion must be used throughout most of the flight trajectory, perhaps as high as Mach 24, which is essentially orbital speeds, in order to obtain a sufficiently high enough average specific impulse that vehicle design closure can be achieved.

The demands that this very high speed flight within the atmosphere place on the airframe and engine are very severe. External leading edges would see radiation equilibrium temperatures in excess of 4000 degrees F. The upper fuselage radiation equilibrium temperatures would exceed 1200 degrees F and the lower, forward fuselage surface, which is the inlet compression surface for the engines, and the lower, aft fuselage surface, which is the engine nozzle expansion surface, would see significantly higher temperatures. These structures are also exposed to high dynamic pressures and high acoustic loads.

In addition, airbreathing propulsion systems above Mach 3 utilize ramjet and scramjet engines. Above approximately Mach 6, hydrogen must be used as the propellant in order to sustain combustion. This places the additional demands on the structure of safely housing a very cold (-427 degrees F) liquid with a very low density (5 lbm/cubic foot). Typical jet fuels have a density of 50 lbm/cubic foot - about ten times more dense than liquid hydrogen. This low propellant density combined with the high propellant fractions (the percentage of the takeoff weight which is propellant) means that most of the vehicle houses propellant tanks.

These demands on the structures for the X-30 are very challenging. They must be very light weight in order to achieve the closure weight ratio, but they must be able to withstand a very severe environment of high temperatures, high dynamic pressures, and high acoustic loads dictated by the flight trajectory required to achieve the closure propulsion efficiency.

STRUCTURAL INTEGRITY ISSUES

The current USAF approach to developing structural integrity in military aircraft is embodied in the USAF Aircraft Structural Integrity Program (ASIP), MIL-STD-1530A, and the Engine Structural Integrity Program (ENSIP), MIL-STD-1783. These two programs were developed to apply a systematic technical management technique to new aircraft and engine development programs to insure that the new products achieved the desired structural utility, safety, and economic cost of ownership. These programs have been successful because they have incorporated the lessons learned from prior development programs, they have been applied to aircraft and engines that are highly similar in design, materials, and use, and there has been a large body of applicable engineering knowledge and test capabilities to support the many technical decisions made throughout the structural development program.

The instances where the structural integrity program for a new system runs into difficulty is typically with the implementation of new structures and materials technologies. New technology brings new problems and requires new engineering solutions. A well laid out structural integrity program recognizes the possibility of unknown problems and expands such areas as materials and component testing to help identify these new problems early to allow time for effective engineering solutions to be developed.

As described above, the demands placed on the X-30 airframe and propulsion system are substantial. To meet these demands much of the airframe and propulsion system will be made from advanced materials utilizing advanced fabrication methods. The government and the NASP contractors have recognized the challenge of successfully transitioning these new technologies to the X-30. To aid this transition, the NASP program has instituted a broad series of technology maturation activities to help identify the unknown problems and to develop the needed effective solutions.

Table 1 lists several of the structural integrity issues that are being addressed for the NASP program. The remainder of this paper discusses these issues in more detail.

Table 1

X-30 Structural Integrity Issues

- Instrumentation for hot structures
- Hot structure material development
- Hot structure testing with hydrogen
- Hot structure design criteria
- Acoustic loading

INSTRUMENTATION FOR HOT STRUCTURES

One of the main concerns in the development of a vehicle of this type is the lack of mature instrumentation for hot structures testing. This concern includes all of the coupon, component, full-scale and flight testing. As has been the case for previous programs, the structural instrumentation is the element that ties all of this testing together for analysis verification and for identification of failure modes in the development and full-scale testing which leads to first flight release. It is the essential basis for expansion of the flight boundaries after the initial clearance for flight.

The current family of structures test instrumentation has been developed for conventional materials operating in moderate temperature ranges. Because of the large temperature extremes and use of advanced materials, the X-30 poses a number of test instrumentation problems that include:

- Gage survival at elevated temperatures and liquid hydrogen cryogenic temperatures. This is especially a problem for flight test instrumentation where the gage location may be inaccessible after the completion of the vehicle assembly.

- Gage bonding techniques that can withstand the temperature extremes but also the rapid temperature changes that induce localized thermal strains at the attachment. For some materials, such as refractory composites, bonding gages is very difficult because of the large difference in the coefficient of thermal expansion (CTE) between the base material and the gage material. On these materials it is also not possible to weld the gage as has been done on some high temperature metallic structures.

- Shielding the attachment wiring from the thermal environment and routing the wiring in such a way as to prevent it from interfering with the temperature, heat flux, or strains that it is intended to measure. This is particular problem in designing component tests that are radiation heated to simulate aerodynamic heating. In the extreme case, gage and wiring effects can induce localized thermal strains and cause local structural failures.

- Gage calibration over a very wide temperature range.

- Protecting the instrumentation wiring from damage due to exposure to the severe temperature environment.

Although there are many areas of the X-30 that have structural temperatures that are within the current strain gage capability, there are large areas that are heated to temperatures in excess of 1200 degrees F. The actively cooled nose cap and leading edges are heated to temperatures in excess of 2000 degrees F and some shock impingement points,

which generate extreme localized heat fluxes and temperatures, could be heated to temperatures in excess of 5000 degrees F if uncooled.

The presentation by M. Lemcoe at the Workshop on High Temperature Structures Testing held at NASA Ames-Dryden on 15-17 November 1988 provided an excellent assessment of the state of the art for strain gages. One of his charts is reproduced in Figure 2. It is evident that the capabilities of the gages listed are not adequate for all of the X-30 testing. It is not evident from this figure that the higher temperature gages are still in the experimental stages and that the cost of these gages will be significantly more than the conventional room temperature stain gages. The NASP program is also assessing the use of fiber optic measurement techniques and advanced remote sensing techniques using various laser applications.

HOT STRUCTURE MATERIALS

The stringent structural weight requirements in conjunction with the heating environment for the X-30 require that the use of materials that are not currently state of the art. The X-30 material goals are shown in Figure 3 relative to a conventional titanium alloy and Rene 41. The materials development is being addressed in the NASP Materials and Structures Augmentation Program. This program, which is a consortium of the NASP airframe and propulsion contractors, is focusing on five classes of materials: titanium aluminides, refractory composites, high specific creep strength, high conductivity composites, and titanium aluminide metal matrix composites. This materials effort will require three years to complete at a cost of 140 million dollars. It is the intent of this work to achieve the following goals:

- Develop and characterize new materials.
- Develop material processes.
- Verify the structural applications.
- Demonstrate producibility.

This work is being accomplished by assigning the primary responsibility for each of the material classes to an airframe or engine contractor. The contractors are sharing their results (and, in some cases, sharing researchers and facilities) with one another. This plan is proving to significantly reduce the overall cost of the material development and to compress the schedule.

Most of the selected materials are moving from the coupon stage to the small component stage. The initial environmental testing of representative small components (one and two feet square) will begin in the fall of 1989. By the end of Phase 2, the materials selected for use on the X-30 will be produced in panel sizes representative of those needed for assembly (4 ft x 8 ft) and similarly tested.

HOT STRUCTURE TESTING WITH HYDROGEN

The X-30 will use hydrogen for its propellant because it is the only effective fuel for high Mach number airbreathing engines. Hydrogen also is an excellent heat transfer agent and can be efficiently used for cooling structural components exposed to high temperatures. Most of the ramjet and scramjet engine flow path structure, the lower fuselage surface inlet compression ramps, the aft lower fuselage nozzle, the engine cowl lip, the wing and control surface leading edges, and the nose cap will be cooled by hydrogen. Besides cooling the structure, this also heats the hydrogen from its normal boiling point temperature of -427 degrees F to the desired injection temperatures which are in excess of 1000 degrees F. This is done to maximize the propulsion efficiency.

Unfortunately we have not yet identified a good simulant for hydrogen for use in structural testing of these actively cooled components. None of the other cryogens are near enough to hydrogen in heat transfer characteristics to provide an adequate simulation. Nitrogen and helium have been used in some design development testing and for analytical code correlation testing, but these cryogens are inadequate for testing the components at the design temperatures and heat fluxes without risking a structural failure. Further, there is a need to properly simulate the panel temperatures, the coolant pressures, and the through-the-wall temperature differentials, which requires the use of hydrogen conditioned to the proper inlet temperatures and pressures. The use of hydrogen, especially at elevated pressures and temperatures, adds significant complexity and cost to running these tests.

In the testing of the hydrogen storage tanks onboard the vehicle, neither nitrogen or helium may be a suitable simulant for liquid hydrogen. Again, the different thermodynamic properties of nitrogen and helium make their use in tank testing difficult. As with component testing, the large scale use of liquid hydrogen will add to test complexity and cost - especially that related to safety.

HOT STRUCTURE DESIGN CRITERIA

The USAF Aircraft Structural Integrity Program (ASIP) has been the basis for verifying that the aircraft structure has adequate strength to provide for a safe and economical operational life. The program to accomplish this is contained in MIL-STD-1530A. The detail requirements and guidance for analyses and ground testing are given in MIL-A-87221. This specification is the result of the evolution in requirements that has taken place over eighty years of powered flight. The detail requirements in this specification are deterministic although the background for the requirements in many cases is probabilistic in nature. For example, the statement that limit loads are the maximum loads which can result from a lifetime of usage of the aircraft is a probabilistic statement since it deals with expectation derived from previous experience with similar aircraft. Also the A and B basis material allowables, which are statistically derived from test data, are used appropriately to make the likelihood of failure remote. There are other examples to be found in the damage tolerance requirements where their basis is tied to a probabilis-

tic concept. The deterministic methods have gained widespread acceptance because they are easy to use and the precedents which they have established over the years have provided confidence in a new aircraft that is to operate in an environment similar to its predecessors. These precedents are important for release of an aircraft for first flight.

The most significant precedent for the X-30 is the X-15 aircraft. This rocket powered aircraft was designed to be released from a B-52, boosted by the rocket to attain its desired trajectory and recovered by means of a landing without power. This program was terminated in the late 1960's. It had been flown almost 200 times and it had reached an altitude of 354,000 feet and a Mach number of 6.7. This vehicle was designed for a maximum temperature of approximately 900 degrees F. The design criteria for the structure is shown in Figure 4. It is seen that there was a different criteria for brittle materials than for ductile materials. It is also seen that at ultimate the ductile materials criteria does not include thermal stresses and the brittle material criteria includes only one times the maximum thermal stress. It should be noted that the X-30 materials will not be the same as the X-15 and the thermal environment will be significantly more severe.

The X-30 aircraft will therefore be exposed to a thermal and mechanical loads environment that is far removed from the experience base from which MIL-A-87221 was derived. Not only are the thermal loads significant in magnitude, but there is an inherent uncertainty in the computational accuracy for the thermal and mechanical loads at high Mach numbers that have not been previously experienced. Also, the X-30 materials which have been developed to withstand the severe thermal environment may not lend themselves to the deterministic procedures that have been traditional used for conventional materials.

These attributes indicate that the use of the current deterministic approach may be unconservative. However, the requirement for a low structural weight fraction for the X-30 indicates that no unnecessary conservatism can be tolerated. Therefore the question is: "How can the aircraft be designed with no more than adequate strength to perform its mission?". The answer to this question may be derived by the use of a probabilistic approach to design.

The Probabilistic Approach

The primary motivation for the adoption of the probabilistic approach is that the designer can decide a priori the risk of failure he is willing to assume and then design the vehicle with that inherent risk. The risk of probability of failure that the designer is willing to accept (reference 1) is, in general, dependent on the proposed usage for that aircraft. For production USAF aircraft there have been two criteria used to determine an acceptable level of risk. The first criterion is that risk of structural failure in a single flight should be no greater than the risk normally accepted in an activity such as driving to work in an automobile. The second criterion is that the risk should be controlled such that the expected number of losses due to structural failure in the lifetime of the fleet should be less than one. The application of these

principles indicate that the single flight probability of failure should be 10^{-7} or less. For experimental aircraft the risk that could be accepted may be somewhat greater. However, it must be kept low enough to not present a significant threat to the goals of the program. It is judged that for an experimental program the probability of failure on any given flight should not exceed 10^{-5}.

Once the acceptable level of risk has been established, the loads calculations and the strength calculations can be made with the goal of deriving the probability distribution function for loads and the probability distribution function for strength. These distribution functions are independent on one another and consequently the joint density function for loads and strength can be derived from the product of the respective marginal density functions. This joint density function can be integrated over the region where the load is greater than the strength to determine the probability of failure. If this calculation does not yield the desired result, then the strength probability distribution can be changed by a structural redesign or the operational envelope can be changed to modify the loads probability distribution function such that the desired risk can be achieved. This process is illustrated in Figure 5.

The Loads Probability Distribution Function

It is evident that the loads on a vehicle such as the X-30 are dependent on many random and non random phenomena. The random events include at least the following:

- Runway roughness.
- Atmospheric density, pressure and temperature.
- Gusts and winds.
- Abort maneuvers.
- Control system errors.
- Initial thermal condition of the vehicle.
- Landing initial conditions.

In addition there are also some phenomena that must be treated as random because they can be determined only to a limited degree of certainty. The following items are included in this category:

- Heat flux in and out of the vehicle.
- Aerodynamic loads at high Mach numbers.
- Engine thrust perturbations.
- Vehicle flight conditions while under direct pilot control.

There are, of course, the non random loads associated with the planned flight path. A time domain procedure will determine the loads for all of these influences. Therefore, the recommended approach is to solve the equations of motion for the vehicle with sufficient degrees of freedom included to adequately determine the load time histories. It is judged that six degrees of freedom will need to be included to represent all important aspects of the rigid body motion. Additional degrees of freedom will be required to represent the deformation of the structure and its dynamic response to abrupt load changes. This can be accomplished through the use of the orthogonal vibration modes of the structure with a suitable boundary condition such as free-free. The number of these modes should be adequate to determine the response due to gusts and landing and to determine the loading changes due to aeroelasticity. These modes will be dependent on the mass of the fuel burned by the engines and on the stiffness changes in the structure due to elevated temperatures. When the deformation degrees of freedom are included, it is desirable to use Lagrange's equations to generate the equations of motion.

The rigid body motion is most easily represented by the body axis components of the aircraft velocity and angular velocity vectors. These velocity and angular components can not, in general, be integrated to orient the body in space and consequently are not generalized coordinates as required by the standard form of Lagrange's equations. This problem can be overcome through a modification of Lagrange's equations which was identified by Whittaker (reference (2)). This modification of Lagrange's equations was used to develop the equations for finding the trajectory of the vehicle.

The abort maneuvers may be the primary source of load on the vehicle. Therefore, is it important that as these maneuvers are developed there is a loads calculation performed with a program such as the one described in this paper. This should help to keep the abort maneuver loads within the loads envelope from loads sources for which there is limited control. Wind and gust loads fall into this category.

The guidance and control equations are also included in the equations of motion. A major challenge is in the determination of the system errors that could influence the applied loads. This must be done on an ad hoc basis since the sensor and gyro technology used has a strong influence on the errors. Another major challenge is to obtain a stabilized version of the guidance and control system. Late changes in the system architecture may have significant effect on the airframe loads.

The heat flux in and out of the vehicle is a source of concern since the analytical tools for this at high Mach numbers are limited. In addition to this complication, the thermal loads calculations are difficult and time consuming to make. Therefore, these calculations will be made in a computer routine that is separate from the trajectory calculations. These calculations will be made at enough points of the trajectory time history to ensure that the thermal effects are properly reflected in the vehicle stiffness and subsequently in the vibration modes.

The engine thrust force determination is complicated by the fact that test facilities are generally inadequate to quantify the thrust for all Mach numbers of interest. Also, the effect of the rigid body and deformed body motion on the thrust may not be well known before the start of flight testing. It is essential that estimates of the expected thrust errors be included in the simulation. Available wind tunnel test results should help in the identification of these thrust perturbations.

Methods of Solution

Since the solutions of the equations of motion required for this problem do not lend themselves to simple probability calculations, the Monte Carlo method is the approach chosen to determine the loads probability distribution function. This may be accomplished in several ways. One approach is to obtain solutions of the equations of motion from repetitive sampling of the boundary conditions of the problem. The critical mechanical loads may then be determined for a number of time segments along the trajectory. These mechanical loads are then combined with the time correlated thermal loads. These loads will then be used to generate an exceedance function for a given time segment in the trajectory. These exceedance functions may then be used as a basis for establishing the probability distribution function. The exceedance functions are modified by dividing each of the ordinates of the exceedance functions by the number of occurrences expected in a single flight. This result is the sample probability of exceeding a given load on a single flight. This sample distribution is approximated by a Weibull, Gumbel, log normal or other distribution for the purpose of obtaining an analytical representation which may be extended to cover remote loading situations. Another approach for doing this is similar to the first except that loads calculations are made for the purpose of defining a solution surface (normally represented by regression equations) which may then be sampled by the Monte Carlo method.

The strength probability distribution is determined for each critical structural element by assessing the features that influence the failure mode(s) for that element. This process is augmented by testing through the building block process which is a procedure of identifying strength and failure modes through testing of progressively more complex specimens.

Benefits from Use of a Probabilistic Design Criteria

The procedure outlined above is believed to have the following benefits:

- The approach will determine the risk of structural failure based on the knowledge of the random inputs.
- These calculations will show where testing can be utilized cost effectively to reduce the risk of failure.
- The approach will identify the need for envelope expansion flight testing.
- The areas of analysis shortcomings will be identified.
- The scope of the materials characterization effort will be defined.
- The approach will be a significant part of the basis for first flight release.

ACOUSTIC LOADING

The acoustic environment on the X-30 is estimated to very severe. D. Mulville at the NASA Ames-Dryden Workshop on High Temperature Structures Testing presented the estimates of the acoustic environment on the X-30 as compared with the AV8-8 measured data. This comparison is shown in Figure 6. The validation of this estimated environment will not be easy since a significant portion of it is engine related and that data will not be available in the near term. The complications with this severe environment are compounded by the fact that much of the affected structure also has a severe thermal environment. This will obviously impose a significant challenge for the test verification of these structures.

CONCLUSIONS

It is evident from the foregoing discussion that the X-30 structural integrity program will provide a significant challenge to the structural engineer. The basic doctrine of the ASIP will need to be used, but as indicated above, the problem will need to be tailored to meet the design goals for this aircraft. The probabilistic approach provides one means for the tailoring to be accomplished in a manner that will provide the best opportunity for the X-30 to meet its design objectives.

REFERENCES

1. Lincoln. J.W., "Risk Assessment of an Aging Military Aircraft," Journal of Aircraft, Volume 22, Number 8, August 1985

2. Whittaker, E.T., "A Treatise on the Analytical Dynamics of Particles and Rigid Bodies," Cambridge University Press, 1961

NASP PROGRAM SCHEDULE

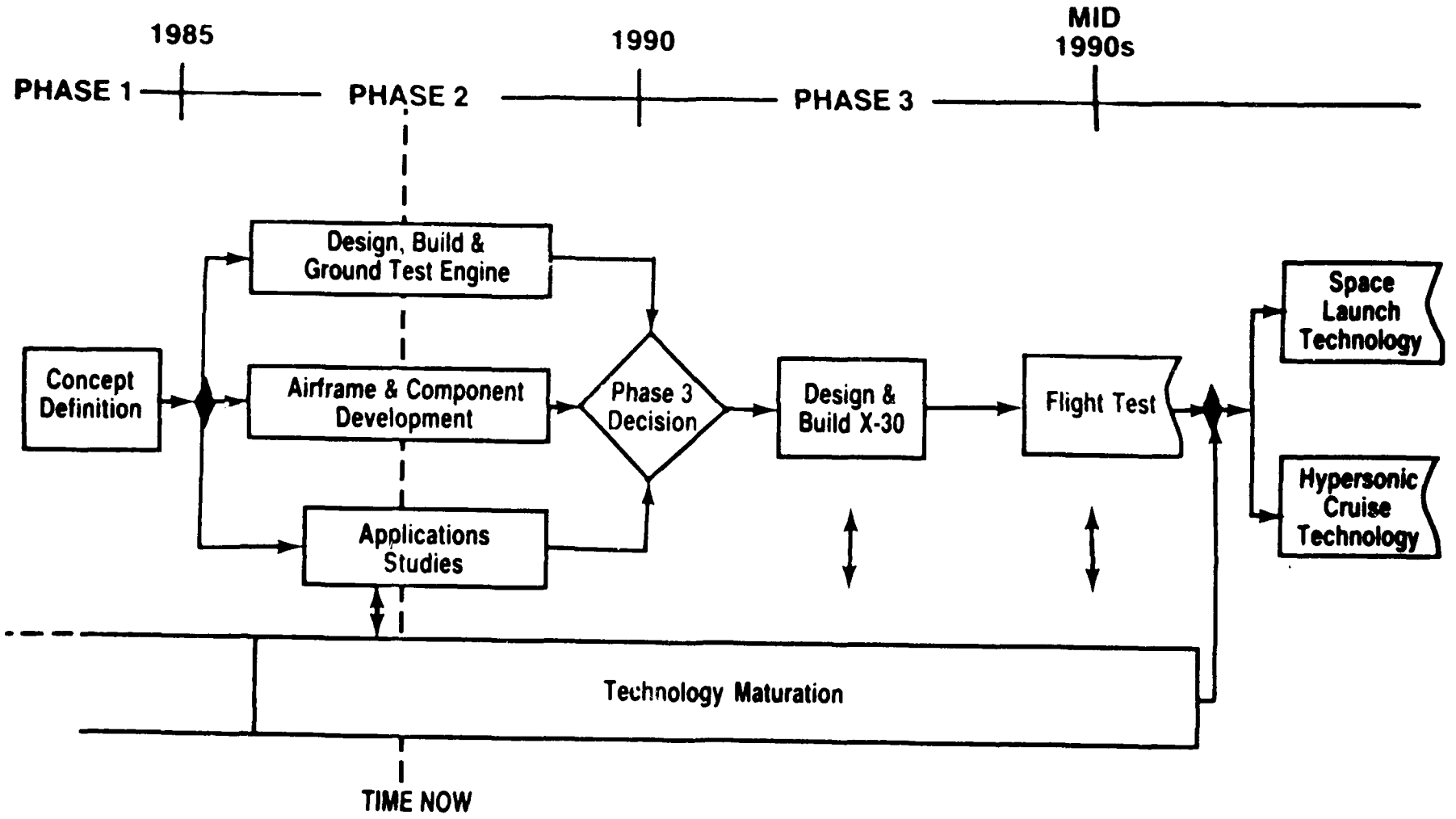


FIGURE 1

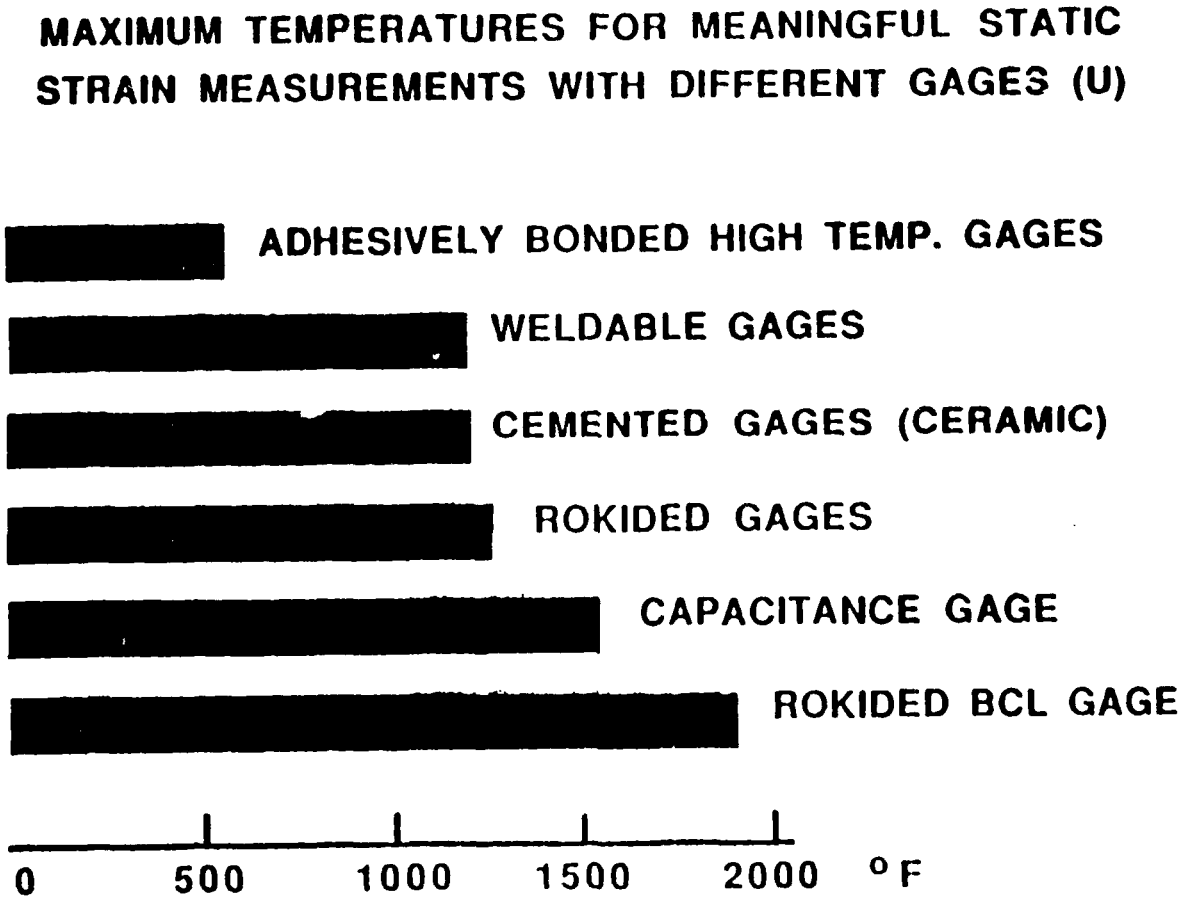


FIGURE 2 - Strain Gage Capabilities

SPECIFIC STRENGTH COMPARISONS

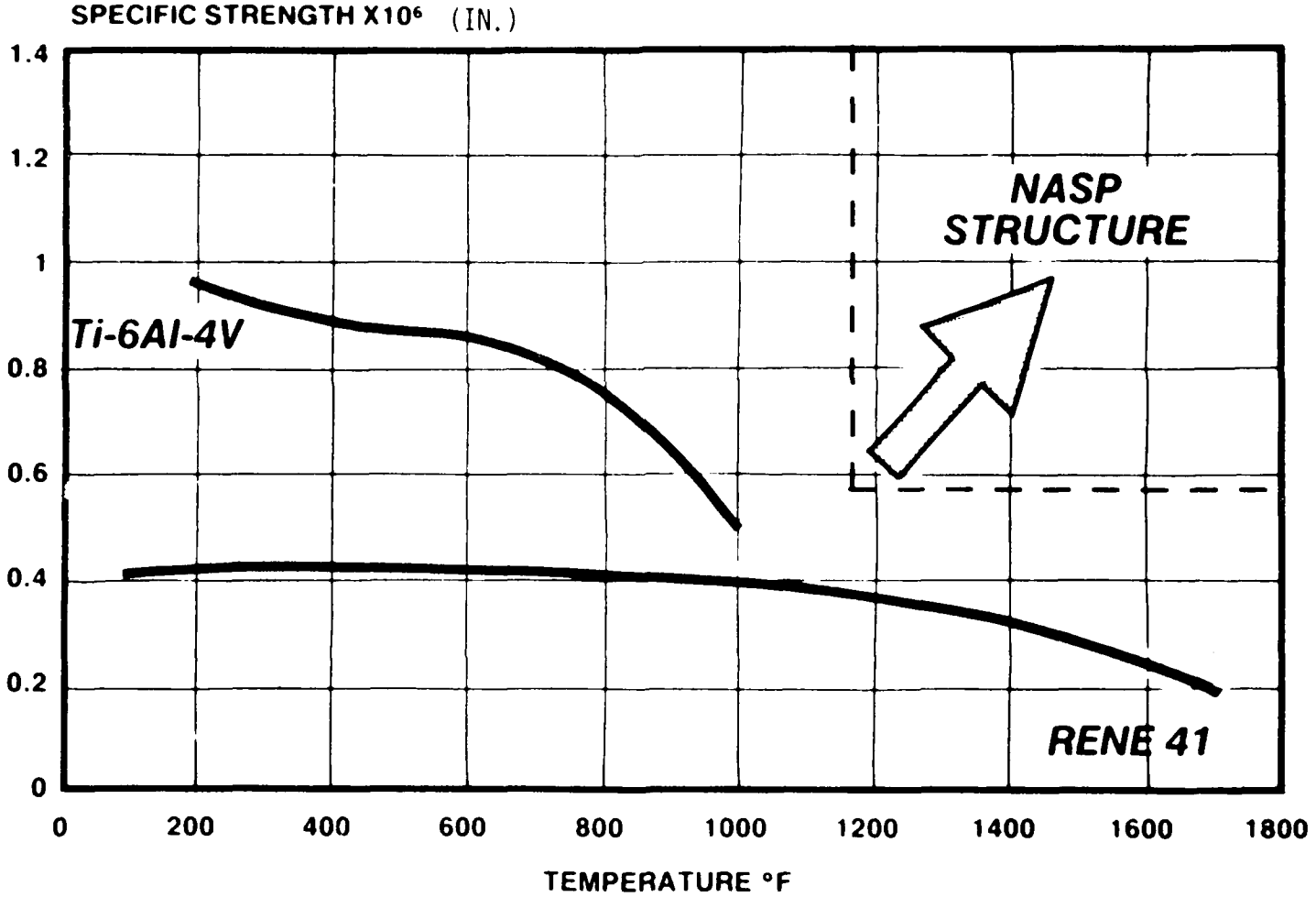


FIGURE 3

STRUCTURAL DESIGN CRITERIA FOR THE X-15

DUCTILE MATERIALS

- AT LIMIT
  - MAX MECH STRESS
    + MAX THERMAL STRESS < MIN YIELD STRESS
- AT ULTIMATE
  - 1.5 MAX MECH STRESS < MIN ULTIMATE STRESS

BRITTLE MATERIALS

- AT LIMIT
  - MAX MECH STRESS
    + MAX THERMAL STRESS < MIN YIELD STRESS
- AT ULTIMATE
  - 1.5 MAX MECH STRESS
    + 1.0 MAX THERMAL STRESS < MIN ULTIMATE STRESS

FIGURE 4

PROBABILISTIC APPROACH

1 ESTABLISH ALLOWABLE FAILURE RATE

2 DETERMINE PROBABILITY DISTRIBUTION FOR LOADS

$$P(X_L < x_L) = F(x_L)$$

3 DETERMINE PROBABILITY DISTRIBUTION FOR STRENGTH

$$P(X_S < x_S) = F(x_S)$$

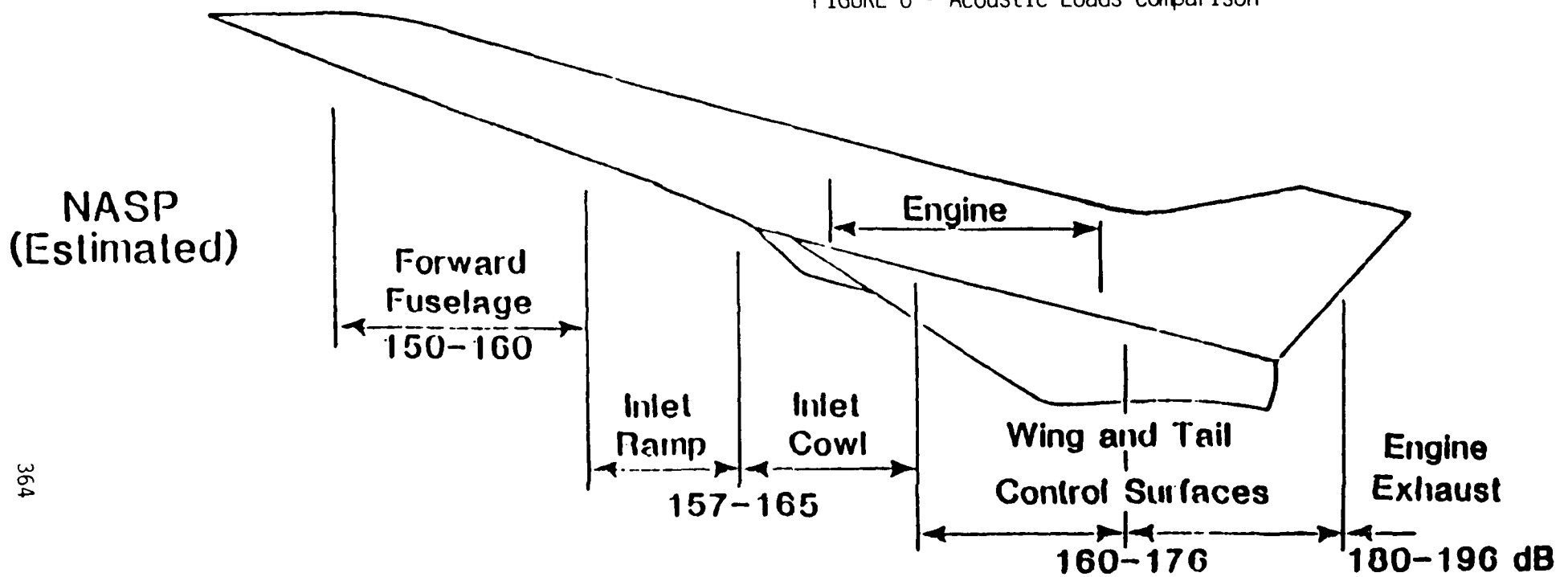
4 CALCULATE FAILURE PROBABILITY P

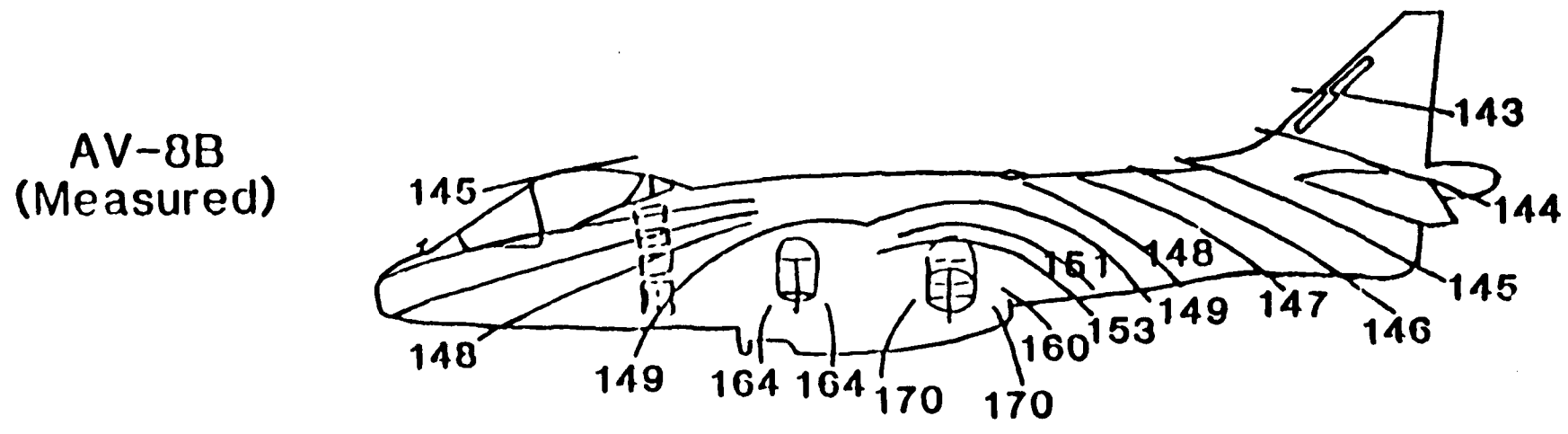
$$P_f = \int_{L>S} F_L' F_S' \, dx_L \, dx_S$$

5 IF P_f IS NOT IN AGREEMENT WITH THE ESTABLISHED FAILURE RATE THEN MODIFY 2 OR 3

FIGURE 5

FIGURE 6 - Acoustic Loads Comparison





STRUCTURAL RISK ANALYSIS IN AGING AIRCRAFT FLEETS

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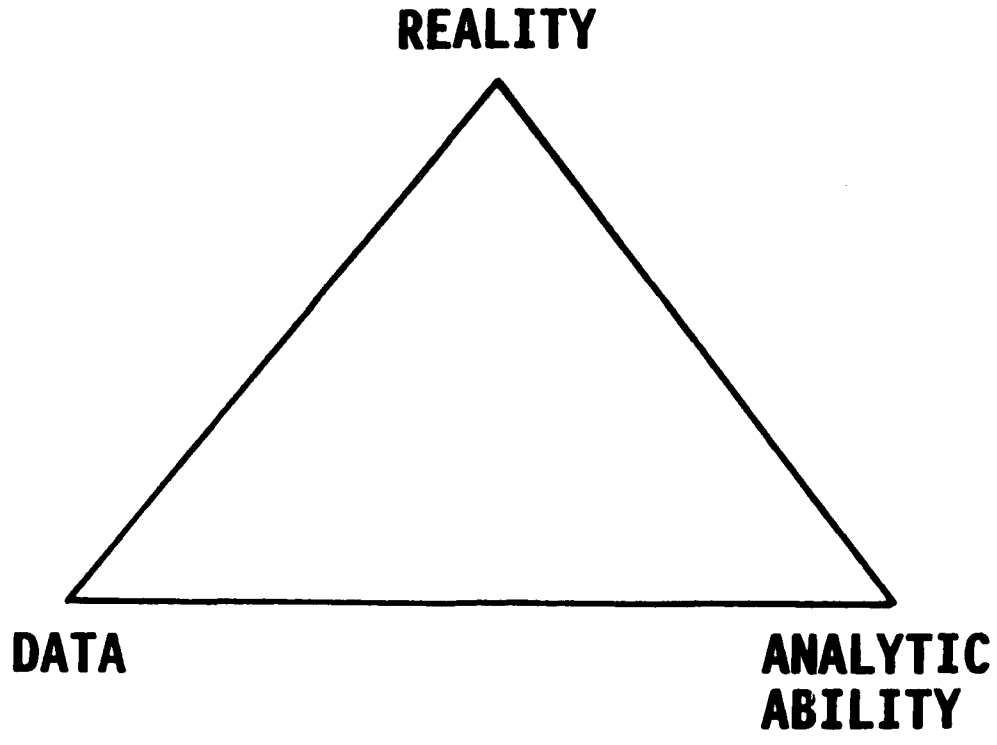
1988 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE

30 NOVEMBER 1988

RISK FACTORS

- CRACK SIZE DISTRIBUTION
- SPECTRUM VARIATION
  - CRACK GROWTH EFFECT
  - MAX STRESS/FLIGHT DISTRIBUTION
- GEOMETRY EFFECT
- CRITICAL STRESS INTENSITY VARIATION
- INSPECTION RELIABILITY

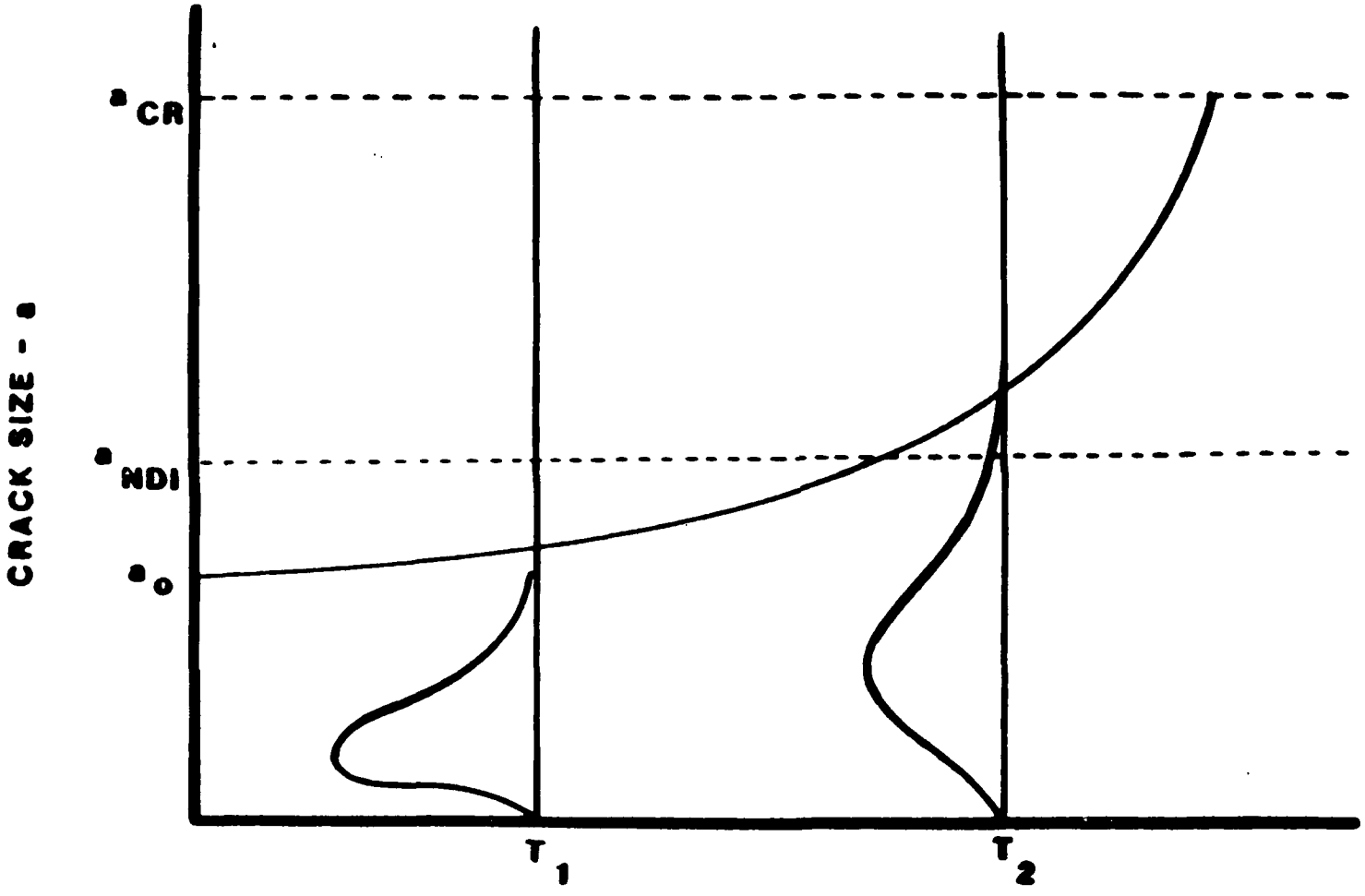
STRUCTURAL RISK ANALYSIS TRADE-OFFS



OUTLINE

- FAILURE PROBABILITY
- BASELINE CONDITIONS
- K_{Ic} DISTRIBUTION EFFECT
- MAX STRESS/FLIGHT DISTRIBUTION EFFECT
- CRACK SIZE DISTRIBUTION EFFECT
- SUMMARY

GROWING POPULATION OF CRACKS

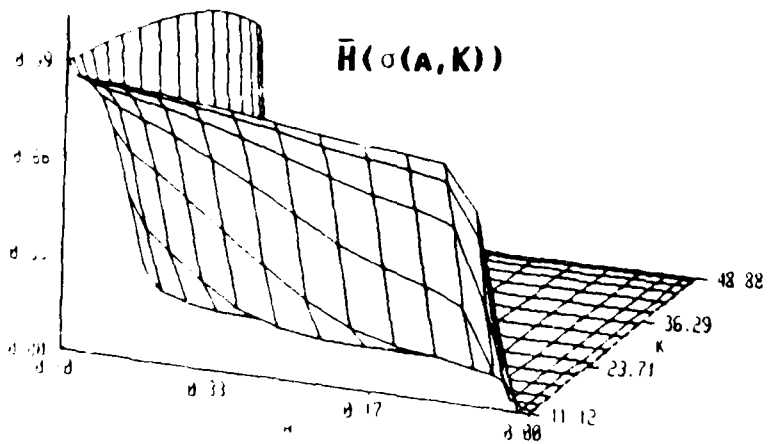


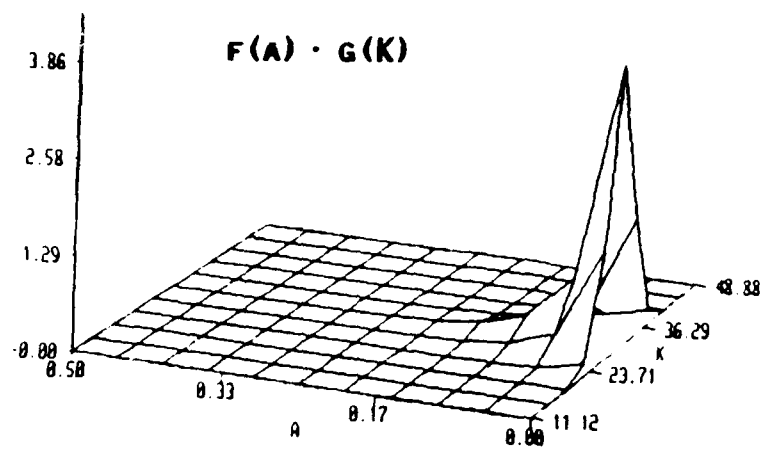
PROBABILITY OF FAILURE

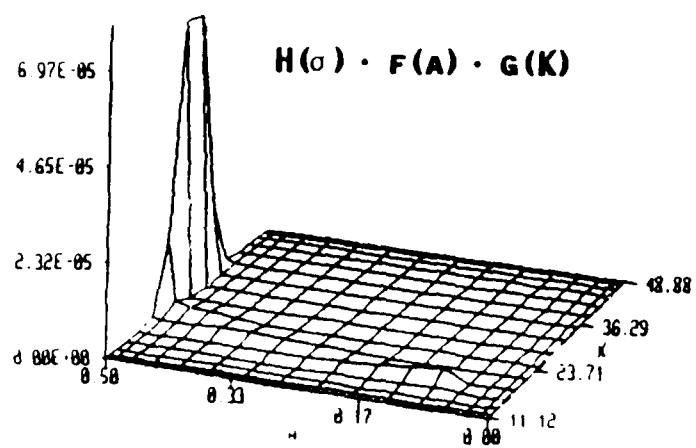
- CRACK SIZE DISTRIBUTION - $F(A)$
- K_{Ic} DISTRIBUTION - $G(K)$
- APPLIED STRESS DISTRIBUTION
  - CONDITIONED ON A AND K_{Ic}

$$P_F = \iint F(A)\ G(K)\ \bar{H}\ (\sigma(A,K))\ DADK$$

SCHEMATIC OF FAILURE PROBABILITY CALCULATION







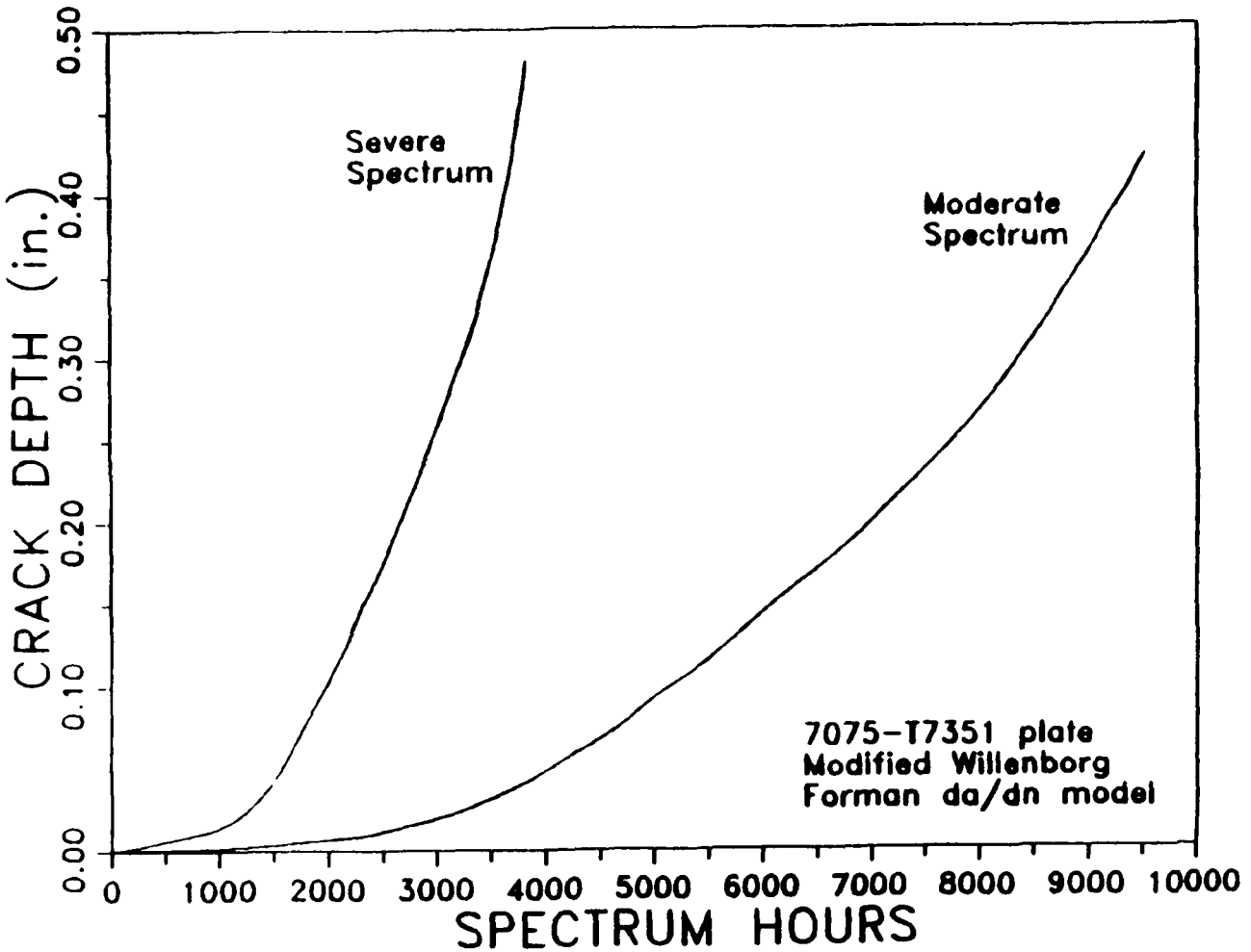
BASELINE CONDITIONS

- ATTACK/FIGHTER/TRAINER (A/F/T) AIRCRAFT
- LOWER WING FASTENER HOLE
- SEVERE USAGE
- NORMAL K_{Ic} DISTRIBUTION ($\mu_{K_{Ic}} = 30$, $\sigma_{KIc} = 3$)
- GUMBEL PEAK STRESS/FLIGHT DISTRIBUTION
  ($A = 3.136$, $B = 22.58$)
- WEIBULL CRACK SIZE DISTRIBUTION
  ($\alpha = 0.9$, $\beta = 0.010$)
- DETERMINISTIC CRACK GROWTH

GEOMETRY FACTOR - FASTENER HOLE
ALPHA (K/σ)
0.20
0.40
0.60
0.80
1.00
1.20
1.40
CRACK DEPTH (IN.)
0.10
0.20
0.30
0.40
0.50
0.60
Dia. = .2755
c
.585
.924

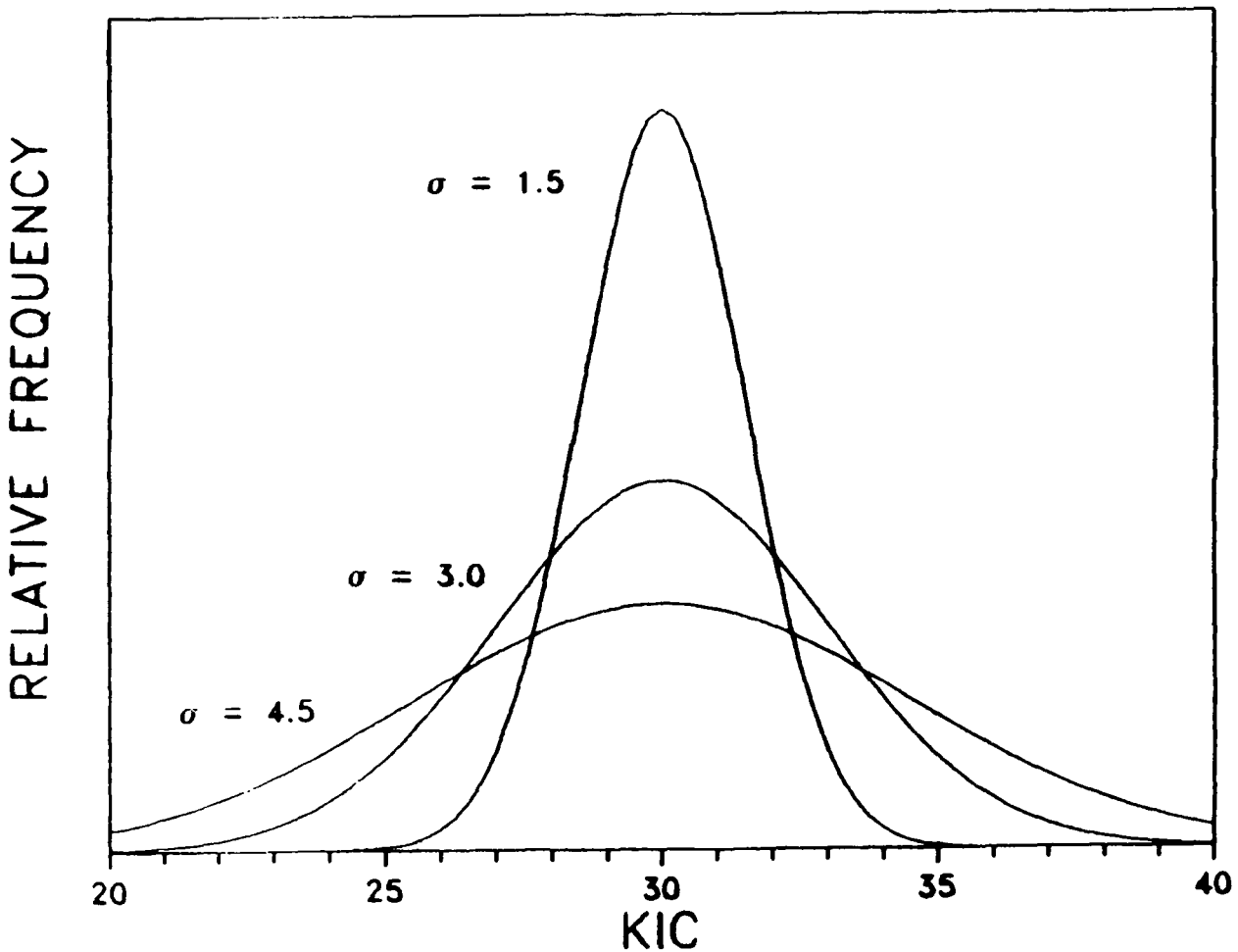
A/F/T - EXCEEDANCE CURVES
EXCEEDANCES /1000 HOURS
STRESS (KSI)
Severe Spectrum Usage
Moderate Spectrum Usage

CRACK GROWTH vs A/F/T SPECTRUM FLIGHT HOURS

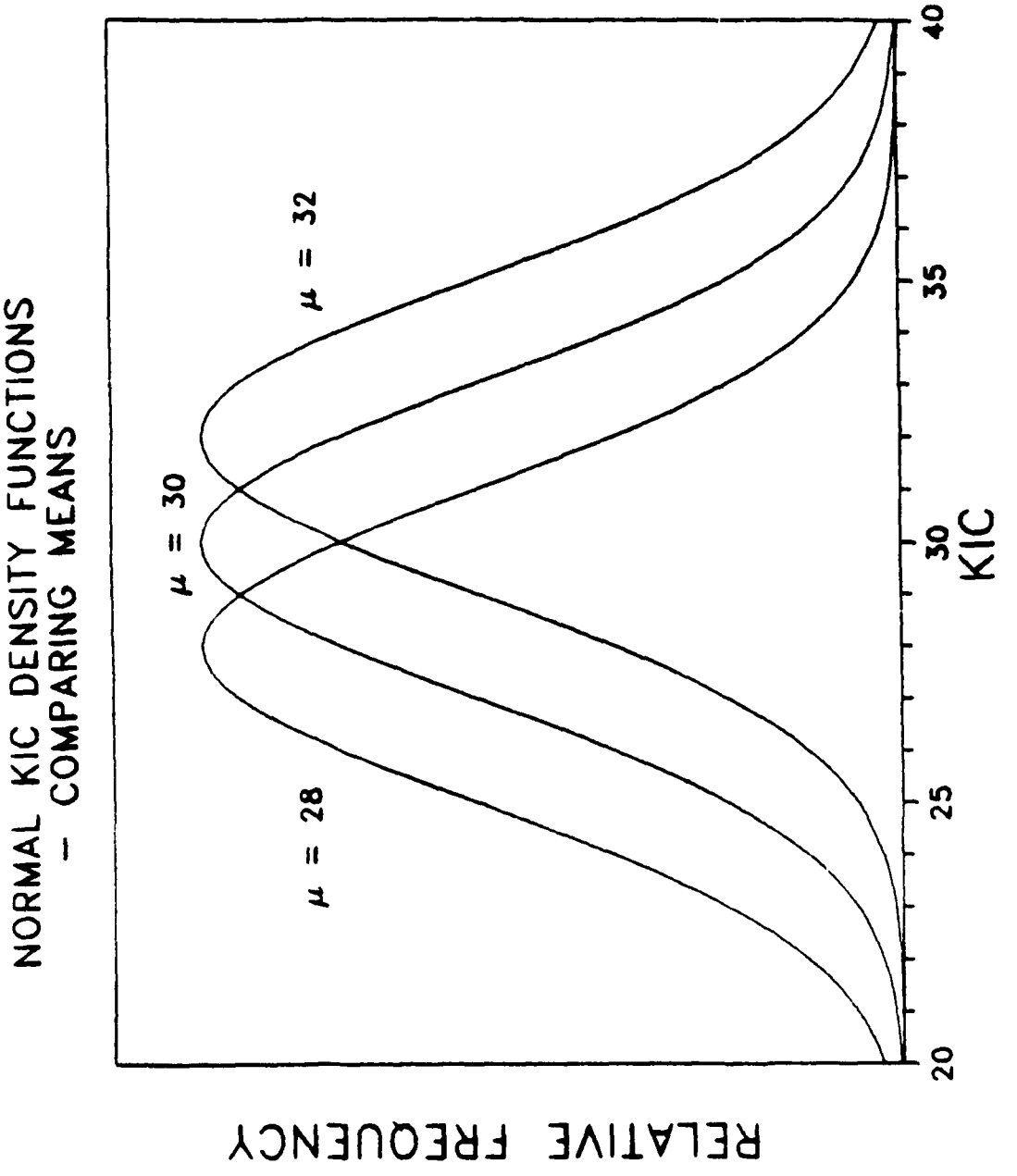


PARAMETRIC STUDY OF RISK PROBABILITIES
- COMPARING A/F/T USAGES
SINGLE FLIGHT PROBABILITY OF FAILURE
SPECTRUM HOURS FROM REFERNCE
WEIBULL CRACKS, α=0.9, β=0.010
NORMAL K_IC, μ_KIC=30, σ_KIC=3
SEVERE A/F/T SPECTRUM
MODERATE A/F/T SPECTRUM

NORMAL KIC DENSITY FUNCTIONS
– COMPARING VARIATION



PARAMETRIC STUDY OF RISK PROBABILITIES
– $\sigma_{K_{IC}}$ EFFECT
WEIBULL CRACKS, α=0.9, β=0.010
NORMAL K_{IC}, $\mu_{K_{IC}}$=30
SEVERE SPECTRUM
$\sigma_{K_{IC}}$ = 1.5 (CV = 5%)
$\sigma_{K_{IC}}$ = 3.0 (CV = 10%)
$\sigma_{K_{IC}}$ = 4.5 (CV = 15%)
SINGLE FLIGHT PROBABILITY OF FAILURE
1
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
10^{-6}
10^{-7}
0
400
800
1200
1600
2000
2400
SPECTRUM HOURS FROM REFERENCE


NORMAL KIC DENSITY FUNCTIONS
– COMPARING MEANS
μ = 28
μ = 30
μ = 32
20
25
30
35
40
KIC
RELATIVE FREQUENCY

PARAMETRIC STUDY OF RISK PROBABILITIES
– $\mu_{K_{IC}}$ EFFECT
SINGLE FLIGHT PROBABILITY OF FAILURE
SPECTRUM HOURS FROM REFERENCE
WEIBULL CRACKS, α=0.9, β=0.010
NORMAL K_{IC}, $\sigma_{K_{IC}}$=3.0
SEVERE SPECTRUM
$\mu_{K_{IC}}$ = 28
$\mu_{K_{IC}}$ = 30
$\mu_{K_{IC}}$ = 32

DISTRIBUTION OF MAX STRESS

- ASSUME EXISTENCE OF EXCEEDANCE CURVE
  - CONVERT TO CDF* - F(s)
- ASSUME FLIGHT J COMPRISES RANDOM SAMPLE OF K_J FROM PEAK STRESS CDF
- $P \{ S_{MAX} < s \} = P \{ \text{ALL } K_J < s \}$

$$= [F(s)]^{K_J}$$

* CUMULATIVE DISTRIBUTION FUNCTION

MAX STRESS/FLIGHT DISTRIBUTION
- OBSERVED VERSUS PREDICTED
PROBABILITY OF EXCEEDING
STRESS (KSI)
0
5
10
15
20
25
30
Severe Spectrum Usage
Moderate Spectrum Usage

BASIS FOR EXTRAPOLATION

- GUMBEL'S EXTREME VALUE THEORY
  - $P \{ S_{MAX} \leq s \}$ CONVERGES TO $F(s)$

    WHERE

$$F(s) = 1 - \exp\left[- \exp\left(\frac{s-B}{A}\right)\right]$$

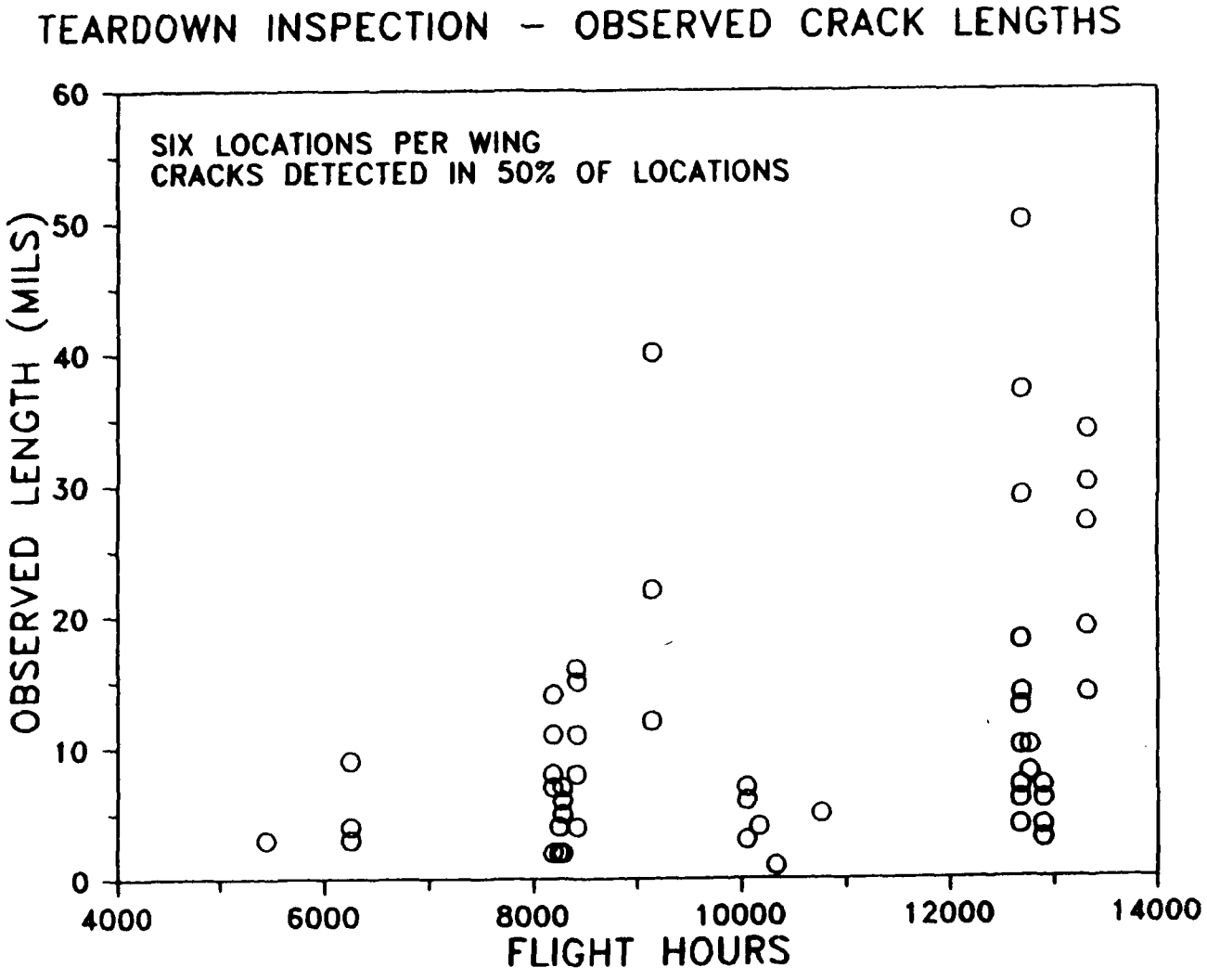
  - $\ln\left[- \ln\left[1 - F(s)\right]\right] = \frac{s}{A} - \frac{B}{A}$

DISTRIBUTION OF MAX STRESS PEAK PER FLIGHT

Distribution of max stress in ? flight
A/F/T Spectrum, Model 2

VARIATIONS OF MAX STRESS/FLIGHT DISTRIBUTION
A/F/T SEVERE SPECTRUM USAGE
EXCEEDANCE PROBABILITY
STRESS - KSI
90% Baseline
95% Baseline
Baseline
105% Baseline
110% Baseline

PARAMETRIC STUDY OF RISK PROBABILITIES
- MAX STRESS/FLIGHT DISTRIBUTION
SINGLE FLIGHT PROBABILITY OF FAILURE
SPECTRUM HOURS FROM REFERENCE
WEIBULL CRACKS, α=0.9, β=0.010
NORMAL K_{IC}, $\mu_{K_{IC}}$=30, $\sigma_{K_{IC}}$=3.0
SEVERE SPECTRUM
90% BASELINE
BASELINE
110% BASELINE

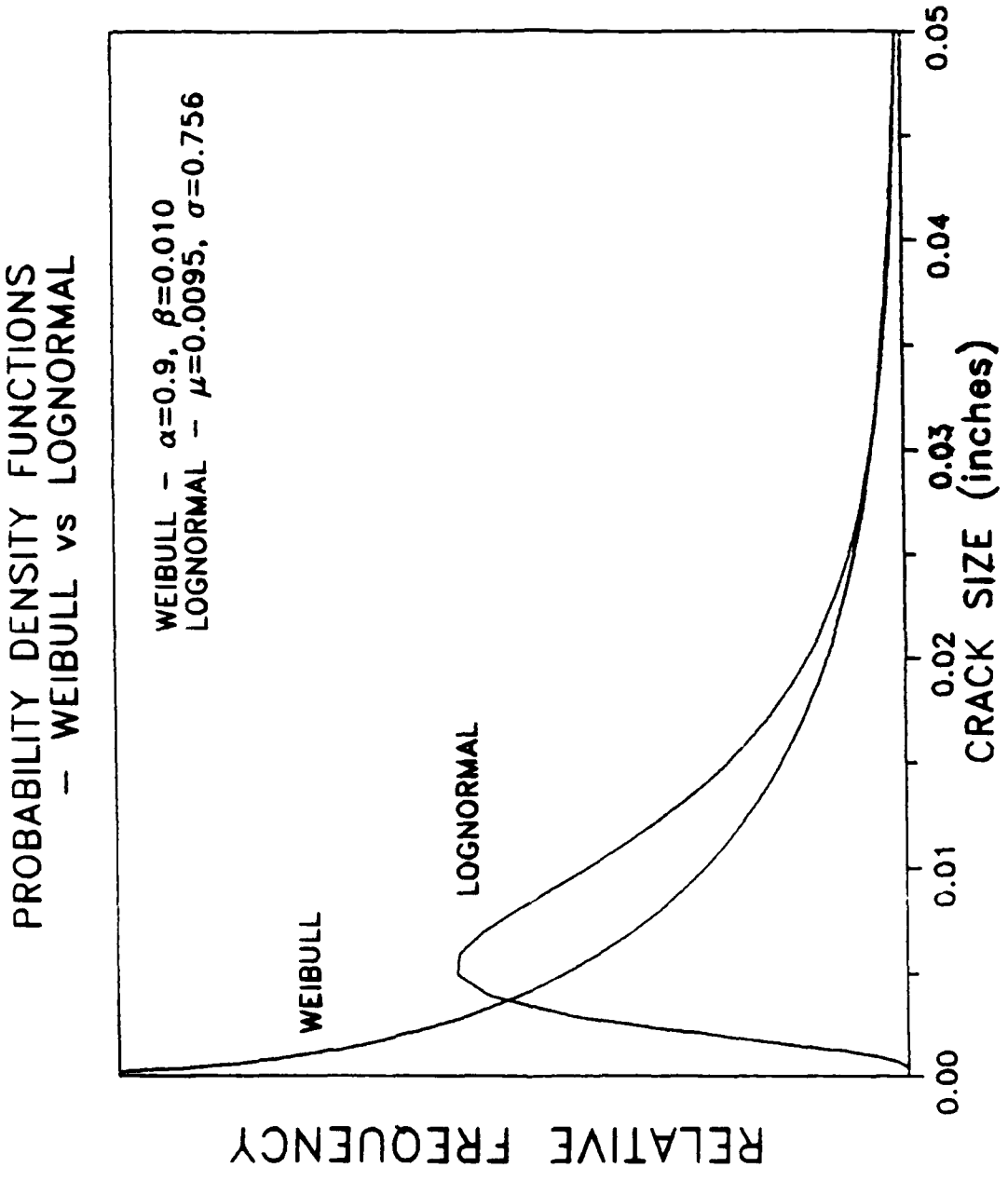

TEARDOWN INSPECTION – OBSERVED CRACK LENGTHS
SIX LOCATIONS PER WING
CRACKS DETECTED IN 50% OF LOCATIONS
OBSERVED LENGTH (MILS)
0
10
20
30
40
50
60
4000
6000
8000
10000
12000
14000
FLIGHT HOURS

MODELS FOR CRACK SIZE DISTRIBUTIONS

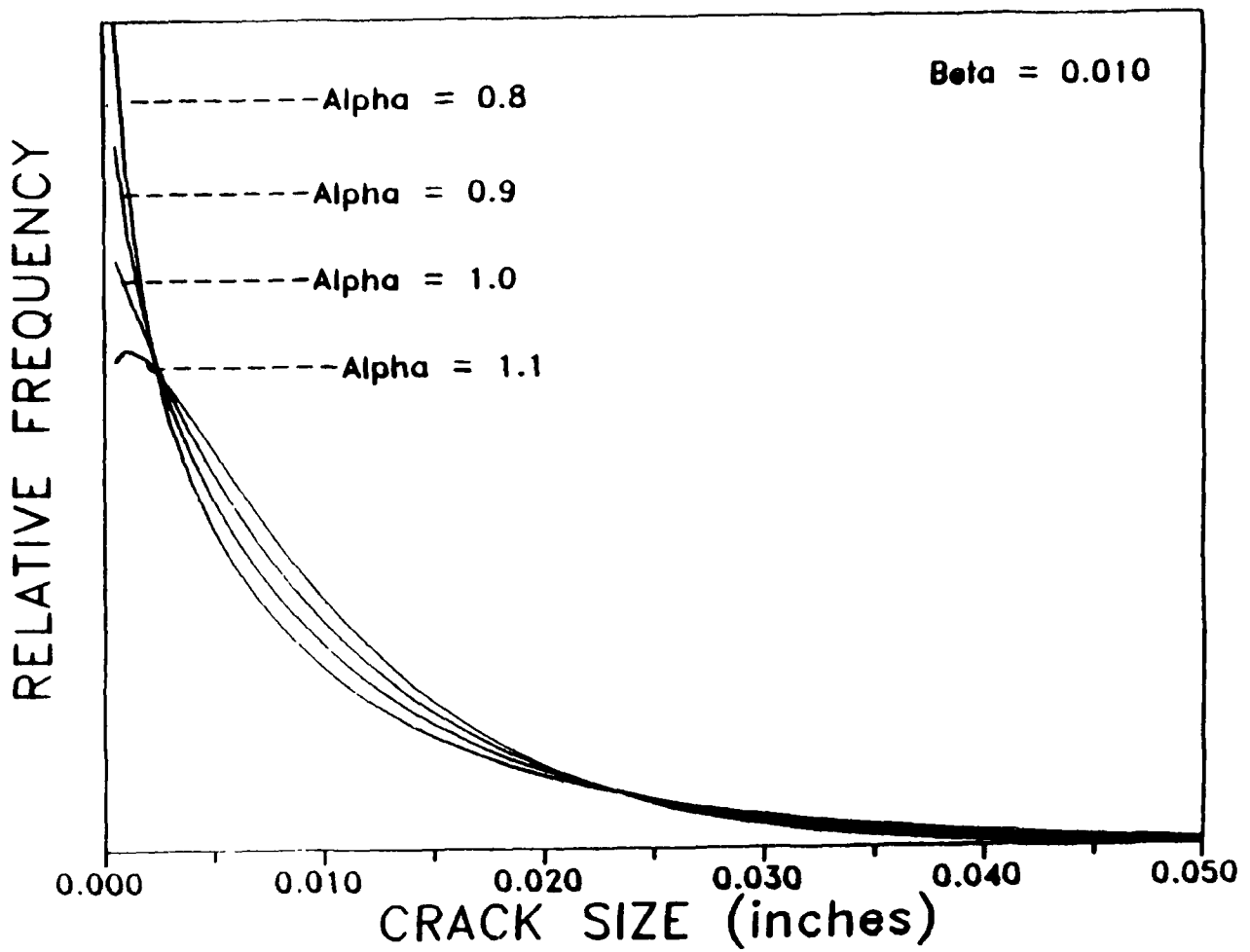
- SIMPLE MODEL
  - EVERY LOCATION HAS A CRACK
    - EQUIVALENT CRACK SIZE CONCEPT
  - PICK A FAMILY OF DISTRIBUTIONS
    - WEIBULL
    - LOGNORMAL
    - BETA
    - JOHNSON S_U
    - WEIBULL COMPATIBLE TIME TO CRACK INITIATION
- INSPECTION AND REPAIR LEAD TO MIXTURE MODELS

MODELS FOR CRACK SIZE DISTRIBUTION

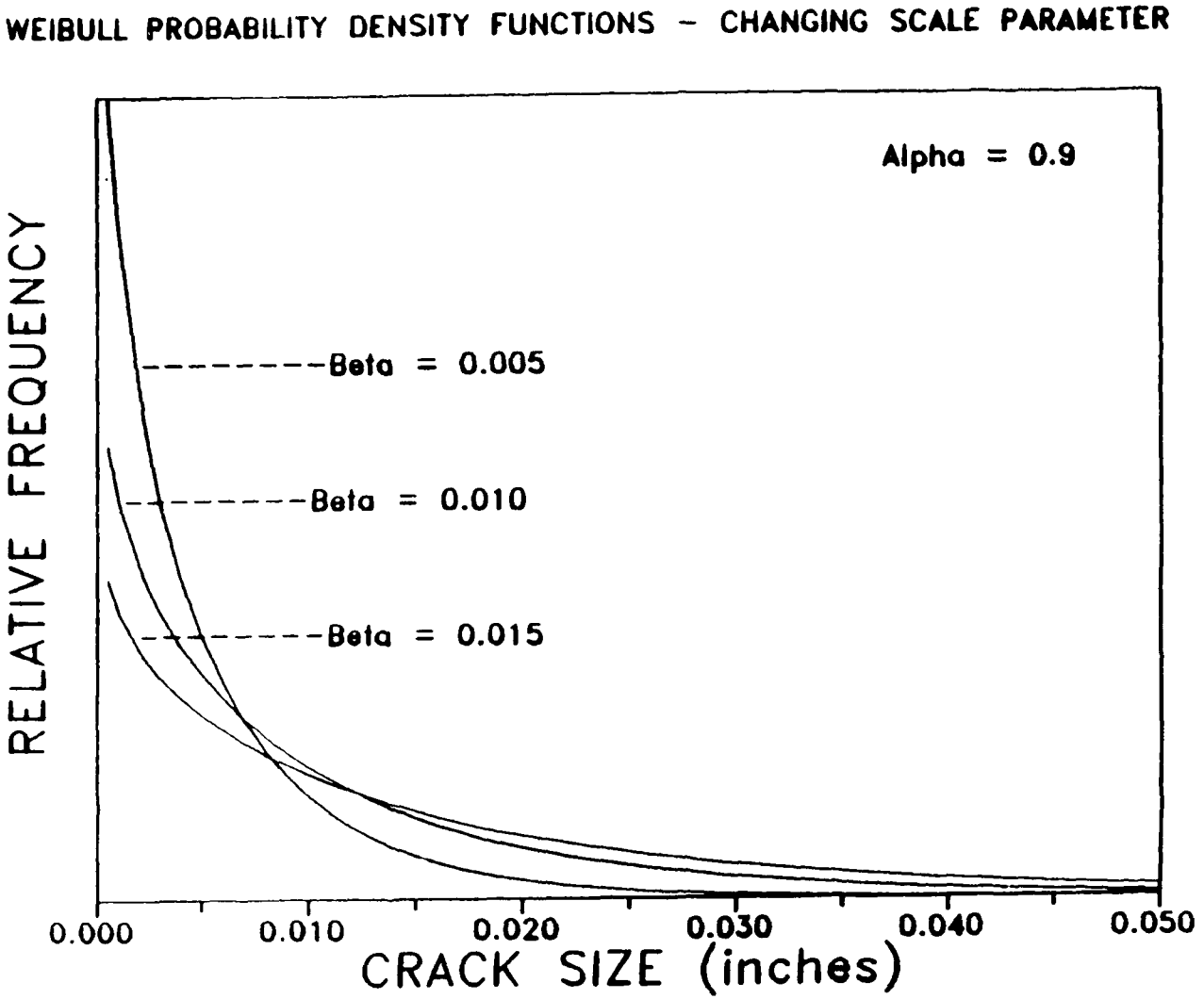
- MIXTURES
  - EVERY LOCATION HAS A CRACK
  - SIZES ARE A MIXTURE FROM DIFFERENT SOURCES
    - EQUIVALENT INITIAL FLAWS
    - "ROGUE" FLAWS
- SPECIAL MIXTURE
  - A PROPORTION, P, OF LOCATIONS HAVE CRACKS OF SIZE ZERO
  - PICK A FAMILY OF DISTRIBUTIONS FOR CRACKED LOCATIONS
  - PROBABILITY OF FAILURE CONDITIONED BY (1 -P)


PROBABILITY DENSITY FUNCTIONS
- WEIBULL vs LOGNORMAL
WEIBULL - α=0.9, β=0.010
LOGNORMAL - μ=0.0095, σ=0.756
WEIBULL
LOGNORMAL
RELATIVE FREQUENCY
0.00
0.01
0.02
0.03
0.04
0.05
CRACK SIZE (inches)

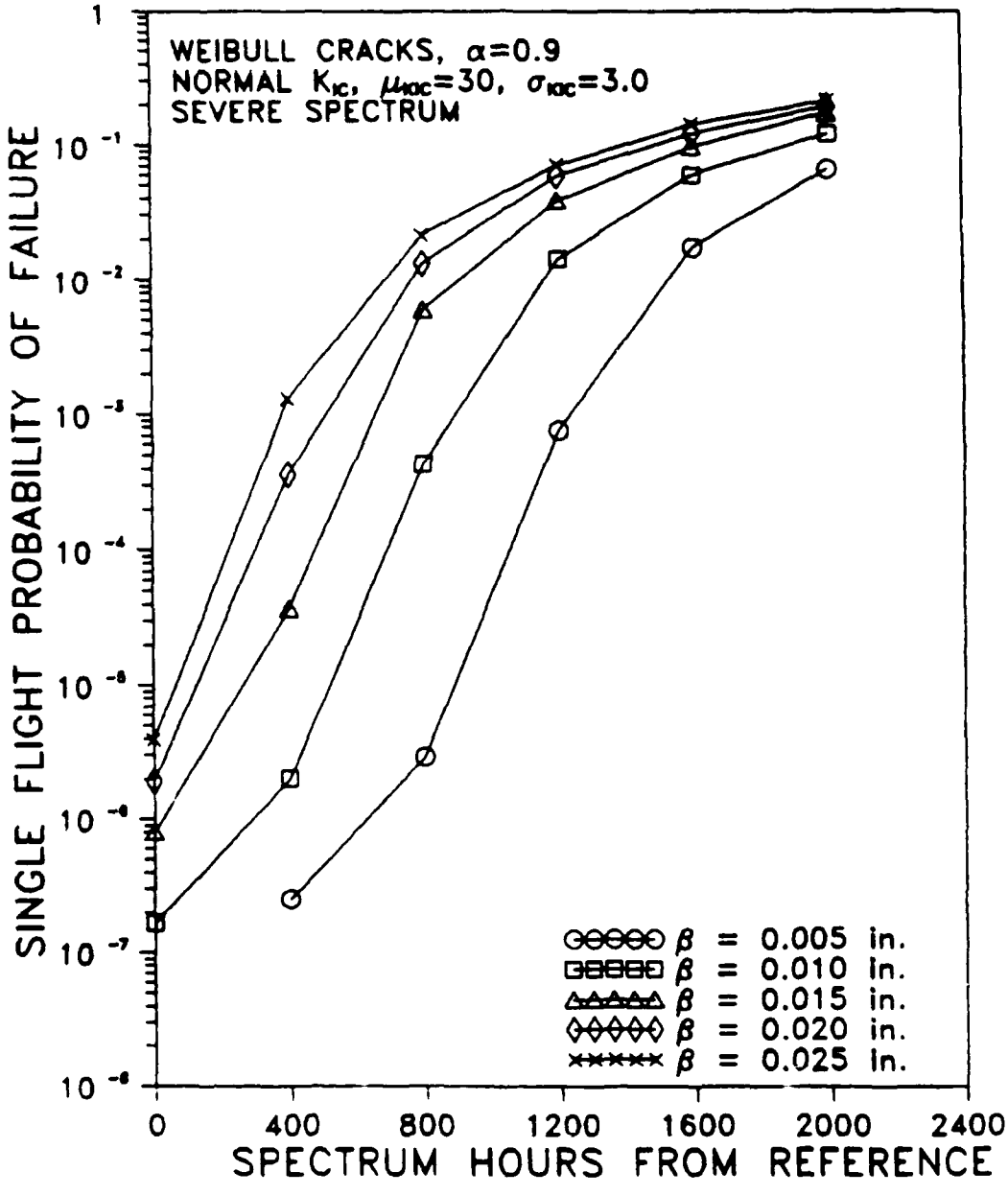
PARAMETRIC STUDY OF RISK PROBABILITIES
– CRACK SIZE DISTRIBUTION MODELS


WEIBULL PROBABILITY DENSITY FUNCTIONS - CHANGING SCALE PARAMETER
Beta = 0.010
Alpha = 0.8
Alpha = 0.9
Alpha = 1.0
Alpha = 1.1
RELATIVE FREQUENCY
0.000
0.010
0.020
0.030
0.040
0.050
CRACK SIZE (inches)

PARAMETRIC STUDY OF RISK PROBABILITIES
- CRACK SIZE DISTRIBUTION SHAPE
SINGLE FLIGHT PROBABILITY OF FAILURE
SPECTRUM HOURS FROM REFERENCE
WEIBULL CRACKS, β=0.010
NORMAL K_IC, μ_KIC=30, σ_KIC=3.0
SEVERE SPECTRUM
α = 0.8
α = 0.9
α = 1.0
α = 1.1


WEIBULL PROBABILITY DENSITY FUNCTIONS - CHANGING SCALE PARAMETER
Alpha = 0.9
Beta = 0.005
Beta = 0.010
Beta = 0.015
RELATIVE FREQUENCY
0.000
0.010
0.020
0.030
0.040
0.050
CRACK SIZE (inches)

PARAMETRIC STUDY OF RISK PROBABILITIES – CHARACTERISTIC CRACK SIZE



SUMMARY

- A/F/T APPLICATION
- GUMBEL'S EXTREME VALUE DISTRIBUTION PROVIDES MODEL FOR DISTRIBUTION OF MAX STRESS PER TIME PERIOD
  - Requires Only Stress Exceedance Rates
- TRADE-OFFS ON CRACK SIZE DISTRIBUTION
  - Method for Modeling Reality
  - Data
- LOW FAILURE PROBABILITIES SENSITIVE TO FACTORS CONSIDERED

PROBABILISTIC METHODS FOR INSPECTION PLANNING AND LIFE EXTENSION OF AN AGING AIRCRAFT FLEET.

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ABSTRACT

The fatigue limit state is the governing limit state for an aging air-frame. The trend of operating air-crafts longer than their originally planned life, calls for extensive testing and inspection. The paper addresses the problem of using this information from inspections and load surveys in a rational way. It is demonstrated how probabilistic fatigue and fracture analysis can provide such a rationale for decisions as to when to inspect, what inspection qualities is required and how long can life be extended in a cost effective way with maintained reliability.

INTRODUCTION

The trend over the past decades has been to operate air-crafts longer than their originally planed life. Continuing service requires extensive analysis and testing to establish airframe inspection and modification action required to ensure the desired new design life.

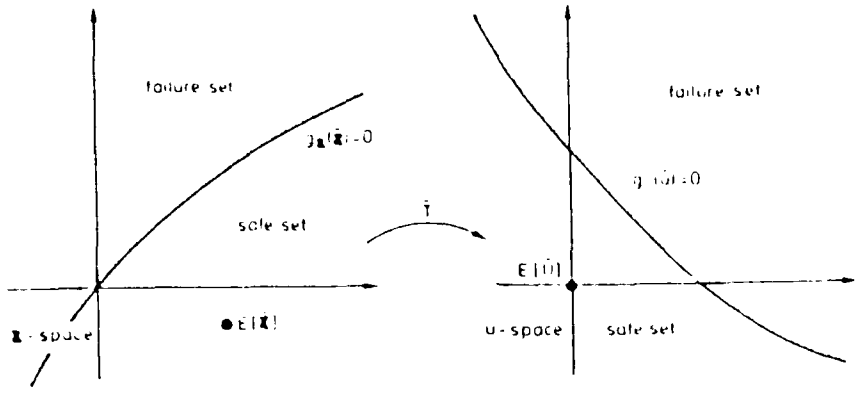
The present durability and damage tolerance design requirements are based on fatigue crack growth predictions and tracking. The durability and damage tolerance requirements assume the existence of a fixed deterministic crack size in the structural details after fabrication. The in-service inspection requirements are than established from further assumptions on deterministic crack growth rates using assumed loads from individual aircraft tracking or loads environmental spectrum survey data. The critical crack size is also established deterministically from the design limit stress.

Contrary to this deterministic approach the paper will address the problem of inspection planning and possible life extension from a probabilistic viewpoint. The initial crack sizes and the crack growth laws will be modeled as stochastic, uncertainties in the load spectrum will be accounted for as well as uncertainties in the critical crack sizes. Inspection methods will also be modeled with its associated uncertainties in detection probabilities. It will thus be demonstrated that probabilistic methods represent a methodology where all aspects of the deterioration process can be modeled, and is also a methodology where in-service inspection results can be accounted for. Trade-off studies can therefore be made between all parameters influencing the fatigue reliability of aging airframes.

RELIABILITY METHODS

Uncertainties are always the main motivation for inspections. Uncertainties governing the fatigue limit state stems from uncertainties in the environmental description, the load and response analysis/models, the detail stress analysis and the material parameters. To treat all uncertainties in a consistent way structural reliability models can be applied ,see e.g Madsen et.al.(1986). In a structural reliability formulation the fundamental notion is the limit state function ($g(\mathbf{z})$), the failure criterion, which divides the set of the random variables $\mathbf{z}$ into the failure set ($g(\mathbf{z})<0$) and the safe set ($g(\mathbf{z})>0$), see figure(1) for illustration. The safety is assured by requiring that the probability content of the failure set P_F is less than a small number. A corresponding reliability measure is the reliability index $\beta_R = -\Phi(P_F)$. Here $\Phi()$ is the standard normal distribution. Fast computer programs for calculating this probability, together with a rich variety of importance ,see e.g Madsen(1987), and sensitivity measure are now available, e.g. Proban(1986). The parametric sensitivity parameters includes such measures as $d\beta/d\tau$,

where r can be any distribution parameter of the random variables or a fixed/deterministic variable. This is an important property of the First Order Reliability Methods (FORM) which are the methods of particular importance in resource allocation models and trade-off studies.



Figure(1): First and Second Order Reliability methods maps the basic stochastic variables into a space of standard normal variables.

FATIGUE CRACK GROWTH MODEL

In a linear elastic fracture mechanics approach the increment in crack size, Δa, during a load cycle is related to the range of the stress intensity factor, ΔK, for the load cycle. A simple relation which is sufficient for most purposes, was proposed by Paris and Erdogan(1963),

$$\Delta a = C(\Delta K)^m, \quad \Delta K > 0 \tag{1}$$

The crack growth equation is used without a positive lower threshold on ΔK below which no crack growth occurs. The equation was proposed based on experimental results, but is also the result of various mechanical and energy based models, see e.g. Wirching et.al.(1987), Madsen et.al.(1987). *C and m are material constants.* The crack increment in one cycle is generally very small compared to the crack size and (1) is consequently written in a 'kinetic' form as

$$\frac{da}{dN} = C(\Delta K)^m, \quad \Delta K > 0 \tag{2}$$

where N is the number of stress cycles. The stress intensity factor K is computed by linear elastic fracture mechanics and is expressed as

$$K = \sigma Y(a)\sqrt{\pi a} \tag{3}$$

where σ is the far-field stress and $Y(a)$ is the geometry function. The geometry function depends on the overall geometry including the geometry of the crack. To explicitly account for uncertainties in the calculation of K, the geometry function is written as $Y(a) = Y(a,\mathbf{Y})$, where $\mathbf{Y}$ is a vector of random parameters. Inserting (3) in (2) and separating the variables leads to the differential equation

$$\frac{da}{Y(a,\mathbf{Y})^m(\sqrt{\pi a})^m} = C\sigma^m dN, \quad a(0) = a_0 \tag{4}$$

where a_0 is the initial crack size. The model can be extrapolated to variable amplitude loading when the appropriate value for S is inserted for each stress cycle. The extrapolation neglects possible sequence effects. Introducing the damage function $\Psi(a) = \int_{a_0}^{a} \frac{dx}{Y(x,\mathbf{Y})^m(\sqrt{\pi x})^m}$ the increment in $\Psi(a)$ in one stress cycle is

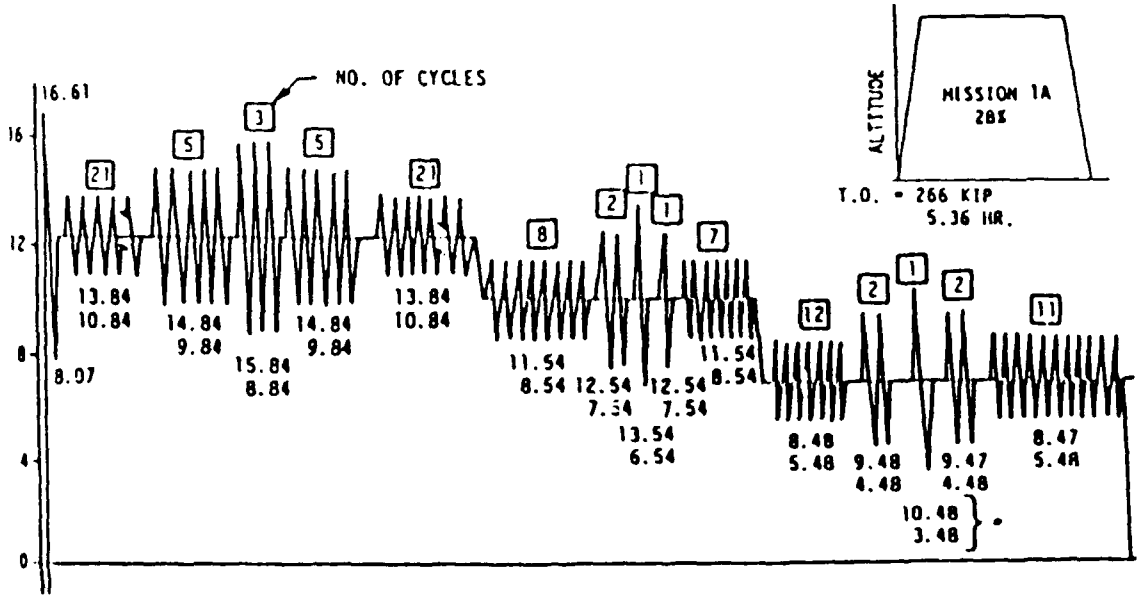
$$\Delta\Psi(a) = CS_i^m \tag{6}$$

The value of $\Psi(a)$ after N cycles is

$$\Psi(a_N) = C\sum_{i=1}^{N} S_i^m \tag{7}$$

The stress process is a random process and each stress range is a random variable. $\sum_{i=1}^{N} S_i^m$ is thus a random variable. When N is a large number the uncertainty in the sum can be neglected and the sum can be replaced by its expected value $NE[S^m]$. Depending on the distribution of the stress ranges, S, the value of the $m'th$ expected moment can be determined by an appropriate expression.

The random stress range must be modeled from the uncertainty in the environmental description, global response analysis and local stress analysis. Environmental description would account for the uncertainties in the load spectra during, climb, cruise, landing, touch and go etc., see figure(2) for illustration.



Figure(2): Illustration of uncertainties in environmental model, leading to uncertain load distribution.

Two failure criteria are of interest to formulate

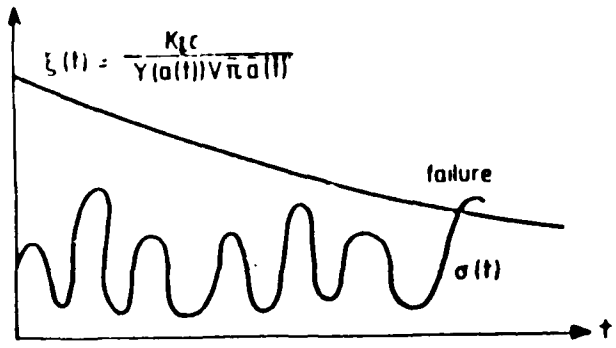
$$a_c - a_i \leq 0 \tag{8}$$

$$K_{IC} - K(a_i) \leq 0 \tag{9}$$

In the first case a critical crack size a_c is selected. In the second case failure occurs when the stress intensity factor exceeds the fracture toughness K_{IC} Then the crack growth becomes unstable and rapid failure occurs if

$$\sigma > \frac{K_{IC}}{Y(a,\mathbf{Y})\sqrt{\pi a}} \tag{10}$$

where $\sigma = \sigma(t)$ is the far field stress. This is illustrated in figure(3), failure does not occur in the time period [0,T] if the stress process $\sigma(t)$ is below the time varying threshold $\xi(t) = K_{IC}/[Y(a(t),\mathbf{Y})\sqrt{\pi a(t)}$ in [0,T] an approximation of this probability can be found in Madsen et.al(1986) and Madsen and Skjong(1987).



Figure(3): Illustration of failure event for brittle fracture under variable amplitude loading.

In the sequel the failure criterion is taken as the exceedance of a critical crack size a_C in time period t_i

$$a_c - a_i \leq 0 \tag{11}$$

since $\Psi(a)$ is monotonically increasing the failure criterion can be reformulated as

$$\Psi(a_C) - \Psi(a_0) = \int_{a_0}^{a_C} \frac{dx}{Y(x,\mathbf{Y})^m (\sqrt{\pi x})^m} - CNE[S^m] \leq 0 \tag{12}$$

The safety margin M is defined as

$$M = \int_{a_0}^{a_C} \frac{dx}{Y(x,\mathbf{Y})^m (\sqrt{\pi x})^m} - CNE[S^m] \leq 0 \tag{13}$$

Since $\mathbf{Y}, m, C$, parameters in the environmental description, the local stress analysis etc. are random M is random and the failure probability P_F is

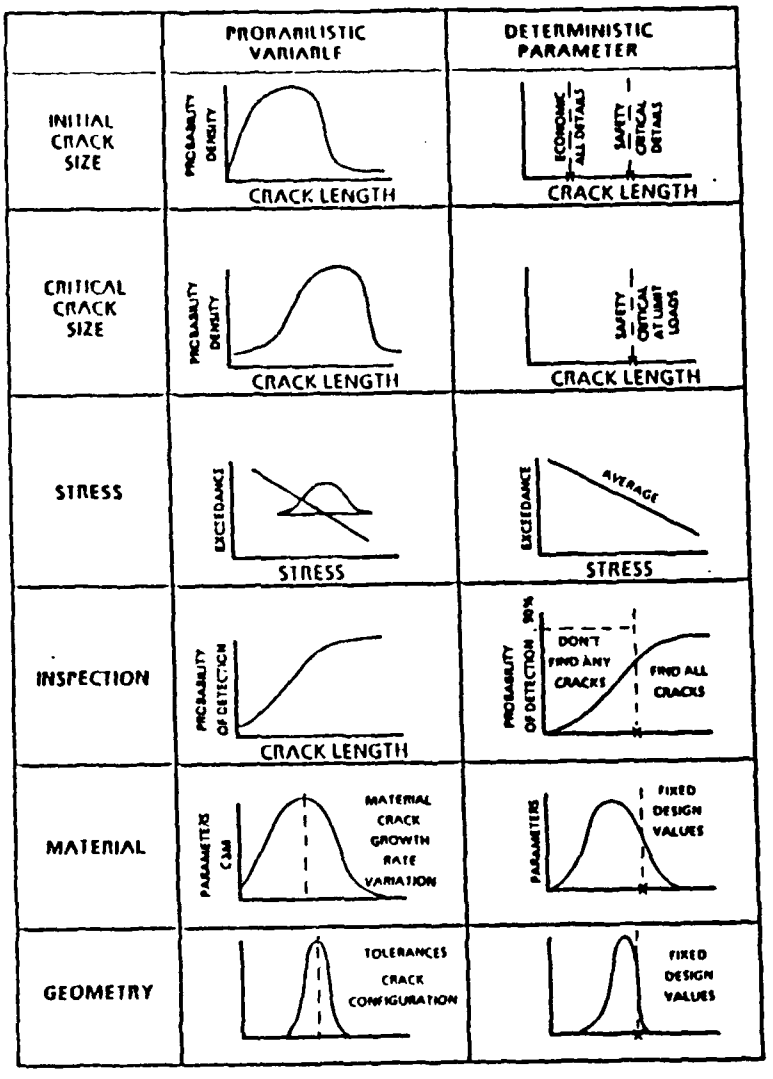
$$P_F = P(M \leq 0) \tag{14}$$

Figure(4) illustrates the most important uncertainties that is modeled in a probabilistic fatigue model compared to deterministic practice. Some of the uncertainties would have to be broken down to a lower level, e.g. the stress distribution, see figure(2).

Crack propagation time histories from experimental studies are shown in figure(5), for constant amplitude loading, illustrating the statistical variability in the crack growth damage accumulation. In the observations the sample curves are all different and irregular and shows a great deal of intermingling. The results indicate that the material is inhomogeneous and that crack growth varies through zones of varying resistance. As a consequence material parameter C can be randomized as

$$C = C(a) = \frac{C_1}{C_2(a)} \tag{15}$$

where C_1 is a random variable modeling variations in C from specimen to specimen, while $C_2(a)$ is a random homogeneous process describing variations from the mean value of C_1 along a crack path. As described in Madsen and Skjong (1987) this gives good agreement with experimental results.



Figure(4): Illustration of difference between probabilistic and deterministic models.

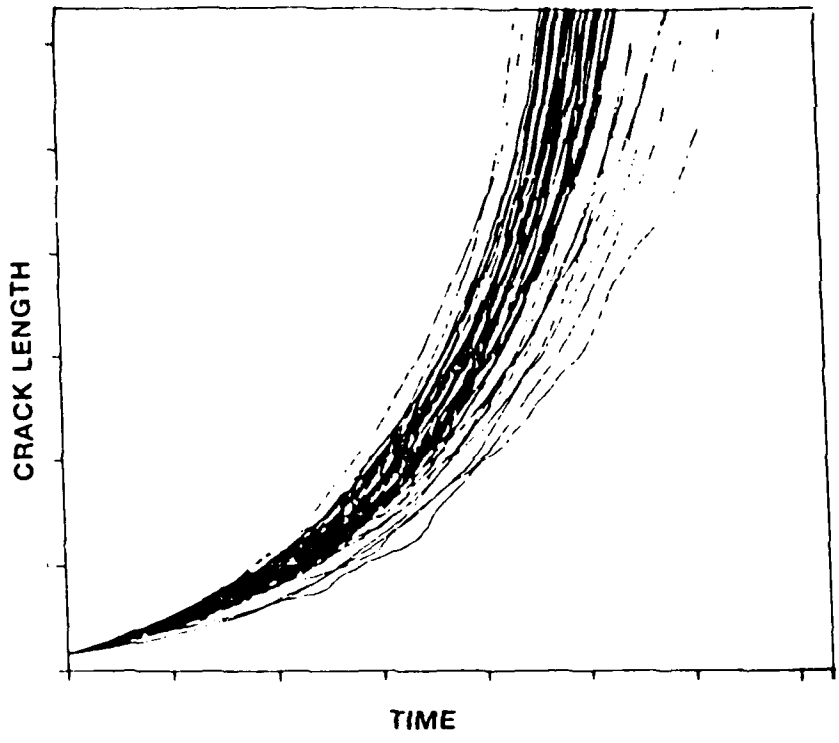
EVENT MARGINS FOR INSPECTION RESULTS

For in-service inspections two types of inspection results are considered

$$a_i \leq A_{di}, \quad i=1,2,\cdots,r \tag{16}$$

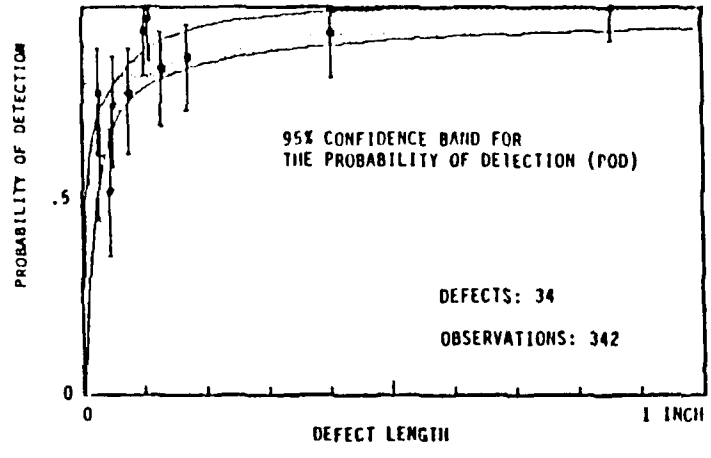
$$a_j = A_j, \quad j=1,2,\cdots,s \tag{17}$$

In the first case, (16), no crack was found in the inspection at time t_i, implying that the crack size was smaller than the smallest detectable crack size A_{di}. A_{di} is generally random since a detectable crack is only detected with a certain probability depending on the crack size and on the inspection method. The distribution of A_{di} is the distribution of non-detected cracks. Information of the type (16) can be envisaged for several times. If A_{di} is deterministic, however, and the same for all inspections, the information in the latest observation contains all the information of the previous ones. In the second case, (17), a crack size A_j is observed after time t_j. A_j is generally random due to measurement error and/or due to uncertainties in the interpretation of a measured signal as a crack size. Measurements of the type (17) can also be envisaged for several times corresponding to several values of t_j. Figure(6) is an illustration of uncertainties in probability of detection (POD). Each inspection type is characterized by its POD curve.



Figure(5): Actual crack growth propagation history mimicked by the proposed model.

usually reliable POD curves must be established from tear-down inspection results.



Figure(6): Typical Probability of detection curve

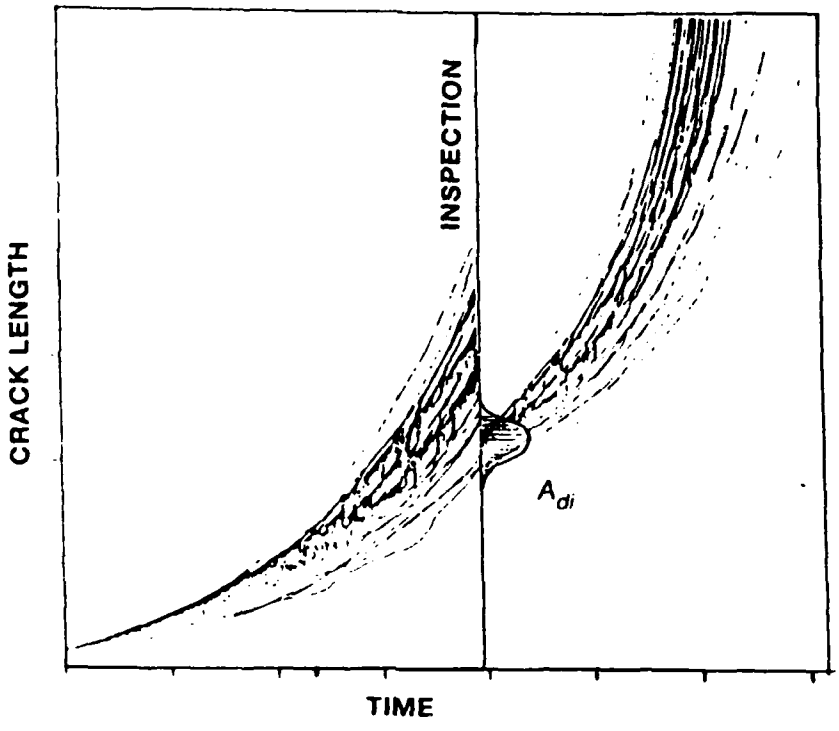
For each measurement (16) an event margin M_i can be defined similar to the safety margin (14) as

$$M_i = NE[S^m] - \int_{a_0}^{A_{d_i}} \frac{dx}{Y(x,\mathbf{Y})^m (\sqrt{\pi x})^m} \leq 0, \quad i=1,2,\cdots,r \tag{18}$$

These event margins are negative due to (16). For each measurement (17) an event margin can similarly be defined as

$$M_j = \int_{a_0}^{A_j} \frac{dx}{Y(x,\mathbf{Y})^m (\sqrt{\pi x})^m} - NE[S^m] = 0, \quad j=1,2,\cdots,s \tag{19}$$

These safety margins are zero due to (17). An inspection where a crack is found is illustrated in figure(6). If the crack was less than predicted by the model, this would still increase the reliability.



Figure(7): Illustration of updating of reliability when a crack is found. The crack growth curves that could be outcomes, will be filtered by the updating.

The situation is envisaged where no crack is detected in the first r inspections at a location, while a crack is detected by the $r+1$'th inspection and its size is measured at this and the following $s-1$ inspections. The updated failure probability is in this case

$$P_F = P(M \leq 0 \,|\, M_1 \leq 0 \cap \cdots \cap M_r \leq 0 \cap M_{r+1} = 0 \cap \ldots \cap M_{r+s} = 0) \tag{20}$$

The calculation of these updated reliabilities can be performed as described in Madsen(1985), Madsen et.al(1987), Proban(1986).

A more general situation involves simultaneous consideration of several locations with potentially dangerous cracks for which inspections are carried out. The updating procedure still applies when due consideration is taken to the dependence between basic variables referring to different locations. The same applies for situations where structural integrity is restored by substituting the detail with a spare part.

If the deteriorating structural part is substituted at time t_i the consequence of this can easily be modeled in terms of limit states and event margins.

The limit state is now

$$Mi,j = \int_{a_0}^{a_C} \frac{dx}{Y(x,\mathbf{Y})^m (\sqrt{\pi x})^m} - C(N_j - N_i)\, E[S^m] \leq 0 \tag{21}$$

modeling the case of failure before time t_j when the component integrity was restored at t_i.

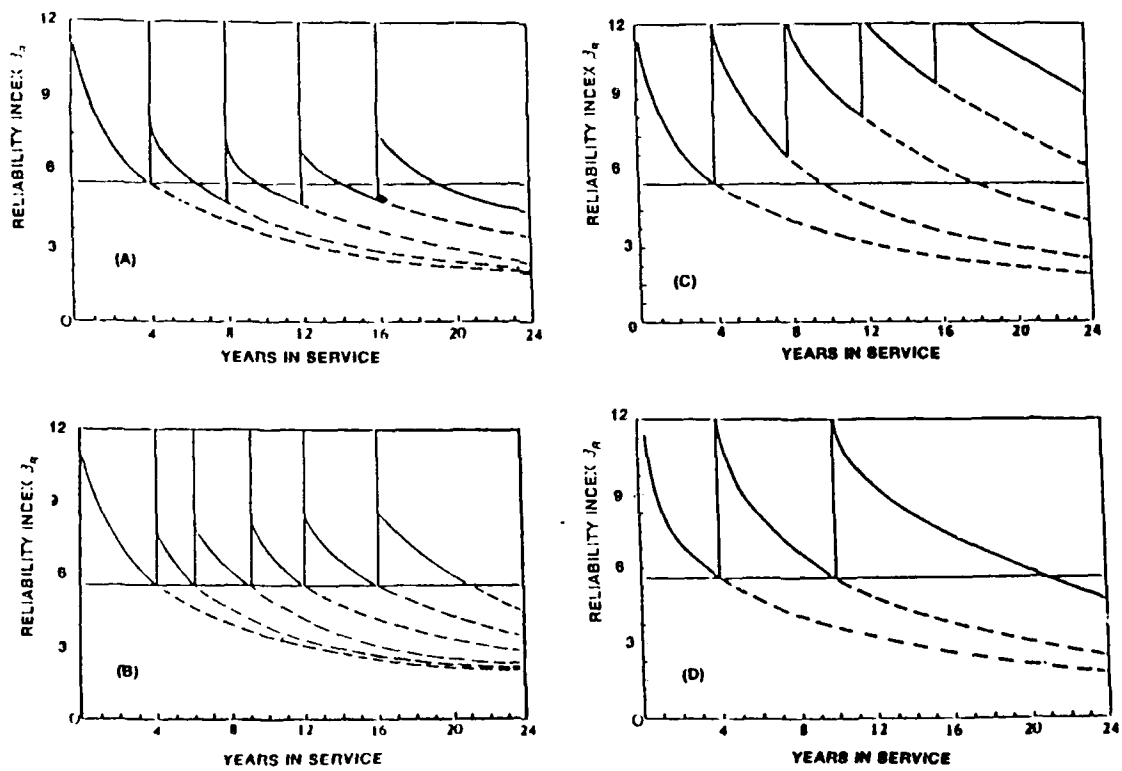
The observed event of failure at time t_i do contain valuable information that can be modeled as

$$M_i = N_i E[S^m] - \int_{a_0}^{A_{di}} \frac{dx}{Y(x,\mathbf{Y})^m (\sqrt{\pi x})^m} \leq 0, \tag{22}$$

where A_{d_i} is the crack size of the detected crack.

Contrary to the previous example the material parameters now are partly or totally independent (some dependence of the to parts if they came from the same batch) in the limit state and the event margin corresponding to the observation. The load parameters would also be independent, while the parameters in the transfer functions and the local stress analysis are the same (fully correlated). If the reason for the failure was model uncertainties in the calculation of the transfer functions the observation contains information with high influence on the updated reliability.

Figure(8) is an illustration of the flexibility obtained by using probabilistic methods in inspection planning. While the traditional approach of equally separated inspections leads to increase in reliability when now crack is found with a reliable inspection method, the probabilistic methods gives an accurate assessment of when the next inspection must be performed to maintain the reliability based on all available (uncertain) information. The method will be equivalent with making maximum use of existing structural integrity reports, data bases and inspection records.



Figure(8): Illustration of probabilistic updating techniques when no crack was found. A and B with a poor inspection method, C and D with a better inspection method. B and C with optimized inspection intervals

A SIMPLE EXAMPLE

Consider a panel with a center crack as in the experiments of Virkler et.al(1979). The loading is a constant amplitude loading leading to a far-field stress range S. The geometry function is modeled as

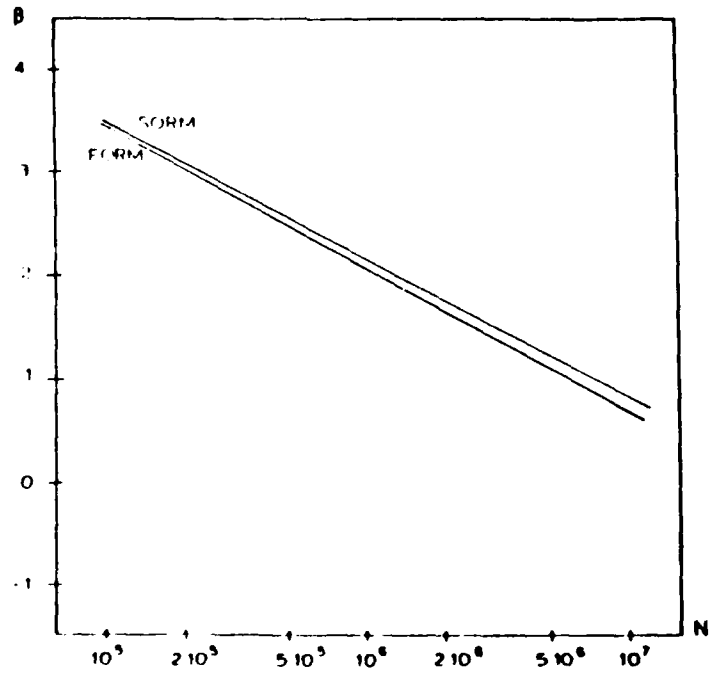
$$Y(a,\mathbf{Y}) = \exp(Y_1(\frac{a}{50})^{Y_2}) \tag{23}$$

The geometry function takes the value one for $a=0$. (Lengths are measured in mm and stresses in N/mm^2). The distribution of the basic variables is taken as

$$\begin{cases} S \in N(60, 10^2) \\ Y_1 \in LN(1, 0.2^2) \\ Y_2 \in LN(2, 0.1^2) \\ a_0 \in EX(1) \\ a_C \in N(50, 10^2) \\ (\ln C_1, m) \in N_2(-33.00, 0.47^2, 3.5, 0.3^2; -0.9) \end{cases} \tag{24}$$

$N(\mu,\sigma^2)$ denotes a normal distribution with mean value μ and variance σ^2. Similarly $LN(\mu,\sigma^2)$ denotes a log-normal distribution with mean value μ and variance σ^2. $N_2(\mu_1,\sigma_1^2,\mu_2,\sigma_2^2;\rho)$ denotes a bi-normal distribution with mean values μ_1 and μ_2, variances σ_1^2 and σ_2^2 and correlation coefficient ρ. $EX(\mu)$ denotes an exponential distribution with mean value μ. Statistics for $C_2(a)$ are taken as those reported in Ortiz and Kiremidjian(1985), see section 4 of these notes.

The first-order and second-order approximations to the reliability index are shown in figure(9) for various life times expressed in terms of the number of stress cycles N. The two approximations are close implying that the curvatures of the limit state surface are moderate at the design point.



Figure(9): First- and second-order reliability index from design calculation.

Statistics for the distribution of life time T can be directly approximated from the results of figure(9). For the mean life times the approximation is

$$E[T] = \int_0^\infty (1 - P(T \le t))dt \approx \int_0^\infty \Phi(\beta(t))dt \qquad (25)$$

For $N=1.5\cdot10^6$ stress cycles the reliability index is $\beta=1.817$ and the α's are shown in Table(1). The α_i^2 can be interpreted as the contribution to the total uncertainty from variable i.

Table(1): Sensitivity Factors
$N=1.5\cdot10^6$, $\beta=1.817$

Variable	α_i	α_i^2
S	0.3577	13%
Y_1	0.0085	0%
Y_2	-0.0060	0%
a_0	0.5514	30%
a_C	-0.0001	0%
m	-0.6141	38%
$C_1 \mid m$	0.4362	19%
$\Psi(a_C) \mid a_0, a_C, m, Y_{1,Y_2}$	-0.0248	0%

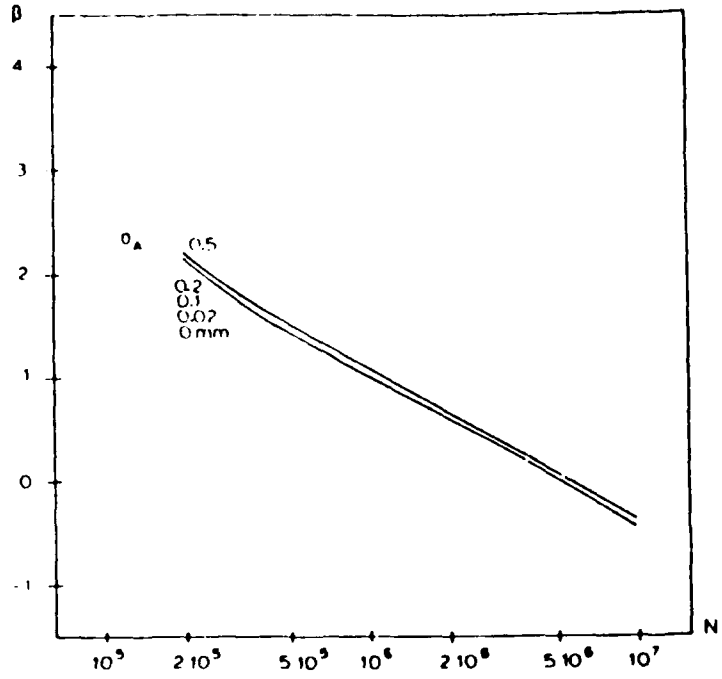
The major sources of uncertainty in this example are thus the material parameters m and C_1, the initial crack size a_0, and the stress range S. The FORM gives a derivative $d\beta/d\mu$ for S as -0.0358 it follows that a change in the mean value of S of $-5MPa$ leads to a change in β of

$$\Delta\beta \approx -0.0358(-5) = 0.179 \qquad (26)$$

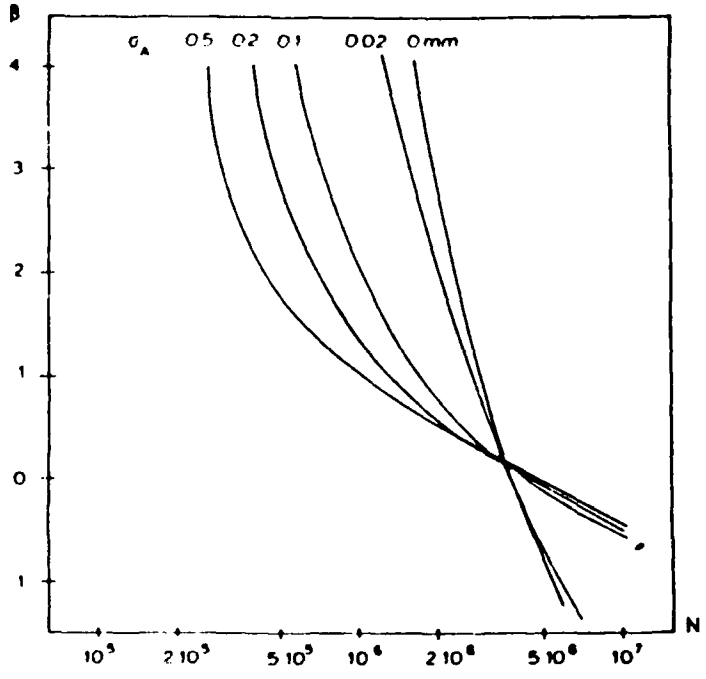
The reliability index is thus changed to approximately 1.996, while re-running the analysis gives $\beta=2.001$. FORM gives derivatives of all input parameters (distribution parameters and fixed variables).

Next, the situation where a crack is found in the first inspection is considered. It is envisaged that the inspection is carried out after $N_1=10^5$ stress cycles and a crack length of 3.9 mm is measured. The measurement error is assumed to be normally distributed with standard deviation σ_A. Figure(10) shows the updated reliability index as a function of σ_A, when (20) has been applied with $(r,s)=(0,1)$. The result is almost independent of σ_A in this example as the uncertainty in the initial crack size is dominating. When the crack is detected a decision has to be made. Two options are present. It may be decided to immediately repair the crack or to leave the crack as it is and base a decision on repair on more inspection results. Due to the large initial crack size uncertainty it is not possible with just one inspection to determine if the crack was initially large but grows slowly enough that repair is not needed, or the crack was initially fairly small but is growing fast and must be repaired. If a requirement on the reliability index in a period without inspections is formulated, e.g., $\beta_R \ge 2$, the latest time of the next inspection is determined from figure(10).

Assume that the crack is not repaired but a second inspection at $N=2\cdot10^5$ stress cycles is required. Let the inspection method be the same as in the first inspection and let the measured crack size be 4.0 mm. The measurement error is again assumed to be normally distributed with standard deviation σ_A and the two measurement errors are assumed to be statistically independent. Figure(11) shows the updated reliability index after this second inspection. Different inspection qualities now lead to very different results. With $\sigma_A=0$ the negative slope of the reliability index curve becomes very large demonstrating that the crack growth behavior is basically determined by two combinations of the basic variables. With a large measurement uncertainty there is an immediate and large increase in reliability, but after some time the curve becomes almost identical to the curve resulting after the first inspection. Due to large uncertainty in both inspections only little information is gained on the crack growth rate. If the inspection quality is very high it may be possible to state that the crack does not grow to a critical size within the design life time. Repair and further inspections are then unnecessary. For a poorer inspection quality a time period until the next inspection can be determined and



Figure(10): Updated first-order reliability index after first inspection with crack measurement 3.9 mm.



Figure(11): Updated first-order reliability index after second inspection with crack measurements 3.9 mm and 4.0 mm.

the decision on repair be further delayed. Figure(11) illustrates how uncertainties effectively can be removed by high quality inspections.

RESOURCE ALLOCATION MODEL

As can be seen from the preceding models the methodology described so far can be used for inspection optimization in the sense that the optimal time to next inspection which maintains the reliability level can be derived as well as how long the life can be extended with the same reliability level. This is thus a highly relevant model for inspection planning when information of the airframe integrity is collected through inspection results.

This model is however, of less interest at the design stage and when considering economic in-service life extension. The model for such decisions has to bee quite different. At the design stage the trade-off study has to be made between design parameters like plate and stiffeners thicknesses and the cost of inspections in service. At the design stage no in-service inspection results are available and only expectations on inspection results can be calculated. The same considerations are valid when life extension is considered. All possible results of future inspections can, however, be modeled as previously described, e.g. Madsen(1988) and Soerensen(1988).

The basic safety margin for the fatigue sensitive detail is written as

$$M = \int_{a_0}^{a_c} \frac{dx}{Y(x,\mathbf{Y})^m (\pi x)^{m/2}} - CNE[S(z)^m] \tag{27}$$

where $S(z)$ now is a function of the design variable z.

If inspections are performed at times T_1 and T_2 with no repair at time T_1 and repair at T_2, the safety margin for failure time $t > T_2$ is

$$M_{01} = \int_{a_R}^{a_c} \frac{dx}{Y(x,\mathbf{Y})^m \sqrt{\pi x^m}} - C(N_t - N_{T_2})E[S(z)^m] \tag{28}$$

where a_R is the crack size after repair, a random variable.

The event margin corresponding to the event that a crack is found and repair is performed at the first inspection T_1 can be formulated as

$$R = \int_{a_0}^{a_{r_1}} \frac{dx}{Y(x,\mathbf{Y})^m \sqrt{\pi x^m})} - C N_{T_1} E[S(z)^m] \tag{29}$$

where a_{r_1} is the smallest detectable crack size.

Similarly the event margin corresponding to the event that repair is performed at the 3'rd inspection (time T_3) given repair at T_1 and no repair at T_2 is written as

$$R_{13} = \int_{a_R}^{a_{r_3}} \frac{dx}{Y(x,\mathbf{Y})^m \sqrt{\pi x^m}} - C(N_{T_3} - N_{T_1})E[S(z)^m] \tag{30}$$

Assuming that I inspections are performed at times T_i, $i \in [1,I]$ the reliability index β for failure before t is

$$\beta(T) = -\Phi^{-1}(P_F(t)) \tag{31}$$

where, for $0 < t \leq T_1$

$$P_F(t) = P(M(t) \leq 0) \tag{32}$$

and for $T_1<t\leq T_2$

$$P_F(t) = P_F(T_1)+P(M(T_1)>0\cap R>0\cap \gamma_0(t)\leq 0)+ \qquad (33)$$
$$P(M(T_1)>0\cap R\leq 0\cap R_1(t)\leq 0)$$

etc. for the paths in figure(12) of repair realizations. From this the expected number of repairs can be calculated $E[R_i]$.

Figure(12): Repair realizations for single elements. 0 denotes no repair while 1 denotes repair.

The inspection quality is modeled by treating the detectable crack size a_{r_i} as a random variable. The POD-curve (probability of detection) is assumed exponential

$$p_i(a) = F_{a_{r_i}}(a) = 1-\exp(aq_i), \qquad a>0 \qquad (34)$$

where q_i is a constant characterizing the reliability of the inspection method.

The resource allocation model is now formulated as

$$\min C(\mathbf{t},\mathbf{q},z) = C_I(z)+\sum_{i=1}^{I}(C_{IN}(q_i)+C_R E[R_i](1-\Delta P_F(T_{i-1},T_i))/(1+r)^{T_i} \qquad (35)$$
$$+ \sum_{i=1}^{I} C_F(T_i)\Delta P_F(T_{i-1},T_i)/(1+r)^{T_i}$$

subject to the reliability constraint $\beta(t)\geq\beta_{min}$, the minimum and maximum time between inspections $t_{min}\leq t_i\leq t_{max}$, the limitations on inspection quality $q_{min}\leq q_i\leq q_{max}$ and the limitations on the design variable $z_{min}\leq z\leq z_{max}$.

Here r is the real rate of return and the cost functions are initial cost $C_I(z)=C_{I0} + C_{I00}(z-z_0)$ (a function of the design variable), the inspection cost $C_{IN}(q_i)$ which could be a function of the inspection quality q_i, the repair cost C_R and the cost of failure C_F.

The control variables in the optimization formulation for a given design (z) are the inspection times $\mathbf{t}=(t_1,t_2,\ldots,t_I)$ and the inspection qualities $\mathbf{q}=(q_1,q_2,\ldots,q_I)$. By performing the optimization for different life times the optimum life extension can be found.

CONCLUSIONS

- A probabilistic fatigue crack growth model has been formulated with uncertainties assigned to material as well as load parameters and the environmental description.
- Methods for reliability updating during in-service has been demonstrated.
- Reliability methods provide a rationale for in service considerations of an aging airframe.
- Reliability analysis combined with a resource allocation technique provides data which are important for reduced life cycle costs of a air-frame considering design, in-service

inspection/repair and consequence costs of a possible failure.

- Cost optimum life extensions can be found by the model.

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Abstract: C-130 STRUCTURAL RISK ANALYSIS - CAPABILITY TO CONSERVATIVELY FORECAST INITIAL CRACK INSTABILITY
H. R. Horsburgh, LASC-Ga
W. O. Greenhaw, WR/ALC, MMSFRA

Risk analysis supplements durability and damage tolerance analysis for C-130 force management. For this purpose only relative risk of proposed alternatives to operations and inspections is required.

The Risky program described in ASD-TR-80-5035 is adapted to interface with the Aircraft Information Retrieval System at Warner Robins ALC, MMSFRA. Risk is calculated as the interaction of five data sets:

1) PDSCB - the applied stress probability distribution
2) SBC - allowable stress as a function of crack size
3) PCB - crack size probability distribution
4) AB - crack growth curve
5) PI - inspection reliability

The last four combine to define the allowable stress probability distribution which is combined with the first to evaluate risk. A zone is defined by a set of structural details for which each of the five data sets are essentially the same for each detail.

Because the relative risk of force management alternatives is required, only the single flight risk for a single aircraft is required where that flight is of the most severe type for the proposed usage. The applied stress probability distribution for that flight expresses the probability that at least one load encountered would exceed a given value. This is based on statistics gathered in C-130, C-141, and C-5 loads surveys. The "correct" allowable stress distribution for that flight must account for variation in material properties, manufacturing tolerances and fit, corrosion, and possible fatigue cracking as accrued up to the time of the flight.

Since the flaw size distribution must account for variation in properties in addition to fatigue crack size, it is expressed as a mathematically equivalent flaw size. Consequently, the appropriate flaw size distribution for the flight at which a risk calculation is

made is a function of the analytical model as well as the physical condition of the structure. Requirements for precision in the five data sets and the analytical model are alleviated by evaluating the relative risk of alternatives. In this case, it is only required that the forecast be conservative, while maintaining enough precision to appropriately reflect the consequences of decision. The model should not forecast a factor of ten when reality is a factor of two.

Lockheed

C-130 RISK ANALYSIS
OBJECTIVE

PROVIDE WARNER ROBINS ALC WITH A TOOL TO SUPPLEMENT DADTA

IN ASSESSING MERITS OF ALTERNATIVE MAINTENANCE ACTIONS

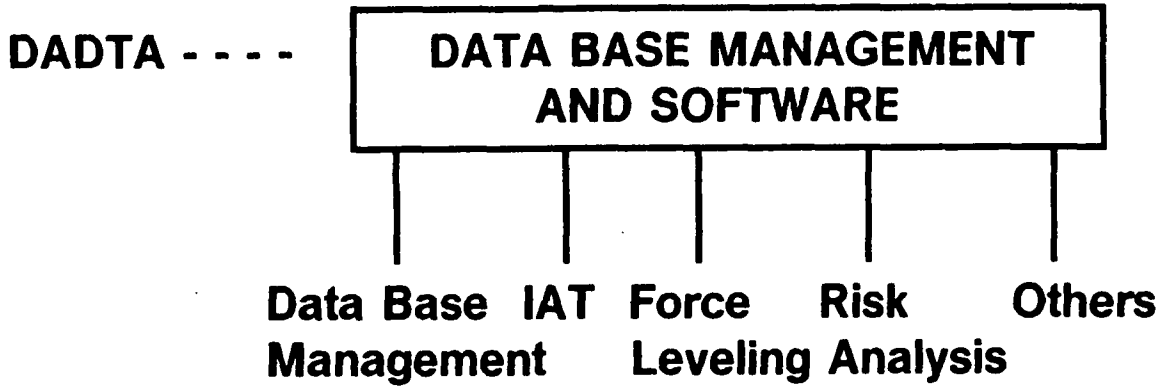
Lockheed

C-130 RISK ANALYSIS DEFINITION

PROBABILITY OF UNSTABLE CRACK PROPAGATION AT ANY SITE DURING THE NEXT FLIGHT

Lockheed

C-130 FORCE MANAGEMENT PRIMARY STRUCTURE



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C-130 RISK ANALYSIS PROCEDURES

- IDENTIFY CURRENT ANALYSIS ZONE	: SETS OF SITES ON THE STRUCTURE WITH THE SAME INPUT DATA
- ESTABLISH CONDITION OF STRUCTURE	: KNOWN.VS.UNKNOWN DAMAGE
- GET STRUCTURAL STATUS AND USAGE FROM DATA BASE	: INSPECTION HISTORY, CRITICAL MISSION, SEVERITY FACTOR
- EVALUATE RISK FOR FIRST ZONE	: PLOT OF RISK.VS.TIME
- ITERATE ALTERNATIVES	: USAGE, INSPECTIONS
- SUPERIMPOSE ADDITIONAL ZONES	

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C-130 RISK ANALYSIS
SINGLE FLIGHT RISK FOR SINGLE POINT

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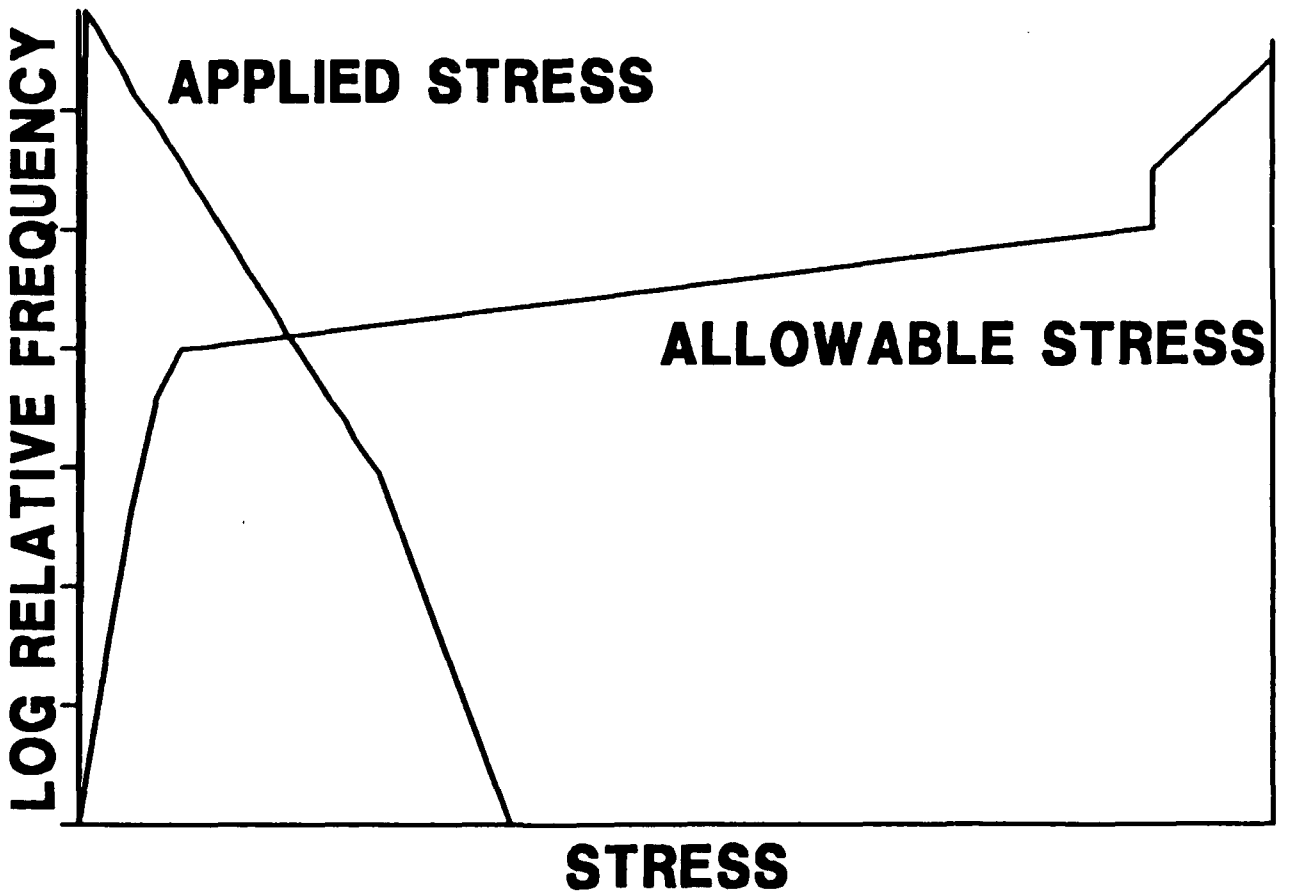
C-130 RISK ANALYSIS
SINGLE FLIGHT RISK FOR SINGLE POINT



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C-130 RISK ANALYSIS
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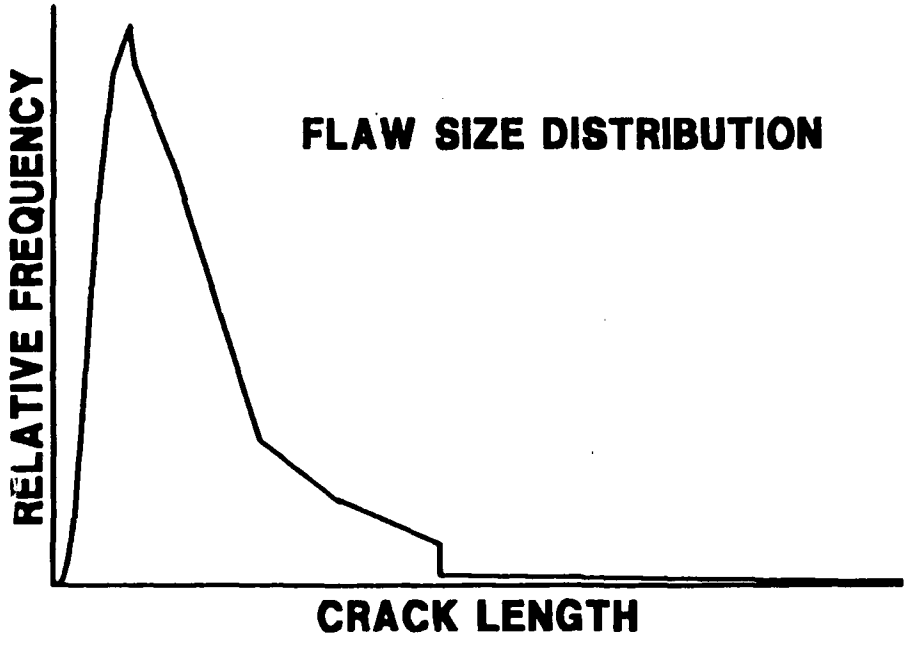

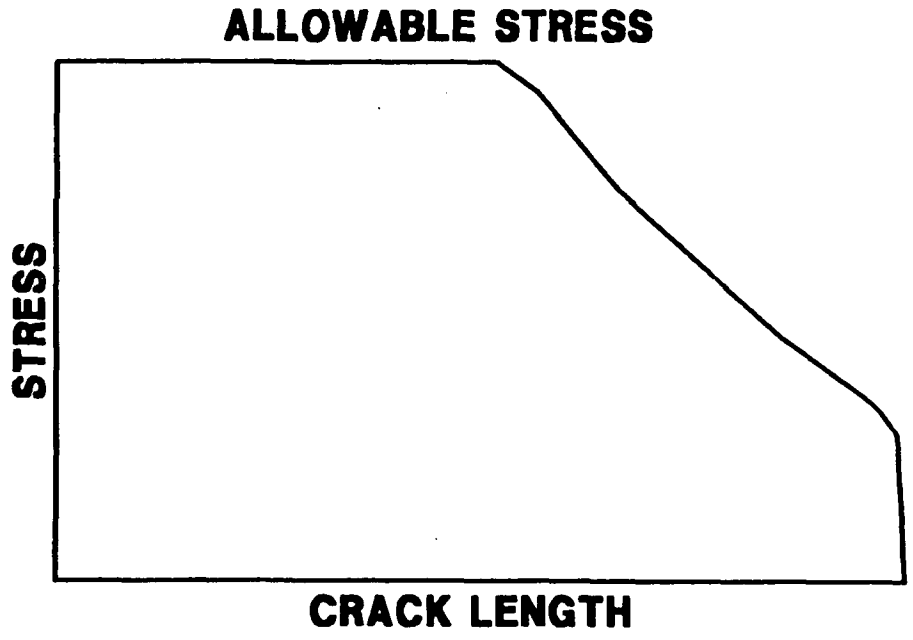

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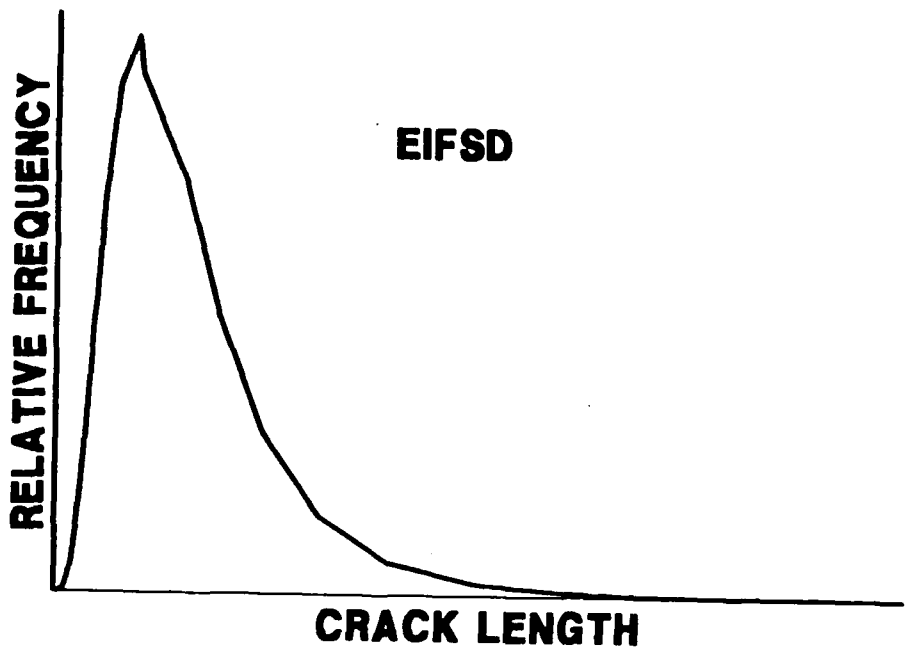

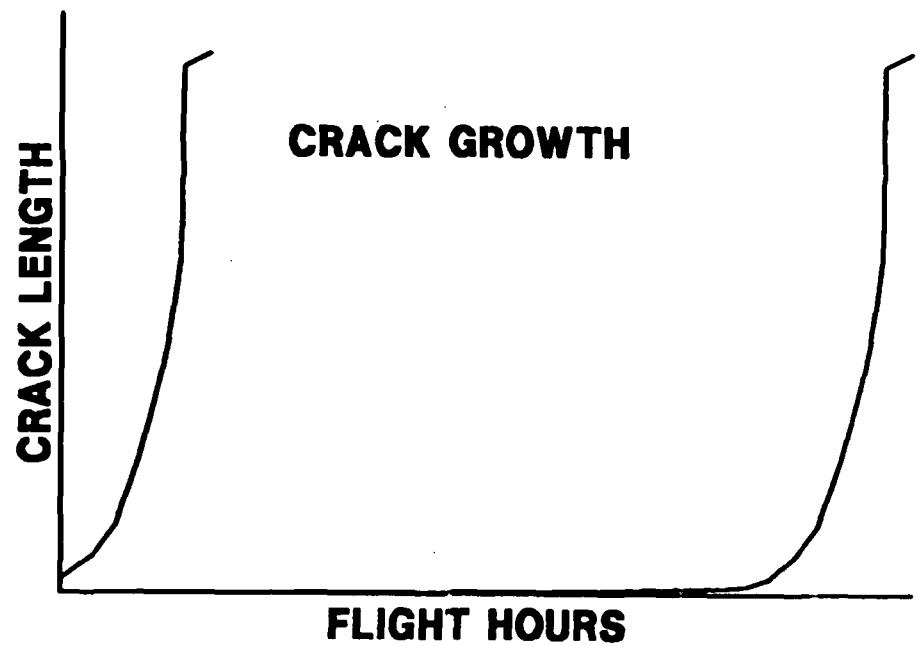

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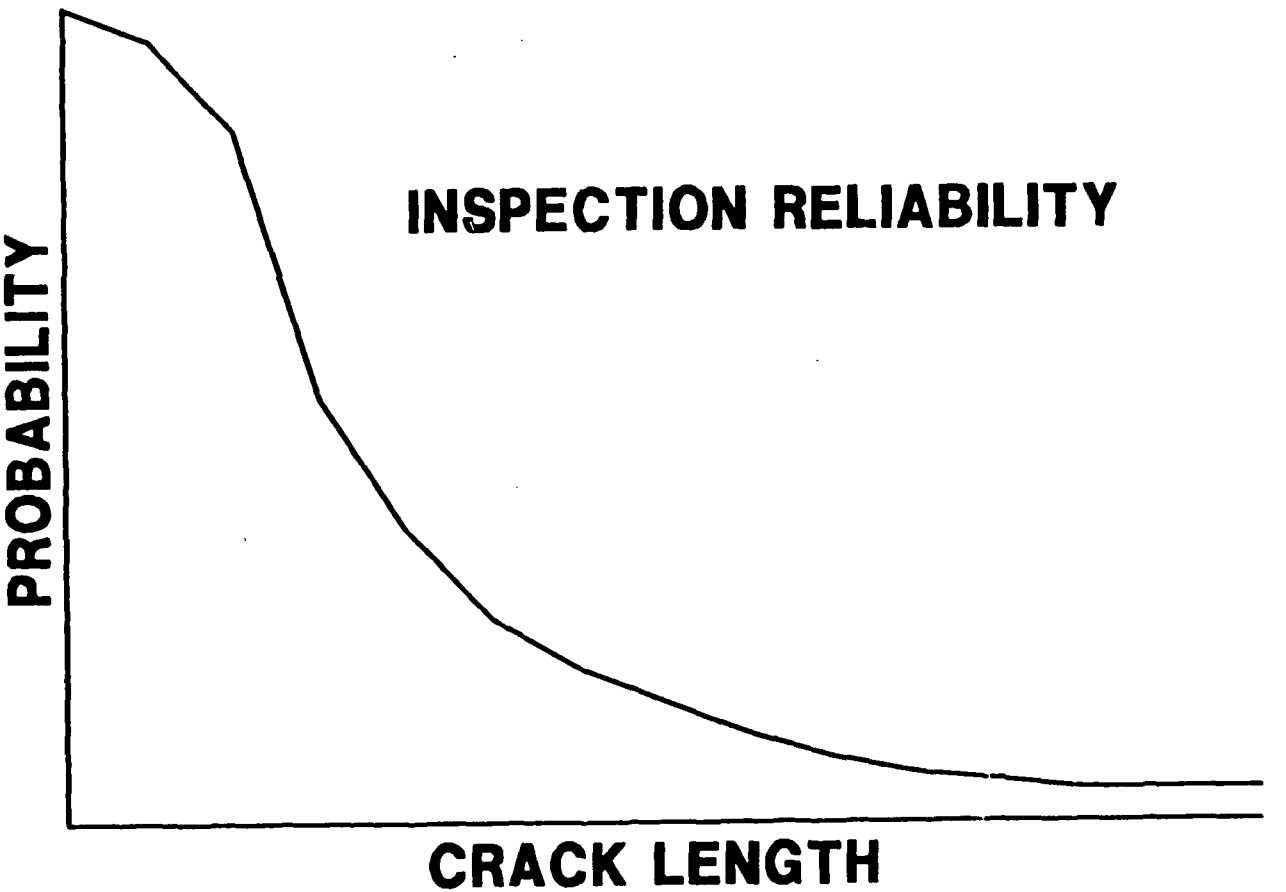

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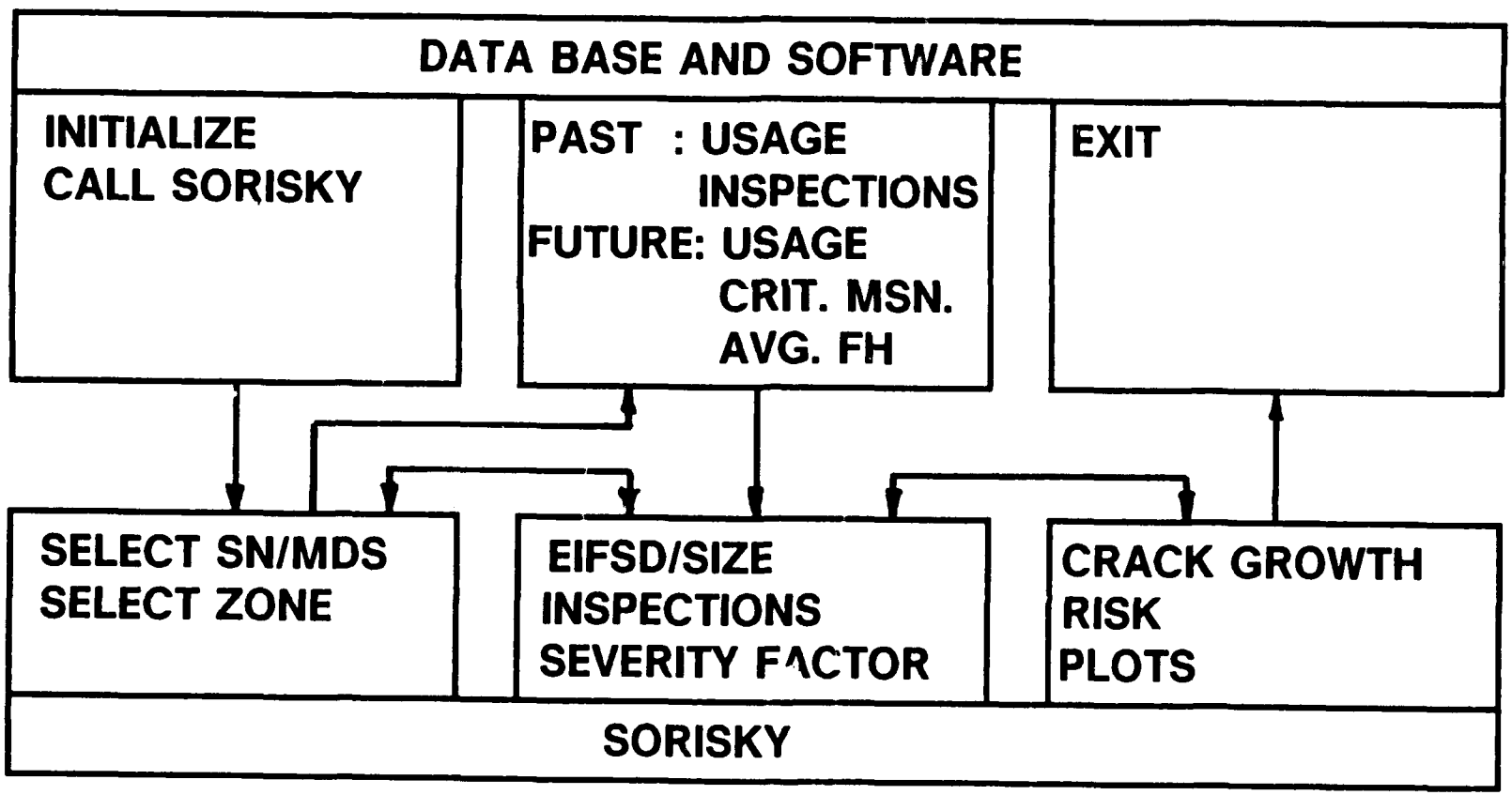
C-130 RISK ANALYSIS
SORISKY - DATA BASE INTERFACE AND FLOW



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C-130 RISK ANALYSIS
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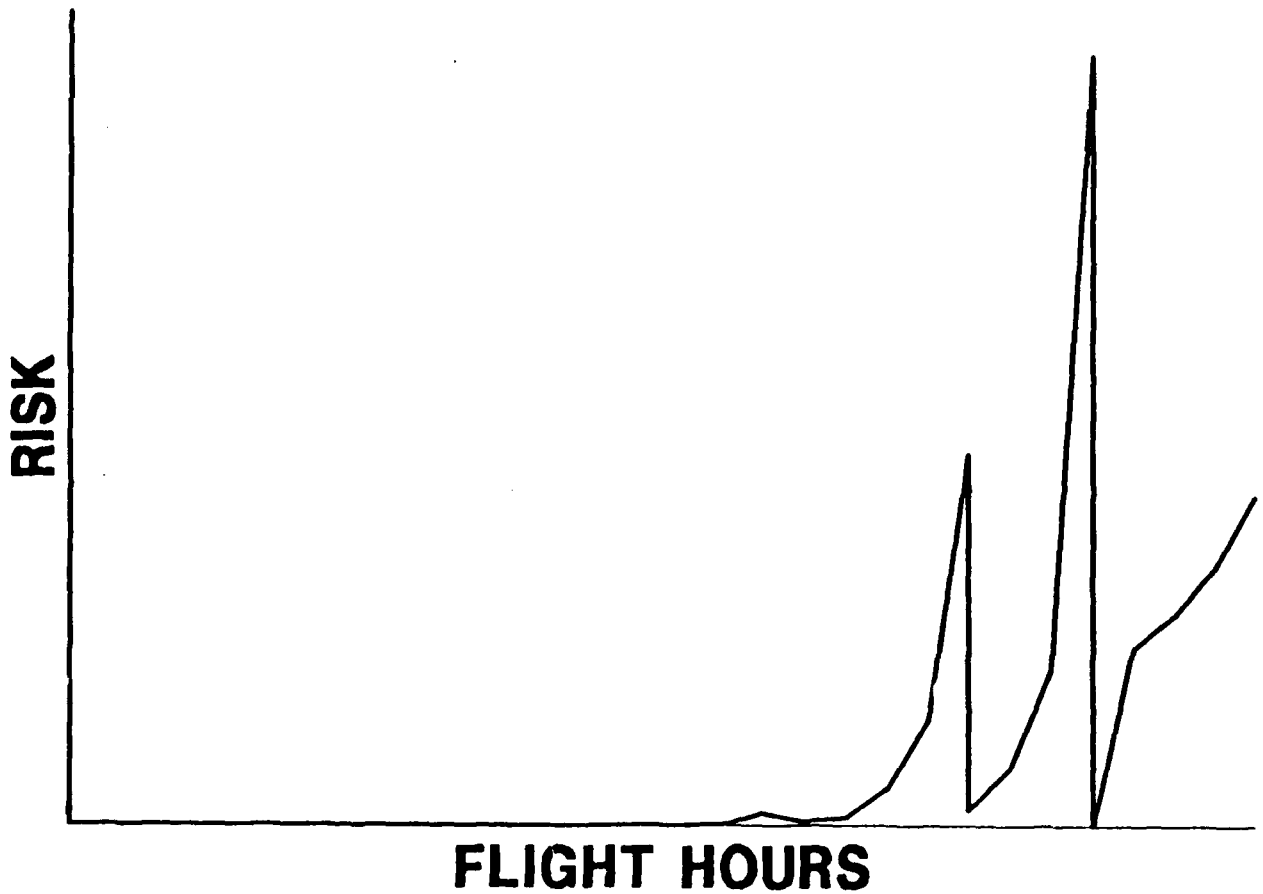

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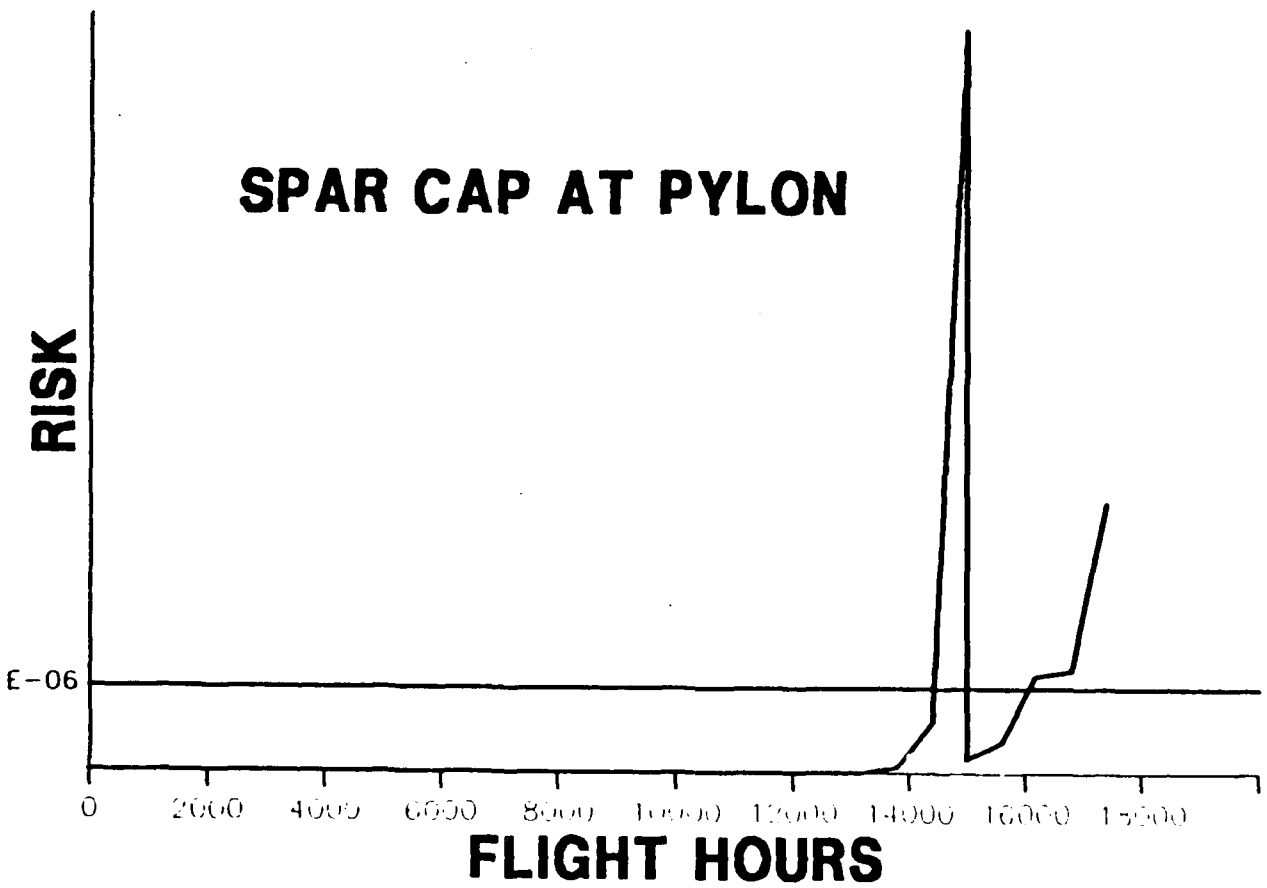

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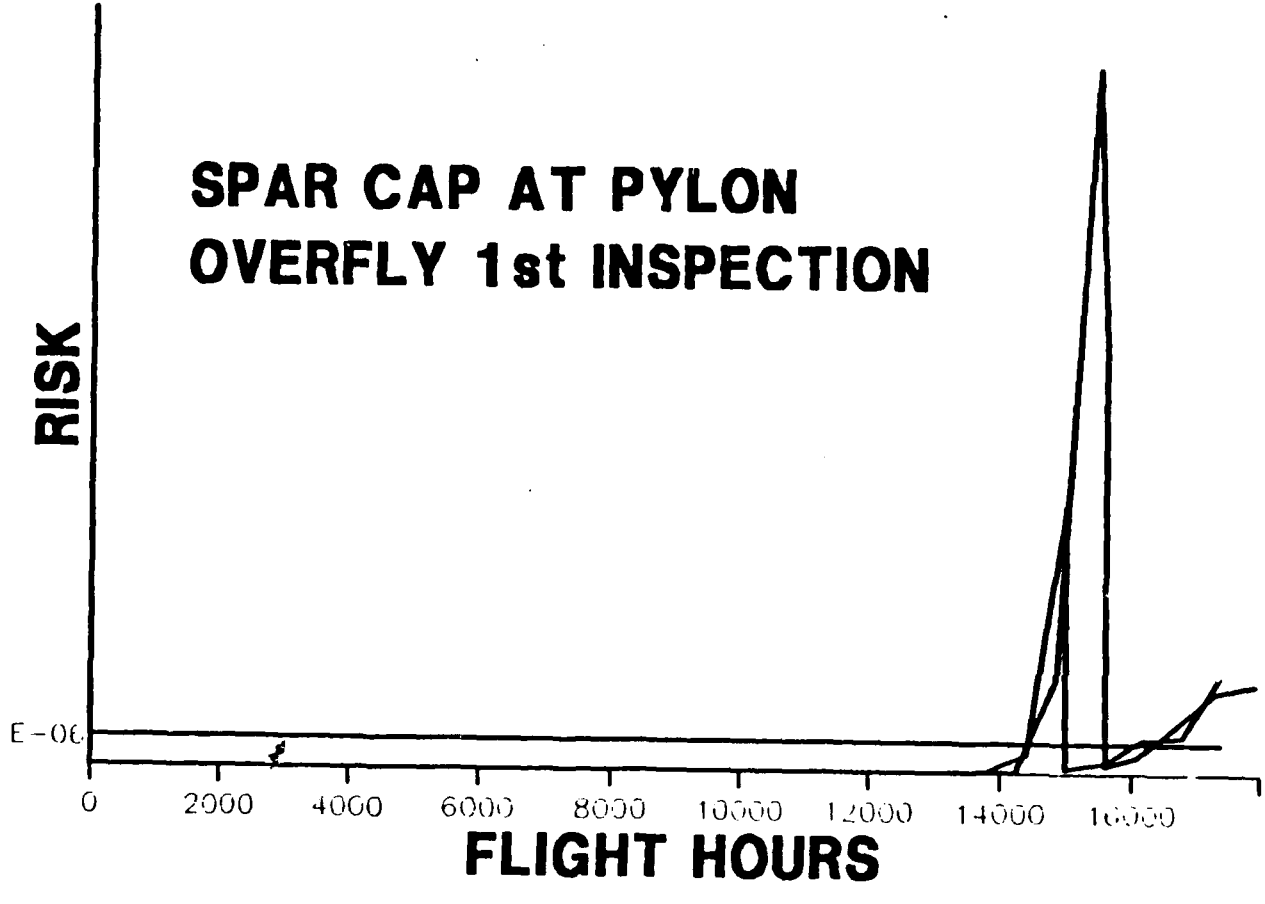

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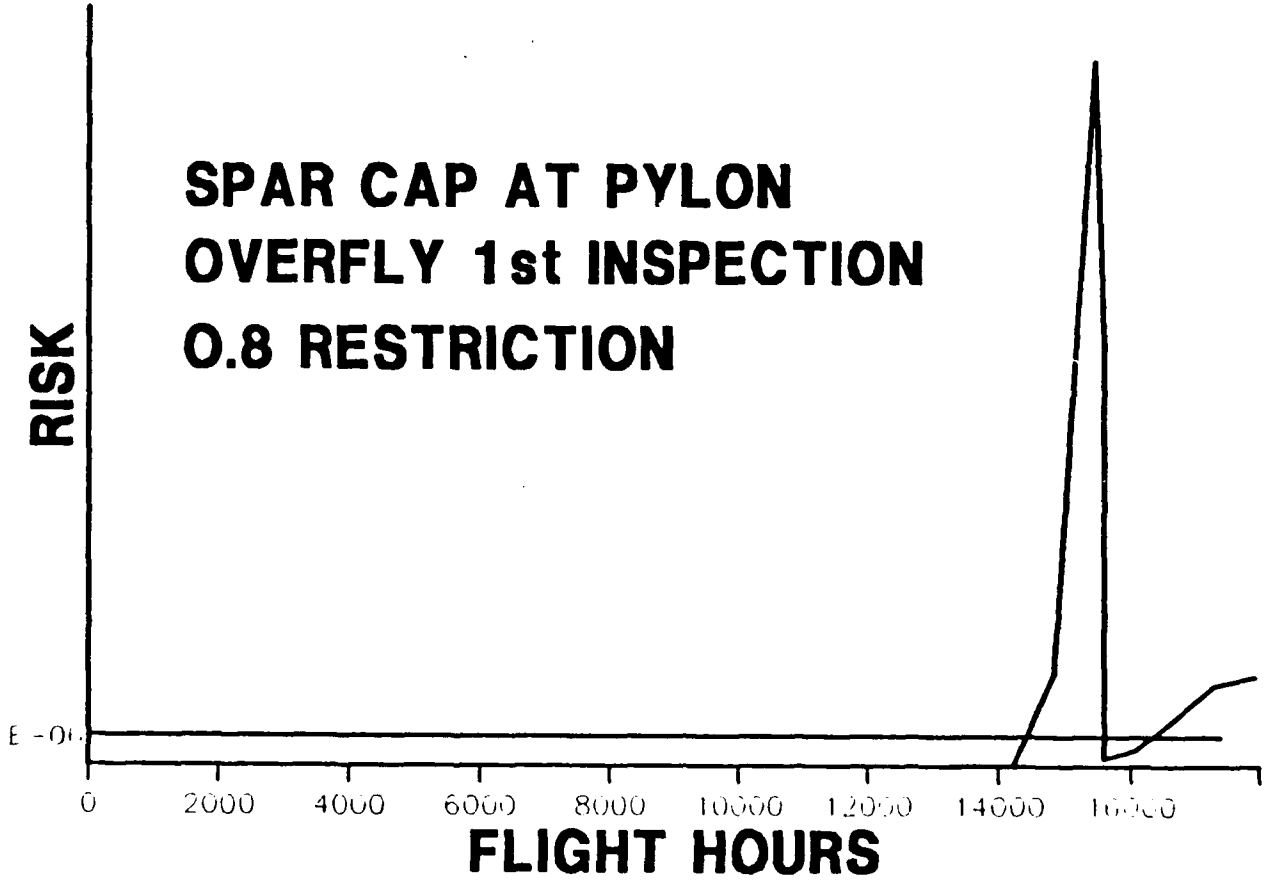

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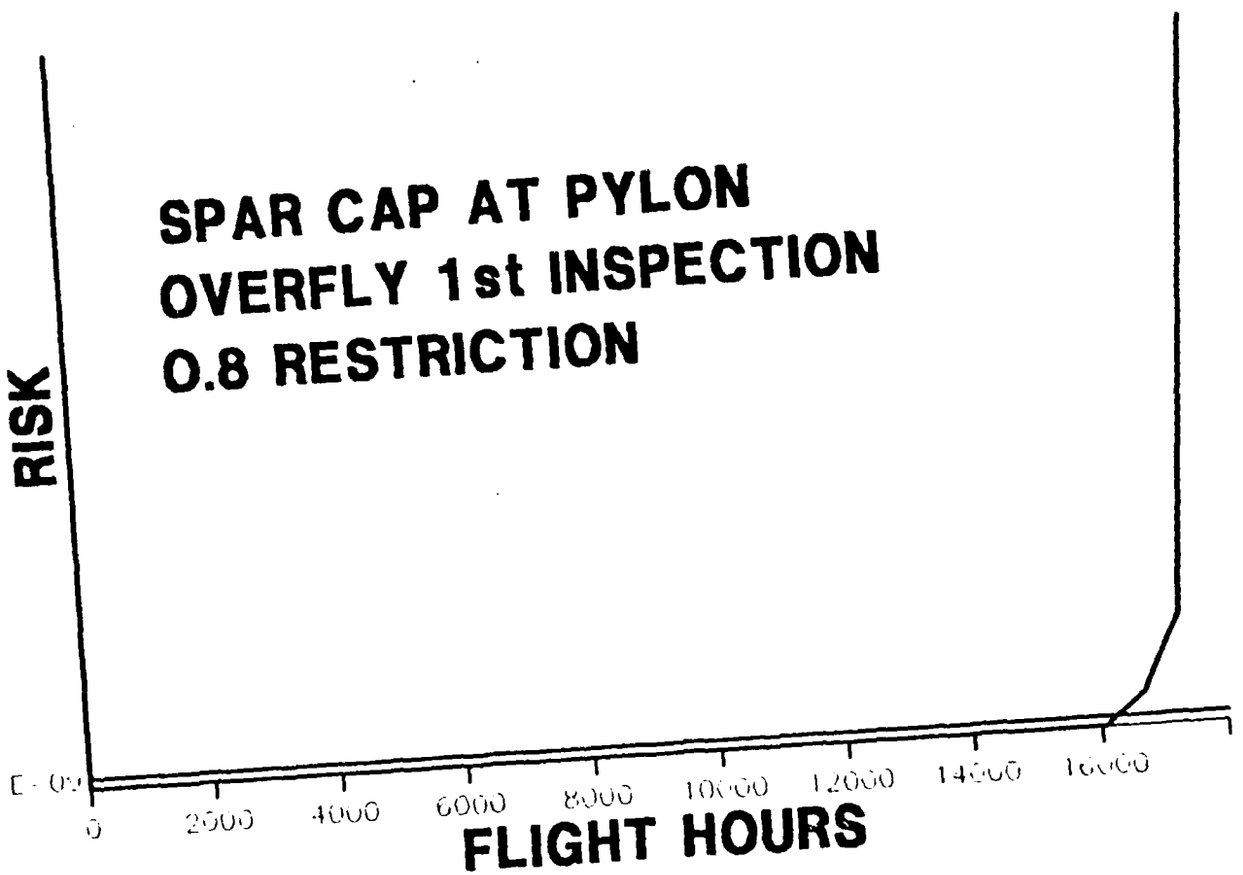
C-130 RISK ANALYSIS
SAMPLE OUTPUT



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C-130 RISK ANALYSIS SUMMARY

AT THE END OF THIS PROGRAM WARNER ROBINS ALC WILL HAVE

- SOFTWARE TO INTERFACE WITH AIRS AND TO CALCULATE RELATIVE RISK
- BASELINE DATA BASE
- CAPABILITY TO DISPLAY THE RELATIVE MERITS OF ALTERNATIVE USAGES AND/OR INSPECTIONS

RISK ASSESSMENT OF THE F-16 WING

Douglas O. Cornog* and John W. Lincoln**

A premature failure of an F-16C wing in static test raised a question on the course of action to be pursued to ensure the continued structural integrity of this aircraft. To address this question a risk assessment was performed that provided insight on the wing strength required to maintain safety of operational aircraft. In addition to the strength problem, an wing upper durability problem surfaced in a durability test article and on operational aircraft. It was found that the wing upper surface modification that was designed to resolve the strength problem also resolved the durability problem.

INTRODUCTION

The F-16A aircraft was the first aircraft to be designed that used the Aircraft Structural Integrity Program (ASIP) requirements in MIL-STD-1530A during its full-scale development. In addition, its structural integrity was enhanced by the decision to establish the stress in the structure such that it was able to tolerate the initial flaw requirements of MIL-A-83444 for two lifetimes without an inspection. The F16A structure met these requirements as demonstrated by analyses, development tests and full-scale tests of strength, durability and damage tolerance. The full-scale static test to ultimate validated the strength of the structure, however, a decision was made to stop the test at ultimate rather than to proceed to failure to determine the capability of the wing. The F-16A was designed to a flight design gross weight of 22,500 pounds. However, in the early eighties it was evident that aircraft weight was increasing

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significantly and that operational squadrons were flying the aircraft to a spectrum different than that used in the design. The historical weight growth of the F-16 aircraft is shown in Figure 1. A line on this Figure with the slope of one pound a day weight growth is seen to be similar to the F-16 experience. To assess the implications on the structural integrity of the F-16 a team of Aeronautical Systems Division and Flight Dynamics Laboratory personnel was formed in March of 1984. The recommendations of this team laid the foundation for a new full-scale durability and static test to be performed. It was decided to perform these tests on an F-16C model produced in the Block 30 configuration. The flight design gross weight for this aircraft is 26,910 pounds. Therefore, the design wing bending moment changed from 3.15 million inch pounds in the initial design to 3.94 million inch pounds in the Block 30 design. Some of this 22 percent change was due to an error in the loads that was corrected through the flight load survey. On 19 October 1987 the left wing failed in static test at approximately 128 percent of limit load. This 9g symmetrical pull up condition was the 22nd test condition for the article. The failure occurred in the proximity of the wing root and was the result of compressive buckling on the upper wing skin and associated substructure. Immediately following this failure, the Air Force imposed restrictions on the F-16C and F-16D aircraft that would maintain a factor of safety of 1.5. At this time there were reports from the field that some of the F-16A aircraft were experiencing cracks in the upper wing skins in the location of the static test failure. Also, the durability test article which had started its flight-by-flight loading 1 September 1987 was found to be cracked in this area after 600 hours of usage testing. The task of resolving these problems was given to an ad hoc team composed of representatives of the Aeronautical Systems Division, the Flight Dynamics Laboratory, the Ogden Air Logistics Center, the Tactical Air Command, and General Dynamics.

DISCUSSION OF THE FAILURES

The wing plan view with the top cover removed is shown in Figure 2. It is noted that there are several holes in the upper surface close to the wing root. There are several holes in the upper surface some of which are numbered in Figure 2. The static test article had numerous strain gages installed. The gage located between holes numbered 2 and 3 is a good indicator of the onset of the upper wing skin buckling. The strain response of this gage for the left and right wing are shown in Figure 3. This figure has several significant features. First, it is noted that the left hand wing strain response when subjected to condition 231.0 is not a linear function of the applied bending moment. The residual strain after this loading was removed was - 3,500

microinches per inch. It is also noted that the right hand wing strain response for the same condition is significantly different. This difference is attributed to the fact that there is a doubler around the inside of hole 2 on the left wing that is not present on the right wing. This doubler had the effect of an eccentricity since it induced a bending moment around the hole. The strain measured on the left wing at condition 208.0A-C3 where it failed was approximately - 19,000 microinches per inch. There was some discussion whether the permanent set in the wing from the loading from condition 231.0 reduced the strength of the wing. It was concluded that it was likely that this happened but the magnitude of this degradation could not be quatified with an acceptable degree of certainty. Consequently, the bending moment at failure was established as the strength of the wing. The failure path of the wing is shown in Figure 4. The metallurgical examination of the wing found that the failure initiated between holes 2 and 3 (i.e. between spars 6 and 7). This examination also found that the chemistry and mechanical properties of the wing were within the allowable variations and that the manufactured dimensions of the wing were within tolerances.

Figure 5 shows typical cracks in the upper wing skin found around hole 3. Cracks were also found on service aircraft emanating from holes 2 and 4, although hole 3 was by far the most evident cracking site. The mechanism for this cracking is that high wing upbending causes a compressive yielding at these upper surface holes which results in a residual tensile stress. When the wing is subjected to downbending the resulting stresses are high enough to initiate a crack. These cracks have been found in some service aircraft with only approximately 800 hours. This residual stress cracking is not unique with the F-16. It has been experienced on the F-15 in a wing upper surface sealing groove and on the F-111 at a stiffener runout on the upper surface of the wing pivot fitting. The details of these failure areas are shown in Reference 1. As seen from Figure 5 the cracks are primarily parallel to the longitudinal axis of the aircraft and predominantly initiate out of fastener holes, although not always. Typically these cracks will either stop or significantly reduce their propagation rate when they progress out of the local residual stress region. Consequently, these cracks are considered an economic problem and not a safety problem. However, if this problem is permitted to persist the economic burden of repairing these cracks would become severe. The results of an inspection of USAF aircraft only is shown in Figure 6.

ACTIONS

After the static test failure occurred the following three

actions were taken by the USAF in conjunction with the contractor:

Flight restrictions were placed on F-16 C/D aircraft.

A risk assessment was conducted to evaluate the magnitude of the required increase in strength.

Structural modifications were developed.

Restrictions

The restrictions placed on the F-16C/D aircraft were designed such that the affected aircraft were still operating with a safety factor of 1.5. These restrictions are summarized in Figure 7.

Risk Assessment

The F-16 was designed based on the the Aircraft Structural Integrity requirements of MIL-STD-1530A and the strength and rigidity requirements of the MIL-A-8860 series specifications. These requirements are deterministic with their validity based on the numerous successful precedents of similar aircraft that were designed by this process. When the wing failed at 128 percent of design limit load, these precedents were violated and consequently there was no basis to clear these aircraft to their full operational envelope. It was believed that the F-16 limiter was effective in preventing loads as high as 128 percent of limit, and conceivably the wing could have adequate strength to perform its intended mission. It was conjectured however, from previous experience that there was some aircraft to aircraft variations in the wing strength. It was known that the limiter permitted some exceedance of limit load (approximately six percent) due to tolerances and that centerline store carriage and roll maneuvers could also cause an exceedance of limit load. The question was then: "was the strength variation in the fleet large enough to cause aircraft to fail that were operating as evidenced by recorded data?" To answer this question there was a decision made to perform an analysis to determine the risk of structural failure.

A probabilistic approach has been used before by the USAF (e.g. reference (2)) to assess the risk of structural failure due to unstable cracking in older aircraft. This has been done mainly to try to provide additional operational capability for aircraft that had reached their calculated damage tolerance limits and had difficult inspection problems. The T-37, T-38, F-5 and C-141

aircraft fell into this category. This approach was also used on the C-5 to examine an alternate approach to the deterministic damage tolerance approach and to determine the number of flight hours at which fail safety of the structure would be degraded below an acceptable level.

There is always the question of what probability of failure is acceptable for operational aircraft. In the assessments cited above two criteria were generally used to determine an acceptable level of risk. The first criterion is that risk of structural failure in a single flight should be no greater than the risk routinely accepted by the public. The second criterion is that the risk should be controlled such that the expected number of losses due to structural failure in the lifetime of the fleet should be less than one. The application of these principles indicate that the single flight probability of failure should be 10^{-7} or less.

The loads calculations and the strength calculations can be made with the goal of deriving the probability distribution function for loads and the probability distribution function for strength. These distribution functions are independent on one another and consequently the joint density function for loads and strength can be derived from the product of the respective marginal density functions. This joint density function can be integrated over the region where the load is greater than the strength to determine the probability of failure.

The Loads Probability Distribution Function

In the case of the F-16 the basic source of loads data is the Flight Loads Recorder data generated from the F-16A/B aircraft. The following missions were identified in this data base:

Transition

Instrument/Ferry

Air Combat Maneuvers (ACM)

Air to Air Weapon Delivery

Surface Attack

Air Defense Intercept

Functional Check

Low Level Navigation

Of these missions the ones that have a predominant effect on the loads probability distribution are the air combat maneuvers, air to air weapon delivery, air defense intercept and functional check. Of the valid data collected there are approximately 8500 hours that could be used to evaluate the F-16A/B aircraft and approximately 7500 hours that could be used to establish the usage for F-16C/D aircraft. However, the data base does not include any actual F-16C/D usage. The actual C/D usage will be available later when the new microprocessor system has been installed on operational aircraft. It is believed that the data base is adequate to develop a meaningful loads probability distribution function.

The exceedance function for the ACM mission is plotted in Figure 8. It is seen that the ordinate is the number of occurrences per 1000 flight hours exceeding a given bending moment. This abscissa can be easily changed to percent limit load since there is a linear relationship between percent limit load and bending moment. The loads probability distribution function can be derived from the exceedance functions as follows: Each ordinate of the exceedance function is divided by the number of flights per 1000 hours to determine the average number of exceedances per flight. Then, if the ordinates that are above one are set equal to one and if it is assumed that the there is no more than one occurrence per flight of a load corresponding to an ordinate less than one, then the resulting function is one minus the loads probability distribution function. The function so derived does not extend to low enough probabilities to be useful for the calculation of the failure probability. A convenient method for extrapolating this function is to use an analytical distribution such as the normal, Weibull or Gumbel. In this assessment the two parameter Weibull distribution was used. For the two parameter Weibull distribution if A is a number defined as the "shape parameter" and B is a number defined as the "scale parameter" then the distribution function can expressed by

$$P(X < x) = F(x) = 1 - \exp\left(\frac{-x}{B}\right)^{A}$$

This distribution has the property that the variance becomes smaller when the shape parameter A is increased. It is noted that when the shape parameter is one then the distribution is the exponential distribution and when the shape parameter is near 3.5 the distribution is approximately the same as the normal distribution.

For the purpose of extrapolating the loads probability distribution the following procedure may be used: The logarithm of the logarithm of 1 - F yields an equation for a straight line

where the slope of the line is the shape parameter. If the double logarithm of one minus the estimate of the loads probability distribution from the flight loads recorder data is plotted versus the percent of limit load then the slope established by this process may be used as the Weibull shape parameter. The Weibull scale parameter is readily obtained by forcing the Weibull distribution through one point of the flight loads recorder derived distribution. It was found that a Weibull shape parameter in the range of eight to ten was typical for the measured data. The loads probability distribution function for the ACM mission is given in Figure 9.

The Strength Probability Distribution Function

For the calculation of the probability distribution for the strength of operational F-16s it is required to establish an estimate of the mean strength and the variability of the strength represented by an analytical distribution function. The static test failure was caused by buckling and consequently the buckling load was influenced by the skin thickness, the boundary conditions on the wing skin and the wing material properties. For the test specimen there were measurements made of the thickness and the material properties and all were found to be nominal. Consequently the judgement was made that the test article failure occurred at the mean. The variability of the failure was evaluated from 28 wing skin thickness measurements taken from service aircraft and from 68 yield strength measurements taken from wing spar material. The wing skin variability was derived from existing MIL-HDBK-5 data. There was no attempt to evaluate the variability of the boundary conditions. Based on the variabilities assessed it was judged that a Weibull probability distribution function could be used to describe the variability and that a Weibull shape parameter of 24 was reasonable. Also, based on work performed by R. Deo from Northrop it was typically found that buckling failures have a coefficient of variation of 4.7 percent which translates into a Weibull shape parameter of approximately 27. Therefore, the use of 24 for the shape parameter appears to be justified.

Failure Probability Calculation

As indicated above the single flight probability of failure can be derived from the probability distribution functions for loads and strength. Since both of these distributions are in an analytical format through the Weibull distribution it is straightforward to determine the associated probability density functions. Further, since the two density functions are independent then the joint density function is their product. With the procedure defined, the probability of failure based on

flying the air combat maneuvering mission, which is generally the most severe of all the missions, was evaluated. It was determined that for a single flight of an aircraft picked at random from the population that the probability of failure was approximately 5×10^{-6}. This is greater than the desired level of risk. Further, it was found that when all of the missions were considered with their measured usage distribution, that the expected number of losses in the F-16 force was approximately eight. If the wing is strengthened then it is found (see Figure 10) that the expected number of losses decreases to less than one when the strength reaches approximately 145 percent of limit load. Consequently, it was found that either the wing needed to be strengthened or aircraft restrictions would be required. It was also found that a safety factor of slightly less than 1.5 was adequate to provide the desired operational safety for the aircraft.

Wing Modification

The modification to the wing is shown in Figures 11 and 12. It consists of the following changes:

The access door cover shelf was removed from holes 2, 3 and 4.

Load bearing plugs were inserted in holes 2, 3 and 4.

Removable reinforcing covers were added over plugs in holes 2, 3 and 4.

Fixed reinforcing plates were added between spars 6 and 7 and between spars 8 and 9.

The concept of this repair was for the plugs to transfer the load through the skin with no load path eccentricity. The reinforcing plates were to stabalize the plugs and increase the local effective skin thickness. This concept was selected after an examination of several alternatives most of which were extremely expensive. Some of the alternatives were fabricated and installed on the wing and for a strain survey. The option shown in Figures 11 and 12 was the result of these efforts. The aerodynamic impact of the chosen repair option were considered to be negligible.

The chosen repair option was fabricated and installed on a wing for a strain survey. To support this survey to 125 percent of limit load considerable effort was expended in the construction of nonlinear finite element models. The modeling effort prior to the development of the nonlinear models was with NASTRAN and an in-house linear buckling analysis program. These models

predicted high strains in the region of failure but did not predict the failure. The nonlinear modeling program used was ABAQUS. It was used to construct three models. One of these which included six of the wing spars was able to predict the static test failure within two percent of the actual failure load. In general, the results of this model when correlated with the strain survey results were acceptable, but the effect of the boundary conditions and the influence of fastener stiffness indicated that there was an uncertainty in the strength prediction of approximately 10 to 15 percent. The predicted strength, however, for the modified structure was in excess of 150 percent of limit. Therefore, the testing of this configuration to ultimate appeared justified.

RESULTS

The modified structure was tested and found to be adequate since it reached 146 percent of design limit load before a failure occurred in the leading edge flap. The analysis of this modified configuration for durability showed that the from the critical hole (i.e. hole 3) the crack growth from an 0.05 inch corner flaw only grew to 0.06 inches in 16,000 flight hours. The durability test article was modified with this repair to substantiate this finding.

CONCLUSIONS

The following conclusions were derived from this study:

A relatively simple modification to the wing resolved a strength and durability problem precipitated by weight growth of the aircraft.

The strength problem was difficult to predict analytically.

The alpha-g limiter used on the F-16 is an effective means to enhance structural integrity.

The risk assessment provided the basis for management actions.

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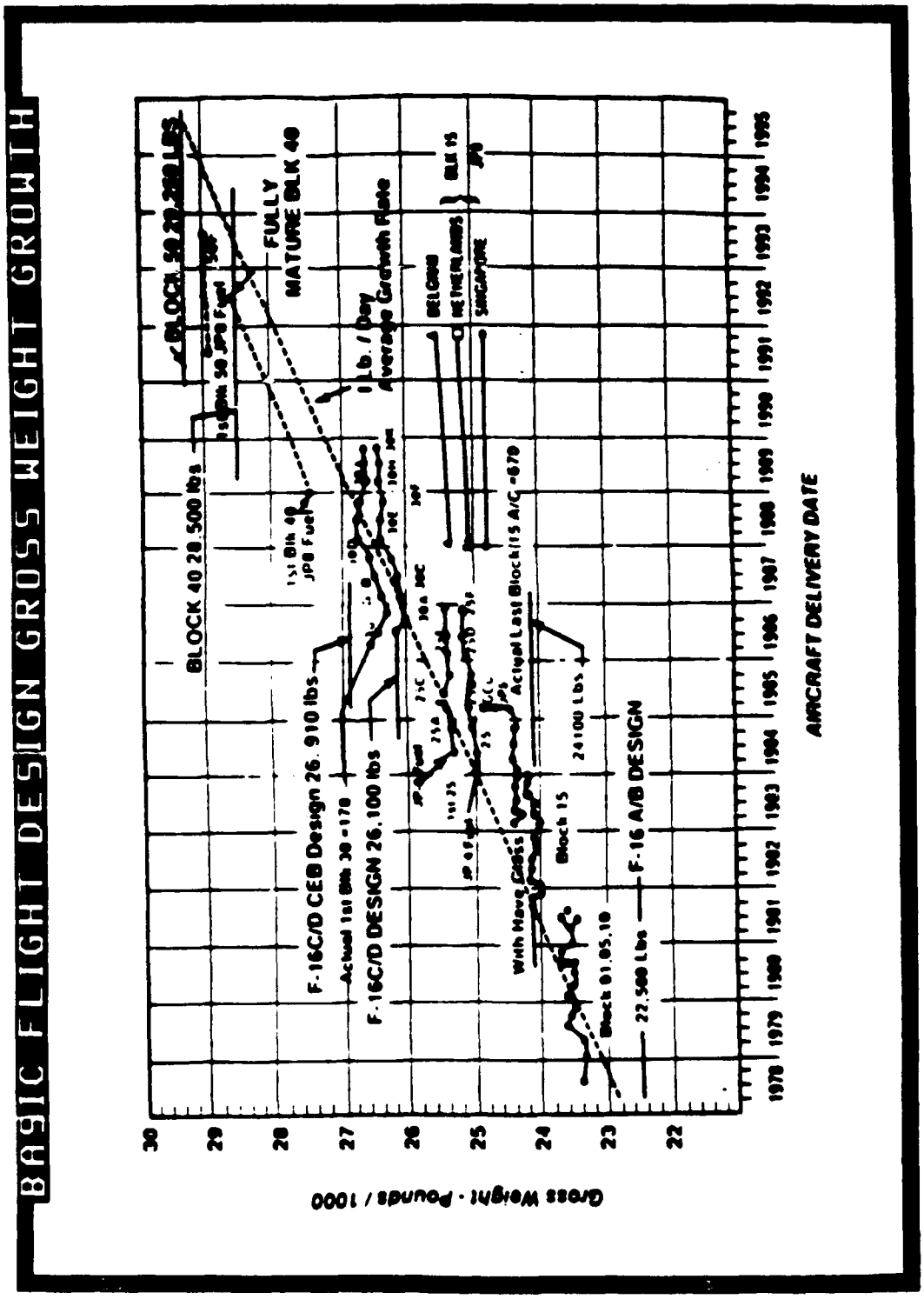


FIGURE 1. F-16 WEIGHT GROWTH

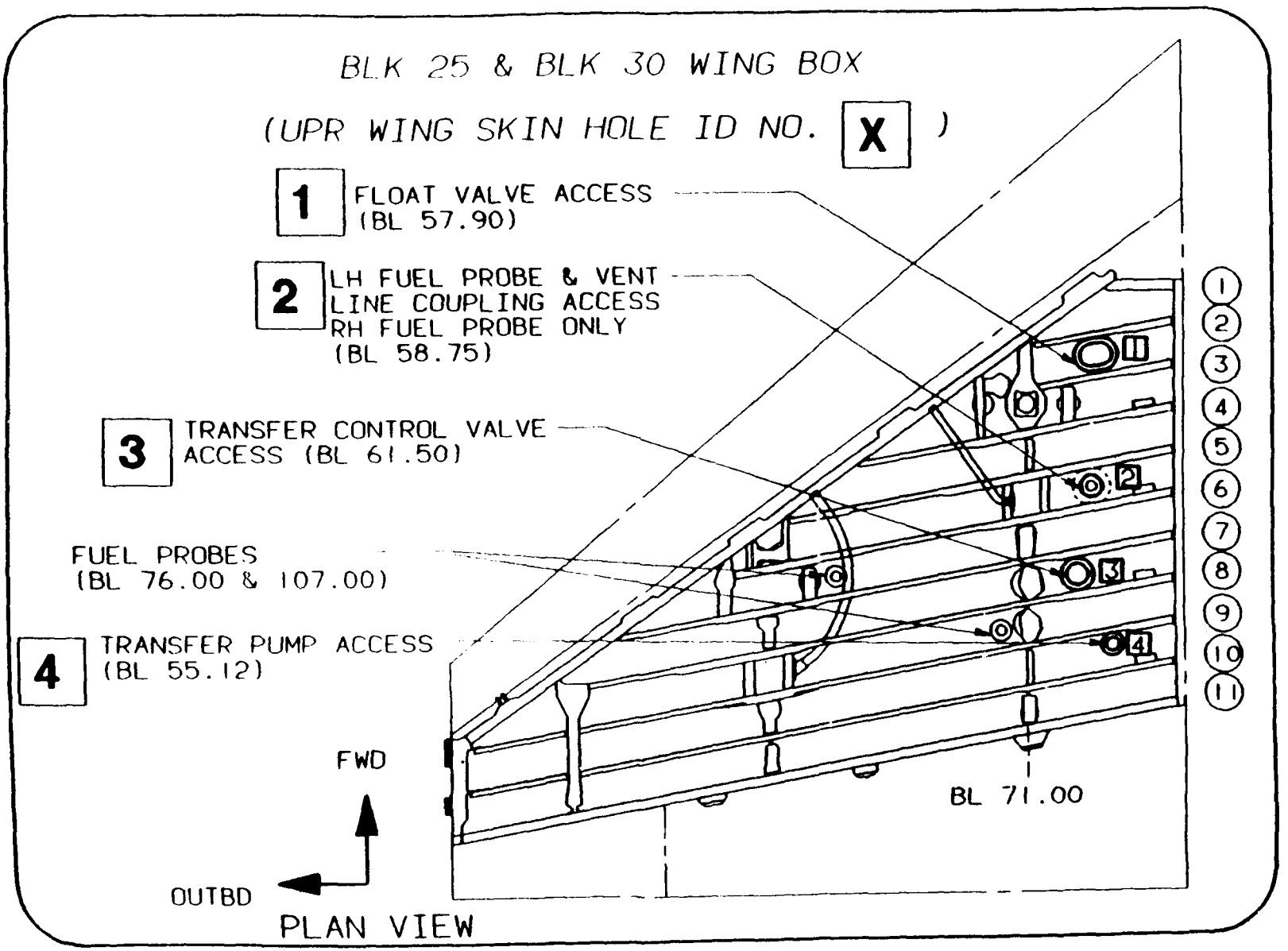


FIGURE 2. F-16 WING STRUCTURAL ARRANGEMENT

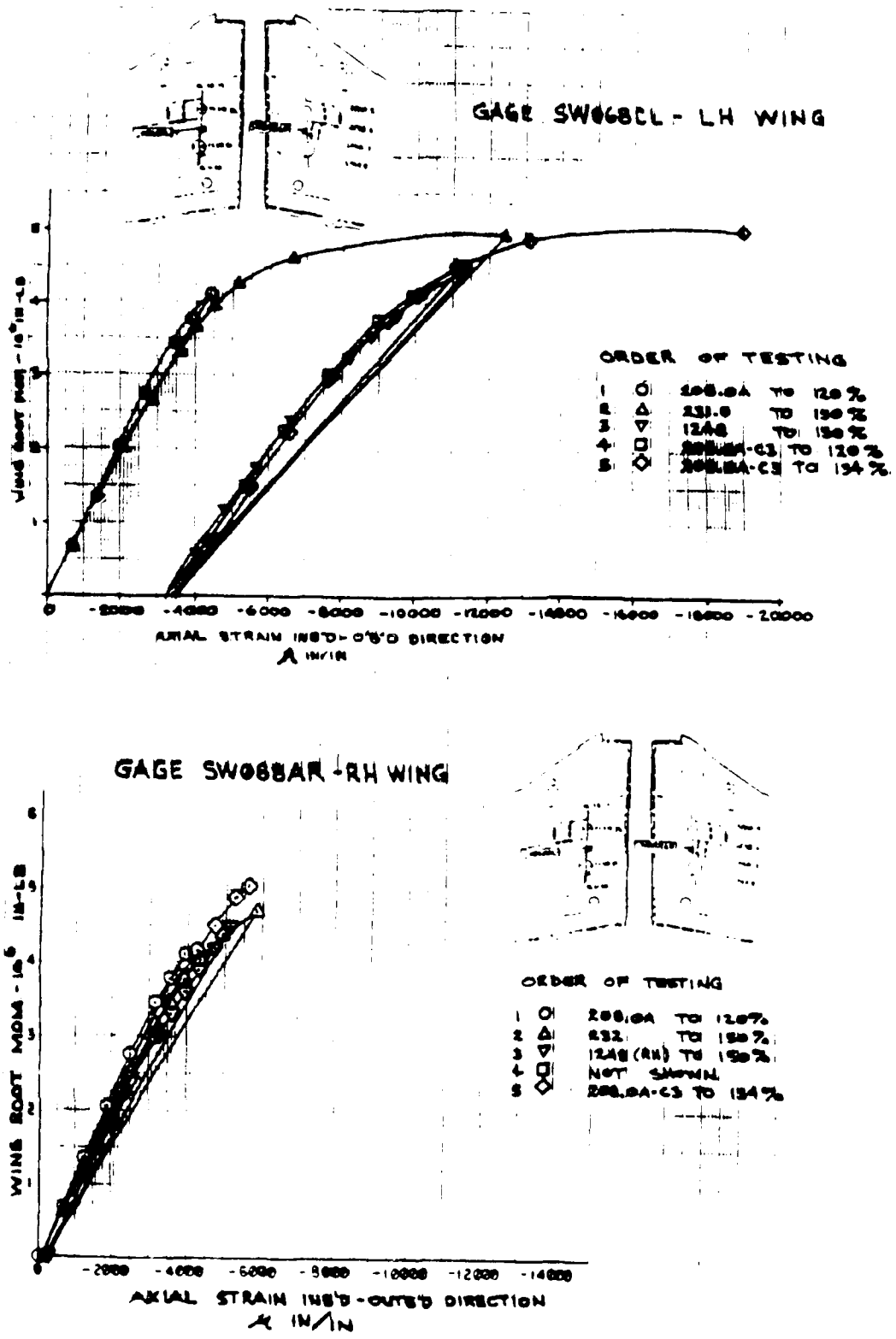


FIGURE 3. STATIC TE[illegible] STRAIN HISTORIES

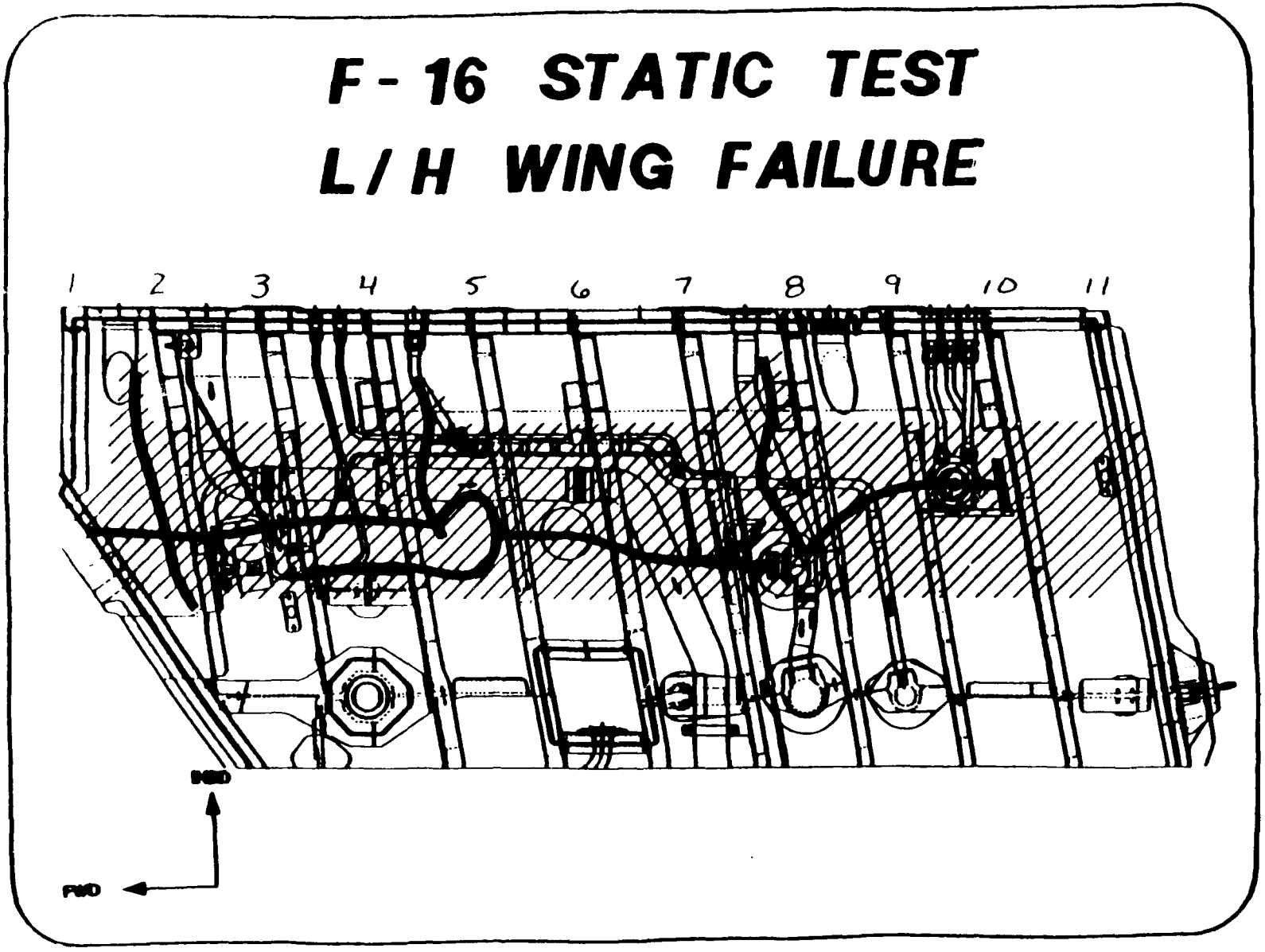


FIGURE 4. LOCATION OF STATIC TEST FAILURE

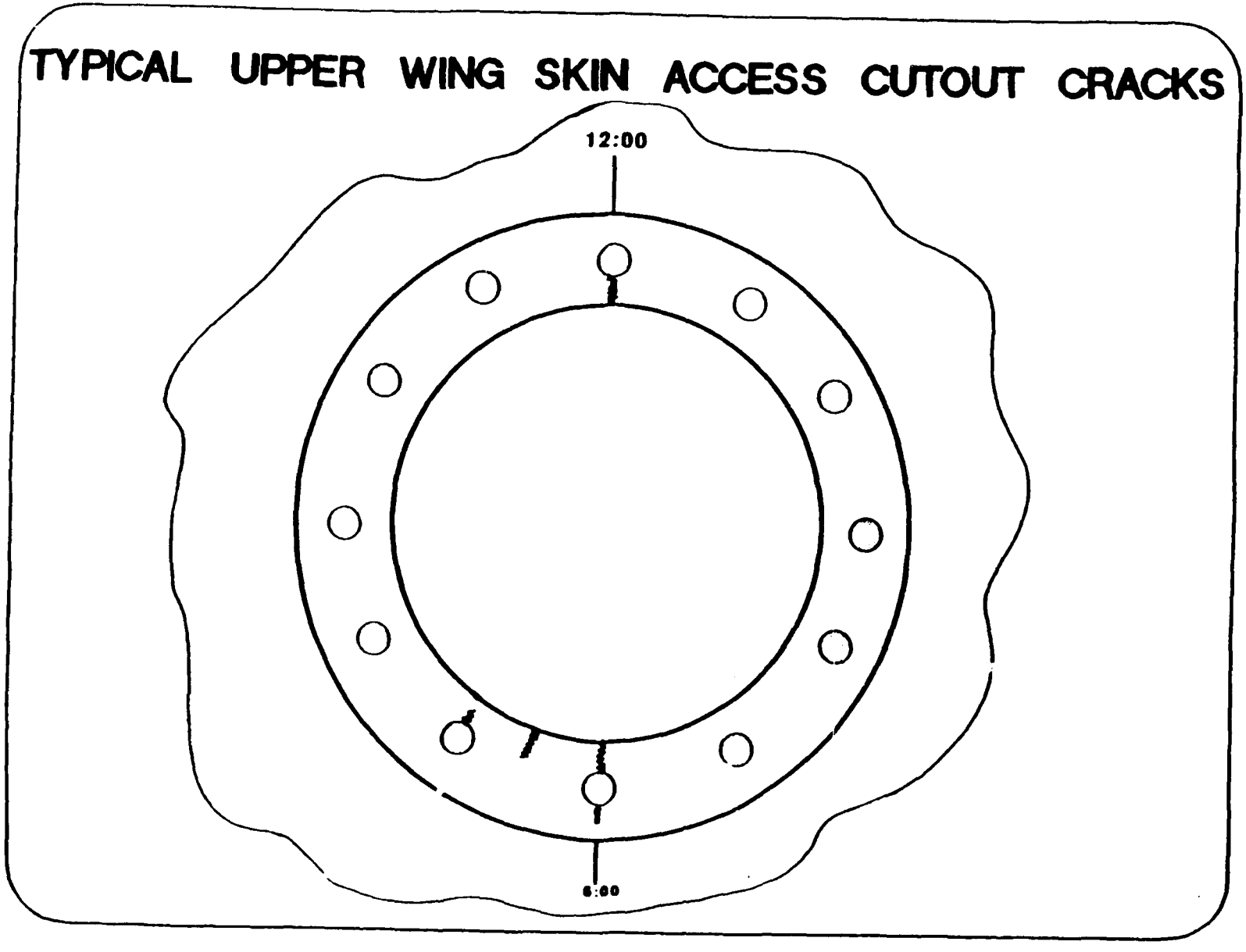


FIGURE 5. WING UPPER SURFACE CRACK LOCATIONS

DURABILITY SERVICE EXPERIENCE

- Upper Wing Skin Access Hole Inspections
  - Inspected A/C: >750 Flt Hrs
  - USAF info only

A/C Block	Number Inspected	Number w/ Cracks	Average* Flt Hrs
A/B Blk 10	236	197	2000
A/B Blk 15	335	133	1400
C/D Blk 25	97	0	N/A

* - Approximate at inspection time

FIGURE 6. SERVICE CRACKING EXPERIENCE IN WING

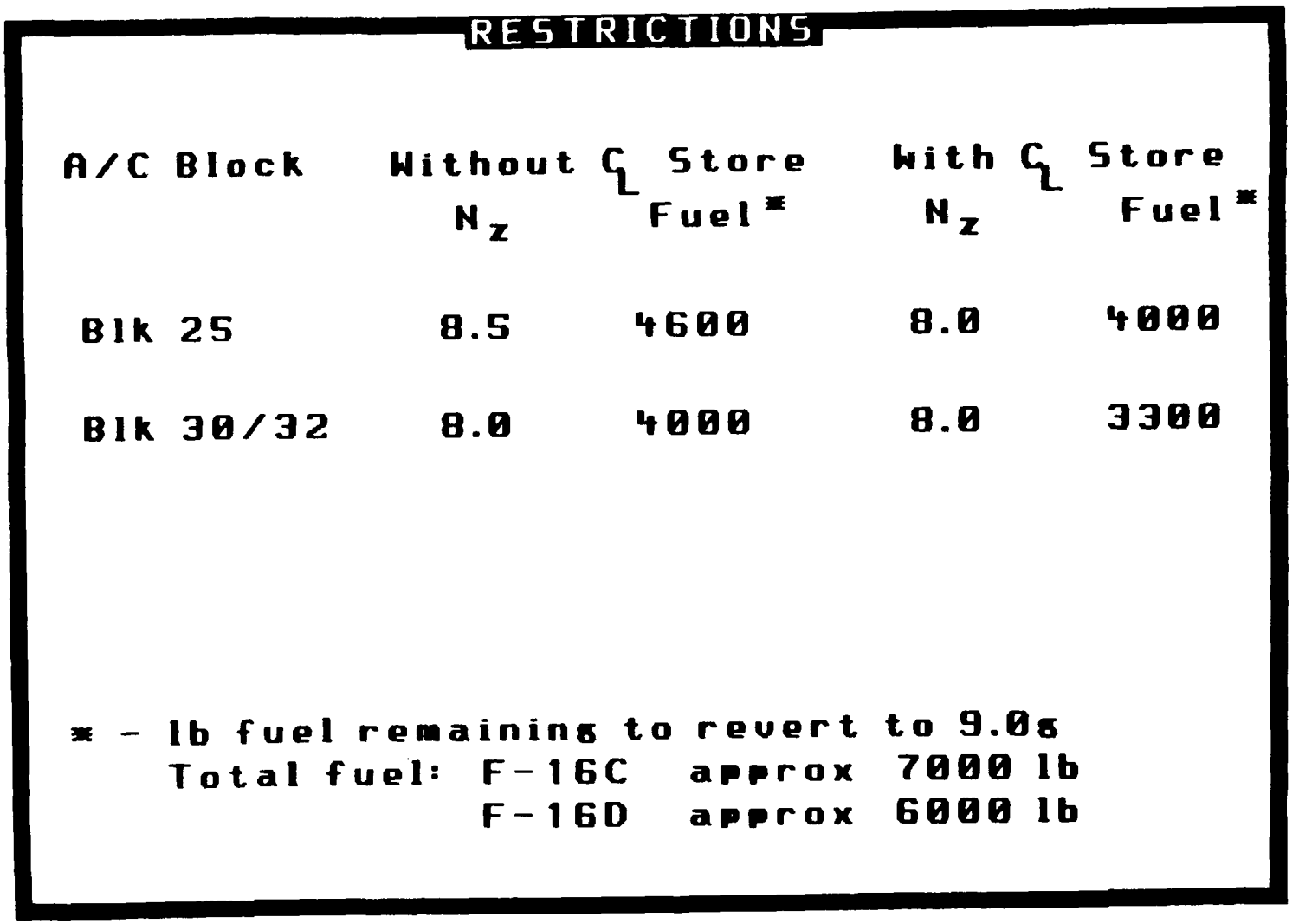


FIGURE 7. F-16C/D INTERIM RESTRICTIONS

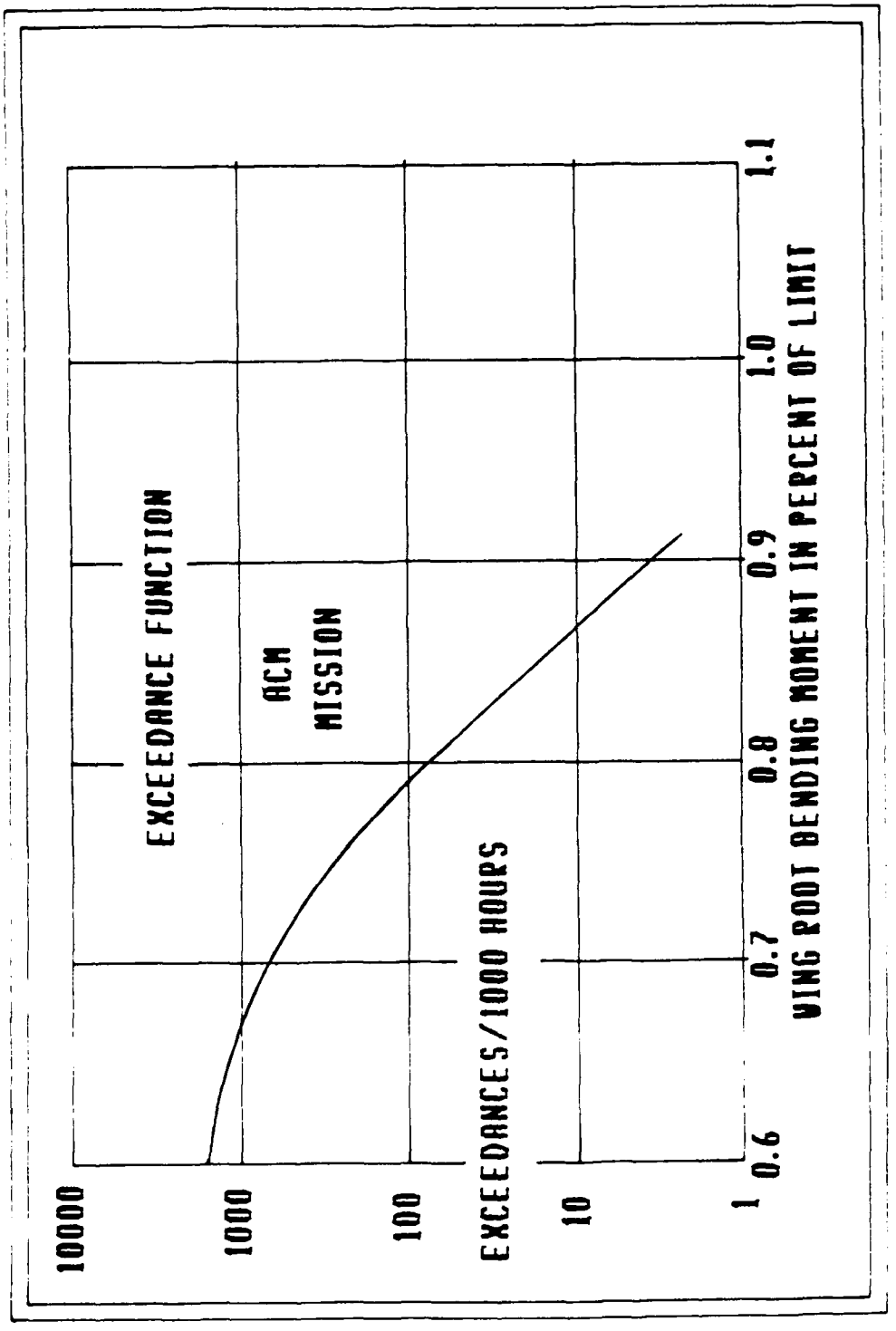


FIGURE 8. BENDING MOMENT EXCEEDANCE FUNCTION

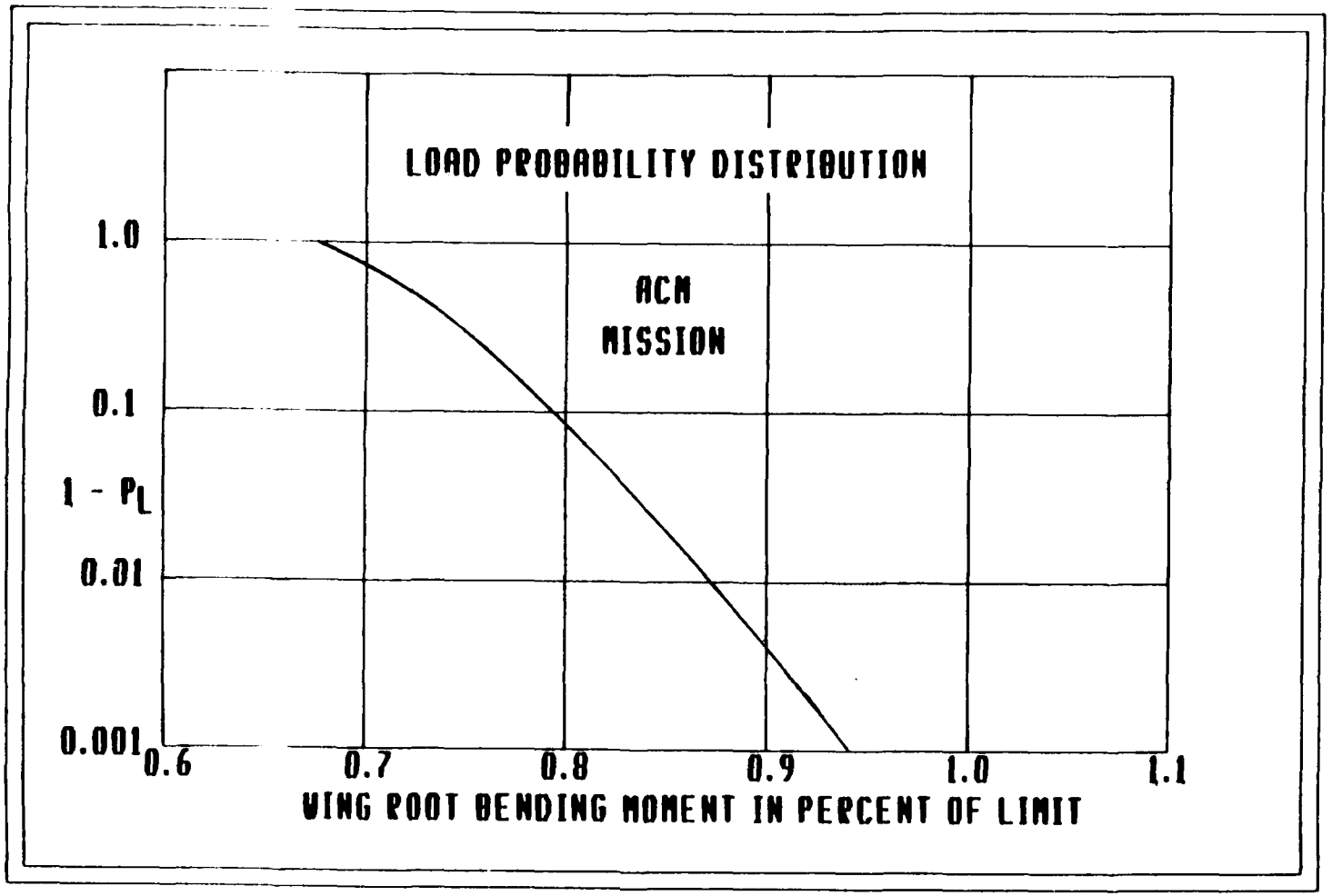


FIGURE 9. BENDING MOMENT PROBABILITY DISTRIBUTION

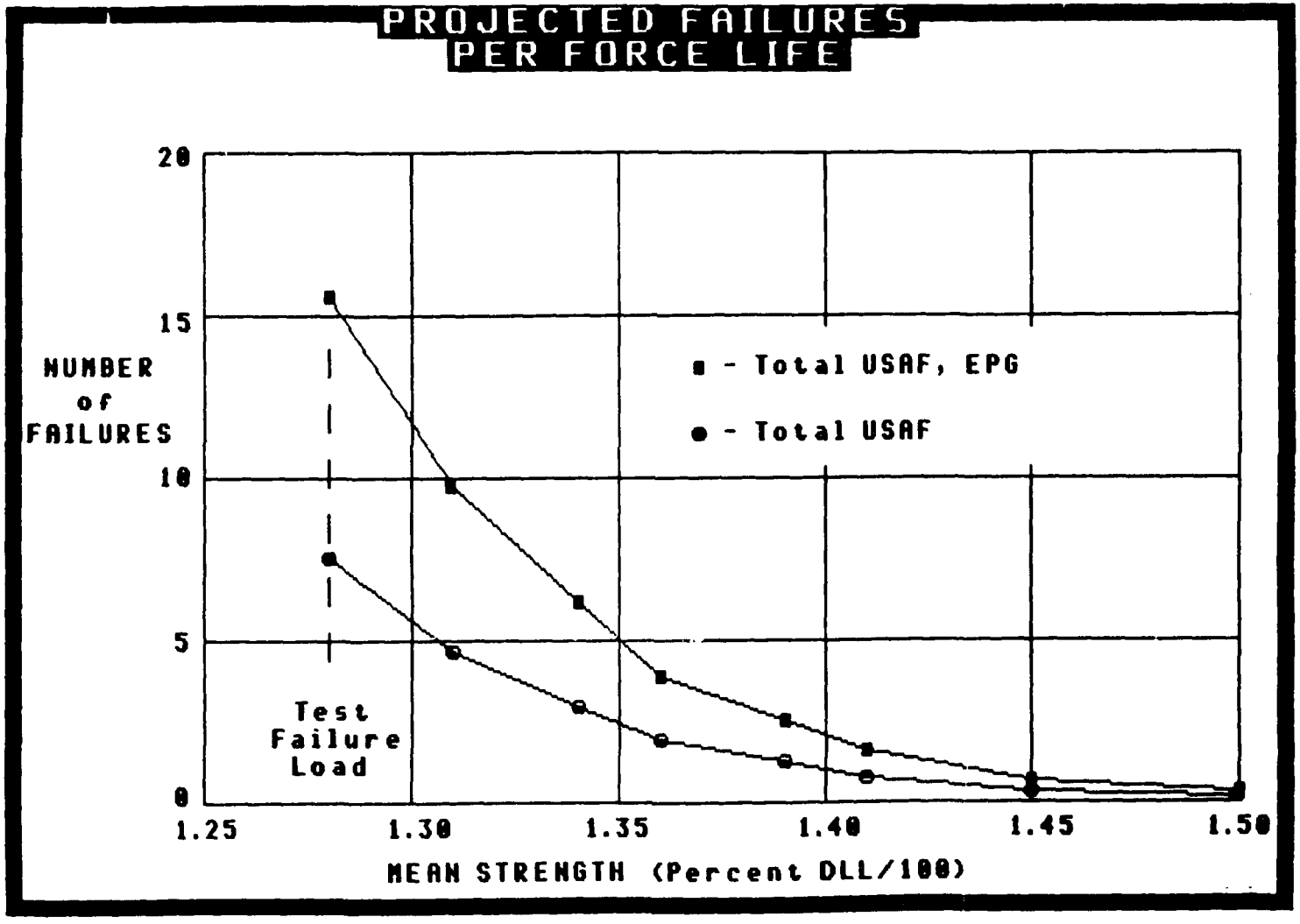


FIGURE 10. EXPECTED FAILURE FOR F-16 OPERATIONAL AIRCRAFT

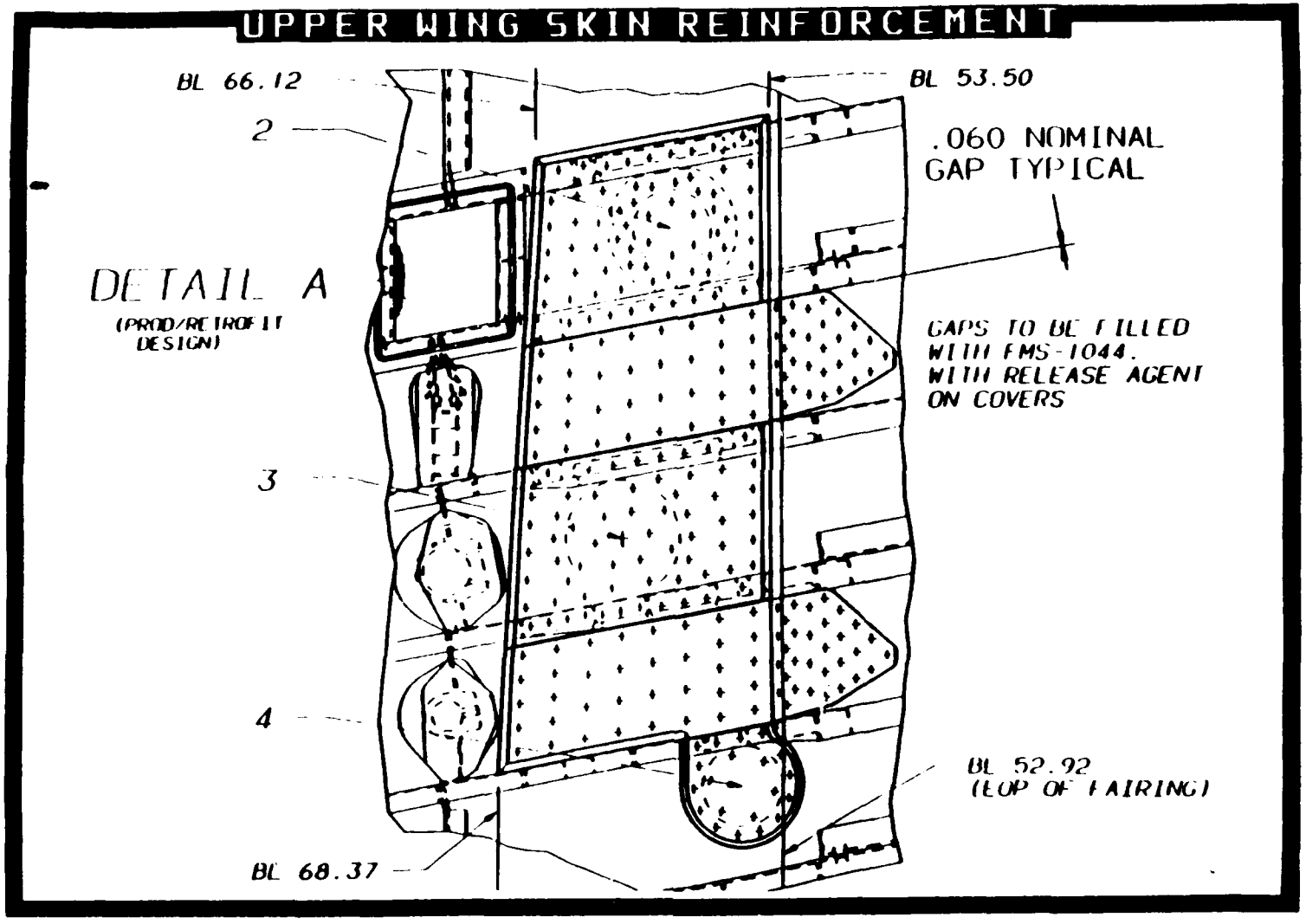


FIGURE 11. UPPER WING SKIN MODIFICATION

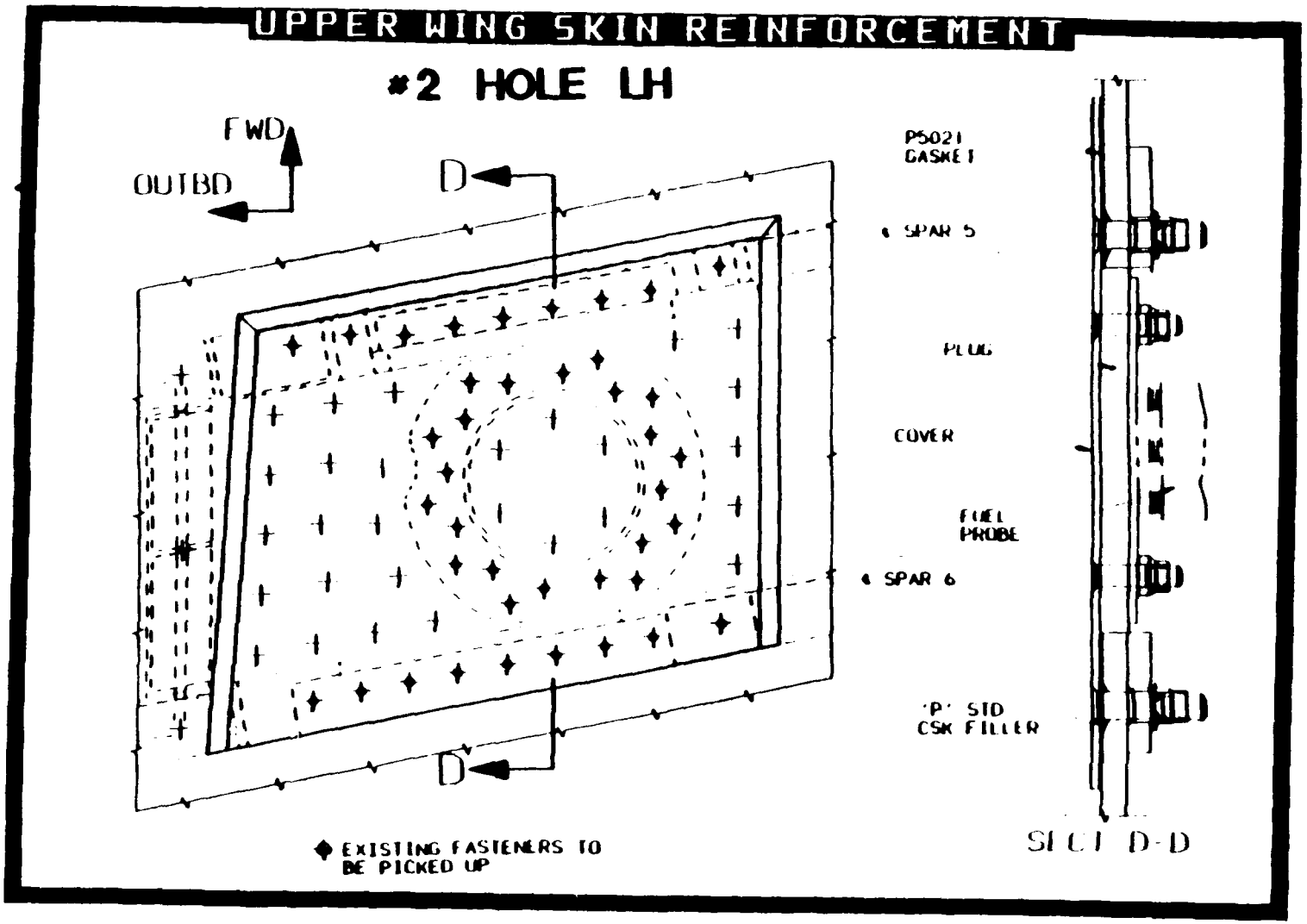


FIGURE 12. DETAIL OF UPPER WING SKIN MODIFICATION

GA 24241.1

ARALL® Laminates Scale-up from R&D to Flying Articles

1988 USAF Structural Integrity Program Conference

Hilton Palacio Del Rio Hotel
San Antonio, Texas
November 29 - December 1, 1988

R.J. Bucci
L.N. Mueller
M.A. Gregory
R.M. Bentley

Alcoa Laboratories, Alcoa Center, PA

ARALL LAMINATES - SCALE-UP FROM R&D TO FLYING ARTICLES

by

R. J. Bucci, L. N. Mueller, M. A. Gregory and R. M. Bentley
Alcoa Laboratories, Alcoa Center, PA 15069

at

1988 USAF Structural Integrity Program Conference
Hilton Palacio Del Rio Hotel, San Antonio, TX
November 29 - December 01, 1988

Abstract

ARALL Laminates are a new family of structural materials developed for fatigue critical applications requiring light gage sheet. The materials are bonded arrangements of thin aluminum alloy sheets and alternating plies of epoxy-adhesive impregnated with reinforcing fibers. The principal benefit of the resulting hybrid composite is ability to impede and self arrest crack growth. ARALL Laminates also display a range of impressive, albeit directional, property improvements over those of monolithic high strength aluminum, and feature performance traits competitive with those of advanced composites. These attractive features include: 15-20% lower density than aluminum; up to 60% higher strength than 7075 and 2024 aluminum at comparable stiffness; fabricability comparable to metal (e.g., can be cut, sawed, drilled, joined and inspected by conventional metal practices); outer metal layers protect fiber/resin system from damage by moisture, thermal attacks, lightning strike and impacts; damping ability superior to monolithic aluminum. Envisioned aircraft usages include: tension dominated fatigue and fracture critical structure (e.g., lower wing and fuselage skins), damping critical structure, lightning strike areas and structure requiring resistance to impacts, where goals fo 15-40% weight savings have been demonstrated to be in reach.

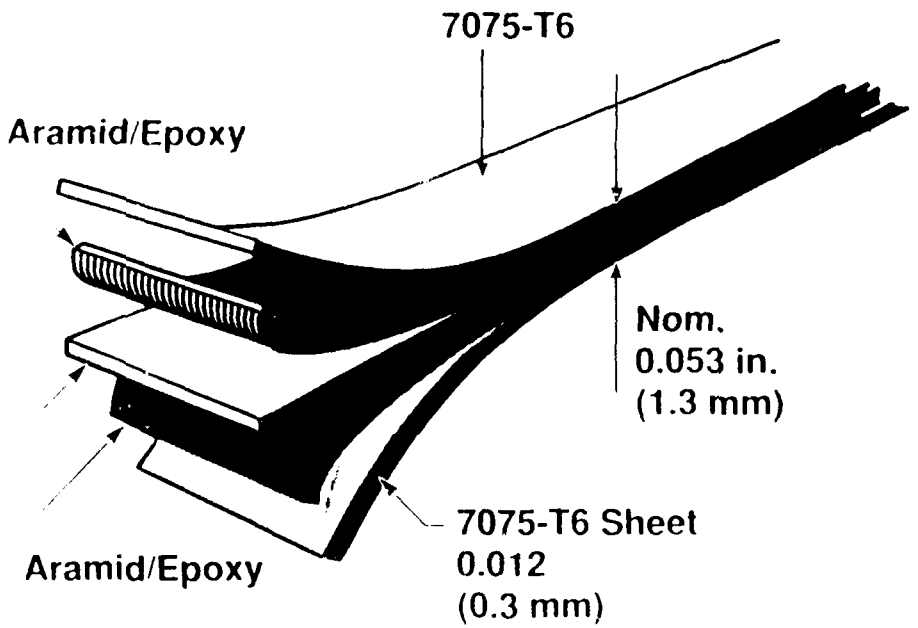
ARALL Laminates are now transitioning from the research/development and test/evaluation stages with the advent of flying articles. Many aircraft companies have internal ARALL Laminates programs ongoing, and industry specifications and certification issues are being brought to the forefront. This presentation will summarize results from more than seven years of extensive testing on these materials and review status of commercialization and implementation efforts.

GA 21737 25

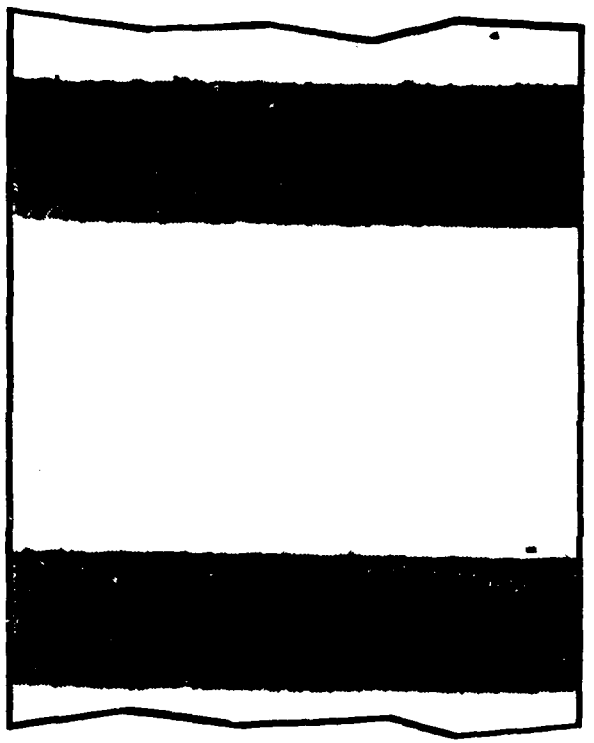
ARALL® Laminates

- Family of new hybrid structural laminates
- Combines advantages of high strength aluminum sheet with strong aramid fibers
- Invented at Delft University (in cooperation with Fokker Aircraft)
- Further developed and commercialized by Alcoa

Alcoa ARALL® Laminates



Schematic of
3/2 ARALL® -1 layup



Magnified cross-section
showing aluminum
middle layer and
aramid prepreg layers

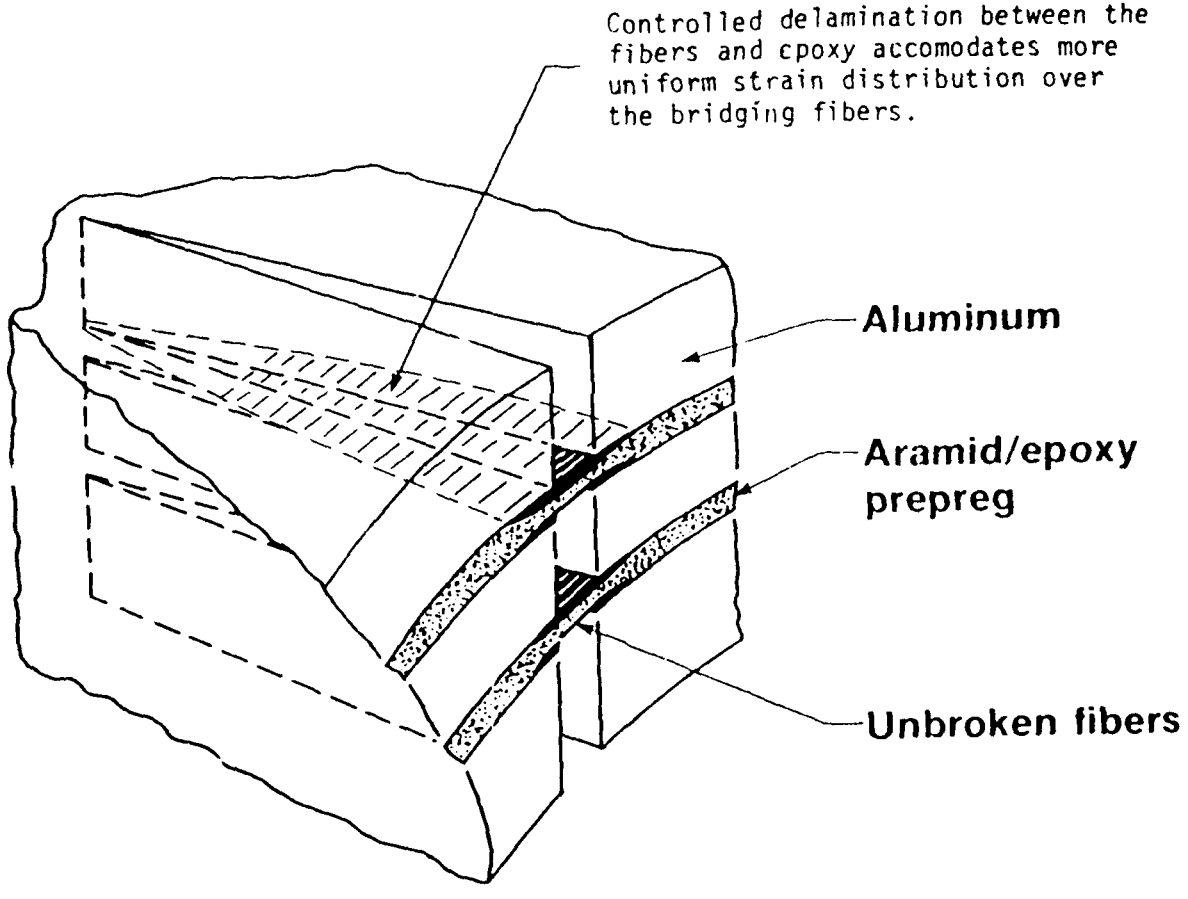
GA 24241 2

ALCOA

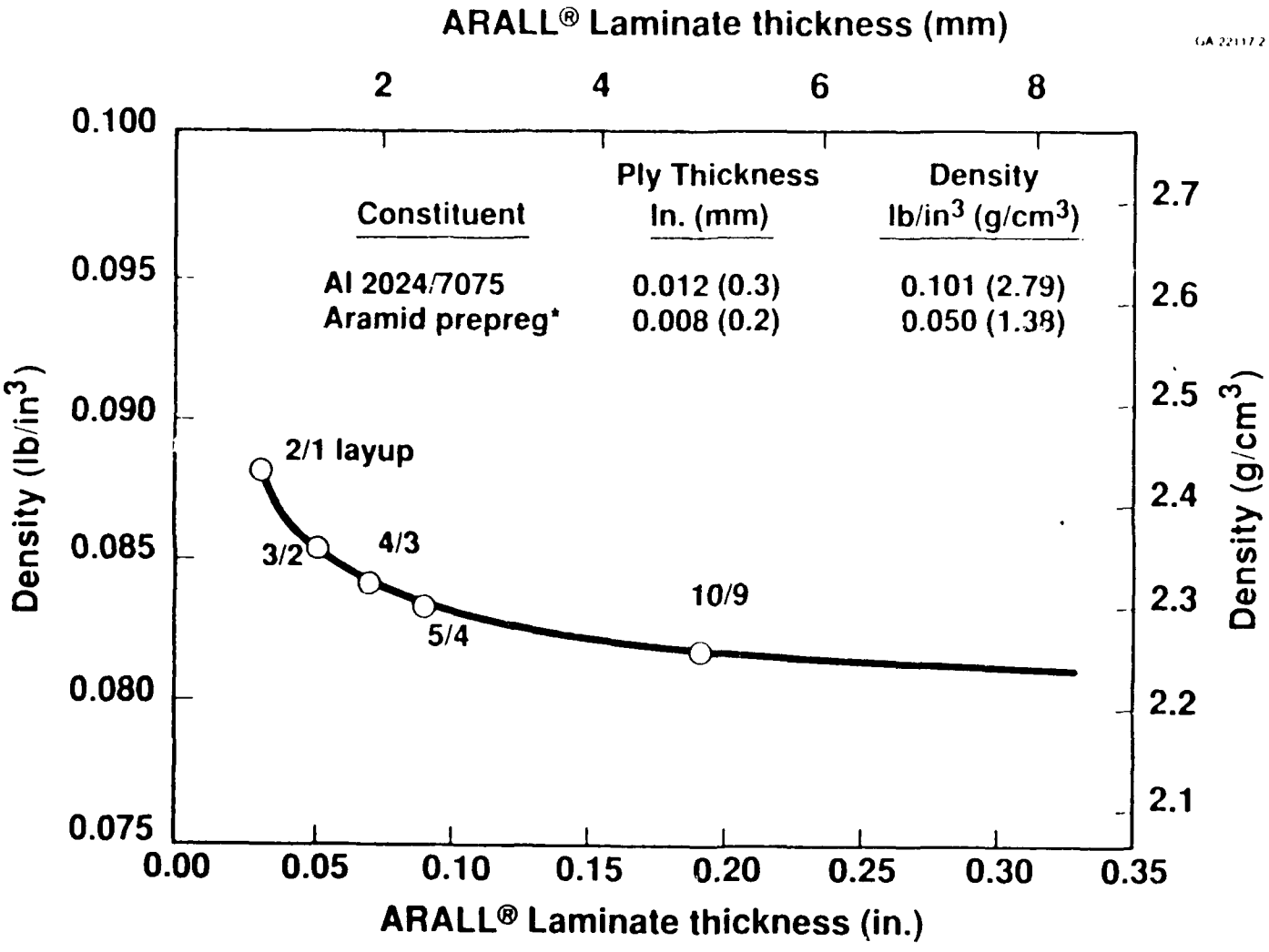
ARALL® Laminates (What Do They Offer?)

- Excellent fatigue resistance
- Up to 60% higher directional strength than aluminum at comparable stiffness
- 15-20% lower density than aluminum
- Retains advantages of metals over pure composites: plasticity, formability, ease of manufacture, lower cost and supportability
- Outer metal layer provides impact resistance, moisture barrier, lightning protection and inspectability
- Vibration damping ability ~ 3X better than aluminum
- Potential for 20-40% structural weight saving

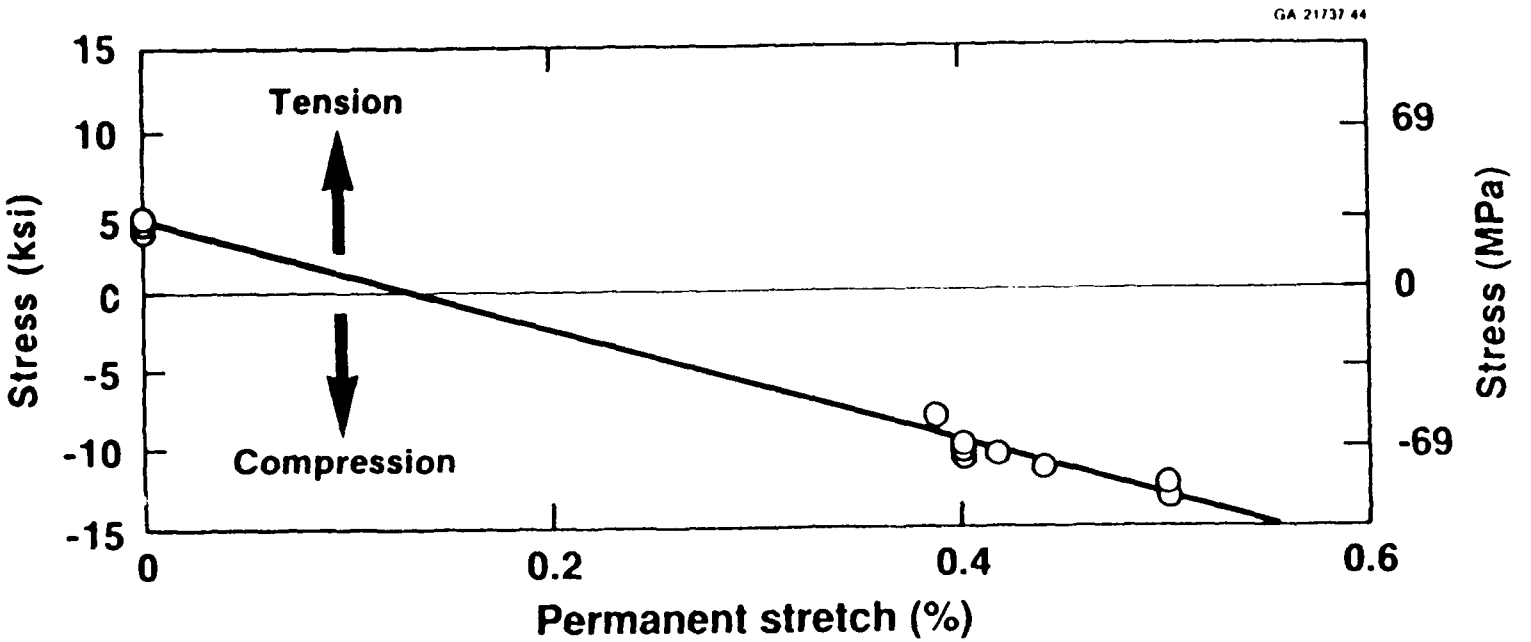
Fatigue Crack in 3/2 ARALL® Laminate



... Drawing illustrates how ARALL derives its good FCG resistance. Cracks which initiate first in the metal layers tend to grow very slow, or self-arrest as load is transferred from the metal to the stronger aramid fibers.



ARALL® Laminate
Calculated Density vs. Product Thickness



Aluminum Layer Residual Stress
3/2 ARALL®-1 Laminate, 0.053 Inch (1.3 mm)

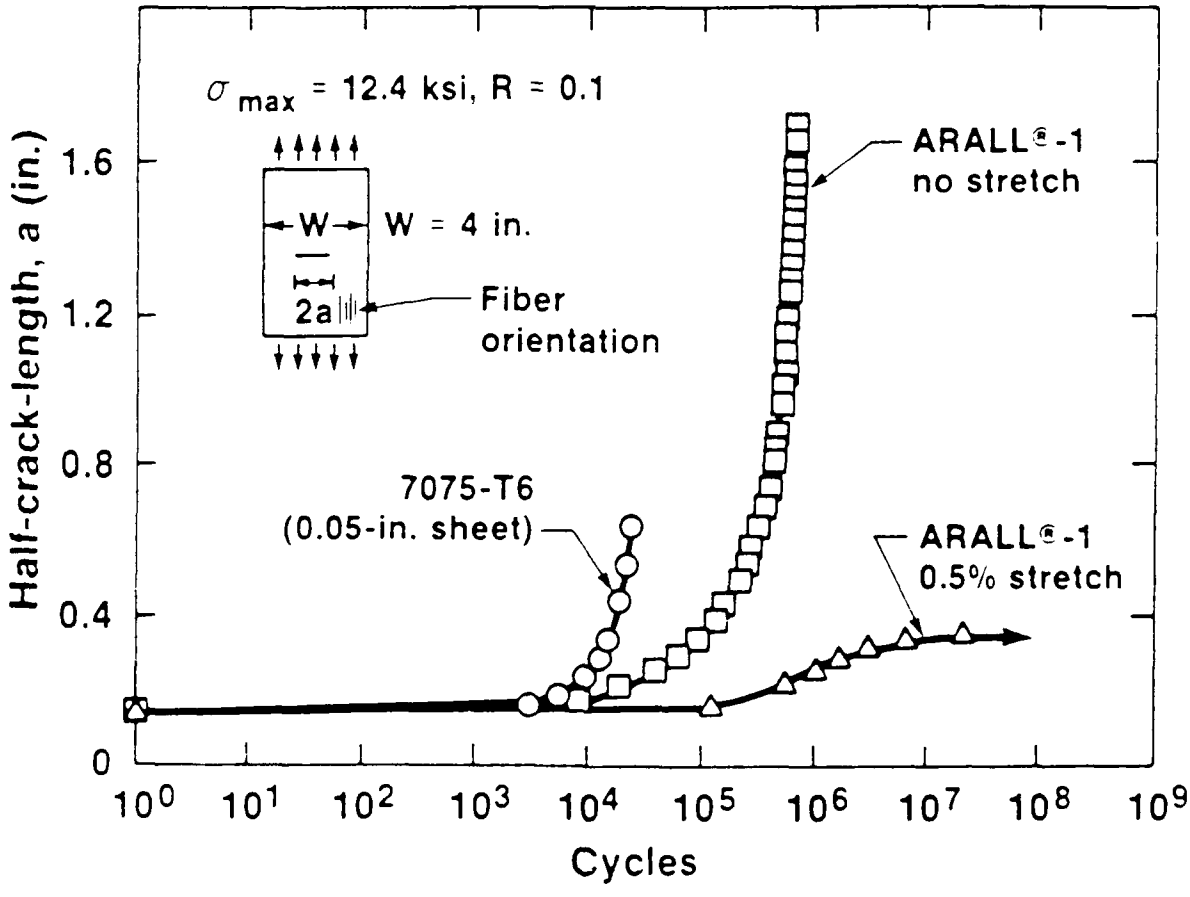
... Post-cure stretching is used to convert ARALL metal layer residual stresses from tension to compression.

3/2 ARALL®-1 Laminate (0.053 in.) Tensile and FCG Properties Before and After Stretch

Tensile (Longitudinal)

Property	No Stretch	0.5% Stretch	7075-T6 0.050 in. sheet
TUS (ksi)	116	116	83
TYS (ksi)	72	93	74
E (msi)	9.3	9.8	10.3
ε_f (%)	2.5	1.9	12

Fatigue Crack Growth



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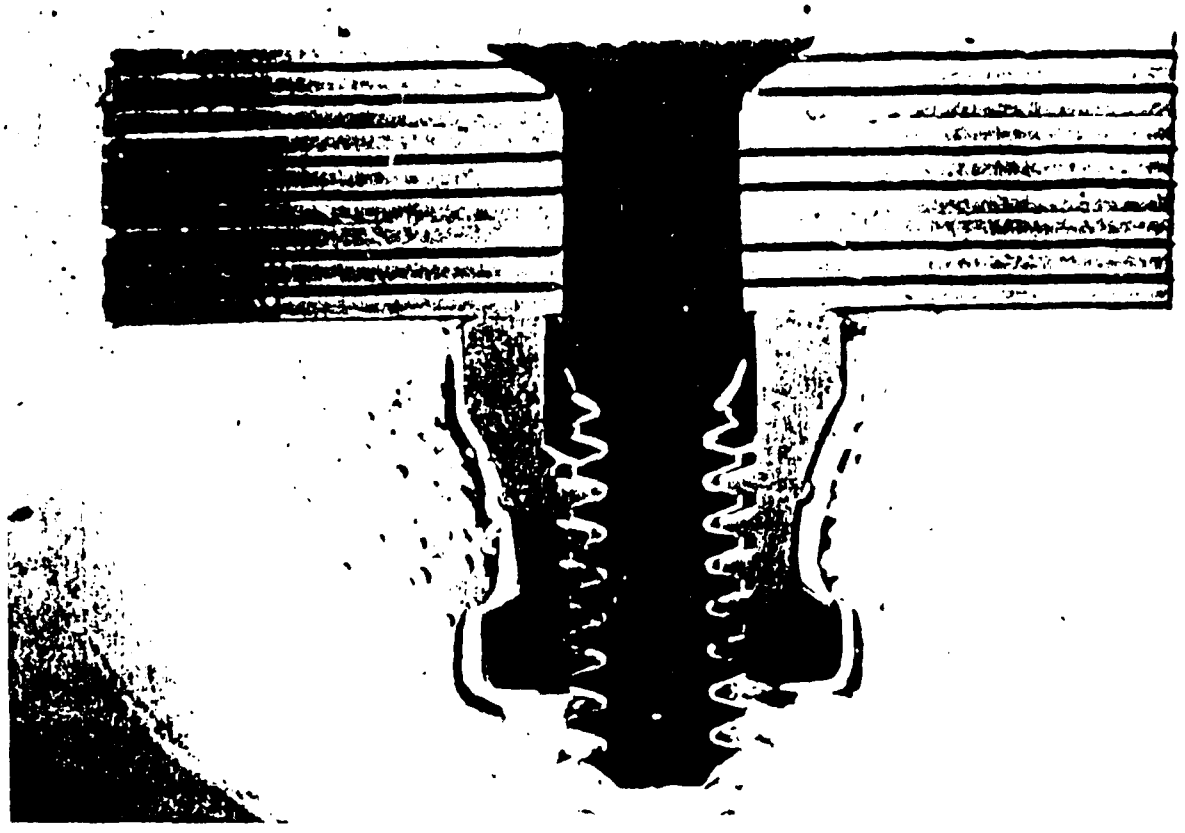

(Courtesy University of DELFT)

ARALL® Laminate Uses Traditional Metal Fabricating Practices

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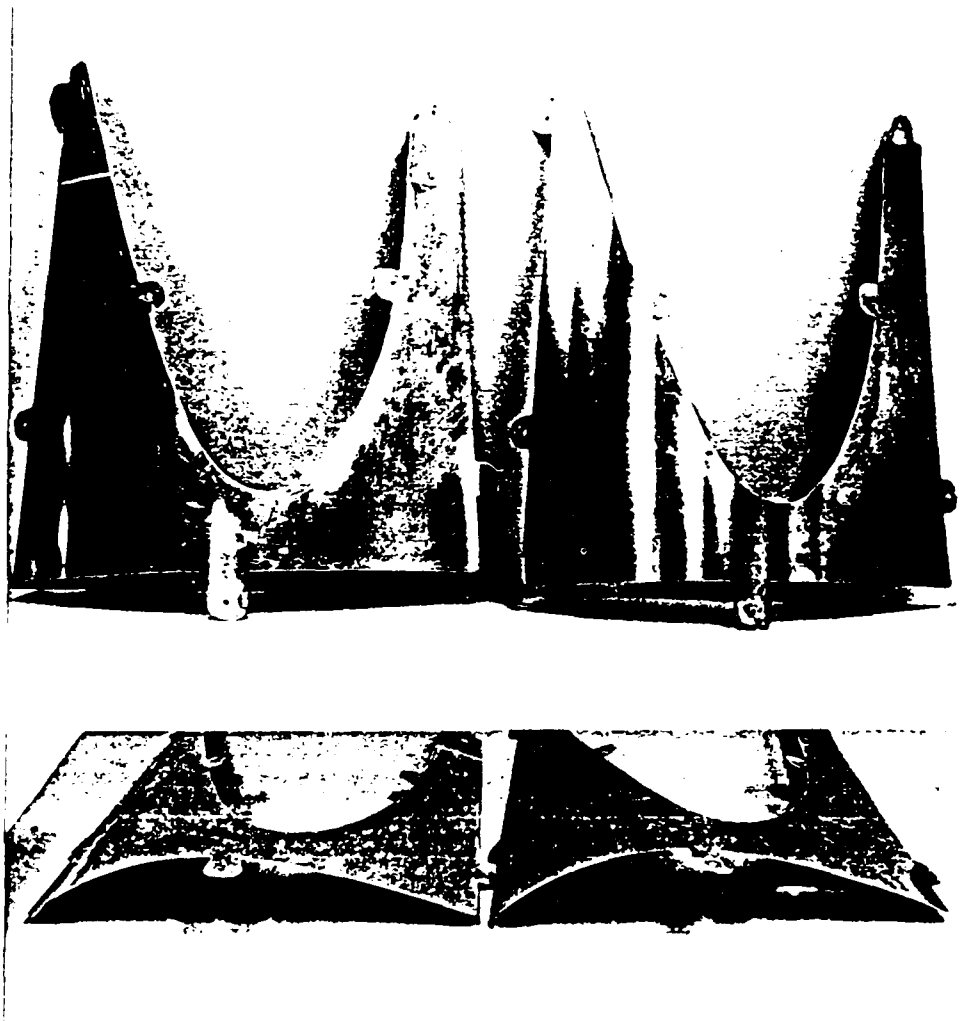
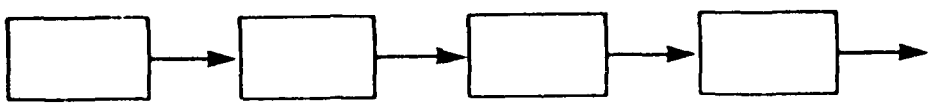


2'1 ARALL-2 2024-T3

Lower Nacelle Cover After Routing Formed Part.

... The figure presents an example stretched formed part (cold stretch after cure), and shows that ARALL can be machined like metal.

ARALL® Laminate Processing Technology. . .



- Aluminum sheet surface treatment
- Prepreg
- Adhesive bonding
- Stretching
- Inspecting

. . .Consistent Quality

GA-24241.3

ALCOA

ARALL® Laminates
Aerospace Product Line

ARALL® Type	Description	Characteristic	Introduction	AMS Specs.	MIL-HDBK-5 Allowables
1	7475-T61 250°F aramid prepreg Unidirectional 0.4% stretched	Highest strength	1984	1988	1988
2	2024-T3 250°F aramid prepreg Unidirectional No stretch	Best formability	1984	1988	1988
3	7475-T761 250°F aramid prepreg Unidirectional 0.4% stretched	Improved toughness, corrosion, formability	1987	1989	1989
4	2024-T8 350°F aramid prepreg Unidirectional No stretch	Elevated temperature	1988	1989	1989
5, 6...	Various aluminum, fiber and cross-ply combinations	Tailored properties	1989 & beyond	1990 & beyond	1990 & beyond

Fatigue resistance good to excellent, all product forms.
ALCLAD available - one or two sides.

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ARALL® Laminate 1-4
Standard Sheet Gauges and Width Capabilities*

Layup Range	Gauge Range in. (mm)	Width in. (mm)	Length in. (mm)
	1988		
2/1 - 5/4	0.033 - 0.094 (0.83 - 2.4)	50 (1270)	90 (2286)
	1989		
No practical limitation		60 (1524)	144 (3658)

* Special sizes and tailored layup combinations are negotiable

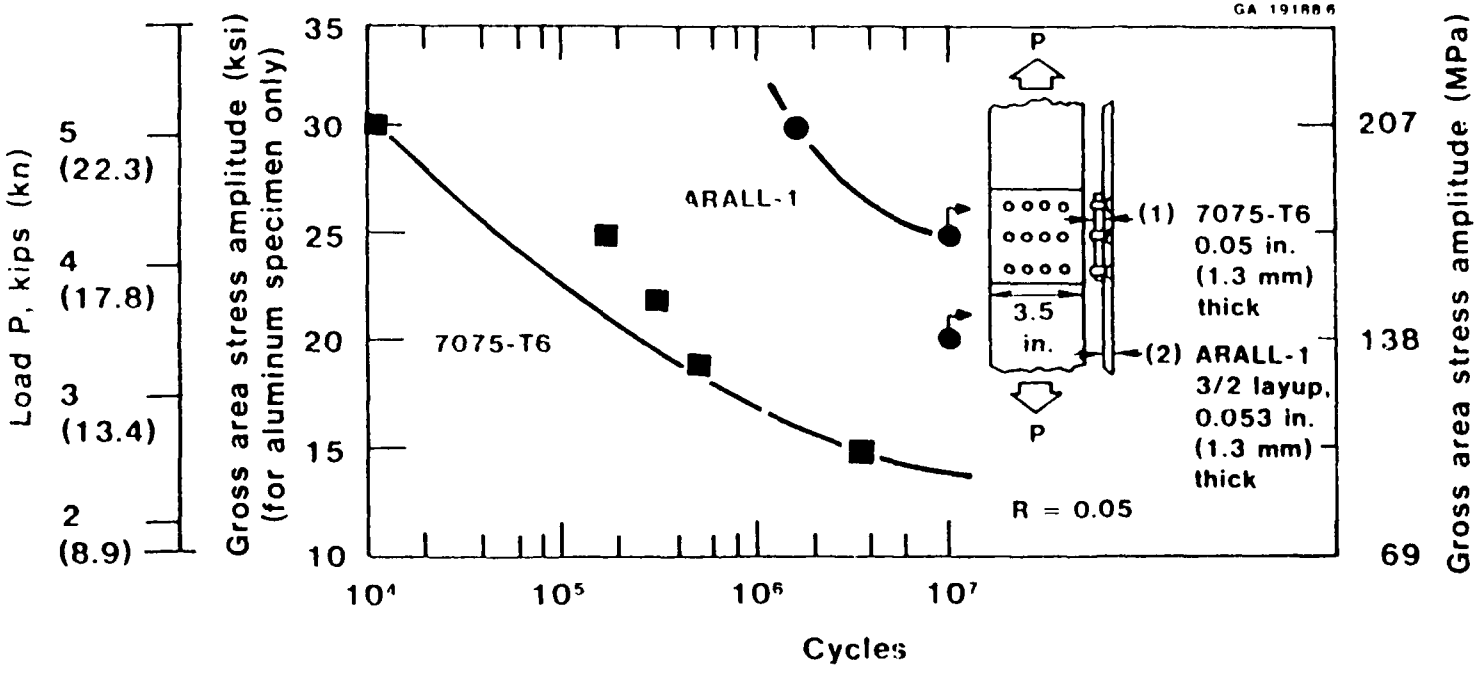
ALCOA

ARALL LAMINATE, ALUMINUM SHEET AND CARBON FIBER COMPOSITE ROOM TEMPERATURE TYPICAL MECHANICAL PROPERTIES

Property	Test Direct.	3/2 ARALL Laminate, 0.053 in. (1.3mm) thick				Aluminum Alloy Sheet, 0.063 in. (1.6mm) thick				Carbon Fiber Composite (a)	
		ARALL-1	ARALL-2	ARALL-3(b)	ARALL-4(b)	7075-T6	2024-T3	7475-T76	2024-T8	Unidirectional (c)	Typical Structure (d)
Tens. Ultimate Strength, ksi (MPa)	L	116(800)	104(717)	120(828)	106(731)	83(572)	66(455)	79(545)	72(497)	180(1241)	95(655)
	LT	56(386)	46(317)	54(373)	49(338)	83(572)	65(448)	80(552)	72(497)	8(55)	40(276)
0.2% Off, Tens, Yld. Strength, ksi (Mpa)	L	93(641)	52(359)	85(587)	54(373)	74(510)	52(359)	69(476)	64(442)	---	---
	LT	48(331)	33(228)	46(317)	46(317)	72(496)	47(324)	69(476)	64(442)	---	---
Tens. Elastic Modulus, msi (GPa)	L	9.8(68)	9.3(64)	9.8(68)	8.9(61)	10.3(71)	10.5(72)	10.0(69)	10.5(72)	21.0(145)	11.0(76)
	LT	7.0(48)	7.1(49)	7.4(51)	7.1(49)	10.3(71)	10.5(72)	10.0(69)	10.5(72)	1.7(12)	5.0(35)
Tens. Elong., %	L	0.7(e)	1.4(e)	1.0(e)	1.4(e)	11	18	12	8	---	---
	LT	7.1(e)	12.0(e)	---	3.9(e)	11	18	12	8	---	---
Tens. Tot. Strain to Failure, %	L	1.9	2.5	2.2	2.6	12	19	13	9	0.9	0.5
	LT	7.9	12.7	---	4.6	12	19	13	9	0.5	0.5
Poisson Ratio (Tension)	L	0.33	0.32	---	---	0.33	0.33	0.33	0.33	0.31	0.48
	LT	0.25	0.26	---	---	0.33	0.33	0.33	0.33	0.017	0.23
0.2% Off. Compr. Yld. Strength, ksi (MPa)	L	54(372)	38(262)	---	---	73(503)	44(304)	64(442)	63(435)	180(1241)(f)	95(655)(f)
	LT	57(393)	34(234)	---	---	76(524)	50(345)	68(469)	64(442)	30(207)(f)	42(290)(f)
Compr. Elastic Modulus, msi (GPa)	L	10.2(70)	9.7(67)	---	---	10.5(72)	10.7(74)	10.5(72)	10.7(74)	21(145)	11.0(76)
	LT	7.5(52)	7.6(52)	---	---	10.5(72)	10.7(74)	10.5(72)	10.7(74)	1.7(12)	5.0(35)
0.2% Off. Shear Yld. Strength, ksi (MPa)	L-LT	---	17(117)(g)	---	---	---	30(207)(g)	---	---	12(83)(h)	37(255)(h)
	LT-L	---	16.5(114)(g)	---	---	---	30(207)(g)	---	---	---	---
Shear Modulus, msi (Gpa)	L-LT	2.4(17)(g)	2.5(17)(g)	---	---	3.9(27)	4.0(28)	3.8(26)	4.0(28)	0.65(4.5)	3.1(21)
	LT-L	---	2.3(16)(g)	---	---	3.9(27)	4.0(28)	3.8(26)	4.0(28)	---	---
Bearing Ult. Strength @ e/D = 1.5, ksi (Mpa)	L	95(655)	77(531)	---	---	129(890)	110(759)	120(828)	107(738)	---	---
	LT	102(703)	79(545)	---	---	129(890)	110(759)	120(828)	107(738)	---	---
Bearing Ult. Strength @ e/D = 2.0, ksi (Mpa)	L	107(738)	82(565)	---	---	166(1145)	137(945)	153(1055)	136(938)	---	---
	LT	105(724)	79(545)	---	---	166(1145)	137(945)	153(1055)	136(938)	---	---
Bearing Yld. Strength @ e/D = 1.5, ksi (MPa)	L	85(586)	56(386)	---	---	110(759)	78(538)	100(690)	88(607)	---	---
	LT	88(607)	56(386)	---	---	110(759)	78(538)	100(690)	88(607)	---	---
Bearing Yld. Strength @ e/D = 2.0 ksi (Mpa)	L	102(703)	66(455)	---	---	128(883)	94(649)	113(780)	100(690)	---	---
	LT	97(669)	64(441)	---	---	128(883)	94(649)	113(780)	100(690)	---	---

(a) DoD/NASA Advanced Composites Design Guide, Vol.1A, generic F180 graphite/epoxy composite, 60% fiber content, 0.056 lb./cu. in. (1.55g/cu. cm); e.g. AS/3501-6, T300/5208.
(b) Single laboratory lot.
(c) 100% fibers in L direction.
(d) Fiber orientations: 42% 0 degree, 50% ± 45 degree, 8% 90 degree.
(e) Plastic strain determined from test record by back extrapolating elastic slope from point of fracture.
(f) Compressive ultimate strength.
(g) Iosipescu shear test method; L-LT means fibers parallel to long specimen axis, LT-L means fibers normal to long specimen axis.
(h) In-plane shear ultimate.

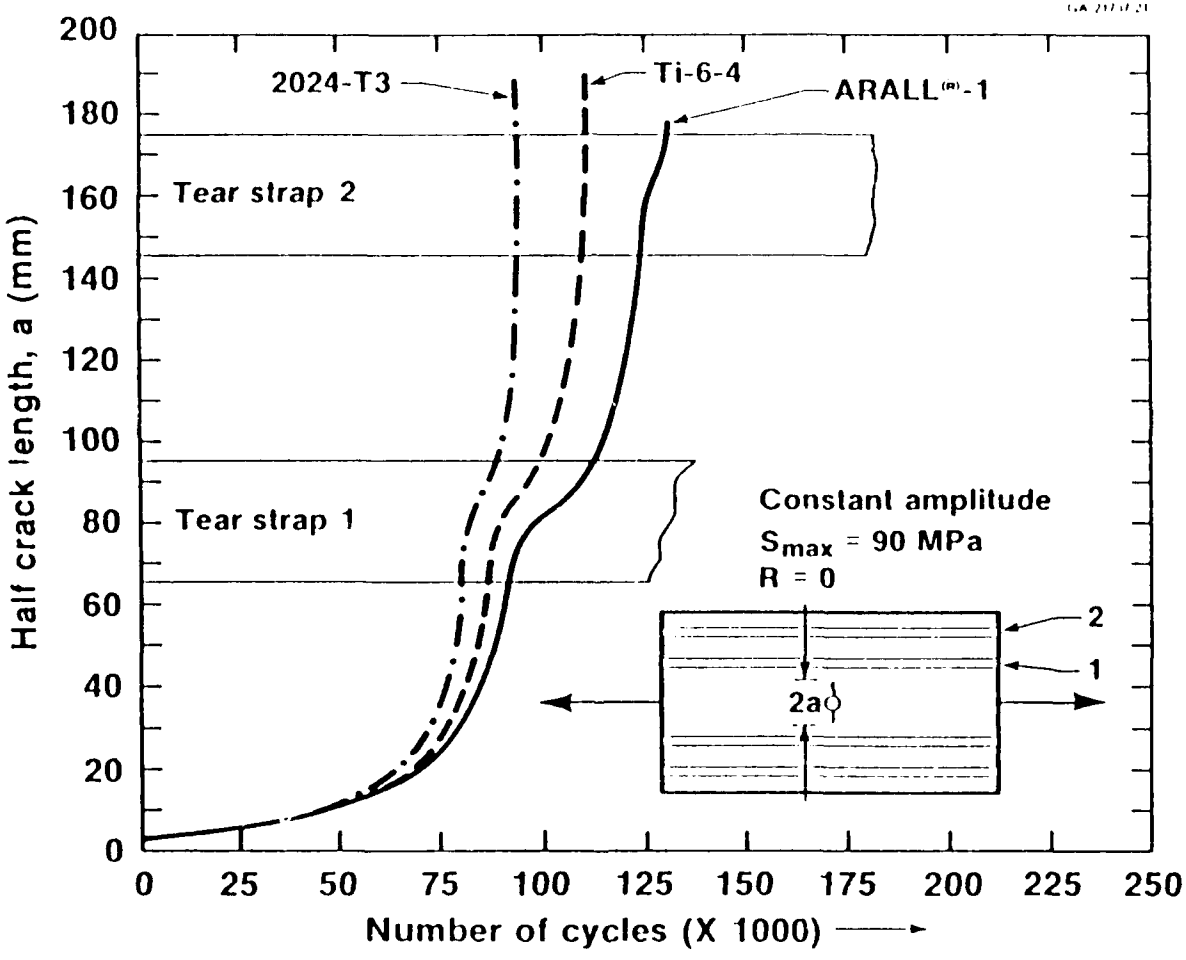
Joint Fatigue Strength of ARALL-1 Laminate and 7075-T6 Aluminum Sheet



(Courtesy McDonnell Douglas, Long Beach, CA)

Crack Growth Comparison of CCT Panels Stiffened with Riveted 2024-T3 Clad, Ti-6Al-4V Annealed and 3/2 ARALL®-1 Laminate Tear Straps

(Courtesy Delft University of Technology)



... In addition to being an effective crack growth retardant, ARALL tear straps would save weight.

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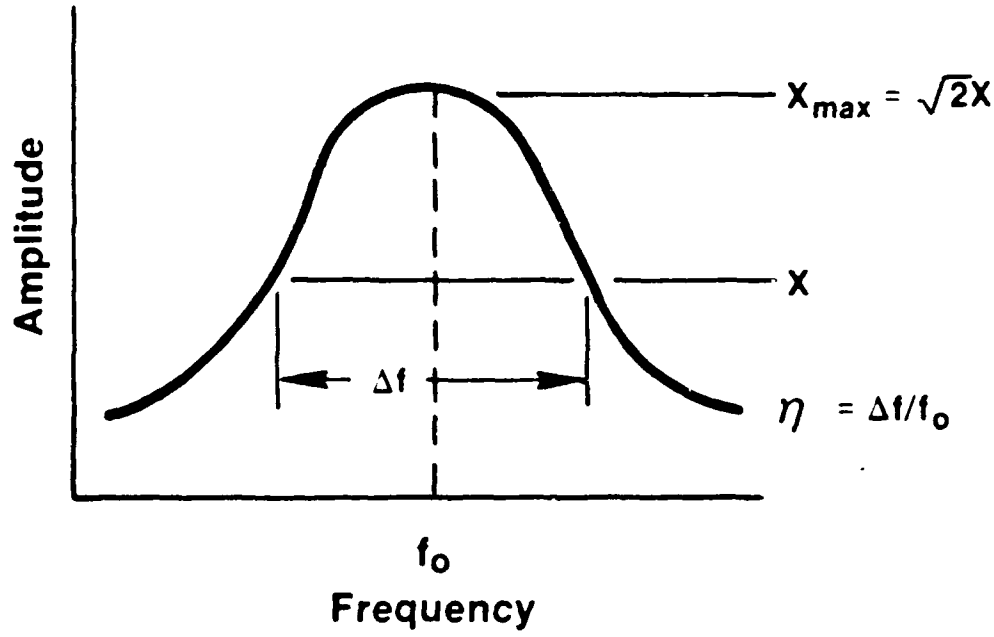
GA-19529 3C

Damping Characteristics
ARALL®-1 Laminate vs. Monolithic Aluminum
(Courtesy Naval Research Labs)

Material	Loss factor η (X 10^{-4})
ARALL®-1 Laminate (longitudinal)	21.0
ARALL®-1 Laminate (transverse)	16.0
2024 aluminum	7.2
6061 aluminum	6.7



... The table shows ARALL damping ability to be about three times better than that of monolithic aluminum.

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GA 21737 2

ARALL® Laminate
Ability to Cope with Lightning Strike

	Response to Zone 2A swept lightning strike (nominal 50 msec. dwell time)	
Gage (in.)	2024-T3 sheet	ARALL®-1 laminate
0.052	—	No molten metal
0.063	Burn through	—
0.080	Molten metal on opposite side	—
0.090	No molten metal	—

Ref: Ioannu et. al., deHavilland, 14th ICAF Symposium

... Table shows monolithic aluminum requires about two times ARALL thickness to get same lightning protection from burn-through.

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ARALL® Laminates
Candidate Aerospace Applications

- Fatigue/fracture critical areas
  - — Lower wings
  - — Fuselage
  - — Tail structure
  - — Tear straps
- Damping critical areas
  - — Rear fuselage on UDF engine transports
  - — Leading edges with buffeting problems
  - — Entire fuselages for lower cabin noise
- Lightning strike protection
  - — Areas around fuel tanks

GA-24241-6

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ARALL® Laminates - Remarks

- Present fatigue critical structure, the application driver.
- Fatigue resistance so good, part gauge (weight saving) controlled by other properties.
- Technical community doing excellent job of highlighting attributes and concerns.
- Users have application programs underway.
- Envisioned certification path similar to that for bonded metal structure.

GA-24241.5

ALCOA

ARALL® Laminates - Usage Issues

- Design :
  - Compressive, shear & bearing allowables
  - Blunt notch & discrete source damage residual strength
  - Transverse properties
  - Early Initiation of fatigue cracks
- Manufacturing:
  - Formability
  - Debonding during manufacture
- Service:
  - Bond durability experience
  - Maximum use temperature 325°F (160°C)

... Resolution possible through:

- Engineering approach
- Manufacturing controls
- Modified ARALL Laminate variants

... Much of the remaining presentation focuses on these issues.

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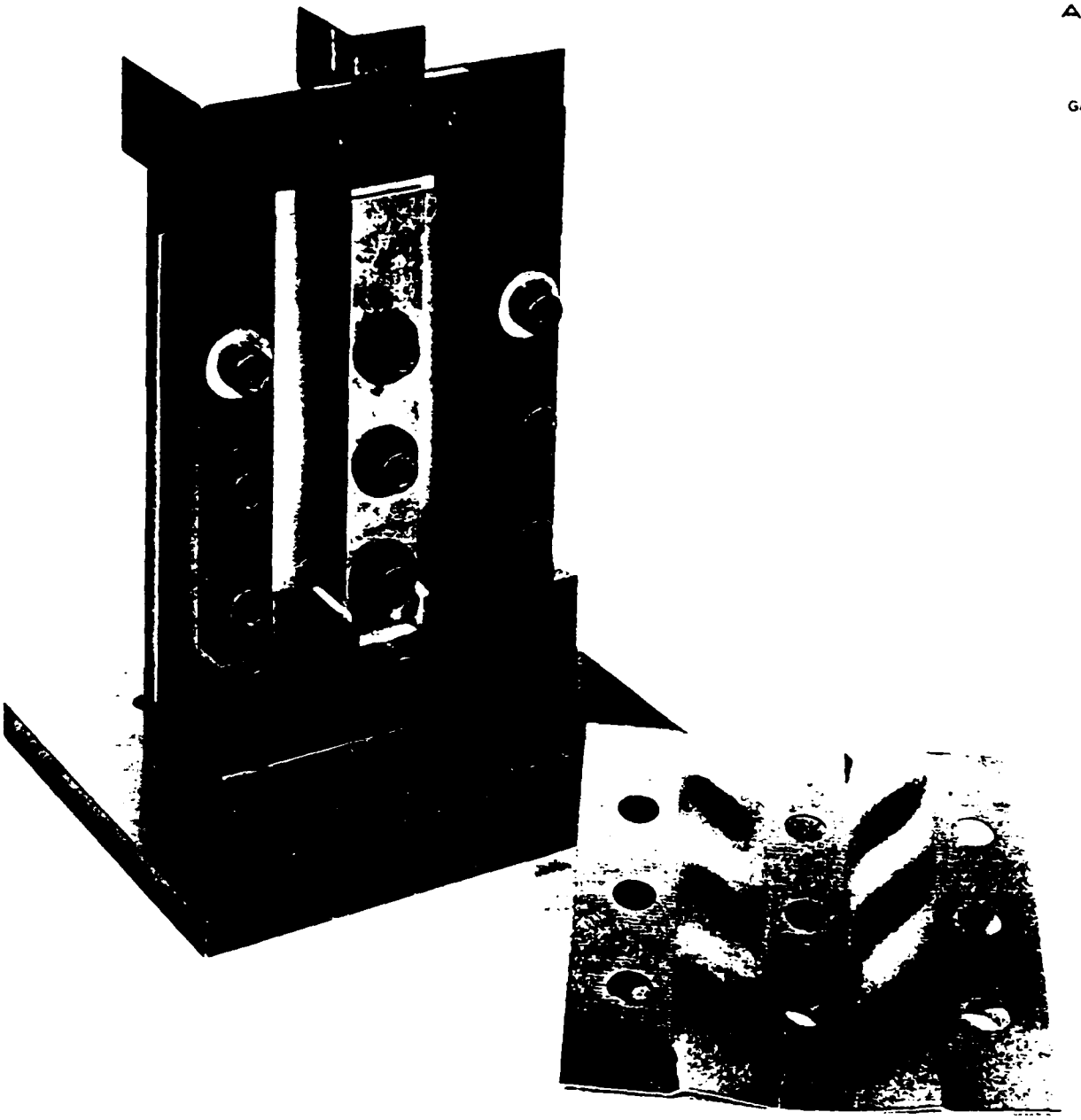
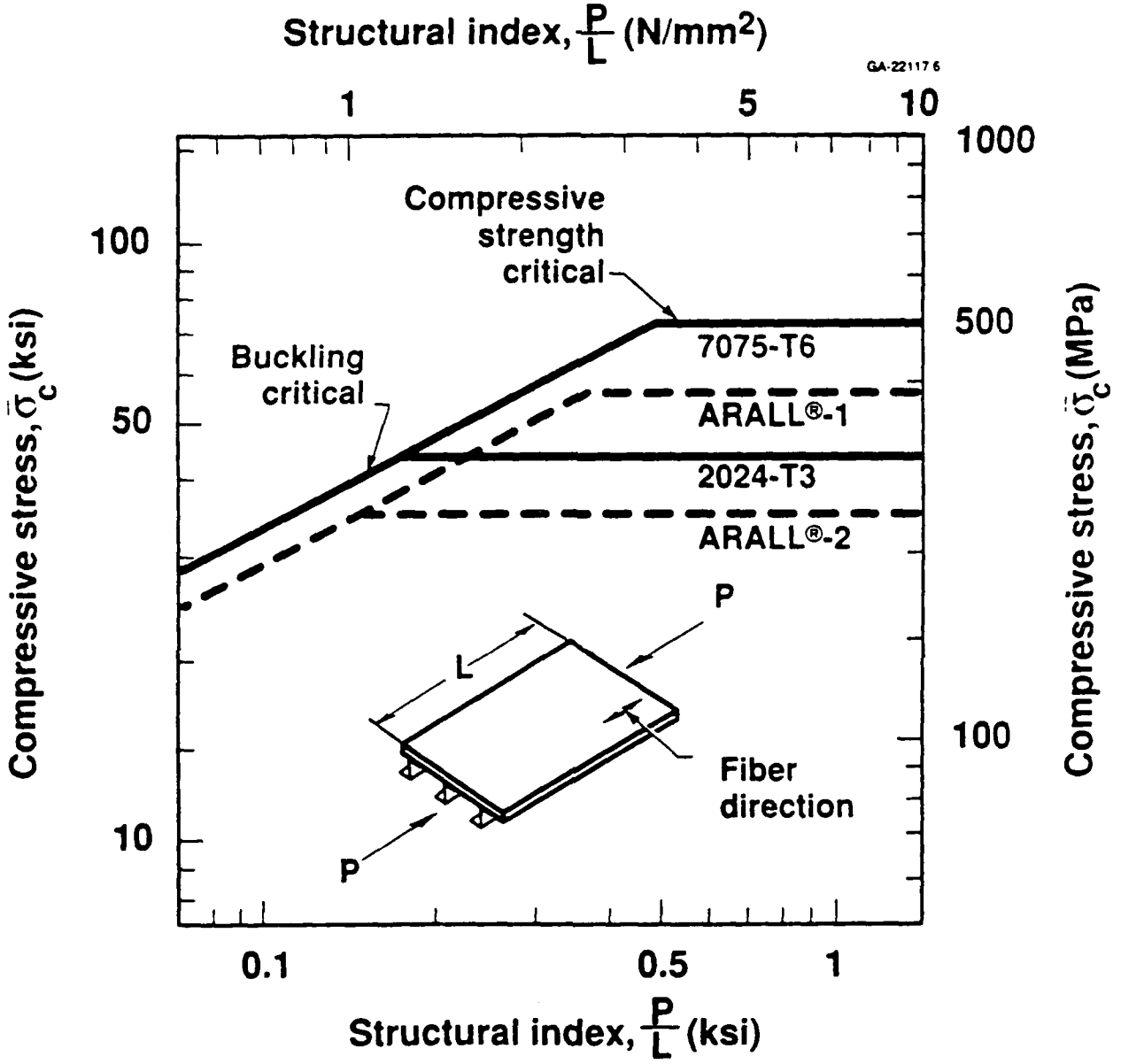
GA 19662 23



2/1 ARALL®-2 Laminate Fails by Plastic Buckling in Three-Rail Shear Test

... Figure illustrates that ARALL shear-buckling/crippling failure mode is similar to that of aluminum. So unlike epoxy-based fiber composites, ARALL provides some warning prior to failure by shear or compressive loadings.



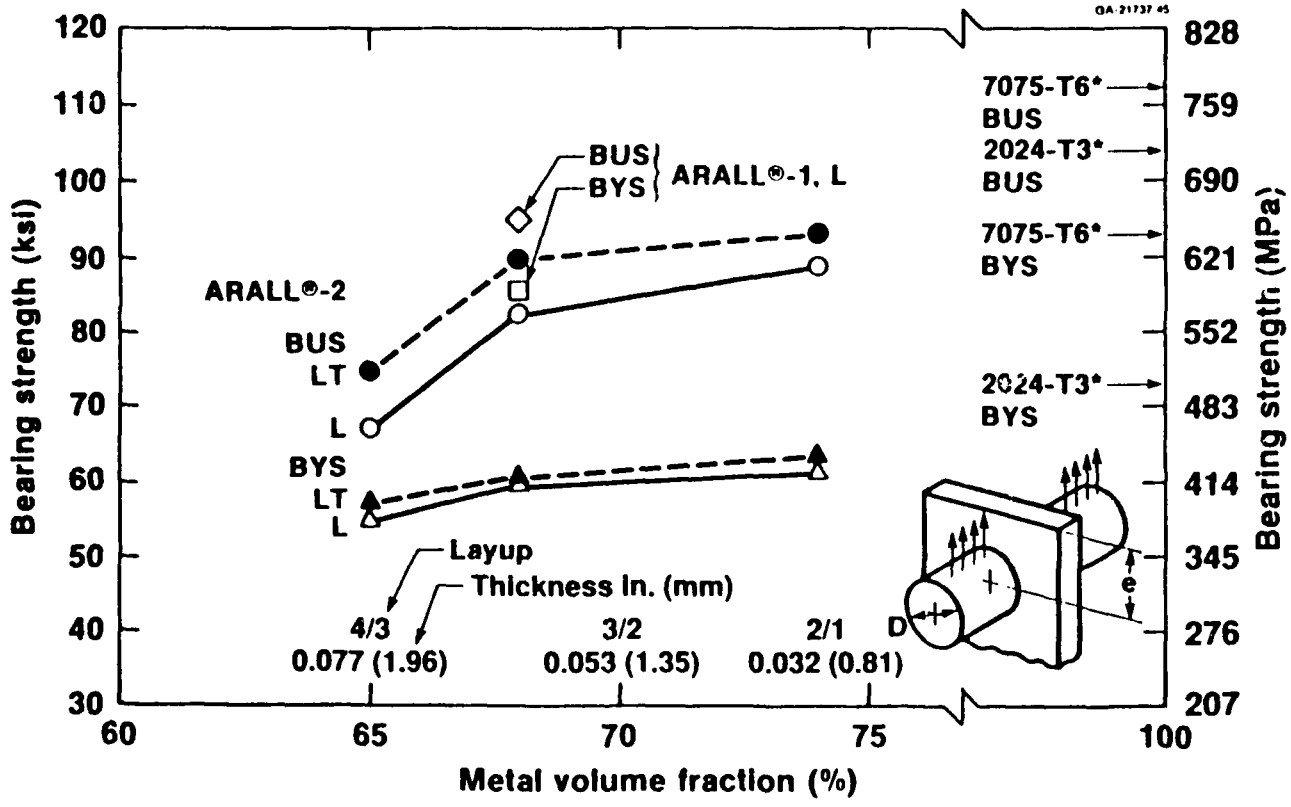
ARALL® Laminate vs. Aluminum Stiffened Panel Compressive Strengths

(Courtesy Delft University of Technology)

... Because aramid fibers contribute little in compression, ARALL metal volume fraction dictates ARALL comressive strength behavior, i.e. rule of mixtures applies. APALL compressive strength is about 15% less than that of monolithic aluminum. However because its density is lower, ARALL compressive structure can be designed to the same structural efficiency as aluminum.

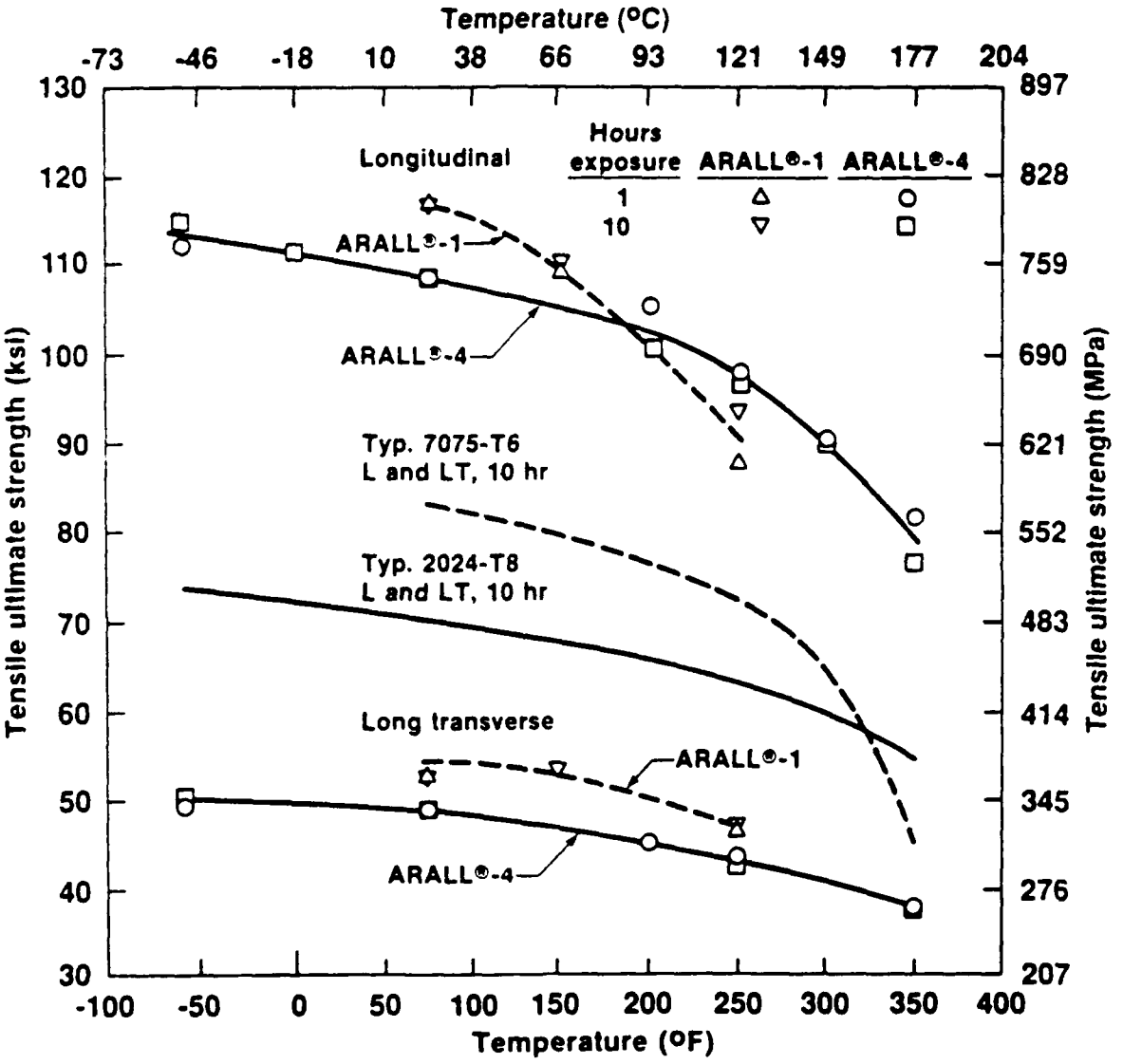
ALCOA

ARALL® Laminate
Pin Load Bearing Test Results (e/D = 1.5)



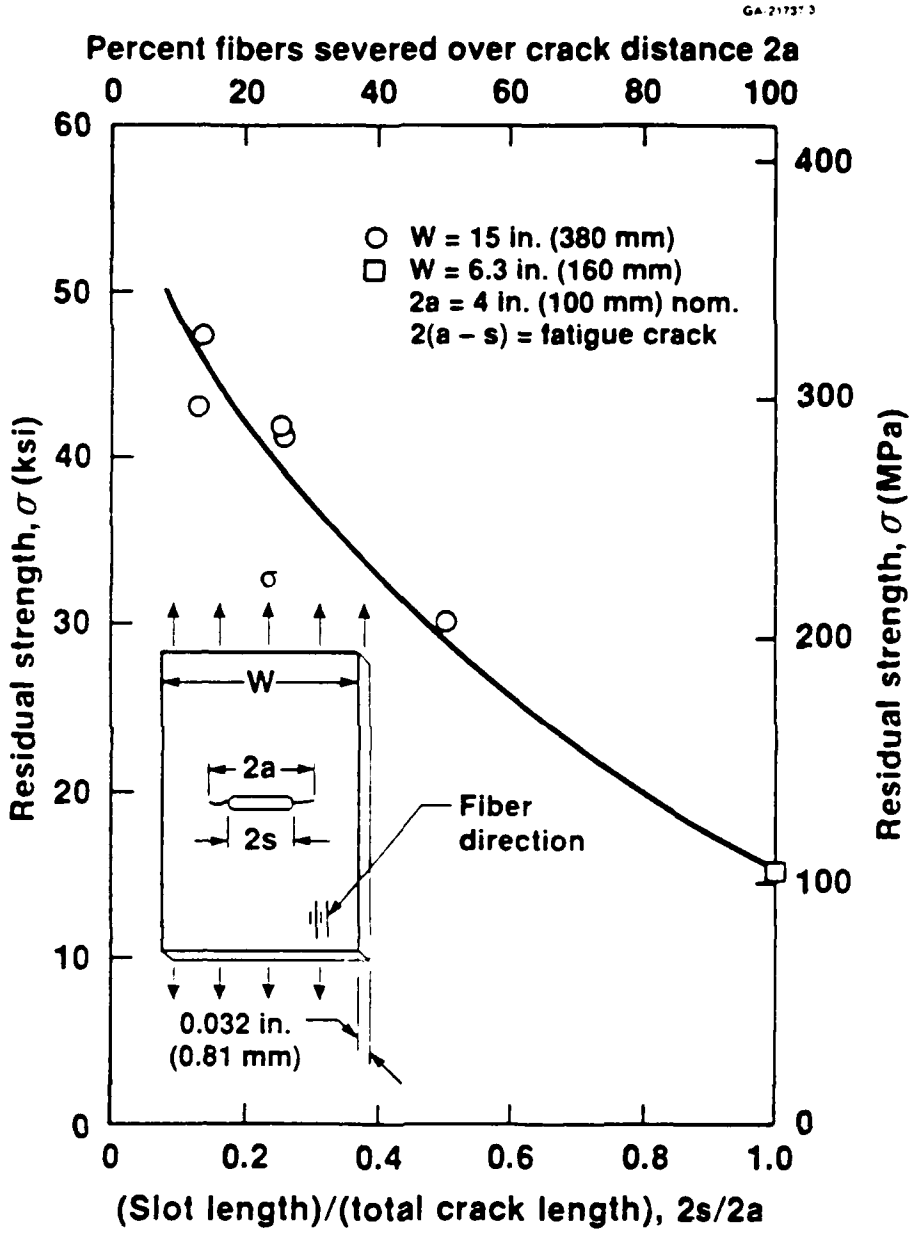
* Mil Hdbk 5 "A" value (0.063 in.) aluminum sheet

... ARALL bearing properties controlled by metal volume fraction, i.e. rule of mixtures applies. ARALL structural bearing properties can be enhanced by employing a combination of adhesive bonding and mechanical fastening.



Elevated Temperature Tensile Strength
ARALL®-1 and ARALL®-4 Laminates
3/2 Layup, 0.053 inch (1.3 mm) Thick

... ARALL-4 with 2024-T8 and 350F cure prepreg increases use temperature by about 100F over that of ARALL variants 1-3, which employ a 250F cure prepreg system.

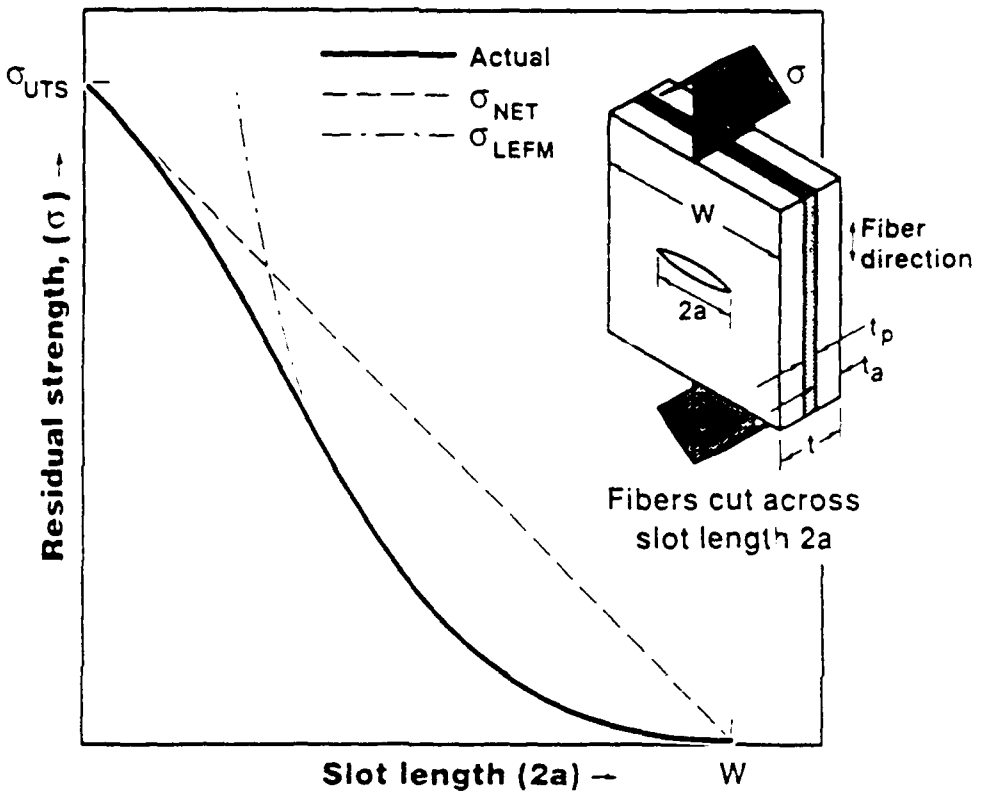


Residual Strength of Fatigue Damaged 2/1 ARALL®-1 Laminate Center Crack Panels

... Figure shows for fixed metal crack length, ARALL panel residual strength increases with percent fibers remaining in-tact over the cracked distance, 2a-2s.

ARALL® Laminate Center Slot Panel Residual Strength Approximation by Rule of Mixtures

ALCOA



Approximation of σ vs. 2a failure relationship from σ_{NET} and σ_{LEFM} failure criteria where:

$$\sigma_{NET} = \left\{ \frac{n_a t_a}{t} [\sigma_{ys}]_a + \frac{n_p t_p}{t} [\varepsilon_f E]_p \right\} \left(1 - \frac{2a}{w}\right)$$

and

$$\sigma_{LEFM} = \frac{K}{\sqrt{\pi a}\, f(a/w)}; \quad K = \sqrt{\left(\frac{n_a t_a}{t} E_a + \frac{n_p t_p}{t} E_p\right)\left(\frac{n_p t_a}{t} \frac{K_{ci}^2}{E_a} + \frac{n_p t_p}{t} G_p\right)}$$

n_a, n_p = number of aluminum and prepreg plys, respectively

t_a, t_p, t = aluminum, prepreg and total thickness, respectively

E_a, E_p = aluminum and prepreg modulus, respectively

σ_{ys} = yield strength, aluminum

ε_f = failure strain, prepreg

K_{ci} = initiation fracture toughness, aluminum

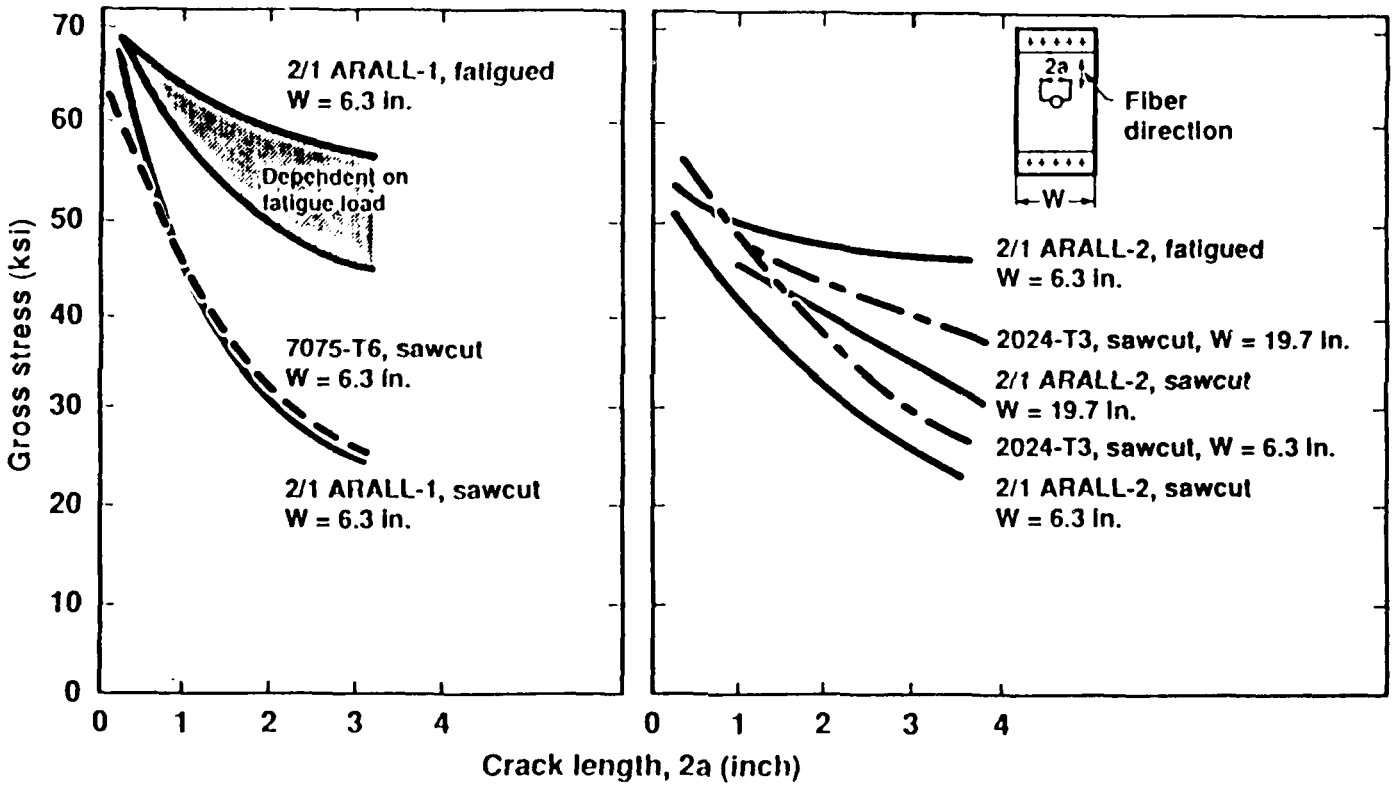
G_p = strain energy release rate, prepreg

...Figure illustrates ARALL center crack panel (fibers cut) residual strength can be estimated as lower bound of net section strength and LEFM failure criteria. Since prepreg properties can be approximated from ARALL properties minus aluminum properties, no specialized prepreg property tests would be required.

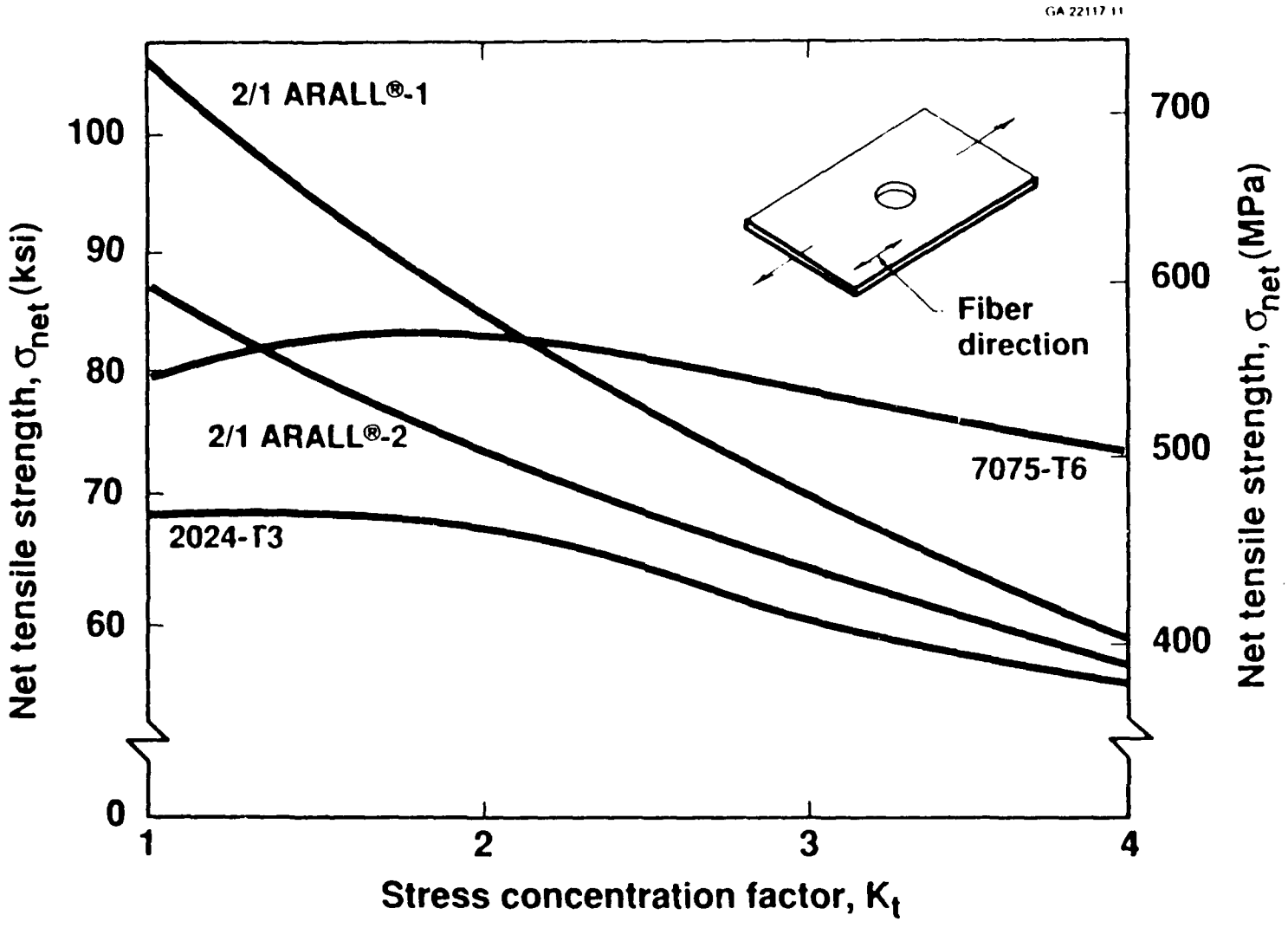
ARALL® Laminate and Aluminum CCT Panel

Residual Strength Variation with Slot Size, Slot Type (Fatigued vs. Sawcut), and Panel Width

(Courtesy Delft University of Technology)



... Figure shows cracked panel residual strength response of ARALL-1 (7075-T6 based) and ARALL-2 (2024-T3 based) variants. For long cracks (fibers cut) the ARALL-1 response is controlled moreso by the LEFM criterion because of the 7075 high strength and low toughness relative to that of 2024. In contrast ARALL-2 failure is more dominated by the net section strength criterion owing to the low strength and high toughness of 2024 relative to that of 7075.



Open Hole Specimen Notch Strength

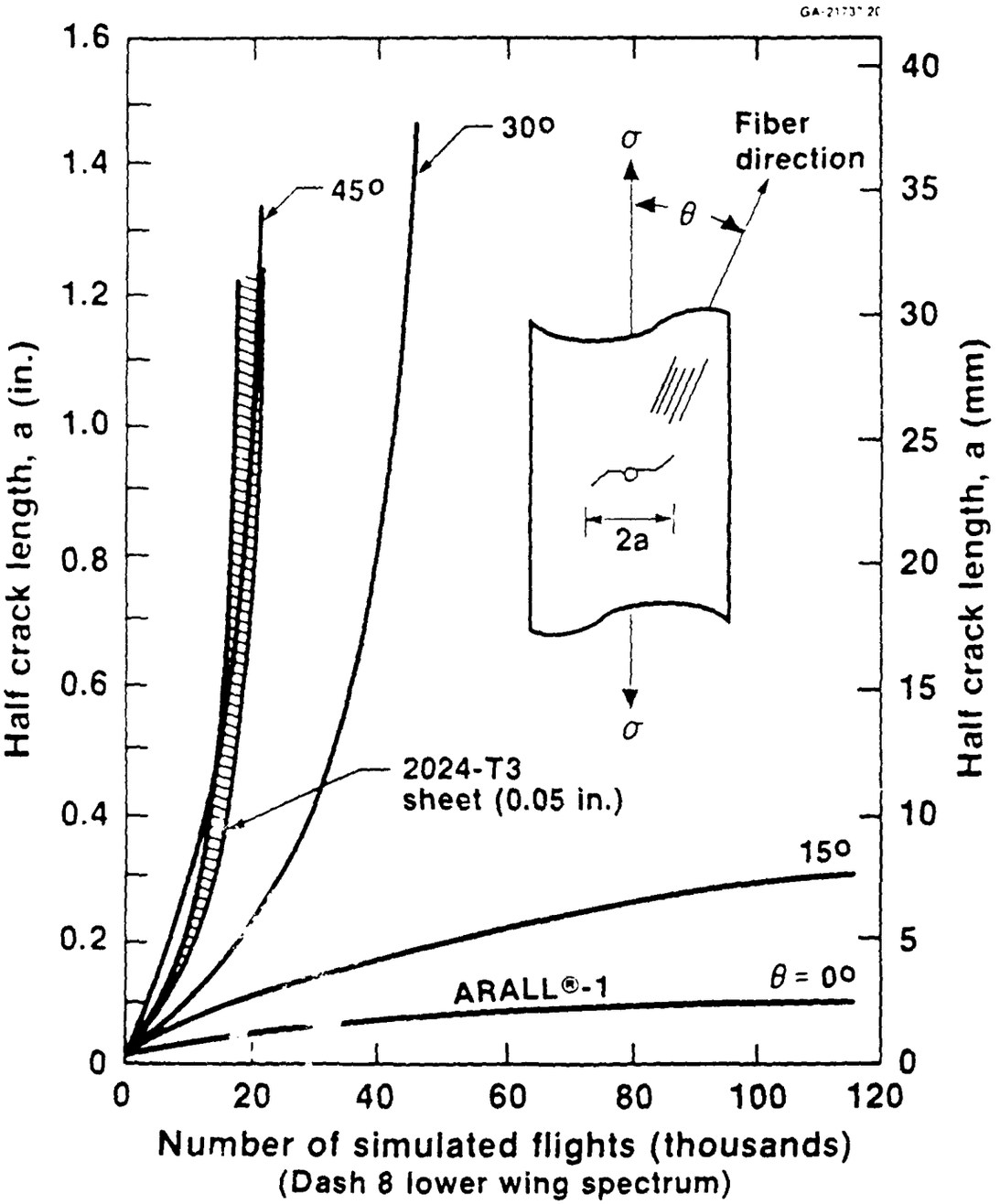
ARALL® Laminate vs. Aluminum Different K_t Factors

(Courtesy Delft University of Technology)

... Figure shows ARALL as more notch sensitive than monolithic aluminum, and ARALL-1 as more notch sensitive than ARALL-2. The latter observation is the result of the fracture toughness differential between 7075-T6 (ARALL-1) and 2024-T3 (ARALL-2).

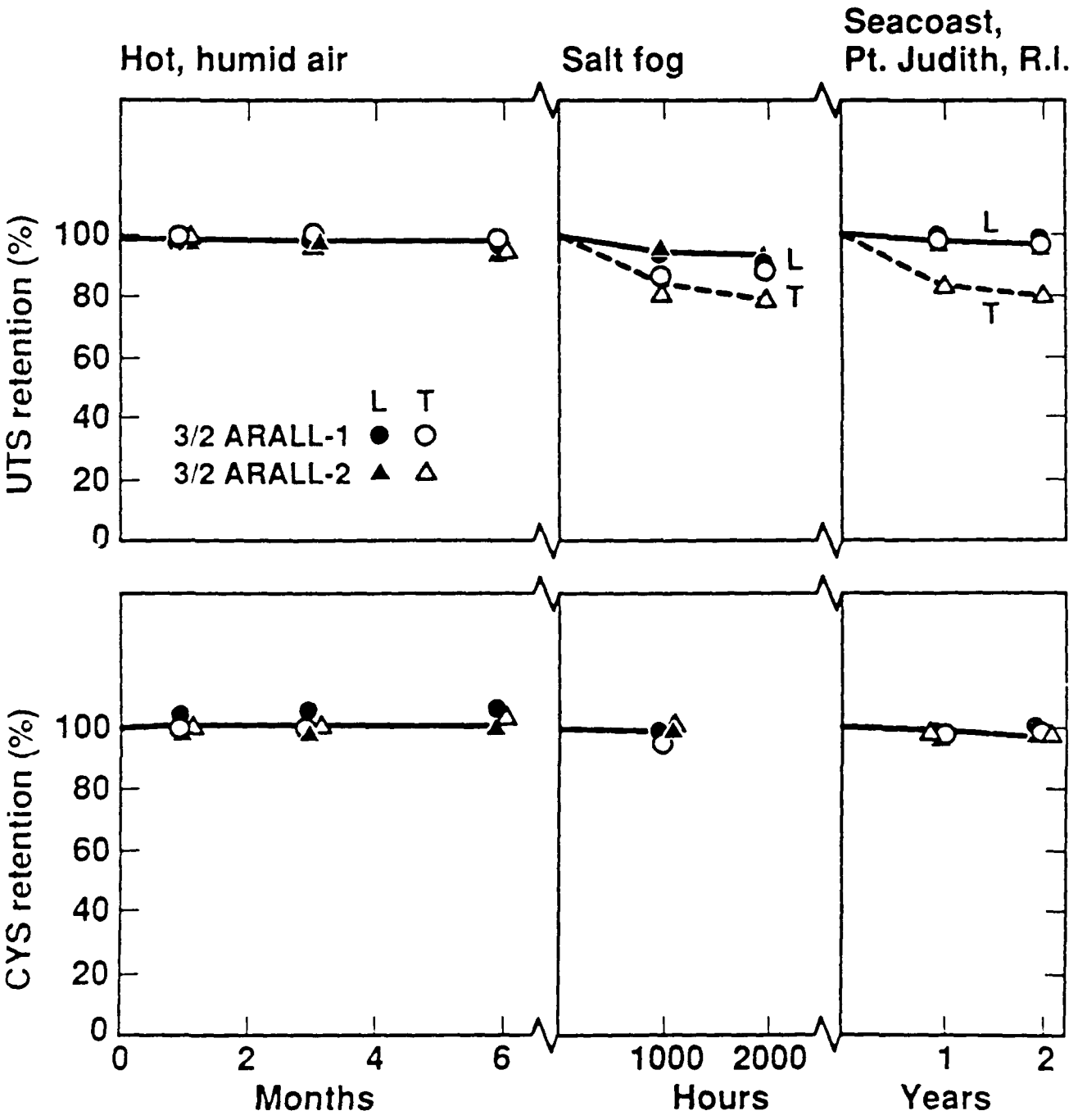
Effect of Fiber Orientation on 3/2 ARALL®-1 Laminate (0.053 Inch) Fatigue Crack Growth

(Courtesy deHavilland Aircraft)



ALCOA

ARALL® Laminate Tensile Ultimate and Compressive Yield Strength Retention After Presoak in Various Environments



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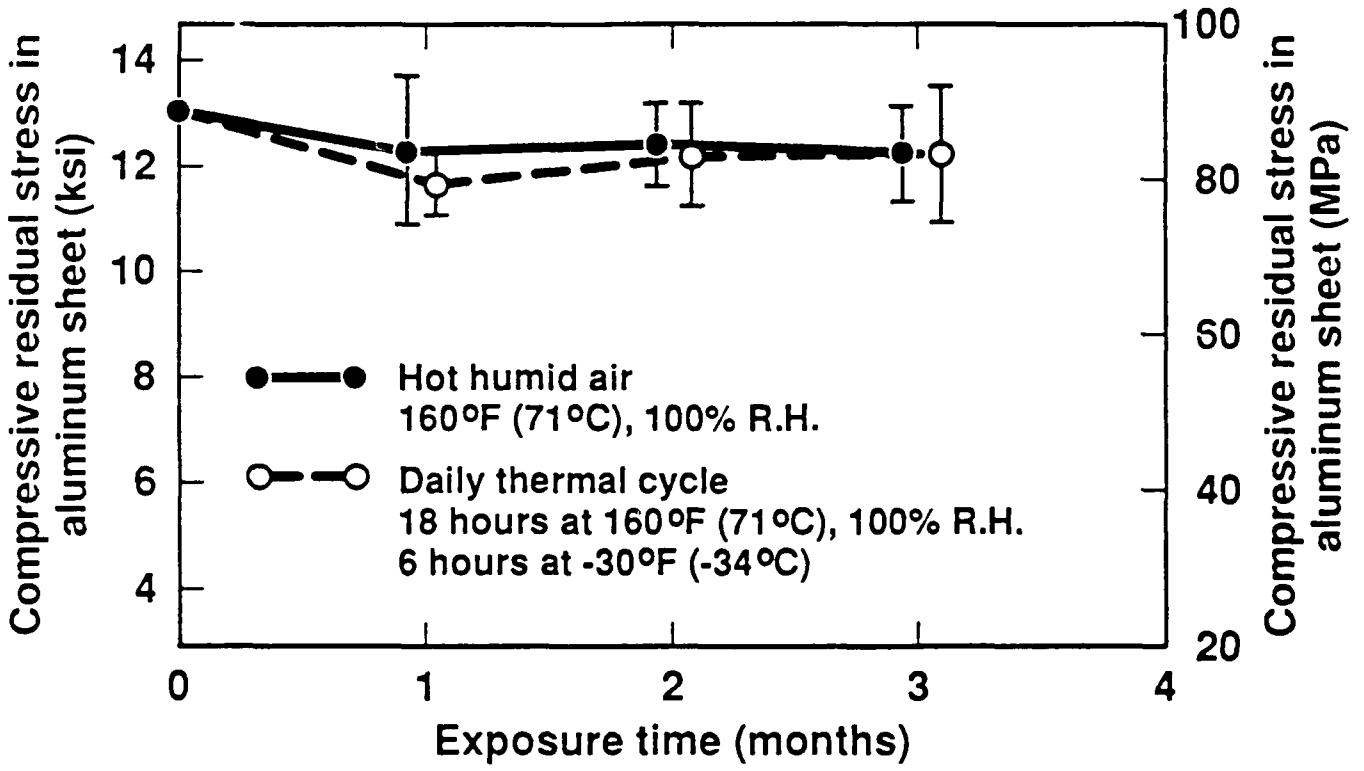
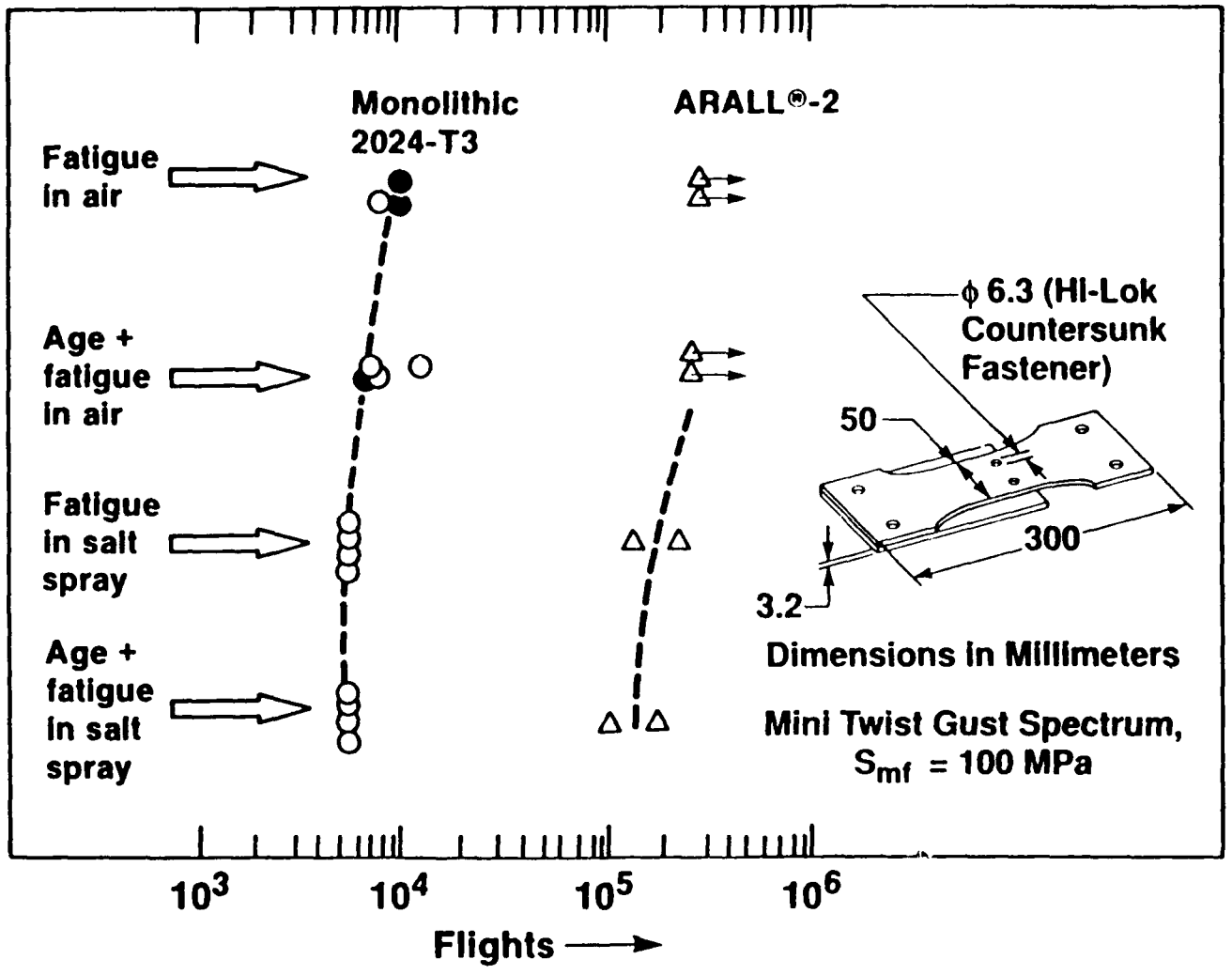
ALCOA

3/2 ARALL® -1 Laminate

Compressive Residual Strength Retention in Aluminum Layers After Presoak in Various Environments



... Figure shows post-stretch compressive residual stress in aluminum layers is retained after accelerated exposure in severe environment.

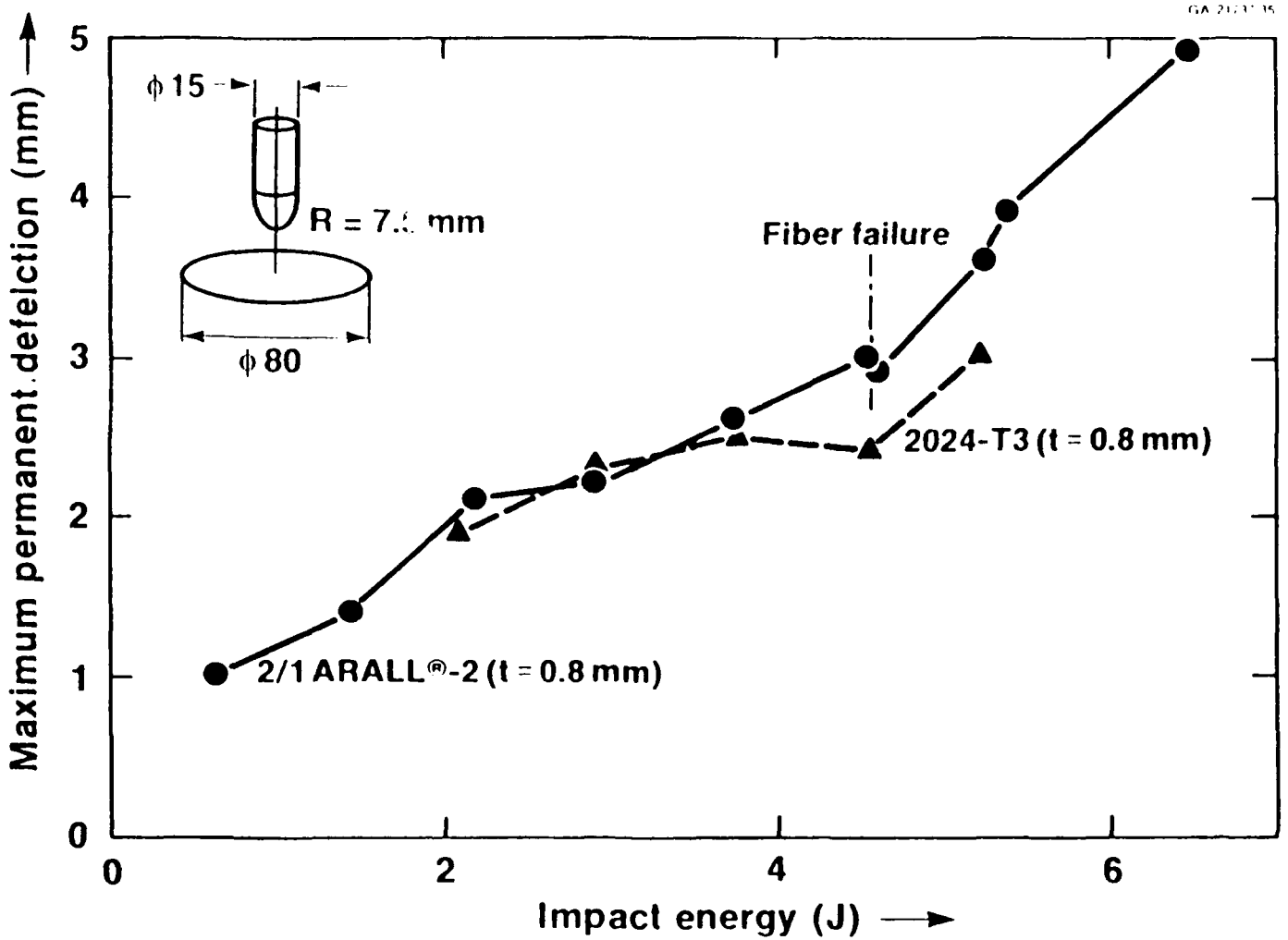


ARALL®-2 Laminate and 2024-T3

Bolted Joint Specimen Fatigue Lifetimes Under Various Exposure Conditions

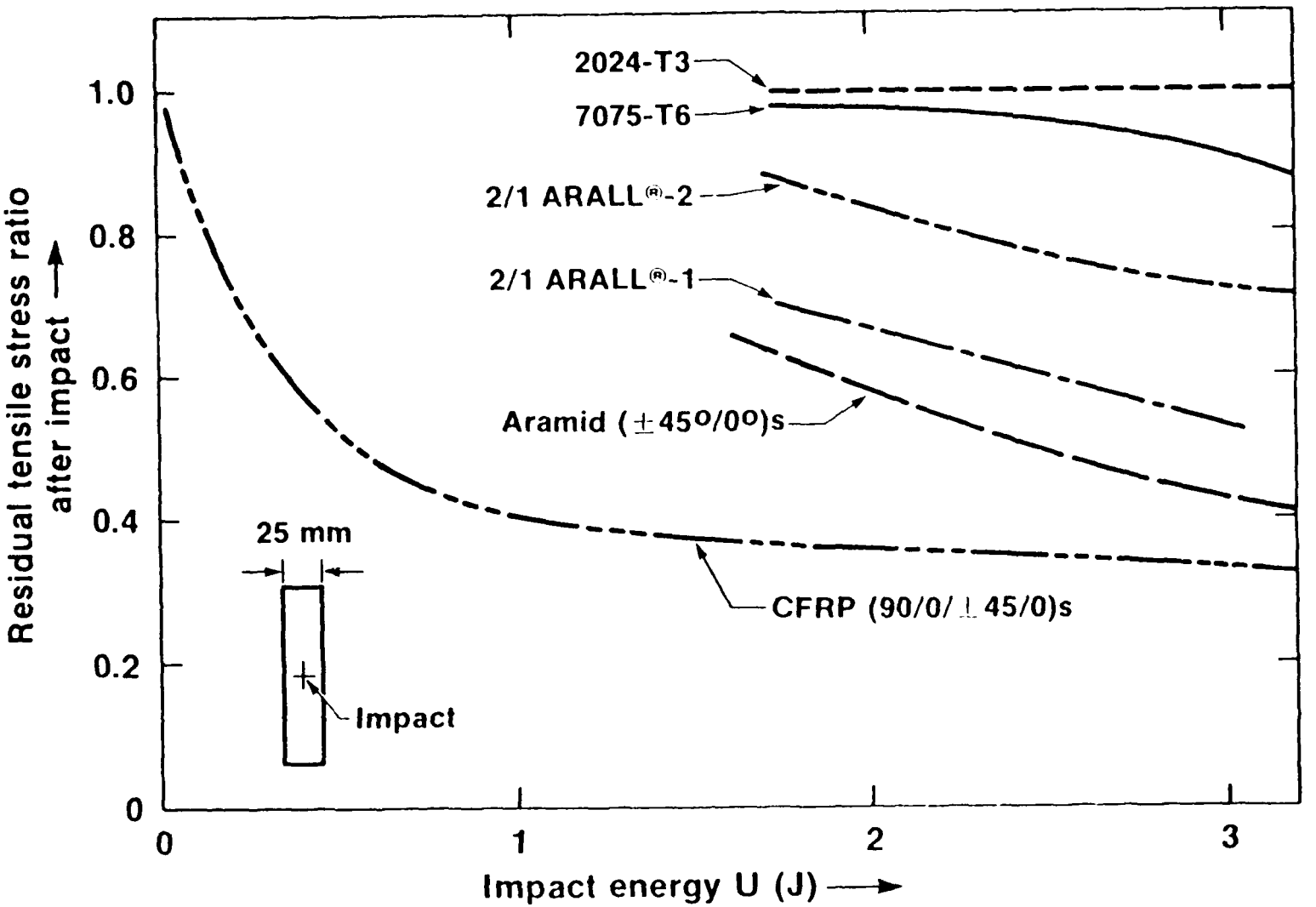
(Courtesy Delft University of Technology)

ARALL®-2 Laminate vs. 2024-T3
Permanent Deflection after Impact
(Courtesy Delft University of Technology)

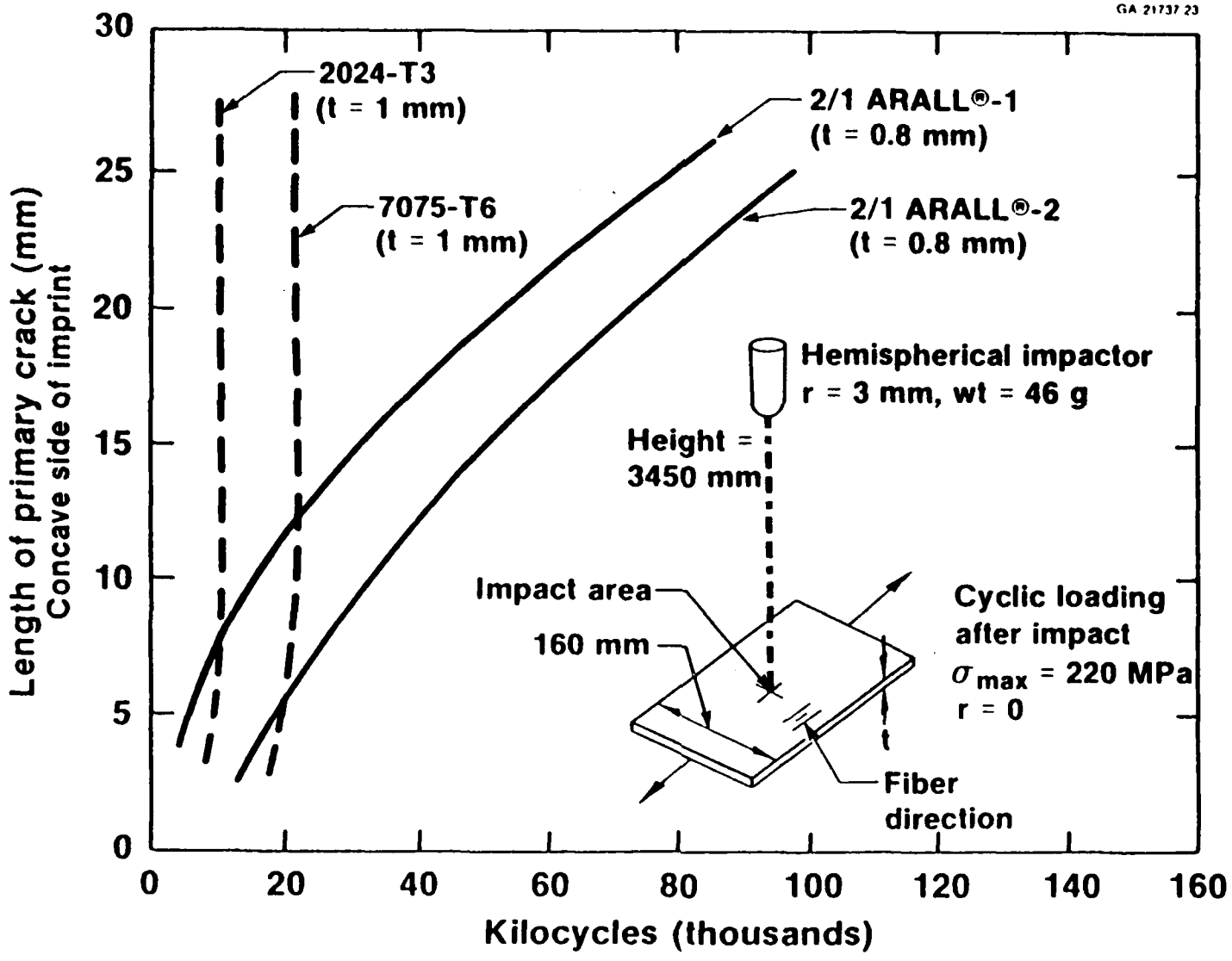


... The figure shows that impact damage inspectability of ARALL is equivalent to that of metal sheet.

Residual Tensile Strength of Different Materials after Impact
(Courtesy Delft University of Technology)

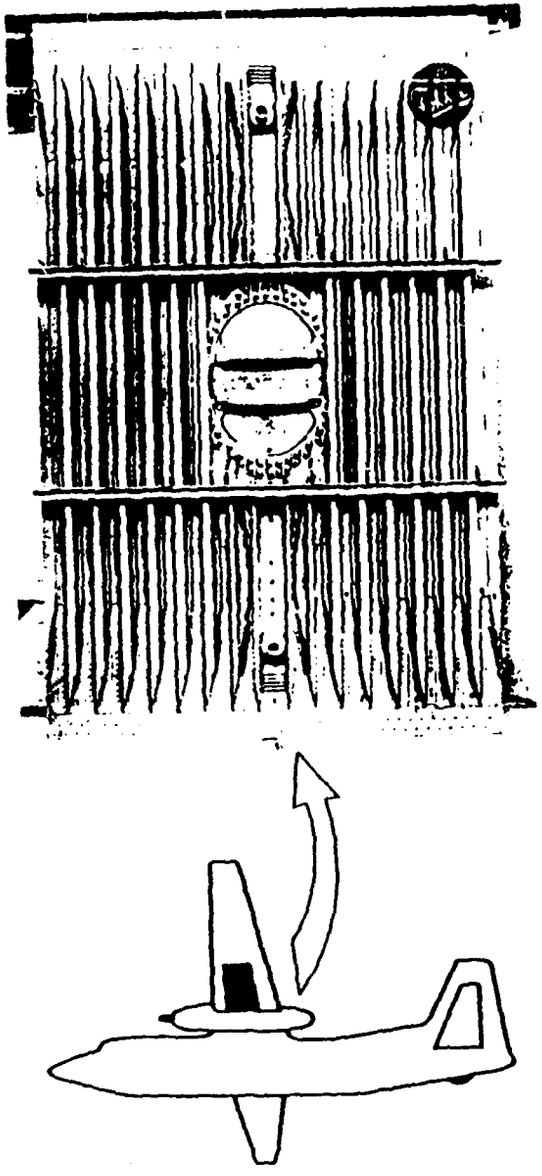


ARALL® Laminates
Fatigue Crack Growth after Impact
(Courtesy G. Roebroeks, Delft University of Technology)



ARALL®-2 Laminate Fokker Aircraft Designed F-27 Lower Wing Panel

- 33% weight saved
- Surpassed 3X operational life requirement
- Damage confined to small areas
  - — Inspectable
  - — Slow crack growth or arrest
  - — No through cracks
  - — Repair not necessary
- Static strength maintained (after 270,000 simulated flights)



Number of Simulated Flights to Initiate 1-3 mm Detectable Crack

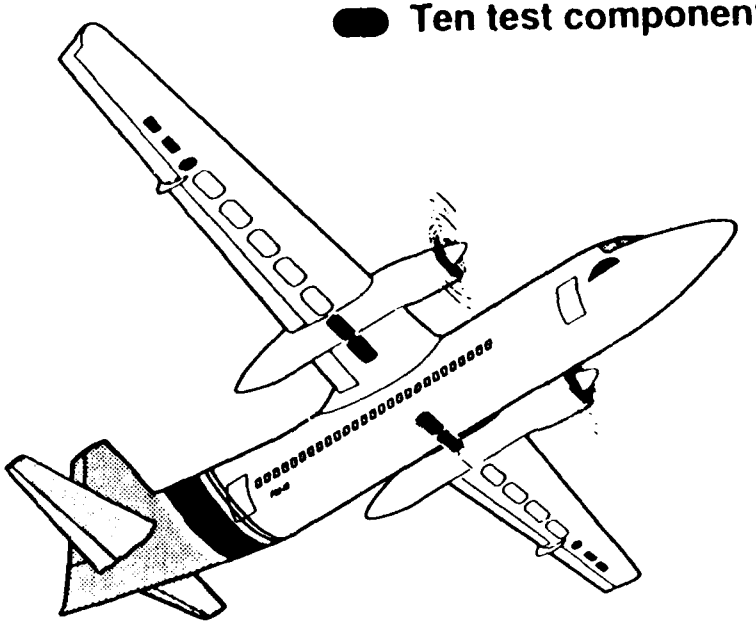
Damage type	All metal F27 panel	ARALL®-2 panel
Crack initiations in rebate (long edges)	10,000 (holes not cold worked)	70,000
	50,000 (holes cold worked)	
Crack initiation in rebate where load transferred to cover	40,000	20,000 (design snag)
Bolt holes at drain hole	30,000	80,000
Rivets and stiffeners at rib joints	45,000	270,000
End of bonded doublers	30,000	270,000
	Various repairs necessary	No repairs

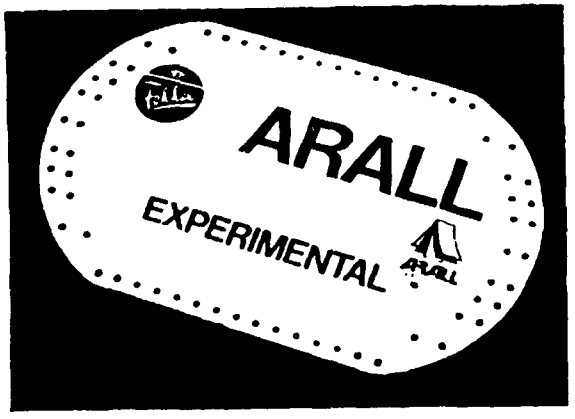
Ref: van Veggel et. al., 14th ICAF Symposium

The First Flying ARALL® Parts
F50 Aircraft Underwing Inspection Covers - October, 1987
Courtesy Fokker Aircraft

ALCOA

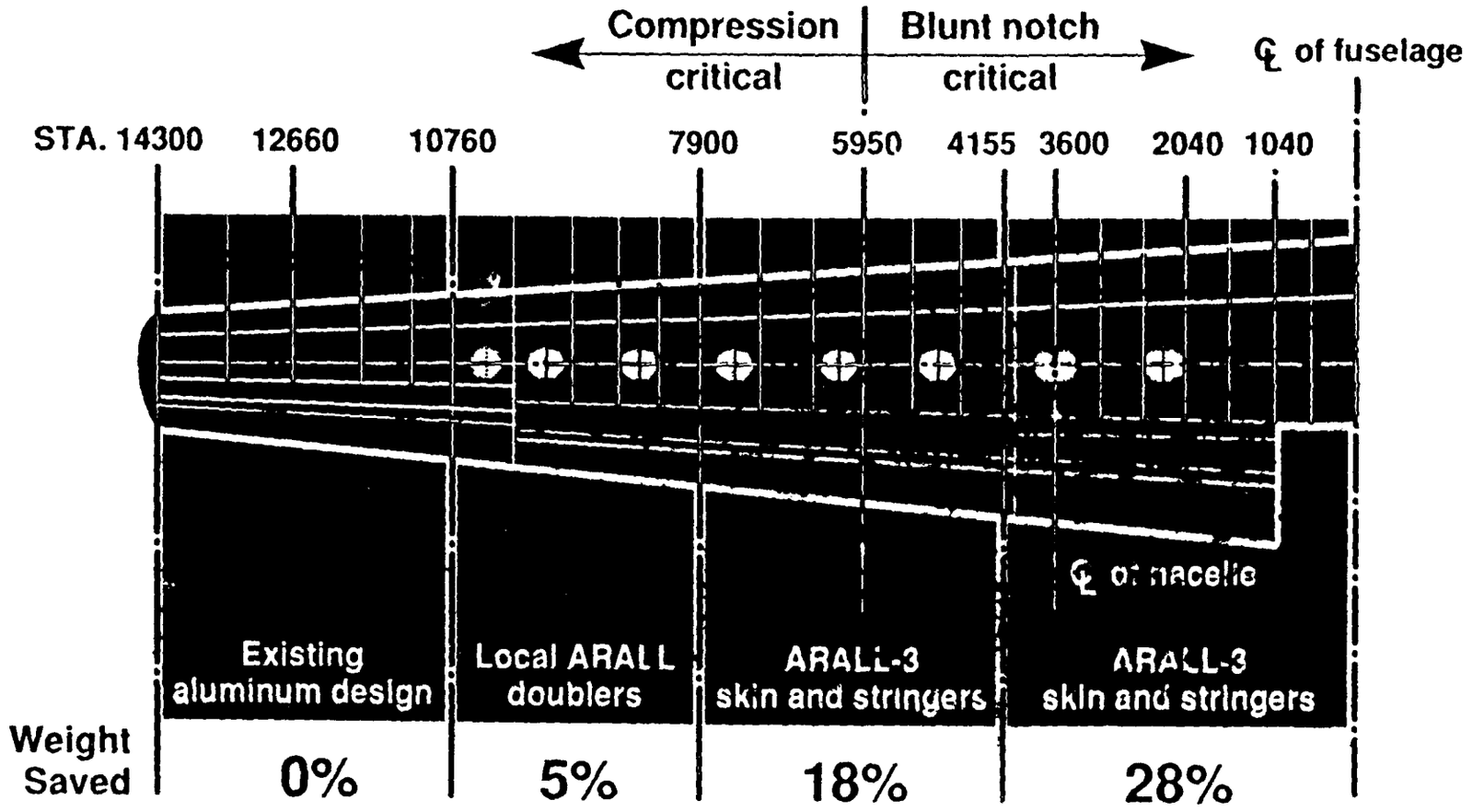
- Ten test components





F50 Wingbox ARALL® Laminate Design Study
Courtesy Fokker Aircraft

ALCOA



Overall weight saved (tip to tip): **20%**

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ALCOA

ARALL® Laminate - Leading Application Programs

- Fokker:
  - F50 hatch covers
  - F50 lower wing
  - F100 emergency door retrofit
  - F100 fuselage
- Douglas:
  - C-17 applications - cargo door
  - MD-11 and MD-90 applications
- Boeing:
  - Tear straps
- Aerospatiale:
  - A330/340 fuselage
- DeHavilland:
  - Dash-8 applications
- General Dynamics:
  - F16 tail light panel
  - ATF Supportable Hybrid Stuctures Program
  - ATA aplications
- USAF AFWAL / Tinker AFB:
  - E3 vertical tail rudder

ALCOA

GA 21737 47

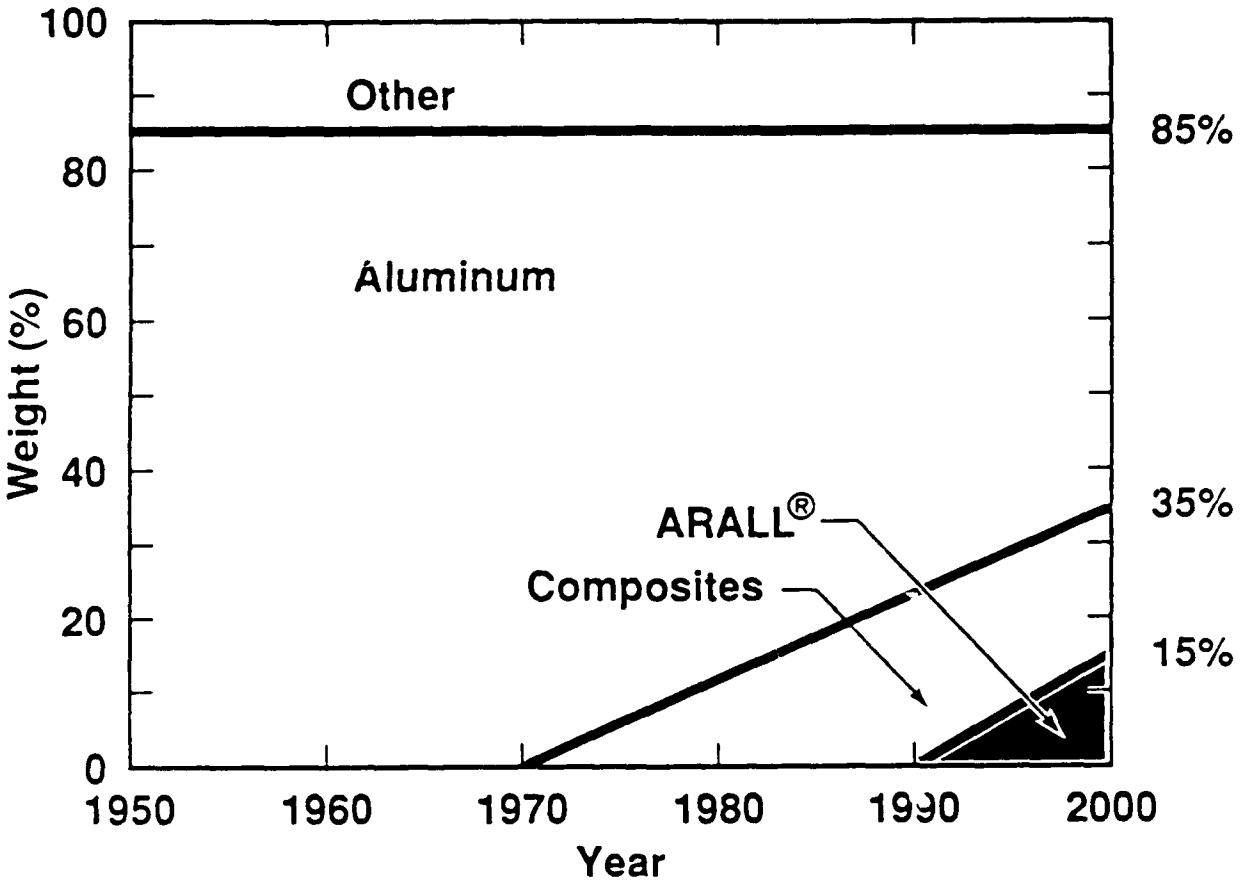
High Performance Material Design Issues

Criteria	Aluminum	Composites	ARALL® Laminates
Strength and rigidity	• Tensile/Comp properties • Buckling	• Notch factors • Environmental knockdowns	• Notch factors • Compression • Buckling
Durability	• Crack initiation by fatigue and/or environment	• Compressive fatigue • Delaminations • Environmental factors	• Environmental at cracks and holes • Thermal cycling
Damage tolerance	• Crack growth • Penetrations • Residual strength	• Compression after impact or other damage modes	• Residual strength after impact, penetration or other damage modes
Supportability	• Inspectable • Repairable • Cracks grow	• Low inspectability • Repair difficult • Requires larger safety margins	• Inspectable • Repairable • Cracks grow very slowly

GA-24241 15

ALCOA

Total Airframe Materials Breakdown



ARALL LAMINATE APPLICATIONS ON THE DOUGLAS C-17 TRANSPORT AIRCRAFT

BY
W. LEODOLTER (MDC)
R. G. PETTIT (MDC)
M. A. GREGORY (ALCOA)

PRESENTED TO:
1988 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE
SAN ANTONIO, TEXAS
NOVEMBER 29 - DECEMBER 1, 1988

DOUGLAS AIRCRAFT COMPANY
MCDONNELL DOUGLAS

CB1646.01

INTRODUCTION

This paper addresses potential use of ARALL laminates on the Air Force's C-17 aircraft, airworthiness qualification issues resulting from its use and its cost versus the benefits to be gained.

The C-17, built by the Douglas Aircraft Company of the McDonnell Douglas Corporation, is the world's first direct-delivery aircraft that satisfies the mobility requirements of the United States Armed Forces. Many of the peacetime missions specified for the C-17 operate in a severe environment. Significant for the integrity of the structure is the relatively high utilization of the aircraft in high-speed cruise at altitudes of 2000 feet or less. Resulting load spectra from wind gusts and maneuvers in this environment have considerable effect on the sizing of the structure. Design stress levels are set to provide durability and damage tolerance levels of the structure resulting in low inspection and maintenance burdens.

The design goal is to meet the in-service noninspectable damage tolerance requirement of MIL-A-83444. The structure, therefore, through redundancy in load paths, will provide substantial tolerance to relatively large damage in the fuselage shell, wing, and empennage.

The Douglas Aircraft Company has recently been in the process of introducing ARALL® laminates into production of the C-17 aircraft. Recognizing the current state-of-the-art of the technology as being a unidirectionally fiber reinforced material system, applicability is limited to structures for which the loads closely match the orthotropic material capability. We have identified such applications; they include various flat panels of the fuselage pressure boundary, the skin of the large cargo door, and the skin panels of the vertical and horizontal stabilizers. ARALL application on the iden-

tified fuselage structure is targeted for the very near-term; stabilizer skins, because of the required additional development work, are projected for later production phase-in.

Airworthiness qualification of ARALL laminate structure follows basically the principles used for aluminum structure, but recognizes the inherent fail safety of ARALL with regard to fracture across the fibers. Environmental durability and composite-like material behavior are being adequately accounted for by coupon level testing.

The operational behavior of ARALL laminates, a characteristic of great importance to the aircraft user, is anticipated to be equal to or better than that of aluminum or composites. Repair of ARALL structure, as any other structure, must provide that the strength and durability characteristics of the virgin and repaired structure are virtually identical.

New technology implementation into production aircraft, after having established an acceptable technical risk position, is also a matter of cost affordability. Here, the scenarios vary depending on the contractual structure of an aircraft program, the maturity of the production program, and other less tangible factors. With focus on ARALL, the higher cost of the material must be overcome by the benefit of the lower weight to demonstrate net benefits over the lifetime of the aircraft.

Douglas, in cooperation with the Air Force, is committed to bringing ARALL onto the C-17 aircraft and is also pursuing applications on other military aircraft and commercial aircraft. Douglas views the ARALL technology, in its generic definition as a hybrid material system, as one of the emerging technologies for future aircraft. ARALL possesses attributes and characteristics that fit well into the perceived trend in aircraft structure of the year 2000 and beyond.

AIR FORCE/MCDONNELL DOUGLAS C-17 AIRLIFTER

The C-17, built by the Douglas Aircraft Company, is the world's first direct-delivery airlifter that satisfies the mobility requirements of the United States Armed Forces.

The aircraft can be refueled in the air and is able to operate out of small austere airfields with runways as short as 3,000 feet. Operational missions include airdrops, low level/high speed cruise, terrain following, and transport of large cargo.

The aircraft, which is smaller than the C-5, has an overall length of 175.2 feet and a wing span of 165 feet.

AIR FORCE/MCDONNELL DOUGLAS AIRLIFTER

AERIAL REFUELING CAPABILITY
COMBAT OFFLOAD
AIRDROP AND LAPES

SMALL AUSTERE AIRFIELD OPERATION
LARGE CARGO TRANSPORT

MAXIMUM TAKEOFF WEIGHT	580,000 LB
MAXIMUM PAYLOAD	172,200 LB
CRUISE SPEED	MACH 0.77
AIRDROP	110,000 LB

LOW LEVEL, HIGH SPEED	CRUISE
LENGTH	175.2 FT
WING SPAN	165 FT

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OVERVIEW

Since 1985, Douglas Aircraft Company's research efforts on ARALL laminates have been directed toward application readiness on new military and commercial aircraft. This effort has recently been focused on the C-17 military airlifter where reduced weight is a paramount parameter in meeting aircraft performance requirements.

A special Douglas and ALCOA task force has been set up and has been collocated in the C-17 program area to most effectively achieve the goal of identifying applications (e.g., solving technical issues, and preparing implementation plans).

Airworthiness certification of ARALL structure, in particular the supporting testing, and the C-17 damage tolerance criteria will be discussed.

The discussion on cost addresses the individual elements contributing to the total cost from development throughout the 20-year life of the aircraft. Differences between military and commercial aircraft applications will be addressed.

The discussion concludes with a status summary with regard to ARALL on the C-17 and a projection of the ARALL's role in the near and distant future.

OVERVIEW

- APPLICATION STUDIES
- PRODUCTION IMPLEMENTATION OPPORTUNITIES
- AIRWORTHINESS CERTIFICATION ISSUES
- COST/BENEFITS
- LONG-TERM PROJECTIONS

L5-08918
FG2048.87

BASIC CRITERIA FOR SELECTING ARALL LAMINATE APPLICATIONS

The first criterion is one of basic engineering optimization whereby the unique properties of the material are most efficiently used for the aircraft structure; dictated by the nature of the laminate, technical and economic benefits may be derived if the structure is unidirectionally loaded, and fatigue and damage tolerance critical. The laminate-controlled limits of temperature capability had no influence in the selection of the C-17 ARALL candidates.

Over the past years, considerable effort has been devoted to determining forming limits and understanding the residual stress system in the laminate after forming (Reference 1). It was our assessment to restrict forming to moderate curvatures with small bidirectionality because of strain limitation imposed by the fibers.

The acceptable risk of applying ARALL laminates in aircraft structures is interpreted in the context of the interrelationship of the probability of technical success and the consequences of failure on cost and program schedule. Sheet applications of ARALL do not constitute any adverse risk.

Thick laminates for winglike structures are currently perceived as high risk technology applications. Those need considerable development work in a number of technology areas.

Cost and schedule impact is usually unique to a particular situation; for the C-17 it's heavily tied in with the contractual segregation of full-scale engineering development and production of operational aircraft. In any case, the economic viability of ARALL versus other candidates must be demonstrated.

BASIC CRITERIA FOR SELECTING ARALL LAMINATE APPLICATIONS

UNIDIRECTIONALLY-LOADED, FATIGUE AND DAMAGE TOLERANCE CRITICAL STRUCTURE

SERVICE TEMPERATURE DOES NOT EXCEED THE LAMINATE CAPABILITY

SKIN APPLICATION REQUIRING NO OR ONLY MODERATE FORMING

ACCEPTABLE TECHNICAL AND ECONOMIC RISK

COST/SCHEDULE IMPACT MUST BE TOLERABLE

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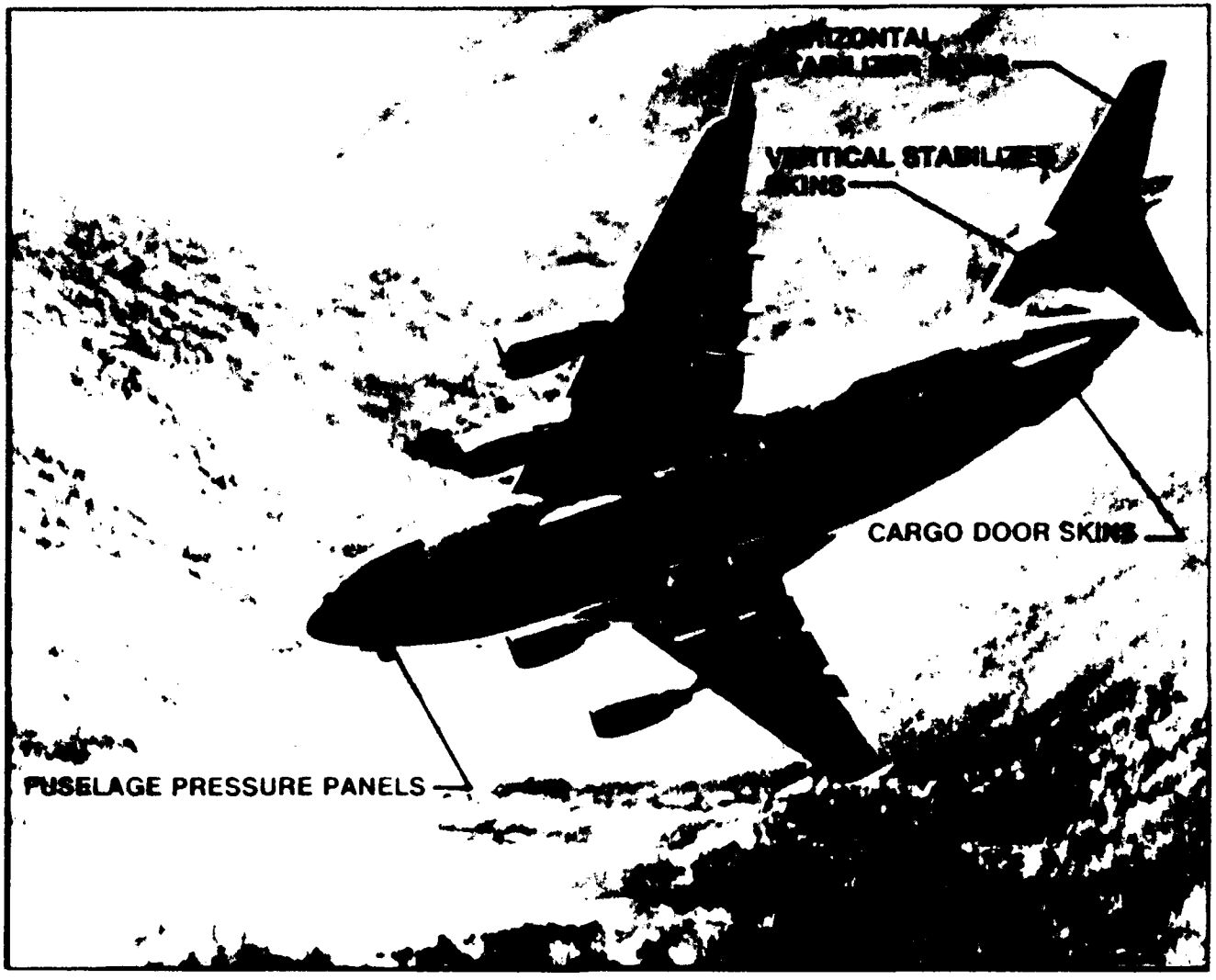
POTENTIAL C-17 ARALL LAMINATE APPLICATIONS

Giving consideration to the current state of the art of ARALL technology and level of industrial capability, namely being a system of unidirectional fiber laminate, application is limited to structures for which the load/stress field closely coincides with the orthotropic capability of the material. C-17 applications that meet that criterion are flat fuselage pressure panels, the cargo door skin, and the horizontal and vertical stabilizer skin panels.

Wing surfaces categorically meet this criterion but are unrealistic candidates because of their size. The majority of the C-17 fuselage skins, because of geometrical irregularities, cutouts, and concentrated load inputs, experience highly complex stress fields for which aluminum is a better solution than ARALL.

POTENTIAL C-17 ARALL LAMINATE APPLICATIONS



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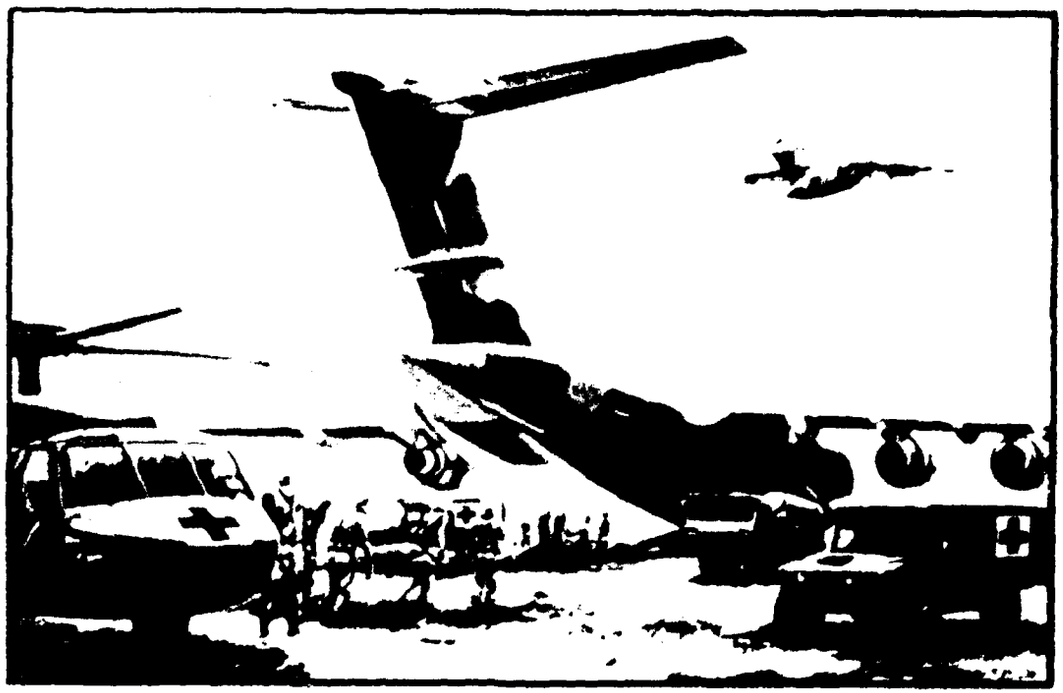
ARALL LAMINATE CARGO DOOR

Loads induced upon the door are primarily due to pressure, and act mostly circumferentially. The front of the door rides on a seal, transmitting little fore and aft loads. Static margins in the baseline aluminum door are adequate to permit downsizing of the ARALL material to derive weight savings greater than that achievable purely due to density.

Preserving minimal cost/schedule impact, the design was tailored to be compatible with the baseline tooling as much as possible. The resulting weight savings is about 130 pounds. ALCOA has formed geometrically comparable parts, verifying the capability to produce the required curvature. Bonded construction of skin subassemblies was considered for cost savings.

The cargo door is the foremost candidate for near-term ARALL implementation on the C-17. The fly-away weight of ARALL laminates is approximately 416 pounds.

ARALL LAMINATE CARGO DOOR



- SIZE: 31.8 FEET LONG; 18.5 FEET TRANSVERSELY BETWEEN HOOKS
- ARALL FIBERS CIRCUMFERENTIALLY COINCIDING WITH PRIMARY LOADS
- FATIGUE AND DAMAGE TOLERANCE CRITICAL; STATIC STRENGTH MARGINS ADEQUATE FOR DOWNSIZING
- ARALL DESIGN TAILORED TO COMPLY WITH BASELINE TOOLING
- TECHNOLOGY IN HAND
- 130 POUNDS WEIGHT SAVING USING ARALL 3

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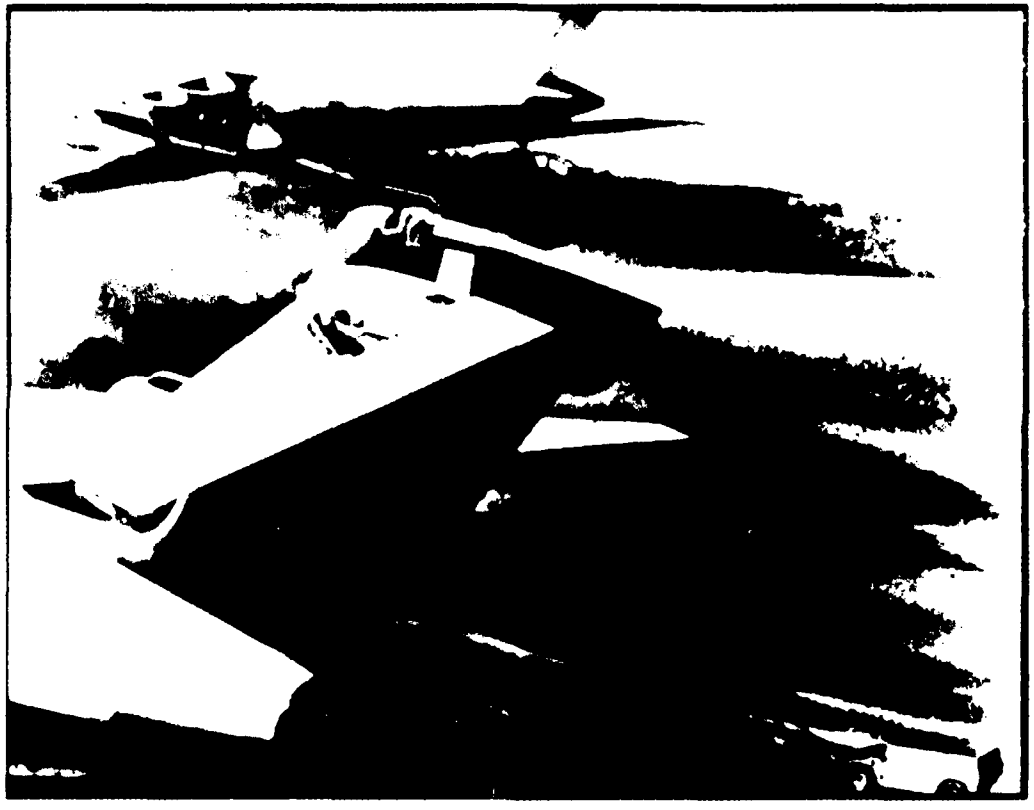
C-17 HORIZONTAL AND VERTICAL STABILIZER

The span of the horizontal stabilizer is approximately 64 feet; the area of the main structural box area considered for ARALL laminate is 333 square feet. The vertical stabilizer is 30 feet long with an area of 255 square feet.

Both surfaces develop a primarily axially oriented stress field due to bending for which ARALL laminates offer a nearly optimum solution. The surfaces are fatigue critical and are sized for compliance with the damage tolerance requirements.

Material build up toward the root section to resist high bending moments and torque present engineering and manufacturing challenges that need to be resolved before production commitment of ARALL for these applications.

C-17 HORIZONTAL AND VERTICAL STABILIZER



- PRIMARY AXIAL LOADS INDUCED BY BENDING
- FATIGUE AND DAMAGE TOLERANCE CRITICAL
- UNRESOLVED ENGINEERING AND MANUFACTURING TECHNOLOGY ISSUES

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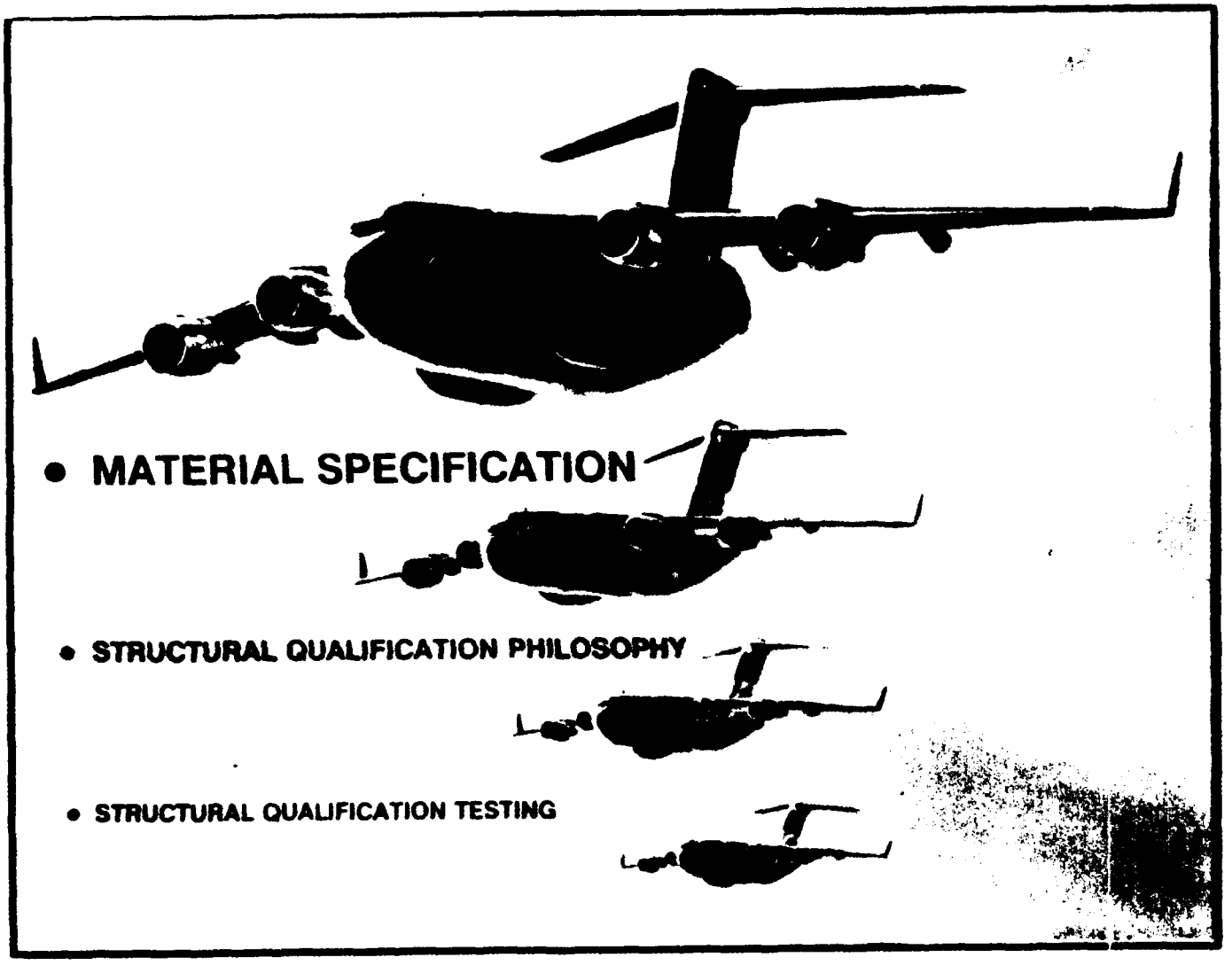
AIRWORTHINESS QUALIFICATION



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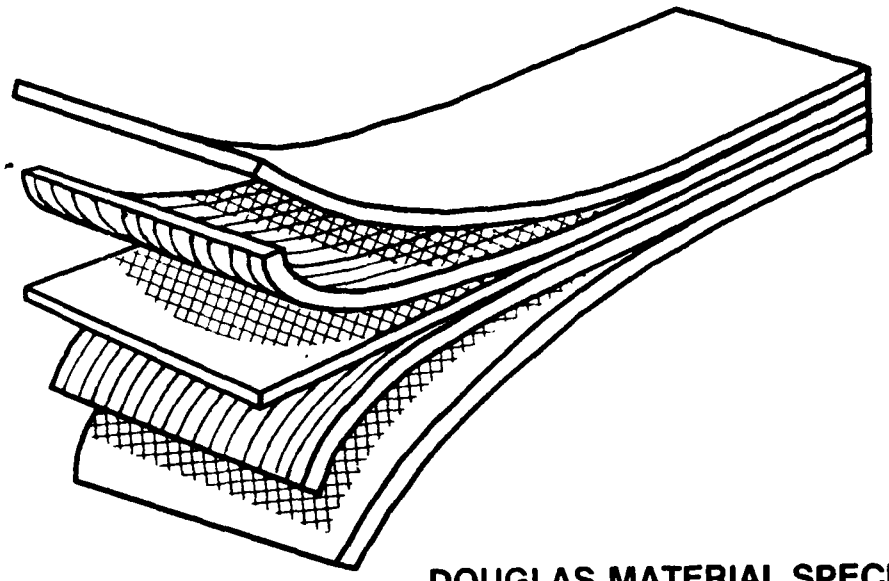
MATERIAL SPECIFICATION

The Douglas material specification of ARALL laminates follows closely the proposed AMS specification. Incorporation of ARALL, being a hybrid material, in MIL-Handbook-5 is unique. The approach to design allowables has paralleled that of aluminum and other monolithic metals with the exception of controlling an extra parameter, the peel strength.

In keeping with the findings of the Primary Adhesive Bonded Structure Program, performed at Douglas during the early 1970s, a primed phosphoric acid anodized bondline and external surfaces has been specified.

MATERIAL SPECIFICATION



DOUGLAS MATERIAL SPECIFICATION FOLLOWS AMS SPECIFICATION

MIL- HANDBOOK-5 ALLOWABLES PROVIDED BY ALCOA

MINIMUM PEEL STRENGTH SPECIFIED

PHOSPHORIC ACID ANODIZED AND BR127 PRIMED SURFACES

CC805.09

ARALL LAMINATE STRUCTURAL QUALIFICATION PHILOSOPHY

The structural qualification philosophy set forth in the C-17 specification includes a slow crack growth criterion requiring at least two lifetimes of safe crack growth from a presumed 0.050-inch manufacturing flaw located critically in the structure. Douglas has also agreed to apply the commercial two-bay fail safety criterion as well. Both criteria are being applied to ARALL laminates.

ARALL LAMINATE STRUCTURAL QUALIFICATION PHILOSOPHY

C-17 SPECIFICATION IS BASED ON SLOW CRACK GROWTH

DOUGLAS HAS OPTED TO SATISFY COMMERCIAL TWO-BAY FAIL-SAFE CRITERION

BOTH REQUIREMENTS WILL BE APPLIED TO ARALL LAMINATES

C-17 SLOW CRACK GROWTH CRITERION

The design goal is to meet the in-service, noninspectable damage tolerance requirement of MIL-A-83444.

In accordance with the slow crack growth criterion, the 0.050-inch manufacturing flaw may be conservatively assumed in all layers; the fibers presumed cut as well. Crack growth across the fibers may be calculated by analytical methods, such as that of Marissen (Reference 2), or by direct comparison to flat sheet crack growth data. In either case, no satisfactory way of accounting for the crack growth inhibition due to the presence of stiffeners is presently available, and it appears appropriate to conservatively neglect them altogether.

It is our present philosophy that crack growth parallel to the fibers may be analyzed in a manner similar to isotropic metals and that the orthotropy of the laminates may be neglected when using existing stiffened plate models.

C-17 SLOW CRACK GROWTH CRITERION

COMPLIANCE WITH MIL-A-83444

0.050-INCH FLAW ASSUMED IN ALL LAYERS, INCLUDING FIBERS

PRESENCE OF STIFFENERS CONSERVATIVELY NEGLECTED WITH REGARD TO CRACK GROWTH PERPENDICULAR TO THE FIBERS

CRACK GROWTH PARALLEL TO FIBERS TREATED SIMILAR TO MONOLITHIC MATERIALS

CC805.03

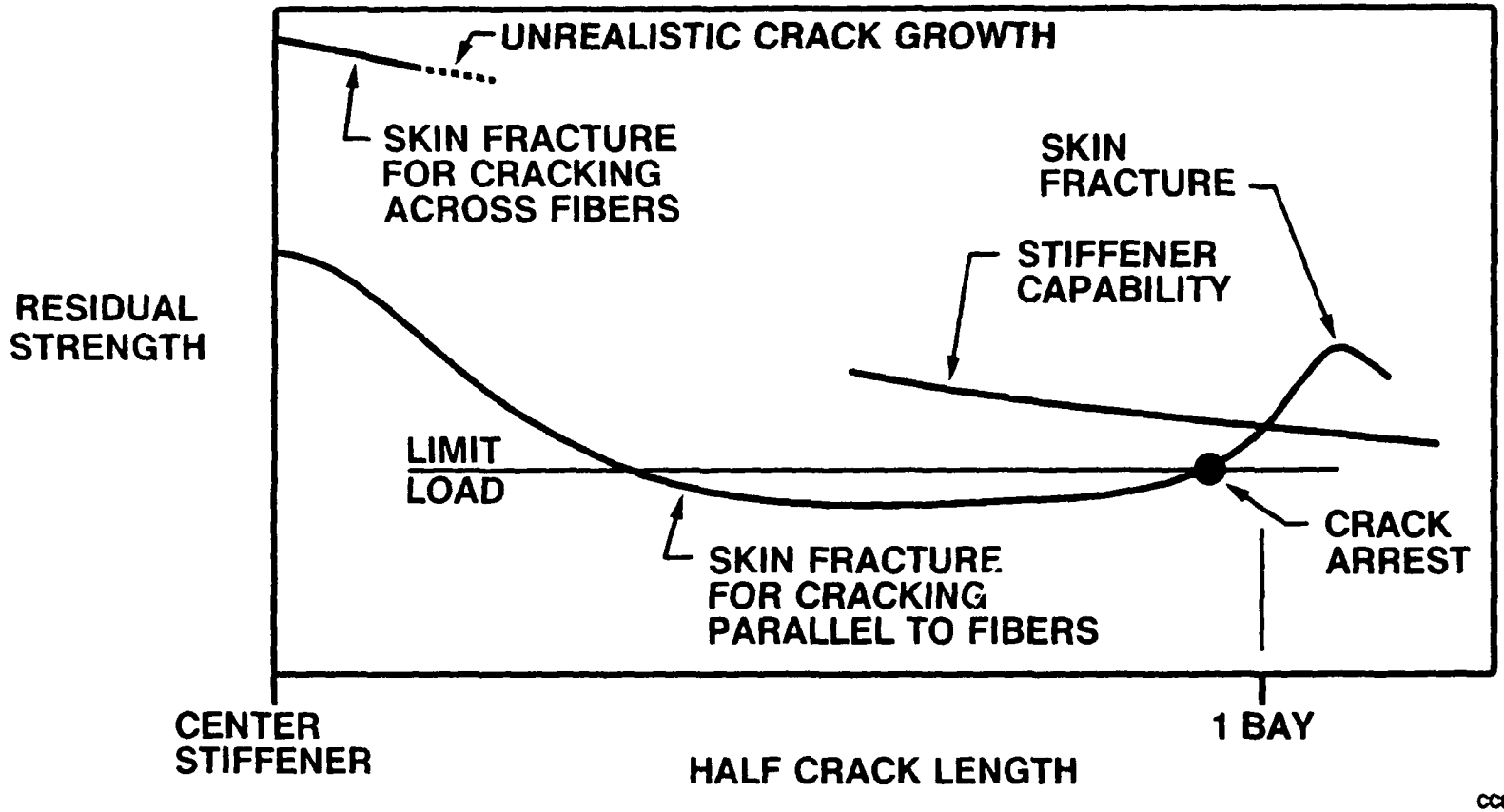
CRACK ARREST CRITERION FOR STIFFENED PANELS

The commercial two bay fail safety criterion stipulates that should a crack grow until instability at operational loads, the structure shall be capable of crack arrest at two bays at limit load (Reference 3).

This criterion can be readily applied to ARALL laminate structure for cracks parallel to the fibers. For fatigue cracks in the other direction, it is our position that the structure is inherently failsafe by virtue of the fibers, which bear load long after the metal is cracked through, thus providing evidence of damage long before the structure becomes critical. In fact, cracks longer than a few inches are quite unrealistic for ARALL with the present loads and life expectency (Reference 4); thus the cracks may be considered to be arrested well before they propagate to a length of two bays between stiffeners.

For ballistic damage, recent tests showed residual strength of ARALL would not be reduced to below limit load requirements. Such damage would presumably be repaired promptly, and would, if left unrepaired, grow more slowly in ARALL than in aluminum, and arrest at the nearest splice, which, at least for the cargo door, is at every other rib.

CRACK ARREST CRITERION FOR STIFFENED PANELS



ARALL LAMINATE CARGO DOOR
STRUCTURAL QUALIFICATION TESTING

Structural qualification testing of ARALL laminate structure is not expected to differ substantially from current practice. A test program developed for the cargo door includes the same types of tests associated with aluminum structure with the addition of some shear fatigue panels which were motivated not only by the directionality of the laminate, but also by the change from rivited to bonded structure.

In contrast, coupon and substructural testing done in the early research and development stages of ARALL laminate technology, both at Douglas and elsewhere, have been in many ways different than for monolithic aluminum. Environmental resilience of the composite laminate, resistance to peel stresses induced by flush fasteners under tensile fatigue loading, and notch sensitivity are among the issues addressed in coupon studies which differ from those of monolithic aluminum.

ARALL LAMINATE CARGO DOOR
STRUCTURAL QUALIFICATION TESTING

STATIC STRENGTH	DURABILITY
SHEAR PANELS	SHEAR PANELS
COMPRESSION PANELS	PRESSURE PANELS
HOOK JOINT	SPLICE SPECIMENS
SPLICE SPECIMENS	FULL-SCALE SECTION OF DOOR

STRUCTURAL QUALIFICATION TESTING SIMILAR TO METALLIC STRUCTURES

RELIANCE ON COUPON TEST DATA FOR ENVIRONMENTAL DURABILITY OF MATERIAL

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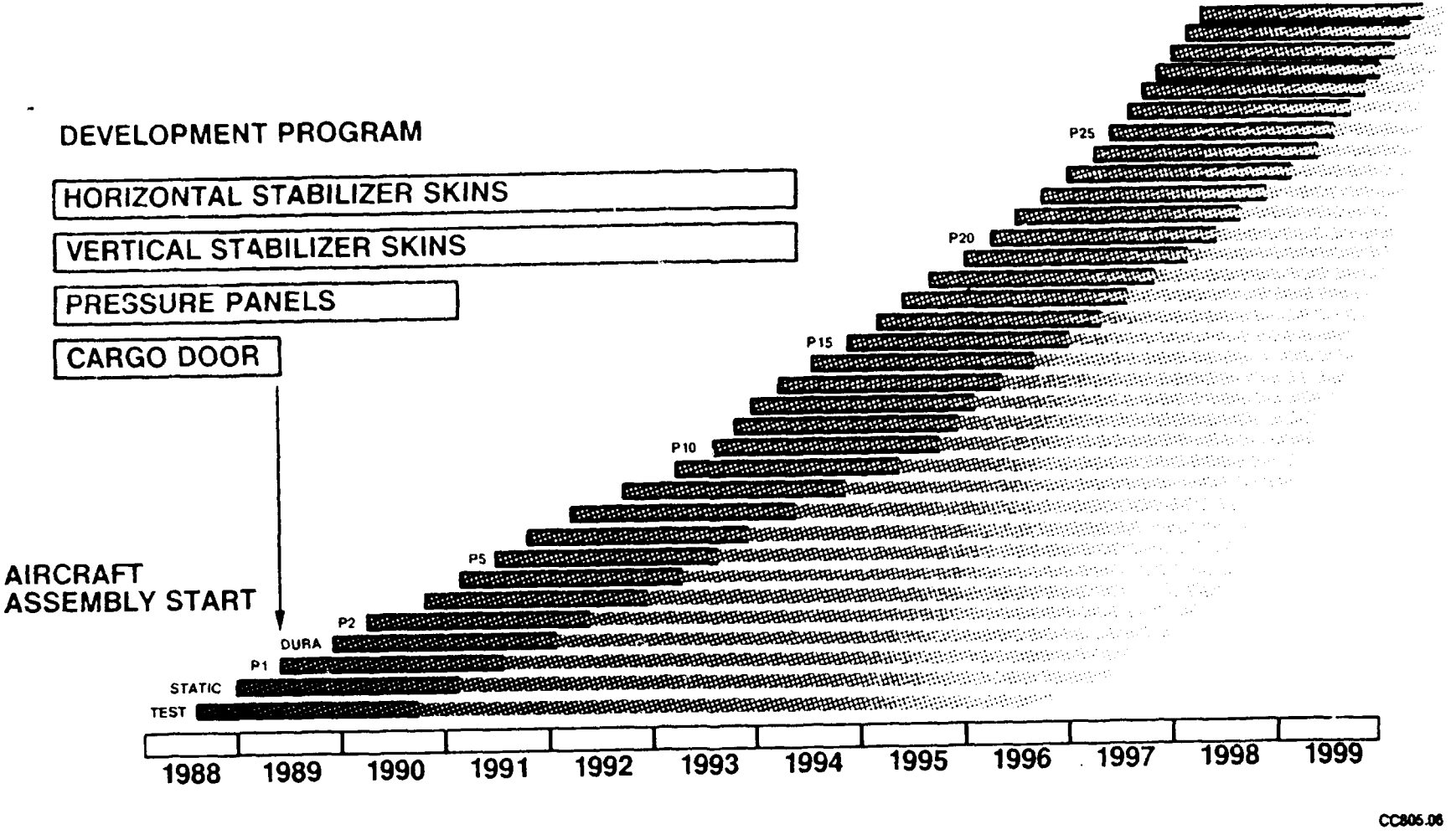
ARALL IMPLEMENTATION PLAN

The C-17 program consists of a full-scale engineering development (FSED) phase and a production aircraft phase. Three aircraft: a flight test aircraft, a static, and a durability test aircraft will be built under the FSED contract. FSED and the production of 210 operational aircraft occur concurrently but time-phased. Production rate build-up is progressive toward the maximum rate by 1992.

Since production has already begun, the introduction of ARALL becomes significantly more difficult than it would have been 1 to 2 years ago. Tooling already fabricated may be rendered obsolete; structural qualification cannot be done on the full-scale aircraft test without major cost implications. Thus, we are confronted with the basic technology phase-in problems; yet the situation is favorable because of the still very early state of production.

Sheet metal applications; i.e., the cargo door skin and the pressure panels are targeted for early aircraft; the horizontal and vertical stabilizer skin panels are years away and depend on the success of the development programs yet to be initiated.

ARALL IMPLEMENTATION PLAN



OPERATIONAL FACTORS

In the interest of low cost for the aircraft operater, maintenance cost must not exceed that of other structural candidates (in this case of aluminum). ARALL laminates, due to their slow crack growth behavior and high residual strength of a damaged structure are believed to require inspections between long intervals which in combination with the increased durability of the structure results in low maintenance burden. This is entirely dependent upon the degree of transverse loading. With regard to impact, the plasticity of the aluminum content renders a far more favorable tolerance to damage than composite structures.

For the particular aircraft under consideration, repairs must be accomplishable at outlying military installations. ARALL structures may be repaired with mechanical fasteners; temporary or permanent repairs with aluminum are acceptable.

In principle, the repaired structure must not alter the residual strength and strength characteristics of the structure locally and in broader dimensions. ARALL and aluminum repairs can be adequately tailored to meet this requirement.

ARALL structures, however, have yet to accumulate service experience to support a maintenance data base, particularly the effects of the orthotropic stiffness and strength on long-time service need yet to be examined.

OPERATIONAL FACTORS

REQUIREMENT	COMPLIANCE
LOW MAINTENANCE	SLOW CRACK GROWTH, HIGH RESIDUAL STRENGTH
EASE OF REPAIRS	MECHANICALLY FASTENED OR BONDED REPAIR PATCHES
RESTORED STRENGTH CAPABILITY	STRENGTH/DURABILITY EQUIVALENCE WITH ALUMINUM OR ARALL PATCHES

COST/BENEFIT ISSUES

Engineering development and tooling costs are regarded to be comparable for ARALL and aluminum structures, with perhaps small deviations due to additional splices required by material size limitations presently imposed on ARALL laminates. However, because the material was not baselined for the C-17, significant up-front costs are associated with the redesign and tooling impact required for ARALL implementation. Nevertheless, these costs appear to be lower for ARALL laminates than for most advanced composites.

The recurring cost of fabrication and assembly of ARALL laminate structures are again comparable to aluminum except as influenced by the above mentioned requirement for more splices. The material cost is an order of magnitude higher, not unlike other advanced materials, and must be considered.

Savings in operational costs associated with ARALL laminate structures are primarily a result of reduced fuel costs due to the achievable weight savings. However, like most military aircraft, the C-17 is designed for far less usage than commercial aircraft, thus the increased cost of the hardware is not fully offset by the fuel savings. Clearly, performance criteria are a significant motivating parameter for the use of ARALL laminates.

COST/BENEFITS ISSUES

LIFE-CYCLE-COST ELEMENTS	COST RELATIVE TO ALUMINUM STRUCTURE
FULL-SCALE ENGINEERING DEVELOPMENT • ENGINEERING DESIGN AND TESTING • TOOLING	GENERALLY COMPARABLE. MAYBE INCREASED COST DUE TO MATERIAL SIZE LIMITATIONS
WEAPON SYSTEMS COSTS FOR 210 AIRCRAFT • FABRICATION AND ASSEMBLY • MATERIAL	COMPARABLE. SIGNIFICANTLY HIGHER
OPERATIONS AND SUPPORT FOR FLEET OVER 20 YEARS • FUEL • MAINTENANCE	REDUCED DUE TO LOWER WEIGHT. PRESUMED COMPARABLE

HIGH MATERIAL COST IS NOT FULLY OFFSET BY FUEL COST SAVINGS FOR THE C-17 AND OTHER MILITARY AIRCRAFT

PERFORMANCE IS A SIGNIFICANT MOTIVATING PARAMETER FOR USE OF ARALL

CC805.08

SUMMARY

In summary, ARALL laminates have progressed to a state of readiness for near-term, cost effective implementation. While such implementation is made more difficult to justify late in a program such as the C-17, we have found it worthy of in-depth consideration due to an attractive potential for weight savings.

Structural qualification of ARALL laminate structures is by an approach similar to aluminum structures, but recognizes the inherent fail safety of the material with regard to fracture across the fibers. Environmental durability and composite-like material behavior can be adequately accounted for using coupon level testing.

Operational characteristics of ARALL laminates are anticipated to be equal or better than aluminum.

Douglas views the ARALL technology as one of the emerging technologies for future aircraft. It possesses attributes and characteristics that fit well into the perceived trend in aircraft structures. Opportunities beyond these first C-17 applications are certainly anticipated.

SUMMARY

- ARALL LAMINATES ARE READY FOR NEAR-TERM IMPLEMENTATION ON THE C-17
- CERTIFICATION OF ARALL STRUCTURES IS SIMILAR TO METALLIC, BUT RECOGNIZES THE INHERENT FAIL SAFETY OF MATERIAL
- OPERATIONAL CHARACTERISTICS OF ARALL LAMINATE STRUCTURES EXPECTED TO BE EQUAL TO OR BETTER THAN THAT OF ALUMINUM STRUCTURE

L6-06459
FG2048.91

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2. Marissen, R. "Fatigue Crack Growth in Aramid Reinforced Aluminum Laminates (ARALL) – Mechanisms and Predictions." DFVLR-FB 84-37, Institut Fur Werkstoff – Forschung, Koln, West Germany, October 1984.

3. Swift, T. "Design of Redundant Structures," Fracture Mechanics Design Methodology, AGARD-NATO Lecture Series; Delft, Netherlands; Munich, Germany; and Lisbon, Portugal; 2-13 October 1988.

4. Gunnink, J. W. "Design Studies of Primary Aircraft Structures in ARALL Laminates," ARALL Laminates Technical Conference, Champion, Pennsylvania, October 1987.

C-130 Advanced Composite Structural Repair Development

W.O. Greenhaw, WR-ALC/MMSFRA
J.J. Grosko, LASC-GA

AF Contract No. F09603-86-G-0455-0029

GA-8347-1

C-130 Advanced Composite Structural Repair Development

Phase I

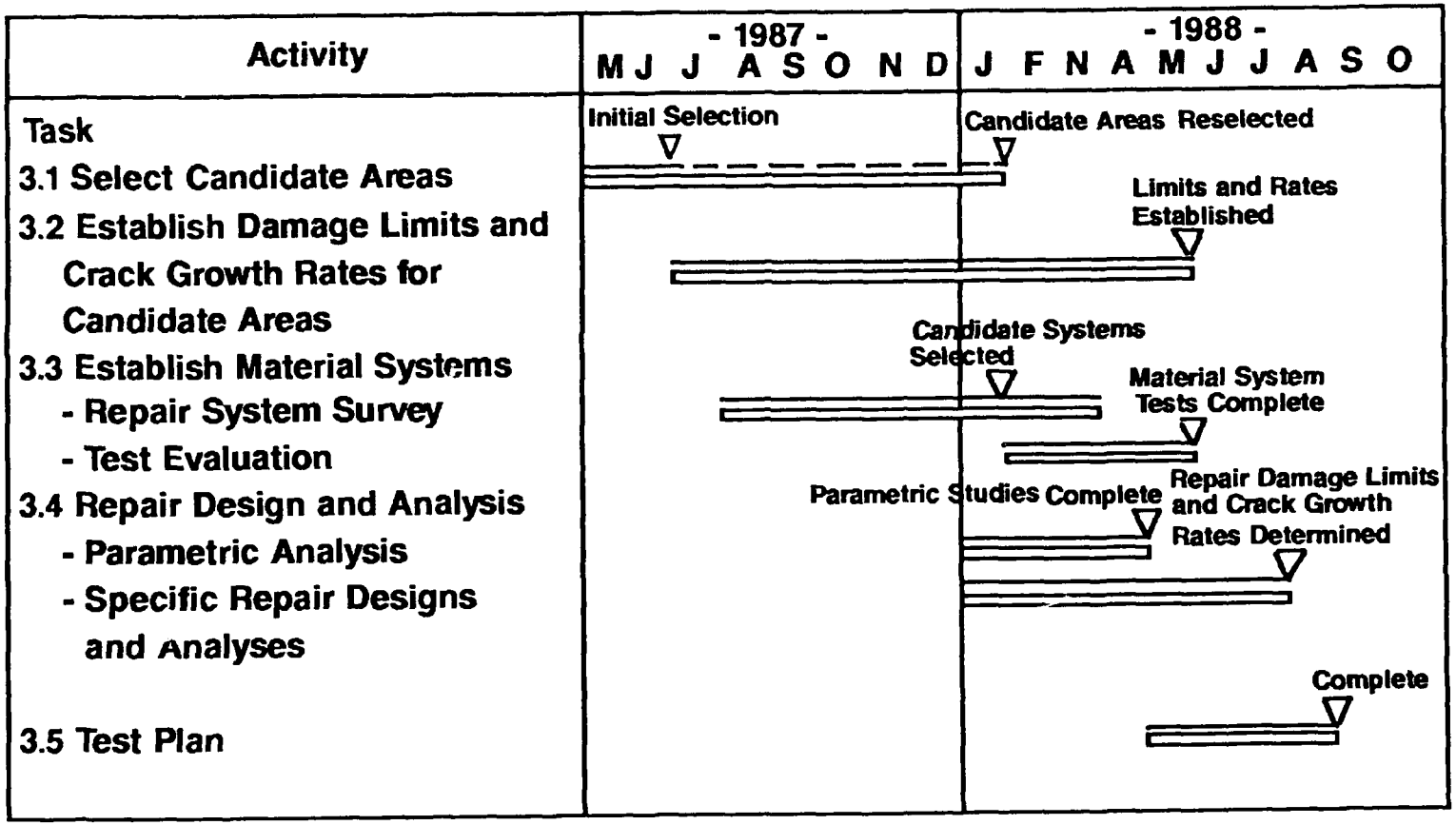
- Select Repair Location
- Establish Damage Limits
- Establish Material Systems
- Repair Design and Analysis
- Test Plan

Phase II

- Component Durability and Static Strength Tests
- Tool Design/Fabrication
- Repair Instllation Trials
- Demonstrate NDI Procedures
- Install Repairs on C-130 Wing Durability Test Article

GA-8347-3

C-130 Composite Repairs Program Schedule - Phase I



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Advantages Of Bonded Repairs

General

- Introduces No New Stress Concentrations
- Efficient Load Transfer
- Produces Sealed Interface

Patch Over Crack

- Dramatically Reduces Stress Intensity Facter At Crack Tip

Repair Materials and Processes

Graphite/Epoxy

- Hercules 3501-6/AS4
- 3M SP377/AS4

Boron/Epoxy

- TEXTRON 5505/4
- TEXTRON 5521/4

Adhesives

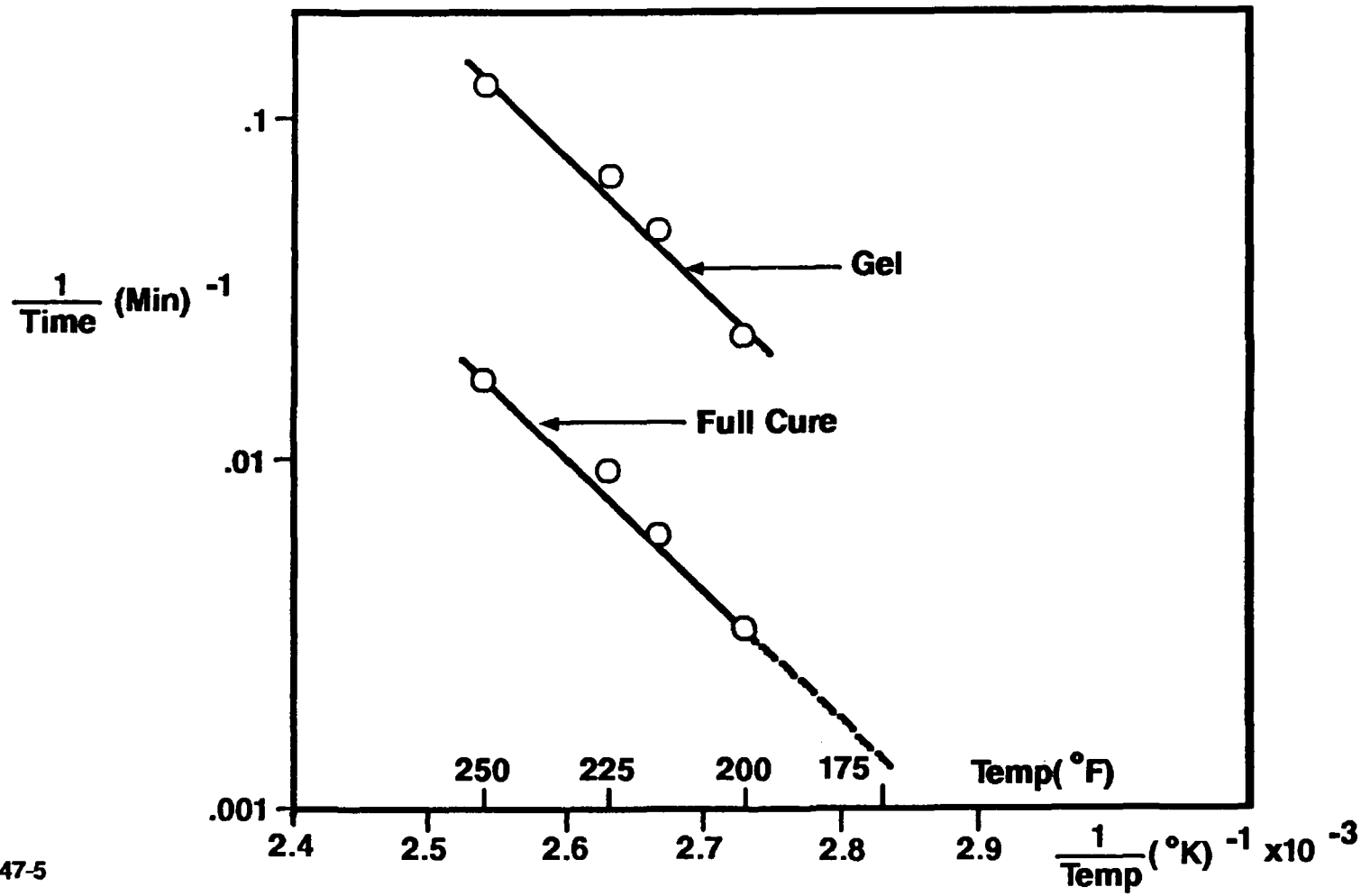
- American Cyanamid FM-73/BR-127
- 3M AF 127-3/3924B

Surface Preparation

- Cold Sulfuric Acid - Sodium Dichromate (FPL) Etch
- Mechanical Abrade/Silane, Union Carbide A-187

Required Time vs Temperature - Texron 5521/4

Source: Textron Speciality Materials



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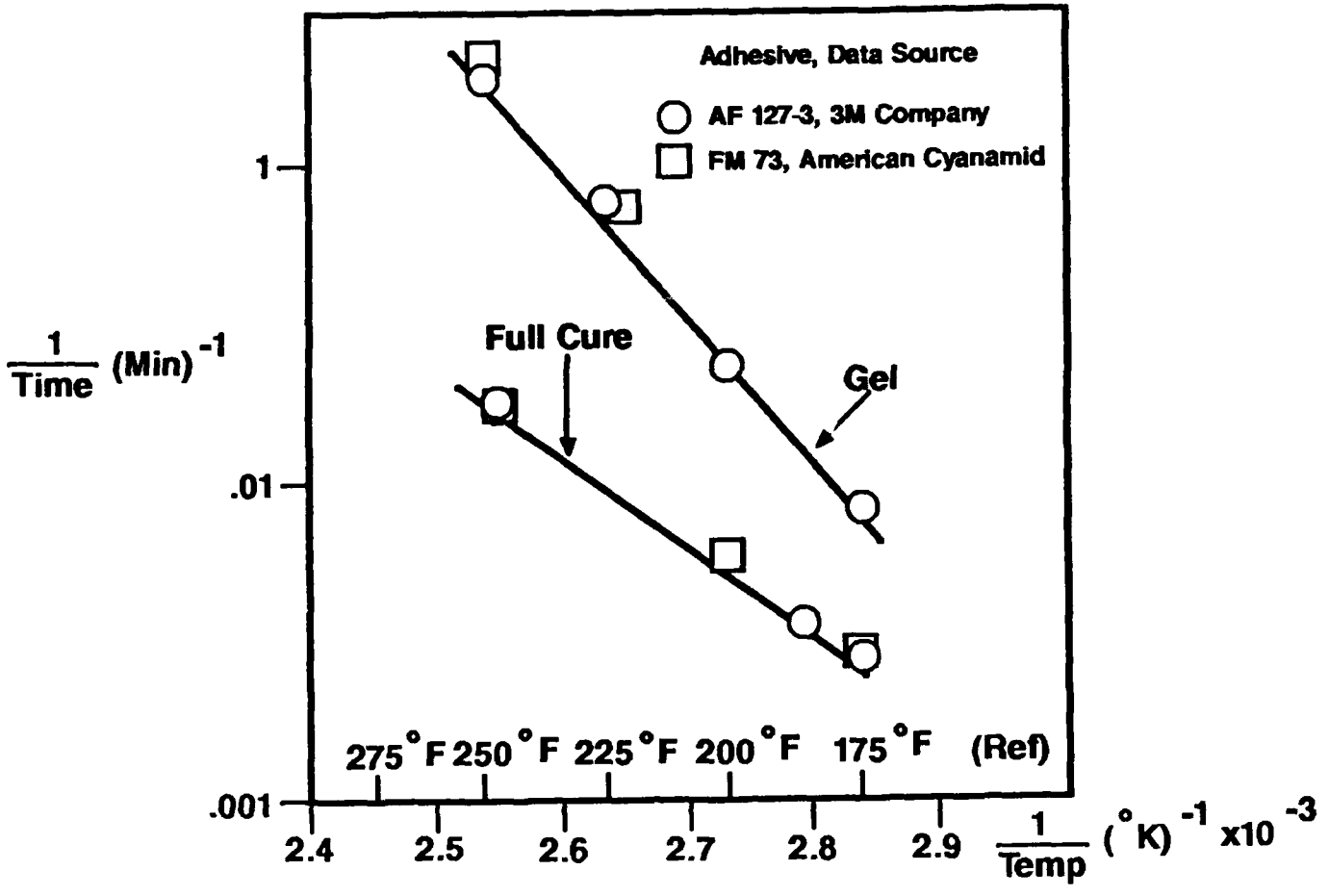
Required Time vs Temperature - Adhesive Systems



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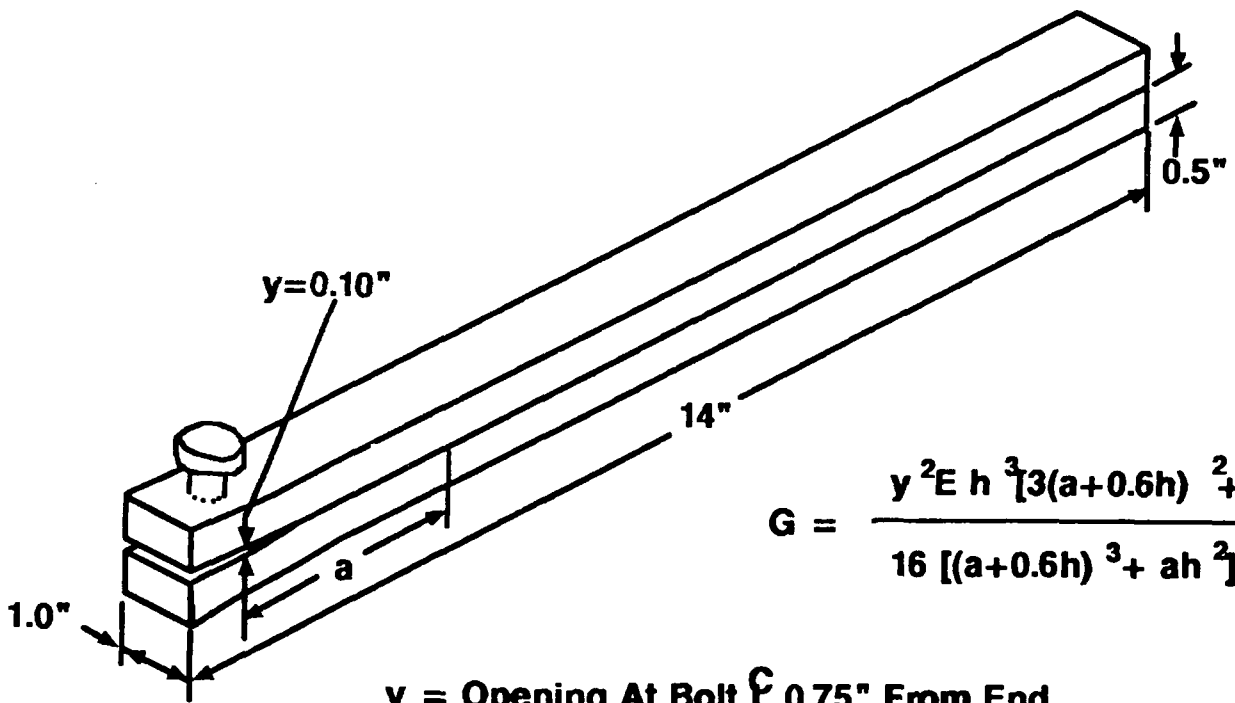
Environmental Crack Extension Test Specimen



$$G = \frac{y^2 E h^3 [3(a+0.6h)^2 + h^2]}{16 [(a+0.6h)^3 + ah^2]^2}$$

y = Opening At Bolt ℄ 0.75" From End

a = Crack Length From Bolt ℄

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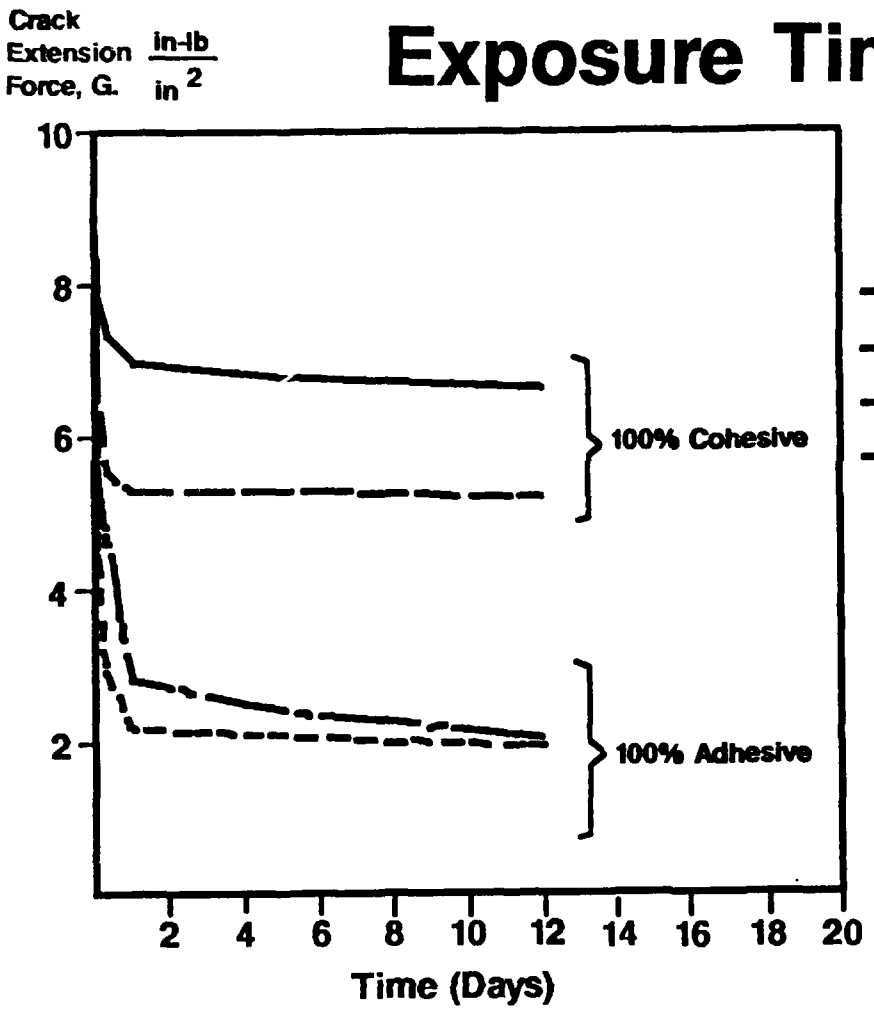
Crack Extension Force, G, vs Exposure Time



Source: LASC-GA

FPL Etch/Br-127 Primer

Grit Blast/A-187 Silane

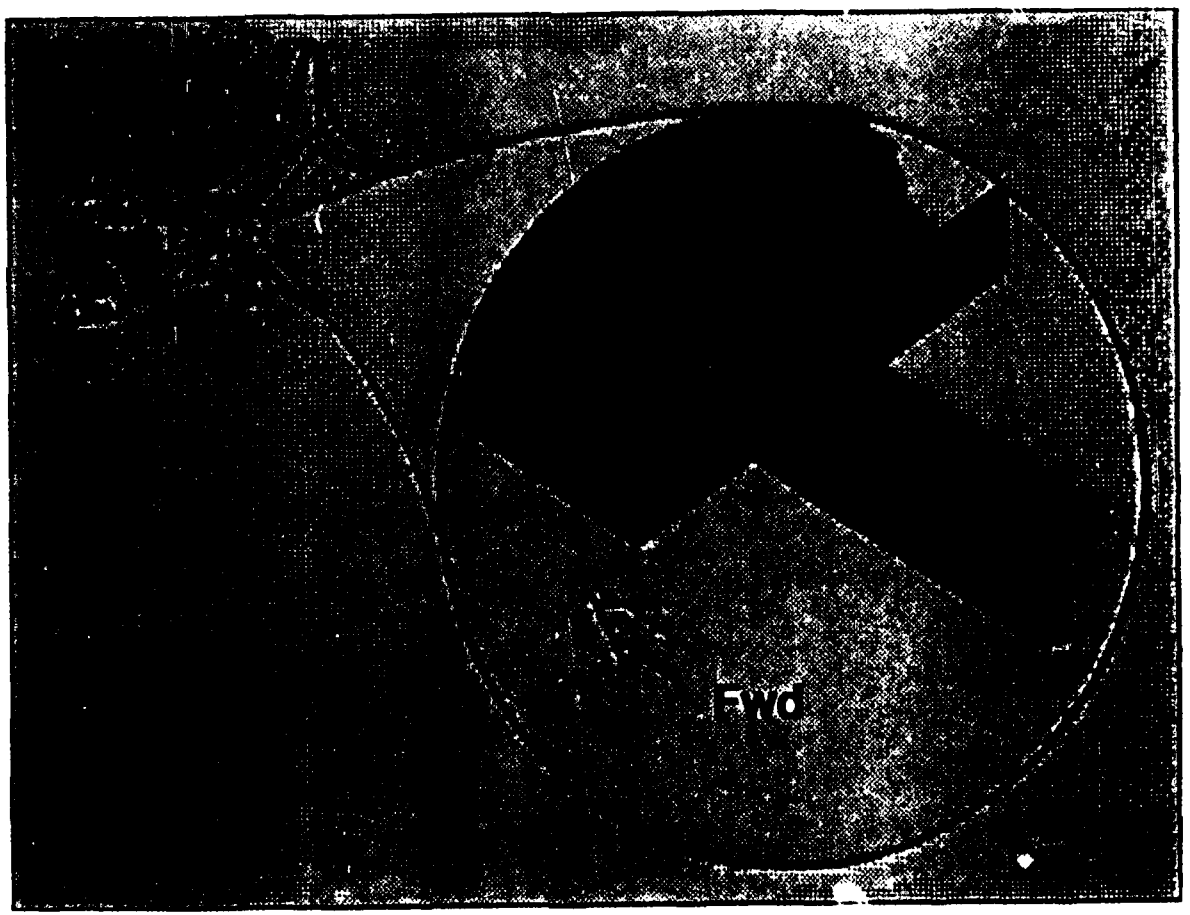
Environment: 90% RH,120° F

Adhesive: Fm 73;0.06 psf

Adherends: 7075-T6

GA-8347-8

C-130 Lower Wing Surface - Damage Site



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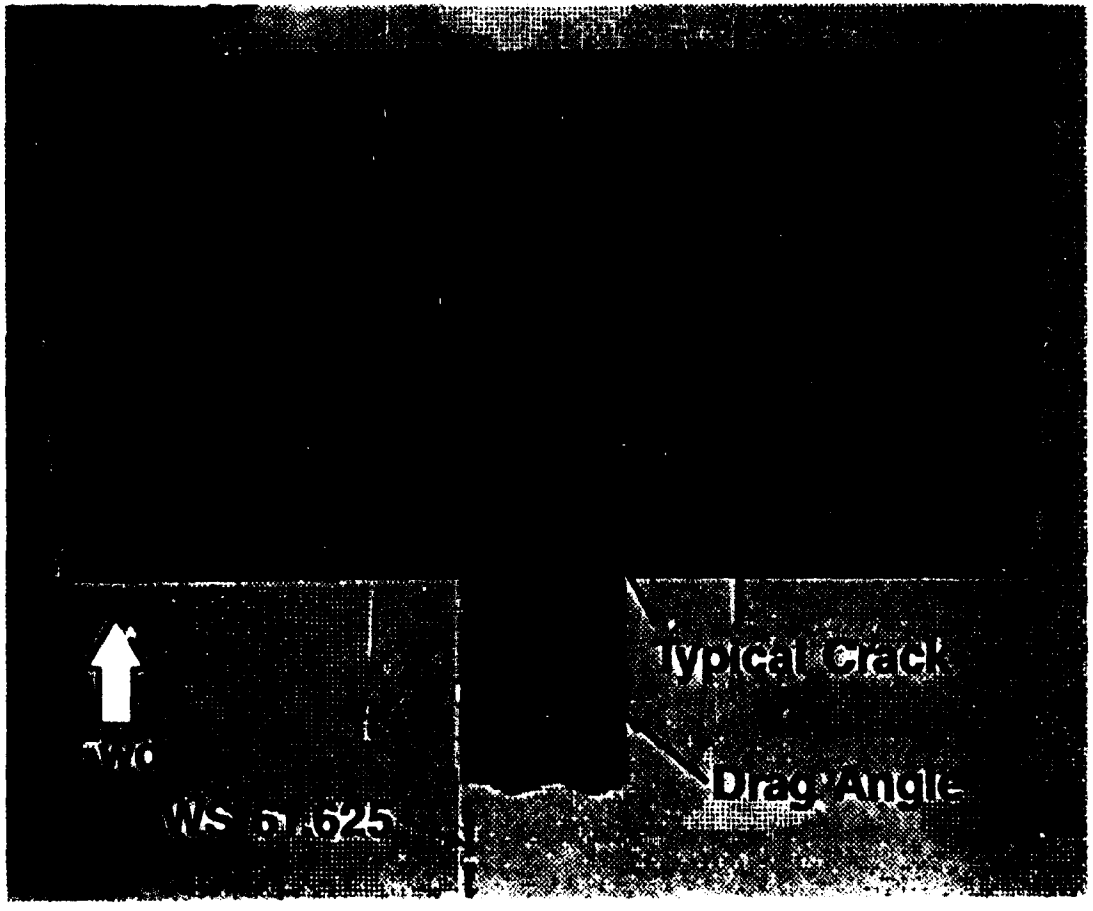
C-130 Lower Wing Surface - Damage Site



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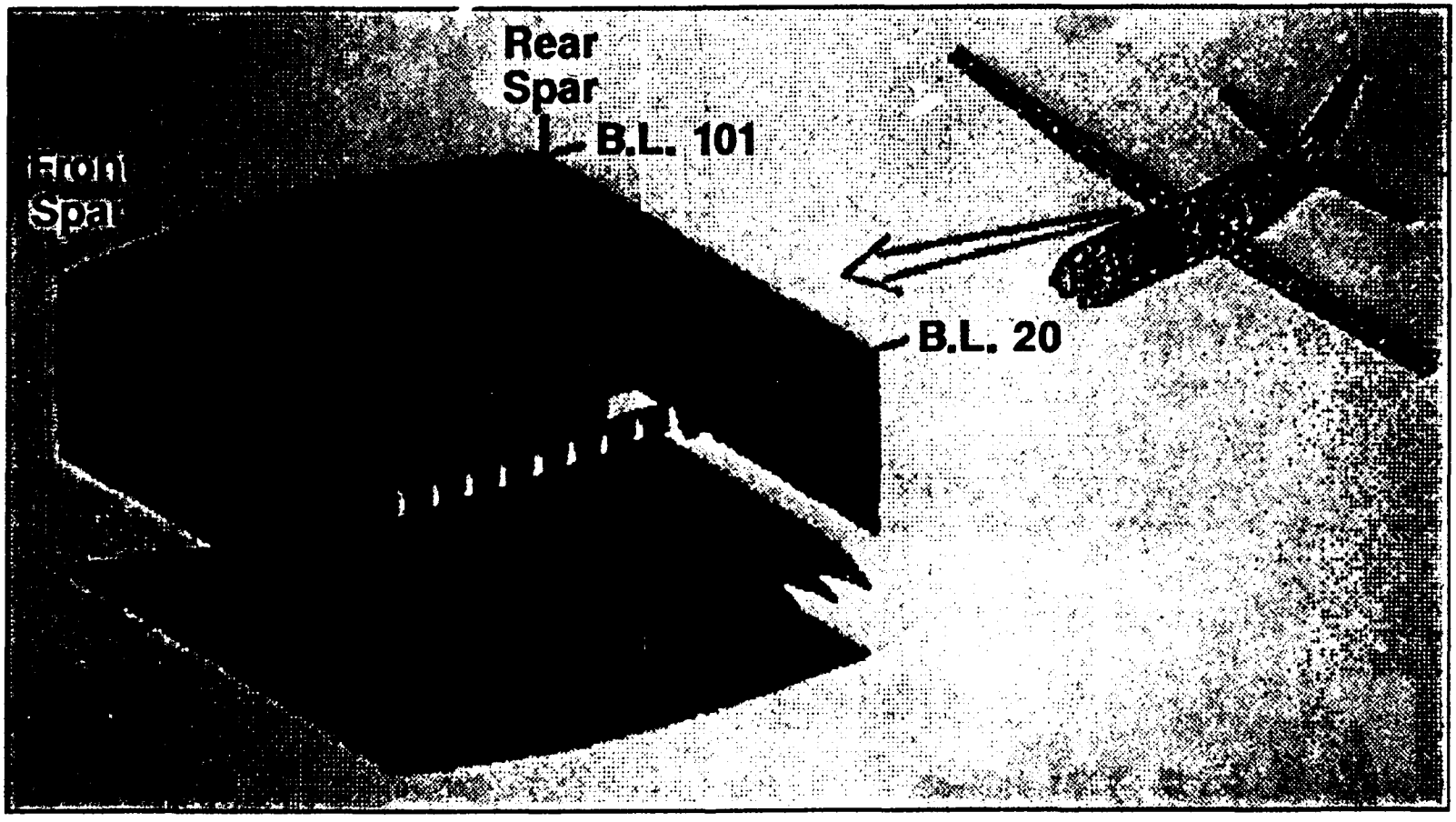
Detailed FEM Analysis - C-130 MLRS Model



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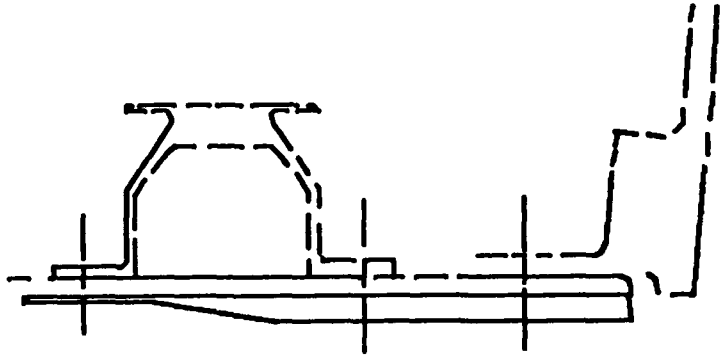
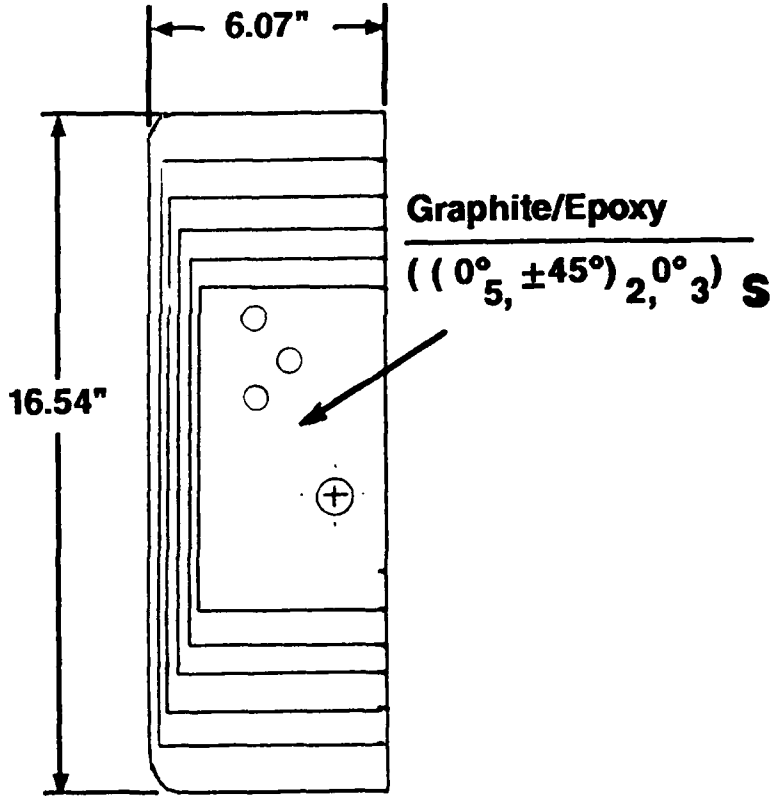
C-130 Lower Wing Surface - Bonded Patch

- Use Precured Graphite/Epoxy Patch To Directly Lower Stress Intensity Factor At Crack Tip





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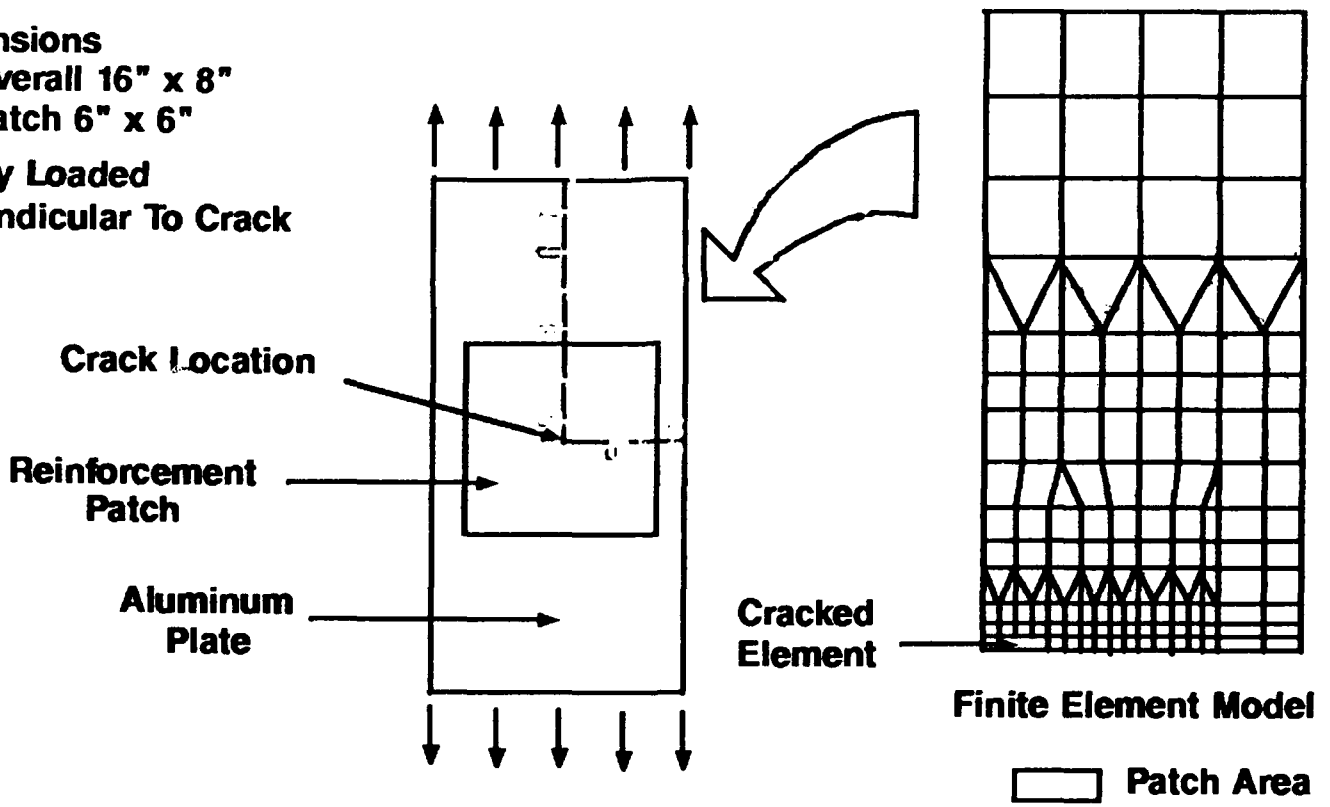
Parametric Analysis Model

- Doubly Symmetric Specimen
- Dimensions
  - Overall 16" x 8"
  - Patch 6" x 6"
- Axially Loaded Perpendicular To Crack



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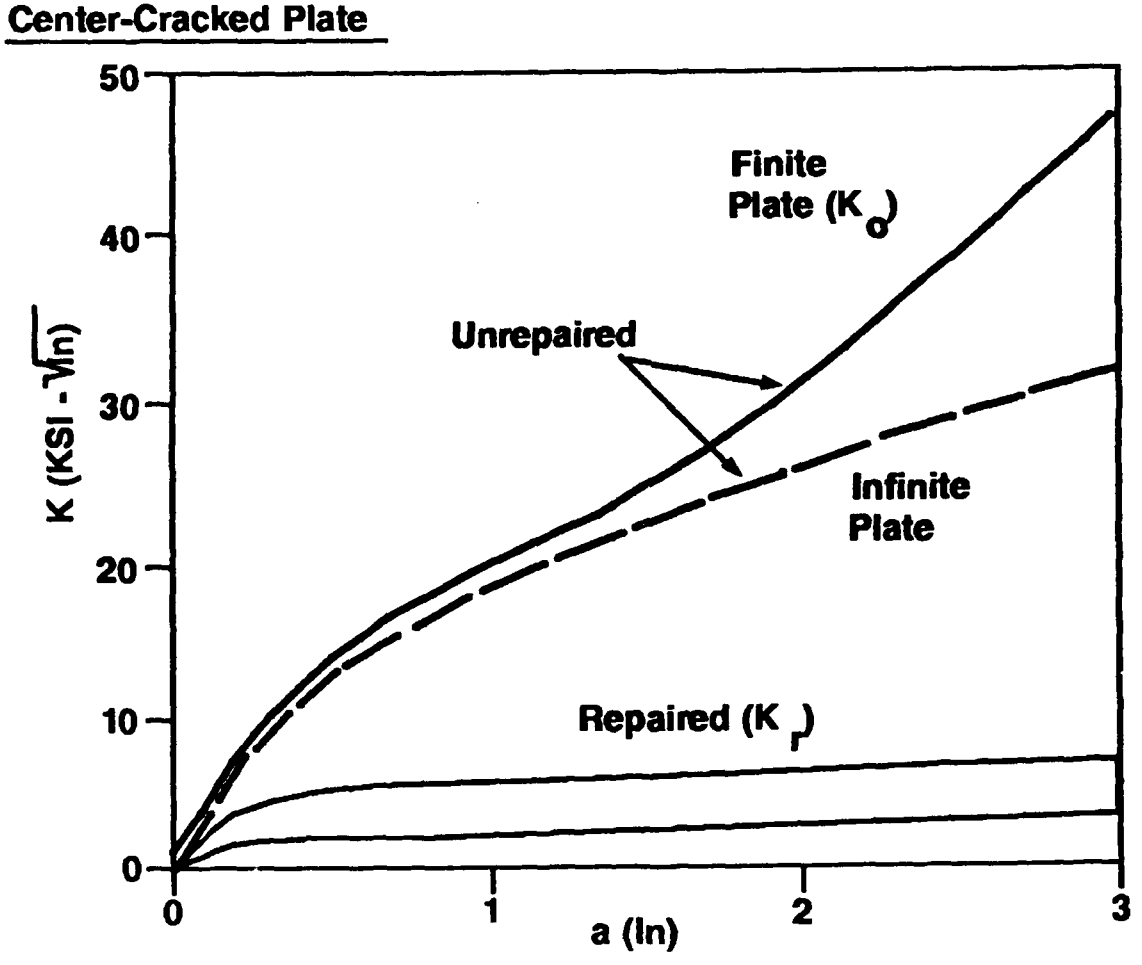
Parametric Analysis Results - K vs a



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General Design Guidelines - Rectangular Patch

- Reinforcing Patch-To-Panel Stiffness Ratio, $S = \frac{E_r t_r}{E_p t_p}$, 1.2 to 1.5
- Untapered Patch Length, $L_r = 6\sqrt{\frac{t_a E_r t_r}{G_a(1+S)}}$
- Untapered Patch Width, $W_r = a_r + \Delta a$ Through Repair Life
- On Edges Subjected to Significant Load Entry
  - Taper All Edges
  - Round All Corners

Patch Material Selection

Boron/Epoxy

- Stiffer
- Less Thermal Expansion Mismatch
- No Galvanic Corrosion
- Easier to Inspect

Graphite/Epoxy

- Easier to Cut/Drill
- More Available
- Easier to Handle
- Easier to Form to Shape
- Less Expensive

GA-8347-28

C-130 Lower Wing Surface - Bonded Patch Detailed Analysis Results

No Reinforcement - Baseline

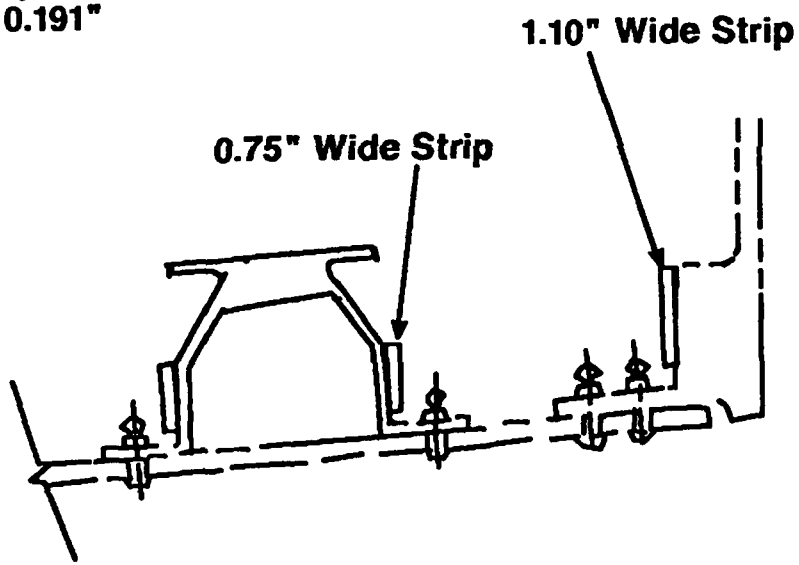
- 3" Crack Grows to a_{cr} = 6.29" in 5500 Flt Hrs Under 29 Mission Operational Flight Spectrum

With Repair

- Composite Patch Strength Margin = 0.597
- Linear Analysis Indicates High Bondline Stress at Crackline at Limit Load (To Be Assessed During Phase II Testing)
- 3" Crack Grows to 6.25" in 200,000+ Flt Hrs Under 29 Mission Operational Flight Spectrum

C-130 Center Wing - Stringer/ Spar Cap Reinforcement

- Use Precured Boron/Epoxy Strips To Lower Stress Level At Crack Tip Site
- All Reinforcement 0° Boron/epoxy 24" Long, Max. Thickness= 0.191"



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C-130 Center Wing - Stringer/Spar Cap Reinforcement Detailed Analysis Results

Load Case: 2.5g Positive Symmetric Maneuver Plus 11.25 psi Fuselage Pressure	Δσ (psi) at BL61/Rear Spar Interface w/3 Straps	w/5 Straps
• Induced During 180°F Cure	-3550	-3550
• Cool Down From Cure Temp 180°F ⟶ 80°F	+4025	+4042
• Stress Reduction at Ultimate Load Due to Repair	-5700	-7450
Net At 80°F	-5225 (9.3%)	-6958 (12.4%)
• Further Cool Down From 80°F To -67°F	+1530	+1860
Net At -67°F	-3695 (6.6%)	-5098 (9.1%)

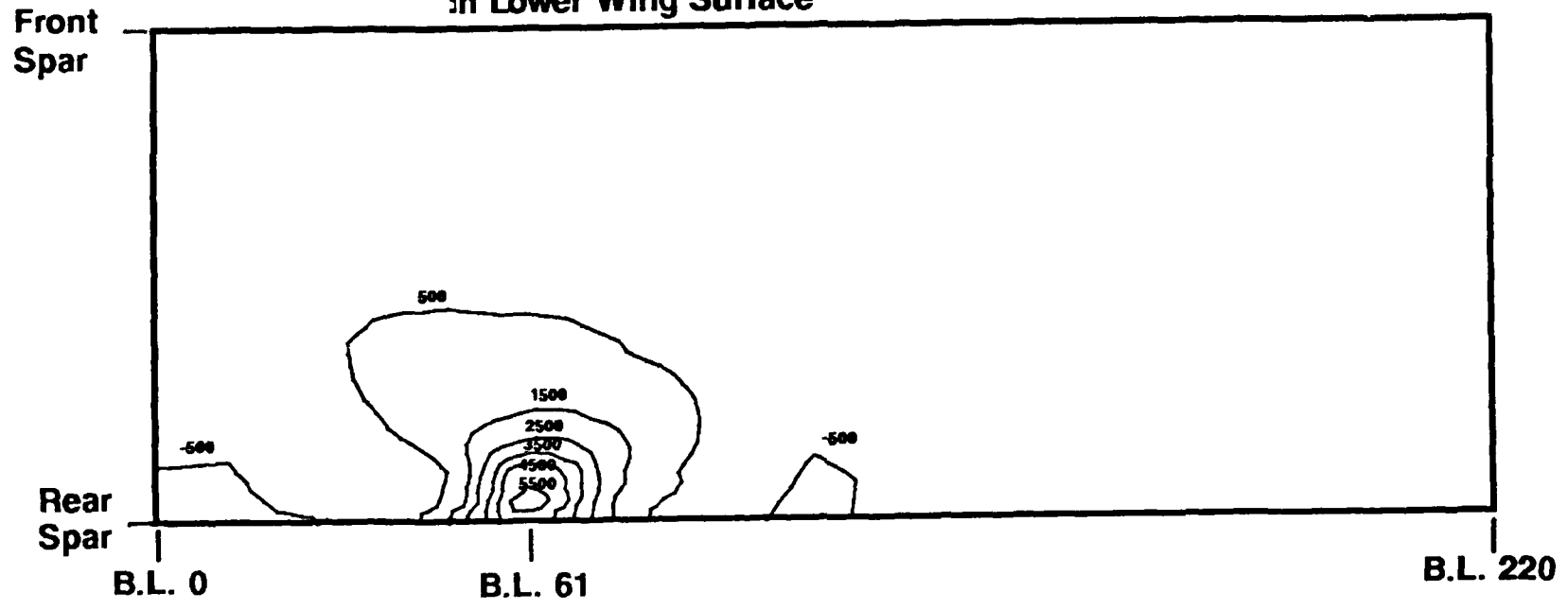
Limit Load at 160°F

- Maximum Adhesive Shear Stress of 5300 psi
- Maximum Axial Stress in Boron/Epoxy Strip of 96000 psi

GA-8347-23

C-130 Center Wing - Stringer/Spar Cap Reinforcement Stress Reduction Contours

- Configuration: 3 Straps, i.e. Rear Beam Cap And Rear Stringer Reinforced
- Load Case: 2.5g Positive Symmetric Maneuver Plus 11.25psi Fuselage Pressure
- Contours: Spanwise Tensile Stress Reduction In Lower Wing Surface



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C-130 Center Wing - Stringer/Spar Cap Reinforcement Detailed Analysis Results

Configuration	Stress Reduction At Wing Cover Critical Location At Indicated Temperature		Safety Limit	Life Increase
	-67°F	80°F		
Baseline - No Straps	—	—	22,000 Hrs	—
Rear Beam Cap And Rear Stringer Reinforced	6.6%	9.3%	33,000 Hrs	50%
Rear Beam Cap And Two Rear Stringers Reinforced	9.1%	12.4%	39,000 Hrs	77%

Notes:

1. Stress Reduction Based On Ultimate 2.5g Positive Symmetric Maneuver Plus 11.25psi Fuselage Pressure
2. Safety Limit Based On Operational Flight Spectrum

GA-8347-25

Phase II Activities

- Start January 1989
- Component Durability Tests
- Component Static Strength Tests
- Tool Design/Fabrication
- Repair Installation Trials
- NDI Demonstration
- Install Repairs on C-130 Wing Durability Test Article

GA-8347-26

LOW VELOCITY IMPACT DAMAGE DETECTOR DEVELOPMENT

Arthur E. Scotese and Dr. Shih Lien Huang

ABSTRACT

This paper describes the development of a low velocity impact surface damage detector for laminated composites and honeycomb sandwich structures. A low cost, hand-held battery-powered damage detector has been developed which can be operated in the field to detect minute surface perturbations resulting from foreign object impact that can cause disbond or delamination or both in aircraft structures. Since the operation of the device is very simple and its detecting capability is not affected by temperature, humidity or vibration, it can be used under field operational conditions by personnel with minimum training. The basic principle used by the device is Shadow Moire interferometry. When the structure is examined through a 4-inch x 5-inch viewing field, surface perturbations become visible in the form of Moire fringe patterns. The entire detector system including the power pack weighs less than 3.5 lbs. It is capable of detecting surface perturbations of less than 0.001 inch. The detector has proven to be practical, effective and reliable in detecting impact damage in composite laminate and honeycomb sandwich structures. Delaminations and disbonds have also been successfully detected. Damage sites can be scanned and identified quickly. In the event further damage characterization becomes necessary, the flaw site can be further analyzed and quantified with other NDE techniques.

INTRODUCTION

Application of cyclic loads to a structure can produce damage which propagates progressively and finally leads to catastrophic failure. The failure modes of composite structures differ considerably from those of metal structures. With metals, fatigue cracks are visible and initiate in the plane normal to the outer surface. With composites, however, damage often initiates and propagates internally, parallel to the laminate plane (References 1,2,3,4,5). Propagation of delamination is a major problem and has been responsible for the ultimate failure of composite laminates. Certain types of flaws in composite structures, such as low velocity impact damage (Reference 6), are not readily observable by visual examination. Other examples include blind-side and subsurface interlaminar cracks.

The increased use of laminated composites and honeycomb sandwich structures on naval aircraft has necessitated the development of inspection techniques to detect the presence and criticality of cracks, flaws, disbonds, delaminations and other defects. These inspections help assure that composite structures provide the same level of structural reliability as comparable metal structures. A variety of non-destructive methods are currently employed to detect internal damage in laminates (References 7,8) Ultrasonics and X-ray are commonly used in industrial applications. Thermographic, laser holography, acoustic emission, and simple hammer

tapping have demonstrated varying degrees of applicability in experimental and field quality control programs. These methods have sometimes proven to be tedious, time consuming and unreliable in field assignments where temperature, humidity and vibration can severely affect the results.

The availability of a more effective means for field detection of impact and resultant delamination is one of the major requirements for the extensive use of structural composites in aircraft today.

DAMAGE DETECTOR DEVELOPMENT

During the development of the damage detector, the key Moire design parameters were identified, e.g., illumination requirements, screen dimensions, line densities, magnification factors and photographic methods. Several candidate sensitivity enhancement techniques were also evaluated.

1. Laboratory Shadow Moire Optical System
   A baseline system was designed for use in the laboratory, with emphasis on the capability to vary and determine the key Moire design parameters.

2. Shadow Moire Out-of-Plane Interferometric (SMOOPI) Damage Detector

   Incorporating the optimum design parameters determined from the Laboratory Shadow Moire Optical System, a portable battery -operated field inspection instrument was developed for the detection of low velocity impact damage and delaminations in laminated composite and honeycomb sandwich structures.

The prototype SMOOPI Damage Detector was used to detect minute surface perturbations resulting from foreign object impact that can cause disbond or delamination or both in aircraft structures. The operation of the device, which is based on Moire Interferometry, is very simple and its detection capability is not affected by temperature and humidity variations or vibrational disturbances. It can be used under field operational conditions by personnel with minimum training. When the structure is viewed through a 4-inch x 5-inch viewing field, surface perturbations become visible in the form of Moire fringes.

MOIRE INTERFEROMETRY

Moire Interferometry (References 9,10,11) has been well developed for in-plane displacements. The basic technique deals with the interference fringes created by the superposition of two grids or systems of finely ruled lines, one on the specimen and the other a master or reference grid. The specimen grid is either printed on, bonded to, or etched on the surface. The superposition of the reference and the specimen grid (Figure 1) is done either by direct contact or by imaging one onto the other via a lens. When subjected to mechanical forces, the test specimen is deformed and the specimen grid along with it. Because of the surface strains and resultant change in pitch (line to line distance) of the specimen grid

rulings, Moire fringes are produced as a result of transmission and obstruction of light passing through the deformed and undeformed grids. The Moire fringes which represent the loci of points of constant displacement in a direction normal to the grid lines appear as black bands, because light transmitted by one set of opaque lines is obstructed in some regions by the other set. When the lines of the master grid coincide in some areas with the spaces of the specimen grid there is obstruction of light and Moire fringes are produced. The grid density in lines per inch (lpi) is usually small, e.g., 150 lpi to 500 lpi, and individual lines are not seen.

Shadow Moire is also based on interferometry and the superposition of two grids, but in this case, the specimen grid is the shadow of the master grid on the specimen surface (References 12,13). The Shadow Moire technique deals with the measurement of out-of-plane displacements or the out-of-plane shape of structures, rather than in-plane deformation. In a typical Shadow Moire set-up (Figure 2), a Master grid, is placed close to the examined surface and illuminated at some angle θ from 30^{o} to 75^{o}, and a shadow of the grid is projected on the surface. The shadows, i.e., the specimen grid, are distorted by the out-of-plane elevation or depth of the surface and when viewed through the master grid produce fringes which are clearly visible. A set of visible fringes is a map of level lines revealing the topography of the observed surface. When the specimen is deformed, the pitch of the shadow grid changes. When the lines on the master grid project onto lines on the specimen, reflection of the light path occurs. In some regions, however, the lines of the master obstruct the light reflected by the specimen and black Moire fringes appear.

DESCRIPTION OF SMOOPI DAMAGE DETECTOR

The SMOOPI Damage Detector shown in Figure 3, consists of an anodized aluminum frame on an extended pistol grip with attachments for easy installation of a 4-inch x 5-inch Moire grid, providing a .030-inch clearance between the grid and the surface to be examined. A 35 watt quartz light source with a .050-inch slit provides a collimating line source, simulating a remote point, and is attached to the frame with a 6-inch support for simple adjustments of illuminator angle and vertical position. The light source can be powered from a portable, shoulder-strap-mounted, rechargable battery-pack or from an AC adapter. The light source with battery-pack can last for one hour of continuous operation before recharging. The entire detector system including the battery-pack weighs less than 3.5 lbs.

Grids with line densities up to 1000 lpi were successfully used with the SMOOPI device. Severe diffraction effects, with the fringe orders being obscured, occurred with grid line densities greater than 1000 lpi. The best results were obtained with grid line densities of 50, 150, and 250 lpi. The 150 lpi grid is for general purpose applications and is used in most cases. A low and high density grid can also be used when required. For cases with too high a signal or poor contrast, the 50 lpi grid should be used. For cases with low displacement fields, the 250 lpi grid is appropriate and provides the best signal. A three-point contact system is

also incorporated into the frame to accommodate inspection of curved surfaces and to produce an out-of-plane rotation of the master grid that creates a grid-bias or carrier fringe pattern (series of parallel lines) for sensitivity enhancement. The surface to be examined should be diffusely reflecting light. For surfaces with poor contrast, a white spray powder (Fluorofinder FD-33 Developer or equivalent) is used to increase contrast during inspection. The powder is then easily cleaned from the surface after inspection.

When compared to other Non-Destructive techniques, the major advantages of the SMOOPI device are:

Low cost

Portable and light weight

Needs no specimen grid.

Scans large areas at a time.

MEASUREMENT AND DETECTION

The sensitivity of the system and "motion" of the shadows is based on the selected illumination angle, grid density and the elevation or depth of the observed point. The classic equation of the Shadow Moire technique is:

$$Z = 1/(P \times \text{Tan } \theta) \qquad \text{Equation 1}$$

where Z is the level difference from one fringe to the next fringe, P is the grid density in lines per inch (lpi) and θ is the angle of incidence between the light source and the normal to the specimen surface. Using a 45° illumination angle, Tan $\theta = 1$
and $Z = 1/P$

The light source is usually set at a 45° angle of incidence for most applications, but other angular settings are possible as indicated on the mounting bracket. When an increase in sensitivity is required, a 63.5° angle of incidence can be selected that will double the output of the 45° selection, or a grid with greater line density can be selected. Table I contains the displacement-per-fringe values for all available grid line densities and angles of incidence, and this information is presented graphically in Figure 4.

Experimental Results and Performance

When using the SMOOPI Damage Detector, the frame is placed in contact with and parallel to the surface to be examined. On a flat surface with no damage, a zero pattern without any fringes is observed. For this condition, the pitch of the master grid and shadow remain the same. If the surface has a slight warpage or curvature, a series of equidistant parallel fringes are observed. Each fringe indicates a change in pitch between the

master grid and specimen grid (shadow). In areas displaying one or more closed fringes (where each fringe or pitch change is a line of constant displacement), the general shape of the damage and a contour map of level lines revealing the surface topography will become visible. For example, spherically shaped damage areas will reveal a fringe pattern of concentric circles. This is shown in Figure 5, where using a-50 lpi grid and a 63.5° incidence angle, three concentric fringes are visible having a .01 inch-per-fringe displacement. In Figure 6, two fringes with slight perturbations and three closed fringes are visible representing a displacement of .0067 inch-per-fringe. In this case, the specimen was viewed with a 150 lpi grid at a 45° angle of incidence. The damage area profile for both impact fringe patterns is shown in Figure 7. This was done by drawing vertical lines thru each fringe that intersected the horizontal line drawn thru the center of the damaged area. The increase in pitch from one fringe to the next is plotted for each line giving a displacement profile of the cross-section.

For a 50 lpi density at a 45° illumination angle, one fringe represents a locus of points of equal out-of-plane displacement of .02 inch, as determined from Equation 1 or shown in Figure 4 or Table I. Using the general purpose grid of 150 lpi density, one fringe is equal to .0067 inch displacement at a 45° illumination angle. The sensitivity may be increased to .0033 inch per fringe at a 63.5° angle of incidence. For a 250 lpi grid at a 63.5° illumination angle, the sensitivity can be increased to .0020 inch of displacement per fringe (see Figure 4 or Table I). On a curved surface, fringes will also reveal the surface topography. For example, a curved surface will show a set of lines parallel to its axis. In this case, surface anomalies will be more difficult to observe, so it is very important to completely understand the shape of the zero or baseline contours.

Carrier Fringes

A simple approach to increase sensitivity and eliminate the ambiguity associated with a low order displacement field, is to use the three-point contact system attached to the frame to form a plane which is tilted slightly from the master grid plane. For this type application, the frame is placed initially at a small angle (a few degrees out-of-plane) to the surface by adjustment of the contact screws on the frame, and on a flat surface, a pattern of equidistant parallel fringes (carrier fringes) is observed. A carrier fringe pattern of approximately two fringes per inch is adequate for most applications. If any surface perturbations are present, the parallelism of the carrier fringes will be disturbed in areas exhibiting less than one fringe, and the fringes will deform out-of-line as shown in Figure 8. A fringe order of 1/2 magnitude can be easily interpolated by eye. In this case, using a 150 lpi grid and a 45° incidence angle, the elevation lines appear in .0067 inch-per-fringe increments. Interpolating 1 and 1/2 fringes, the depth sensitivity in a direction perpendicular to the grid is .01 inch. Using a high density grid of 250 lpi and a 63.5° incidence angle, Figure 9, the depth increase for one fringe is .0020 inch and the visual sensitivity with a 1/4 fringe interpolation is .0005 inch. The contrast between a low damaged area and a high damaged area is shown in Figure 10, where a 150 lpi density grid and a 63.5 angle of incidence were used. For this case, one and one-half

fringes and four fringes are displayed with a .0033 inch displacement per fringe.

Damage Quantification

Surface damage detection (References 14,15,16,17) with SMOOPI provides a means for fast inspection of large areas. For example, a 36 inch x 36 inch area can be easily inspected in a few minutes. After surface perturbations have been identified, the area can be A-Scanned or C-Scanned, if necessary, to further quantify flaw extent. This approach is demonstrated in Figures 11 and 12. A 25 ply graphite/epoxy specimen was inspected with the SMOOPI Damage Detector and three damage points were identified. One of the points, which is typical of the three, is shown in Figure 11, where three fringes are visible indicating an out-of-plane displacement profile of .002 inch per fringe. A 250 lpi grid and a 63.5^{o} illumination angle were used. The specimen was then C-Scanned (Figure 12), revealing subsurface damage and the size of the delamination at each of the three points.

Fatigue Damage and Blind-Side Damage Assessment

Figure 13 shows the Moire fringe pattern and fatigue damage (delamination) around a hole in a 25-ply Gr/Ep specimen when viewed with a 500 lpi grid and a 63.5^{o} illumination angle. Three distinct fringes are visible corresponding to an out-of-plane displacement of .003-inch. Figure 14, shows a typical pattern associated with back-surface shattering, where the damage is not visible on the front surface. The damage in this area produced an out-of-plane displacement on the top surface of less than .001-inch. The fringe pattern shows the deformation created by the split on the back surface.

CONCLUSIONS

The SMOOPI Damage Detector has proved to be practical, effective and reliable in detecting impact damage in composite laminate and honeycomb sandwich structures. Delaminations and disbonds have also been successfully detected with a surface perturbation detection capability of .001 inch or less. Damage sites can be scanned and identified quickly. In the event further damage characterization becomes necessary, the flaw sites can be further analyzed and quantified with other NDE techniques.

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The authors would like to thank their colleagues Joseph Minecci, Armando Gaetano, Glenn Werczynski and Gwynn McConnell of NAVAIRDEVCEN for their technical assistance and thoughtful comments, and through whose collaboration many of the results presented herein were obtained.

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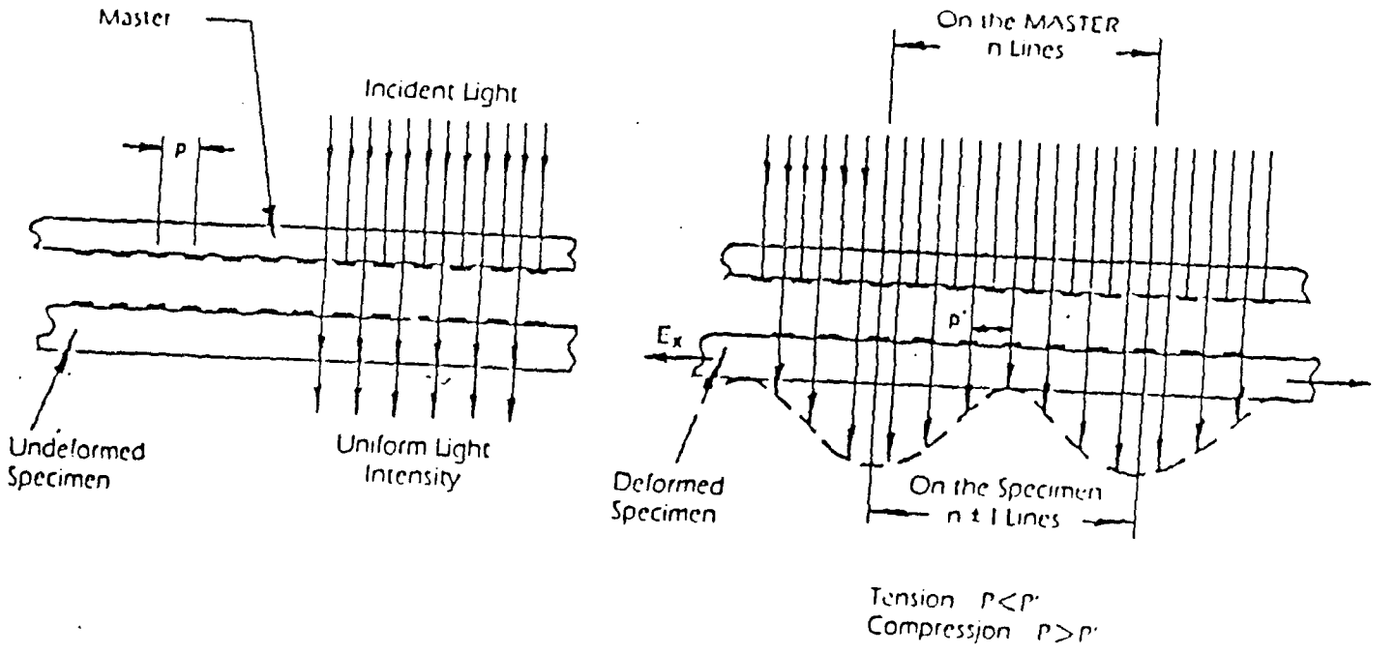


Figure 1. Moire Fringe Analysis: Superposition of Master and Specimen Grids.

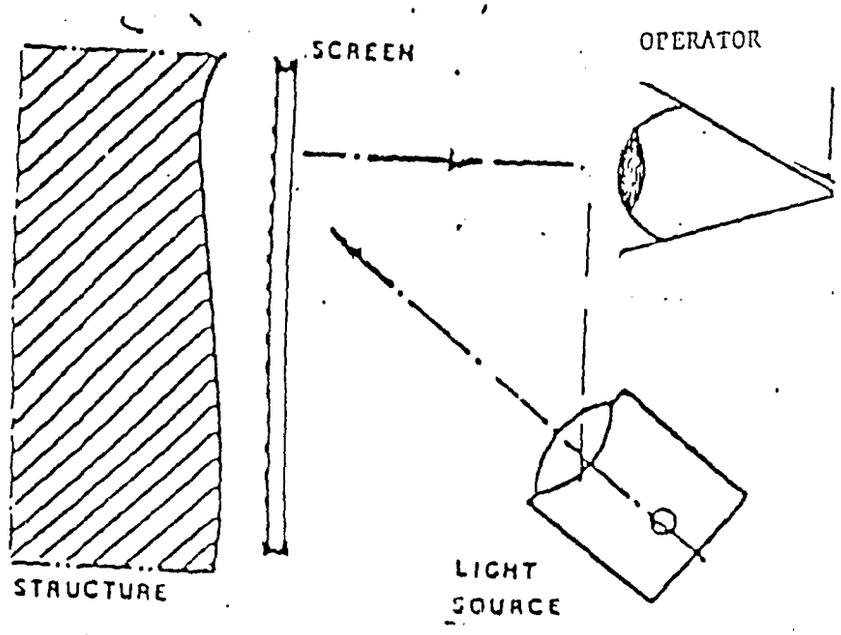


Figure 2. Shadow Moire: Schematic Diagram of Setup.

Table 1 Displacement Per Fringe For SMOOPI Damage Detector

ANGLE OF INCIDENCE DEGREES / GRID LINE DENSITY LPI	50	150	250
30	.0345	.0118	.0069
45	.020	.0067	.004
63.5	.010	.0033	.002
75	.0034	.0018	.0011

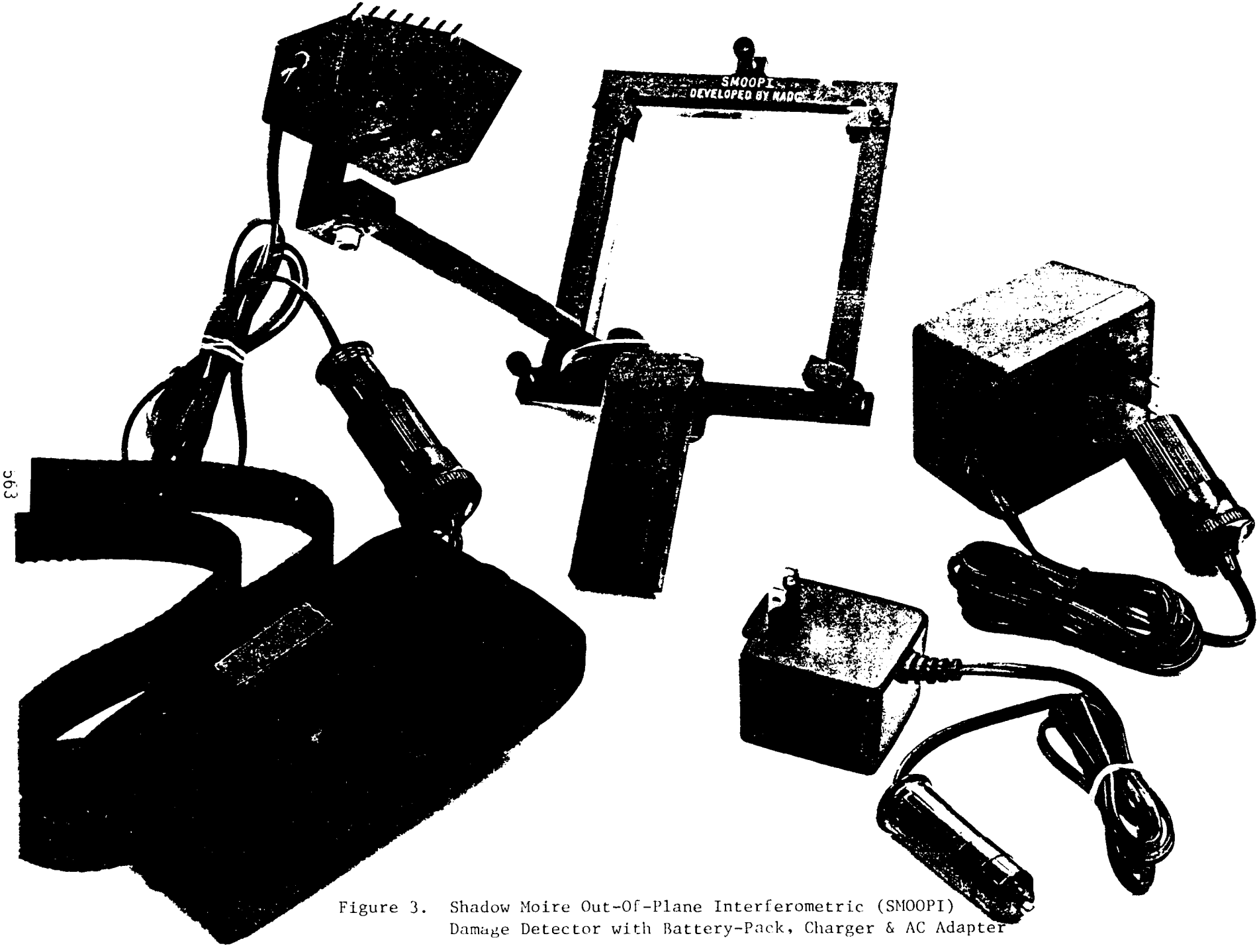
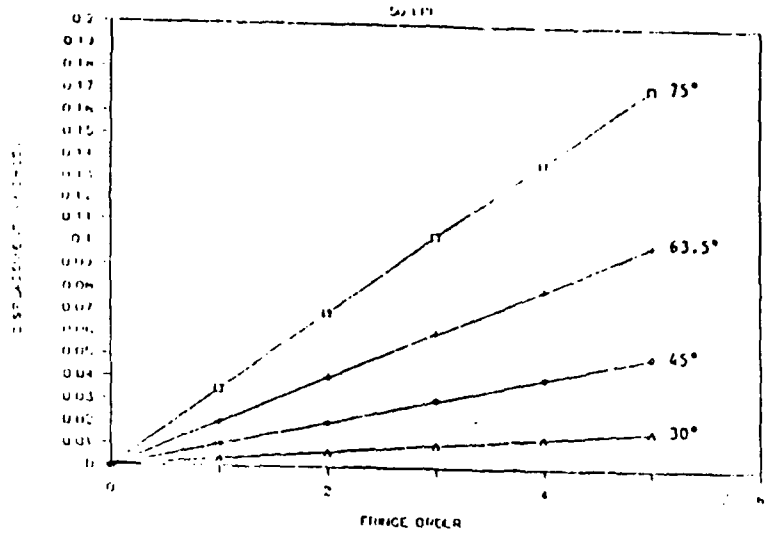
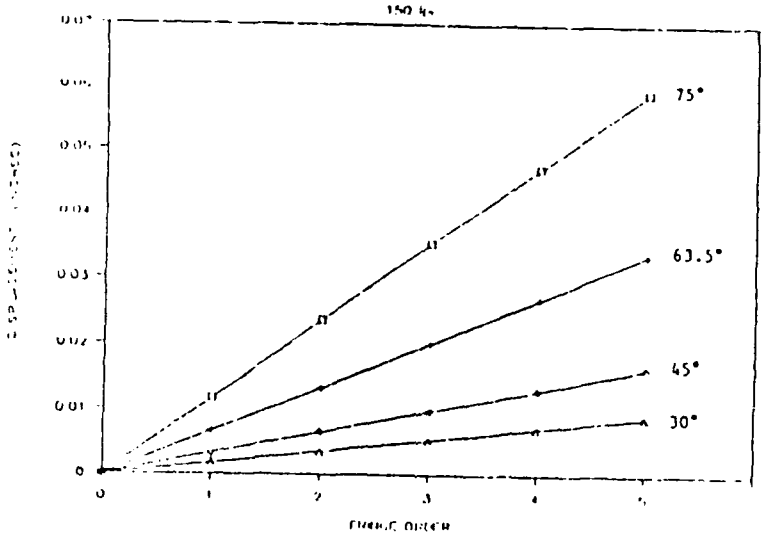


Figure 3. Shadow Moire Out-Of-Plane Interferometric (SMOOPI) Damage Detector with Battery-Pack, Charger & AC Adapter





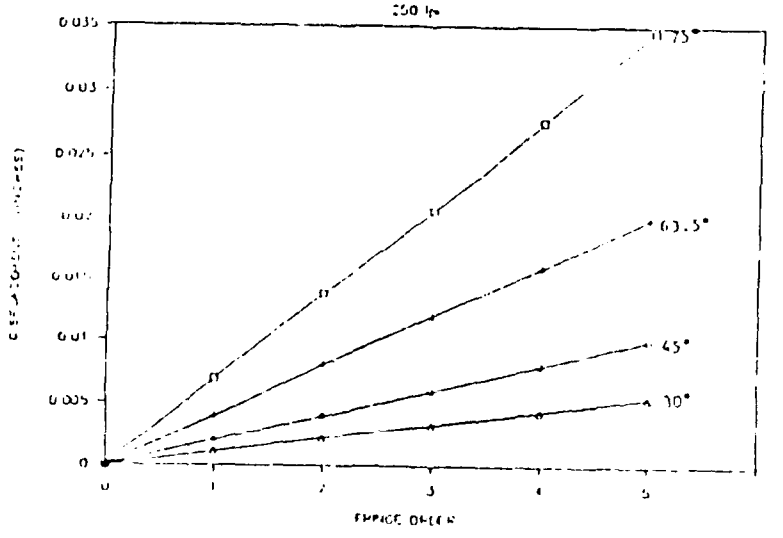


Figure 4. . Displacement Per Fringe for SMOOPI Damage Detector

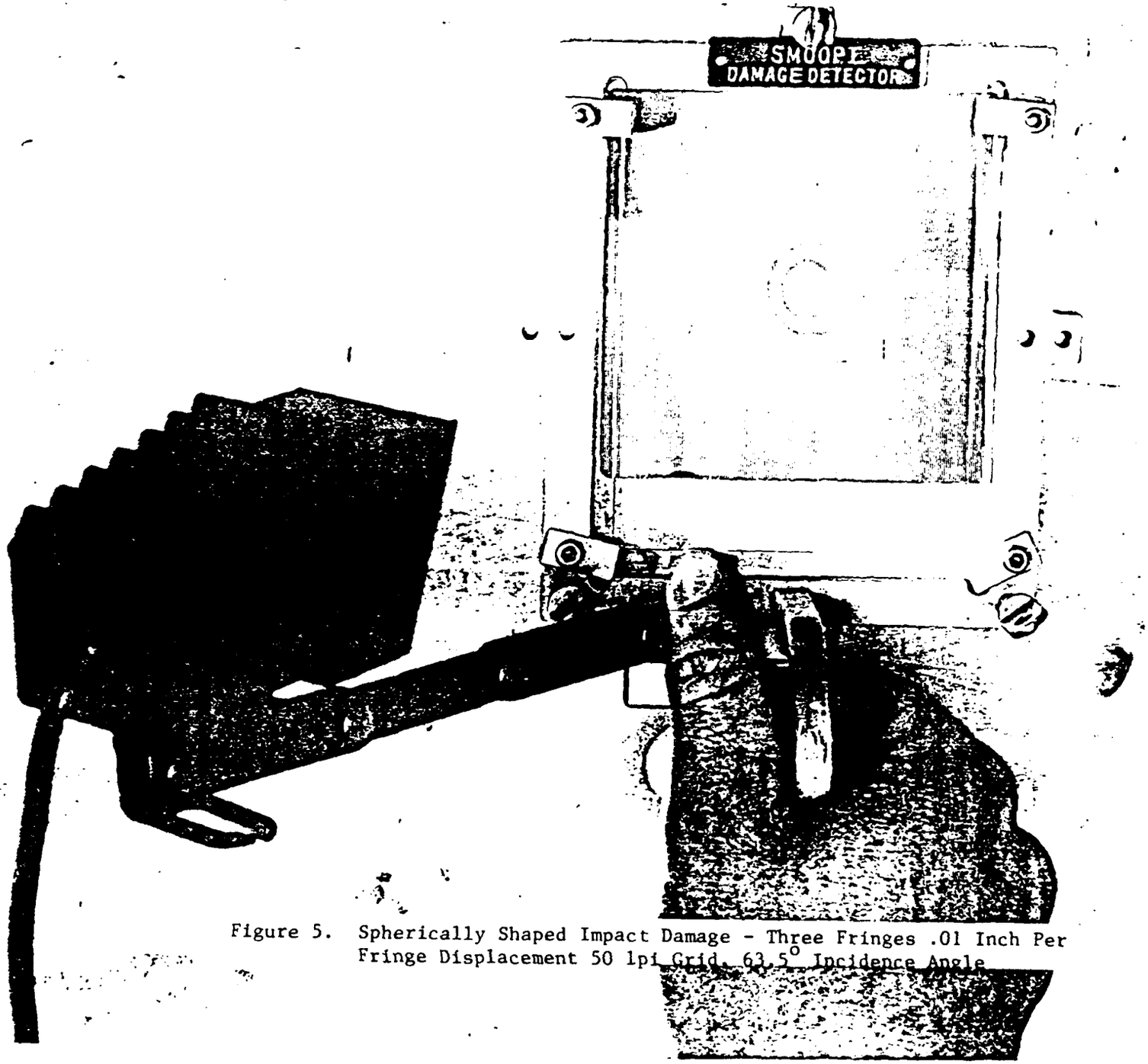


Figure 5. Spherically Shaped Impact Damage - Three Fringes .01 Inch Per Fringe Displacement 50 lpi Grid, 63.5° Incidence Angle

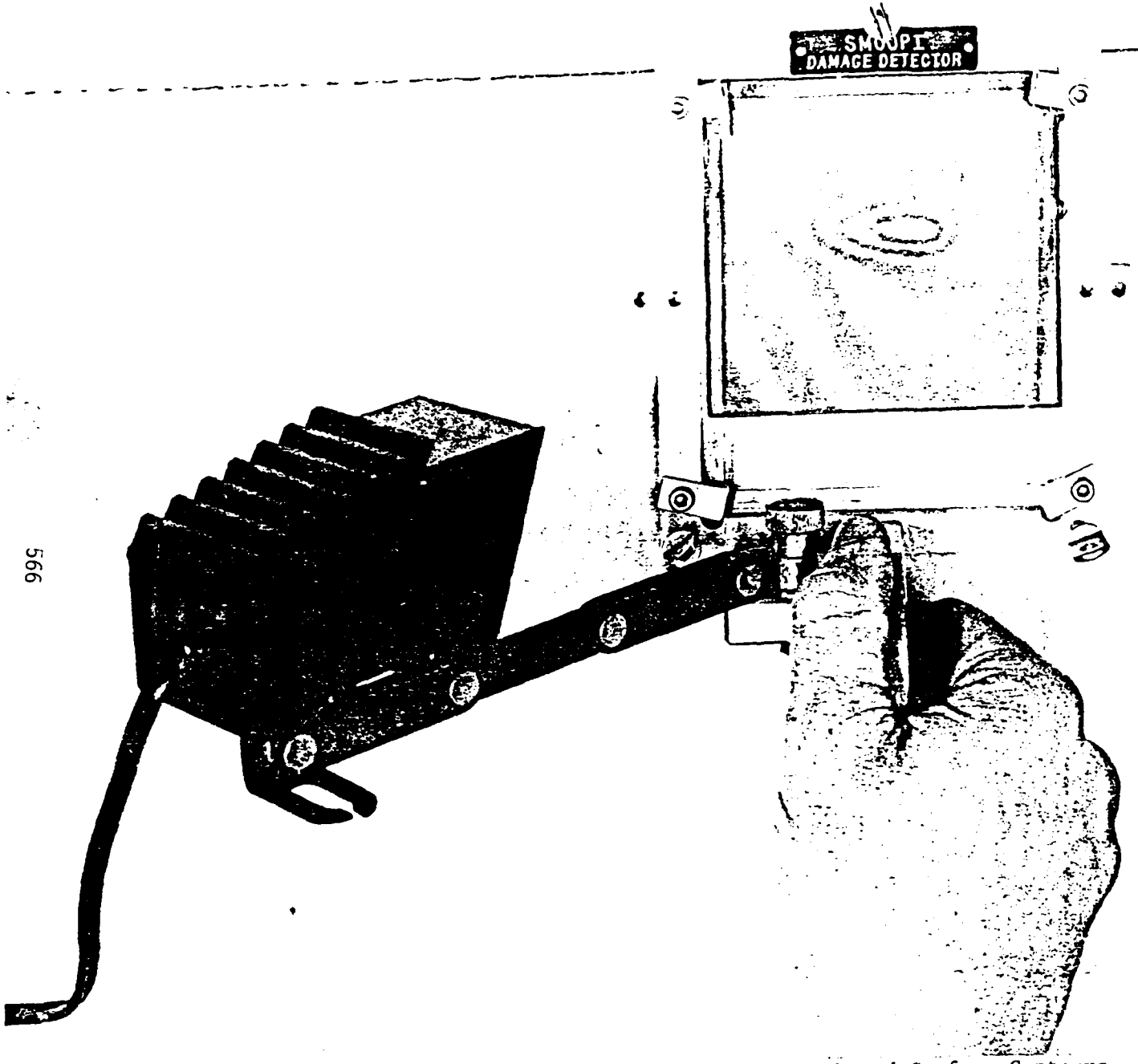


Figure 6. Three Fringes - Elliptically Shaped Surface Contours - 150 lpi Grid, 45° Incidence Angle, .0067 Inch Per Fringe Displacement Viewed After Impact

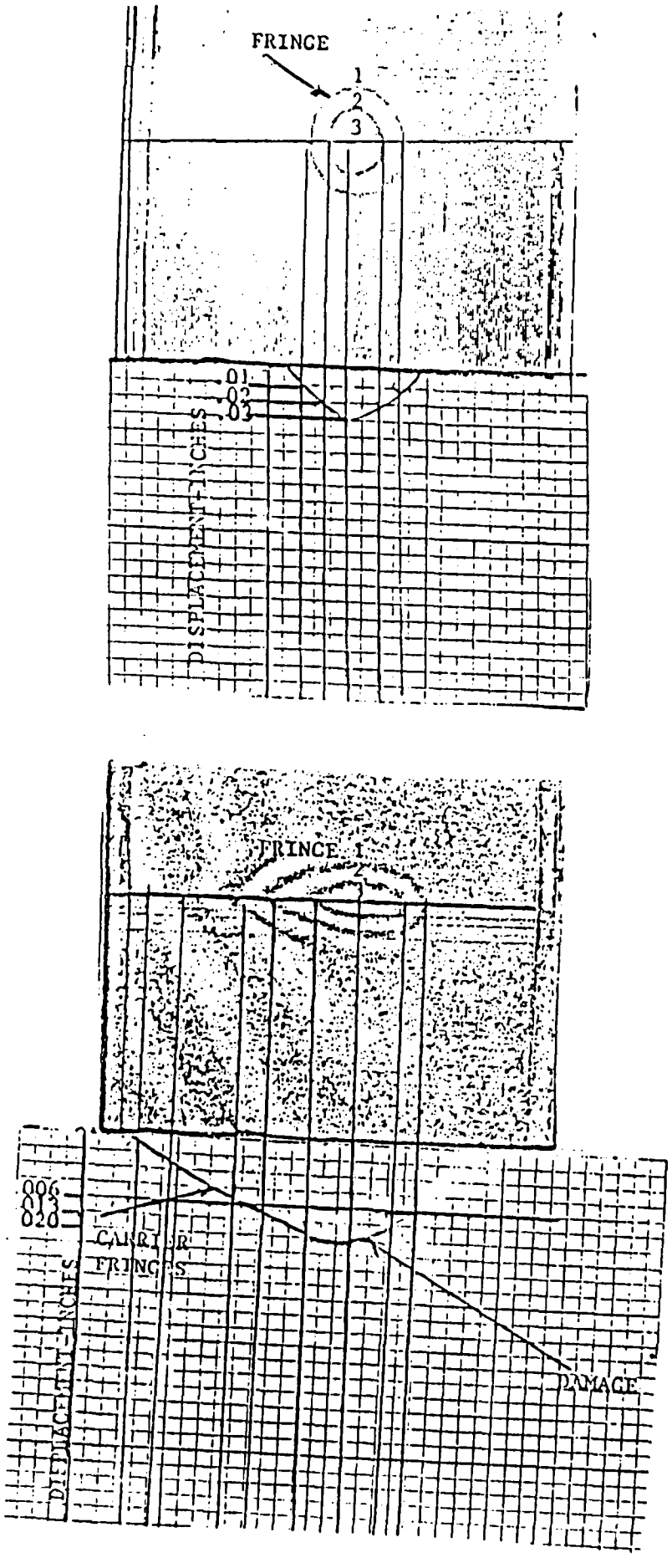


Figure 7. Damage Displacement Profiles for Figures 5 & 6

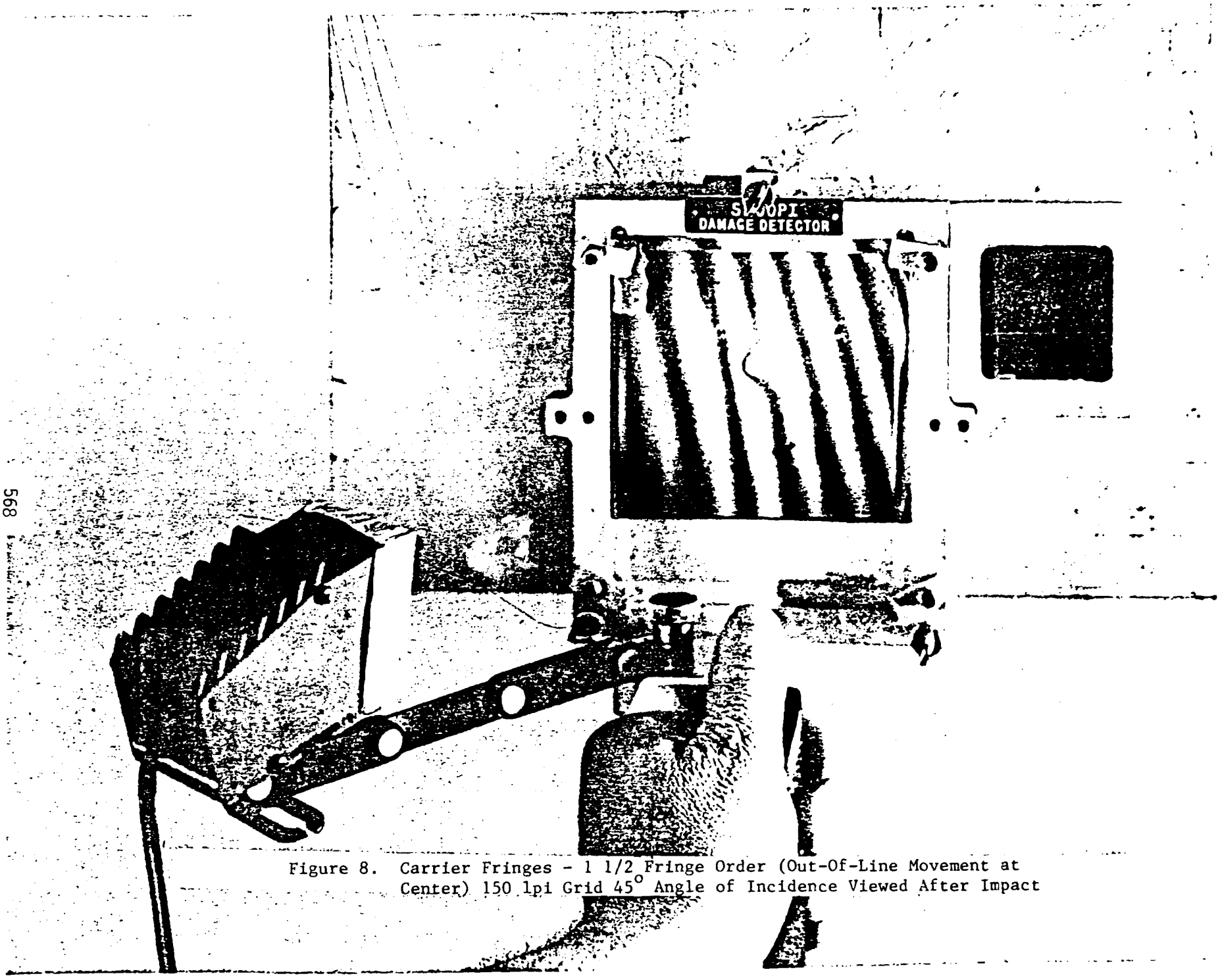
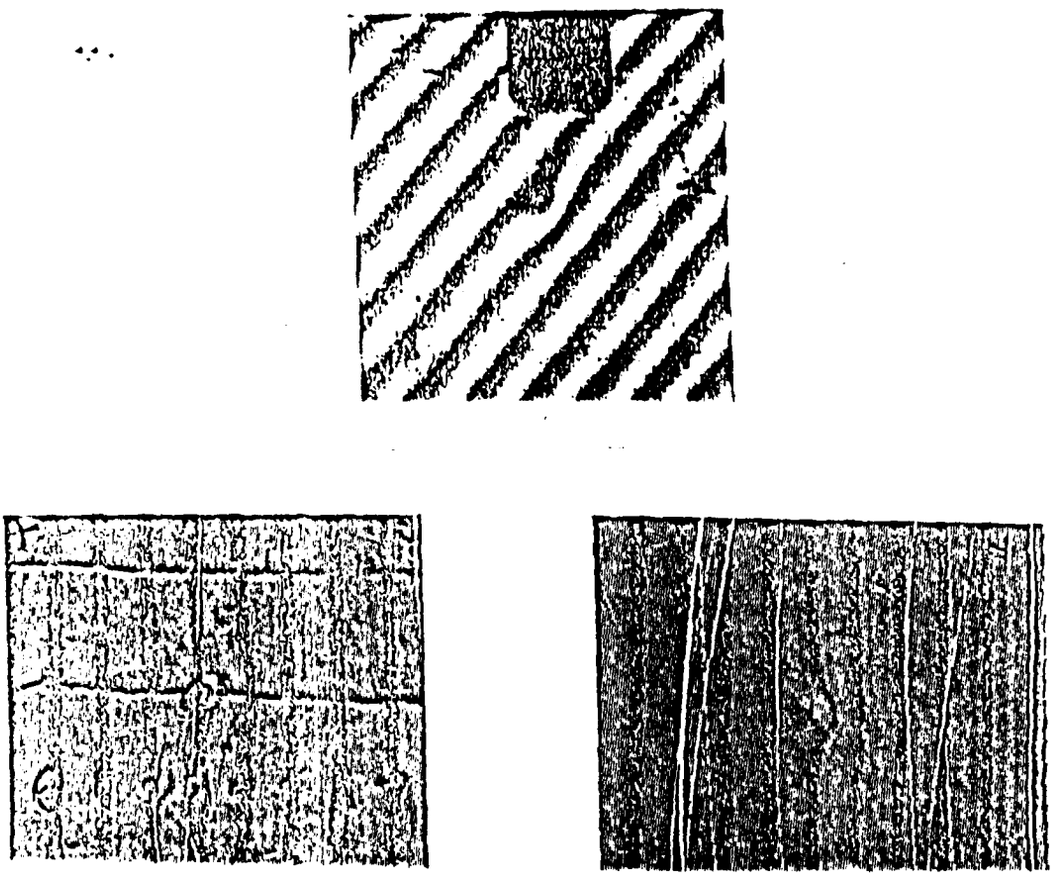


Figure 8. Carrier Fringes - 1 1/2 Fringe Order (Out-Of-Line Movement at Center) 150 lpi Grid 45° Angle of Incidence Viewed After Impact



Note: Out-Of-Plane Movement of Parallel Carrier Fringes - Visual Interpolation of Change in Fringe Spacing from 1/10 to 1/2 Fringe.

Figure 9. Grid-Bias Linear Carrier Fringes: 40-Ply GR/EP Specimen Viewed After Impact - 250 lpi Grid, 63.5° Angle of Incidence. Visual Sensitivity for 1/4 Fringe Interpolation Equals .0005 Inch.

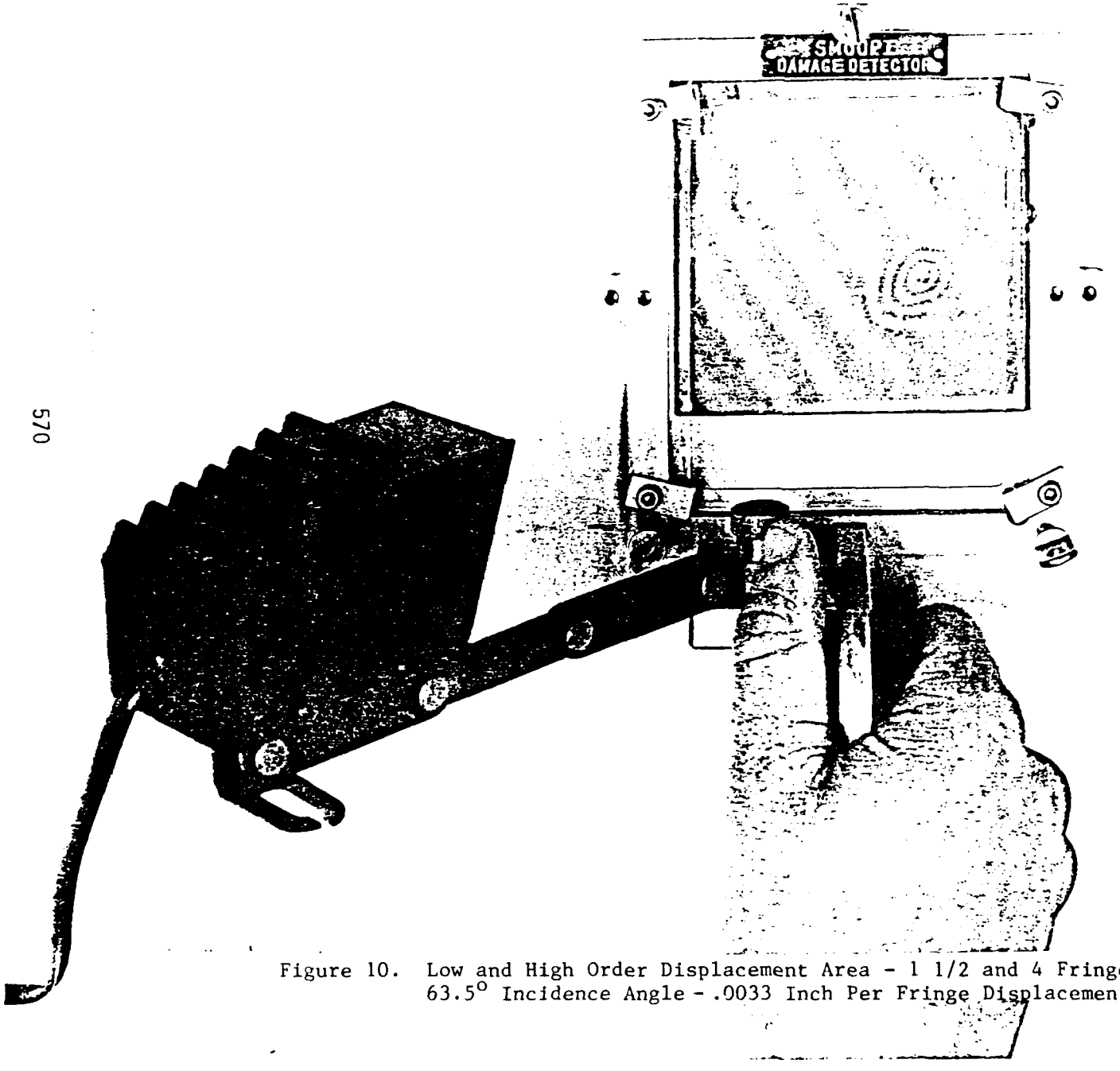


Figure 10. Low and High Order Displacement Area - 1 1/2 and 4 Fringes, 150 lpi Grid, 63.5° Incidence Angle - .0033 Inch Per Fringe Displacement Viewed After Impact

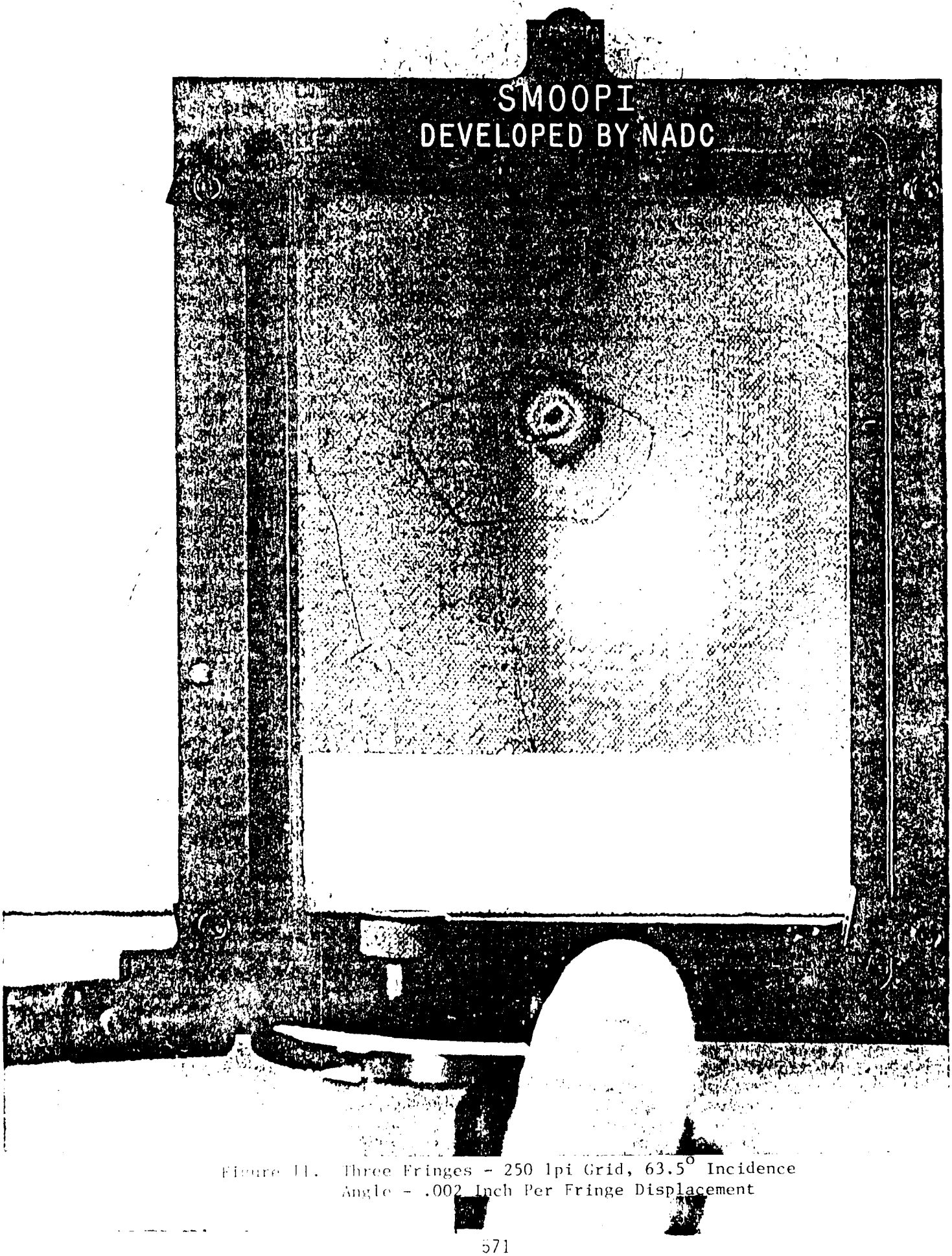


Figure 11. Three Fringes - 250 lpi Grid, 63.5° Incidence Angle - .002 Inch Per Fringe Displacement

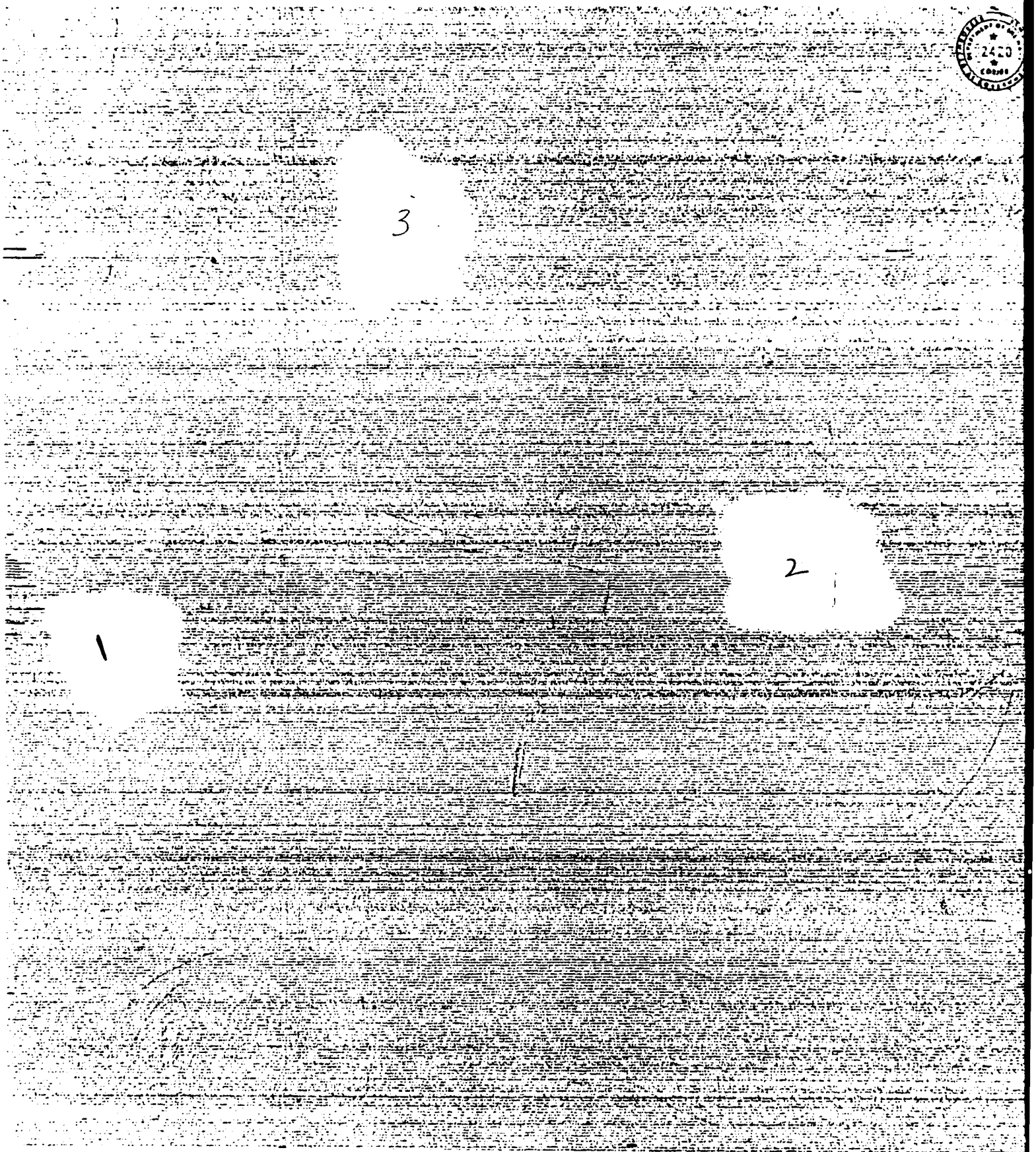


Figure 12. Ultrasonic C-Scan of Specimen in Figure 11

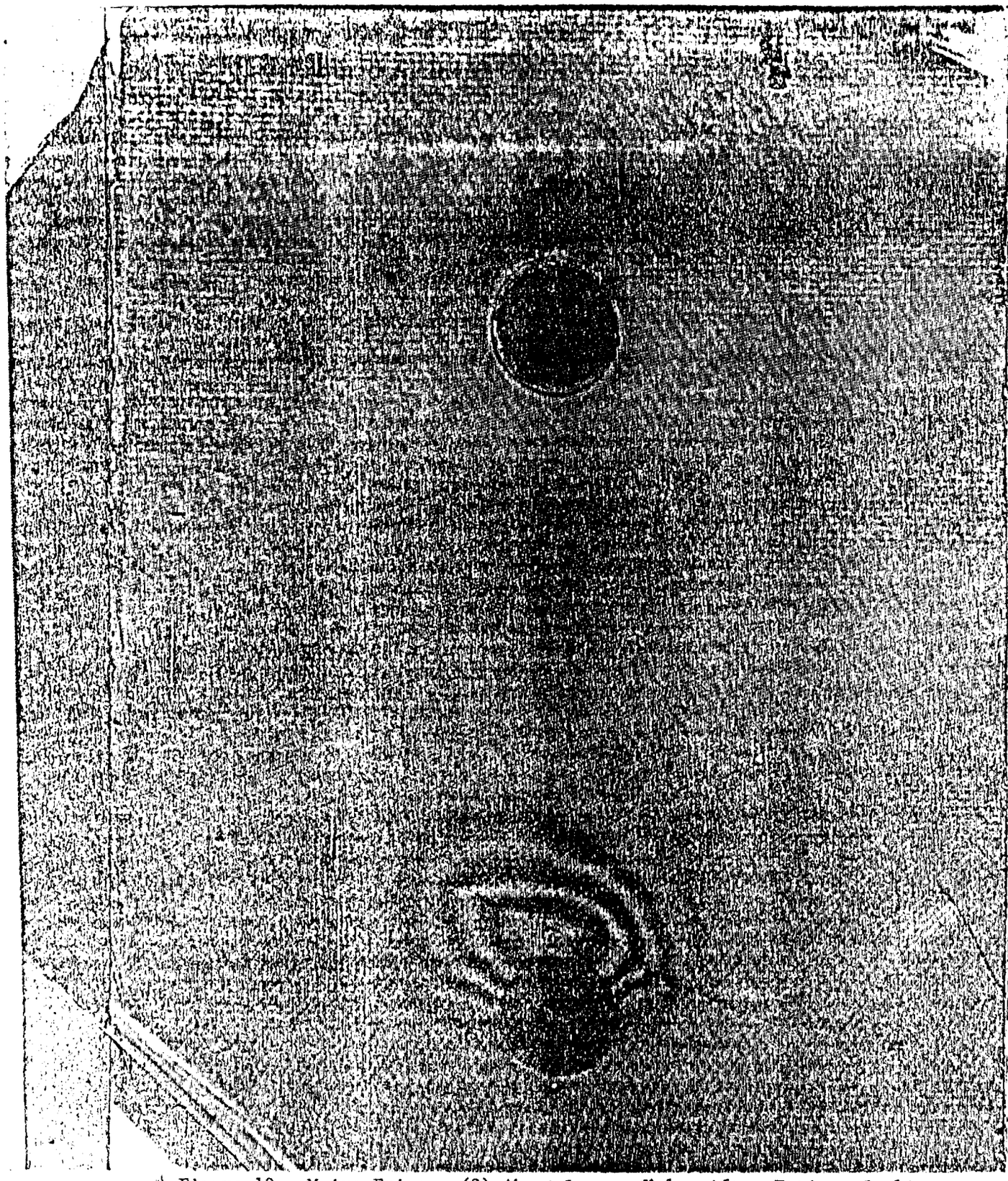


Figure 13. Moire Fringes (3) About Lower Hole, After Fatigue Cycling, Inspected with 500 lpi Grid; 25-Ply GR/EP Specimen; Out-Of-Plane Displacement 0.003 Inch

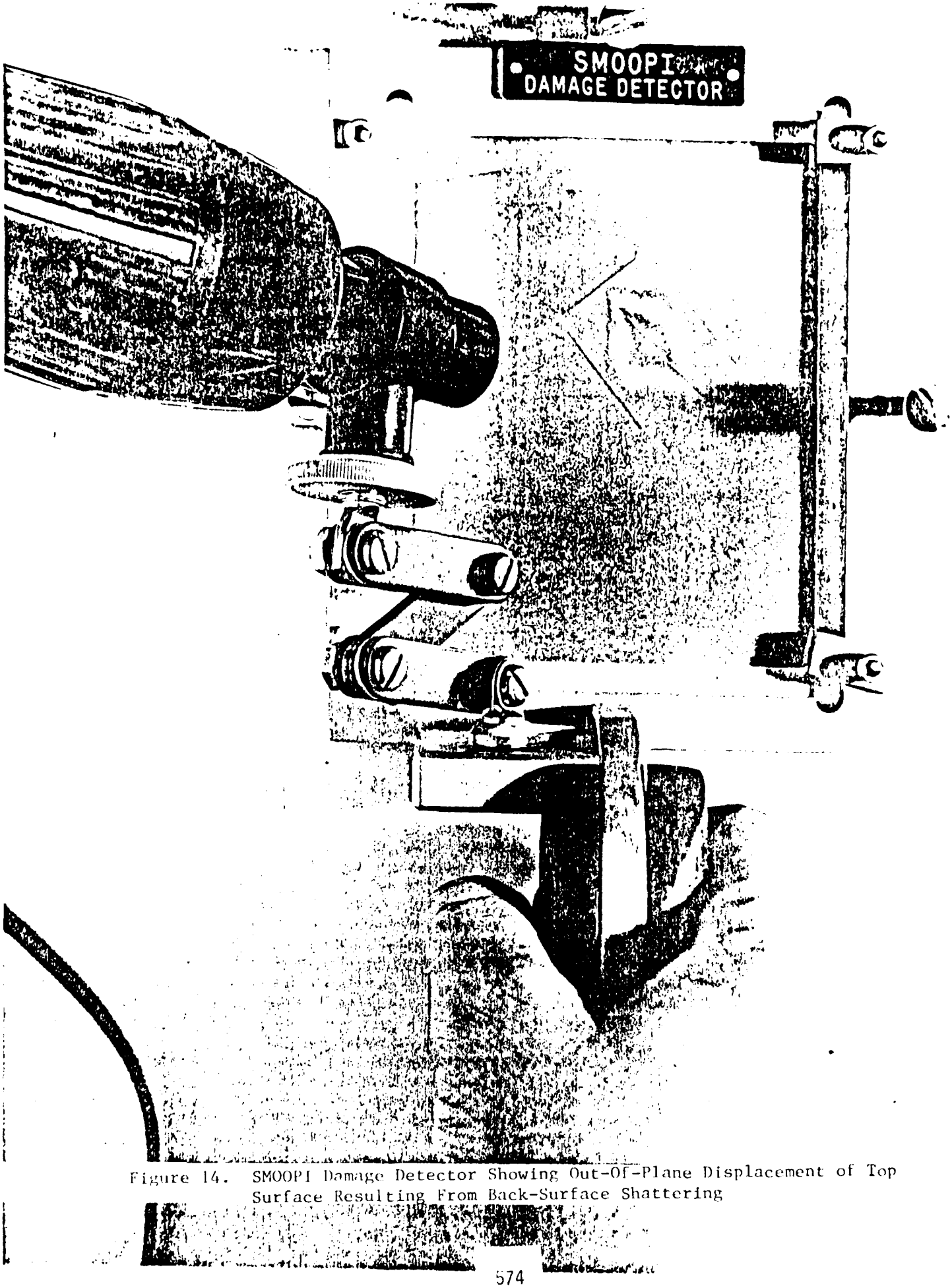


Figure 14. SMOOPI Damage Detector Showing Out-Of-Plane Displacement of Top Surface Resulting From Back-Surface Shattering

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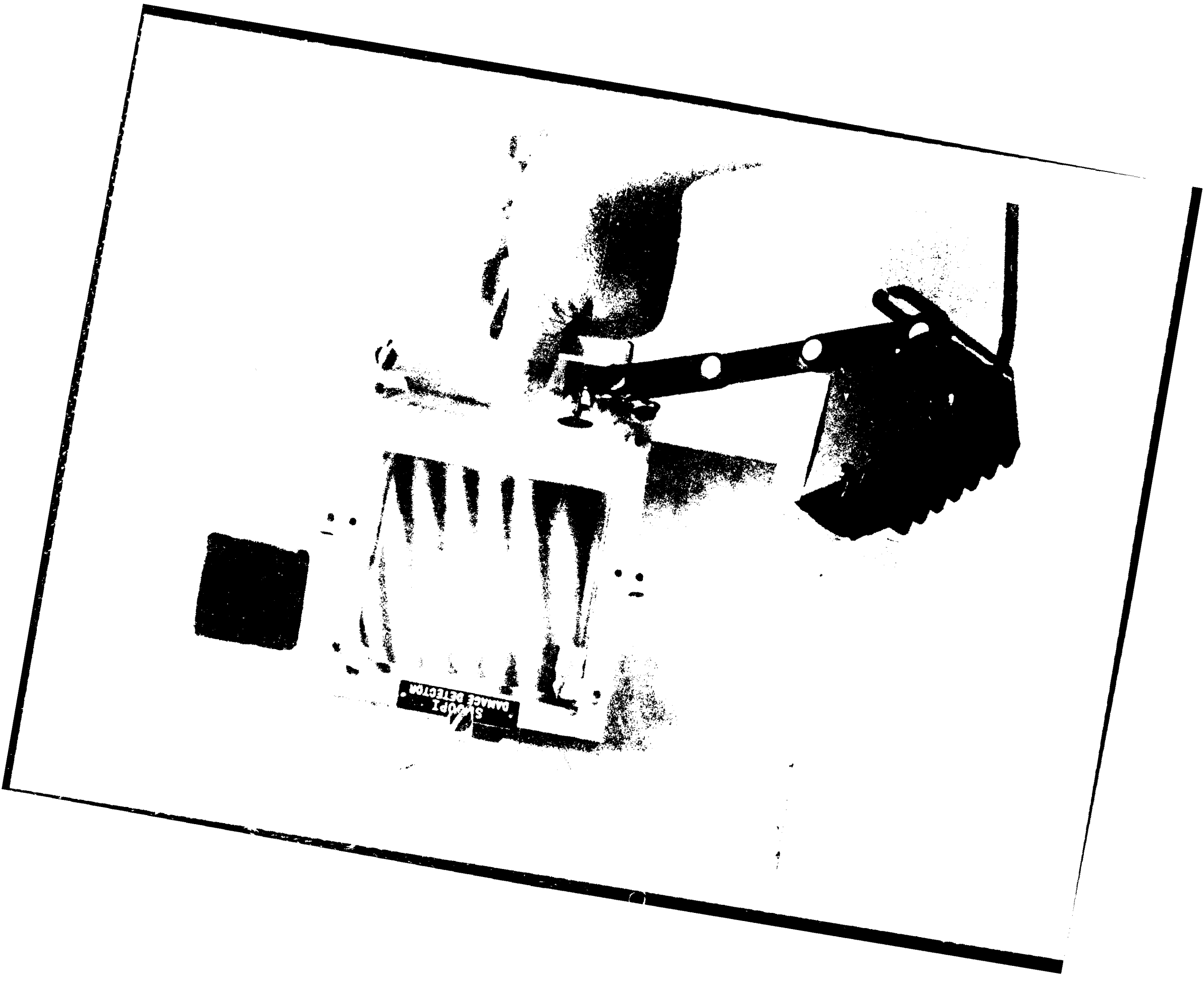
DAMAGE DETECTOR

SMOPI
DAMAGE DETECTOR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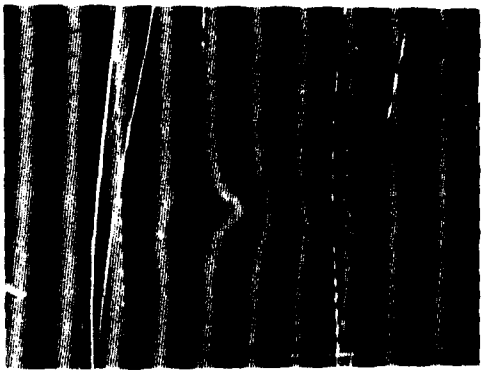

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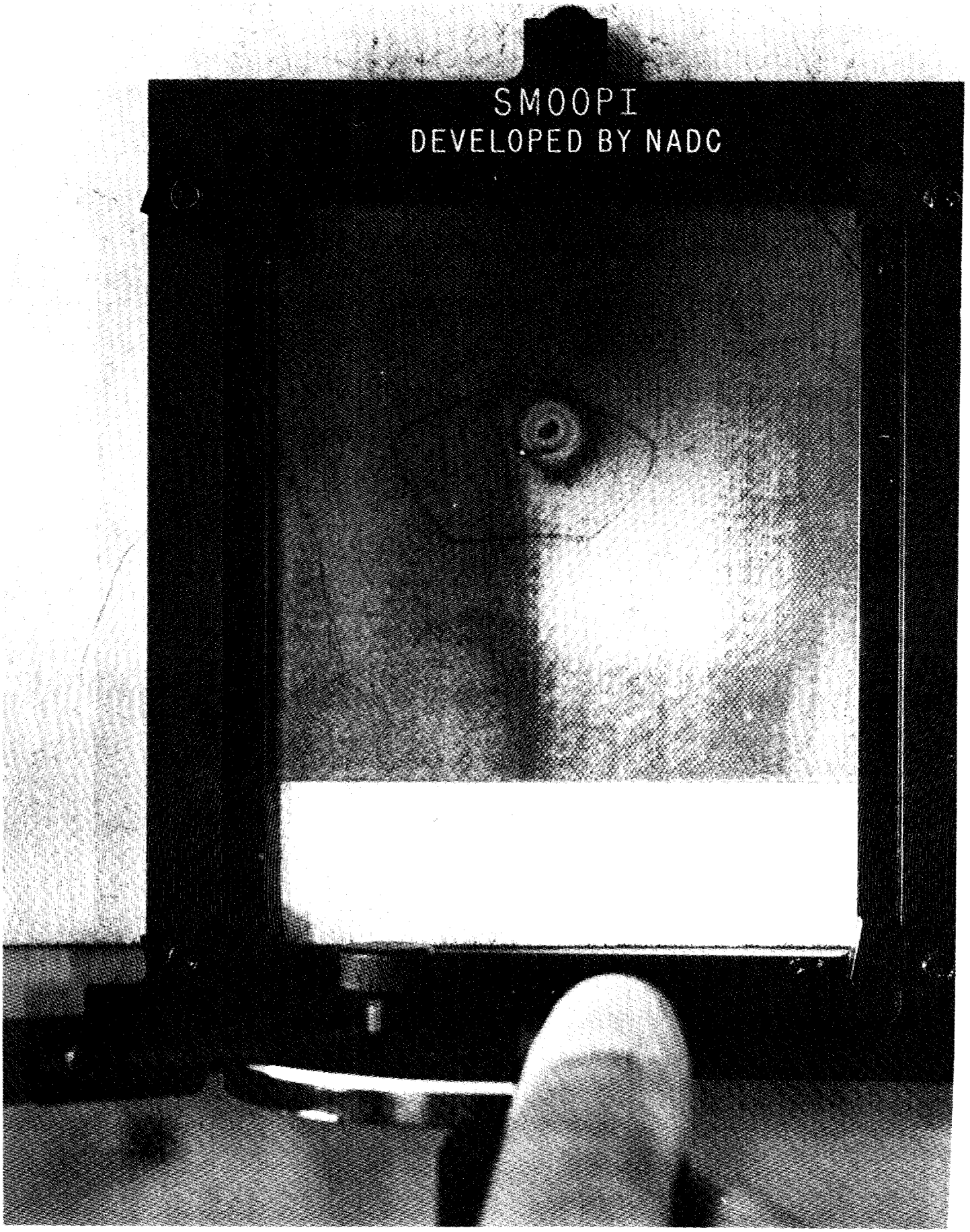
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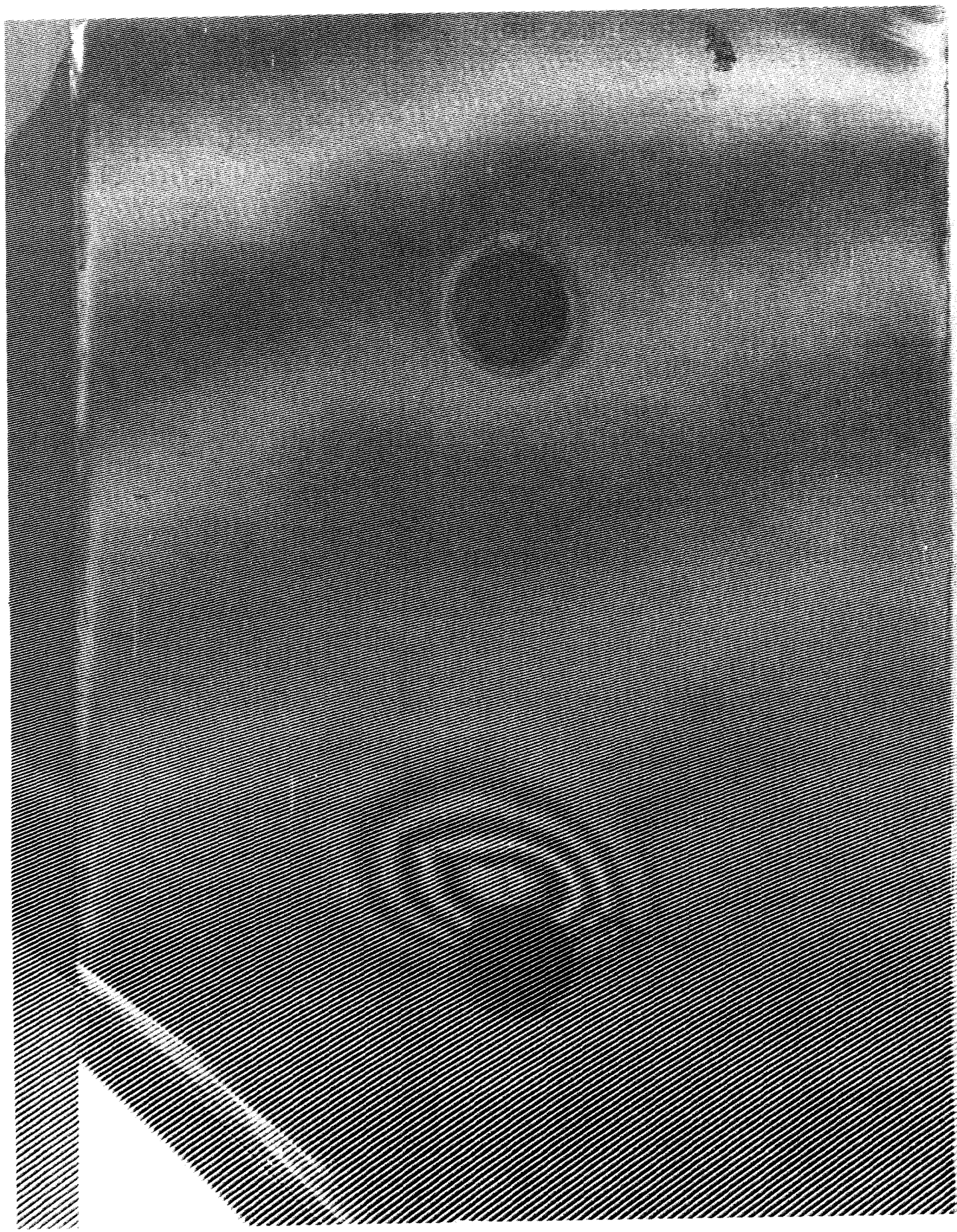
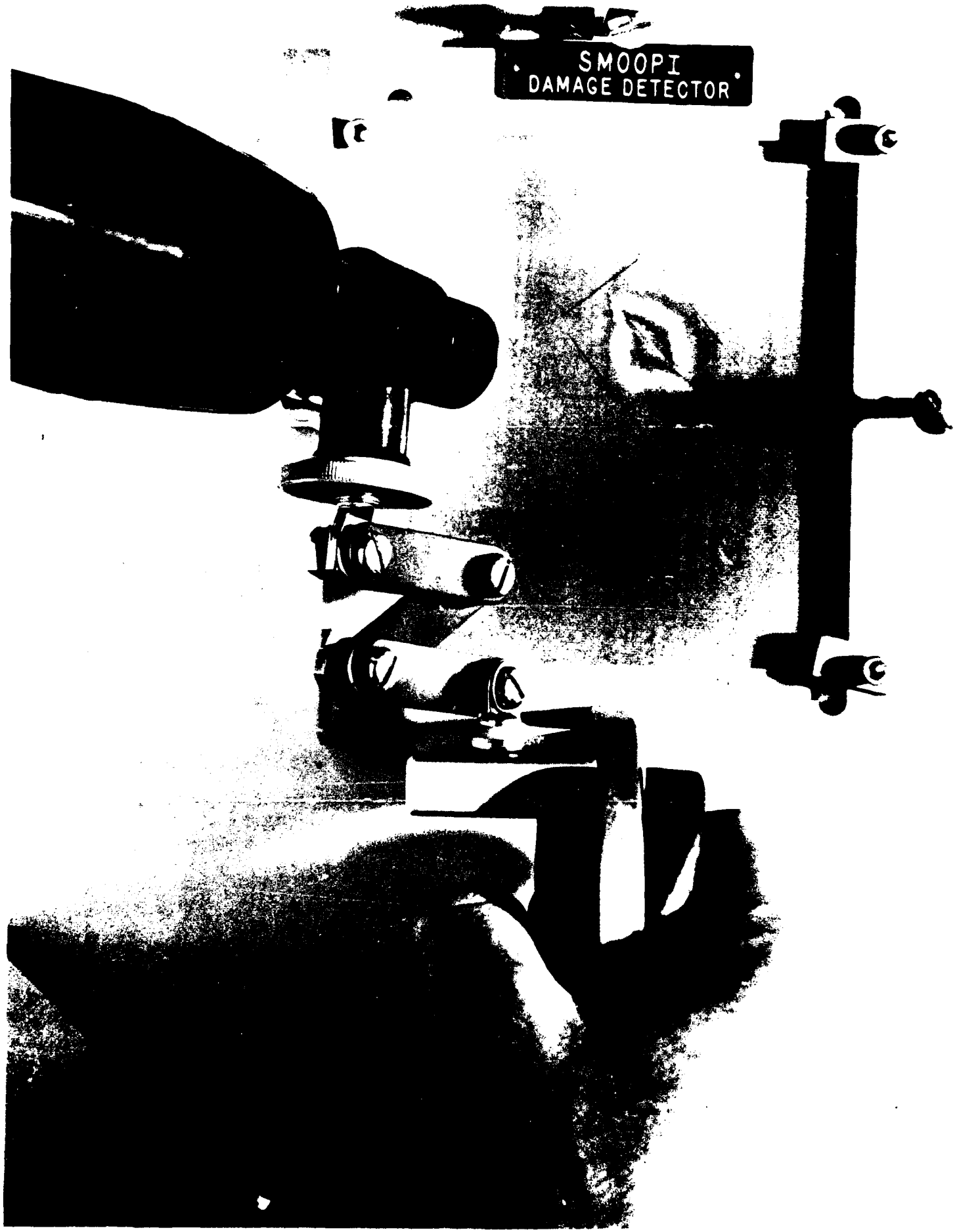



SMOOPI
DAMAGE DETECTOR

FATIGUE TECHNOLOGY INC.

CX SYSTEMS

Recent Advancements In Fatigue Enhancement Technology

Eric T. Easterbrook
Michael A. Landy
Leonard F. Reid

Presentation to:

1988 USAF Aircraft Structural Integrity Program Conference
Nov. 29th-Dec. 1st, 1988

The purpose of this presentation is to provide the conference with a short review of recent developments in fatigue enhancement technology. Two of these developments were in the R & D phase when they were presented to this conference about three years ago, the other had only been suggested as a possibility. They are all now fully systematized and in production use on USAF aircraft. All of the technology that will be discussed today is based upon advanced applications of the widely used and well-proven Cold Expansion systems, for fatigue enhancement of holes in metal structures.

FATIGUE TECHNOLOGY INC.

CX SYSTEMS

Outline

- Background of fatigue enhancement technology
- Life enhancement of bushed holes/ lugs (ForceMate)
- Life enhancement of stop-drilled holes (StopCrack)
- Life enhancement of non-round holes
- Summary of USAF applications

The presentation will begin with a brief review of the history of fatigue enhancement for holes in metallic structure, to show how we got to where we are today, and to focus our attention on the most important aspects of the technology. We will then take a close look at three of the more significant developments of this technology:

First, we will examine fatigue life enhancement for bushed holes using the ForceMate system - an alternative to traditional bushing installation systems. The ForceMate system also provides some significant manufacturing cost reductions as compared to traditional installation processes such as shrink fitting using cryogenic liquids.

Next, we will look at a method for enhancing the life of stop drilled holes called StopCrack. Although this technique has been in use for some time now, its effectiveness in thinner gauge materials has now been demonstrated, and thus its applicability is significantly wider.

Next, we will examine brand new technology for providing fatigue and crack growth life enhancement for unusual, non-round holes. Previously, FTI had always required a round hole in order to use our cold expansion systems. Now the life of existing non-round holes can be lengthened. We will describe this technology as well as methods for extending the life of non-round holes before they are produced.

Last, we will provide a quick summary of applications of cold expansion technology on USAF aircraft.

FATIGUE TECHNOLOGY INC.

CX™ SYSTEMS

Evolution of Hole Fatigue Enhancement

1940's-1950's	Mandrelizing, broaching
1960's	Improved materials, surface treatments
1970's	Interference fit fasteners
1980's	Sleeve cold expansion

Modern hole fatigue enhancement began in the 40's and the 50's using mandrelizing and broaching, along with the mistaken belief that the so-called "perfect" holes produced by these methods would have greater life. In fact, just the opposite was true. Hole scoring and rifling actually degraded fatigue performance.

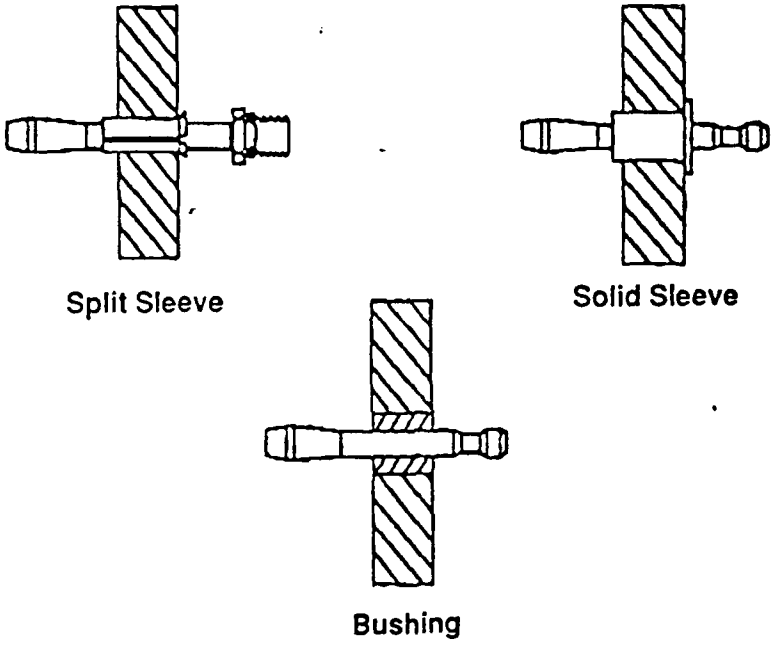
In the 60's, improved materials arrived. These "improved" materials, including 7075-T6, 7079-T6, 7178-T6, soon proved to be highly crack sensitive. Surface treatment methods, such as shot peening or ballizing, were used without much success for fatigue improvement in the holes. In the 70's, interference fit fasteners were introduced in an attempt to get the industry out of the trouble they had gotten themselves into with the "improved" materials.

In the 80's, cold expansion and, in particular, split sleeve cold expansion, have become the most widely used fatigue enhancement systems in the world. During this presentation we will be demonstrating some of the aspects of this technology that have resulted in its wide acceptance.

FATIGUE TECHNOLOGY INC.

CX™ SYSTEMS

High Interference Cold Expansion Methods



High interference cold expansion is generally performed using one of three systems. Specific details of the processing will be shown later.

1. Split Sleeve Cold Expansion: The most widely used fatigue enhancement system. It's used mainly for fastener and other plain holes.
2. Solid Sleeve Cold Expansion: Used much less due to process imitations, but effective as a hole resizing method.

3. ForceMate: Used as an alternative to traditional bushing installation methods.

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Life Enhancement Using Sleeve Cold Expansion

- Sleeve cold expansion creates a massive zone of compressive residual stress.
- Residual compressive stresses extend fatigue and crack growth lives.
- Reliable fatigue enhancement is maximized through systematic approach.

Sleeve cold expansion creates a massive zone of compressive residual stress. The magnitude of the compressive stress zone is at least two-thirds of the material's tensile yield strength and extends radially outward from the edge of the hole to a distance at least equal to the radius of the hole, and more typically, to about one diameter. Since it's unusual for applied cyclic tensile stresses to exceed this value, the hole is effectively shielded from tensile loading. From a fracture mechanics viewpoint, the effective stress intensity factor is significantly reduced. Typically, minimum life improvement factors of 3:1 (cold expanded life:non-cold expanded life) have been shown for both fatigue and crack growth, with higher factors common. Reliable fatigue and crack growth life enhancement is maximized through a systematic approach to processing, involving technology, process procedures, quality assurance, and tooling.

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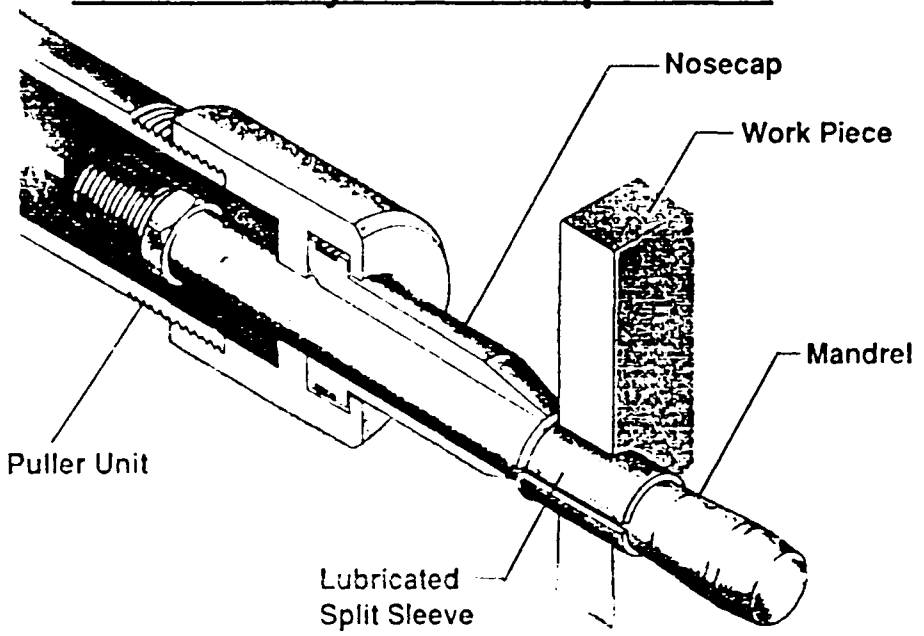
Basic Cx System Components



Typically, high interference cold expansion is accomplished by drawing an oversized mandrel through a fastener hole, using a prelubricated stainless steel split sleeve. The sleeve protects the hole from damage, insures the hole expansion is in a radial direction, minimizes the pull force required to cold expand the hole, and allows one sided (blind) processing. Optimal fatigue performance is achieved when the hole is expanded at least 3% for aluminum or mild steel, and at least 4.5% for titanium and high strength steel.

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	System	Material
SsCx	Split Sleeve Cold Expansion	Aluminum/Titanium/Steel
Cx2s	Cold Expansion to Size	All
CsCx	Countersink Cold Expansion	Aluminum
FmCx	**ForceMate** System (with McDonnell-Douglas)	All
ScCx	**StopCrack** Enhanced Stop Drill Repair (ESDR)	All

Fatigue Technology's family of Cold Expansion systems includes the following:

1. Split Sleeve Cold Expansion (SsCx): The most widely used joint fatigue enhancement process in the world today. It offers the most flexibility of any cold expansion method in terms of hole sizes, materials, and application suitability.

2. Cold Expansion to Size (Cx2s): Reduces cold expansion processing to only one step by providing the required final hole diameter after cold expansion, eliminating the need for final reaming.

3. Countersink Cold Expansion (CsCx): Provides the capability to cold expand, or cold expand to size, previously countersunk holes.

4. ForceMate (FmCx): An alternative to traditional bushing installation processes, providing consistent high interference, fatigue/crack growth life enhancement, and reduced bushing installation cost.

5. StopCrack (ScCx): Provides a means to enhance the life of the stop drill repair technique.

All of the above systems have found wide acceptance throughout aerospace and other industries.

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CX™ SYSTEMS

Advanced Applications

Materials:

- -Nickel based alloys
- -Ultra high strength steel
- -Austenitic stainless steel

Configurations:

- -Non-round cutouts
- -Slots
- -Stop drill holes

Unusual Structures:

- -Thick tubes
- -Cylinder heads
- -Wind turbines
- -Bridges

Cold expansion systems have been in use throughout aerospace and other industries for over eighteen years. In the last few years, however, these processes have found new uses in advanced applications. Many of these applications are in structure previously thought to be unlikely to benefit from cold expansion. Some of these newer applications include:

Advanced materials: Cold Expansion systems are now being used routinely in materials such as nickel based alloys, ultra high strength steel (up to 300 KSI), and austenitic stainless steels.

Configurations: Cold expansion systems are now being used to provide life enhancement for non-round holes, slots, and stop drill holes.

Unusual structures: Cold expansion systems are now benefitting any non-aerospace structures, including thick tubes, cylinder heads, wind turbines, and bridges.

Some of these advanced applications are the primary focus of this presentation.

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CX SYSTEMS

Benefits Of Cold Expansion

- Improves fatigue life
- Retards crack growth
- Insensitive to manufacturing variables
- Inexpensive
- Cost effective
- Reliable and proven systems

Before discussing the specific advancements it worth summarizing some of the most important and well proven benefits of cold expansion:

1. It improves the fatigue and crack growth life of structure,
2. It is insensitive to most manufacturing variables,
3. It is inexpensive and, more importantly, has demonstrated cost effectiveness, and
4. It is reliable.

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CX SYSTEMS™

ForceMate™

A System for High Interference Bushing Installation and Fatigue Life Enhancement

The first advancement in fatigue enhancement technology to be discussed is the ForceMate System. ForceMate is an alternative to traditional bushing installation methods such as press fit or shrink it using cryogenic fluids.

FATIGUE TECHNOLOGY INC.

ForceMate™

Background

- ForceMate concept originally developed by McDonnell Aircraft Company for production of F/A-18 Aircraft

- FTI engineered and developed FmCx System as an alternative to traditional bushing installation processes.
  - Engineering handbook (EH-3)
  - Standard system
  - Special applications

The ForceMate concept was originally developed by Coughenour and Hoeckleman of McDonnell-Douglas Corp for specific beryllium-copper bushing installations in the F/A-18. FTI engineered and developed the concept into the ForceMate System, as a general use alternative to the traditional bushing installation processes. The system now includes:

1. An Engineering Handbook providing all engineering, manufacturing, and tooling requirements,
2. A Standard System which is sized to be a direct replacement for NAS, or equivalent, bushing installations,
3. A Special System that provides ForceMate technology for unique applications.

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ForceMate™

Traditional Bushing Installation

- Inconsistent interference fit
  - --corrosion/fretting
  - --walk-out/complex bushing design
- Manufacturing problems
  - --labor intensive
  - --hole/bushing damage
- In-service problems
  - --warranty/product quality
  - --costly recovery

The ForceMate System minimizes or eliminates many of the problems associated with traditional bushing installation processes, such as press fit or shrink fit using cryogenic fluids. These problems include:

1. Inconsistent interference fit allows corrosion inducing products to enter the bushing to hole wall interface, and also leads to bushing movement within the hole causing fretting damage. Bushing retention is a problem and often results in complex pinned bushing designs.

2. Manufacturing of traditional bushed holes is labor intensive and hole/bushing damage can occur.

3. In-service problems result with resulting warranty and product quality implications. Costly recovery programs may follow.

FATIGUE TECHNOLOGY INC.

ForceMate™

The FmCx System Provides:

- Consistent high interference fit
  - –as high as .000 to .010 inches
  - –fretting/corrosion are minimized
- Manufacturing cost reduction
  - –accomplished in seconds
  - –safe
- Fatigue enhancement
  - –fatigue initiation sites minimized
  - –synergistic effect from cold work and high interference

FmCx eliminates custom sizing of bushings to holes.

The ForceMate System eliminates virtually all of the problems associated with the traditional bushing installation processes, and provides reliable, consistent fatigue life improvement. The benefits include:

1. Consistent high interference fit, as high as .010 inch, thereby minimizing fretting/corrosion and maximizing bushing retention.

2. Manufacturing cost reduction. Installation is accomplished in seconds and the use of cryogenic fluids (e.g., liquid nitrogen) is eliminated.

3. Consistent fatigue and crack growth life improvement results from the synergistic effect of cold expansion and high interference.

4. Elimination of custom sizing of bushings to holes.

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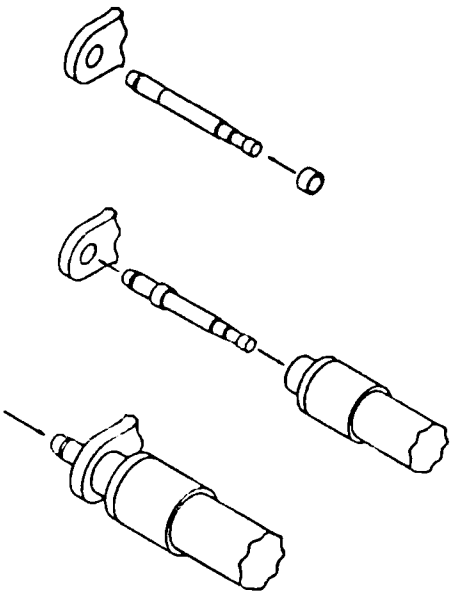
ForceMate™

System Overview

Place insert onto mandrel.

Place mandrel into puller unit. Place mandrel and insert into workpiece (one sided operation).

Install insert and cold expand hole by pulling mandrel through insert (using hydraulic puller unit).



The ForceMate System is extremely easy to implement. A specially sized ForceMate insert, with a proprietary dry film lubricant on the inside surface is placed on a tapered expansion mandrel. The attachment end of the mandrel is placed into a hydraulic puller unit; the mandrel/insert is placed into the hole; and the puller unit is activated to pull the mandrel through the insert. The expansion of the insert by the mandrel cold expands the hole material while the insert is simultaneously installed with high interference. A subsequent reaming operation is performed to size the insert inside diameter to the final size and to remove lubricant residue. If the post-ForceMate insert inside diameter meets application requirements, the final reaming operation may be eliminated (FmCx to size).

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Current Applications

Company	Application	Insert Material	Lug Material
Aeritalia	Tornado Wing Refueling Port Holes	13-8 PH	7075-T73
Beech Aircraft	Wing Attach Fitting	4130 Steel	7075-T6
Bell Helicopter	V-22 Rotor Hub Spindle Engine Links	15-5 PH 17-4 PH	13-8 PH
Boeing	B-52 Nacelle Strut Attachment	4340 Steel	7079-T6
British Aerospace	A320 T-45	BeCu BeCu	Ti 6AL-4V 2014-T6
DeHavilland	Dash 8 Series 300	431 Stainless	7175-T73
McDonnell Douglas	F-18	BeCu	Al/Ti
USAF	T-37	4340 Steel	7075-T73
USN	T-2	4130 Steel	7075-T76

Current production applications of the ForceMate System are shown. Note the wide variety of combinations of insert and lug materials.

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Current Investigations

Company	Application	Insert Material	Lug Material
BCAC (MR&D)		Al-Ni-Br	Al, Ti, Steel
DeDietrich	Portable Bridge	17-4 PH	7020-T6
FTI	R & D	Al-Ni-Br 4340 Be-Cu	7075-T6 Ti 6Al-4V
General Dynamics	F-111	D6AC Steel	D6 AC 220-240KSI
McDonnell	AH-64 Main	4130	7075-T73

As of the date of this presentation, a large number of corporate R & D programs are being performed, investigating the benefits of the ForceMate System. Again note the wide variety of insert and lug combinations being studied, and the international acceptance of this process.

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Current Investigations

Company	Application	Insert Material	Lug Material
Aerospatiale La Courneve	Helicopter Blade Attachment	4340	Steel & Composite
Allison	TF-41 Disk Blade Attachment	Jethete (Cr-Mo)	Jethete (Cr-Mo)
British Aerospace	A330/340 Evaluation	BeCu	GAL44Ti
Cessna	T37 Evaluation	4130 Steel	7075-T76
Dassault	Test Program	Inconel 718	2214-T6
Rockwell International	B-1B Attachment Links	13-8 PH	7075-T73
Pratt & Whitney	TF-33 Compressor	AMS 5667	AMS 5667

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ForceMate™

FmCx Standard System

- -- Engineering Handbook provides all engineering, manufacturing, and tooling requirements
- -- Equivalent to NAS
- -- Sized by customary bore diameters (.187" to 1.250")
- -- Complete tooling system
- -- Common insert alloys (4340, Be-Cu, ANB, SS)

The ForceMate Standard System makes it easy to implement the process on a production basis. The system, equivalent in inside diameters to the NAS standards for press fit bushings, is completely specified in FTI Engineering Handbook, EH-3. The tooling is systematized for ease of use, similar to other Cold Expansion systems and all required components are provided (i.e., cutting tools, inserts, mandrels, gages, puller units/PowerPaks). Inserts are available in all commonly used aerospace alloys (4340, Be-Cu, ANB, SS).

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ForceMate™

Special Designs And Other Applications

- Point design configurations
- Holes in composites
- Oversized holes, to resize hole to original diameter
- Access panel mounting holes, to maintain consistency of hole diameters
- Transfer holes through bulkheads, ribs, etc., to facilitate hose and wire bundle routing

The ForceMate System includes special design applications and uses other than the traditional lug or lug-type configurations. These include:

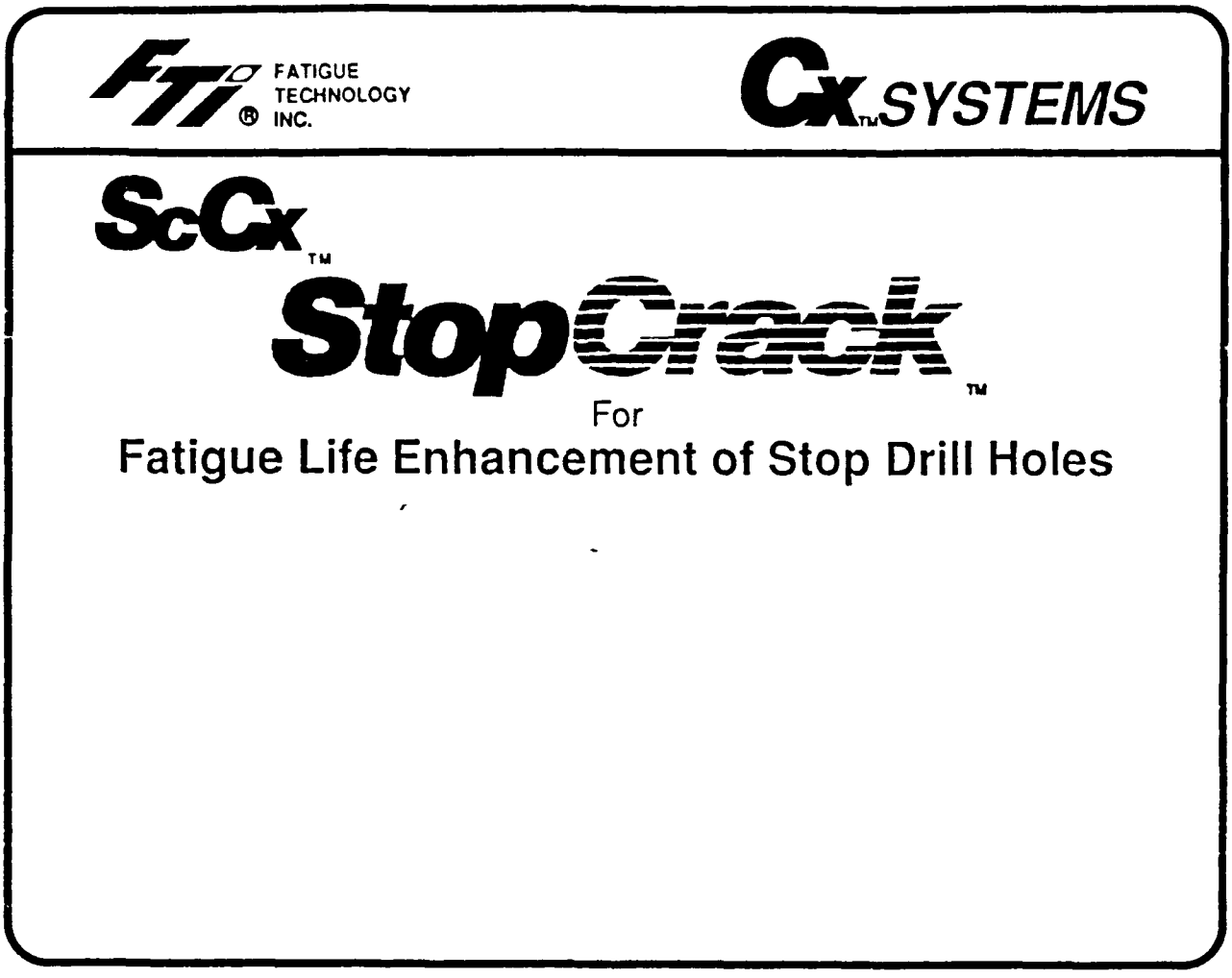
1. The ForceMate Special System which permits point design use of the process for unique configurations.

2. Fastener holes in composites. The ForceMate System allows the installation of an insert into a hole in a composite. Process parameters can be tailored so as to ensure extremely tight fit of the insert with minimal local damage. Testing has verified that such an installation provides life improvement of the hole while allowing the use of a removable fastener.

3. Repairing oversized holes, to resize holes to original diameters.

4. Maintaining consistency of hole diameters in access panel mounting holes.

5. Life improvement of transfer holes through bulkheads, ribs, etc., used to facilitate hose and wire bundle routing.



The second advancement to be discussed is the StopCrack System, used to provide fatigue or crack growth life enhancement for the traditional stop drill repair process.

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StopCrack™

Problems:

- Stop drilling of cracks has traditionally been used as an interim structural repair.
- Cracks can, and often do, reinitiate from other side of stop drill hole.
  - -- Stress concentration remains high.
  - -- Missed crack tip.
- Consequences of crack reinitiation include:
  - -- More complex repair.
  - -- Increased maintenance costs.
  - -- Aircraft downtime.
  - -- Safety.

Stop drilling of cracks has traditionally been used as an interim repair throughout the aerospace industry. The technique is based upon using a hole to remove and, in effect, blunt the crack tip. The fact is, however, that cracks can, and often do, reinitiate from the other side of the stop drill hole. This is due to the stress concentration of the stop drill geometry remaining high and, most importantly, the crack tip is frequently missed during the stop drill process. Some obvious consequences of crack reinitiation include:

1. More complex repair required,
2. Increased maintenance costs,
3. Aircraft downtime, and
4. Safety concerns.

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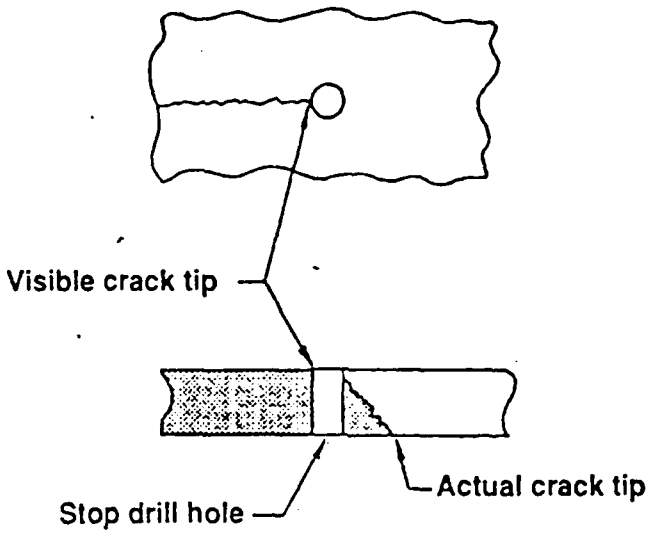
StopCrack™

Missed Crack Tip Problem



Missing the crack tip during stop drill repair is the most significant problem associated with the procedure. The cause of this problem is illustrated. Crack tips do not grow as straight lines perpendicular to the surface. They typically grow obliquely or, more frequently, they tunnel with the actual crack tip far beyond the visible crack on the surface. This makes locating the actual crack tip difficult or, in some cases, impossible.

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StopCrack™

Solution

- Use proven split sleeve cold expansion system to enhance stop drill repair process (StopCrack).
- Used as improved interim repair until depot level repair/rework can be performed.
- Applicable to materials as thin as .030 inch.

The solution to these problems associated with traditional stop drill repair is to use the proven split sleeve cold expansion system to enhance the stop drill repair. The increase in life of the repair has been shown to be significant, even if the crack tip is missed. Use of the Enhanced Stop Drill Repair (StopCrack) System insures that cracks will not reinitiate until the proper depot level repair/rework can be performed. The StopCrack System is applicable to materials as thin as .030 inch.

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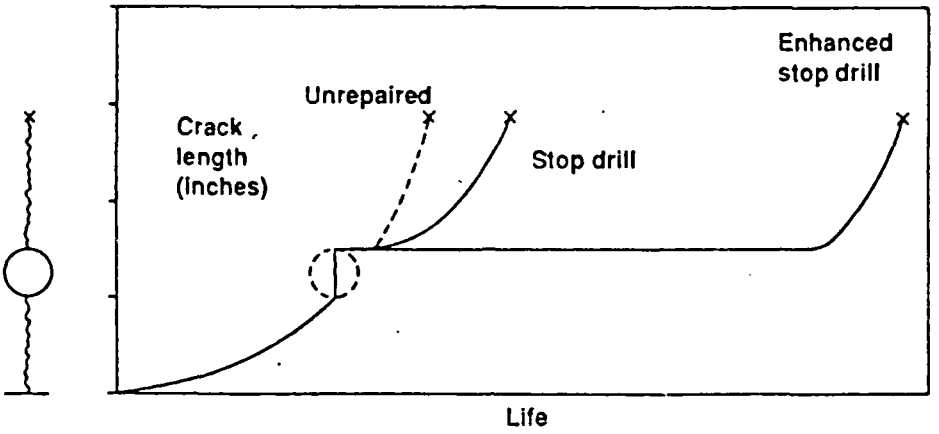
Crack growth of stop drilled hole



The crack growth behavior of a conventional and enhanced stop drill repair can be presented graphically. A crack forms and grows. After discovering the crack, a stop drill repair is performed. If the crack tip is missed, the crack growth life will be no better than that of the unrepaired structure. If the stop drilling does remove the crack tip, the increase in life can be minimal. If the stop drill hole is cold expanded, the increase in life has been shown to be significant, even if the crack tip was missed.

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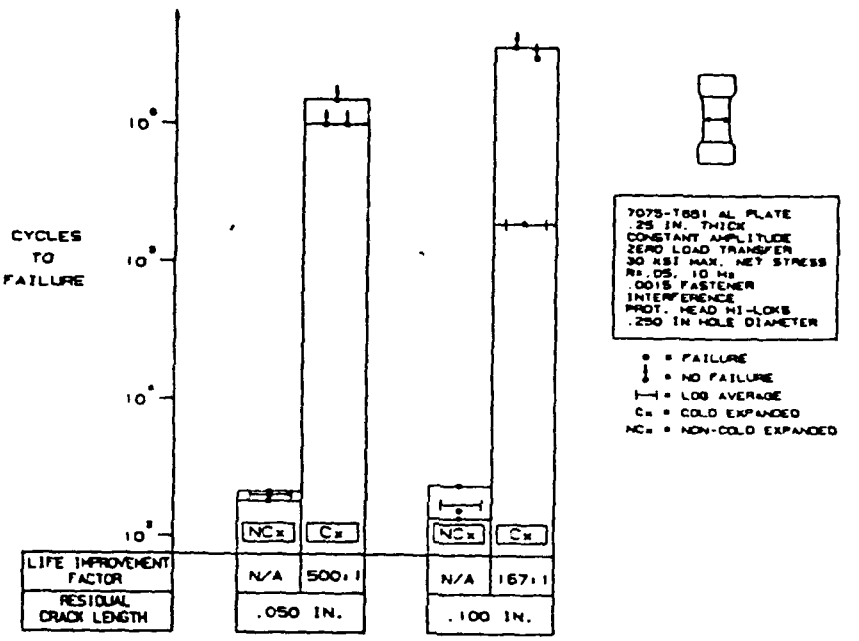
StopCrack™

Effect Of Missed Crack Tip



The effectiveness of the StopCrack System when the crack tip is missed has been dramatically demonstrated. These results have been presented to this conference before but are worth repeating.

Residual, sharp, natural fatigue cracks of two different lengths were purposefully introduced in test specimens in order to simulate the situation. With a residual crack of .050 inch (through crack), use of the StopCrack System resulted in better than a 500:1 life improvement, for the conditions shown. With a residual crack of .100 inch, a 167:1 life improvement factor was demonstrated.

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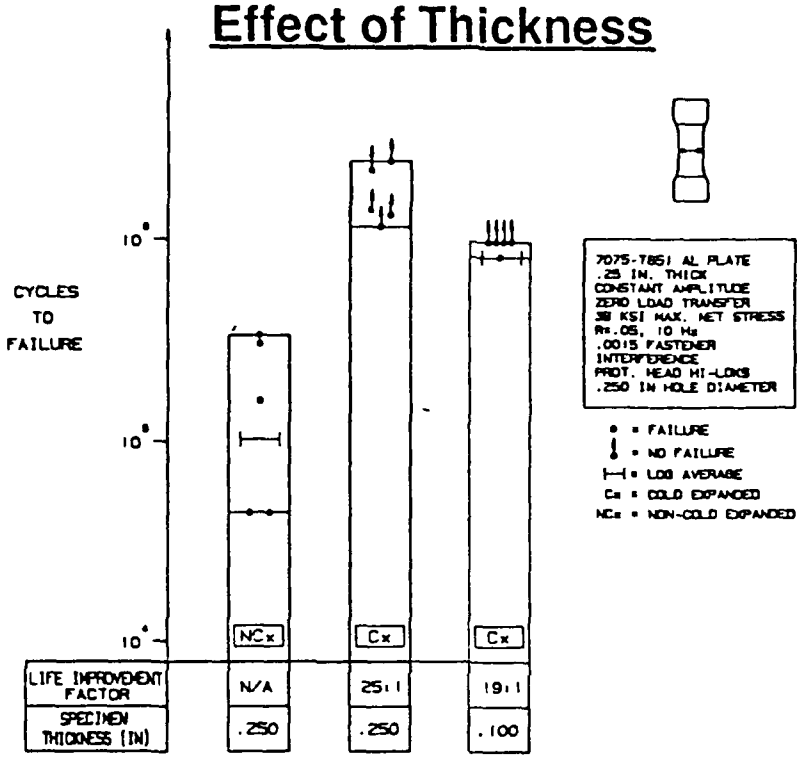
StopCrack™

Effect of Thickness



There is a great deal more test data available concerning the effect of cold expansion on the life of a stop drill repair. A major development has been the demonstration of the system's effectiveness in a broad range of material thicknesses, including gauges as thin as .030 inch.

The effect of StopCrack in .25 inch and .10 inch material is shown. For the conditions tested, use of cold expansion provided a 25:1 life improvement (enhanced:not enhanced) for .25 inch thick 7075-T6, and 19:1 life improvement for .100 inch material.

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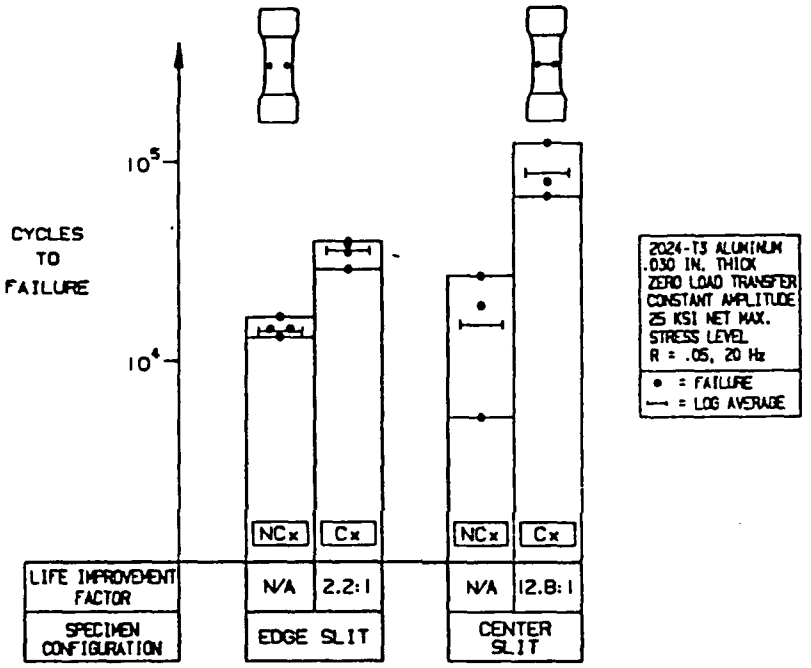
StopCrack™

Effect of StopCrack System in Thin Sheet



Testing with .030 inch material also shows effectiveness. The data shows that use of the StopCrack System provides better than 2:1 life improvement, even in thin, high toughness 2024-T3 aluminum. This data extends the usefulness of the StopCrack System to applications in commercial aircraft fuselage structure, general/light aviation aircraft, and thin non-primary structure in all aircraft.

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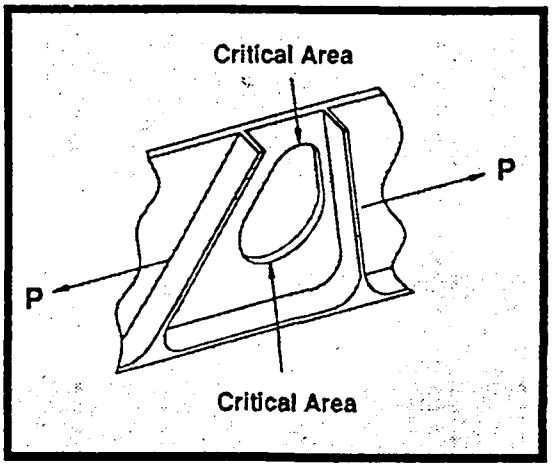
CX SYSTEMS

SSCX™

Split Sleeve Cold Expansion™

For

Fatigue Life Enhancement of Non-Round Holes



The last advancement to be discussed is the fatigue life enhancement of non-round holes, the most significant development to date. Previously, only holes which were circular could be cold expanded.

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SSCX™

Fatigue Life Enhancement of Non-Round Holes

Re-work (existing non-round hole):

-"Install" residual compressive stresses after detail is machined.

-Special technology, tooling and applications criteria required.

-Verification test programs have demonstrated benefit.

New production:

-"Install" compressive residual stresses before final machining of detail.

-Use existing standard cold expansion technology.

-Feasibility studies have demonstrated potential.

There are two categories of non-round holes which can be processed using the Split Sleeve Cold Expansion System. They include:

1) Existing non-round holes (re-work).

2) New production non-round holes.

The method by which the holes are treated is different in each case.

1) Rework of existing holes requires special tooling, technology and application criteria.

2) New production uses existing cold expansion technology and tooling.

Verification testing has confirmed the benefit of this process for rework of existing non-round holes, and it is currently being used in a particular aircraft application. Feasibility studies and fatigue testing have demonstrated the potential for fatigue life improvement for production of non-round holes.

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Fatigue Life Enhancement of Non-Round Holes

Fatigue improvement of existing detail:

- Potential critical holes in primary load path structure on aircraft
- High strength aluminum
- Unusual "pear"-shaped, "D"-shaped, and oval holes
- FTI tasked to develop new technology to improve fatigue performance (increased inspection interval)
  - --Cyclic test program, unusual specimen design
  - --Tooling and procedures development

The concept of enhancing the fatigue life of non-round holes came from a particular application. The detail was in primary load path structure, in high strength aluminum. This particular hole was "pear-shaped," but other shapes have been evaluated, including "D-shaped" and oval holes.

FTI was tasked to develop new technology to improve the fatigue performance in order to increase the inspection interval. A tooling evaluation was initiated to assess the effectiveness of each tooling concept.

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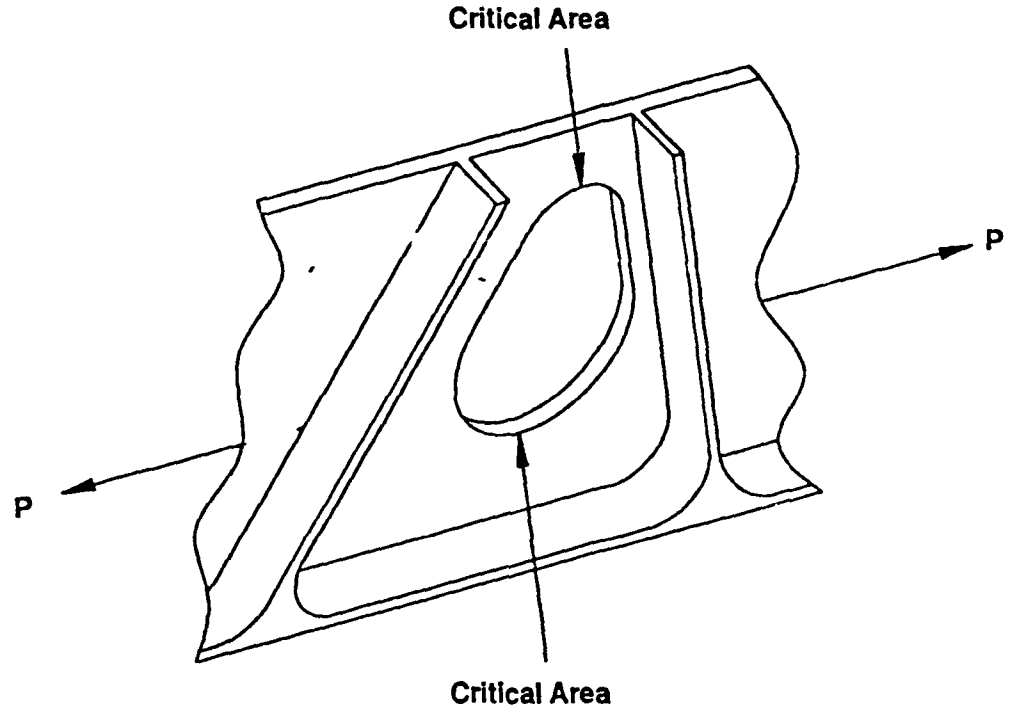
Non-Round Hole Configuration



The hole and surrounding structure look something like this sketch; note the load direction and potential critical areas.

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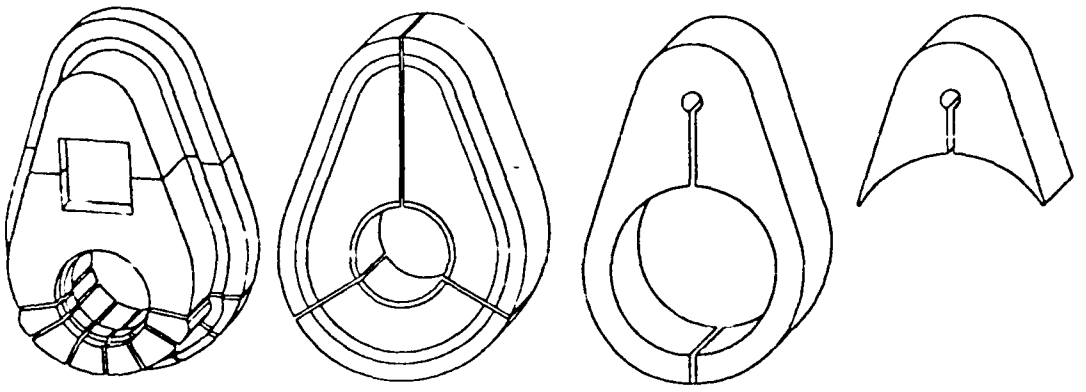
Evolution of "Pear"-shaped Hole Cold Expansion Fixture



The evolution of tooling concepts for the cold expansion of a "pear"-shaped hole are shown. The tooling was originally conceived as adjustable so as to fit a wide existing hole tolerance. The first concept was adjustable; the mandrel used to cold expand the fixture was based on the measurement of the hole width. This concept proved impractical as the pull force was exceedingly high. A test program using birefringent material (photostress) was performed to assess process effectiveness.

The second concept/prototype was designed to reduce the cost and the number of pieces, i.e. splits. The adjustability was controlled by the mandrel used to cold expand the fixture. The hole in the fixture was increased in diameter to increase the expansion energy efficiency. The pull force required was less than that required for the first but remained relatively high. The hole size was increased to further reduce the pull force.

The third prototype reduced pull force to an acceptable level and increased the depth of cold expansion to the small radius, now found to be equally critical as the large radius. The design effort now concentrated on production tool design, reducing cost, and increasing ease and simplicity of operation.

The forth and final concept reduced the pull force to about 1/4 the pull force required on the first prototype. In addition, the zone of compressive stress was much larger and more uniform than any of the early prototypes.

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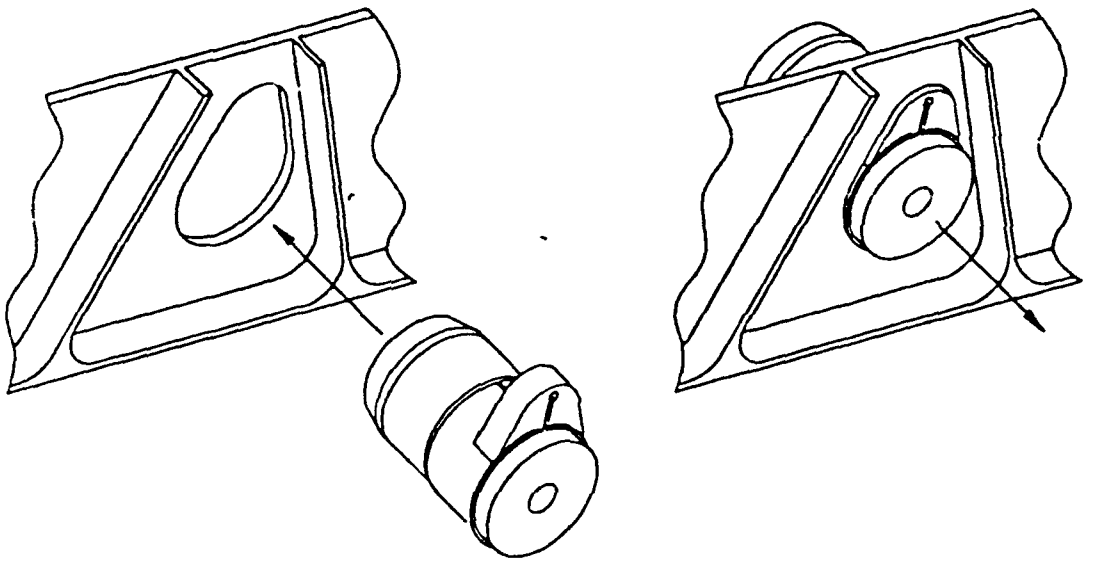
System Overview:

"Pear"-shaped Hole Cold Expansion



Production use of the tooling is shown. The mandrel/sleeve/cam assembly is inserted into the hole. The mandrel is pulled through the hole by a puller unit while a nosecap retains the sleeve and cam.

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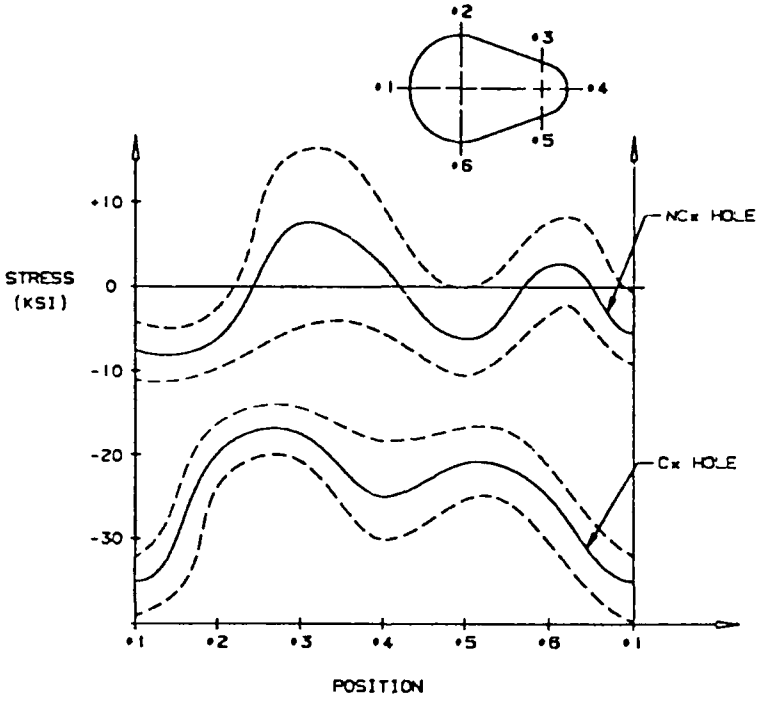
SsCx™

X-Ray Diffraction Results



This graph shows the residual stress pattern around the hole, from x-ray diffraction data. The work was performed by TEC(Technology for Energy Corp.). The graph plots location on the hole versus tangential stress for both the as-machined hole and cold expanded hole. The compressive stress was highest at the ends of the hole; approximately 38 ksi in compression. All other regions around the hole were in a compressive stress state.

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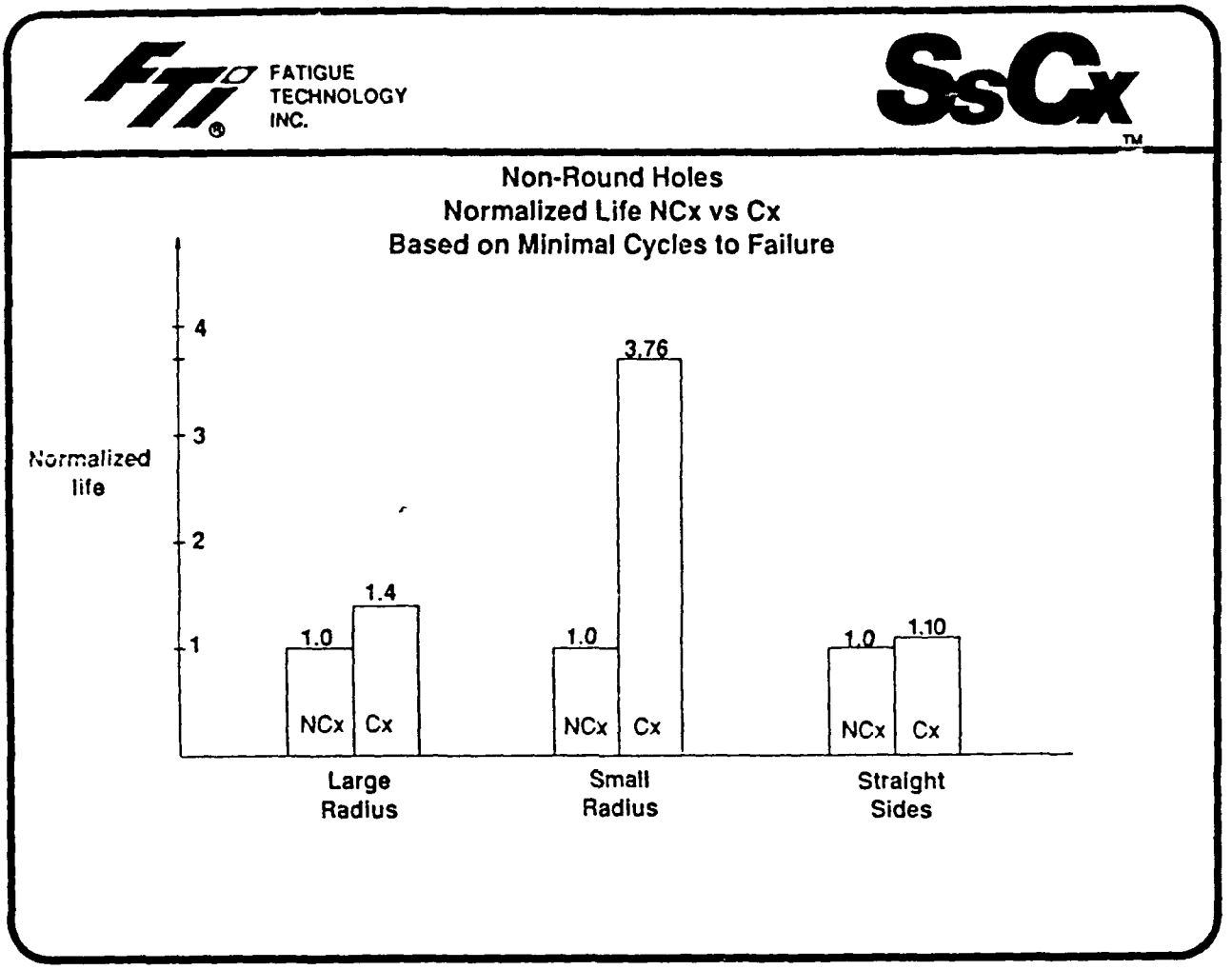

The normalized lives of non-cold expanded and cold expanded specimens for the large and small radii and the straight sides are shown, under actual in-service load conditions. A 1.4:1 life improvement was demonstrated for the large radius, 3.7:1 for the small radius and little improvement for the sides. It is important to note that the straight sides were not considered critical.

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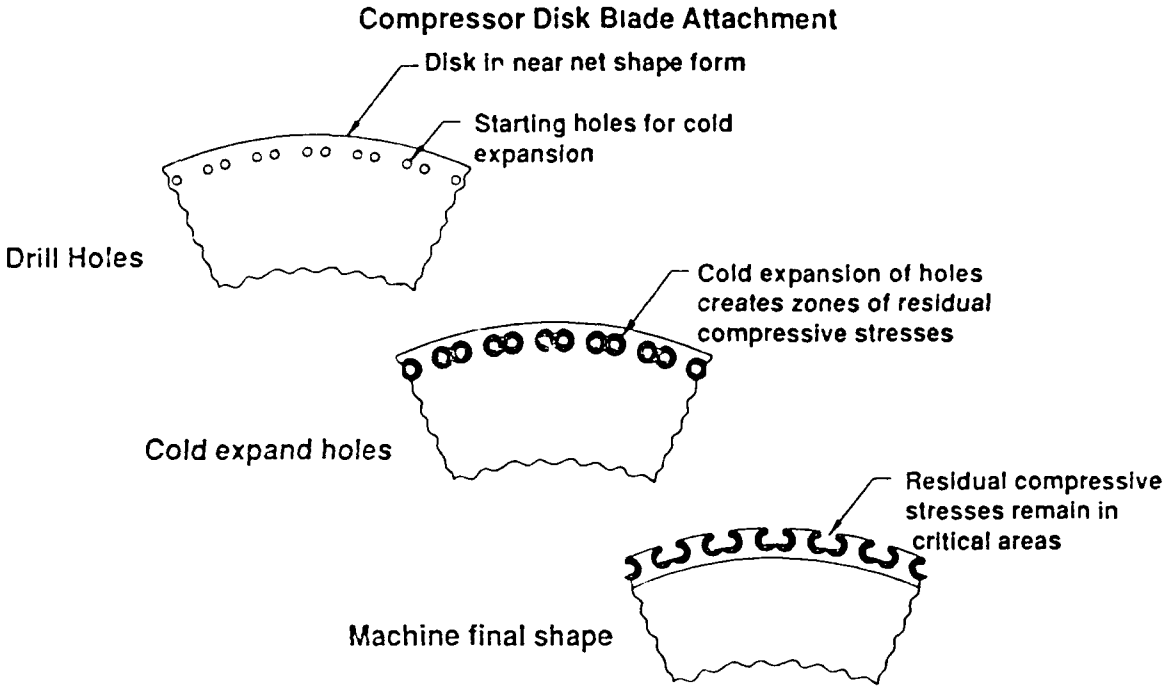
SsCx™

Fatigue Life Enhancement of New Production Detail



The process steps that could be used for providing fatigue enhancement for new production of non-round holes are shown. The steps are illustrated using a compressor disk as an example. This method of cold expansion could be readily applied to many other configurations. The process steps are:

1) Machine starting holes into a disk, or other structure, in near net shape condition.
2) Cold expand the holes to induce residual compressive stress into the part.
3) Machine the detail to the final shape.

The final machining operation does not eliminate all the compressive residual stress. The results of fatigue testing using this method in both aluminum and titanium have been encouraging.

FATIGUE TECHNOLOGY INC.

CX SYSTEMS

USAF Usage

- Split Sleeve Cold Expansion is being used on new production or rework on virtually all USAF aircraft

A-7	F-4	KC-135	T-37	Air Force One
A-10	F-5	C-130	T-38	E-3
B-1	F-15	C-141		E-4
B-2	F-16	C-5		
B-52	F-111			

- StopCrack System included in KC-135 SRM (-3 TO) and being evaluated for general SRM use
- ForceMate being used on B-52 and being evaluated for applications on B-1, F-111, T-37, TF-33, TF-41
- Fatigue enhancement of non-round holes being used on a USAF aircraft system

In conclusion, a brief review of the usage of Cold Expansion on USAF aircraft is shown:

1. The Split Sleeve Cold Expansion and/or Split Sleeve Cold Expansion to Size Systems are being used on new production or rework of virtually all USAF aircraft.

2. The StopCrack System is included in the KC-135 SRM(-3 T.O.) and being evaluated for general SRM use.

3. The ForceMate System is being used on B-52 and being evaluated for applications on the B-1, F-111, T-37, TF-33 and TF-41.

4. Fatigue enhancement of non-round holes being used on a USAF aircraft.

FATIGUE TECHNOLOGY INC.

Cx SYSTEMS

Summary

- Advances in fatigue enhancement technology have been shown:
  - -ScCx
  - -FmCx
  - -Non-round holes
- Fatigue enhancement technology has been applied to unusual materials, geometries, and structures.
- USAF is now using Cx Systems for life improvement and increasing inspection intervals.

In summary, advances in fatigue enhancement technology have been shown:

1. ForceMate - An alternative to traditional bushing installation methods,
2. StopCrack - Enhanced Stop Drill Repair, and
3. Fatigue enhancement of non-round holes.

Fatigue enhancement technology has been applied to unusual materials, geometries, and structures.

The USAF is now using Cold Expansion Systems for life improvement and increasing inspection intervals.

LIFE CYCLE COST REDUCTIONS REALIZED FROM IMPLEMENTATION OF DAMPING

by
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INTRODUCTION

Under various Air Force contracts, the University of Dayton Research Institute (UDRI) has developed and implemented passive damping designs which have successfully eliminated serious vibration problems on aerospace systems. The elimination of these vibration problems has had a significant impact on the life cycle cost of the system. In all cases, the damping design was cost effective to procure and install, and the development costs were competitive with alternative design solutions.

The purpose of this paper is to present the impact passive damping designs can have on reducing vibration induced life cycle maintenance problems, thereby reducing life cycle costs. This purpose is met through describing damping systems which are, or soon will be, installed and which have proven successful in both eliminating the maintenance problems and reducing life cycle costs. The damping projects presented are the:

- TF-41 jet engine inlet extension
- TF-30 jet engine inlet guide vane (IGV)
- TF-33 jet engine IGV
- KC-135 aircraft aft fuselage

The TF-41 and TF-30 damping systems have been installed and in service for several years. The TF-33 and KC-135 are just now entering implementation. A brief discussion of the specific vibration problem, damping design, test evaluation, and implementation process is given for each problem. For the two designs which have been in service the life cycle cost reductions are presented. For the two designs just entering implementation, the projected cost savings are given.

TF-41 Inlet Extension

The TF-41 inlet extension connected the engine case and the engine bay air seal. The extension, shown in Figure 1, is an eight inch wide ring of sheet steel shaped in a truncated cone with one inch flanges welded to each side.

Fatigue failures in the extension encounterd during engine development were eliminated by wrapping several layers of fiberglass reinforced epoxy around the extension.

In service, whenever debonds occurred between the fiberglass epoxy layers, the severe vibratory environment would cause frictional heating which eventually burnt the fiberglass. Fatigue cracks occurred in the extension directly beneath the burnt area. The repair procedure for the extension was to check the burnt area to determine the extent of delamination and to apply a patch. If the fiberglass was sufficiently delaminated, the old fiberglass was stripped off, the extension weld repaired if required, and new fiberglass epoxy applied. These repairs were very expensive and time consuming.

UDRI initiated an effort to develop a vibration damping design which would replace the fiberglass wrap and eliminate the fatigue by reducing the vibratory stress levels.(1)

For the extension, the maximum operating temperature range extended from -65°F (-53.98°C) to 225°F (107.22°C). Flight temperature data indicated that for approximately 90 percent of the total flight hours the extension saw a temperature in the range of 50°F (10°C) to 200°F (93.33°C).

The final damping design was a two layer, constrained layer damping system consisting of ISD-110 damping material and an aluminum constraining layer, and ISD-112 damping material and an aluminum constraining layer in the dimensions shown in Figure 2.

A plot of the maximum response during the engine test cell evaluation with the fiberglassed extension and the corresponding damped extension data is given in Figure 3. Overall, the response levels recorded from the damped inlet were lower than those from the fiberglass wrap configuration by a factor of five to six.

Field service data has shown the damping system eliminated all maintenance problems on the extension.

TF-30 Inlet Guide Vanes

The inlet guide vane (IGV) case for the TF-30-P100 jet engine, shown in Figure 4, was a very high maintenance cost item because of fatigue cracks occurring in less than 50 engine hours. The cause of the cracks was identified as resonant vibration of the vanes, excited by pressure fluctuations caused by first-stage fan blade passage.

Several standard design changes such as reducing stress concentrations, changing the vane alloy, and relocating the inlet case were evaluated, but none of these were successful in reducing the vane cracking in a cost effective manner. The passive damping approach provided a cost effective fix.(3)

Based on flight data the damping design needed to be effective in a temperature range from 0°F to 125°F which accounted for 98 percent of engine operation time and the most significant damage accumulation. Occasional excursions to higher and lower temperatures, due to infrequent flight conditions, were not important for fatigue damage, but were crucial in designing the damping system and installation procedure. The engine test data showed that the modes occurring between 3000 and 4000 Hz generated the highest vibration stresses; therefore, the damping design was optimized for these modes.

The TF-30-P100 IGV damper wrap (shown in Figure 5) was fully qualified for use. The vibratory stress reduction and attendant crack abatement has been proven by strain gage data in the test cell and confirmed qualitatively by accumulation of high engine run times without failures in service. The systems integration aspects of engine performance, distortion tolerance, anti-icing effectiveness and durability were thoroughly and successfully addressed by test cell and field evaluations.

TF-33 Inlet Guide Vane

The effort on the TF-33 IGV was initiated because: 1) the TF-33-P3 IGV cracking was a high cost maintenance problem, 2) IGV

cracking impacted operational readiness, and 3) the program on the TF-30 engine proved successfully in alleviating a similar HCF problem.(4)

The comparison of vane damping with and without the damping design is shown in Figure 6 which indicates an increase in damping of the vane at 73°F (23°C) of more than a fact of seven. Significant damping improvement is shown over the entire temperature range.

The laboratory test success led a field evaluation. A damped case went into service in November of 1979 and was overhauled in June of 1985. This time frame represented at least a 33 percent increase in time between repairs.

KC-135 Aft Fuselage

Early in the life of the KC-135 aircraft, a sonic fatigue problem in the skins was corrected by adding "belly bands" to the skin panels which were placed between the frames, running parallel with the frames around the fuselage. This successful fix was both costly and heavy.

Recently, sonic fatigue cracks appeared in the KC-135A aft of fuselage body station (BS) 1370 which is the frame location where the belly bands stopped. In an effort to find a more cost and weight effective solution to the fatigue problem, UDRI undertook a program to design and evaluate a passive damping system.(5)

The modal data indicated one mode dominated the band under study. This mode at 300 Hz was a coupled mode of the skin, stringers, and frames. Resonances at other frequencies between 300 and 800 Hz were plate modes in the skin and structural modes of the skin stringer structure.

The frequency response data from the points surveyed during the engine run showed modal frequencies which agreed closely with those acquired during the modal test. The SPL recorded was 157 dB.

The structural baseline data combined with an operational temperature range of 20°F to 100°F lead to the following final damping designs:

- 0.004 inch 3M ISD113 and 0.012 inch aluminum constraining layer, or
- 0.006 inch Soundcoat MN and 0.015 inch aluminum constraining layer.

Two sources for the damping system were developed to avoid a sole source procurement situation. Figure 7 illustrates the effectiveness of the damping design.

Based upon the curves presented in Figure 8, implementation of the recommended skin damper designs would result in an expected ten-fold increase in fatigue life of the skin and at least a doubling of fatigue life for the frame/stringer substructure.(6) The predicted fatigue life increases were obtained at a weight penalty of less than 21 pounds added weight.

COST ANALYSIS

In the following cost analysis, the total cost savings is calculated from the difference between the maintenance cost of the undamped part and the damped part. The costs associated with the development, evaluation, procurement and installation of the damping system are included in the total maintenance costs of the damped part. Therefore, the presented cost savings are true cost savings.

TF-41 Inlet Extension

Cost analysis in 1978 of the fiberglassed extension showed that the maintenance costs were $398,811 annually.(2) By installing the damping wrap during standard overhaul, this cost data indicated that five years after the fleet was damped, a cost savings of $2,537,873 would be realized. The cost of purchase and installation of the damping system was subtracted from the savings so that the real savings is over 2.5 million dollars.

Using the appropriate inflation factors and current overhaul schedules, $525,379 per year would be spent to maintain the undamped extension. The 1987 figures for the damped inlet indicate a maintenance cost of $22,140; therefore, a $503,239 cost savings per year is realized as a result of damping the TF-41 inlet extension.

Damping system implementation started in May 1980; and current TF-41 operation schedules indicate the engine will fly until the year 2000. Using this information, the total life cycle cost savings can be calculated as:

• Implementation plus five years May 1980 to June 1988	$2,537,873
• Damped maintenance July 1988 to June 2000	$6,038,868
TOTAL SAVINGS	$8,576,741

TF-30 Inlet Guide Vanes

The cost analysis of the TF-30-P100 IGV in 1978 revealed that the maintenance costs for the undamped cases were $1,506,675 per year. Cost analysis on the other versions of the TF-30 IGV (P3, P7 and P9) showed an combined maintenance cost of $1,994,148 per year. By incorporating the damping design into the standard overhaul procedure, the cost analysis indicated a total savings on all the TF-30 IGV's of $14,778,177 five years after fleet implementation was complete. Implementation started in February 1978 and was completed in 28 months.

Using the current overhaul schedules and proper inflation rates, $6,027,522 in 1986 dollars per year would be spent to maintain the undamped TF-30 IGV cases. Cost figures show that the maintenance expenditure for 1987 for the damped IGV's was $109,853; therefore, a $5,917,669 savings per year is realized from damping the TF-30 IGV's.

The TF-30's are expected to fly until the year 2010; therefore, the total life cycle cost impact of the damped IGV's can be shown to be:

• Implementation plus five years Feb. 1978 to April 1985	$14,778,177
• Damped Maintenance May 1985 to April 2010	$147,941,724
TOTAL SAVINGS	$162,719,902

TF-33 Inlet Guide Vane

The yearly repair cost for the TF-33-P3 IGV cases is $1,465,708. It would take five years to damp all the IGV cases in the inventory during standard engine overhaul. The cost of purchasing and installing the damping wraps, and maintaining the damped IGV's for a total of ten years (five years to install and five years to maintain the damped IGV's) was projected to be $9,503,973. The cost to maintain the undamped IGV's for the same ten year period would be $14,657,080. The cost to develop and evaluate the TF-33 damping design was $150,000. Therefore, the Air Force can realize a $5,003,108 maintenance cost savings over a ten year period if the TF-33 IGV's were damped. An additional $1,435,488 savings will be realized for each year the TF-33-P3 is flown beyond the ten year period used in the cost analysis above.

The total life cycle cost savings for the TF-33 IGV's are:

• Implementation plus five years Jan. 1988 to Dec. 1997	$5,003,108
• Damped Maintenance Jan. 1998 to 2010	$17,225,856
TOTAL SAVINGS	$22,228,964

KC-135 Aft Fuselage

The total cost for the KC-135 damping system was:

• Design and evaluation of the damping system	$ 60,000
• Development of installation and procurement specifications	$ 24,500
• Purchase and installation of the damping system on all 741 KC-135 aircraft	$713,135
TOTAL	$797,635

The estimated cost for the "belly band" solution was $16,000,000 not including installation; therefore, a minimum $15,202,365 savings was realized by implementing the damping system.

SUMMARY AND CONCLUSIONS

The four projects discussed are excellent examples of passive damping systems alleviating resonant vibration problems, surviving severe operational environments and providing significant cost savings in life cycle maintenance and redesign implementation costs. Table 1 summarizes the cost savings data for all four projects. As Table 1 presents, UDRI efforts in designing and implementing these damping systems have resulted in life cycle cost savings of $208,727,948.

TABLE 1
COST SAVINGS SUMMARY
(ALL FIGURES IN DOLLARS)

Project	Cost savings total, 5 yrs after fleet implementation	Savings per year after implementation	Total life cycle cost savings	Alternate redesign & implementation costs
TF-41 Inlet Extension	$ 2,537,873	$ 503,239	$ 8,576,741	N/A
TF-30 IGV	$14,778,177	$5,917,669	$162,719,888	$15,000,000 plus implementation
TF-33-P3 IGV	$ 5,003,108	$1,435,488	$22,228,964	N/A
KC-135 aft fuselage	$15,202,365	N/A	$15,202,365	$16,000,000 plus implementation
TOTAL	$37,671,523		$208,727,948	

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5. Drake, M. L., "TF-41 Inlet Extension Damping Wrap Program," Final Report No. UDR-TR-80-34, prepared for the Aeronautical Systems Division PRAM Office, Wright-Patterson AFB, OH, March, 1980, available from University of Dayton Research Library.

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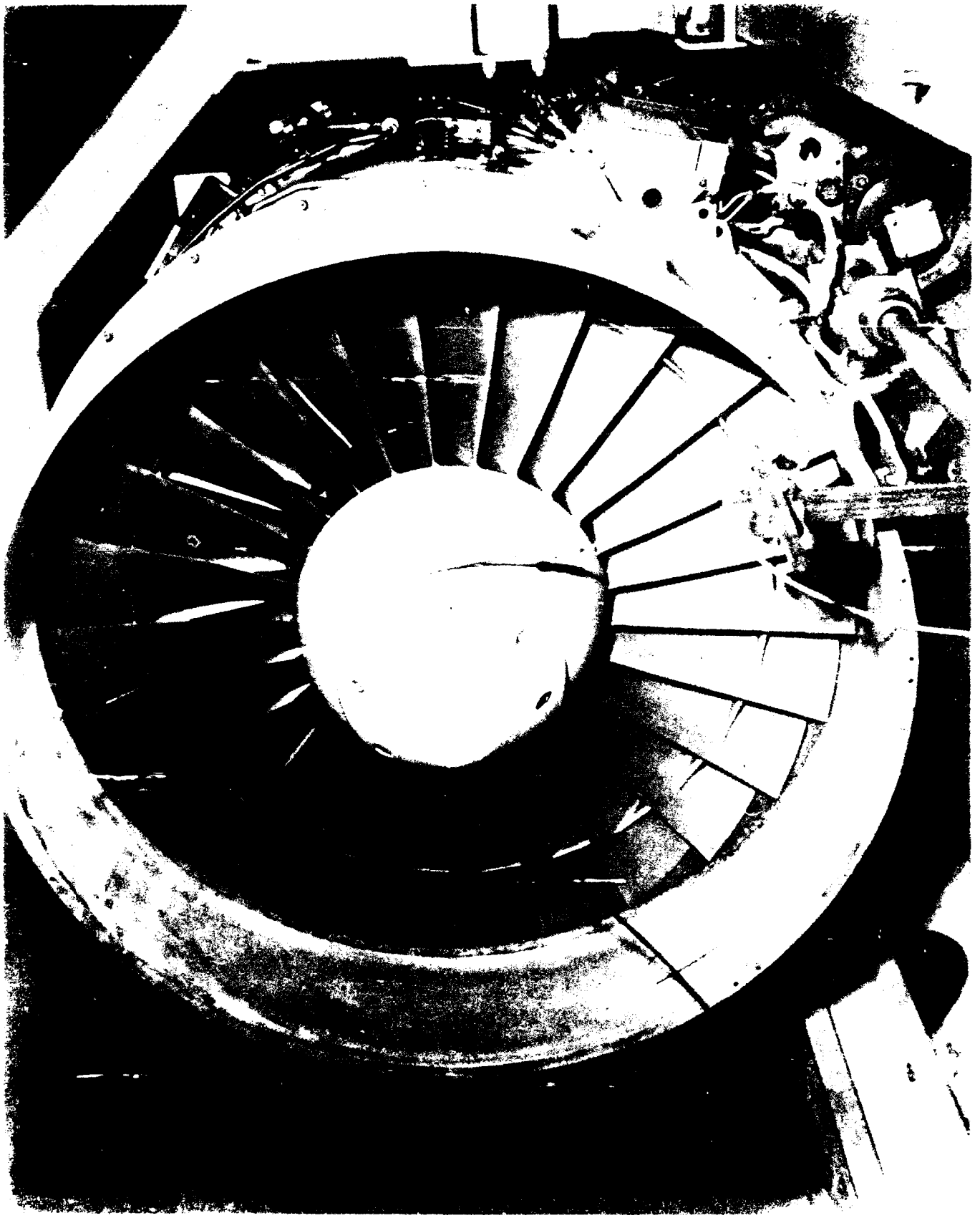


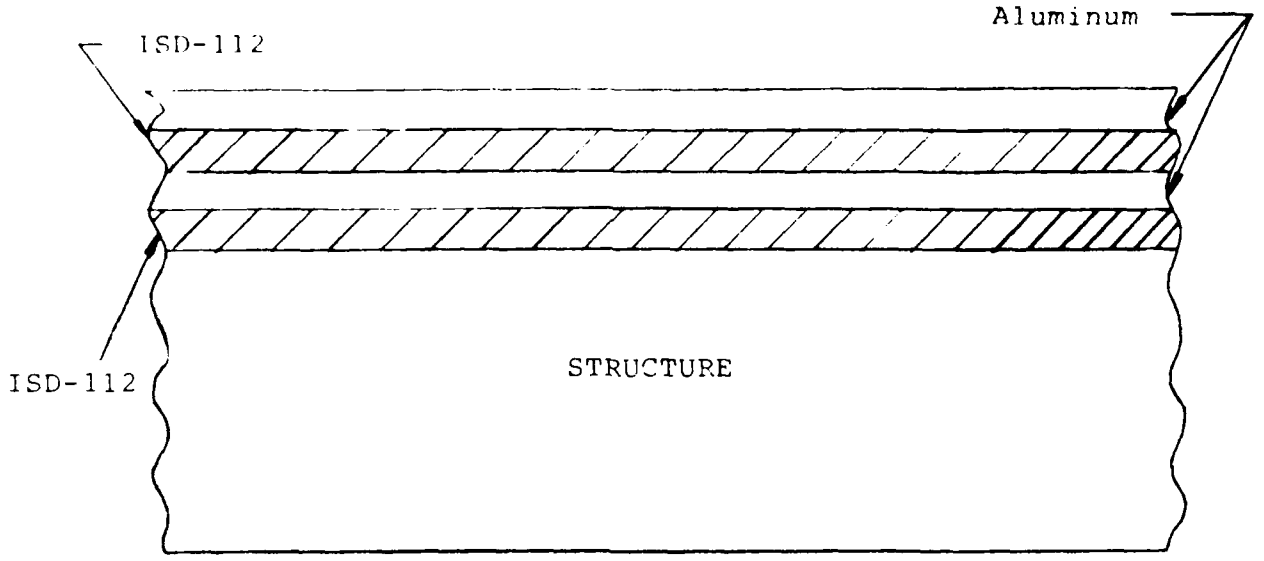


Fig. 2 Schematic of TF-41 Damping Design.

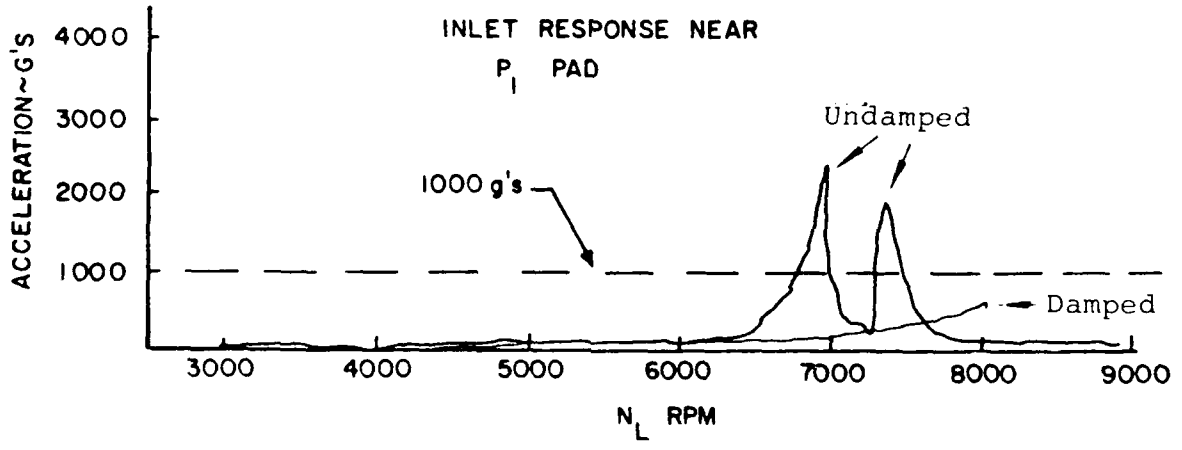


Fig. 3 TF-41 Undamped Response.

Fig. 4 TF-30 Inlet Guide Vane Case.

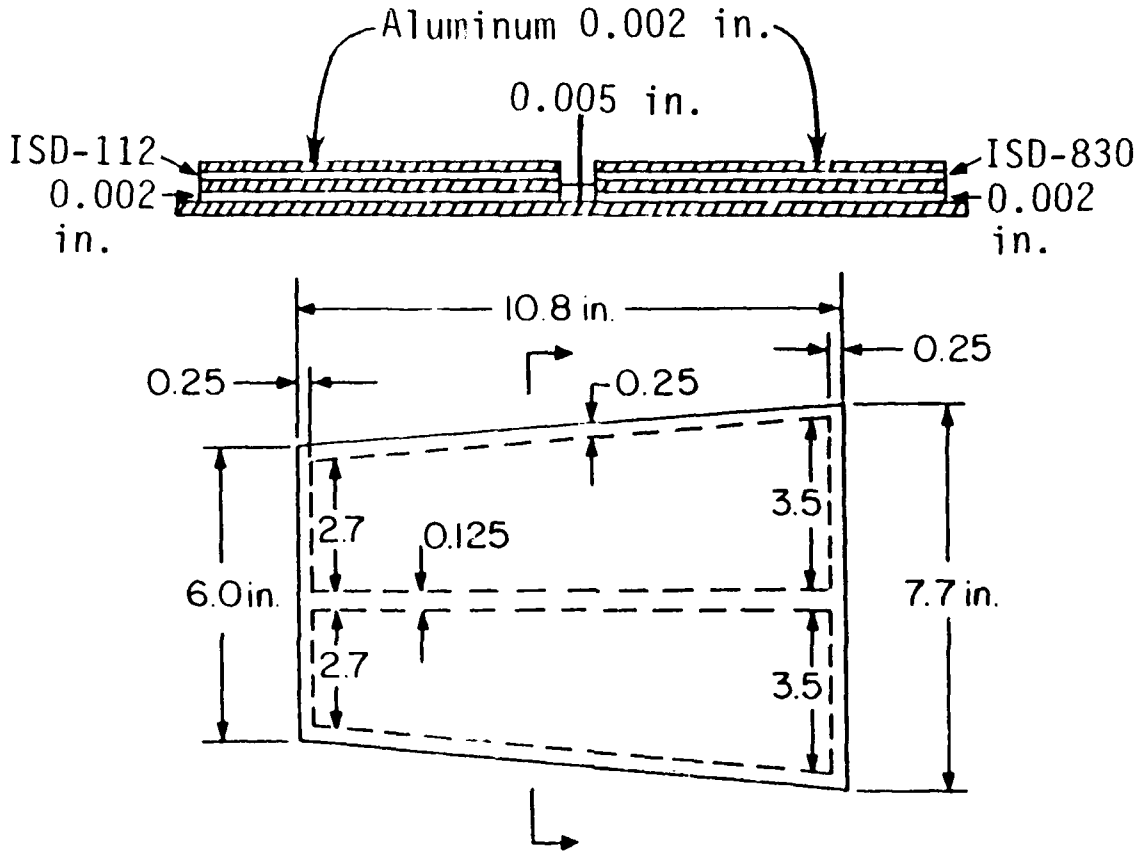


Fig. 5 TF-30 Damping Design.

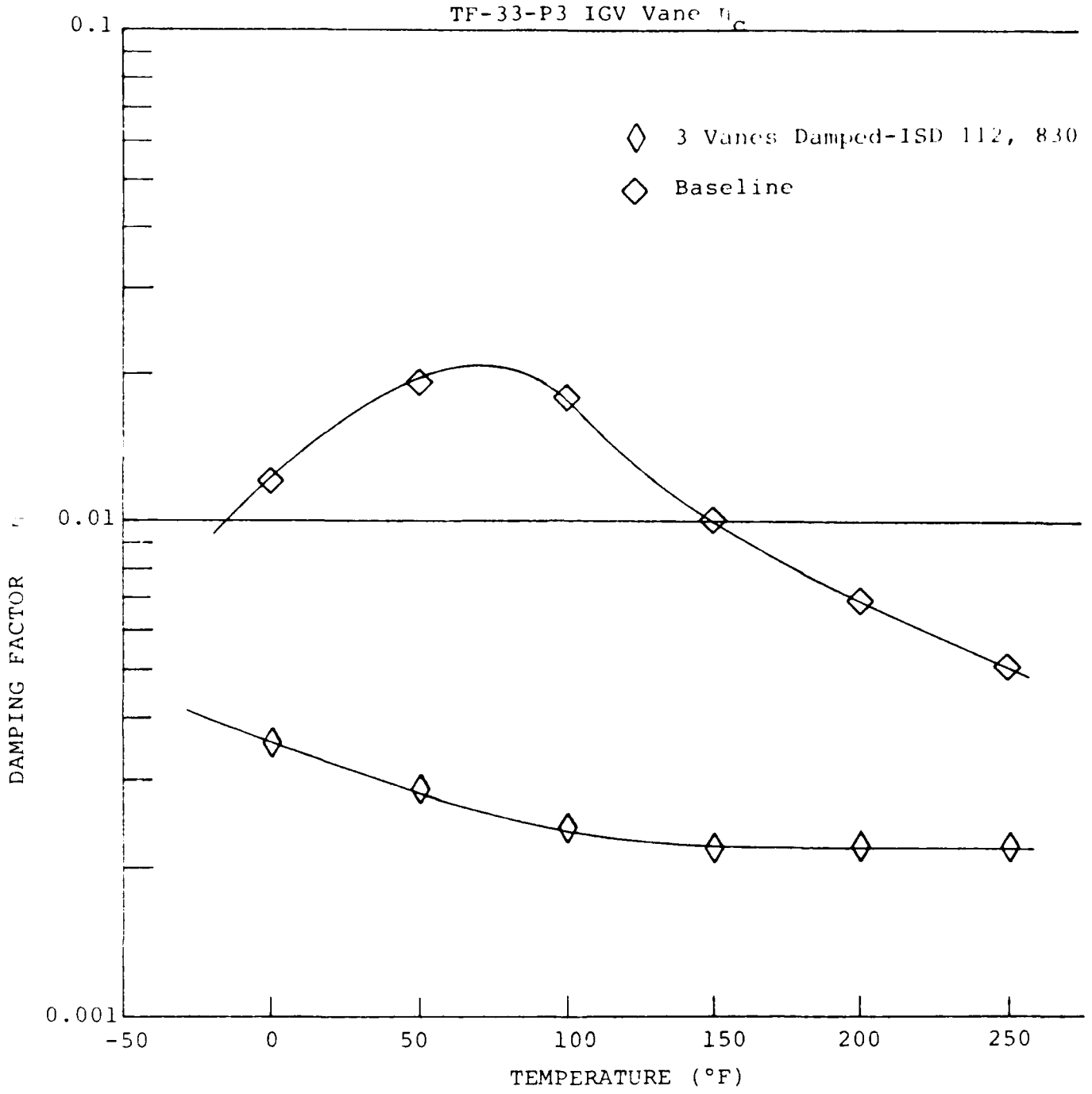


Fig. 6 TF-33-P3 IGV Case Vane Laboratory Test Results - Damping Factor Versus Case Temperature.

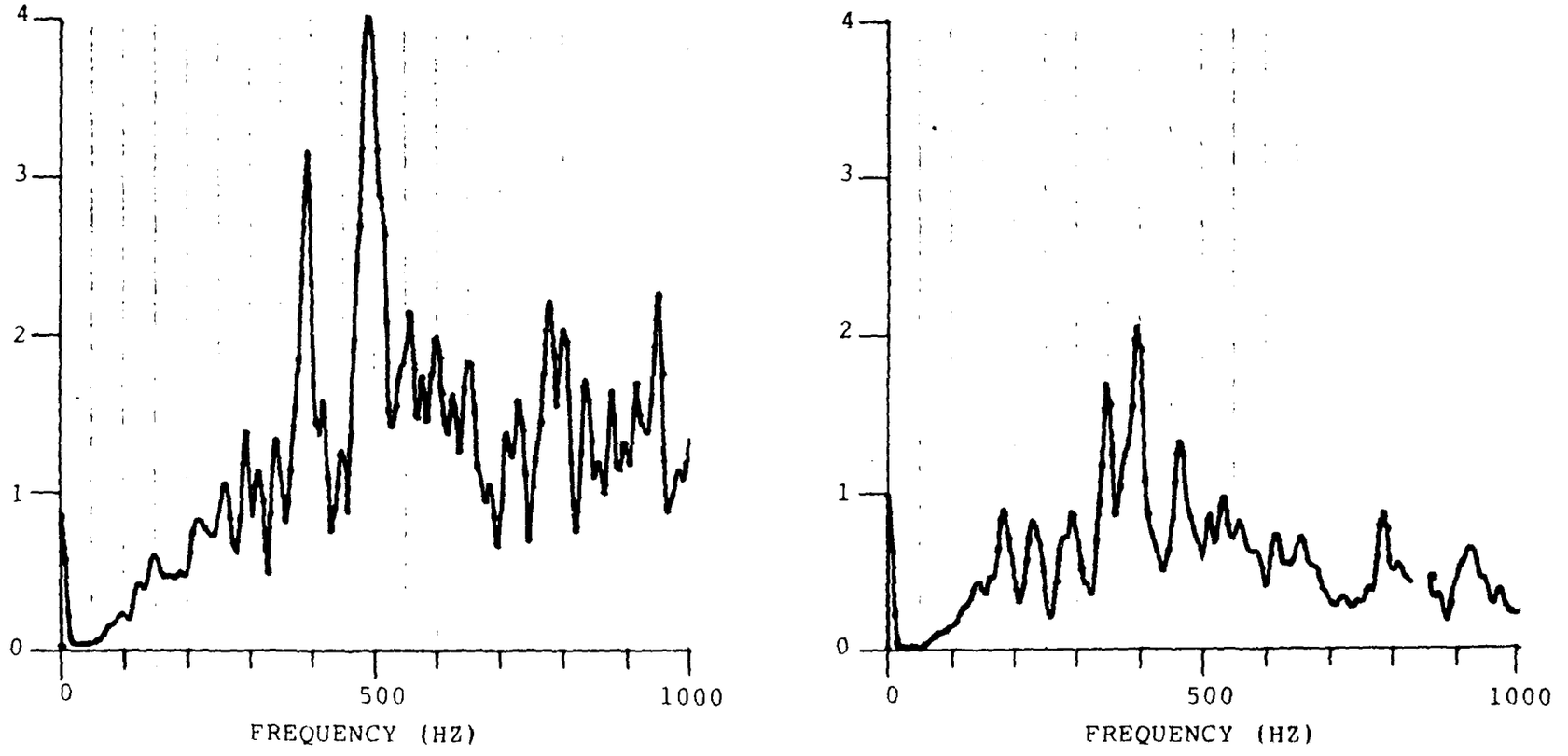


Fig. 7 Response of Test Point 77 to Sonic Excitation Before (Left) and After Damping Installation.

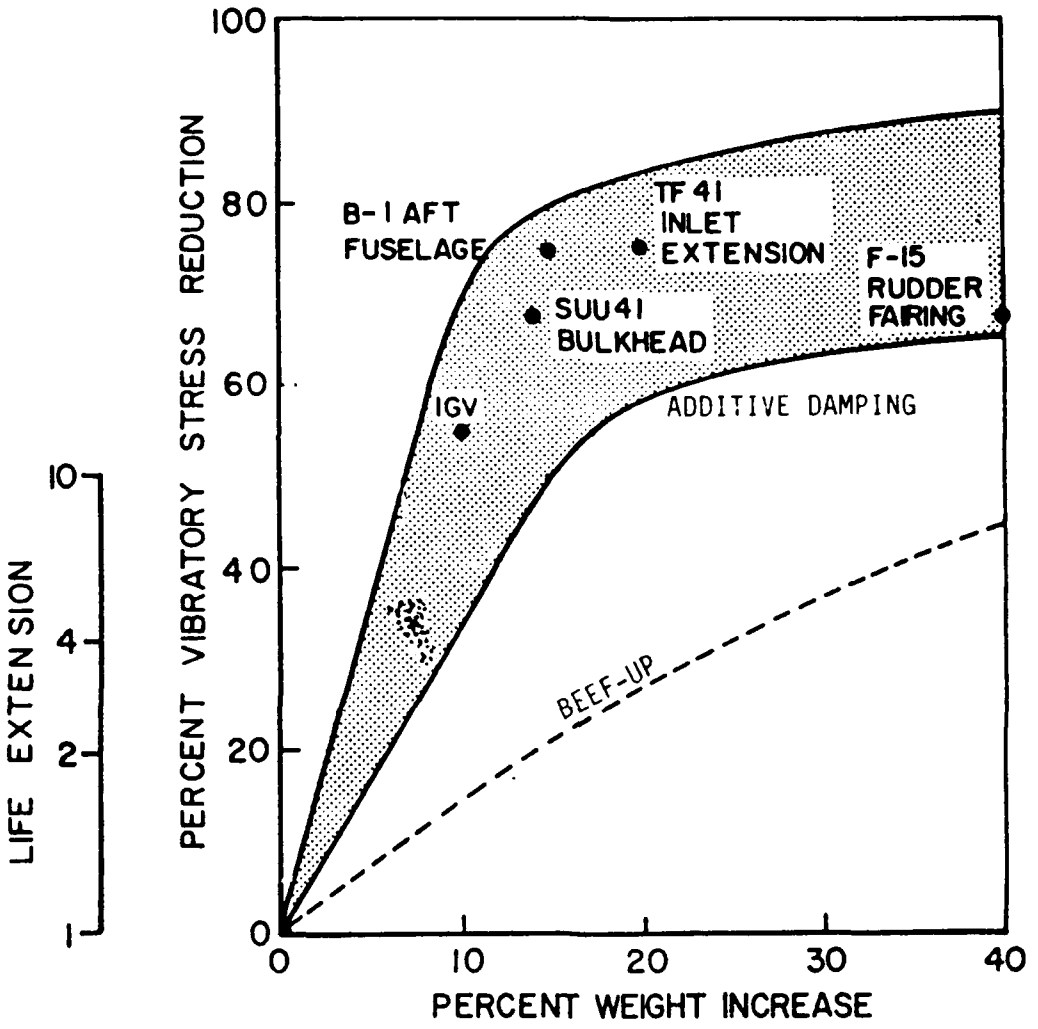


Fig. 8 Life Extension as a Function of Increased Weight.

Initiating and Integrating AVIP with ENSIP

1988 USAF Structural Integrity Conference

November 29 - December 1, 1988

San Antonio, Texas

Louis B. Allen, Jr.
General Electric Company
Aircraft Engines
Cincinnati, Ohio

AVIP Purpose

- Reduce development risk
- Improve system availability
- Reduce life cycle cost
- Assure specified operating life

AVIP Approach

- More formal feedback mechanisms
- Find problems earlier
  - Easier to recover

Lower Schedule Risk

AVIP Structure

These stages run concurrently

- Stage I — Design information
- Stage II — Preliminary planning
- Stage III — Design
- Stage IV — Compliance and production
- Stage V — Life managment

Glossary

Damage tolerance - Reponse to system defect, manufacturing or service induced

Durability - The ability of hardware to function without mission or safety failure

Design criteria - Design rules and restrictions

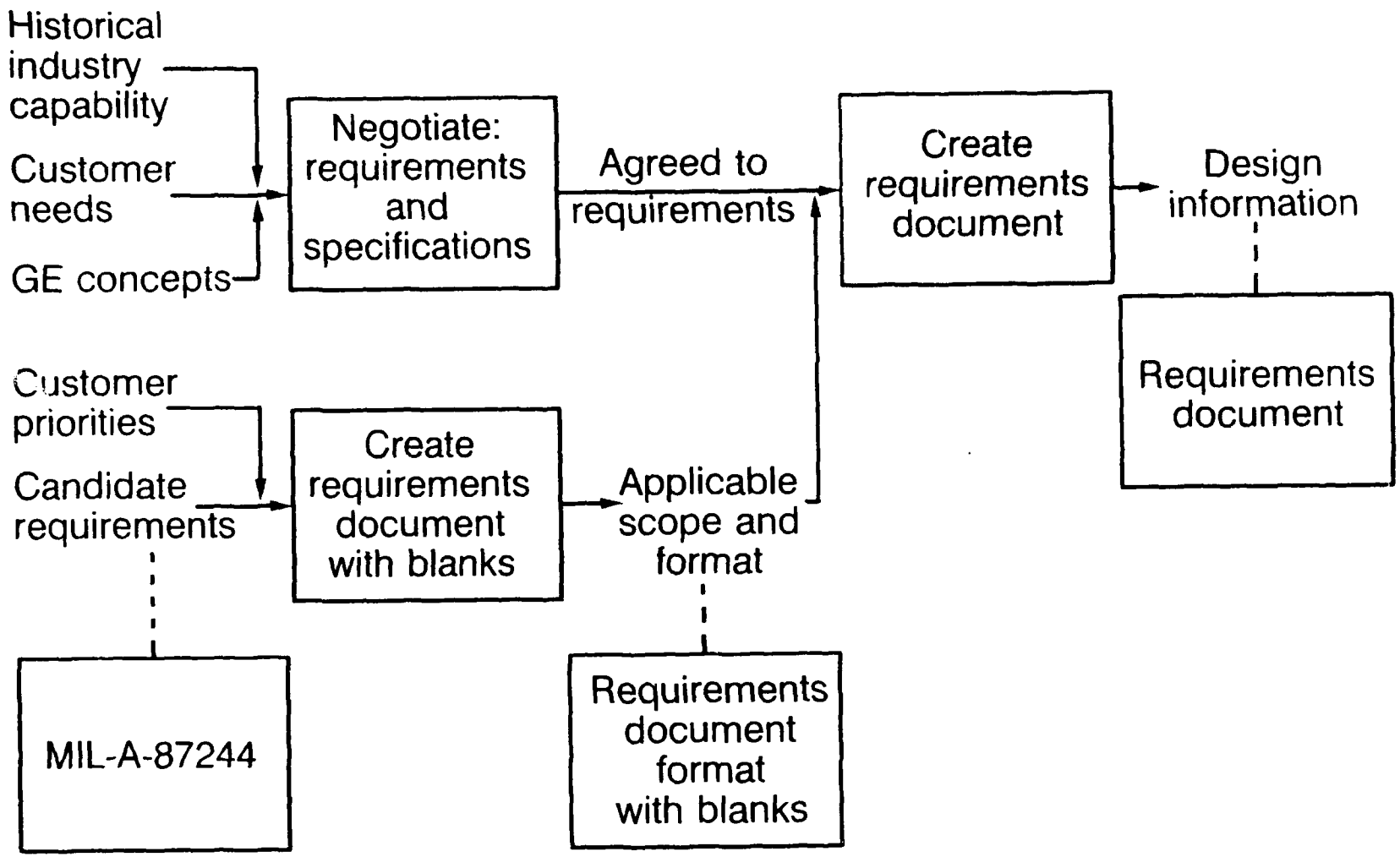
Sensitivity and tolerance model - Computation model relating performance to tolerance

Failure review board - Directive board for recovery from failures

Stage I - Design Information

- Operational requirements
- Maintenance concepts
- Mission profiles
- Environmental data
  - Operational
  - Logistical
- Usage constraints
- Durability requirements
- Damage tolerance requirements

AVIP Stage I - Design Information Process Flow



Stage II - Preliminary Planning

- Develop avionics integrity master plan
  - Shows how AVIP compliance will be demonstrated
  - Defines analyses
  - Defines inspections
  - Defines demonstrations
  - Defines trade studies
  - Defines tests
  - Defines durability/economic life plan
  - Defines materials characterization plan
  - Defines design criteria (design rules)
  - Defines damage tolerance control plan
  - Defines quality control/compliance plan
  - Defines corrosion prevention and control plan
  - Defines life manageament plan

Stage II - Preliminary Planning (Continued)

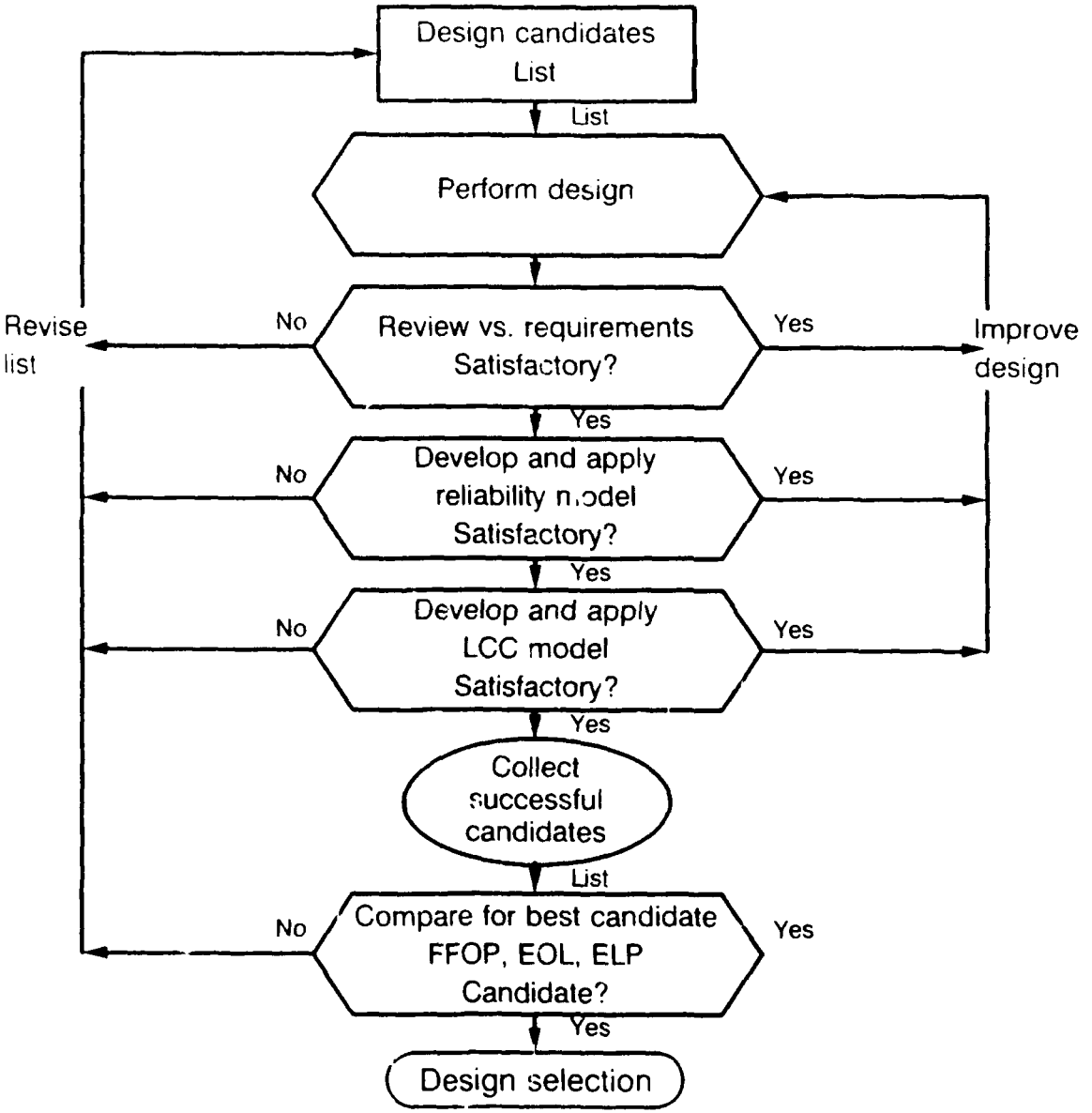
- Develop AVIP task descriptions
  - Determine required activities
    - Include requirement from MIL-STD-1796 by name or function
    - Maintains traceability
  - Develop milestones
    - Timely interactions and activities identified
    - Compliance check by thorough review of MIL-STD-1796 and MIL-A-87244

Stage II - Preliminary Planning (Continued)

- Develop program plan
  - Develop detailed schedules
  - Write work statements
  - Estimate resources required by AVIP
- Initiate task activities as necessary

Stage II - Preliminary Planning

(Continued)
AVIP Design
Process Flow



Stage III - Design

- Design criteria being applied
- Part selection policy established
- Trade studies being performed
- Design practices documented
- Design/development analysis being conducted
- Development tests initiated
- Manufacturing development initiated
- Recorders definition
- Avionic integrity data base initiated
- Electronic failure diagnosis studies underway

Stage III - Design (Continued)

Example of Early Discovery and Recovery

Discovery (Typical failure source)	Recovery (Areas for design solutions)
Thermal (cycles, extremes, gradients)	Material selection, matching layering, geometry, device selection, electronic technology family, thermal conduction and insulation
Shock and vibration	Mounting, structure (chassis, micro-electronic), mass distribution, external protection
Corrosion	Coatings, sealing, material selection, manufacturing processes
Damage	Redundancy, design margin, producibility, geometry
All of the above	Stress sensor, nonvolatile memory

AVIP Tasks Completed Prior to Preliminary Design Review

- Generate environmental analysis
- Design criteria established
- Life cycle cost model generated
- Sensitivity and tolerance model generated
- Several concepts evaluated
  - Requirements
  - Reliability
  - Life cycle cost
- Propose conceptual design
  - Preliminary reliability estimate and failure modes

AVIP Tasks Completed Prior to Detailed Design Review

- Component stress tests initiated
- Component stress response modeled
- Manufacturing processes documented
- Damage tolerance analysis complete
- Failure model finalized
- Reliability model finalized
- Economic life prediction
- Data base established

AVIP Tasks Completed Prior to First Delivery Milestones

- Failure review board established:
  - Design improvement
  - Update failure rate
  - Reliability strategy
- Design verification:
  - "Design verification" cycle tests
  - "Stress tolerance" tests
  - "Damage tolerance" verification
- Models updated:
  - Component stress
  - Tolerance analysis
  - Failure model
  - Reliability model
  - Life cost model
  - Economic life prediction

Stage IV - AVIP Compliance and Production

- Compliance verification
- Failure/reliability design improvements, stress recorders, acceptance test cycle, support definition, improved economic life prediction
- Manufacturing management
  - Quality control/compliance plan application

Stage V - Life Management

- Repair cycle definition, Expected Operating Life (EOL) prediction
- Data base operation

Avionics Integrity Data Base

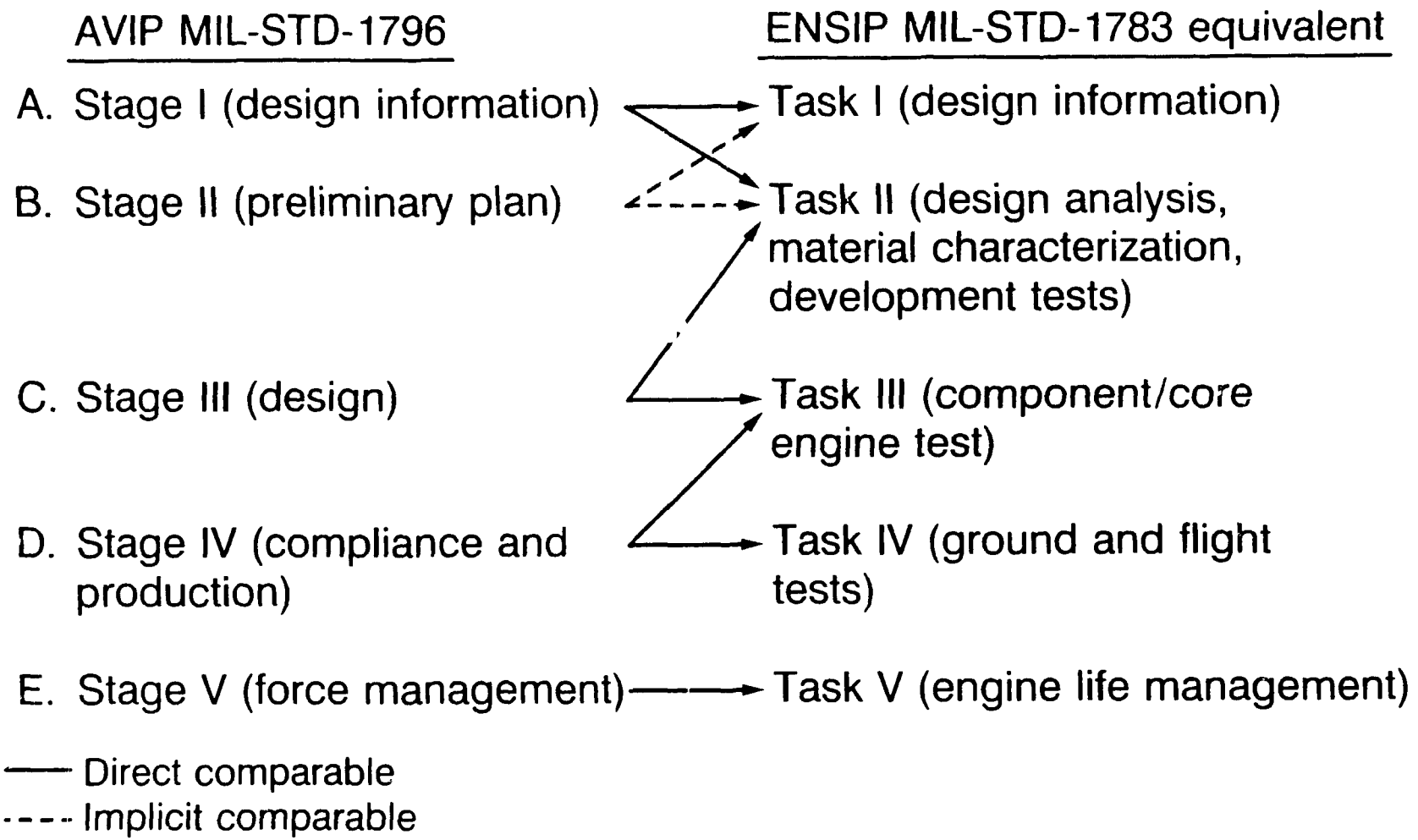
- Information
  - Design option trade study summaries
  - Procurement, assembly, and test records
  - All design information
  - All failure and operation trend data

- Uses
  - Design critique
  - Improve FFOP, EOL estimates
  - Data base for extending lifetimes
  - Data base for expanding manufacturing rate

Summary

- The AVIP development process is well ordered
- AVIP initiation proceeds through stage III
- AVIP parallels ENSIP during stages IV and V

Structural Comparison of AVIP to ENSIP



Comparison of ENSIP to AVIP Process

ENSIP	AVIP
• Few failure modes - Well known	• Many failure modes - Not all known - Many features
• Less complex design philosophy - Fewer design options - Fewer design solutions - Trade studies optimize design process - Tests validate selected design	• Complex design philosophy - Few design constraints - Many potential solutions - Trade studies optimize design process - Tests validate selected design
	• Life vs. stress and reliability vs. stress models
• Options for life/LCC/REL problems - Change physical parameters	• Options for life/LCC/REL problems - Use experience to update models - Redistribute tolerances - Change part selection process - Redesign part

Conclusion

- ENSIP and AVIP can naturally combine
  - ENSIP adds an experience base and stable structure to AVIP
    - AVIP tailors to ENSIP structure
  - AVIP adds flexibility and enhanced future responsive capability to ENSIP

GE Aircraft Engines

THE APPLICATION OF ENSIP TO THE F101-GE-102 AND F110-GE-100 ENGINES PRACTICAL PERSPECTIVES AND FUTURE NEEDS

1988 USAF
STRUCTURAL INTEGRITY CONFERENCE
NOVEMBER 29 THROUGH DECEMBER 1, 1988
SAN ANTONIO, TEXAS

R.D. Graman

AGENDA

- PRACTICAL ASPECTS OF THE SUCCESSFUL INCORPORATION OF ENSIP INTO THE F101-GE-102 AND F110-GE-100 ENGINES FOR 3 AREAS:
  - PRODUCTION ENHANCED INSPECTION IMPLEMENTATION
  - OVERHAUL DEPOT ENHANCED INSPECTION IMPLEMENATION
  - CONTROLS, ACCESSORIES & EXTERNALS (CA&E) APPLICATIONS
- HOW THESE APPLICATIONS OF ENSIP PAVE THE WAY FOR EXPANDED INTEGRITY PROGRAMS ... PPSIP
- FUTURE APPLICATIONS FOR MAXIMIZING BENEFITS

STEPS IN SUCCESSFUL PRACTICAL APPLICATION

1. CLEAR COMMUNICATION OF NEED

2. DEMONSTRATE COMMITMENT FOR SUPPORT - ADDED VALUE

3. ACTIVATION ⟶ ACCEPTANCE

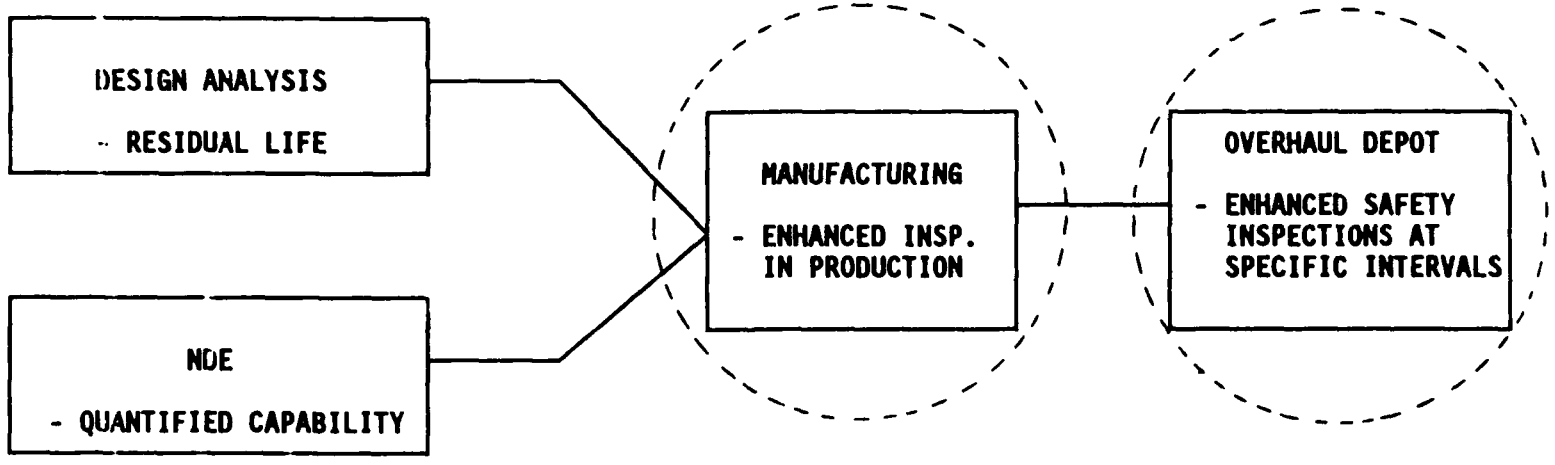
   - MULTI DISCIPLINED TEAMS

4. PAYOFF

   - IMPROVED PRODUCT
   - ESTABLISHED BASE TO PURSUE FUTURE INTEGRITY GOALS

DAMAGE TOLERANCE - THE MOST VISIBLE APPLICATION

- "CULTURAL" CHANGE REQUIRED



PRODUCTION INSPECTIONS - CHALLENGES

INSPECTION

- MULTIPLE SITES REQUIRED IMPLEMENTATION
- EQUIPMENT PROCUREMENT
- OPERATOR TRAINING, INSPECTION DEVELOPMENT

COMPONENT INSPECTABILITY

- PART SURFACE CONDITIONS
- PROCESSING REQUIREMENTS

REQUIRED DEDICATED EFFORT FROM MANUFACTURING, QUALITY, NDE, ENGINEERING

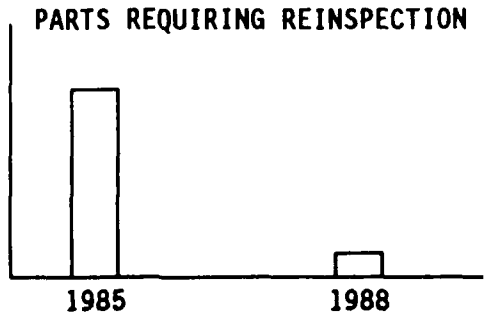
PRODUCTION INSPECTIONS - RESULTS

- BY ADDRESSING BOTH THE MANUFACTURING PROCESS AND NDE, SIGNIFICANT IMPROVEMENTS OBTAINED ...

MANUFACTURING PROCESS

- TOOL/FIXTURE DESIGNS & MATERIALS
- MULTI OPERATION MACHINING
- TOOL CHANGE FREQUENCY
- ETC.

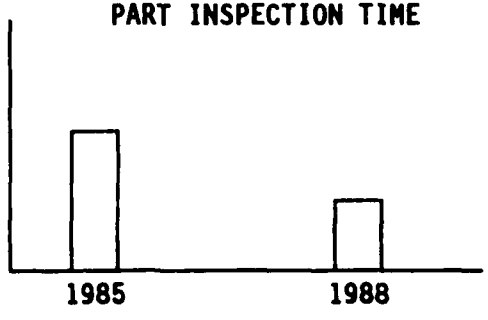
= %



NDE (EDDY CURRENT) PROCESS

- INCREASED SCANNING SPEEDS
- IMPROVED PROBE QUALITY
- IMPROVED INSTRUMENTS

= TIME



FOR TYPICAL PART, INSPECTION TIME DECREASED 50%, WHILE THE NUMBER OF PARTS REQUIRING REINSPECTION REDUCED BY FACTOR OF ~5

- AND STILL IMPROVING

PRODUCTION INSPECTION - PAYOFFS

- MANUFACTURING/PARTS HANDLING METHODS IMPROVED AND IMPLEMENTED AS STANDARD PROCEDURE

- PRODUCTION IMPLEMENTATION HELPED FACILITATE THE IMPLEMENTATION AT THE DEPOT

- PRODUCTION ENHANCED INSPECTIONS ESTABLISHED AS A WAY OF LIFE

- PRODUCT QUALITY IMPROVED

DEPOT INSPECTION IMPLEMENTATION - CHALLENGES

- RAPID FIELD CYCLIC ACCUMULATION FORCED COMPRESSED IMPLEMENTATION SCHEDULE

- JOINT OCALC/ASD/GE LEARNING PROCESS ... FIRST TIME REQUIREMENTS FOR:
  - TECH ORDER MANUALS FOR ENHANCED INSPECTIONS
  - SUPPORT EQUIPMENT APPLICATIONS

- FACILITIZATION OF DEPOT PRESENTED UNIQUE INSPECTION PROBLEMS
  - INSPECTION OF WELDED SPOOLS - INDIVIDUAL DISKS INSPECTED IN PRODUCTION

DEPOT INSPECTION IMPLEMENTATION - SOLUTIONS

- DEDICATED TEAM FORMED FROM ASD, OCALC, AND GE
- CLEAR GOALS OUTLINED AND INDIVIDUAL TASKS PERFORMED
- BUILT UPON PRODUCTION AND TF34 INSPECTION IMPLEMENTATION EXPERIENCE
  - EARLY INVOLVEMENT OF TECHNICAL PUBLICATIONS PERSONNEL
- USAF AND GE MANAGEMENT COMMITMENT TO MAKE THIS HAPPEN

DEPOT INSPECTION IMPLEMENTATION - RESULTS/BENEFITS

- VALIDATED/VERIFIED TECH ORDERS AND QUALIFIED SUPPORT EQUIPMENT FOR THE F110 SUPPLIED TO OCALC IN 14 MONTHS - REDUCED FROM A 24+ MONTH INITIAL SCHEDULE
  - 14 DIFFERENT COMPONENTS
  - EDDY CURRENT AND SURFACE WAVE ULTRASONIC

- PAYOFFS
  - F110 DEPOT ENHANCED INSPECTIONS CURRENTLY BEING PERFORMED
  - CAPABILITY AVAILABLE TO SUPPORT F110 PACER PROGRAM
  - ESTABLISHED DEPOT AWARENESS OF INSPECTIONS

CONTROLS, ACCESSORIES & EXTERNALS - CHALLENGES

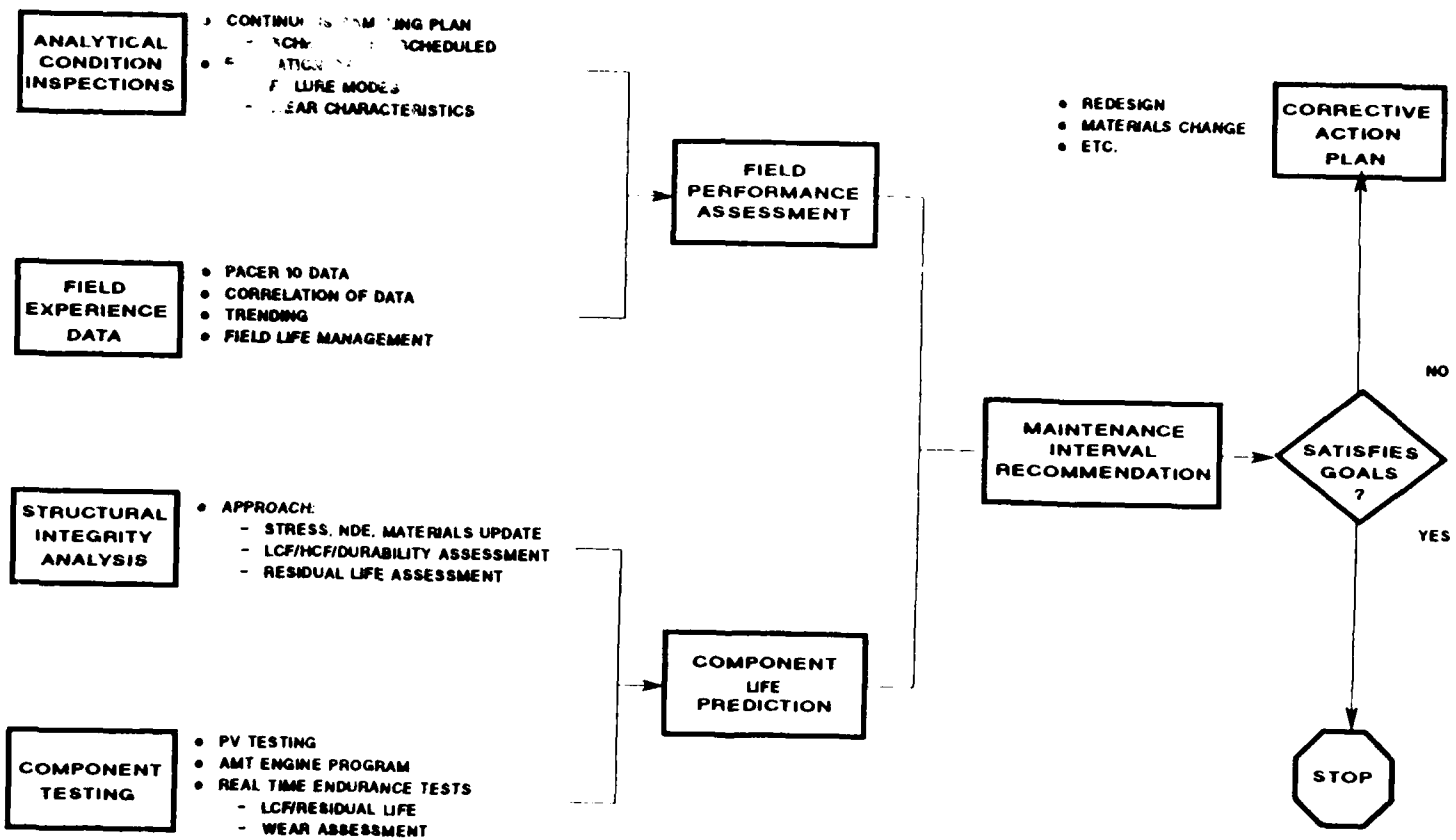
- INITIAL INVOLVEMENT RESULTED FROM FIELD INVESTIGATIONS

- REALIZED EARLY ON THAT MODIFIED ENSIP APPROACH WAS REQUIRED
  - DIFFERENT TYPES OF FAILURE MODES
  - SYSTEMS, NOT JUST INDIVIDUAL COMPONENTS MUST BE ADDRESSED

- TECHNICAL APPROACH NEEDED TO ENCOMPASS GENERAL WORKSCOPE PLUS COMPONENT SPECIFIC EFFORTS

CONTROLS, ACCESSORIES & EXTERNALS - ACTIONS

- DEVELOPED WORK SCOPE BALANCING A COMBINATION OF ANALYSIS AND TESTING



CONTROLS, ACCESSORIES & EXTERNALS - RESULTS

- APPLICATION OF FRACTURE MECHANICS
  - ACCESSORIES
  - TUBING
- DEVELOPMENT OF COMPREHENSIVE COMPONENT TEST PLANS, EMPHASIZING FIELD MISSION
- NDE CAPABILITY ASSESSMENTS
- GENERATION OF MATERIALS CHARACTERIZATION DATA
- INITIATED "PROTOTYPE" VENDOR INVOLVEMENT PLANS

CONTROLS, ACCESSORIES & EXTERNALS - BENEFITS

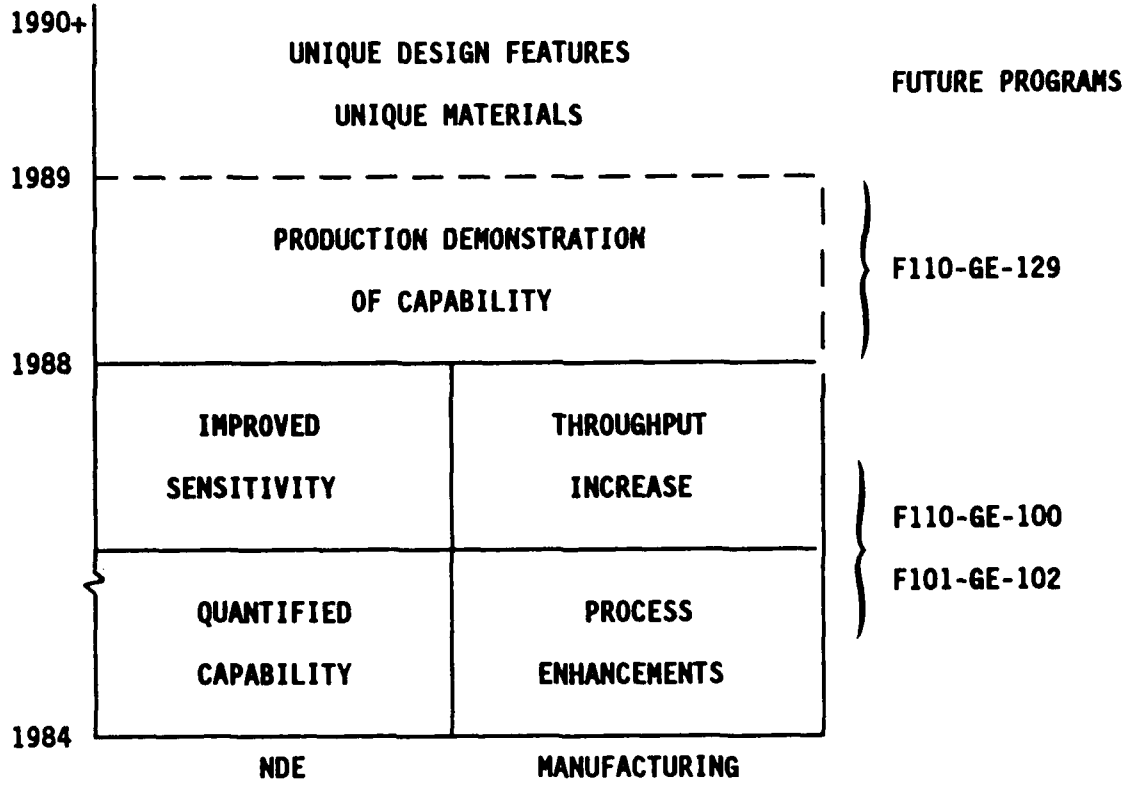
- CA&E ACTIVITIES ON THE F110-GE-100 TO DATE HAVE SUPPLIED;
  - SUPPORT FOR LIFE LIMIT GROWTH
  - QUALITY STRUCTURAL DATA FOR FIELD PROBLEM RESOLUTION

- HAVE TARGETED AREAS FOR IMPROVEMENT IN FUTURE PROGRAMS WITHIN THE BASIC DESIGN PROCESS
  - MATERIAL SELECTION/CHARACTERIZATION
  - QUALITY CONTROL/NDE
  - "SMART" TESTING
  - CONSISTENT VENDOR LIFE ANALYSIS PRACTICES

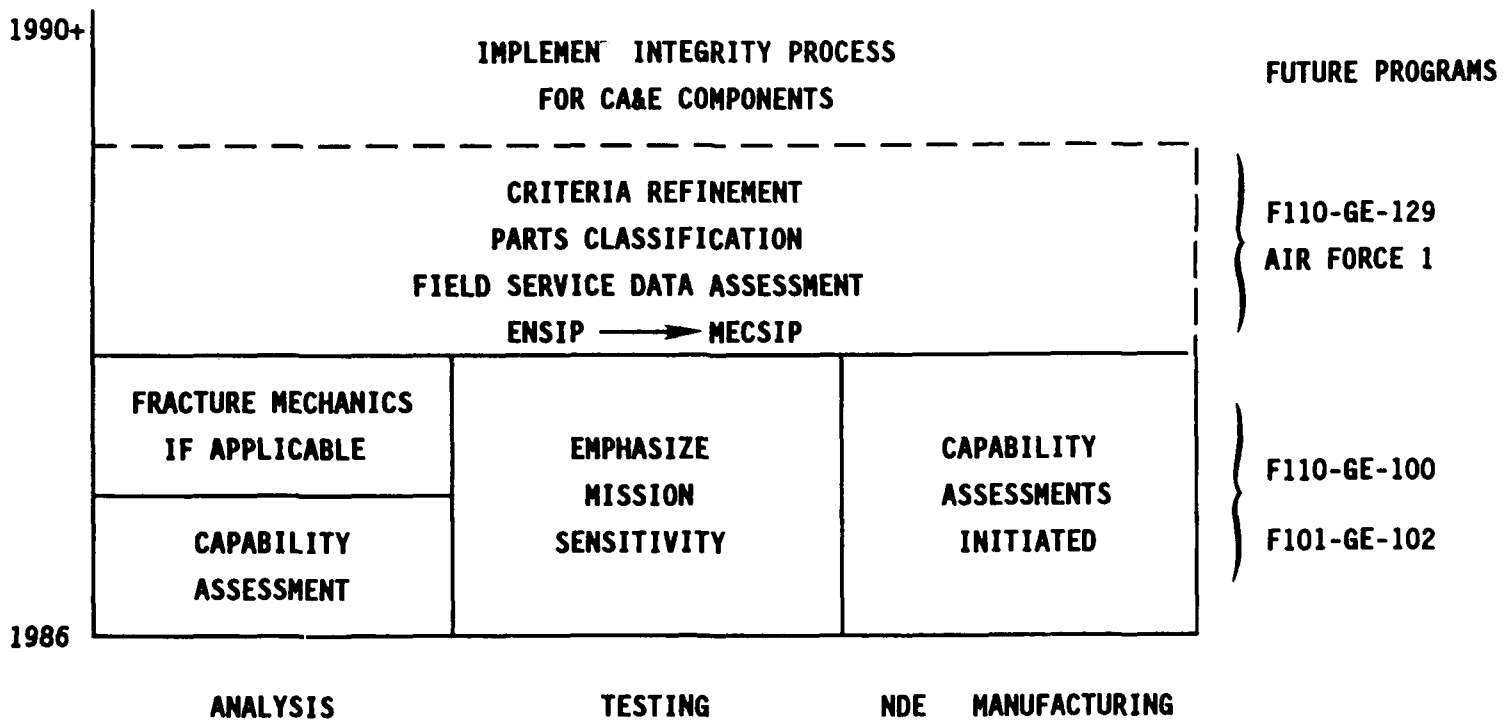
- HAVE INITIATED GEAE - SUPPLIER INTEGRATION

- ENSIP APPLICATION HAS BUILT SOLID FOUNDATION FOR CONTINUING EFFORTS TO MEET FUTURE CHALLENGES FOR;

"STRUCTURAL" COMPONENTS

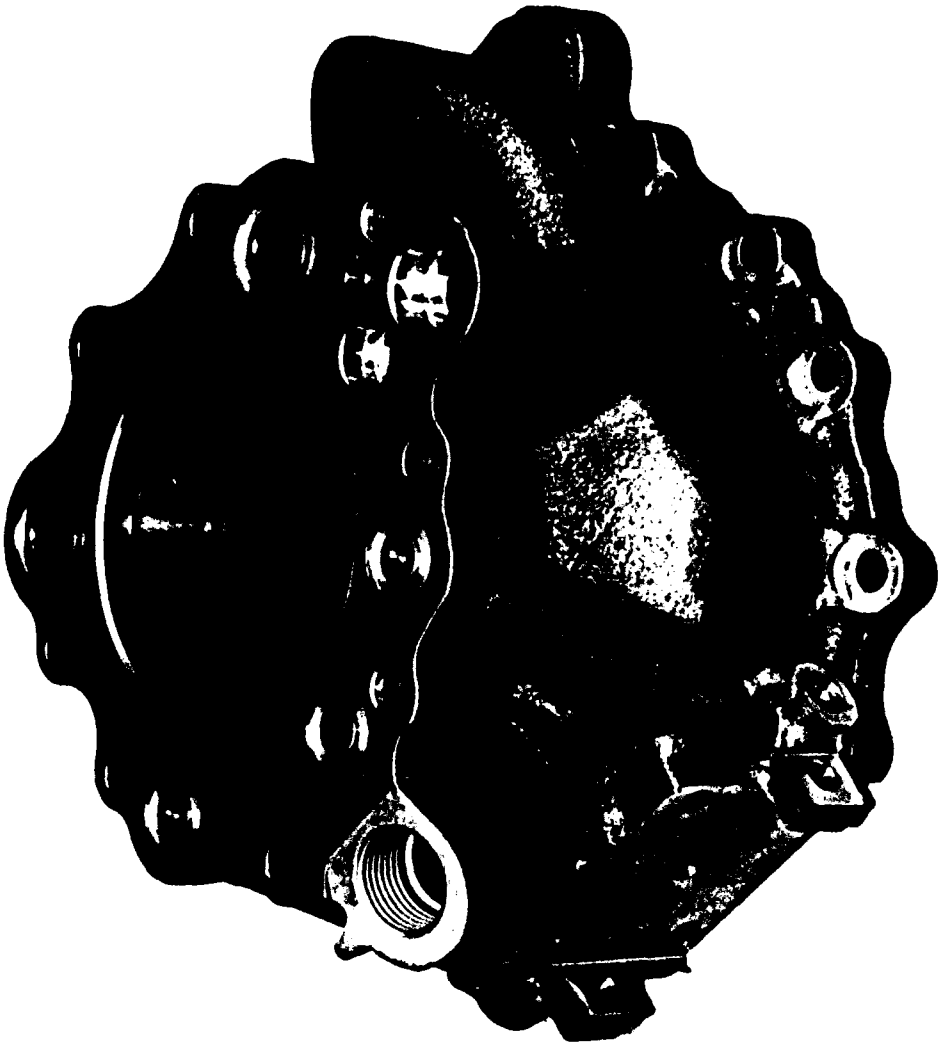


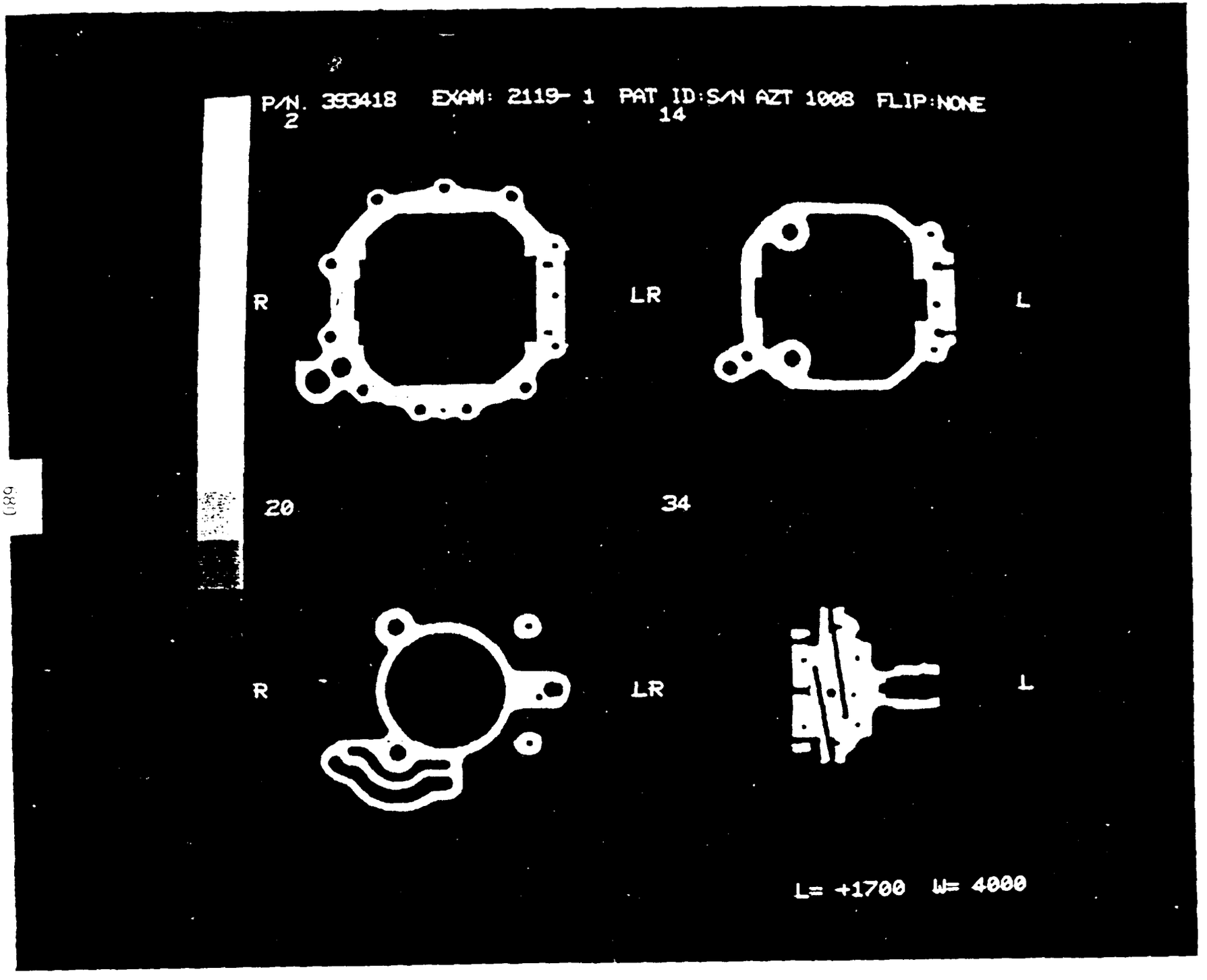
AND FOR CONTROLS, ACCESSORIES AND EXTERNALS



FUTURE NEEDS/APPLICATIONS

- EDDY CURRENT SIGNAL PROCESSING/FLAW CHARACTERIZATION
  - NECESSARY TO ALLOW INSPECTION OF SOME GEOMETRIC FEATURES
  - REDUCE THE BURDEN OF INTERPRETATION OF INSPECTION RESULTS
- APPLICATION OF CATSCANNING TO COMPLEX CASTINGS
  - QUALITY CONTROL
  - ANALYSIS
- CONTINUING NEED FOR CUSTOMER - SUPPLIER INTERFACE TO ENSURE MAXIMIZED BENEFITS
  - USAF-GEAE
  - GEAE-SUPPLIERS

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SUMMARY

- STRUCTURAL INTEGRITY PHILOSOPHY CAN, AND HAS BEEN, APPLIED IN A PRACTICAL MANNER TO PRODUCTION ENGINES

- THIS APPLICATION HAS RESULTED IN:

  - LESSONS LEARNED

    EARLY INVOLVEMENT BY ALL PARTIES INVOLVED IS KEY

  - FURTHER NEEDS/APPLICATIONS IDENTIFIED

    BEING PURSUED AS PART OF CONTINUING LIFE MANAGEMENT EFFORTS

  - PAYOFFS FOR TODAY AND THE FUTURE

    IMPROVED CURRENT PRODUCT QUALITY
    ESTABLISHING A FOUNDATION FOR FUTURE INTEGRITY PROGRAMS

- GE AIRCRAFT ENGINES IS WORKING TO APPLY STRUCTURAL INTEGRITY PROGRAMS TO MAXIMIZE CUSTOMER LIFE CYCLE COST BENEFITS

STRUCTURAL INTEGRITY PROGRAM

- NDE APPLICATION AND DEVELOPMENT -

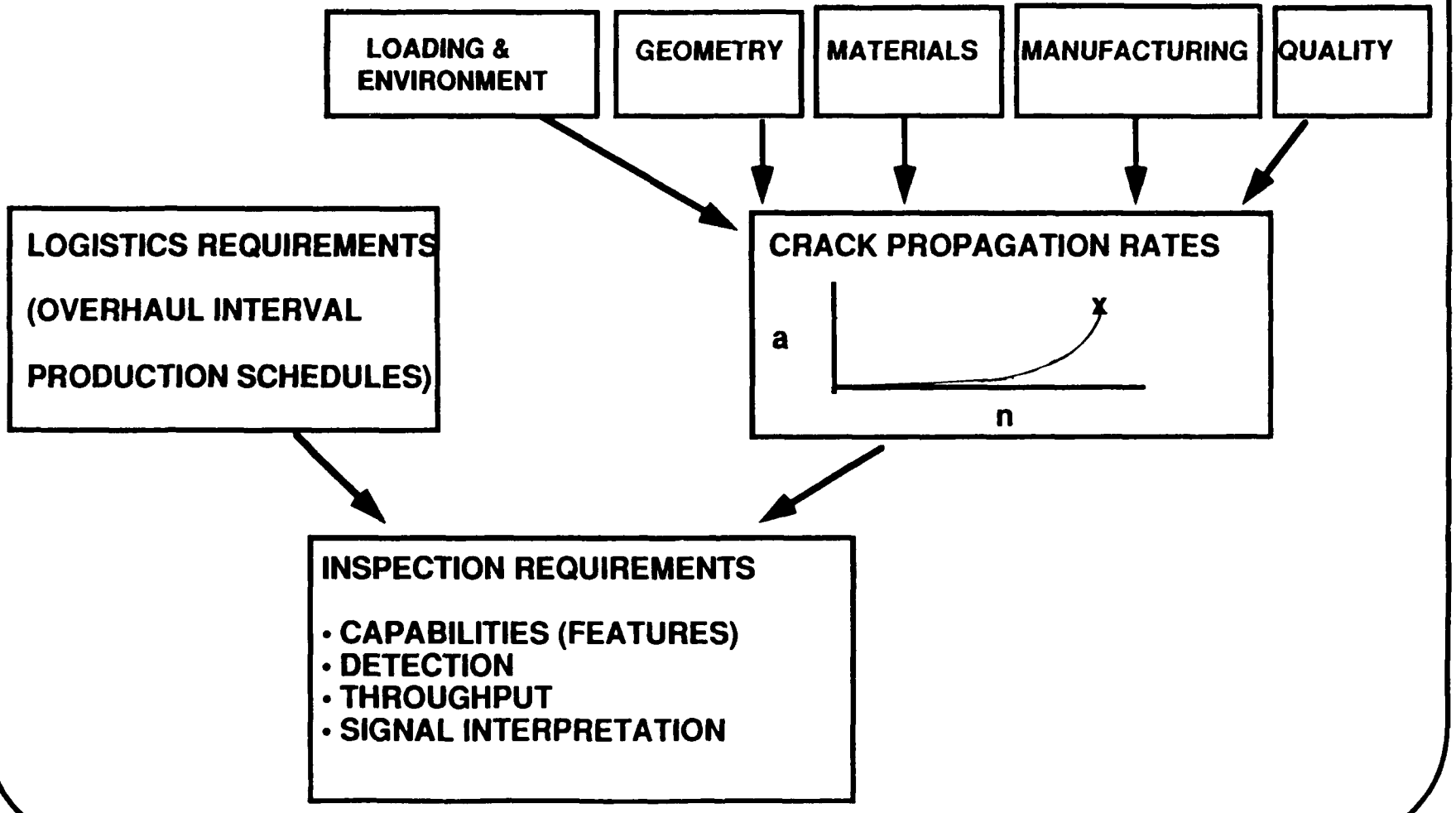
1988 USAF STRUCTURAL INTEGRITY CONFERENCE
NOVEMBER 29 - DECEMBER 1, 1988
SAN ANTONIO, TEXAS

W.L. HERRON
GENERAL ELECTRIC COMPANY
AIRCRAFT ENGINES
CINCINNATI, OHIO

OUTLINE

- INSPECTION REQUIREMENTS
- NDE PROGRESS:
  - CAPABILITIES
  - DETECTABILITY
  - THROUGHPUT
  - INTERPRETATION
- IMPACT OF MECSIP ON NDE
  - (AND VICE-VERSA)

INSPECTION DEFINITION



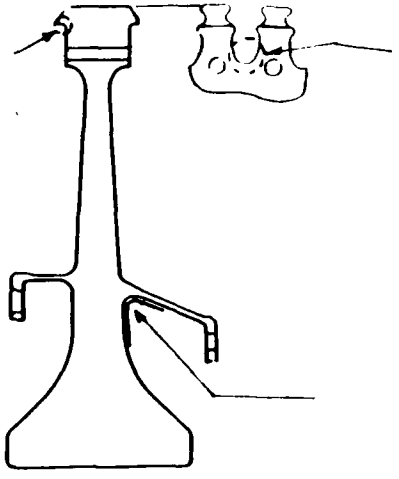
CHALLENGE

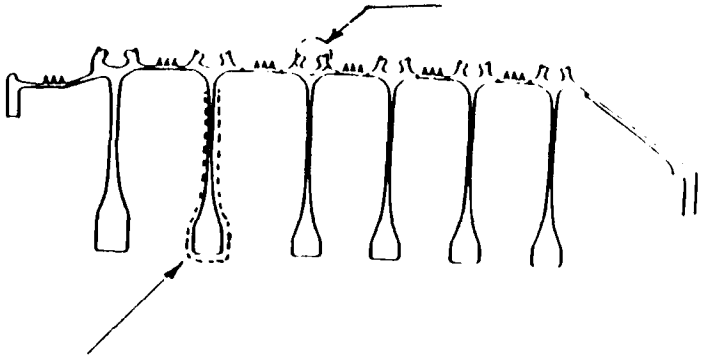
MEET THESE INSPECTION REQUIREMENTS FOR

INCREASINGLY DEMANDING APPLICATIONS

CAPABILITIES (FEAURES)

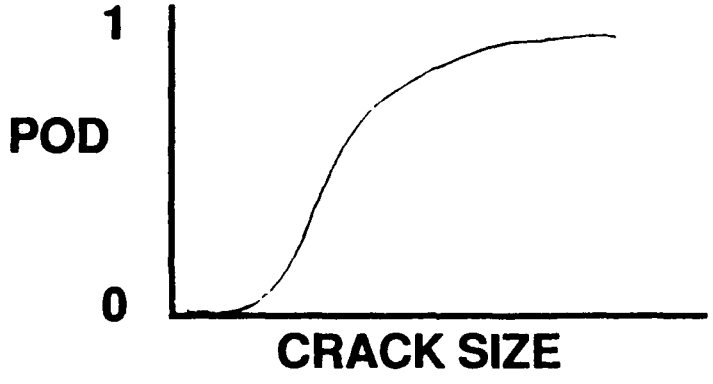
- PART FEATURES MAY PRESENT DIFFICULTIES WITH NON-RELEVANT (GEOMETRY) SIGNALS, MANIPULATION REQUIREMENTS, ACCESSIBILITY
- SOLUTIONS MAY ENTAIL PROBE DESIGN, INSPECTION PROCEDURE, SIGNAL PROCESSING, OR A COMBINATION OF ALL THESE





DETECTION

- DETECTION CAPABILITIES ARE EXPRESSED IN TERMS OF PROBABILITY OF DETECTION (POD)

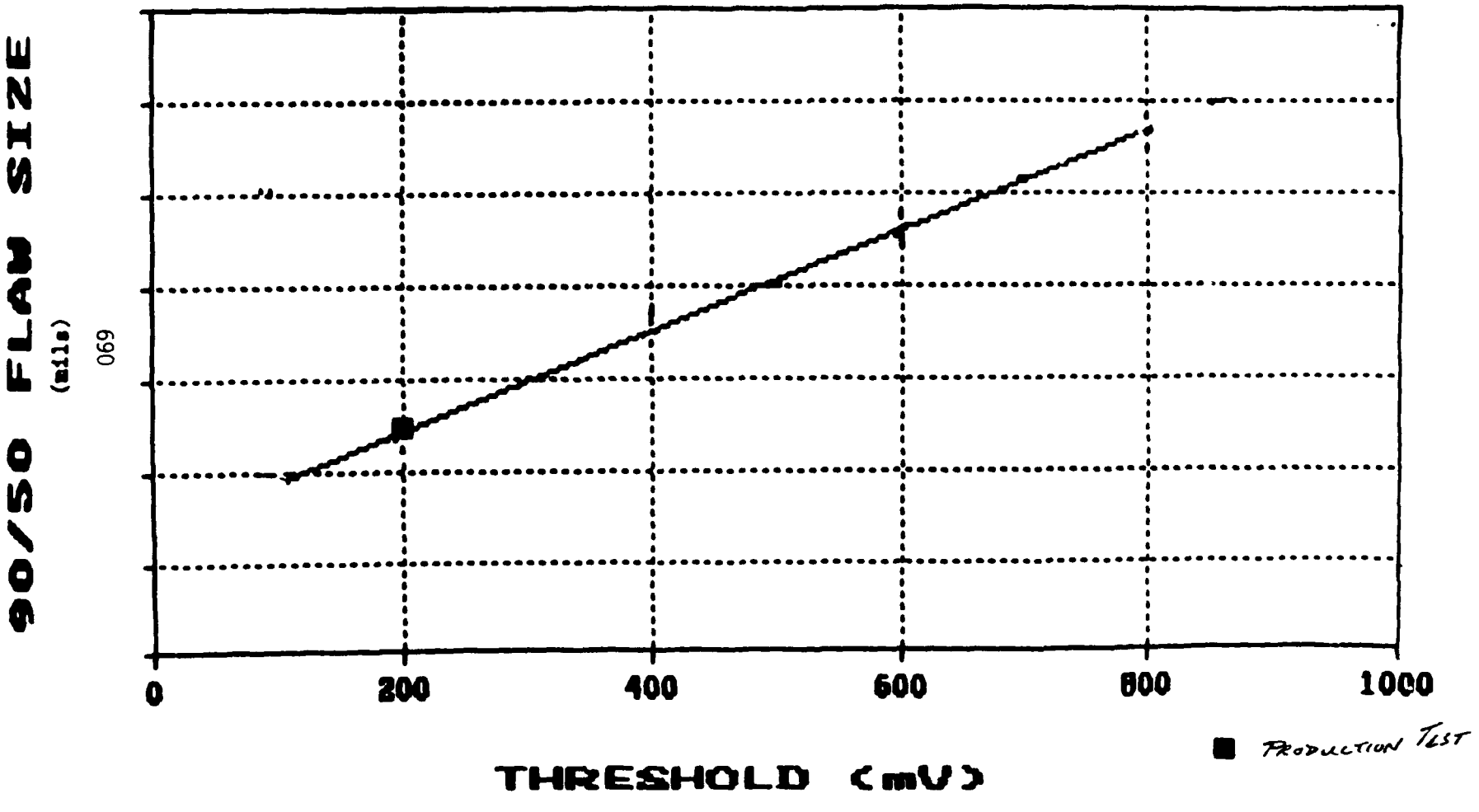


- FLAW SIZE DETECTION REQUIREMENTS VARY WITH PART AND FEATURE
- CONVENTIONAL FLUORESCENT PENETRANT INSPECTION (FPI) IS SUFFICIENT FOR MOST
- AUTOMATED EDDY CURRENT INSPECTION IS USED WHERE FPI IS NOT ADEQUATE

DETECTION

- PRODUCTION CAPABILITY IS DEMONSTRATED BY RIGOROUS TEST PROGRAM CONSIDERING ALL INSPECTION VARIABLES:
  - OPERATOR
  - INSPECTION SITE
  - FEATURE GEOMETRY AND TOLERANCES
  - PART FIXTURING
  - INSTRUMENTATION/MATERIALS
  - PART MATERIAL
- GE IS NOW CONDUCTING THIS DEMONSTRATION PROGRAM IN ITS PRODUCTION FACILITIES FOR FPI, EC AND UT

GEAE EDDY CURRENT INSPECTION CAPABILITY



DETECTION

- SINCE THE TF34 PROGRAM, GE HAS SUBSTANTIALLY IMPROVED EDDY CURRENT DETECTION CAPABILITIES:
  - MANY PROBE AND COIL TYPES HAVE BEEN EVALUATED:

SIDE MOUNT COILS	REFLECTION
WIDE COILS	DUAL ABSOLUTE
TORROIDAL	NON-CONTACT
QUAD DIFFERENTIAL	SHEATHED

SOME OF THESE HAVE BEEN ADOPTED, OTHERS ARE INADEQUATE

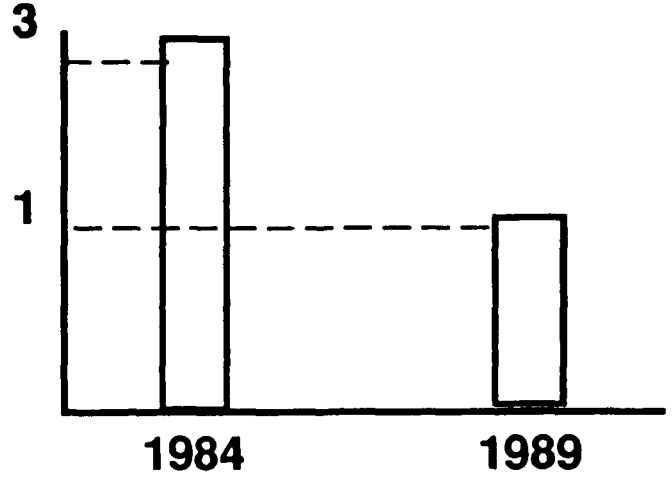
  - TIGHTER CONTROL OF PROBE QUALITY HAS ENHANCED RELIABILITY:

LIST OF APPROVED VENDORS
PROBE BUYING SPECIFICATION

DETECTION

- OTHER IMPROVEMENTS HAVE BEEN IDENTIFIED AND ARE BEING IMPLEMENTED:
  - NEW INSTRUMENTATION HAS BEEN INTRODUCED
  - FULLY AUTOMATIC DATA HANDLING IS BEING INSTALLED ON SEMI-AUTOMATED STATIONS
  - IMPROVED PARTS PROCESSING/HANDLING ALLOWS REVISED CALIBRATION
- KEY CRITERION: WHAT CAN BE PRACTICALLY IMPLEMENTED

NORMALIZED 90/50 POD (PRODUCTION)



THROUGHPUT

- SIGNIFICANT EDDY CURRENT INSPECTION AND PARTS PROCESSING IMPROVEMENTS HAVE BEEN IMPLEMENTED:
  - SCAN SPEED INCREASED 3X (5X HAS BEEN APPROVED)
  - SCAN INDEX INCREASED 50% FOR SOME FEATURES
  - CALIBRATION PROCEDURES/REQUIREMENTS REVISED
  - TAPELESS CONTACT PROBES IMPLEMENTED
  - PROTOTYPE SHIMLESS CONTACT PROBE FOR BOLTHOLES DEVELOPED
  - OTHER APPROVED ENHANCEMENTS NOT YET FULLY IMPLEMENTED

THROUGHPUT

- AUTOMATIC FPI
  - GOAL: .030" 90/50 POD (CRACK LENGTH)

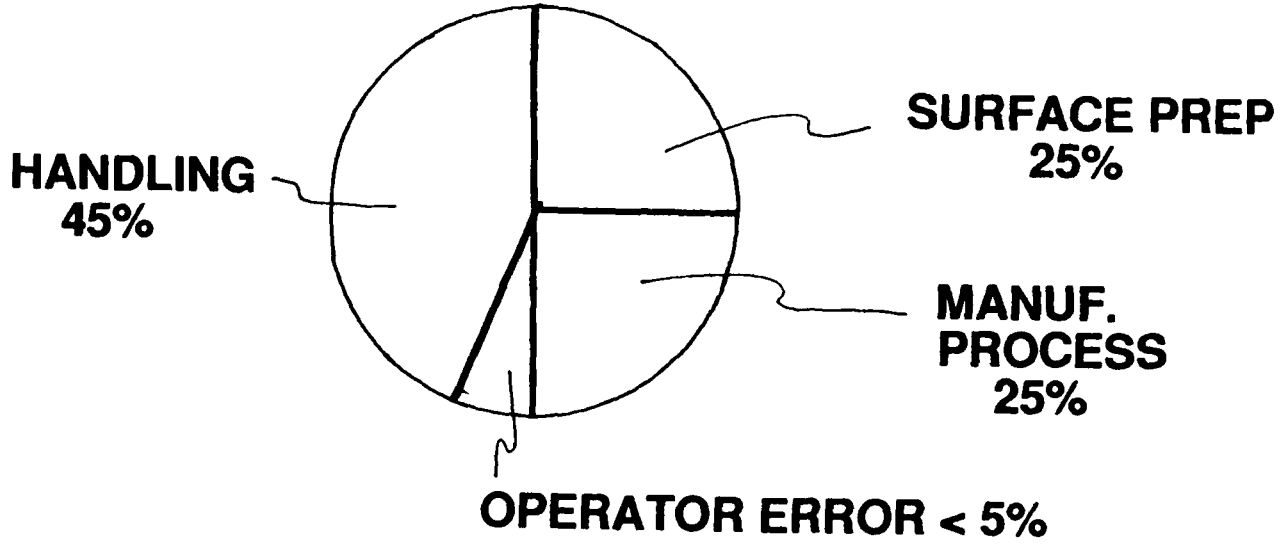
    4 SQUARE INCHES/SEC INSPECTION
  - STATUS: PROTOTYPE SYSTEM IN TEST

    POD IN RANGE OF GOALS

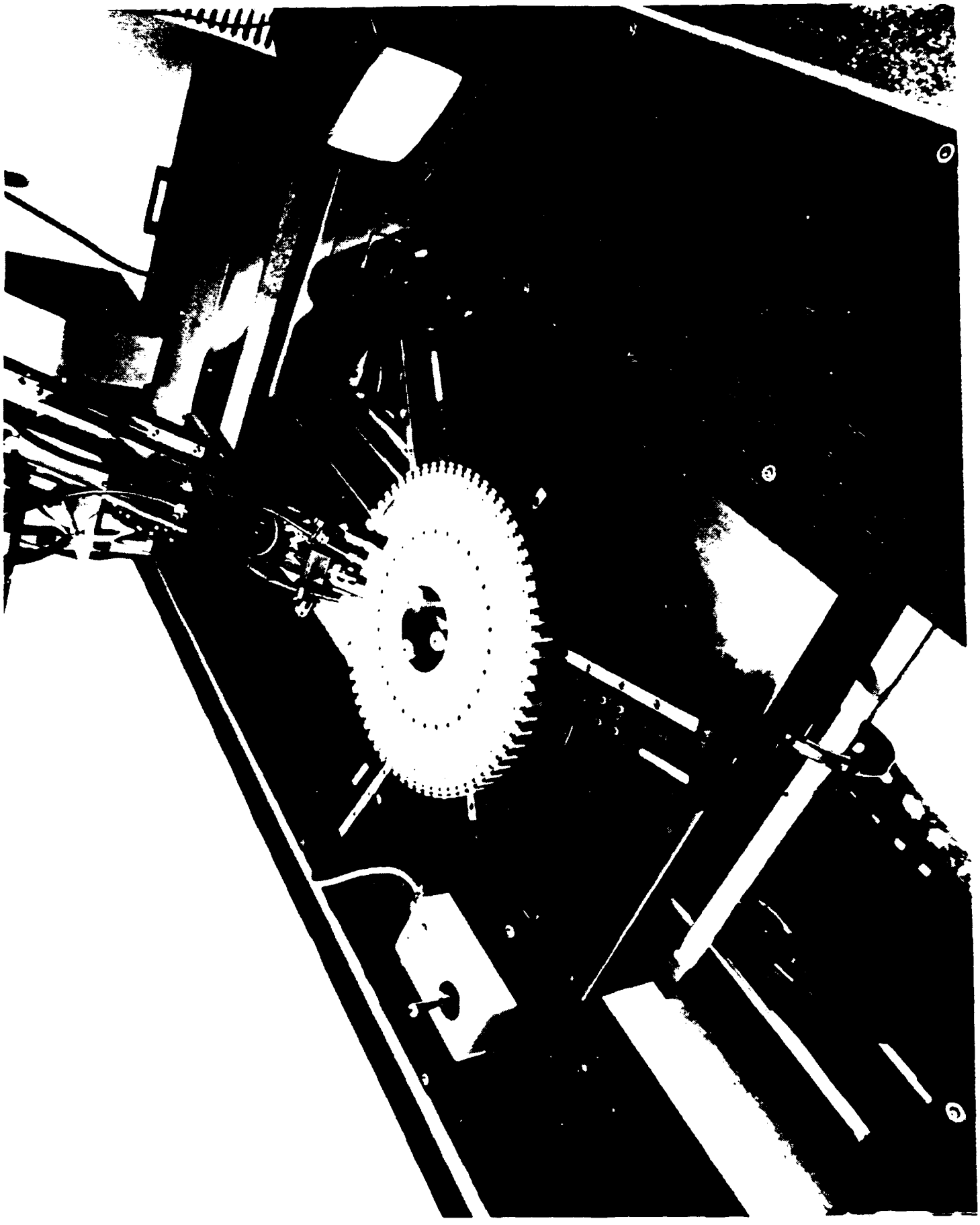
    ~ 2% OVERCALL RATE ON LCF SPECIMENS
  - PRODUCTION SYSTEM DEMO SCHEDULED 4TH QUARTER 1989

SIGNAL PROCESSING

- MANUFACTURING EXPERIENCE (EDDY CURRENT REJECTS)

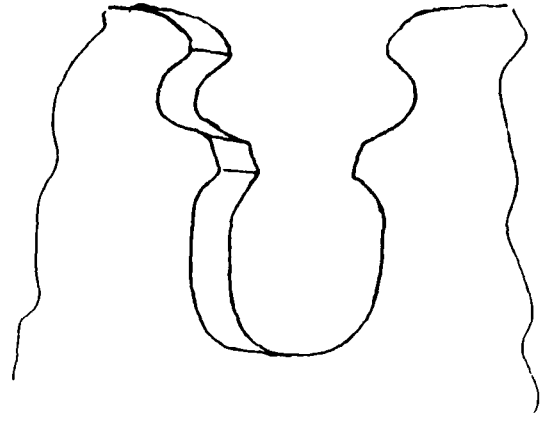


- GOAL: CENSOR NON-RELEVANT SIGNALS
  - PART GEOMETRY
  - IMMATERIAL MACHINING MARKS
  - PART CLEANLINESS



SIGNAL PROCESSING

- PROCEDURES: SIGNAL PATTERN RECOGNITION
  - 1 DIMENSIONAL (WAVEFORM ANALYSIS)
  - 2 DIMENSIONAL (IMAGING)
- EXAMPLE:



MENU
IMAGE DISPLAY
COLORBAR

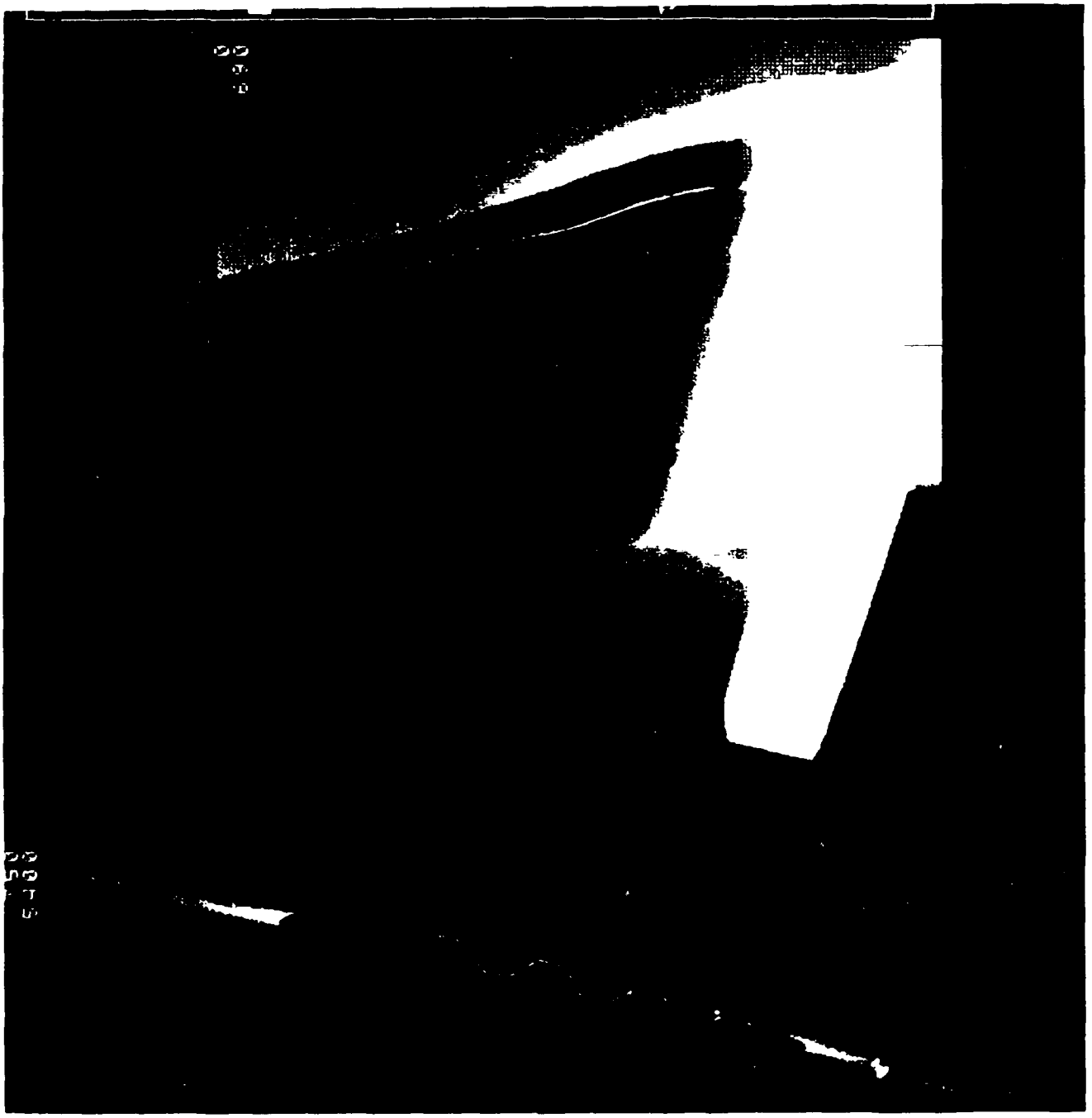
MECSIP

- INCLUSION OF C&A IN STRUCTURAL INTEGRITY PROGRAMS

  RE-EMPHASIZES NDE:

  - TUBE INSPECTIONS EC, UT, FPI
  - MPI, UT FOR GEARS
  - DIMENSIONAL, THICKNESS TESTING
  - CT SCANNING OF BLADES, PUMPS, GEARBOXES

- SOME NEW DESIGN FREEDOMS OPENING UP



SUMMARY

- IMPLEMENTATION OF ENSIP HAS HAD SUBSTANTIAL IMPACT ON NDE:
  - IMPROVED CAPABILITY
    - DETECTION
    - THROUGHPUT
    - INTERPRETATION
  - PRODUCTION EVALUATIONS CRITICAL FOR REALISTIC ASSESSMENT
- NDE DEVELOPMENT HAS INTRODUCED CONSTRAINTS AND FREEDOMS TO THE DESIGN:
  - SOME FEATURES NOT YET PRACTICALLY INSPECTABLE
  - NEW CAPABILITIES ALLOW NEW DESIGNS

ALLIED-SIGNAL AEROSPACE COMPANY
GARRETT ENGINE DIVISION

F109-GA-100 ENGINE ACCELERATED MISSION (AMT) TESTING AND CONDITION MONITORING

1988 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE

HANS MAERTINS
TFE/F109 PROJECT

Allied-Signal Aerospace Company
Garrett Engine Division

AGENDA

- F109 ENGINE OVERVIEW
- ENGINE AMT VERIFICATION TESTING
- ENGINE CONDITION MONITORING

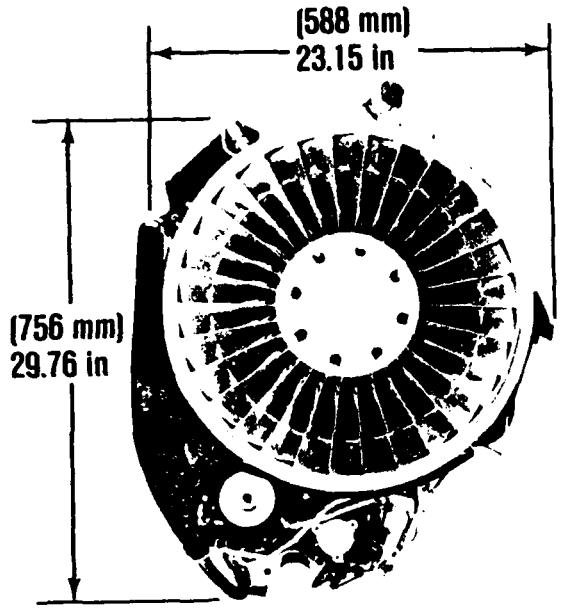
703

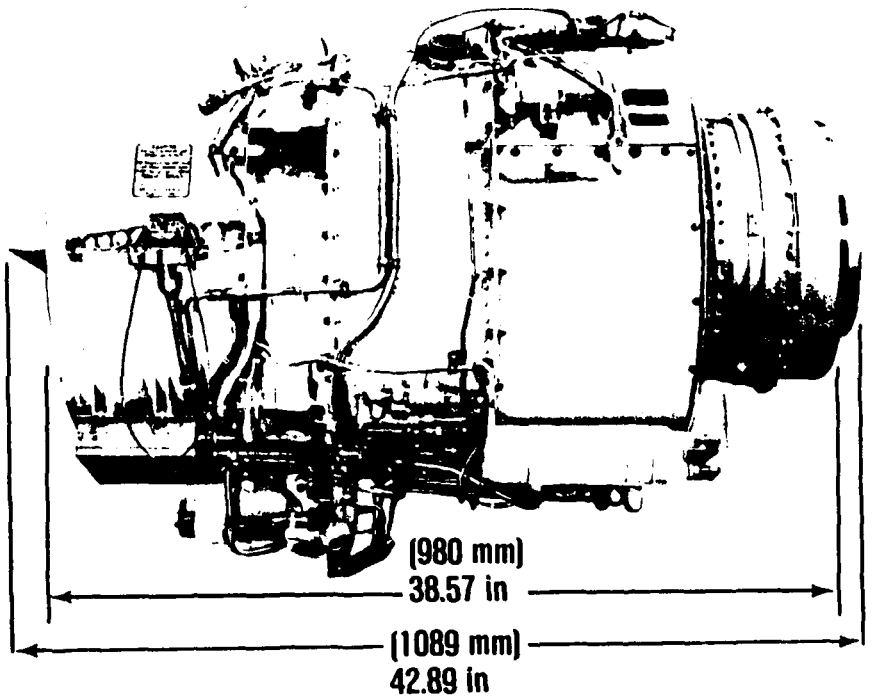
F109 ENGINE OVERVIEW

F109 ENGINE DEVELOPED TO MEET NEEDS OF AFFORDABLE JET TRAINERS

- PROVIDES JET OPERATIONAL CHARACTERISTICS
- FUEL ECONOMY EXCEEDS THAT OF TURBOPROP TRAINER
- 18,000-HOUR DESIGN LIFE
- DEMONSTRATED DURABILITY/RELIABILITY/AVAILABILITY
- UNPARALLELED SAFETY
- LOW OPERATING COSTS WITH FLEXIBLE MAINTENANCE OPTIONS

F109 ENGINE DESIGNED SPECIFICALLY FOR NEW PRIMARY TRAINER





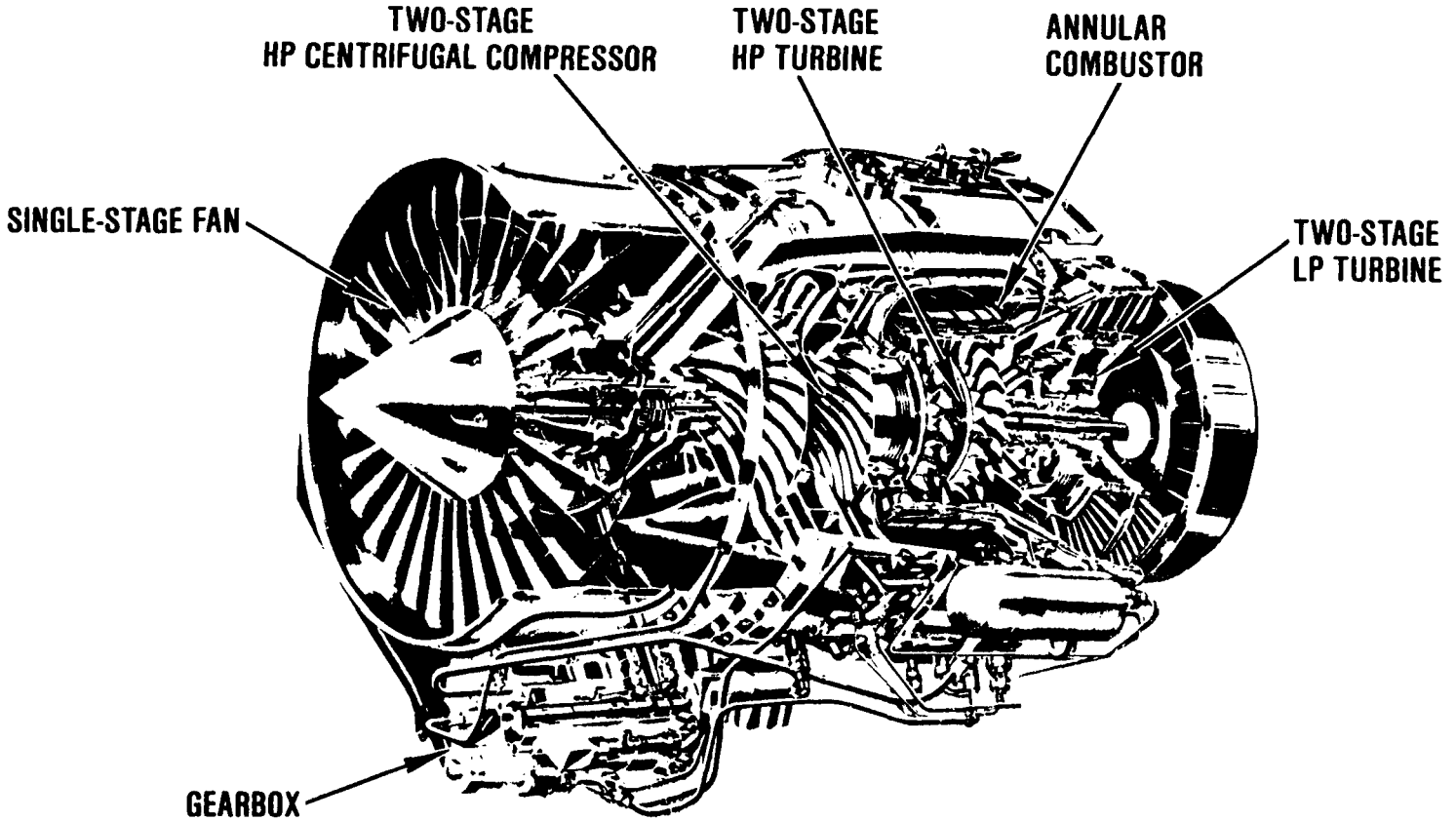
- THRUST — 1330 LB SLS-ISA
- TSFC — 0.392 LB/HR/LB
- BYPASS RATIO — 5:1
- ENGINE WEIGHT — 439 LB* (199 KG)

*INCLUDES EFCU WEIGHT OF 15.2 LB

G8223-5A

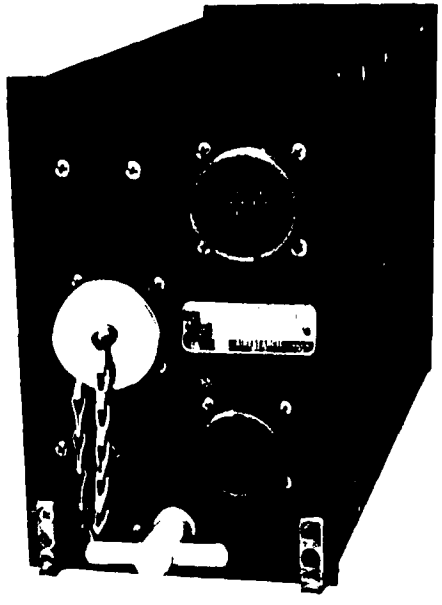
Allied-Signal Aerospace Company
Garrett Engine Division
Allied Signal

F109 CONFIGURATION COMBINES HIGH PERFORMANCE WITH EXCELLENT DURABILITY

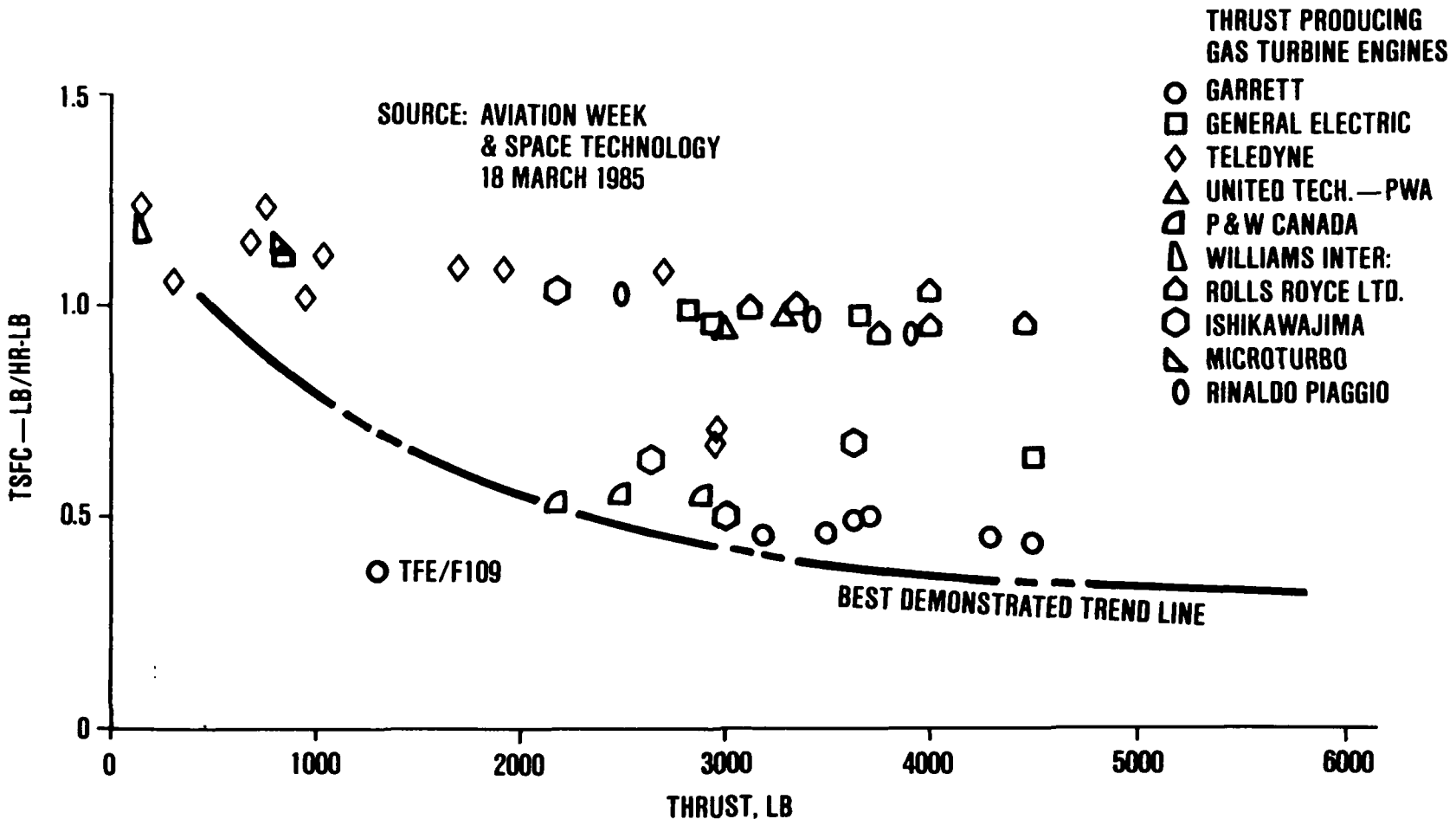


ELECTRONIC FUEL CONTROL UNIT (EFCU) MINIMIZES PILOT WORKLOAD AND PROVIDES MAINTAINABILITY BENEFITS

- EASY TO USE—
  - AUTOMATIC START AND RESTART
  - AUTOMATIC OVERSPEED/TEMPERATURE LIMITING
  - SIMPLE POWER MANAGEMENT
  - AUTOMATIC THRUST TRIM
- EASY TO MAINTAIN—
  - EXTENSIVE BUILT-IN TEST
  - FAILURE ACCOMMODATION
  - FAILURE LOGGING
- SUPPORTS EFFECTIVE LIFE MANAGEMENT—
  - OPERATIONAL HISTORY
  - TREND MONITORING
  - GROUND STATION COMPATIBILITY



F109 FUEL CONSUMPTION IS IN A CLASS BY ITSELF



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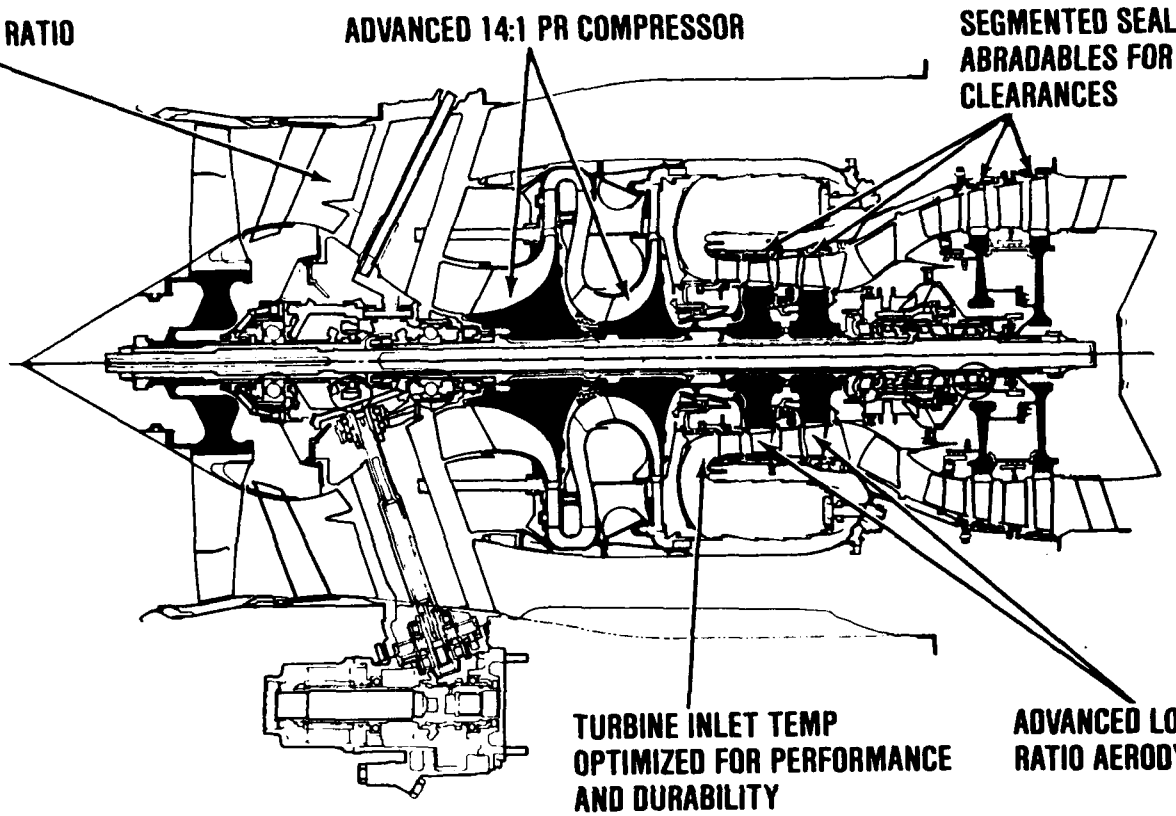
F109 IS DESIGNED FOR OPTIMUM FUEL ECONOMY

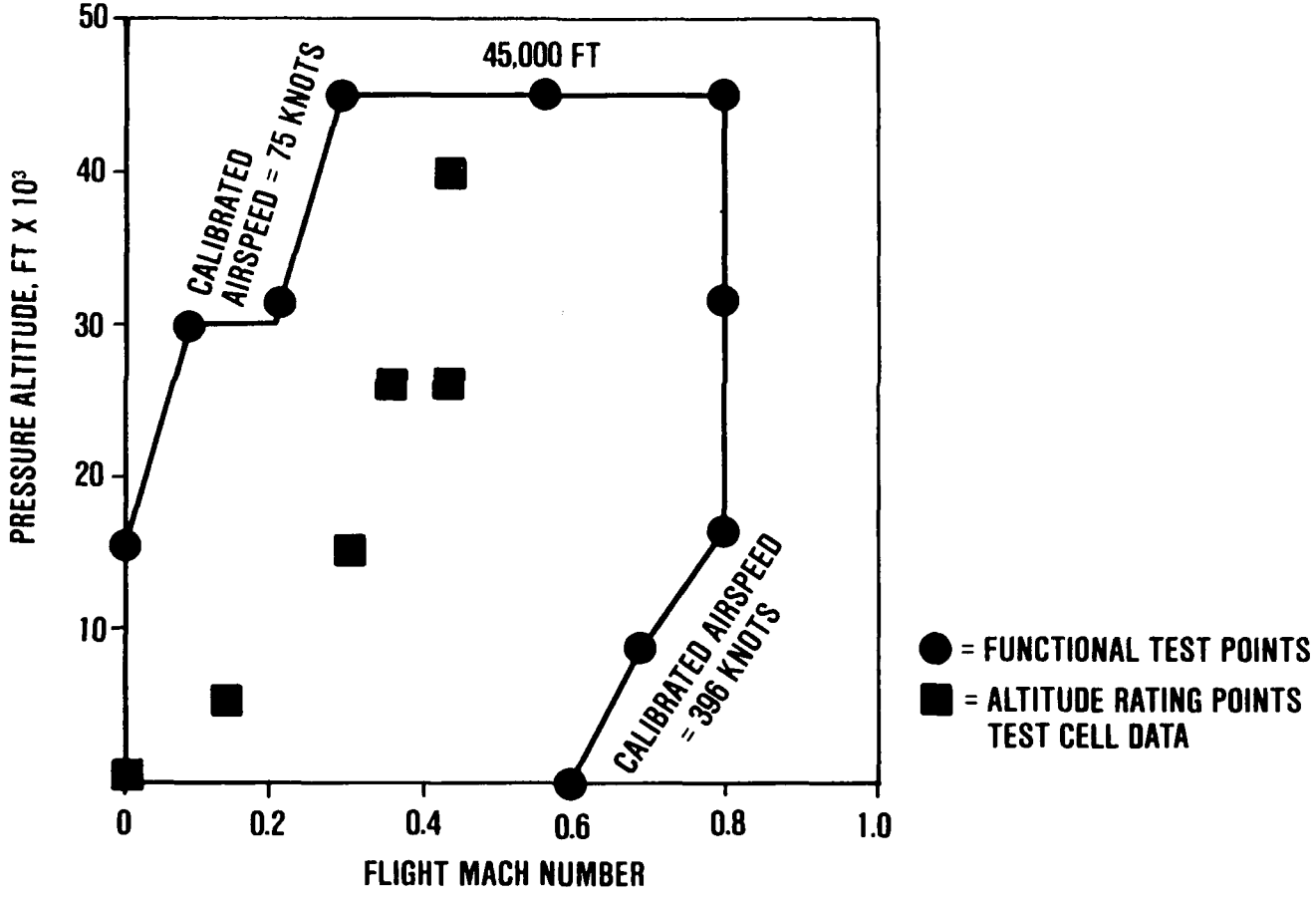


Allied-Signal Aerospace Company
Garrett Engine Division

Allied Signal

68223-89A

F109 PROVIDES FOR EXPANDED TRAINING ENVELOPE



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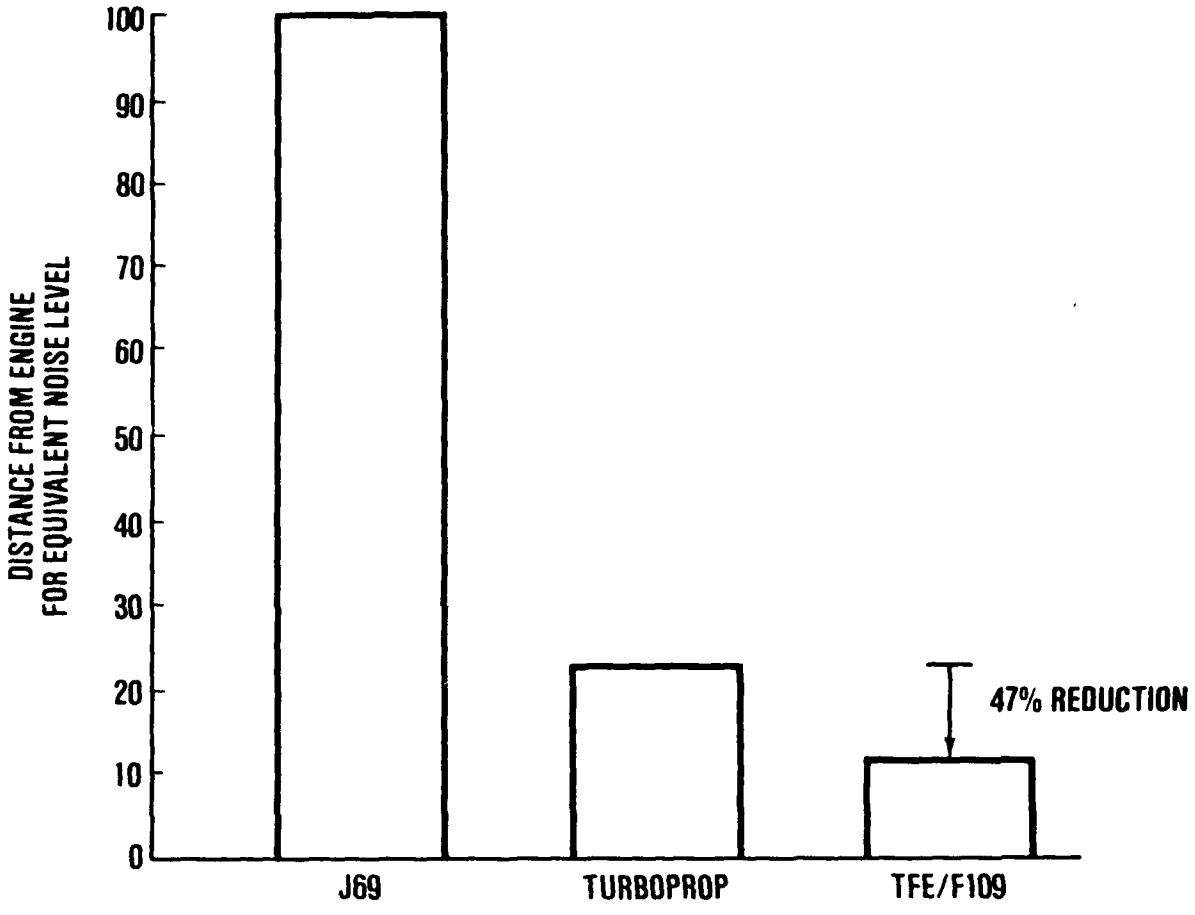
F109 ENGINE OFFERS DRAMATIC NOISE REDUCTION



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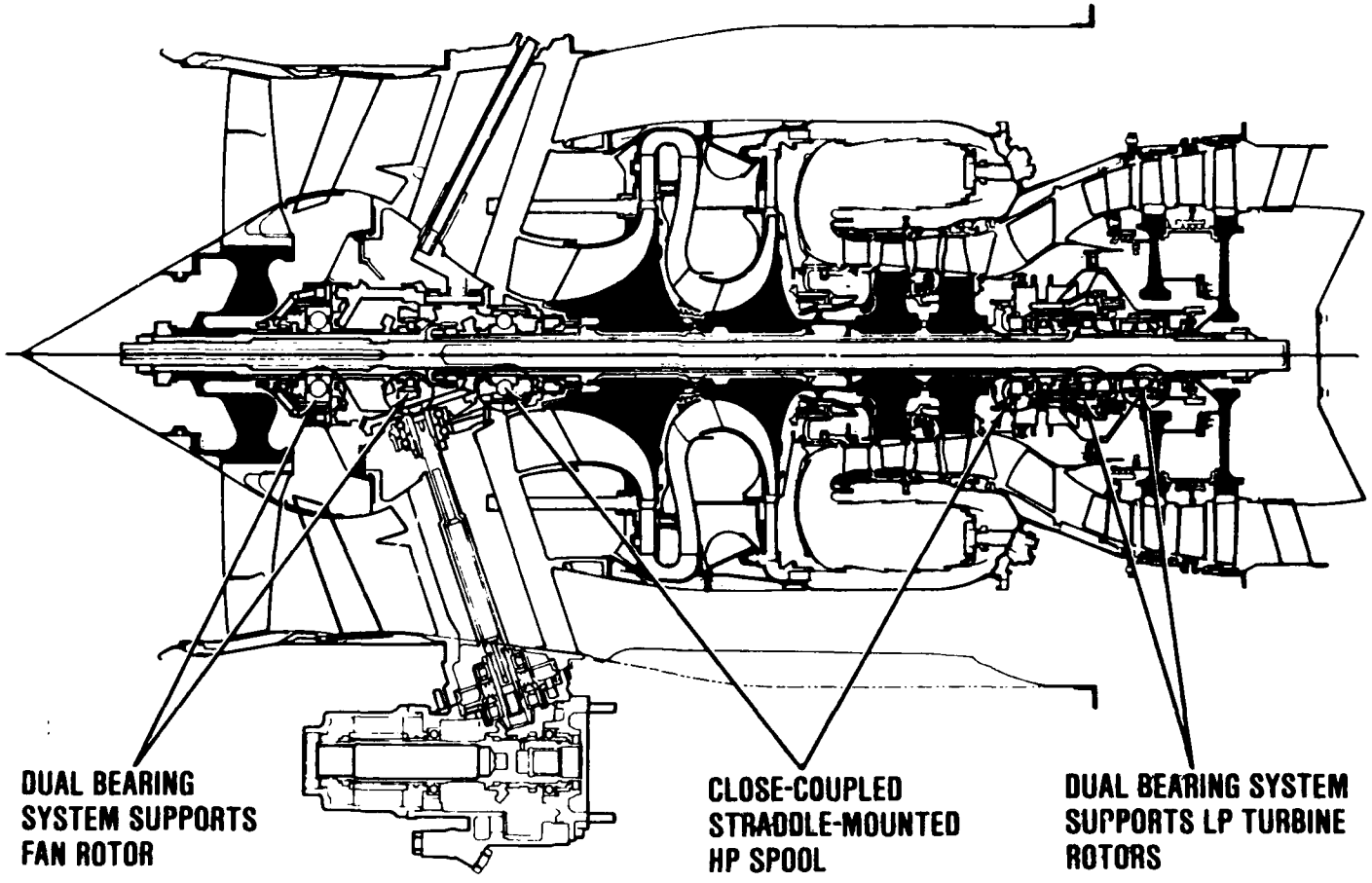
Allied-Signal Aerospace Company

Garrett Engine Division

Allied Signal

F109 ENGINE DESIGNED FOR TRAINER 3.5 RAD/SEC MANEUVER



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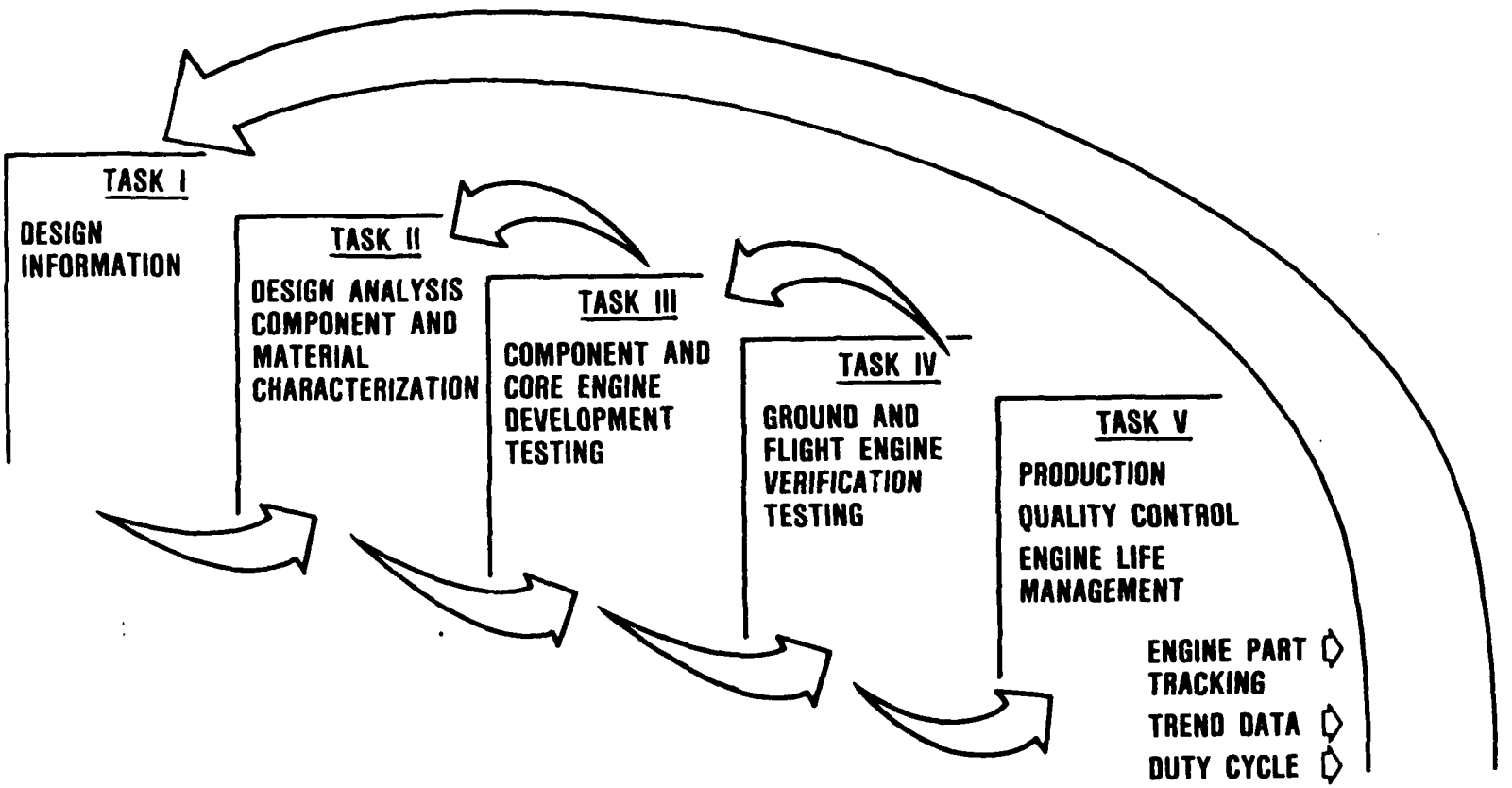
F109 IS FIRST ENGINE TO BE DEVELOPED FROM ITS INCEPTION UNDER ENSIP



68223-33A

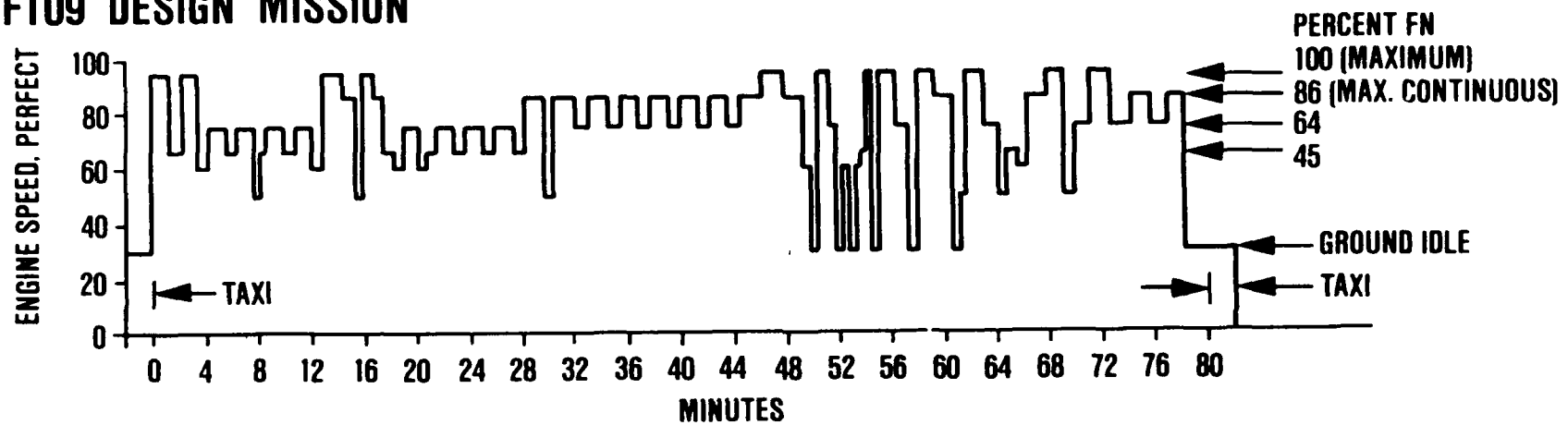
ENGINE AMT VERIFICATION TESTING

ENGINE COMPONENT DESIGN CRITERIA IS TWO-FOLD

- ANALYTICAL COMPONENT DESIGN LIFE BASED ON
  - LOW-CYCLE-FATIGUE (LCF)
  - STRESS RUPTURE
  - HIGH-CYCLE-FATIGUE (HCF)
  - CREEP
- ENSIP DESIGN CRITERIA ESTABLISHES REASONABLE CRITICAL HARDWARE INSPECTION INTERVALS
  - DAMAGE-TOLERANCE (SURFACE FLAWS)
  - CYCLIC CRACK PROPAGATION LIFE
  - RETIREMENT FOR CAUSE (RFC)

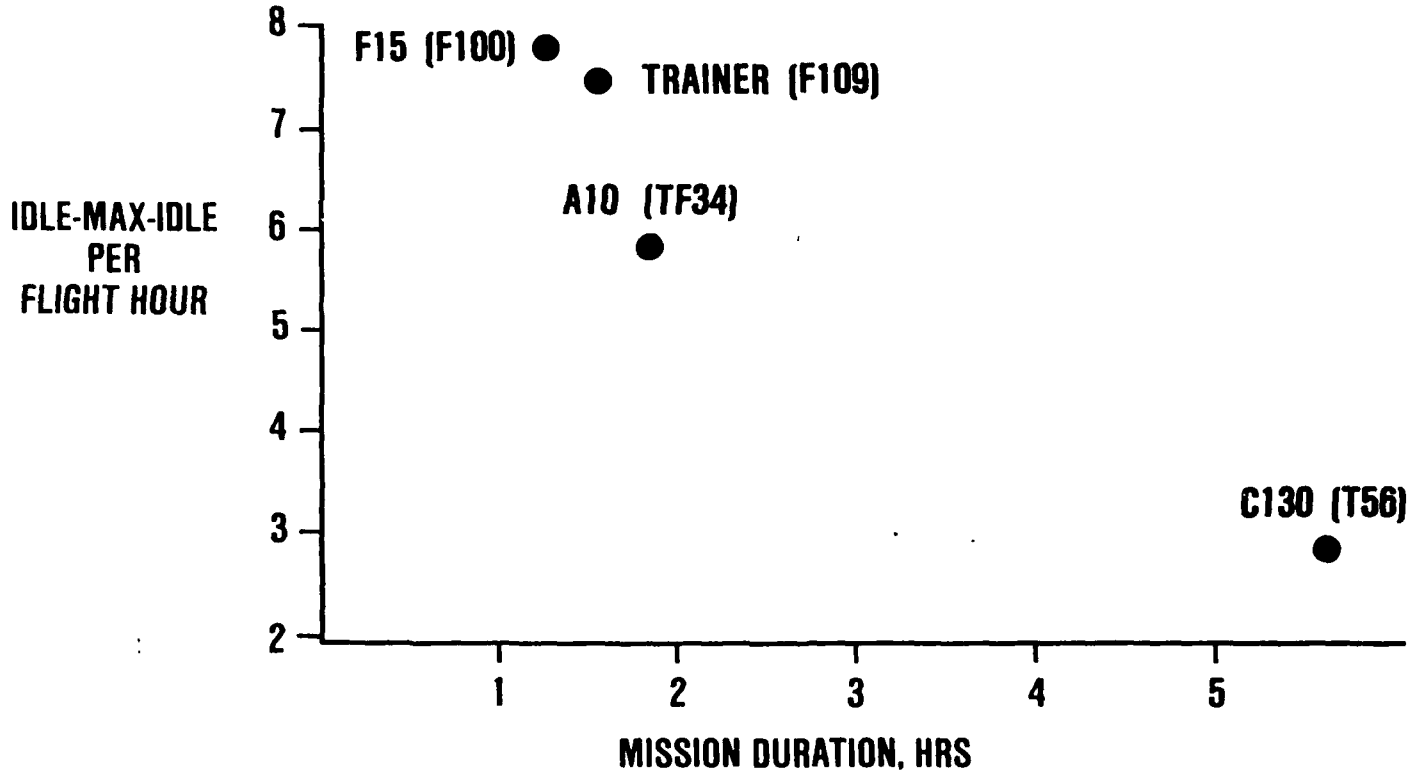
DESIGN DUTY CYCLE IS BASED ON ACTUAL T37 USAGE

F109 DESIGN MISSION



ENGINE DESIGNED FOR 18,000 FLIGHT HOURS	
TOTAL TIME AT MAXIMUM POWER	20%
TOTAL STARTS (ZERO-MAX-ZERO)	14,550
TOTAL TRANSIENTS (IDLE-MAX-IDLE)	148,000

USAF TRAINER MISSION SEVERITY SIMILAR TO F15 FIGHTER MISSION



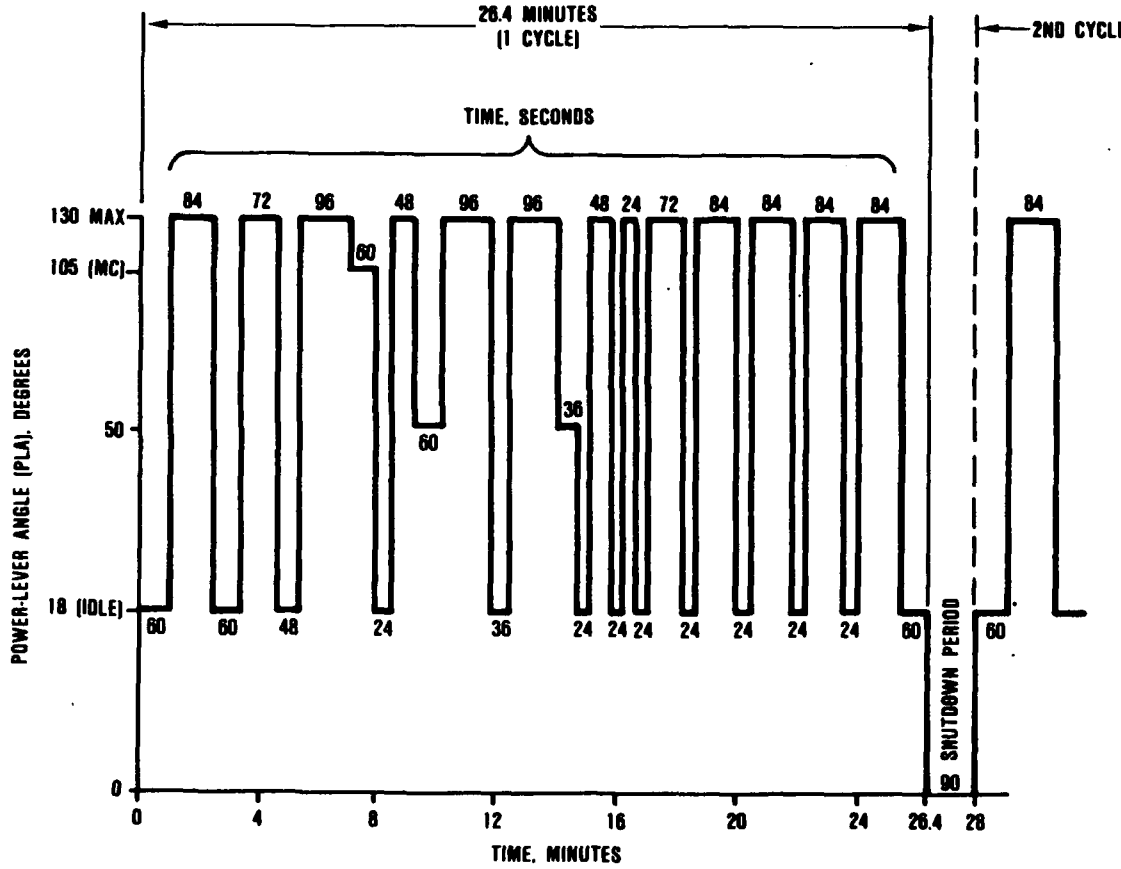
THREE-STEP DEVELOPMENT PROGRAM DESIGNED TO MAXIMIZE MATURITY AT PRODUCTION

- IFR (INITIAL FLIGHT RELEASE):
  - SAFETY OF FLIGHT
  - OK TO START FLIGHT TEST
- FFR (FULL FLIGHT RELEASE):
  - AEROTHERMO MARGIN FOR PRODUCTION, FIELD USAGE, MAINTENANCE
  - INITIAL LIVES DEMONSTRATED FOR MAJOR COMPONENTS (HSI)
  - OK FOR PRODUCTION START-UP
- ISR (INITIAL SERVICE RELEASE):
  - LIFE DEMONSTRATIONS
  - HSI/CSI INTERVAL VERIFICATION
  - ENGINE DIAGNOSTICS

AMT TEST PROVIDES STRINGENT ENGINE EVALUATION

- AMT CYCLE
  - CYCLE SEVERITY IS 3 TO 1
    80-MINUTE MISSION COMPRESSED INTO 26.4 MINUTES
  - ENGINE ACCELERATIONS AND DECELERATIONS CONDUCTED AT MAXIMUM RATES
- ENGINE TEMPERATURE MARGIN SET TO A MINIMUM ENGINE CONDITION WITH CUSTOMER BLEED EXTRACTION
- ADDITIONAL 0.15 LB/SEC CUSTOMER BLEED EXTRACTED AT MAXIMUM POWER
- 8 HORSEPOWER LOAD EXTRACTED FROM ACCESSORY GEARBOX; 2 HORSEPOWER FROM HYDRAULIC PUMP PAD AND A MINIMUM OF 6 HORSEPOWER FROM STARTER/GENERATOR PAD

ACCELERATED MISSION TEST DEFINITION



TIME, SECONDS

LEGEND	
TOTAL TIME	26.4 MIN
TIME MAX	16.2 MIN
TIME MAX CONT (MC)	1.0 MIN
I-MAX-I	11 CYCLES
0-MAX-0	1 CYCLE

*ACTUAL VALUES OF AN AMT CYCLE

FFR AMT TEST DEMONSTRATED INITIAL LIVES

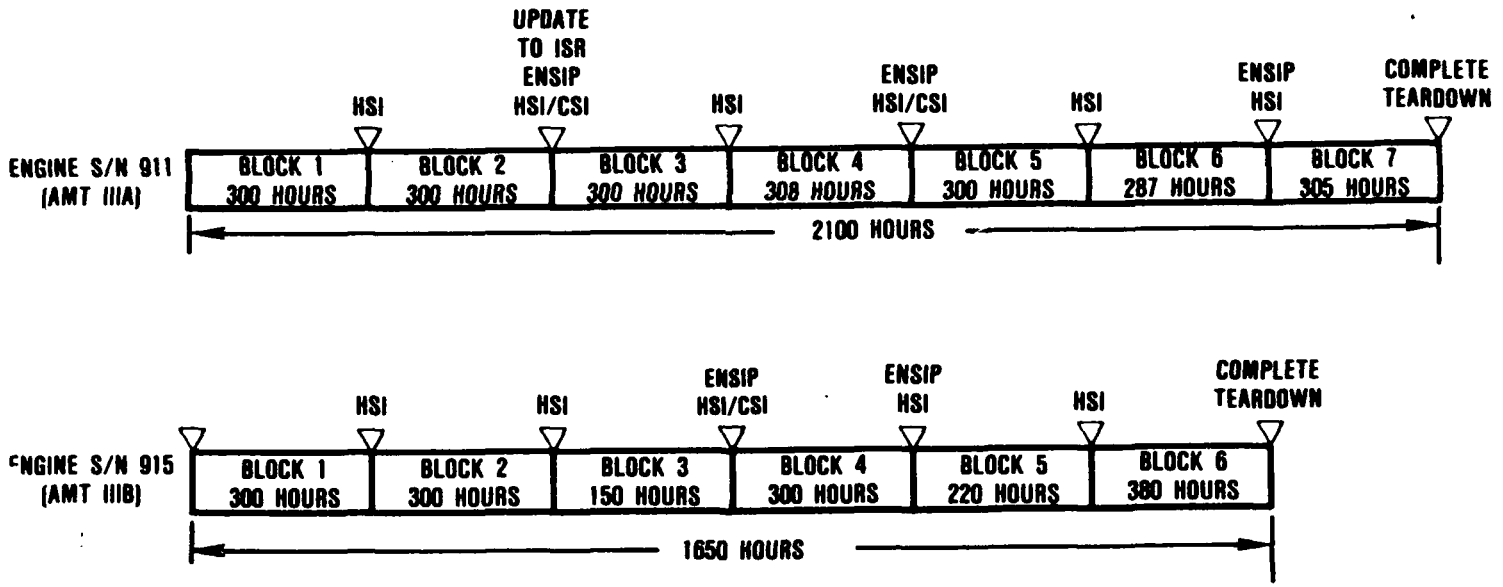
FFR TEST SUMMARY

- 750 HOURS OF AMT CYCLES COMPLETED ON ENGINE S/N 911
  - 1705 AMT CYCLES
  - 20,460 IDLE-MAX-IDLE CYCLES
  - 460 HOURS AT MAX POWER
  - 6394 TAC CYCLES

- ALL PERFORMANCE AND DURABILITY GOALS WERE MET

ISR AMT TESTS DESIGNED TO DEMONSTRATE ENGINE LIFE AND R&M GOALS

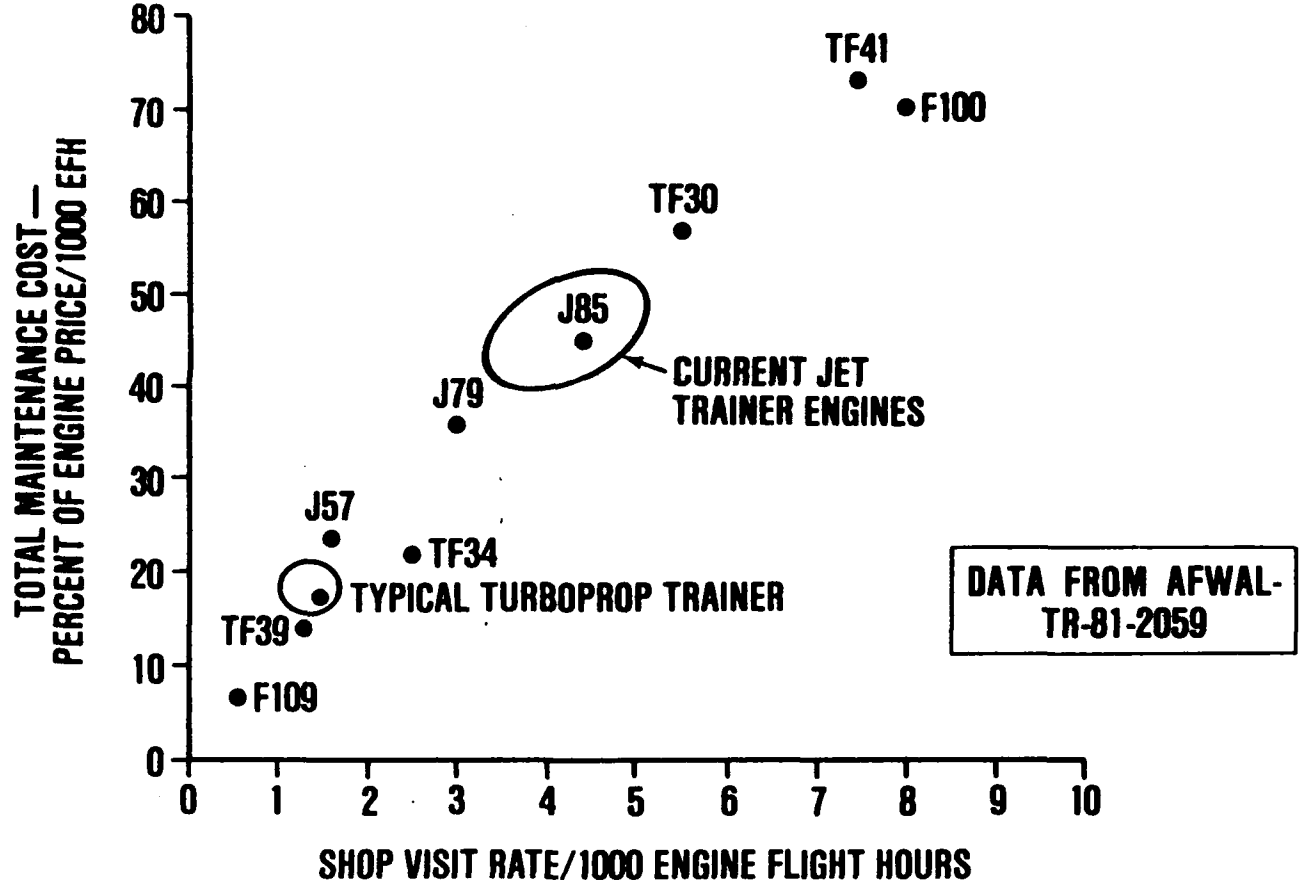
ISR TEST SUMMARY



ISR TESTS DEMONSTRATE THAT F109 ENGINE MEETS PIDS SUCCESS CRITERIA

- PERFORMANCE DETERIORATION WAS WITHIN SPECIFIED LIMITS
- VIBRATION, OIL PRESSURE, TEMPERATURES, MAX-RATING N_L SPEED, AND IDLE RATING THRUST WERE WITHIN SPECIFIED LIMITS
- ALL RELIABILITY AND MAINTAINABILITY GOALS WERE MET
- OVERALL CONDITION OF ENGINES WAS EXCELLENT

AMT TEST RESULTS DEMONSTRATE REDUCED MAINTENANCE COSTS



USAF QUALIFICATION IS COMPLETE THROUGH ISR

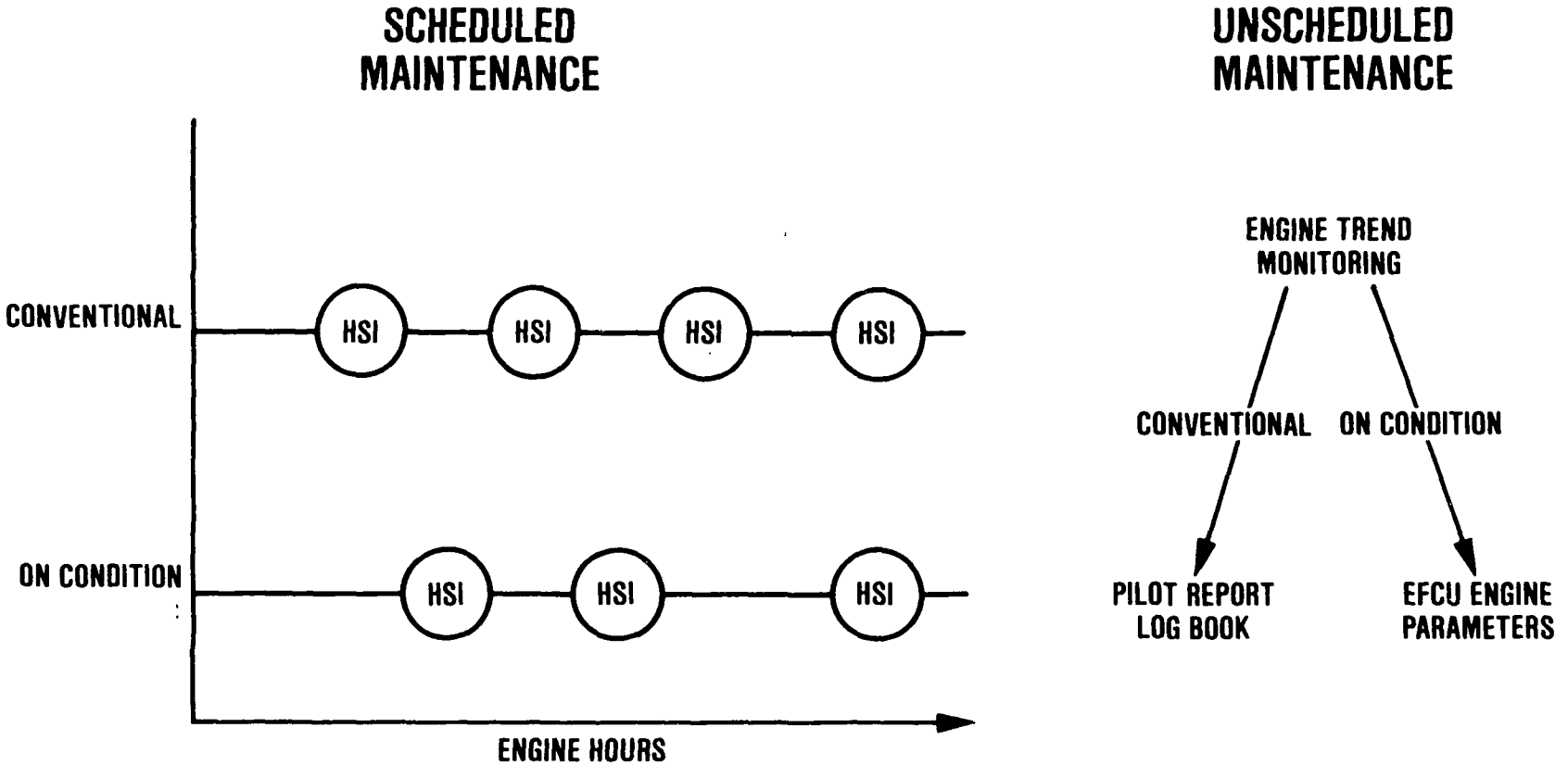
	INITIAL FLIGHT RELEASE (IFR)	FULL FLIGHT RELEASE (FFR)	INITIAL SERVICE RELEASE (ISR)
NUMBER OF TESTS	41	52	4
ENGINE, HOURS	1758	4062	6905
LENGTH OF PROGRAM, MO.	27	13	25
STATUS	COMPLETED	COMPLETED	COMPLETED

ENGINE CONDITION MONITORING

727

Allied-Signal Aerospace Company
Garrett Engine Division
Allied Signal

F109 OFFERS MAINTENANCE CHOICES



68223-56

ENGINE ELECTRONIC CONTROL UNIT IS KEY ELEMENT OF AIRBORNE MONITORING SYSTEM

- DATA LOGGER—PARTS LIFE TRACKING
- PERFORMANCE TRENDING — TRACKS ENGINE HEALTH
- ENSIP PARAMETERS TO AIRBORNE DATA RECORDER—ENGINE DUTY CYCLE SURVEY
- EVENT DETECTION—OVERSPEED/OVERTEMP
- BUILT-IN TEST—ISOLATES CONTROL SYSTEM FAULTS

ANALYSIS OF ENGINE DATA ALLOWS PARTS LIFE TRACKING

EFCU LOGS THESE DATA:

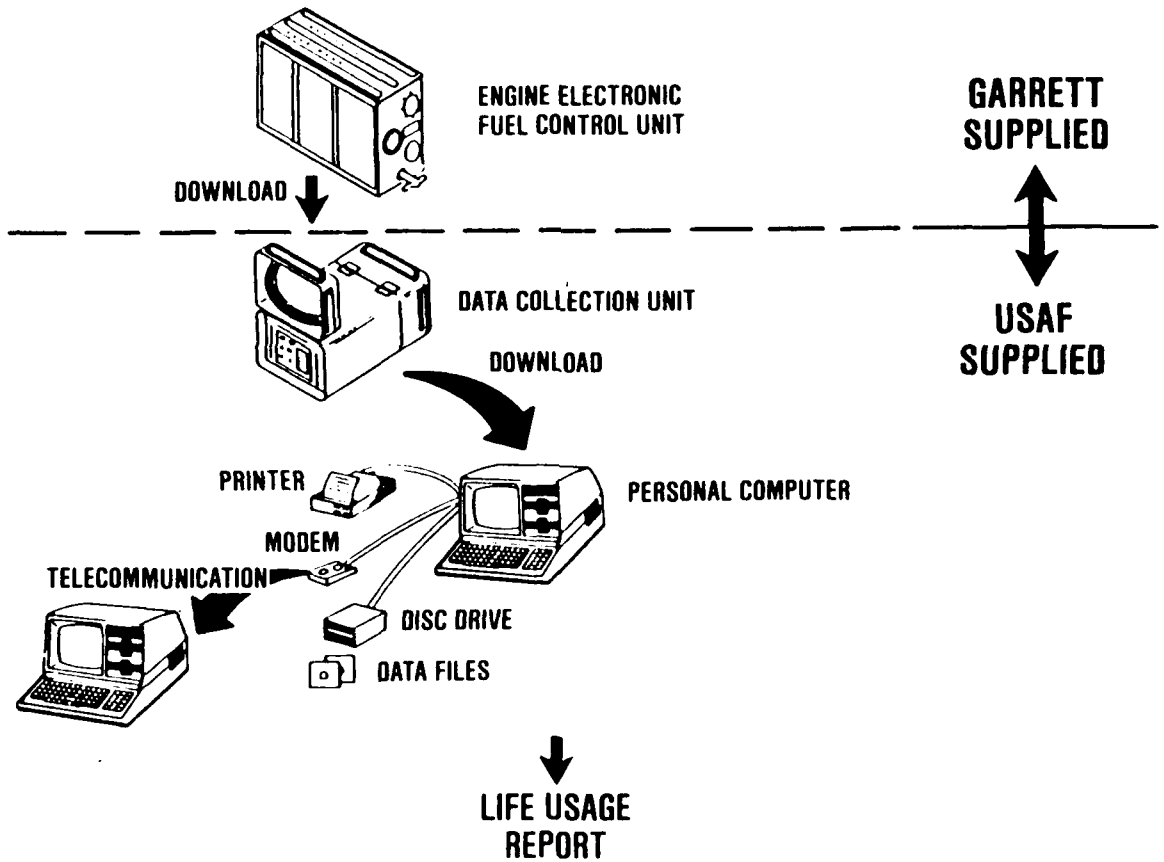
DOCUMENTARY — ENGINE S/N
EFCU S/N

EACH FLIGHT — TIME ABOVE SELECTED N_L (4)
TIME ABOVE SELECTED N_H (4)
TIME ABOVE SELECTED $T_{4.5}$ (4)
TIME ABOVE START TEMPERATURE (4)

CUMULATIVE — ENGINE STARTS
TIME AT MAX THRUST
TOTAL ENGINE TIME
TIME OVER START TEMP (4)
N_L LCF COUNTER (8)
N_H LCF COUNTER (8)
TIME ABOVE SELECTED $T_{4.5}$ (7)
P_3 LCF COUNTER (4)

ENGINE MAINTENANCE MANAGEMENT SYSTEM WAS DEVELOPED FOR USAF

EMMS DATA FLOW



SIX MAINTENANCE REPORTS GENERATED BY EMMS

- ENGINE HISTORY
- CORE AUTOMATED MAINTENANCE SYSTEM (CAMS) DATA
- INTERTURBINE (ITT) TREND PLOTS
  - ENGINE ITT PLOT
  - ENGINE ITT VS BASE FLEET AVERAGE
  - ENGINE ITT TEMPERATURE MARGIN COMPARISON
- TREND DATA LIST
- EVENTS DATA
- BUILT-IN TEST (BIT) FAULT IDENTIFICATION

CORE AUTOMATED SYSTEM DATA

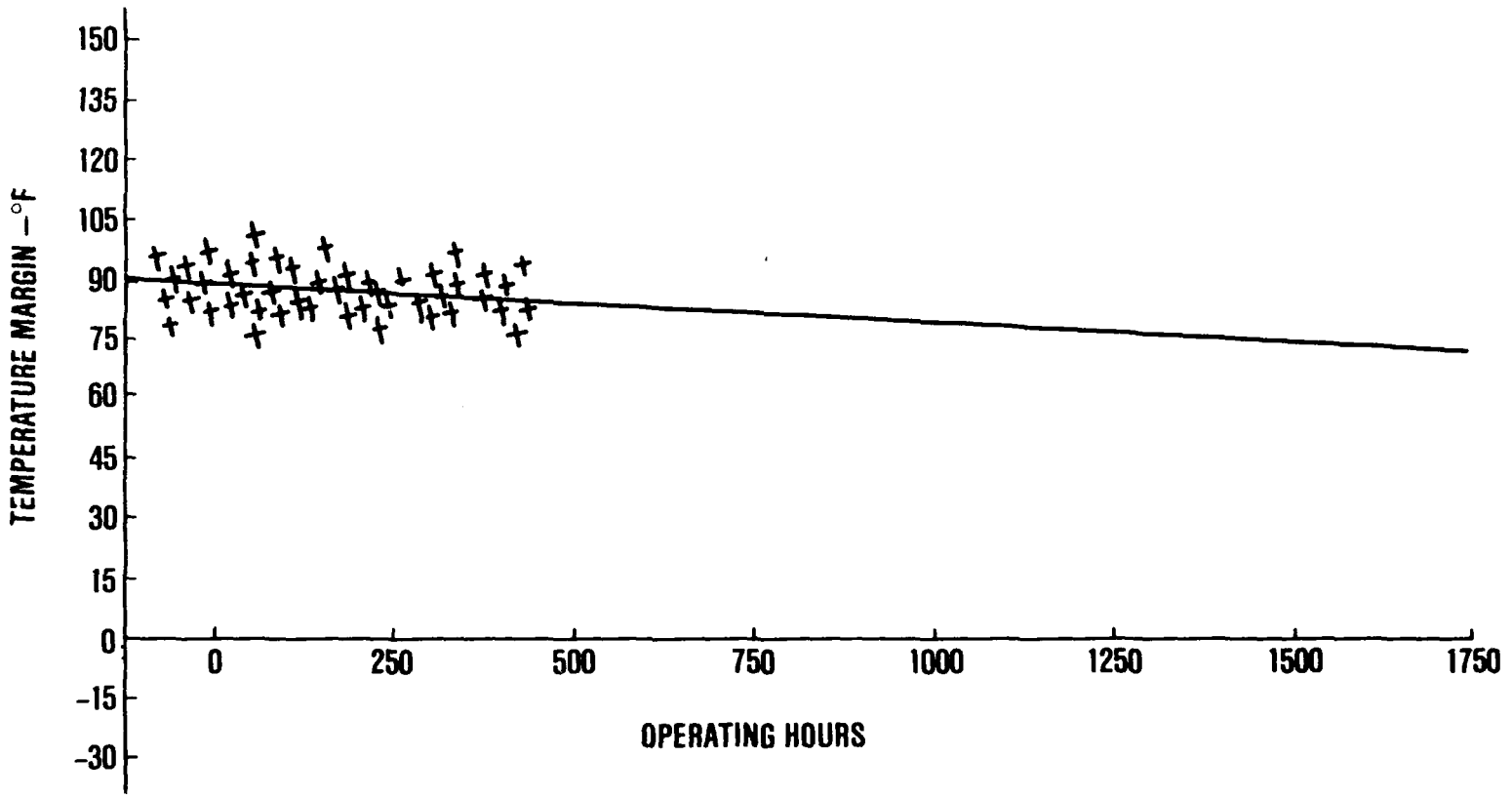
Engine CAMS Data Report For Engine Number 911 as of 11/17/87

DATE/TIME= 87313/1110 EFCU#= 1 UNIT= A ENGINE TIME= 94.38 HRS

NH RPM Ranges	Cycles	NL RPM Ranges	Cycles
0-45,306	1	0-15,500	1
0-44,200	173	0-14,200	172
31,600-44,200	1100	6,400-15,500	142
31,600-40,000	750	6,400-14,200	1710

T45 Temp Traps	Time (HRS) logged	PS3 Pressure Ranges	Cycles
1250	50.96	0-200-0	173
1300	45.83	80-200-80	2552
1350	38.04		
1395	1.84		
1438	.01		
1465	.01		
1490	.01		
1550	.01		

INTERTURBINE TEMPERATURE TREND PLOTS USED TO MONITOR ENGINE HEALTH



68223-60

Allied-Signal Aerospace Company
Garrett Engine Division

Allied Signal

BUILT-IN-TEST (BIT) INDENTIFIES FAULTS

ENGINE S/N 915 BUILT IN TEST FAULT DATA

DOWNLOAD DATE/TIME	EFCU	FLIGHT #	FAULT	LRU
87120/1027	45	41	PMG POWER FAULT	ECU
87120/1027	45	41	NO CROSS EFCU INPUT RECEIVED	ECU
87120/1027	45	42	NO CROSS EFCU INPUT RECEIVED	ECU
87120/1027	45	43	PMG POWER FAULT	ECU
87120/1027	45	43	NO CROSS EFCU INPUT RECEIVED	ECU
87120/1027	45	44	NO CROSS EFCU INPUT RECEIVED	ECU
87120/1027	45	45	PMG POWER FAULT	ECU
87120/1027	45	45	NO CROSS EFCU INPUT RECEIVED	ECU

ENGINE MISHAP DATA ANALYSIS
ASD RESERVE PROJECT
87-254-SEA

DATE: 29 NOVEMBER 1988

BY: AERONAUTICAL SYSTEMS DIVISION
WILLIAM COWIE ASD/YPEE
THOMAS STEIN ASD/HRMR

CONTENTS/OVERVIEW

- BACKGROUND/APPROACH
- MISHAP REPORT FORMAT
- MISHAP DATA F-16, F-15, A-7, A-10
- GRAPHICS/RESULTS
- OBSERVATIONS

BACKGROUND/APPROACH

- A.F. INSPECTION AND SAFETY CENTER ENGINE RELATED MISHAP DATA 1975-1987

- FIGHTER & FIGHTER/ATTACK MODELS

  F-16

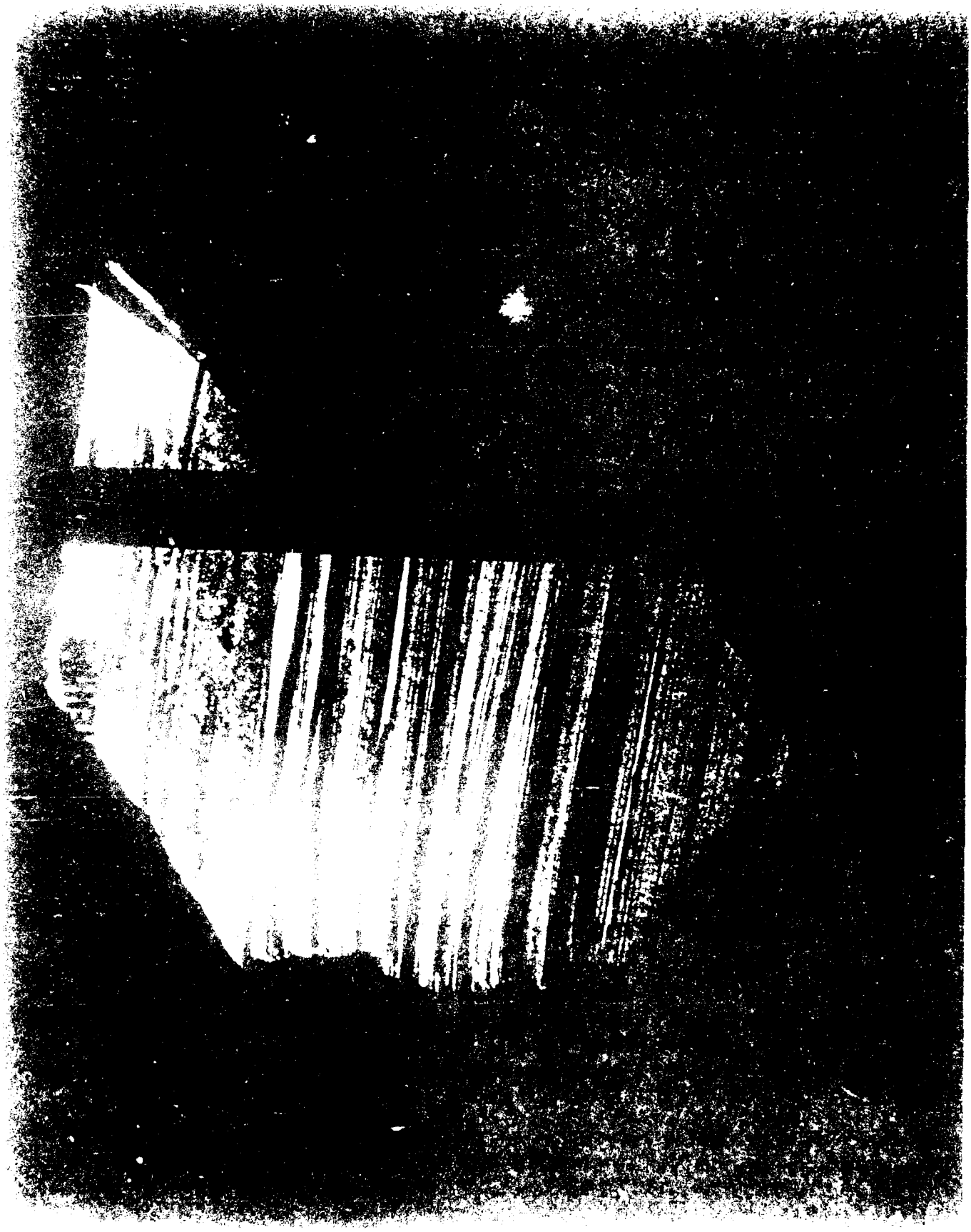
  F-15

  A-7

  A-10

  TOTAL ENGINE MISHAPS EXAMINED 1685 MISHAPS

- CLASS A, CLASS B & CLASS C DEF BASE

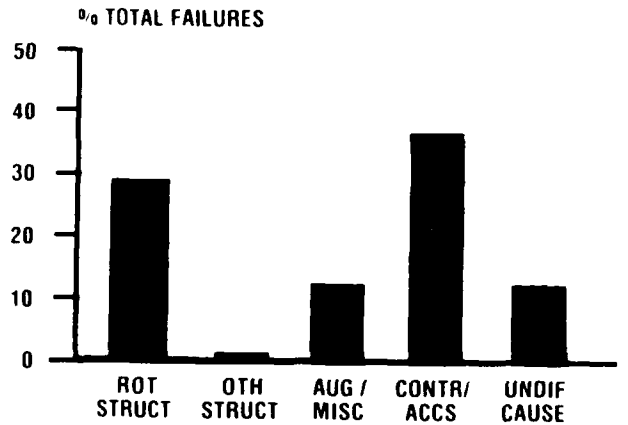


ANALYSIS & CATEGORIZATION

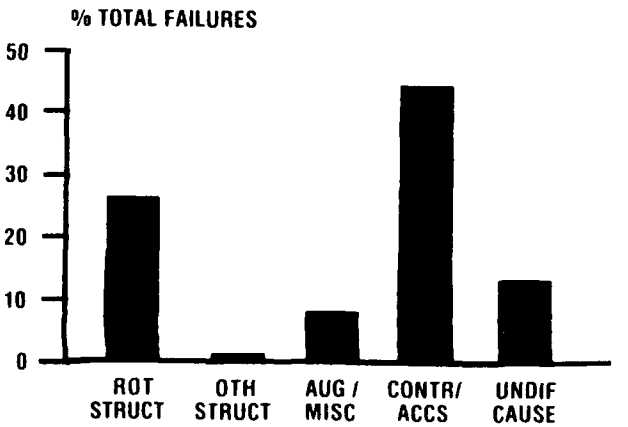
- CATEGORIZATION:
  - ROTATING STRUCTURE
    - FAN
    - COMPRESSOR
    - TURBINE
    - BEARING FAILURE (SUBTOTAL)
  - OTHER STRUCTURE
  - AUGMENTOR/MISCELLANEOUS
  - CONTROLS & ACCESSORIES
    - HYDRAULIC
    - DRIVES
    - BLEED SYSTEM
    - FUEL SYSTEM
    - TUBE INTEGRITY
    - OIL SYSTEM (SUBTOTAL)
  - UNDEFINED CAUSE

CATEGORIZATION OF MISHAP CAUSE FACTORS

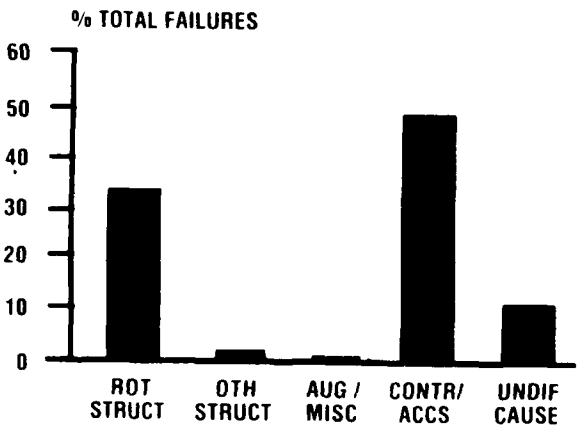
F-16 ENGINE FAILURE COMPARISONS



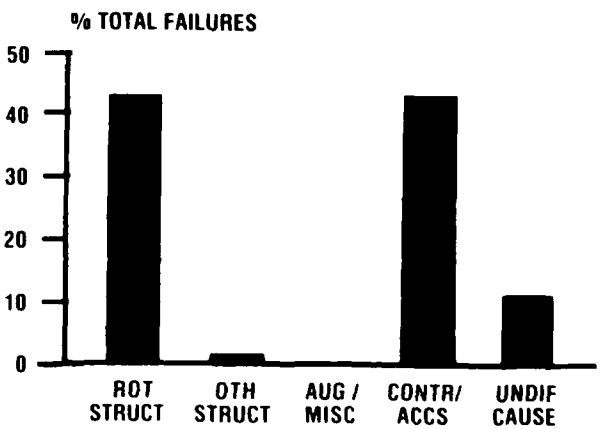
F-15 ENGINE FAILURE COMPARISONS



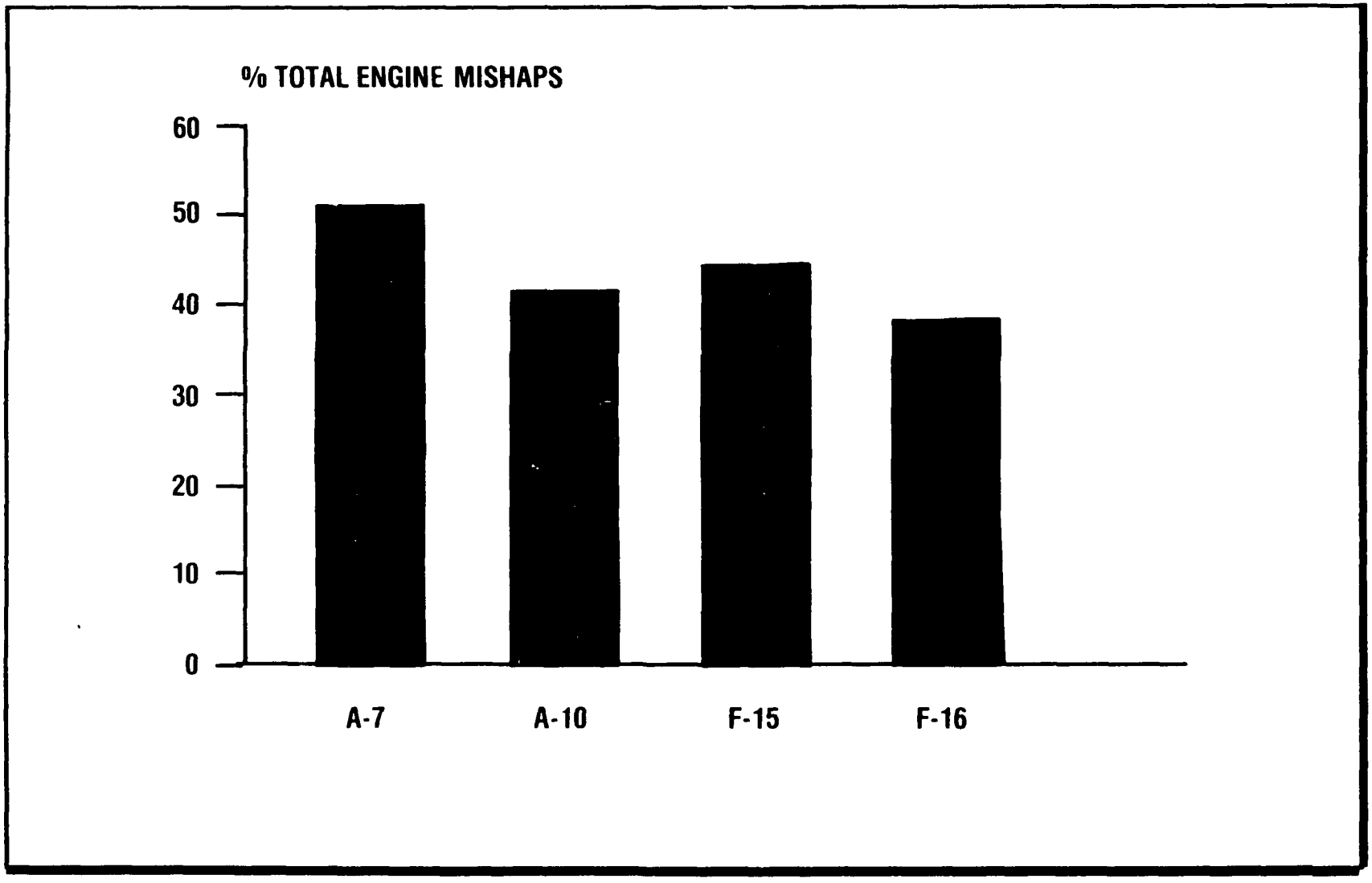
A-7 ENGINE FAILURE COMPARISONS



A-10 ENGINE FAILURE COMPARISONS



C & A DAMAGE COMPARISON



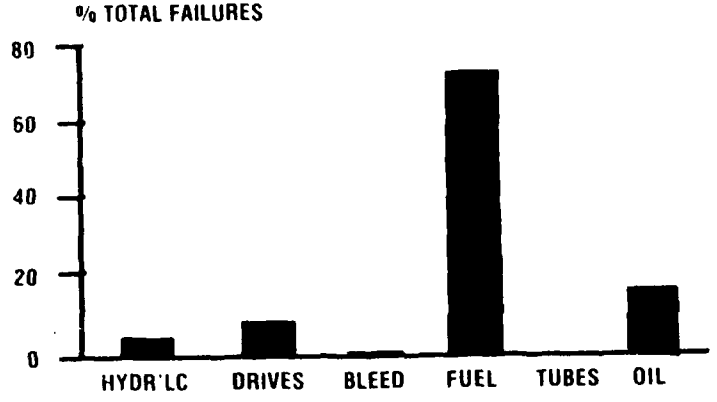
OBSERVATIONS

- CONTROLS AND ACCESSORIES ARE A SIGNIFICANT CONTRIBUTOR TO USAF ENGINE MISHAPS FAR EXCEEDING THOSE PROBLEMS THAT ENSIP SCREENS OUT

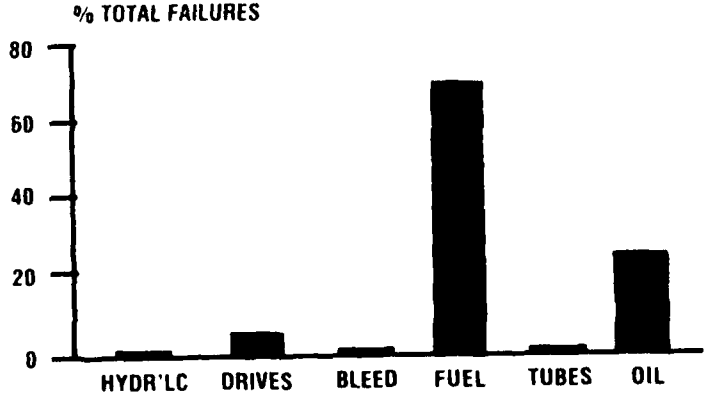
- MUST CONTINUE TO APPLY ENSIP TO ENGINE PRIMARY STRUCTURE . . . HOWEVER IN ADDITION, THERE IS A BIG PAYOFF IF GAINS CAN BE MADE TO ENHANCE THE SERVICE LIFE INTEGRITY OF ENGINE CONTROLS & ACCESSORIES

CATEGORIZATION OF C & A MISHAP CAUSE FACTORS

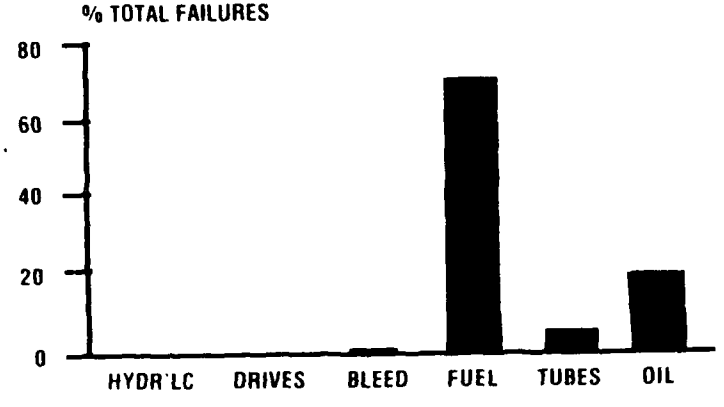
F-16 ENGINE CONTROLS & ACCESSORIES FAILURES



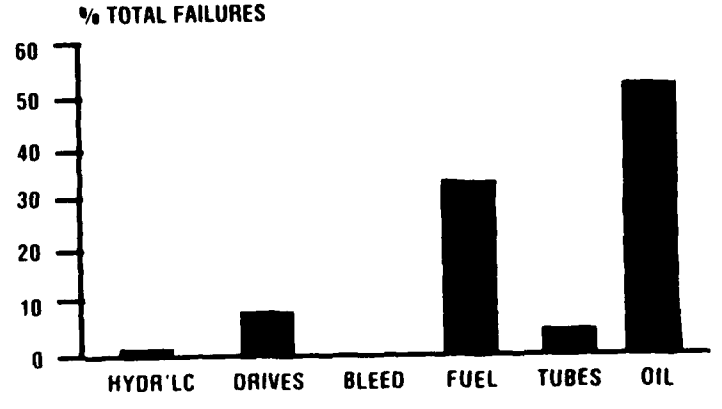
F-15 ENGINE CONTROLS & ACCESSORIES FAILURES



A-7 ENGINE CONTROLS & ACCESSORIES FAILURES



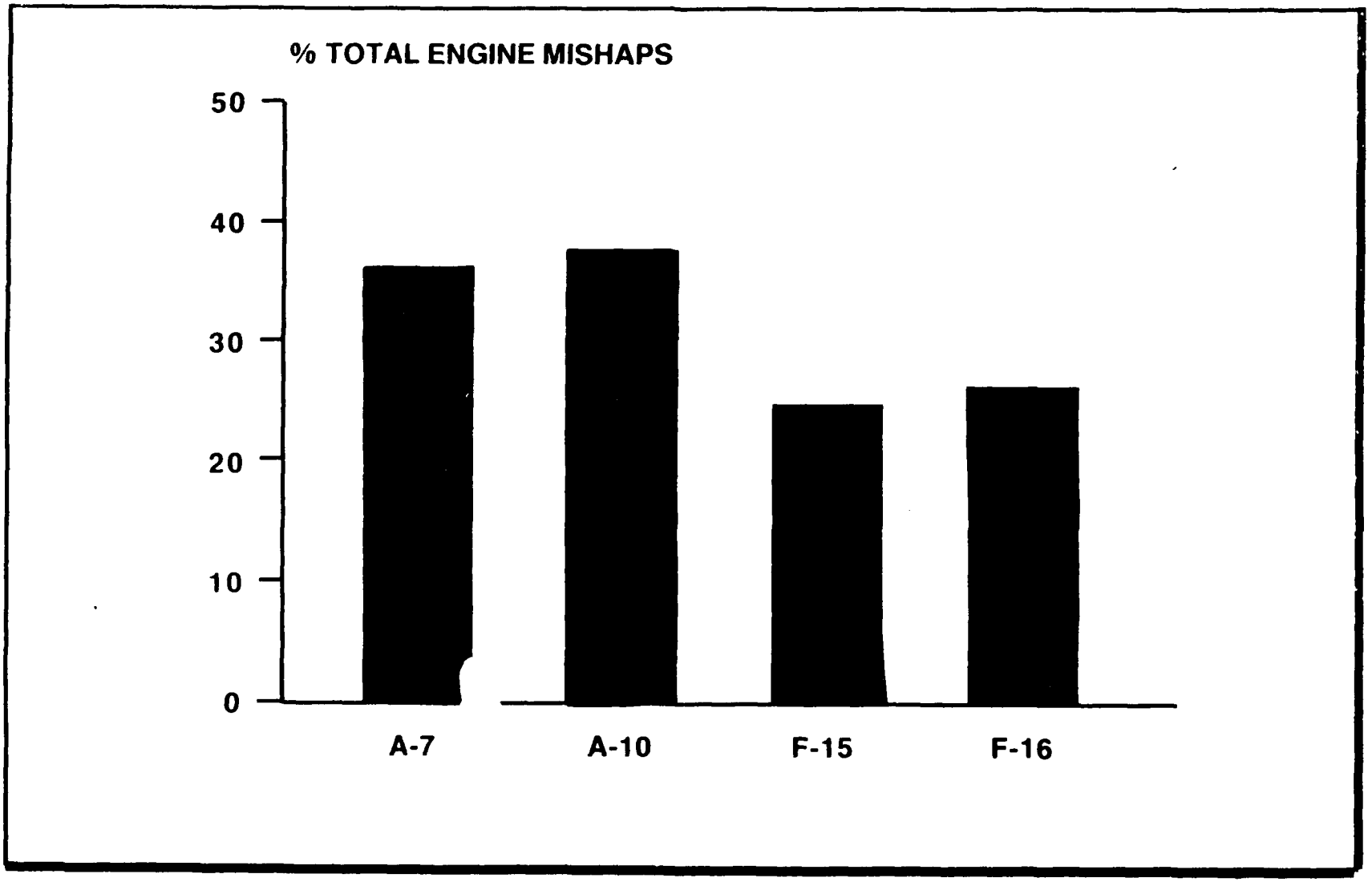
A-10 ENGINE CONTROLS & ACCESSORIES FAILURES



OBSERVATIONS (CONT'D)

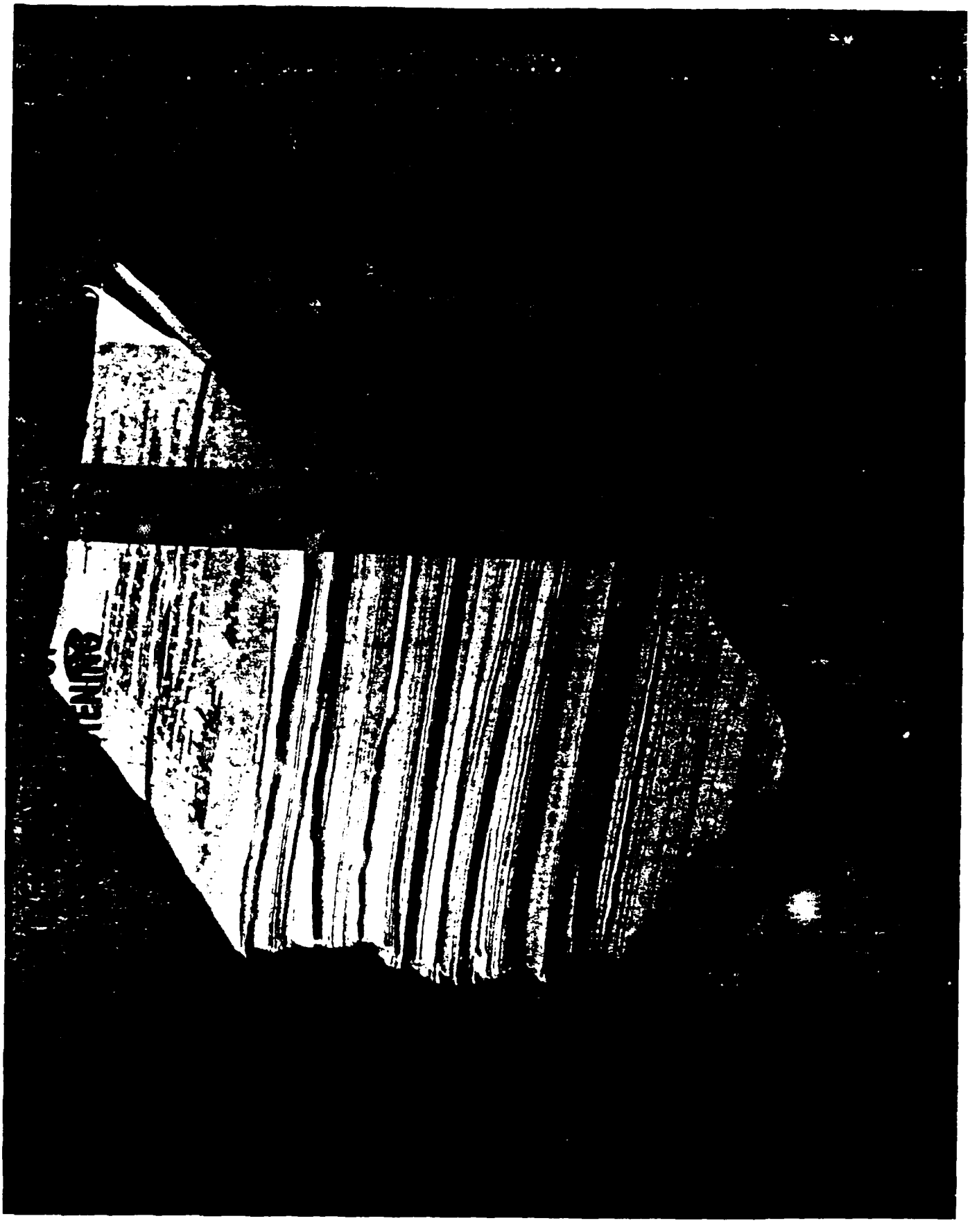
- FUEL CONTROL IS LARGEST PROBLEM ITEM WITHIN C & A AREA
- FOD CONTINUES TO BE A SIGNIFICANT ENGINE MISHAP CAUSE FACTOR

FOD DAMAGE COMPARISON



CONCLUSIONS & FUTURE DIRECTION

- LARGE POTENTIAL PAYOFF FOR C & A INTEGRITY IMPROVEMENTS
- ASD INITIATING INTEGRITY PROGRAM FOR ENGINE CONTROLS AND ACCESSORIES



ISRAEL AIRCRAFT INDUSTRIES LTD
AIRCRAFT DIVISION

MATAN - CIVIL AIRCRAFT

ASTRA

DAMAGE TOLERANCE
PRESENTATION

NOV. 1988

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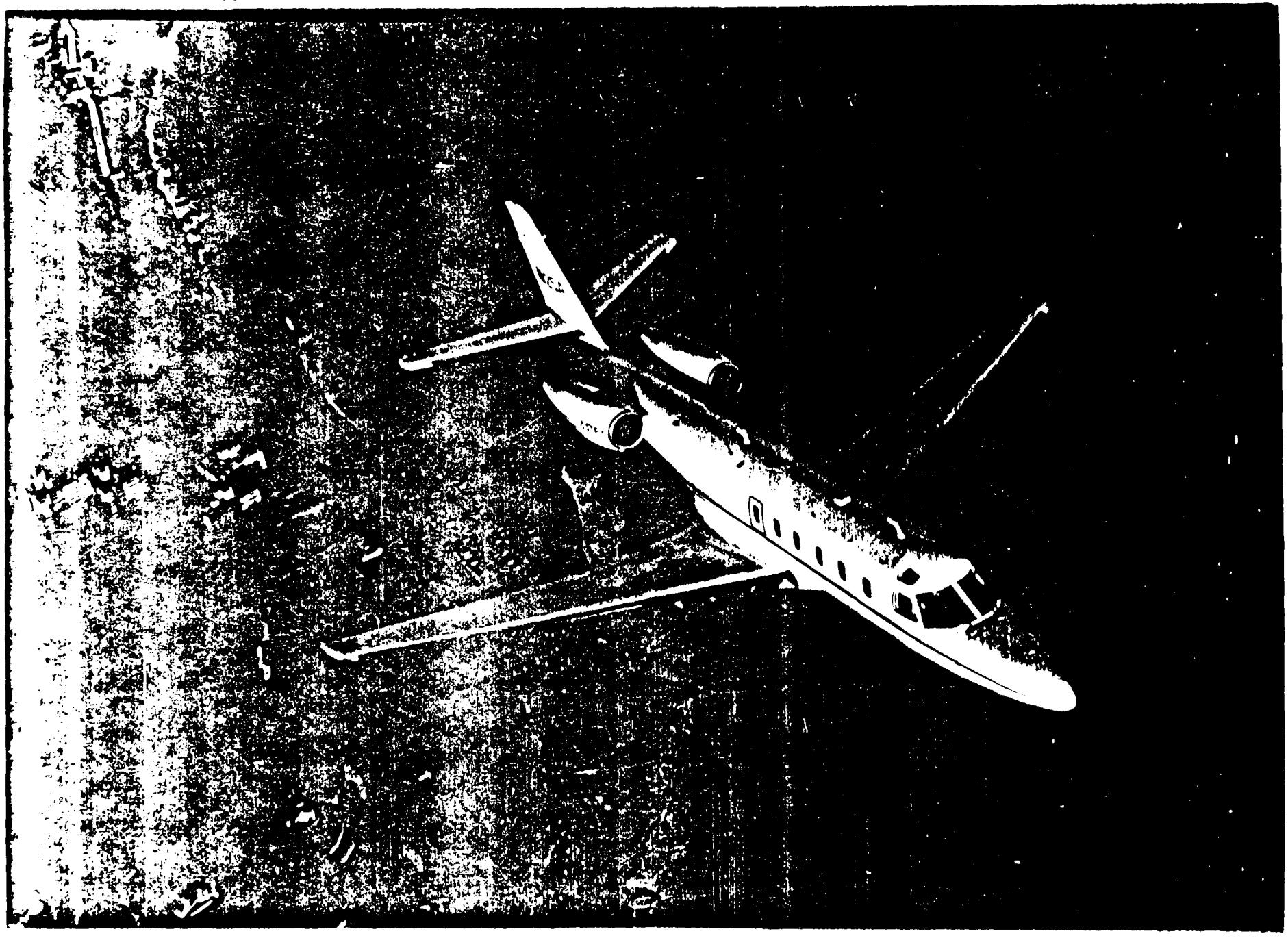
ASTRA



ISRAEL AIRCRAFT INDUSTRIES LTD
BEN-GURION INTERNATIONAL AIRPORT ISRAEL ZIP CODE 70100
TEL. (03) 9713111 TELEX: ISRAV - IL 381002,381014,381033
CABLES : ISRAVIA, TELEFAX: 9712290, 9713131, 9711762

THE ASTRA A/C IS A COMMERCIAL EXECUTIVE JET.

FROM STRUCTURAL POINT OF VIEW IT IS A FULLY FAIL SAFE/MULTIPE LOAD PATH STRUCTURE.

THE FRONT FUS. IS DESIGNED WITH CLOSELY SPACED FRAMES AND 4 MAIN BEAMS, TWO UPPER AND TWO LOWER.

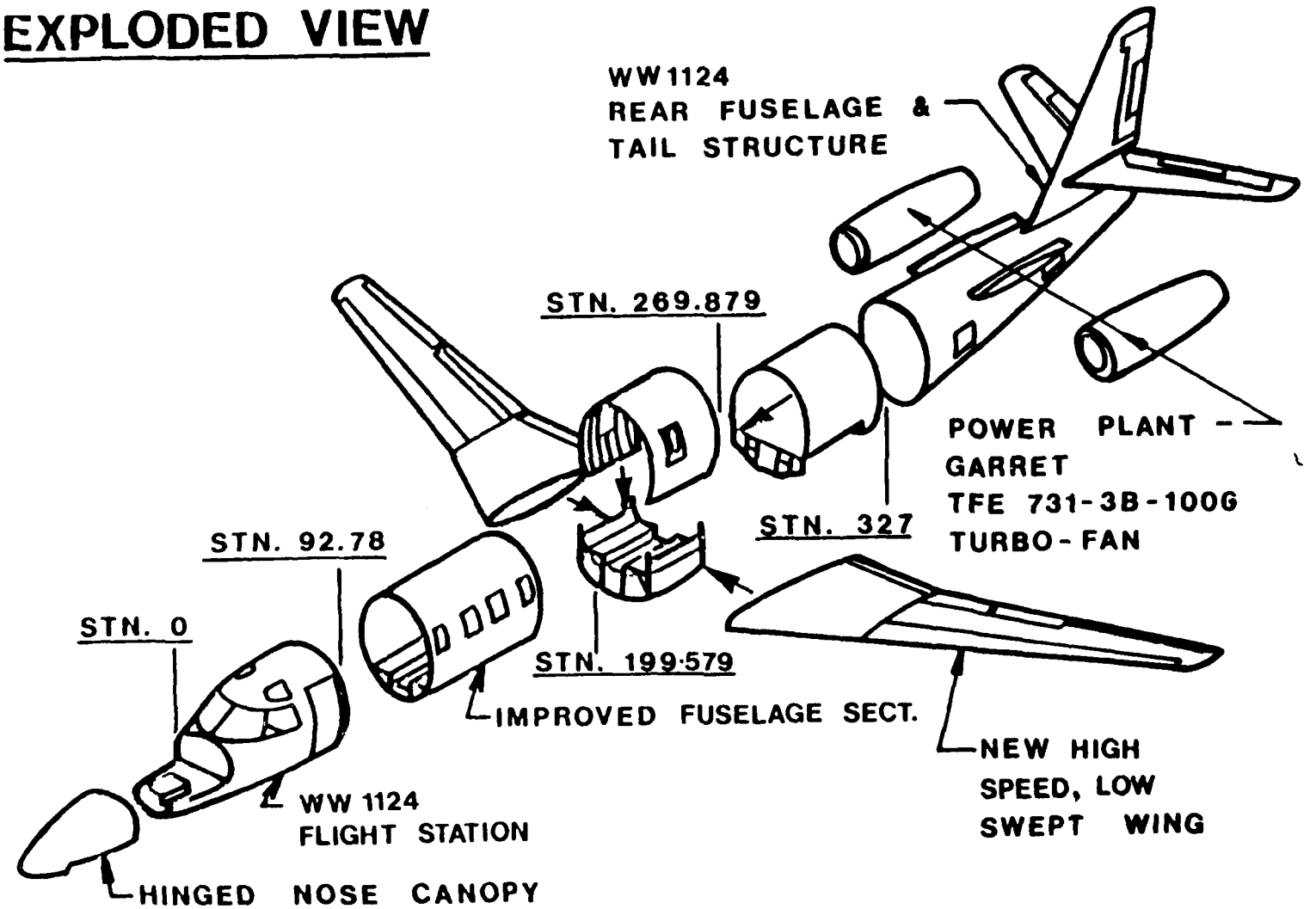
THE AFT FUSELAGE CONTAINS 26 STRINGERS AND CLOSELY SPACED FRAMES.

THE INNER WING CONTAINS 3 SPARS, FRONT, CENTER AND AFT AND THREE SKINS.
THE OUTER WING CONTAINS 2 SPARS AND TWO SKINS SPLICED SPANWISE.

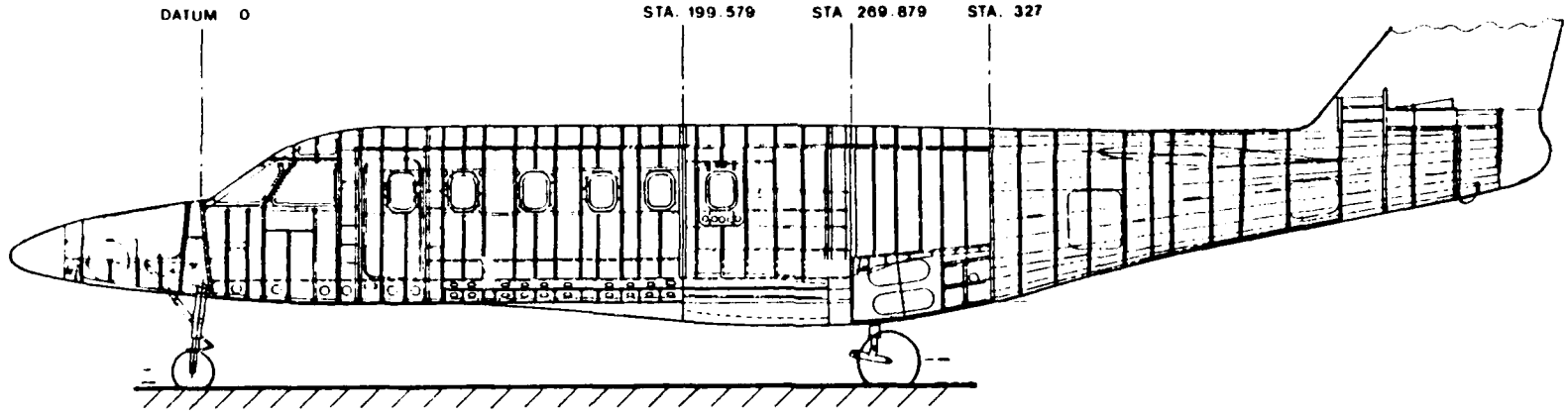
THE SLAT MOVES ON 6 RAILS AND 3 ACTUATORS.

THE FLAP MOVES ON 6 RAILS AND 3 ACTUATORS.

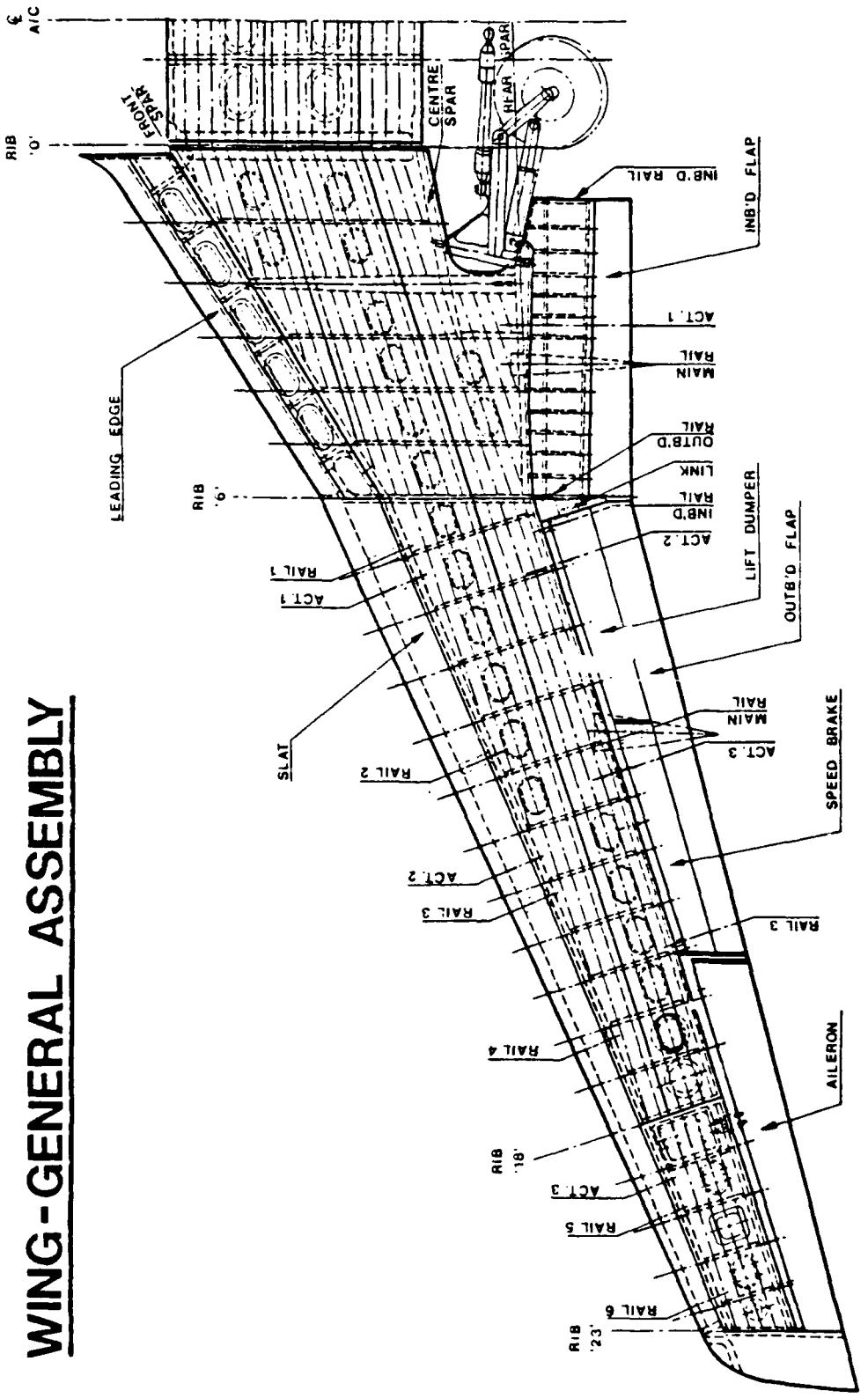
EXPLODED VIEW



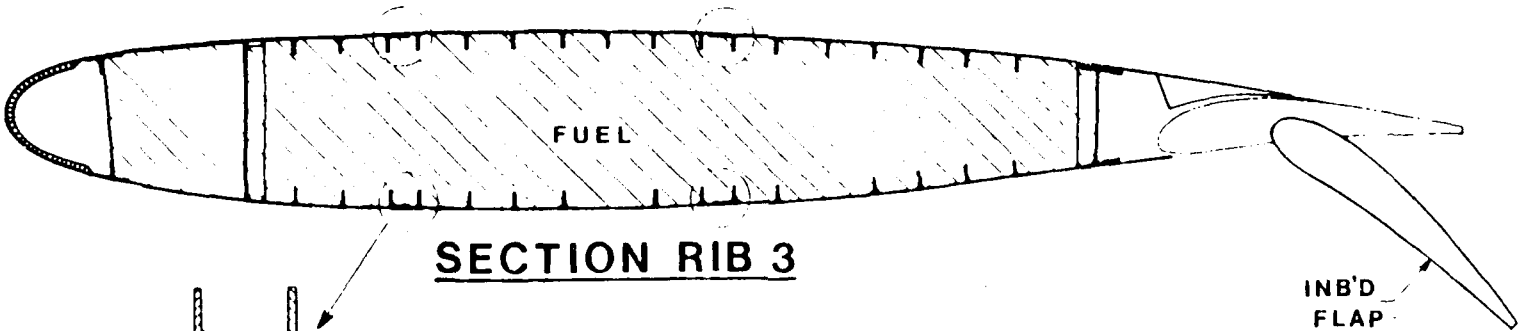
FUSELAGE STRUCTURAL ASSY.



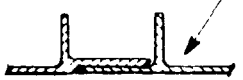
WING - GENERAL ASSEMBLY

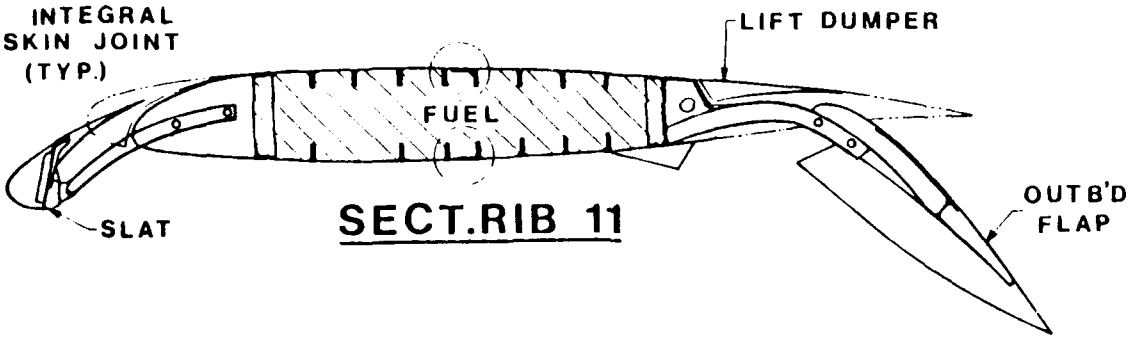


WING SECTIONS



SECTION RIB 3





SECT.RIB 11

SECT.RIB 17

DAMAGE TOLERANCE PRINCIPLES

IN ALL PRIMARY AIRCRAFT STRUCTURE THERE ARE INITIAL CRACKS (DUE TO FATIGUE, CORROSION OR ACCIDENTAL DAMAGE)

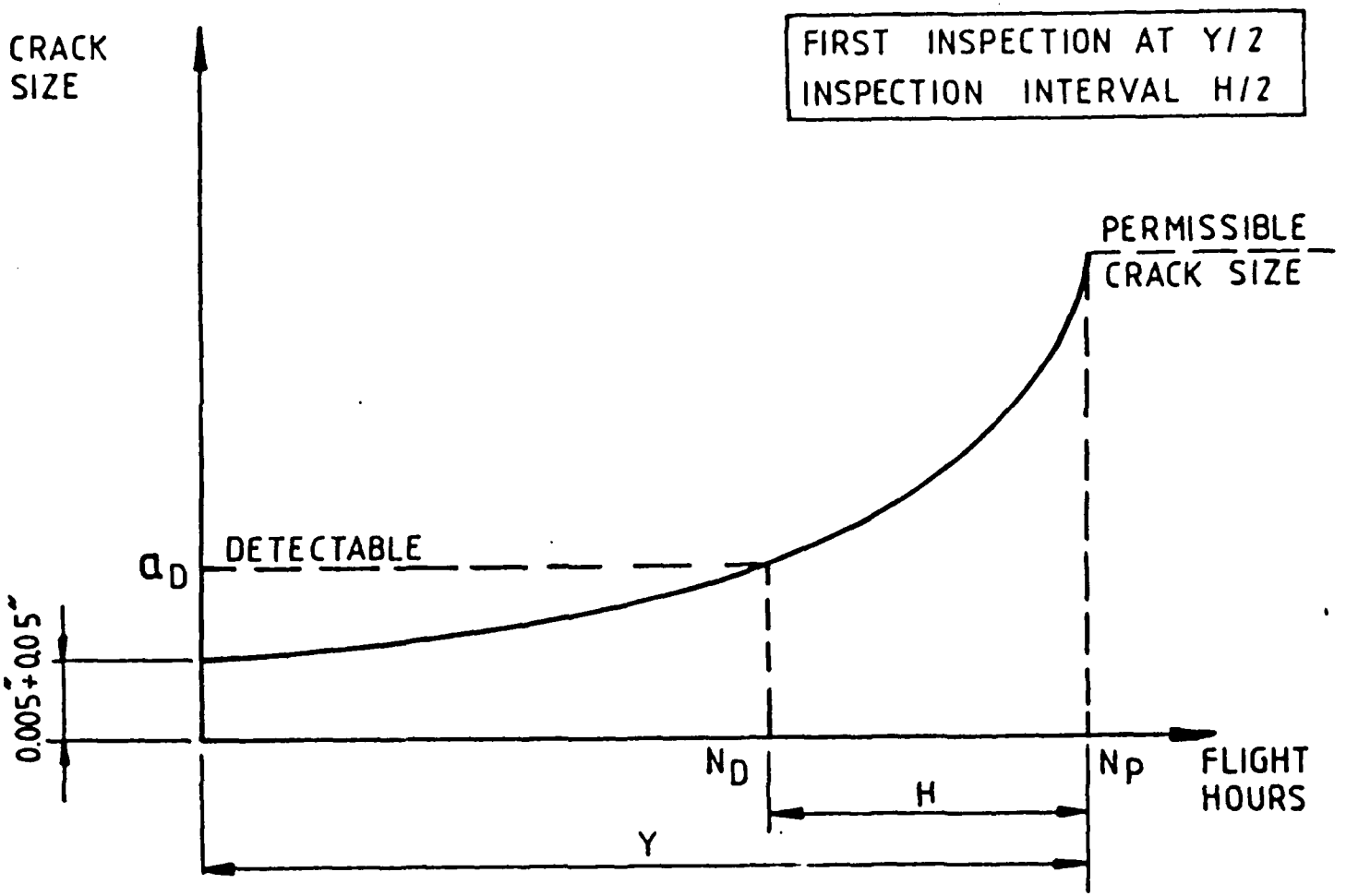
DAMAGE TOLERANCE REQUIREMENTS ATTEMPT TO PREVENT THAT CRACKS IN AIRCRAFT STRUCTURES CAUSE FRACTURE AND LOSS OF THE AIRPLANE

THE RESIDUAL STRENGTH IS TAKEN EQUAL TO LIMIT LOAD

IT SHOULD BE SHOWN THAT A CRACK WILL BE ELIMINATED BEFORE IT REACHES SIZE a_p

A PERIODIC INSPECTION PROGRAM BASED ON CRACK GROWTH UNDER SPECTRUM LOADING IS NECESSARY

CRITERIA FOR THRESHOLD AND INSPECTION INTERVAL (SINGLE CRACK)



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DAMAGE TOLERANCE CRITERIA

- INSPECTION INTERVALS GOAL
  - FIRST INSPECTION (THRESHOLD) 1/2 LIFE TIME
  - FOLLOWING INSPECTIONS 1/4 LIFE TIME

- MULTIPLE SITE DAMAGE CRITERION
  - STRESS LEVELS WERE KEPT LOW
  - STRESS CONCENTRATION WAS AVOIDED
  - COLD WORKED HOLES IN HIGH LOAD TRANSFER
  - SHOT PEENING

- INSPECTABILITY & ACCESSIBILITY CRITERION
  - PROVISIONS FOR ACCESSIBILITY IN MOST CRITICAL AREAS
  - ALL INSPECTIONS ARE VISUAL
  - MOST INSPECTION ARE EXTERNAL

- MATERIALS
  - HIGH TOUGHNESS
  - LOW CRACK GROWTH

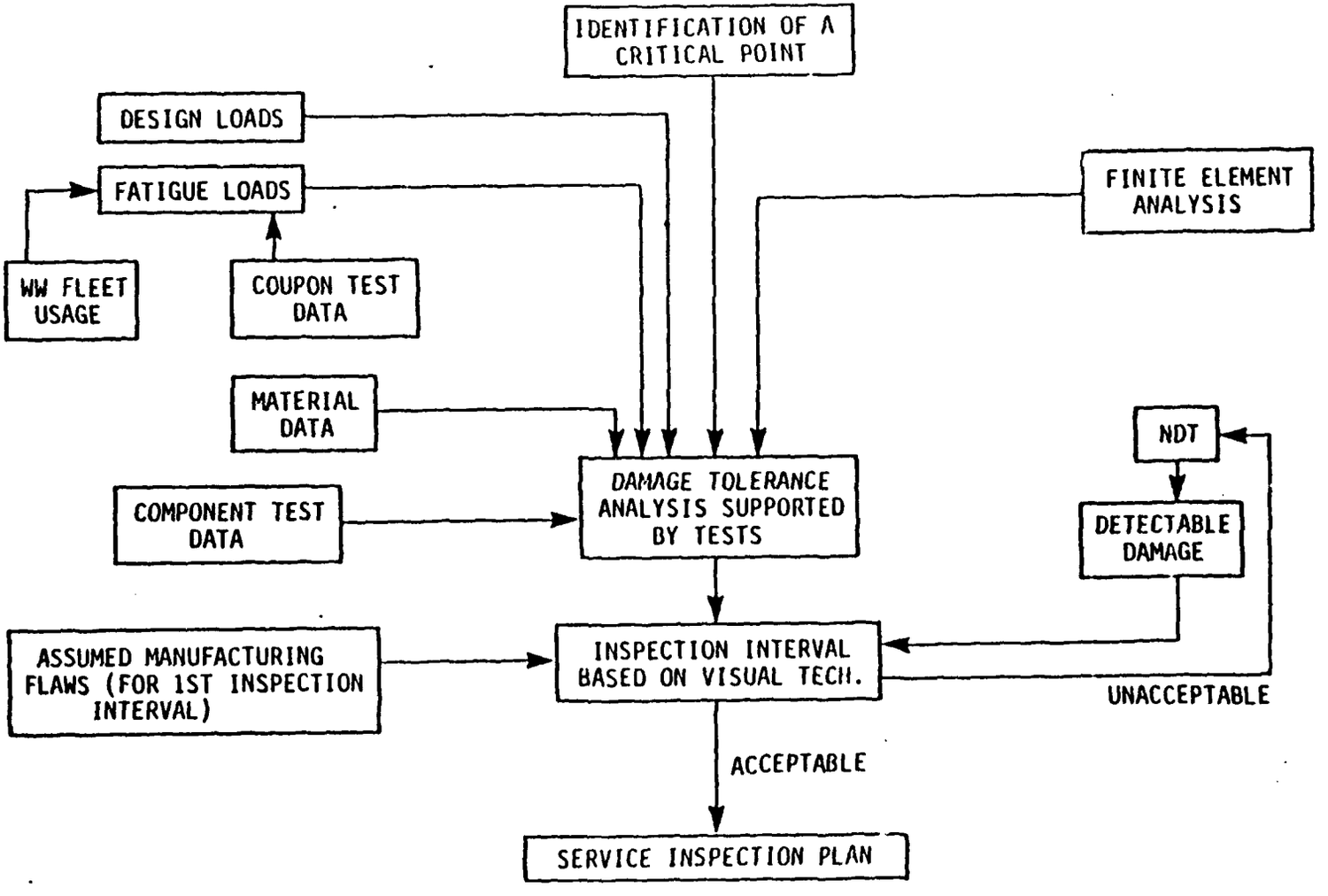
- CORROSION PREVENTION PROGRAM
  - PROVISIONS FOR DRAINAGE
  - PROTECTION OF DISSIMILAR MATERIALS
  - PROTECTIVE COATINGS

COMLIANCE WITH SPECIAL CONDITIONS FOR FLIGHT ABOVE 41,000 FT

MAX PERMISSIBLE CRACK LENGTH = MIN of:
- MAX CRACK LENGTH, DICTATED BY LEAKAGE REQUIREMENTS
- MAX PERMISSIBLE CRACK LENGTH FROM DAMAGE TOLERANCE ANALYSIS

INSPECTION INTERVAL = INSPECTION INTERVAL, CALCULATED BY PROBABILITY METHOD, DEVIDED BY FACTOR 2

DAMAGE TOLERANCE SUBSTANTIATION PROCEDURE



IDENTIFICATION OF CRITICAL POINTS

BASED ON:

1. DETAILED STATIC STRESS ANALYSIS (DETERMINATION OF AREAS SUBJECTED TO TENSION/SHEAR)
2. DETAILED FATIGUE ANALYSIS (CRACK INITIATION, HIGH STRESS CONCENTRATION LIKE CUTOUT CORNERS, SMALL RADIUS, ETC.)
3. STRUCTURAL ELEMENTS THAT ARE PART OF A MAJOR LOAD PATH
4. STRUCTURAL LOCATIONS PRONE TO IN-SERVICE ACCIDENTAL DAMAGE
5. LOCATIONS WHERE STRESSES WOULD BE HIGH IN SECONDARY MEMBERS AFTER PRIMARY MEMBER FAILURE
6. IN-SERVICE AND TEST DATA IN PREVIOUS WESTWIND AIRCRAFT
7. DATA OBTAINED FROM COMPONENT FATIGUE AND D.T. TESTS

FATIGUE LOAD & STRESS SPECTRA DEVELOPMENT

METHODOLOGY PHASES

- PRIOR WESTWIND MODELS SERVICE USAGE DATA PROCESSING
- DEFINITION OF TYPICAL MISSION PROFILES (T.M.P.)
- DEFINITION OF PRESSURIZATION CONDITIONS
- DEFINITION OF THE MISSION MIX
- DETERMINATION OF FLIGHT STAGES AS PER ACTUAL FLIGHT OF THE AIRCRAFT
- DEFINITION OF WEIGHT & LOAD CONFIGURATIONS FOR FATIGUE
- GENERATION OF LOAD FACTOR SPECTRA (L.F.S.) FOR ALL FLIGHT STAGES

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TESTING PROGRAM

- COUPON TESTS
- COMPONENT TESTS
- FULL SCALE TESTS

COUPON TEST PROGRAM

- BASELINE INPUT DATA
  - MATERIAL FRACTURE TOUGHNESS
    - THICK SECTION K_1c
    - THIN SECTION Kc
  - MATERIAL FATIGUE - CRACK GROWTH
    - "R" EFFCT
    - ENVIRONMENT EFFECT
    - EQUATION CONSTANTS

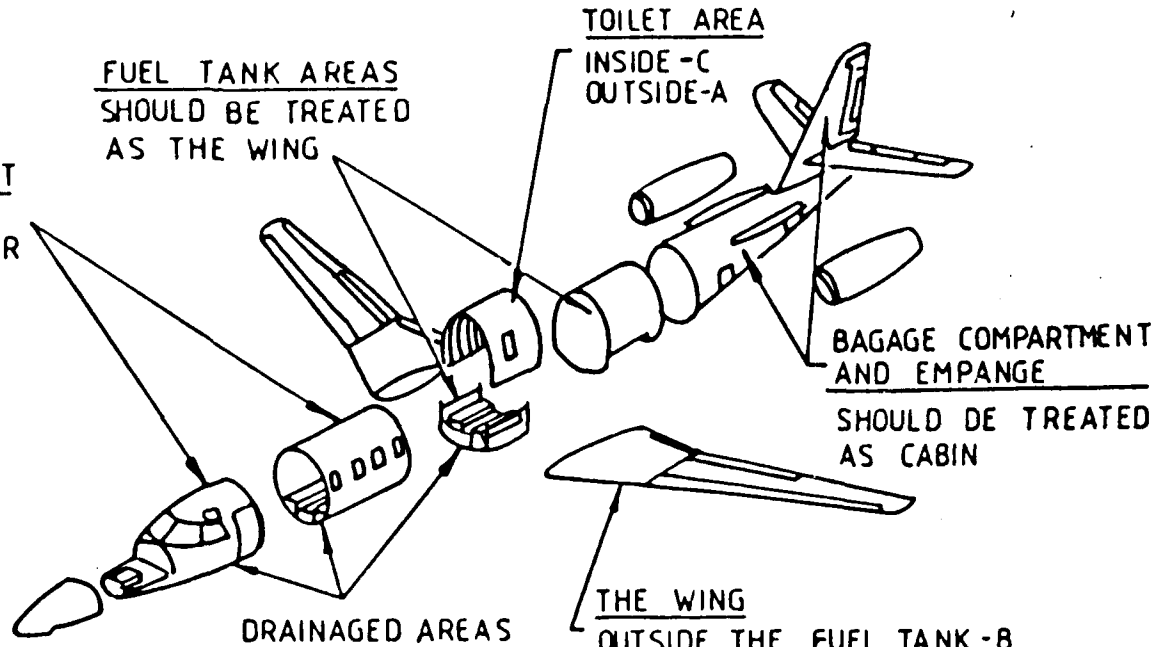
- SUBSTANTIATION TESTING
  - SPECTRUM LOADING CRACK GROWTH
  - SENSITIVY STUDY - TRUNCATION & CLIPPING
  - RETARDATION COEFFICIENTS
  - [redacted]
  - MARKER LOAD EFFECTS

- COLD WORK TEST PROGRAM

ENVIRONMENTAL EFFECTS

PARALLEL TESTING IN GROUND TEST AND MATERIAL DEPARTMENTS INCLUDE THE FOLLOWING TESTS:

- SUMP TANK WATER
- HIGH HUMIDITY AIR
- 3.5% NaCl SOLUTION
- TEMPERATURE EFFECTS



AIR PLANE ENVIRONMENT / LABORATORY ENVIRONMENT	A	B	C	D
LAB AIR	30%	30%	15%	
WET AIR	65%	25%	50%	
-65% F AIR		40%		
3.5% SALT	5%	5%	15%	
SUMP TANK WATER			20%	20%
JET FUEL				80%

COLD WORK TEST PROGRAM

TO QUANTIFY BENEFICIAL EFFECT OF COLD WORKING

TO VALIDATE ANALYTICAL PROCEDURES

I A I PREVIOUS EXPERIENCE

IN PRODUCTION AND RETROFIT ON K F I R FIGHTER

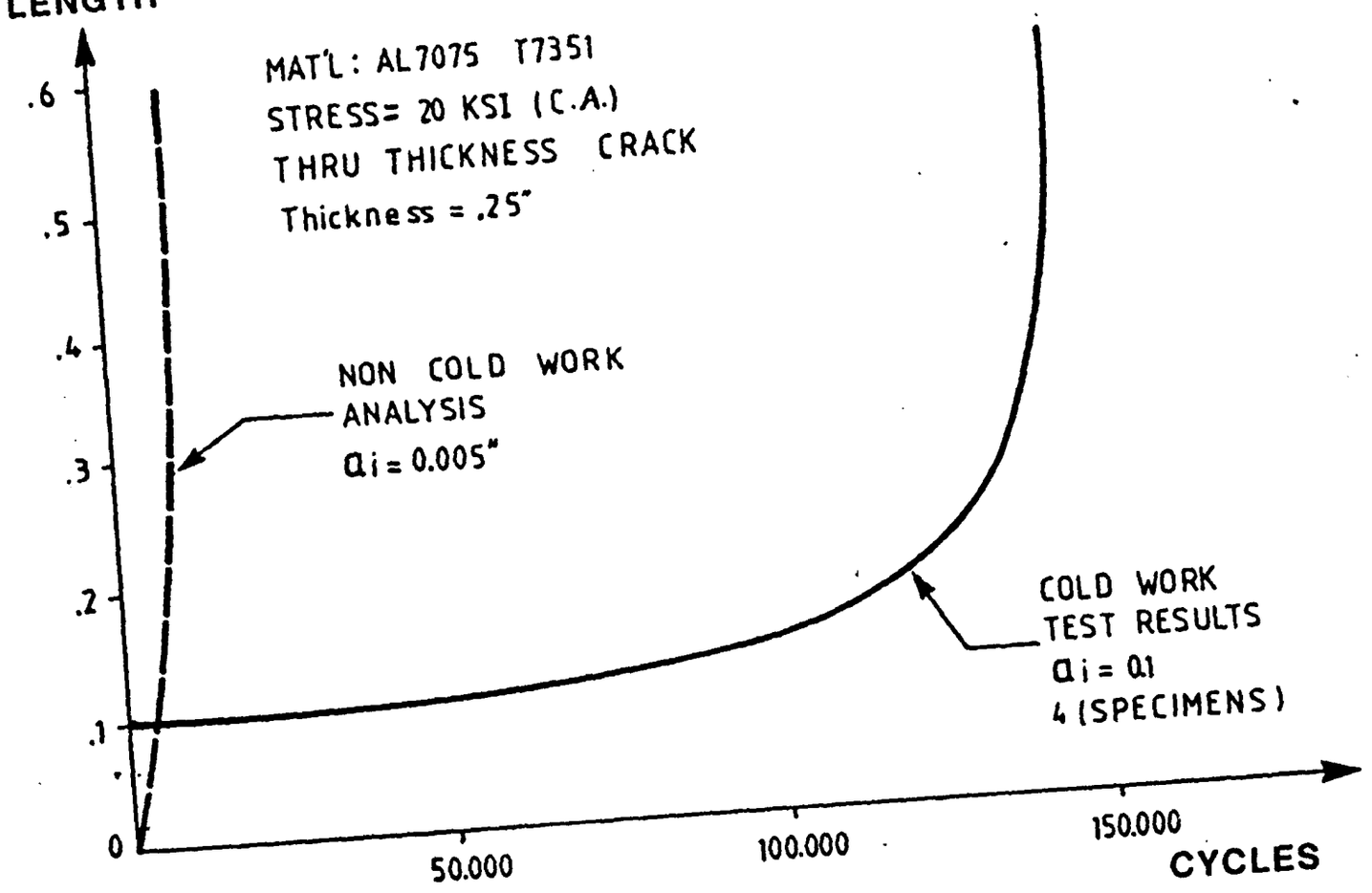
- COMPONENT TESTS
- FULL SCALE FATIGUE TEST

NUMEROUS SPECIMEN TEST PROGRAM

- INITIATION
- PROPAGATION

MATAN - CIVIL AIRCRAFT

COLD WORK TEST RESULTS (ai=0.1") VS. NON COLD WORK ANALYSIS (ai=0.005")



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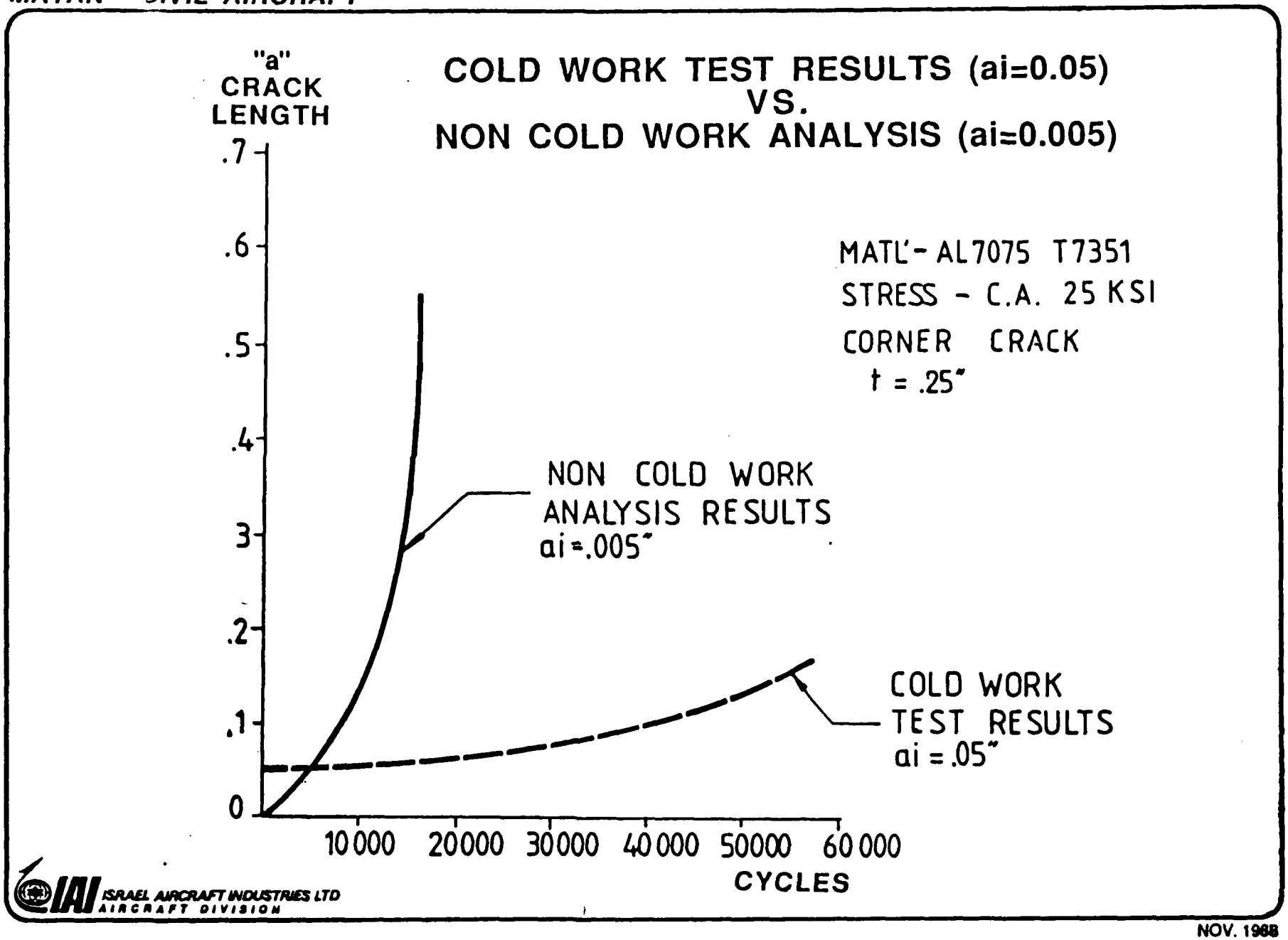
COLD WORK TEST RESULTS (ai=0.05)
VS.
NON COLD WORK ANALYSIS (ai=0.005)
"a"
CRACK
LENGTH
.7
.6
.5
.4
3
.2
.1
0
MATL'- AL7075 T7351
STRESS - C.A. 25 KSI
CORNER CRACK
t = .25"
NON COLD WORK
ANALYSIS RESULTS
ai=.005"
COLD WORK
TEST RESULTS
ai = .05"
10000
20000
30000
40000
50000
60000
CYCLES
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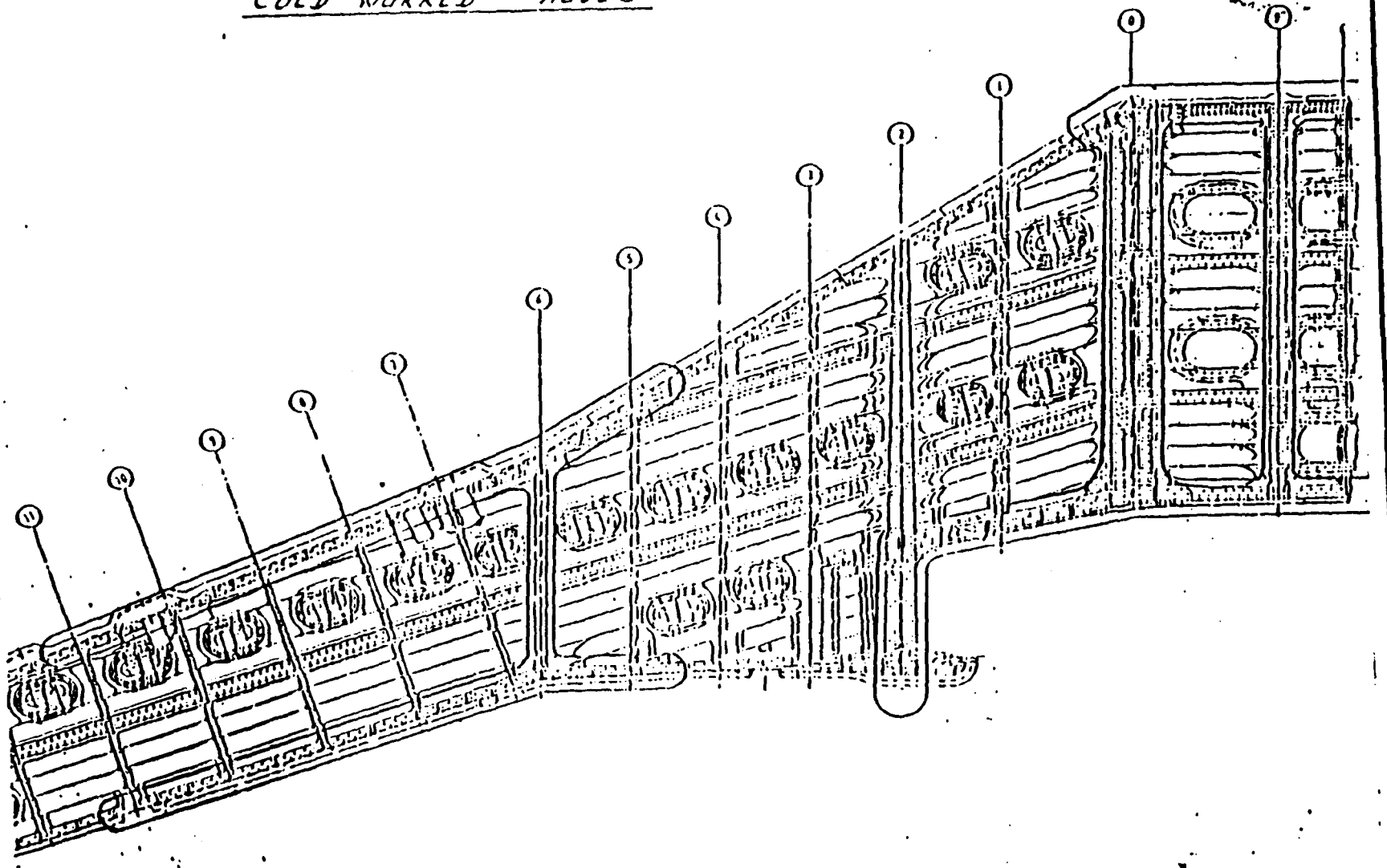
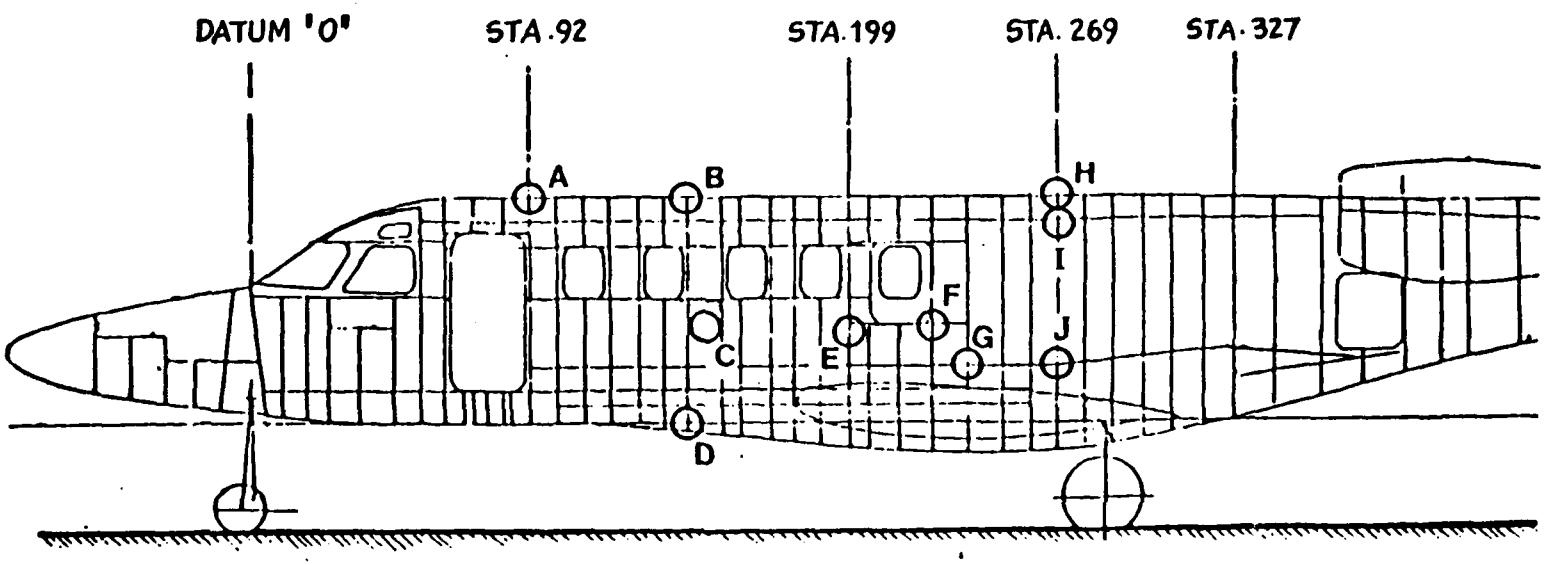
COLD WORKED HOLES

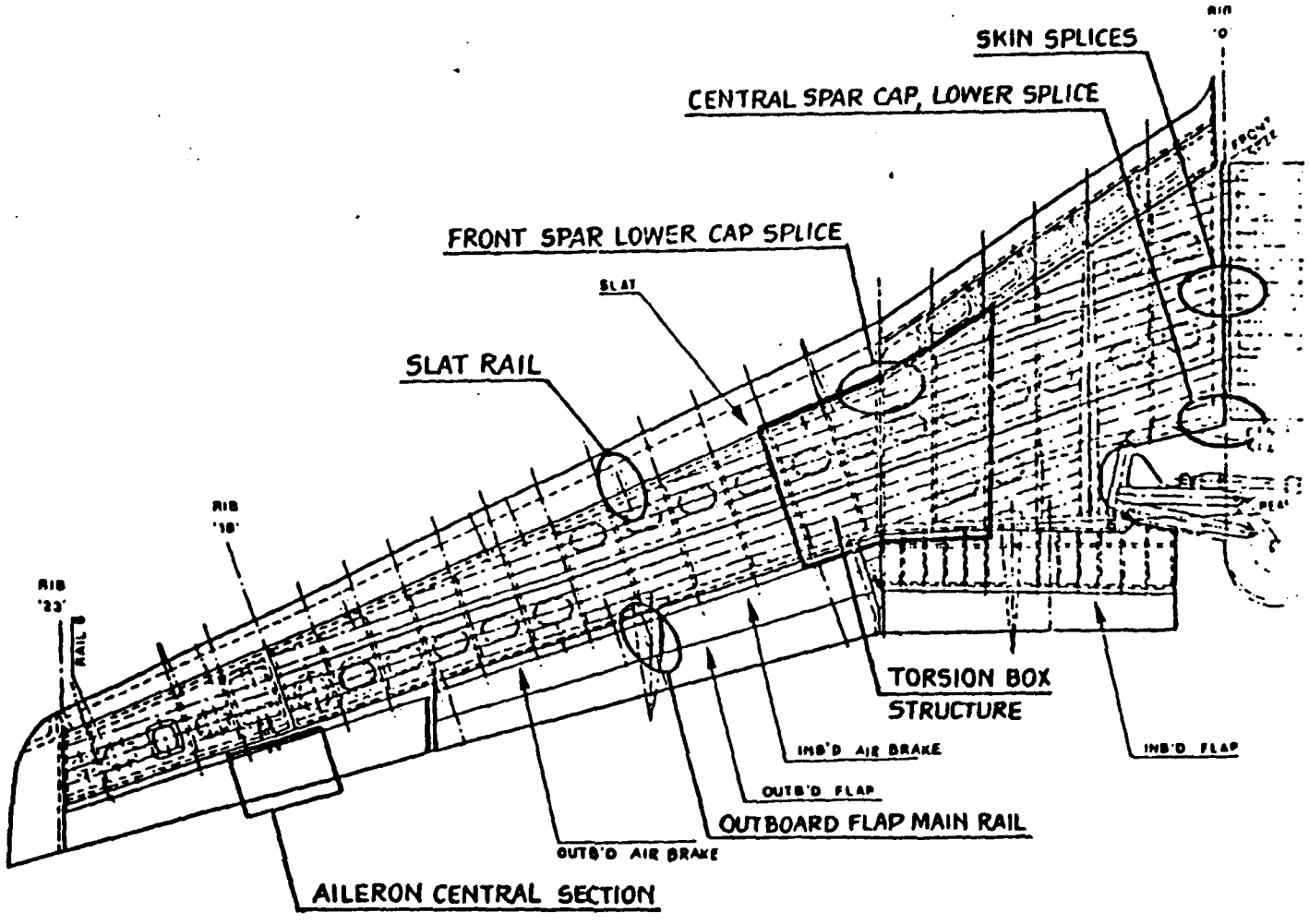


LOCATIONS OF FUSELAGE DAMAGE TOLERANCE TEST COMPONENTS

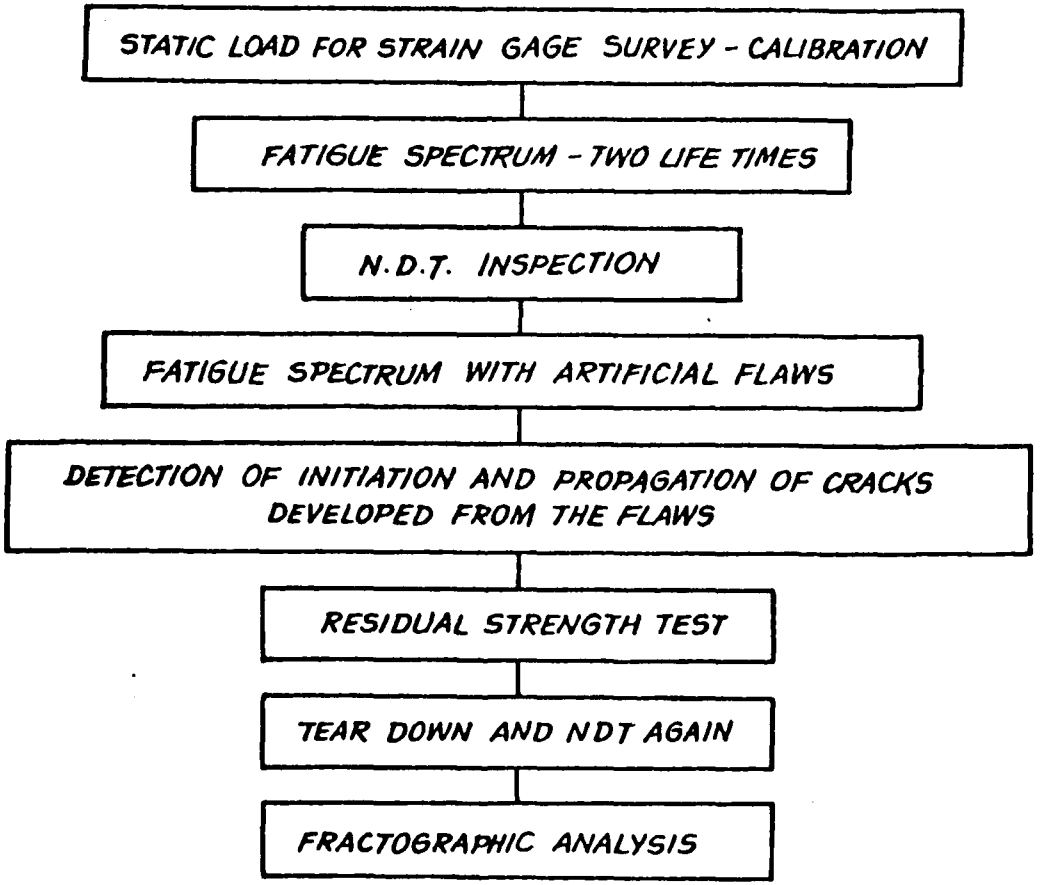
A. Door Frame Upper Segment
B. Typical Frame Upper Segment
C. Fuselage Curved Panel
D. Typical Frame Lower Segment Splice
E. Main Frame Splice
F. Emergency Exit Corner
G. Frame Splice
H. Upper Longitudinal Skin Splice
I. Upper J-Beam Attachment
J. Aft Pressure Bulkhead - Stiffener Attachment to Machined Part



LOCATIONS OF WING DAMAGE TOLERANCE TEST COMPONENTS

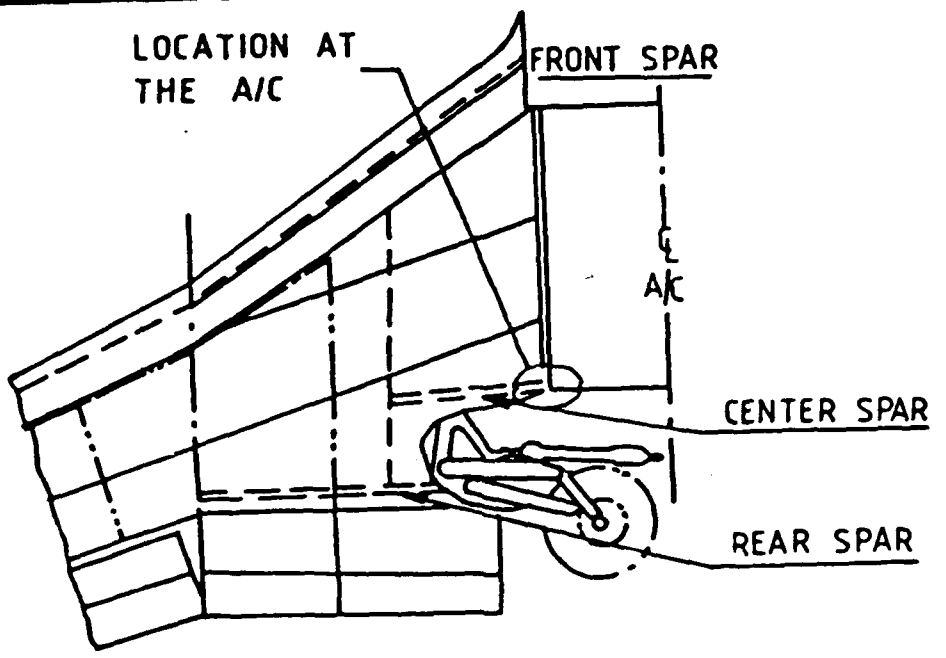


STRUCTURAL COMPONENTS TEST PROCEDURE

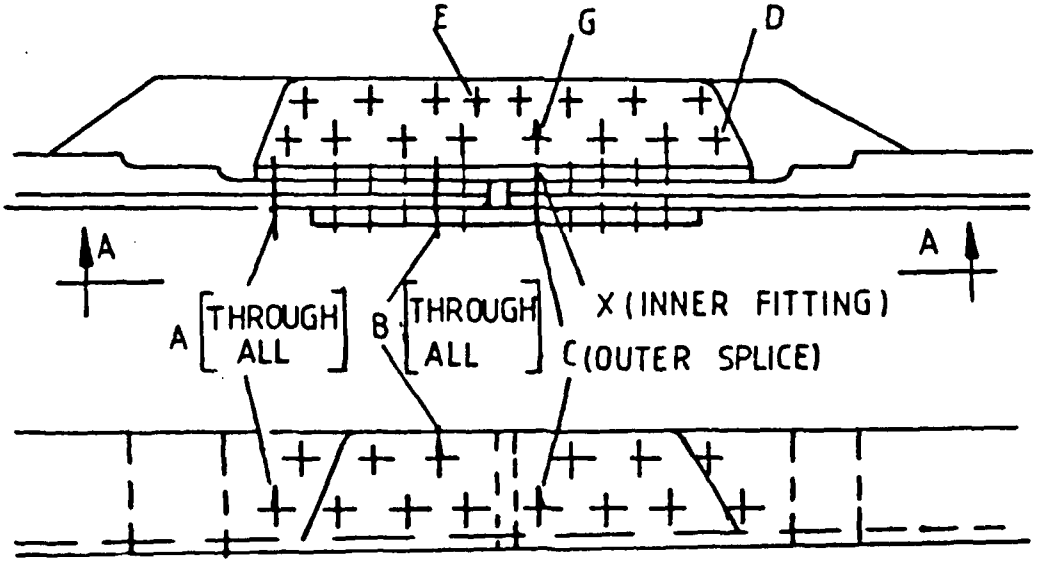


DATA FROM THE TEST

- CRACK GROWTH RATE
- RESIDUAL STRENGTH
- ESTABLISHING METHODS FOR N D T INSPECTIONS
- IDENTIFYING NEW CRITICAL LOCATIONS



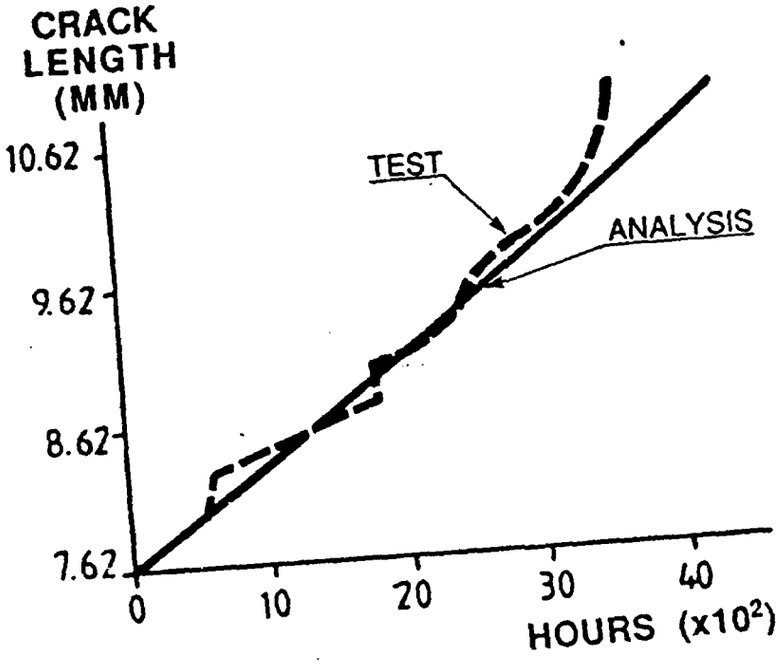
LOCATION OF THE ANALYSED ZONE IN THE A/C

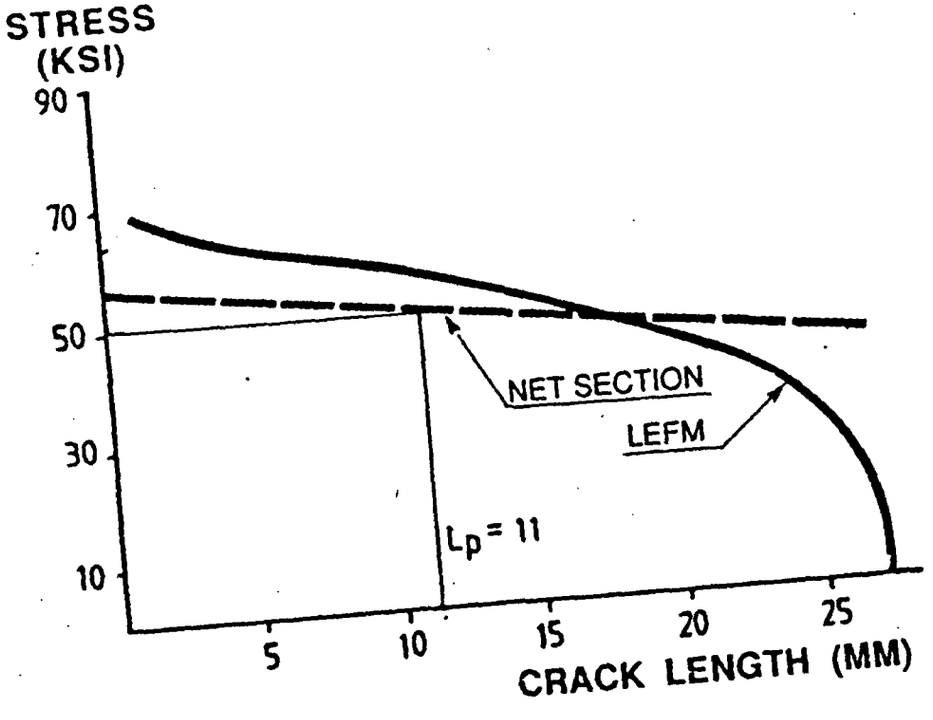


VIEW A-A

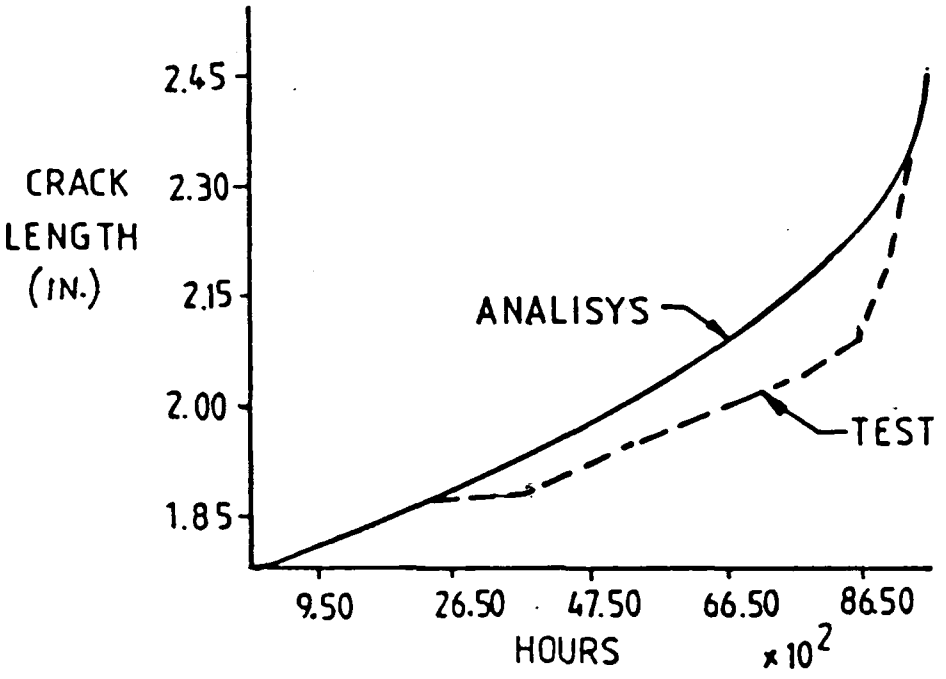
TEST SPECIMEN AND ARTIFICIAL FLAW LOC.

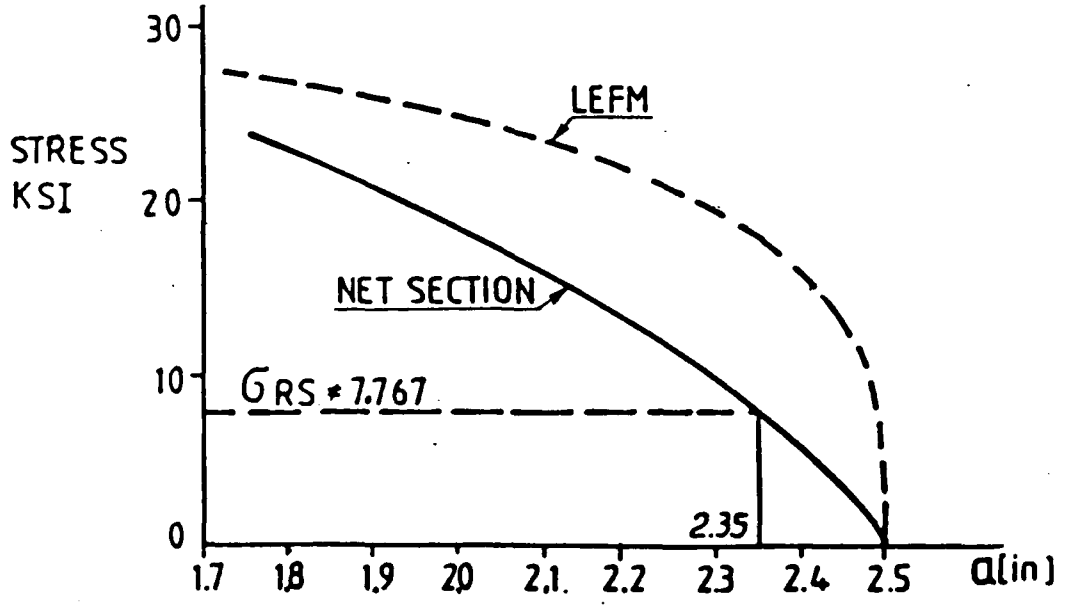
"ASTRA" DAMAGE TOLERANCE TEST CENTER SPAR LOWER CAP ATTACH





"ASTRA"- DAMAGE TOLERANCE TEST-CENTER SPAR LOWER CAP ATTACH.





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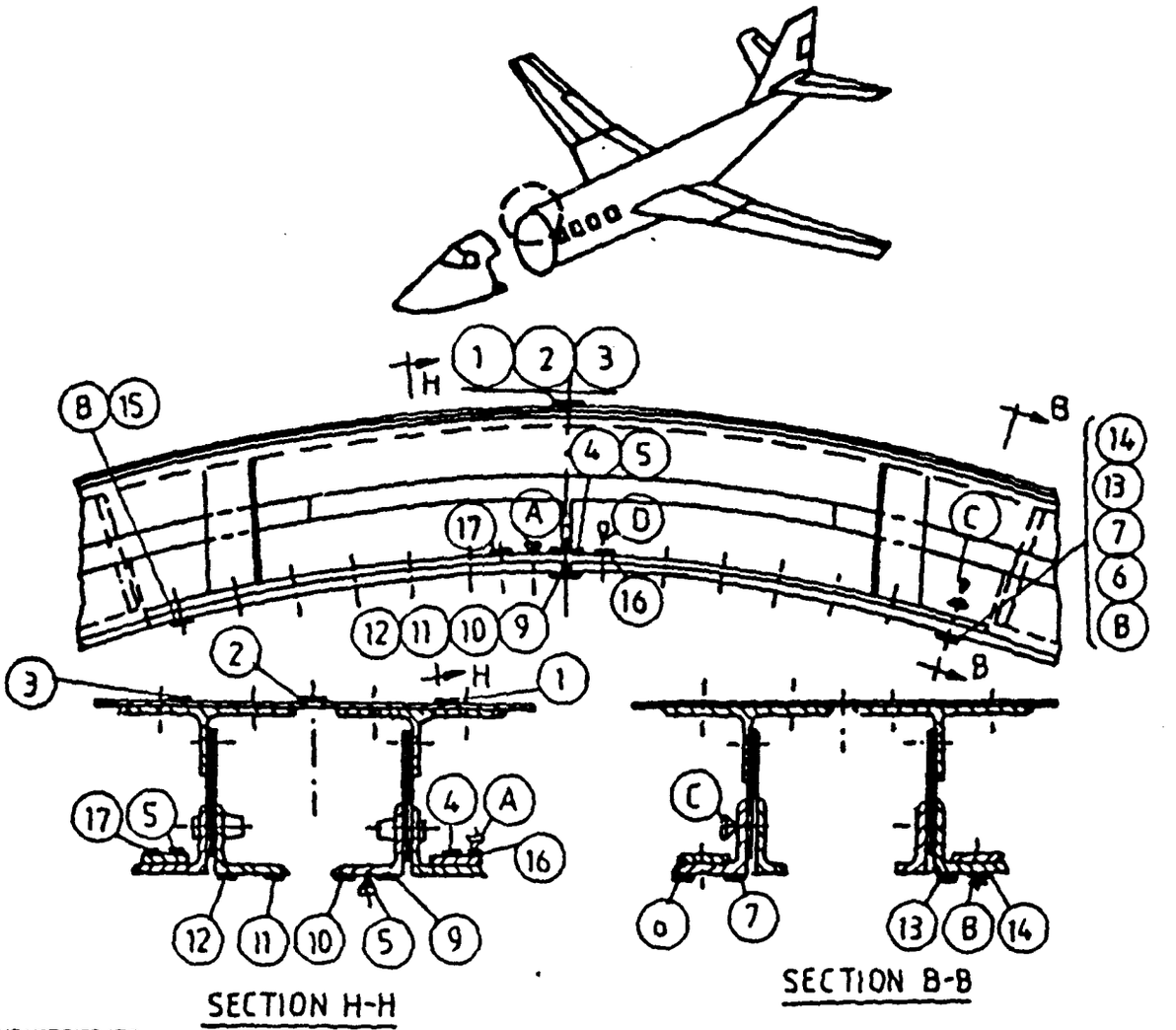
DOOR FRAME AS STA. 92.78 UPPER SEGMENT
DAMAGE TOLERANCE TEST COMPONENT



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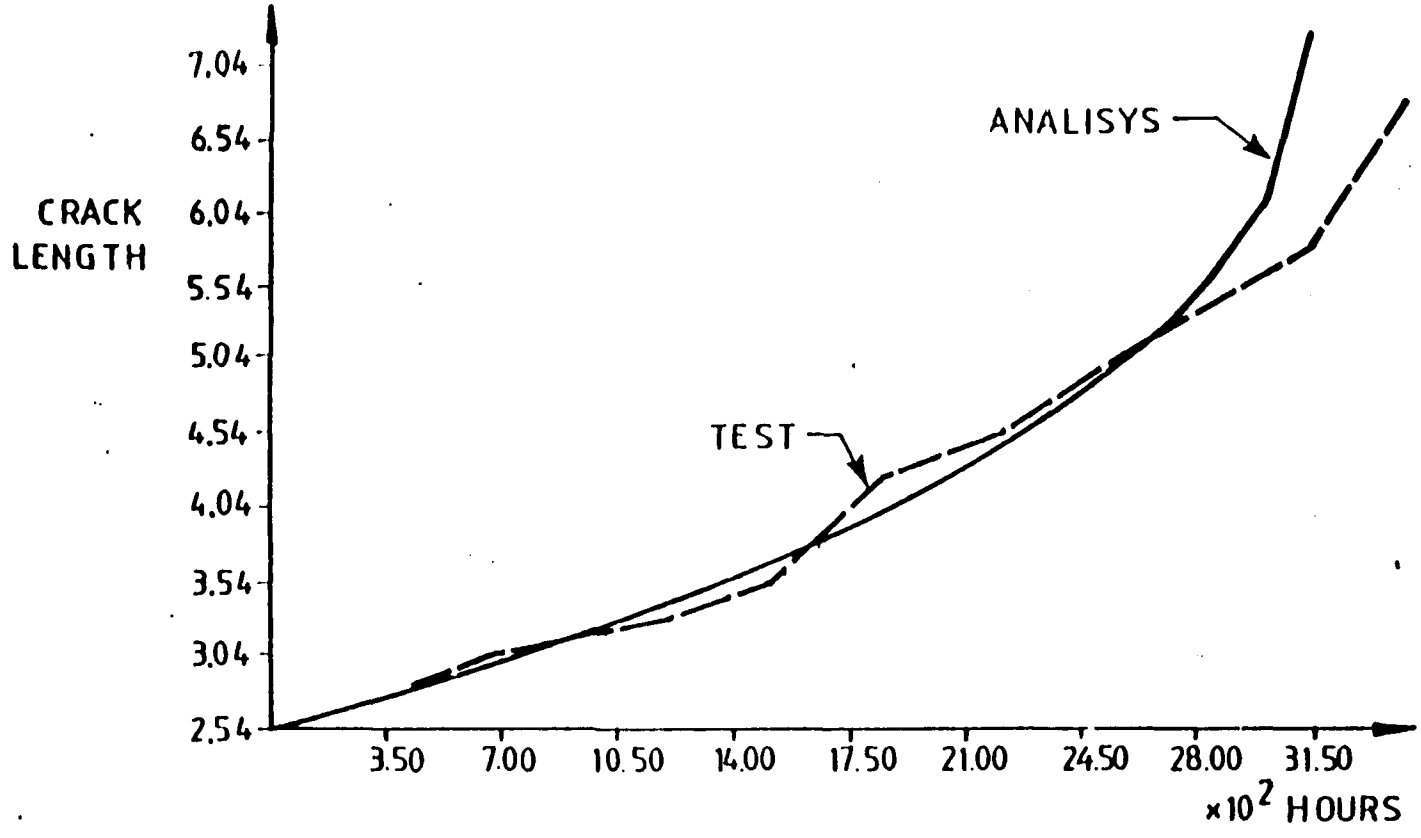
DOOR FRAME 92.78 DAMAGE TOLERANCE TEST



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FULL SCALE DAMAGE TOLERANCE TESTS (F.S.D.T.T.)

F.S.D.T.T. IS DESIGN TO EVALUATE:

- PRODUCTION MANUFACTURING DETAILS
- FATIGUE - CRACK GROWTH BEHAVIOR UNDER ANTICIPATED SERVICE CONDITIONS
- THE MOST POTENTIAL FRACTURE CRITICAL LOCATIONS
- CONTINUEING DAMAGE
- DETAIL STRUCTURAL DESIGN AND CRACK ARREST FEATURES
- STRUCTURAL WEAK SPOTS NOT FOUND BY ANALYSIS
- INSPECTABILITY OF CRITICAL LOCATIONS
- RESIDUAL STRENGTH AND QUALIFICATION OF STRUCTURE

FULL SCALE DAMAGE TOLERANCE TEST PROCEDURE

STATIC CALIBRATION LOADING

FATIGUE TEST LOAD CYCLING (2-3 LIFES)

N.D.T. INSPECTION

FATIGUE SPECTRUM WITH CRACKS AND/OR FLAWS

CRACK PROPAGATION MEASUREMENT

STATIC RESIDUAL STRENGTH TEST

TEAR DOWN AND N.D.T.

FRACTOGRAPHIC ANALYSIS

FULL SCALE FUSELAGE DAMAGE TOLERANCE TEST

TEST SPECIMEN

- FULL SCALE PRESSURIZED CABIN

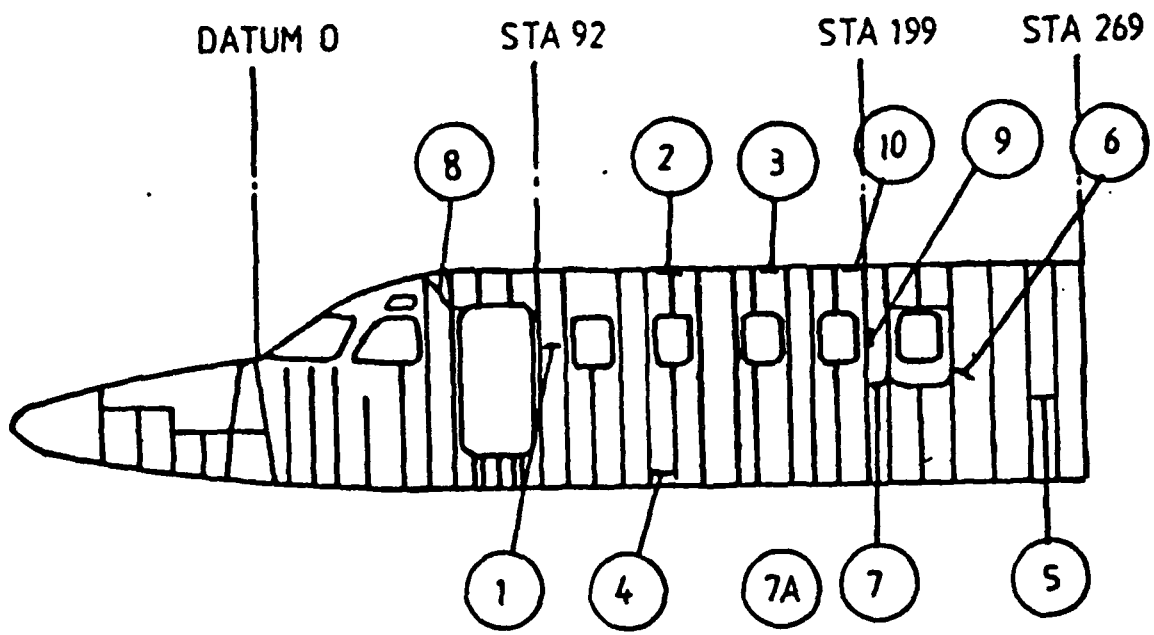
TEST LOADING

- GUST AND MANEUVRE
- PRESSURIZATION
- CRUISE
- DEPRESSURIZATION
- LANDING

ANALYSIS

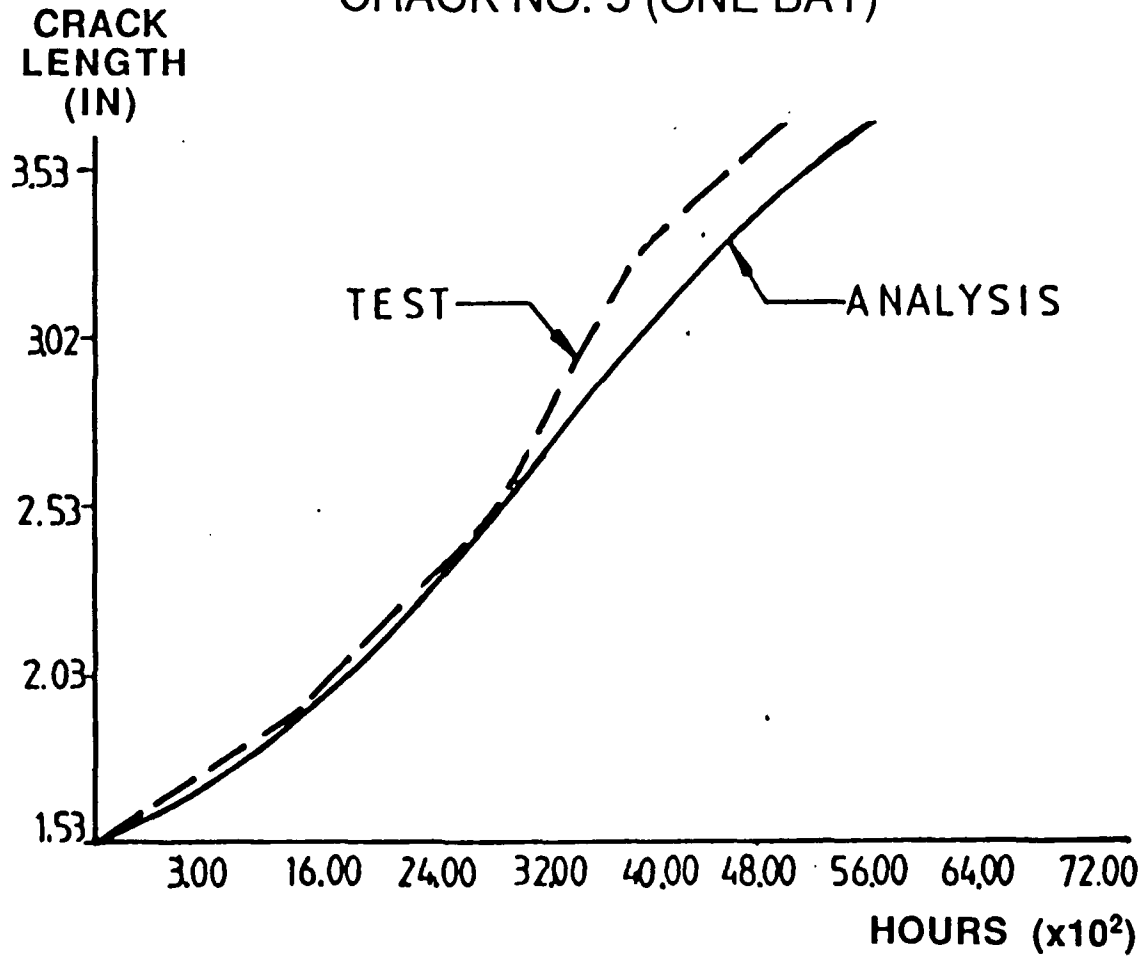
- "CRACK 4" PROGRAM WAS EMPLOYED
- WALKER EQUATION FOR CRACK GROWTH ANALYSIS
- WHEELER EQUATION FOR RETARDATION EFFECT
- BULGING EFFECTS WERE TAKEN INTO ACCOUNT

CRACK LOCATION



TEST VS. ANALYSIS
FULL SCALE FUSELAGE TEST

CRACK NO. 5 (ONE BAY)



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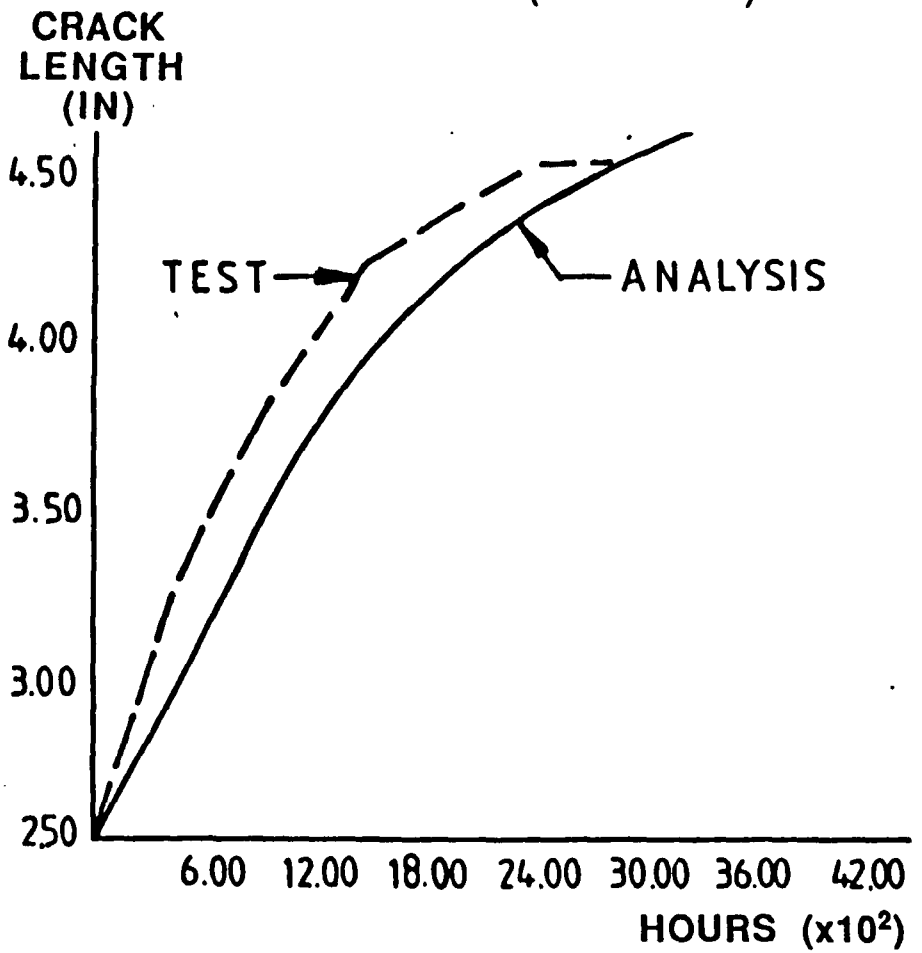
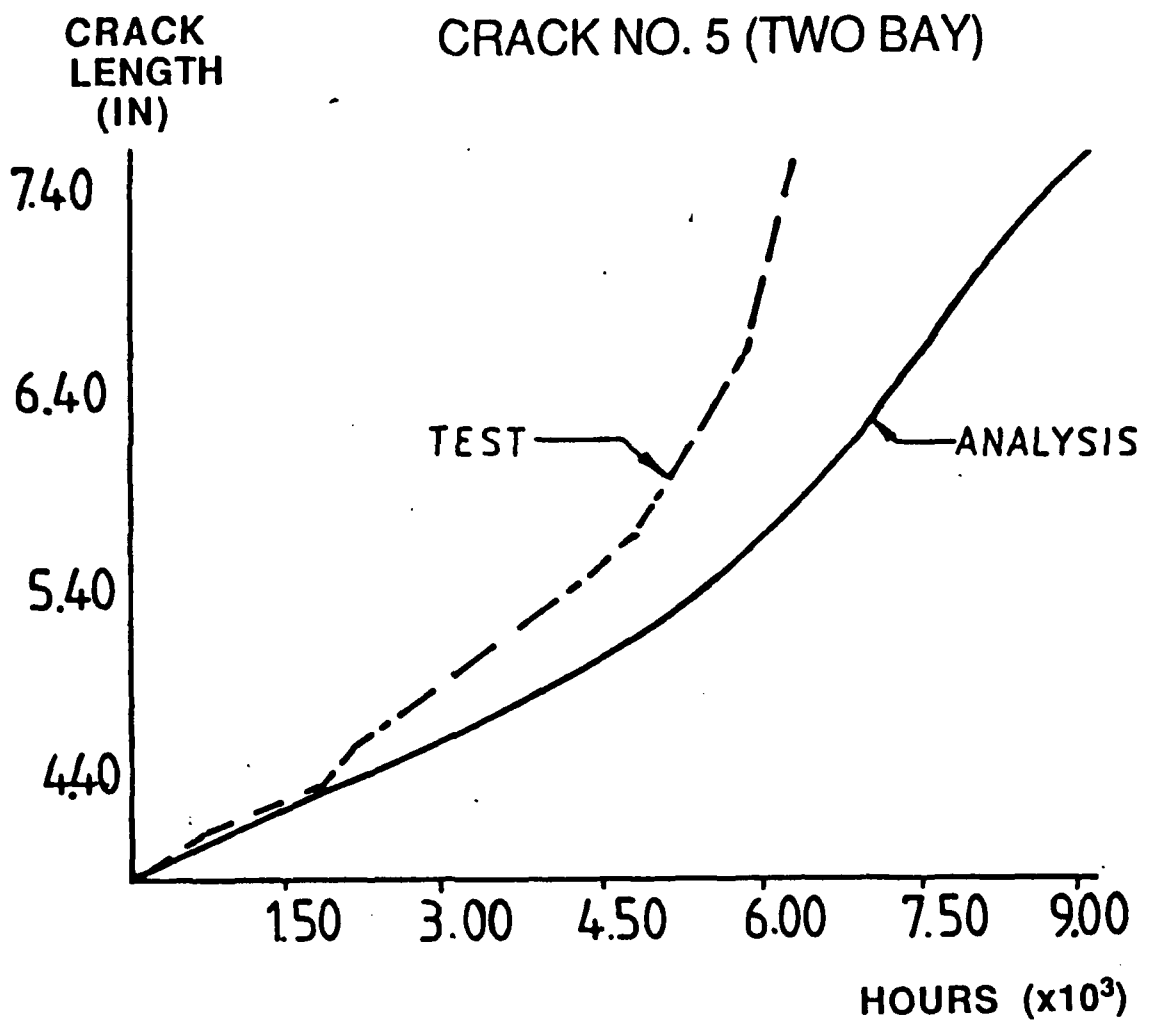
TEST VS. ANALYSIS
FULL SCALE FUSELAGE TEST

CRACK NO. 3 (ONE BAY)



TEST VS. ANALYSIS
FULL SCALE FUSELAGE TEST



FULL SCALE WING DAMAGE TOLERANCE TEST

TEST SPECIMEN

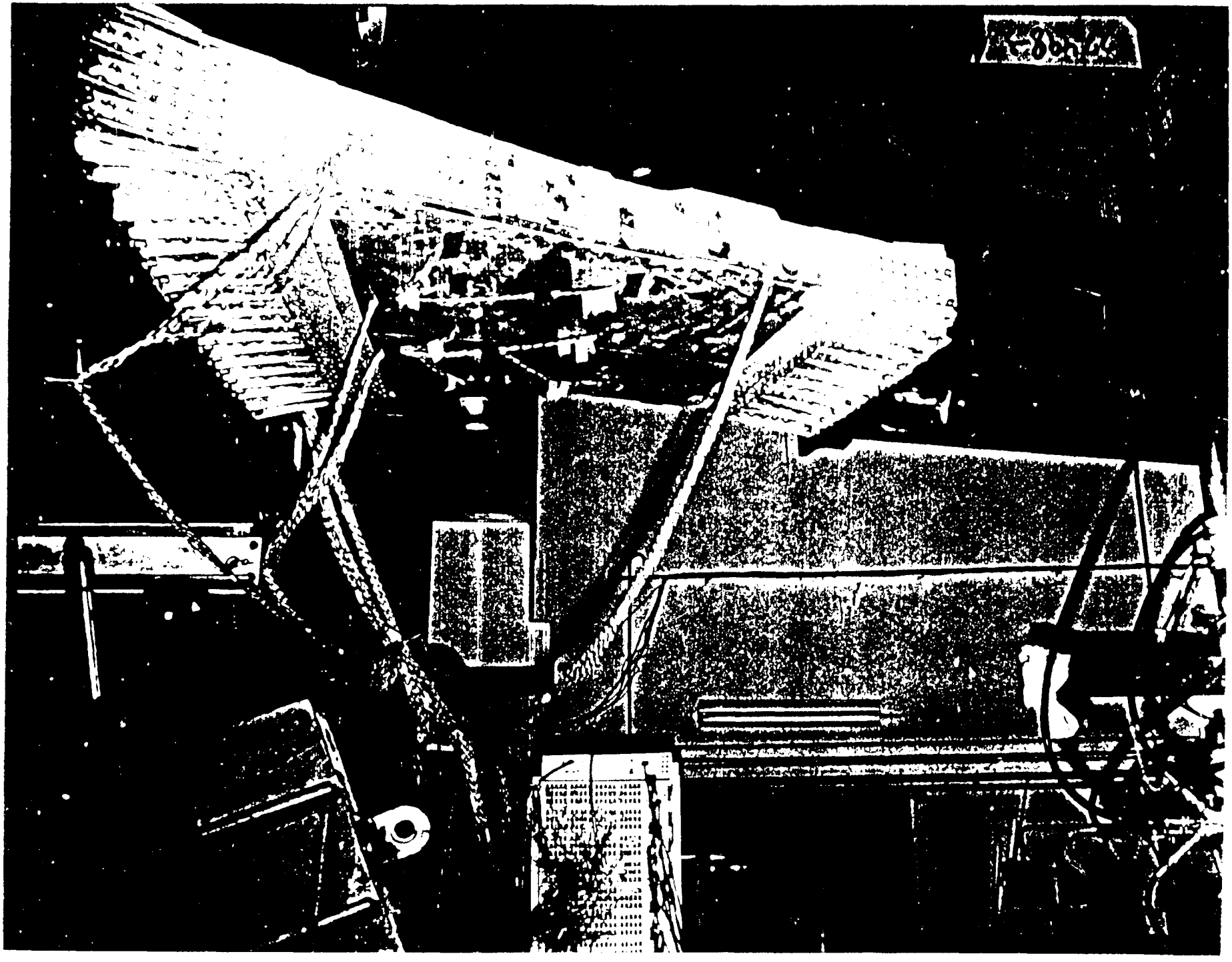
- FULL SCALE OF WING TORSION BOX MOST CRITICAL SECTION (RIBS 4-8)

TEST DURATION

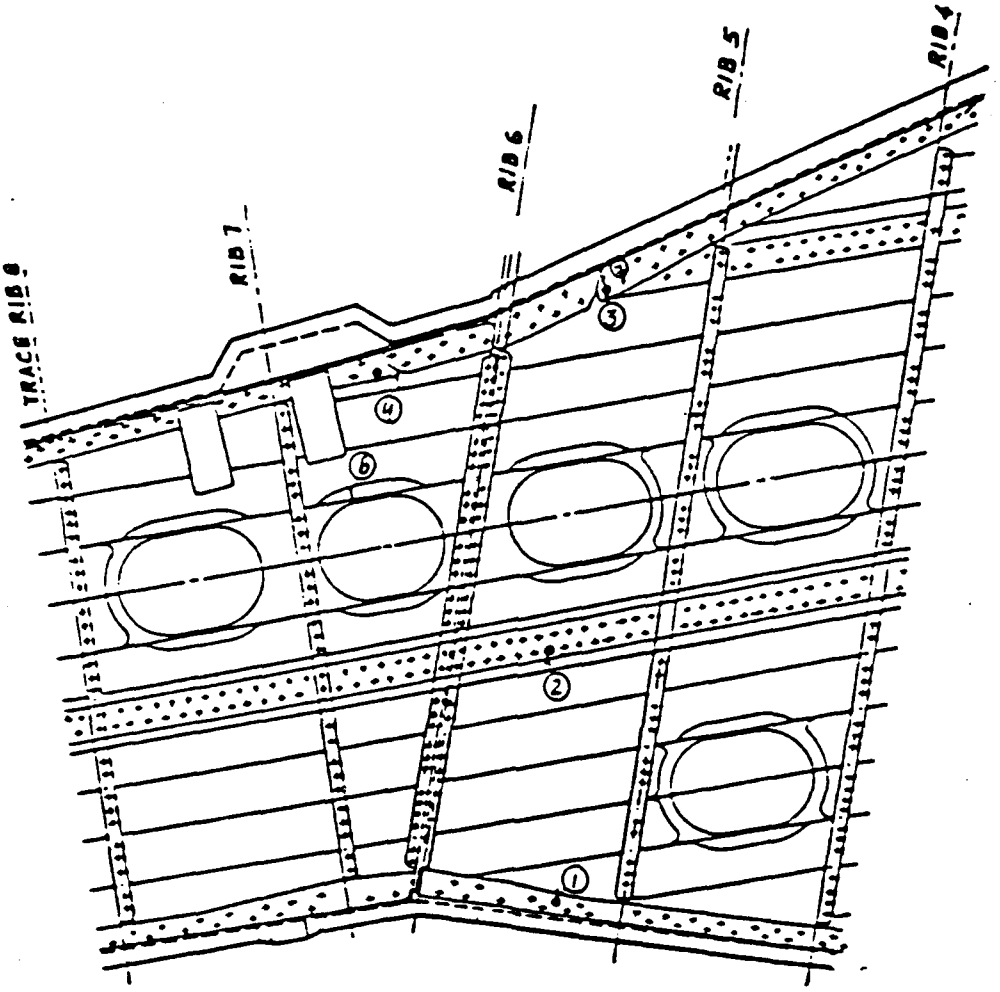
- FATIGUE TESTING 2 LIFE TIMES (40,000 FLIGHT HOURS)
- DAMAGE TOLERANCE TESTING 3 LIFE TIMES (60,000 FLIGHT HOURS)

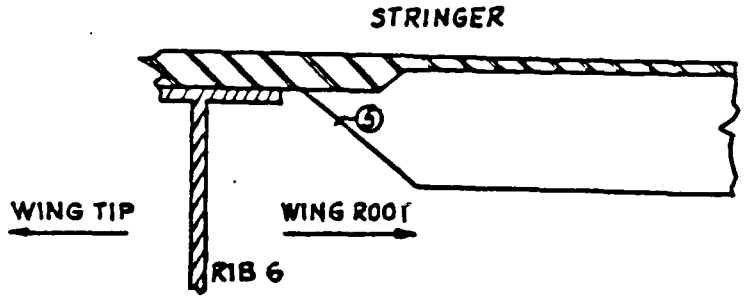
ANALYSIS

- "CRACK 4" PROGRAM WAS EMPLOYED
- WALKER EQUATION FOR CRACK GROWTH ANALYSIS
- WHEELER EQUATION FOR RETARDATION EFFECT
- ADVANTAGE OF COLD WORKING EFFECTS



LOCATION OF ARTIFICIAL FLAWS IN THE LOWER SKIN AND IN STRINGER RUNOUT OF THE WING





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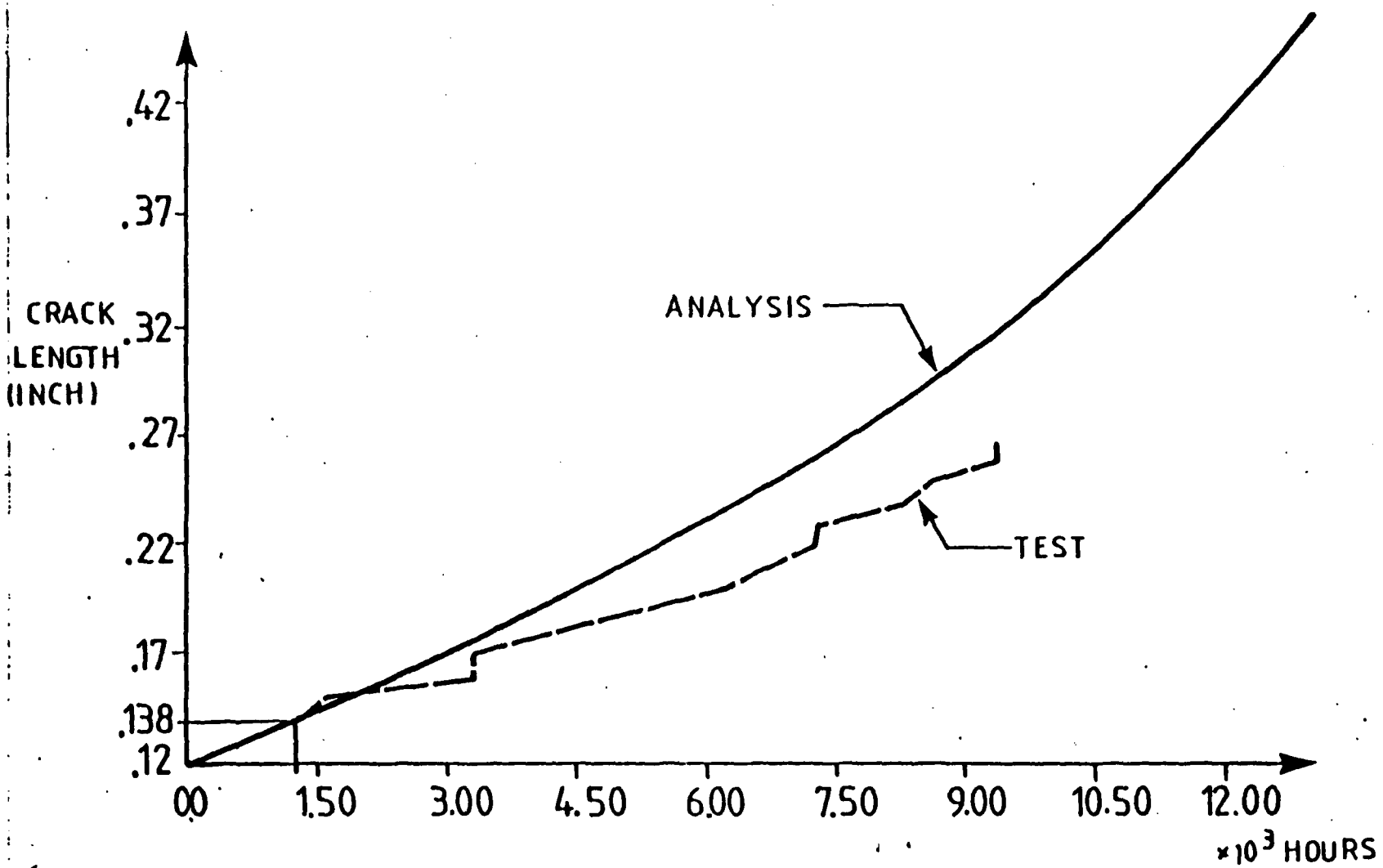
LOWER REAR SKIN BETWEEN RIBS 5-6 AT REAR SPAR



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CRACK PROPAGATION AND RESIDUAL STRENGTH ANALYSIS

- ▶ STATE-OF-THE-ART FRACTURE MECHANICS TECHNIQUES
  - ▷ GEOMETRY FACTORS FOR THE DESIGN DETAILS FROM LITERATURE AND TEST DATA
  - ▷ MATERIAL PARAMETERS FROM RELIABLE SOURCES SUCH AS MIL-HDBK-5C AND TEST DATA
  - ▷ STRESS VARIATIONS FROM DETAILED FINITE ELEMENT MODELS
- ▶ STATE-OF-THE-ART CRACK RETARDATION TECHNIQUES
- ▶ FINITE ELEMENT APPLICATION FOR STRESS INTENSITIES INC. PLASTICITY BY STRAIN ENERGY RATE

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BEN-GURION INTERNATIONAL AIRPORT ISRAEL ZIP CODE 70100
TEL. (03) 9713111 TELEX: ISRAV - IL 381002,381014,381033
CABLES: ISRAVIA, TELEFAX: 9712290, 9713131, 9711762

THE LENGTH OF THE INSPECTION INTERVAL IS VERY SENSITIVE TO THE CHOISE OF THE DETECTABLE CRACK SIZE.

ARBITRARY CHOICH OF THE DETECTABLE CRACK SIZE LEADS TO DIFFERENT PROBAILITIES OF DETECTION FOR DIFFERENT INSPECTION METHODS.

BASED ON A LARGE BODY OF INSPECTION DATA GENERATED DURING AN EXTENSIVE RESEARCH PROGRAM SPONSORED BY THE USAF IN WHICH STRUCTURAL COMPONENTS WERE INSPECTED BY HUNDREDS OF INSPECTORS, I.A.I DEVELOPED A PROBABILISTIC METHOD FOR THE DETERMINATION OF INSPECTION INTERVALS.

THE CRITERION USED FOR THE RATIONALE IS THAT THE PROBABILITY OF CRACK DETECTION MUST ALWAYS BE THE SAME, REGARDLESS OF THE INSPECTION PROCEDURE, THE SPECIFICITY OF THE INSPECTION PROTOCOL, AND THE ACCESSIBILITY OF THE LOCATION.

I A I CRITERIA FOR DETERMINATION OF INPECTION INTERVALS

PROBABILISTIC METHOD BASED ON A LARGE BODY OF INSPECTION DATA (PROGRAM SPONSORED BY U.S.A.F.)

DETECTABLE CRACK SIZE BASED ON

- **ACCESIBILITY**
- **SPECIFICITY**
- **N.D.I. METHOD**
- **SKILL OF THE INPECTOR**

THE PROBABILITY OF CRACK DETECTION IS ALWAYS THE SAME

- **95% FOR MULTIPLE LOAD PATH**
- **99% FOR SINGLE LOAD PATH**

PROBABILITY OF DETECTION BASED ON INSPECTABLE CRACK SIZE (NOT TRUE CRACK SIZE)

INSPECTION CATEGORY CLASSIFICATION

ACCESSIBILITY / SPECIFICITY	GOOD	REASONABLE	POOR
ONE LOCATION	A(1)	B(2)	C(3)
SMALL AREA	B(2)	C(3)	D(4)
GENERAL AREA	C(3)	D(4)	E(5)

a_0 FOR VARIOUS INSPECTION METHODS AND CATEGORIES

METHOD / CATEGORY	ULTRASONIC	PENETRANT	EDDY CURRENT	X-RAY	VISUAL
A(1)	0.036	0.04	0.056	0.1	0.222
B(2)	0.073	0.08	0.112	0.2	0.444
C(3)	0.109	0.12	0.168	0.3	0.666
D(4)	0.145	0.16	0.224	0.4	0.888
E(5)	0.182	0.20	0.280	0.5	1.111

DIMENSIONS IN INCHES

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CONCLUSIONS

THE ASTRA STRUCTURE WAS A PIONEER IN FULLY SUBSTANTIATION ACCORDING TO THE D.T REQUIREMENTS.

THE WHOLE PROCESS WAS CLOSELY FOLLOWED UP BY DR. DAVID BROEK A WELL KNOWN EXPERT IN FRACTURE MECHANICS AND BY MR. TOM SWIFT THE NATIONAL RESOURCE SPECIALIST IN FRACTURE MECHANICS .

AT THE COMPLETION OF THE D.T CERTIFICATION OUR EVALUATION WAS PRAISED BY MR. TOM SWIFT.

SUMMARY

An extremely detailed presentation was made by IAI engineers. It was obvious from the material covered that IAI, with guidance provided by Dr. David Broek, an internationally well-known authority in fracture technology, are conductiong an excellent damage tolerance evaluation of the 1125 structure. The analysis methods used are conventional and are well-supported by a comprehensive element and component test program. It is the opinion of this NRS that the scope of the evaluation is equivalent to any made so far and in many areas is more detailed than for other programs.

Sicerely yours,

Tom Swift
National Resource Specialist,
Fracture Mechanics/Metallurgy
ANM-101N

AUTOMATIC RECORDING OF OIL ANALYSIS

NOW AVAILABLE AT NDI LABS

Mr. Lawrence Miller

Mr. A.B Souza

Ms. Marilyn Anderson

Mr. Jeff Frenster

BACKGROUND

The Joint Oil Analysis Program (JOAP) was established by a joint Army, Navy, and Air Force regulation as a combined effort to establish and maintain a standard program for all three services. The Oil Analysis Program is based upon the use of Spectrometric Oil Analysis as a means to identify the wearmetal content of oil samples from jet engines, transmissions, gearboxes and hydraulic systems. The wearmetal readings are analyzed and compared against predetermined limits and trend values. The presence of unusual concentrations of an element in the fluid sample can indicate abnormal wear of the equipment. The identification of abnormal wear results in management's decision to repair or remove the equipment from service. The JOAP program is a major preventive maintenance tool and a major factor in the jet engine diagnostic process.

Early identification of potential failures in the modern high technology turbine engine is a vital element in the prevention of catastrophic engine failure. The JOAP program wearmetal readings are just one of the factors that are considered in a comprehensive engine diagnostic program.

The analysis of wearmetal readings from the OAP Laboratory assist in identifying incipient mechanical failures and determining the quality and useful life of the oil. Consequently, potential component failure or failure of the lubricating qualities of the oil can be detected prior to major equipment failure and expensive repair/rebuild procedures.

Oil analysis may also be used to identify inadequate or improper maintenance procedures. Taken to its logical conclusion, early identification of potential failures can prevent the loss of aircraft and aircrew.

The normal procedure that is followed in collecting and processing the oil samples include:

- Collecting a small sample of oil immediately after flight;
- Delivering the sample to the OAP Lab within 1 hour;
- Performing the spectrometric burning process;
- Recording the wearmetal readings;
- Comparing the wearmetal readings with engineering limits;
- Advising Engine Technician of abnormal readings; and
- Reporting the readings to the BOO3 Central Data Bank.

SYSTEM DESCRIPTION

The Automatic OAP recording capability is provided in conjunction with the installation of a standard base level engine diagnostics system.

The standard Engine Diagnostic System used in the Air Force today is the Comprehensive Engine Management System, Increment IV (CEMS IV). CEMS IV has the capability to collect and correlate data from the aircraft Turbine Engine Monitoring System (TEMS), the Maintenance Data Collection System (MDCS), The Maintenance Management Information and Control System (MMICS), the Core Automated Maintenance System (CAMS), and the Oil Analysis Program Laboratory. The entire CEMS IV system is composed of a network of standard USAF microcomputers, Z100/Z248, which are hosted on a Harris MCX-3 minicomputer. The microcomputer workstations are located on the flight-line, OAP Lab and in the Engine Management Branch (EMB). Figure 1 is a system diagram of the typical CEMS IV installation at a Tactical Air Force Base. Figure 2 shows the detailed breakout of the Oil Analysis Laboratory.

The availability of the Z100/Z248 microcomputers in the work areas accelerates the data collection process and provides the engine technician with real time data to determine the proper course of maintenance action. The installation of a CEMS IV system at an Air Force Base provides the Oil Analysis Laboratory with a completely automated data collection and analysis capability. The CEMS IV installation includes an RS 232 interface that allows the Spectrometer to automatically transfer oil wearmetal readings to the Z100/Z248 microcomputer. Software programs developed under the CEMS IV Program are resident on the microcomputer.

The workstation located at the OAP Lab allows the laboratory to monitor up to 800 separate engines and maintain history on the 30 most recent records for each engine. The T.O. 33-1-37 limit data is stored in the microcomputer. The ability to store data and compare oil readings against the T.O. limits, gives the operator automatic analysis capability.

The OAP workstation software has the capability to assist the technician with:

- Daily standardization and calibration procedures.
- The preparation and documentation for the monthly laboratory certification.
- Entry and/or modification of oil analysis data with appropriate error checking.
- Entry and/or modification of oil feedback maintenance data with error checking.
- Preparation of DD2027 forms and the transient engine logs as required.
- Automatic preparation of correctly formatted B003 records from oil and maintenance records, which may be sent to B003 through the host or sent on floppy disks if the workstation is deployed.
- The automatic archiving of all data as it passes the purge window of 30 records.
- The automatic loading of data collected by a remote base.

These functions are implemented through a series of menu driven options. The functions are available on the OAP workstation in two versions:

The host version ties the Z100/248 in the OAP Lab with the CEMS IV minicomputers. This version provides unlimited storage capacity for oil sample history and integrates the oil data with parametric data gathered from other engine monitoring devices. The host version provides the engine technician with complete diagnostics capability.

CEMS IV OVERVIEW

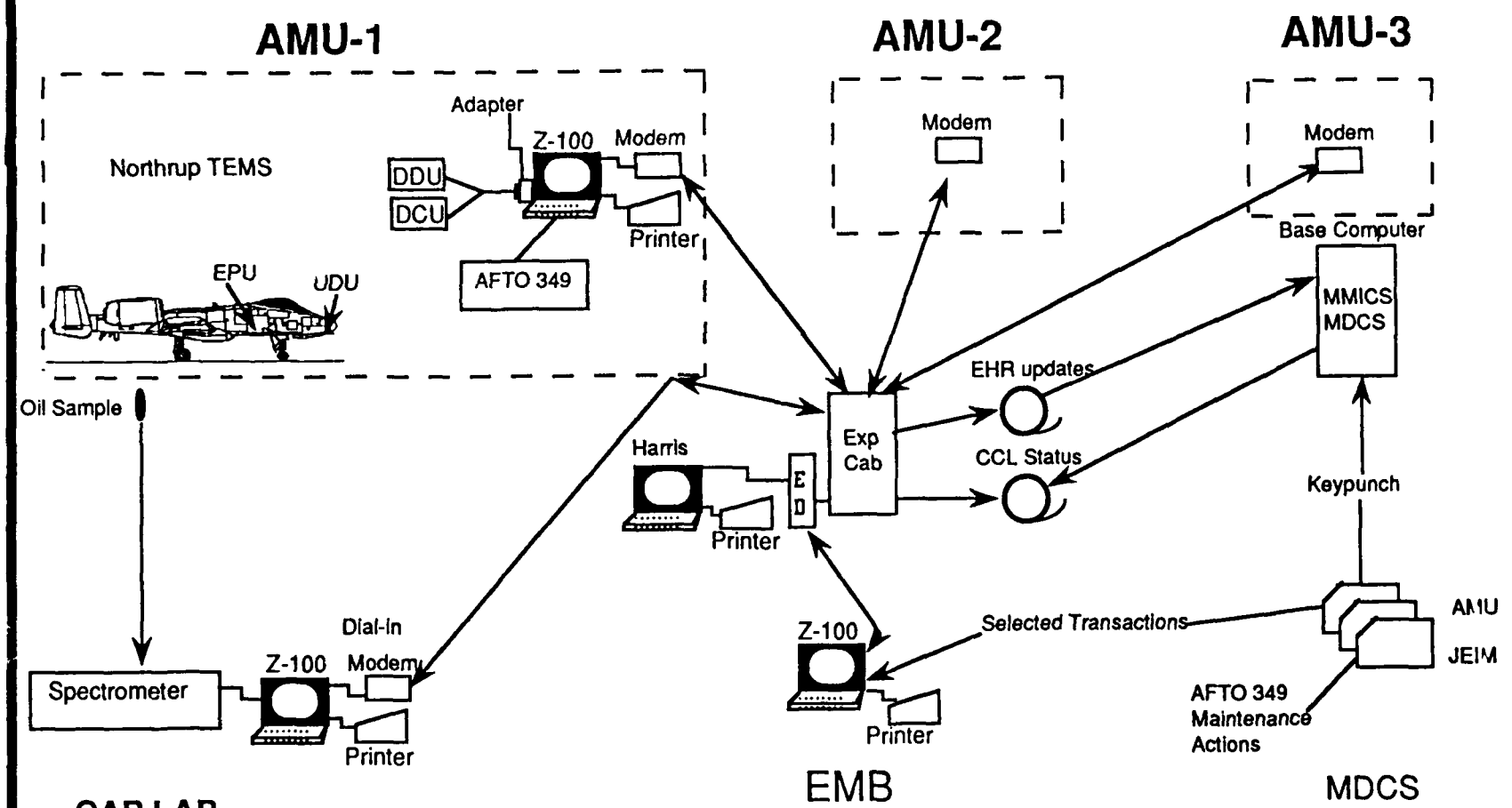


Figure 1: Typical CEMS IV System

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

L9-0987.32d2

OIL ANALYSIS PROGRAM

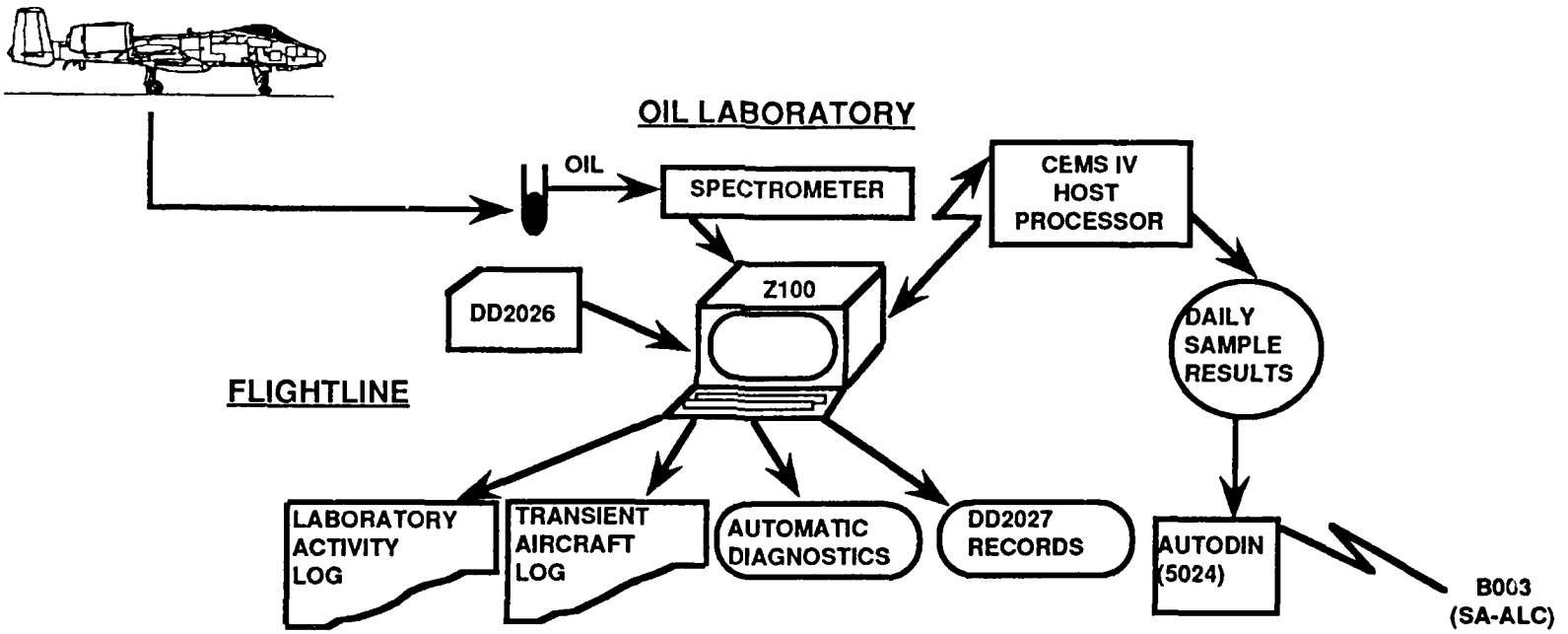


Figure 1: Automated Oil Laboratory

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

The standalone version permits automatic processing of spectrometer data, limits the number of oil samples to thirty, and does not provide for integration of other engine data with OAP data. It permits transferring of burn data to the host at the end of each burn session. The standalone version was developed primarily to support unit deployment.

The automation of the OAP Laboratory is accomplished with the addition of the RS232 interface card to the Baird A/E35U-3 spectrometer. The RS232 card permits the spectrometer to interface directly with the Z100/Z248 microcomputer. The microcomputer is connected to the CEMS IV host directly or through a leased line modem. The OAP Laboratory workstation includes a printer to allow hardcopy documents to be produced at the OAP Laboratory.

The software to automate the OAP Laboratory is provided through the USAF CEMS IV Program Office and consists of a set of floppy disks that are used to configure the microcomputer and initialize the OAP database. Once the microcomputer database is initialized, individual engine burn records are added to the database. Transient engines are loaded to a separate database in the microcomputer. The Technical Order data is loaded as part of the initialization process and is contained on the floppy disks that are provided as part of the system. Users Manuals are provided as part of the original installation and training is available through the CEMS IV Program Office at Kelly AFB.

After an engine is loaded to the system and the first oil sample is burned, only the database ID and the Engine Serial Number are required to retrieve the engine history. The historical data includes previous burn data (up to 30 records). The OAP operator determines if the wearmetal readings are entered automatically or manually, as selected. The computer program analyzes the wearmetal readings by comparing them to the T.O. limits and identifies those readings which exceed the T.O. parameters. A computer recommendation is made based on those findings; however, the final recommendation is left to the technician's expertise.

Oil Analysis Laboratories are required to maintain voluminous records as a result of the number of oil samples that are processed on numerous pieces of equipment. The record keeping process has always been labor intensive and error prone due to manual transcription of wearmetal readings and equipment identification data. With the installation of CEMS IV workstations and the automation of the OAP Laboratory, the record keeping workload has been reduced, transcription errors eliminated and the technician is able to more profitably spend his efforts diagnosing problem engines. It is not uncommon to have error rates of 0% at CEMS IV bases.

OAP And ENGINE DIAGNOSTICS

USAF aircraft have traditionally relied upon the results of oil analysis to assist engine technicians in determining the health of their engines. As newer, more expensive, high performance and high tolerance turbines are introduced into the USAF inventory, early identification of potential failures becomes more critical. Engine Diagnostic Systems have evolved to the point where on-board sophisticated processors monitor the engine in flight and selected parametric data is downloaded for additional processing on the ground. Ground processing stations assimilate data from various sources, including the OAP Lab, to give the technician a total picture of the health of the engine. In the future automated procedures will be able to better integrate all factors into a system whereby the engine mechanic will be able to pin-point a trouble spot before a failure occurs.

Although the automated OAP capability exists, and is installed at those bases where the CEMS IV is installed, this capability has not been expanded beyond the CEMS IV equipped bases. The CEMS IV expansion program is in source selection and should be on contract by January 1989. Award of the CEMS IV contract will bring the automated OAP capability to 60 additional Air Force Bases. The CEMS IV program is managed at SA-ALC/MMECD, Kelly AFB, Texas. The OAP portion of CEMS IV is coordinated between the CEMS IV Program office and SA-ALC/MMEI.

OIL ANALYSIS PROGRAM

AUTOMATIC RECORDING OF OIL ANALYSIS NOW AVAILABLE AT NDI LABS

L. Miller
A.B. Souza
M. Anderson
J. Frenster

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

OIL ANALYSIS PROGRAM

PROCEDURE

Collect Sample

Deliver to OAP Lab

Spectrometer Burn

Record Wearmetal Readings

Analysis (T.O. Comparison)

Advise Engine Technician

Report Readings to B003

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

OIL ANALYSIS PROGRAM

PRESENT

SPECTROMETRIC OIL ANALYSIS

Automatic Burn
Manual Analysis (T.O. 33-1-37)
Manual Data Transcription
Key Punch
Autodin to B003

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

OIL ANALYSIS PROGRAM

SHORTFALLS

MANUAL ANALYSIS --

- Slow
- Relies on Manual T.O. References
- Time Consuming

MANUAL TRANSCRIPTION --

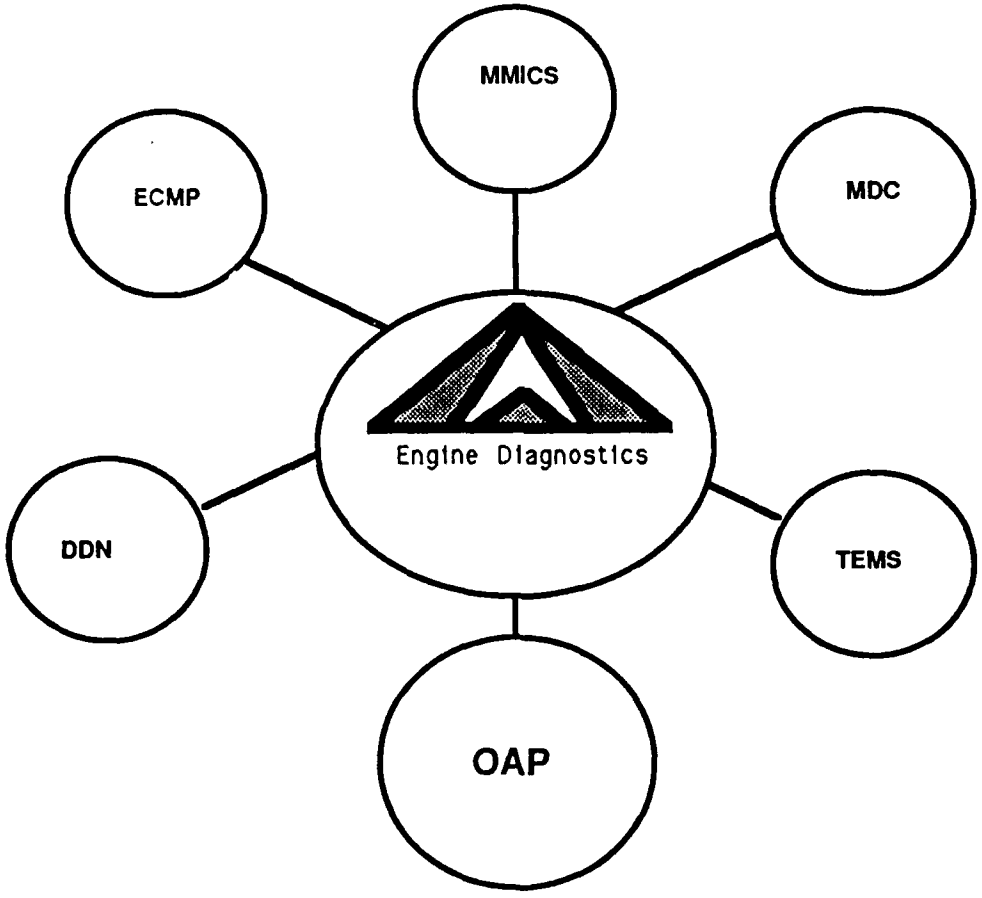
- Slow
- Error Prone
- Time Consuming

KEY PUNCH --

- Error Prone
- Time Consuming

SCT SYSTEMS CONTROL TECHNOLOGY, INC

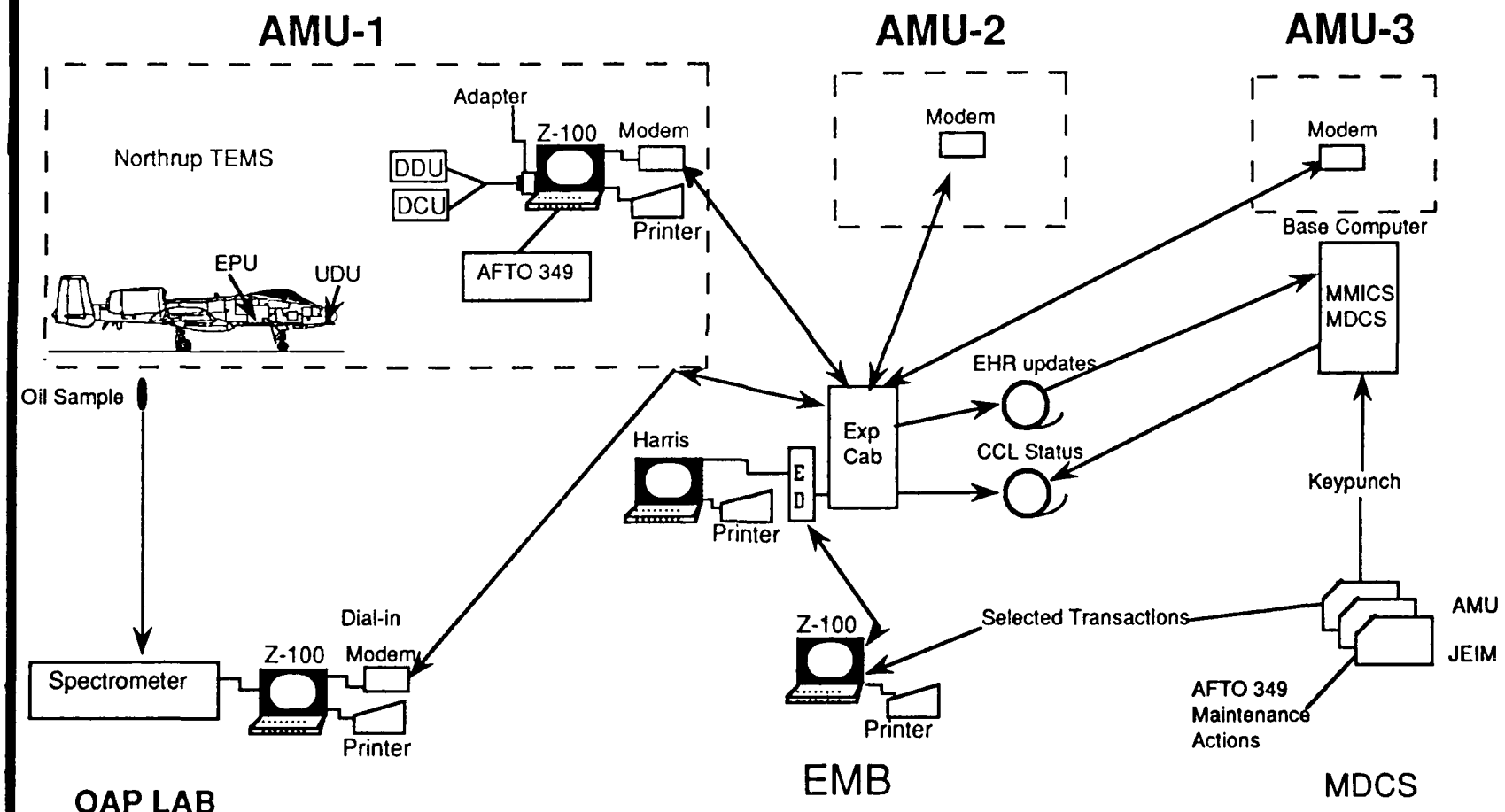
OIL ANALYSIS PROGRAM



SCT SYSTEMS CONTROL TECHNOLOGY, INC.

Oil Lab Briefing Slide 5

CEMS IV OVERVIEW



Typical CEMS IV System

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

L9-0987.32d2

OIL ANALYSIS PROGRAM

CEMS IV PROVIDES

- RS 232 Interface Card
- OAP Software
- OAP Operator Training

GFE

- Z100/Z248

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

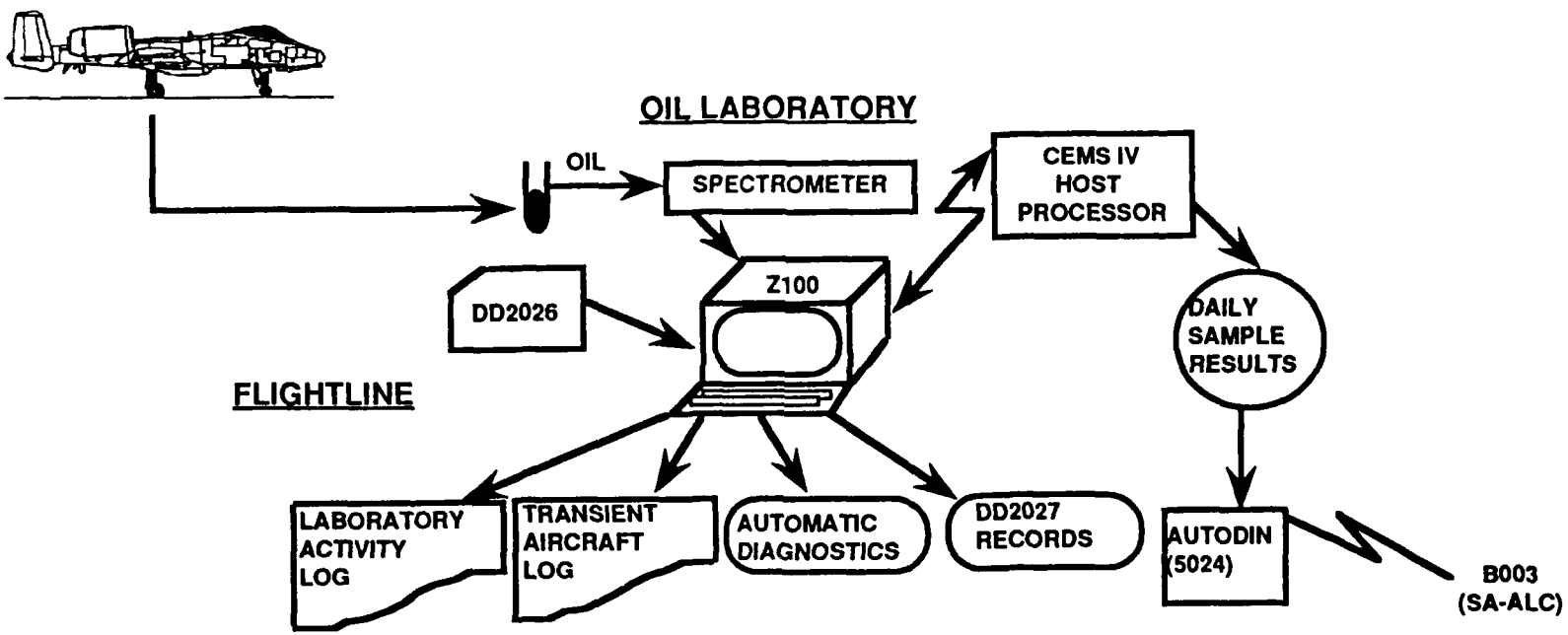
OIL ANALYSIS PROGRAM

SOFTWARE ASSISTS OAP TECHNICIAN WITH

- Daily Standardization and Calibration
- Preparation and Documentation for Monthly Lab Certification
- Entry/Modification of OAP Data With Error Checking
- Entry/Modification of Oil Feedback Maintenance Data With Error Checking
- Preparation of DD 2027
- Automatic B003 Reporting
- Automatic Archiving of Data
- Automatic Loading of Data From Remote Lab

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

OIL ANALYSIS PROGRAM



Automated Oil Laboratory

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

Oil Lab Brief slide 4

OIL ANALYSIS PROGRAM

BENEFITS

AUTOMATIC WEARMETAL RECORDING
- Increased Accuracy
- Reduced Workload

AUTOMATIC ANALYSIS OF WEARMETAL READINGS
- Automatic T.O. Deviation Notices
- Elimination of Errors

AUTOMATIC REPORTING TO B003
- Elimination of Key Punch
- Significantly Reduced Error Rate

SUPPORTS DEPLOYABILITY

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

OIL ANALYSIS PROGRAM

CEMS Program Manager

SA-ALC/MMECD

OAP - CEMS IV Coordination

SA-ALC/MMEI

SCT SYSTEMS CONTROL TECHNOLOGY, INC.

T-38 Severe Usage Spectra Development/Update

Contract No. F41608-86-C-C070

Prepared by: M.P. Kaplan
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**SA-ALC/MMS
Kelly AFB, TX 78241

I. INTRODUCTION

The use of flight monitoring equipment to record the usage of individual aircraft has been existence since the middle1960's. The devices that were first used were limited in the data that they collected. These devices, primarily electronic in nature, included counting accelerometers and VGH recorders. The second generation of these devices include multi-channel recorders such as the MXU-553/A, and those that use solid state recording electronics. None of these methods directly measure the stress (or strain) of the aircraft component. Rather, they measure external parameters (altitude, velocity, n_z, time into the flight, etc.) and through the use of derived mathematical relationships, the investigator can determine the stresses, the crack growth behavior, or the inspection procedures required for many of the aircraft's critical components.

An alternative route to determine the life of an aircraft component is a mechanical device which can measure either the crack growth behavior of a component, or the stresses the components experience. Devices of this sort include crack gages, and mechanical strain recorders (MSR's). Both of these devices are rigidly fastened to the aircraft and experience the same load time history as their underlying structure. The crack gage, actually a mechanical device containing a crack of known size and well defined mathematical properties, will continue to crack as a result of the loads (stresses) that it experiences. By using straightforward mathematical procedures, the crack growth behavior of other structure can be easily determined.

The MSR uses a different approach. It actually measure the deflection of a particular component. It does this by having a foil (wrapping and unwrapping about different spools) rigidly attached to the aircraft structural component. Also contained within this device are two stylii that imprint marks upon the foil. One stylus reacts to the structural deformation of the component to which it is attached, and that movement is directly proportional to the magnitude of the component deflection. The second stylus places a reference line upon the foil. Contained within the MSR is a ratcheting device that drives the wind spool as a function of the deflection. Thus each mark imprinted upon the foil by the stylus is in a new location. Figures 1 through 4 illustrate the MSR device.

The foils are placed in and removed from the aircraft on a periodic basis

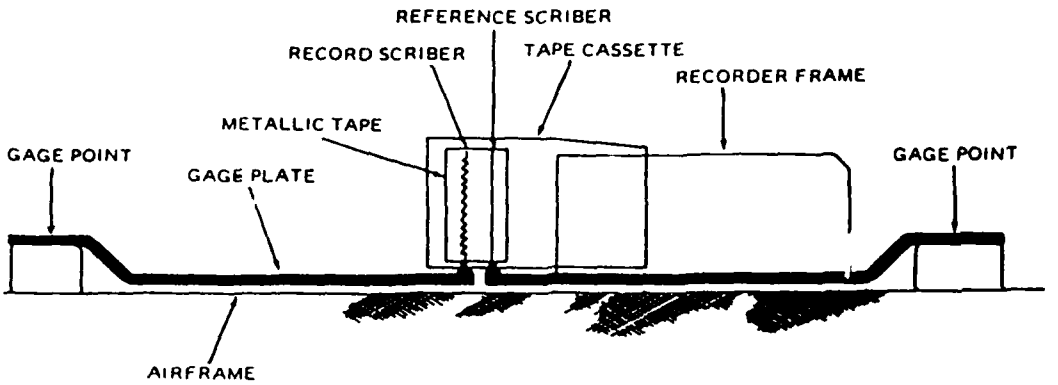


Figure 1 - MSR Recorder

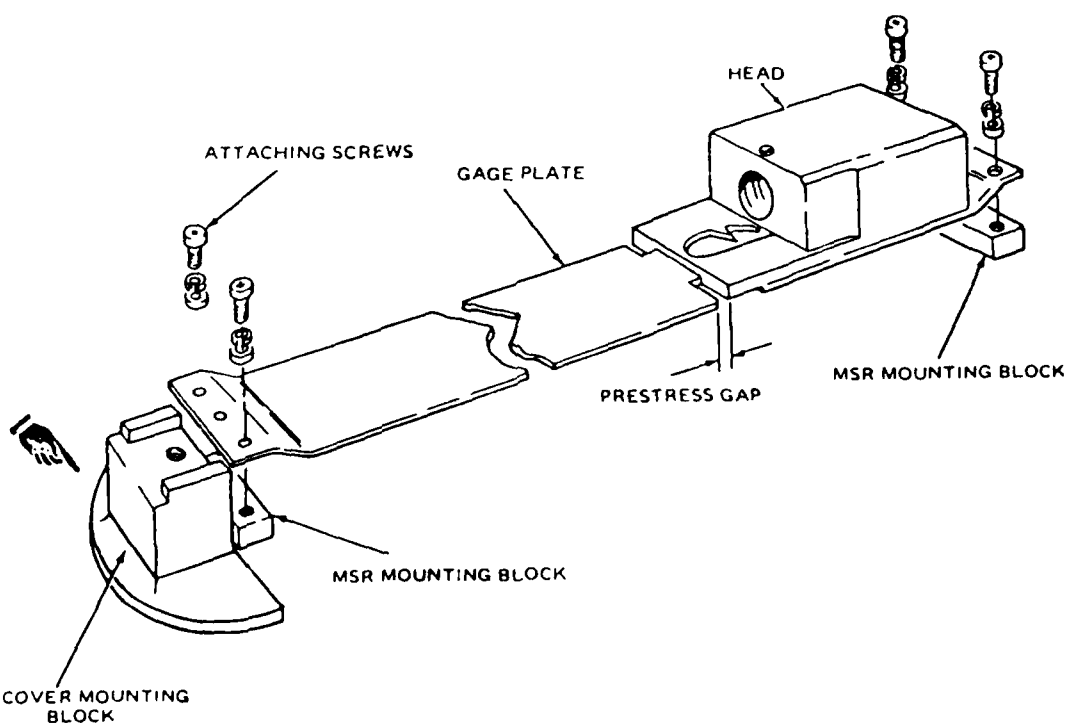


Figure 2 - MSR Mounting

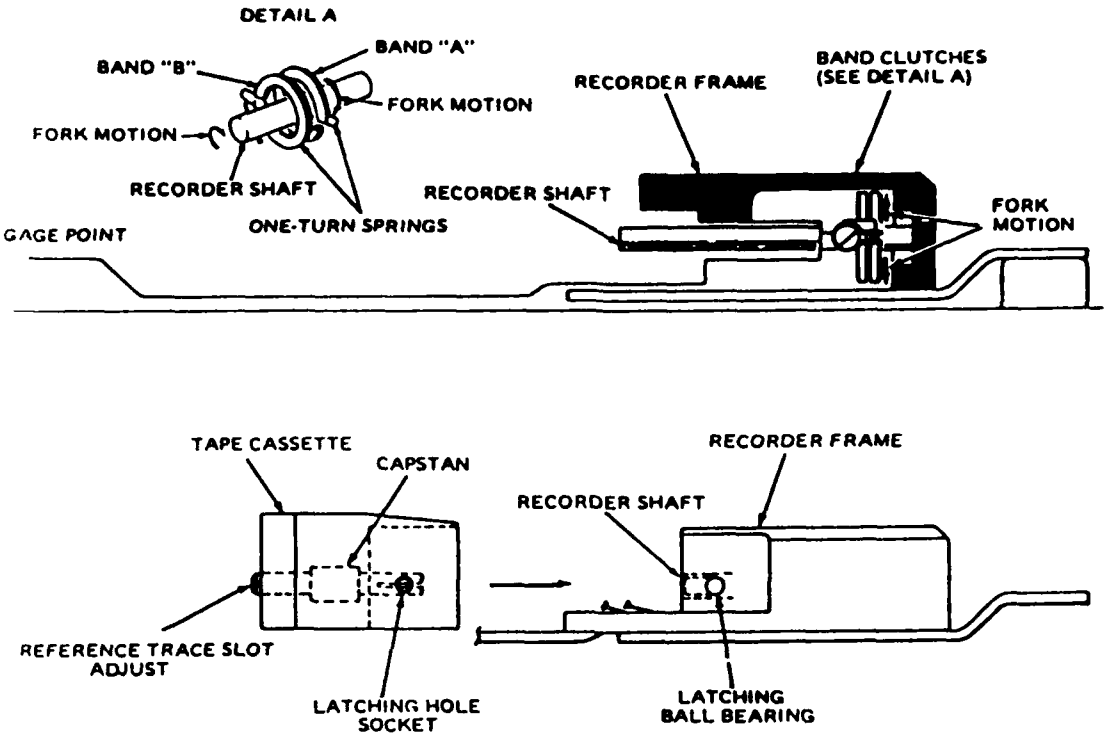


Figure 3 - MSR Cassette Installation and Operation

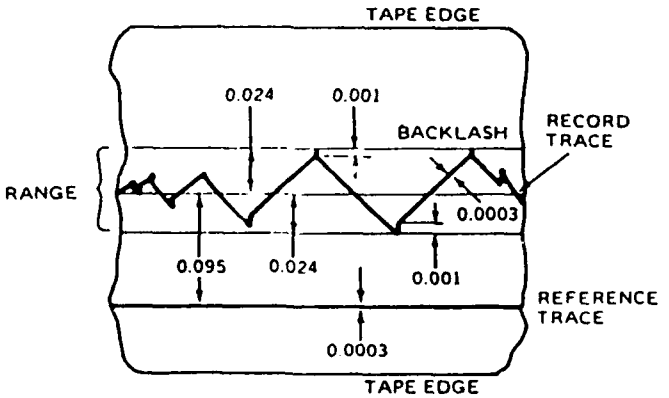


Figure 4 - Recording Pattern

(approximately 50 flight hours). The removed foils filled with flight data are sent to Oklahoma City Air Logistics Center (OK-ALC) where the foils are mechanically placed into a foil reader, and the results are read and digitized. The digitized deflection results, along with the reference line for each cassette, are stored on a magnetic tape. It is from this magnetic tape that the flight behavioral patterns for individual aircraft and usages may be determined.

In the work reported here, digitized MSR data were obtained for four different usages for the years between 1982 and 1985. These were: lead-in fighter (LIF), air combat maneuver (ACM) , Detachment-6 (DET6), and dissimilar air combat training (DACT). For the entire program, there were 45459.9 hours of LIF data, 1339.9 hours of ACM, 2182.1 hours of DET6, and 376.8 hours of DACT. It may be seen that for the amount of supplied data, numerical methods using computer procedures to read, scan, and process the material is essential.

It was the objective of this program of work to read the digitized tapes, transform the deflections into stresses for a selected location on the aircraft, and prepare a 1000 hour stress-time history (stress spectrum) for each usage and year required. Exceedence curves would be prepared comparing the 1000 hours of selected data with the total data for that time period. In addition, a computer program to accomplish these tasks would be written that would allow SA-ALC personnel to organically obtain similar results on data that has not yet been either received or processed. From these stress spectra, either those prepared as a result of this effort, or those organically determined, SA-ALC personnel would ascertain the safety limit and inspection intervals of the T-38 subjected to these various usages.

II. ANALYSIS

The data that is provided by the MSR is, in itself, not useful for determining the life of any aircraft component. For that data to become useful for this purpose, functional changes must occur. The changes that are required include restructuring the form of the data from that of deflection to stress, mathematical determination of the proper 1g stress level, and transfer of these newly derived stresses from the location of the recorder to the location of interest. In addition, a methodology was developed to rid the data of invalid tape cassettes. It is the purpose of this section to explain the

mechanism used to obtain these ends.

A. Deflection to Stress Determination

The raw data recorded by the MSR is a set of peaks and valleys. The first data point, in every case, was the value of the reference line that was obtained from one of the stylii. Table 1 shows an example of the data obtained in this format. It should be noted that this reference point was different for each MSR cassette. As this reference point was algebraically included to each of the data points, it was necessary to subtract this value from each of the recorded data points. Table 2 shows an example of the recorded data with each of the reference points subtracted.

As can be seen by studying Table 2, there now exists an ordered set of peaks and valleys, and these are in terms of deflections. It is now necessary to change these deflections first into strains, and then into stresses. To change these deflections into strains, we first must note that these deflections are recorded in microinches. Then, we must note that the gage length of the tape is eight inches, and finally that the reader constant is 12. Using the format from a report published by Northrop Corporation, Equation 1 allows us to transform the deflection data to strain.

$$e = ((m-d)/K)(10^3/G) \quad \text{inches/inch} \tag{1}$$

where e = engineering strain (inches/inch),
m = MSR recorded value in microinches during flight,
d = reference value of the MSR recorded data in microinches,
K = reader constant = 12, and
G = tape gage length = 8 inches.

Another report that discusses the same type of transformation is that put forth by University of Dayton Research Institute (UDRI). In their report, UDRI used the tape reader system of Aeronautical Systems Division (ASD) at Wright-Patterson AFB. In this instance, it appears that the reader constant was 10 instead of the 12 that was reported by the ASIMIS at OK-ALC. To transform the strain into stress, it is necessary to use Hooke's Law. This is shown in Equation 2.

$$S = Ee \quad \text{psi} \tag{2}$$

Data Set Header
FID RSEQN ACSN ACTYP ACNFG RECSN CASSN ACBASE
DIK 1 6413288 T38B 09051 05331 KWRD
DINSTL HINSTL DREMOV HREMOV TRDATE GAGL INTINC OPINIT
820820 07550.0 821124 07607.6 830204 0080 80 MKM
FIDB RSEQB NORPSI NORZRO
DIK 2 01104
COMMENT:
Data Record: DIK000010
1104 1118 1072 1115 1070 1110 1065 1108 1063 1109 1059 1104
1053 1096 1059 1095 1063 1106 1063 1103 1061 1107 1064 1104
1064 1104 1064 1102 1056 1100 1057 1121 1061 1158 1059
Data Record: DIK000020
1145 1072 1136 1059 1147 1103 1142 1077 1107 1076 1133 1076
1139 1070 1101 1050 1086 1052 1113 1070 1139 1050 1144 1051
1126 1060 1103 1067 1102 1066 1112 1066 1116 1056 1097
Data Record: DIK000030
1066 1142 1076 1129 1066 1104 1064 1126 1088 1130 1067 1122
1069 1120 1069 1120 1064 1121 1063 1129 1066 1105 1058 1104
1062 1123 1085 1147 1059 1101 1053 1108 1076 1124 1066
Data Record: DIK000040
1100 1059 1119 1066 1103 1060 1101 1063 1123 1057 1108 1071
1122 1050 1127 1066 1113 1055 1109 1047 1082 1051 1154 1063
1102 1060 1091 1060 1111 1060 1093 1059 1137 1059 1160
Data Record: DIK000050
1063 1129 1067 1163 1091 1124 1072 1122 1059 1104 1059 1104
1064 1112 1060 1101 1066 1101 1059 1114 1060 1135 1055 1149
1050 1161 1041 1139 1070 1132 1069 1116 1073 1118 1055
Data Record: DIK000060
1111 1059 1107 1065 1114 1052 1151 1068 1110 1050 1171 1056
1118 1087 1120 1078 1123 1072 1123 1063 1112 1061 1097 1056
1097 1058 1113 1055 1135 1047 1087 1050 1141 1059 1134
Data Record: DIK000070
1069 1104 1048 1092 1050 1101 1064 1116 1062 1157 1079 1116
1081 1125 1053 1147 1044 1122 1055 1155 1084 1127 1053 1149
1078 1116 1060 1099 1058 1098 1065 1123 1048 1164 1047
Data Record: DIK000080
1087 1029 1089 1053 1116 1054 1125 1077 1119 1059 1139 1049
1116 1055 1090 1055 1127 1055 1139 1053 1135 1053 1141 1056
1143 1061 1129 1079 1132 1081 1132 1074 1125 1047 1144
Data Record: DIK000090
1093 1142 1056 1101 1062 1100 1067 1125 1089 1151 1046 1135
1053 1149 1032 1142 1052 1099 1066 1099 1049 1112 1053 1128
1065 1044 1050 1110 1059 1135 1079 1122 1066 1150 1047
Data Record: DIK000100
1142 1053 1086 1054 1087 1050 1110 1062 1104 1060 1104 1063
1101 1066 1101 1060 1107 1063 1101 1066 1113 1049 1135 1056
1123 1046 1108 1055 1113 1054 1111 1058 1114 1081 1127

Table 1 - Raw MSR Data

Data Set Header:
FID RSEQN ACSN ACTYP ACNFG RECSN CASSN ACBASE
DIK 1 6413288 T38B 09051 05331 KWRD
DINSTL HINSTL DREMOV HREMOV TRDATE GAGL INTINC OPINIT
820820 07550.0 821124 07607.6 830204 0080 80 MKM
FIDB RSEQB NORPSI NORZRO
DIK 2 01104
COMMENT:
Data Record: DIK000010
0 14 -32 11 -34 6 39 4 -41 5 45 0
-51 -8 -45 -9 -41 2 -41 -1 43 3 40 0
40 0 -40 -2 -48 -4 47 17 43 54 45
Data Record: DIK000020
41 -32 32 -45 43 -1 38 27 3 28 29 28
35 -34 -3 -54 -18 -52 9 -34 35 -54 40 -53
22 -44 -1 -37 -2 -38 8 38 12 -48 -7
Data Record: DIK000030
-38 -38 -28 25 -38 0 -40 22 16 26 -37 18
-35 16 -35 16 -40 17 -41 25 -38 1 -46 0
-42 19 -19 43 -45 -3 -51 4 -28 20 -38
Data Record: DIK000040
-4 -45 15 -38 -1 -44 -3 -41 19 -47 4 -33
18 -54 23 -38 9 -49 5 -57 -22 -53 50 -41
-2 -44 -13 -44 7 -44 -11 -45 33 -45 56
Data Record: DIK000050
-41 25 -37 59 -13 20 -32 18 -45 0 45 0
-40 8 -44 -3 -38 -3 -45 10 -44 31 -49 45
-54 57 -63 35 -34 28 -35 12 -31 14 -49
Data Record: DIK000060
7 -45 3 -39 10 -52 47 -36 6 -54 67 -48
14 -17 16 -26 19 -32 19 -41 8 -43 -7 -48
-7 -46 9 -49 31 -57 -17 -54 37 -45 30
Data Record: DIK000070
-35 0 -56 -12 -54 -3 -40 12 -42 53 -25 12
-23 21 -51 43 -60 18 -49 51 -20 23 -51 45
-26 12 -44 -5 -46 -6 -39 19 -56 60 -57
Data Record: DIK000080
-17 -75 -15 -51 12 -50 21 -27 15 -45 35 -55
12 -49 -14 -49 23 -49 35 -51 31 -51 37 -48
39 -43 25 -25 28 -23 28 -30 21 -57 40
Data Record: DIK000090
-11 38 -48 -3 -42 -4 -37 21 -15 47 -58 31
-51 45 -72 38 -52 -5 -38 -5 -55 8 -51 24
-39 40 -54 6 -45 31 -25 18 -38 46 -57
Data Record: DIK000100
38 -51 -18 -50 -17 -54 6 -42 0 -44 0 -41
-3 -38 -3 -44 3 -41 -3 -38 9 -55 31 -48
19 -58 4 -49 9 -50 7 -46 10 -23 23

Table 2 - Raw MSR Data with Reference Value Subtracted

where S = stress (psi), and
E = Young's Modulus for aluminum = 10.3×10^6 psi.

Therefore, the final equation for this portion of the analysis is shown in Equation 3 below;

$$S = 107.292\,(m\text{-}d) \quad psi \tag{3}$$

B. Mathematical Determination of the Proper 1g Stress Level

Strain gages accurately determine the range of strains to which the material they have been attached are exposed. They do not, however, give us any information regarding the absolute value of these strains. This information is necessary as we must know not only ΔK, but also K_{max} to conduct our analysis. Changing the form of the data from strain to stress does not further our information in this respect. What is necessary is to determine the value of the 1g stress level independently of whatever values are read from the MSR.

To accomplish this task, determination of the level of the 1g stress, we must first remove the unknown location of the stress levels inherent in the data itself. This was accomplished in the following manner. We know, that on average, the stresses return to a 1g level after each maximum excursion. It is important that we look at this as an average representation as there are many instances where the stress levels do not return to a 1g condition. The stresses may, because of the particular maneuvers flown, return to a load factor higher than 1g, or in other instances, fall to a stress level that is less than the 1g level. But, on the average, by looking at a graphical ouput of the data, Figure 5, or a tabular representation of the unedited data, Table 3, we note that the stresses return to a particular lower level. It was our purpose to determine this lower level, and remove it from the data. This was accomplished by averaging the minimum values, i.e., the valleys, and subtracting them from the data. This is accomplished by rewriting Equation 3 as follows.

$$S = 107.292(m\text{-}d\text{-}mv) \quad psi \tag{4}$$

where mv is the mean valley of the data

We now will have the data in a format such that the minimum values are

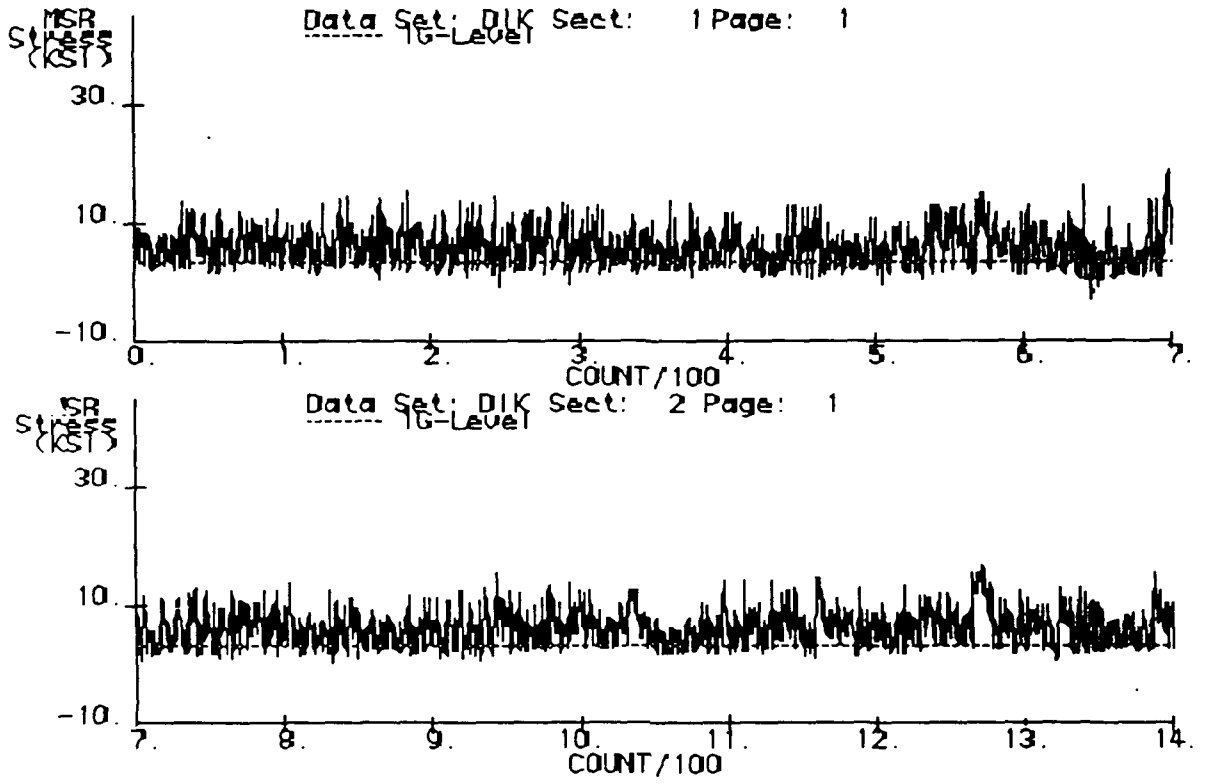


Figure 5 - Graphical Data Output

DATA SET: DIK25331
----- Data Set Header:
FID RSEQN ACSN ACTYP ACNFG RECSN CASSN ACBASE
DIK 1 6413288 T38B 09051 05331 KWRD
DINSTL HINSTL DREMOV HREMOV TRDATE GAGL INTINC OPINIT
820820 07550.0 821124 07607.6 830204 0080 80 MKM
FIDB RSEQB NORPSI NORZRO
DIK 2 01104
COMMENT:

1	3457.	5031.	-140.	4693.	-364.	4131.	-926.	3907.	-1151.
10	4019.	-1601.	3457.	-2275.	2558.	-1601.	2445.	-1151.	3682.
19	-1151.	3345.	-1376.	3794.	-1039.	3457.	-1039.	3457.	-1039.
28	3232.	-1938.	3007.	-1826.	5368.	-1376.	9527.	-1601.	8065.
37	-140.	7054.	-1601.	8290.	3345.	7728.	422.	3794.	310.
46	6717.	310.	7391.	-364.	3120.	-2612.	1434.	-2388.	4469.
55	-364.	7391.	-2612.	7953.	-2500.	5930.	-1488.	3345.	-702.
64	3232.	-814.	4356.	-814.	4806.	-1938.	2670.	-814.	7728.
73	310.	6267.	-814.	3457.	-1039.	5930.	1659.	6379.	-702.
82	5480.	-477.	5255.	-477.	5255.	-1039.	5368.	-1151.	6267.
91	-814.	3569.	-1713.	3457.	-1264.	5593.	1021.	8290.	-1601.
100	3120.	-2275.	3907.	310.	5705.	-814.	3007.	-1601.	5143.
109	-814.	3345.	-1488.	3120.	-1151.	5593.	-1826.	3907.	-252.
118	5480.	-2612.	6042.	-814.	4469.	-2050.	4019.	-2950.	984.
127	-2500.	9077.	-1151.	3232.	-1488.	1996.	-1488.	4244.	-1488.
136	2221.	-1601.	7166.	-1601.	9751.	-1151.	6267.	-702.	10089.
145	1996.	5705.	-140.	5480.	-1601.	3457.	-1601.	3457.	-1039.
154	4356.	-1488.	3120.	-814.	3120.	-1601.	4581.	-1488.	6941.
163	-2050.	8515.	-2612.	9864.	-3624.	7391.	-364.	6604.	-477.
172	4806.	-27.	5031.	-2050.	4244.	-1601.	3794.	-926.	4581.
181	-2388.	8740.	-589.	4131.	-2612.	10988.	-1938.	5031.	1546.
190	5255.	535.	5593.	-140.	5593.	-1151.	4356.	-1376.	2670.
199	-1938.	2670.	-1713.	4469.	-2050.	6941.	-2950.	1546.	-2612.
208	7616.	-1601.	6829.	-477.	3457.	-2837.	2108.	-2612.	3120.
217	-1039.	4806.	-1264.	9414.	647.	4806.	872.	5817.	-2275.
226	8290.	-3287.	5480.	-2050.	9189.	1209.	6042.	-2275.	8515.
235	535.	4806.	-1488.	2895.	-1713.	2783.	-926.	5593.	-2837.
244	10201.	-2950.	1546.	-4973.	1771.	-2275.	4806.	-2163.	5817.
253	422.	5143.	-1601.	7391.	-2725.	4806.	-2050.	1883.	-2050.
262	6042.	-2050.	7391.	-2275.	6941.	-2275.	7616.	-1938.	7841.
271	-1376.	6267.	647.	6604.	872.	6604.	85.	5817.	-2950.
280	7953.	2221.	7728.	-1938.	3120.	-1264.	3007.	-702.	5817.
289	1771.	8740.	-3062.	6941.	-2275.	8515.	-4636.	7728.	-2388.
298	2895.	-814.	2895.	-2725.	4356.	-2275.	6155.	-926.	7953.
307	-2612.	4131.	-1601.	6941.	647.	5480.	-814.	8627.	-2950.
316	7728.	-2275.	1434.	-2163.	1546.	-2612.	4131.	-1264.	3457.
325	-1488.	3457.	-1151.	3120.	-814.	3120.	-1488.	3794.	-1151.
334	3120.	-814.	4469.	-2725.	6941.	-1938.	5593.	-3062.	3907.
343	-2050.	4469.	-2163.	4244.	-1713.	4581.	872.	6042.	-1488.

Table 3 - Stress Transformation of MSR Data

DATA SET: DIK25331
----- Data Set Header:
FID RSEQN ACSN ACTYP ACNFG RECSN CASSN ACBASE
DIK 1 6413288 T38B 09051 05331 KWRD
DINSTL HINSTL DREMOV HREMOV TRDATE GAGL INTINC OPINIT
820820 07550.0 821124 07607.6 830204 0080 80 MKM
FIDB RSEQB NORPSI NORZRO
DIK 2 01104
COMMENT:

1	7680.	9253.	4083.	8916.	3858.	8354.	3296.	8129.	3071.
10	8242.	2622.	7680.	1947.	6781.	2622.	6668.	3071.	7905.
19	3071.	7567.	2847.	8017.	3184.	7680.	3184.	7680.	3184.
28	7455.	2285.	7230.	2397.	9590.	2847.	13749.	2622.	12288.
37	4083.	11276.	2622.	12513.	7567.	11951.	4645.	8017.	4533.
46	10939.	4533.	11614.	3858.	7343.	1610.	5657.	1835.	8691.
55	3858.	11614.	1610.	12176.	1723.	10152.	2734.	7567.	3521.
64	7455.	3409.	8579.	3409.	9029.	2285.	6893.	3409.	11951.
73	4533.	10490.	3409.	7680.	3184.	10152.	5881.	10602.	3521.
82	9703.	3746.	9478.	3746.	9478.	3184.	9590.	3071.	10490.
91	3409.	7792.	2509.	7680.	2959.	9815.	5544.	12513.	2622.
100	7343.	1947.	8129.	4533.	9928.	3409.	7230.	2622.	9366.
109	3409.	7567.	2734.	7343.	3071.	9815.	2397.	8129.	3971.
118	9703.	1610.	10265.	3409.	8691.	2172.	8242.	1273.	5207.
127	1723.	13300.	3071.	7455.	2734.	6219.	2734.	8467.	2734.
136	6443.	2622.	11389.	2622.	13974.	3071.	10490.	3521.	14311.
145	6219.	9928.	4083.	9703.	2622.	7680.	2622.	7680.	3184.
154	8579.	2734.	7343.	3409.	7343.	2622.	8804.	2734.	11164.
163	2172.	12738.	1610.	14086.	599.	11614.	3858.	10827.	3746.
172	9029.	4195.	9253.	2172.	8467.	2622.	8017.	3296.	8804.
181	1835.	12962.	3633.	8354.	1610.	15210.	2285.	9253.	5769.
190	9478.	4757.	9815.	4083.	9815.	3071.	8579.	2847.	6893.
199	2285.	6893.	2509.	8691.	2172.	11164.	1273.	5769.	1610.
208	11838.	2622.	11052.	3746.	7680.	1385.	6331.	1610.	7343.
217	3184.	9029.	2959.	13637.	4870.	9029.	5095.	10040.	1947.
226	12513.	936.	9703.	2172.	13412.	5432.	10265.	1947.	12738.
235	4757.	9029.	2734.	7118.	2509.	7005.	3296.	9815.	1385.
244	14424.	1273.	5769.	-750.	5994.	1947.	9029.	2060.	10040.
253	4645.	9366.	2622.	11614.	1498.	9029.	2172.	6106.	2172.
262	10265.	2172.	11614.	1947.	11164.	1947.	11838.	2285.	12063.
271	2847.	10490.	4870.	10827.	5095.	10827.	4308.	10040.	1273.
280	12176.	6443.	11951.	2285.	7343.	2959.	7230.	3521.	10040.
289	5994.	12962.	1161.	11164.	1947.	12738.	-413.	11951.	1835.
298	7118.	3409.	7118.	1498.	8579.	1947.	10377.	3296.	12176.
307	1610.	8354.	2622.	11164.	4870.	9703.	3409.	12850.	1273.
316	11951.	1947.	5657.	2060.	5769.	1610.	8354.	2959.	7680.
325	2734.	7680.	3071.	7343.	3409.	7343.	2734.	8017.	3071.
334	7343.	3409.	8691.	1498.	11164.	2285.	9815.	1161.	8129.
343	2172.	8691.	2060.	8467.	2509.	8804.	5095.	10265.	2734.

Table 4 - MSR Stresses at W.S. 0.0, 44% Spar

equal to zero (0 psi).

The next step in our analysis is placing in the appropriate 1g stress level. To accomplish this task. Northrop, in their report, uses 2756 psi as their 1g stress level. This was based upon strain at a particular location when the aircraft was a rest. We were unsure of both the location and the dash number of the wing for which this value was obtained. Therefore, we found it necessary to determine more information regarding the 1g stress level.

To accomplish this task, data was obtained from SA-ALC/MMSA personnel regarding the 1g stress level using MXU data for the area in which we were interested, i.e., W.S. 0 0 at the 44% spar on the -29 wing. It was necessary to use the equations that were derived for M_x and M_y at W.S. 0.0 and the bending moments that were obtained from W.S. 26.6. This analysis gave us a stress level for the 1g stress approximately equal to 3790 psi.

There were two values of stress at the 1g location. One was obtained using analytical techniques, and the other was obtained using experimental procedures. As the methods utilized in this study, examination of the MSR data, did not allow for an independent corroboration of either of them, it was decided to average them and use a 1g stress level approximately equal to this averaged value. For this reason, it was decided that 3300 psi would be used as the 1g stress level. To place the 1g stress level into the equation, we now add this term to Equation 4 and get;

$$S = 107.292(m\text{-}d\text{-}mv) + 3300 \quad psi. \tag{5}$$

We now have an equation that can transform the displacements from the MSR into stresses, and included in this transformation is the determination of the 1g stress level. Table 4 shows these stresses.

C. Stress Transfer Function Determination

The next aspect of the analysis required transferring the stress-time history data from the location of the MSR to the location where the analyst would like to perform his crack growth studies. According to T.O. 1T-38-570, the center line of the MSR is located 3-1/16 inches from the 39% spar fastener line in a direction toward the 44% spar and between W.S. 9.6 and W.S. 16. It was assumed that the actual location of the stylii themselves were at W.S. 12.8. By reviewing the update of the DTA on the -29 wing (11), data

was obtained from the finite element models and the exceedence curves that Northrop had prepared detailing the difference in stress levels between the position of the MSR and the critical location where the crack growth analysis would be accomplished. As the location of the MSR was not directly determined, it was necessary to interpolate between the data that was available, i.e., W.S. 28.6 and W.S. 0.0. The result of that interpolation indicated that the stress level at W.S. 0.0 was 1.0476 greater then at the location of the MSR. It was then necessary to factor this transfer function into the general equation. We would get;

$$S = \{107.292(m\text{-}d\text{-}mv) + 3300\}1.0476 \quad psi \qquad (6)$$

where all the terms are the same as those defined previously. This equation now allows the investigator to transform the data from the MSR into stresses at a particular critical location. In this instance, this location is W.S. 0.0 at the 44 % spar. To use this stress-time history as input to the crack growth program used by SA-ALC, it was necessary to reformat the data. An example of the reformatted data is shown in Table 5.

D. Invalid Data Determination

It was determined by SA-ALC/MMSA personnel that if data were placed on the MSR tape at a rate greater than 10 inches/hour, the data would be declared invalid. Unfortunately, only OK-ALC ASIMIS technicians actually view the MSR tapes and they do not identify the density of any of the rate data. They do, however, prepare log sheets which contain much of the header data. From this information, the investigator is able to determine the correct aircraft and cassette number. In addition, the ASIMIS transcriber places many comments on the log sheet noting the validity of the reference line, the tracking of the stylii, the length of the tape, etc. An example of a log sheet is shown in Table 6, which shows the log sheets for March 1985. If the ASIMIS technician declares the data to be invalid, then the computer program declared that data invalid, and it is no longer considered in subsequent operations.

In the computer program that was written, all tapes that had more than 10 hours per inch for the entire length of tape were declared invalid. These cassettes would be declared invalid by the computer program itself. It was not possible to determine if there were sections of the tape that had more than 10 hours per inch as the tape was not furnished to anyone other than

1 1862 325 408 891 385 835 329 812 307 824 262 767 194 678
262 666 30 790 307 756 284 801 318 767 318 767 318 745 228 723
239 959 284 1374 262 1228 408 1127 262 1251 756 1195 464 801 453 1093
453 1161 385 734 161 565 183 869 385 1161 161 1217 172 1015 273 756
352 745 340 857 340 902 228 689 340 1195 453 1048 340 767 318 1015
588 1060 352 970 374 947 374 947 318 959 307 1048 340 779 250 767
295 981 554 1251 262 734 194 812 453 992 340 723 262 936 340 756
273 734 307 981 239 812 397 970 161 1026 340 869 217 824 127 520
172 1329 307 745 273 621 273 846 273 644 262 1138 262 1397 307 1048
352 1431 621 992 408 970 262 767 262 767 318 857 273 734 340 734
262 880 3 1116 217 1273 161 1408 59 1161 385 1082 374 902 419 925
217 846 262 801 329 880 183 1296 363 835 161 1521 228 925 576 947
475 981 408 981 307 857 284 689 228 689 250 869 217 1116 127 576
161 1183 262 1105 374 767 138 633 161 734 318 902 295 1363 486 902
509 1004 194 1251 93 970 217 1341 543 1026 194 1273 475 902 273 711
250 700 329 981 138 1442 127 576 -75 599 194 902 205 1004 464 936
262 1161 149 902 217 610 217 1026 217 1161 194 1116 194 1183 228 1206
284 1048 486 1082 509 1082 430 1004 127 1217 644 1195 228 734 295 723
352 1004 599 1296 116 1116 194 1273 -41 1195 183 711 340 711 149 857
194 1037 329 1217 161 835 262 1116 486 970 340 1285 127 1195 194 565
205 576 161 835 295 767 273 767 307 734 340 734 273 801 307 734
340 869 149 1116 228 981 116 812 217 869 205 846 250 880 509 1026
273 633 295 711 205 678 183 610 205 869 183 1374 194 588 205 1071
228 666 228 621 262 711 205 666 250 1329 93 1195 531 1026 543 1060
228 745 228 779 262 801 183 902 262 801 385 756 250 666 273 869
250 1161 531 1138 262 1127 498 947 554 959 262 981 194 588 149 644
239 655 284 767 228 711 194 902 217 767 82 498 127 498 116 453
116 509 93 475 127 779 194 767 205 599 205 1273 374 745 127 1285
835 1262 205 790 138 1251 486 902 71 1228 543 1015 59 1228 543 1037
71 1262 161 655 116 981 217 655 228 689 217 678 262 790 295 869
138 666 194 666 239 655 205 610 239 790 307 790 183 599 205 734
194 666 239 1060 295 700 318 1082 138 599 239 666 -52 801 161 621
127 902 475 914 464 891 520 981 205 576 172 981 149 689 172 700
149 734 194 700 172 599 59 1161 509 959 262 1273 138 1285 408 1262
723 1262 588 1183 161 1127 520 1116 520 1228 217 1228 824 1240 588 1195
621 1262 363 925 194 621 262 992 104 1352 172 1352 947 1487 914 1498
779 1307 262 734 194 1004 655 1161 408 767 419 857 453 857 453 981
318 666 284 846 149 576 104 576 116 1161 554 1127 127 1285 453 902
374 981 475 959 228 992 610 1004 397 981 161 711 205 723 228 655
250 801 194 767 172 947 127 1138 531 1015 374 914 116 869 71 869
59 1610 250 723 14 621 172 576 -299 678 116 464 -86 509 26 498
14 846 486 1228 307 869 93 531 93 588 161 554 127 520 138 520
161 936 138 621 127 486 138 509 127 509 93 543 161 1352 756 1341
161 621 172 1374 71 801 93 1386 464 1779 779 1858 644 1015 475 812
194 565 71 1172 385 902 183 576 172 588 194 576 228 576 228 1116
307 1082 397 745 262 711 262 801 363 1082 397 1093 374 801 183 565
217 1183 250 1105 723 1228 161 1296 194 689 194 1093 419 1240 498 846
217 633 217 1060 228 947 127 947 554 947 194 621 183 857 127 1240
273 1093 576 1071 183 1082 700 1048 284 1026 509 992 509 1060 666 1037
531 1060 374 779 217 1240 318 891 329 981 509 1015 161 1228 363 902

Table 5 - MSR Stresses at W.S. 0.0, 44% Spar, Formatted as Input for Crack Growth Program

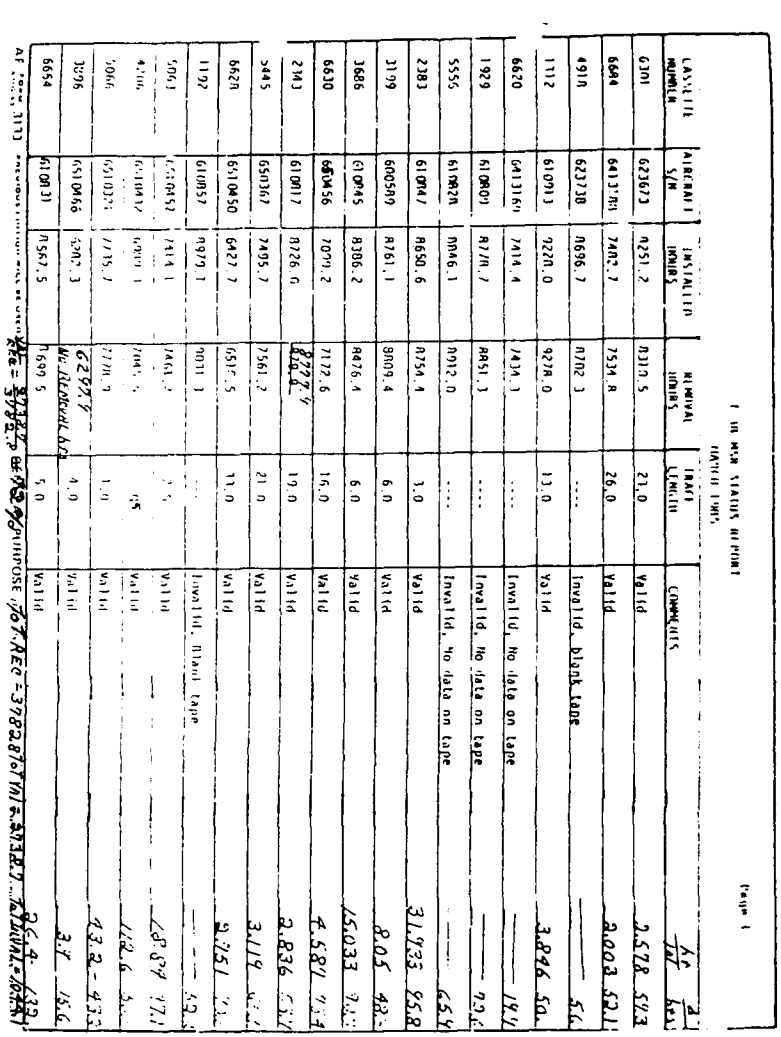


Table 6 - Log Sheet Example

the ASIMIS personnel.

By studying the log sheets in conjunction with the output of the header information of the computer program, it was noted that the majority of the invalid data occurred if there were less than 25 pts/hr on the MSR tape or greater than 250 pts/hr on the MSR tape. Because in many instances, the ASIMIS technician neglected to identify tape length, it was decided that all MSR tapes that had more than 250 pts/hr or less than 25 pts/hr would be declared invalid. These data cassettes would be declared invalid by the computer program itself.

The final mechanism of determining the validity of the data is the discretion of the investigator himself. The investigator has the ability to search through the data for anomalous behavior. An example may be a very high stress. Stresses in excess of 40000 psi have been found. This was probably due to a transcription error on the part of the ASIMIS technician as it was only relegated to a few high stresses and not the entire MSR tape. However, it was necessary to declare the entire MSR tape invalid as there is no way of identifying or removing errant stresses. In other instances, an entire tape may be invalid because of the location of the reference line.

III. DESCRIPTION OF THE COMPUTER PROGRAM

This section provides a general description of the set of computer programs developed during this project to process data from the transcribed foil tape data provided in digital form on magnetic tapes by OK-ALC. The general flow of these operations begins with data tapes, generated at a non-VAX facility, containing digital images of data transcribed and collected from a large number of MSR recording tapes. A small utility program (ASCDMP) is used for a mass copy of the data from this original tape (an IEBCOPY tape) to an intermediate unformatted file with no attempt at data conversion of any kind. A second small program (CONVERT) then transcribes this intermediate data file to another unformatted data file in form that is readily accessible by normal, VAX FORTRAN I/O operations. Normally, the copy and convert operations described above are performed when the original data tape is first installed on the VAX facility with subsequent, day-by-day data processing carried on the permanent, converted data files. Finally, the MSRPROC program, which is the principal data processing program, is employed for a variety of operations specifically related to the analysis and use of this converted MSR data. The general flow of these

operations is illustrated in Figure 6.

A. MSRPROC

All additional operations on the MSR data are conducted by means of the MSRPROC program, which is a multi-purpose data processing program designed for these operations. A functional flow diagram of the program is given in Figure 7, which shows all of the principal sub-programs and the overall flow of the operations. The program is menu and command driven from the terminal, or alternately, for batch-type operation through corresponding command files and the VAX Batch processor. The program operates in three major phases as follows:

(1) A setup phase in which the main menu is displayed and a variety of data operations and related mode settings and parameters are enabled for application to individual data sets selected during subsequent passes through the data;

(2) A data set select phase which consists of one or more passes through the MSR data file. Data sets, each of which corresponds to a collection of data from a single MSR cassette, may be selected for processing in the following ways:

(a) By numerical order of occurrence within the file;

(b) by the three character alphabetical identification code established by the collecting agency;

(c) by aircraft serial (tail) number;

(d) by the four (4) character alphabetical identification code associated with the aircraft base; and

(e) by the four (4) digit cassette number of the original foil data tape.

The select mode is arranged so that data sets may be selected one at a time for examination at the terminal or all together as in batch process operation by taking a sufficiently large sequence range. Data sets selected for processing in this manner are passed sequentially to several modules, each

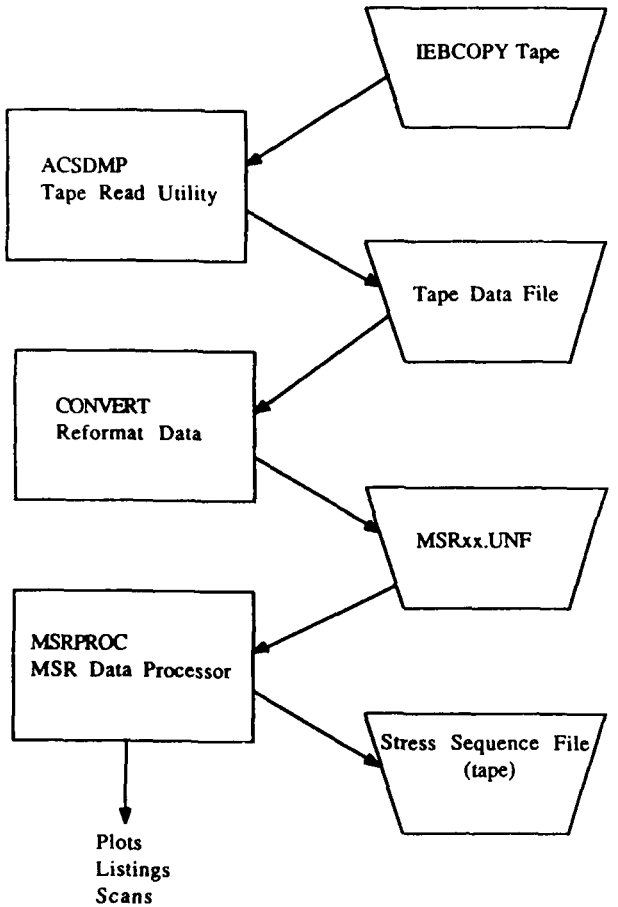


Figure 6 - Overall Data Flow

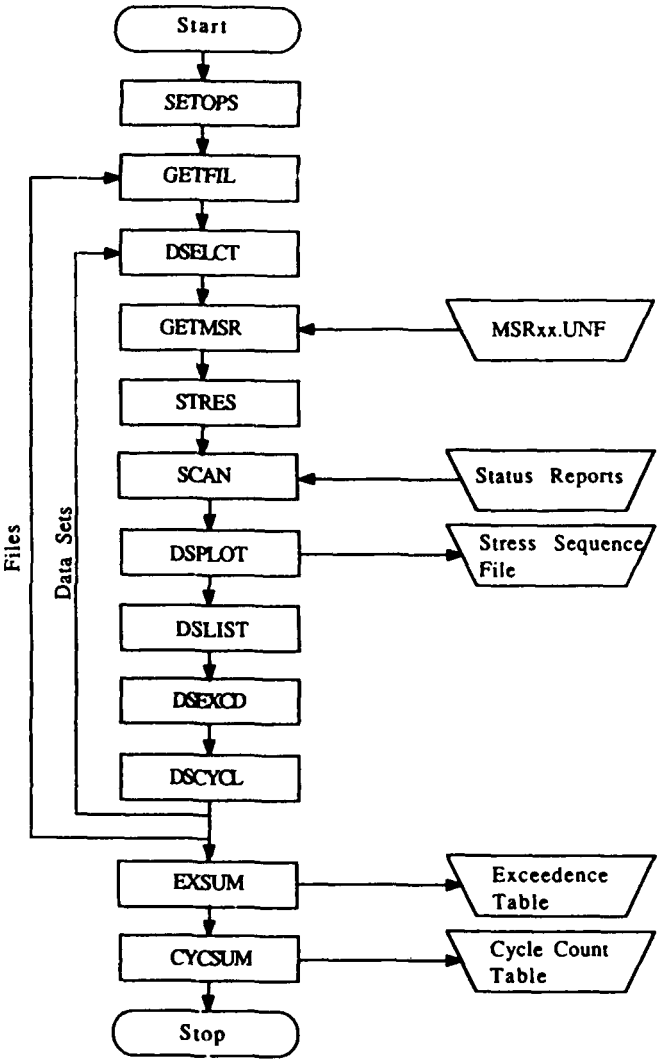


Figure 7 - MSRPROC Program Flow

corresponding to a particular data operation if enabled during the initial setup phase. Options are available during this phase to change select modes, make multiple passes through a particular data file or to change files between successive data passes;

(3) Finally, a post select phase in which various types of summary data (summary exceedence tables, cycle counts, etc.) are reported and tabled.

Additional information relating to each of the principal operating modules follows:

SETOPS

The SETOPS module displays a menu offering a set of operating options and additional flags and parameters which enable and control subsequent operations on selected data sets. The main SETOPS menu has the form:

MSRPROC Vers. 1.2 Aug. 1987
Display and Set MSR In-Line Operations

Oper	On/Off	Description	O P T I O N S				
STRS	On	Stress Scaling	0	2	0	0	0
SCAN	On	Data Set Scan	0	0	0	1	0
DPLT	Off	Plot Data Set	1	0	0	0	0
IOP	On	Set I/O Mode	0	1	0	0	0
ECHO	Off	Data Set Echo	1	0	0	0	0
LIST	Off	Data Set List	1	1	1	0	0
EXCD	Off	Exceedence Tables	1	1	0	0	0
CYCL	Off	Cycle Count - Rainflow	1	1	0	0	0

The present version of the program offers 8 operating options (STRS, SCAN, DPLT, IOP, ECHO, LIST, EXCD, and CYCL), the current status (on/off) of each, and the current value of up to five auxiliary option flags (the first of which is the on/off switch (0/1)). The user is prompted to enter an operation name after which the status of that operation is displayed in detail

and additional prompts are offered to turn the operation on or off and to provide additional flag settings and parameters. Briefly, the principal use of each operation code is as follows:

STRS: Control the scaling of the MSR to stress units.

SCAN: Govern the scanning of each data set for errors, comparison with status report data (see below), and the generation of certain statistical data.

DPLT: Enable and control the graphical display of individual data sets.

IOP: Direct the flow of main program output data and reports to the terminal screen and program output file.

ECHO: Enable and control the generation of a data file containing an ASCII list of the data as it is read from the file in a form reasonably close to the actual encoded data (essentially a data dump).

LIST: Enable and control the generation of a list of the data after scaling to stresses both as a data set list file and as a contribution to an ensemble stress sequence file.

EXCD: Enable the generation of an exceedence table for selected data sets and a contribution to the ensemble exceedence table.

CYCL: Enable and control the generation of a rainflow cycle count table for selected data sets and a contribution to an ensemble cycle count table.

GETFIL

This routine prompts for an MSR data file and corresponding status report file, inquires for the existence of these files from the system and opens the files for access by the data set search routines.

DSELCT

This routine prompts, first, for a data set select mode and then controls the data set selection procedure for subsequent selections. The main select

mode prompt is as follows:

```
Data Set Select Modes are:
Select by a Range of File Sequence Numbers:    SEQ;
Select by Data Set Id:                         FID;
Select by Cassette Number:                     CAS;
Select by Aircraft Serial Number:              ACS;
Select by Base ID:                             BAS;
Stop Selecting and Terminate Run:              <> or STO;
ENTER Select Mode Id: SEQ,FID,CAS,ACS,BAS,<STO>:
```

GETMSR

This routine reads sequentially through the data file until it encounters the next data set which satisfies the data set selection established by the DSELCT routine. In additon, the routine provides the data set ECHO file if requested and checks for internal consistency from record to record within the data. Data sets which lack the required internal consistency are declared DISABLED and, except for the preparation of an ECHO file are not passed to subsequent operations modules or counted as having been "selected".

STRES

This module applies stress scaling and transfer factors to the basic MSR as described elsewhere in this report.

SCAN

This module scans both the header and numerical data within a selected data set and establishes validity of the data. Particular functions of this module are:

* Calculate maximum, minimum, mean and standard deviation values for the complete data and for peak and valley subsets.

* Declare the data set INVALID based on the data rate (points per hour), net flight (i.e., greater than zero value) or indecipherable cassette number.

* Scan the status report file for an entry with matching cassette

number, examine the entry for number of data points per inch of foil tape, and the transcription operators Valid mark and declare the data set INVALID for an excessive value of the former or an "I" mark by the operation.

* Provide a report of the data set header, data statistics and status report contents.

DSPLOT

This module prepares and displays plots of stress versus data count for selected data sets.

DSLIST

This program provides a list of the scaled stress data either as a separate file in readable form for each selected data set or as a contribution to an ensemble stress sequence output file.

DSEXCD

This program counts the number of stress exceedences for pre-set stress levels for peak and valley data points separately and contributes the exceedence counts to the ensemble exceedence tables.

DSCYCL

This program carries out a rainflow cycle counting algorithm for each selected data set and contributes these cycle counts to the ensemble cycle count table.

EXCSUM

This program generates a normalized (i.e., per 1000 hours) exceedence table from the ensemble table after completion of the data scan, reports both tables to the program output file and to a separate exceedence table file.

CYCSUM

This program generates a normalized (i.e., per 1000 hours) rainflow

cycle count table from the ensemble cycle count table and reports both to the program output file and to a separate cycle count table file.

B. Status Files

In addition to the main MSR data file obtained by converting the original IEBCOPY data tape, the program employs a status report file obtained from the log sheets filled in by the operator at the time the data is transcribed from the original foil MSR tapes. The contents of these log sheets are entered into simple text files using any available, simple system editor progam. The file contains one line per cassette and up to 80 characters of data, entered as six fields of data separated by five commas, such as:

```
2657,60-0573,7080.8,7178.0,4.5,Valid data
|    |       |      |      |   |
1    2       3      4      5   6
```

The vertical bar and numeral marks the beginning of each field which contain the following:

Field	Content
1	Cassette number
2	Aircraft Serial (Tail) Number
3	Hours at installation
4	Hours at removal
5	Foil tape length (inches)
6	Operators comment field, of which the first letter must be "V" for valid data

This free form file of status log reports is read by an auxiliary computer program, TAPECHK.FOR, developed as part of this investigation. This program has the following functions:

* Scan the report data line by line for unacceptable entries. Such entries are those lines which have either an indecipherable cassette number entry (i.e., cannot be interpreted an integer of up to 4 digits) or have less than 6 fields (i.e., less than 5 commas). These entries are declared REJECTED and ignored.

* All other lines are accepted for inclusion in the formatted status

report file, with anomalous field entries reported as warnings. Note that data entries are accepted here on the basis of their legibility only and are not necessarily equivalent to valid data cassettes.

* A strictly formatted file (TAPESTATUS.DAT) is then produced for subsequent reference by the MSRPROC program.

These status report files, in addition to being a useful compilation of the information on the operators log sheets, have also been found to be useful as means of defining subsets of the MSR data and of eliminating as invalid such anomalous data sets as those containing duplicate copies and spurious but otherwise valid stress data. In particular, the status report files STATLIF83.DAT, STATLIF84.DAT, and STATLIF85.DAT contain invalid entries for all cassettes which correspond to aircraft serial numbers identified as non-LIF usage by SA-ALC personnel. These cassettes were first identified by searching the MSR file for all occurrences of these tail numbers and appropriate entries, containing cassette number in field 1 and "I" in field 6, were then entered in the status report files. These status report files thus serve as pre-selected set of data definitions, in this case, LIF usage.

IV. STRESS SPECTRUM DEVELOPMENT

As stated earlier, one of the purposes of this program was to develop a 1000 hour stress-time history that was representative of the total time the selected aircraft were spending in a particular mission. The amount of data available was instrumental in determining the selection of the data. In certain instances, there may have been as little as 175.2 hours of valid data (ACM, 1984), whereas in other instances, (LIF, 1983-1985), there were 32961.4 hours of data from which to choose.

To accomplish the task of selecting the appropriate data sets to ensure that the 1000 hours of flight data chosen is representative of the total population requires that the investigator examine the exceedence curve of the data that represents the total population. The object of the exercise is to choose a collection of flights whose exceedence curve compares favorably with the total population.

There are three criteria which the investigator must examine. These are:

(1) The total number of hours must approximately equal to 1000 hours. Each of the MSR tapes vary in length. Many of them are approximately 50 hours, however, they may represent as little as ten (10) hours of flight time or as much as 130 hours of flight time;

(2) The total number of cycles per 1000 hours of flight time for each of the MSR tapes be approximately equal to the total number of cyles per 1000 hours for the entire population. This may be accomplished in three ways:

(a) The investigator may select only those individual tapes which have the points per hour, i.e., number of peaks (this is equivalent to the sum of both the maximum and minimum peaks) approximately equal the the points per hour of the entire population. This information, in points per hour, may be obtained from the header information;

(b) the investigator may select MSR tapes with both high and low points per hour, but ensuring that the number of hours for each gives a weighted average of the same number of points per hour as the entire population;

(c) finally, a combination of (a) and (b) above.

This drives the upper end of the exceedence curve (low stress end) for the 1000 selected hours to become equivalent to the total population exceedence curve; and

(3) The lower end of the exceedence diagram (the high stress end) is equal to the total population both in stress level and number of occurrences. This was accomplished by studying the header information for the magnitude of the maximum stress.

To accomplish this successfully when there is a great deal of data to choose from becomes an iterative process. There is no close formed or pre-selected option on the computer program to accomplish this feat. It remains a trial and error method. The computer program has as an option and the ability to select certain cassettes and then print out the total number of hours and the exceedances per 1000 hours for those selected cassettes; thus there is no difficulty in establishing representative spectra.

V. DATA COMPARISON

In this effort, flight load recorded data was obtained for the four usages discussed earlier. To determine the severity of the various usages, hence the inspection intervals and safety limits of the aircraft, a stress-time history for each of the usages for each of the years was desired. Historically, the bases for developing these life time histories is 1000 hours of usage. Therefore, it was decided that stress spectra of approximately 1000 hours would be developed for each of these usages. Because of the wealth of data, it was also decided that 1000 hour spectra be developed for each of the individual years as well as a composite spectra for all of the years combined. Table 7 shows each of the spectra developed, and the number of hours of total/valid/invalid flight data for each of the years and the composite year.

A comparison can be made between the individual years for each of the usagesis that the year to year variation for each of the usages is not statistically significant. It is understood that to determine this effect quantitatively, a crack growth analysis must be conducted. This was not accomplished. This observation was based upon a study of the exceedence curves only.

A second set of comparisons may be made by comparing each of the individual usages. A comparison of this sort indicates that the DACT usage was the most severe, the ACM the least severe, and the DET6 and LIF similar to each other and somewhere between the DACT and ACM usages in severity.

A third comparison which can be made is the stress exceedence [illegible]s for the LIF using both MSR data and using MXU data for the same location. This comparison is shown in Figure 8, with the corresponding crack lengths vs. time shown in Figure 9. It may be seen that the data obtained from the MXU is substantially more severe. This would result in a significantly lower life. The reason for this difference is not totally understood. There are, however, a number of reasons that exist that may account for the non-congruence of these curves. These include reasons that attack the credibility of each of these methods. These include:

(1) The inability of the MSR to respond quickly enough to the abrupt changes in flight, hence stresses, that may occur;

(2) The MSR may not be sensitive enough to changes in strain. For

Table 7 - Spectra Developed

Usage	Year(s)	Total Hours	Valid Hours	Invalid Hours
LIF	1983	12610.0	7833.1	4776.9
LIF	1984	14984.2	11211.5	3772.7
LIF	1985	17865.6	14587.1	3278.5
LIF	1983-1985	45459.9	32961.4	12498.5
LIF	1983*	1000.2	1000.2	N/A
LIF	1984*	1000.0	1000.0	N/A
LIF	1985*	1001.0	1001.0	N/A
LIF	1983-1985*	1000.0	1000.0	N/A
ACM	1983	357.8	220.0	137.8
ACM	1984	325.9	175.2	150.7
ACM	1985	656.2	355.9	300.3
ACM	1983-1985	1339.9	751.1	588.8
ACM	1983*	995.0	995.0	N/A
ACM	1984*	1004.5	1004.5	N/A
ACM	1985*	1008.0	1008.0	N/A
ACM	1983-1985*	1001.1	1001.1	N/A
DET6	1983	674.6	552.3	122.3
DET6	1984	904.2	802.2	102.0
DET6	1985	603.3	503.3	100.0
DET6	1983-1985	2182.1	1857.8	324.3
DET6	1983*	997.0	997.0	N/A
DET6	1984*	1000.1	1000.1	N/A
DET6	1985*	1006.6	1006.6	N/A
DET6	1983-1985*	999.7	999.7	N/A
DACT	1983-1985	376.8	327.4	49.4
DACT	1983-1985*	993.0	993.0	N/A

* = 1000 hours of selected data

N/A = not applicable; no invalid data selected

Figure 8 - Comparison of Occurences for LIF Usage

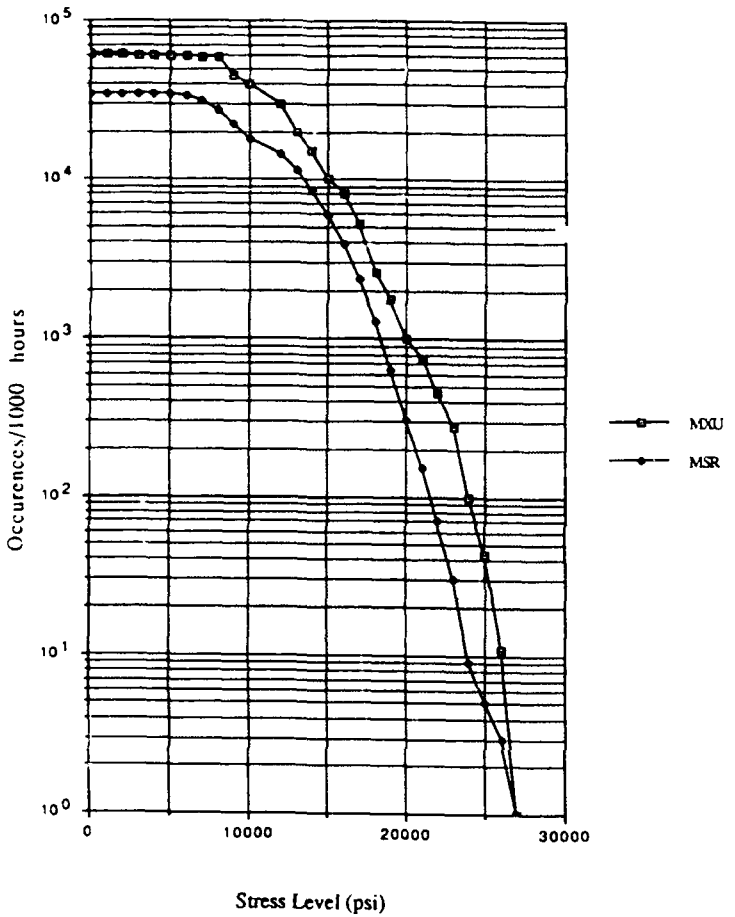
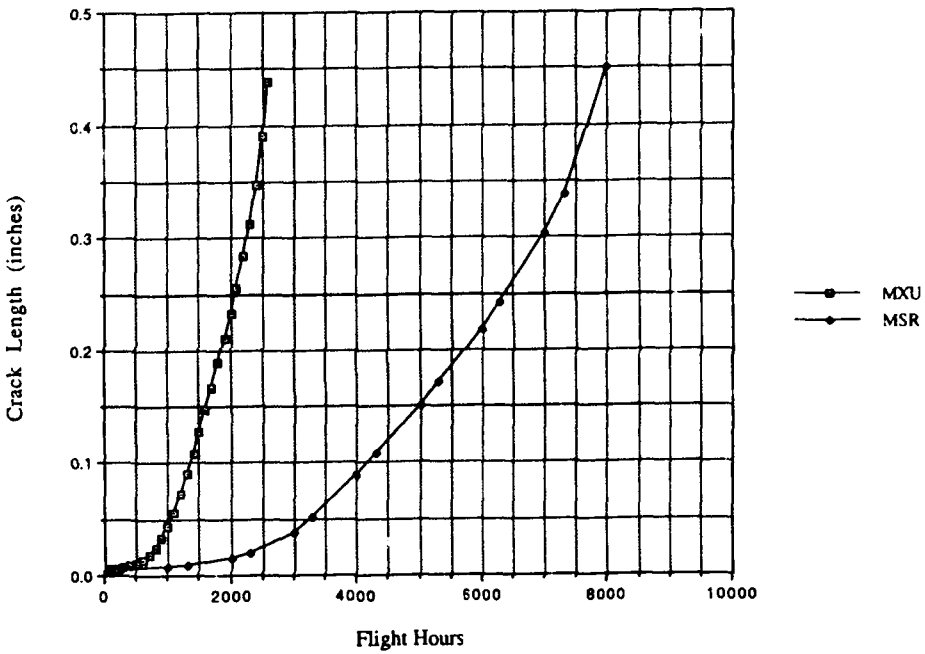


Figure 9 - Comparison of Crack Growth for LIF Usage



different regions in the flight envelope, the change in stress that triggers the MSR to count a cycle may be greater than the change in stress level that can cause damage;

(3) The MXU may count all stresses as maximum stresses. After each count of the MXU, the investigator may elect to return to a 1g stress level instead of having the minimum peak 2g's or more; and

(4) The MXU may be programmed to count too many peaks. The definition of a peak may require a fall of 0.25g, 0.50 g, or 1/2 the g level at that particular instant. Each of these definitions of a peak gives a substantially different number of cycles. Hence, there will be a vastly different life.

VI. SUMMARY

A. A computer program has been written that will allow SA-ALC/MMSA personnel to organically process MSR data obtained from ASIMIS at OK-ALC.

B. The computer program allows a systematic investigation of the MSR data in many formats including raw, edited, transformed to stresses, etc.

C. The computer program allows the investigator to view the data either in a tabular form or a graphical form.

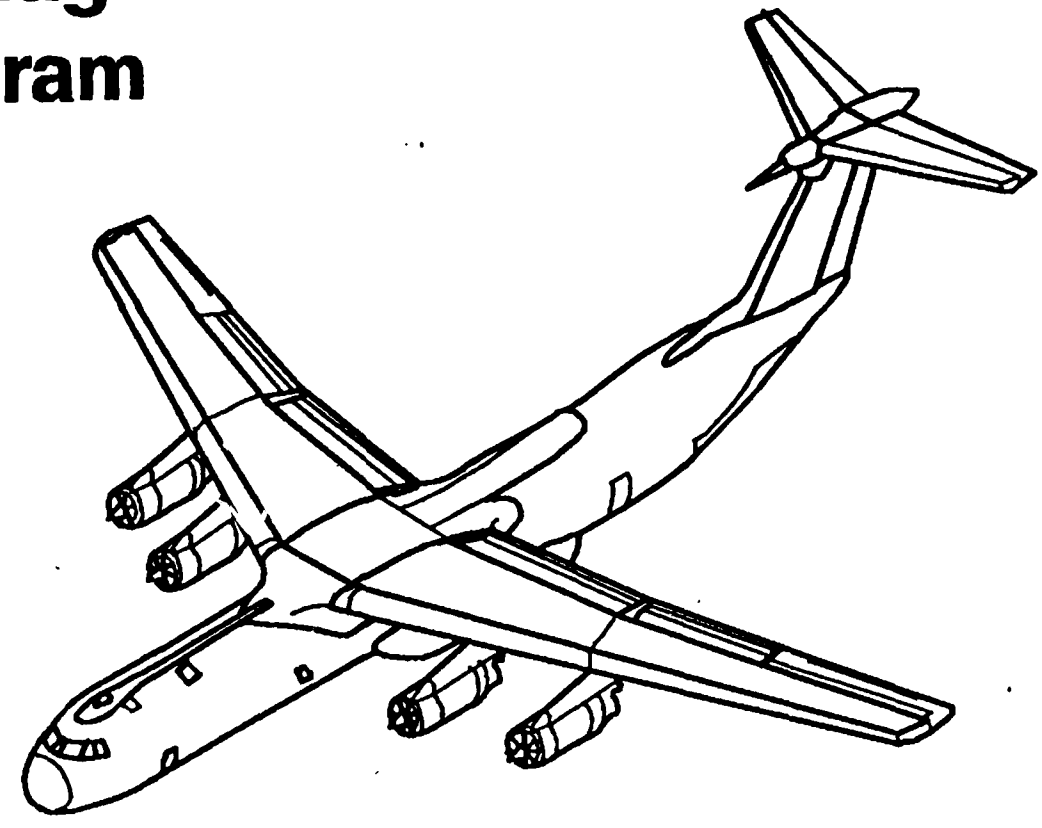
D. The computer program allows the investigator to change the valid/non-valid criteria, change the stress transfer function, prepare new spectrum tapes, etc.

E. Mathematical equations were derived to transfer the displacements obtained from the MSR to stresses at W.S. 0 0 and the 44% spar.

F. 1000 hour spectra were provided for LIF, DET6, and ACM usages for 1983,1984, 1985, and 1983-1985, and 1983-1985 for the DACT usages.

G. Exceedence curves were provided for LIF, DET6, and ACM usages for a 1000 hour spectrum and for all the valid MSR data for 1983, 1984, 1985, and 1983-1985, and for a 1000 hour spectrum and all the valid data for the DACT usage.

C-141
Force Management Program



GA-8092-01

C-141 Force Management Program

Purpose

Provide the C-141 System Program Manager (SPM) With a Single Coordinated, Dynamic, Proactive Program That Satisfies USAF Requirements in Terms of MTM/D, MMH/FH, MC Rate, Manpower Resource and Budget Constraints

GA-8092-02

C-141 Force Management Program

Objectives

- Provide a Fielded System That Cost-Effectively Meets Air Force Requirements
- Provide an Optimized Structural and Functional Inspection Program by Aircraft Tail Number and Work Unit Code
- Provide Current Data Files Consistent With Weapon System Master Plan (WSMP) Requirements
- Provide Structured, Prioritized Task (Organic and Contracted) Visibility in Support of All Force Management Activities

GA-8092-03

Why "Single Program" Concept

- Unifies and Coordinates Management Activities
- Provides Efficient SPM Control
- Easier To Plan and Execute
- Provides Common Goal for All Activities
- Avoids "Piece-Mealing" Numerous Tasks
- Allows for Single Line Item in USAF Budget
- Improves Weapon System Effectiveness in Terms of MTM/D, MMH/FH and MC Rate

C-141 Structural Programs

"Point Paper"

Task ID:

Title:

Objective:

Approach:

Organic/Contract:

Estimated Cost:

Schedule:

Payoff:

Impact (If Not Done):

TASK ID: 3013

TITLE: C-141 COMPOSITE REPAIRS STUDY

OBJECTIVE: PRELIMINARY EVALUATION OF POTENTIAL COMPOSITE REPAIR APPLICATIONS FOR METAL STRUCTURE.

APPROACH: IDENTIFY LOCATIONS/TYPES OF REPAIRS. PERFORM INITIAL DESIGNS, ANALYSES, PERFORM TRIAL FAB/INSTL OF FIVE REPAIRS. REPORT CONCLUSIONS, RECOMMENDATIONS.

MEANS: ORGANIC _____ CONTRACT F09603-86-G-0455-0011

ESTIMATED COST: 880K

SCHEDULE: ESD 4-86 ECD 9-88

PAYOFF: SIGNIFICANT SAVINGS IN EFFECTIVENESS, COST, DOWNTIME, INSPECTION REQMTS FOR REPAIRS. TAPM, RAMTIP, GENERIC BENEFITS.

RELATED TASKS: 3014

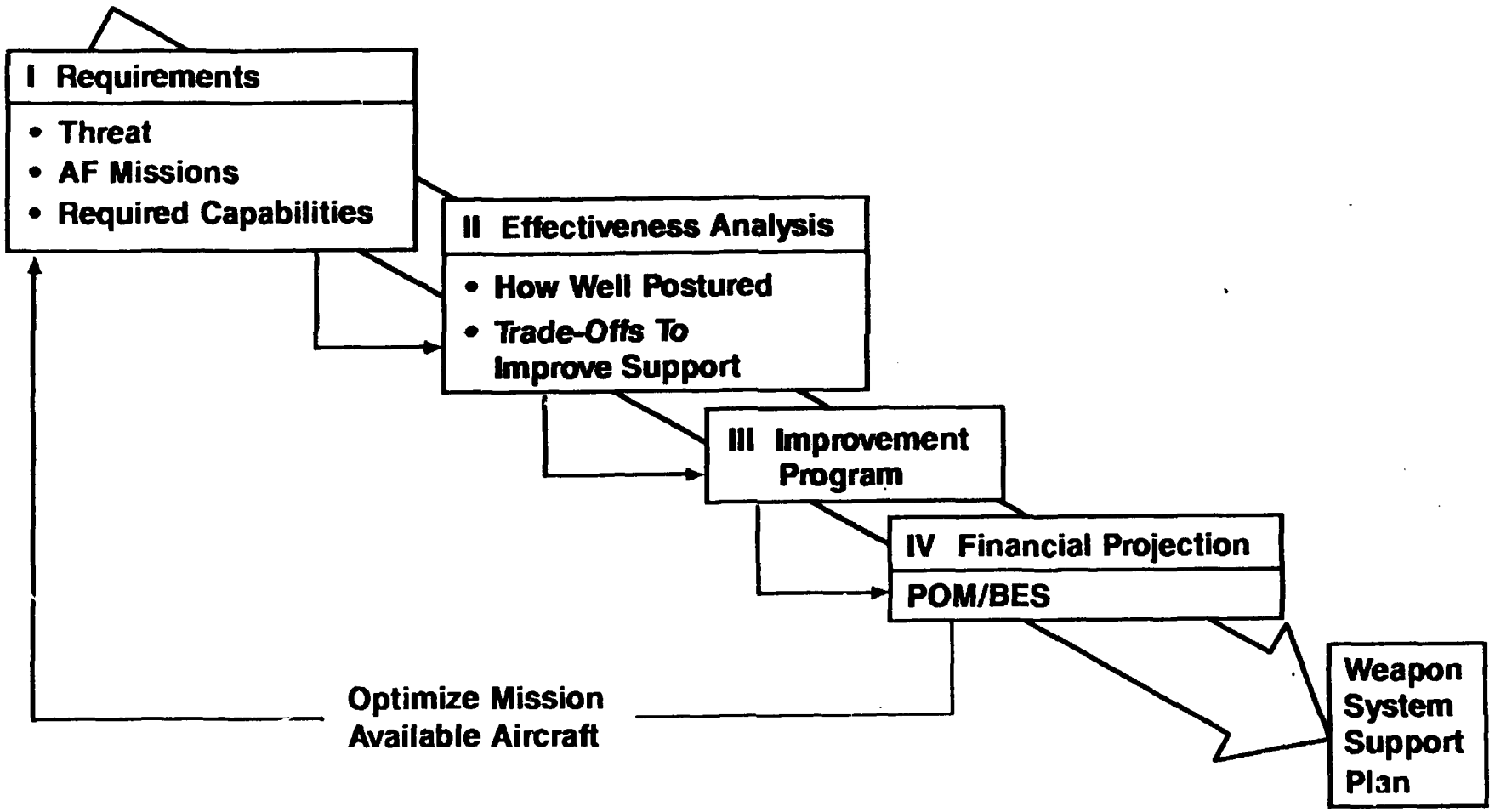
ARIES OUTPUT MODULE: TA ___ WS X FM X PM X IM ___

IMPACT: SIGNIFICANT SAVINGS, ALREADY DEMONSTRATED ON A/C 6145 COMPOSITE REPAIR AS SPINOFF OF STUDY.

PREPARED BY: MORCOCK

C-141 Force Management Program

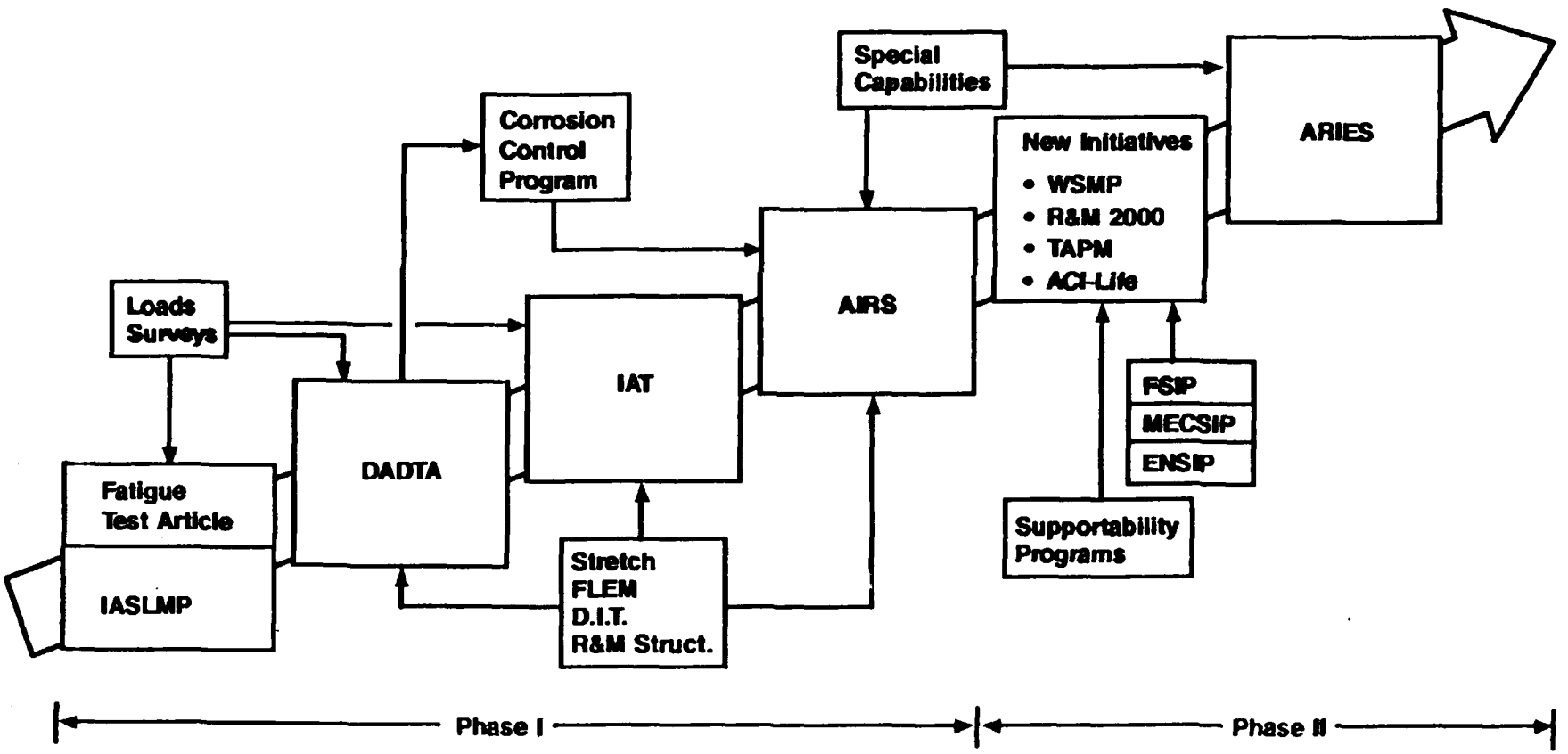
Weapon System Master Planning System



GA-8092-05

C-141 Force Management Program

Evolution of Force Management Technology



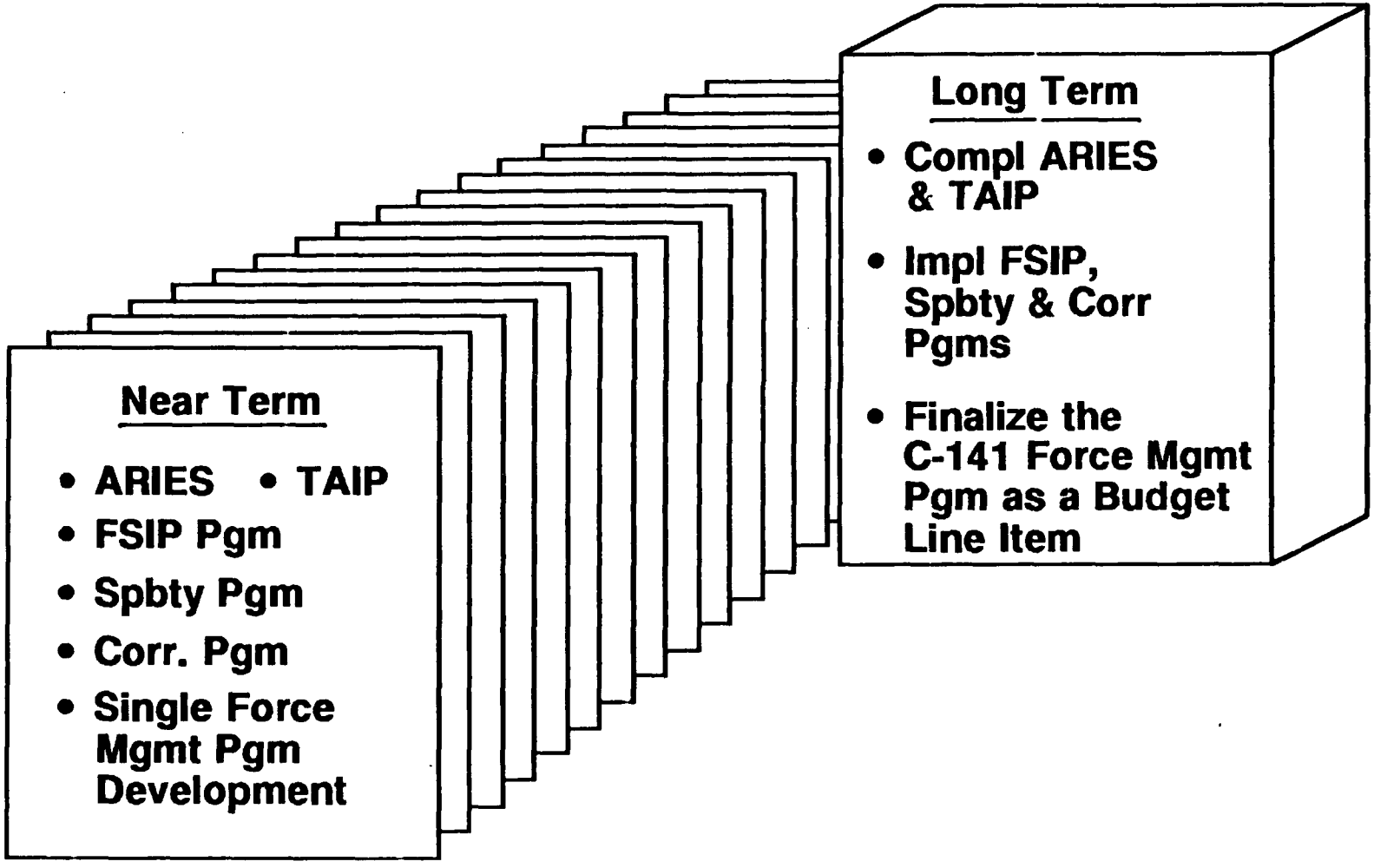
GA-8092-06

Where We Are Today

- AFLC Taskings
  - R&M 2000, TAPM, ACI Life Cycle, WSMP
- Established Databases
  - IAT, AIRS, Maintenance Records, Usage, Etc.
- Awareness of Need for FSIP
  - MECSIP
  - ENSIP
  - AVSIP

C-141 Force Mgmt Program

Where We Are Going



GA-8098-35

C-141 Force Management Program

Meeting the Challenge

- System Program Manager
  - Organizational Alignments To Function in ALL Aspects of Force Management Activities
  - Initiate Innovational Planning Towards All-Encompassing Force Management Concept
  - Design Game Plan To Improve Funding Process in Terms of Single Program Concept
  - Recognize Contractual Support as a Resource That Complements Rather Than as a Cost Burden That Hinders

GA-8092-11

Structural Programs

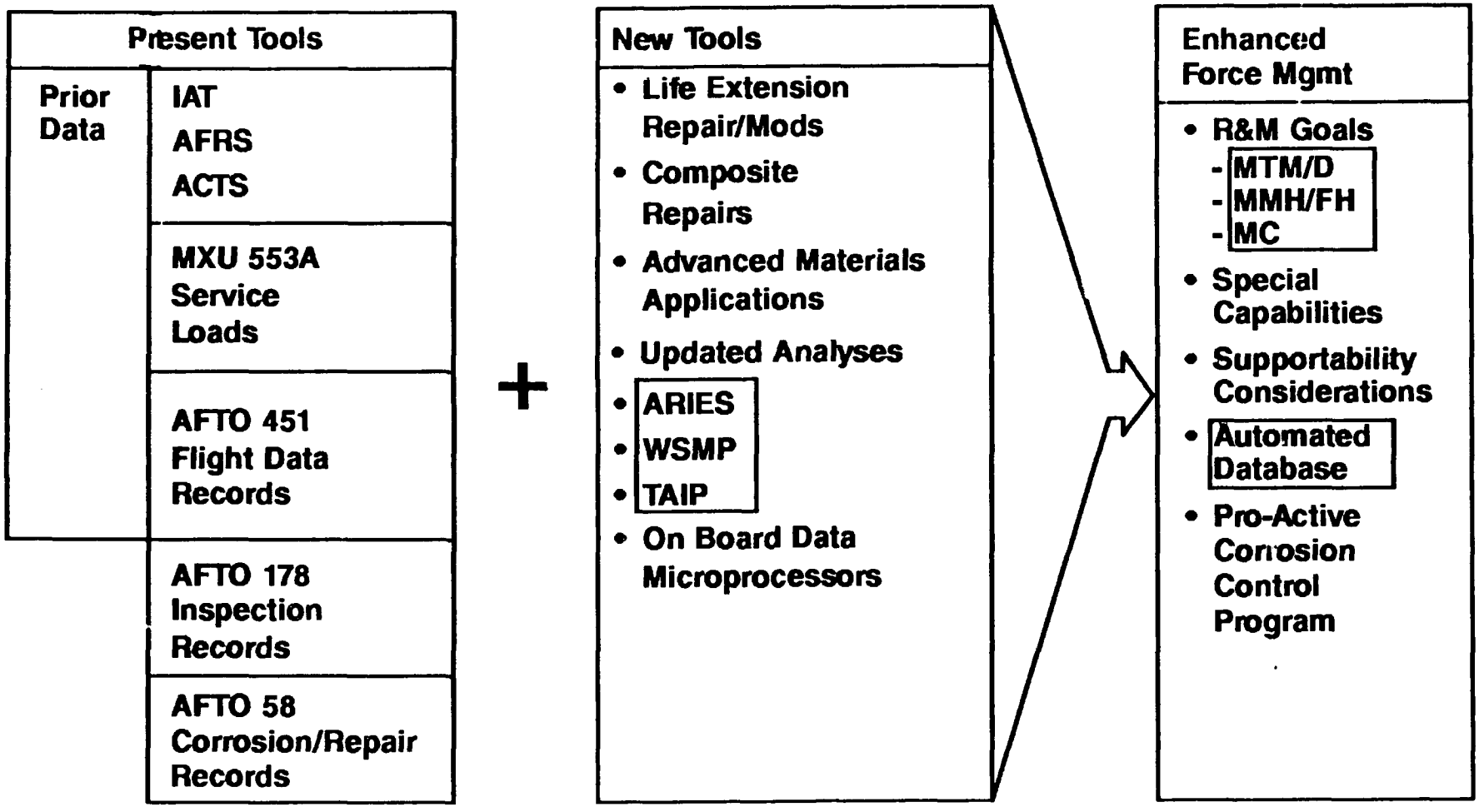
C-141 Structural Programs

Objective

Provide SPM With Proactive Program To Manage Structural Integrity of C-141 Airframe Through a Series of Preventive Actions That Will Provide a Basis for Life Extension Commensurate With Primary Goals of R&M 2000

C-141 Structural Programs

Where We're Going



GA-8092-22

C-141 Structural Programs

How Are We Getting There?

A - Aircraft Information Retrieval System (AIRS)

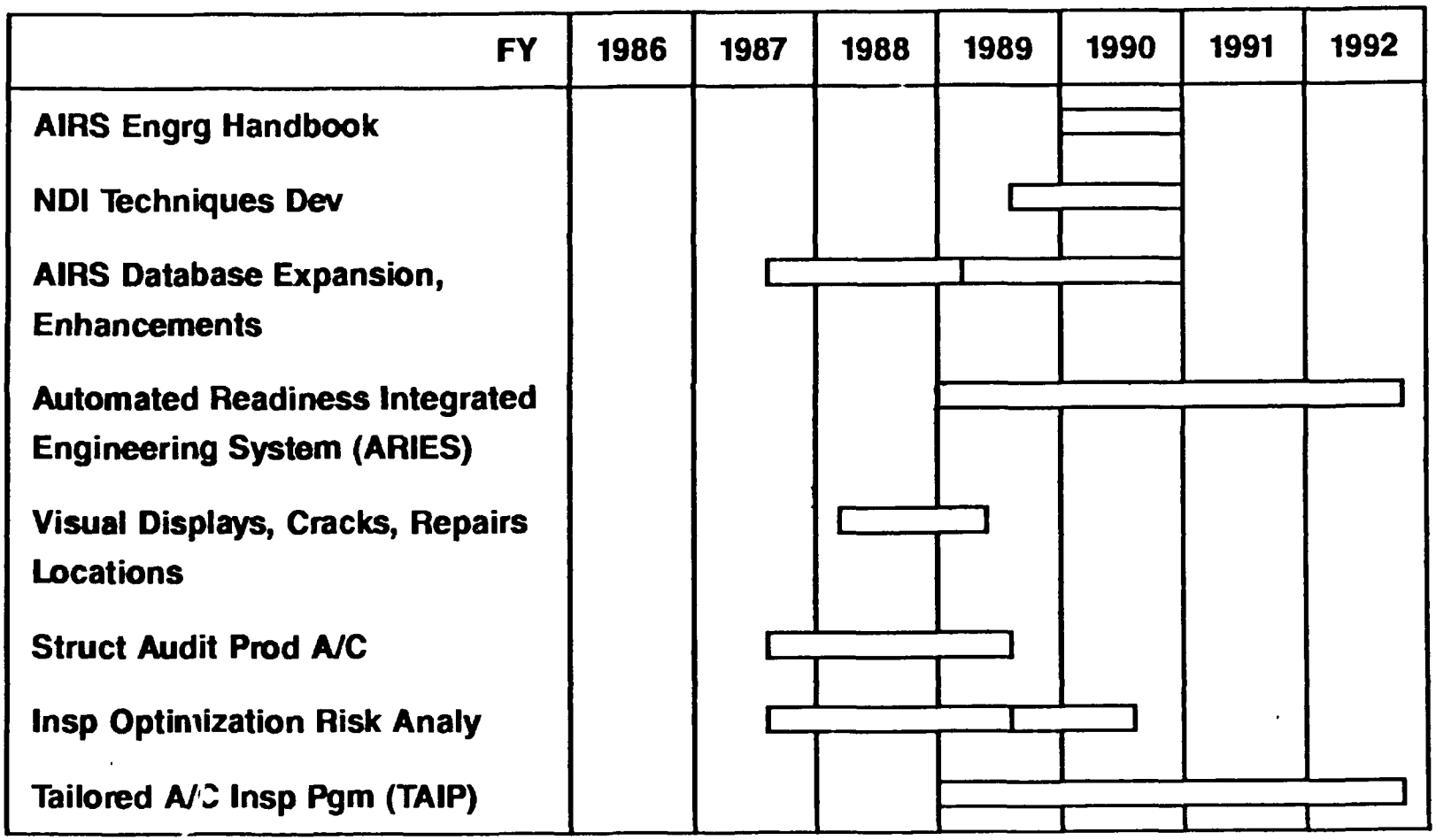
T - Individual Aircraft Tracking (IAT)

L - Loads

A - Analyses

S - Special Programs

C-141 Structural Programs



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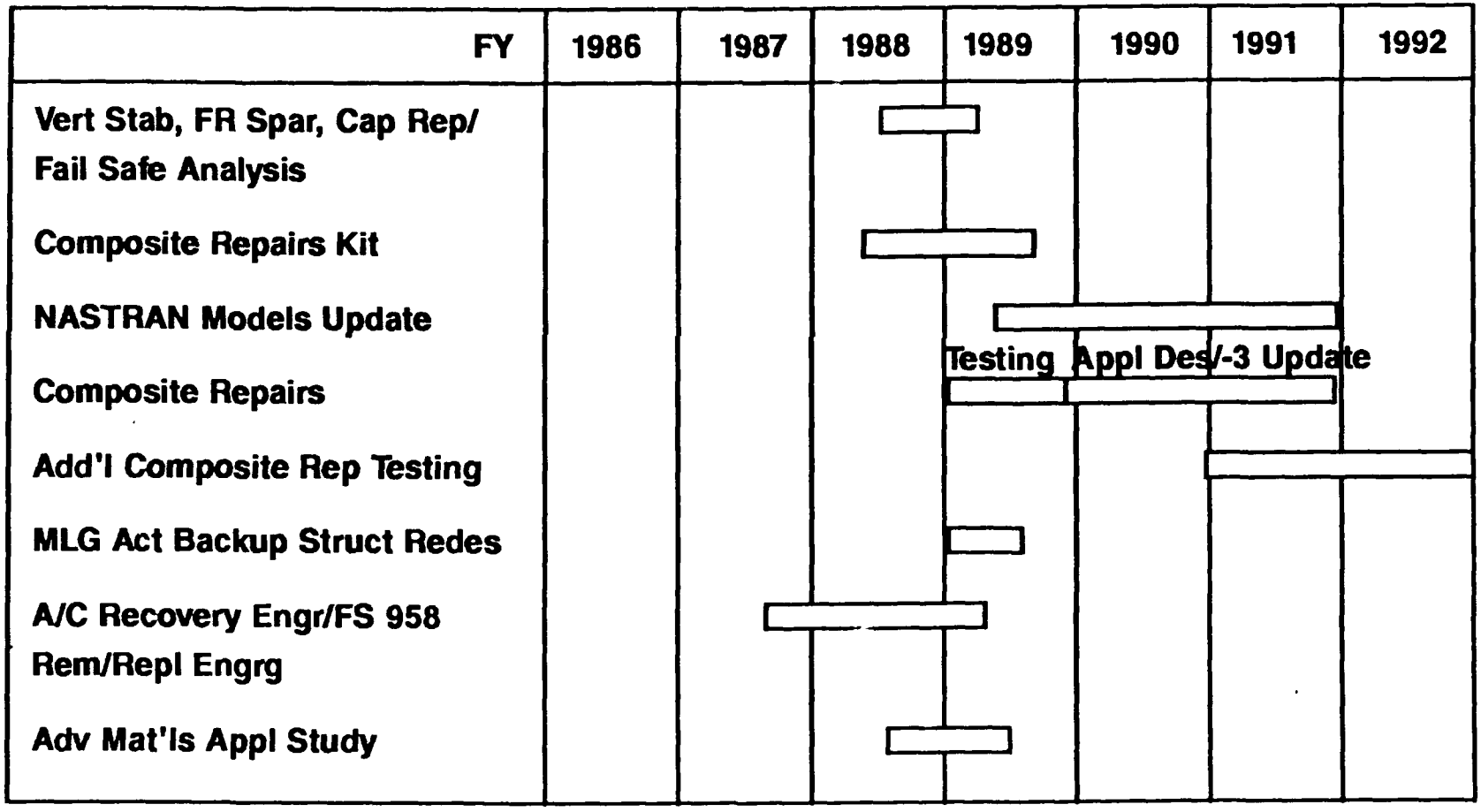
C-141 Structural Programs (Cont'd)



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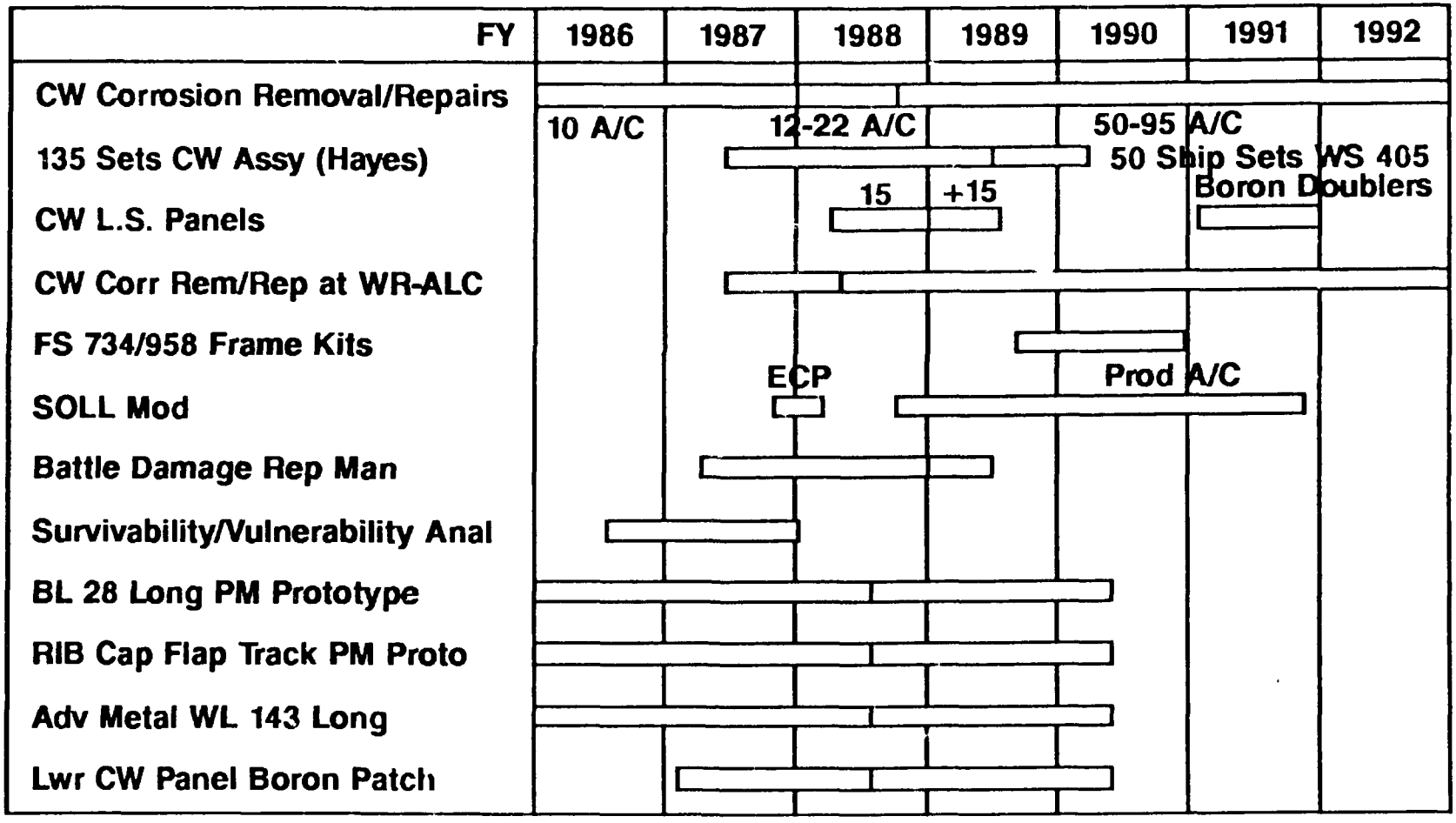
C-141 Structural Programs (Cont'd)



GA-8098-30

C-141 Structural Programs

Summary

- Structural Force Management Is the Foundation of the Integrity Process
- R&M 2000 Goals Are Real in Structural Integrity
- C-141 Structural Force Management Provides an Excellent Blueprint for Managing Aging Aircraft

GA-8092-25

C-141
Functional Systems Force Management Program

GA-8094-21

C-141 Functional Systems Program

Objective

- Provide Structured, Prioritized Tasks (Organic and Contracted) Visibility in Support of Air Force Management Activities

- Provide a Functional Systems Integrity Program (FSIP) That Is Consistent With Weapon System Master Plan (WSMP) Requirements

C-141 Functional Systems Program

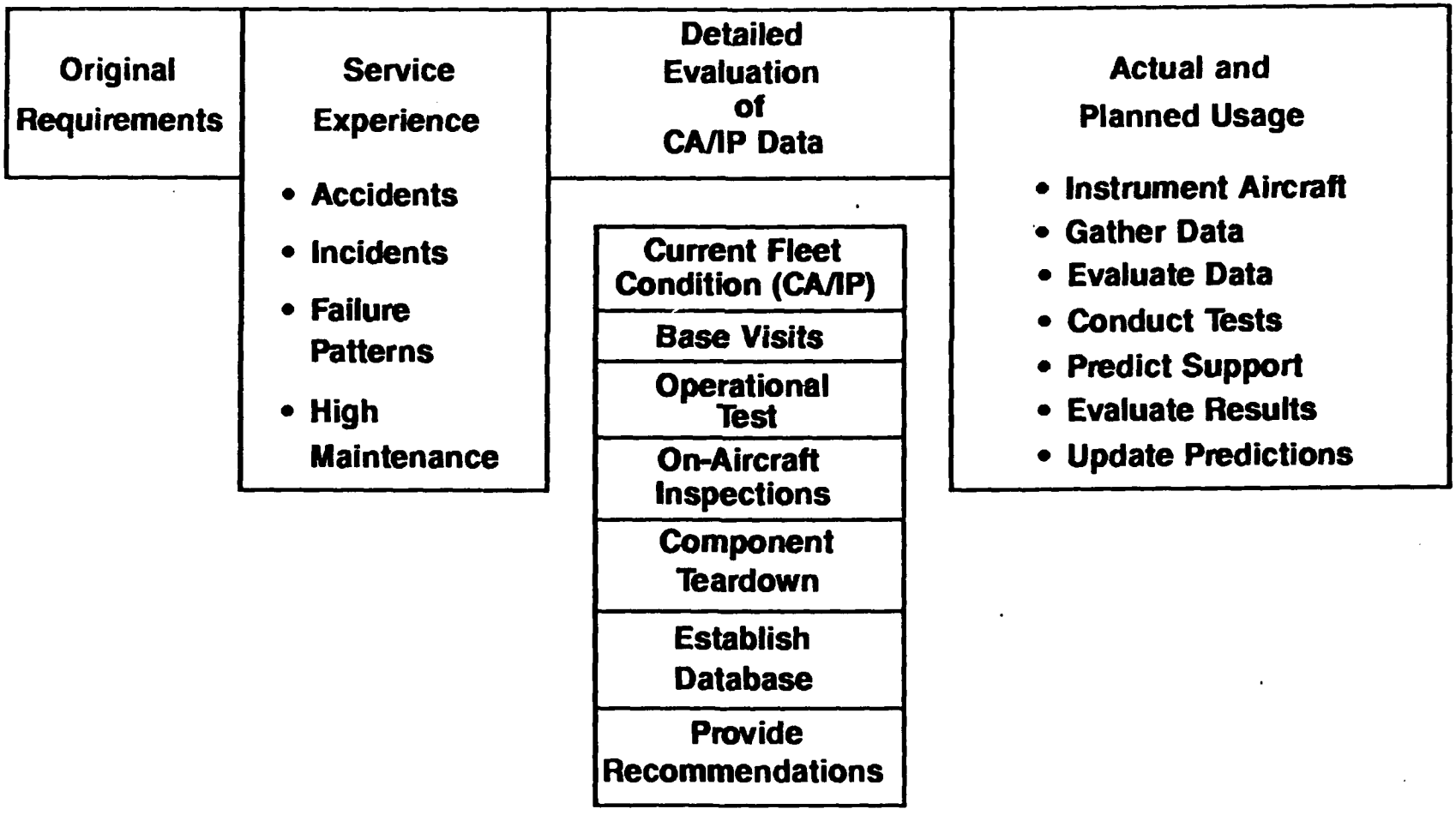
What Is Today's Challenge

- Expand the Successful "Integrity Process" Approach to Functional Systems

- Institutionalize the Process
  - Specifications
  - Standards
  - Air Force Regulations

C-141 Functional Systems Program

Functional Systems Integrity Program



GA-8094-25

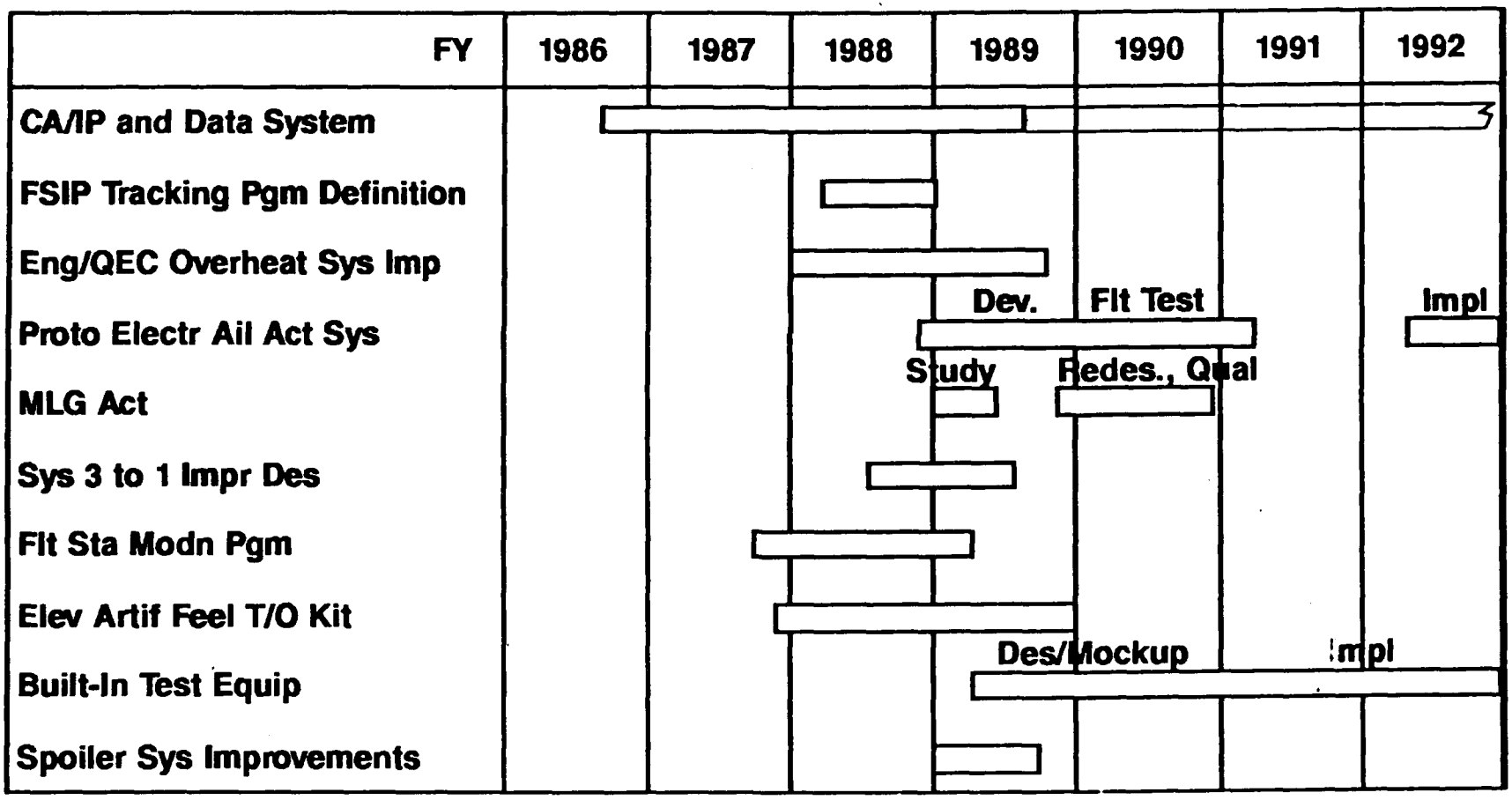
C-141 Functional Systems Program

Functional Systems Plan

- Establish Baseline Assessment and Identify Current Air Force Practices for Development, Qualification and Maintenance of Functional Systems
- Identify Potential Improvements
- Recommend Spec/Std Changes

CA/IP Currently Underway To Aid in This Part of the Plan

C-141 Functional Systems Programs



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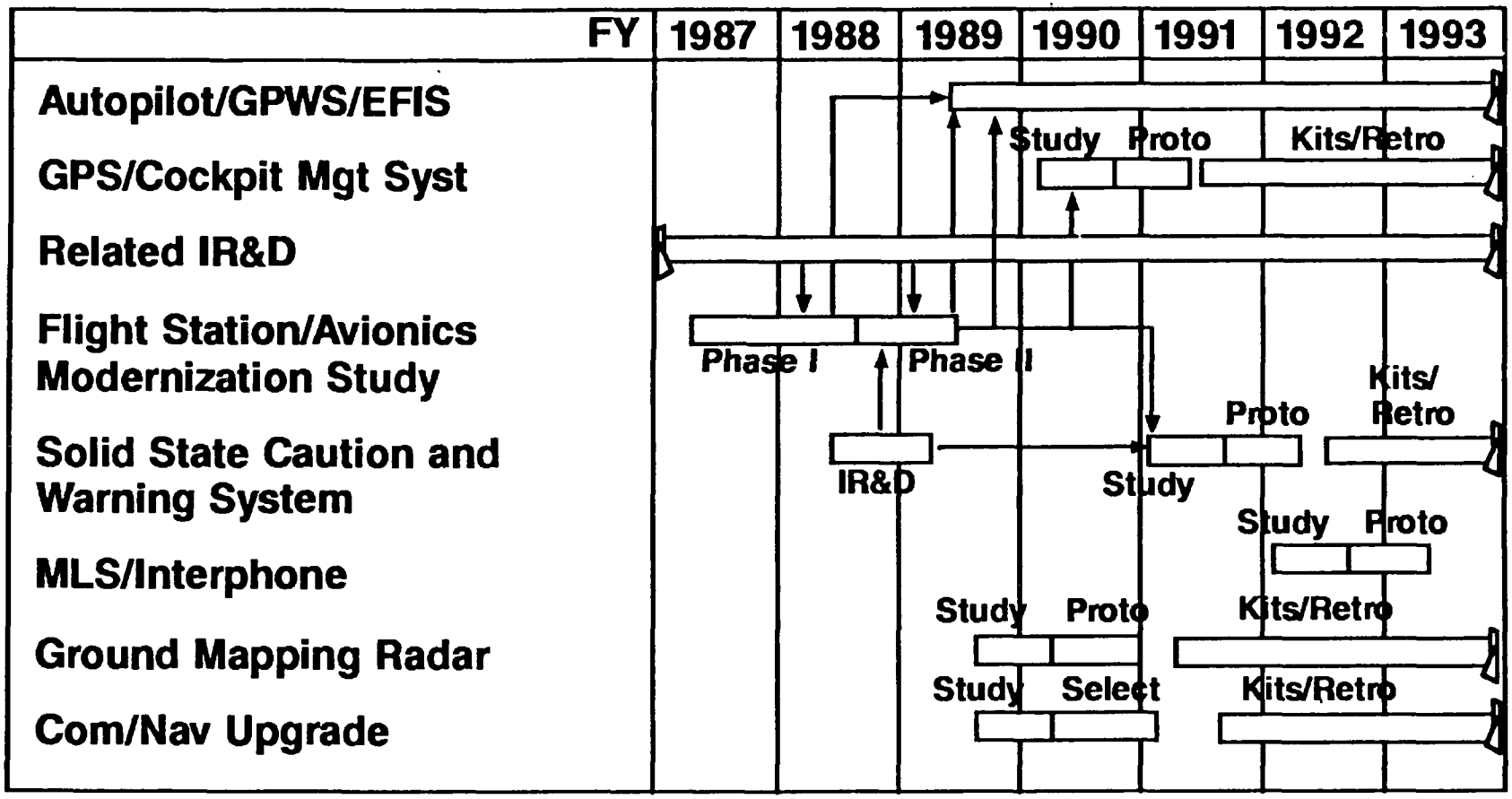
C-141 Avionics Systems Programs



GA-8098-36

C-141 Corrosion

Prevention/Control Program

GA-8094-01

C-141 Corrosion Prevention/Control Program

Why a Corrosion Prevention/Control Program?

- Corrosion Is The Prime Life Limiter
- Corrosion Is Costly
  - Repairs
  - Downtime
  - Safety
  - Readiness
- A Program Is Required To Enhance Service Life

GA-8094-03

C-141 Corrosion Prevention/Control Program

What Is a Corrosion Prevention/Control Program?

- A Plan of Action
- Tailored to Each Airplane Type/Base
- Emphasis Must Be on Prevention
- Requires Support for Life of Airplane
- Will Require Periodic Change

GA-8094-04

C-141 Corrosion Prevention/Control Program

Program Content

- Provides Policy/Technical Visibility
- Outlines Responsibilities at All Levels
  - Field
  - Headquarters
  - Depot
  - Contractor
- Designates Contacts (Includes Avionics, Systems, Structure, etc.)
- Includes Corrosion Prevention Requirements for/in Other Programs
- Commitment to Performance

C-141 Corrosion Prevention/Control Program

Program Implementation

- Tools in Place
  - AFR 400-44
  - MAJCOM/Depot Requirements
  - Technical Orders
  - CPAB
  - Command Surveys
  - Contractors' Expertise

GA-8094-06

C-141 Corrosion Prevention/Control Program

Program Implementation (Cont'd)

- Training
  - Review/Update Tech School Course
  - Review Base Training
  - Use NACE Technician Courses
- Corrosion Inspection & Repair
  - Normally at ISO (Most Bases on 24-Hour Schedule)
  - Long Span Work May Be Deferred (781K, Tub File, etc.)
  - Take Advantage of Airplane Ground Time
  - Only Corrosion People Inspect/Repair

GA-8094-07

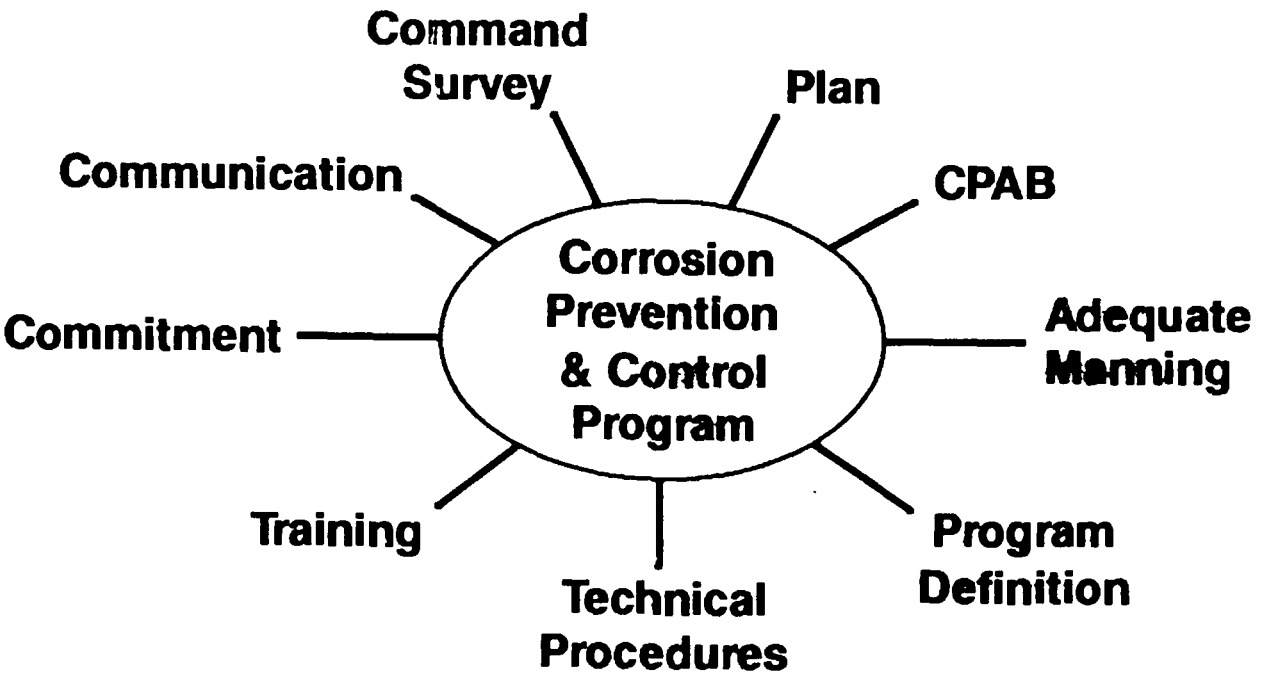
C-141 Corrosion Prevention/Control Program

Program Implementation (Cont'd)

- Emphasis Is on Appearance
  - Corrosion People Mostly Paint
- Manning
  - Most Corrosion Shops Are Undermanned
  - Change Method of Manning Specialist Shops

C-141 Corrosion Prevention/Control Program

Contents of a Good Corrosion Prevention/Control Program



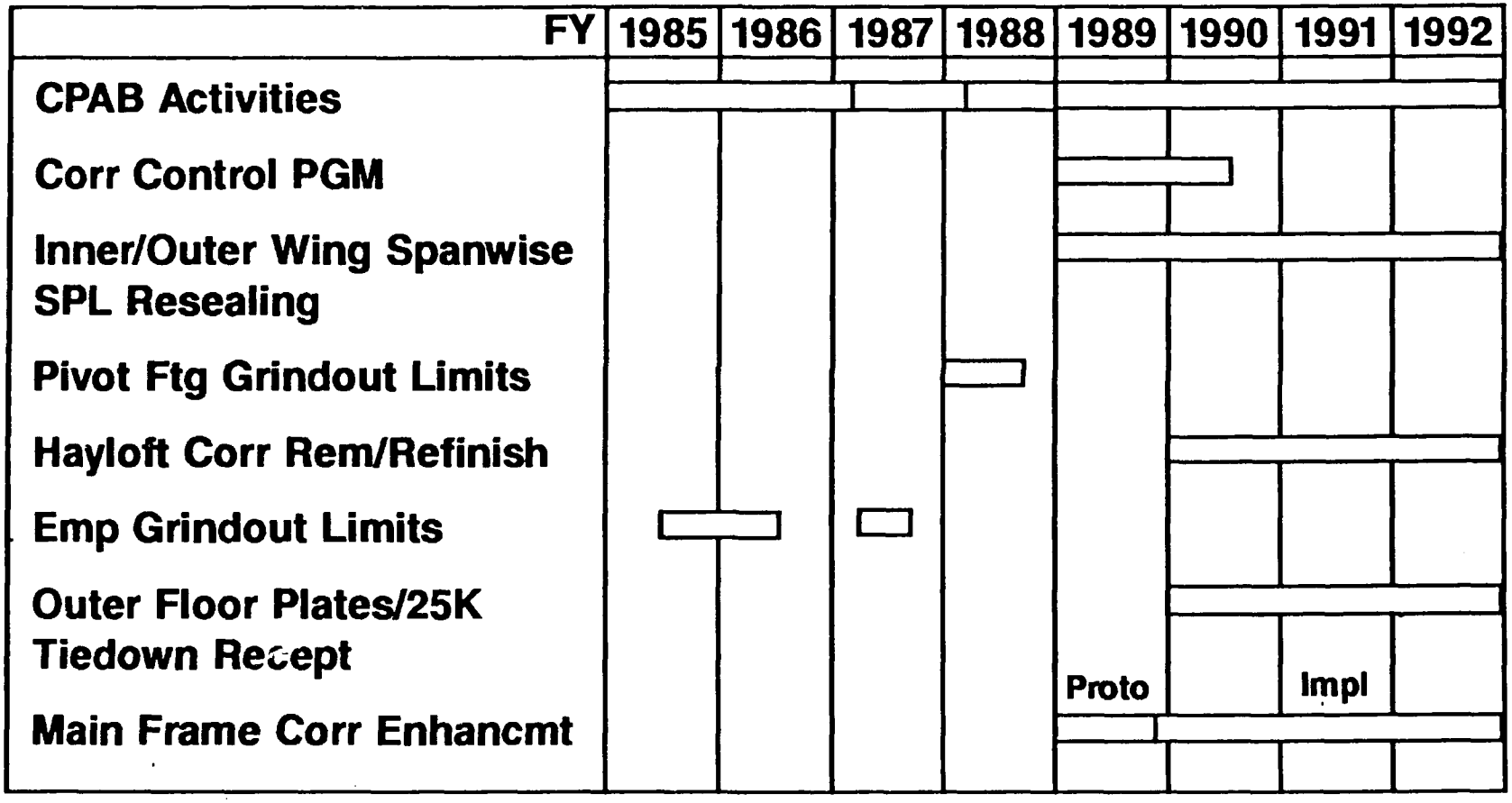
GA-8094-10

C-141 Corrosion Prevention/Control Program

Summary

- Corrosion Is the Prime Life Limiter
- Corrosion Is Expensive
- Program Required - With Emphasis on Prevention
- Changes Are Required
  - Attitude
  - Methodology
- Benefit = Aircraft Longevity

C-141 Corrosion Control Programs



* = On-Going

GA-8098-33

C-141
Supportability Programs

Supportability Engineering

Supportability Goals for C-141

- Achieve R&M 2000 Goals
  - Increase Warfighting Capability
  - Increase Survivability of Combat Support Structure
  - Decrease Mobility Requirements
  - Decrease Manpower Requirements
  - Decrease Costs

GA-8094-15

Supportability Engineering

Programs Under Way or Completed in Recent Years

Operational Support

- Field Visits to MAC Bases (Preliminary to CAIP) - Completed
- Organizational Maintenance Manual Sets (OMMS)
  - FRM/FIM - Complete Except for Lighting System and Engine in a Test Cell
  - General Equipment Manuals - Complete
- Computer Monitored Inspection Program - On-Going

Supportability Engineering

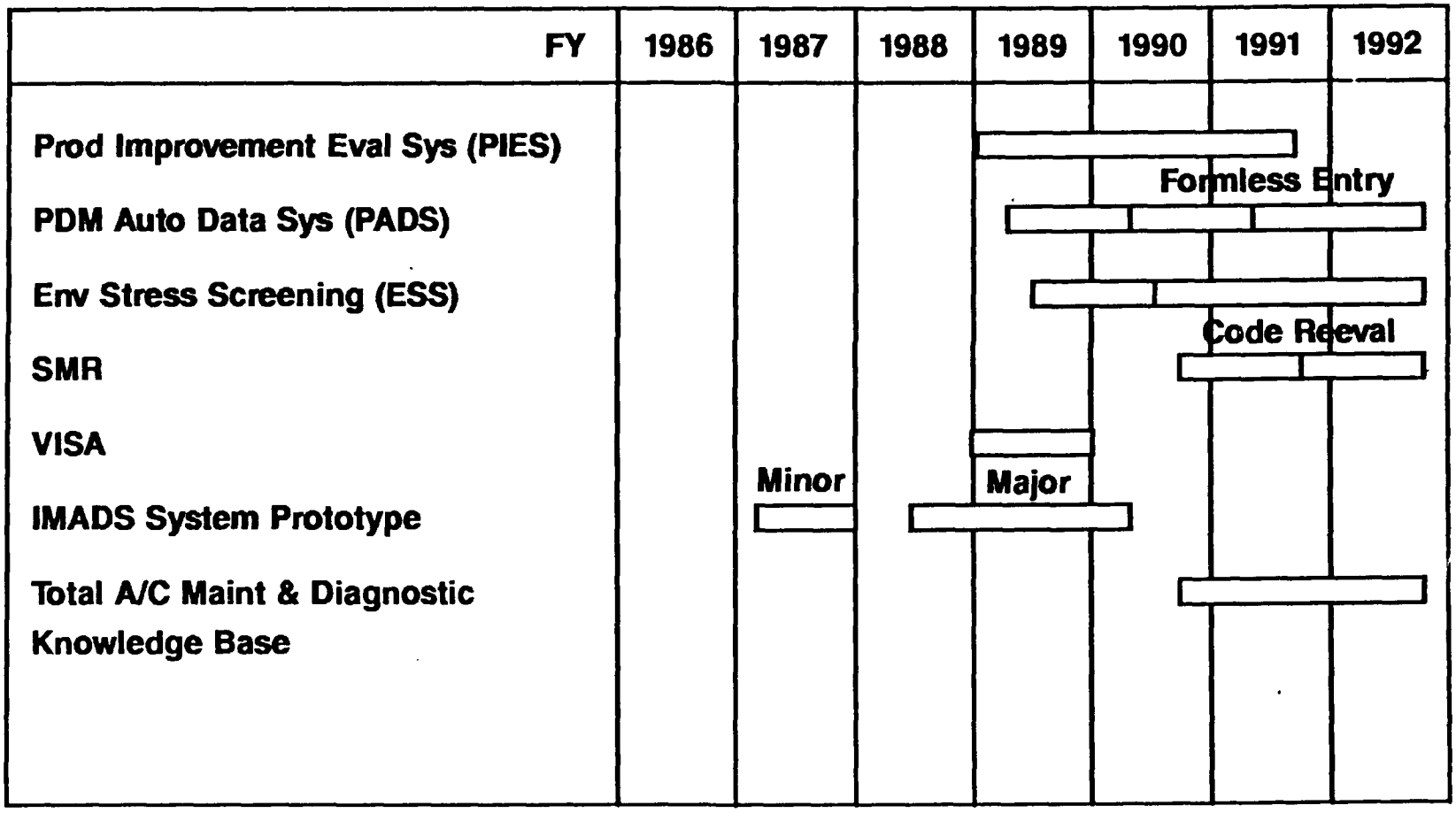
Summary of Proposed Programs

Operational Support

Reliability & Maintainability

- R&M 2000 Product Improvement Evaluation System
- Programmed Depot Maintenance Automated Data System (PADS)
- Environmental Stress Screening (ESS) for Selective Repairables
- Source Maintenance and Recoverability (SMR) Code Re-Evaluation

C-141 Supportability Programs



GA-8098-25

Summary

Summary

- C-141 StarLifter Provides an Unprecedented Blueprint for "An All-Encompassing" Force Management Program Definition
- Although Approaching a Force Average of 30,000 Hours, the C-141 Built With 1950/60's Technology Adapts Well to the Force Management Technology of the 1980s

Summary

- The Concept of a Single "All-Encompassing C-141 Force Management Program" Needs To Be Developed
- Programs Presented Today Provide the Means for Force Management
  - Proactive
  - Meets Requirements of MTM/D, MMH/FH and MC Rate
  - Provide Potential for Increased R&M Benefits

GA-8098-21

SOFI:
NAVY'S STRUCTURAL ONLINE FATIGUE INFORMATION SYSTEM

by

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ABSTRACT

SOFI is the U.S. Navy's Structural Online Fatigue Information System that is currently under development, testing and implementation. Among the salient objectives of SOFI are:

(a) Effective, optimum, and phased replacement of the counting accelerometer group employed over the past two decades on the majority of the Navy's fixed-wing aircraft to manage their structural integrity vis-a-vis structural fatigue;

(b) Optimization of the record-to-report cycle of naval aircraft currently under the Structural Appraisal of Fatigue Effects (SAFE) program, by considering all aspects of data acquisition, processing, and dissemination; and

(c) Modular and uniform development/application of state-of-the-art structural fatigue methodologies with emphasis toward commonality across aircraft types.

This paper presents SOFI in terms of its three subsystems --- the Acquisition Subsystem including the newly developed Structural Fatigue Data System (SFDS), the Processing Subsystem, and the Dissemination Subsystem.

Due to its vast scope, the subject matter is addressed through the highlights of its relevant aspects.

GLOSSARY

ACC	Aircraft Controlling Custodian
AIR-01	NAVAIR Acquisition Executive & Deputy Commander for Operations
AIR-411	Logistics and Maintenance Policy Division
AIR-43	Deputy Assistant Commander for NADEPs
AIR-530	Air Vehicles Division
AIR-5302	Branch Head, Structures, NAVAIR
APML	Assistant Program Manager, Logistics
APMSE	Assistant Program Manager, Systems & Engineering
ASLS	Aircraft Structural Life Surveillance program
CAG	Counting Accelerometer Group
CFA	Cognizant Field Activity
DTU	Data Transfer Unit
FLE	Fatigue Life Expended
ISM	Inertial Sensing Module

GLOSSARY
(Continued)

MM	Memory Module
NADC	Naval Air Development Center
NADEP	Naval Aviation Depot
NADOC	NADEP Operations Center
NAMO	Naval Aviation Maintenance Office
NAVAIR	Naval Air Systems Command
OPNAV	Office of the Chief of Naval Operations
PMA	Program Manager, Air
RC	Reporting Custodian
SAFE	Structural Appraisal of Fatigue Effects program
SFDR	Structural Fatigue Data Recorder
SFDS	Structural Fatigue Data System
SLAP	Structural Life Assessment Program
SLEP	Structural Life Extension Program
SLL	Service Life Limits
SOFI	Structural Online Fatigue Information system
T/M/S	Type/Model/Series of Navy aircraft
WSM	Weapon System Manager

INTRODUCTION

The art of aircraft fleet management is a multifaceted endeavor that constantly seeks to optimally compromise between the operational requirements, the structural capabilities, and the economic viability. Although the emphasis among these three aspects strongly depends upon the applications, the

circumstances and the philosophies --- e.g., military vs. commercial; Navy vs. Air Force; troubled times vs. peace times; safe life vs. fail safe --- a common denominator of cardinal interest to all is the information regarding the structural usage of individual aircraft of the fleet. Such information forms the basis for subsequent estimations of usable structural life remaining in individual fleet aircraft.

Figure 1 globally summarizes NAVAIR's objectives and approach regarding the Navy's aircraft life management program. Among the foremost objectives are the flight safety of the crew and the aircraft, and the operational readiness of the fleet. An effective, timely and reliable life management program also helps ensure for OPNAV the availability of a variety of planning options including flexibility with respect to establishing force levels and mixes.

In order to achieve these objectives, the Navy's approach can be categorized as two-fold in that

(a) structural life limits, in terms of flight hours, catapults, arrestments, etc., are established for all fatigue critical components, and locations of the airframe for every T/M/S aircraft, and

(b) every individual aircraft and its components are tracked to ensure their usage within the established life limits.

FIGURE 1

NAVY'S AIRCRAFT LIFE MANAGEMENT PROGRAM

NAVAIR OBJECTIVES

- ENSURE FLIGHT SAFETY - CREW & AIRCRAFT
- ENSURE OPERATIONAL READINESS
- PRESERVE OPNAV PLANNING OPTIONS
- ENSURE OPNAV FORCE LEVEL/MIX FLEXIBILITY
- ASSESS FUTURE NEEDS

NAVAIR APPROACH

- ESTABLISH STRUCTURAL LIFE LIMITS - AIRFRAME & COMPONENTS
- ENSURE USAGE AGAINST THESE LIMITS

FIGURE 2

NAVAIR APPROACH

- ESTABLISH LIMITS

TESTING

- FATIGUE TESTING DEMONSTRATES SPEC LIFE

- ENSURE LIMITS

TRACKING

- FLEET USAGE TRACKING ENSURES SAFE STRUCTURAL LIFE LIMITS ARE NOT EXCEEDED

The structural life limits are determined (Figure 2) by extensively fatigue testing the full scale aircraft, its major assemblies, components, elements, and coupons. For all of Navy's fixed-wing aircraft, the structural life limits are published by NAVAIR through Reference 1.

The implementation of the above approach to Navy's aircraft life management is embodied in the Aircraft Structural Life Surveillance (ASLS) program (Figure 3). The program comprises three major life cycle phases --- continuous tracking (SAFE), timely assessment (SLAP), and necessary extension (SLEP). SLAPs are generally conducted to address the structural problems experienced in the fleet, and can include flight loads surveys to coupon tests to analyses. Identification of problems/causes/solutions leads to Service Life Extension Programs (SLEPs). Structural modifications/repairs/alterations are designed, implemented and tested, for forward and retroactive fleetwide implementation under SLEPs. Salient details of the life tracking SAFE program are presented next.

SAFE

The key objectives of the SAFE program are summarized in Figure 4. Due to NAVAIR's approach of tracking every individual aircraft, these objectives apply to every Navy fleet aircraft. Guided by the comprehensive list of structural life limits (Reference 1), the operational usage and maintenance actions on airframe and components are recorded and reported. This information is periodically monitored, processed for fatigue life expenditures, and reported for fleetwide dissemination.

FIGURE 3

NAVAIR IMPLEMENTATION

AIRCRAFT STRUCTURAL LIFE SURVEILLANCE PROGRAM

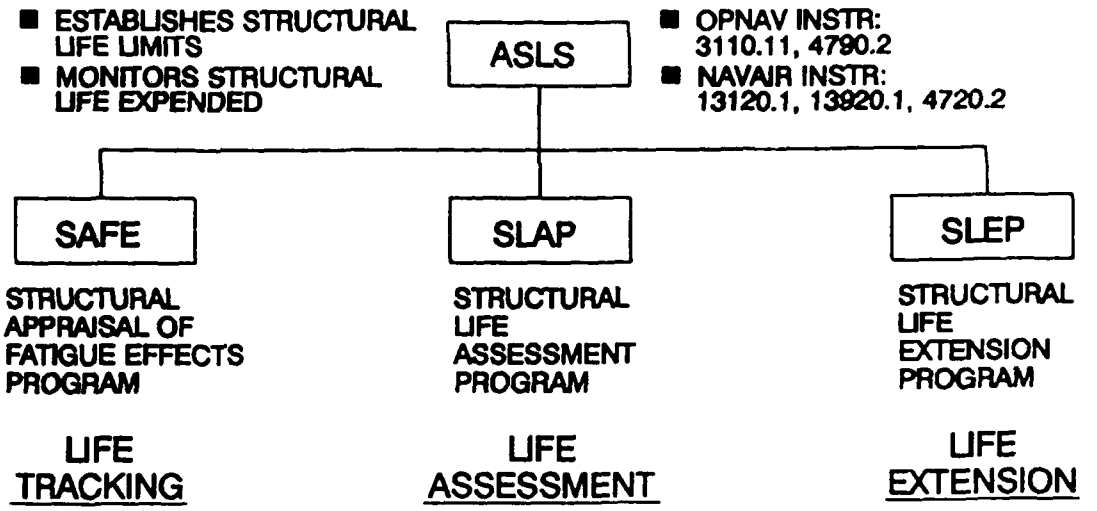


FIGURE 4

SAFE OBJECTIVES

FOR EVERY AIRCRAFT

- COMPREHENSIVE LIST OF STRUCTURAL LIFE LIMITS
  - AIRFRAME
  - CRITICAL COMPONENTS
  - ALL MODES OF NAVY & MARINE OPERATION
- RECORD & REPORT OPERATIONAL USAGE
- RECORD & REPORT MAINTENANCE
  - ROUTINE
  - MODS/ALTERATIONS
  - TIME COMPLIANCE
- COMPONENTS TRACKING (STRUCTURAL)
- MONITOR & REPORT FATIGUE LIFE EXPENDED

Figure 5 illustrates the primary flow of structural usage and life tracking information along with the principally responsible entities within the Navy. The life limits are established by AIR-5302, the Structures Branch in the AIR-530 Air Vehicles Division of NAVAIR. The fleet aircraft are owned and controlled by the Aircraft Controlling Custodians (ACCs). The aircraft users/operators are the squadrons (Reporting Custodians) that periodically report the individual aircraft usage to NADC, for subsequent processing and dissemination.

For the past quarter of a century, the SAFE program has successfully addressed the Navy's needs for individual aircraft tracking. It has evolved and accommodated the growth of the fleet both in numbers and types of aircraft; it has recognized and taken initiatives to keep pace with advances in several of its aspects such as data gathering, transmission, processing, reporting, and updates (Figures 6 through 8).

A majority of the present fleet aircraft are equipped with the Counting Accelerometer Group (CAG) system which principally employs a uni-axial accelerometer located near the center of gravity of the airplane to sense normal acceleration, and a four-level exceedance counter to incrementally (and mechanically) record the Nz excursions of the aircraft. Highlights of the CAG system are summarized in Figures 9 and 10.

In a major endeavor to upgrade the CAG system in its entirety, and to also equip those fleet aircraft with no onboard fatigue monitoring systems, the Navy recently initiated the SOFI system under its ASLS program (Figure 11). Salient aspects of SOFI are presented in the following section.

FIGURE 5

SAFE RESPONSIBILITIES

NAVAIR HQ
- AIR-01
- PMA'S, WSM'S
- APMSE'S
- APML'S
- AIR-530
- AIR-411
- AIR-43

NAMO
CFA'S
NADEP'S
ACC'S
RC'S
NADC
NADEP OP CTR

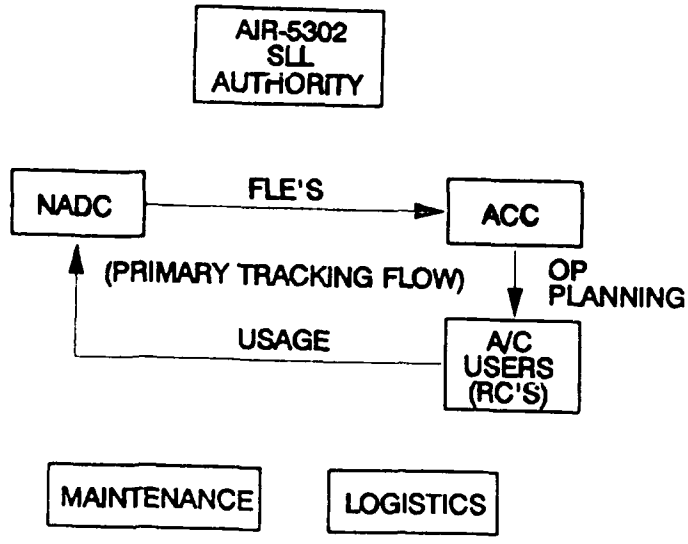


FIGURE 6

SAFE
(TRACKING)

- DATA GATHERING
  - FATIGUE DATA
    - SINGLE - PARAMETER (N_z)
    - MULTI-PARAMETER (N_z , V, h, STRAINS, ...)
    - LANDINGS/ARRESTMENTS
    - CATAPULTS
  - APPROACH
    - CUMULATIVE EXCEEDANCES
    - LOAD/STRAIN CYCLE HISTOGRAMS
    - LOAD/STRAIN CYCLE SEQUENCE

FIGURE 7

SAFE

(TRACKING)

CONTINUED

- DATA GATHERING
  - EQUIPMENT
    - AIRBORNE
      - TRANSDUCERS (ACCELS, STRAIN GAGES)
      - RECORDERS (ELECTRO-MECHANICAL COUNTERS, ELECTRONIC, ...)
    - ON-GROUND
      - GROUND STATIONS
        - DOWNLOADING
        - UPLOADING
        - ERROR INDICATION
        - MAINTENANCE ACTIONS
- DATA TRANSMISSION
  - POST CARDS (MAIL)
  - MAGNETIC MEDIA (MODEM, MAIL)

FIGURE 8

SAFE

(TRACKING)

CONTINUED

- DATA PROCESSING
  - DATA QUANTITY CHECKS
  - DATA QUALITY CHECKS
  - TRACKING METHODOLOGIES
  - FATIGUE ALGORITHMS
  - DATA ARCHIVING
- DATA REPORTING
  - PERIODIC REPORTS
  - SITUATION REPORTS/MESSAGES
  - STATISTICAL REPORTS
- UPDATES
  - METHODS/PROGRAMS
  - INSTRUMENTATION/HARDWARE

FIGURE 9

COUNTING ACCELEROMETER GROUP SYSTEM (CAG)

(PRESENT TRACKING)

- ON MAJORITY OF (EXPERIENCED) NAVY AIRCRAFT
- SOME AIRCRAFT WITHOUT ANY SYSTEMS
- DATA GATHERING
  - FATIGUE DATA
    - SINGLE-PARAMETER (N_z)
    - LANDINGS/ARRESTMENTS (LOGBOOK)
    - CATAPULTS (LOGBOOK)
  - APPROACH
    - CUMULATIVE N_z EXCEEDANCES
  - EQUIPMENT
    - AIRBORNE
      - N_z ACCEL
      - ELECTRO-MECHANICAL COUNTER
    - ON-GROUND
      - NONE REQUIRED

FIGURE 10

CONTINUED

COUNTING ACCELEROMETER GROUP SYSTEM (CAG)

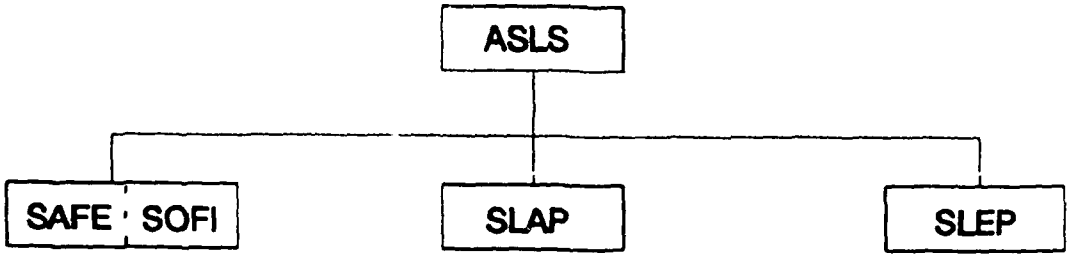
(PRESENT TRACKING)

- DATA TRANSMISSION
  - POST CARDS (MAIL)
- DATA PROCESSING
  - EXTENSIVE QUANTITY & QUALITY CONTROL CHECKS
  - SIMPLE/CONSERVATIVE FATIGUE LIFE ESTIMATION (MINER'S RULE)
- DATA REPORTING
  - SAFE, FLAG, STATISTICAL SUMMARIES, MESSAGES
- UPDATES
  - METHODS/PROGRAMS
    - MULTIPLE POINTS-IN-SKY
    - SEQUENCE-DEPENDENT FATIGUE ALGORITHMS

FIGURE 11

SOFI

(TRACKING)



- A/C WITH CAG'S
- A/C WITHOUT ANY FATIGUE MONITORING SYSTEM

FIGURE 12

SOFI OBJECTIVES

- PHASED-IN REPLACEMENT FOR CAG
- PHASED-IN SYSTEM FOR A/C WITH NO ONBOARD SYSTEM
- IMPROVED ACCURACY
  - -USAGE RECORDS
  - -FLE'S
- REDUCED HUMAN ERROR
- REDUCED USAGE (RECORD)-TO-REPORT CYCLE TIME
- COMMONALITY OF METHODS
- MODULARITY OF METHODS
- ONLINE

SOFI

The principal objectives of SOFI are grouped in Figure 12. As stated above, the origins of SOFI lie in upgrading and improving upon the CAG system. A detailed list of planned improvements is compiled and presented in Figures 13 and 14. As seen in these figures, all aspects of structural usage and life tracking are being addressed in developing SOFI.

SOFI can be described as an information system designed to gather, process, and channel the flow of structural fatigue-related information on Navy aircraft to the users, owners, planners and maintainers. By its very nature, SOFI relies on interactions between people, aircraft, information, and computer hardware and software. A clear understanding and acceptance of these interactions is, therefore, imperative and necessary in accomplishing SOFI's objectives.

From the standpoint of functional clarity, the SOFI system is divided into three distinct but interfacing subsystems (Figure 15):

(a) Acquisition Subsystem (Figures 16, 17),

(b) Processing Subsystem (Figures 18, 19), and

(c) Dissemination Subsystem (Figure 20).

FIGURE 13

CAG IMPROVEMENTS → SOFI

- DATA GATHERING
  - FATIGUE DATA
    - SINGLE-PARAMETER → MULTI-PARAMETER
    - LANDINGS/ARRSTS/CATS: LOGBOOK → ON-BOARD RECORDER
  - APPROACH
    - CUM EXCEEDANCES → LOAD/STRAIN CYCLES & SEQUENCE
  - EQUIPMENT
    - AIRBORNE: ONE ACCEL, EL-MECH COUNTER → MULTI-ACCEL, MULTI-STRAIN GAGE ELECTRONIC RECORDER & STORAGE
    - ON-GROUND: NONE, PRESENTLY → ELECTRONIC DATA TRANSFER UNIT

FIGURE 14

CAG IMPROVEMENTS → SOFI

CONTINUED

- DATA TRANSMISSION
  - POST CARDS, MAIL → DISKETTES, MODEM, MAIL
- DATA PROCESSING
  - EXTENSIVE QUANTITY & QUALITY CHECKS → (HOPEFULLY!) REDUCED CHECKS
  - CONSERVATIVE FLE'S → REALISTIC, MULTIPLE POINTS-IN-SKY, STRAINS-SEQUENCE BASED FLE'S
- DATA REPORTING
  - HARDCOPY → HARDCOPY, ONLINE, SELECTIVE
- UPDATES
  - METHODS/PROGRAMS: MOST SOFTWARE IN NEED OF UPDATE → ADVANCES IN DATABASE, FATIGUE, INFORMATION SOFTWARE, MODULARITY, COMMONALITY

FIGURE 15

SUBSYSTEMS

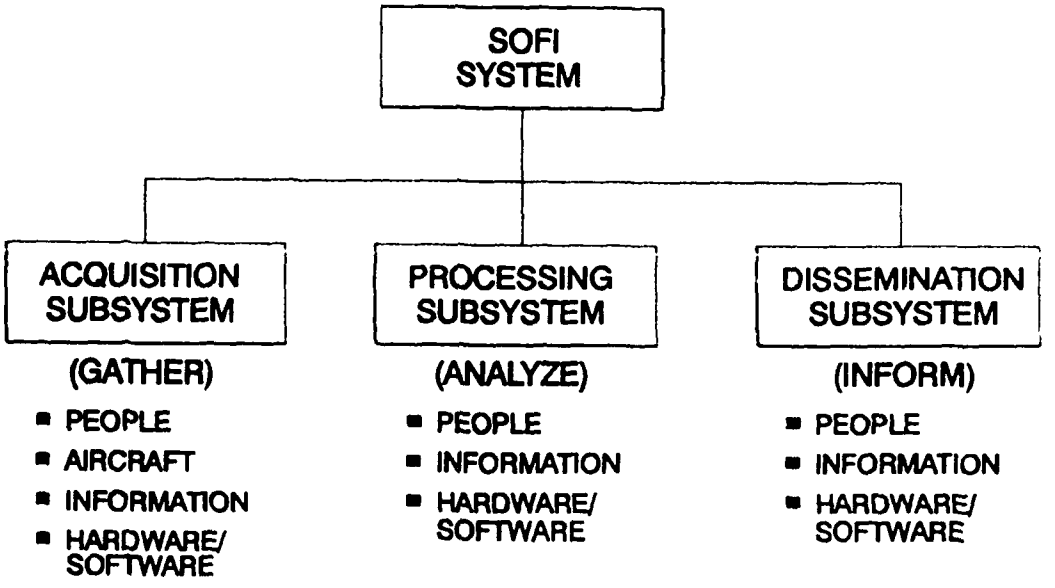


FIGURE 16

ACQUISITION SUBSYSTEM

PRIMARY FUNCTION

- DATA GATHERING
- DATA TRANSMISSION

ELEMENTS

- PEOPLE
  - REPORTING CUSTODIANS (SQUADRONS, OTHER USERS)
    - GATHERING & TRANSMISSION
- AIRCRAFT
  - ALL AIRCRAFT ADDRESSED BY SOFI
- INFORMATION
  - INDIVIDUAL AIRCRAFT USAGE
    - MONTHLY BASIS
    - SITUATION BASIS
- HARDWARE/SOFTWARE
  - STRUCTURAL FATIGUE DATA SYSTEM (SFDS)
  - 3 1/2" DISKETTES
  - MODEMS
  - ONBOARD SOFTWARE (SFDR)
  - ON-GROUND SOFTWARE (DTU)

FIGURE 17

CONTINUED

ACQUISITION SUBSYSTEM

PEOPLE - SOFTWARE INTERFACES

- DTU-BASED
  - INITIALIZE SFDR
  - DOWNLOAD MM
  - DATA ONTO 3 1/2" DISKETTES
  - MODEM DATA OUT
  - RECEIVE DATA (MODEM, DISKETTES)
    - SOFTWARE UPDATES
    - DTU-BASED INFO UPDATES
- CENTRAL PROCESSING FACILITY-BASED
  - RECEIPT/ACKNOWLEDGMENT OF USAGE DATA

FIGURE 18

PROCESSING SUBSYSTEM

PRIMARY FUNCTION

- DATA RECEIPT/ACKNOWLEDGMENT
- DATA PROCESSING
  - QC
  - FATIGUE
  - OTHER
- DATA PREPARATION FOR DISSEMINATION SUBSYSTEM
- DATA ARCHIVING

ELEMENTS

- PEOPLE
  - CENTRAL PROCESSING FACILITY
    - DEVELOPMENT
    - PRODUCTION
    - COMPUTER SYSTEMS

FIGURE 19

CONTINUED

PROCESSING SUBSYSTEM

- INFORMATION
  - INDIVIDUAL AIRCRAFT USAGE
  - INDIVIDUAL AIRCRAFT FLE'S
  - STRUCTURAL LIFE LIMITS
  - INDIVIDUAL AIRCRAFT CONFIGURATION
  - MAINTENANCE DATA/RECORDS
- HARDWARE/SOFTWARE
  - CENTRAL COMPUTER SYSTEM
  - SYSTEM SOFTWARE
  - TRANSMISSION SOFTWARE
  - QC SOFTWARE
  - FATIGUE SOFTWARE
  - DATABASE SOFTWARE

PEOPLE - SOFTWARE INTERFACES

- WITH ACQUISITION SUBSYSTEM
- WITH DISSEMINATION SUBSYSTEM

FIGURE 20

DISSEMINATION SUBSYSTEM

PRIMARY FUNCTION

- INFORMATION DISSEMINATION
  - AIRCRAFT USAGE DATA
  - AIRCRAFT FLE'S
  - REPORTS (HARDCOPY & ONLINE)
  - OTHER (MAINTENANCE, STRUCTURAL LIFE LIMITS,...)

ELEMENTS

- PEOPLE
  - ALL USERS
- INFORMATION
  - RECEIVED, PROCESSED, STORED (USAGE, FLE'S, MAINT., SLL'S,...)
- HARDWARE/SOFTWARE
  - SIMILAR TO PROCESSING SUBSYSTEM

PEOPLE - SOFTWARE INTERFACES

- ONE INTERFACE FOR ALL USERS WITH SELECTIVE ACCESS TO VARIOUS INFORMATION

In these figures, each of the subsystems is described in terms of its primary function, principal elements and interfaces. One of the most substantial advantages of defining SOFI through its subsystems and interfaces is that the developments, implementations, modifications, and upgrades within any subsystem can be carried out independently, with the other subsystem(s) kept informed only through the interfaces. This modularity, by definition and design, also provides for an inherent mechanism for timely keeping up with state-of-the-art advances in, e.g., fatigue damage algorithms, and data transmission hardware/software.

Figure 21 illustrates the primary flow of tracking information through the three SOFI subsystems. This flow path is essentially identical to the present information flow path shown in Figure 5.

The next subsection discusses one of the most salient elements of the entire Acquisition Subsystem.

Structural Fatigue Data System

The Structural Fatigue Data System (SFDS, Figure 22) is divided into two groups --- the airborne portion of the system and the ground based portion.

The airborne portion of the system is composed of three major components (Weapon Replaceable Assemblies), the Structural Fatigue Data Recorder (Table I), Inertial Sensing Module (Table II), and the Memory Module (Table III). The ground based portion of the system is the Data Transfer Unit.

FIGURE 21

OVERVIEW OF PRIMARY TRACKING FLOW

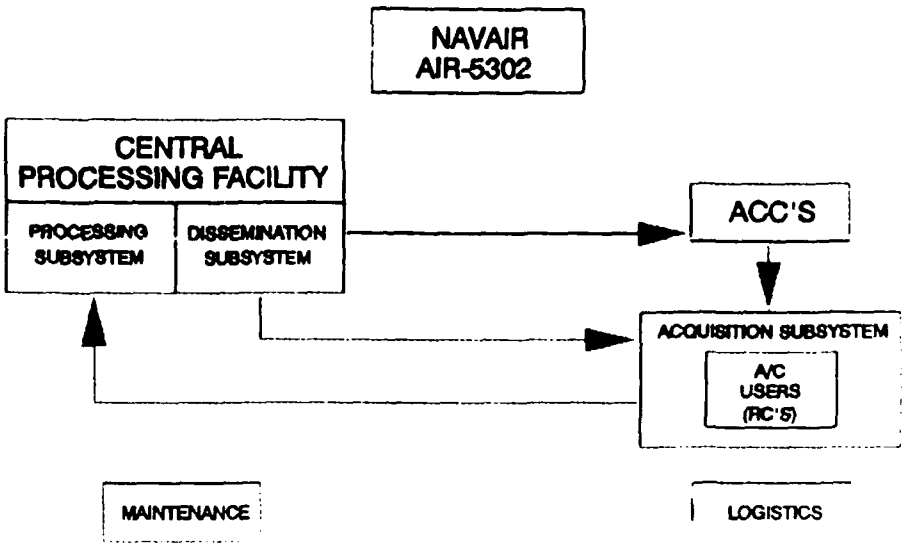


FIGURE 22

SFDS

(STRUCTURAL FATIGUE DATA SYSTEM)

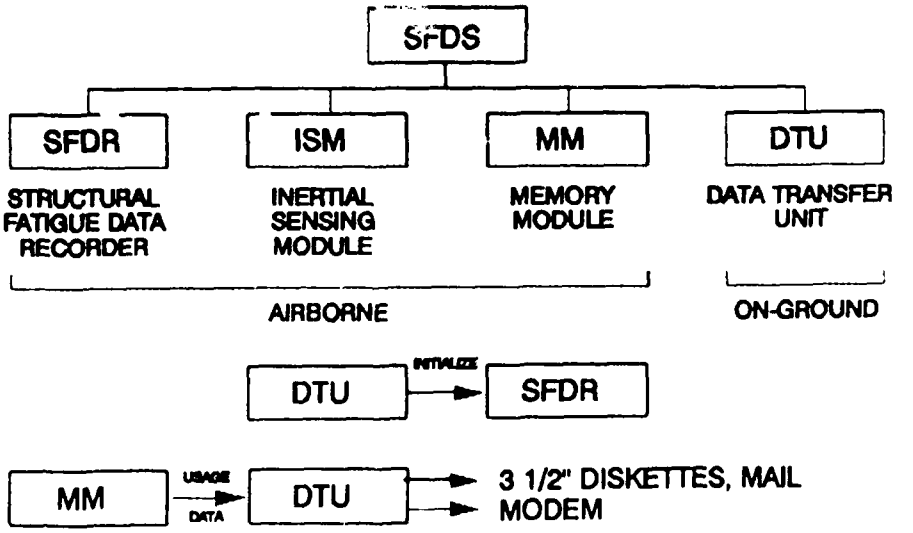


TABLE I

Structural Fatigue Data Recorder
(Recorder-Converter RO-601/ASH-37)

Power Requirements:	
Input Voltage	+28 VDC per MIL-STD-704
Power	Less than 10 watts
Insulation	Power & Signal - 50 Megohms
Performance:	
Input Channels	2 pressure, 3 accelerations and 7 electrical
Band width:	12 Hz
Input Impedance	200,000 ohms
Accuracy	± 1 % of full scale
Digital Bus	RS232
Sampling Rate	Programmable to 512 samples second
Algorithm Choices	Peak-valley, fixed band
Each Channel	Usage Matrix or combinations
Recording Format	Uses Configurable by DTU
Built-in-Test (BIT)	End to End
Indication	Visual Flag and BIT status in Recorded Data
Mean Time Between Failures (MTBF):	
Predicted MTBF	21,961 Hours

TABLE II

Inertial Sensing Module (ISM)
Transducer Motional Pickup TR 352/ASH-37

Performance:	
Acceleration	3 axis sevo type Accelerometer
Normal Acceleration (Nz)	± 10g
Longitudinal Acceleration (Nx)	± 10g
Roll Angular Acceleration $\dot{p}$	± 40 radians/sec^3
Accuracy	0.5 % of full scale typical
Mean Time Between Failure	41,889 Hours
Predicted MTBF	

TABLE III

Memory Module
Memory Unit, MU 983/ASH

Memory Module	Removable, Non-Volatile with No Batteries
Capacity	Expandable to 200K bytes
Indicators	Flys for BIT & Memory Usage
Download method	Data Transfer Unit
Aircraft Identification	(Recorder-Reproducer) RD-608/ASH-37
Special Purpose Data	Programmed into SFDR by DTU
Mean Time Between Futures	Number of flights, number of arrestments, number of catapults
Predicted MTBF	17,914 Hours

The Structural Fatigue Data Recorder is designed to measure 20 channels of data (Figure 23). These 20 input channels are divided into eight discrete inputs (on-off) such as store release, inflight refueling, weight-off/on wheels, flaps up/down, etc. The other 12 channels are designed for analog inputs; five of which are currently dedicated to the system for the measurement of three acceleration signals from the Inertial Sensing Module and the other two channels are dedicated for the measurement of pitot and static pressures to calculate airspeed and altitude. The remaining seven channels are divided into two signal level categories. Three of the seven remaining channels accept electrical signals of ± 10 volts while the remaining channels accept ± 60 millivolt signals. All low level channels have five pole 12 Hz filters. The signal conditions for the two linear accelerations contain five pole 12 Hz filters while the signal conditioning for the angular acceleration has a three pole 12 Hz filter. The total system accuracy, which includes the instrumentation amplifier, the active filters, the track and hold amplifier and the analog to digital converter errors, does not exceed ± 1 percent of full-scale over the temperature range of -54° to 71°C.

The SFDR has a programmable sampling rate for each channel with a total rate of 512 samples per second. The SFDR records data by employing three algorithms or a combination of these algorithms. These algorithms are peak/valley, fixed band and usage matrix, and are user selectable by employing the DTU. During the recent flight test effort on the A-6E aircraft, the peak/valley algorithm was used. The other unique recording features are aircraft takeoff and landing algorithms.

FIGURE 23

SFDR

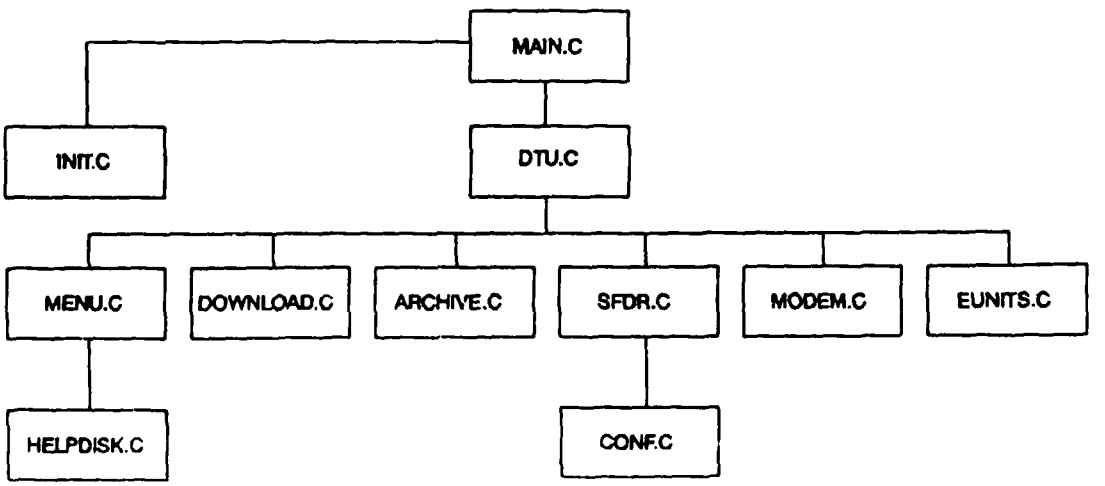
(STRUCTURAL FATIGUE DATA RECORDER)

- FATIGUE PARAMETERS RECORDED
  - N_z ($\pm$ 10 g)
  - N_x ($\pm$ 10 g)
  - $\dot{p}$ ($\pm$ 40 rad/sec/sec)
  - V (0-850 Kts, MSL)
  - h (-1K to 70K feet)
  - Elapsed Time
  - 4 Strains (-3000 to 6000 micro-inch/inch)
  - 3 Additional Accels ($\pm$ 10 VDC)
  - 8 DISCRETE (WEIGHT-ON-WHEELS, FLAPS UP/DOWN, ...)

TOTAL 20 CHANNELS, PLUS ELAPSED TIME

FIGURE 24

DTU SOFTWARE STRUCTURE



Built-In-Test (BIT) is performed when power is applied to the system. This end-to-end test is documented in the Memory Module as a BIT word in the header record.

The system was installed on A-6E aircraft at the Naval Air Test Center, Patuxent River, Maryland for flight test evaluation. The flight test aircraft was instrumented with a telemetry system and SFDS. The Navy's telemetry instrumentation system was employed to evaluate the SFDS. Both systems measured the airspeed, altitude, Mach number, roll rate, roll acceleration, Nx, Nz, main gear position, flap position, hook position and weight on/off wheels. The flight test data from both instrumentation systems was obtained for six catapults, 17 arrestments, 11 touch and go's and 53 high Nz maneuvers. The evaluation of the data exhibited good correlation between both instrumentation systems.

The DTU consists of two major subsystems: the laptop computer and the memory unit interface. The DTU provides a means of extracting the recorded flight data from the memory module. The recorded data can be transmitted to the data processing facility either by 3 1/2" floppy disk or by modem. The menu-driven DTU software requires a minimum number of key strokes to download the flight data from the memory module. The DTU is also used to upload the proper configuration of each type aircraft from the data files stored on its hard disk. Figure 24 illustrates the DTU software structure.

CONCLUDING REMARKS

This paper has attempted to summarily review the highlights of the Navy's SOFI system in terms of its background, objectives and approach. The system is presently under various stages of development, testing and implementation phases.

ACKNOWLEDGEMENTS

All aspects of SOFI are under the sponsorship of the Structures Branch (AIR-5302) of the Naval Air Systems Command in Washington, D.C.

REFERENCE

1. NAVAIR Instruction 13120.1B, "Fixed Wing Aircraft Structural Life Limits," published by Commander, Naval Air Systems Command, Washington, D.C.

The SPECTRAPOT System: Multichannel On-Board Fatigue Monitoring.

H. R. Wilms, SPECTRALAB, Brunnenmoosstrasse 7
CH - 8802 Kilchberg, Switzerland

The new multichannel SPECTRAPOT System is a compact 16-bit micro-computer equipped with a real-time, multitasking, multiprocessing operating system for use as a continuous autonomous on-board fatigue monitoring computer.

1. General

The **key element** of the system is the **Data Collector**, which is installed in the aircraft. The Data Collector's task is to **sample** measurement data over a period of time, varying from several hours to some months, to **process** it and, after data reduction has been completed, to **store** it. The storage medium is a removable 512 kByte Memory Cassette, which can be read in a ground based Data Processor or on site the aircraft's Data Collector. This Memory Cassette can be exchanged by mail, which can be dealt by assistant personnel.

The multichannel Data Collector is resistant to environmental factors such as low/high temperatures, heavy accelerations, high shocks, vibration. It is **MIL-STD810C approved.** As a result of consequent RF shielding under EMI influence, it is **MIL-STD 461B approved.**

Figure 1: The SPECTRAPOT Data Collector

Equipped with classification programs, such as **Realtime Rainflow** (RTRF), **Time-at-Level** (TAL), etc. or with other programs such as

Sequential Peak Valley (SPV/SPVc), **Smoothed Raw Values** (RAW), **Fast Fourier Analysis** (FFT), etc. the SPECTRAPOT System mainly acquires data on load cycles of mechanical construction parts relevant to **material fatigue.** Within the use of the operating system it is possible to run more than one program on each channel. An efficient **16/32-bit computer** (the Data Processor) equipped with two floppies and/or a harddisk, a terminal and a graphic printer is the ground based processing station. Furthermore, the processing system can be connected on-line with a host-computer, where the **structural analysis program** of the aircraft manufacturer runs.

The initialisation program in the processing system is used to prepare the Memory Cassette for a new measuring serie. The Memory Cassette will be formatted, the configuration and the programs for the Data Collector are loaded. At the time of initialization, it is **not** necessary to know in which aircraft the respective memory is fitted. When fitting the memory in a specific aircraft, the correct calibration values are activated from the **resident memory** (512 kBytes) of the Data Collector. Through this processing concept, it is possible to install **program-updates** as well as completely **new** programs in the aircraft, without changing any hardware components. Taking care of the software can be done centrally, which is an enormous advantage.

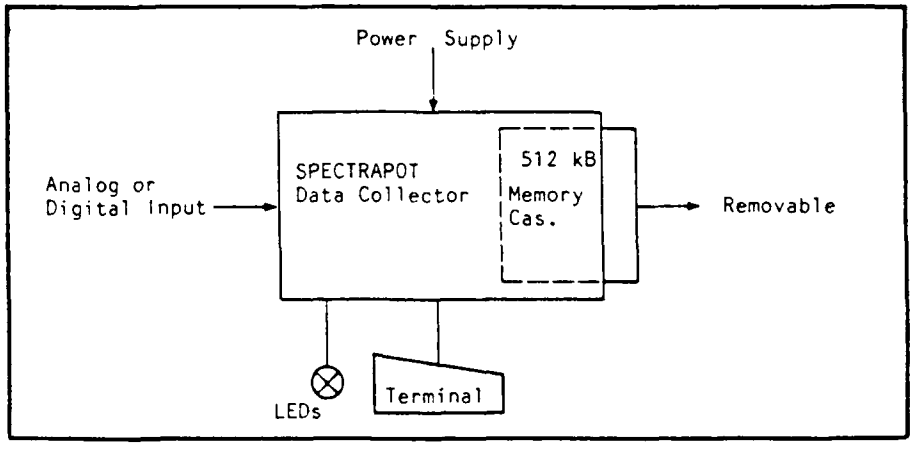


Figure 2: The Data Collector key elements (measuring concept)

The principles of the SPECTRAPOT System can be summarized as follows:
- multi-channel, multi-tasking, real-time measurement with the SPECTRAPOT Data Collector
- Storage after data reduction, classification or processing

- Data compression and storage in solid state memory with a capacity of 512 kByte
- No moving parts
- Exchangeable Memory Cassettes which can be evaluated centrally in the Data Processor
- Control of configuration, scaling and results on site the aircraft
- Central software update: processing program is transferred into the Data Collector by a solid state Memory Cassette: RAMdisk concept
- Data filing and complete measurement listing with graphics
- Fleet and operating statistics is possible centrally in connection with a mainframe (host)

2. The SPECTRAPOT Data Collector and its environment

The SPECTRAPOT Data Collector is connected to its environment over different interfaces (modules). **Power** is supplied by means of a two-wire cable from a battery, an adapter or from the aircraft's +28 VDC supply. The sensors or measurement data sources are connected by means of the appropriate voltage or current interfaces. *For sensors, producing only small signals* (e.g. strain gages), sensor-amplifier modules are available.

For the SPECTRAPOT Data Collector there are two different **analog** and **analog/digital** modules:
- The standard **Analog** Module is a 12 bit analog/digital converter, ready to connect 4 external low-level signal preamplifier or suitable sensors, such as accelerometers, directly. The selftest voltages are also provided and software supported from this board.
- The **UNIO** (UNiversal Input/Output) Module is the second one, and can be connected with up to 16 input channels (digital and/or 12 bit analog). The module has a **local** 16 bit 68'070 and dual ported RAM for powerful input preprocessing and for easy communication with the CPU module. A local DC/DC converter supplies voltages for external sensors, such as strain gages, etc. The configuration of the input channels (analog or digital) is made with an adapter, to be defined at time of order.

It is possible to **stack** two of the mentioned modules together in one SPECTRAPOT Data Collector.

The **serial V-24/RS-232** interface of the SPECTRAPOT Data Collector can be connected to a terminal for servicing purposes, for the acquisition of static measuring and control signals or to

download the results when not removing the Memory Cassette. In other words this serial interface is supported multilaterally.

Each of the two **LED's** have three brilliant colors which can also blink in a fast or slow mode. This allows a quick test to check e.g. that none or a faulty Memory Cassette is fitted, or that the power is too low, or that the processing unit does not run correctly. These LED's indicate also what the Data Collector is doing at the moment: measuring, self-test, booting, etc.

Finally, the **Memory Cassette** can be regarded as an interface, the operating programs and the measuring parameters (configuration) are transferred into the Data Collector, and after the measurement has been completed, the measurement data (results) are located in the Memory Cassette. By exchanging the Memory Cassette, the measurement data can be transferred to the SPECTRAPOT Data Processor, where they can be further processed, filed and represented. The Data Processor, a kind of personal computer with 2 floppy drives and/or harddisk, is served by a standard terminal. An additional serial interface in the processor allows communication with a host computer. Also other

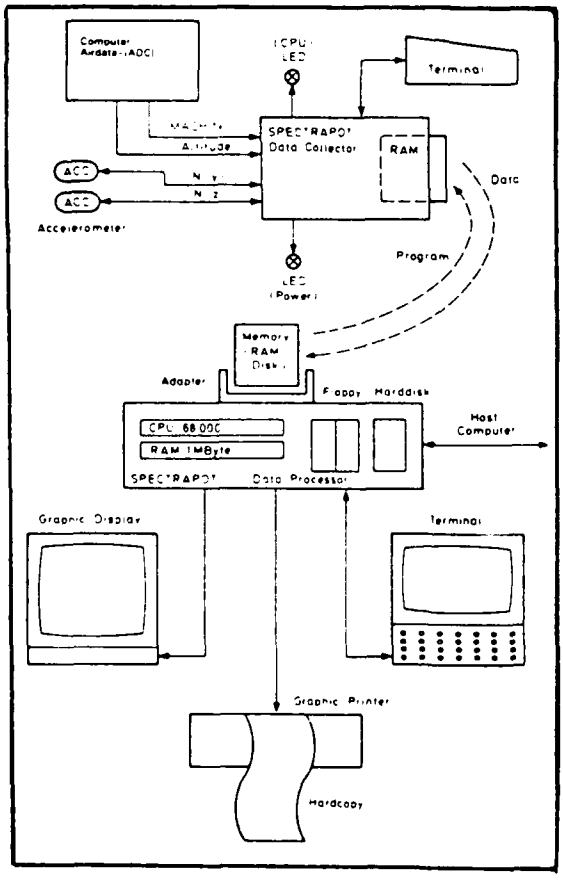


Figure 3: Complete system lay-out: several mobile Data Collectors at the measurement locations, sensors and sensor amplifiers, data transfer to the central Data Processor via the solid state Memory Cassette, operating program back to the Data Collectors, central data evaluation, possible further processing by host computers

interfaces can be offered, such as Ethernet, Cheapernet, etc. to speed up the transfer.

3. Measuring with the SPECTRAPOT Data Collector

Before the SPECTRAPOT Data Collector is started, it must be **prepared**, calibrated for the measurement. For this purpose, there is a **maintenance program** (maint) for the Data Collector with which the analog inputs can be scaled. The possibility of universal **scaling in physical units** is one of the main advantage of the SPECTRAPOT System.

The program '**Maint**' is normally loaded automatically into the execution directory of the Memory Cassette (/c0/cmds = execution directory) when initializing a Memory Cassette, with the command 'cas'. Initialization provides the complete information about what to measure, which algorithm is used for each channel, a part of the OS9 operating system, scaling, etc. all this modules are transferred into the Memory Cassette.

There are series of varied configurations, corresponding to the different requirements involved in the data acquisition.
For example, all 8 channels can operate independently, or channels 2 and 3 are recorded correlated to channel 1, and the other channels are computed independently. Or only one channel is activated, whereby the full computing power and memory is available for this channel only. The operating system permits also that different programs can operate at the same time on the same channel(s).

When the power is switched on, a **self-test program** runs; this program varies according to the particular mode of application. It may take up some time (seconds) to complete this, at the end of which the LED's indicate readiness to operate.

A **measurement (start/stop) switch** must be **closed** to **start** the data tracking. Normally the power-on do not coincide with start of measurement. This decoupling can be useful because, for example, in an aircraft recording should only be performed during flight (landing gear switch closed). During measurement the signal channels are continuously sampled with 128 samples per channel per second (or more). The timeseries are stored in the main memory, where they are evaluated by **appropriate algorithms**. Programs like SPV and RTRF are tailored to material fatigue analysis. This is essentially a question of **detecting, counting and classifying** load changes (double amplitudes). It can also be applied in other fields. Other programs like, e.g. 'TAL' joint time at level classification, 'RAW' data acquisition, 'FFT' frequency analysis, etc ... can be used in different applications.

Many different measurements can be stored in one Memory Cassette. The **start** and the **end** of each measurement is logged precisely and entered with the time and date. Between these fixed events, user selected **time marks** can be recorded. The period until the memory is full depends on the measurement configuration, how many channels are activated, the algorithms used and the degree of data reduction on the Memory Cassette. In the RTRF, or TAL program, the measurement duration is **practically unlimited** because it counts different events in a histogram. The maximum recording time of the SPV or TAL program is also influenced by the selected hysteresis and of course, for SPV, by the variability of the measuring signal itself.

From time to time, the **Memory Cassette** is taken out and transferred to the SPECTRAPOT Data Processor. Here are the programs for accepting and storing the preprocessed datafiles on **floppy disk** or harddisk, and for the final evaluation and display of measurement listings. The Memory Cassette can be re-initialized after this. During initialization, the possible updated program configuration is loaded again into the Data Collector via cassettes. The **program update** can be carried out centrally and by a single specialist. The assistant staff must not know anything about the individual Data Collectors. The Data Collector does **not** have to be disassembled in order to carry out the update, as this is with EPROM's.

It is also possible to check or download the results on site the aircraft via the serial input/output interface of the Data Collector.

4. Application examples

Equipped with the classification programs Real Time Rainflow (RTRF) and Sequential Peak Valley (SPV), the SPECTRAPOT System mainly acquires individual data on load cycles of mechanical construction parts relevant to material fatigue. For this purpose, the Data Collectors and contingent preamplifiers, together with the necessary sensors, are installed in the aircraft or vehicle. As a rule, data on complete movements are acquired in statistically representative form,

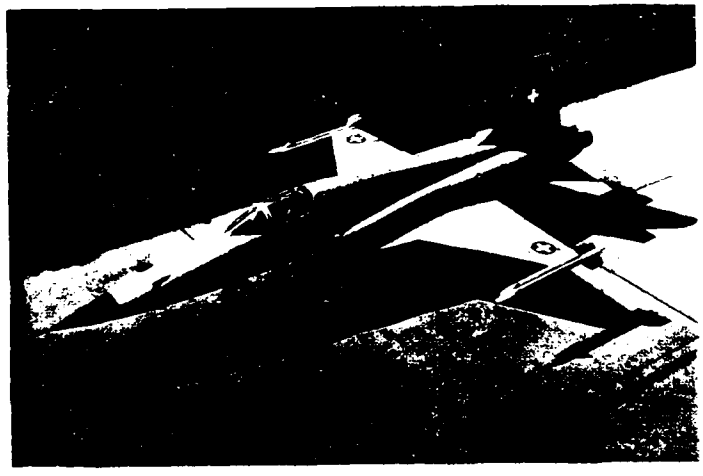


Fig. 4 The SPECTRAPOT-4C data collector has been installed in the «Tiger» aircraft of the Swiss Airforce. Thus fatigue load tracking und structure monitoring has been established at an excellent level in Switzerland.

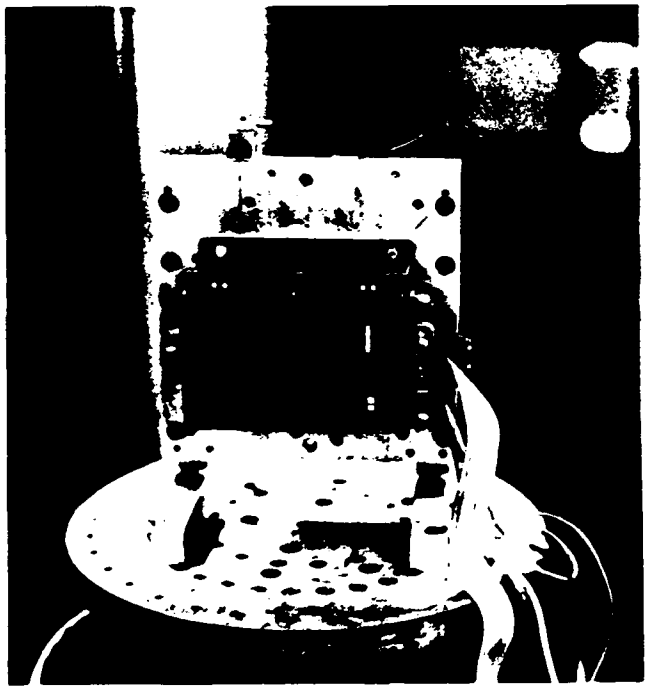
and with the help of interchangeable Memory Cassettes, collected in stationary Data Processors, where they are subject to further statistical processing.

In a well organized concept, it is possible to establish continuous fatigue monitoring, calculate fatigue indexes and prognose the lifetime for many aircrafts.

Alongside to this principal application for fatigue analysis, new fields with other evaluation programs (RAW, TAL, FFT, SPVC, ...) are investigated for the SPECTRAPOT System. All these fields of application could be placed under the general title of **'front-end computer system'** for service load tracking. Other applications are for example:

- Examination of driving behaviour and motor parameters for the further development of automobiles
- Quality check on vehicles of material testing plants
- Month-long monitoring of revolution figures, pressure and temperature in turbo chargers in ship diesel engines on the open sea or diesel locomotives.
- Examination of light weight railroad cars and pivot bogies in rail vehicles
- Spectral signature of runway textures by Fast Fourier transform
- etc ...

Paper with coloured overhead projection slides (25 minutes)



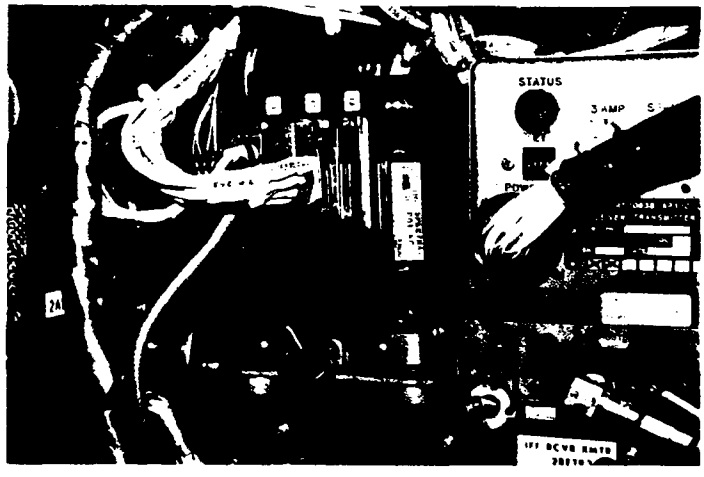


Fig. 5 Data collector in an extreme environment.

Figure 6: (left)
Vibration test with SPECTRAPOT Data Collector. Part of the tests to approved MIL-STD 810C

ASIP Concepts for the Standard Flight Data Recorder

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ASIP Concepts for the Standard Flight Data Recorder

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Abstract

Successful implementation of the USAF Aircraft Structural Integrity Program (ASIP) requires acquisition of critical aircraft parameter data via a flight recording system. Recent advances in solid state memory technology and microprocessor capabilities, coupled with innovations in data reduction/compression methods have made feasible the development of the Standard Flight Data Recorder (SFDR). Essentially, the SFDR is a microprocessor-driven solid state flight recorder which processes and compresses selected aircraft data inflight, to reduce both the volume of recorded data and the subsequent amount of data editing performed during ground processing. This approach presents a more reliable and cost effective alternative for collecting information than early flight recorder systems, which are limited by conservative sample rates and employ a magnetic tape recording medium which is often prone to failure in severe operating environments. The purpose of this paper is to provide an overview of the SFDR system - its functions, capabilities, and growth potential, with an emphasis on ASIP.

1 Introduction

The Aircraft Structural Integrity Program (ASIP) represents a methodology for predicting aircraft component life, fatigue, crack growth, and impending failure based on analysis of data collected during ground and flight operations [1]. The program consists of two elements: Individual Aircraft Tracking (IAT) and Loads/Environment Spectra Survey (L/ESS). Implementation of the ASIP methodology requires acquisition of pertinent aircraft parameter data via a flight recording system.

In 1958 the USAF undertook an initiative to determine the feasibility of monitoring the actual aircraft structure usage by a dynamic collection of key flight parameters. Prior to this time, the aircraft structural analysis was based primarily on component inspection and life expectancy predictions (i.e., on the basis of theoretical data derived from simplistic simulation models). The USAF theorized that analysis of actual flight loads data would provide for early detection of worn critical components, thus allowing for timely maintenance actions (repair and/or replacement). Such preventative maintenance would potentially save the service millions of dollars by reducing the number of aircraft lost due to component failure in flight, as well as minimizing the number of aircrew fatalities. The results of the initiative produced the Aircraft Structural Integrity Program

Since the inception of ASIP, the USAF has procured and utilized a large collection of flight data recorders to fulfill the need to provide on-board flight loads data collection. Typically, separate recorders have been used for IAT and L/ESS data collection. These recorders consist of the simple VGH-counter design, IAT G (normal acceleration) counters, mechanical strain recorders, tape recorders, and flight logs. Although these devices produce valuable data for fleetwide structural analysis, they also possess some inherent limitations. Most notably:

- Large quantities of non-pertinent data are recorded. Consequently, the data is filtered during the ground processing stage of analysis.
- Extensive ground processing support is required. Ground processing performs such tasks as data reformatting/transcribing, editing and validity checks, non-pertinent data filtering.
- Data sample rates are non-programmable. Structural analysis personnel are unable to alter the sample rates when empirical data dictates a change.
- Mechanical and electro-mechanical devices are prone to high failure rates in the dynamic environment of military aircraft.
- Inconvenient recording media are employed. Commonly used media are foil cartridges, tape cartridges, handfilled flight logs, punch cards, and records. This burdens the personnel at both the base level and ASIMIS.
- Airborne equipment costs are high. The typical costs of electro-mechanical systems grow quickly due to the rugged design requirements imposed by the operating environment, the costs of frequent maintenance, and the associated costs of supplemental aircraft equipment (converter-multiplexers, stain gage amplifiers, pulse rate integrators).
- Limited duty applications. Multiple recorders are required for tasks such as mishap reconstruction, IAT, L/ESS, and ENSIP.

Initially, the recorder procurement effort was characterized by designing a recorder for each unique aircraft (SDRS for the F-111, MADARS for the C-5, etc.). As procurement costs for military electronics systems escalated, the USAF acquired a generic electro-mechanical recorder system (MXU-553) which found application on at least 13 different existing aircraft types (including bomber, cargo, fighter, tanker, trainer). However, while this effort was primarily a retrofit program, the USAF continued to procure uniquely designed recorders for new weapon systems.

A recent addition to the Air Force recording equipment inventory is the Standard Flight Data Recorder (SFDR). Undergoing development at the SLI Avionics Systems Corporation (SLIASC) of Smiths' Industries since 1983, the SFDR has the potential and the capabilities to overcome the limitations of the previous generation of recording systems. In the following section, the significant features and flexibilities of the SFDR are presented.

2 SFDR System

In the early 1980's the USAF Aeronautical Systems Division (ASD) helped construct a Department of Defense Tri-Service Specification for a Standard Flight Data Recorder. The intent of this specification is to define a new generation standardized solid state data recorder system applicable to all military branches.

ASD developed a more specific version of this specification in 1987 to procure the SFDR for 16 different USAF aircraft. This standardized recorder has a lower acquisition cost and a much higher reliability than existing electro-mechanical flight recorders which utilize magnetic or foil tape recording media. It is capable of recording mishap, structural, and engine data and is flexible for many future data recording needs.

2.1 System Description

The SFDR is a fully solid state airborne data recording system with no moving parts and which requires no alignment or calibration. The system is much smaller than the tape recording systems it replaces, fitting easily into existing data recorder mounting racks. It is designed to record several types of data. Among them:

Mishap data: Data which aids in the investigation of a mishap. This data is recorded into a medium which withstands the rigors of an aircraft crash.

Individual Aircraft Tracking (IAT) data: Data which tracks the structural history of an individual aircraft and which is used to determine maintenance intervals of that aircraft.

Loads/Environment Spectra Survey (L/ESS) data: Strategic structural parameter data used to identify trends within the aircraft fleet. This data is processed in such a manner to support the analysis of IAT data.

Engine data: Data which tracks the performance of the aircraft's engines.

Three Line Replaceable Units (LRUs) are used to configure the various airborne SFDR systems. Figure 1 shows the interconnection of these LRUs. Depending on the data recording requirements and interface characteristics of a particular application, all three of these LRUs may not be required.

The Signal Acquisition Unit (SAU) is the major LRU within the system and is necessary for all possible SFDR configurations. It acquires raw mishap, structural, and engine parameters from the various aircraft sensors and the MIL-STD-1553B MUX bus, if one is available. The SAU then performs data reasonableness tests on the acquired data and compresses this data using various data compression techniques unique to the type of recorded data.

Compressed mishap data is transmitted over an RS-422 serial channel to a Crash Survivable Memory Unit (CSMU). The CSMU is a specially hardened unit designed to protect its recorded data from destruction in the event of an aircraft crash.

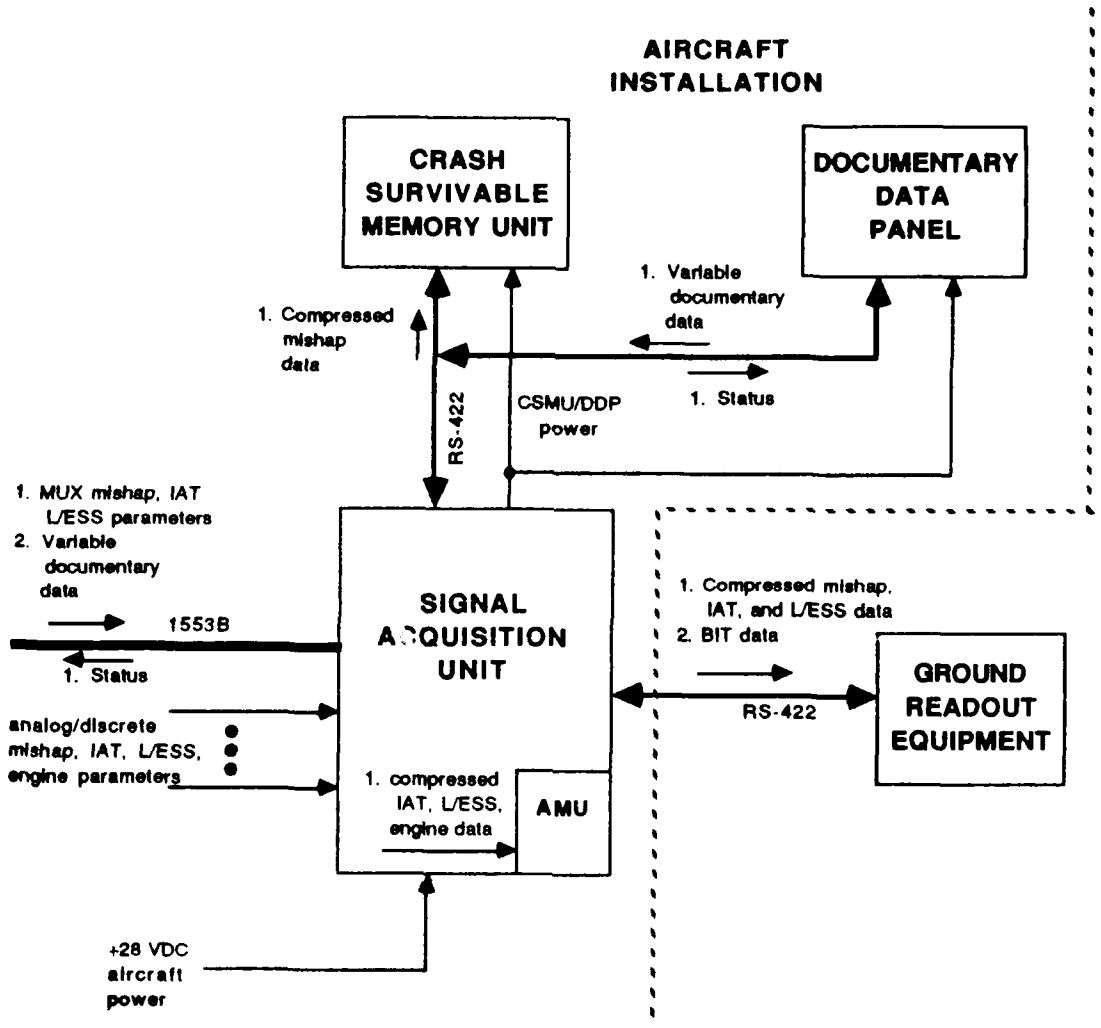

Figure 1: SFDR Airborne System

Compressed IAT, L/ESS, and Engine data are recorded into an Auxiliary Memory Unit (AMU) which resides within the SAU. The AMU contains up to 1M word of EEPROM memory, in increments of 256K words.

After the AMU memory is filled with recorded data, a special ground readout equipment (GRE) device is used to electrically communicate with the SAU. Compressed data from the AMU and CSMU is then downloaded over an RS-422 serial channel into the GRE and stored on $3\frac{1}{2}$ inch floppy disks. These disks are sent to the ASIMIS facility at OC-ALC for further ground processing.

The Documentary Data Panel (DDP) transmits aircraft and mission specific variable documentary data (such as aircraft gross weight and calendar date) to the SAU for ASIP recording purposes in those cases where the MIL-STD-1553B multiplexed data bus is not used for this function. This variable documentary data is manually entered by ground personnel prior to the flight. The DDP also displays SFDR system status and allows manual initiation of recorder BIT.

The SFDR system possesses a minimum acceptable Mean Time Between Failures (MTBF) of 1500 hours and a predicted MTBF of 4500 hours. This high reliability feature permits a two level maintenance concept (organizational and depot), thus eliminating the intermediate level of maintenance.

2.2 System Flexibility

The key feature of the SFDR is its flexibility; both in hardware and software. For example, hardware flexibility is inherent in the system design. Only a minimal amount of hardware differences between all the Air Force SFDR systems exists. AMU memory capacity is configured between 256K and 1M word in 256K word increments. Further growth is possible when higher density EEPROM memory devices become available.

The software flexibility revolves around an Operational Flight Program (OFP) which programs a 1750A microprocessor, the computing heart of the SAU. Each aircraft application possesses a unique OFP which controls the entire operation of the SFDR including:

- Parameter acquisition (including sampling rates)
- Data reasonableness checks
- Data compression
- Data storage
- Built in test

This software flexibility allows common hardware to be utilized across many applications by simply loading the specific OFP.

The major advantage of the SFDR over electro-mechanical tape recording systems is in data compression. While tape recording systems record a tremendous quantity of data, the data which is recorded is raw and uncompressed. Time consuming data reasonableness checks and data compression must then be performed at the ground processing facilities.

The SFDR, on the other hand, performs data reasonableness checks and data compression during flight, in real time, and only records the pertinent information necessary for further data analysis. This enables the use of solid state memory, thus lowering the acquisition and maintenance costs in comparison to tape recording systems.

Recording compressed data also requires less ground data processing to produce the final engineering results. Data compression techniques are tailored to each individual aircraft application, and thus support many different structural recording philosophies. Finally, archiving data in a compressed format helps alleviate the evergrowing problem of long term data storage.

3 Data Processing Concepts

There are a host of tradeoffs and constraints which come into play when considering data processing requirements for a standard recorder Some of these are

- Microprocessor capability and attributes
- Software considerations
- Recording memory size
- Overall system costs

In addition, each aircraft is unique with respect to its design, purpose, and function Fortunately, data compression plays a major role in permitting the SFDR to accommodate the diverse cross-section of aircraft applications The next paragraphs describe the data compression methods within the capabilities of the SFDR that are applicable to ASIP

3.1 Data Compression

Data compression techniques are primarily employed to reduce the volume of recorded data This is accomplished by identifying and retaining the pertinent signal features and eliminating the unnecessary elements Besides this, compression techniques tend to reduce: mass memory size and costs: the level of ground processing: and the amount of corrupted data, when used in conjunction with data reasonableness checking

In the context presented here, phrases such as data reduction, data condensation, data editing, and data compaction are used synonomously with data compression The key data compression methods useful for ASIP purposes are as follows:

Compressed Time History Selected data (typically from those parameters which depict the flight profile versus time) are passed through a zero order predictor algorithm, and the data are designated for recording only if a threshold is exceeded The recording period and data content are selected to insure that each parameter's time history is reconstructed to the limits of its particular acquisition rate and recording resolution specifications

Occurrence Histograms Occurrence histograms are matrices composed of counting information The counts may represent the occurrence of an event or elapsed time Update of a particular element in an occurrence matrix is initiated by a pre-defined condition such as a peak or valley detection, or the transition of parameters from one operating region to another Some typical occurrence histograms are Time-at-Mach-Altitude Matrices, From/To Usage Matrices, and Overload Tables

Peak-Valley Time History Those parameters designated for peak-counted recording are subject to a peak/valley search algorithm. This compression technique detects the local peaks and valleys based on the sign reversal of the parameter's slope, and further reduces the data to condensed local peaks and valleys by applying a threshold criterion. Usual parameters reduced in this manner are strains, accelerations and control surface positions.

For many of the attack/fighter/trainer aircraft applications, a time-hack (which consists of the simultaneous value of a subset of all parameters) is recorded at the time of occurrence of a condensed local peak or valley. With these applications, the number of condensed peaks/valleys is on the order of a few hundred per flight hour, which fits well within the SFDR solid state memory capacity. Furthermore, the more intense data analysis for ASIP purposes is performed postflight on the decompressed data. Typically, this analysis is based upon the recorded time-hack information, and a careful mix of occurrence histogram and compressed time history data.

Conversely, for some of the larger bomber/tanker/transport aircraft, the number of condensed local peaks/valleys is on the order of several thousand per flight hour, and additional reduction techniques are employed to further eliminate unnecessary data prior to recording. Extensive time-hack information is often not required for the strain, acceleration, and surface position local peaks/valleys. However, other quantities are computed to accompany the local condensed peaks and valleys and to aid in the second phase of data reduction. Generally, these are:

- **Mission Segments**
  The aircraft operates in one and only one mission segment at a time. By logical operations on flight parameters, the start and stop times of specific mission segments are determined.

- **Calculated Mean Level**
  The slowly varying mean level for each measured peak-counted parameter is computed. This mean level algorithm is reset at noncrossable boundaries, such as mission segment changes.

- **Gust-Maneuver Response Indication**
  Counted peaks and valleys are separated into those resulting from gust encounters and those due to a maneuver by a simple logic test performed on the NZCG parameter.

- **Breakpoint Determination**
  The breakpoint determination process accounts for major shifts in the mean due to extended maneuvers within selected mission segments.

Further data compression is accomplished for the larger aircraft by refined peak processing. That is, the local peaks/valleys identified within the dead band about the calculated mean level are discarded, and all consecutive local peaks/valleys that are outside of the dead band are eliminated, except for the maximum peak (or minimum valley). This reduces the volume of data to be recorded while still maintaining the pertinent information for further ASIP analysis.

It is possible to perform a third level of flight data reduction. This processing involves the onboard generation of cumulative occurrence exceedance spectra. Using this approach, a peak-counted primary spectra is accumulated as a function of mission segment, altitude

band, peak valley trigger parameter, and gust maneuver indicator. In addition, a peak-counted auxiliary spectra is accumulated during breakpoints.

It is noted that spectra generation may be performed on the ground during decompression since all of the necessary information is stored in memory. In this way, the onboard microprocessor workload is reduced, thereby providing an opportunity for a greater level of data reasonableness checking. Overall consideration is given to the increase in recording efficiency (i.e., total flight hours stored in memory with/without spectra generation) versus any additional system costs (and reduced benefits) due to processing implementation.

Besides the above compression techniques, other elements in the recorder system influence performance and efficiency, and therefore require further design consideration. For example, time history data designated for recording is generally assigned an identification tag, and decisions are made about the number of bits to record, and whether distinct parameters should share the same word space. Also, it is possible to perform certain binary data compaction operations on the bit stream to make efficient use of the storage space. These and other issues figure into the overall recorder system design. More detail may be found in Reference [2].

Finally, once the compressed data is downloaded onto a floppy disk, it is ready to be decompressed via a ground-based computer. That is, data decompression algorithms are employed to reconstruct the desired data sequences. For each compression method employed, there is a corresponding decompression technique. These software algorithms transform the compressed data into engineering unit data files, which are then suitable for supporting more extensive data processing and analysis.

3.2 Data Reasonableness Checks

With regards to the recording function, data reasonableness checking refers to interrogating the output from sensors and/or aircraft subsystems in order to determine the validity of the information. Reasonableness testing tends to:

- Increase the detection of sensor failures
- Enhance the ability of maintenance personnel to identify and correct problems in a timely fashion
- Improve the overall quality of the recorded data

The SFDR may perform a variety of reasonableness tests, several of which are listed below.

BIT Flag Test: Digital data (received via the multiplex data bus) is checked by examining the appropriate BIT status words.

Limit Test: Selected parameters are checked by comparing them to given maximum and minimum values.

Activity Test: Data is be checked by determining if it has changed by at least a minimum amount, or a minimum number of times

Rate of Change Test: Selected parameters are checked by comparing the rate of change against a maximum acceptable value

Chatter Test: Certain data is tested to determine if during a given time frame, it has changed more than a maximum number of times

The SFDR is capable of following several courses of action when a parameter fails one or more reasonableness tests For example, if a critical parameter fails a test for a pre-determined consecutive number of times, the SFDR may be instructed to: replace the bad data with the last good value and continue recording; record the bad value and attach a discrete parameter within the compressed datastream indicating a failure; record the bad data and place the failure information in BIT status; or stop recording altogether. The exact option selected is based upon a user's preference and overall philosophy for data recording

As a minimum, for reasonableness testing to be useful, the results of the tests must be recorded and made available to the appropriate systems and personnel. The sensor failure annunciation may be embedded in the compressed flight data, or via the DDP lamps and/or 1553 MUX Bus In addition, it is desirable to keep a record of the bad data for the maintenance personnel to see under what conditions the system failed (i.e., failure environment recording)

Additional tests and background material may be found in Reference [3].

4 Current and Advanced Applications

The SFDR is being utilized for both present and advanced multifunction data recording applications Several of the applications are highlighted in this section.

CSFDR The Crash Survivable Flight Data Recorder (CSFDR) represents the earliest version of the SFDR Development of the CSFDR for application to the F-16 C/D aircraft commenced in January, 1984 To date, over 600 units have been built and delivered.

ACTES In August 1984, SLIASC was chosen to adapt the CSFDR to host an embedded training program for USAF attack/fighter aircraft. This program, known as Air Combat Training & Evaluation System is an aircrew debriefing system which permits accurate reconstruction of formation flight maneuvers.

SFDR SLIASC was awarded a contract in July 1988 to supply the SFDR to the USAF for application to sixteen different aircraft. Many of the ASIP data concepts presented in this paper are under consideration for these applications (c.f., Reference [4]).

EDS The primary goal of the Enhanced Diagnostic System (EDS) is to eliminate inspections, and thereby schedule maintenance actions based on usage rather than flight hours or calendar date

GCAS The purpose of the Ground Collision Avoidance System (GCAS) is to provide visual and aural warnings to flight crew members Also, GCAS enhances radar altimeter inputs and uses multiple modes of operation for various flight conditions

Other The SFDR is being considered for use as an Engineering Development Tool (EDT) In the area of ground testing, the SFDR may be used to: record digital flight control system response for software model analysis; record engine trim data for engine run analysis; or debug avionics integration and MIL-STD-1553 data bus controller software Possibilities also exist in the flight test environment, where validation of aircraft performance predictions may be made, and diagnosis of unpredicted flight dynamics anomalies may be studied and resolved

5 Summary

In this paper, basic features of the SFDR system are presented, with primary emphasis on the flexible nature of both hardware and software It is noted that data compression represents the main advantage of the SFDR over other recording systems Prevalent data compression techniques within the capabilities of the SFDR and applicable to ASIP are given Comments are made concerning the different levels of onboard data reduction which may be selected, particularly for the larger application aircraft A brief mention of data reasonableness checking and annunciating is offered, which serves to support the flexible character of the SFDR Finally, some current and advanced applications of the SFDR are cited, further demonstrating the versatility of this recorder

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F-15 AIRCRAFT INFORMATION RETRIEVAL SYSTEM (AIRS)

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ABSTRACT

The F-15 AIRS is a state-of-the-art information management system for the F-15 weapon system which greatly reduces the information collection workload of the ASIP engineer. It presents information in the form of easy-to-read reports, charts, and graphs. It is a centralized system allowing many users to share a common database.

AIRS employs a relational database management system, a database query language and a data dictionary to store and retrieve information. The majority of the system is written in FORTRAN with an interface to the database query languages. It is designed to operate on Tektronix color graphics terminals and uses a VAX 11/780 as the host computer. By definition, the system is turn-key but an expert mode has also been provided which allows the experienced user to rapidly access any of the programs in the system.

The system includes programs broken down into the following categories:
- AFTO Forms Data Processing
- G033B Interrogation
- Component Tracking
- Aircraft History
- Maintenance Requirements Review Board Brochure
- Service Aircraft Fatigue Estimate (SAFE) Report
- Signal Data Recorder (SDR) / Exceedance Counter
- System / Production Management

The system is currently 75% complete (38 of 49 programs in place and operational). Future directions include:
- increased sharing of data elements among the programs. This increased connectivity will reduce the database maintenance effort of the various users.
- application of Artificial Intelligence technology. This will include the development of several prototype expert systems to aid the ASIP engineer in making scheduling and technical decisions.

INTRODUCTION

Management of a fleet of military aircraft involves a huge amount of usage and maintenance data that must be available to the ASIP engineer. This information comes in many forms, some automated and some not. There is often conflicting information among the various sources. Sorting through the various data sources and resolving conflicts for the purpose of assembling useful and accurate information is often a very slow and painful task.

This WR-ALC based state-of-the-art information management system is user-friendly and menu driven with persons of minimal computer training in mind. It organizes and provides access to large amounts of usage and maintenance information. An interactive query capability allows users to obtain information not provided in the standard reports. A powerful "what if" capability is provided by several of the programs.

Systems are currently in place and operational for the C-130, C-141, and F-15 weapon systems. In addition to the local area network at Robins, dial-in access for remote users is provided.

SYSTEM OVERVIEW

The AIRS for the F-15 weapon system is currently under development. It includes programs broken down into the following categories:

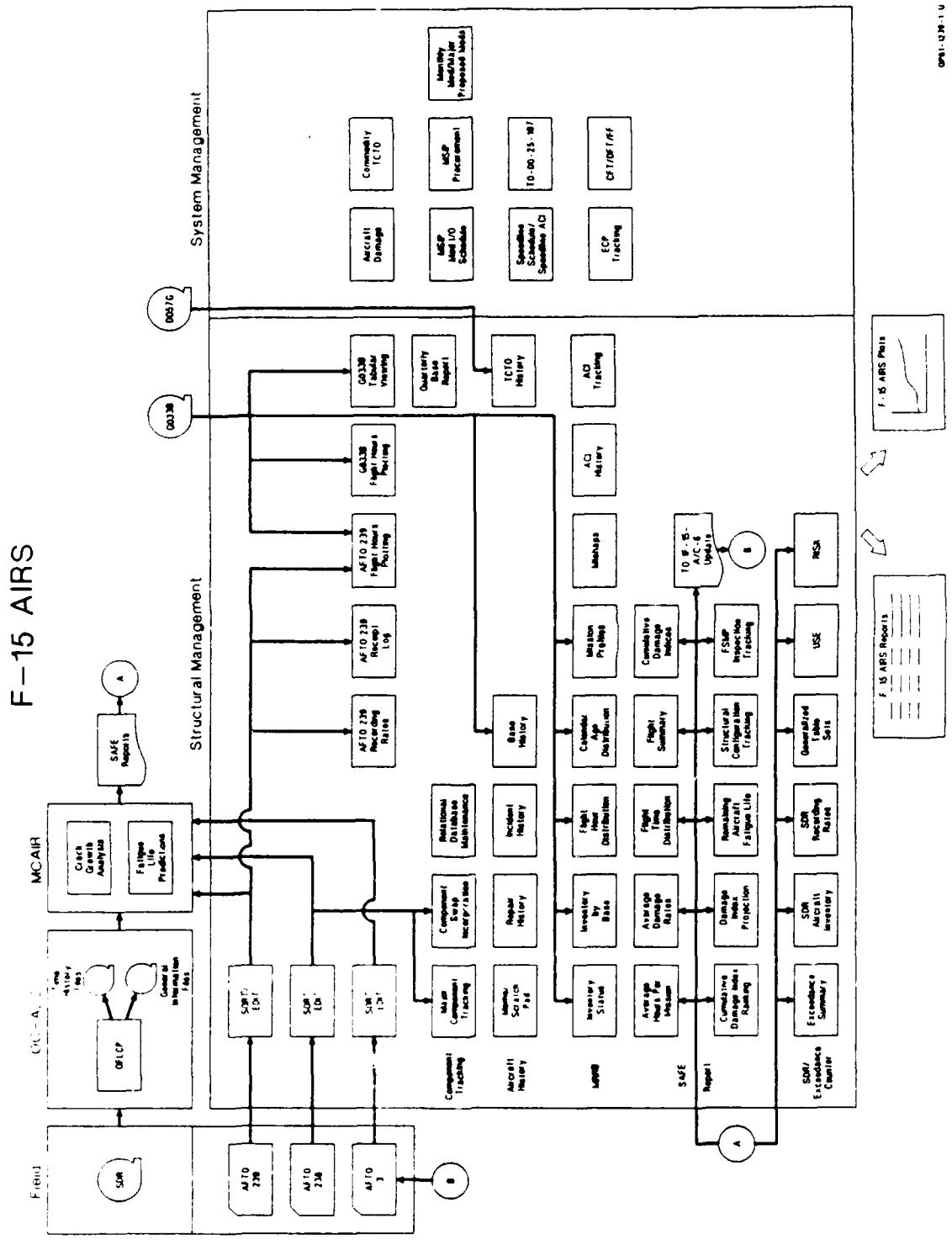
- AFTO Forms Data Processing
- G033B Interrogation
- Component Tracking
- Aircraft History
- Maintenance Requirements Review Board Brochure
- Service Aircraft Fatigue Estimate (SAFE) Report
- Signal Data Recorder (SDR) / Exceedance Counter
- System / Production Management

Figure 1 shows a system flow chart for the F-15 AIRS. It shows the great wealth of information that is accessible through the system. As the chart shows, automated data systems (G033B and D057G) are accessed by many of the programs in the system. Appendix I gives a brief description of the capabilities provided under each program.

Hardware used by the system includes a VAX 11/780 mainframe computer, Tektronix 4105 color graphics terminals, Tektronix 4696 color plotters, and Zenith 248 personal computers. Software includes Fortran 77, Rdb (Relational Database Management System), Datatrieve, CDD (Common Data Dictionary), FMS (Forms Management System), and Plot-10 Graphics.

The system employs a powerful relational database (Rdb) which serves as a common storage location for data elements common to many of the programs. Since it is a common storage location, updates and changes are

Figure 1 F-15 AIRS SYSTEM FLOW CHART



automatically reflected by all programs which access it. This greatly simplifies the database maintenance task. The relational database provides fast access to the desired data which may be easily queried, modified, or restructured. It has its own query language (Relational Data Manipulation Language) which may be used to "manually" interrogate the database without the aid of the application programs. This provides ultimate flexibility in terms of the combinations of data elements that may be collected and summarized to create non-standard reports.

Many of the application programs use Datatrieve, a fourth-generation data manipulation language, to store and retrieve data. It contains a powerful report writing facility which can access the relational database to create summary reports. Datatrieve also has an interactive query language which allows the user to access the relational database without the aid of the application programs to create non-standard reports.

The Common Data Dictionary (CDD) acts as a common storage location for the definitions of records, domains, indices, and relations used by the database. It may be used to modify data definitions without having to alter programs which use the definitions. It also allows the system manager to establish system security by assigning access privileges.

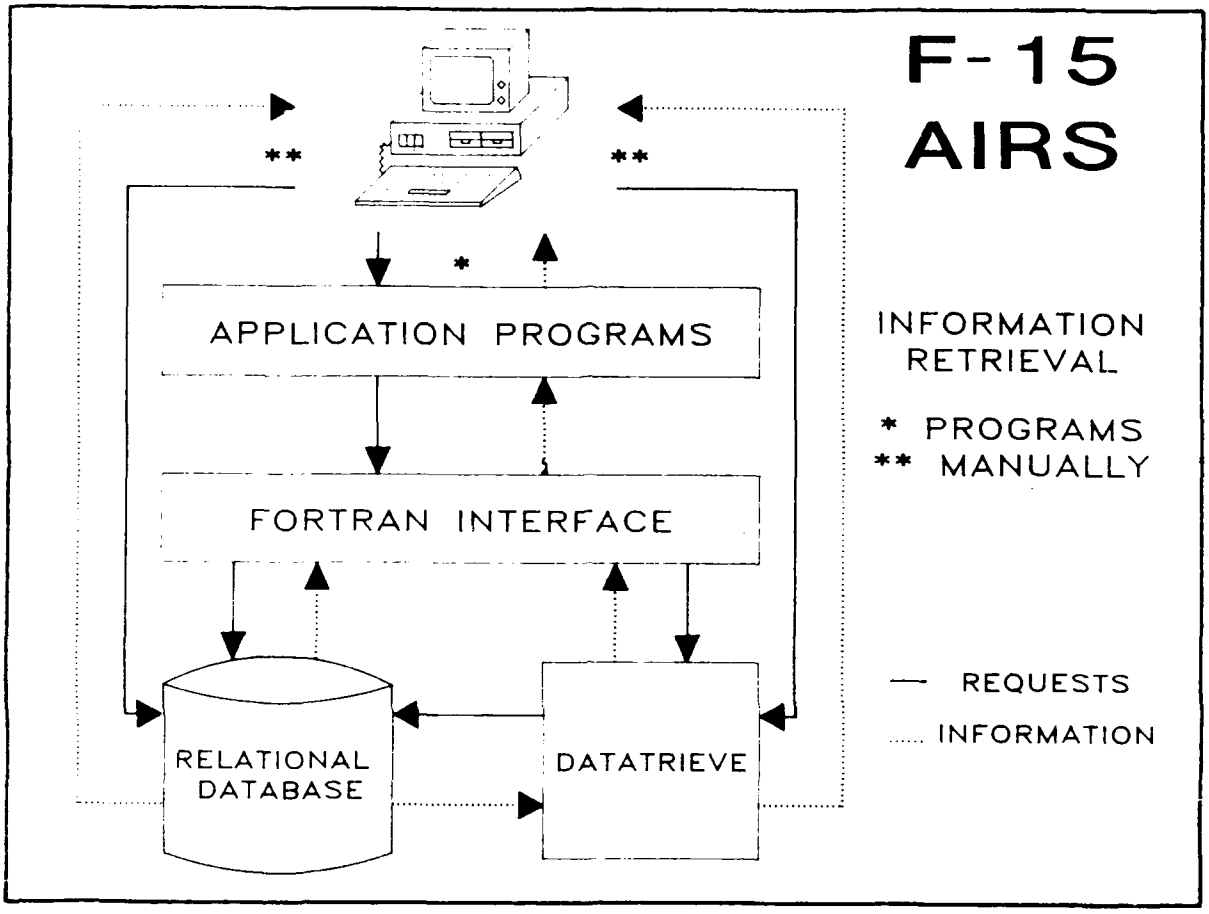
All of the programs use the Forms Management System (FMS) to prompt the user to select program options, input data, input report generation parameters, and display resulting information on the screen. It allows the developer to design the display format by specifying reverse video, bold, blinking, underlined, double size, and/or double height characters. It also provides for the use of scrolled regions on the screen so that many lines of data may be viewed without changing screens. It also includes an input validation capability and help screens.

All graphical reports provided by the system are programmed with the Plot-10 Graphics software. All are done in color to enhance the visual presentation and highlight trends.

An expert mode allows the experienced user to execute any of the programs directly without running through the multi-level menu structure. Once this has been mastered, the user may rapidly move about the system to obtain desired information.

Figure 2 shows how requests for information flow from the user to the database, and how the resulting information flows back to the user. The application programs allow the novice user to access information in a structured and well documented manner. The figure also shows how the experienced user may bypass the application programs and query the database directly. While this method provides much more flexibility in terms of the types of reports that may be created, it requires a thorough knowledge of the structure and content of the database.

Figure 2 F-15 AIRS INFORMATION FLOW



PROGRAM STATUS

Figure 3 shows the current schedule for the program. The development task has been broken down into five phases as shown. The first four are currently under contract and awarding of the contract for the final phase is imminent.

Figure 4 shows the current program status. To date, 38 of the 49 programs (about 75%) are in place and operational. A brief description of each program is given in Appendix I.

Figure 3 F-15 AIRS PROGRAM SCHEDULE

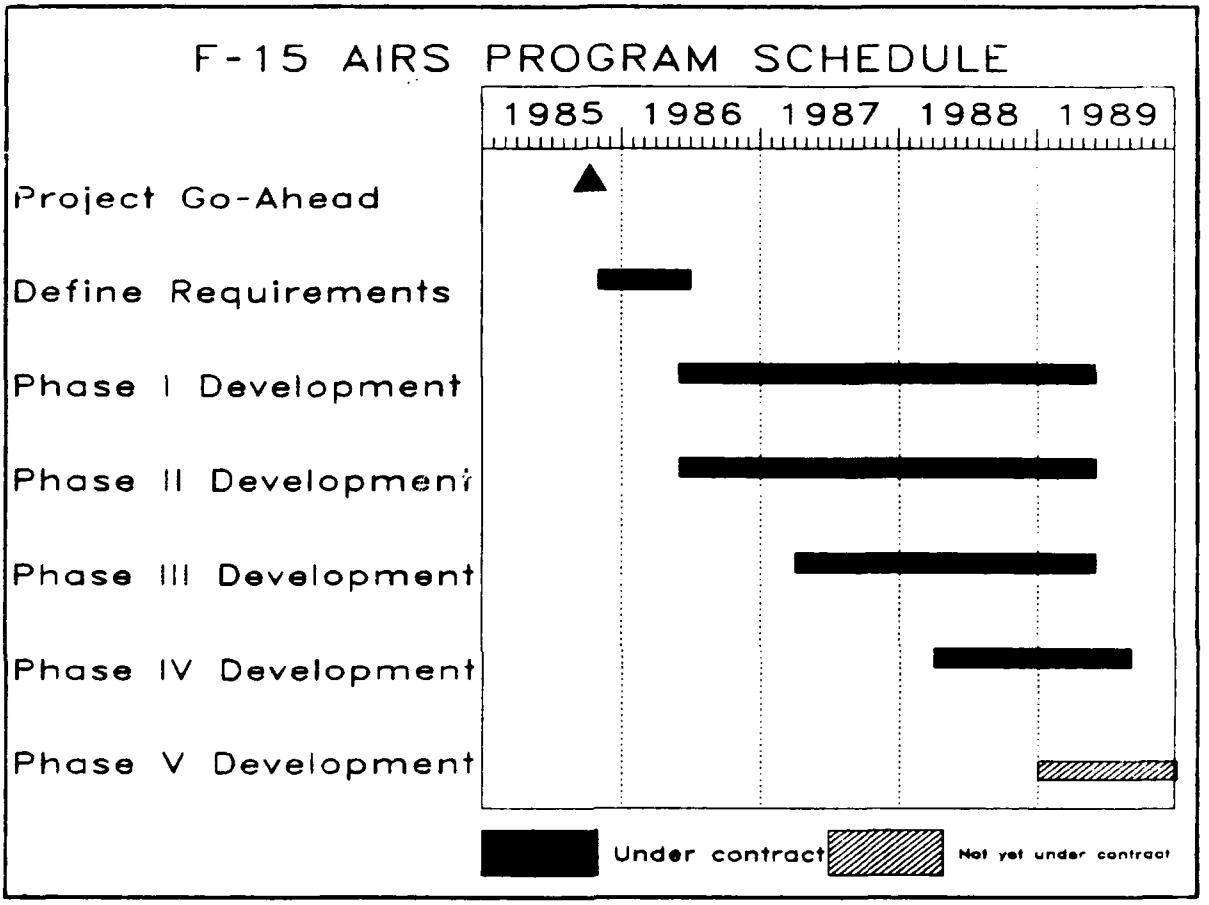


Figure 4 F-15 AIRS IMPLEMENTATION STATUS

F-15 AIRS SOFTWARE REQUIREMENTS BY PHASE

	STRUCTURAL MANAGEMENT	SYSTEM MANAGEMENT
PHASE I	* AFTO 3 SORT/EDIT	* AIRCRAFT DAMAGE REPORT
	* AFTO 238 SORT/EDIT	* COMMODITY TCTO REPORT
	* AFTO 239 SORT/EDIT	
	* AFTO 239 RECEIPT LOG	
	* AFTO 239 VS GO33B FLIGHT HOURS	
	* GO33B FLIGHT HOURS PLOTTING	
	* GO33B TABULAR DATA VIEWING	
	QUARTERLY BASE REPORT	
PHASE II	* MAJOR COMPONENT TRACKING	* MSIP MOD I/O SCHEDULE
	* COMPONENT SWAP INCORPORATION	* MSIP PROCUREMENT
	* RELATIONAL DATABASE MAINTENANCE	* MONTHLY MOD / MAJOR PROPOSED MODS
	* MEMO / SCRATCH PAD	* SPEEDLINE SCHEDULE / SPEEDLINE ACI
	* REPAIR HISTORY	* T.O.-00-25-107
	* INCIDENT HISTORY	
	* BASE HISTORY	
	* STRUCTURAL CONFIGURATION TRACKING	
	FSMP INSPECTION TRACKING	
PHASE III	* INVENTORY STATUS	* ECP TRACKING
	* INVENTORY BY BASE	* CFT/DFT/FF
	* FLIGHT HOUR DISTRIBUTION	
	* CALENDAR AGE DISTRIBUTION	
	* MISSION PROFILES	
	* MISHAPS	
	* ACI HISTORY	
	* ACI TRACKING	
	* EXCEEDANCE SUMMARY	
	* SDR AIRCRAFT INVENTORY	
	* SDR RECORDING RATES	
	* AFTO 239 RECORDING RATES	
	TCTO HISTORY	
PHASE IV	* AVERAGE HOURS PER MISSION	
	AVERAGE DAMAGE RATES	
	* FLIGHT TIME DISTRIBUTION	
	* FLIGHT SUMMARY	
	CUMULATIVE DAMAGE INDICES	
	CUMULATIVE DAMAGE INDEX RANKING	
	DAMAGE INDEX PROJECTION	
	REMAINING AIRCRAFT FATIGUE LIFE	
PHASE V	GENERALIZED TABLE SETS	
	USAGE SIMULATION AND EVALUATION (USE)	
	STRUCTURAL RISK ASSESSMENT	

NOTES: (1) * = PROGRAM HAS BEEN DELIVERED
(2) SEE APPENDIX I FOR A BRIEF DESCRIPTION OF EACH PROGRAM

LESSONS LEARNED

Several important lessons were learned in the process of designing and developing the F-15 AIRS. They are as follows:

1. Spend a sufficient amount of time to design the system. The key word here is sufficient. How does one know how much time is sufficient? No one knows. but a good guideline is that when you think you've done enough work here, you're only about two thirds of the way there. Spend about 50% more time to refine the requirements and integrate the system as much as possible up front.

Another point here is that a task group should be formed to develop requirements on the customer's side. Ideal qualifications for this group include a thorough knowledge of both the applications and the capabilities of the computer system.

2. When estimating and scheduling the job, allow for revisions and enhancements to the software after it is delivered. This is necessitated by the lack of communication that often exists between the end-user and the developer. Also, when the program is delivered and demonstrated, seeing what can be done stirs ideas for new capabilities which expand on the original requirements of the program.

3. Develop thorough test plans. We ran into several problems with incompatibilities with the computer systems. Also, the potential for well-hidden bugs always exists and a thorough test plan will likely uncover it.

The system we devised for implementing and testing is as follows:

a) Each program will be delivered and demonstrated to the appropriate user(s).
b) A two-month trial period follows delivery during which the programs are used and critiqued. At the completion of two months, the desired enhancements are discussed and agreed upon between the user and developer.
c) The changes are then made and at six months from the initial delivery, the final version of the program and its documentation are delivered.

At times, we found the two month period to be inadequate to fully test and critique the program. Also, requirements change over time. Although the programs provided what was initially agreed upon, some did not satisfy the needs of the user.

4. Develop an on-going implementation plan. This is a critical stepping stone to the system's long term success. It needs spell out specifics about responsibilities for maintaining the system. It is important to keep it up to date as the personnel situation changes.

5. System integration is sometimes best performed after the programs are in place and operational. This is especially true in the case where there are very few people who understand the "big picture". In this case, the expertise to properly design and integrate parts of the system

up front was just not available. Some programs were developed as separate entities with the intention to integrate them after they have been operational for some time.

6. Success in developing a large information management system such as this not only depends on the quality of the programs but perhaps more on the commitment to use and maintain the system. The most sophisticated programs in the world will not compensate for an obsolete data source.

FUTURE DIRECTIONS

In the future we plan to further enhance the system's capabilities. It will continue to grow and remain flexible to meet the needs of the users. You will notice from Figure 4 that there are no new development tasks for the System Management area under Phases IV and V. The task to be accomplished there will be to further enhance and integrate the programs already delivered.

We will also pursue applications of Artificial Intelligence applications within the scope of ASIP information management. Two areas for AI prototype work have been selected and we are developing prototype expert systems in diagnostics and scheduling areas. We are very excited about the possibilities here and will have a detailed report in the future.

SUMMARY

The F-15 AIRS provides the user access to a great wealth of usage and maintenance data for the F-15 weapon system in a flexible and easy to use manner. It allows the user to monitor aircraft on a flight-by-flight basis as well as provide fleet management reports including a wide variety of information about past, present, and future aircraft activities.

The advent of AIRS has expanded the capabilities of the ASIP engineer and Production Management Specialist dramatically. It provides a single data source for many reports and reduces the time required to produce routine reports, thus allowing the individual to do a better, more thorough job at a lower cost to the government.

The possibilities for enhancing and expanding AIRS is limited only by one's imagination. Plans are currently in place for significant improvements to the system. Further enhancements will be made as funding is available.

The writers plan to make an update to this presentation at the 1989 ASIP Conference. By that time, installation of the current system will be nearly complete.

REFERENCES

1. Sullentrup, M.G., and McCall, M.R., F-15 AIRS Functional Description, February 1988 revision.

2. Sullentrup, M.G., and Beck, A.M., F-15 AIRS User's Manual, Volumes I and II, May 1988.

3. Scheidter, R.A., Sullentrup, M.G., and Grizzle, P.M., "F-15 AIRS System Overview", a presentation given at WR-ALC, 2 June 1987.

APPENDIX I

BRIEF DESCRIPTION OF F-15 AIRS SOFTWARE

PHASE I

AFTO 3 Sort/Edit	- Provides interactive error checking and data correction for FSMP Inspection Feedback data
AFTO 238 Sort/Edit	- Provides interactive error checking and data correction for Component Swap data
AFTO 239 Sort/Edit	- Provides interactive error checking and data correction for Flight Log and Exceedance Counter data
AFTO 239 Receipt Log	- Provides a log of all forms received by base and utilization period
AFTO 239 vs G033B Flight Hours	- Provides a graphical measure of the flight log data reporting rate by base
G033B Flight Hours Plotting	- Provides a graphical display of either cum-to-date or monthly flight hours by tail number
G033B Tabular Data Viewing	- Provides a tabular display of monthly and cumulative flight hour totals as well as squadron assignments and mission codes
Quarterly Base Report	- Provides a quarterly report to the individual bases including the number of forms received for the reporting period and data recording rates
Aircraft Damage Report	- Tracks labor, material, and related costs incurred for damaged aircraft
Commodity TCTO Report	- Provides cross reference of Commodity TCTOs with related aircraft TCTOs

PHASE II

Major Component Tracking	- Provides cross reference of tail numbers and serialized components and also component swap history data

PHASE II (continued)

Component Swap Incorporation	- Provides component swap validation for data received on AFTO Form 238
Relational Database Maintenance	- Provides capability to add new aircraft and squadrons to the fleet as well as edit incorrect entries in the database
Memo / Scratch Pad	- Provides capability to store general information by tail number
Repair History	- Provides capability to track the history of structural repairs by tail number
Incident History	- Provides capability to maintain incident history information (Password protected)
Base History	- Provides capability to view base history of an aircraft or aircraft history of a base
Structural Configuration Tracking	- Provides capability to monitor structural configuration by tail number and tracked location
FSMP Inspection Tracking	- Provides capability to view past, pending, and future inspections by tail number as well as produce fleet summary reports
MSIP Mod I/O Schedule	- Generates input and output schedule for MSIP retrofit based on input parameters and provides automated schedule maintenance
MSIP Procurement	- Determines the purchase ordering / procurement list of the components comprising the MSIP retrofit kit for each airplane
Monthly Mod / Major Proposed Mods	- Provides monthly production management reports on modifications made to the aircraft including graphical summaries
Speedline Schedule/Speedline ACI	- Provides capability to monitor Speedline input and output as well as retrofit/repair activities by tail number

PHASE II (continued)

T.O.-00-25-107	- Provides capability to track requests for repairs to damaged aircraft including cost tracking

PHASE III

Inventory Status	- Summarizes aircraft service status by model for base or fleet, including graphics
Inventory by Base	- Provides aircraft totals by command or base as well as a tail number inventory for a chosen time period, including graphics
Flight Hour Distribution	- Provides flight hour histogram for base, command, or fleet, as well as capability to search for aircraft within a chosen flight hour range, including graphics
Calendar Age Distribution	- Similar to Flight Hour Distribution but in terms of years, including graphics
Mission Profiles	- Provides breakdown of number of aircraft by mission profile for each command
Mishaps	- Provides capability to track Class A, B, and C mishaps by fiscal year for each of the aircraft systems, including graphics (Password protected)
ACI History	- Provides information regarding aircraft that have been selected for ACI
ACI Tracking	- Provides capability to monitor ACI results by tail number and task number
Exceedance Summary	- Provides capability to compare normalized exceedance curve for an aircraft to the base and fleet averages, including graphics

PHASE III (continued)

SDR Aircraft Inventory	- Provides capability to view SDR tail numbers by base and command, including graphics
SDR Recording Rates	- Provides capability to track SDR recording rates by command and reporting period, including graphics
AFTO 239 Recording Rates	- Provides capability to track flight log and exceedance counter recording rates by base and reporting period, including graphics
TCTO History	- Provides capability to monitor compliance and non-compliance listings by TCTO number and also provide TCTO history for an aircraft
ECP Tracking	- Provides capability to monitor Engineering Change Proposals (ECPs)
CFT/DFT/FF	- Provides capability to track TCTO compliance by method of accomplishment

PHASE IV

Average Hours Per Mission	- Provides capability to track average flight hours per mission by base and mission type, including graphics
Average Damage Rates	- Provides capability to monitor average fatigue damage accumulation rates by base and tracked location, including graphics
Flight Time Distribution	- Provides capability to monitor flight time distribution by base and mission type. including graphics
Flight Summary	- Similar to Flight Time Distribution but based on number of flights, including graphics
Cumulative Damage Indices	- Provides capability to monitor cumulative damage index by tail number and tracked location, including graphics

PHASE IV (continued)

Cumulative Damage Index Ranking	- Provides capability to rank tail numbers by cumulative damage index and tracked location
Damage Index Projection	- Provides capability to project future damage indices based on current damage index and current damage accumulation rate
Remaining Aircraft Fatigue Life	- Provides capability to estimate remaining aircraft fatigue life using several different methods

PHASE V

Generalized Table Sets	- Provides capability to view summary tables of various flight parameters (Mach, Altitude, Nz, Wing and Fuselage bending moments, etc.) by base
Usage Simulation and Evaluation	- Provides capability to estimate fatigue damage accumulation rates for proposed mission scenarios
Structural Risk Assessment	- Provi[illegible]apability to estimate probability of finding flaws based on past inspection results

*U.S.Government Printing Office: 1989 648 056/04506