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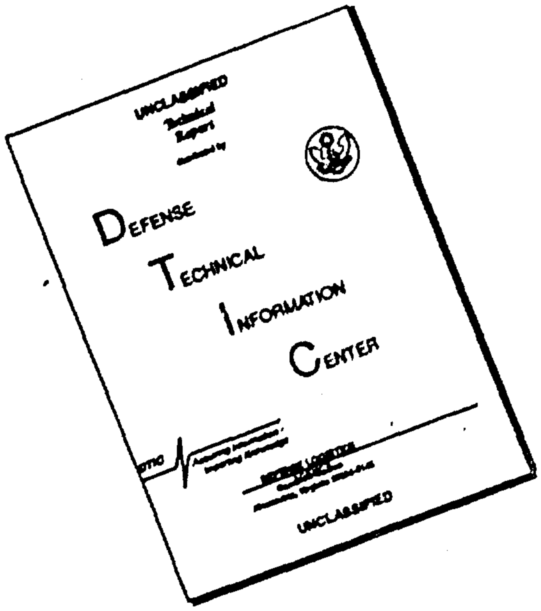

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1986 USAF STRUCTURAL INTEGRITY PROGRAM (ASIP, ENSIP)

2-4 December 1986

Capitol Plaza Holiday Inn
Sacramento, California

Sponsored by
Headquarters
Air Force Logistics Command

Hosted by
Sacramento Air Logistics Center
Aircraft System Management
Divisions, MMS and MMT

Chairman
Fred Chuang
HQ AFLC/MMEEP
Wright-Patterson AFB, OH
45433

P·R·O·C·E·E·D·I·N·G·S

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DIRECTORATE OF FLIGHT SYSTEMS ENGINEERING
AERONAUTICAL SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433-6503

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THIS TECHNICAL REPORT HAS BEEN REVIEWED AND IS APPROVED FOR PUBLICATION.

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1986
USAF
STRUCTURAL INTEGRITY
PROGRAM
(ASIP, ENSIP)

A FEW WORDS FROM CONFERENCE COMMITTEE

The 1986 US Air Force Structural Integrity Program (ASIP/ENSIP) Conference, hosted by Sacramento Air Logistics Center (SM-ALC), 2-4 December 1986, at the Capital Plaza Holiday Inn in downtown Sacramento, Califronia, was a success. The theme of this year's ASIP/ENSIP conference was "The Impacts of Nondestructive Inspection (NDI), Corrosion Prevention, and Advanced Composites Usage on Weapon Systems Structural Integrity." The theme was recommended by the Air Force General Officers' Steering Group for the Air Force NDI, Corrosion, and Advanced Composites Programs. We are thankful to the steering group members for this recommendation. Also, this is the first year we have officially incorporated the Engine Structural Integrity (ENSIP) Program with the ASIP conference. We appreciate the support from the engine community.

We thank Brigadier General James W. Hopp, Vice Commander, SM-ALC, for his opening remarks, and Mr. Earl W. Briesch, Assistant Deputy Chief of Staff for Material Management, Headquarters Air Force Logistics Command, for his remarkable keynote speech. We want to apologize to those who submitted presentation proposals which were not selected for presentation at the conference. In trying to hold the conference to three days, there was insufficient time available to present them all. We also want to thank each participant in this year's conference, especially those outside the US Air Force. The quality of the presentations were absolutely outstanding. We are looking forward to seeing you at next year's conference.

CONFERENCE COMMITTEE

Fred Chuang, Chairman Shiu-Nan Chuang
Jack Lincoln
Tom Cooper
Grover Hardy
Dave Bell, Secretary

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B. G. Hopp, Commander, Sacramento Air Logistics Center

Keynote Speaker
Mr. Earl W. Briesch, Assistant Deputy Chief of Staff, Material Management Headquarters, Air Force Logistics Command

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John W. Lincoln, ASD/ENFS

2. **Impact of Structural Integrity Program on Nondestructive Inspection**
Maj David J. Jirele, Air Force NDI Program Office

3. **Air Force Corrosion Control Program**
Lt Col Ronald C. Hoch, WR-ALC/MMEM

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FIRST SESSION

OVERVIEWS

Chairman

Michael Findley

SM-ALC/MMSR

Keynote Address To The Aircraft Structural Integrity Conference

By

Mr. Earl W. Briesch
Asst. Deputy Chief of Staff
For Material Management
HQ, Air Force Logistics Command

It's a real pleasure to be here this morning. Aircraft and engine structural integrity are certainly important links in maximizing aircraft availability and flight safety. I know each of you has an important role to play in aircraft readiness. Today, I hope I can share some of my thoughts with you on the importance of readiness issues in the Air Force mission. Certainly, these missions in maintaining the security of this country deserve the very best that each of us can contribute.

First, let me set the stage by sharing with you the magnitude of the challenge that we face today. Despite a significant and sustained build-up over the past six years, our forces still face a potential adversary that has been and continues outproducing us by almost any measure of military capability. For example, over the past twelve-year period, while we were building 3,500 new fighter/interceptor aircraft, the Soviets were building 8,400, outproducing us by a 2.4 to 1 ratio. Similar comparisons exist today in almost all categories of weapon systems.

Despite the statistics, there are already signs of erosion in the public support for the continuation of our defense build-up. The state of the economy and the Gramm-Rudman-Hollings Deficient Reduction Law make defense expenditures an inviting target for those who would ignore the threat. Clearly, the path to peace through deterrence is built on the foundation of a strong defense. This philosophy is as valid today as it was upon the founding of this republic, when president Washington observed that "To be prepared for war is the most effectual way for peace." Unfortunately, many in our society have lost sight of, or choose to ignore the wisdom in that observation.

I am pleased to report to you that our six-year defense build-up has not been wasted, and that we have made significant progress since 1980 in correcting many of the serious short falls that had accrued in our defense posture. The increased emphasis on readiness and sustainability has allowed us to improve aircraft mission capabilities--to significantly increase flying hours and to improve combat readiness through more realistic peacetime scenarios. Another very positive sign is the continued lowering of the Class A aircraft accident rate to its all time low in 1985. Today, with increased sustainability and availability, we can surge both tactical and airlift sorties to much higher levels, and can sustain those surges for much longer periods of time. Despite these improvements, challenges exists throughout the whole logistics spectrum. Our weapon systems-support requirements have never been greater. Today, we face some truly awesome tasks. One problem which should be of particular concern to this group is the growing age of our weapons systems. Our total Air Force aircraft inventory averages 16.5 years old. This figure is increasing by almost six months per year. Our strategic component of that force, the B-52 bomber, averages more than 25 years of age. Many of these aircraft were bought under the assumption that they would be replaced after ten years or so. Up until the 1960's, that was the pattern, with few aircraft remaining in service beyond a ten-year life. More recently, however, we have been unable to replace aircraft within these time frames and, as a result, have kept them in service much longer than anyone ever expected at the time they were designed or built. It is therefore, of paramount importance that we gear our aircraft structural integrity programs to continuing support of this aging force. We must fully exploit all available technological improvements to enhance readiness and sustainability of these forces.

I am certainly aware of the highly recognized successes which have occurred in the past few years in the field of aircraft structural integrity. Our shift from cumulative fatigue damage technology to durability and fracture mechanics technology has provided very significant, positive results for the Air Force. Certainly, continuation in service of the only outsize airlift capability, the C-5A, was made possible only through your development of advanced structural analysis and tracking techniques. This technology allowed us to control and conserve remaining service life of these unique and vital airlift resources until the wing modification program was in place. In a similar fashion, the durability and damage tolerance assessment technology, developed by the aircraft industry, provided the Air Force with the capability to conduct a detailed analytical review of the structural status of the C-141. The results of that analysis provided the necessary visibility to justify what otherwise would have been a very risky decision to proceed with the C-141B "stretch" modification. The continued successes attained by that program certainly attest to the validity of both the modification decision and the analysis techniques used in justifying that decision. Additionally, the Air Force is now routinely accomplishing critical structural modifications such as the one we accomplished on the F-4, following an extensive durability and damage tolerance analysis. This analysis accurately predicted widespread fatigue cracking in the lower torque box skin attachment to the main spar. Modification action was taken to cold work fastener holes and install taper lock fasteners in that area, thus ensuring structural integrity of that critical wing area until planned phase-out of those aircraft. This put us in a proactive, fatigue-life-extending mode, rather than our previous reactive mode of waiting for significant fatigue cracking to develop before taking necessary action.

I congratulate each of you for these and similar developments over the past decade. However, there is much that can be and should be done to improve Air Force weapon systems readiness. We need to continually review the lessons of the past and strive for usable technology developments in making further strides toward higher levels of readiness and sustainability. From my perspective, opportunities are unlimited for further significant advances in this field. Let's look at just a few areas which would appear to hold potential for additional improvements.

First, materials. Although we have made significant progress in this area, much remains to be done. We have developed new aluminum alloys with improved fracture toughness, and with reduced susceptibility to corrosion. We are seeing increased acceptance and more widespread usage of advanced composite materials in both secondary and primary aircraft structures. Many of our older aircraft, however, still suffer problems, such as stress corrosion, which severely reduce their readiness posture. Keep in mind that these are problems that were not envisioned at the time the aircraft were build, but which now are costing us dearly in terms of aircraft readiness today. Certainly, corrosion and its insidious effects constitute one of our major challenges to readiness in our aging aircraft. Newer, more effective and less costly methods and materials for controlling and preventing corrosion are absolutely essential in the near term.

One of the key elements in a successful aircraft structural integrity program is nondestructive inspection. Improvements have been and continue to be made in NDI technology. And, I believe, the downtimes required to accomplish NDI inspections can be further reduced. These downtimes adversely affect aircraft readiness. The complexity and subjectivity of many existing procedures require highly-skilled and experienced technicians. We must keep in mind that we may not always have the luxury of a laboratory environment in which to conduct our required NDI inspections. We should have our aircraft structural inspection program designed and ready for the wartime scenario. Our goal, therefore, is clear. We should strive for NDI techniques that require minimal training, are as near foolproof as possible, require minimal aircraft downtime to accomplish, and provide highly-accurate inspection results. Eddy current II and Retirement for Cause/NDI for engine components are examples of the NDI equipment designed along these lines. These improvements are absolutely necessary to improve aircraft readiness through reduction of inspection downtimes.

My final point concerns quality of workmanship during manufacture and repair. The science of fracture mechanics has taught us a great deal over the past several years about the origin and development of structural fatigue failures. It has been shown that clear correlation exists between quality characteristics of fastener holes and fatigue crack initiation. Even though much progress has been made in improving fastener hole workmanship in critical structure, I am convinced that more opportunities exists for further dramatic improvements that will minimize the problem of fatigue crack initiation from fastener holes. As a corollary to this development, we need better quality assurance techniques to reduce the size of the "rogue flaws" which drive too many inspections and are responsible for too high aircraft downtimes. These improvements, coupled with necessary advanced and wider acceptance of structural damage analysis techniques, will provide significant and much-needed enhancements to aircraft readiness throughout the Air Force.

The objectives of these challenges which lie before use are clear. Aircraft structural integrity support, particularly to fielded systems, is at present, a logistics burden which is too cumbersome to provide the necessary combat capability to out operational commanders. Therefore, we in the logistics business must exercise every opportunity to simplify that logistics support and thus improve combat capability. We must constantly strive for a logistics infrastructure that allows air power to pursue its greatest attribute - flexibility. Let us keep that thought firmly in mind as we design the aircraft of tomorrow, and as we develop new and improved techniques to deal with the problems of the aircraft today. The challenges are there. We must meet these challenges. We have come along way together. Let us press on to even greater achievements to do our part in ensuring the security of this great nation.

Again, it is a great pleasure to be with you today.

Impact of Structural Integrity Program On Nondestructive Inspection

By

Major D. Jirele
Air Force NDI Program Office

Impact of Structural Integrity Program on Nondestructive Inspection

Major D. Jirele

Air Force NDI Program Office

I. INTRODUCTION

This paper provides a broad overview of the Air Force Nondestructive Inspection (NDI) program as it is affected by the Aircraft and Engine Structural Integrity Program requirements. It briefly addresses what is NDI, and how we should work together to support the present ASIP/ENSIP technology thrusts.

II. DISCUSSION

First of all, Nondestructive Inspection (NDI) is the process of inspecting materials and components for the presence of flaws, discontinuities, and/or cracks without in any way altering or changing the material's properties or characteristics. Although numerous NDI methods exist, the major methods in use today are X-ray, Ultrasonics, Eddy Current, Magnetic Particle, and Fluorescent Penetrant. Each of these methods has been used extensively by the Air Force through a comprehensive NDI program for all major weapon systems, commodity items, and support equipment.

The Air Force NDI Program Office, located in the Engineering Division at the San Antonio Air Logistics Center (ALC), is responsible for managing and supporting all NDI activities throughout the Air Force. The office operates under the authority of AFR 66-38 (AF NDI Program) and is assisted by NDI managers at each major command, each AFSC product division, and at each of the five Air Logistics Center (ALCs).

Having presented this background information, let me proceed to the main objective of this presentation. NDI methods traditionally have been applied throughout industry as a method of quality control. In the Air Force, NDI is part of the maintenance function. Basically, we use it to inspect components for flaws/defects and provide a cost effective way of determining whether the part is to be repaired/rejected. If no flaw is detected the part is placed back into service. NDI is also used to verify that a repair has been properly completed, which is a process control function. Example, welds on engine oil coolers are x-rayed before repair to determine where defects are and afterward to verify the welds are repaired.

The present use of NDI extends beyond maintenance of the weapon system. It is now into extending the service life of aircraft and missiles beyond what was originally planned. Life extension has placed a greater responsibility on NDI to identify pending structural failures thereby preventing a catastropic fracture. These inspections are usually in limited access areas requiring special NDI fixtures for the detection of small defects. In some cases, we are inspecting to determine that small flaws are not present, thereby extending the life. This is clearly illustrated in the T-38 wing inspection. To extend the wing life until a new wing could replace it, an inspection of the lower wing skin fasteners was required. It is inspected using a special rotational ultrasonic scanning system. The inspection is for .100 inch cracks around fasteners.

Not finding this defect can lead to a catastrophic failure. This inspection provides for both life extension and safe flight. Prior to the life extension program the size of flaw detected was not considered a serious factor, inspection was just to determine their existence. In the seventies NDI began to deal with extending service life of aging aircraft and the newer systems with higher stresses and smaller defects. The Air Force NDI Program office decided to evaluate the capability of NDI on aircraft structures. The program, completed in 1978, revealed that using the manual method, we had a 50% detection probability of 1/2 inch cracks with a confidence of 95%. However, using semiautomatic eddy current and ultrasonic, which was incorporated late into the program, indicated a 90-95% reliability of detecting smaller than 1/2 inch cracks.

This NDI evaluation became the catalyst for two other programs, engine reliability and technician proficiency. The engine reliability program, completed in 1981, showed an improvement in detection capability, however, there was an opportunity to do better. Capability determined at the engine ALCs is illustrated in the table below.

ALC FACILITIES NDI DETECTION CAPABILITIES
(CAPABILITY/CONFIDENCE LEVEL)

INSPECTION METHOD	BEST	DEPOT
ULTRASONIC SURFACE	0.180 / 90%	0.375 / 80%
MAGNETIC PARTICLE	0.125 / 65%	0.300 / 60%
EDDY CURRENT - STATIC	0.150 / 30%	0.200 / 30%
EDDY CURRENT - DYNAMIC	0.130 / 98%	0.090 / 90%
PENETRANT - AUTOMATED LINES	0.175 / 85%	0.230 / 75%
PENETRANT - HAND PROCESSING	0.125 / 90%	0.240 / 85%
PENETRANT - ELECTROSTATIC	0.175 / 90%	0/220 / 90%
PENETRANT - BATCH PROCESSING	0.125 / 90%	0.190 / 75%

Technician proficiency is now an on-going program. Prototype inspection kits have been developed for all five methods. These kits are devised to be similar to aircraft structure with inspection sites arranged in a holding container or rack. The rack consists of ten 2 inch wide by 12 inch long, 1/4 inch thick aluminum plates with 10 fasteners arranged in a row. A collection of 10 or more are put into a rack to allow for inspection similar to an aircraft. Fatigue cracks are initiated at selected fastener holes for the defect simulation. Technicians will practice on the racks to improve their proficiency in detecting flaws.

These programs have also prompted the development of a Process Assurance function. It will relate to all NDI procedures; equipment, personnel, and process control. Process assurance will provide the means to assess the NDI cabilities and provide an immediate solution to bring the NDI back "on line" should there be shortcomings. This means immediate feedback on how well the job is being performed, and therefore will create an incentive to do better and thus improve capability. Improving capability leads to improved reliability.

Having explained some background of the NDI process and how it is affected by maintenance, life extension, and safe flight, let's relate this to structural integrity. The Aircraft Structural Integrity Program (ASIP) and Engine Structural Integrity Programs (ENSIP) provide great benefit to the newer weapon system developments. Through the use of fracture mechanics engineering, we can predict usable life and determine allowable flaw sizes. MIL-A-83444, "Airplane Damage Tolerance Design Requirements" assumes that flaws are inherent in any material and the design must account for them. Designs with flaw sizes smaller than specified in MIL-A-83444 require the contractor to demonstrate his capability to find them with a documented and controlled NDI method. This specification also defines in-service flaw detection limits. And MIL-I-6870, "NDI Requirements for Aircraft and Missile Materials and Parts" requires the manufacturer to classify and inspect components and materials using NDI for acceptance. Therefore, NDI supports the aircraft system during design, development, and acquisition, as well as throughout its service life. Similarly ENSIP uses the same approach. For these programs to mesh with the operational Air Force, it will require a great deal of teamwork between the NDI community and ASIP/ENSIP community.

An example of how we are working together toward a common goal is illustrated with the requirement to do whole field inspection of engine disks. The engine designer wanted to inspect the whole disk and reliabily find a 0.030 inch flaw using fluorescent penetrant inspection (FPI). Our assessment showed the best we could achieve was 0.20 inches and greater. In the dialogue and communication that followed we, the Air Force NDI community, stated that with a diligent effort we could detect 0.100 inch flaw. The designer agreed to live with the 0.100 inch detection capability without too much extra risk for now, but a 0.030 inch flaw detection size capability will be needed in the 1990's. The NDI community agreed to work towards an R&D program to find a 0.030 inch flaw by the early 1990's. The basis of this agreement came as a result of communication and dialogue with the Air Force NDI community and ENSIP.

In summary, NDI is an important technology tool which must be recognized and used if we are to achieve ASIP/ENSIP goals, both now and for the future. The Air Force must undertake a continuing, systematic NDI capability assessment for airframes, continue its efforts for engines, and provide feedback on both to the acquisition community early during the design/development phase. ASIP and ENSIP must also continue to work toward introduction of materials with improved damage tolerance. Finally, we must become smarter and more effective in implementing and managing a true life cycle NDI program for all systems and engines. In this regard, NDI Advisory Boards and other AF NDI Program Office objectives can be of significant value in enhancing our management approach. We must all work together as a team with the same objectives for program improvement.

IMPACT OF STRUCTURAL INTEGRITY PROGRAM ON NONDESTRUCTIVE INSPECTION

BRIEFER: MAJ DAVE JIRELE

AF NDI PROGRAM MANAGER

KELLY AFB, TX.

OVERVIEW

- BACKGROUND
- ASIP/ENSIP
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- AF NDI PROGRAM
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  - LESS THAN ½ IN.
  - FATIGUE CRACK
- DEVELOPED
  - TECHNICIAN PROFICIENCY
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  - MINI KITS
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  - FEED BACK

FY87/253/5

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- MIL-A-83444
  - INHERENT FLAWS
  - MIL-I-6870
  - TEAMWORK

FY87/253/6

WHOLE FIELD INSPECTION

- FPI
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  - 0.20 IN. +
- DIALOGUE
  - AGREEMENT
  - R & D

FY87/253/7

SUMMARY

- NDI TECHNOLOGY
- FEEDBACK
- LIFE CYCLE MANAGEMENT
  - — NDI ADVISORY BOARDS

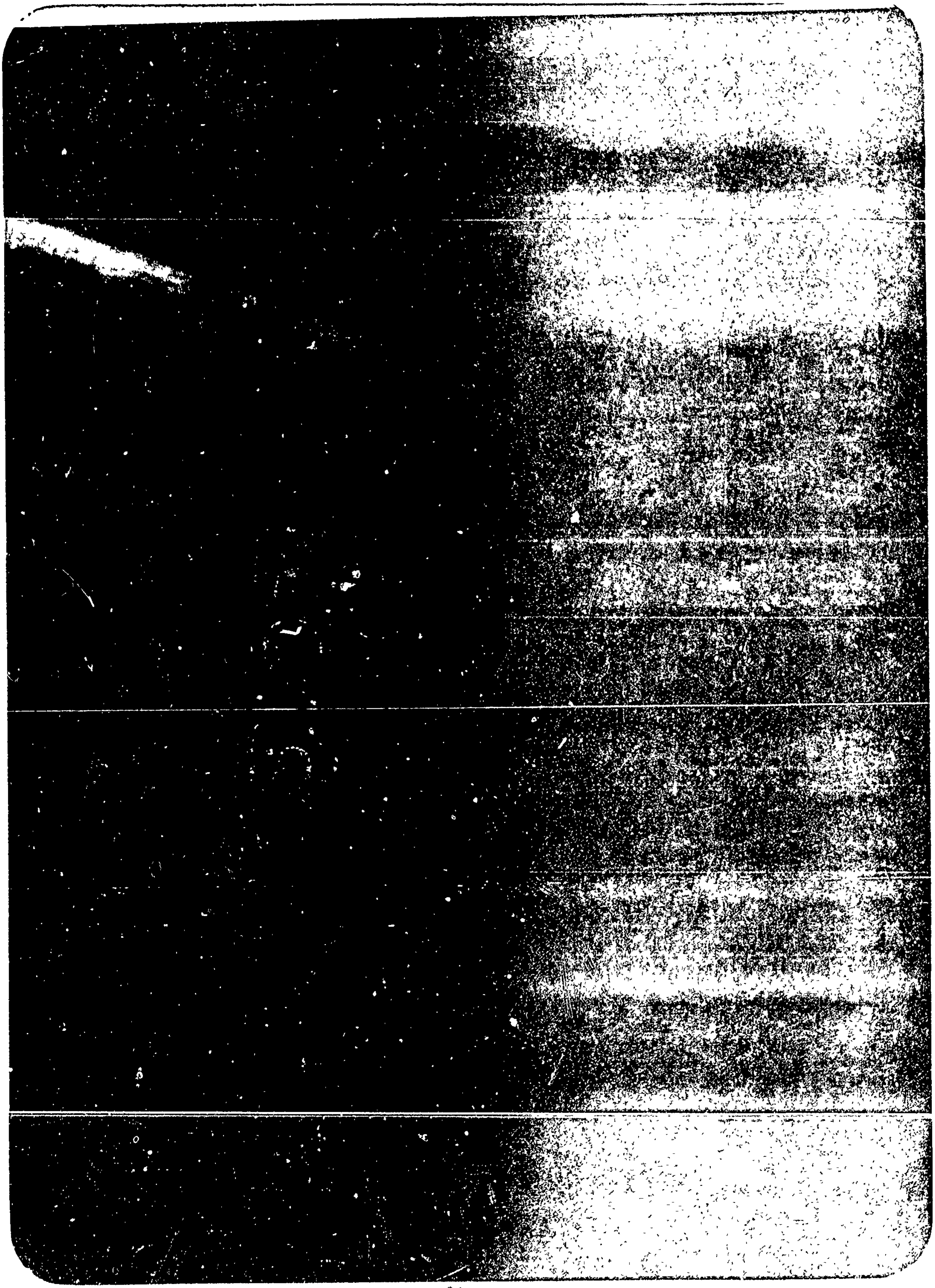
FY87/253/8

Air Force Corrosion Control Program

By

Lt Col Ronald C. Hoch
WR-ALC/MMEM

AIRCRAFT PAINTING/STRIPPING



SLIDE 2

I WILL DISCUSS FOUR MAJOR TOPICS CONCERNING AIRCRAFT PAINTING AND STRIPPING.

WARNER ROBINS ALC

OVERVIEW

MM

WR-ALC

AF AIRCRAFT PAINT POLICY

AIR QUALITY STANDARDS

AIRCRAFT SURFACE PREPARATION

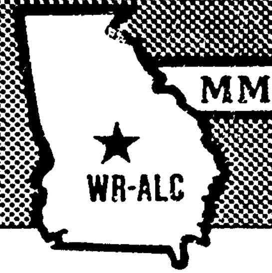
AIRCRAFT PAINT STRIPPING

SLIDE 3L/R

THE U.S. AIR FORCE'S AIRCRAFT PAINT POLICY IS ESTABLISHED UNDER AFR 66-34 AND T.O. 1-1-4. THE PRIMARY PURPOSE FOR PAINTING AIRCRAFT IS FOR CORROSION PREVENTION AND CONTROL. ALSO, RECENTLY ESTABLISHED IN AF POLICY IS THE REQUIREMENT TO PRESERVE A PROFESSIONAL PAINT APPEARANCE AS AN INTEGRAL PART OF A WELL-MANAGED CORROSION CONTROL PROGRAM. THE AF AIRCRAFT PAINT POLICY REQUIRES EACH WEAPON SYSTEM PROGRAM MANAGER TO DEVELOP A WEAPON SYSTEM SERVICE LIFE PAINT PLAN WHICH LAYS OUT THE CRITERIA FOR PAINTING, THE TYPE OF PAINT, COLOR SCHEME, WHEN WILL A COMPLETE STRIPPING AND REPAINTING BE DONE OR A SCUFF SANDING AND OVERCOATING. THE SLIDE ON THE RIGHT GIVES A BREAKDOWN OF THE INTERVAL FOR PAINTING U.S. AIRCRAFT. THE STRIPPING AND REPAINTING AND SCUFF SANDING AND OVERCOATING ARE PRIMARILY DONE AT A DEPOT FACILITY (ORGANIC OR CONTRACT). THE DEPOT PAINTING CURRENTLY COST $50 MILLION AND WILL EXPAND $95 MILLION WITH NEW PAINT PLANS. SECTIONALIZED PAINTING (WING OR STABILIZER) AND MAINTENANCE TOUCH-UP PAINTING ARE ACCOMPLISHED AT OPERATIONAL UNITS.

WARNER ROBINS ALC

AIRCRAFT PAINTING/STRIPPING



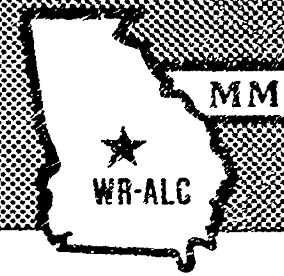
AF AIRCRAFT PAINT POLICY

- O AFR 66-34/T.O. 1-1-4
- O PRIMARY PURPOSE OF PAINTING:

  <u>CORROSION PREVENTION/CONTROL</u>

- O PROFESSIONAL PAINT APPEARANCE
- O SYSTEM PROGRAM MANAGER
  - OO WEAPON SYSTEM SERVICE LIFE PAINT PLAN

AIRCRAFT PAINTING/STRIPPING



AF AIRCRAFT PAINT POLICY

O WEAPON SYSTEM SERVICE LIFE PAINT PLANS

6 YRS	7 YRS	8 YRS	9 YRS
F-15	*C-130	*F-4D	*F-4E/G
F-16	(42 mo. PDM)	*F-111	*RF-4C
A-10		*KC-135	*C-9A
*C-135B		C-141	C-5
*RC-135		E-4	
E-3		*C-18	
B-1B		*B-52	
		*C-130	
		(24 mo. PDM)	
		*C-130	
		(48 mo. PDM)	

*MID-LIFE OVERCOAT

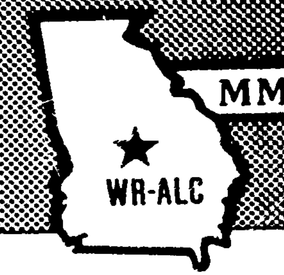
SLIDE 4L/R

THE AF AIRCRAFT PAINT POLICY AND THE MILITARY STANDARD 1568A STATES THAT THE STANDARD PAINT SYSTEM FOR U.S. AIRCRAFT IS TWO COATS OF EPOXY PRIMER (MIL-P-23372) AND TWO COATS OF POLYURETHANE TOPCOAT (MIL-C-82836). THIS COATING SYSTEM HAS BEEN USED ON U.S. AIRCRAFT SINCE 1972 AND HAS PROVIDED EXCELLENT CORROSION PROTECTION. SOME EXCEPTIONS DO EXIST.

THE EPOXY PRIMER AND ACRYLIC LACQUER SYSTEM IS USED ON THE F/FB/EF-111 FLEET. PRIMARY REASON FOR ACRYLIC LACQUER IS TO PREVENT HYDROGEN EMBRITTLEMENT OF THE STEEL STRUCTURES ON THE AIRCRAFT BY NOT USING HARSH CHEMICAL STRIPPERS TO REMOVE POLYURETHANE IF IT WERE USED INSTEAD OF ACRYLIC LACQUER.

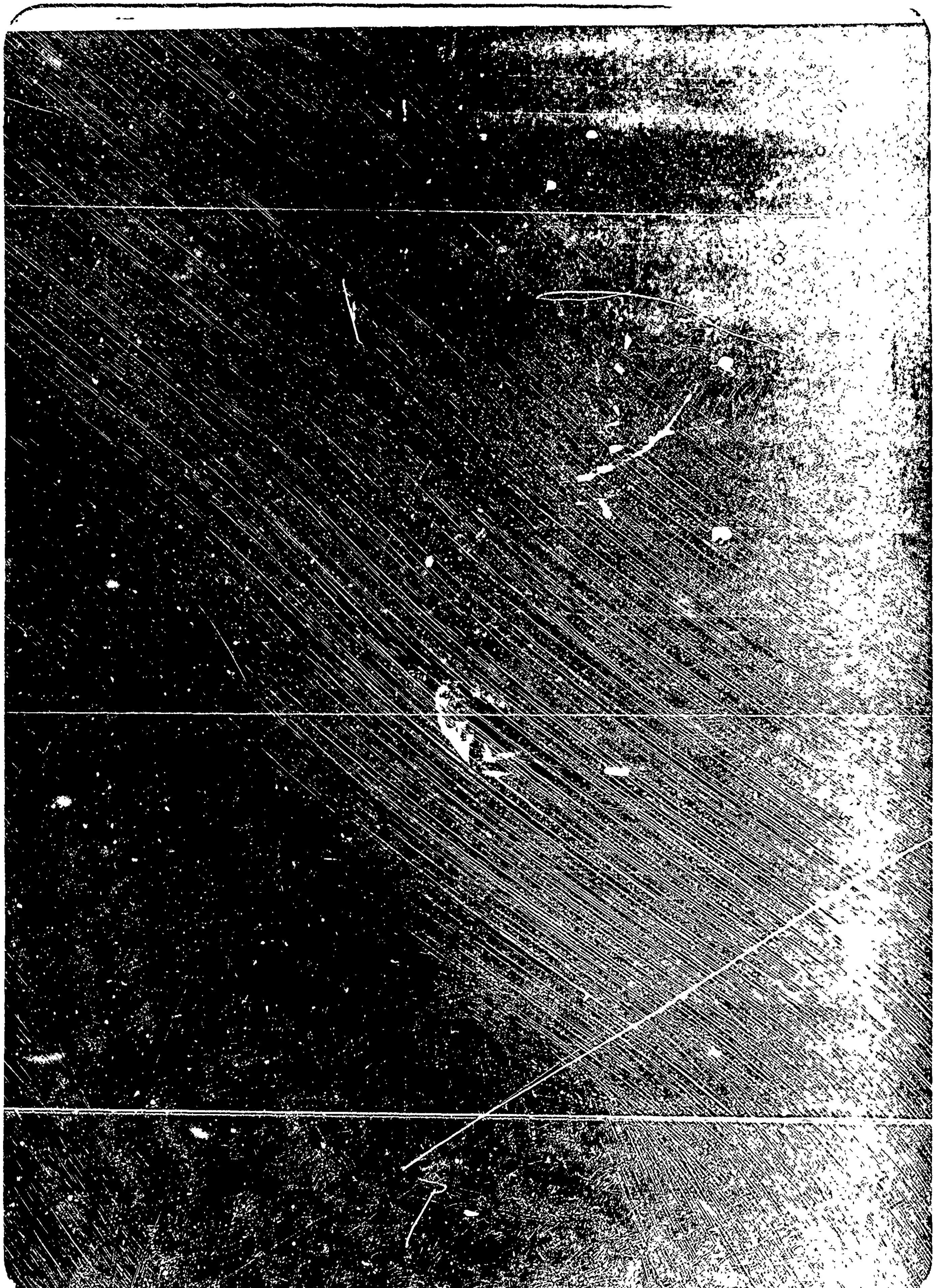
POLYSULFIDE PRIMER ALONG WITH POLYURETHANE TOPCOAT IS USED ON THE B-52, KC-135, C-141 AND C-130 AIRCRAFT. THE POLYSULFIDE PRIMER IS A MORE FLEXIBLE COATING THAN THE EPOXY PRIMER AND DURING TESTS IN 1974-76 SHOWED IT PROVIDED BETTER PROTECTION ON THE UPPER WING SURFACES OF LARGE AIRCRAFT.

AIRCRAFT PAINTING/STRIPPING



AF AIRCRAFT PAINT POLICY

- O STANDARD PAINT SYSTEM (1972)
  - OO EPOXY PRIMER/POLYURETHANE TOPCOAT
- O EXCEPTIONS
  - OO EPOXY PRIMER/ACRYLIC LACQUER
    - OOO F/FB/EF-111
    - OOO HYDROGEN EMBRITTLEMENT
  - OO POLYSULFIDE PRIMER/POLYURETHANE
    - OOO KC-135, C-130, B-52, C-141
    - OOO UPPER WING CORROSION



SLIDE 5L/5R

THE DEPOTS FOR THESE AIRCRAFT (B-52, KC-135, C-141, C-130) ARE EVALUATING THE USE OF A POLYURETHANE KOROFLEX PRIMER TO REPLACE THE POLYSULFIDE PRIMER. THE KOROFLEX PRIMER HAS SEVERAL ADVANTAGES.

SINGLE COMPONENT PAINT

INDEFINITE POTLIFE

REMOVABLE WITH EXISTING CHEMICAL STRIPPERS

USEABLE IN ELECTROSTATIC PAINTING

THE KOROFLEX PRIMER HAS BEEN USED ON SEVERAL TEST AIRCRAFT BY OKLAHOMA CITY ALC SINCE 1984 AND WARNER ROBINS ALC RECENTLY PAINTED C-141/C-130 AIRCRAFT.

AIRCRAFT PAINTING/STRIPPING

AF AIRCRAFT PAINT POLICY

O STANDARD PAINT SYSTEM (1972)

OO EPOXY PRIMER/POLYURETHANE TOPCOAT

O EXCEPTIONS

OO EPOXY PRIMER/ACRYLIC LACQUER

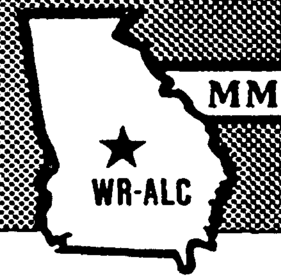
OOO F/FB/EF-111

OOO HYDROGEN EMBRITTLEMENT

→ OO POLYSULFIDE PRIMER/POLYURETHANE

OOO KC-135, C-130, B-52, C-141

OOO UPPER WING CORROSION



AIRCRAFT PAINTING/STRIPPING

AF AIRCRAFT PAINT POLICY

O KOROFLEX PRIMER

OO ADVANTAGES

- SINGLE COMPONENT
- INDEFINITE POTLIFE
- FILM FLEXIBILITY
- REMOVABLE WITH EXISTING CHEMICAL STRIPPERS
- REDUCE MAN-HOURS
- USEABLE IN ELECTROSTATIC PAINTING

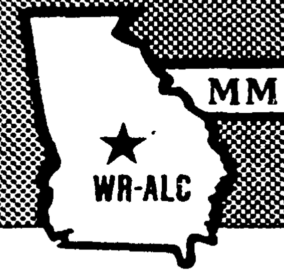
OO TEST AIRCRAFT - B-52, KC-135, C-141, C-130

SLIDE 6L

THE NEXT TOPIC I WILL DISCUSS IS THE U.S. GOVERNMENT'S (ENVIRONMENTAL PROTECTION AGENCY AND LOCAL DISTRICTS) REQUIREMENT TO MEET CERTAIN AIR QUALITY STANDARDS TO REDUCE AMOUNT OF OZONE IN THE AIR. THE CLEAN AIR ACT OF 1977 IS REQUIRING THE DEPARTMENT OF DEFENSE AND AEROSPACE CONTRACTORS TO REDUCE THE AMOUNT OF SOLVENTS (VOLATILE ORGANIC COMPOUNDS) EMITTED FROM ORGANIC AND CONTRACT DEPOT AIRCRAFT PAINTING.

WARNER ROBINS ALC

AIRCRAFT PAINTING/STRIPPING



AIR QUALITY STANDARDS

- O CLEAN AIR ACT 1977
- O DOD/AEROSPACE CONTRACTORS
- O PAINT SOLVENT EMISSIONS

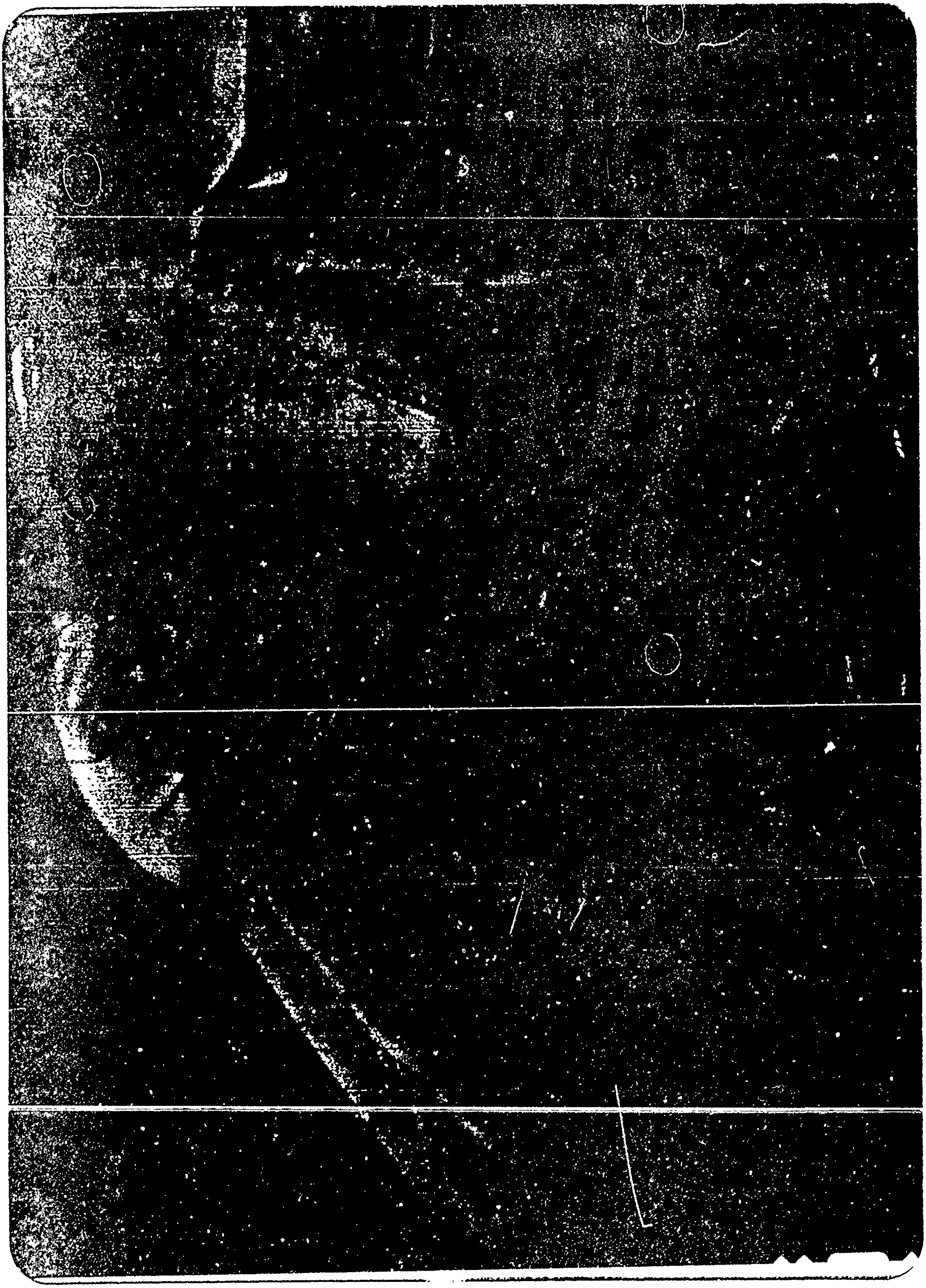
WARNER ROBINS ALC

AF CORROSION PROGRAM

WR-ALC MATERIEL MANAGEMENT ★ CENTER OF EXCELLENCE ★ WE MAKE IT HAPPEN WITH PRIDE

- ISSUE
  - SOLVENT EMISSIONS FROM AIRCRAFT PAINTING
    - PRIMERS 600 -> 340 GRAMS/LITER
    - TOPCOAT FINISH 580 -> 420 GRAMS/LITER
  - ACTIONS
    - HQ AFLC/MM 5 JUN 86
      - EPA WAIVER
    - HQ USAF/LEY 30 JUN 86
      - AWAITING DOD GUIDANCE FOR AIRCRAFT
    - HQ USAF/LE 18 SEP 86 HQ AFLC/AFSC/CV
      - AF PLAN SAF/MI/AL
      - LOW VOC PAINTS
        - VEHICLES; FACILITIES; REAL PROPERTY EQUIPMENT; ROAD MARKINGS

1662



AF CORROSION PROGRAM

- ACTION (CONTINUED)
  - HQ USAF/LE -> MAJOR AIR COMMANDS 23 SEP 86
    - SOURCES OF SOLVENT EMISSIONS
    - LOCAL AIR QUALITY LAWS
    - DEVELOP PLAN FOR COMPLIANCE
    - LOCALLY NEGOTIATE WAIVERS
    - NO WAIVERS NEGOTIATED AT NATIONAL LEVEL

SLIDE 6R

THE SOLVENT REDUCTION MAY BE ACHIEVED THROUGH SEVERAL ACTIONS:

A. INSTALLATION OF ABATEMENT DEVICES ($1.8 MILLION PER DEPOT IN AFLC)

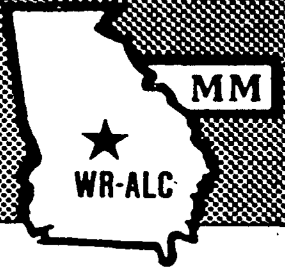
B. CHANGE OUR CURRENT PAINT FORMULAS - PRIMERS FROM 600 GRAMS/LITER OF SOLVENT TO 340 GRAMS/LITER OF SOLVENT AND POLYURETHANE TOPCOAT FROM 580 GRAMS PER LITER OF SOLVENT TO 420 GRAMS PER LITER OF SOLVENT. IN THE PAINT FORMULA CATEGORY, THE AIR FORCE MATERIALS LABORATORY HAS TESTED AND APPROVED A WATERBASE PRIMER WITH A CORROSION INHIBITOR WITH 340 GRAMS OF SOLVENT PER LITER TO MEET THE EPA STANDARD. A HIGH SOLIDS POLYURETHANE WITH 420 GRAMS/LITER OF SOLVENT HAS BEEN TESTED AND APPROVED IN THE LABORATORY. A B-1B AIRCRAFT AT ROCKWELL'S PALMDALE PLANT WAS PAINTED WITH THE HIGH SOLIDS POLYURETHANE IN JUNE 86.

AIRCRAFT PAINTING/STRIPPING

AIR QUALITY STANDARDS

- O ACTIONS
  - OO ABATEMENT DEVICES
    - OOO SOLVENT RECOVERY SYSTEM
  - OO PAINT FORMULAS
    - OOO WATERBASE PRIMER
    - OOO HIGH SOLIDS POLYURETHANE
  - OO ELECTROSTATIC PAINTING

AIRCRAFT PAINTING/STRIPPING



AIR QUALITY STANDARDS

O ACTIONS

OO ABATEMENT DEVICES

OOO SOLVENT RECOVERY SYSTEM

OO PAINT FORMULAS

OOO WATERBASE PRIMER

OOO HIGH SOLIDS POLYURETHANE

OO ELECTROSTATIC PAINTING

AF CORROSION PROGRAM

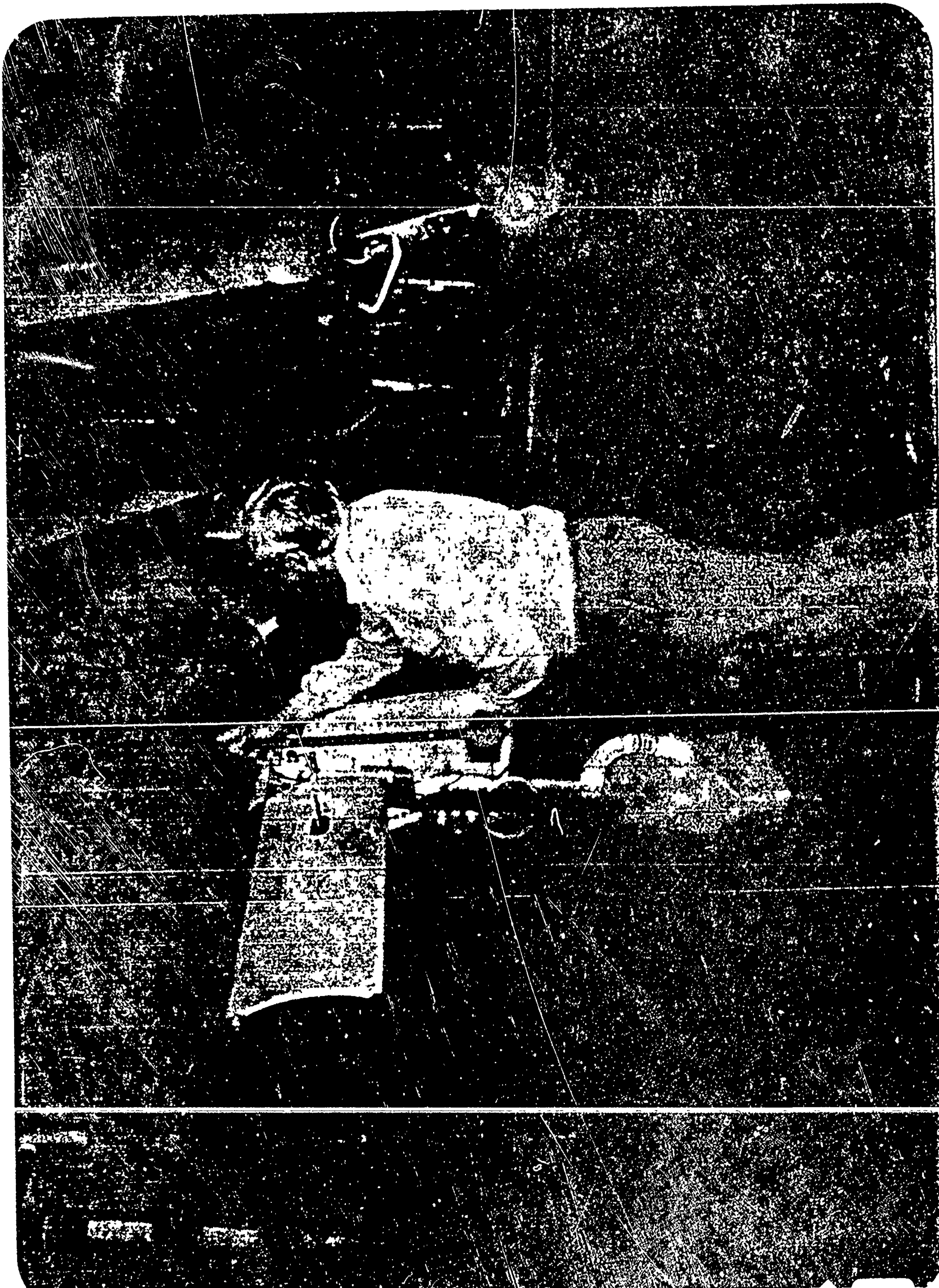
WARNER ROBINS ALC

WR-ALC MATERIEL MANAGEMENT ★ CENTER OF EXCELLENCE ★ WE MAKE IT HAPPEN WITH PRIDE

- ISSUE
  - SOLVENT EMISSIONS
    - PAINT TECHNOLOGY DEVELOPMENT
      - WATERBASE PRIMER   CORROSION INHIBITOR
      - HIGH SOLIDS POLYURETHANE TOPCOAT
        - B–1B   PALMDALE   JUN 86
        - EQUIPMENT/APPLICATION/COLOR UNIFORMITY
      - C–17 FINISH SYSTEM
        - TEMPORARY RELIEF   S. CALIF EPA DISTRICT
        - AF STANDARD PAINT SYSTEM VS CHLORINATED PAINTS
    - EQUIPMENT DEVELOPMENT
      - OC–ALC ELECTROSTATIC PAINTING
        - JUN 86   AFLC/IG   WAIVER JP–5 FUEL
        - OPERATIONAL SUPPLEMENT TO 1–1–8
        - B–52; KC–135; A–7; E–3

SLIDE 7L/R

C. ANOTHER MEANS TO REDUCE THE SOLVENT EMISSIONS IS THE USE OF ELECTROSTATIC PAINTING. ELECTROSTATIC PAINTING USES AN ELECTRIC CHARGE ON THE AIRCRAFT TO ATTRACT CHARGED ATOMIZED PAINT AND PROVIDES BETTER PAINT TRANSFER EFFICIENCY FROM THE PAINT GUN TO THE AIRCRAFT WHICH REDUCES SOLVENT LOSS INTO THE ENVIRONMENT. OKLAHOMA CITY ALC HAS SUCCESSFULLY USED THE ELECTROSTATIC PAINTING EQUIPMENT TO PAINT AN A-7, KC-135, AND B-52 AIRCRAFT. HQ AFLC INSPECTOR GENERAL HAS, RECENTLY, APPROVED USING ELECTROSTATIC PAINTING ON JP-5 FUELED AIRCRAFT. THEREFORE, AIRCRAFT PRODUCTION WILL NOT HAVE TO DEFUEL AIRCRAFT FOR PAINTING. OKLAHOMA CITY ALC PLANS TO USE ELECTROSTATIC PAINTING IN THEIR PRODUCTION ENVIRONMENT.



SLIDE 8

NEXT, I'LL DISCUSS THE STEPS THE AIR FORCE GOES THROUGH TO PREPARE AN AIRCRAFT FOR PAINTING AND ADDRESS SOME OF THE CHANGES OR IMPROVEMENTS THAT COULD OCCUR IN THE PROCESS.

THE PRIMARY PROCEDURES FOR AIRCRAFT SURFACE PREPARATION ARE ADDRESSED IN T.O. 1-1-8. THE SEQUENCE OF ACTIONS ARE:

- WASH AIRCRAFT TO REMOVE HEAVY SOILS (GREASE, HYDRAULIC FLUIDS, ENGINE OIL)
- CHEMICALLY STRIP PAINT FROM THE AIRCRAFT
- WASH BARE AIRCRAFT SURFACE WITH ALKALINE SOAP
- PERFORM MAINTENANCE TO INCLUDE MECHANICAL CORROSION REMOVAL AND REPAIR, AND APPLY CHEMICAL CORROSION REMOVER FOR LIGHT CORROSION
- WASH THE AIRCRAFT
- APPLY CHEMICAL CONVERSION COATING (REFERRED TO ALSO AS ALODINE) - AIRCRAFT MUST BE PAINTED WITHIN 48 HOURS OF APPLICATION OF CONVERSION COATING
- PERFORM WATER BREAK TEST TO CHECK CLEANLINESS OF SURFACE
- SOLVENT WIPE DOWN OF AIRCRAFT (50% MEK AND 50% TOLUENE) (USE OF CLEAN RAGS IS PARAMOUNT IN THE WIPE DOWN PROCESS)
- PRIME/PAINT AIRCRAFT

AIRCRAFT PAINTING/STRIPPING

AIRCRAFT SURFACE PREPARATION

- O T.O. 1-1-8 PROCEDURES
  - OO WASH AIRCRAFT
  - OO CHEMICALLY STRIP AIRCRAFT
  - OO WASH BARE AIRCRAFT SURFACE
  - OO PERFORM MAINTENANCE (MECHANICAL CORROSION REMOVAL/REPAIR)
  - OO WASH AIRCRAFT
  - OO APPLY CHEMICAL CONVERSION COATING
  - OO WATER BREAK TEST
  - OO SOLVENT WIPE DOWN OF AIRCRAFT
  - OO PRIME/PAINT AIRCRAFT

WARNER ROBINS ALC

AIRCRAFT PAINTING/STRIPPING

MM

WR-ALC

AIRCRAFT SURFACE PREPARATION

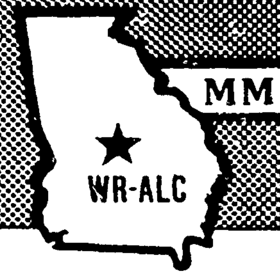
- O T.O. 1-1-8 PROCEDURES
  - OO WASH AIRCRAFT
  - OO CHEMICALLY STRIP AIRCRAFT
  - OO WASH BARE AIRCRAFT SURFACE
  - OO PERFORM MAINTENANCE (MECHANICAL CORROSION REMOVAL/REPAIR)
  - OO WASH AIRCRAFT
  - OO APPLY CHEMICAL CONVERSION COATING
  - OO WATER BREAK TEST
  - OO SOLVENT WIPE DOWN OF AIRCRAFT
  - OO PRIME/PAINT AIRCRAFT

SLIDE 9L/R

THE AIR FORCE CORROSION PROGRAM OFFICE CURRENTLY HAS THREE CONTRACTS AMOUNTING TO $500,000 THAT ARE FOCUSING ON IMPROVEMENTS IN AIRCRAFT CLEANING OR AIRCRAFT WASHING.

1. ONE CONTRACT IS REVISING THE MILITARY STANDARDS FOR AIRCRAFT CLEANING COMPOUNDS, AND THE QUALIFIED PRODUCTS LIST. ALSO, EXPERIMENTING WITH FOAMERS, HAND SCRUBBERS, AND A HAND HELD VIBRATING CLEANER/ WASHER.
2. ANOTHER AGENCY, TRACOR-HYDRONAUTICS, HAS DEVELOPED A PULSE JET HAND HELD CLEANER/WASHER WHICH WILL REMOVE HEAVY SOILS FROM THE AIRCRAFT SURFACE. TESTS WERE DONE ON A C-5, KC-135 AND F-4 AIRCRAFT WITH THIS EQUIPMENT.
3. ALSO, FIELD TESTS ARE UNDERWAY AT THREE AIR FORCE BASES USING HOT WATER/HIGH PRESSURE PORTABLE CLEANERS/WASHERS. A MAJORITY OF OUR WASH FACILITIES DO NOT HAVE HOT WATER AVAILABLE TO ENHANCE THE AIRCRAFT CLEANING. THIS PORTABLE SUPPORT EQUIPMENT WILL PRODUCE HOT WATER AND MIX THE CLEANING COMPOUNDS WITH THE WATER TO ALLOW MORE EFFICIENT AIRCRAFT WASHING.

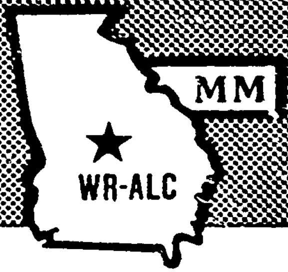
AIRCRAFT PAINTING/STRIPPING



AIRCRAFT SURFACE PREPARATION

- O T.O. 1-1-8 PROCEDURES
  - ➡ OO WASH AIRCRAFT
  - OO CHEMICALLY STRIP AIRCRAFT
  - OO WASH BARE AIRCRAFT SURFACE
  - OO PERFORM MAINTENANCE (MECHANICAL CORROSION REMOVAL/REPAIR)
  - OO WASH AIRCRAFT
  - OO APPLY CHEMICAL CONVERSION COATING
  - OO WATER BREAK TEST
  - OO SOLVENT WIPE DOWN OF AIRCRAFT
  - OO PRIME/PAINT AIRCRAFT

AIRCRAFT PAINTING/STRIPPING



AIRCRAFT SURFACE PREPARATION

- ● AIRCRAFT CLEANING/WASHING
  - ●● MILITARY STANDARDS/QUALIFIED PRODUCTS LIST
    - ●●● FOAMERS, HAND SCRUBBERS, HAND HELD VIBRATING CLEANER/WASHER
  - ●● TRACOR–HYDRONAUTICS
    - ●●● PULSE JET CLEANER/WASHER
  - ●● THREE PORTABLE HOT WATER/HIGH PRESSURE WASHERS

SLIDE 11

ANOTHER AREA THE AIR FORCE IS EXPANDING SOME EFFORT IN IS TO REPLACE CHEMICAL PAINT STRIPPING WITH ANOTHER TECHNOLOGY.

AIRCRAFT PAINTING/STRIPPING

AIRCRAFT SURFACE PREPARATION

O T.O. 1-1-8 PROCEDURES
- OO WASH AIRCRAFT
- ➡OO CHEMICALLY STRIP AIRCRAFT
- OO WASH BARE AIRCRAFT SURFACE
- OO PERFORM MAINTENANCE (MECHANICAL CORROSION REMOVAL/REPAIR)
- OO WASH AIRCRAFT
- OO APPLY CHEMICAL CONVERSION COATING
- OO WATER BREAK TEST
- OO SOLVENT WIPE DOWN OF AIRCRAFT
- OO PRIME/PAINT AIRCRAFT

AIRCRAFT PAINTING/STRIPPING

WARNER ROBINS ALC

WR-ALC MM

AIRCRAFT SURFACE PREPARATION

- O T.O. 1-1-8 PROCEDURES
  - OO WASH AIRCRAFT
  - → OO CHEMICALLY STRIP AIRCRAFT
  - OO WASH BARE AIRCRAFT SURFACE
  - OO PERFORM MAINTENANCE (MECHANICAL CORROSION REMOVAL/REPAIR)
  - OO WASH AIRCRAFT
  - OO APPLY CHEMICAL CONVERSION COATING
  - OO WATER BREAK TEST
  - OO SOLVENT WIPE DOWN OF AIRCRAFT
  - OO PRIME/PAINT AIRCRAFT

AIRCRAFT PAINTING/STRIPPING

AIRCRAFT PAINT STRIPPING

- O CHEMICAL STRIPPERS
  - OO HAZARDOUS WASTE DISPOSAL
  - OO MANPOWER INTENSIVE
- O DRY STRIPPING CANDIDATES
  - OO LASER BEAM
  - OO XENON FLASH LAMP
  - OO CARBON DIOXIDE PELLETS
  - OO PLASTIC MEDIA BLASTING

SLIDE 12

THE MAJORITY OF OUR AIRCRAFT TODAY ARE STRIPPED AT ORGANIC AND CONTRACT DEPOT FACILITIES USING CHEMICAL PAINT STRIPPERS. THE ENVIRONMENTAL PROTECTION AGENCY IS TIGHTENING THE RULES ON THE DISPOSAL OF HAZARDOUS CHEMICAL WASTES FROM AIRCRAFT PAINT STRIPPING. CHEMICAL STRIPPING IS A MANPOWER INTENSIVE METHOD OFTEN REQUIRING THREE OR FOUR COATS OF STRIPPER TO REMOVE THE NUMEROUS LAYERS OF PAINT. ALSO, THE STRENGTH OF THE STRIPPERS HAVE BEEN REDUCED WITH THE ELIMINATION OF PHENOLS AS AN INGREDIENT IN STRIPPERS. THEREFORE, AF LOGISTICS COMMAND IS TESTING SEVERAL DRY STRIPPING CANDIDATES TO REMOVE PAINT FROM AIRCRAFT, COMPONENTS, ENGINES, AND RADOMES. THE CANDIDATES ARE:

LASER BEAM

XENON FLASH LAMP

CARBON DIOXIDE PELLETS

PLASTIC MEDIA BLASTING

AF CORROSION PROGRAM

WARNER ROBINS ALC

WR-ALC MATERIEL MANAGEMENT ★ CENTER OF EXCELLENCE ★ WE MAKE IT HAPPEN WITH PRIDE

- ISSUES
  - AIRCRAFT PAINT SOLVENT EMISSIONS
  - DRY STRIPPING METHODS
  - PLASTIC MEDIA BLASTING
  - DPML AWARENESS
  - JOINT SERVICE PROJECTS
  - SUSTAINING ENGINEERING PROJECTS
  - MAJOR AIR COMMAND SUPPORT

SLIDE 13

THE LASER BEAM PAINT STRIPPING PROJECT IS UNDERWAY AT WARNER ROBINS ALC. SOME TESTING OF THE LASER BEAM AS A STRIPPING DEVICE HAS BEEN DONE AT THE AF MATERIALS LABORATORY AND BATTELLE COLUMBUS LABORATORY. THE CONTRACT FOR THE LASER EQUIPMENT SHOULD BE AWARDED SHORTLY. THE WARNER ROBINS PROJECT WILL USE THE LASER TO STRIP PAINT FROM AIRCRAFT COMPONENTS.

AT OGDEN ALC, A PROJECT USING ROBOTICS AND A LASER BEAM WILL BE TESTED TO REMOVE PAINT FROM THE AIRCRAFT AND AIRCRAFT COMPONENTS.

BOTH LASER BEAM PROJECTS SHOULD BE FUNCTIONAL IN 1988.

SLIDE 14

TWO OTHER TECHNOLOGY CANDIDATES ARE:

A. XENON FLASH LAMP

SACRAMENTO ALC IS WORKING THIS PROJECT. A FLASH LAMP PROTOTYPE HAS BEEN USED SINCE OCT 85 TO DEMONSTRATE THE REMOVAL OF PAINT FROM AIRCRAFT COMPONENTS AND COMPOSITE MATERIALS.

B. CARBON DIOXIDE PELLETS

OKLAHOMA CITY ALC HAS INITIATED A $210,000 PROJECT WITH LOCKHEED SERVICES TO PROVIDE A CO PELLET MAKING MACHINE AND A STRIPPING BOOTH. TESTS IN THE NEXT SEVERAL MONTHS WILL BE PERFORMED ON AIRCRAFT COMPONENT AND ENGINE ACCESSORIES TO DEMONSTRATE THE EFFECTIVENESS OF THE METHOD TO REMOVE PAINT WITHOUT LEAVING RESIDUE, ESPECIALLY ON ENGINE PARTS.

WARNER ROBINS ALC

AF CORROSION PROGRAM

WR-ALC MATERIEL MANAGEMENT ★ WE MAKE IT HAPPEN WITH PRIDE ★ CENTER OF EXCELLENCE

- ISSUE
  - DRY STRIPPING METHODS
    - LASER SYSTEM WR-ALC

      PURCHASE DESCRIPTION TO PROCUREMENT 1 NOV 86

      PM RELEASE TO CONTRACTORS 1 DEC 86

      CONTRACT AWARD DATE 1 SEP 87
    - CO_2 PELLET PAINT REMOVAL SYSTEM OC-ALC
      - MST FUNDS $210K LOCKHEED
    - FLASH LAMP SM-ALC

SLIDE 15

ONE OF THE CANDIDATES WHICH MAY PROVIDE US WITH A NEAR TERM SOLUTION IS PLASTIC MEDIA BLASTING. A PRAM PROJECT WAS APPROVED FOR OGDEN ALC AND THE BLASTING BOOTH CONSTRUCTION COMPLETED IN JUN 85. THE AF EXECUTIVE STEERING GROUP APPROVED PMB FOR REMOVING PAINT FROM F-4 AIRCRAFT AT THE DEPOT IN MAY 85. THIS APPROVAL IS FOR FOUR CYCLES OF PAINT STRIPPING USING THE PMB METHOD.

PLASTIC MEDIA BLASTING FOR GROUND SUPPORT EQUIPMENT WAS ALSO APPROVED BY THE STEERING GROUP. SAN ANTONIO ALC HAS DEVELOPED THE PROCUREMENT PACKAGE FOR A SMALL PMB BOOTH FOR GROUND SUPPORT EQUIPMENT AT OUR OPERATIONAL WINGS.

AT THE TIME OF PMB APPROVAL FOR THE F-4 AIRCRAFT, THE OTHER AIRCRAFT SYSTEM PROGRAM MANAGERS WERE REQUESTED TO IDENTIFY CRITICAL/NON-CRITICAL MATERIALS ON THEIR AIRCRAFT FOR TESTING. A $400,000 CONTRACT WAS INITIATED WITH BATTELLE COLUMBUS LABORATORY TO DO VARIOUS TESTS ON THE EFFECTS OF PLASTIC MEDIA BLASTING ON EIGHT METALLIC AIRCRAFT MATERIALS.

SLIDE 18

THE BATTELLE PLASTIC MEDIA BLASTING STUDY WAS COMPLETED 3 JUL 86. THE REPORT WAS REVIEWED BY AN AFLC/AFSC WORKING GROUP ON THE 28-29 JUL. A FORMAL BRIEFING ON THE REPORT IS BEING PREPARED AND WILL BE PROVIDED TO THE AF EXECUTIVE STEERING GROUP IN EARLY SEPTEMBER. FROM THE RESULTS, A DECISION WILL BE MADE TO AUTHORIZE PLASTIC MEDIA BLASTING FOR OTHER USAF AIRCRAFT BY OCT 86 AND FOR DEPOT IMPLEMENTATION BY OCT 87 OR CONTINUE TO DEVELOP OTHER TECHNOLOGIES FOR AIRCRAFT PAINT STRIPPING.

AF CORROSION PROGRAM

- ISSUES
  - AIRCRAFT PAINT SOLVENT EMISSIONS
  - DRY STRIPPING METHODS
  - PLASTIC MEDIA BLASTING
  - DPML AWARENESS
  - JOINT SERVICE PROJECTS
  - SUSTAINING ENGINEERING PROJECTS
  - MAJOR AIR COMMAND SUPPORT

AF CORROSION PROGRAM

- ISSUE
  - PLASTIC MEDIA BLASTING
    - OGDEN ALC
    - AF EXECUTIVE STEERING GROUP MAY 85
      - GROUND SUPPORT EQUIPMENT
      - F-4 AIRCRAFT
    - MFP 7 SUSTAINING ENGINEERING FUNDS
      - SEP 85 BATTELLE COLUMBUS LAB
        $460,000

AF CORROSION PROGRAM

- ISSUES
  - AIRCRAFT PAINT SOLVENT EMISSIONS
  - DRY STRIPPING METHODS
  - PLASTIC MEDIA BLASTING
  - DPML AWARENESS
  - JOINT SERVICE PROJECTS
  - ➡ SUSTAINING ENGINEERING PROJECTS
  - MAJOR AIR COMMAND SUPPORT

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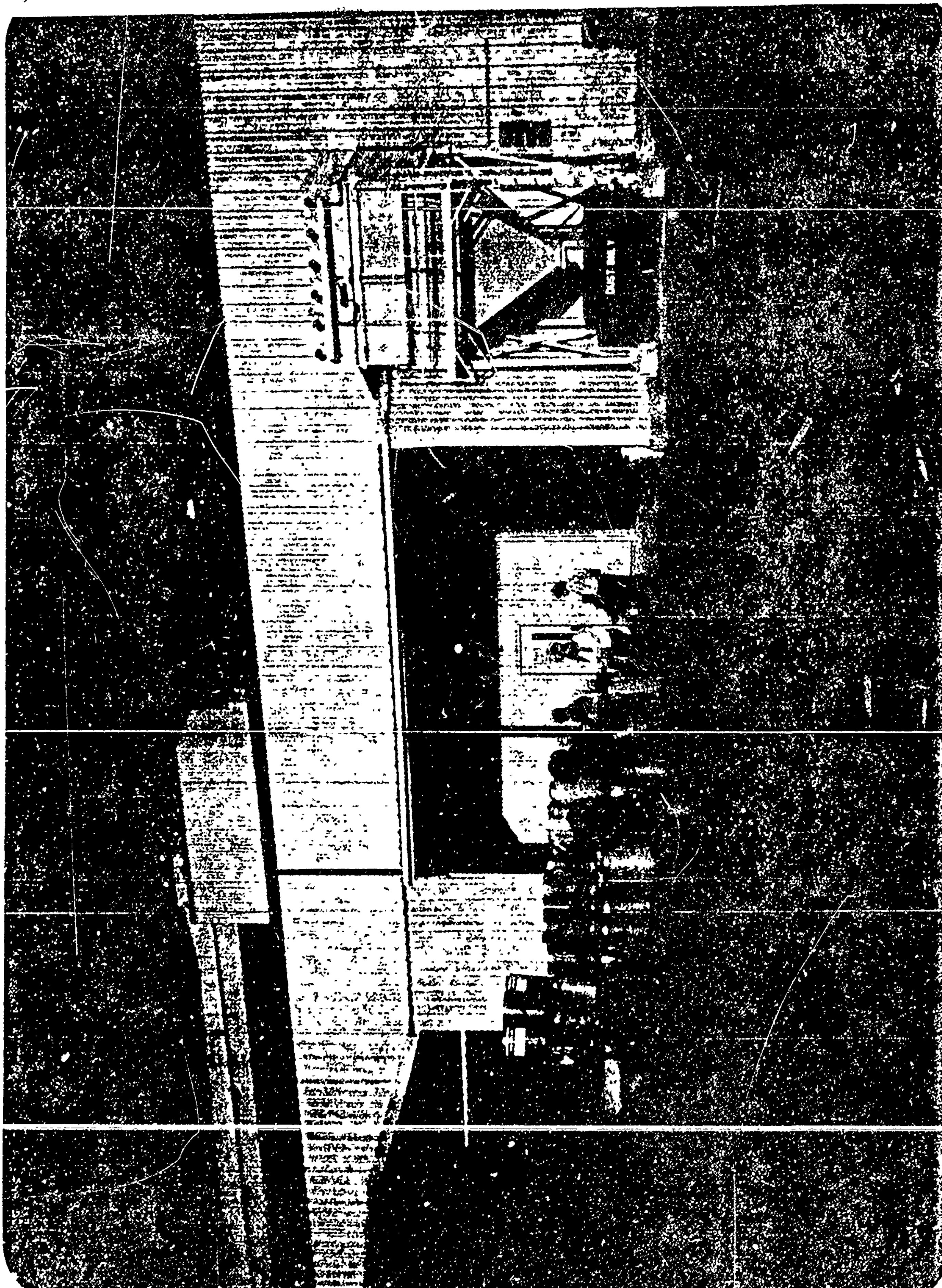
AF CORROSION PROGRAM

WARNER ROBINS ALC

WR-ALC MATERIEL MANAGEMENT ★ WE MAKE IT HAPPEN WITH PRIDE ★ CENTER OF EXCELLENCE

- ISSUE
  - MFP 7 SUSTAINING ENGINEERING PROJECTS
    - FUNDED

PLASTIC BEAD MATERIAL CHARACTERIZATION STUDY	$260K
AIRCRAFT CLEANING/WASHING	$303K
EMP/EMI/CORROSION SUSCEPTIBILITY	$122K
EPA COMPLIANT COATING SYSTEMS	$150K
AIRCRAFT CLEAR WATER RINSING	$150K
TOTAL	$985K



AF CORROSION PROGRAM

- ISSUE
  - MFP 7 SUSTAINING ENGINEERING PROJECTS
    - TERMINATED
      - PULSE WATER JET CLEANER $50K
    - UNFUNDED
      - AF FACILITY PLAN $400K
      - CORROSION DETECTION DEVICES $150K

TOTAL $550K

AF CORROSION PROGRAM

WARNER ROBINS ALC

WR-ALC MATERIEL MANAGEMENT ★ CENTER OF EXCELLENCE ★ WE MAKE IT HAPPEN WITH PRIDE

- ISSUES
  - AIRCRAFT PAINT SOLVENT EMISSIONS
  - DRY STRIPPING METHODS
  - PLASTIC MEDIA BLASTING
  - DPML AWARENESS
  - ➡ JOINT SERVICE PROJECTS
  - SUSTAINING ENGINEERING PROJECTS
  - MAJOR AIR COMMAND SUPPORT

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AF CORROSION PROGRAM

WR-ALC MATERIEL MANAGEMENT ★ CENTER OF EXCELLENCE ★ WE MAKE IT HAPPEN WITH PRIDE

- ISSUE
  - JOINT SERVICE PROJECTS
    - TECHNICAL DATA
      - AVIONICS – IMPLEMENTED
      - CLEANING/WASHING – JUN 87
    - CORROSION INFORMATION CENTER
      - NO DLA FUNDS
    - TRI-SERVICE COATING REMOVAL CONFERENCE JAN 87
    - TRI-SERVICE CORROSION CONFERENCE MAY 87
  - JOINT AERONAUTICAL GROUP OCT 87
    - FUNDS CORROSION RESEARCH

SUMMARY

AF AIRCRAFT PAINT POLICY

AIR QUALITY STANDARDS

AIRCRAFT PAINT STRIPPING

USAF
Turbine Engine Structural Integrity Program
(ENSIP)

By

William D. Cowie

ASD/YZEE

AN OVERVIEW OF THE USAF TURBINE ENGINE STRUCTURAL INTEGRITY PROGRAM (ENSIP) - MIL.. STND. 1783

WM D. COWIE
ASD/YZEE
DEC 1986

OUTLINE

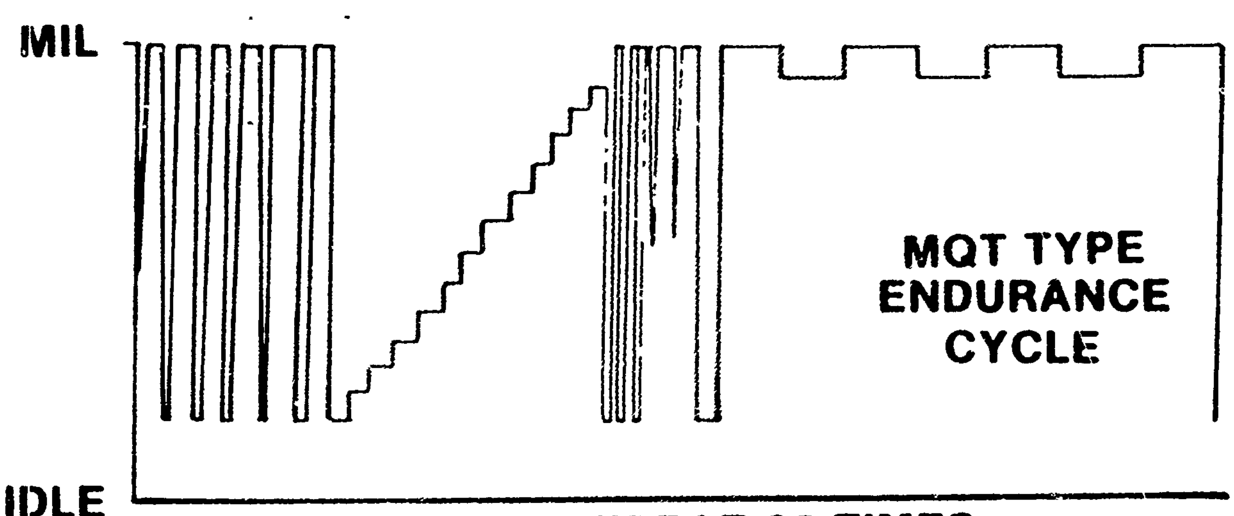
- HISTORICAL PERSPECTIVE (STRUCTURAL DURABILITY)
- ENGINE STRUCTURAL FAILURE MODES
- OVERALL FACTORS AFFECTING ENGINE DURABILITY
  - DESIGN VARIABLES
- TESTING FOR ENGINE DURABILITY
- INTRODUCTION TO ENSIP (ENGINE STRUCTURAL INTEGRITY PROGRAM)

HISTORICAL PERSPECTIVE

- 1946 - EARLY TURBINE ENGINES HAD 25 HOURS LIFE
- 1952 - UP TO 160 HOURS LIFE
- PRE-1969 - ENGINE SPECIFICATIONS WERE DEFICIENT IN THE STRUCTURAL DURABILITY AREA
  - LIFE REQUIREMENTS
  - DUTY CYCLE
  - ANALYSIS
  - TESTING - NOT MISSION RELATED
- IMPROVED CRITERIA APPLIED IN 1969 TO F101 AND TF34 PROGRAMS
- 1973-74 - SPECIFICATION UPDATED
- 1976 - SAB REVIEW
- 1978 - ENSIP DEVELOPED
- 1979 - DURABILITY AND DAMAGE TOLERANCE ASSESSMENT ON F100
- 1984 - ENSIP MIL PRIME SPECIFICATION

HISTORICAL PERSPECTIVE

MANY ENGINE DEVELOPMENTS IN PAST DID NOT REVEAL
SERVICE RELATED PROBLEMS
TYPICAL 150 HOUR TEST

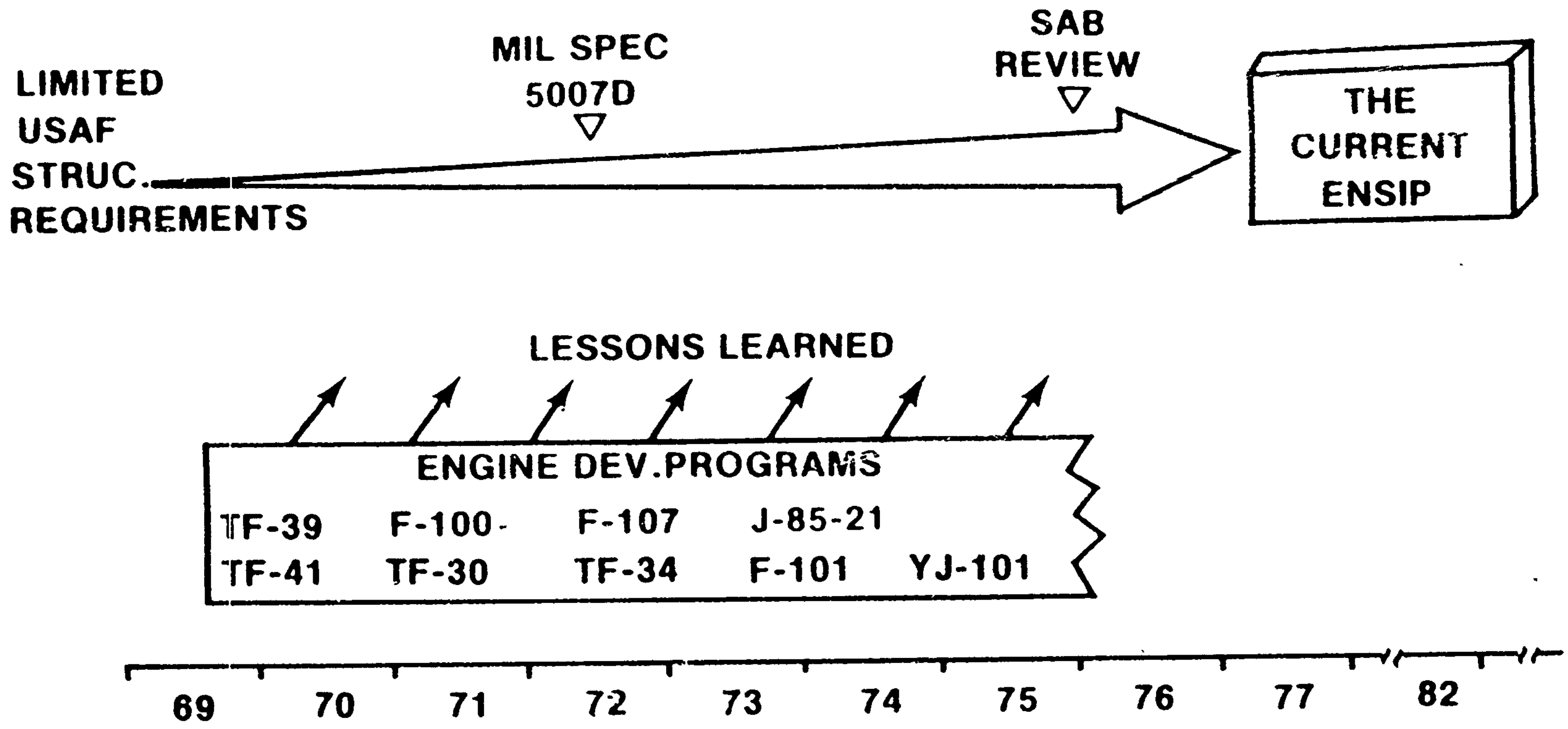


6 HOURS - REPEAT 25 TIMES
CYCLE DEFINITION

	150 HR TEST	2,000 HR USAGE
IDLE-MIL-IDLE	300	20,000
O-MIL-O	25	1,900

HISTORICAL PERSPECTIVE

THE EVOLUTION OF ENSIP



- STRUCTURAL REQUIREMENTS HAVE BEEN TRANSITIONING INTO NEW ENGINE DEV. FOR PAST TWELVE YEARS

HISTORICAL PERSPECTIVE
SOME SPECIFIC LESSONS LEARNED

- IT IS UNREALISTIC (AND CAN BE DANGEROUS) TO ASSUME DEFECT-FREE STRUCTURE IN SAFETY OF FLIGHT COMPONENTS
- CRITICAL PARTS (& PART DETAILS) AND POTENTIAL FAILURE MODES MUST BE IDENTIFIED EARLY AND APPROPRIATE CONTROL MEASURES IMPLEMENTED
- INTERNAL THERMAL & VIBRATORY ENVIRONMENTS MUST BE IDENTIFIED EARLY IN THE ENGINE DEV
- PREDICTED ANALYTICAL STRESSES MUST BE VERIFIED BY TEST FOR COMPLEX COMPONENTS
- MATERIALS & PROCESSES MUST BE ADEQUATELY CHARACTERIZED (PARTICULARLY, THE FRACTURE PROPERTIES)

HISTORICAL PERSPECTIVE

SOME SPECIFIC LESSONS LEARNED (CONT'D)

- DESIGN STRESS SPECTRA, COMPONENT TEST SPECTRA, AND FULL SCALE ENGINE TEST SPECTRA MUST BE BASED ON THE ANTICIPATED SERVICE USAGE OF THE ENGINE: (I.E., ACCELERATED MISSION RELATED TESTING)
- POTENTIAL ENGINE/AIRFRAME STRUCTURAL INTERACTIONS MUST BE DEFINED AND ACCOUNTED FOR
- CLOSED LOOP FORCE MANAGEMENT PROCEDURES MUST BE DEFINED AND ENFORCED
  - REALISTIC INSPECTION AND MAINTENANCE REQUIREMENTS
  - INDIVIDUAL ENGINE TRACKING PROCEDURES
  - DEFICIENCY REPORTING
  - UPDATES IN PROCEDURES BASED ON ACTUAL USAGE

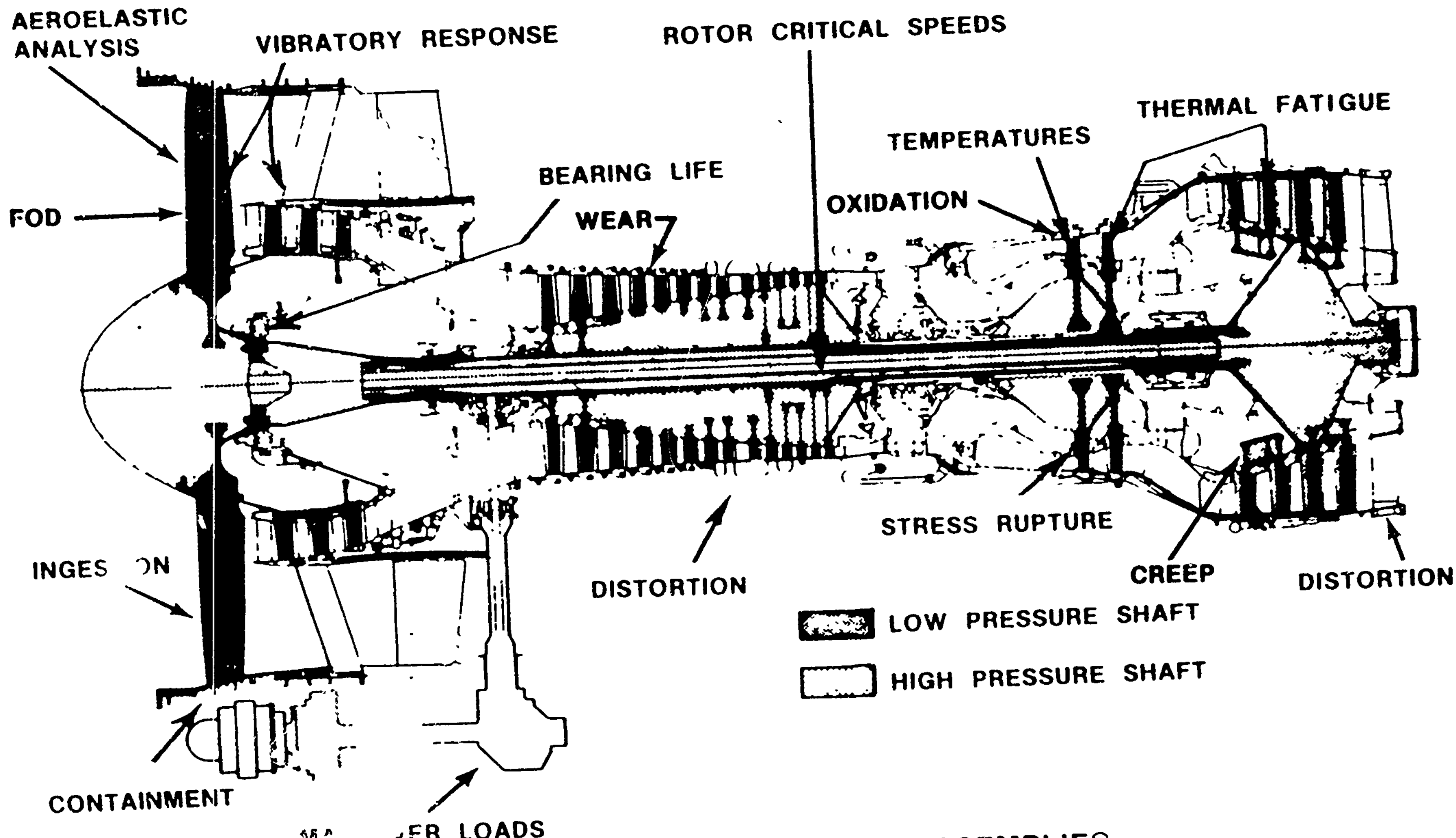
ENGINE STRUCTURAL FAILURE MODES

ENGINE STRUCTURAL FAILURE MODES
TYPICAL STRUCTURAL FAILURE MECHANISMS

COMPONENT	TYPICAL MECHANISM	DESCRIPTION
FRAMES CASES	LOW CYCLE FATIGUE (LCF)*	CRACKING DUE TO REPETITIVE APPLICATION OF CENTRIFUGAL LOADS, APPLIED PRESSURES, THERMAL STRESS AND/OR FLIGHT LOADS
BLADES, DISKS BEARINGS		
VANES THIN SHELL STRUCT.	HIGH CYCLE FATIGUE (HCF)*	CRACKING DUE TO HIGH FREQUENCY STRESS OSCILLATIONS CAUSED BY AERODYNAMIC, SONIC, OR MECHANICAL VIBRATORY ESCITATION FORCES
TURBINE BLADES	STRESS RUPTURE*	DEFORMATION & CRACKING DUE TO PROLONGED APPLICATION OF LOAD AND TEMP.
VANES NOZZLES AUGMENTOR LINERS	OVER-TEMP (BURN-THRU)	LOCAL MELTING DUE TO EXCESSIVE TEMP.

* PRE-EXISTING AND/OR SERVICE INDUCED DEFECTS CAN ACCELERATE CRACKING

ENGINE STRUCTURAL FAILURE MODES



CF6-50 ROTATING ASSEMBLIES

OVERALL FACTORS AFFECTING ENGINE DURABILITY

OVERALL FACTORS AFFECTING ENGINE DURABILITY

DURABILITY

DESIGN	MANUFACT-URING	MAINTENANCE	OPERATIONS
LOW CYCLE FATIGUE	QUALITY	MAINT. CONCEPTS	USAGE
WEAR	INSPECTIONS	SPARES	FOD
DAMAGE TOLERANCE	MISASSEMBLY	TECH ORDERS	ENVIRONMENT
STRESS RUPTURE	MFG METHODS	INSPECTIONS	TACTICS
DYNAMICS		REPAIRS	TRACKING
THERMAL ENVIRONMENT		TRAINING	
MATERIALS			
DESIGN USAGE			

DESIGN VARIABLES

DESIGN VARIABLES
ENGINE DESIGN LIFE GOALS

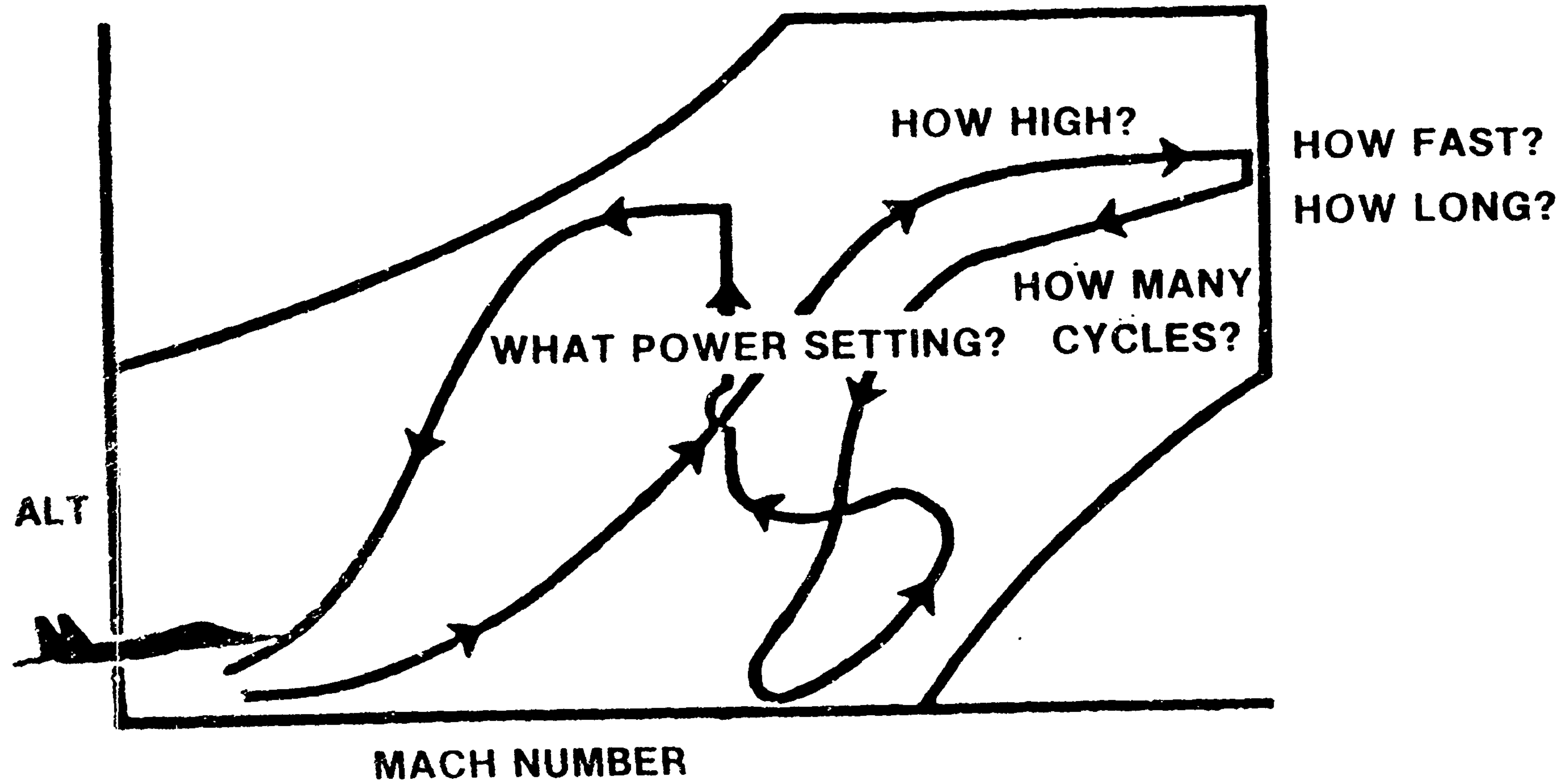
- ENGINE SEPARATED INTO TWO BASIC DIVISIONS
  - HOT PARTS - PARTS IN HOT GAS STREAM (TURBINE VANES, BLADES, COMBUSTOR, ETC.)
  - COLD PARTS - ALL OTHER PARTS (SHAFTS, DISKS, ETC.)
- ENGINE DESIGN LIFE REQUIREMENTS (PAST 12 YEARS)
  - COLD PART LIFE - SAME AS AIRFRAME → SUBJECT TO TRADES
  - HOT PART - ½ COLD PART LIFE

EXAMPLES

	COLD PARTS LIFE	HOT PARTS LIFE
F100	4,000 HOURS	2,000 HOURS
TF-34 (A10 ENGINE)	6.000 HOURS	3,000 HOURS
NGT	18.000 HOURS	9,000 HOURS

DESIGN VARIABLES

WHAT IS A DUTY CYCLE FOR AN AIRCRAFT ENGINE?



- DESCRIPTION OF THE ENGINE USAGE IN THE AIRCRAFT

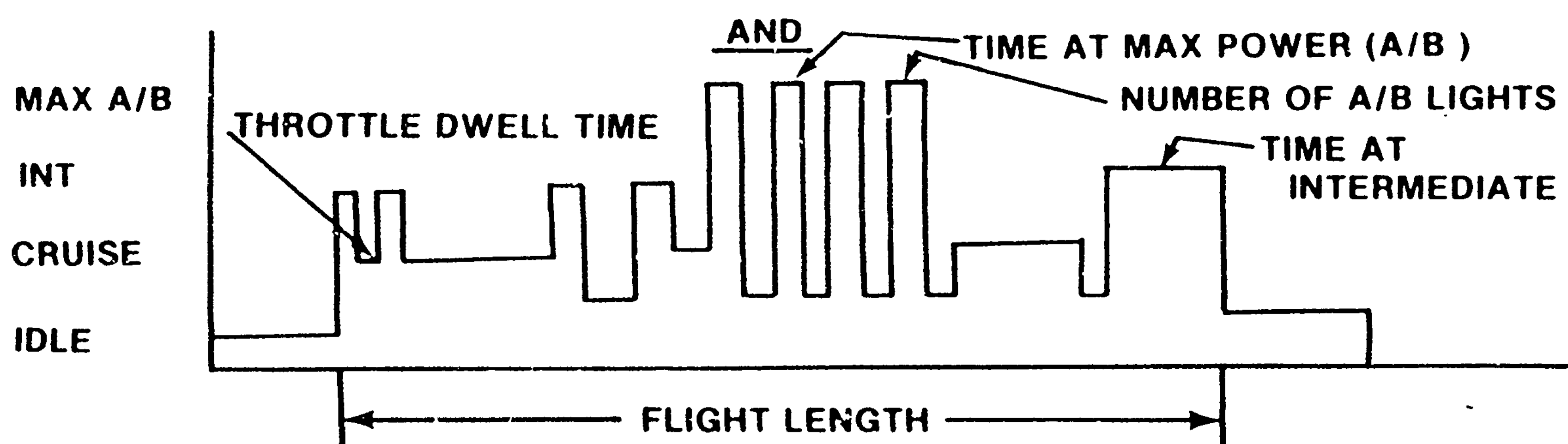
DESIGN VARIABLES

MISSION AND MISSION MIX EFFECTS

- TYPE OF MISSION
  - AIR-TO-AIR
  - AIR-TO-GROUND
  - NAVIGATIONAL

MACH NO.
ALTITUDE
THROTTLE SETTING

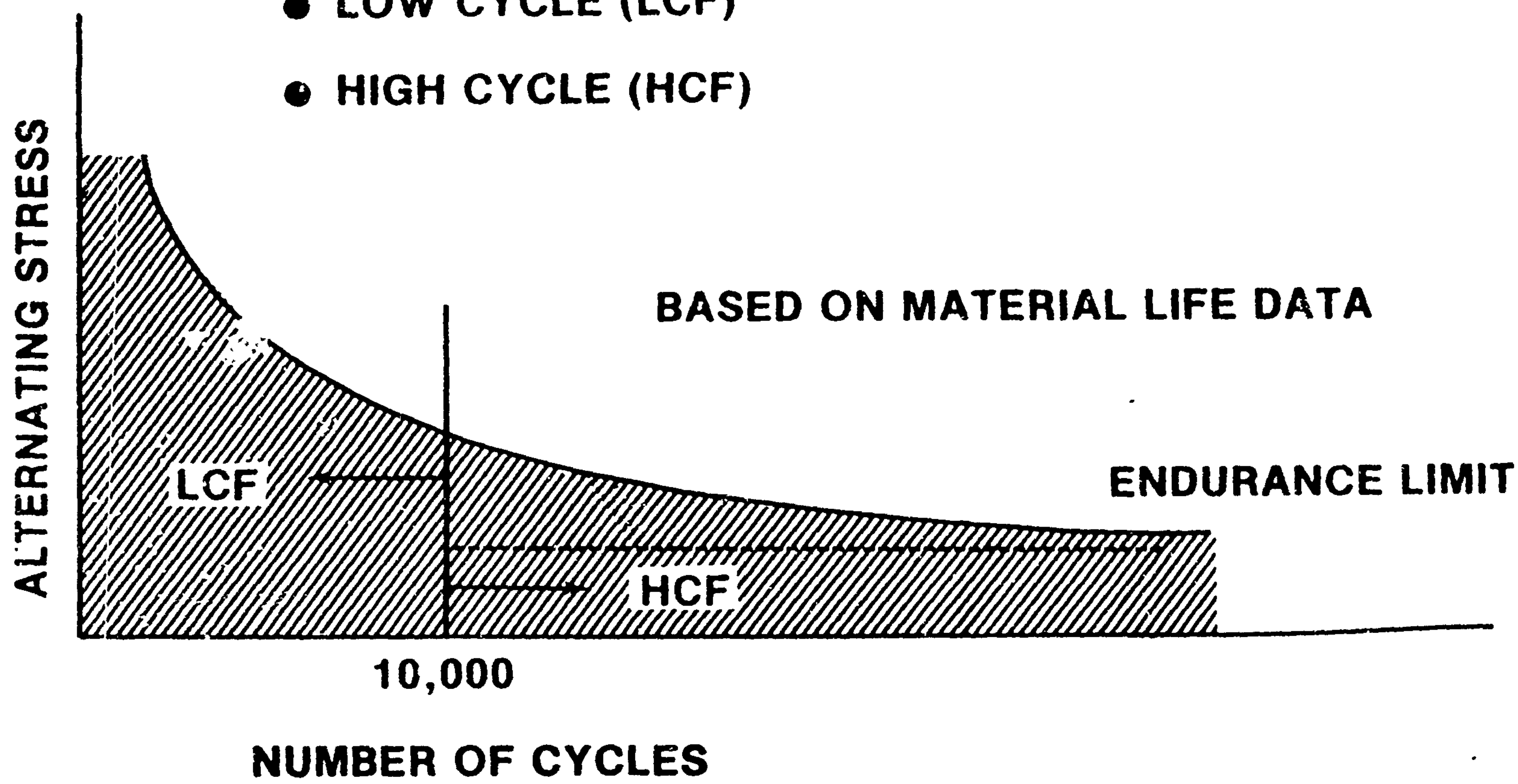
- ESTABLISHES - DESIGN DUTY CYCLE
  - NUMBER OF O-MAX-O CYCLES
  - NUMBER OF PARTIAL CYCLES

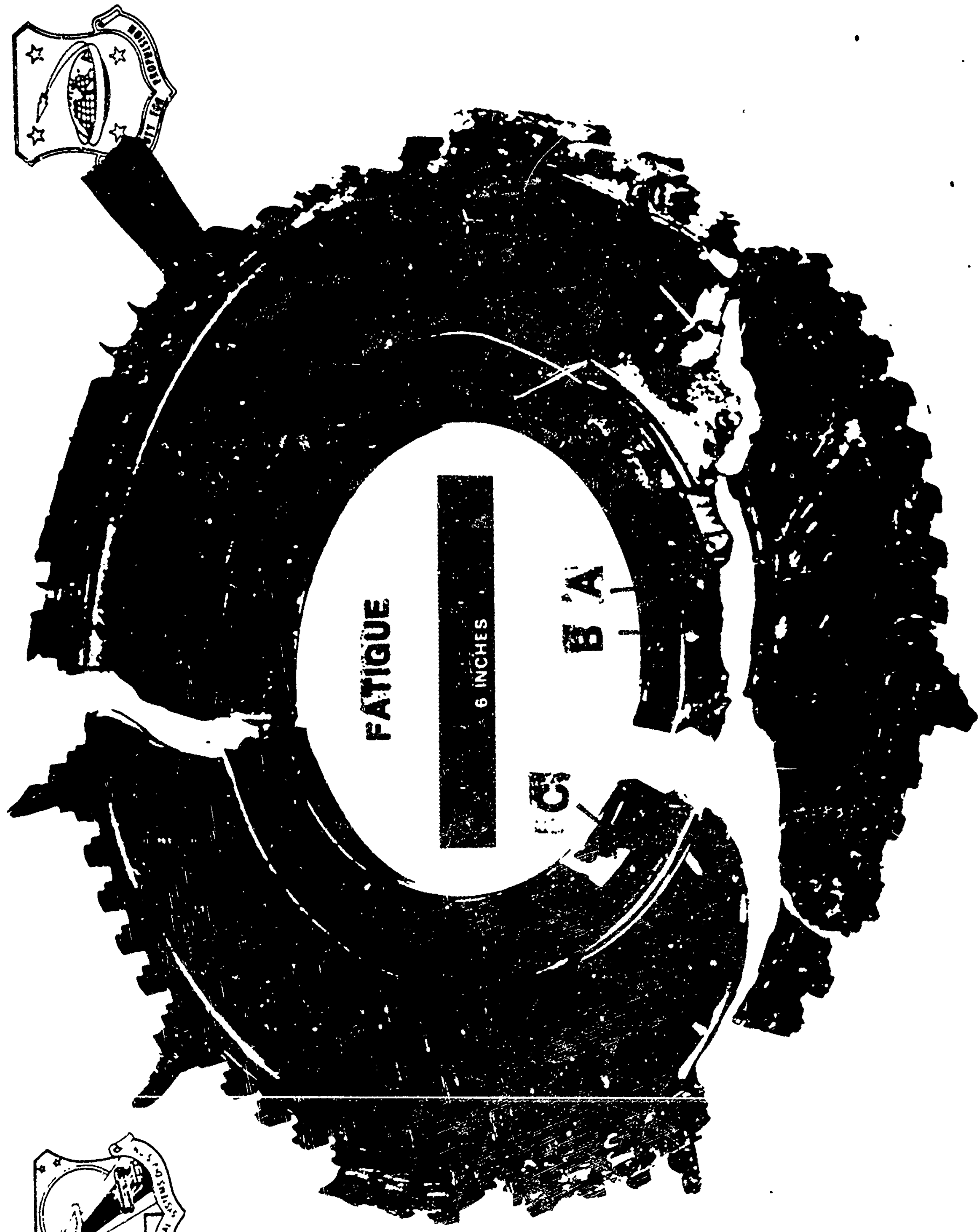


DESIGN VARIABLES

FATIGUE DESIGN

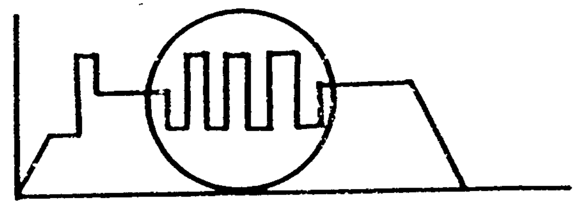
- LOW CYCLE (LCF)
- HIGH CYCLE (HCF)

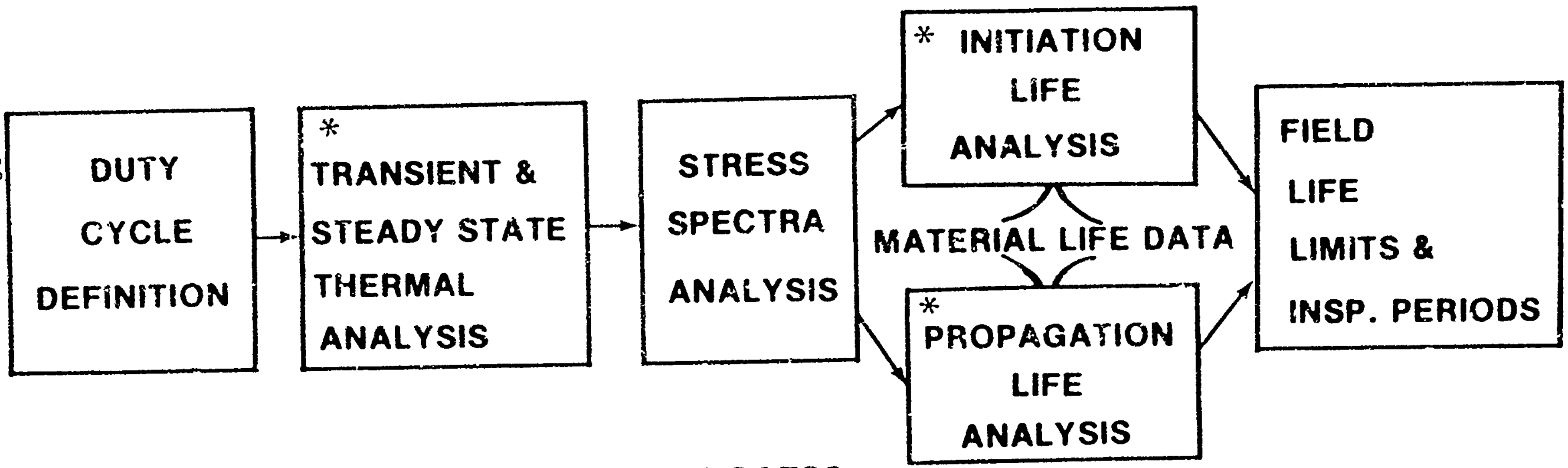



FATIGUE
6 INCHES
A
B
C

DESIGN VARIABLES
LOW CYCLE FATIGUE DESIGN

● DRIVER - MISSION THROTTLE TRANSIENTS





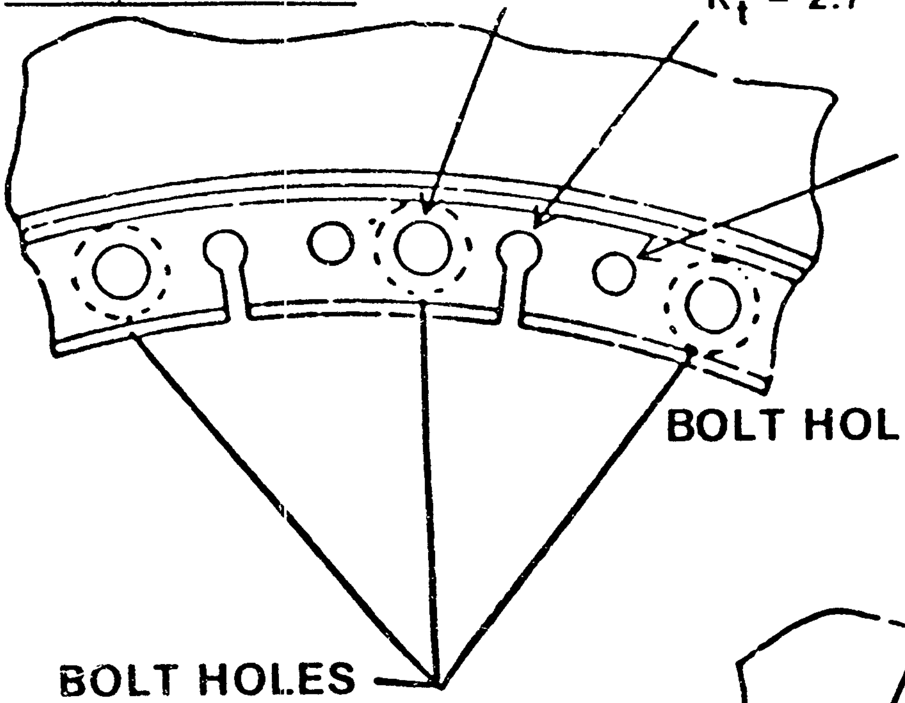
FATIGUE DESIGN PREDICTON PROCESS

* CONFIRM BY TEST

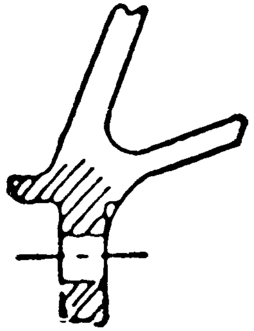
DESIGN VARIABLES
FATIGUE DESIGN

- FATIGUE LIFE - VERY SENSITIVE TO DETAIL DESIGN

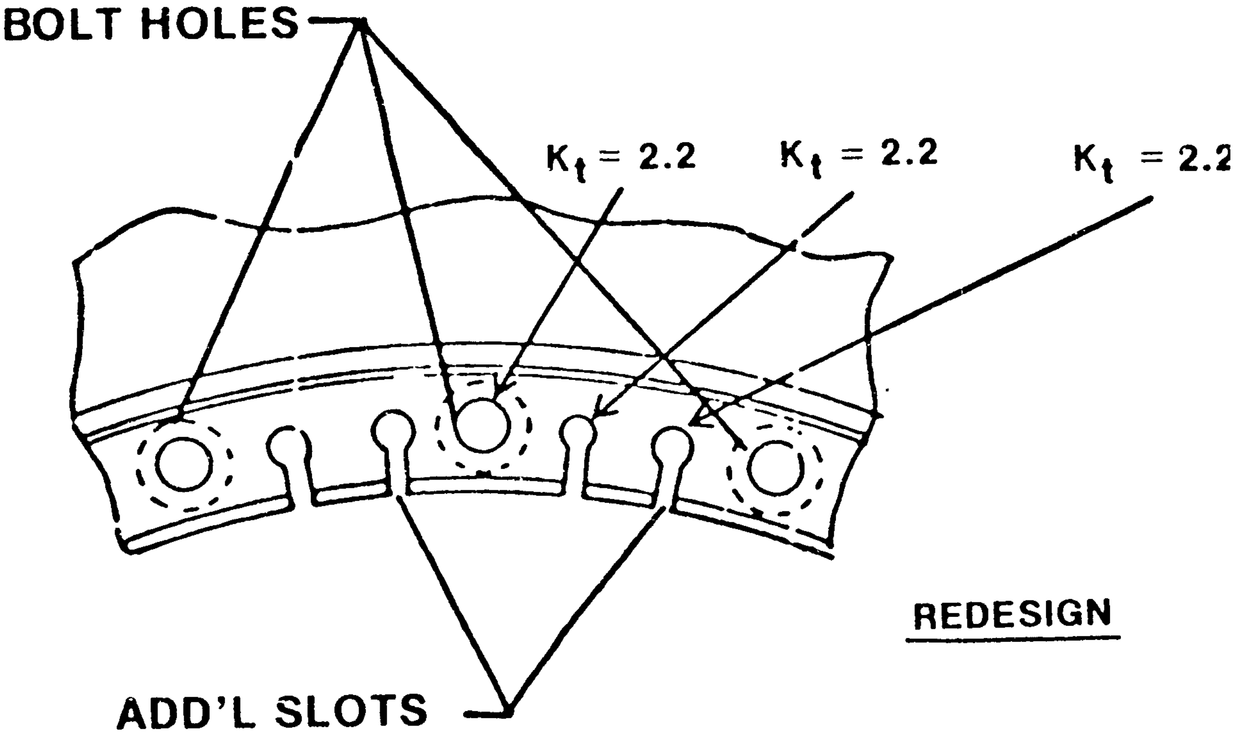
<u>ORIGINAL DESIGN</u>



LOCAL STRESS = K_t X GROSS AREA STRESS



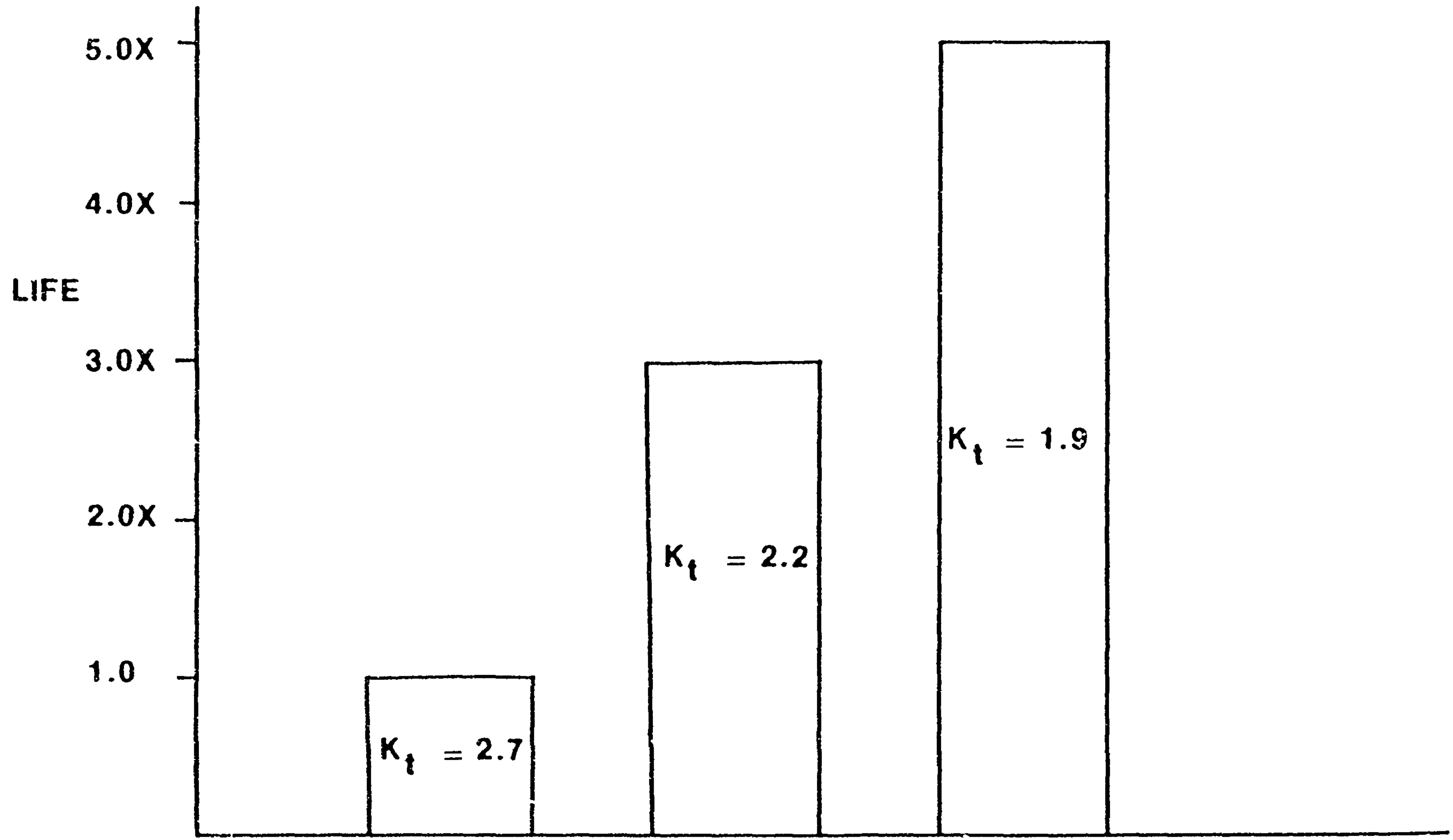
(K_t) - STRESS CONCENTRATION



<u>REDESIGN</u>

DESIGN VARIABLES

LIFE VERSUS STRESS CONCENTRATION FACTOR (K_t)



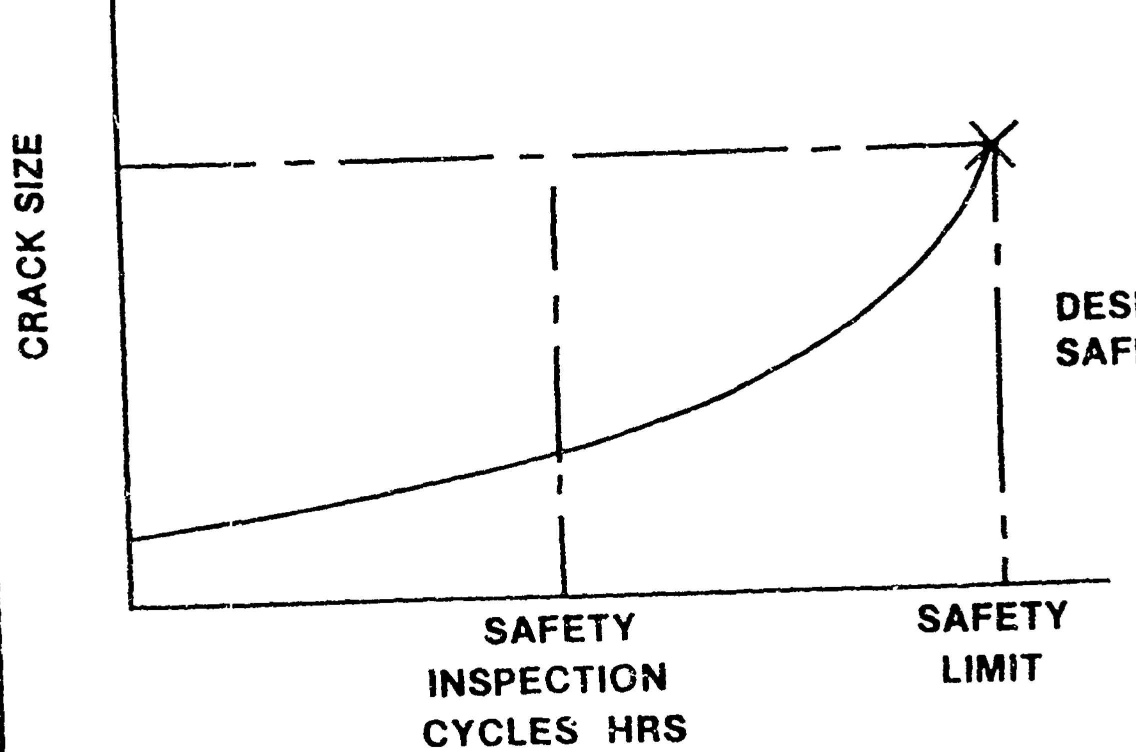
DESIGN VARIABLES
DAMAGE TOLERANCE

- THE ABILITY OF THE ENGINE TO RESIST FAILURE DUE TO THE PRESENCE OF FLAWS, CRACKS, OR OTHER DAMAGE FOR A SPECIFIED PERIOD OF TIME

DESIGN VARIABLES

DAMAGE TOLERANCE DESIGN CONCEPT

- FRACTURE CRITICAL PART
  - A PART, IF FAILURE OCCURS, WOULD JEOPARDIZE FLIGHT SAFETY



DESIGN VARIABLES AFFECTING SAFETY LIMIT

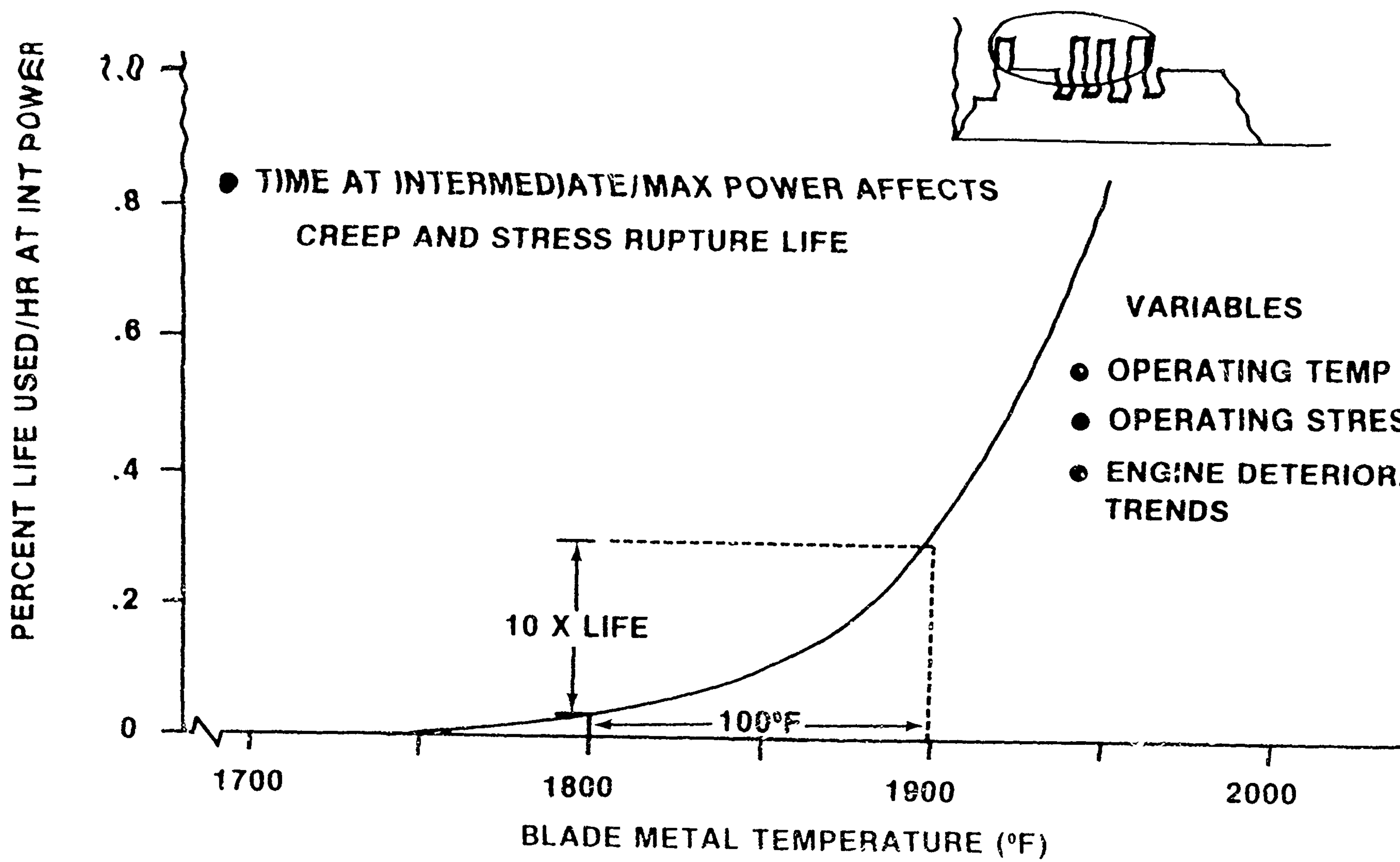
- INITIAL FLAW SIZE
- STRESS LEVELS
- DUTY CYCLE (SPECTRUM)
- MATERIAL CHARACTERISTICS
- DESIGN DETAILS
- ENVIRONMENT
  - VIBRATION
  - TEMPERATURE

DESIGN VARIABLES

TEMPERATURE EFFECTS

ON LIFE

DESIGN VARIABLES
BLADE LIFE SENSITIVITY TO TEMPERATURE



- TIME AT INTERMEDIATE/MAX POWER AFFECTS CREEP AND STRESS RUPTURE LIFE

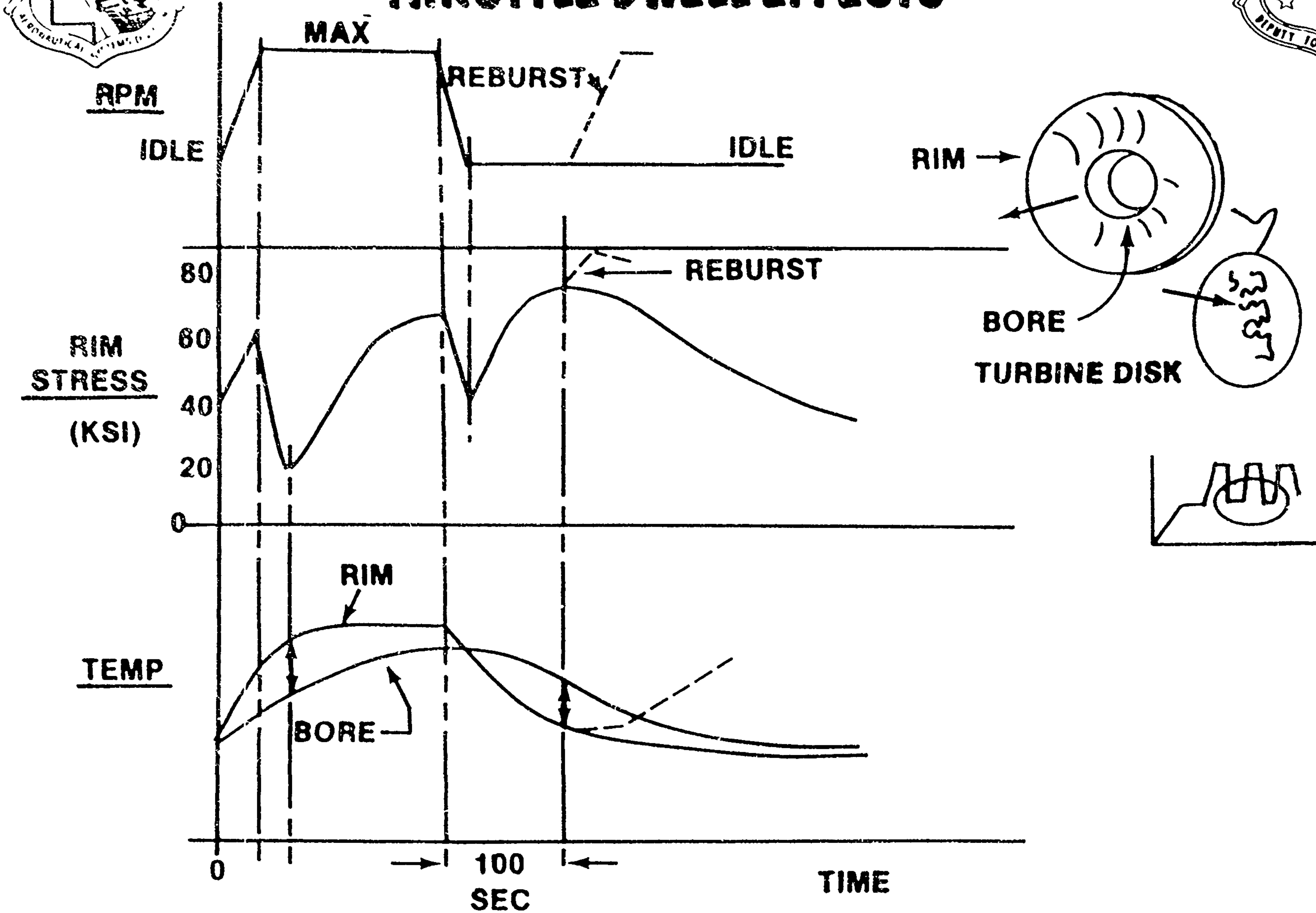
VARIABLES

- OPERATING TEMP
- OPERATING STRESS
- ENGINE DETERIORATION TRENDS

DESIGN VARIABLES

DWELL EFFECTS

DESIGN VARIABLES
THROTTLE DWELL EFFECTS

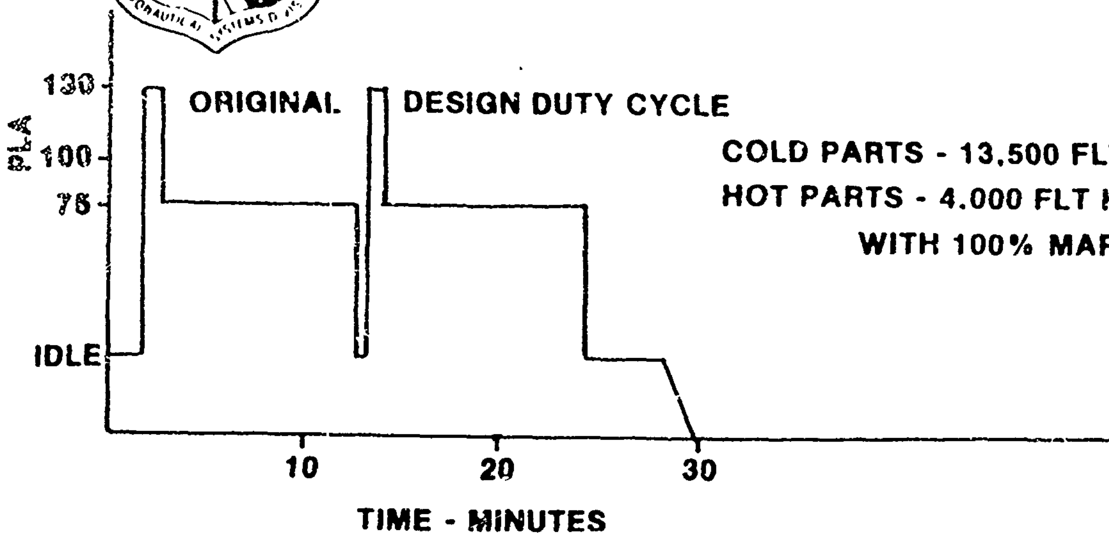


DESIGN VARIABLES

EXAMPLE OF MISSION

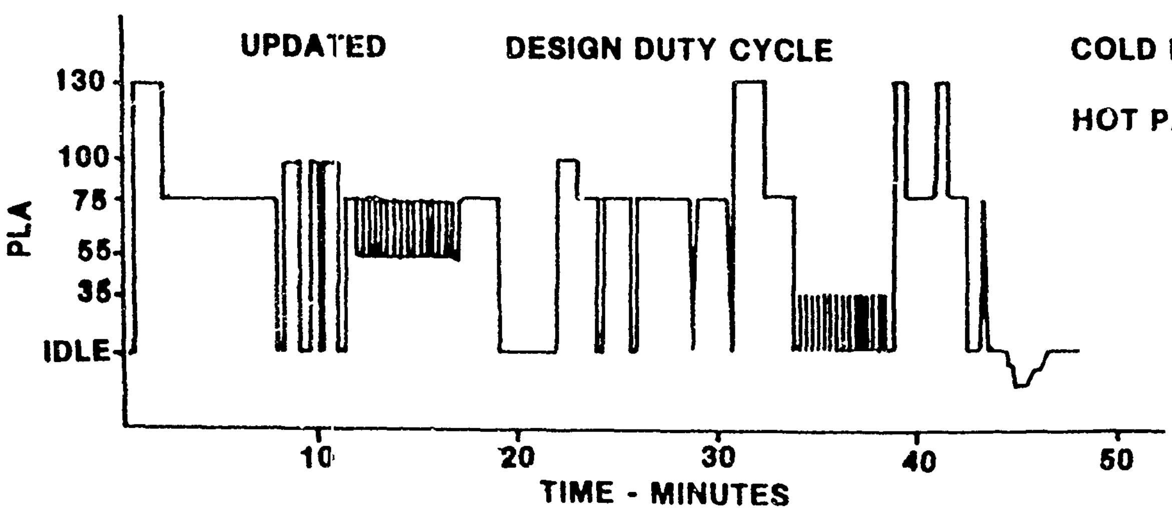
EFFECTS ON LIFE

DESIGN VARIABLES
MISSION EFFECTS ON LIFE (EXAMPLES)



COLD PARTS - 13,500 FLT HRS
HOT PARTS - 4,000 FLT HRS
WITH 100% MARGIN

	ORIGINAL	UPDATED
TAMT (HRS)	285	424 ←
LCF (CYCLES)	2700	2540
FTC (CYCLES)	5400	30000 ←
A/B (CYCLES)	2400	4578 ←
A/B TIME (HRS)	60.8	7[illegible]6.



COLD PARTS - 10,000 FLT HRS
HOT PARTS - 3,000 FLT HRS

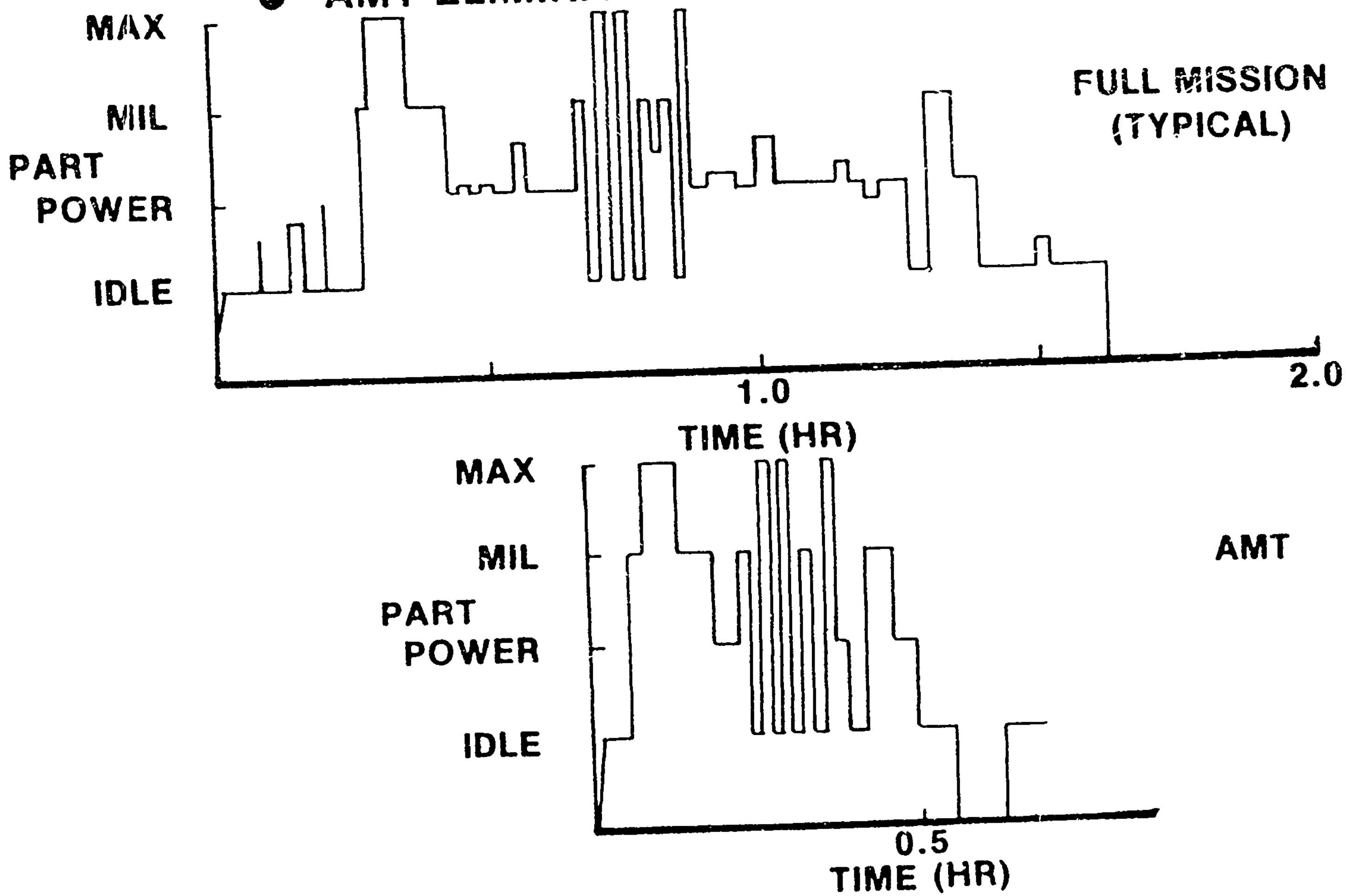
TESTING FOR ENGINE DURABILITY
AMT TESTING

- AN IMPORTANT ADVANCEMENT IN ENGINE DEVELOPMENT

- ACCELERATED MISSION TESTING REFLECTS ACTUAL FIELD USAGE AND FAILURE MODES

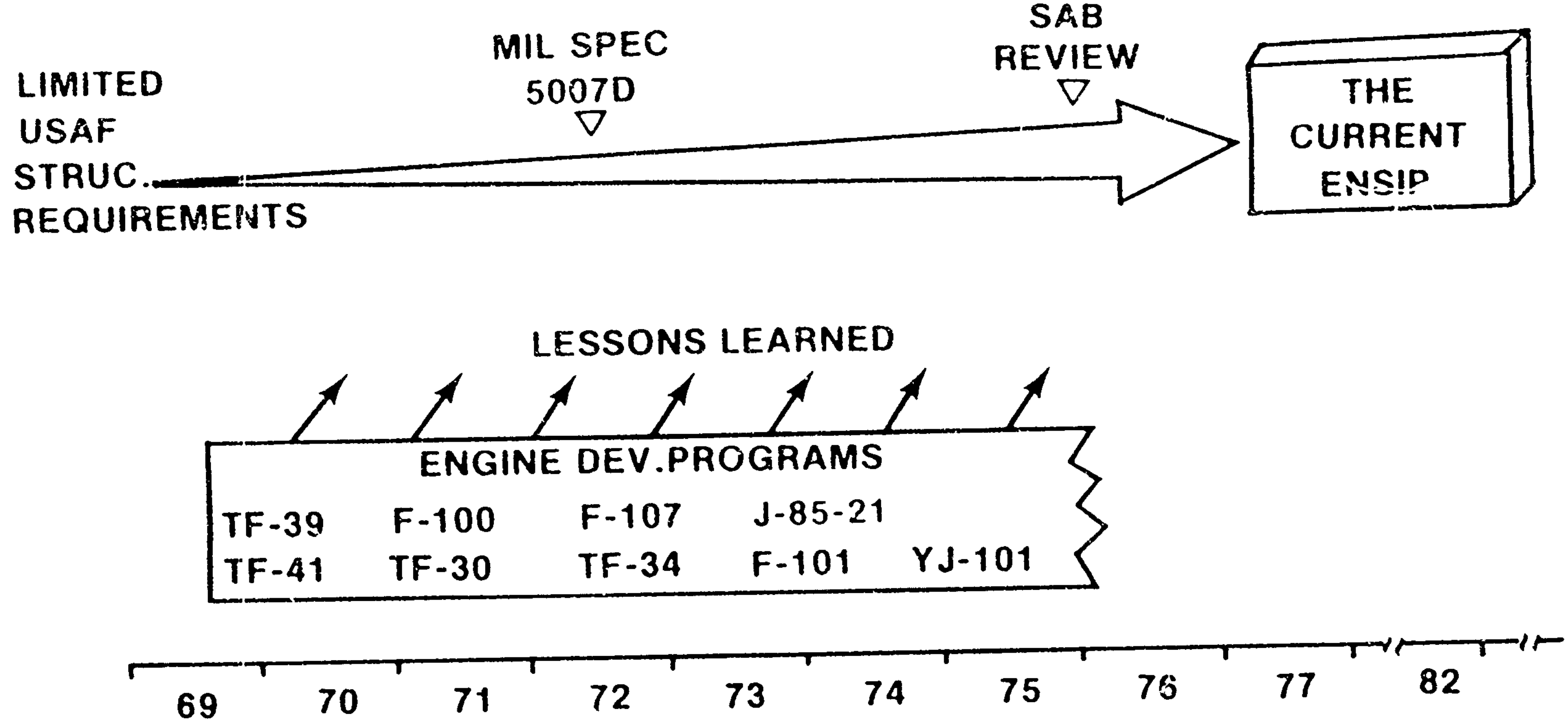
TESTING FOR ENGINE DURABILITY
ACCELERATED MISSION TESTING (AMT)

- AMT ELIMINATES NONDAMAGING TESTING



...BUT RETAINS DAMAGING CYCLES AND HOT TIME

HISTORICAL PERSPECTIVE
THE EVOLUTION OF ENSIP



- STRUCTURAL REQUIREMENTS HAVE BEEN TRANSITIONING INTO NEW ENGINE DEV. FOR PAST TWELVE YEARS

B938

INTRODUCTION TO ENSIP
THE 1976 S.A.B. ASSESSMENT

........WE NEED TO APPLY A SYSTEM OF DISCIPLINE TO OUR DEVELOPMENT PROCEDURES.

........AIR FORCE SHOULD DEFINE AN AGGRESSIVE PROGRAM FOR ENGINE MECHANICAL AND STRUCTURAL INTEGRITY AND DURABILITY. ....THIS PROGRAM SHOULD BE REQUIRED BY REGULATION.

........DURABILITY & DAMAGE TOLERANCE ASSESSMENTS SHOULD BE PERFORMED ON FLEET ENGINES ANALOGOUS TO THOSE BEING PERFORMED ON SEVERAL WEAPON SYSTEM AIRFRAMES.

B938-16

INTRODUCTION TO ENSIP
ENSIP
(ENGINE STRUCTURAL INTEGRITY PROGRAM)

- ENSIP IS AN ORGANIZED AND DISCIPLINED APPROACH TO THE STRUCTURAL DESIGN, ANALYSIS, DEVELOPMENT, PRODUCTION, AND LIFE MANAGEMENT OF GAS TURBINE ENGINES WITH THE GOAL OF ENSURING:

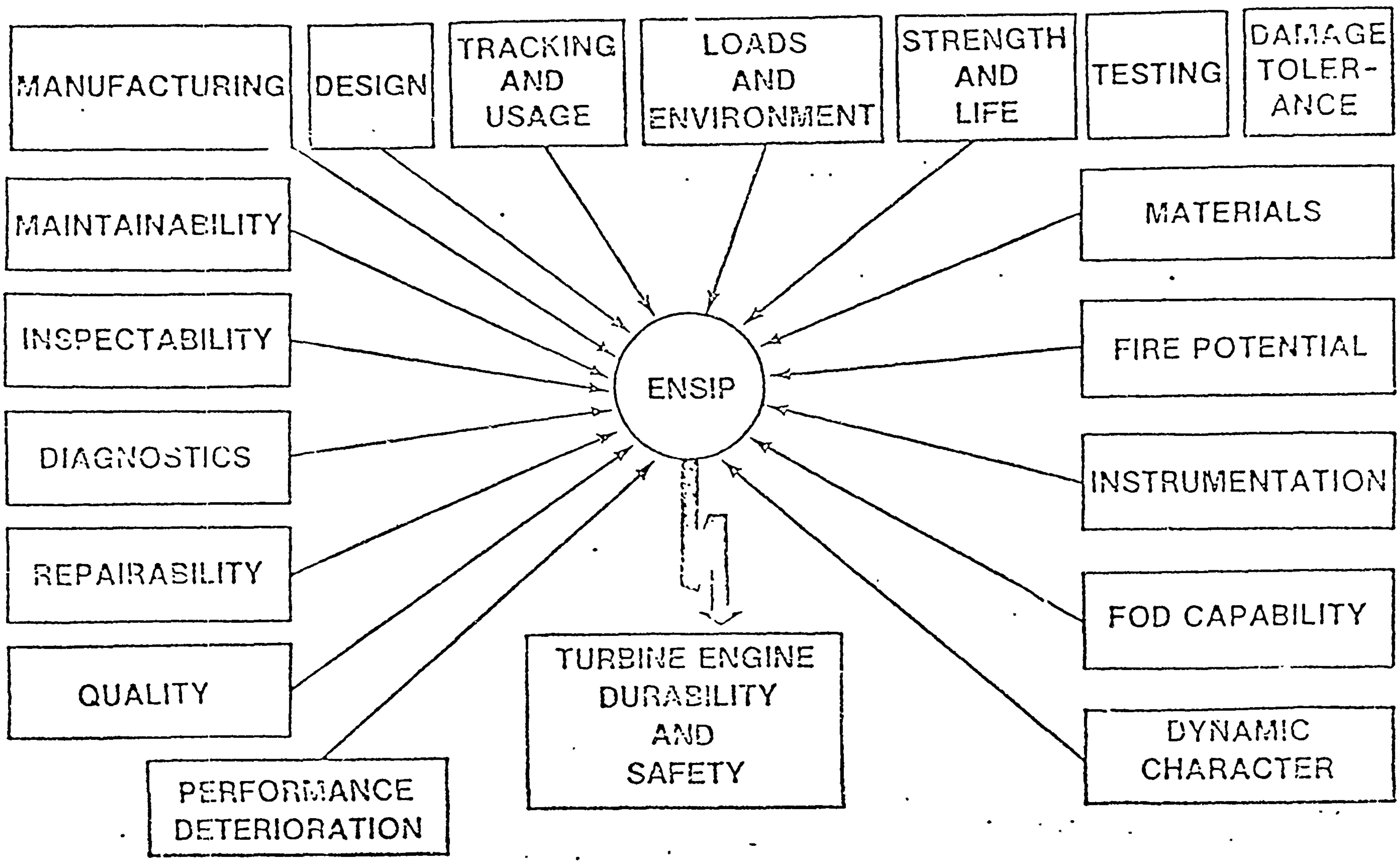
  A) ENGINE STRUCTURAL SAFETY,

  B) INCREASE SERVICE READINESS AND,

  C) REDUCE LIFE CYCLE COSTS

THE SCOPE

(MANY SPECIFIC DISCIPLINES)



INTRODUCTION TO ENSIP

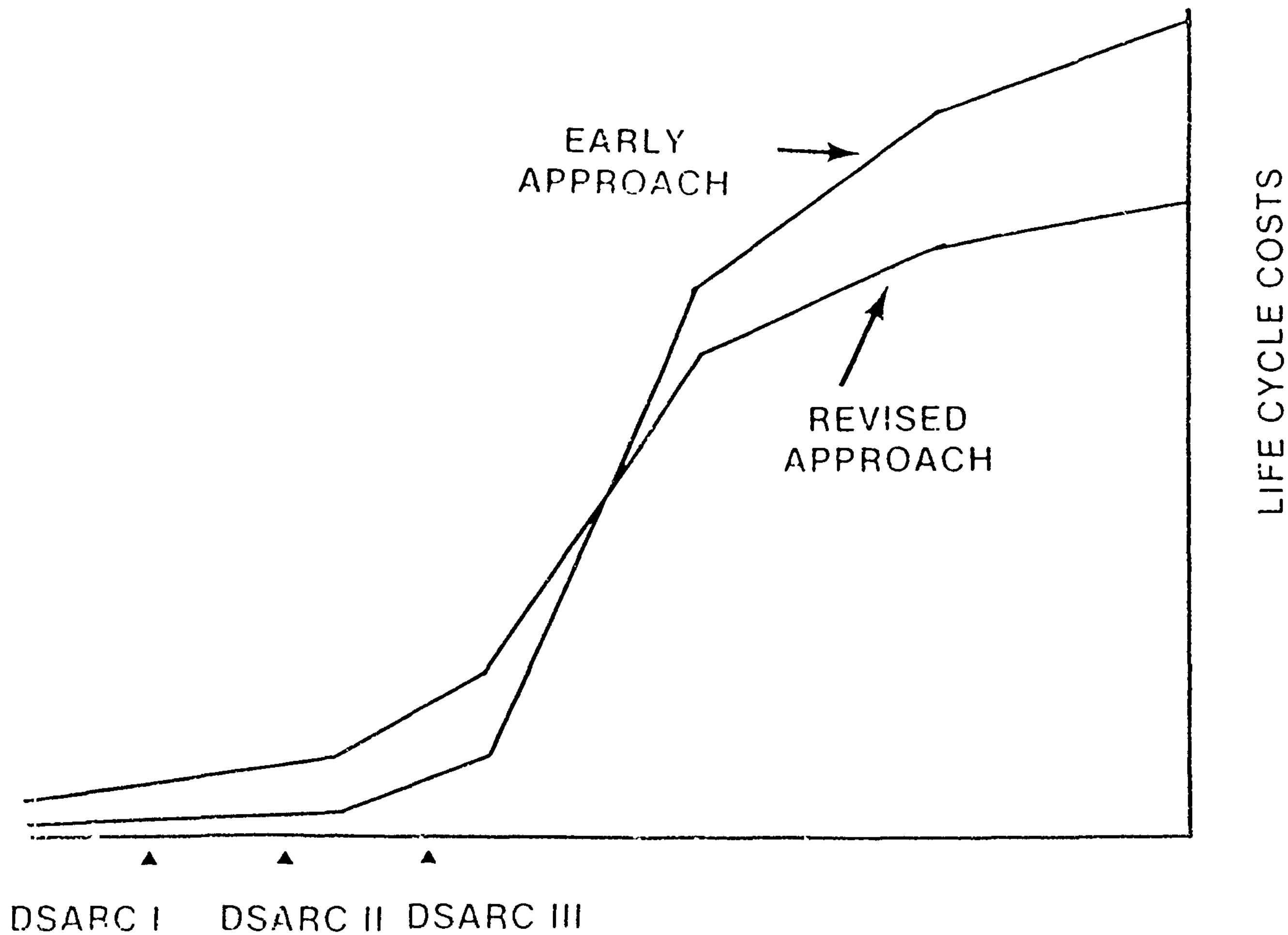
THE ENSIP TASKS

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANAL. COMPNT & MAT. CHARAC.	COMPONENT & CORE ENG. TESTING	GROUND & FLIGHT ENG. TESTS	PROD. QUAL. CONTROL & ENG. LIFE MGT.
• ENSIP MASTER PLAN • DESIGN SERV. LIFE & USAGE REQUIREMENTS • DESIGN CRITERIA	• DESIGN DUTY CYCLE • MAT'LS AND PROCESSES DESIGN DATA CHARACTERIZED • STRUCTURAL/ THERMAL ANALYSIS • MFG. AND QUALITY CONTROL	• STRENGTH TESTING • DAMAGE TOLERANCE TESTS • DURABILITY TESTS • THERMAL SURVEY • VIBRATORY STRAIN & FLUTTER BOUNDARY SURVEY	• ENVIR. VERIF. TESTING • (AMT) TEST SPEC. DERIV. • DURABILITY TESTS (AMT) • DAMAGE TOL. TESTS • FLIGHT TEST STRAIN SURVEY • UPDATED DURA. & DAM. TOL. CONTROL PLAN • PERFRM. DETERIOR. STRUC. IMPACT ASSESSMENT • CRITCL. PART UPDATE	• PROD. ENG. ANALYSIS • STRUC. SAFETY & DURAB. SUM. • ENG. STRUC. MAINT. PLAN • INDIV. ENG. TRACKING • LEAD THE FORCE PROG. (USAGE) • DURA. & DAM. TOL. CONTROL PLAN IMPL. • TECHNICAL ORDER UPDATE

INTRODUCTION TO ENSIP
SPECIAL FEATURES

- STRUCTURAL SAFETY (DAMAGE TOLERANCE) DESIGN CRITERIA
- DURABILITY/ECONOMIC LIFE CRITERIA
- MAINTAINABILITY CRITERIA
- MATERIAL AND PROCESS CHARACTERIZATION PLAN
- ENVIRONMENTAL DEFINITION
- COMPREHENSIVE GROUND TEST POLICY
- USAGE AND TRACKING PROGRAM
- ENGINE STRUCTURAL MAINTENANCE PLAN

LIFE CYCLE COST PROJECTIONS



INTRODUCTION TO ENSIP
THE IMPLEMENTATION PLAN

- FUTURE ENGINES
  - DEVELOP & PUBLISH
    - USAF REG. ON ENSIP
    - PRIME SPEC. ON ENSIP
    - DESIGN HANDBOOK

MIL-STND-1783

- PRIMARY PURPOSE
  - DEFINE RESPONSIBILITIES
  - DEFINE REQMT'S
  - PROVIDE RATIONALE AND METHODS OF COMPLIANCE

- CURRENT DEVELOPMENT ENGINES
- EXISTING ENGINES
  - RECOMMEND CANDIDATE ENGINES AND PRIORITIES FOR POSSIBLE DURABILITY AND DAMAGE TOLERANCE ASSESSMENTS (DADTA'S)
    - F-100 - COMPLETED IN 1979
    - TF-34 - COMPLETED IN 1982

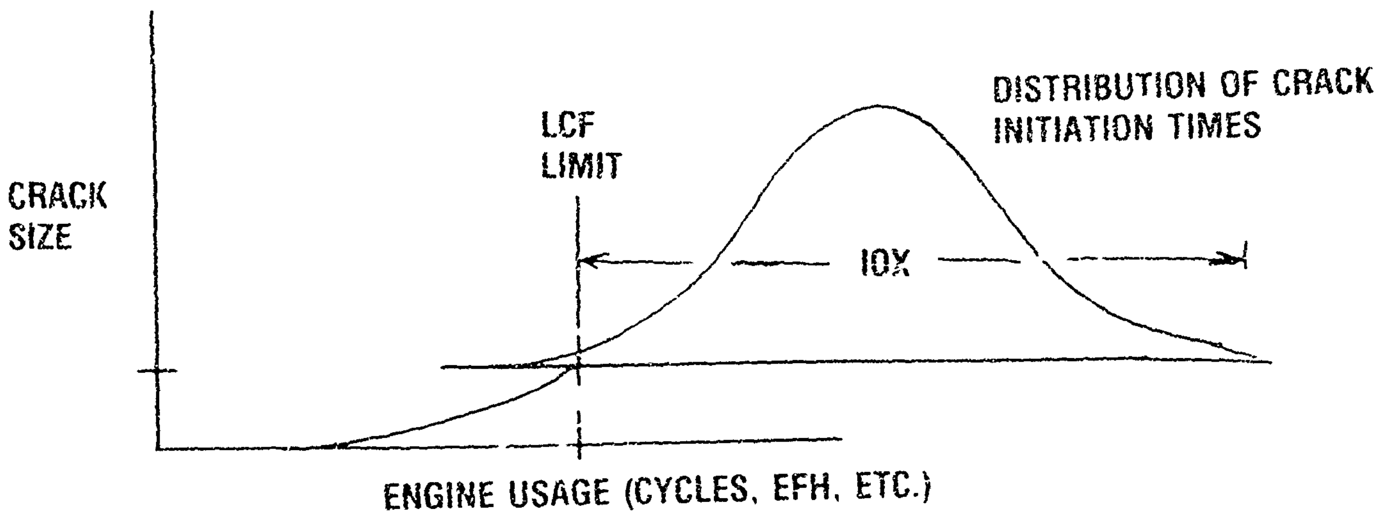
AERONAUTICAL SYSTEMS DIVISION

DAMAGE TOLERANT DESIGN CONCEPTS FOR MILITARY ENGINES

THE ENSIP TASKS

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANAL MAT'L CHARACT & DEV TESTS	COMPONENT & CORE ENGINE TESTS	GROUND & FLIGHT ENGINE TESTS	ENGINE LIFE MANAGEMENT
DEVELOPMENT PLANS • ENSIP MASTER • DURABILITY & DAM TOL CONTROL • MAT'L & PROCESS CHARACTERIZATION • CORROSION PREV & CONTROL • INSPECTION & DIAGNOSTICS OPERATIONAL RQMTS • DESIGN SERVICE LIFE & DESIGN USAGE RQMTS • DESIGN CRITERIA	• DESIGN DUTY CYCLE • MATERIAL CHARACTERIZATION • DESIGN DEV TESTS • STRUCTURAL/ THERMAL ANALYSIS • INSTALLED ENGINE INSPECTABILITY • MFG AND QUALITY CONTROL	COMPONENT TESTS • STRENGTH • VIBRATION • DAMAGE TOLERANCE • DURABILITY CORE ENGINE TESTS • THERMAL SURVEY • VIBRATION STRAIN & FLUTTER BOUNDARY SURVEY	GROUND ENGINE TESTS • STRENGTH • DAMAGE TOLERANCE • ACCELERATED MISSION TEST (AMT) • THERMAL SURVEY • VIBRATION STRAIN & FLUTTER BOUNDARY SURVEY FLT ENGINE TESTS • FAN STRAIN SURVEY • THERMAL SURVEY • INSTALLED VIBRATION • DETERIORATION	• UPDATED ANALYSES • STRUCTURAL MAINTENANCE PLAN • OPERATIONAL USAGE SURVEY • INDIVIDUAL ENGINE TRACKING • DURABILITY & DAMAGE TOLERANCE CONTROL ACTIONS (PRODUCTION)

CONVENTIONAL APPROACH TO LIFE MANAGEMENT OF CYCLIC LIMITED ENGINE COMPONENTS



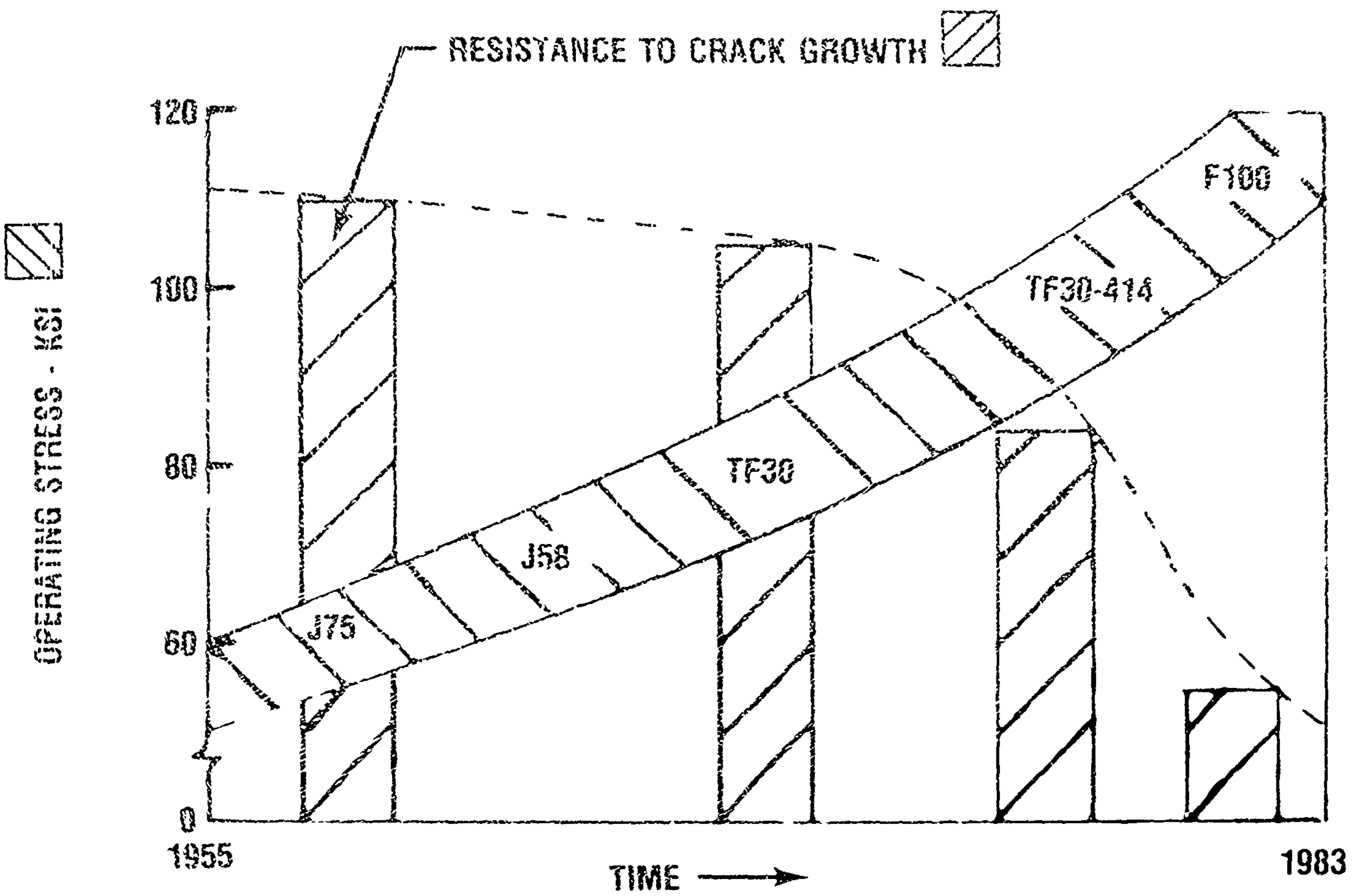
CRITERIA: LOW CYCLE FATIGUE (LCF) LIMIT BASED ON LOWER BOUND (-3σ OR 1/1000) DISTRIBUTION OF CRACK INITIATION TIME

ACTION: 100% PART REPLACEMENT AT LCF LIMIT

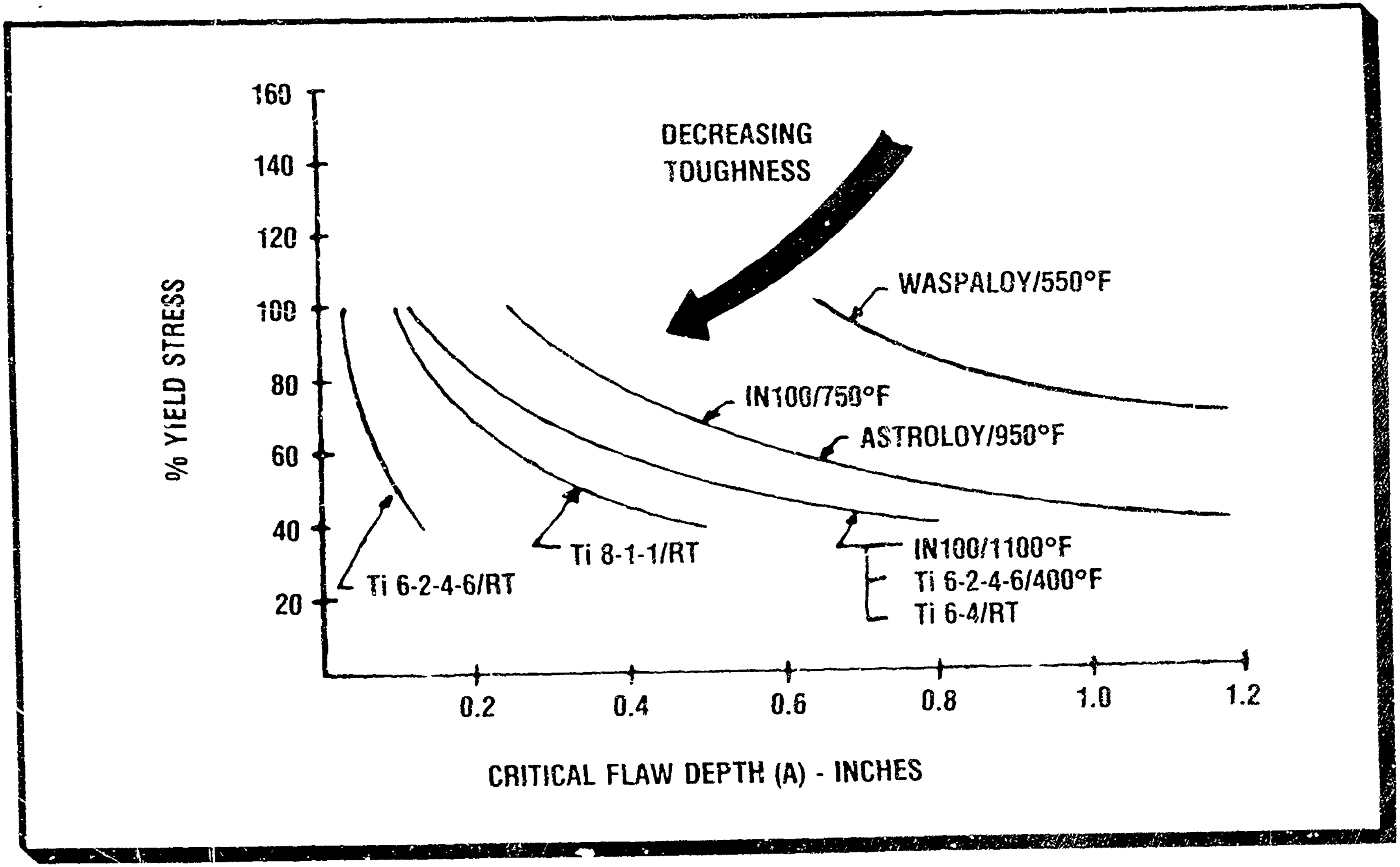
CONCERNS: NO RECOGNITION OF IMPACT INITIAL DEFECTS CAN HAVE ON TOTAL PART LIFE (PART FAILURE CAN OCCUR PRIOR TO LCF LIMIT)

MAJORITY OF PARTS MAY BE DISCARDED PRIOR TO REACHING THEIR INDIVIDUAL CRACK INITIATION TIME (I.E., IF LCF LIMIT IS LESS THAN FULL LIFE)

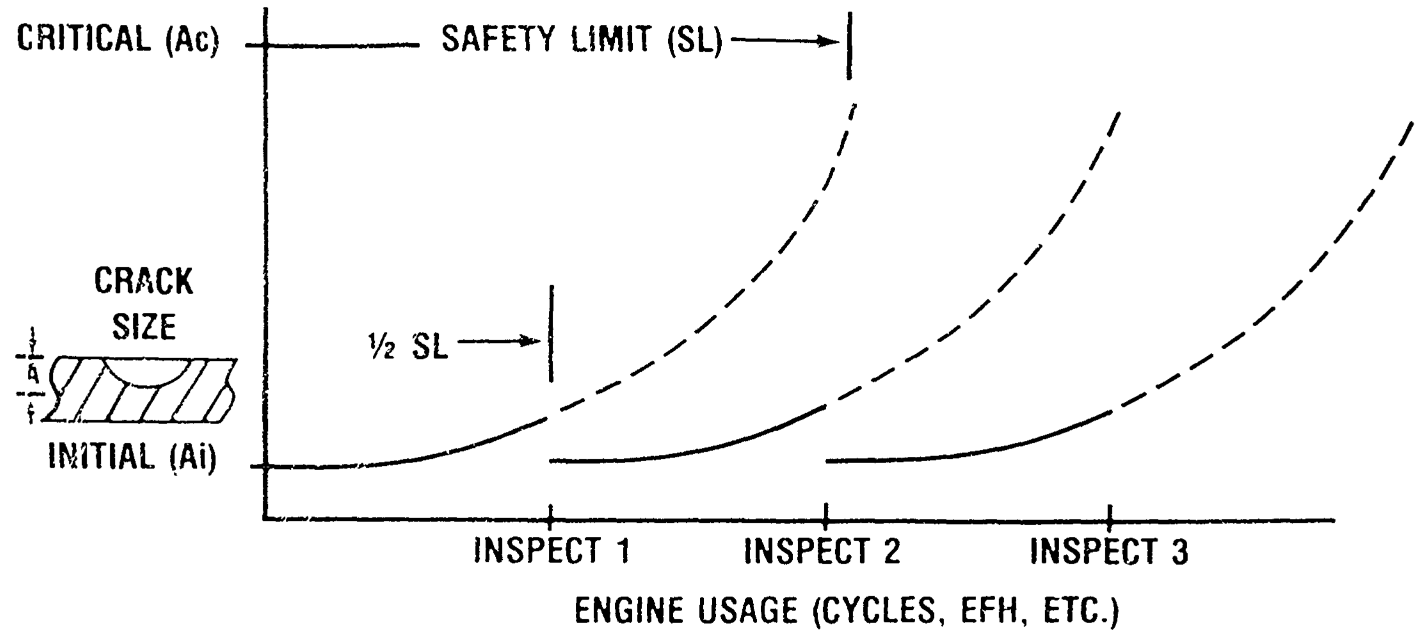
SOME MATERIAL TRENDS



SOME MATERIAL TRENDS



DAMAGE TOLERANCE APPROACH TO LIFE MANAGEMENT OF CYCLIC LIMITED ENGINE COMPONENTS



<u>CRITERIA:</u>

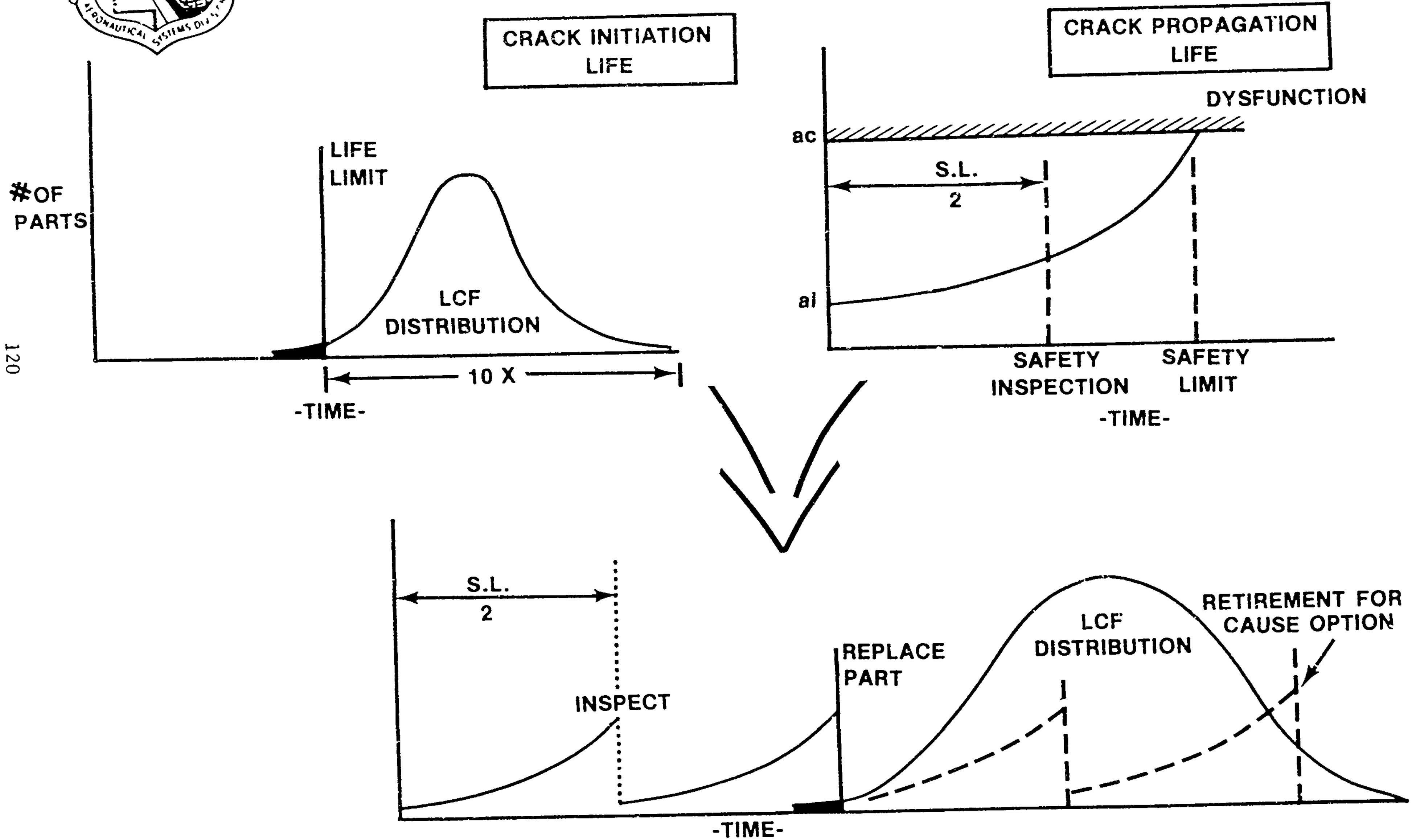
SAFETY LIMIT OR RESIDUAL LIFE IS TIME FOR INITIAL FLAW (Ai) TO GROW AND CAUSE PART FAILURE

INSPECT AT ½ SL. INSPECTION INTERVAL ≥ 1 LT (DESIGN GOAL) OR 1 DEPOT INTERVAL (MIN DESIGN REQMT)

APPLIES ONLY TO FRACTURE CRITICAL PARTS

Ai P^^ED ON NDI METHOD OR MATERIAL DEFECT DISTRIBUTION (FOR IMBEDDED DEFECTS)

ENSIP STRUCTURAL DESIGN PHILOSOPHY
(FRACTURE CRITICAL PARTS)



RETIREMENT FOR CAUSE
COST SAVINGS F-100 ENGINE ONLY

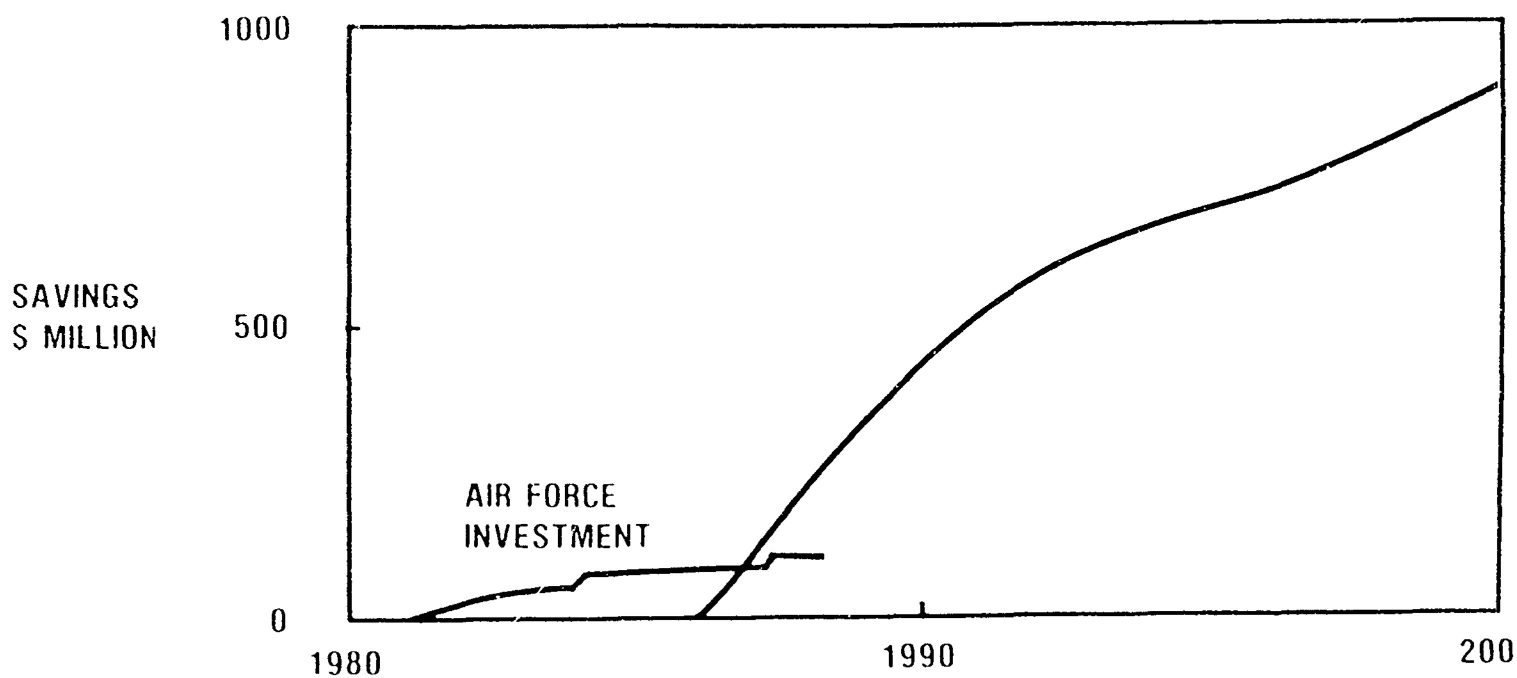


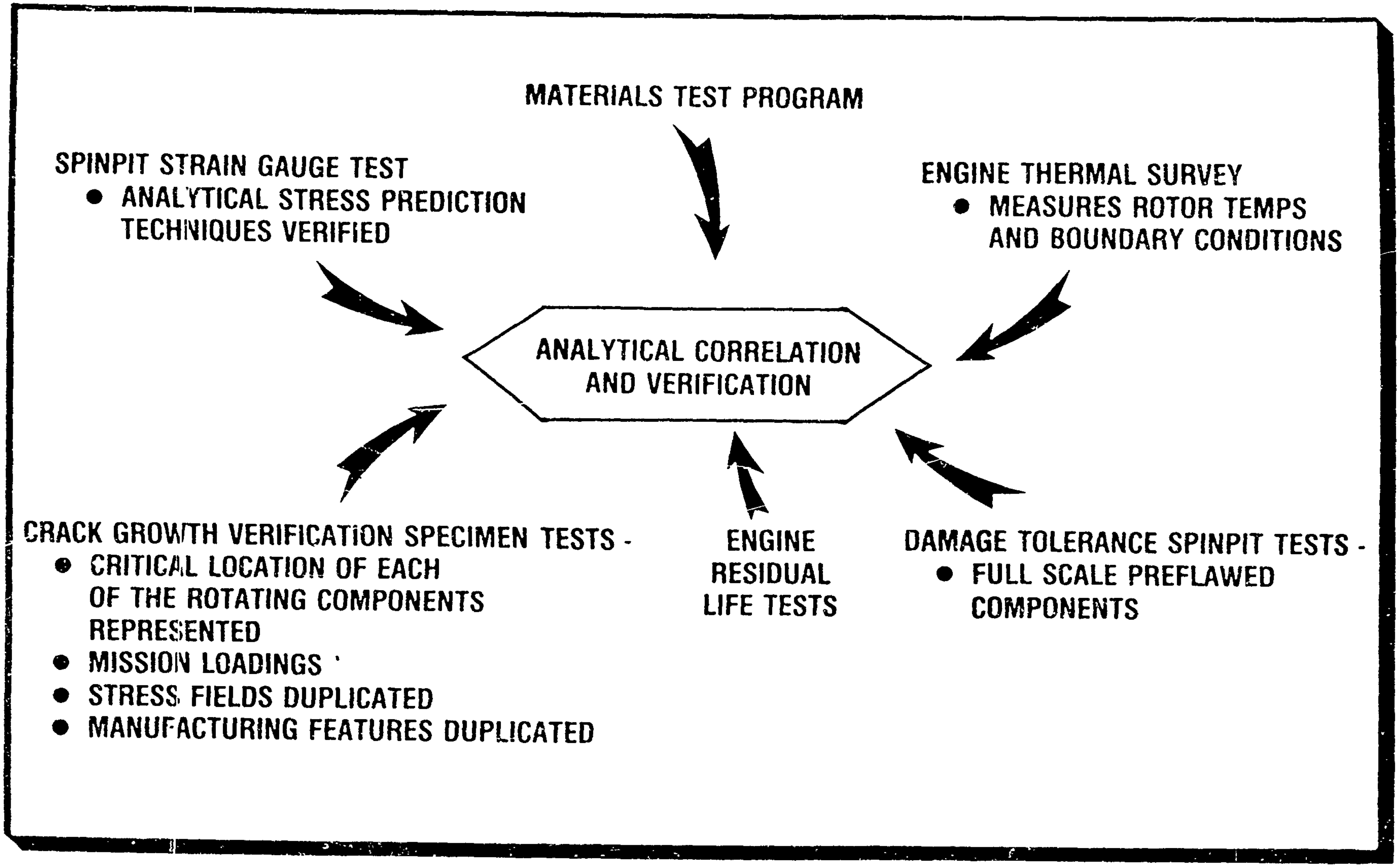
FIGURE 22

VERIFICATION

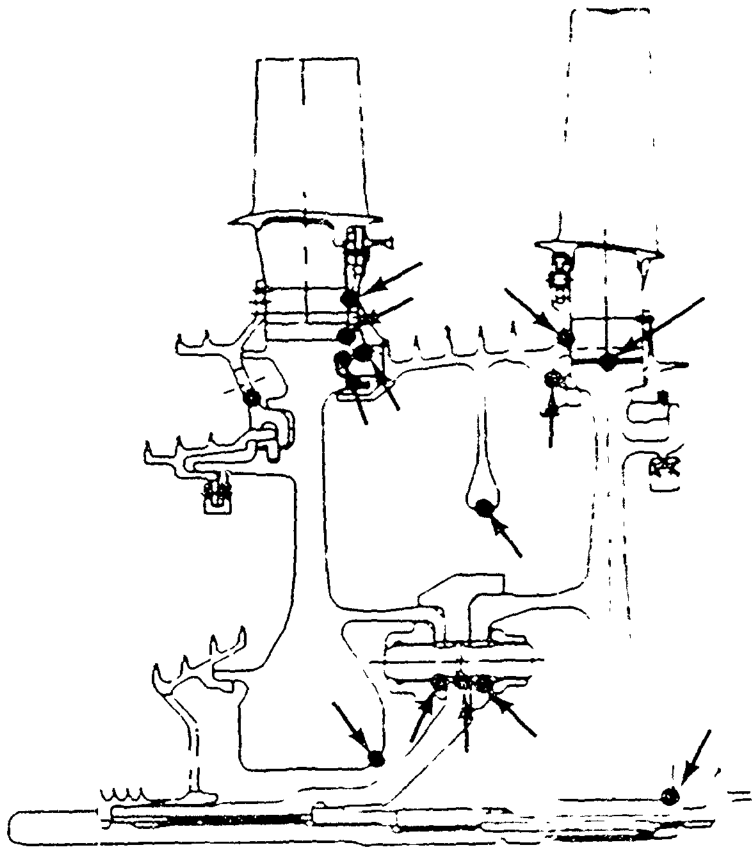
TASKS - DAMAGE TOLERANCE CONTROL PLAN

1. TRADE STUDIES - DESIGN CONCEPTS/MATERIAL/WEIGHT/ PERFORMANCE/COST
2. ANALYSIS
3. DEVELOPMENT AND QUALIFICATION TESTS
4. FRACTURE CRITICAL PARTS LIST
5. ZONING OF DRAWINGS
6. BASIC MATERIALS FRACTURE DATA
7. MATERIAL PROPERTIES CONTROLS
8. TRACEABILITY
9. NONDESTRUCTIVE INSPECTION (NDI) REQUIREMENTS

RESIDUAL LIFE ANALYSIS AND TEST PROCEDURE



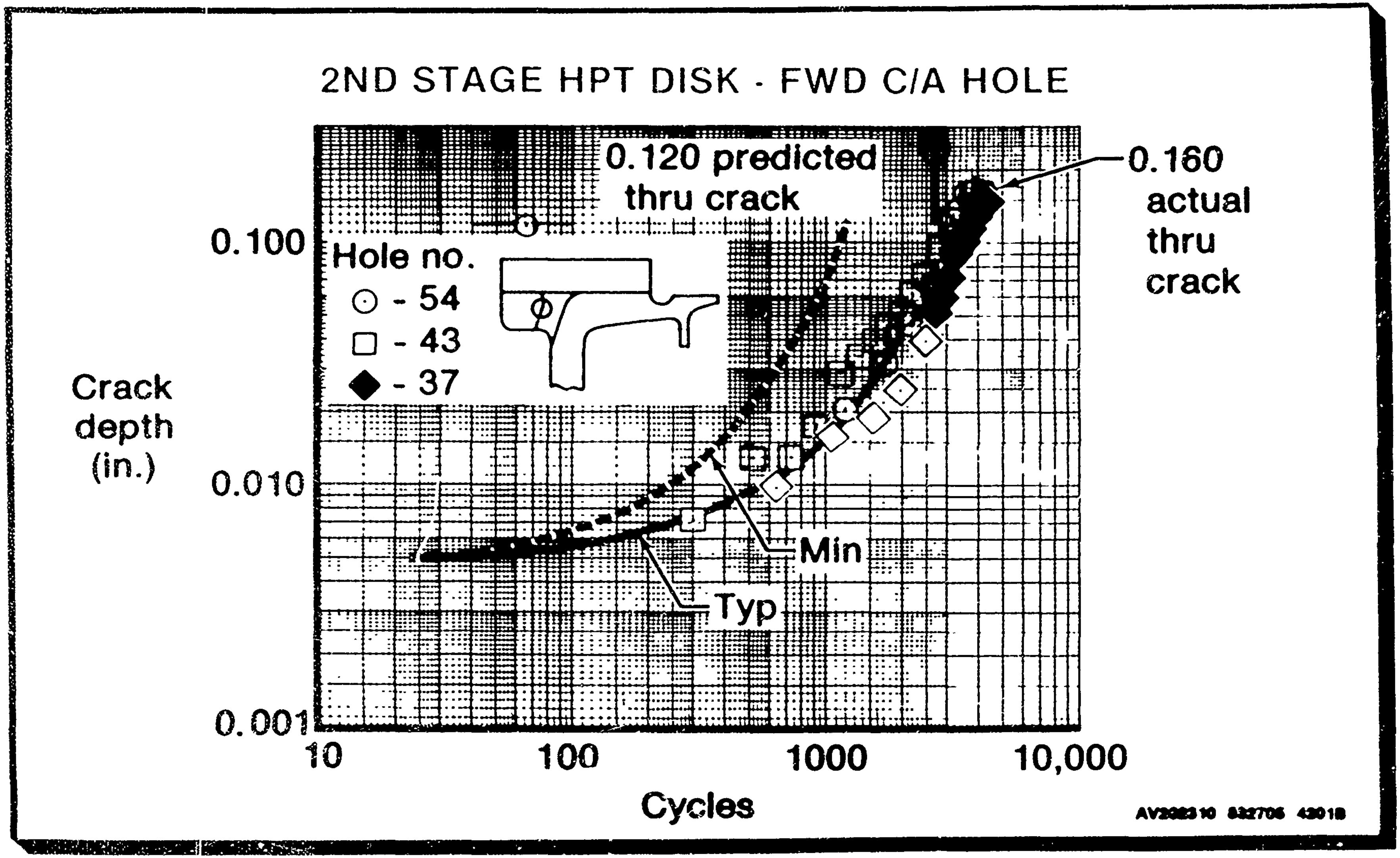
TURBINE ROTOR CYCLIC SPIN TEST



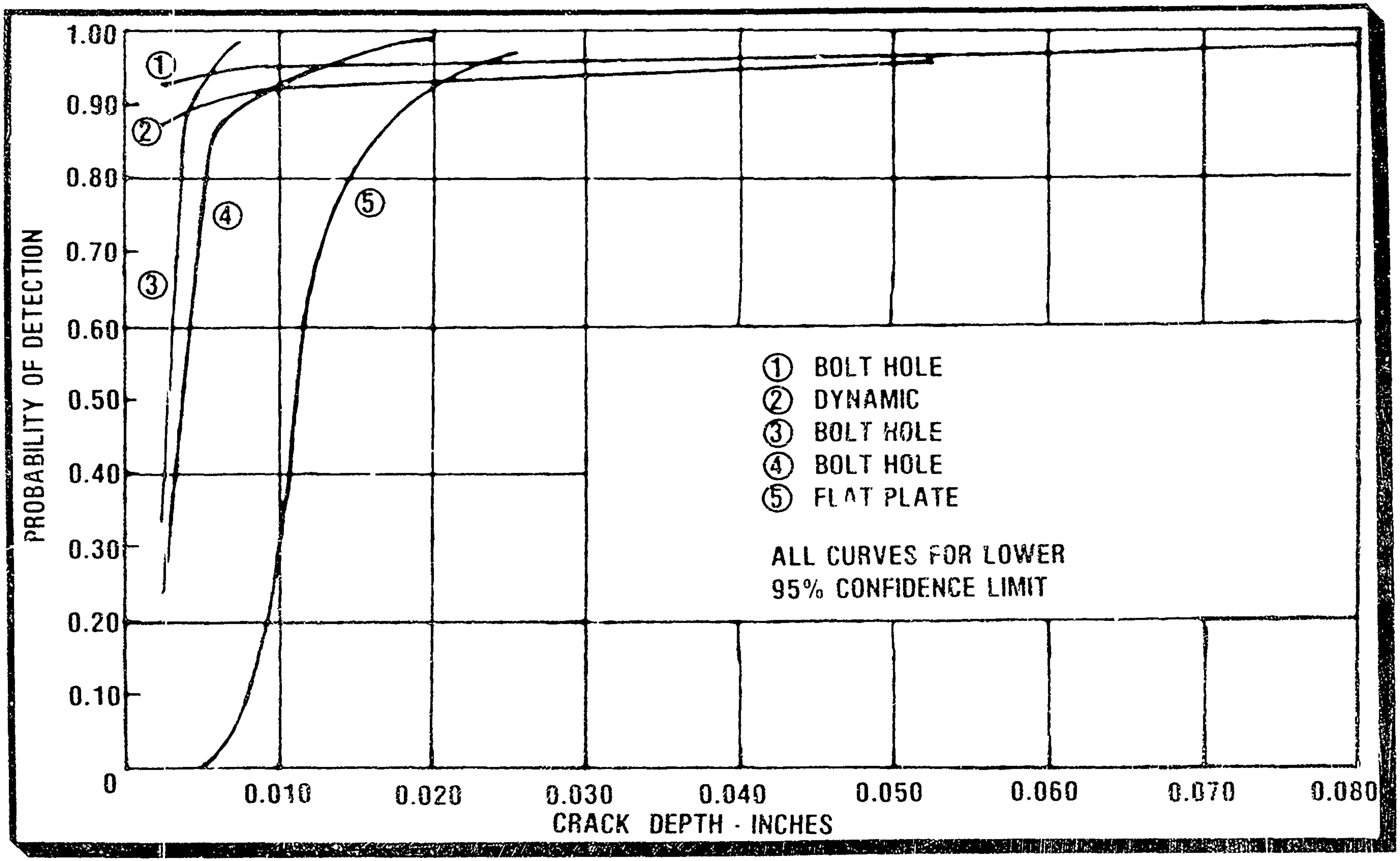
- HEATED SPIN PIT
- PREFLAWED AT CRITICAL LOCATIONS TO SIMULATE WORST EXPECTED DAMAGE

ACTUAL AND PREDICTED FLAW GROWTH

(F100 ENGINE TEST)



PROBABILITY OF DETECTION - EDDY CURRENT



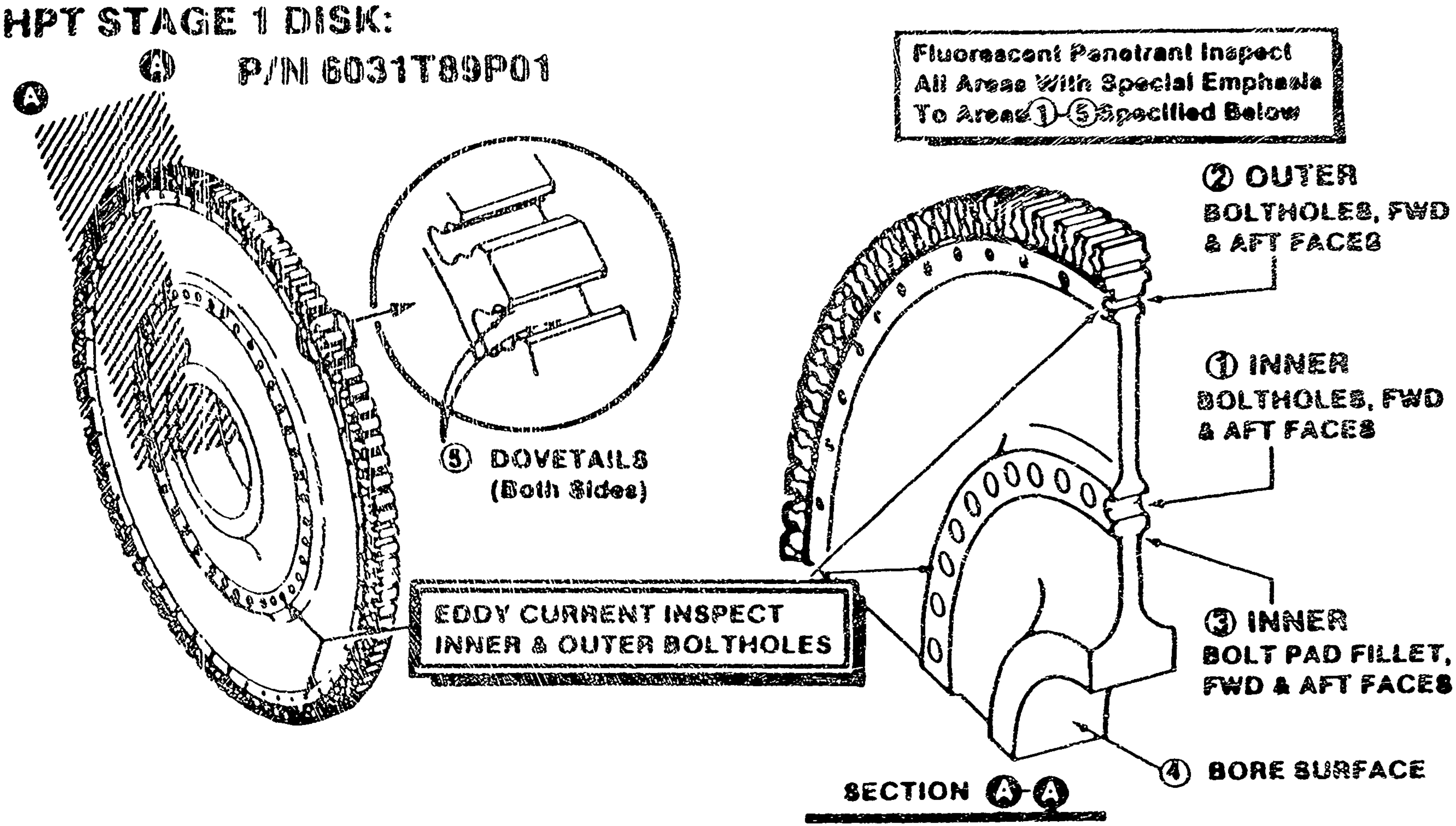
STATUS

IMPLEMENTATION OF UPGRADED NDI REQUIREMENTS

(INCLUDES FPI IMPROVEMENTS)

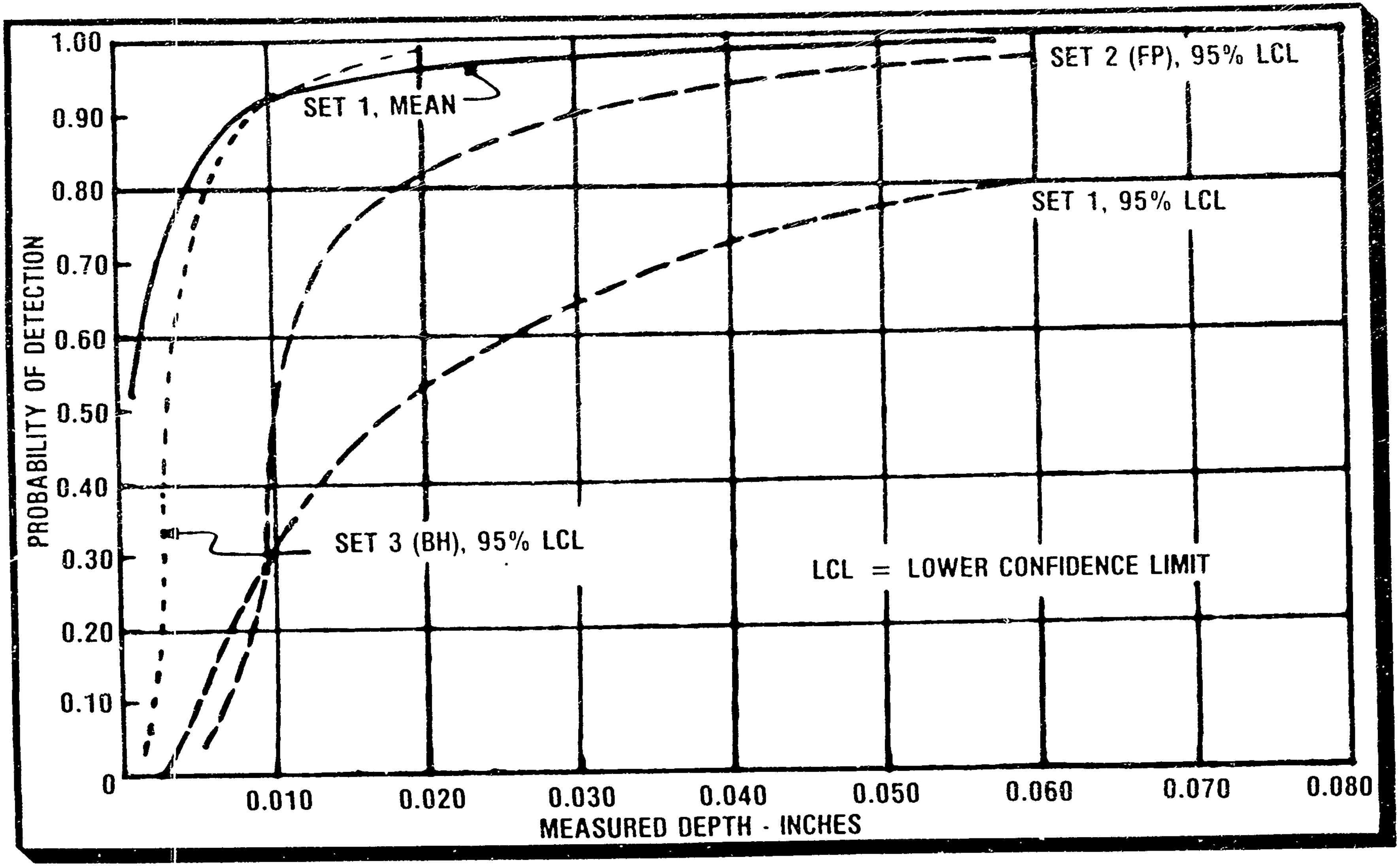
F100 ENGINE	
EDDY CURRENT @ DEPOT	3/80
EDDY CURRENT @ PRODUCTION	11/82
FAN DISK CRYOGENIC PROOF TEST	1/84
TF34 ENGINE	
EDDY CURRENT @ DEPOT	1/84-12/84
F101 ENGINE	IMPLEMENTATION
EDDY CURRENT @ PRODUCTION & DEPOT	PLAN DEVELOPED
FUTURE IMPLEMENTATIONS	BEING DEFINED
F110	
F109	
ATFE	

TF34-100 INSPECTION SHEET

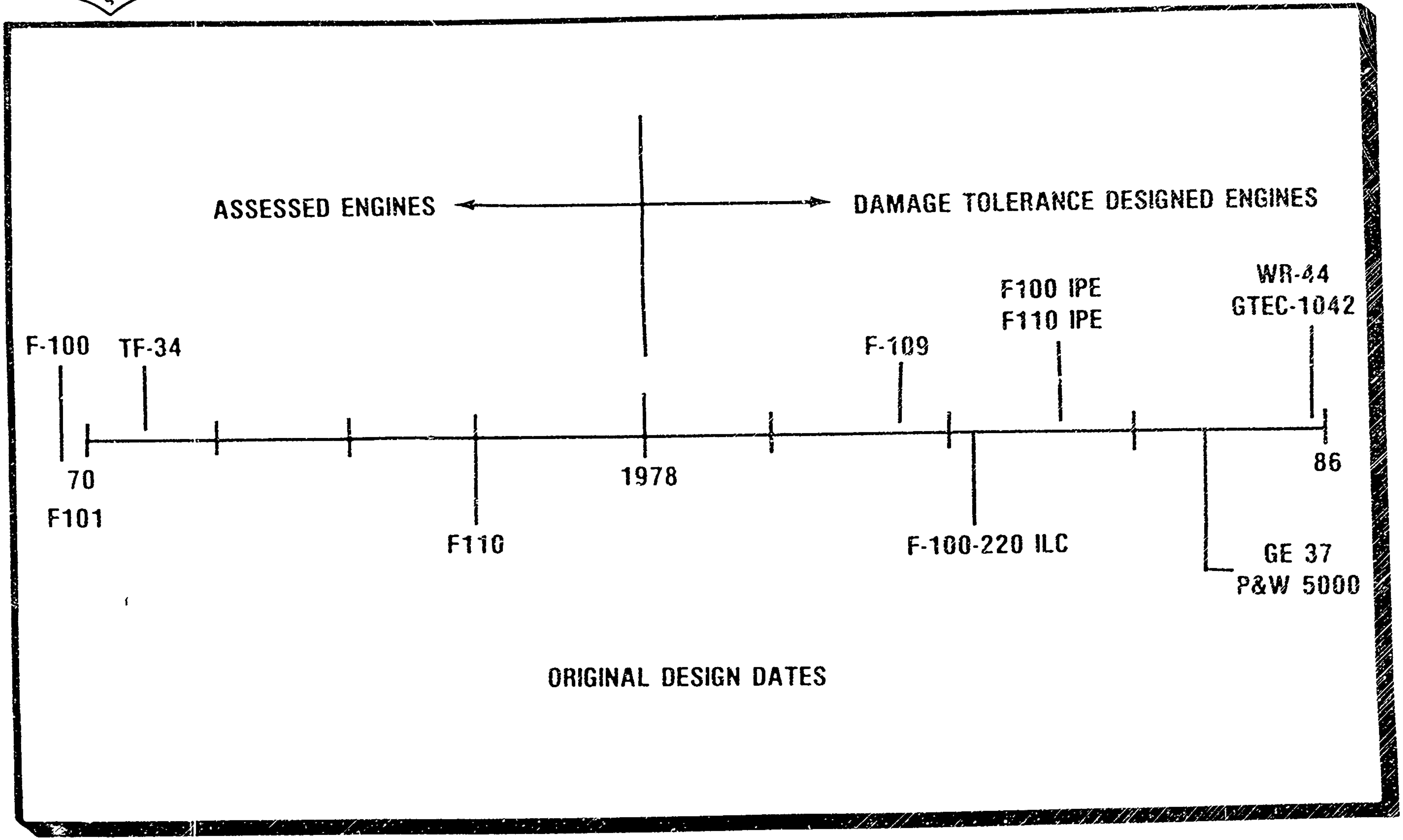


FOCUSED INSPECTIONS: LOCATIONS 1 - 5; FPI Group VI B (ZL30A*/ZR30A*/approved developer) *or equivalent
LOCATIONS 1 & 2: EDDY CURRENT

PROBABILITY OF DETECTION - FLUORESCENT PENETRANT INSPECTION (FPI)



ENSIP ENGINES



SUMMARY

- DAMAGE TOLERANCE REQUIREMENTS FUNDAMENTAL FEATURE OF ENSIP
  - MAINTAINS SAFETY IN PRESENCE OF QUALITY VARIATIONS (INITIAL DEFECTS)
- DAMAGE TOLERANCE IS COST EFFECTIVE
  - ADDS SOME UPFRONT COSTS
  - SUPPORT COSTS SIGNIFICANTLY REDUCED
  - TOTAL LIFE CYCLE COST BENEFIT
- FLUORESCENT PENETRANT INSPECTION (FPI) PROCESS IMPROVEMENTS MUST BE IMPLEMENTED
  - IMPROVE SMALL FLAW DETECTION RELIABILITY

SECOND SESSION
AIRFRAME STRUCTURES

Chairman

C. 'Pete' Petrin

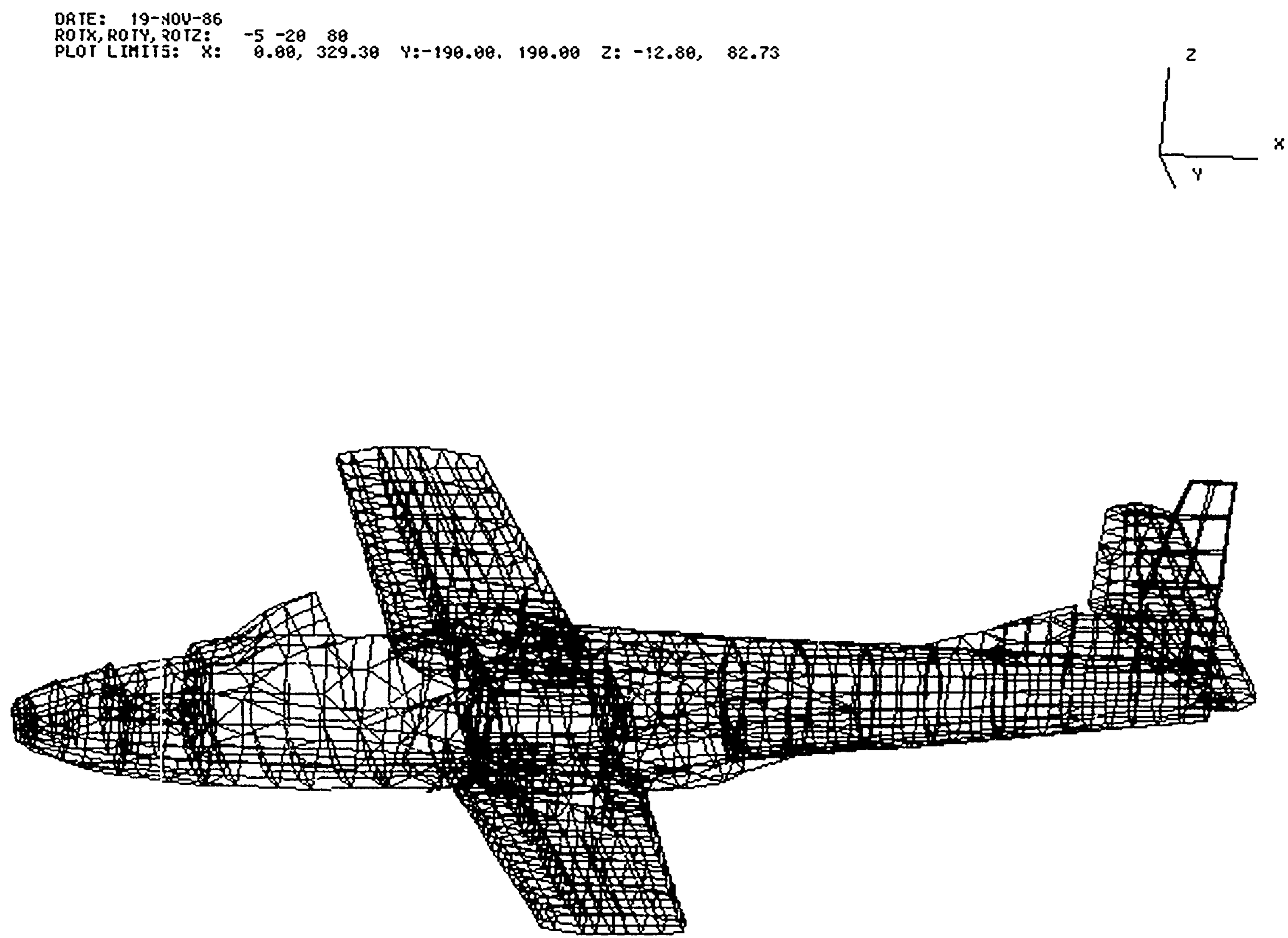
ASD/ENFS

T-37 Structural Life Extension Program

By

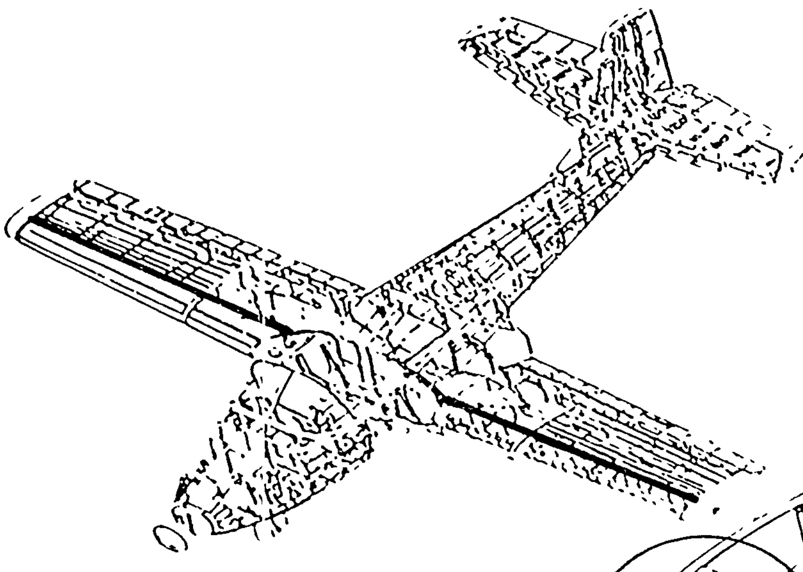
1 Lt Brian J. Duddy,
Fighter/Tactical/Trainer Systems
Program Management Division
Directorate of Materiel Management
San Antonio Air Logistics Center

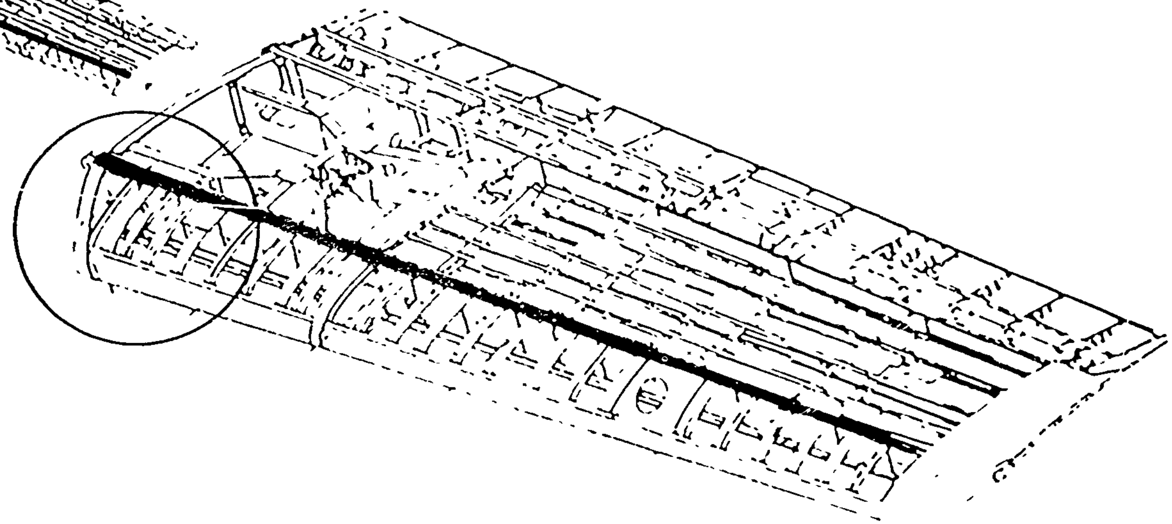
T—37 COARSE GRID
FINITE ELEMENT MODEL


DATE: 19-NOV-86
ROTX,ROTY,ROTZ: -5 -20 80
PLOT LIMITS: X: 0.00, 329.30 Y:-190.00. 190.00 Z: -12.80, 82.73
Z
X
Y

FRONT SPARS

- **PAST ACTION**
  - REPLACE FRONT SPARS
  - COLD-WORK ATTACH LUG
- **PROPOSED ACTION**
  - REDESIGN SPARS
  - REPLACE SPARS

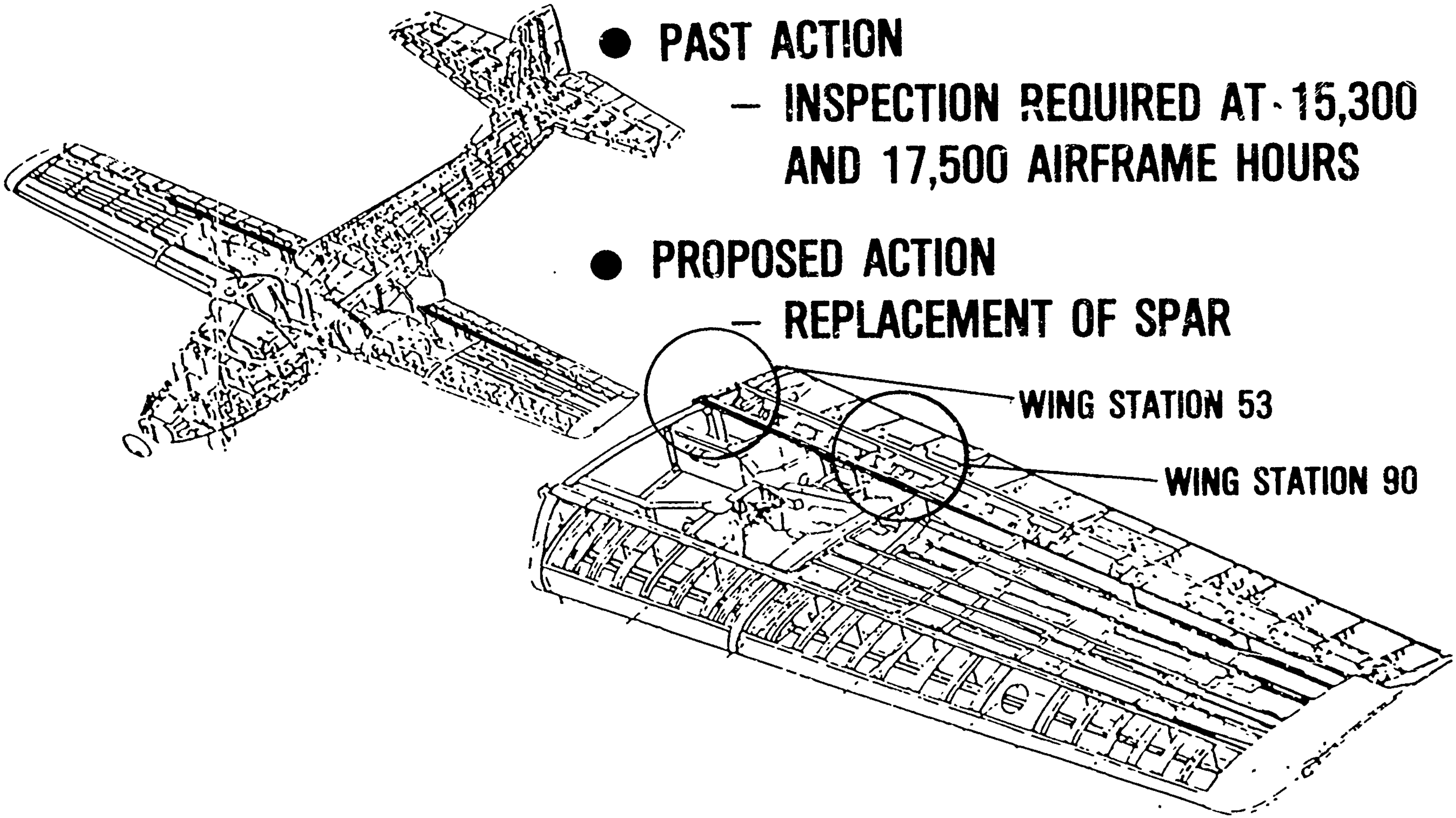




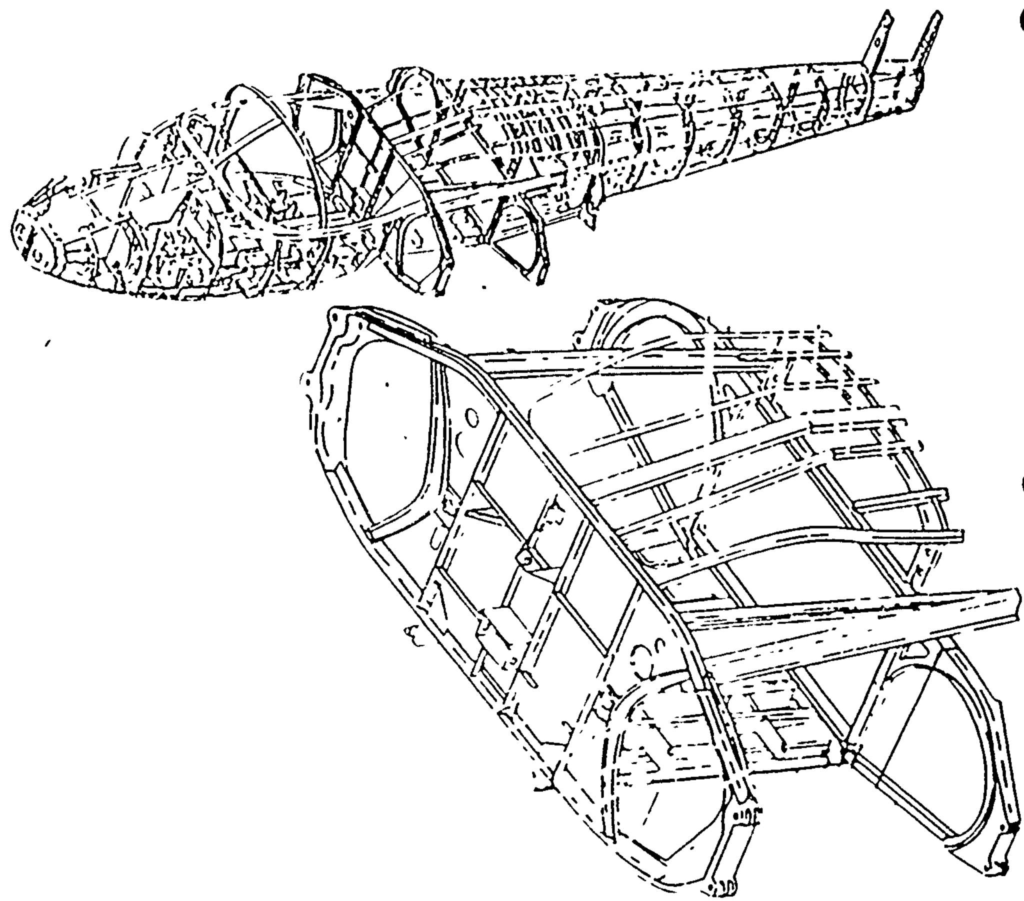
FATIGUE CRITICAL

REAR SPAR

- PAST ACTION
  - INSPECTION REQUIRED AT 15,300 AND 17,500 AIRFRAME HOURS
- PROPOSED ACTION
  - REPLACEMENT OF SPAR



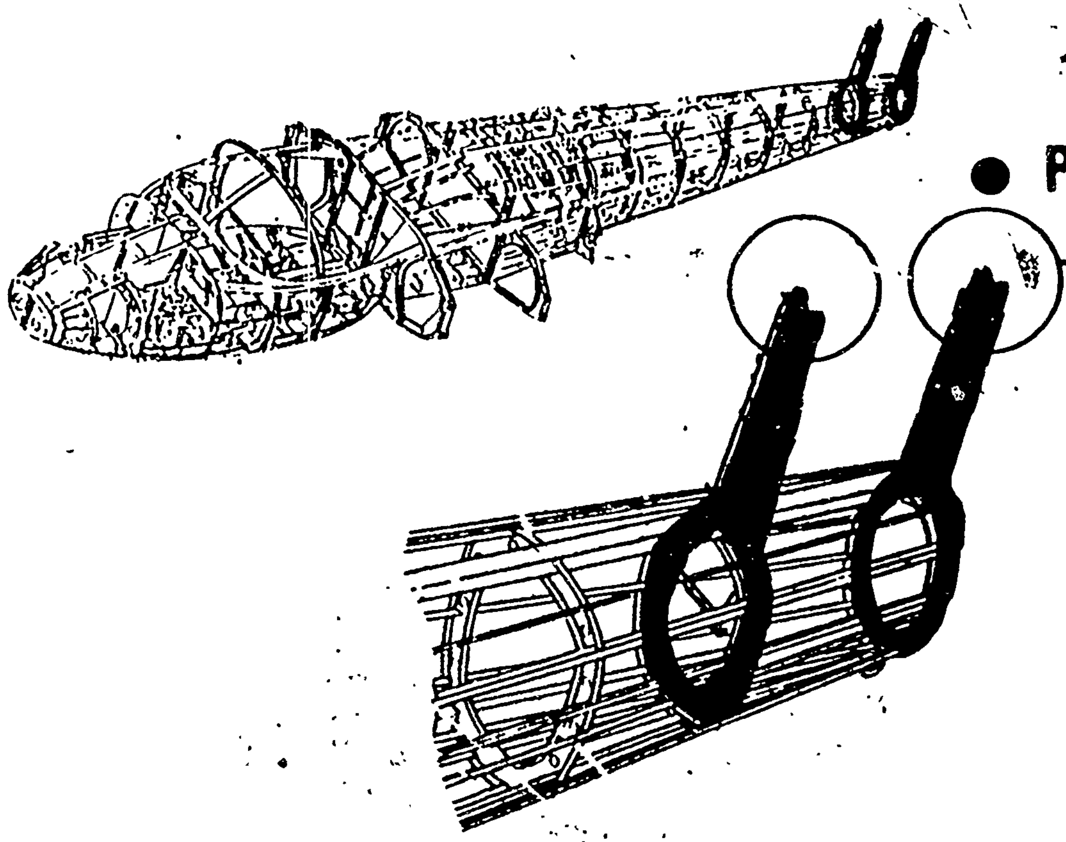
CARRY THROUGH



- PAST ACTION
  - ADDED STRAPS
  - SHOT PEENED OUTBOARD FITTING
  - REPLACE OUTBOARD FITTING
- PROPOSED ACTION
  - REPLACE CARRY THROUGH

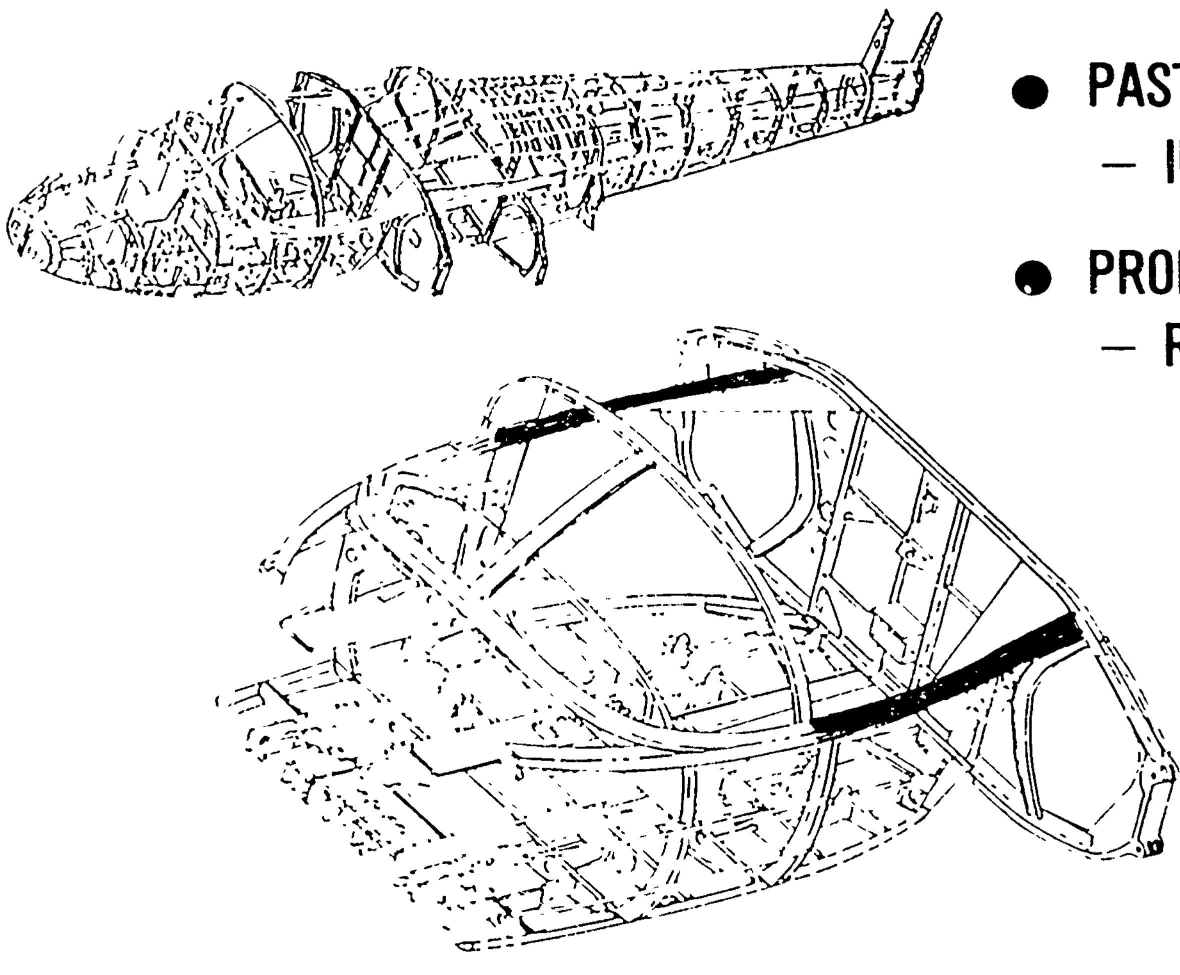
BANJO FITTINGS

- PAST ACTION
  - 60 FITTINGS REPLACED
- PROPOSED ACTION
  - REPLACE WITH FITTING WITH IMPROVED MATERIAL



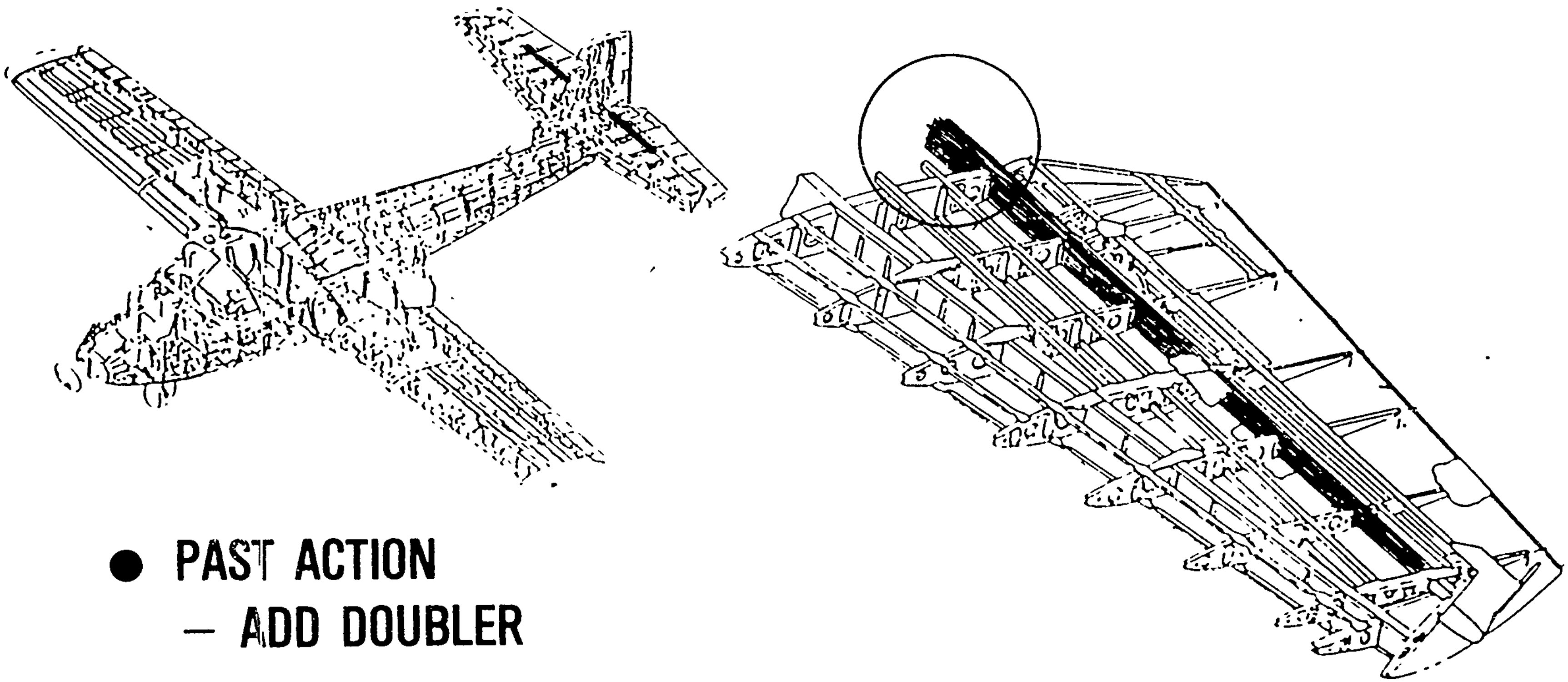
6

CANOPY RAIL



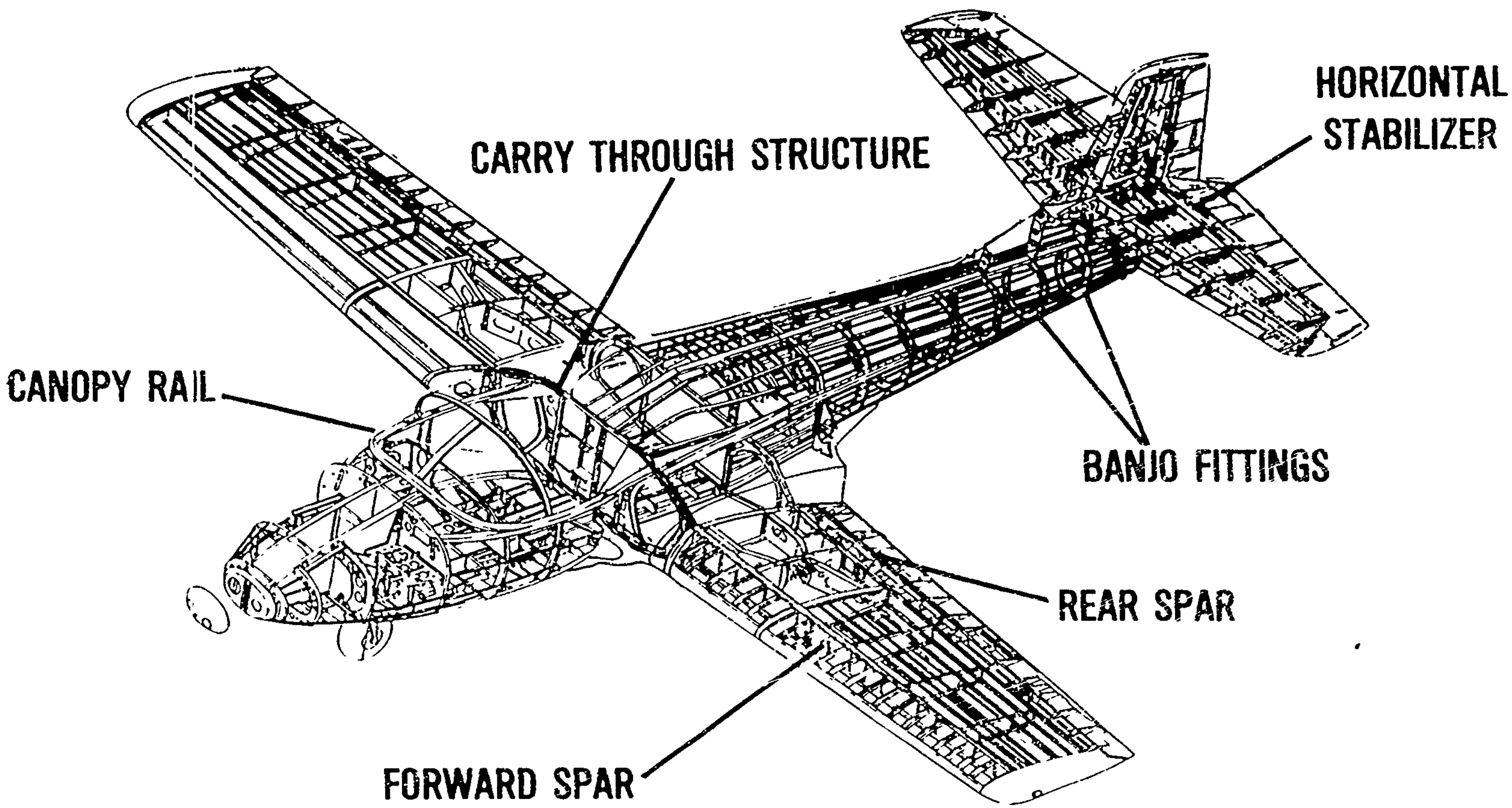
- PAST ACTION
  - INSTALL DOUBLERS
- PROPOSED ACTION
  - REPLACE CANOPY RAIL

HORIZONTAL STABILIZER



- PAST ACTION
  - ADD DOUBLER
- PROPOSED ACTION
  - REPLACE HORIZONTAL STABILIZER

CRITICAL STRUCTURAL AREAS (SIX PACK)



T-37 STRUCTURAL LIFE EXTENSION PROGRAM

1Lt Brian J. Duddy

Aircraft Structural Integrity Branch
Fighter/Tactical/Trainer System Program Management Division
Directorate of Materiel Management
San Antonio Air Logistics Center

INTRODUCTION

As can be seen from recent news reports over the last several months, there has been much uncertainty in the future of the selection of a primary trainer aircraft for the U.S. Air Force. The Fairchild T-46A, once designated as the Next Generation Trainer (NGT), now is the center of much controversy. This uncertainty and delay has brought to the forefront the need to reexamine the life of the aging Cessna T-37. This paper will present the strategy being implemented by the T-37 System Manager at San Antonio Air Logistics Center to extend the life of the aircraft in the event that it remains in the active Air Force inventory.

Several different options have been proposed for the aircraft, ranging from purely structural modifications to operational improvements such as reengining. This paper will mainly cover the structural aspects and those other limited improvements that have been authorized by Air Staff. Included will be a brief background of the aircraft and its usage, structural history, the current Durability and Damage Tolerance Analysis program, and a summary of the planning for life extension.

AIRCRAFT HISTORY/BACKGROUND

The T-37 has been operational since 1956, and has earned the best safety record of any jet aircraft that the Air Force has had. The fleet has accumulated over nine million flying hours; averaging almost 12,000 hours per aircraft. The low time aircraft has exceeded 7300 hours while the high time aircraft is nearing 16,000 hours. This means that at the present utilization rate of 45 hours per month, the high time aircraft by the year 1990 will reach 18,000 hours.

The T-37 aircraft are primarily used by Air Training Command in their undergraduate pilot and navigator training programs. In addition, ATC has 57 aircraft authorized for SAC's Accelerated Copilot Enrichment Program. 51 T-37s are being used for the Euro/NATO Joint Jet Pilot Training Program at Sheppard AFB. ATC has recently transferred 29 T-37s to TAC as part of the O-2A replacement program, "Pacer Swap."

The structure is of the semi-monocoque type, with sheet metal skins, built up spars, ribs, and longerons. There are major forged fittings at at the primary connection points such as wing to fuselage, and vertical/horizontal stabilizer to fuselage.

The 18,000 hour figure mentioned earlier refers to the current "service life" of the T-37. As we know, the accepted engineering design criteria of the 1950s was the service life concept; which, through a program of fatigue tests resulted in an airframe that was considered safe to fly for a finite period of time. In the case of the T-37, the initial design goal for airframe life was 8000 flight hours, with the structure only tested to meet this target. Later on, the airframe was further tested for 60,000 hours, resulting in 15,000 hours of service life, then to 72,000 hours of testing resulting in an 18,000 hour service life using this fatigue testing concept. Some components of the fatigue test article, such as the canopy rails and wing spars, required repair prior to reaching 18,000 hours. This in turn led to modifications of the aircraft to ensure safe operation to reach the 18,000 hour target. Beyond this figure, however, nothing is known about the remaining life of the T-37 primary structure.

STRUCTURAL HISTORY

Durability testing and field problems have revealed six areas of concern on the aircraft structure. They are: forward and rear wing spars, carry through structure, canopy rails, horizontal stabilizer, and banjo fittings. Each of these critical areas will now be discussed separately.

The forward spar lower cap on the wing in the area of W.S. 46 has been identified as a critical area. In fact, ATC lost an aircraft in 1968 due to fatigue failure of this spar. The spar was redesinged, tested, and replaced fleetwide as a result of this accident. In addition, the lower forward wing attachment lugs have been coldworked to extend the fatigue life. Now the service life of the replacement spar is nearing its end; the efforts mentioned here only served to get the spar safely to the 18,000 hour safe life point. A new redesigned spar will be required for life extension, unless completion of the DTA shows inspections will suffice.

The second critical location is the wing rear spar lower cap in the area of W.S. 53 and W.S. 90. To certify these areas to 18,000 hours, an inspection of the spar at W.S. 53 has to be performed at 15,300 and 17,500 hours. This inspection has been performed on eight aircraft that have reached the first inspection interval, and 22 Analytical Condition Inspection (ACI) aircraft. So far no cracking has been found in the area. The second location at W.S. 90 will soon require a similar inspection and/or rework. Replacement of the lower spar cap is needed to extend the life and eliminate these costly inspections.

The third critical component is the wing-fuselage carry through structure. The forward carry through has shown to be fatigue critical and work has already been accomplished in the area of the outboard wing attach fitting, to include adding doublers, shot peening, and even replacement of some cracked fittings (4.5% of the fleet) during the coldwork program. The problem is that the center portion of the carry through is buried deep in the aircraft and would require extensive maintenance (approximately 800-1000 manhours/aircraft) in disassembly just to accomplish the required recurring inspection in order to safely fly the aircraft. For life extension it is more economical to replace the forward carry through rather than accomplish the required inspections, but, the replacement time will be based on the DTA rather than the tests conducted with a 20 year old usage.

The two empennage attachment or "banjo" fittings in the aft fuselage are experiencing stress corrosion cracking in the horizontal stabilizer attachment area. Over 60 of the fittings have already been replaced by the field. We have also experienced an in-flight failure of a cracked banjo fitting. Under most circumstances, this would have meant a loss of the aircraft and possibly the crew, however, a safe recovery was made. The only way to alleviate this problem is to replace all of these fittings with new components made of superior material which is less susceptible to stress corrosion. At present this area is being inspected every 500 hours.

The fifth area of major concern is the cockpit longeron/canopy rail. The current canopy rails are actually repair fittings that were installed on top of the original longerons because these longerons were experiencing fatigue cracks. We are now seeing some cases of fatigue cracking in these repair rails. The only repair possible merely slows crack growth. These repair rails are nearing the end of their design lifetime and installation of a newer, more durable, rail is recommended for life extension.

The last structural component of concern is the horizontal stabilizer. An earlier modification which added a doubler to the stabilizer extended its life to 18,000 hours. At this time it is cheaper to install a new stabilizer rather than overhaul the old one to extend its life.

These so called "six pack" areas have been identified by all the testing and analysis to date. While they do incorporate most of the airframe's highly loaded areas, we cannot assure that these are the only structural members that will require attention to extend airframe life. Because of this, SA-ALC is currently performing a comprehensive Durability and Damage Tolerance Analysis (DADTA) with contractual support from Cessna Aircraft Co. This may also show that the 18,000 hour point is not the time to begin all of the modifications.

DURABILITY AND DAMAGE TOLERANCE ANALYSIS (DADTA)

As mentioned before, the 18,000 hour limit is the only known life figure for the aircraft. To assure that the modifications are indeed necessary for the aircraft, it was imperative that a complete DADTA be accomplished. The current DADTA on the T-37 is the first such full-scale program ever performed on the aircraft. Limited analysis has been performed on the wing front spar and wing attach lugs, but the current program addresses the condition of the entire aircraft and is a major factor in the modification planning for the fleet. Completion of the entire program is scheduled for November 1987, at least two full years before the first T-37 reaches the 18,000 hour point.

The analysis will be based on MXU-553 Flight Recorder data gathered from T-37s at Columbus and Williams AFB in the Air Training Command usage and from Sheppard AFB in the Euro/NATO usage. Subsequent DADTA updates will be possible for the Shaw AFB usage by the use of innovative usage analysis software being developed for SA-ALC by Alamo Technology Inc. and Southwest Research Institute.

The Cessna program involves several conventional DADTA tasks such as spectrum development, finite element modeling, and crack growth analysis, as well as a comprehensive airloads/strain survey flight testing to gather data on the T-37 that was previously unavailable. All in all, it is a sizable project, but one that is critically important to the future of the aircraft as well as to the mission of the Air Training Command.

PLANNING FOR THE T-37 LIFE EXTENSION PROGRAM

The main thrust of the T-37 Life Extension Program is to enable us to determine inspection requirements based on the current ATC usage (and eventually the TAC usage as well) and determine what structural mods will be necessary beyond the arbitrary 18,000 hour point. Several other mods and operational improvements have been proposed to bring the aircraft up to the requirements desired by ATC. Among these were cockpit pressurization, improved avionics, air conditioning, improved ejection seat, liquid oxygen system, single point refueling, and most importantly, improved performance through an engine upgrading or replacement program. At present, the only tasks approved for the T-37 Life Extension are the structural six-pack mods, ejection seat upgrade, and a Depot Economy Repair Program or DERP. The DERP tasks, such as minor structural repair, corrosion control, and rewiring would not stand alone because of the expense of disassembling the aircraft exclusively for these efforts. Historically, however, we save about three unscheduled maintenance man hours for each hour spent on these tasks. In a DERP, SA-ALC will look at some areas of the aircraft that have not been touched for the thirty years that the T-37 has been in service.

The modification program for the T-37 is scheduled to commence in 1990. This timing is planned to avoid potential grounding of the aircraft as they reach the 18,000 hour point. Throughout this planning period, SA-ALC has been working closely with Air Training Command to insure that the needs of the user are met, and at the lowest cost to the Air Force. The options examined cover the spectrum of what can be done from simply extending the airframe life to full scale modification of the airframe with state-of-the-art engines, avionics, and equipment. The program we have presented will be providing a safer, more reliable aircraft to the user while awaiting the acquisition of a new primary trainer for the U.S. Air Force.

Aircraft Structural Integrity Program Force Management Using The F-16 Crash Survivable Flight Data Recorder System

By

Janet Weiss

ASD/YPEF

F-16 Program Manager/Project Engineer

CSFDR/Force Management

AIRCRAFT STRUCTURAL INTEGRITY PROGRAM

REQUIREMENTS

- MIL-STD-1530 DEFINES THE OVERALL REQUIREMENTS NECESSARY TO ACHIEVE AND MAINTAIN STRUCTURAL INTEGRITY
- AFR 80-13 REQUIRES A STRUCTURAL INTEGRITY PROGRAM AN USAF AIRCRAFT IAW MIL-STD-1530 AND DEFINES ASIP RESPONSIBILITIES
  - SYSTEMS COMMAND INITIATES
  - LOGISTICS COMMAND CONTINUES
  - USING COMMAND SUPPORTS
- TAC REGULATION 66-5 ASSIGNS RESPONSIBILITIES AND DEFINES PRODECURES FOR IMPLEMENTING AND MAINTAINING ASIP WITHIN TAC
- F-16 ASIP MASTER PLAN DOCUMENTS AND IMPLEMENTS ASIP REQUIREMENTS
  - 16PP029A: INITIAL F-16 ASIP REPORT
  - 16PR4452: UPDATED F-16A/B ASIP REPORT
  - 16PP388: INITIAL F-16C/D ASIP REPORT

F-16 AIRCRAFT STRUCTURAL INTEGRITY PROGRAM

FULL SCALE DEVELOPMENT EFFORT			FORCE MANAGEMENT PLAN	
DESIGN INFORMATION	DESIGN ANALYSES & DEVELOPMENT TESTS	FULL SCALE TESTING	FORCE MANAGEMENT DATA PACKAGE	FORCE MANAGEMENT
ASIP MASTER PLAN STRUCTURAL DESIGN CRITERIA DAMAGE TOLERANCE & DURABILITY CONTROL PLANS SELECTION OF MAT'LS. PROCESSES & JOINING METHODS DESIGN SERVICE LIFE AND DESIGN USAGE	MATERIALS AND JOINT ALLOWABLES LOAD ANALYSIS DESIGN SERVICE LOADS SPECTRA DESIGN CHEMICAL THERMAL ENVIRONMENT SPECTRA STRESS ANALYSIS DAMAGE TOLERANCE ANALYSIS DURABILITY ANALYSIS SONIC ANALYSIS VIBRATION ANALYSIS FLUTTER ANALYSIS DESIGN DEVELOPMENT TESTS	STATIC TESTS DURABILTY TESTS DAMAGE TOLERANCE TESTS FLIGHT & GROUND OPERATIONS TESTS SONIC TESTS FLIGHT VIBRATION TESTS FLUTTER TESTS INTERPRETATION & EVALUATION OF TEST RESULTS	FINAL ANALYSES STRENGTH SUMMARY FORCE STRUCTURAL MAINTENANCE PLAN LOADS/ENVIRONMENT SPECTRA SURVEY INDIVIDUAL AIRPLANE TRACKING PROGRAM	LOADS/ENVIRONMENT SPECTRA SURVEY INDIVIDUAL AIRPLANE TRACKING DATA INDIVIDUAL AIRPLANE MAINTENANCE TIMES STRUCTURAL MAINTENANCE RECORDS

F16AC
DISK #3

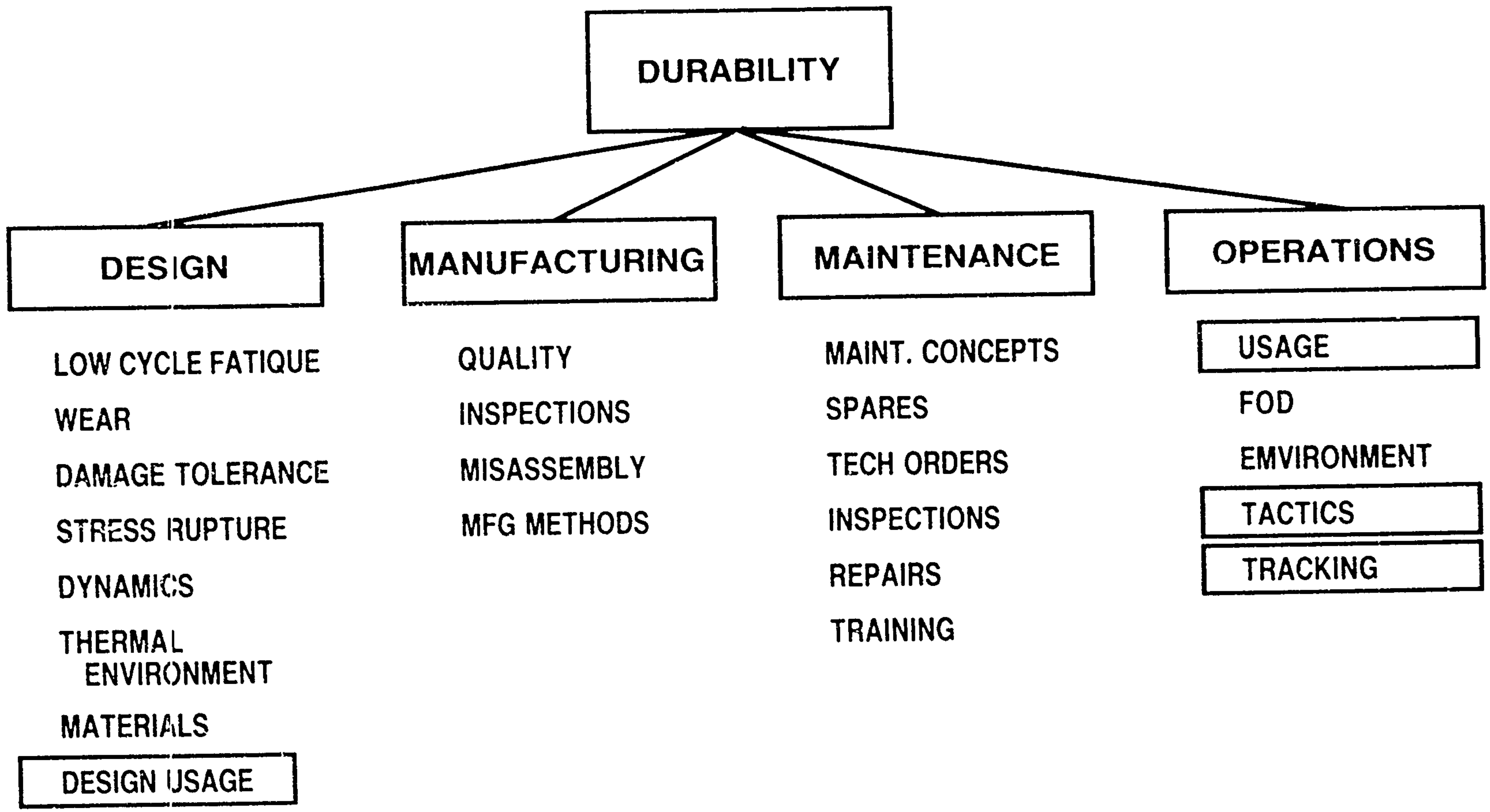
ENGINE STRUCTURAL INTEGRITY PROGRAM (ENSIP)

- DEVELOPMENT INITIATED 1970's
  - REVISION TO AF REG 80-13
- REQUIRED BY MIL-STD-1783
- ANALOGOUS TO ASIP
  - THE GOAL OF ENSIP IS TO ENSURE ENGINE STRUCTURAL SAFETY, DURABILITY, REDUCED LIFE CYCLE COSTS AND INCREASED SERVICE READINESS.
- ASIP AND NOW ENSIP RESPONSIBILITIES DEFINED FOR:
  - HQ USAF
  - AFSC
  - AFLC
  - ASIP AND ENSIP MANAGERS
  - OPERATING COMMANDS
  - AIRFRAME AND ENGINE MANUFACTURES
- ENGINE LIFE MANAGEMENT IS A DEFINED TASK
- F-16 ENSIP RECORDING SYSTEMS
  - F-16A/B FLIGHT LOADS RECORDER (MXU-553A)
  - F-16C/D TYPE 4 CSFDR SYSTEM

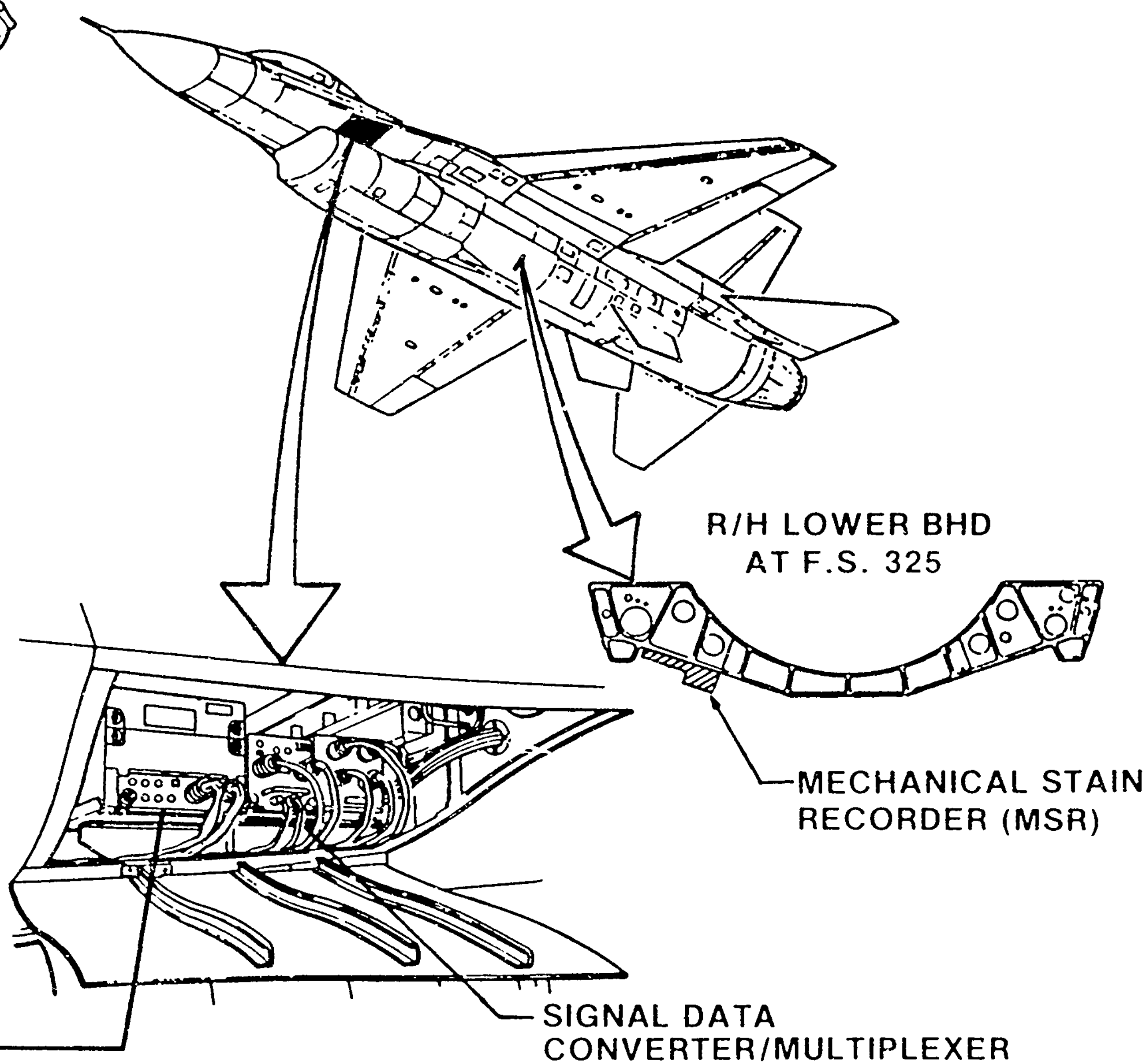
ENSIP TASKS

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANAL COMPNT & MAT CHARAC	COMPONENT & CORE ENG TESTING	GROUND & FLIGHT ENG TESTS	PROD QUAL CONTROL & ENG LIFE MGT
• ENSIP MASTER PLAN	• DESIGN DUTY CYCLE	• STRENGTH TESTING	• ENVIR VERIF TESTING	• PROD ENG ANALYSIS
• DESIGN SERV LIFE & USAGE REQUIREMENTS	• MAT'LS AND PROCESSES DESIGN DATA CHARACTERIZED	• DAMAGE TOLERANCE TESTS	• (AMT) TEST SPEC DERIV	• STRUC SAFETY & DURAB SUM
• DESIGN CRITERIA	• STRUCTURAL/ THERMAL ANALYSIS	• DURABILITY TESTS	• DURABILITY TESTS (AMT)	• ENG STRUC MAINT PLAN
	• MFG AND QUALITY CONTROL	• THERMAL SURVEY	• DAMAGE TOL TESTS	• INDIV ENG TRACKING
		• VIBRATORY STRAIN & FLUTTER BOUNDARY SURVEY	• FLIGHT TEST STRAIN SURVEY	• LEAD THE FORCE PROG (USAGE)
			• UPDATED DURA & DAM TOL CONTROL PLAN	• DURA & DAM TOL CONTROL PLAN IMPL
			• PERFRM DETERIOR STRUC IMPACT ASSESSMENT	• TECHNICAL ORDER UPDATE
			• CRITCL PART UPDATE	

FACTORS AFFECTING DURABILITY



FLR AND MSR LOCATIONS



FLR PARAMETERS

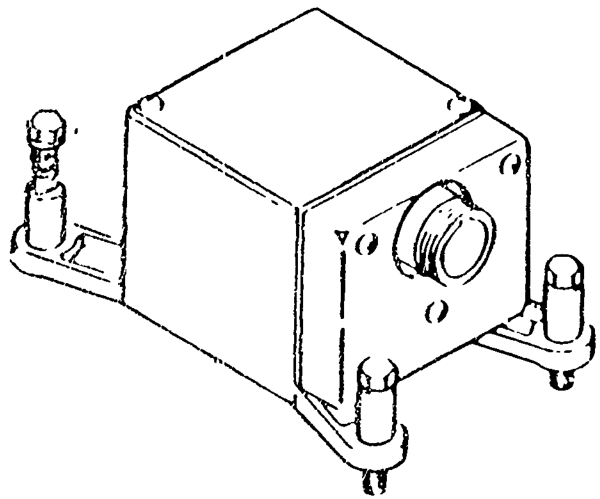
ITEM	SYMBOL	SAMPLE RATE
PRESSURE ALTITUDE	H_P	1
CALIBRATED AIRSPEED	V_C	1
PITCH RATE	Q	15
YAW RATE	R	15
ROLL RATE	P	30
ROLL ACCELERATION	P	30
NORMAL ACCELERATION	N_Z	15
LATERAL ACCELERATION	N_Y	15
LONGITUDINAL ACCEL (1)	N_X	5
FUEL QUANTITY	F_Q	1
ENGINE ROTOR SPEED	N_2	1
RUDDER POSITION (1)	Δ_R	15
L/H H.T. POSITION (1)	Δ_{HL}	15
R/H H.T. POSITION (1)	Δ_{HR}	15
LEFT FLAPERON POSITION (1)	Δ_{FL}	15
RIGHT FLAPERON POSITION (1)	Δ_{FR}	15
ENG POWER LEVER ANGLE (2)	PLA	5
STRUCTURAL STRAIN (1)	σ	15
WEIGHT ON WHEELS EVENT	E_1	1
LDG GEAR POSITION EVENT	E_2	1
WEAPONS RELEASE EVENT	E_3	1
DOCUMENTARY DATA	DD	1
TIMING WORD/GAP	T	1

(1) FLR PECULIAR GROUP B SENSORS (2) BLOCK 11 & ON AIRPLANES

CRASH SURVIVABLE FLIGHT DATA RECORDER (CSFDR)

- DEVELOPED BY THE F-16 SPO ACCORDING TO TRI-SERVICE EXHIBIT
- MULTIPURPOSE, SOLID STATE RECORDER DESIGNED TO MEET
  - MIL-STD 1530A ASIP FORCE MANAGEMENT REQUIREMENTS
  - F-16 PROGRAM MANAGEMENT DIRECTIVE (PMD) FOR CRASH INVESTIGATIVE DATA
- CSFDR IS A TWO BOX CONCEPT
  - SIGNAL ACQUISITION UNIT (SAU) - TO INTERFACE AIRCRAFT SIGNALS, RECEIVE AND PROCESS DATA AND DISTRIBUTE DATA TO THE APPROPRIATE MEMORY
    - AUXILIARY MEMORY UNIT (AMU) - AN OPTIONAL MEMORY DEVICE TO BE PART OF THE SAU FOR RETENTION OF STRUCTURAL LOADS (L/ESS) AND ENGINE USAGE DATA
  - CRASH SURVIVABLE MEMORY UNIT (CSMU) - AN ARMORED HOUSING CONTAINING SOLID STATE MEMORY DEVICES FOR STORING FLIGHT DATA. THIS UNIT IS TO BE CRASH SURVIVABLE.

FLIGHT DATA RECORDER COMPONENTS

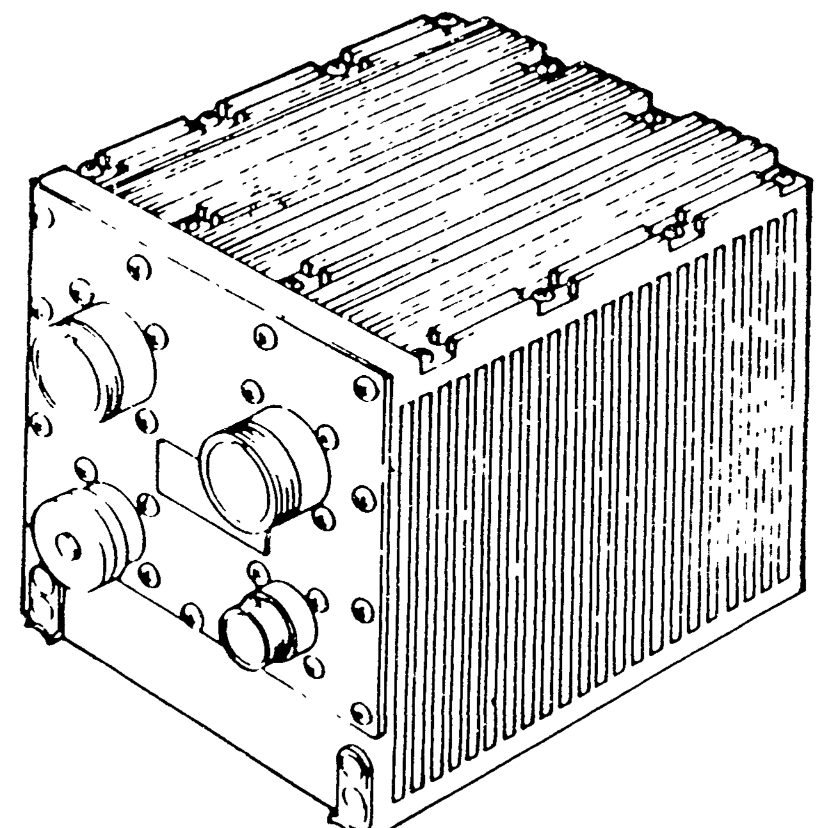


SAU/AMU

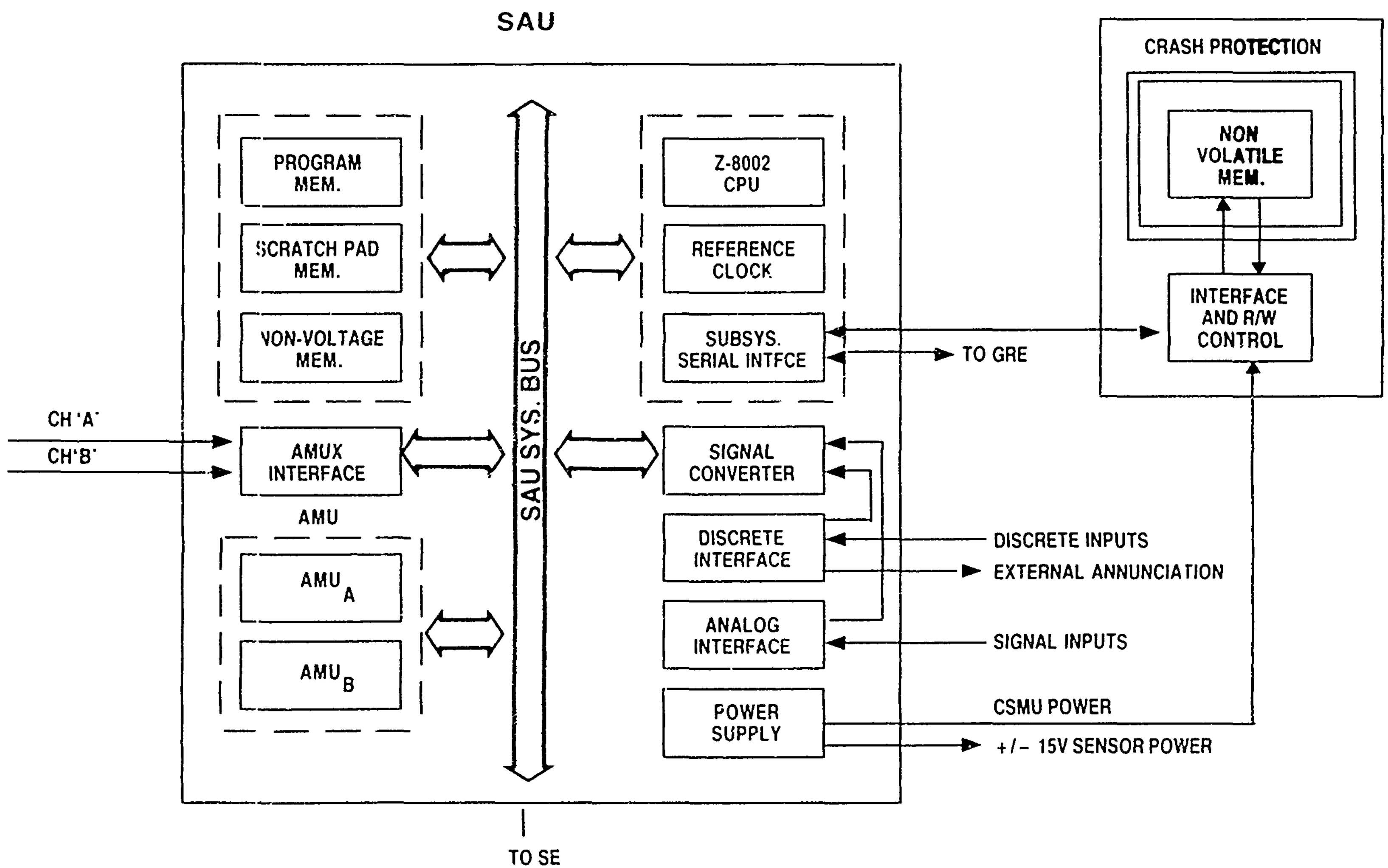
- Z8000 MICROPROCESSOR
- PROGRAM MEMORY - 16K, GROWTH TO 24K
- SCRATCHPAD MEMORY - 2K, GROWTH TO 8K
- NONVOLATILE MEMORY - 2K, GROWTH TO 4K
- AUXILIARY MEMORY UNIT - 256K
- INTERFACE - ANALOG
  - DISCRETE
  - 1553
- WEIGHT - 14.35 LBS
- SIZE - 6.2'' x 7.0'' x 7.25''

CSMU

- CRASH SURVIVABILITY FOR MILITARY AIR
- WEIGHT - 3.48 LBS
- SIZE - 2.9'' x 3.0'' x 4.5''
- CRASH SURVIVABLE MEMORY - 28K, GROWTH TO 64K



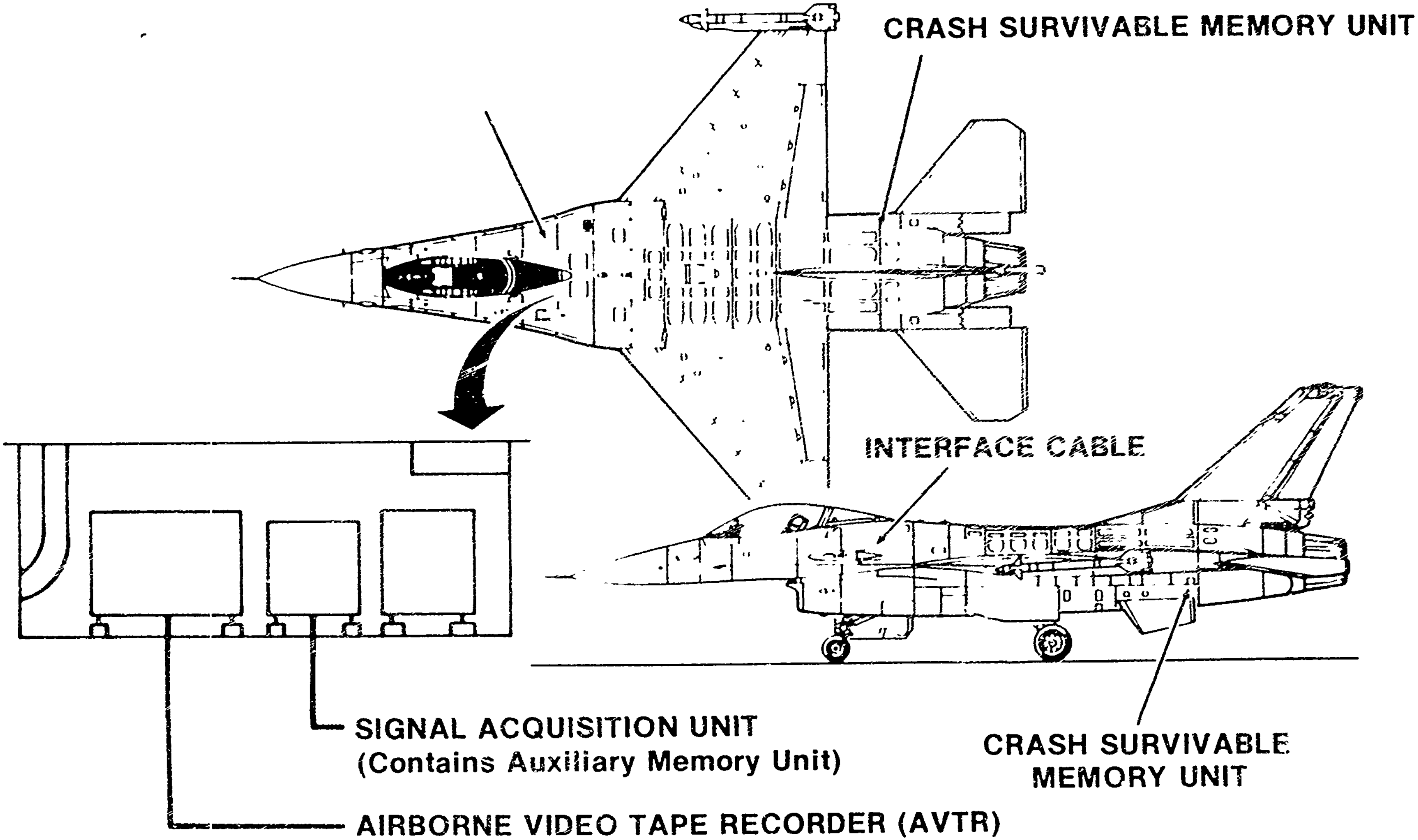
CONCEPTUAL CSFDR DESIGN



CSMU SURVIVABILITY

- **IMPACT SHOCK** - 200G PEAK, 115 MSEC DURATION LOW LEVEL SHOCK
  - 1700G PEAK, 5-8 MSEC DURATION HIGH LEVEL SHOCK
- **PENTRATION** - WITHSTAND IMPACT FORCE EQUAL TO A 500LB STEEL PIN DROPPED FROM 10 FEET ON TO EACH SIDE
- **STATIC CRUSH** - 5000 LBS APPLIED TO EACH AXIS FOR 5 MINUTES
- **FIRE** - 1100°C MIN AT A THERMAL FLUX OF 50,000 BTU/FT^2/HR FOR 30 MINUTES MIN
- **FLUID IMMERSION** - SEAWATER FOR 48 HRS
  - JP1 FOR 24 HRS
  - JP4 FOR 24 HRS
  - LUBRICATION OIL FOR 24 HRS
  - HYDRAULIC FLUID FOR 24 HRS
  - WATERGLYCOL FIRE EXTINGUISHING FLUID FOR 24 HRS
  - CO_2 FOAM FIRE FIGHTING MATERIAL FOR 24 HRS
  - FIRE EXTINGUISHING FOAM CONCENTRATE FOR 24 HRS

F-16 FLIGHT DATA RECORDER SYSTEM INSTALLATION
USAF F-16C/D AIRCRAFT



F-16 FORCE MANAGEMENT RECORDING SYSTEMS

F-16C/D PRODUCTION SYSTEM

- CRASH SURVIVABLE FLIGHT DATA RECORDER (CSFDR)
  - SIGNAL ACQUISITION UNIT (SAU) ON EVERY AIRPLANE
  - AUXILIARY MEMORY UNIT (AMU) WITHIN THE SAU ON EVERY AIRPLANE
  - CRASH SURVIVABLE MEMORY UNIT (CSMU) ON EVERY AIRPLANE
  - PERFORMS ONBOARD DATA COMPRESSION AND DATA STORAGE
    - TYPE 1 - DATA FOR MISHAP INVESTIGATION ANALYSIS
      - LAST 15 MINUTES OR MORE OF FLIGHT DATA
      - UP TO 5 SPECIAL EVENTS
      - BASELINE DATA AT LIFTOFF
      - DATA EXTRACTED AS REQUIRED
    - TYPE 2 - DATA FOR INDIVIDUAL AIRPLANE STRUCTURAL TRACKING (NVM IN SAU)
      - NzW EXCEEDANCES AND OTHER USAGE INFO
      - DATA EXTRACTED AT 150-HOUR PHASED INSPECTIONS
    - TYPE 3 - DATA FOR STRUCTURAL LOADS & SERVICE LIFE ANALYSIS (AMU)
      - RETAINS DATA FOR LAST 8 TO 10 FLIGHTS
      - DATA EXTRACTED AT 150-HOUR PHASED INSPECTION
    - TYPE 4 - DATA FOR ENGINE USAGE ANALYSIS (AMU)
      - DATA EXTRACTED AT 150-HOUR PHASED INSPECTIONS
- GD STUDY TO DEFINE CSFDR FOR F-16A/B RETROFIT DUE DECEMBER 1986

F-16C/D CSFDR SAU COMPUTED PARAMETERS

No.	Parameter Name	Label	Units	Samples/Sec	Data Types		
1.	Computed CG - Lateral	BLCG	Inches	1/Flt	2U	3U	4U
2.	Computed CG - Long.	FSCG	Inches	1/Flt	2U	3U	4U
3.	Computed CG - Vertical	WLCG	Inches	1/Flt	2U	3U	4U
4.	CG Offset For NY - Long.	YXCGO	Feet	1/Flt		3U	
5.	CG Offset For NY - Lateral	YYCGO	Feet	1/Flt		3U	
6.	CG Offset For NY - Vertical	YZCGO	Feet	1/Flt		3U	
7.	CG Offset For NZ - Long	ZXCGO	Feet	1/Flt	2U	3U	4U
8.	CG Offset For NZ - Lateral	ZYCGO	Feet	1/Flt	2U	3U	4U
9.	CG Offset For NZ - Vertical	ZZCGO	Feet	1/Flt	2U	3U	4U
10.	Differential Aileron (Flaperon)	DA	Degrees	16		3U	
11.	Differential Stabilizer (HT)	DHD	Degrees	16		3U	
12.	Symmetric Stabilizer (HT)	DE	Degrees	16		3U	
13.	Normal Force Coefficient	CN	-	16		3U	
14.	Pressure @ Altitude	PA	Lbs/Ft^2	1		3U	
15.	Roll Acceleration	PDOT	Rad/Sec^2	16	2U	3R	
16.	Pitch Acceleration	QDOT	Rad/Sec^2	16	2U	3R	
17.	Yaw Acceleration	RDOT	Rad/Sec^2	16	2U	3R	
18.	Lateral Load Factor @ CG	NYCG	G's	16		3R	
19.	Normal Load Factor @ CG	NZCG	G's	16	2RU	3R	4D
20.	Normal Load Factor @ CG *G W	NZCG *W	Lbs	16	2U		
21.	Dynamic Pressure	QBAR	Lbs/Ft^2	1		3R	
22.	CG - % MAC	CGX	Percent	1/Flt		3R	
23.	Horiz Tail Bending Moment - Left	HTBML	In-Lbs	16		3U	
24.	Horiz Tail Bending Moment - Right	HTBMR	In-Lbs	16		3U	
25.	Vertical Tail Bending Moment	VTBM	In-Lbs	16		3U	

Type 1: Mishap Investigation Data
Type 2: Individual Airplane Tracking Data
Type 3: Structural Loads/Enviroment Data
Type 4: Engine Usage Data

R: Recorded In Memory Device
U: Used By SAU Algorithm
D: Decompressed For Types 3 & 4 Files

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F-16C/D CSFDR SAU DIGITAL/ANALOG INPUT PARAMETERS

No. Discrete Name	0 =	1 =	Data Types	
40. LEF Asymmetry Lockup	Not Asymm	L&R Asymm	1R	
41. SEC Caution	Normal	Caution	1R	
42. JFS "Start 2"	Not St 2	Start 2	1R	
43. HYD A Low	Not Low	Low	1R	
44. HYD B Low	Not Low	Low	1R	
45. Main Generator Fail	Normal	Fault	1R	
46. Standby Generator Fail	Normal	Fault	1R	
47. Air Command (EPU)	Not Com	Command	1R	
48. FLCS Power Test Switch	Not Engd	Engaged	1R	
49. FLCS Branch A INV "GOOD"	Out	In	1R	
50. FLCS Branch B INV "GOOD"	Out	In	1R	
51. FLCS Branch C INV "GOOD"	Out	In	1R	
52. FLCS Branch D INV "GOOD"	Out	In	1R	
53. FLCS Disch LT Branch A	Not Disch	Disch	1R	
54. FLCS Disch LT Branch B	Not Disch	Disch	1R	
55. FLCS Disch LT Branch C	Not Disch	Disch	1R	
56. FLCS Disch LT Branch D	Not Disch	Disch	1R	
57. FLCS Caution Reset	Not Depd	Depressed	1R	
58. Dual FLCS Fail	Not Fail	Dual Fail	1R	
59. CAT III Selected	Cat I	CAT III	1R	3R
60. Control Stick Switch Pos'n	Aft Pos	Fwd Pos	1R	
61. Fwd Paddle Switch Depress	Not Depd	Depressed	1R	
62. AFT Paddle Switch Depress	Not Depd	Depressed	1R	
63. Elec Engine Control	Normal	Caution	1R	
64. Fwd Fuel Low	Not Low	Low	1R	
65. Aft Fuel Low	Not Low	Low	1R	
66. Auto TF Select	Not Select	Selected	1R	3R
67. Man. TF Select	Not Select	Selected	1R	3R
68. TF Fail Light	Not Fail	Fail	1R	
69. OBS Warn Lt	No OBS	OBS	1R	

F-16C/D CSFDR SAU DIGITAL/ANALOG INPUT PARAMETERS (CONT)

No.	Parameter Name	Units	Samples/Sec	Data Types	Range
31.	Long. Stick Force	Lbs	4	1R	+ or - 32
32.	Lat. Stick Force	Lbs	4	1R	+ or - 20
33.	Power Lever Angle (PLA)	Deg	2	1R 4R	0 to 130
34.	Core RPM (N2)	Percent	2	1R 4R	0 to 100 (0 to 110)
35.	Fan RPM (N1)	Percent	2	1R 4R	0 to 120 (10 to 100)
36.	Fan Turbine Inlet Temp (FTIT)	Deg C	2	1R 4R	200 to 1200
37.	Nozzle Position	Percent	1	1R	-10 to 110
38.	GE F110 Pyrometer	Deg C	2	1R 4R	650 to 1040
39.	5KVA PMG Freq	Percent	2	1R 4R	0 to 160

NOTES: *1 - SAU Conversion From Deg/Sec To Rad/Sec
*2 - SAU Conversion From Semicircles To Degrees (+ or -180)
( ) - Denotes Values For GE F110 Engine In Range Column

F-16C/D CSFDR SAU DISCRETE INPUT PARAMETERS

No. Parameter Name	Units	Samples/Sec	Data Types				Range
1. Reference Clock Time	Sec	16	1R	2U	3R	4R	-
2. Altitude (HP)	Feet	1	1R		3D	4R	-1500 to 80K
3. Calibrated Airspeed (VCAL)	Knots	1	1R		3R		50 to 1000
4. True Freestream Air Temp	Deg K	1	1R		3D	4R	173 to 323
5. Mach Number (M)	-	1			3D	4R	0.1 to 3.0
6. Radar Altitude	Feet	1	1R				-50 to 50K
7. Radar Alt Low Set	Feet	1	1R				0 to 50K
8. External Stores Weight	Lbs	1			3R		0 to 32767
9. Total Fuel Quantity	Lbs	1	1R	2U	3R		0 to 20 K
10. Gross Weight (GW)	Lbs	1		2RU	3R	4R	16K to 37.5K
11. Fuel Flow Rate	Lbs/Hr	1	1R				0 to 80 K
12. Fwd Fuel Quantity	Lbs	1/5	1R				0 to 4200
13. Aft Fuel Quantity	Lbs	1/5	1R				0 to 4200
14. Roll Rate (P)	Rad/Sec *1	16	1U	2U	3R		+ or – 5.24
15. Pitch Rate (Q)	Rad/Sec *1	16	1U	2U	3R		+ or – 1.05
16. Yaw Rate (R)	Rad/Sec *1	16	1U	2U	3R		+ or – 1.05
17. Long. Load Factor (NXMEAS)	G's	4	1R		3R	4D	+ or – 3
18. Lateral Load Factor (NXMEAS)	G's	8	1R		3U		+ or – 1
19. Normal Load Factor (NXMEAS)	G's	16	1R	2U	3U		+ or – 12
20. Rudder Position (DR)	Deg	8	1R		3R		+ or – 30
21. Left HT Position (DHL)	Deg	16	1R		3R		+ or – 22.15
22. Right HT Position (DHR)	Deg	16	1R		3R		+ or – 22.15
23. Left Flaperon Position (DFL)	Deg	16	1R		3R		-20 to +23
24. Right Flaperon Position (DFR)	Deg	16	1R		3R		-20 to +23
25. LEF Position (DLEF)	Deg	4	1R				-2 to +25
26. Angle of Attack (ALPHA)	Deg	8	1R		3R		-5 to +40
27. Pitch Attitude	Deg *2	4	1R				+ or– 110
28. Roll Attitude	Deg *2	4	1R				+ or– 180
29. True Heading	Deg *2	1	1R				+ or– 180
30. Great Circle Steering Error	Deg *2	1	1R				+ or– 180

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F-16C/D CSFDR SAU DISCRETE INPUT PARAMETERS (CONT)

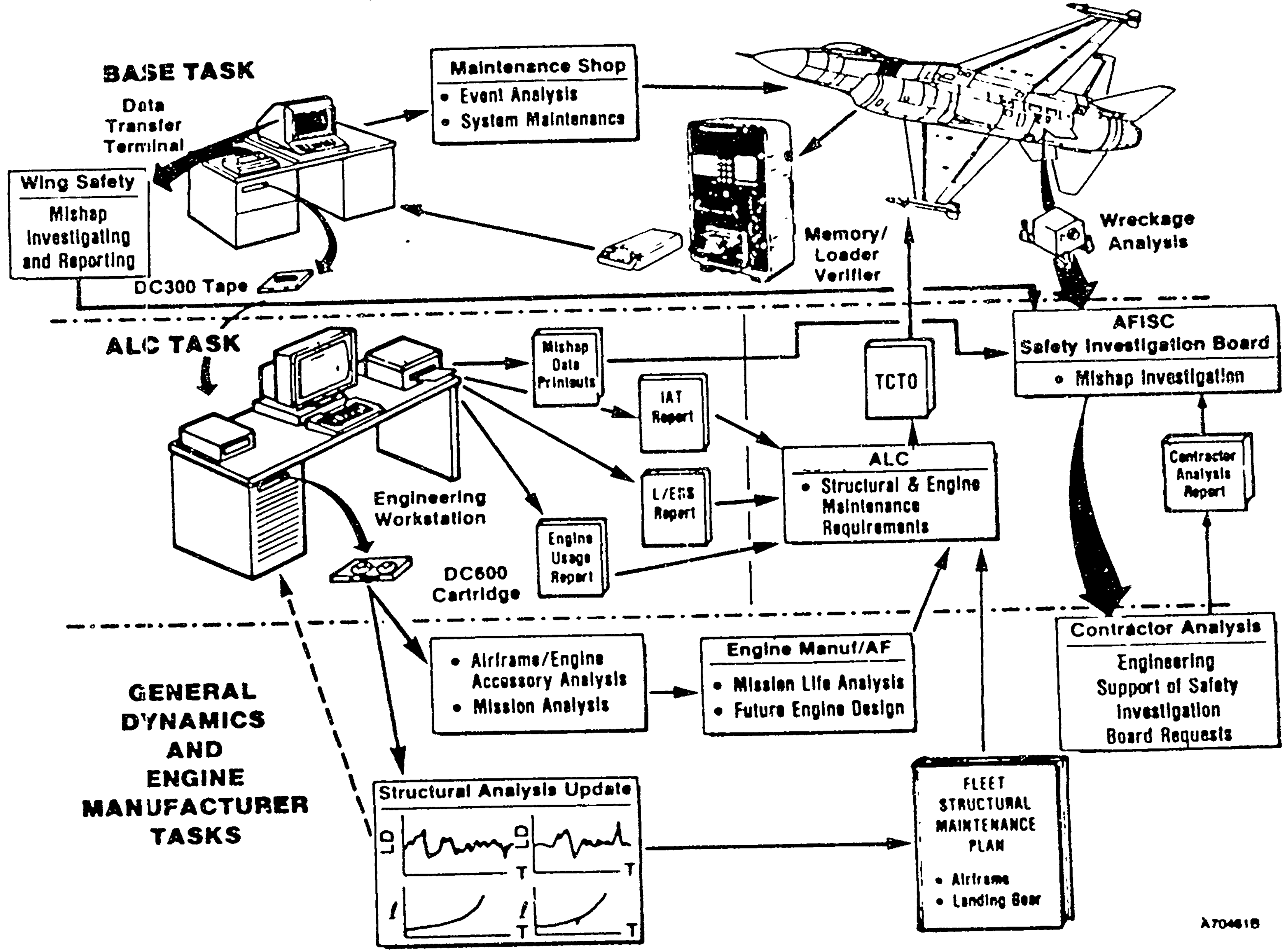
No.	Discrete Name	0 =	1 =	Data Types
70.	ATF Not Engage Light	Engaged	Not Engd	1R
71.	Valid Weapon Release	Not WR	WR	1R 3R
72.	SIM Weapon Release	Not WR	WR	1R
73.	MLG Weight-On-Wheels	Air	Gnd (WOW)	1R 2U 3R 4U
74.	Landing Gear Down Command	Not Down	Cmd Down	1R 2U 3R
75.	UHF/VHF Radio Transmit	Not Keyed	Keyed	1R
76.	Overheat Caution	Not Ovrht	Overheat	1R
77.	Roll Heading Select	Not Hold	Hold	1R
78.	Roll Attitude Hold	Not Hold	Hold	1R
79.	Pitch Altitude Hold	Not Hold	Hold	1R
80.	Pitch Attitude Hold	Not Hold	Hold	1R

SPECIAL EVENTS PRESERVED

MAIN GENERATOR FAIL	$NZ > +9.5G$
LEADING EDGE FLAP LOCKUP	$NZ < -3.0G$
DUAL FLT CONTROL FAIL	5KVA PMG FREQ $> 109\%$
N_2 RPM $> 97\%$	FTIT $> 1015°C$
N_1 RPM $> 115\%$	$N_2 < 55\%$, PLA $> 14°$
$N_1 > 110\% + D1/DT > 30\%/SEC$	PLA $< 14°$ W/WOW
AOA $> 29°$	FLCS CAUTION RESET
AOA $< -5°$	OVERHEAT DISCRETE

- THIRTY (30) SECONDS OF DATA PRESERVED PER EVENT
- EVENT OCCURRENCE REPORTED
- DATA PRESERVED UNTIL START OF NEXT FLIGHT

F-16C/D SAFETY/ASIP/ENSIP DATA FLOW

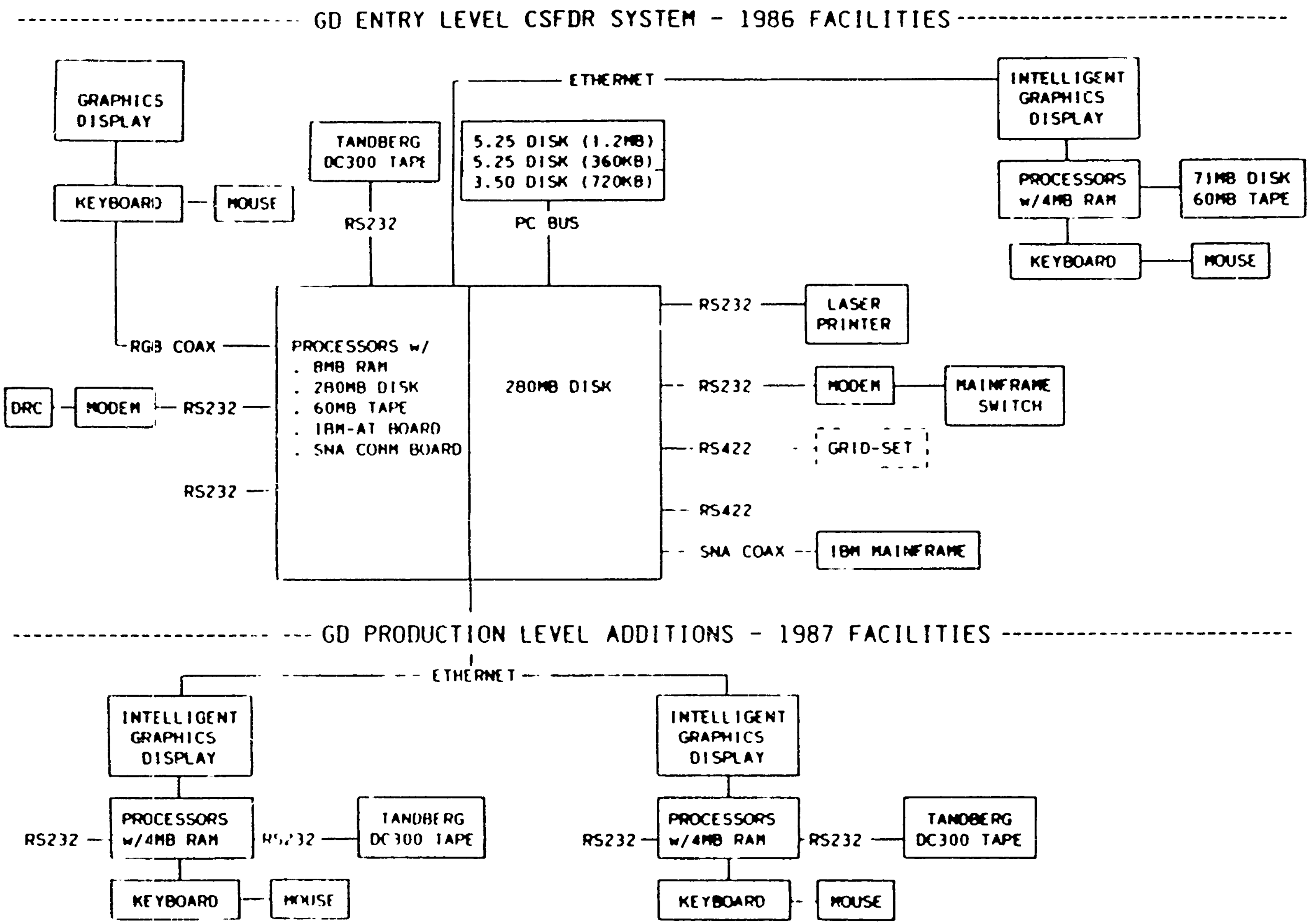


CSFDR ENGINEERING WORKSTATION HARDWARE AND SOFTWARE

FEATURES AND BACKGROUND

- CSFDR EWS SYSTEM TAILORED TO:
  - EFFICIENTLY PROCESS LARGE QUANTITIES OF COMPRESSED RAW CSRDR DATA
  - PROVIDE FIRST LINE DATA VALIDATION AND EDITING BY TECHNICIANS
  - CONDUCT MISHAP/INCIDENT ANALYSIS AND REPORTING
  - CONDUCT INDIVIDUAL AIRCRAFT TRACKING ANALYSIS AND REPORTING
  - CONDUCT LOADS/ENVIRONMENT SPECTRA SURVEY ANALYSIS AND REPORTING
  - CONDUCT ENGINE USAGE ANALYSIS AND REPORTING
  - PROVIDE GREATLY IMPROVED DOCUMENT PREPARATION CAPABILITY
- INITIAL SYSTEM HARDWARE AND SOFTWARE REQUIREMENTS DEFINED BY GD FROM:
  - DATA BASE MANAGEMENT AND REPORT PREPARATION EXPERIENCE GAINED DURING DEVELOPMENT AN REVISED INDIVIDUAL AIRCRAFT TRACKING PROCEDURE
  - ENGINEERING WORKSTATION STUDY CONDUCTED PRIOR TO FMS EWS PROPOSAL INCLUDING:
    - REVIEW OF DETAILED TECHNICAL PROPOSALS FROM TEN (10) VENDORS
    - LIMITED HANDS-ON EVALUATION OF THREE (3) SYSTEMS
    - DETAILED HANDS-ON EVALUATION OF THREE (3) SYSTEMS
- INITIAL CSFDR DATA PROCESSING SOFTWARE DEVELOPMENT BY GD UNDER 2ND MULTI-YEAR BUY

CSFDR ENGINEERING WORKSTATION HARDWARE AND SOFTWARE



CSFDR ENGINEERING WORKSTATION HARDWARE AND SOFTWARE

USAF SYSTEM HARDWARE REQUIREMENTS

- USAF SYSTEM(S) SHOULD BE COMPARABLE TO GD SYSTEM IN:
  - PROCESSING POWER (32BIT MICROPROCESSOR WITH FPA)
  - COMMUNICATION AND INTERFACE CAPABILITIES (MODEM AND ETHERNET NETWORK)
  - INPUT/OUTPUT DEVICES (DC300 TAPE DRIVE AND LASER PRINTER)
  - MASS STORAGE DEVICED (500+ MBYTE DISK(S) AND 60+ MBYTE BACKUP TAPE)
  - INTELLIGENT HIGH RESOLUTION GRAPHICS DISPLAYS (PREFERABLY CAPABLE OF DISK-LESS OPERATION IN THE NETWORK ENVIRONMENT)
- MAINFRAME COMMUNICATION AND PROCESSING IS NOT REQUIRED

USAF SYSTEM SOFTWARE REQUIREMENTS

- USAF SYSTEM(S) SHOULD HAVE THE SAME SOFTWARE REQUIREMENTS AS THE GD SYSTEM PARTICULARLY THE FOLLOWING:
  - UNIX OPERATING SYSTEM (EITHER UNIX 4.2 OR V)
  - C AND FORTRAN 77 COMPILERS, LINKER AND DEBUGGER
  - CORE AND/OR CGI GRAPHICS LIBRARIES
  - DOCUMENT PREPARATION ENVIRONMENT

CSFDR ENGINEERING
WORKSTATION CAPABILITY TRANSFER
WORKSTATION TERMINALS AND MANPOWER REQUIREMENTS

- USAF EWS SYSTEM MASS STORAGE REQUIREMENTS CONTROLLED AND LIMITED BY CONFIGURATION SELECTED
- THE GD, USAF, AND FMS EWS SYSTEMS SHOULD BE COMPATIBLE TO:
  - MAINTAIN COMMON SOFTWARE AND OPERATING SYSTEM
  - REDUCE CAPABILITY TRANSFER EFFORT AND COST
  - PERMIT DIRECT TRANSFER OF TYPE 3 L/ESS AND TYPE 4 ENGINE USAGE FLIGHT-BY-FLIGHT DATA BASES FOR FUTURE F-16 STRUCTURAL SERVICE LIFE ANALYSIS
- SELECTION OF EWS SYSTEM(S) SIMILAR TO GD'S (CENTRAL SERVER AND DISK-LESS NODES) WOULD BE THE MOST EFFICIENT AND WOULD REQUIRE THE FOLLOWING MASS STORAGE:
  - UNIX OPERATING SYSTEM WITH ALL UTILITIES - 30 MBYTES
  - PROGRAMMING ENVIRONMENT AND DOCUMENT PREPARATION SOFTWARE - 40 MBYTES
  - SYSTEM OVERHEAD (SWAP SPACE, USER SPACE, ETC) - 20 MBYTES PER USER
  - GD DEVELOPED DATA PROCESSING SOFTWARE INCLUDING SOURCE CODE AND EXECUTABLE FILES - 50 MBYTES
  - CSFDR DATA BASE FILES (BASED ON 1000 AIRCRAFT PER DATABASE):
    - TYPE 1 MISHAP/INCIDENT SUMMARIES PER AIRCRAFT - 20 MBYTES TOTAL
    - TYPES 2, 3 AND 4 USAGE PER AIRCRAFT - 100 MBYTES TOTAL
    - IAT, L/ESS AND ENGINE USAGE ANALYSIS PER AIRCRAFT - 80 MBYTES TOTAL
    - TYPE 3L/ESS FLIGHT-BY-FLIGHT BUILDUP - 50 MBYTES (400 FLTS)
    - TYPE 4 ENGINE USAGE FLIGHT-BY-FLIGHT BUILDUP - 50 MBYTES (400 FLTS)
  - TOTAL MASS STORAGE REQUIRED - 420 MBYTES + 20 MBYTES PER USER

CSFDR ENGINEERING WORKSTATION HARDWARE AND SOFTWARE

CSFDR DATA PROCESSING SOFTWARE REQUIREMENTS (GD DEVELOPED)

- MULTIPLE PROCEDURES TO CONDUCT THE FOLLOWING PROCESSING:
  - RAW DATA COPY FROM MULTIPLE DEVICES (DC300 AND FLOPPY DISK) AND BACKUP
  - RAW DATA DECOMPRESSION FOR ALL FOUR (4) DATA TYPES
  - TYPE 1 MISHAP INVESTIGATION/SYSTEM MAINTENANCE ANALYSIS AND REPORT PREPARATION
  - TYPE 2 INDIVIDUAL AIRCRAFT TRACKING ANALYSIS AND REPORT PREPARATION INCLUDING THE FOLLOWING INPUTS:
    - STRUCTURAL INSPECTION INFORMATION
    - OPERATIONAL AIRCRAFT USAGE INFORMATION
  - TYPE 3 LOADS/ENVIRONMENT SPECTRA SURVEY ANALYSIS AND REPORT PREPARATION AND TYPE 3 L/ESS DATABASE BUILDUP AND MANAGEMENT
  - TYPE 4 ENGINE USAGE ANALYSIS AND REPORT PREPARATION AND TYPE 4 ENGINE USAGE DATA BASE BUILDUP AND MANAGMENT
- USER INTERFACE PROCEDURE TO AUTOMATE DATA PROCESSING PROCEDURES
- MAJOR DATA PROCESSING PROCEDURES WILL BE WRITTEN IN FOURTRAN 77 AND WILL BE EASILY TRANSPORTED TO SIMILAR COMPUTER ENVIRONMENTS
- DEVICE DRIVERS AND USER INTERFACE PROCEDURE MAY BE UNIQUE TO EACH DIFFERENT SYSTEM (MOST WILL BE WRITTEN IN C)

CSFDR ENGINEERING

WORKSTATION CAPABILITY TRANSFER

WORKSTATION TERMINALS AND MANPOWER REQUIREMENTS

- REQUIREMENTS FOR GRAPHICS WORKSTATIONS (DISPLAYS AND TAPE DRIVES NETWORKED WITH SERVER) LIMITED BY SYSTEM PROCESSING CAPABILITIES AND MANPOWER LOADING
- PROCESSING CAPABILITY AND MANPOWER LOADING CALCULATIONS BASED ON THE FOLLOWING ASSUMPTIONS:
  - 25 HOURS PER MONTH PER AIRCRAFT
  - 150 HOURS (PHASED INSPECTION) PER DOWNLOADING PER AIRCRAFT
  - TWO (2) DOWNLOADINGS PER YEAR PER AIRCRAFT
  - 2000 DOWNLOADINGS PER YEAR PER 1000 AIRCRAFT
  - TWO (2) HOURS FOR DATA PROCESSING (EXCLUDING DOCUMENT PREPARATION) PER DOWNLOAD (ONE DC300TAPE)
  - ONE (1) HOUR PER REPORT FOR IAT, L/ESS SUMMARY AND ENGINE USAGE SUMMARY
  - EIGHT (8) HOURS PER REPORT FOR L/ESS ANALYSIS AND ENGINE USAGE ANALYSIS
  - 40 TO 80 HOURS PER REPORT FOR DOCUMENT ASSEMBLY AND MASTER COPY
  - 2000 MANHOURS PER YEAR PER COMPUTER OPERATOR
  - TOTAL MANPOWER LOADING - APPROX. 5000 MANHOURS PER YEAR PER 1000 AIRCRAFT PLUS PROGRAM MAINTENANCE AND ADMINISTRATIVE SUPPORT

WORKSTATION TERMINALS AND MANPOWER REQUIREMENTS (CONT'D)

- BASED ON THE PREVIOUS ASSUMPTIONS GD RECOMMENDS THE FOLLOWING EQUIPMENT AND MANPOWER AVAILABLE AND DEDICATED TO THE USAF CSFDR DATA PROCESSING TASK:
  - ONE (1) EWS SYSTEM SERVER WITH 500+ MBYTES MASS STORAGE PER 1000 AIRCRAFT
  - FOUR (4) GRAPHICS WORKSTATIONS PER SYSTEM SERVER (1000 AIRCRAFT)
  - ONE (1) COMPUTER OPERATOR PER GRAPHICS WORKSTATION (8 HOURS PER DAY)
  - ONE (1) COMPUTER PROGRAMMER PER SYSTEM (WORKSTATIONS AND SERVER)
  - ONE (1) ADMINISTRATIVE SUPPORT PERSON (LEADMAN)

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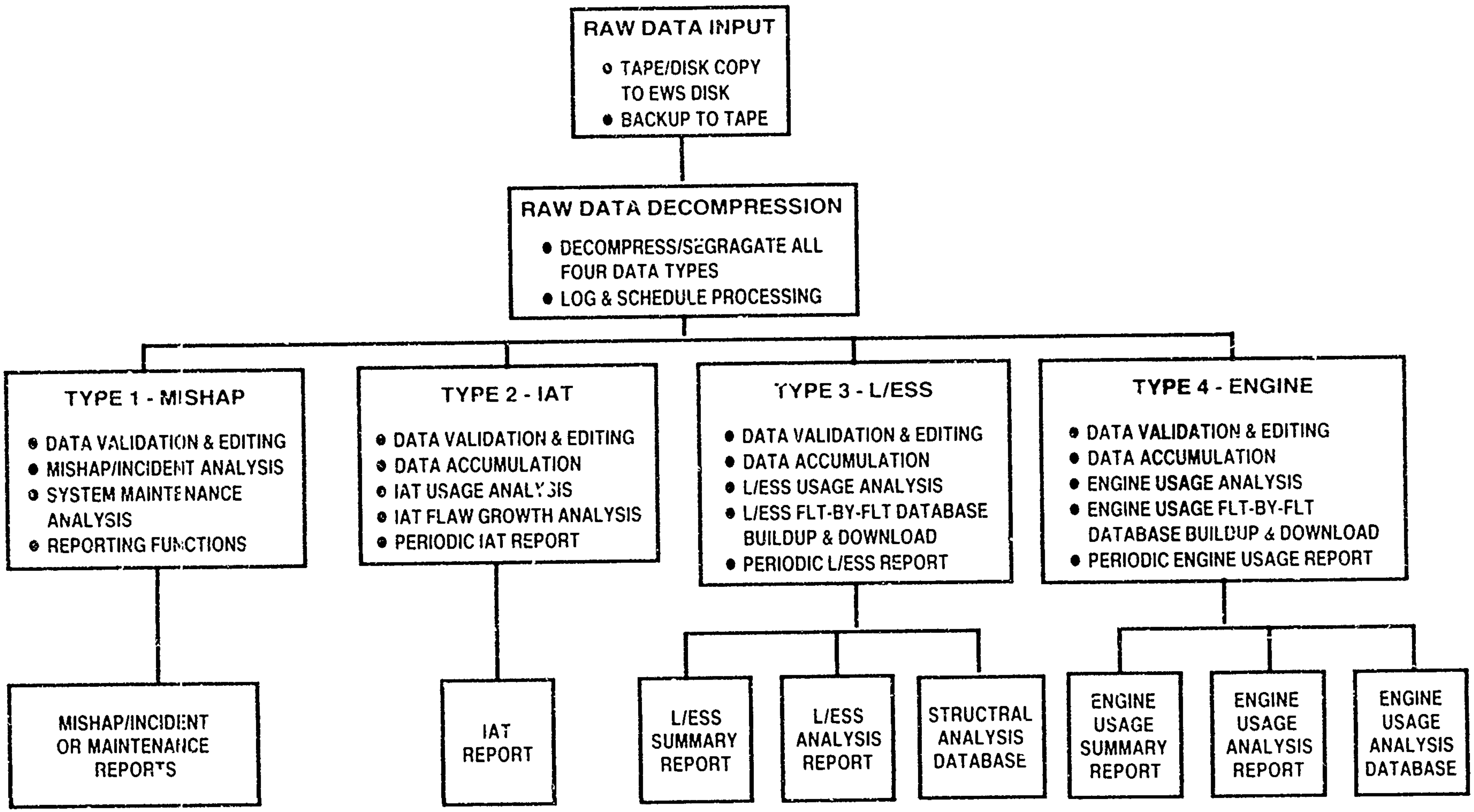
CSFDR ENGINEERING WORKSTATION HARDWARE AND SOFTWARE
CSFDR DATA PROCESSING SOFTWARE REQUIREMENTS (GD DEVELOPED)



An Innovative Approach To An OV-10 Usage Survey

By

Kurt H. Schrader
Southwest Research Institute
San Antonio, Texas

Today I will be discussing an innovative approach being used to define current OV-10 operational usage and mission profiles. This approach is believed to be the first of its kind and is a logical, timely improvement over the older oscillographic recording systems. What I want to describe is a small, simple, dedicated system being used for this usage survey. Micro-processors are being used during the airborne recording and data reduction phases on this program.

This OV-10 survey program for San Antonio Air Logistics Center at Kelly AFB, Texas, demonstrates this approach and highlights a new tool to meet ASIP needs and can be tailored for use in a wide variety of applications.

USAF ASIP REQUIREMENTS

OV-10 Description

USAF Role

Reasons for New Survey

A brief description of the OV-10 and its role in the USAF inventory is in order. This survey program satisfies some basic needs for the U.S. Air Force as they begin a full damage tolerance program for the OV-10.


USAF
13556

The OV-10 first entered the Air Force fleet in 1966. It is a two-place, twin engine, high wing aircraft in the 10,000 - 15,000 pound weight class. Its primary role is that of an forward air controller but it can also be used in a ground attack mission and can carry a variety of weapons on its external pylons. It can fly a range of missions and with external fuel tanks installed can remain in the air for as long as 5 hours.

This aircraft was built to U.S. Navy specifications and its service life was based on safe-life fatigue analyses. Generally this approach is more conservative than the damage tolerance analysis methods used today. In order to begin a total damage tolerance program, some knowledge of the use of the aircraft must be determined.

Typically, the Air Force performs a usage survey every 6-10 years in order to assess any changes in the aircraft's flying environment. The last survey for the OV-10 was completed during 1976 with data recorded from aircraft stationed at two bases in Florida - Eglin and Patrick.

In light of the 10 years since this survey and the need for current airload information on the OV-10 for upcoming programs, a new survey was required.

TECHNICAL TEAM

Southwest Research Institute (San Antonio,Tx)

Manage program
Final data reduction
Compilation of usage data

ESPRIT Technology, Inc. (Walnut Creek, Ca)

Design and install recorder systems
Provide long term system support

On-site Representative

Base level data collection
Preliminary data reduction

A technical team was put together to effectively and efficiently carry out this survey program. The primary organizations involved in this program are Southwest Research Institute (San Antonio, Tex.) who provide overall management of the program. They also perform the final data reduction, compilation, and presentation of the analyzed data to the Air Force. ESPRIT Technology, Inc. (Walnut Creek, Ca.) is responsible for design and installation of the recording systems. ESPRIT supplied all transducers and performed the complete installation in each aircraft including hookup to the pitot static system. They provide long term support during the recording period and at the conclusion, will remove the systems and return each aircraft to its original condition.

A final member of the team provides support services as on-base representative for SwRI and ESPRIT. This individual performs data collection and preliminary data reduction functions.

OBJECTIVE

Survey current OV–10 operational use

CONUS
PACAF

Compare results with previous surveys

Compare results between CONUS and PACAF

Collect 750 valid flight hours in 9 months per base

The main objective of the program is to perform a survey of current OV-10 operational use and mission profiles at both a CONUS and PACAF base.

The results of this survey will be compared with previous surveys to determine if the use of the OV-10 has changed significantly enough to warrant more detailed examinations of its predicted life. In addition, the results gathered at the two bases will be compared to each other to assess the flying conditions imposed by the different operating environments. The Air Force ultimately plans to use this usage data in on-going damage tolerance analysis programs conducted by San Antonio Air Logistics Center.

The requirement is to collect a minimum of 750 valid flight hours during a 9 month period from each base.

GOALS

Reduce survey costs

Improve retrieval rate

Reduce manpower requirements

Improve consistency in analyzing data

One of the main goals of this program is to reduce the cost of performing usage surveys. This can be accomplished in two ways:

1) improve the retrieval rate which means that the required data can be collected in a shorter time period or that more data can be collected in a required time period. The former means the results are available more quickly if time is critical while the latter provides a broader statistical base from which to prepare the final results.

2) reduce manpower requirements both at the base level and during the analysis stages.

A natural by-product of using the micro-processors will be improved consistency in analyzing the flight data because the human factor has been eliminated.

APPROACH

On–board micro–processor and transducers

- Peak data – V, G, H, and time
- Periodic data – V, G, H, and time

Supplementary data forms

- Mission
- Configuration
- Weight

Assemble/sort data using desk top computer

In order to gather the data, an on-board micro-processor records significant peak/valley data and also records periodic data on a time slice basis. For every data value recorded, the micro-processor retains the velocity (V), normal acceleration (G), altitude (H), and real time.

In addition to the automatically recorded data, supplementary data is provided on specially prepared forms. These forms are filled out by the pilot during debriefing immediately after each flight. The data include mission, configuration, and gross weight values that are used to correlate each recorded flight.

Desk top personal computers (PCs) are used to download the data from recorder memory, convert the data to engineering units and assemble and sort the data for final presentation.

RECORDING SYSTEM

Hardware

ELAPS IIB – airborne recorder
Interrogator/transcriber – ground based
Removable data cartridge

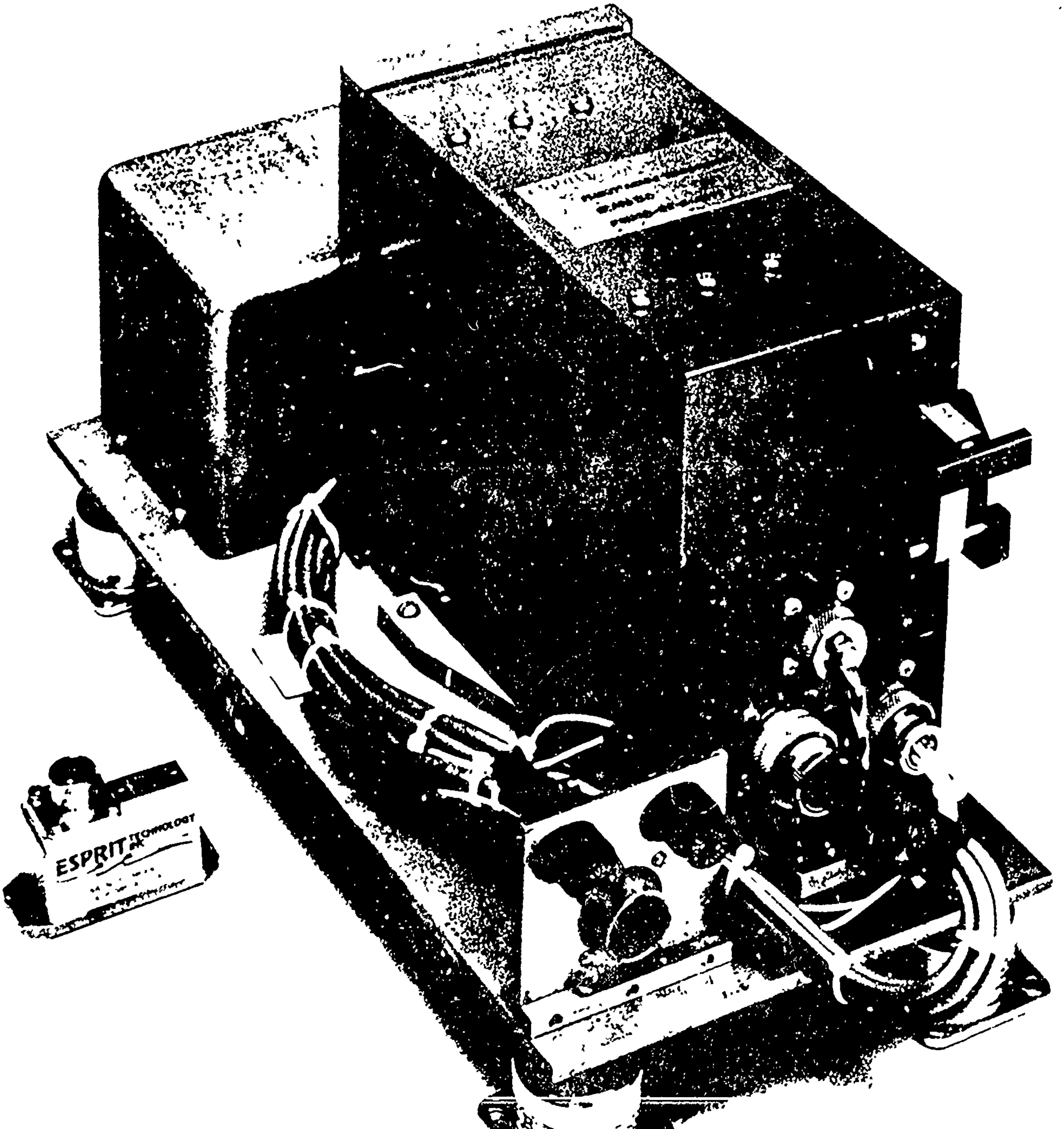
Software

Peak–valley algorithm stored in recorder memory
Multiple flight storage capability
Menu driven data reduction programs
Output in engineering units on floppy disk

The heart of the data recording package is the ESPRIT Load Assessment and Profile System (ELAPS) IIB which consists of a multichannel airborne recorder, ground-based interrogator, storage and transport medium, and a data transcriber. The recorder can be configured to interface with any of a number of transducers to measure, for instance, acceleration, shock, temperature, velocity, and altitude. The system ranges and algorithms are software controlled and can be changed in the field. In this application, the recorder serves as a VGH package capturing normal acceleration, velocity, and altitude on a time slice basis (every 1/100th of an hour) and also peak/valley accelerations (and associated velocity and altitude) under control of the algorithm stored in memory. The flight data is stored in a removable data cartridge and the whole system is compact and lightweight, weighing less than 15 pounds.

As mentioned, the peak-valley algorithm is stored on-board and thus only significant data will be saved. The system has multiple flight storage capability; depending on the amount of channels being recorded, up to 20 flights can be stored to accommodate cross-country or TDY flying. Menu-driven programs running on a PC lead the field-level operator through the preliminary data reduction process. No special operator skills are required to perform any of the tasks during the data collection phase.

The output consists of flight data in engineering units on a floppy disk which is easily accessible by a variety of software programs or by programs written to read the flight data by the analyst.



ELAPS IIIB L/ESS SYSTEM
AS INSTALLED ON USAF AIRCRAFT
FOR FLIGHT LOADS SURVEY
INSTALLATION INCLUDES Nz, VELOCITY,
AND ALTITUDE TRANSDUCERS
VIBRATION ISOLATORS ARE TO PROTECT TRANSDUCERS,

Here is a picture of the recorder as installed in the cargo area of the OV-10. The system consists of the recorder (shown with cartridge in place), base pallet, vibration isolators (to protect the transducers), and a typical vertical acceleration transducer. The module seen behind the recorder contains pressure transducers which are connected to the pitot static system to provide altitude and velocity pressure sources. If the OV-10 had an air-data computer, these sensors would not be required. All equipment is designed to meet military specifications and is electrically isolated to prevent damage to the aircraft electrical system. The removable cartridge is about the size of a video cassette.

ON–SITE DATA COLLECTION

Cartridge removed from recorder in cargo bay

Mission and configuration data collected

Run ELAPS software on PC

Remove diskette and prepare for mailing

It is the responsibility of the field-level operator to remove the cartridges from the recorder on a periodic basis and install fresh cartridges. In addition, he collects the necessary supplemental data forms after pilot debriefing. Menu driven programs that run on a PC dump the data from cartridge memory onto raw data files and clears the cartridge memory for later use. These programs lead the operator through the process of converting the raw data to engineering units files on floppy disks. The disks can be easily mailed to any location for additional analysis. The data can also be transmitted via the modem provided with the PC over conventional telephone lines.

FINAL DATA REDUCTION

Use speadsheet program (SYMPHONY) on PC

Analysis steps

- Plot time histories
- Flight phase identification
- Accumulate new occurrence data in data base

Prepare final report

The final reduction process begins with the use of a spreadsheet program (such as SYMPHONY) on a PC to assemble the flight data into a working file.

The processing steps are:

1) plot time histories on the same time scale one below the other. The plots are of peak normal acceleration (Nz), periodic altitude and velocity.

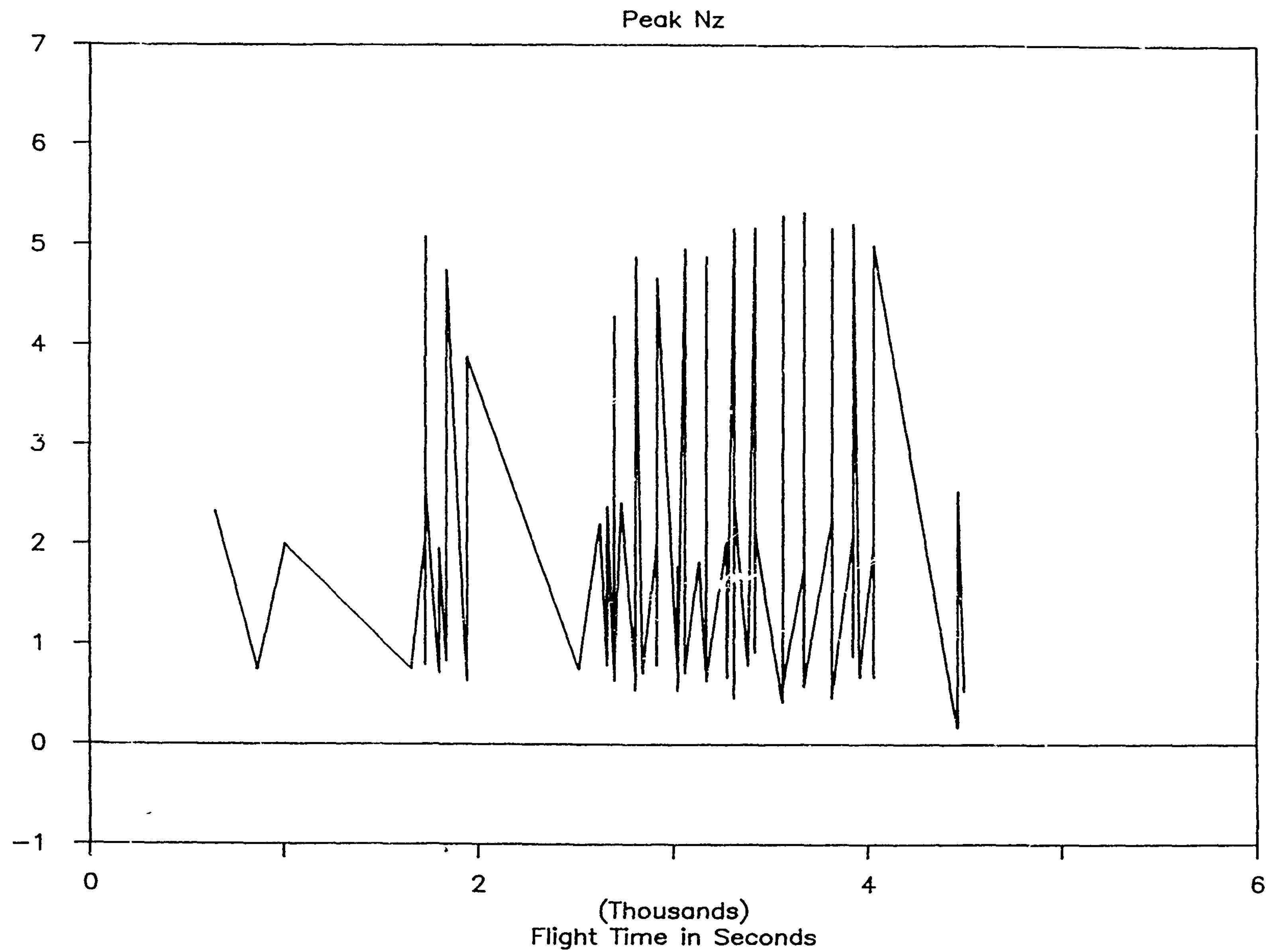
2) identify phases of flight (climb, cruise, etc.) on the plot. Basic guidelines are used to help define when the aircraft transition from one phase to another.

3) when the data has been properly marked, the flight data is added to the data base of occurrence data and associated parameters for each base.

Once the 9 month collection period has concluded, the data is further analyzed and compiled in a final report.

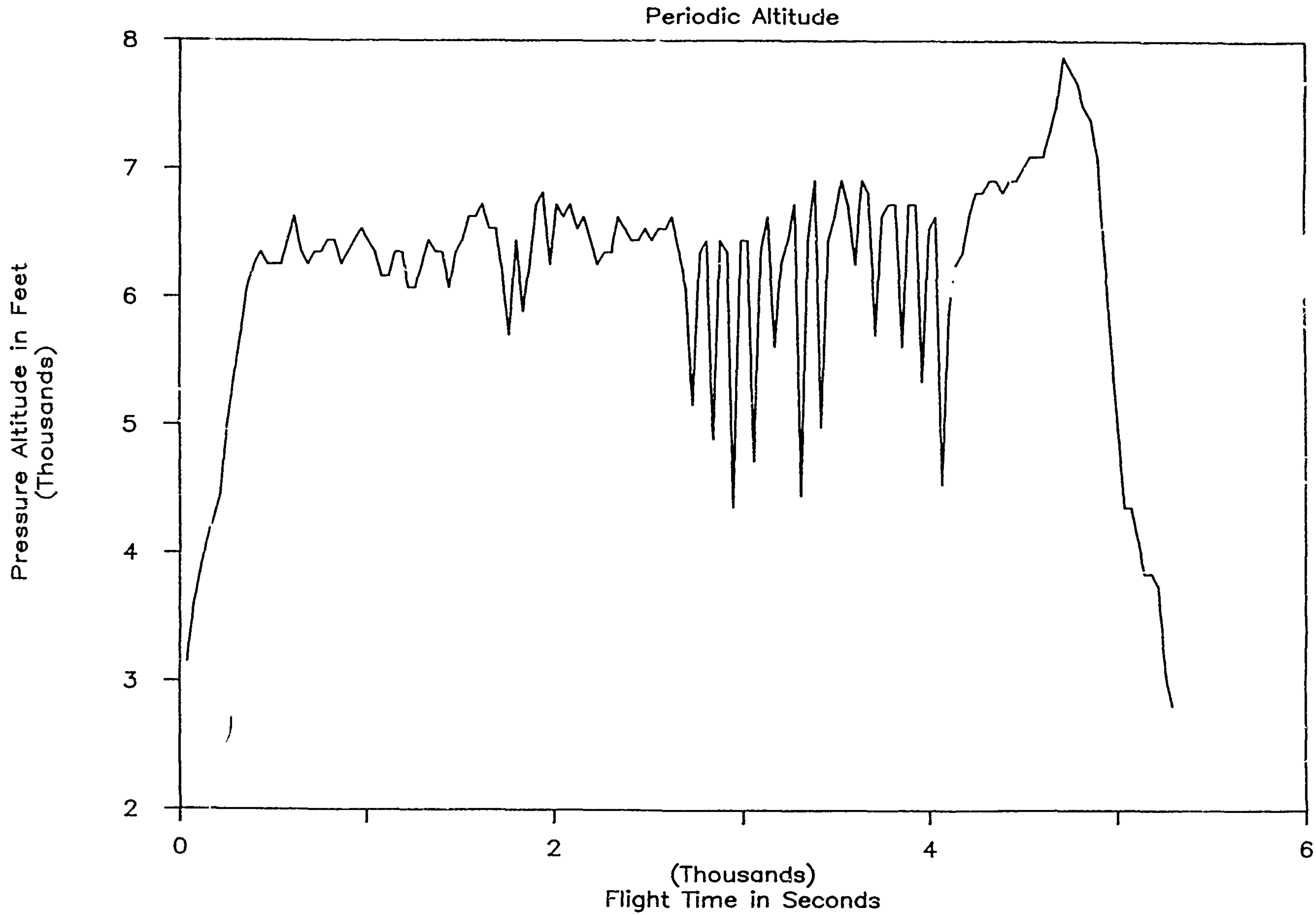
The following charts give an indication of the quality of the data being recorded and also illustrates the phase marking process of data reduction.

07850403.08

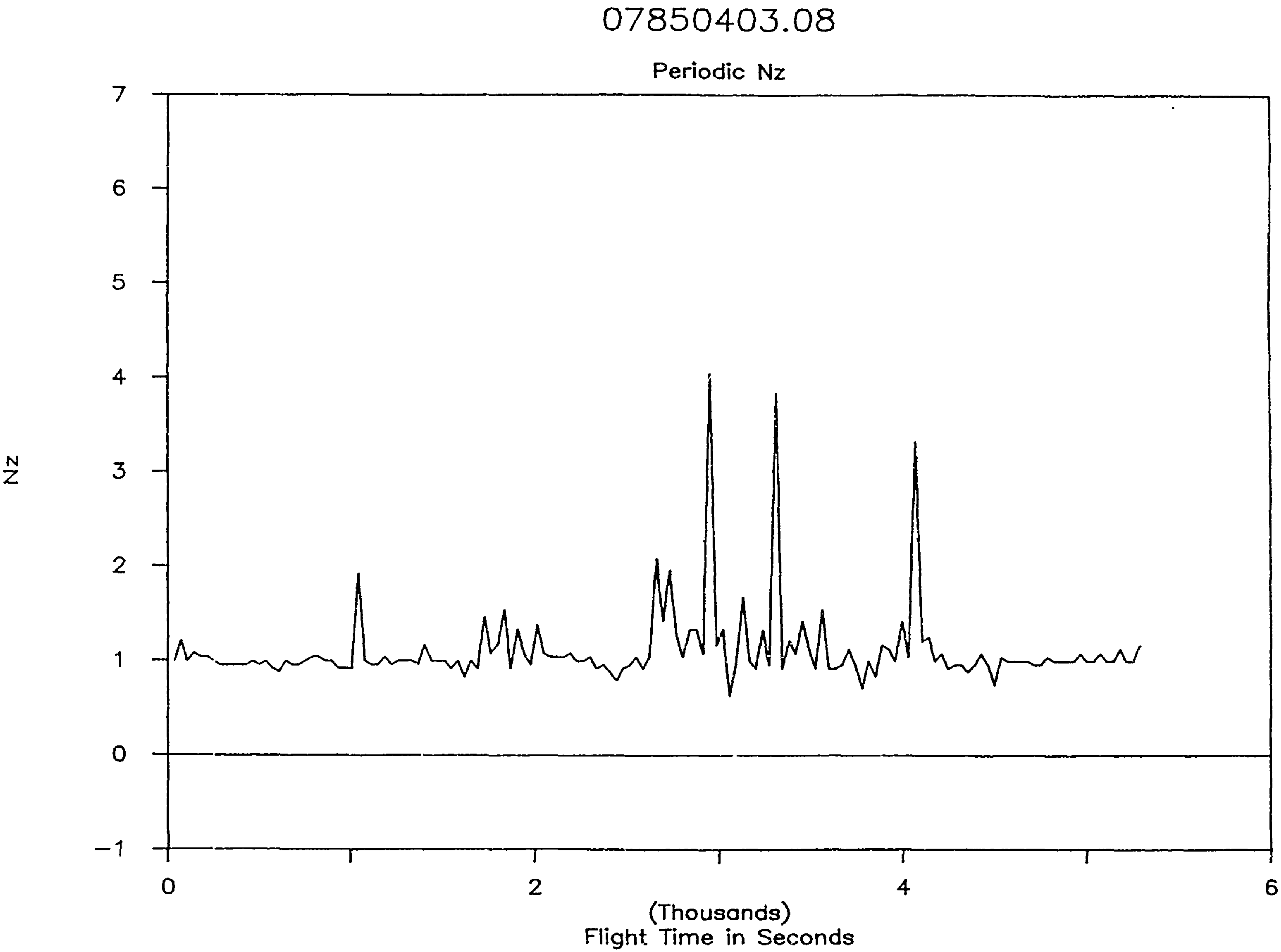


Shown is a typical plot of significant Nz occurrences during a given flight. Note the Nz activity at about 1800 seconds into the flight and again during the flight at 2600 seconds through 4000 seconds. The approximate constant level of peak Nz of 5 gs and valleys of 1 g is indicative of repetitive store-release passes over a weapons range.

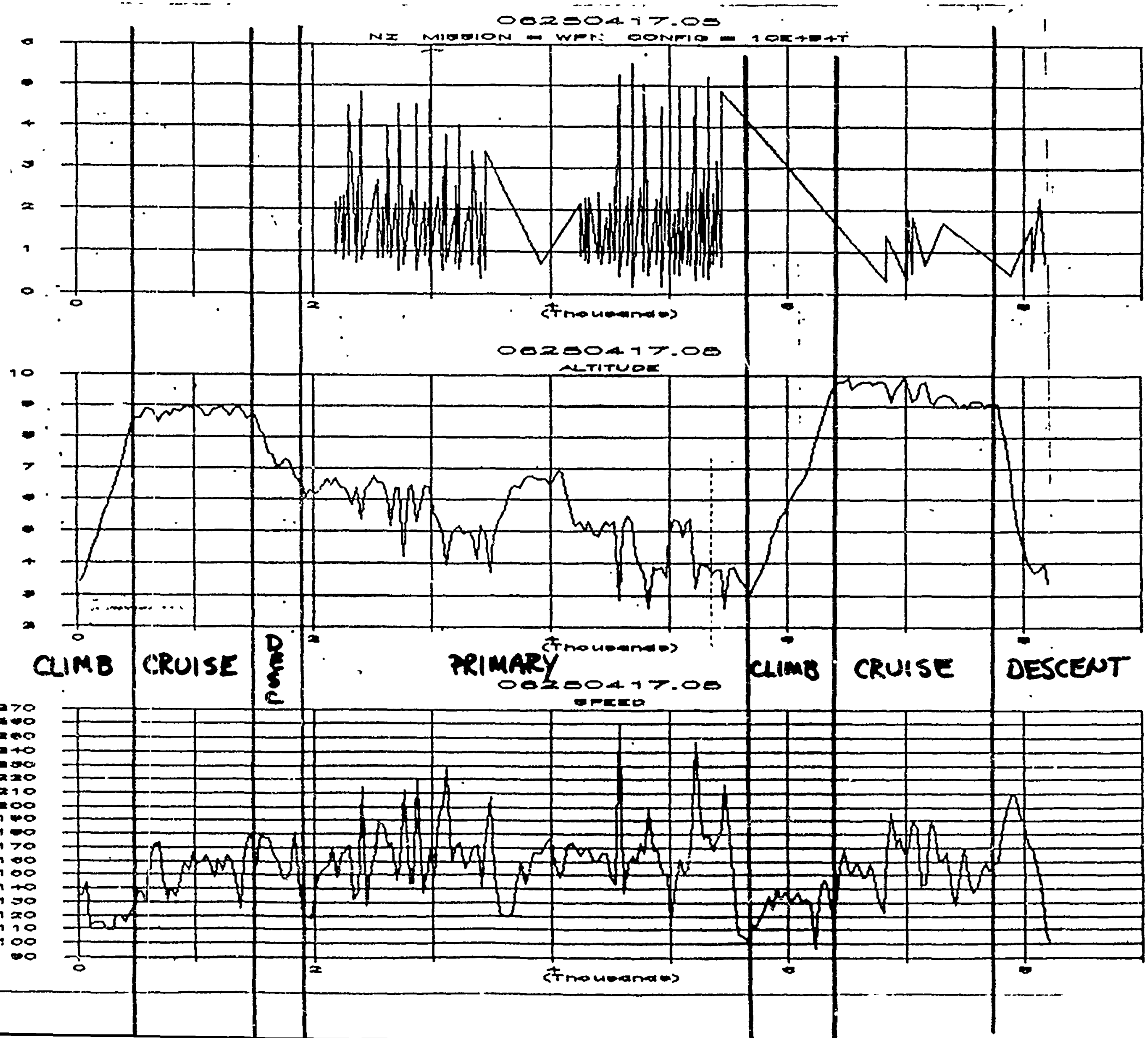
07850403.08


Periodic Altitude
Pressure Altitude in Feet
(Thousands)
8
7
6
5
4
3
2
0
2
4
6
(Thousands)
Flight Time in Seconds

This plot shows the altitude of the aircraft at periodic intervals throughout the flight. There are large altitude excursions near 1800 seconds and between 2600 and 4000 seconds into the flight.

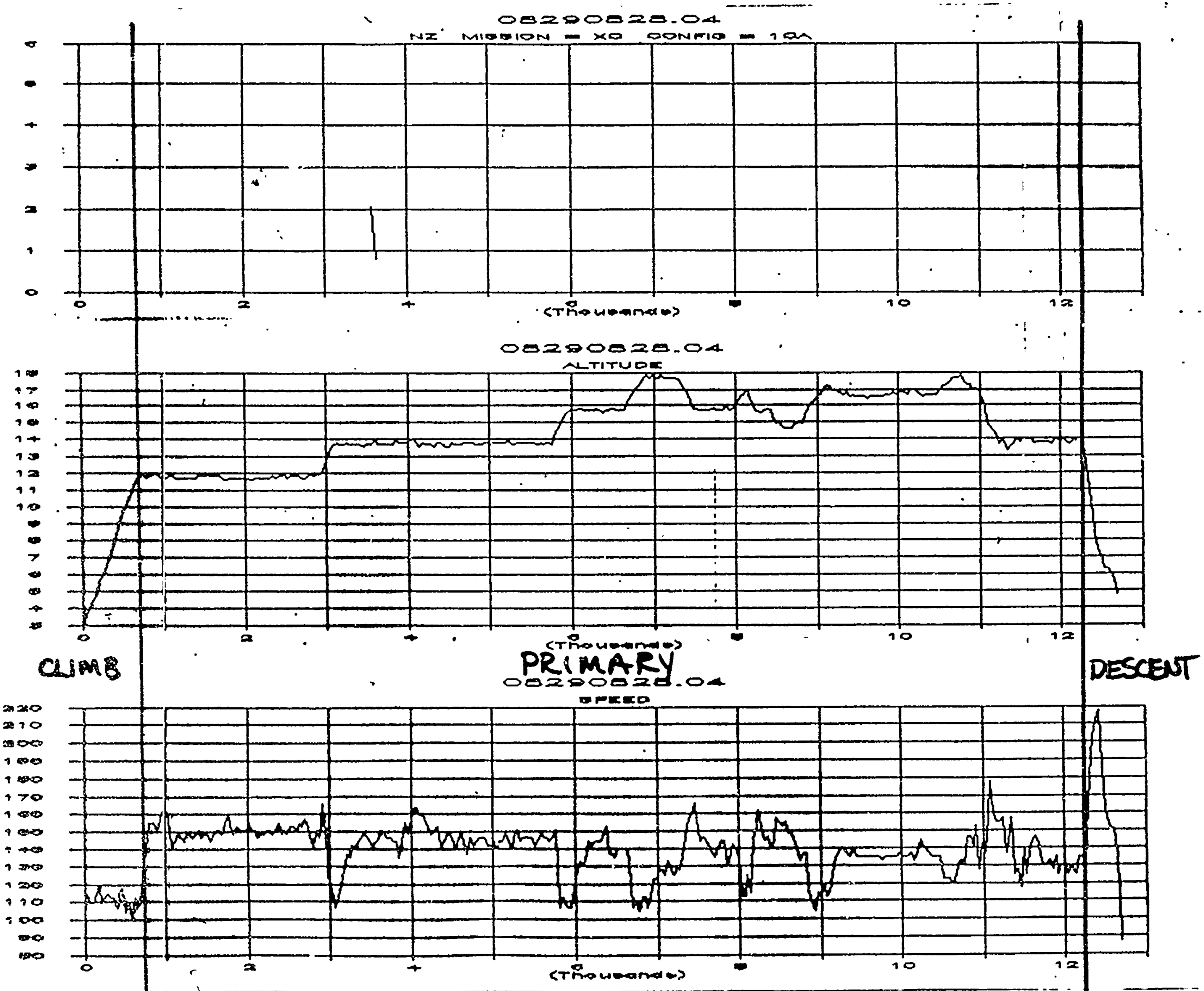

07850403.08
Periodic Nz
Nz
7
6
5
4
3
2
1
0
−1
0
2
4
6
(Thousands)
Flight Time in Seconds

The next plot further illustrates that at times other than when the pilot was performing apparent store-release passes, he was maintain approximately 1 g and not stressing the airframe.


06250417.08
NZ MISSION = WPN CONFIG = 10E+B4T
(Thousands)
06250417.08
ALTITUDE
(Thousands)
CLIMB
CRUISE
DESC
PRIMARY
CLIMB
CRUISE
DESCENT
06250417.08
SPEED
(Thousands)

Generally the flight data is plotted as indicated on this chart for use during the phase marking portion of the data reduction. The top trace is the peak Nz time history and the bottom two traces show the period altitude (middle trace) and velocity (lower trace). The information at the top of the plot details that the aircraft had a serial number of 0625 and the data of flight was April 17 and engine start was between 5:00 and 6:00 am. The pilot indicated that he flew a weapon mission with a given configuration.

Using the basic guidelines, the phase of flight are as indicated. The aircraft took off and climbed to a cruising altitude prior to a quick descent to set up for the weapons passes on the firing range. After the pilot had completed his primary mission, he climbed up to the cruising altitude for a return-to-base followed by a descent for landing.


08290828.04
NZ MISSION = XO CONFIG = 10A
(Thousands)
08290828.04
ALTITUDE
(Thousands)
CLIMB
PRIMARY
DESCENT
08290828.04
SPEED
(Thousands)

The previous chart showed an active mission for the OV-10 but occasionally the missions are benign as indicated by this plot of a cross-country. Notice that there was one significant peak and valley which occurred at about 3500 seconds into the flight. This mission consists almost entirely of the primary phase, constant altitude flying between destinations.

Pre-flight Health Data and Flight Time Summary

07:57:26	POWER ON	- POWER ON, 8:00 AM
	SELF TEST RAN	
	AVG. START G = 1.00000 EXP. 0.9 to 1.1	AUTOMATIC CALIBRATION OF NZ ACCELEROMETER AND ALTITUDE AND VELOCITY TRANSDUCERS
	AVG. START DYN. PRESS. = 0.00000 EXP. 0.0	
	AVG. START STAT. PRESS. = 27.14002 EXP. 26 to 28	
	GROUND PREFLIGHT 1757 SECONDS FOR FLIGHT 1	
08:26:43	TAKE OFF FLIGHT 1	FLIGHT 1, 2 HOURS
10:27:03	LAND FLIGHT 1	
	DURATION OF FLIGHT 1 WAS 7220 SECONDS	
	GROUND TIME 7 SECONDS BEFORE FLIGHT 2	- TOUCH AND GO, 7 SECONDS
10:27:10	TAKE OFF FLIGHT 2	FLIGHT 2 AROUND THE PATTERN FOR 3.5 MINUTES
10:30:37	LAND FLIGHT 2	
	DURATION OF FLIGHT 2 WAS 207 SECONDS	
	GROUND TIME 6 SECONDS BEFORE FLIGHT 3	- TOUCH AND GO, 6 SECONDS
10:30:43	TAKE OFF FLIGHT 3	FLIGHT 3 AROUND THE PATTERN FOR 4 MINUTES
10:34:30	LAND FLIGHT 3	
	DURATION OF FLIGHT 3 WAS 227 SECONDS	
	GROUND TIME 424 SECONDS BEFORE FLIGHT 4	- FULL STOP LANDING, 7 MINUTES
10:41:34	TAKE OFF FLIGHT 4	FLIGHT 4 RETURN-TO-BASE WITH FULL STOP LANDING, 30 MINUTES
11:11:56	LAND FLIGHT 4	
	DURATION OF FLIGHT 4 WAS 1822 SECONDS	

In addition to the flight data being recorded, all parameters about the health of the system and major activities within the flight are also available. Information in this health file include the results of the self-test performed by the recorder, and time associated with take-off, landing, the duration of flight and the number of touch-and-goes flown during the flight.

This chart shows a typical health file from a flight. After power came on, the recorder performed a self-test and calibration of the transducers. The aircraft was involved in ground operations for 30 minutes prior to take off for flight 1. That flight lasted for about 2 hours at which time the pilot performed a series of touch and goes at an auxiliary base. On the third landing, he performed a full stop landing and spent 7 minutes taxiing into position for the take off to return to his home base.

PRELIMINARY RESULTS

Six instrumented aircraft

After 7 and 1/2 months (Mar 31 – Nov 21)

Total Air Force flight hours	1770.3
Total recorded hours	895.4 (50.6%)
Total valid hours	793.7 (44.8%)

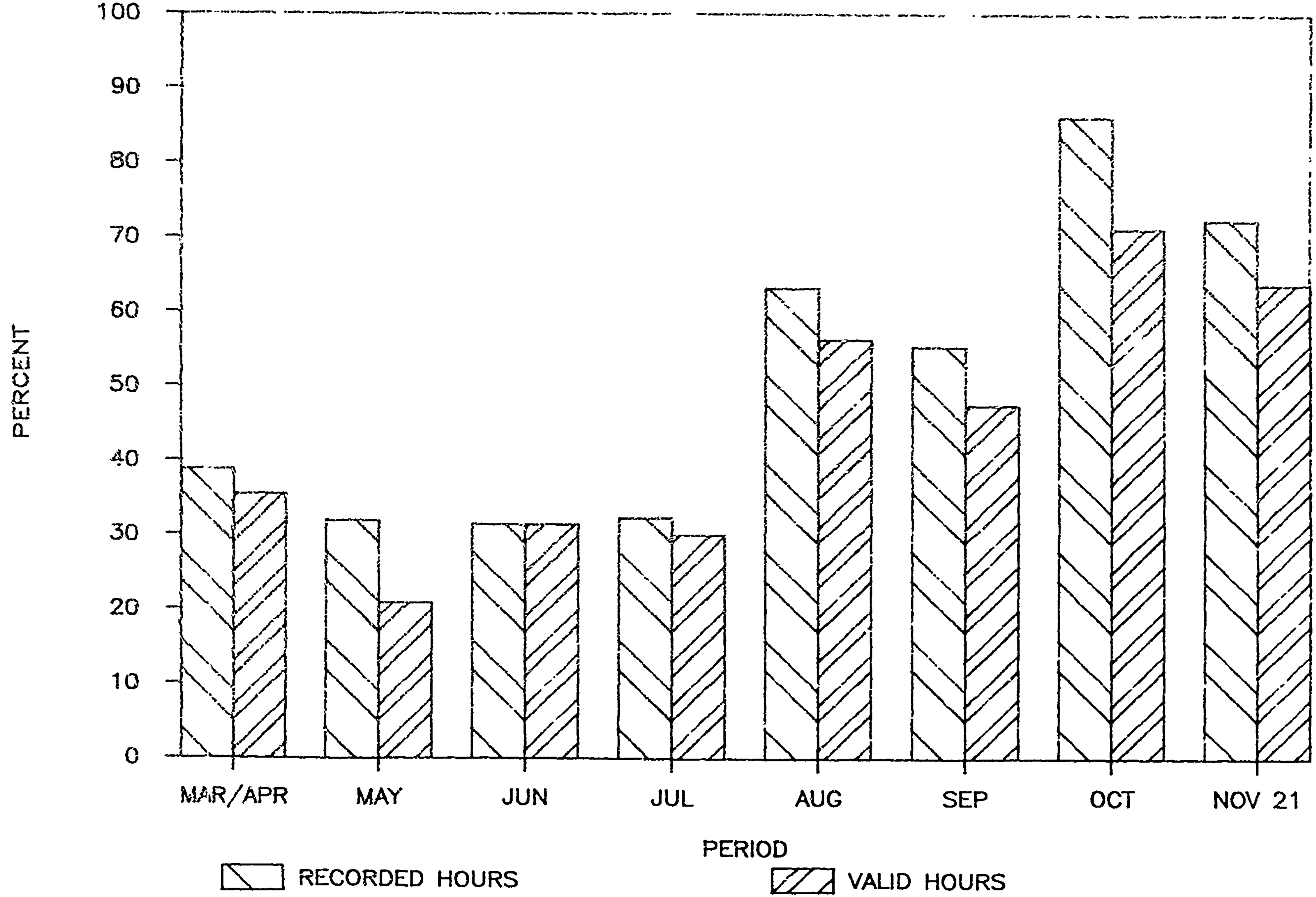
During the 9 month recording period, 6 aircraft were equipped with recorders and after about 7 and 1/2 months of data collection (Mar 31 - Nov 21, 1986), the preliminary results are:

Total Air Force flight hours	1770.3
Total recorded hours	895.4 (50.6%)
Total valid hours	793.7 (44.8%)

A better way to see these results is on the following bar charts that show the recording history of all aircraft on a month-by-month basis.

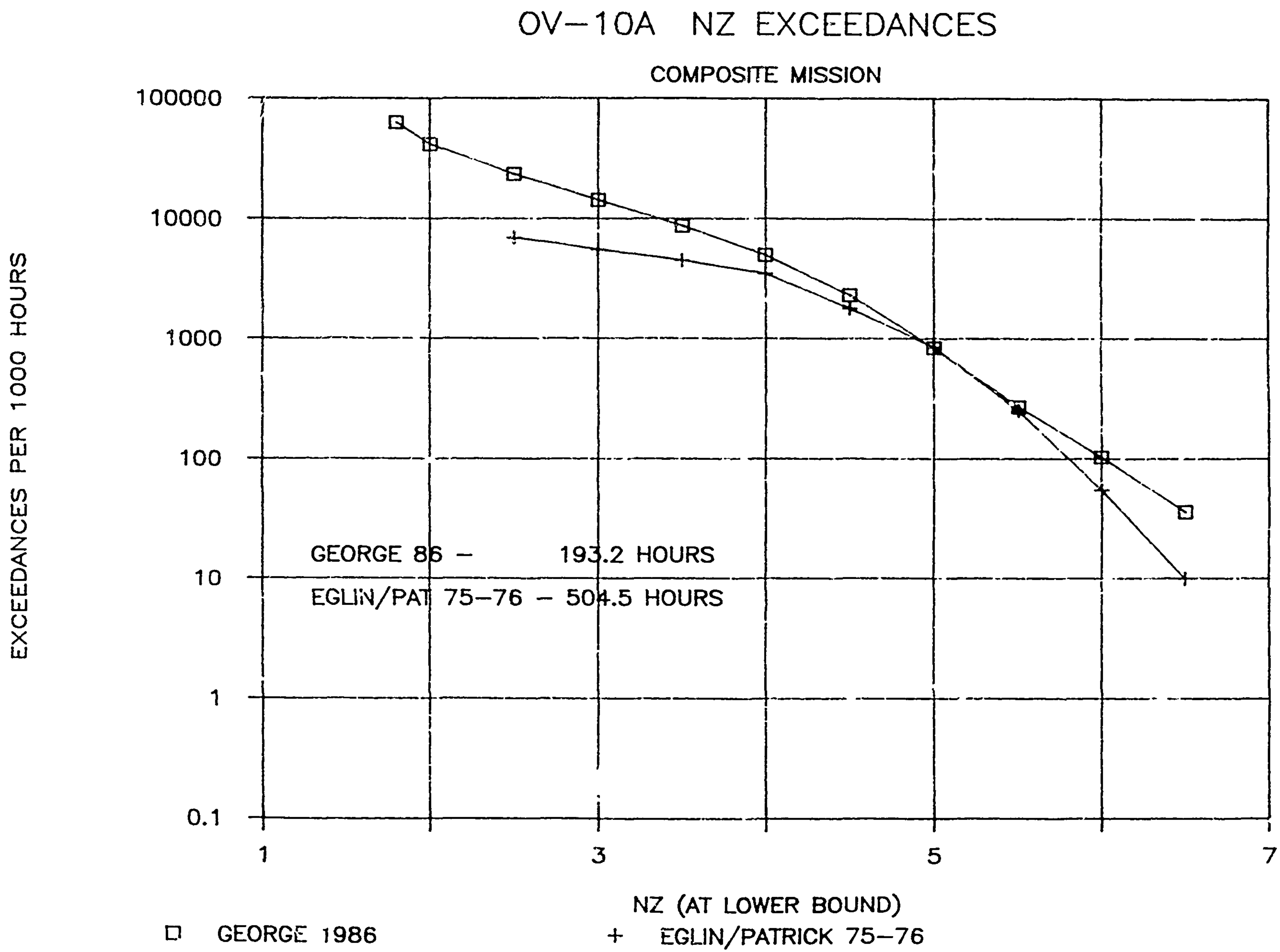
ALL A/C

PERCENT OF AF FLIGHT HOURS



The first four months exhibit lower than expected retrieval rates due to a variety of problems. Infant mortality problems with the new system rendered several installations inoperable for extended periods. In addition, several recorders were experiencing larger that expected power surges from the aircraft input power line. These surges were greater than military specification and the internal circuitry of the recorder absorbed these surges in order to protect the sensitive electronics. The end result of this protection is to prevent power from reaching the recorder and thus no data is recorded.

Efforts by ESPRIT during August and September greatly improved the retrieval rates as evidenced by the near doubling of the percentages. Excellent recovery rates are shown for the months of October and November where between 70-75% of valid data was recorded. Problems encountered early on have been remedied and the retrieval rate is approaching system capabilities.


OV–10A NZ EXCEEDANCES
COMPOSITE MISSION
100000
10000
1000
100
10
1
0.1
EXCEEDANCES PER 1000 HOURS
GEORGE 86 – 193.2 HOURS
EGLIN/PAT 75–76 – 504.5 HOURS
1
3
5
7
NZ (AT LOWER BOUND)
GEORGE 1986
+ EGLIN/PATRICK 75–76

One of the strong points about this approach to flight data recording is the ease and quickness with which data can be analyzed using available software. For example, an indication of the severity of flying can be quickly determined by examining an Nz exceedance plot. This plot shows the number of Nzs that exceed a given Nz value in 1000 hours of flying. Preliminary exceedance data for the OV-10 fleet at George AFB is compared with data taken from Eglin AFB and Patrick AFB. The amount of composite mission hours for the 1975-1976 data is 505 hours and for the 1986 data, 193 hours.

This preliminary analysis shows tnat the current flying environment for the aircraft located at George A.F.B. is more severe than indicated by the previous survey. No attempt has been made to consider the mission mix associated with each spectra. A more detailed analysis will be completed and presented in the final report to SA-ALC.

CONCLUSIONS

Achieving objectives and goals

Advantages

- Reduce human judgment
- Instantaneous recording
- Recording only significant data
- Eliminate dependency on mainframes
- Retain accuracy and reliability
- Increase data retrieval rate
- Reduced manpower requirements
- Data stored on tape

Wide variety of applications

In conclusion, the objectives and goals of this program are being met. The early reliability of the system was not as expected but the trend was reversed once the individual problems were identified and corrected.

It can be seen that there are many advantages to solid-state on-board recorders:

1) reduced human judgment in analyzing dynamic traces. The recorder captures the exact peak and/or valley and all associated data.
2) the dynamic parameters are recorded instantaneously.
3) only significant data is recorded.
4) dependency on large mainframe computers has been eliminated while maintaining accuracy and reliability.
5) increased retrieval rates.
6) reduced manpower requirements.
7) data can be transferred to magnetic tape for future analysis.

The main points to remember are that data reduction and direct use of the flight data can begin almost immediately after installation and retrieval of first flight data. This application is a simple, dedicated VGH recorder system to meet ASIP needs. Because the system is a multichannel recorder capable of interfacing with a variety of transducers, the system can be expanded to function in a wide range of ASIP activities.

Use Of ASIP Instrumentation For Ground Loads Testing of F111

By

Tony G. Gerardi
AFWAL/FIBE

Abstract for Paper

on

"Use of ASIP Instrumentation for Ground Loads Testing of F-111"

An ongoing program known as HAVE BOUNCE has been established to determine the capability of Air Force aircraft to operate on rough, battle damaged runways. The approach is to:

1. Develop a computer program capable of simulating the aircraft traversing the rough surfaces during taxi, takeoff and landing.

2. Validate the computer program with measured aircraft test data.

3. Use the validated computer program to determine the roughness capabilities of the aircraft.

Generally, under the HAVE BOUNCE (HB) Program, each aircraft is fully instrumented and a test dedicated totally to the runway roughness problem is conducted for the purpose of validating the ground loads computer programs. This is an expensive venture requiring support from many Air Force organizations and many times the aircraft manufacturer is contracted to perform the test.

For the F-111A a different approach was taken. Approximately 20% of the F-111 fleet is equipped with ASIP (Aircraft Structural Integrity Program) instrumentation and recording equipment. By utilizing one of the ASIP equipped aircraft on a paved surface of known roughness, it was possible to obtain the measured data necessary for validating the HAVE BOUNCE computer programs and consequently avoid a costly dedicated F-111 test program.

The test was performed at Mountain Home AFB ID the week of 5 Nov 84. Two ASIP equipped F-111A's belonging to the 366 TFW were tested (Serial Numbers 67-0102 and 67-0086. The tests were performed on a non-interference basis prior to routine training missions. Although the quantity and quality of the data obtained were not as good as that for a dedicated HAVE BOUNCE test, it was sufficient to validate the computer models.

The purpose of this proposed paper is to illustrate a profitable spin off benefit of the ASIP program and encourage similar test programs.

The test was the result of a cooperative effort between three Air Force commands, (1) AFSC: the R&D laboratory requiring the test data, (2) TAC: the F-111A operators and resident of Mountain Home AFB, ID, and (3) AFLC: the ASIP experts who provided the instrumentation, recording and data reduction.

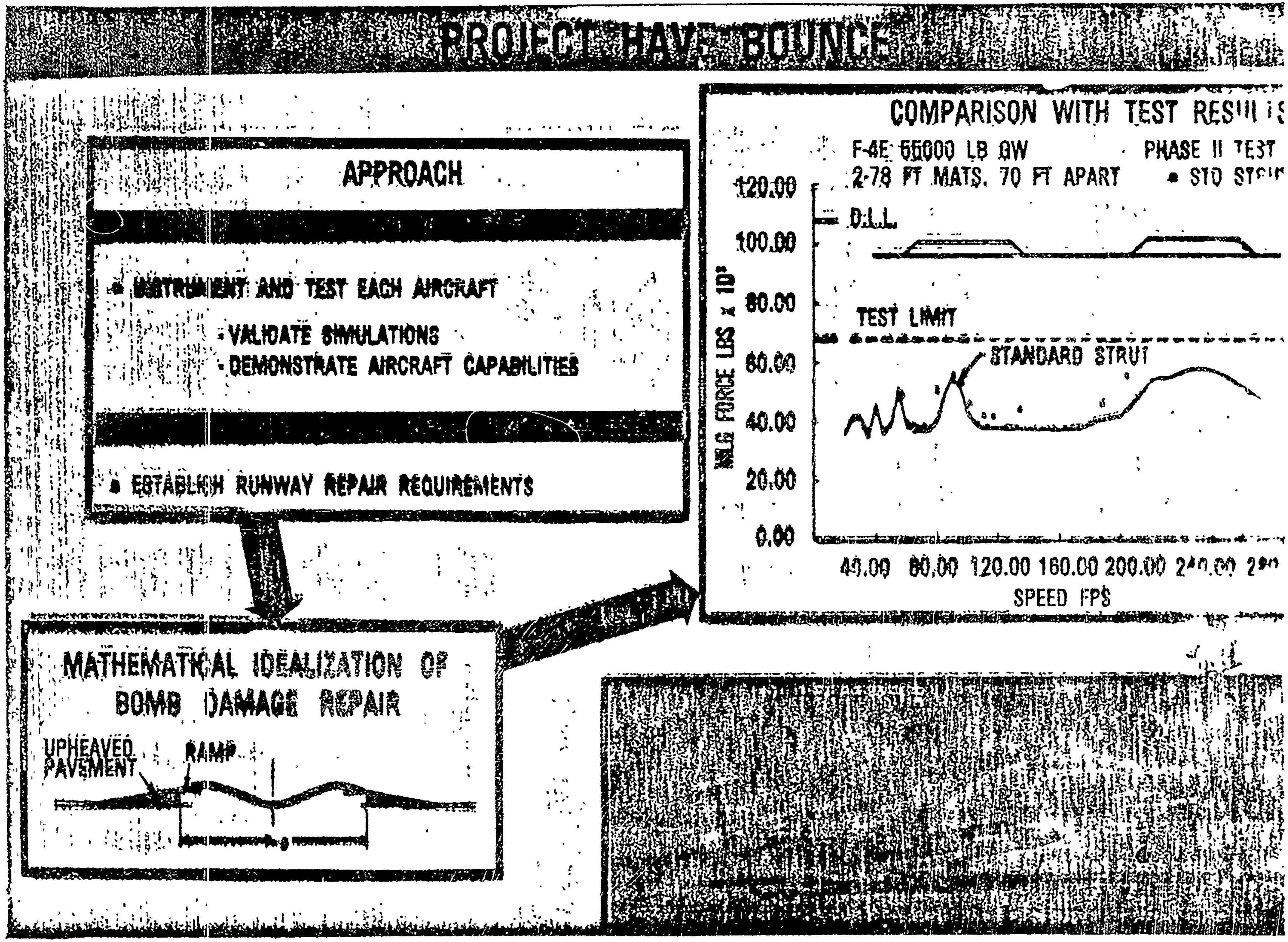
STRUCTURAL INTEGRITY BRANCH

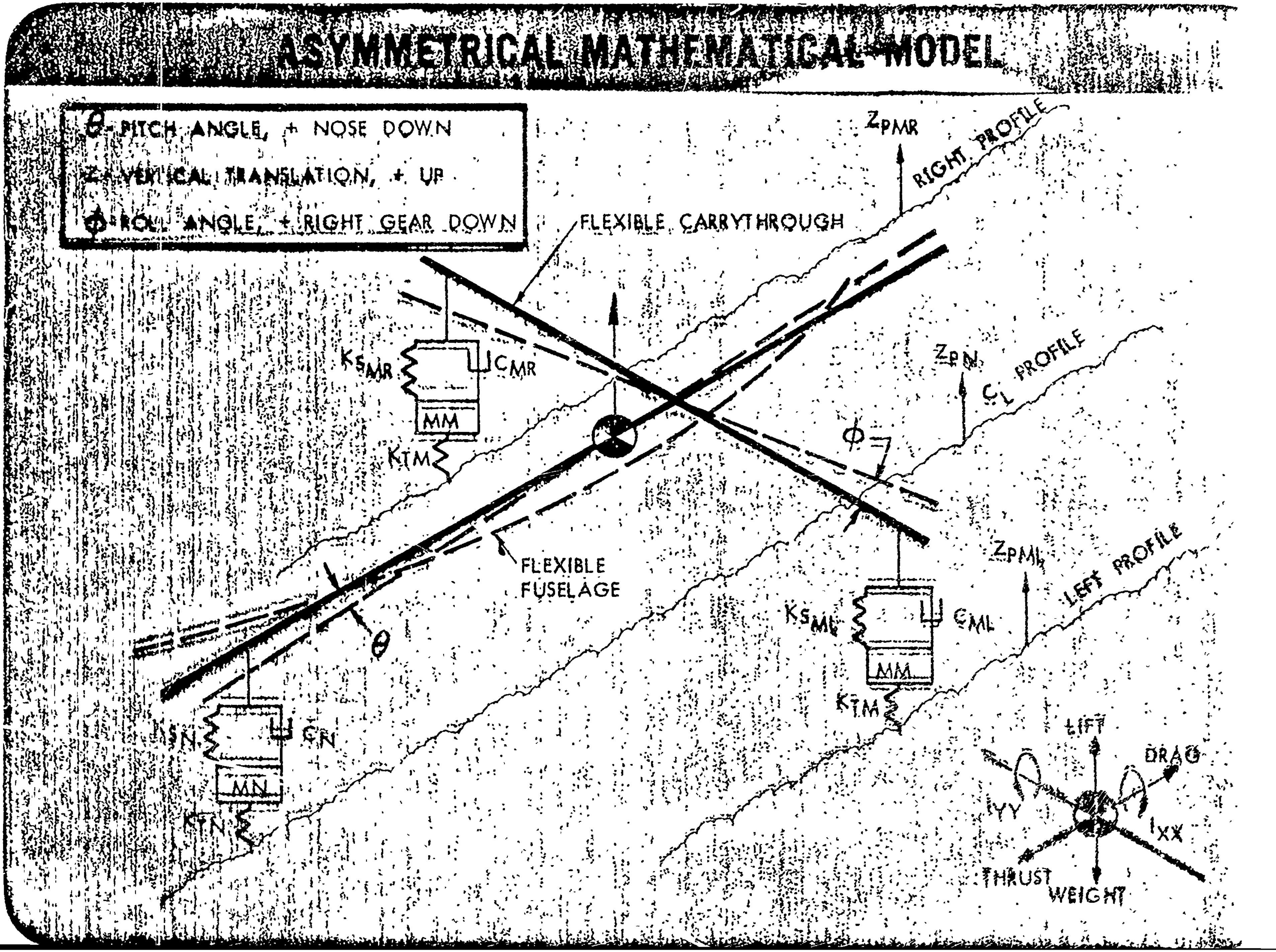
ASD / AFWAL / FIBE

(CHIEF ENGINEER)
TONY GERARDI

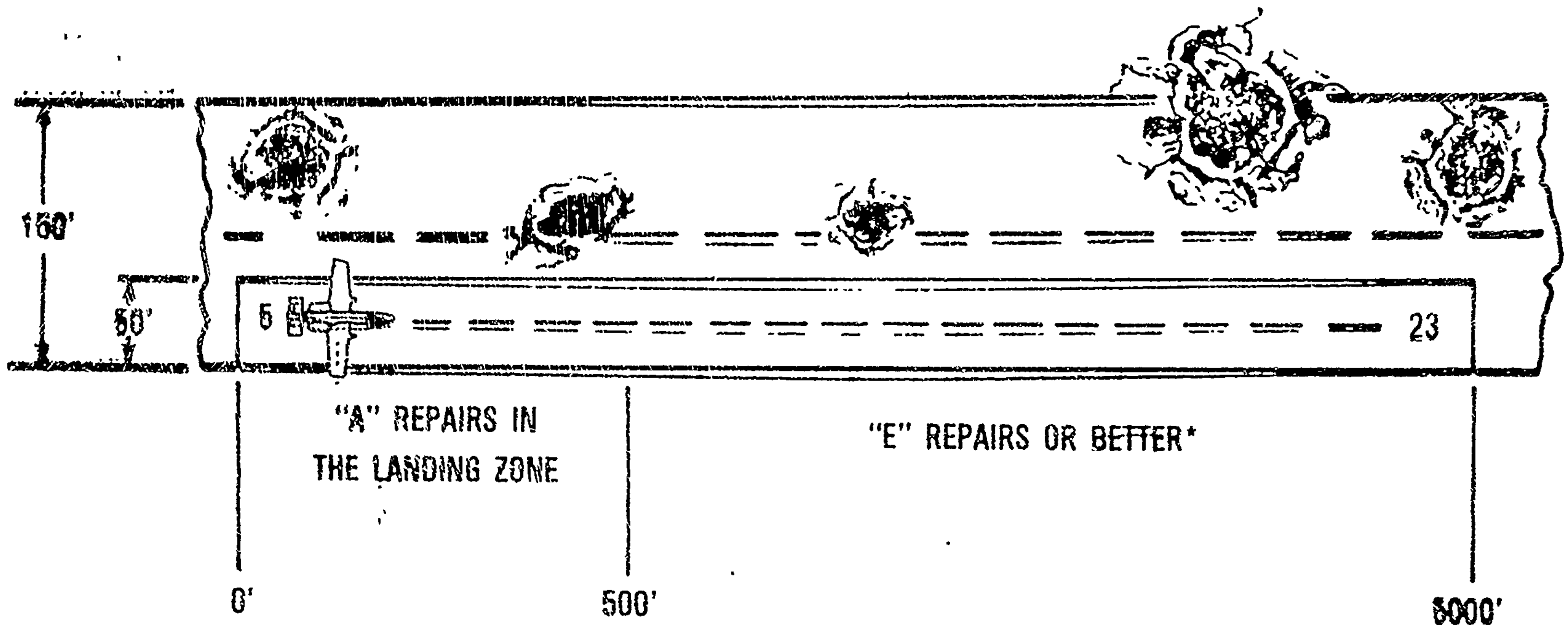
Commercial - (513)255-2544
Autovon - 785-2544

BOMB DAMAGED RUNWAYS
PAKISTANI AIR FIELD CIRCA 1974


PROJECT HAVE BOUNCE
APPROACH
• INSTRUMENT AND TEST EACH AIRCRAFT
• VALIDATE SIMULATIONS
• DEMONSTRATE AIRCRAFT CAPABILITIES
• ESTABLISH RUNWAY REPAIR REQUIREMENTS
MATHEMATICAL IDEALIZATION OF
BOMB DAMAGE REPAIR
UPHEAVED PAVEMENT
RAMP
COMPARISON WITH TEST RESULTS
F-4E 55000 LB GW
2-78 FT MATS, 70 FT APART
PHASE II TEST
• STD STRUT
D.L.L.
TEST LIMIT
STANDARD STRUT
MLG FORCE LBS x 10³
120.00
100.00
80.00
60.00
40.00
20.00
0.00
40.00
80.00
120.00
160.00
200.00
SPEED FPS


ASYMMETRICAL MATHEMATICAL MODEL
θ - PITCH ANGLE, + NOSE DOWN
Z - VERTICAL TRANSLATION, + UP
Φ - ROLL ANGLE, + RIGHT GEAR DOWN
FLEXIBLE CARRYTHROUGH
FLEXIBLE FUSELAGE
Z_{PMR}
RIGHT PROFILE
Z_{PN}
℄ PROFILE
Z_{PML}
LEFT PROFILE
$K_{S_{MR}}$
C_{MR}
MM
K_{TM}
$K_{S_{ML}}$
C_{ML}
MM
K_{TM}
K_{SN}
C_N
MN
K_{TN}
Φ
θ
LIFT
DRAG
I_{YY}
I_{XX}
THRUST
WEIGHT

UNIDIRECTIONAL MINIMUM OPERATING STRIP (MOS)



- * "A" QUALITY REPAIRS ARE REQUIRED IN FIRST 500 FEET (LANDING ZONE)
- MULTIPLE "E" QUALITY REPAIRS MUST BE SPACED 300 FEET APART.
- AN "E" REPAIR FOLLOWED BY A "C" REPAIR CAN BE SPACED 120 OR MORE FEET APART
- MULTIPLE REPAIRS CLOSER THAN 120 FEET APART SHOULD BE MADE AS A SINGLE REPAIR
- TAXIWAY REPAIRS MAY USE "E" QUALITY REPAIRS. TAXI SPEEDS OVER "E" REPAIRS SHOULD BE LIMITED TO 15 KNOTS

F-111 MODEL VALIDATION

- GENERAL DYNAMICS F-111 MODEL DEVELOPED
- AFWAL F-111 MODEL DEVELOPED
- NO MEASURED DATA AVAILABLE FOR VALIDATION
- INSUFFICIENT FUNDS ($3M) TO CONDUCT HAVE BOUNCE TESTS

SEQUENCE OF EVENTS

- PRELIMINARY DISCUSSIONS WITH OK CITY ALC
- OBTAIN TAC Hq APPROVAL
- IDENTIFY TEST SITE AND POC
- WRITE AND COORDINATE TEST PLAN
- CONDUCT TEST AT MOUNTAIN HOME
- ALC REDUCE ASIP DATA
- WRITE FINAL REPORT (AFWAL-TM-85-214-FIBE)

ALTERNATE APPROACH

- 20% OF F-111 FLEET HAVE ASIP (AIRCRAFT STRUCTURAL INTEGRITY PROGRAM) INSTRUMENTATION AND RECORDING SYSTEMS INCLUDING STRUT PRESSURE GAGES

- CONDUCT LIMITED TESTS USING ASIP EQUIPPED F-111 OPERATIONAL AIRCRAFT AT ITS HOME BASE

- OKLAHOMA CITY ALC PROVIDE ASIP SUPPORT, DATA TAPES AND DATA REDUCTION

- TAC HQS PROVIDE AUTHORITY FOR TEST AND SELECT TEST SITE AND TAC POC

F-111 ASIP RECORDED PARAMETERS

	PARAMETER	RECORDING RATE (SAMPLES PER SECOND)
1.	MACH NUMBER	1
2.	PRESSURE ALTITUDE	1
3.	OUTSIDE AIR TEMPERATURE	1
4.	WING POSITION	1
* 5.	ACCELERATION, Z AXIS	30
6.	ACCELERATION, X AXIS	15
7.	ACCELERATION, Y AXIS	10
8.	ROLL RATE	15
9.	YAW RATE	15
10.	PITCH RATE	15
11.	FLAP POSITION	1
12.	LANDING GEAR POSITION	1
13.	SINK RATE	5
14.	LH HORIZONTAL TAIL POSITION	15
15.	RH HORIZONTAL TAIL POSITION	15
16.	RUDDER POSITION	15
*17.	FUEL FLOW, RIGHT ENGINE	1
18.	FUEL FLOW, LEFT ENGINE	1
19.	TRUE ANGLE OF ATTACK	5
20.	RIGHT OUTBOARD SPOILER POSITION	30
21.	LEFT OUTBOARD SPOILER POSITION	30
*22.	LEFT MAIN LANDING GEAR OLEO PRESSURE	1
*23.	RIGHT MAIN LANDING GEAR OLEO PRESSURE	1
*24.	NOSE LANDING OLEO PRESSURE	15

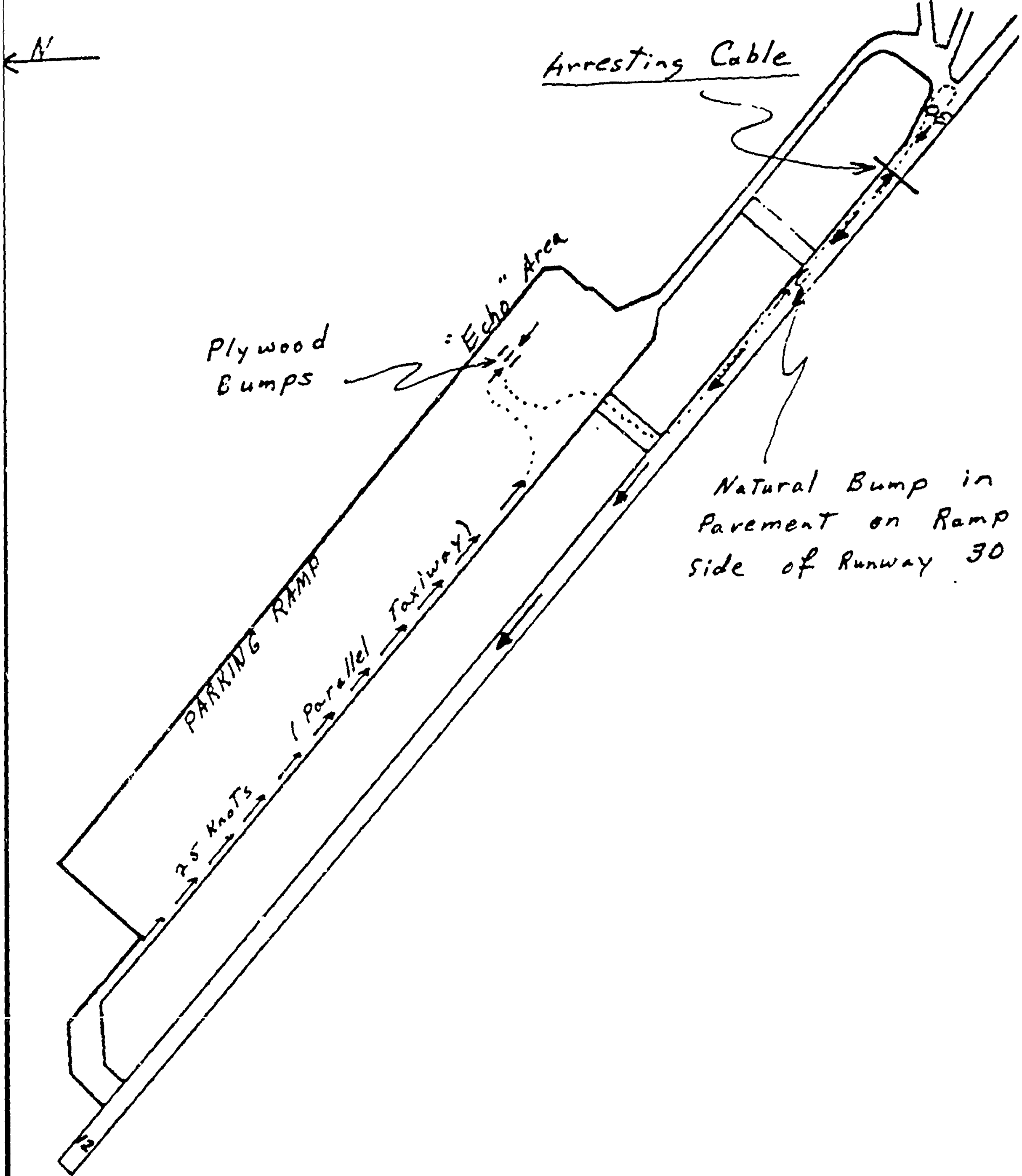
* USED IN THE F-111 GROUND LOADS TEST.

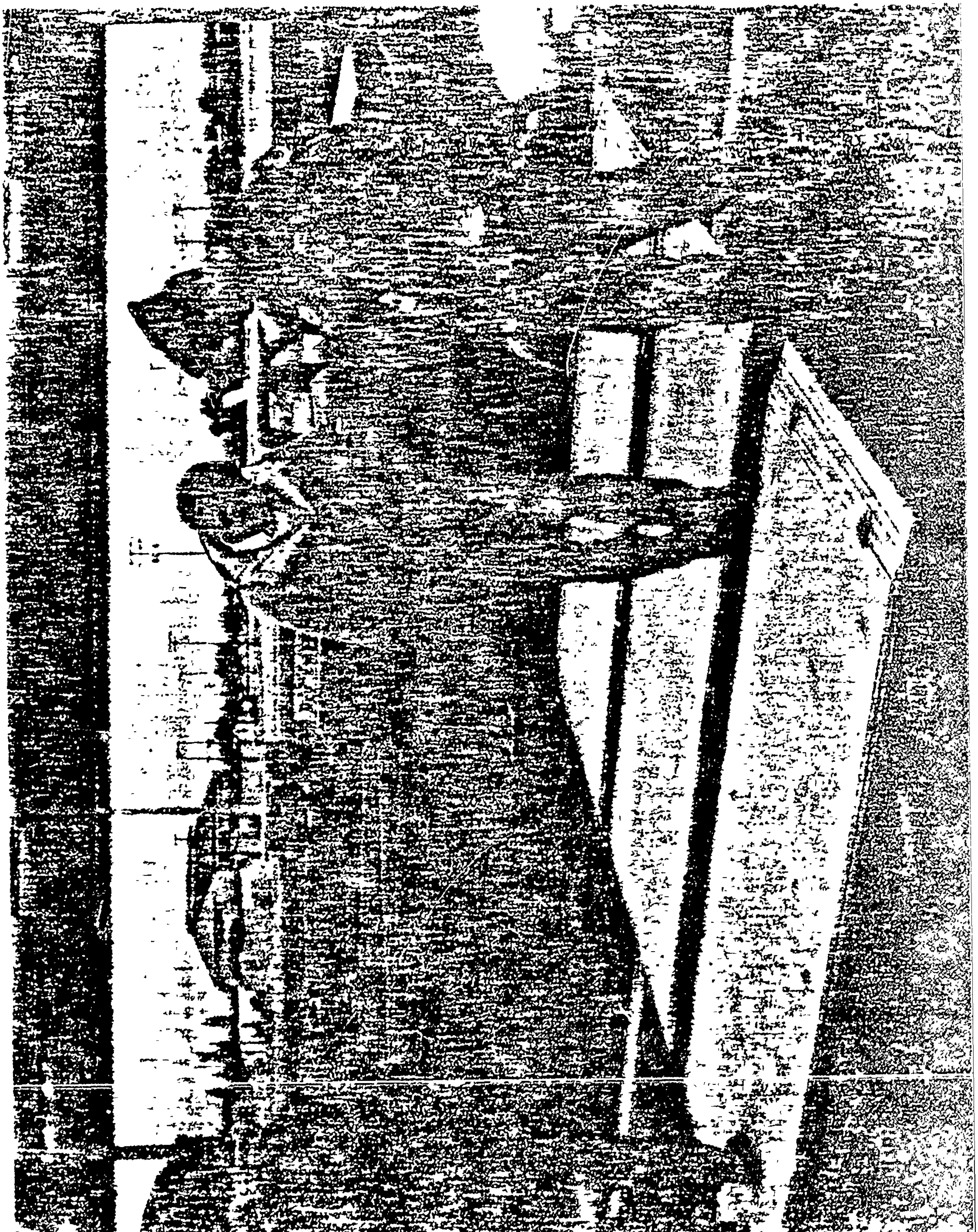
TEST SITE

- MOUNTAIN HOME AFB, IDAHO 366TFW SELECTED AS TEST SITE

- TWO F-111A WERE SELECTED FOR THE TEST BASED ON HISTORY OF INSTRUMENTATION RELIABILITY AND AIRCRAFT STATUS

Mountain Home AFB





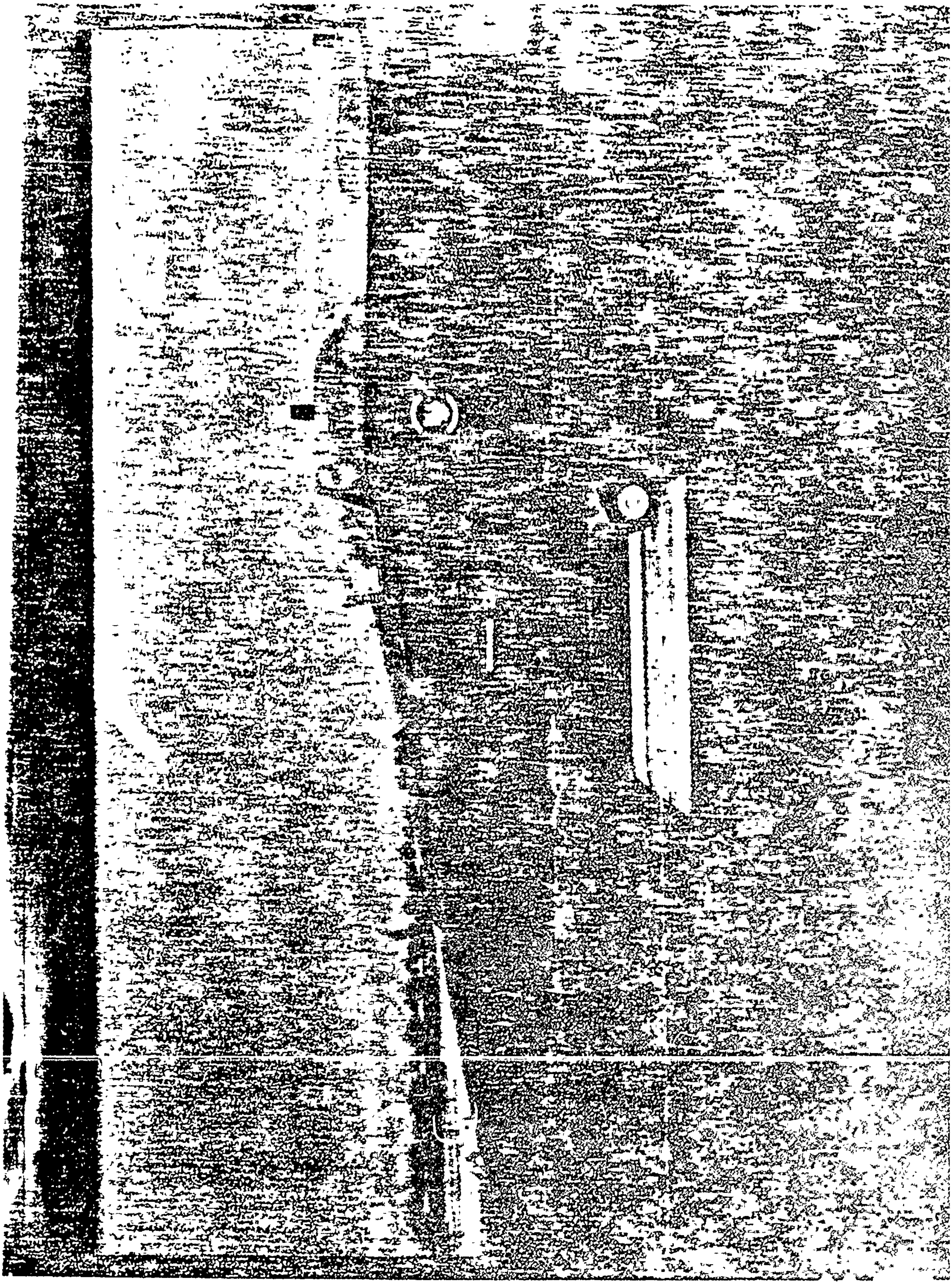
TEST AIRCRAFT CONFIGURATION

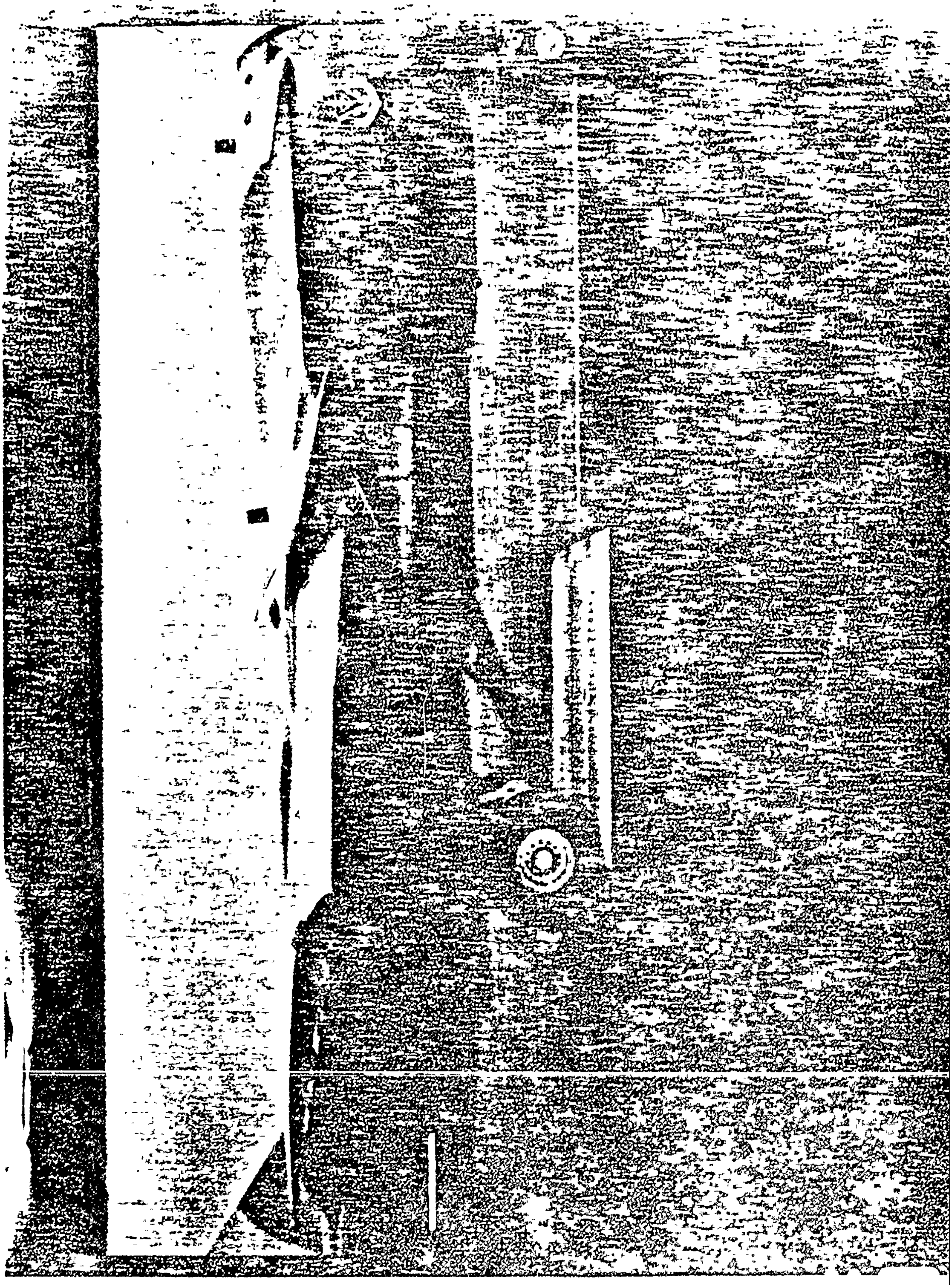
- TYPICAL TRAINING WEIGHT FLOWN AT MT HOME AFB

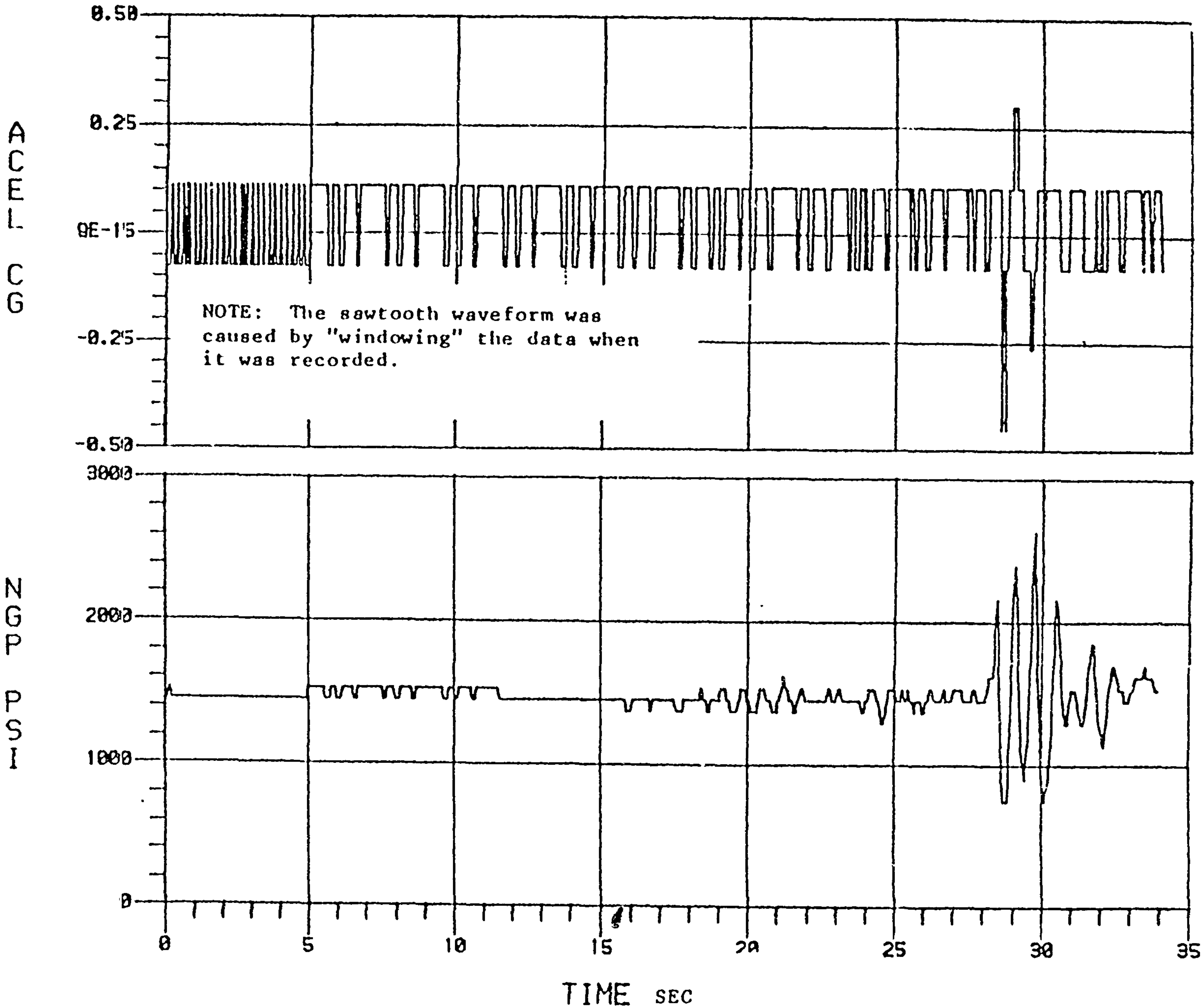
  GW = 83000
  CG = 27% MAC

- STRUTS AND TIRES WERE SERVICED IN ACCORDANCE WITH TECH ORDER PRIOR TO TESTING

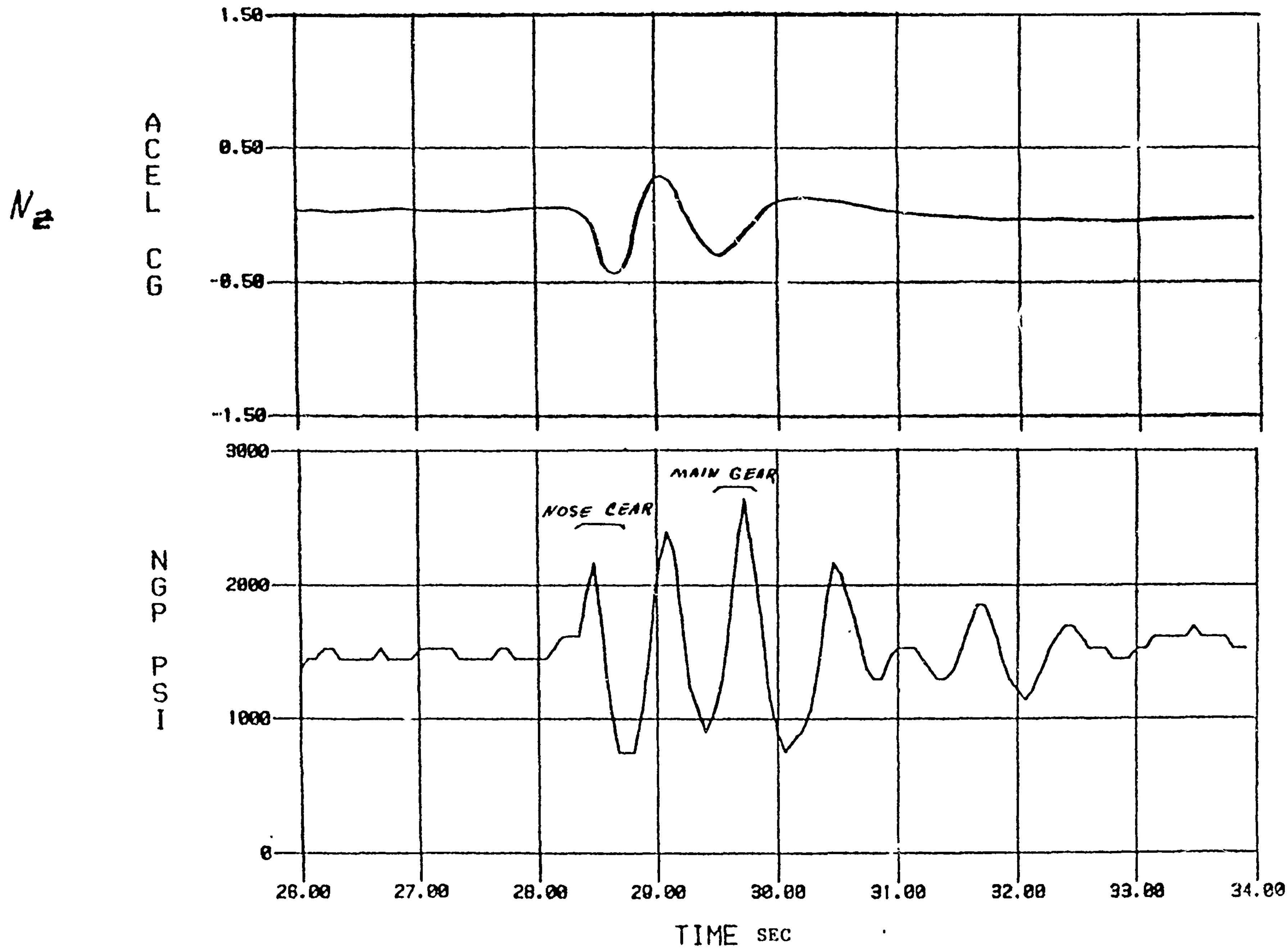
- NEW ASIP TAPE INSTALLED







Time History Plot for Test 5, 15 Knot Taxi Over Plywood Bumps in the 30 Direction



Time History Plot for Test 5, 15 Knot Taxi Over Plywood Bumps in the 30 Direction

VALIDATION

- NLG STRUT MODEL (PRESSURE, DAMPING, FRICTION)
- PITCH FREQUENCY (IYY, TUNING, FLEXIBILITY EFFECTS)
- PLUNGE FREQUENCY
- LIMITED MLG STRUT MODEL VERIFICATION
- NO BRAKING RUNS
- ONE HIGH SPEED RUN (TAKEOFF) NO PROFILE DATA

CONCLUSIONS

- GOOD NOSE LANDING GEAR STRUT DATA MEASURED
- FAIR MAIN LANDING GEAR STRUT AND CG LOAD FACTOR DATA MEASURED
- ROD & LEVEL SURVEY DATA COLLECTED FOR MOST TEST SURFACES
    - MEASURED DATA SUFFICIENT FOR MODEL VALIDATION:
      (BRAKING NOT INCLUDED)
      (SPALLS NOT INCLUDED)
- A FINAL REPORT WITH A DESCRIPTION OF THE AIRCRAFT AND SPEED TESTED AND THE TEST DATA IN ENGINEERING UNITS PUBLISHED (AFWAL-TM-85-214-FIBE)
- THE MEASURED RUNWAY PROFILE DATA IS CONTAINED IN (AFWAL-TM-85-0156-FIBE)

Demonstration Of A Durable Honeycomb Control Surface

By

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LTV Aerospace & Defense Co.

And

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AFWAL

DEMONSTRATION OF A DURABLE HONEYCOMB CONTROL SURFACE

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Air Force Wright Aeronautical Laboratories

ABSTRACT

Honeycomb structures have, in the past several years, received a bad reputation due to problems with corrosion and early failures in service. This bad reputation is mostly undeserved, because the problems experienced in service are not problems inherent in honeycomb itself, but rather problems that result from design configurations that did not consider the full range of environmental conditions to which a honeycomb structure may be exposed. When a honeycomb structure's design considers all possible environmental conditions, it can operate in a durable manner. This paper outlines the environments that cause durability and corrosion problems, and discusses the design elements that must incorporated due to these environments.

The application of improved structural designs, materials, and manufacturing methods to achieve improved durability is discussed. The durability, cost, and weight of an advanced design honeycomb component is compared to a baseline component by analysis and testing. This comparison demonstrates the ability to increase the durability of honeycomb structures without significant increases in cost or weight.

INTRODUCTION AND BACKGROUND

Honeycomb structures have gained prominence in applications such as aircraft control surfaces because of their high structural efficiency and design versatility. However, in recent years, honeycomb control surfaces have earned a bad reputation due to problems with corrosion and early failures in service. Most of this bad reputation is undeserved because the problems experienced in service are not problems inherent in honeycomb itself, but rather problems that result from design configurations that did not consider the full range of environmental conditions to which a honeycomb control surface may be exposed.

A durable honeycomb control surface that was designed with consideration of the full range of environmental conditions has been demonstrated by LTV Aerospace and Defense Company under Air Force Contract Number F33615-81-C-3219, the Structural Improvement of Operational Aircraft Program. In this program, it is shown by comparison between a baseline and an advanced component that major increases in durability can be achieved without significant increases in component cost or weight. This paper presents the critical design criteria and a comparison of the baseline and advanced components.

The baseline component is an F-111 outboard spoiler of conventional design and fabrication. It utilizes chem-milled aluminum skins, bare aluminum core, multiple piece edge closures with fiberglass patch doublers, and conventional adhesive system with no surface treatment. The configuration is typical of many honeycomb control surfaces found on active fleet aircraft and offers an ideal opportunity to apply generic advanced design concepts. Figure 1 illustrates the baseline spoiler.

The baseline spoiler was redesigned and remanufactured using advanced metallic design concepts to achieve major durability and reliability characteristics. The advanced spoiler incorporates integrally damped laminated aluminum skins, an unconstrained layer damping treatment, a superplastically formed lower skin that incorporates edge closure and spar details, a CRIII coated aluminum microcell honeycomb core, and an improved surface treatment and adhesive system. Figure 2 illustrates the advanced spoiler.

HONEYCOMB CONTROL SURFACE DESIGN CRITERIA

A major contributor to honeycomb control surface damage is the fluctuating pressure (or aeroacoustical environment) produced by turbulent boundary layer flow. This fluctuating pressure can induce damaging vibrations at mechanical resonances of the structure. Thus, once identified, these fluctuating pressure airloads must be fully considered in the design process to achieve a durable honeycomb control surface. Fluctuating pressure loads from jet engine exhaust blast are generally recognized in the design process, but the fluctuating pressure loads that result from turbulent boundary layer phenomena have been largely ignored. This can be a major oversight in the design criteria, because it has been shown that these fluctuating pressures cause fatigue of honeycomb structures. To illustrate the level of fluctuating pressure that may be present in a turbulent boundary layer, consider the analytical flow realms illustrated in Figure 3. Here, the fluctuating pressure, P, is defined as a fraction of the free stream kin tic energy per unit volume, Q ($1/2 \rho v^2$), for various turbulent boundary layer flow realms. For an attached turbulent boundary layer of fluctuating pressure of up to 0.6 percent of the free stream kinetic energy per unit volume can be imposed on a structure. When there is a local separation in the turbulent boundary layer, the level of fluctuating pressure can increase ten times, up to six percent of the free stream kinetic energy. If the locally separated flow interacts with a shock wave, the level of fluctuating pressure can increase five times, up to thirty percent of the free stream kinetic energy. The level of fluctuating pressure is often expressed in terms of decibels. For the case of a locally separated turbulent boundary layer interacting with a shock wave, the analytically predicted fluctuating pressure is equivalent to a 183 dB overall broadband random sound excitation. The levels of fluctuating pressure discussed above are for grazing flow. For incident flow, such as a deployed spoiler, a fluctuating pressure of up to fifty percent of the free stream kinetic energy can be imposed on a structure. It should be noted that these predicted flow realms exist for only a small fraction of the total flight time, but can still be extremely significant in terms of the design criteria for a structure.

To illustrate why these fluctuating pressures can be so significant and why damping is effective in increasing structural durability, consider the resonant response of a single degree of freedom system as illustrated in Figure 4. Figure 4 illustrates the response of a single degree of freedom mass, spring, and damper system in terms of a acceleration versus frequency normalized with respect to the natural frequency of the system. The response of the system is illustrated for varius sample loss factors, n. The sample loss factor is the damping coefficient of the system normalized wit respect to the critical damping coefficient of the system. At frequencies below the natural frequency of the system, the response is controlled by the stiffness of the system, and at frequencies above the natural frequency of the system the response is controlled by the mass of the system. At the natural frequency of the system the response is controlled by the damping of the system. Notice that at the natural frequency the response is inversely proportional to the sample loss factor. Additionally, the fatigue life of structure exposed to resonant response is an exponential function of the sample loss factor. Identical phenomena exist for continuous systems such as honeycomb control surfaces. Thus, if the frequency of the fluctuating pressures from a turbulent boundary layer occurs near a resonant frequency, the structure will respond in a resonant manner. Since typical honeycomb structures have low sample loss factors (η=0.02), they often respond to fluctuating pressure from a turbulent boundary layer in a damaging manner.

Damaging resonances can be effectively controlled by increasing the damping of the structure. Changing the weight or stiffness of a structure merely shifts its resonant frequencies to new frequencies at which fluctuating pressures may still be present. While this may reduce stresses to some extent, this reduction comes at the expense of additional component weight.

Problems with water entrapment in and corrosion of the honeycomb core are aggravated by the resonant vibrations. These vibrations can open or crack bondlines and sealers allowing moisture to accumulate inside the component. Later, if a structural repair is necessary due to skin or core damage from resonant vibrations, corrosion may be found inside the structure and reported as the source of structural failure. However, in many cases both the corrosion and the structural damage are a result of the damaging resonant vibrations.

Damping can be incorporated into a honeycomb structure by the use of an integrally damped laminated skin and an unconstrained layer damping treatment. Here, a laminated skin is defined as skin material (typically aluminum sheet) that is bonded together with a viscoelastic adhesive material as illustrated by Figure 5. In a laminated skin, the adhesive material is constrained by the skin material. Thus, when the laminated skin is subjected to cyclic bending, the skins constrain the adhesive and force it to deform in shear as shown by Figure 5. It is believed this shearing action of the adhesive material dissipates energy of the vibrations. Integral damping is provided by using an adhesive that has good damping properties in addition to adequate structural properties. Integrally damped laminated skins have proven quite effective in damping high frequency vibrations in honeycomb control surfaces. However, the integrally damped laminated skin is not effective in controlling low frequency vibrations.

One method found to be effective for controlling low frequency, high amplitude vibrations of honeycomb control surfaces is the use of an unconstrained layer damping treatment. Here, a damping material is applied to the surface of the structure so that whenever the structure is subjected to cyclic bending, the damping material will be subjected to extensional deformation as illustrated in Figure 6. The extensional action of the damping material apparently dissipates the energy of the vibrations.

Now with an understanding of how a honeycomb control surface can be affected by various flow realms and why damping is effective in increasing structural durability, consider how to design a honeycomb control surface to increase durability and reduce corrosion. The key to the design of a honeycomb control surface for improved durability and corrosion resistance is threefold. First, the design criteria must define the full range of environmental conditions that a honeycomb structure may be exposed to. This definition must consider not only maximum static load conditions, maneuver fatigue cycles, and thermal cycles, but must also consider fluctuating pressure loads that may be introduced by turbulent boundary layer phenomena or from jet engine exhaust blast. Once identified, these fluctuating pressure airloads must be fully considered in the design process to achieve an integrally damped structure that resists damaging resonant responses.

Second, the design must be of an improved configuration to reduce stress concentration effects, provide improved environmental sealing, and improve the bond strength between the skin and core. The use of laminated skins, superplastically formed assemblies, and microcell honeycomb to achieve these goals are discussed in the following section.

Finally, the "state-of-the-art" material and process systems need to be incorporated. This includes a complete surface treatment before bonding, a corrosion resistant honeycomb core, and an improved adhesive system. The following section discusses the integration of these three key areas into the design of a durable honeycomb control surface.

TECHNOLOGIES FOR IMPROVED HONEYCOMB DURABILITY

To illustrate the integration of the technologies for improved honeycomb durability, consider a comparison between the designs of the baseline and advanced F-111 outboard spoilers. As mentioned in the introductory section, the baseline spoiler uses design and fabrication practices that were an accepted norm in the early 1960's. The baseline spoiler has a history of service problems that include delamination of the skins from the core, corrosion, cracking of the spar and overhangs on the inboard and outboard ends of the spoiler, and cracking of the skin and core. The baseline spoiler often experiences service failures in less than 1,000 flight hours. Although the baseline spoiler used what is now considered poor surface preparation and obsolete adhesive bond processes, it should be noted that just a switch to the best available adhesive bonding technology would not solve all the service problems of the baseline spoiler. This is exemplified by the service problems experienced by honeycomb structures used on the latest generations of aircraft. As stated earlier, the design must use not only the latest adhesive bond technology, but must also incorporate damping and design configuration improvements to achieve a durable honeycomb structure. Now, consider how these technologies were integrated together into the design of the advanced spoiler.

The first area considered in the design of the advanced design spoiler was the incorporation of damping. Analysis and testing had shown that the baseline spoiler had several damaging resonances that could be excited in flight by turbulent boundary layer phenomena. Especially damaging was the observation that the body of the spoiler, and the inboard and outboard overhangs respond at the same frequency but 180° out of phase. The resonant response of the advanced spoiler was controlled through the use of integrally damped laminated skins. The damped skins not only controlled the resonant responses, but additionally detuned the response of the body and overhand of the spoiler resulting in a stress reduction of 2.5 times as compared to the baseline spoiler at the root of the overhang. Critical in the design of an integrally damped laminated skin is the choice of an adhesive that provides both adequate structural strength and damping in the temperature range of interest. For the advanced spoiler, AF-32, a 3M Company product, was found to be effective. A side benefit of the adhesively bonded laminated skins was the reduction of stress concentrations associated with chem-milled steps in the baseline skins. While the integrally damped laminated skins were effective in controlling high frequency resonant responses, they were found to be less effective at low frequencies that would be excited in flight. To control the low frequency responses, an unconstrained layer damping treatment was applied to a portion of the lower surface of the advanced spoiler. The material used was LD-400 which is a product of United McGill Corporation. This material was quite effective at reducing the low frequency resonant responses and resulted in a stress reduction of two for all the resonant modes.

The next area considered in the design of the advanced spoiler was improving the design configuration. The baseline spoiler uses several detail parts to close out the honeycomb core from the lower skin to the upper skin. This results in poor load paths in areas and difficulty in keeping the core sealed from the elements. The advanced design combined the details of the lower skin, the lower skin to supper skin edge closures on the inboard and outboard ends of the spoiler, and the spar into a single detail, the lower pan. This was done by using an advanced manufacturing process, superplastic forming. The lower pan was formed from Supral 220, a material similar to 2024, by Superform (a leader in superplastic forming). The lower pan provides a big improvement in load path from upper skin to lower skin and a large decrease in the amount of exposed bond line (or sealant) in terms of linear inches.

Another area of improved design configuration on the advanced design spoiler was the use of microcell honeycomb core to improve the bond strength between the skin and core. The increase in bond strength due to the "footprint" of the core is shown dramatically by the climbing drum peel specimens shown in Figure 7. Two important observations are made from Figure 7. First, notice the difference in peel strength between the baseline spoiler adhesive system (type I and baseline spoiler specimen) and the advanced design spoiler adhesive system (type II). The baseline spoiler adhesive system, without surface treatment before bonding, does not fully utilize the cohesive properties of the adhesive. This was shown dramatically by the type I specimen that experienced adhesive failure. Although this was not considered to be unusual for an adhesive system without surface treatment, an attempt to check the type I specimen results was made by taking a climbing drum peel specimen out of a section of a baseline spoiler. Unfortunately, because the piece of baseline spoiler had a core density almost half that of the type I specimen, the results of this test were not conclusive because the specimen failed in the core. However, the baseline spoiler specimen did indicate somewhat better results than the type I specimen. The second observation from Figure 7 is the increase in core to skin bond strength attributable to microcell honeycomb core. The major difference between the type II and type III specimens in the change from a 1/8 inch cell size in the type II specimen to a 1/16 inch cell size (microcell) in the type III specimen. The core density of the microcell core remains essentially the same because the foil thickness has been reduced. The improved adhesive "footprint" due to the smaller cell size improves the peel strength by over sixty percent.

The final area considered in the design of the advanced design spoiler was the use of the latest technology available for adhesive bonding. All bonded parts are produced from bare aluminum alloys. The parts are prepared for bonding by chemical cleaning using the standard aerospace industry sulfuric acid-sodium dichromate (FPL) etch. Following the cleaning, detail parts are phosphoric acid anodized and spray coated on the appropriate surfaces with a corrosion inhibiting adhesive primer that is heat cured. Finally, assemblies are laid up and autoclave cured with the appropriate adhesive. For corrosion protection, all bonded details and subassemblies are protected with a heat cured corrosion inhibiting adhesive primer; a CRIII coated honeycomb core is used; drilled holes and threaded inserts are protected by filling, draining, and coating cavities with a chromated polysulfide sealant; and all faying surfaces of the hinge and hinge shim assemblies are protected with a polysulfide sealant.

DURABILITY

Two advanced spoiler components have been tested. The first advanced component was subjected to a static and maneuver fatigue test. Results of the static and maneuver fatigue test are compared to previous testing of a baseline component. The second advanced component was subjected to an acoustical fatigue test. In addition, a baseline component was obtained and subjected to an acoustical fatigue test. This allows for a direct comparison between the acoustical durability of advanced and baseline spoiler.

STATIC AND MANEUVER FATIGUE TESTING

The advanced spoiler was subjected a three part static and maneuver fatigue test. First, the spoiler was static proof tested to 100 percent design limit load of both stowed and extended flight conditions. Second, the spoiler was maneuver fatigue tested for two lifetimes (8,000 flight hours) using a fatigue spectrum composed of stowed and extended flight loads. Finally, the spoiler was static ultimate tested to 150 percent design limit load for both stowed and extended flight conditions.

Tables 1 and 2 present reduced strain gage data at 100 percent design limit load and provide comparisons to the baseline spoiler at comparable loads. Drawing conclusions from the stress comparison is difficult due to internal geometric and design differences between the baseline and advanced design spoiler. However, examining the load on the links (dummy actuators) proves that similar loads were applied to the baseline and advanced spoilers.

The purpose of the static and maneuver fatigue tests was to qualify the advanced design spoiler for flight evaluation. No structural damage occurred during the proof test, the maneuver fatigue test, or the static ultimate test, so the advanced design spoiler was qualified for flight.

Acoustical Fatigue Testing

The spectrum and levels of aeroacoustical excitation on which the acoustical fatigue tests were based were derived from a combination wind tunnel tests, flight tests, NASA data, and Air Force data. During the acoustical fatigue test, the spoilers (both baseline and advanced) were excited with a simulated aeroacoustical fluctuating pressure for a typical operational mode. The spoilers were mounted (one at a time) in a progressive wave tube and excited by electro-pneumatic transducers. Because damaging aeroacoustical excitation occurs for only a small fraction of the baseline spoilers life, no accelerated testing of the baseline spoiler was required. Accelerated testing was required on the advanced spoiler to compress several aircraft lifetimes into a few days of laboratory testing. The "state-of-the-art" electro-pneumatic source of sound in the laboratory could not generate enough acoustic energy to simulate the most severe excitations that a spoiler encounters in service. In fact, no facilities in the US can provide the power level needed. However, because of the low duty cycle of the excitation of the spoiler on a parent F-111, it was possible to compress an airplane lifetime into a few days of testing, using accumulated exposure to simulate the highest excitation levels.

The baseline spoiler exhibited several levels of failure at different times during testing. First, after 200 equivalent flight hours, the adhesive along the spar failed cohesively. Rivets in the spar held the part together; but in actual service, water could penetrate the failed adhesive and initiate corrosion of the honeycomb core. After 1,360 equivalent flight hours of excitation, the inboard overhang lost its stiffness. That loss in stiffness would constitute failure in service, because static airloads and flutter of the overhang would quickly break the part once stiffness is lost. In fact, with static airloads superimposed upon the aeroacoustical excitation as it is in flight, this failure would probably have occurred in less than 1,360 flight hours. The outboard overhang cracked at an equivalent of 1,760 hours of flight, while the inboard overhang cracked after 1,900 hours. Most breaks of the overhang occur in service in the range of 1,360 to 1,900 hours.

An advanced spoiler was subjected to acoustical fatigue testing. The advanced spoiler was tested for the equivalent of 98,000 flight hours before a crack developed along the edge of the outboard hinge fitting. This takes into account the "accelerated" testing which occurred on the advanced spoiler. Comparing the acoustic fatigue life of the advanced design spoiler to that of the baseline shows a tremendous improvement in the durability of the advanced spoiler.

Additional acoustic fatigue testing of an advanced component was performed using the static and maneuver fatigue specimen. This additional acoustic testing was performed to demonstrate the repairability of the advanced design spoiler. The specimen was intentionally damaged and repaired with a plate and core patch. The durability of the repaired advanced spoiler was demonstrated by testing for over 2,600 equivalent flight hours (over 1/2 an airplane lifetime), at which time the test was suspended due to a crack in the skin outside of the repair area.

The static and maneuver fatigue test of the advanced spoiler verified the structural integrity of the advanced spoiler and thus qualified it for flight. Additionally, the acoustic fatigue test proved that there is a tremendous increase in acoustic durability for the advanced spoiler (98,000 flight hours) over the baseline spoiler (1,360 flight hours).

IN-SERVICE EVALUATION

Between February and March 1986, six advanced design F-111 outboard spoilers were installed on TAC aircraft at the 27TFW, Cannon AFB, NM, for a six-month service evaluation. The spoilers were installed on F-111 aircraft performing routine mission categories; for example: low level, low level bomb, and instrument missions. During the six-month period, maintenance personal visually inspected the spoilers after each mission for defects: inspection results were unremarkable. Comprehensive inspections took place after the third and sixth month of spoiler flight time.

After three months of service, the spoilers were removed from the aircraft and subjected to a Bondascope inspection. The Bondascope is an ultrasonic inspection device suitable for field use. No anomalies in the honeycomb structure of the spoilers were located. In addition, the removed spoilers were visually inspected for defects. There was evidence of interference with machine screws on the vane guide near the inboard end of the spoiler. It was determined that incorrect machine screws were used on the vane guide and the problem was corrected. The spoilers were then reinstalled on the aircraft for further flight testing.

The advanced spoilers completed their six-month service evaluation in September 1986. The outboard spoilers had accumulated over 950 flight hours when they were removed from their respective aircraft and shipped to LTV for inspection. There were three methods used to inspect the six spoilers: visual, Bondascope, and real time x-ray. The real time x-ray is a filmless radiographic technique that employs an extremely small point source focal spot (10-100 microns) which allows the honeycomb structure to be magnified several times without any geometric distortion. The findings of the Bondascope and real time x-ray inspections were excellent; no areas of disbond were found on any of the six service evaluation spoilers. The visual inspection revealed a small discrepancy on one of the spoilers: the forward outboard end of the spoiler where the lower skin had peeled back due to interference with the wing skin. The part was repaired by re-bonding the peel area with 3M EC-1614 adhesive. The spoiler trim was corrected to prevent further interference with the wing skin. After finishing the inspections, the six spoilers were returned to the 27TFW at Cannon AFB for reinstallation.

WEIGHT

Table 3 provides comparison between the weights of the advanced spoiler and the baseline spoiler. The advanced spoiler has a small weight increase over the baseline spoiler (10.9 percent). This increase is due to improvements in the design of the skin, core, and spar, and to the addition of an unconstrained layer of damping. The skin is heavier because it incorporates spar and edge closure details into one-piece superplastically formed pans. The use of laminated skins vs. chemilled skins also accounts for a weight increase due to the addition of adhesives. The increased weight of the skin is offset by the advantages in damping and reduction in stress concentrations. Also, the baseline core is composed of three different densities, while the advanced core uses a one-piece core of single density which weighs more than the baseline core. The decreased weight of the advanced spar is accounted for by the incorporation of much of the baseline spar into the SPF skin pans. A layer of unconstrained damping, LD400, was added to the underside of the spoiler for low frequency vibration control. The LD400 is responsible for approximately one-half of the weight increase of the advanced design spoiler. It should be noted that while there is a slight weight increase, the weight comparison is make between a design that fails in service in a short time (baseline - 1,360 flight hours) to a design that has demonstrated by testing a tremendous durability improvement (advanced - 98,000 flight hours). If the baseline was "beefed-up" by increasing its weight by ten percent, it still would not match the durability exhibited by the advanced spoiler.

COST

Two comparisons between the costs of the baseline and advanced spoilers were made. The first comparison considers the difference in production costs (or the cost of the "new" technology). The second comparison presents projected cost savings (life cycle cost) for a possible retrofit effort where baseline spoilers are replaced, as they fail, with advanced spoilers.

PRODUCTION COST

This engineering cost analysis presents a comparison of production cost between the baseline and advanced spoiler. The following assumptions made in the analysis: First, a manufacturing technology estimate was made of the production procedures for fabrication of the advanced spoiler, and procurement costs for spare spoilers manufactured by General Dynamics were obtained from Sacramento Air Logistics Center. Then industrial engineering standard hours were applied to these production procedures. The advanced spoiler's cost uses 1985 labor rates and factors. Both the baseline and the advanced spoiler prices were based on a lot size of fifteen, as requested by Sacramento ALC. The advanced spoiler production cost utilized a ninety-eight percent material learning curve slope and an eighty-five percent labor learning curve slope. This analysis indicates that the advanced spoiler costs 4.8 percent less than the baseline. This decrease is not surprising when the baseline spoiler's makeup is compared to the advanced spoiler's makeup. The chem milling procedure used in the baseline skins is labor intensive and is as costly as the adhesively bonded laminated skins. Because parts of the spar and close-out structures are incorporated in the superplastically formed pans, the advanced spoiler has fewer detail parts which decreases costs. Also, rivets are not used in the advanced spoiler which eliminates the cost of rivet installation. From a production cost comparison, it is evident that the advanced spoiler is cost effective.

PROJECTED LIFE CYCLE COST

The cost benefits to be derived from the durability improvements of the advanced spoiler consider three cost factors. These factors are the production costs (both recurring and nonrecurring costs), maintenance costs (in-service), and replacement/repair costs (component removal and reinstallation). These three cost inputs are used to compare the life cycle cost of the baseline and the advanced spoiler.

The following guidelines were used in the life cycle cost analysis. There are 383 aircraft in the F-111 fleet which accumulate approximately 103,788 flight hours per year. The mean time between replacement for the baseline spoiler is approximately 1,000 hours. The spoiler hinges, a high cost item, are salvaged ninety percent of the time. The fleet attrition was assumed to be two percent per year. A production cost analysis gave costs for the advanced spoiler, with and without new hinges. It was conservatively assumed that an advanced spoiler would last 5,000 flight hours before needing replacement.

The analysis projected a total cost savings of $5.0 million for the advanced spoiler over a ten year period. Figure 8 illustrates the ten year cost trends of the baseline and advanced spoilers. The sharp break in the advanced spoiler cost line occurs when the fleet is completely retrofitted with the more durable advanced design, near the three-year point.

CONCLUSIONS

The Advanced Design F-111 Outboard Spoiler demonstrates a generic technological advancement that is applicable to the entire Air Force fleet. The major generic benefit is the demonstration of a durable honeycomb control surface design that incorporates integral damping and improved design concepts. The advanced spoiler has demonstrated the utility of those concepts as a design tool, rather than an after the fact fix.

The advanced spoiler illustrates that future designs of honeycomb structure must consider the total design environment to demonstrate adequate life. Design criteria must consider not only static load conditions and maneuver fatigue cycles, but must also include the aeroacoustic environment (or fluctuating pressure loads). After the environment has been completely defined, the design can include those elements necessary for adequate component life, such as integral damping.

Finally, the technology employed on the advanced flap is applicable to any honeycomb structure. This technology could be incorporated on numerous fleet components that are experiencing premature failures. Additionally, this technology could be applied to the next generation fighter/attach aircraft or to honeycomb structure utilized on large aircraft.

Demonstration of a Durable Honeycomb Control Surface

1Lt R S Fredell AFWAL/FIBAA

S N Vacca LTV Aerospace

AF Contract F33615-81-C-3219

LTV Aerospace and Defense
Vought Aero Products Division

SECONDARY STRUCTURE PROBLEMS

- Thin gage skin and rib structure:
  - Skin cracking
  - Rib cracking
  - Loose or damaged fasteners

- Honeycomb structure:
  - Delamination
  - Corrosion
  - Skin cracking
  - Core damage

- Premature failures are affecting both combat readiness and life-cycle cost

AP4-3207-2

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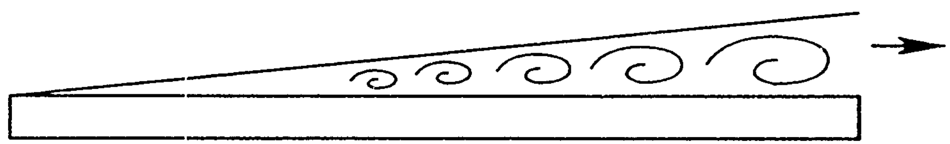
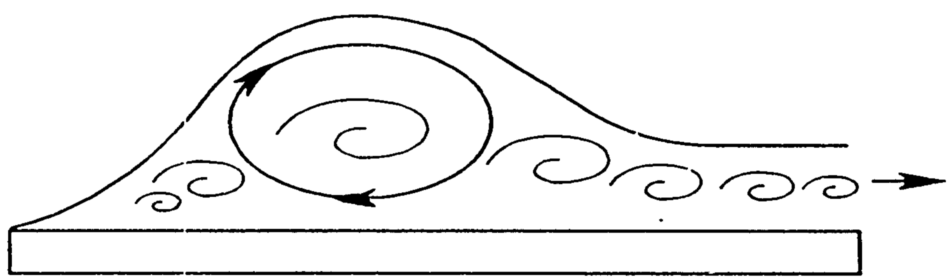
AEROACOUSTICAL FLUCTUATING PRESSURES CAUSE FATIGUE



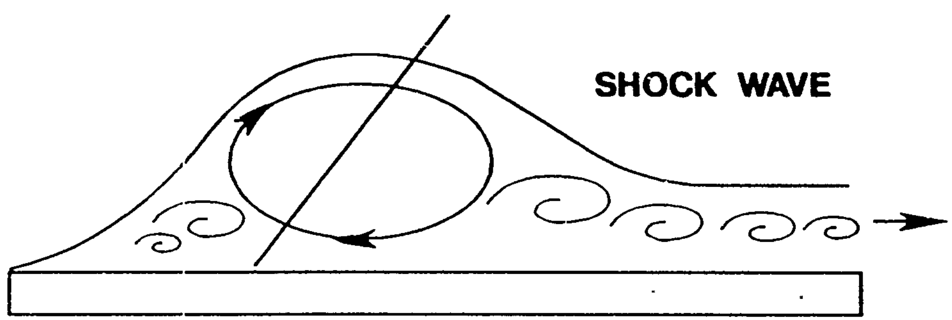
P = 0.006Q

ATTACHED TURBULENT BOUNDARY LAYER



P = 0.06Q

LOCALLY SEPARATED TURBULENT BOUNDARY LAYER



P = 0.1 TO 0.3Q
183 dB OVERALL
BROADBAND RANDOM
(3 PSI RMS)

LOCALLY SEPARATED TURBULENT BOUNDARY LAYER INTERACTING WITH A SHOCK WAVE

GP5-1906-4

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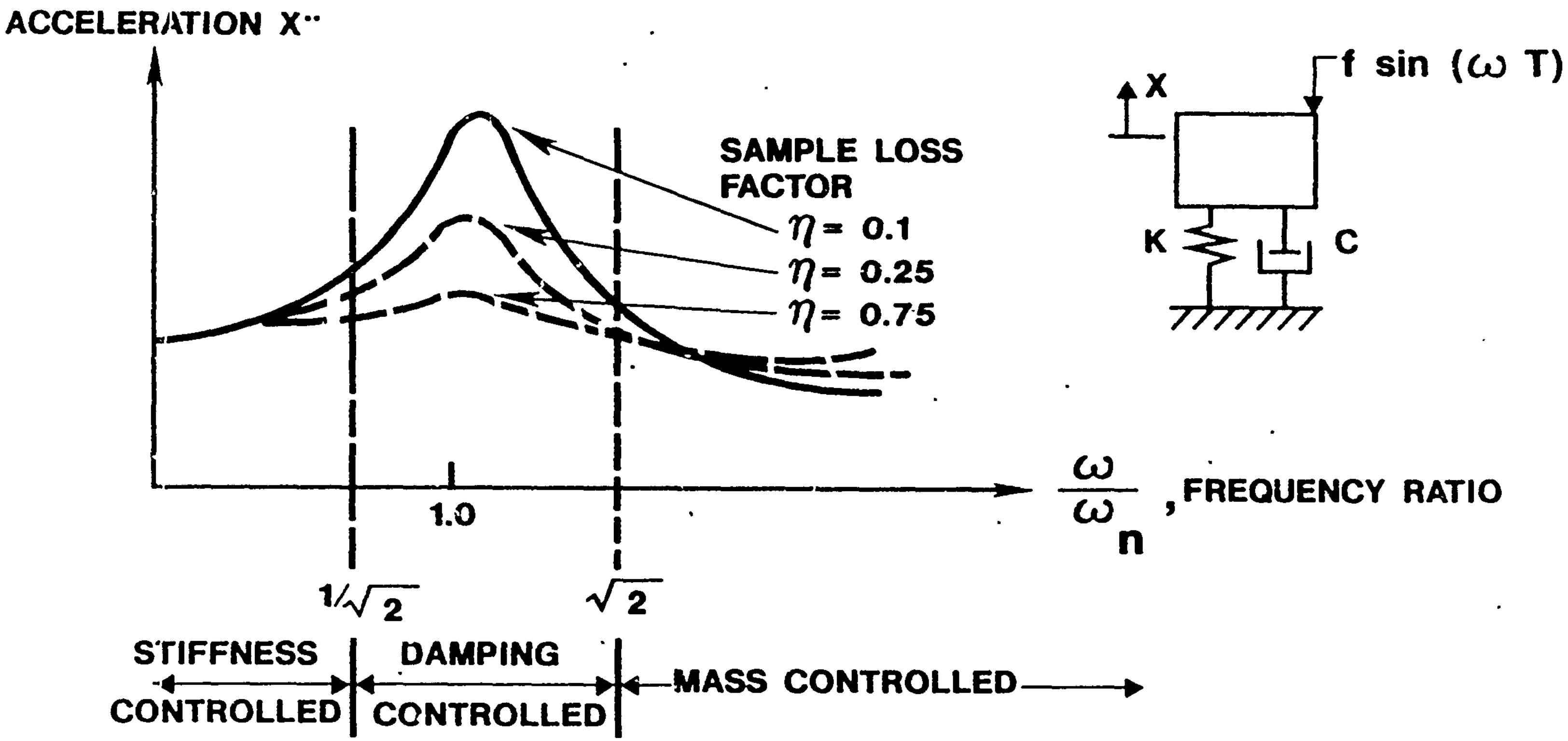
INFLUENCE OF DAMPING ON RESONANT RESPONSE



at $\frac{\omega}{\omega_n} = 1$, $\ddot{X} \propto \frac{1}{\eta}$, FATIGUE LIFE $\propto e^{\eta}$

- ☐ Identical phenomenon in continuous systems

GP5-1906-8

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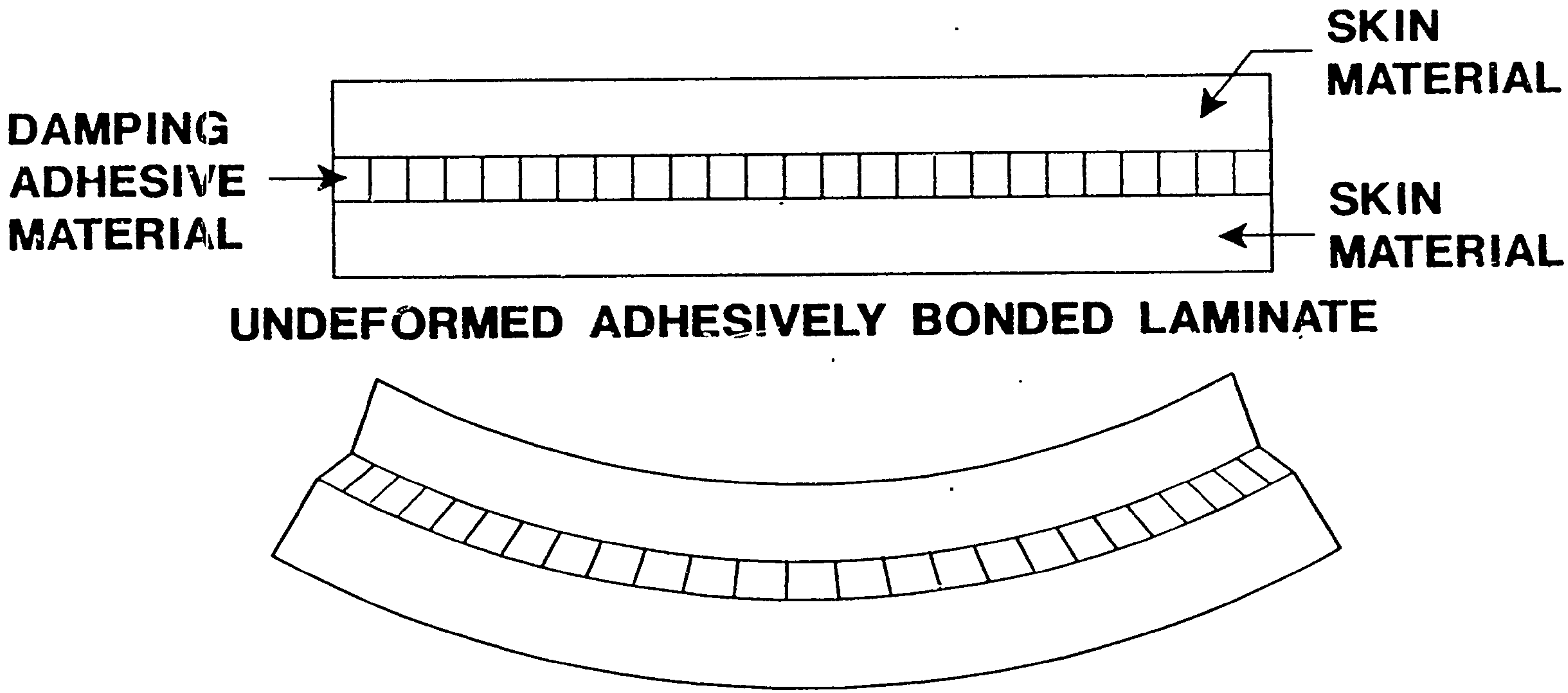
INTEGRALLY DAMPED LAMINATED SKIN



UNDEFORMED ADHESIVELY BONDED LAMINATE

DEFORMED ADHESIVELY BONDED LAMINATE

- Shearing action of adhesive material dissipates vibrational energy

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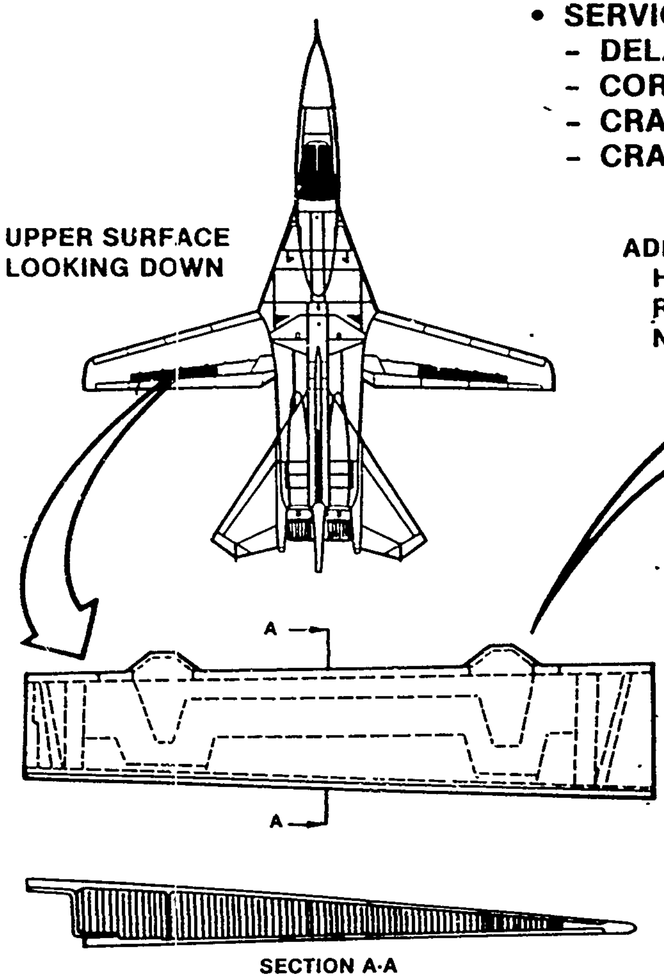
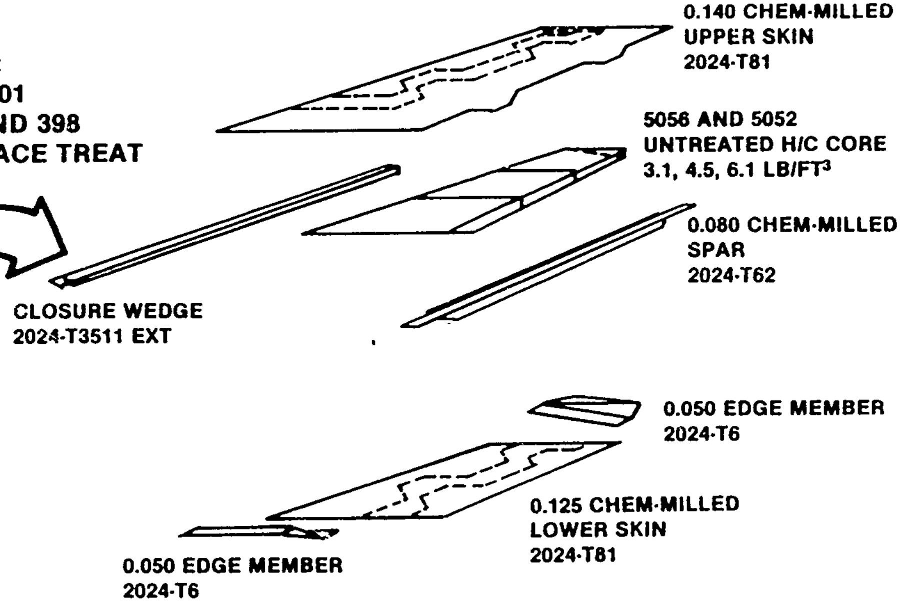
GP5-1906-5

BASELINE F-111A OUTBOARD SPOILER

- SERVICE PROBLEMS
  - DELAMINATION
  - CORROSION
  - CRACKED SPAR AND OVERHANG
  - CRACKED SKIN

ADHESIVES:
HYSOL 9601
RELIABOND 398
NO SURFACE TREAT





AP5-2604-11

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BASELINE F-111 OUTBOARD SPOILER

ACOUSTICAL FATIGUE FAILURE - DEBONDED AND CRACKED SKIN

AP4-2313-1

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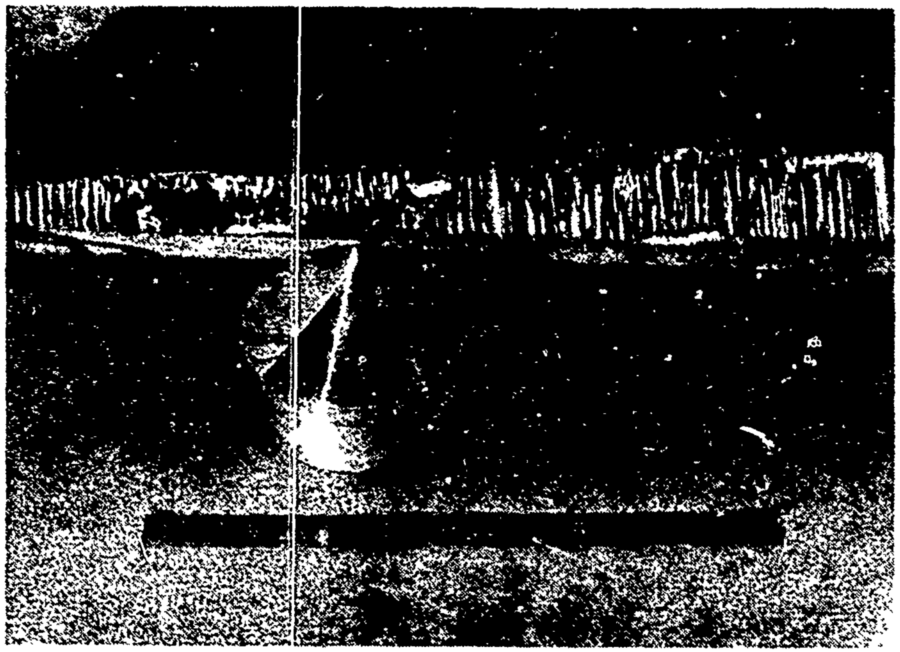
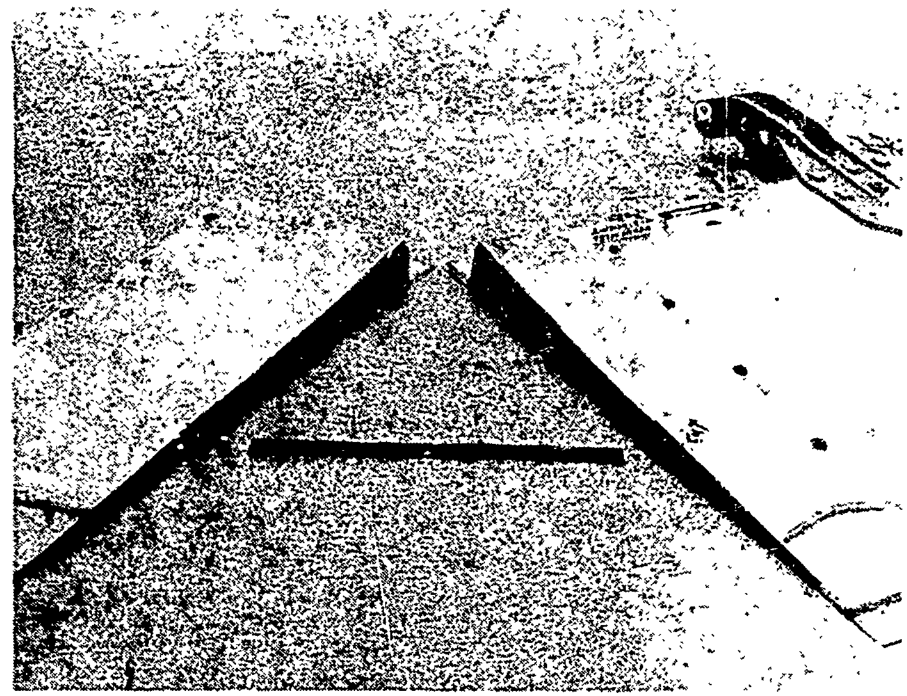
BASELINE SPOILER FAILURES



CRACKED SKIN AND DAMAGED CORE



CRACKED SKIN AND DAMAGED CORE

AP5-2604-13

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F-111 OUTBOARD SPOILER REPAIRS/REPLACEMENTS

- MTBF average for last 2 years

  RH-880 flight hours LH-1166 flight hours

- Majority of spoilers are remanufactured
  - Use old hinges
  - Replace skins and cores
  - Replace spar ~25% of the time
- Based on 1980 ECOs, 50% of the spoilers are replaced in 2 years

AP5-2604-49

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F-111 OUTBOARD SPOILER HISTORY

CHANGE ORDER NUMBER	DATE	PART OF DESIGN AFFECTED	DESCRIPTION OF CHANGE
ECP-1994	April 1970	Spoiler, Linkage Support Fitting	Shortened the Inboard End of the Spoiler, Redesigned the Brackets and Rear Spar. (Incorporated on TAC No. 200 and Up, Reworked on Pre-ECP-1994 Aircraft after 800 Flight Hours).
ECO80C543 ECO80C544	Aug 1980	Inboard and Outboard Closure Skin	Thickness of Skin Was Changed from 0.020 in. to 0.050 in., and Bend Radius Was Adjusted.
ECO80C0542	Aug 1980	Upper Skin	Chem Mill Line on Upper Skin Moved Away from Tapering Honeycomb.

AP5-2604-34

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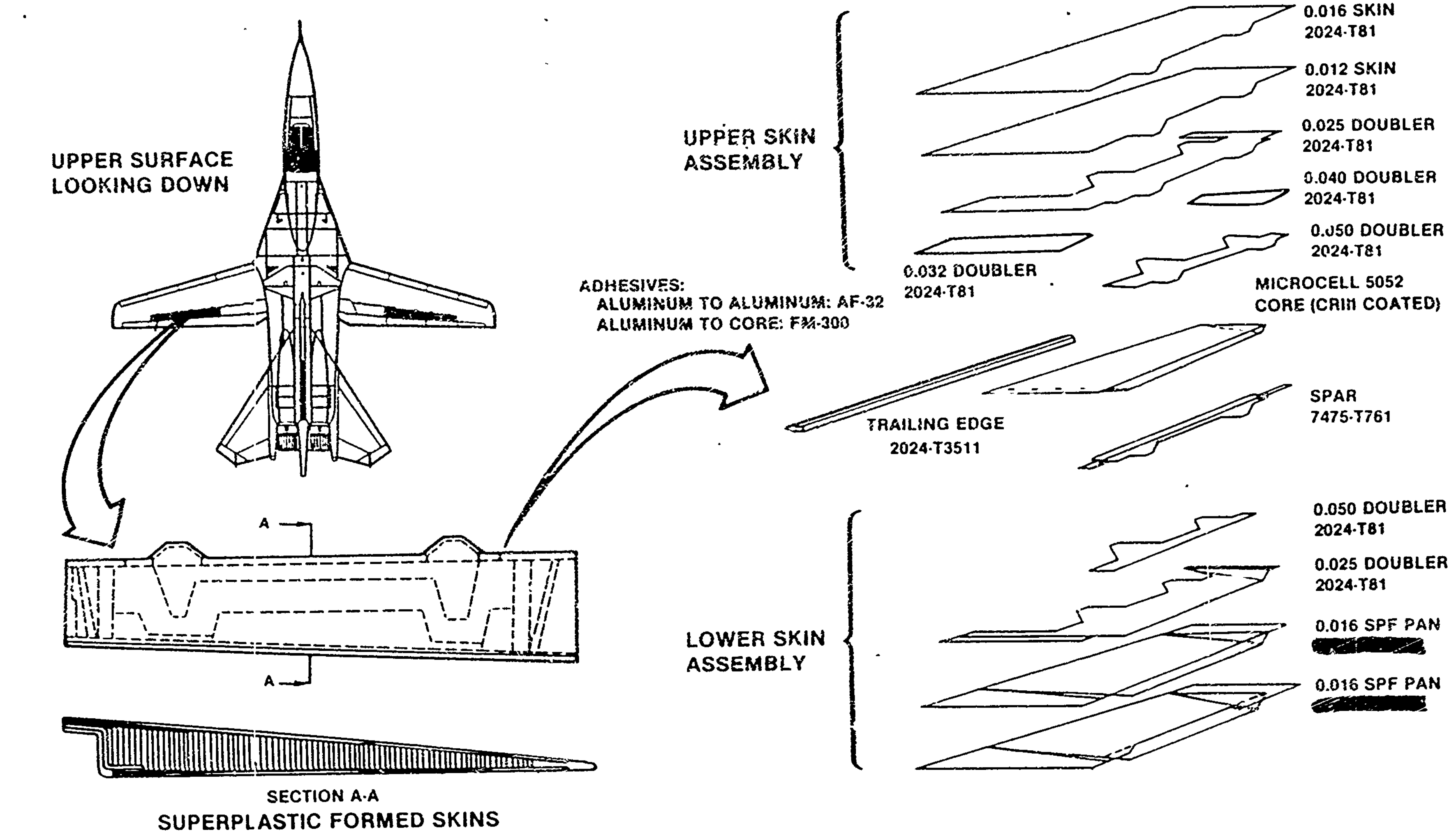
ADVANCED DESIGN F-111A OUTBOARD SPOILER



AP4-2043-1A

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TECHNOLOGIES FOR IMPROVED HONEYCOMB DURABILITY

- Constrained Layer Damping
  - Laminated skins with AF-32 bond line
- Improved Design Configuration
  - Nested one-piece SPF lower skin
  - Microcell core
  - Laminated skin
- Improved Adhesive System
  - Sulfuric acid-sodium dichromate (FPL) etch
  - Phosphoric acid anodize
  - Adhesive primer
  - Aluminum to aluminum: AF-32
  - Aluminum to core: FM-300

APS 2604-15

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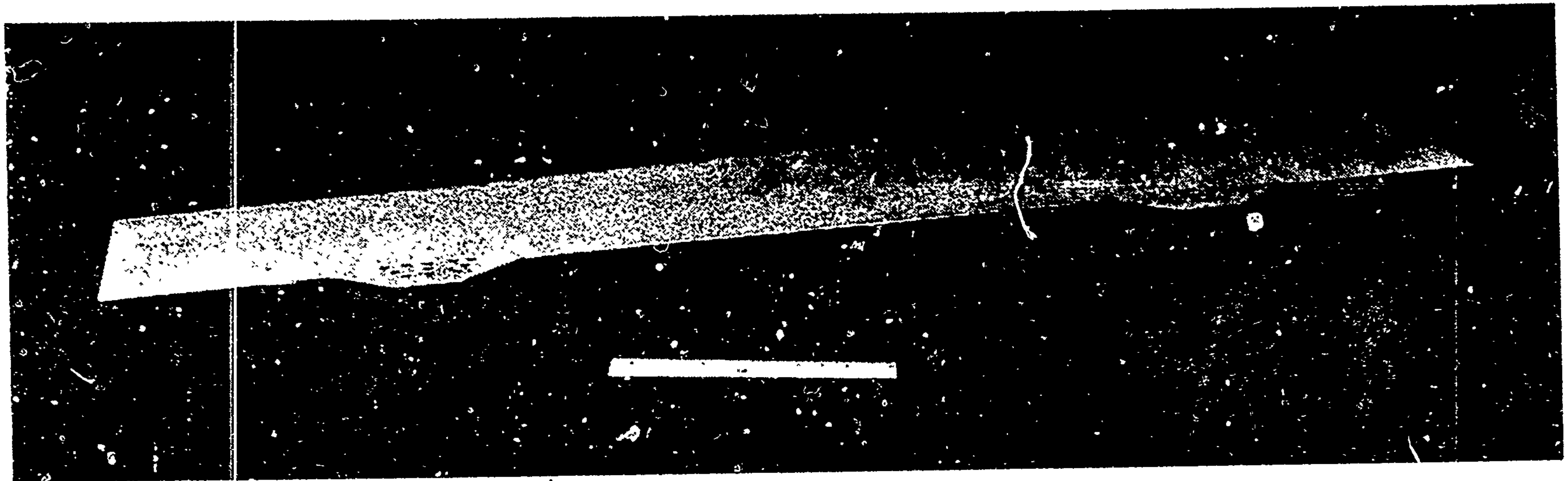
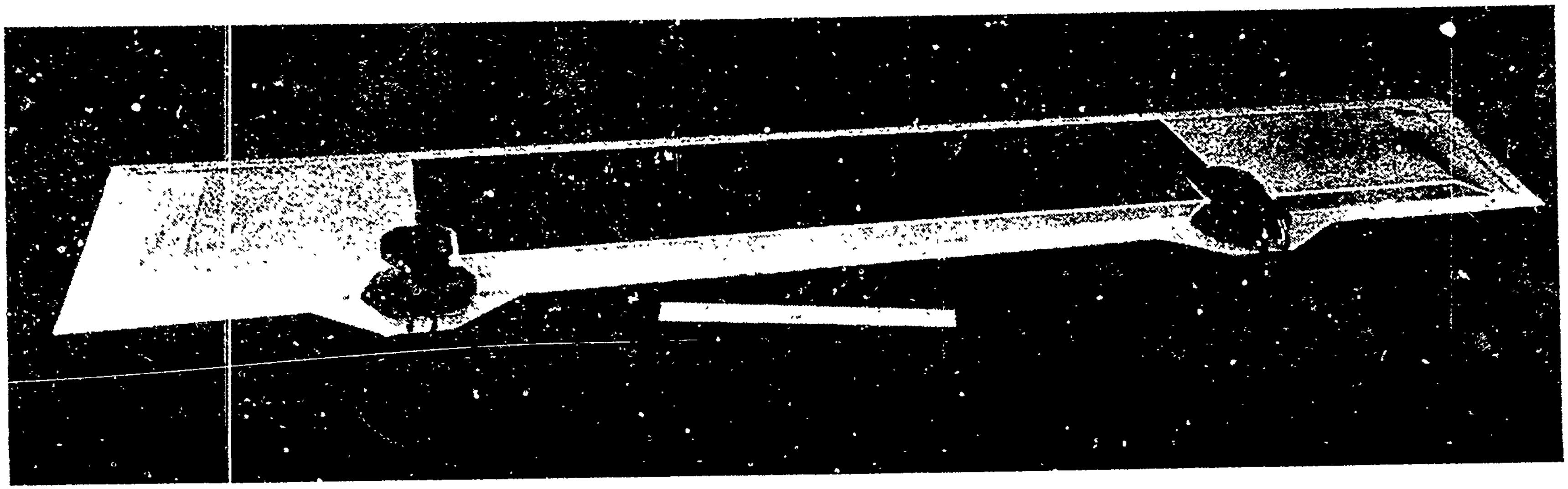
ADVANCED DESIGN F-111 OUTBOARD SPOILER



UPPER SURFACE



LOWER SURFACE

AP5-2606-7

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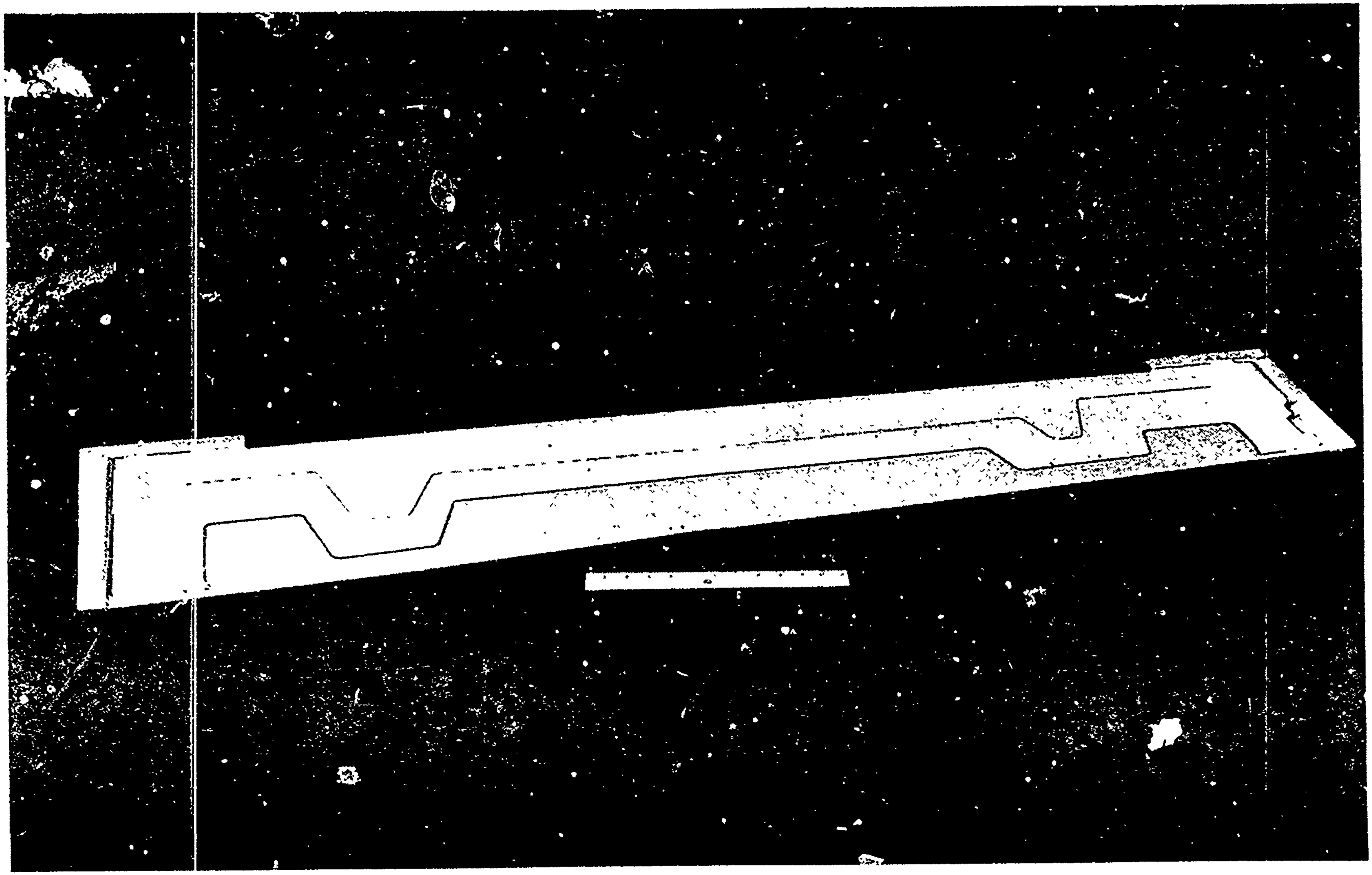
COMPLETED LOWER SKIN SUBASSEMBLY



AP5-2606-4

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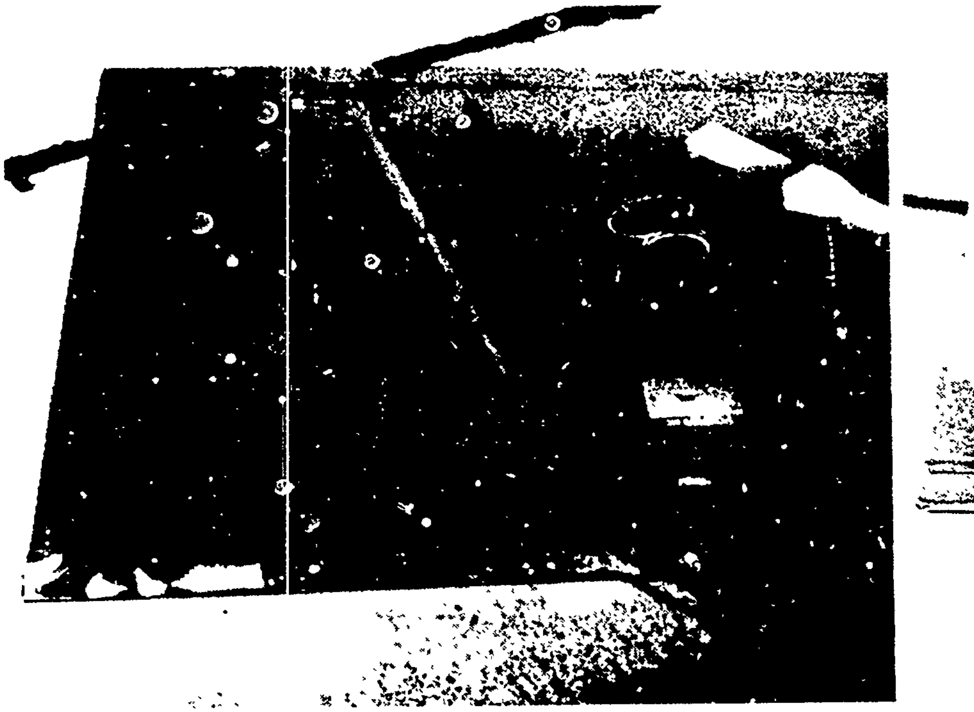
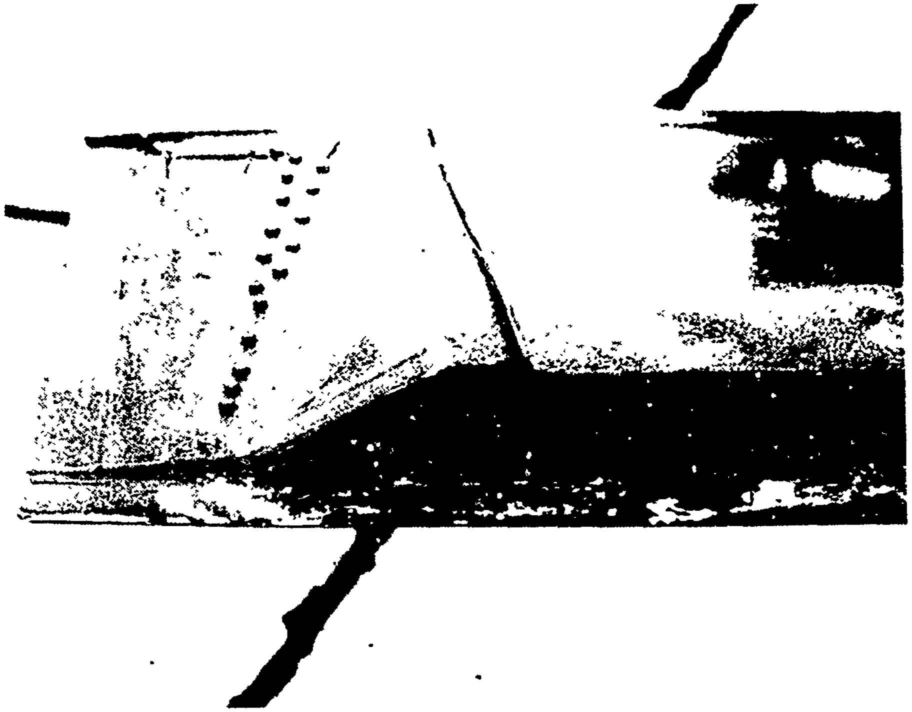
F-111 SPOILER BASELINE END CLOSE-OUTS

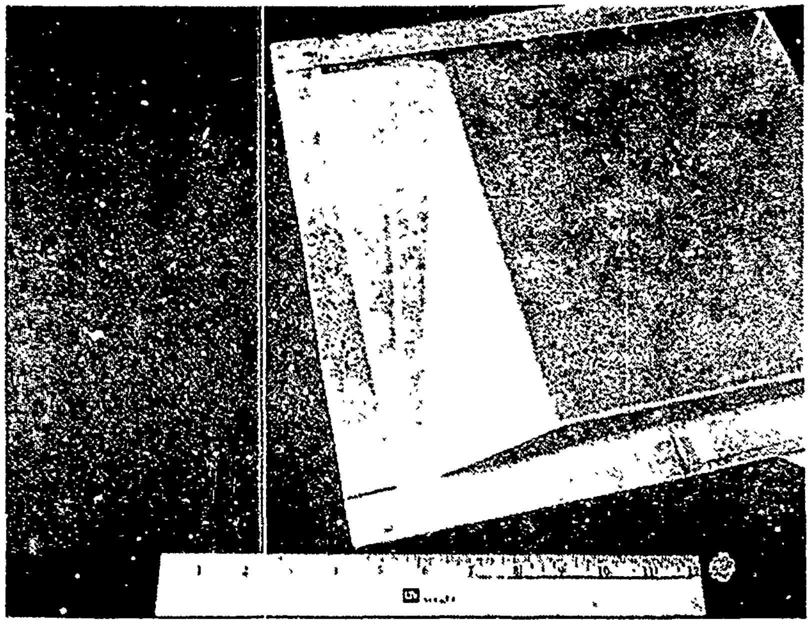




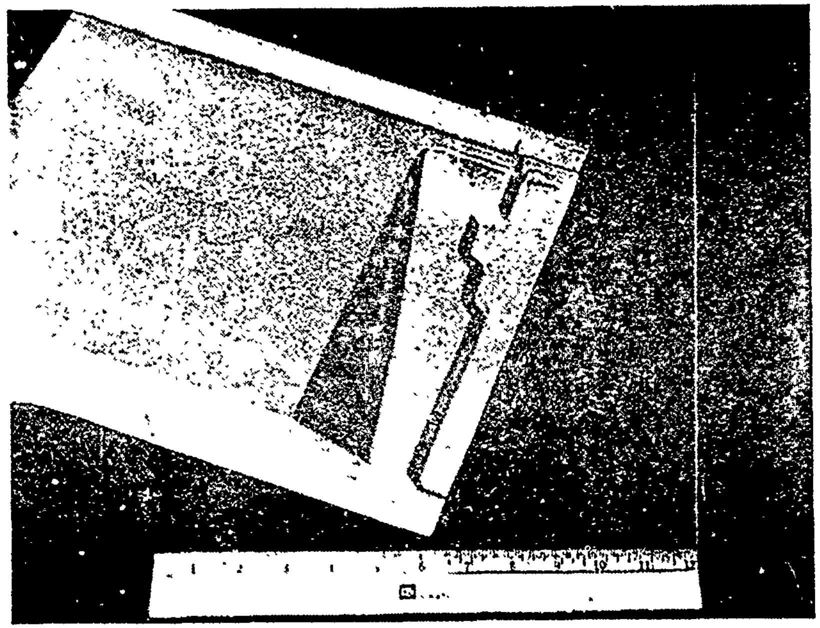
AV6-3272-4

ADVANCED DESIGN F-111 OUTBOARD SPOILER

CLOSE-UP OF OVERHANGS



INBOARD



OUTBOARD

AP5-2606-8

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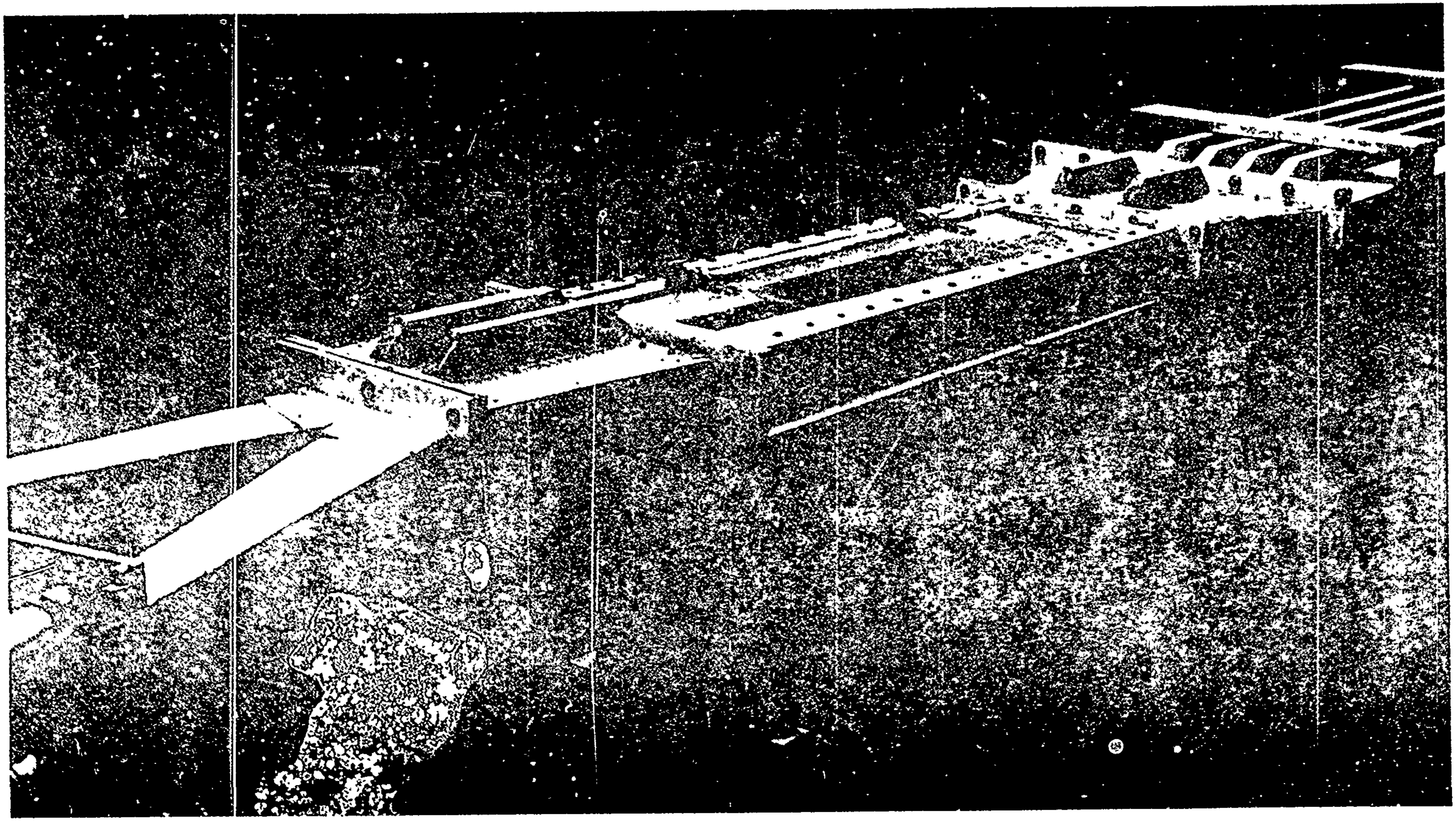
ADVANCED F-111 SPOILER ACOUSTICAL FATIGUE TEST (TEST FIXTURE)



AP5-2604-36

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FAILURE SEQUENCE OF ACOUSTIC TEST SPOILERS

SPOILER	FAILURE	SERVICE HOURS
Baseline	• Failure of spar adhesive — allows moisture penetration	200 hr
	• Loss of stiffness of inboard overhang	1360 hr
	• Inboard overhang cracked — skin of body cracked — core torn	1760 hr
	• Outboard overhang cracked	1900 hr
Advanced	• Lower skin crack at outboard hinge fitting	98,000 hr

AV6-4020-14

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CORROSION

- WR-ALC/MMEMC (Air Force Corrosion Office) has completed a 500-hour salt fog exposure test on a section of the acoustic fatigue spoiler
  - No moisture intrusion
  - No visual signs of corrosion
  - Ultrasonic inspection is in progress
- Another 500 hours of salt fog exposure is planned

AV6-4020-16

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IN-SERVICE EVALUATION

- 6 spoilers in-service at Cannon AFB for six months

S/N	Flight Hr
1	178.5
2	178.9
3	194.6
4	130.0
5	148.3
6	121.6
	951.9

- Bondascope Inspection at three months
- Real Time X-Ray and Bondascope Inspection at six months

AV6-4020-15

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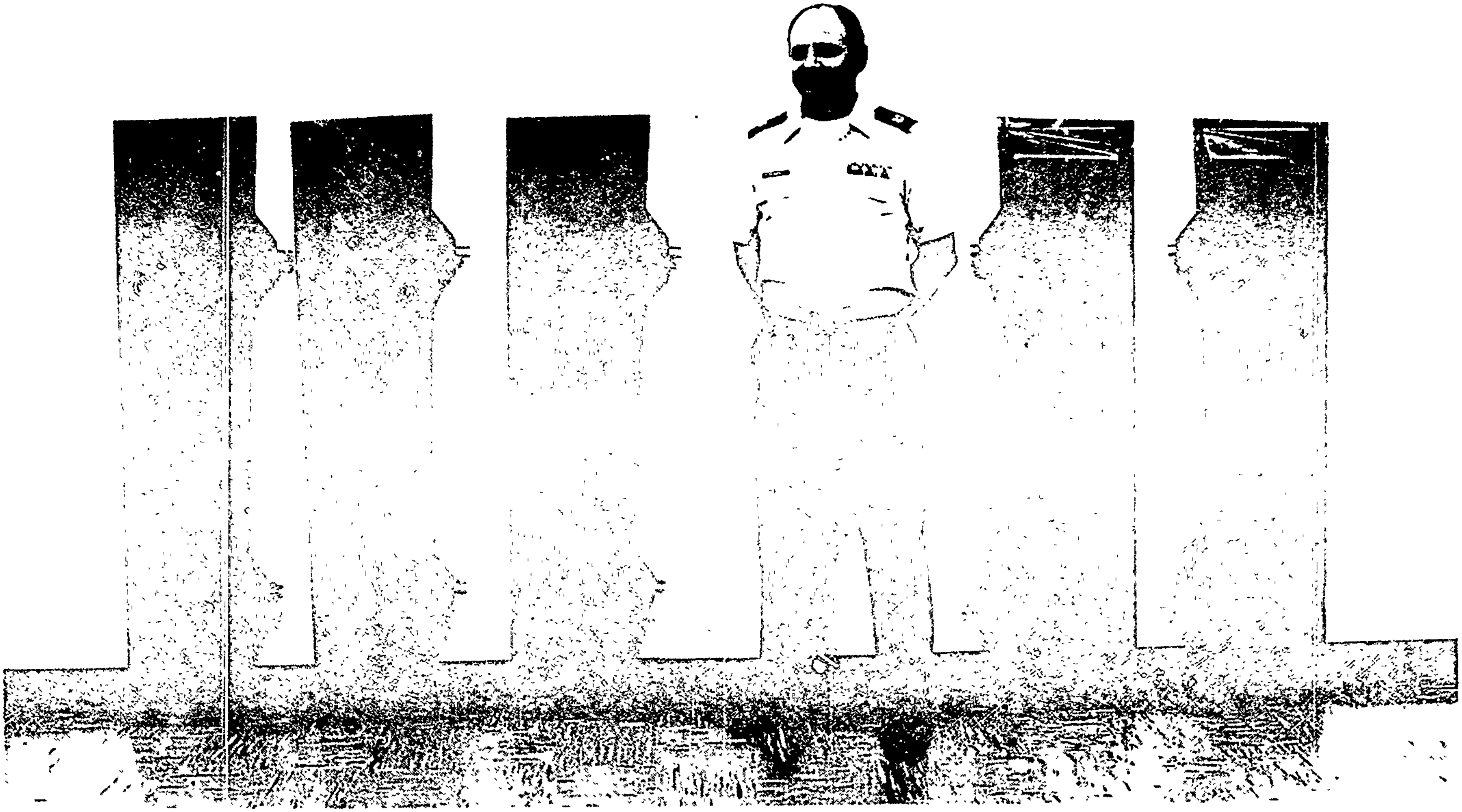
ADVANCED SPOILERS SIX MONTH INSPECTION



AV6-4020-7

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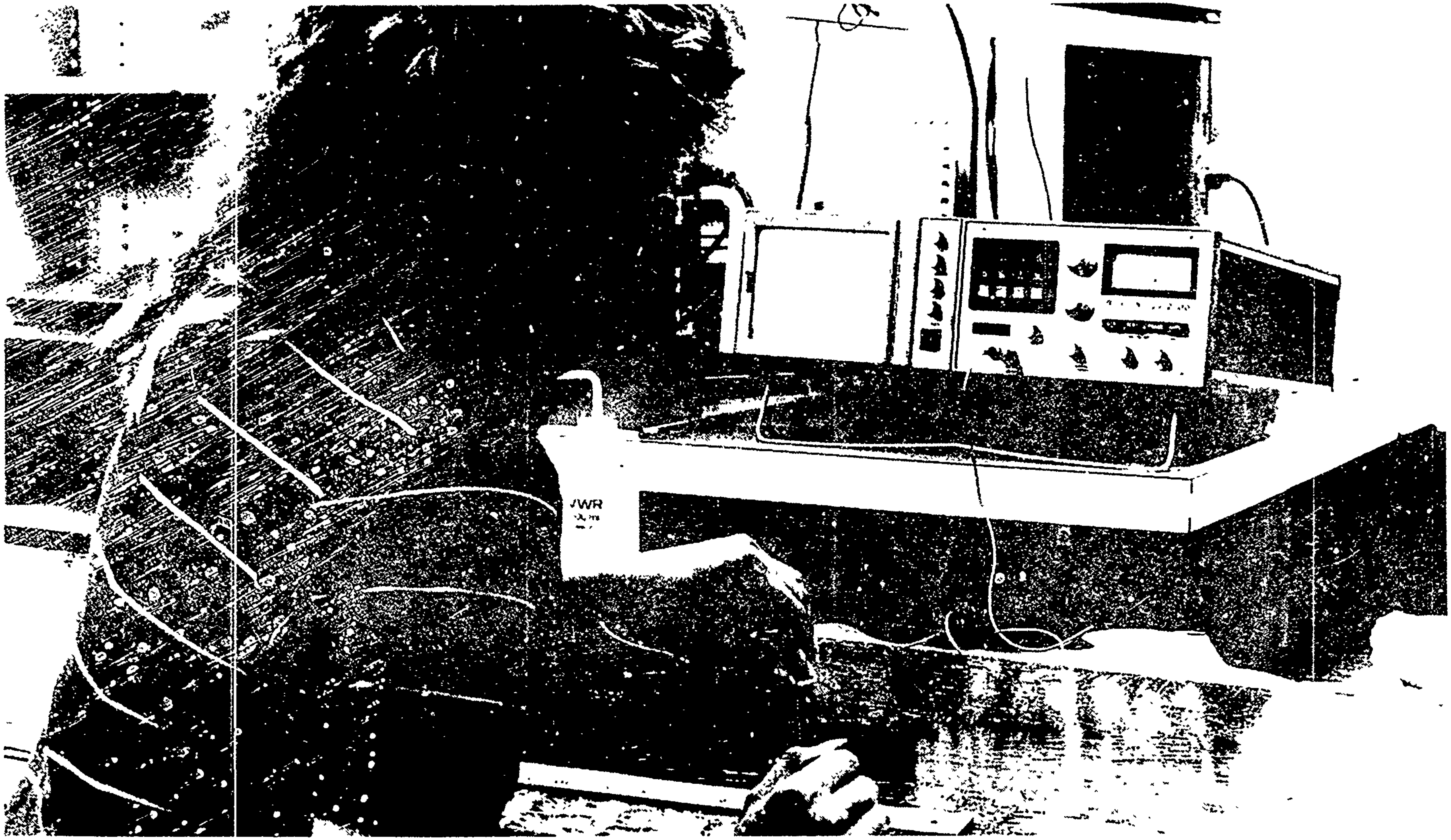
ADVANCED SPOILER BONDASCOPE INSPECTION



AV6-4020-6

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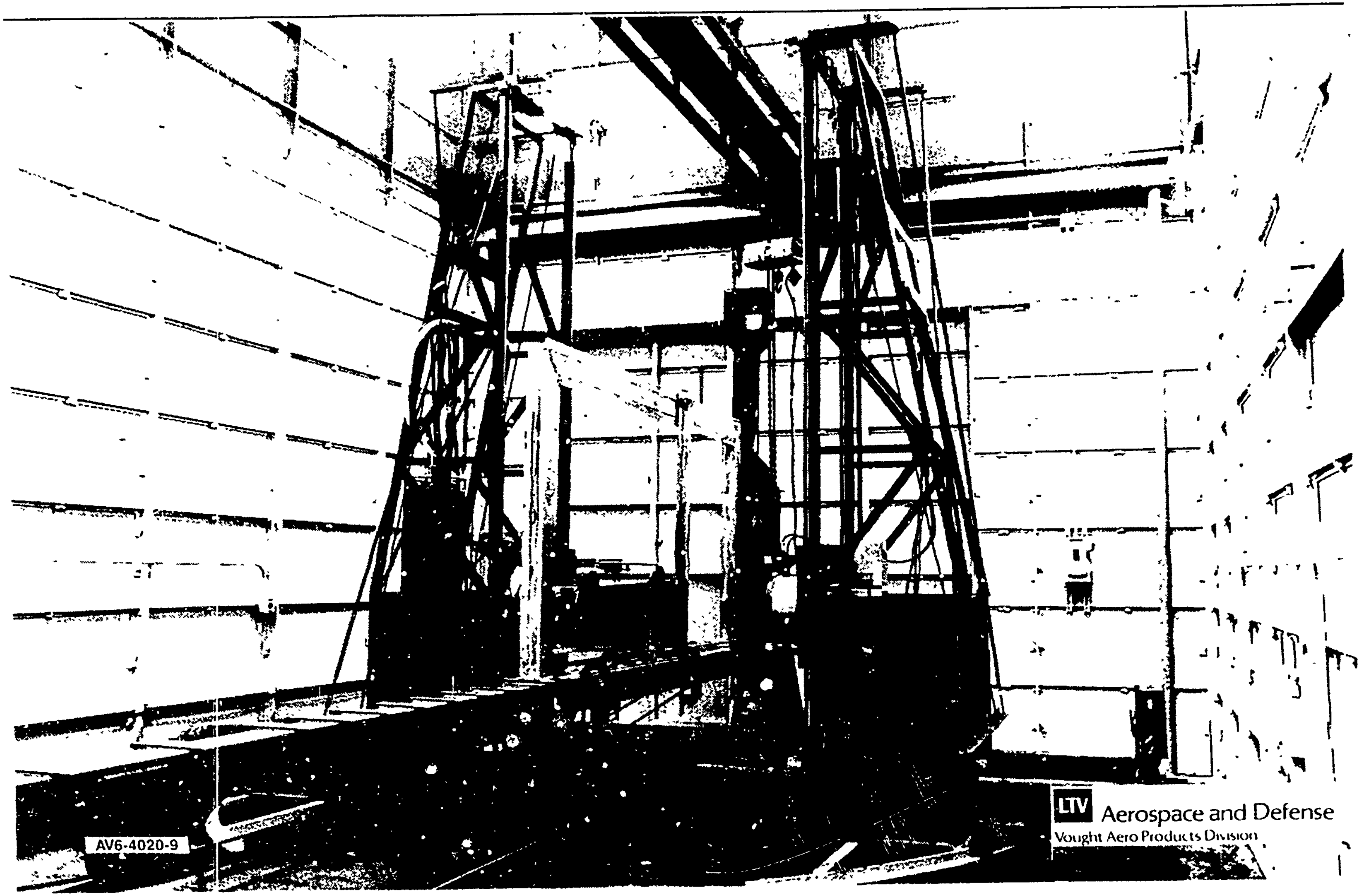
REAL TIME MICRO-FOCUS X-RAY FACILITY



TECHNICAL SUMMARY
F-111 OUTBOARD SPOILER

	BASELINE	ADVANCED
• Life	1360 hr	98,000 hr
• Cost (Retrofit - 1000 MTBF)	$12.5M (10 yr)	$7.4M (10 yr)
– 4.8% Production Cost Decrease		
• Weight	17.4 lb	19.3 lb

The advanced design spoiler demonstrates a generic technological advancement - applicable to the entire Air Force fleet

AP5-2604-51

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PROGRAM CONCLUSIONS

- Design goals for advance components have been met or exceeded
  - Major durability increase
  - Minimal weight increase
  - Minimal production cost increase
- The advanced components demonstrate a generic technology for durability improvement
- This technology is applicable to other aircraft in the Air Force fleet
- DoD R&M emphasis requires this technology on the next generation aircraft

THIRD SESSION
NDI CORROSION CONTROL

Chairman

Warren Johnson

AFWAL/MLS

Nondestructive Inspection Engineering And Nondestructive Inspection Reliability Studies

By

Ward D. Rummel
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Martin Marietta Corporation
Denver Aerospace
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- ABSTRACT -

This paper describes some applications of nondestructive inspection technology and nondestructive inspection engineering technology as applied to structural integrity concepts, design engineering and systems management. Important concepts discussed are: (1) the need for nondestructive inspection (NDI) technology awareness, with respect to applications, capabilities and applications boundary conditions; (2) the use of (NDI) technology in providing confidence in structural integrity and fitness for purpose; (3) that NDI measurements are not absolute or single valued; (4) The methodology of NDI application and process analysis and validation techniques; (5) the boundary conditions in NDI applications; and (6) the rapidly changing nature of the NDI science and engineering base.

The paper concludes that NDI is an integral and essential technology to the design, production, operation and management of modern engineering structure and systems.

NONDESTRUCTIVE INSPECTION ENGINEERING

INTRODUCTION:

Nondestructive inspection (NDI) technology is becoming increasingly important to the development and management of engineering components, structures and systems. The objectives of this paper are: (1) to communicate a need for awareness of NDI technology, its capabilities, applications and limitations; (2) to impart a recognition that NDI is a distributed technology and that no single NDI technique is capable of providing either the necessary data that may be required to characterize all engineering properties of concern or of solely providing confidence in fitness for purpose; (3) to establish recognition that the results of NDI are not absolute or single valued; (4) to provide familiarization with NDI application, measurement, analysis and validation methodologies; (5) to identify some of the uncertainties and boundary conditions in analysis and application; and (6) to identify that the science and engineering base of NDI technology is dynamic and rapidly changing.

NDI AWARENESS

NDI is a multi-disciplined, broad based technology, that is applied to indirectly characterize and/or measure a property of interest, without affecting the fitness of the test object, structure, or system for service. Familiar nondestructive inspection methods include: visual inspection; X-radiography; liquid penetrant inspection ("Zyglo"); magnetic particle inspection ("Magnaflux"); ultrasonic inspection; and leak testing. Less familiar methods include: eddy current inspection, neutron radiography; thermography; holography; acoustic emission analyses; and nuclear magnetic resonance inspections. Advancements in NDI technology include not only increased understanding of known techniques, but also the discovery of new techniques and methods.

Familiarity with NDI methods is often made by observations (or experiences) in medical testing, testing in automotive and machine shops and by industry use in process control and process assurance applications. NDI is frequently used in process assurance to provide an unquantified measure of added confidence that a process has been completed in a desired manner. Such applications are frequently carried out by relatively unskilled ("quality inspectors") who follow recipe type instructions and procedures. Process assurance applications constitute a large portion ofNDI technology utilization and are important to various elements of our economy. The characteristic of primary importance is that of providing an unquantified measure of added confidence in process adequacy.

NONDESTRUCTIVE INSPECTION ENGINEERING

Modern engineering structures have incorporated NDI measurements and assessment into the quantitative measurement and validation of materials properties or into the validation of structures or system configuration. The requirement for quantitative measurement, in turn, has required quantification of the methods, procedures, and application capabilities and uncertainties (reliability). Indeed quantification of NDI performance capabilities and reliability are mandatory for the life-cycle management of critical materials, structures and systems.

NDI CAPABILITIES

NDI methods are used to directly or indirectly measure materials, components, structures or systems properties that can be related to a functional capability of interest. NDI methods involve making a measurement of the application of a process that will result in a capability to make a measurement and/or decision concerning a property of interest. The types, and required precision of measurements, to be made and the required application environment, constitute the basic requirements that must be met by the application of an NDI procedure. For example, visual inspection may be applied to confirm the presence or absence of an object (ie. an aircraft on a taxiway), or may be applied to measure the surface texture of a hardware component (ie. examination of a landing gear for surface corrosion). The methods, and complexity of the procedure to be applied, vary widely with the requirements and operating conditions for application.

The most important facts governing all NDI applications are: (1) that no method or technique is universally applicable to meet all test objectives or to characterize all properties of concern; (2) that the results of NDI are measurements and are thus neither absolute nor single valued; and (3) that variations in either the test object of the NDI process parameters may require requalification and revalidation of the specific procedure to be applied.

To emphasize these points, consider the application of a liquid penetrant inspection process to detect surface connected cracks in gas turbine engine blades. Application of a liquid penetrant procedure will <u>not</u> provide assurance that the blades do not contain any cracks, but can provide assurance that the blades do not contain cracks greater the size for which the procedure was validated (typically 0.150 inch long). Further, if the liquid penetrant process is not carefully controlled, the detection (screening level) capability may be considerably greater than the validated value. The threshold detection limit does not constitute the smallest fla· that can be detected by the procedure, but rather the largest flaw that might be missed by the procedure.

In developing NDI capabilities and requirements, it is important to recognize the dynamic nature of the state of the art of nondestructive inspection technology. A large portion of NDI procedures are carried out to provide an unquantified measure of confidence in process control or process output. Such applications are procedure based and are often managed by quality assurance personnel. The capability of such procedures is usually unknown and the level of performance is often accepted by testimonials of compliance with a specification. Sure rote applications often by-pass the necessary NDI engineering steps and result in the application of less capable procedures. A less capable procedure may, in-turn, project an unjustified level of confidence even though the procedure is applied to specifications requirements. Variations in control of procedures applications are also made for the convenience of the production process and/or available personnel. Since the applications are unquantified, no overt variance is observed or recorded. The results cannot be considered to meet the requirements of quantitative NDI.

Quantitative nondestructive inspection must be knowledge and data based. Application, management and accountability must be considered and must be disciplined as an engineering technology. Variations in control of a procedure or application must be based on analysis of NDI data and on impacts on the verification of design / operating margins. Control parameters and outputs must be quantified and variations must be observed and recorded. Quantitative NDI measurement technology has been rapidly increasing to meet the challenges of modern engineering requirements. Technology used experience from past "quality assurance" applications cannot be considered to be a basis for judging quantitative NDI applications. NDI technology is maturing and new methods are emerging. Awareness of such developments are important to obtain producible, reliable and economical designs; to initiate production using modern engineering materials and design concepts; and to enable life-cycle management of modern engineering materials, structures and systems.

QUANTITATIVE NDI PROCESS VALIDATION

All NDI methods involve measurement or measurement of process parameters to produce a measureable output. The NDI detection / measurement process (for example, crack detection) does not produce and absolute, single valued, or binary output. Detection is dependent on the distribution of measurement responses (signal and noise) that are generated by the interaction of the interrogating energy field with the target (flaw). Data interpretation and the decision process constitutes a problem in conditional probability, in which the outcome is not a simple positive or negative response, but may also be a false position (false alarm) or false negative (miss) response. The outcome is governed by the statistical distribution of signal plus noise (S+N) and noise (N) responses, by the acceptance level applied to the signal (S+N) and noise (N) distributions, and by the precision and consistency in applying the acceptance criteria threshold to the signal response data. The signal (S+N) and noise (N) response distributions will vary with the size of crack to be detected. For a large crack, the signal and noise responses may be distinctly separated as shown in FIGURE 1. Little problem in separating the crack signal response from the noise (application noise) response would be expected in this case and a high probability of detection (POD) would be expected with a low probability of false alarms (POFA). Signal and noise response distributions from an intermediate size crack will overlap as shown in FIGURE 2 and a lower probability of detection (POD) and a higher probability of false alarms (POFA) would be expected. As the signal (S+N) response distribution approaches the noise (N) response distribution (ie. small cracks) as shown in FIGURE 3, the probability of detection would be expected to be low and the probability of false alarms would be expected to be high. For all cases (large cracks to small cracks), the actual probability of detection (POD) is not governed solely by the signal and noise response distributions, but is also governed by the acceptance level applied to the process.

Consider the response distribution from an intermediate size crack as shown in FIGURE 4. If the acceptance criteria level is set too low (A-A), all cracks will be found (high POD) but all parts will be rejected (high POFA). If the acceptance criteria level is set too high (B-B), all cracks will not be found (lower POD), but no good parts will be rejected (low POFA). The optimum acceptance level (C-C) is that which produces the highest probability of detection with the lowest probability of false alarms. The point to be made from this discussion is that the optimum acceptance criteria level is inherent to both the nondestructive inspection process being applied, and the noise response, that is characteristic to the test object. Acceptance criteria cannot be an arbitrarily set value. FIGURES 5 and 6 show the probability of detection by an eddy current procedure at two different acceptance criteria levels. Two different acceptance levels were applied to the same set of inspection response data to produce the results shown in FIGURES 5 and 6. It is clear from this data, that the NDI procedure, being applied, is capable of screening smaller flaws with no increase in inspection cost.

Process validation may be accomplished by measurement of the signal (S+N) and noise (N) response distributions from a few cracks of a size near that to be screened. Acceptance criteria are then applied to calculate the probability of detection (POD) and probability of false alarms (POFA). The probability of detection (POD) as a function of flaw size may be plotted by one of several methods. For an inspection process that produces only detection or miss type outputs (ie. liquid penetrant inspection), the probability of detection may be plotted by the maximum likelihood method [REF 1], by the "Probit" method [REF 2], or by the moving average method [REF 3]. Plotting by the moving average method (FIGURE 6) is accomplished by passing a large number of test objects, that contain a large number of flaws of varying sizes (large to small), through an inspection process and recording the output in terms of detection (+) or failure to detect (0). The probability of detection, as a function of crack size, is obtained by: (1) ordering the data from the largest flaw size to the smallest flaw size; (2) selecting a sample size (from sampling tables) that is consistent with the desired confidence level of the output (for example, 29 observations are required to attain a 95% confidence level that the calculated value is correct); (3) counting down sequentially from the largest flaw size to obtain a selected sample group of NDI observations; (4) calculating and plotting the point estimate of detection for the sample group (ie. the number of detections divided by the number of observations); (5) calculating and plotting the lower confidence limit based on the detection success in the sample group; (6) dropping the largest flaw in the sample group and adding the largest flaw from the unsampled data; and (7) repeating the analytical and plotting processes until the data are exhausted. Curve fitting may then be accomplished by regression analysis or by fitting the data to a predetermined function. The shape of the resultant curve is a measure of the degree of control of the process. The detection threshold for the process is the flaw size at the inflexion point of the curve.

For inspection processes that produce quantitative and discrete outputs, alternative analysis procedures can be applied and more inferences can be made for the process. A large number of test objects, containing a large number of flaws of varying size, are passed through an NDI process. Quantitative (discrete) data values for both the flaws and the background (test object material and surface responses) noise are recorded. The data may be analyzed by plotting the flaw response and background response distributions to estimate an acceptance level.

The probability of detection curve may be plotted by the moving average method or by an a/a' method as described by Berens et al [REF 4]. The a/a' analysis method is initiated by plotting the flaw responses as a function of flaw size (FIGURE 7) and using a regression method to obtain a best fit, linear relationship. A test of the normality of the data (degree of control of the process) is obtained by a measure of the scatter of the data along the a/a' regression line. Acceptance criteria are then applied to the data and the probability of detection curve is plotted based on the scatter of data about the regression line [REF 5]. The result is a calculated probability of detection curve as a function of flaw size (FIGURE 8). The a/a' method requires less data then other methods and is useful in estimating detection capabilities when a linear NDI response can be obtained.

The probability of detection data are useful in both establishing engineering acceptance criteria and in the validation of inspection process performance (demonstration of fitness for purpose). Care must however be exercised with respect to the indicated probability of detection and the inspection acceptance level applied. A given procedure may be capable of finding very small flaws, but if a small flaw is present in a part that exhibits high surface response noise (due to service pits, machining texture or other surface texture) all parts submitted for inspection will be rejected. The inherent surface response noise for the part to be inspection may thus be the limiting boundary condition for establishing a detection capability.

The relationship between signal (S+N) and noise (N) may be graphically presented in the form of a signal/noise plot as shown in FIGURE 9 or may be described by a detection specificity plot as shown in FIGURE 10. The advantage of the specificity plot is that of presenting both the probability of detection (POD) and the probability of false alarms (POFA), as a function of acceptance criteria, on the same plot. Such plots enable optimum production decisions to be made in terms of both quantitative error rates and design margins [REF 6].

Nondestructive inspection technology is being developed to provide predictive modeling tools based on the calculation of energy propagation, energy partition, energy scattering and flaw interactions. Geometry corrections may be added to predict the signal and noise responses that are used in generating probability of detection outputs. The utility and practicality of such models are dependent on correct modeling of both the material / flaw interactions and the application test conditions. In such modeling, it is necessary to account for the noise that is inherent to each hardware application by either predictive methods based on propagation and scattering theories or by the incorporation of experimental data into the model. Each individual application of a nondestructive inspection procedure requires validation, under the conditions of application to assure the screening of critical defects (flaws).

SUMMARY AND CONCLUSIONS:

Nondestructive inspection technology has a long history of successful application to engineering hardware to provide added confidence and validation of material / process control in production and in added confidence in the continuing fitness for service of engineering materials, structures and systems. Modern engineering and life prediction methods require both quantitative outputs and precision in the measurement of those materials properties that are used to establish design margins, confidence in fitness for service and life predictions based on fixed maintenance intervals that are used to revalidate continuing fitness for service. Materials measurement science and engineering have advanced to meet the challenges of implementing improved design and production tools and improved life prediction tools. The effectiveness of implementation is dependent, in part, on a clear understanding of the data / information requirements necessary to establish and validate nondestructive inspection procedures; on an understanding of the capability and operating boundary conditions of nondestructive inspection procedures applications; on an understanding of the nondestructive inspection engineering time, materials and test specimen requirements; on the boundary conditions of access, specialized tooling and equipment; and on the operating conditions necessary to make the measurements.

This paper has briefly identified some of the nondestructive inspection methods and procedures that have been successfully implemented to add confidence to the fitness for purpose of modern engineering hardware. Some of the methodology, data requirements and implementation boundary conditions have been discussed with the objectives of increasing a broad awareness and understanding of modern nondestructive inspection technology and dispelling ideas and past experiences gained from procedures based nondestructive testing that were implemented as procedure based applications. Significant points of the discussions included:

1. Recognition that no single nondestructive inspection technique or measurement is capable of providing confidence in fitness for purpose in all applications. Nondestructive inspection measurement tools and procedures are as varied and complex as the other engineering design tools that are used to establish the varied conditions and requirements for measurement of materials, structures or systems properties.

2. Recognition that nondestructive inspection measurements are not absolute or single valued. If a nondestructive inspection measurement is required, the measurement criteria and precision must be accurately stated. Designs which call for measurement with a "no flaw" or "unattainable small flaw" criteria are incomplete and are not acceptable for implementation.

3. The concepts, rationale and methodology used in nondestructive inspection measurements, analyses and validation were discussed. No single methodology or validation procedure is applicable to all nondestructive inspection methods or applications.

4. Some of the uncertainties, error sources and operating boundary conditions of nondestructive inspection implementation were discussed. Each application has unique constraints and boundary conditions.

5. Finally, the dynamic state of nondestructive inspection measurement science and technology was discussed in terms of the requirements, methods and practical considerations for implementation.

Nondestructive inspection measurement is an essential element in design, production, operating and maintenance of safe, reliable and efficient engineering structures and systems. Integration of nondestructive inspection requirements, as primary considerations in conception, design and test, are necessary to attain maximum engineering structures and systems efficiency, reliability and service.

NONDESTRUCTIVE INSPECTION ENGINEERING

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5. Ibid.

6. Rummel, Ward D., Brent K. Christner, Steve J. Mullen, and Donald L. Long, Characterization of Structural Assessment Testing, SA-ALC/MMEI-1-86, USAF NDI Program Office, Kelly AFB, TX, January, 1986.

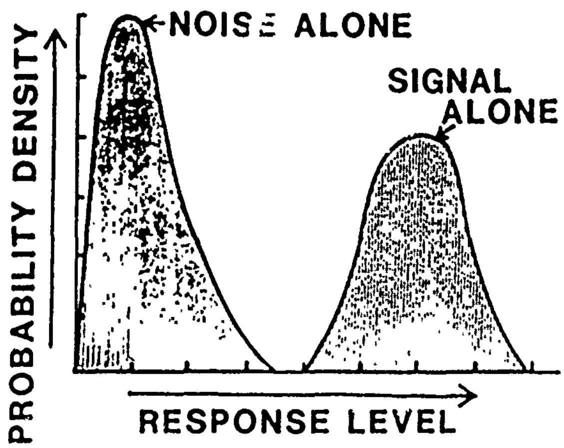


FIGURE 1. SIGNAL / NOISE RESPONSE FOR DISCRIMINATION WITH A HIGH DEGREE OF SPECIFICTIY

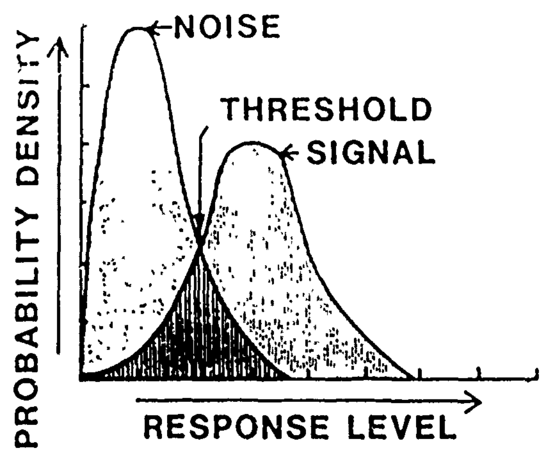


FIGURE 2. SIGNAL / NOISE RESPONSE FOR DISCRIMINATION WITH OVERLAPPING SIGNAL AND NOISE STIMULI

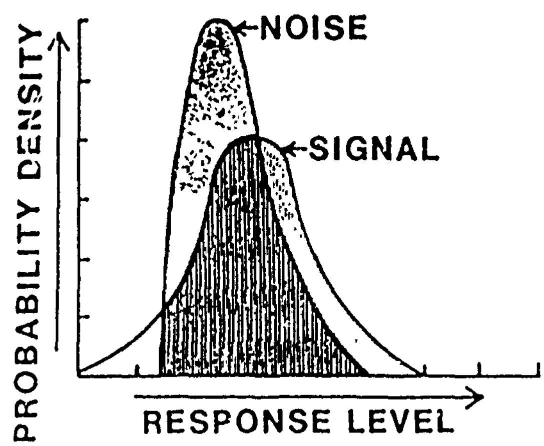
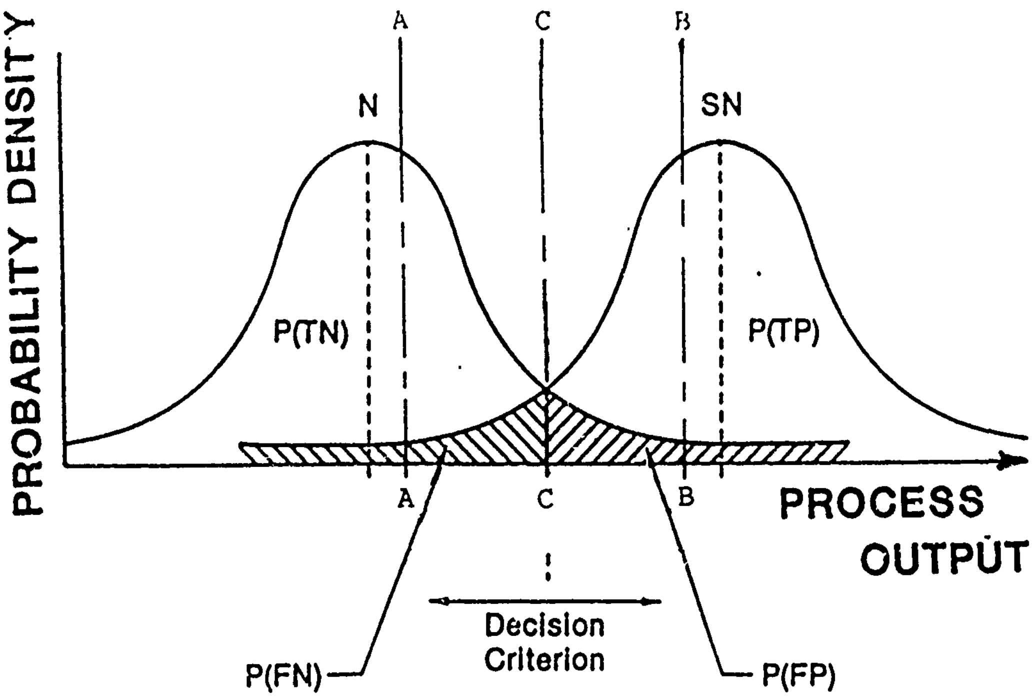


FIGURE 3. SIGNAL / NOISE RESPONSE FOR COINCIDENT STIMULI



- Null Hypothesis "N" = No Flaws
- Alternate Hypothesis "SN" = Flaws
- P(FP) = Type I Error: "Significance" of Test
- P(FN) = Type II Error

FIGURE 4. INFLUENCE OF DECISION CRITERIA ON THE PROPABILITY OF DETECTION FOR FIXED SIGNAL (S+N) AND NOISE (N) DISTRIBUTIONS

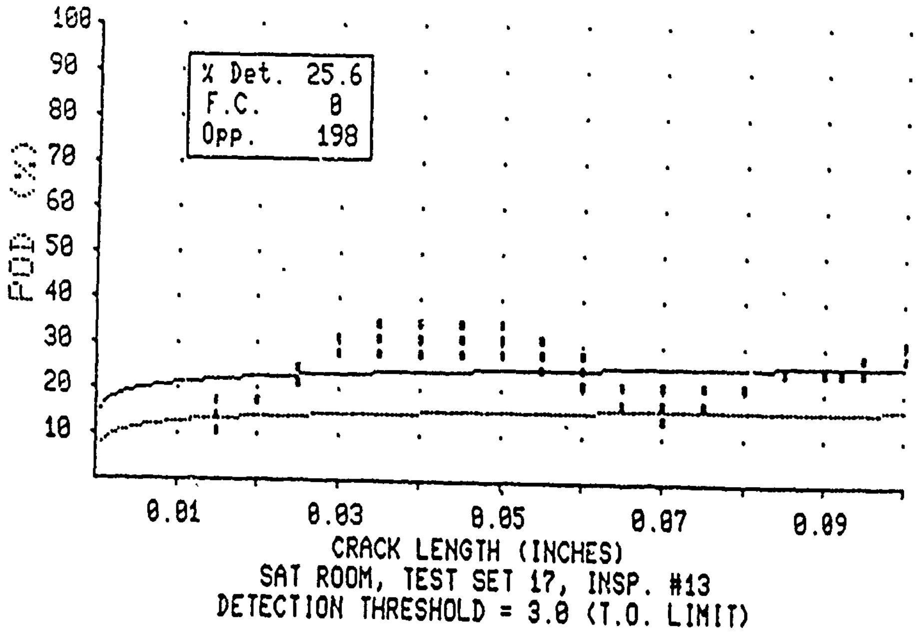


FIGURE 5. PROBABILITY OF DETECTION (POD) CURVE FOR TEST SET #17, INSPECTION SEQUENCE #13 PLOTTED BY THE MOVING AVERAGE METHOD 3.0 millivolts - ACCEPTANCE THRESHOLD

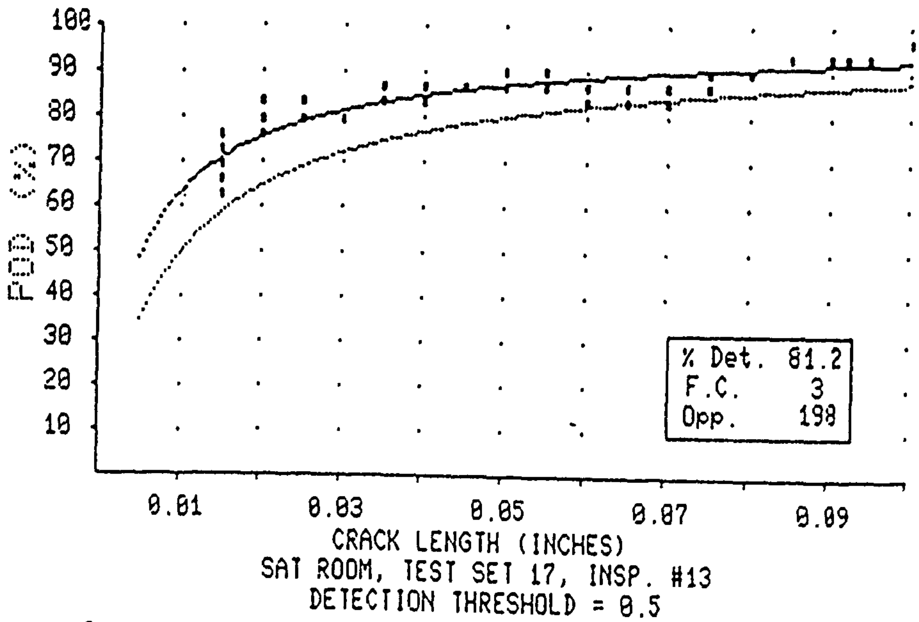


FIGURE 6. PROBABILITY OF DETECTION (POD) CURVE FOR TEST SET #17, INSPECTION SEQUENCE #13 PLOTTED BY THE MOVING AVERAGE METHOD 0.5 millivolts - ACCEPTANCE THRESHOLD

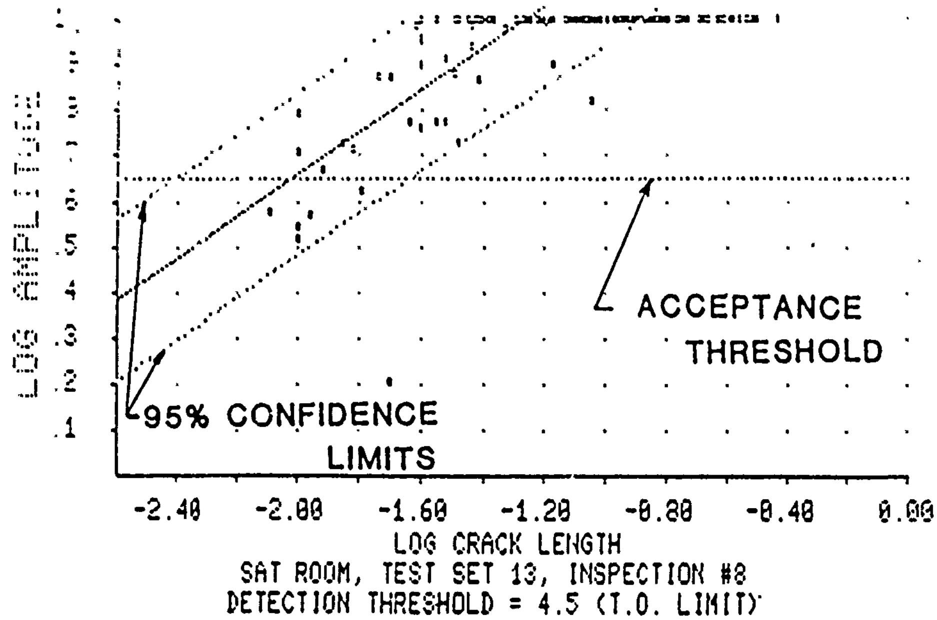


FIGURE 7. a/a' REPONSE RELATIONSHIPS FOR TEST SET #13, INSPECTION SEQUENCE #8

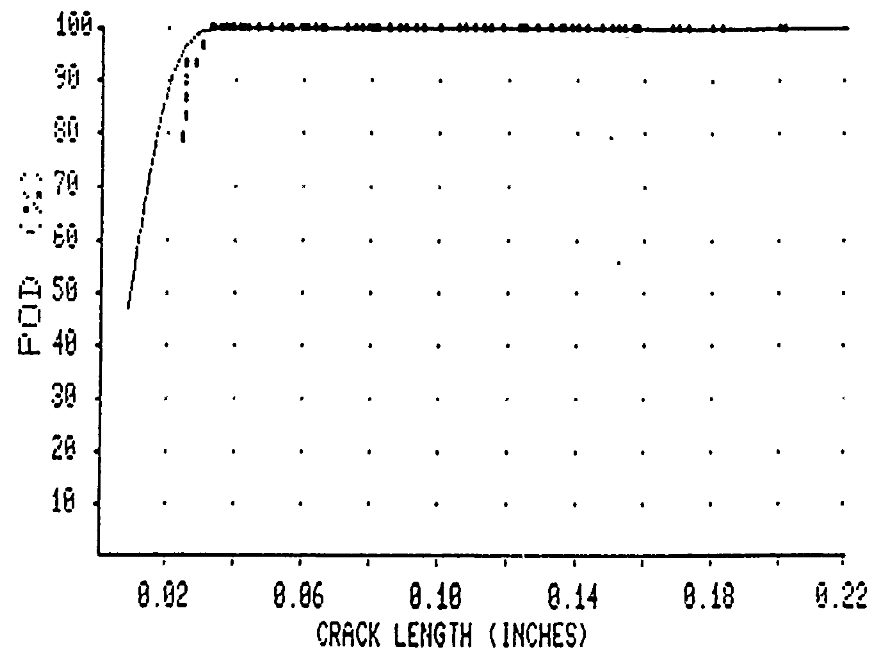


FIGURE 8. PROBABILITY OF DETECTION (POD) CURVE FOR TEST SET #13, INSPECTION SEQUENCE #8 PLOTTED BY THE a/a' METHOD

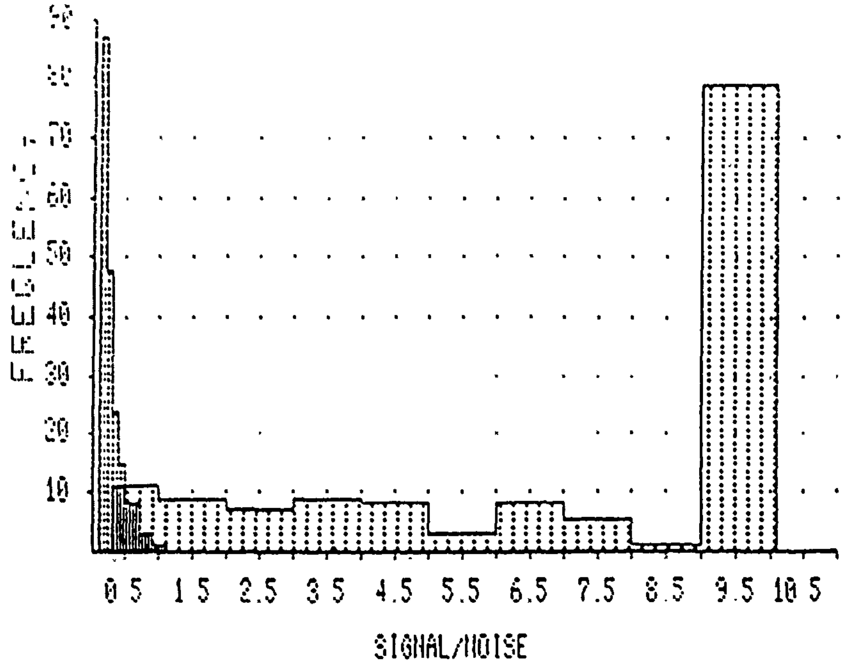


FIGURE 9. SIGNAL (PLUS NOISE) AND NOISE RESPONSE PROBABILITY DENSITY DISTRIBUTIONS FOR TEST SET #14, INSPECTION SEQUENCE #17

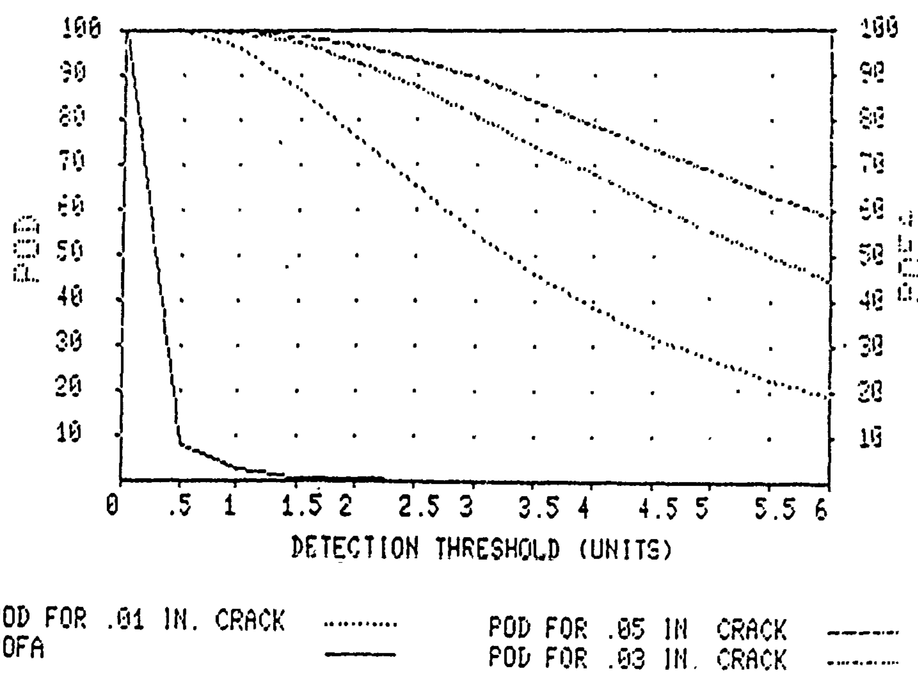


FIGURE 10. DETECTION SPECIFICITY CURVE FOR TEST SET #14, INSPECTION SEQUENCE #17

Plastic Bead Blast Materials Characterization Study

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PLASTIC BEAD BLAST MATERIALS CHARACTERIZATION STUDY

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ABSTRACT

The U.S. Air Force, in recognition of the need to determine the effect that paint removal by plastic bead blasting (PBB) will have on aircraft materials, initiated a comprehensive study of a pilot process currently being used at Ogden Air Logistics Center (ALC) to remove paint from F-4 aircraft and numerous small aircraft parts.

Two-foot-square panels of newly acquired material were painted, artificially cured, and stripped up to 4 times by the PBB process at the Ogden ALC. Tension tests, center notched and unnotched fatigue tests, fatigue crack growth tests, and Almen type strip tests were conducted on both the as-received and stripped materials.

The PBB process was found to significantly lower fatigue life in unnotched specimens and to increase fatigue crack growth rates, particularly in the 0.016-inch and 0.032-inch thick materials tested. The process was found to have a lesser effect on materials with a thickness of 0.063-inches or greater. No decrease in fatigue life was observed for notched specimens with a stress concentration of 2.43.

Materials included in this study were 7075-T6, 7075-T6 (clad), and 2024-T3 aluminum sheet in a thickness range from 0.016 to 0.190 inches; one thickness of 2024-T81, 2219-T81, and 7475-T761 (one side clad) aluminum sheet in a thickness range from 0.063 to 0.080 inches; and 0.063 inch thick Ti-6Al-4V sheet.

An investigation of the stripped material found the blasted surface to be covered with small crater like defects. An inspection of

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the fatigue coupon fracture surfaces has shown the point of crack initiation to be generally at one of these craters. These craters are the likely mechanism for the decrease in fatigue life.

During the study, residual stresses produced by the plastic bead blast process were measured. Based on preliminary analysis, the magnitude of these residual stresses are high enough to account for the increase in crack growth rates observed.

A follow-on study is in progress to reduce the cause of surface craters and to reduce the residual stresses produced. The program will investigate the process parameters with the goal of establishing an operations envelope which will provide fatigue and crack growth rates that are similar to those of virgin materials.

Key words: paint removal, plastic bead blasting, fatigue, fatigue crack growth, tension tests, Almen strips, aluminum, titanium, surface roughness, fastener preload.

INTRODUCTION

In recognition of the need to determine the effect that paint removal by plastic bead blasting (PBB) will have on fatigue-critical areas of aircraft, the U.S. Air Force has initiated a plastic bead blast characterization study at Battelle. A limited earlier investigation, conducted by AFWAL's Materials Laboratory[3], indicated the need for a comprehensive study of the effects of PBB.

The plastic bead blast paint removal process is one method currently under consideration as a possible alternative to the present chemical paint removal method. The present chemical removal process produces in excess of 10,000 gallons of toxic waste per fighter type aircraft. Therefore it is important that an alternative stripping process be developed.

The PBB process at Ogden ALC involves a pressurized system through which a plastic medium is propelled through a hose having a 3/8-inch diameter nozzle. A 40 psi nozzle pressure is currently used along with a "Type II" plastic bead medium having a 30-40 mesh size (0.023-0.015 inches), a hardness of 3.5 on the Moh scale, and a density of 1.5 g/cc. The medium is manufactured by U.S. Technology; in appearance, it looks like a multicolored sand with sharp edges. Nozzle standoff distance and angle relative to the part being stripped are not controlled but are generally 18 to 24 inches and 60 to 90 degrees, respectively. Velocity of the medium, 18 inches from the nozzle measured by high speed photography was found to be between 525 and 625 ft/sec.

TEST PROGRAM AND PROCEDURES

Aluminum alloy materials studied in this program were purchased in sheets 4 by 12 feet. The titanium sheet was 3 feet by 8 feet. The large sheets were cut into manageable sizes, typically 24 by 25 inches, and designated for testing in the as-received condition or for submittal

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[3] Childers, S., Watson, D. C., Stumpff, P., and Tirpak, J., "Evaluation of the Effects of a Plastic Bead Paint Removal Process on Properties of Aircraft Structural Materials", AFWAL-TR-85-4138, December 1986.

to one of several PBB conditions consisting of a number of paint-strip cycles. Table 1 lists the materials that were obtained together with their surface preparations and thicknesses. Also shown are the number of paint-strip cycles to which each material was exposed.

Panels designated for the paint-strip cycles were sent to Ogden ALC for painting and stripping. All panels were cleaned and alodined, then primed with epoxy, painted with polyurethane, air cured for 72 hours, and finally cured for 96 hours at 200 F. Each panel was then stripped by the PBB process described above. This paint-strip procedure was repeated for the number of cycles listed in Table 1.

Each of the as-received material types were cut into Almen strip specimens as shown in Figure 1. Almen strips, made from spring steel, are used frequently in shot peening to provide a relative measurement of the residual stresses being produced. The magnitude of residual stress is indicated by the amount of curvature generated in the strip and is measured in terms of arc height over a set span. Attempts at using standard N gauge spring steel Almen strips, having a thickness of 0.031 inches, showed that the residual stresses produced by the plastic bead blast process did not produce a measurable curvature. In an attempt to find an adequate Almen strip material, two Almen strips made of the material and thickness being stripped and two Almen strips each of .032-inch-thick 7075-T6 bare and 2024-T3 bare aluminum were located around the panel perimeter and blasted. The panels and Almen strips were mounted on a support fixture as shown in Figure 2 during the stripping process. Arc heights were measured following each strip cycle with the intent of establishing, 1) a low cost, quick method of measuring the amount of residual stress being introduced into the material and 2) a method of checking the effect on the surface finish prior to stripping an actual part. The results showed that the 0.032-inch-thick bare aluminum was adequately sensitive in providing measurable arc heights and is a candidate device for future process control.

The as-received and the painted and stripped panels were cut into tensile, fatigue, and crack growth coupons as shown in Figures 3 through 7, respectively. Three types of fatigue coupons were produced; unnotched, notched with a 1/4-inch-diameter button head cap screw (NAS

1578), and notched with a 1/4-inch-diameter countersunk head screw (NAS 1580). Materials with a thickness less than or equal to 0.080 inches contained the button head cap screw and materials with a thickness of 0.160 inches or greater contained the countersunk head screw. A 1/4-inch centrally located hole in a 1.0-inch wide plate in tension provides a 2.43 stress concentration based on net section stress (local stress = 2.43 x net section stress). Each fastener was finger tightened with a nylon washer between the specimen and nut to prevent a secondary source of fatigue damage. In addition, all fatigue coupon edges were polished with 600 grit aluminum oxide paper, including the corners along the test section edges, to decrease the chances of crack initiation at flaws produced by the machining process.

Fatigue and crack growth tests were conducted in electrohydraulic and servocontrolled closed-loop test systems. Each system had a load cell mounted in direct line with the specimen. The calibration of all systems is traceable to the National Bureau of Standards.

TEST RESULTS SUMMARY

Tensile Tests

Tensile ultimate strength, yield strength (at 0.2 percent strain), percent elongation to failure, and Young's modulus measurements were made for all of the as-received materials and for the materials exposed to the various PBB cycles. Typically, the as-received tensile ultimate and tensile yield strengths were slightly higher than the reported values listed in MIL-HDBK 5D. This was expected, since the reported values are statistically derived minimum values. Static properties for the PBB materials were comparable to the properties measured for the as-received materials. All static properties are based on results obtained from three samples for each of the conditions tested.

Fatigue Tests

A major focus of this study was to determine the effects of the Ogden ALC PBB process on the fatigue properties of metallic materials common to U.S. aircraft. Figures 8 through 22 summarize the fatigue results obtained for the materials and PBB conditions tested. Based on the data, it appears that the PBB process significantly decreases (in several cases up to one log life) the fatigue life for materials with a 0.032 inch thickness or less. For materials having a thickness of 0.063 inches or greater, with the exception of the 0.063-inch thick 2024-T3 bare aluminum, the fatigue life loss for 4 PBB cycles is significantly less and is, in most cases, within the scatter for the as-received data.

An investigation of the PBB fatigue coupon fracture surfaces found that the point of crack initiation was at a surface defect created by the PBB process. A typical surface defect is shown in Figure 23 (Site Number 3 in Figure 24) at a 400X magnification. These defects were observed to be widely scattered on the PBB coupons, as shown in Figure 24 at a 15X magnification.

Samples of the plastic bead medium were collected during the blasting of the test panels. A portion of the medium was mixed with a solution of trichlorotrifluoroethane allowing dense particles to be separated. A chemical analysis of the particles separated on the basis of density showed a high concentration of silicon (probable SiO_2 characteristic of sand) and titanium together with traces of other elements. It is these contaminants which are possibly responsible for the surface damage and observed loss in fatigue life.

Crack Growth Rates

Figures 25 through 27 summarize the percent change in crack growth rate produced by PBB for the three materials tested. Consistent with the fatigue results, an increase in crack growth rate was most evident in materials with a thickness of 0.032 inches or less. For example, as shown in Figure 26, a mean increase in crack growth rate in excess of 160 percent was observed for the 0.016-inch-thick 7075-T6 bare aluminum at a stress intensity range of 6 ksi $in.^{1/2}$ Figure 27 shows a 106 percent increase in crack growth rate for a stress intensity range of 15 ksi $in.^{1/2}$ Crack growth rate increases are less than 50 percent for materials having a thickness of 0.032-inch or greater. For a material with a thickness of 0.063 inch or greater (with the exception of the 0.071-inch-thick 7075-T6 bare aluminum), the effect of PBB seems to have a minimal effect on crack growth rates.

During the program, 0.032-inch-thick 7075-T6 aluminum Almen strips were instrumented with strain gages to measure the back surface (surface opposite the bead blasted side) residual strain resulting from the PBB process. An average microstrain in excess of 200 was obtained after 1 PBB cycle and an average microstrain of 250 was obtained after 4 PBB cycles. These two strain values relate to a residual stress of 2000 and 2500 psi, respectively. Assuming that this residual stress increases the stress ratio, a simple analysis, based on the crack-growth data in Figure 3.7.4.1.9 of MIL-HDBK-5D, indicates that such an increase in crack growth rate could be expected.

CONCLUSIONS

The plastic bead blast process to strip paint from aircraft materials, as in operation at the Ogden ALC pilot facility, produces measurable damage in the form of lower fatigue life and increased crack growth rates particularly in thin materials.

(1) A reduction in fatigue life and an increase in crack growth rates for less than 0.063 inch-thick materials occurred as a result of plastic bead blasting of aircraft materials for paint removal according to the pilot process at Ogden ALC.

(2) For materials greater than 0.063 inch thick, the damage observed is significantly less and may be within the scatter expected for the data excpet for the 0.063-inch thick 2024-T3 bare aluminum which showed definite fatigue life loss.

A follow-on study is in progress to determine the exact source of the surface damage, to reduce the cause of the surface damage, and to conduct a parametric study of the PBB process with the intention of decreasing residual stress effects.

TABLE 1 - MATERIALS INVESTIGATED

Material Type	Surface Preparation	Thickness in.	Number of Paint-Strip Cycles
7075-T6	Bare	0.032	4
		0.071	1,2,3,4
		0.190	4
7075-T6	Clad	0.016	4
		0.032	4
		0.071	1,2,3,4
		0.160	4
2024-T3	Bare	0.016	4
		0.032	4
		0.063	4
		0.190	4
2024-T81	Bare	0.080	4
2219-T81	Bare	0.063	4
7475-T761	Clad	0.071	4
Ti-6Al-4V	Bare	0.063	4

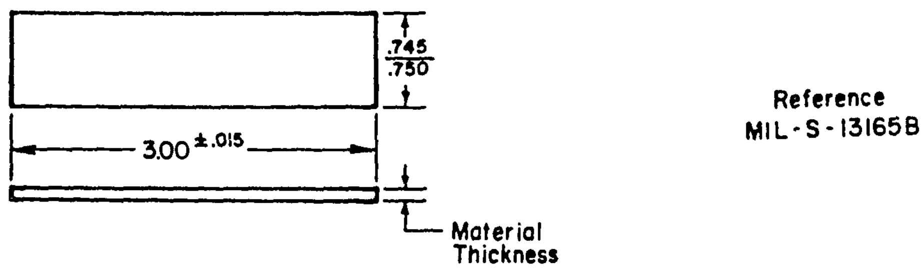


Figure 1 - Almen Strip Specimen

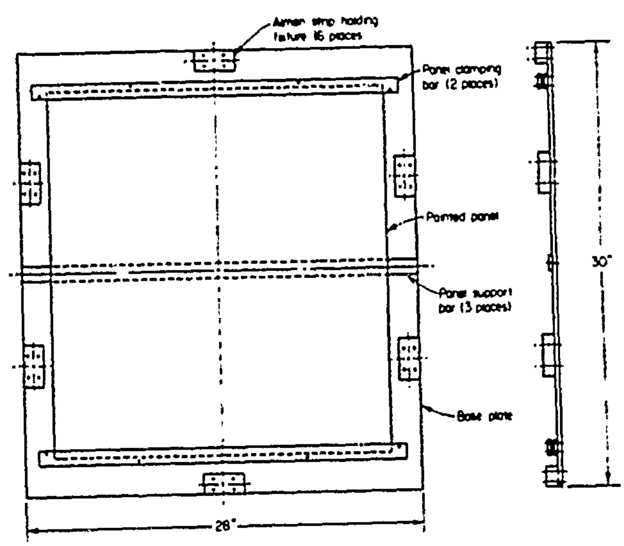


Figure 2 - Plastic Bead Blast Support Fixture

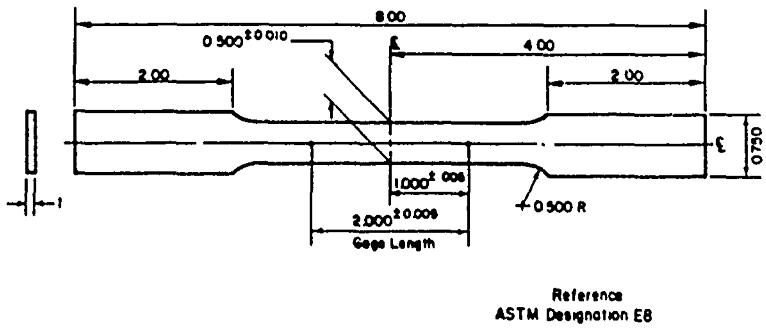


Figure 3 - Tensile Specimen

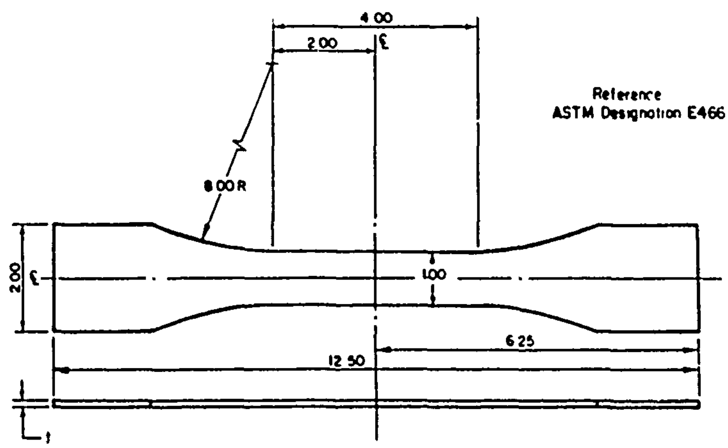


Figure 4 - Unnotched Fatigue Specimen

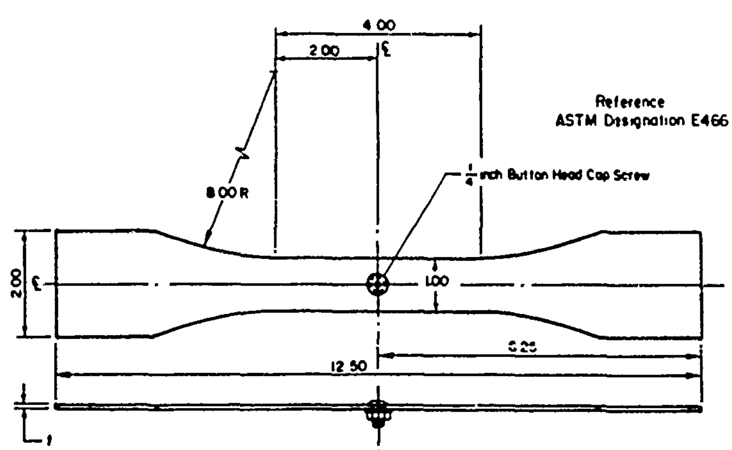


Figure 5 - Notched Fatigue Specimen With Button Head Cap Screw

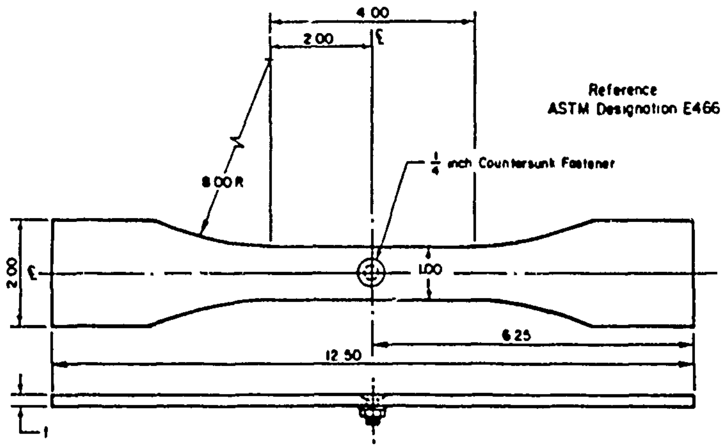


Figure 6 - Notched Fatigue Specimen With Countersunk Cap Screw

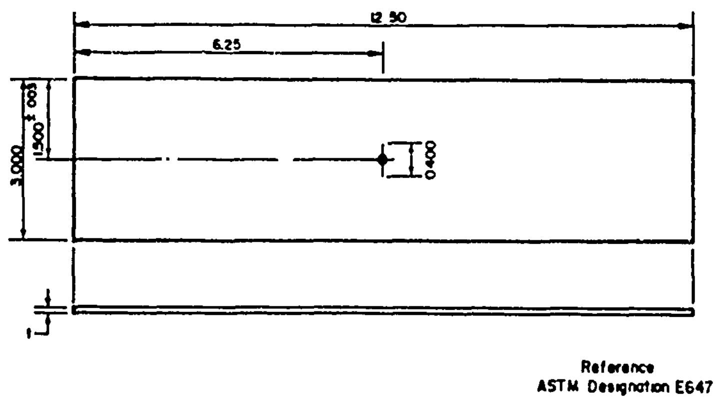


Figure 7 - Fatigue Crack Growth Specimen

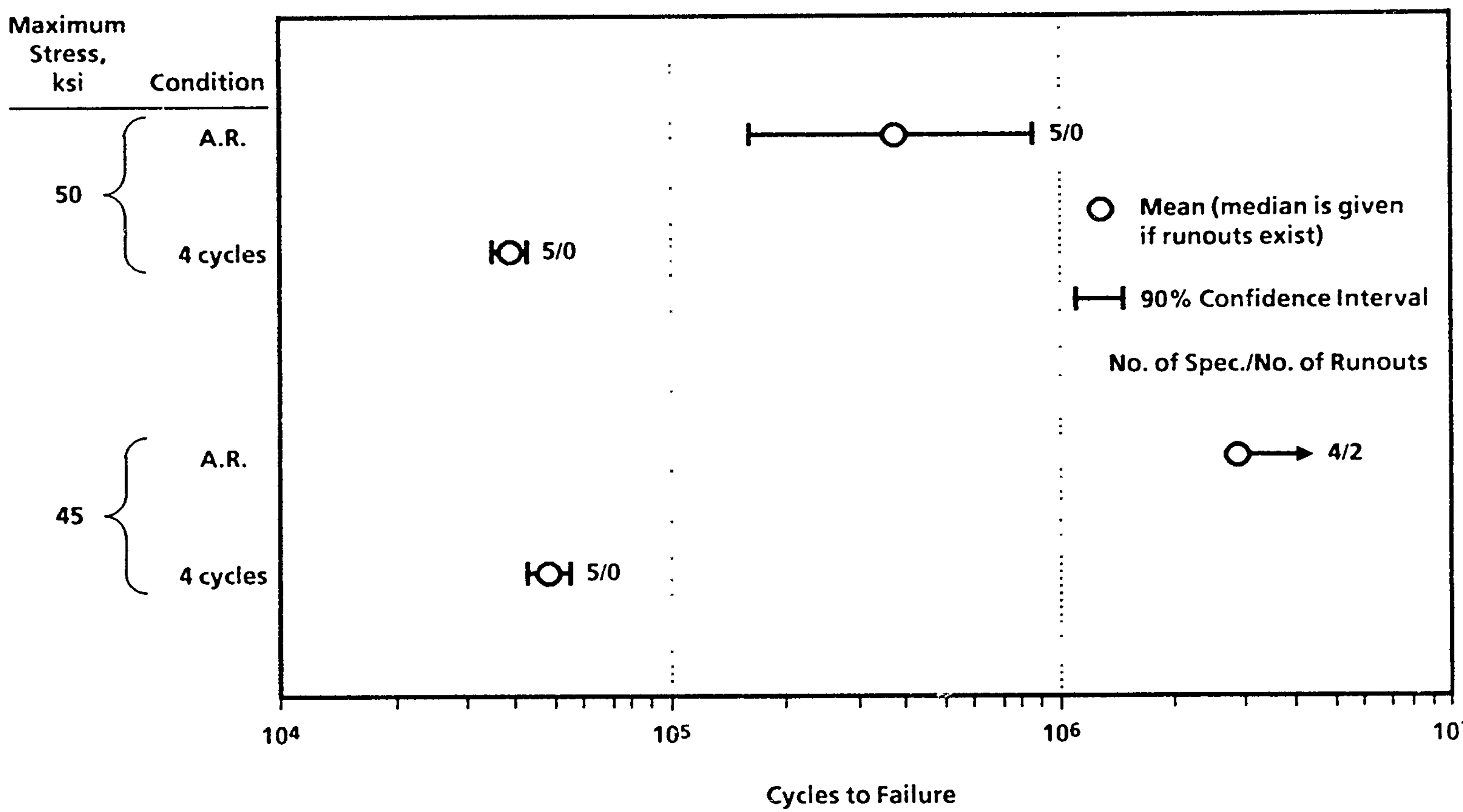


FIGURE 8. 7075-T6 BARE ALUMINUM FATIGUE RESULTS, t = 0.032 in., R = +0.3

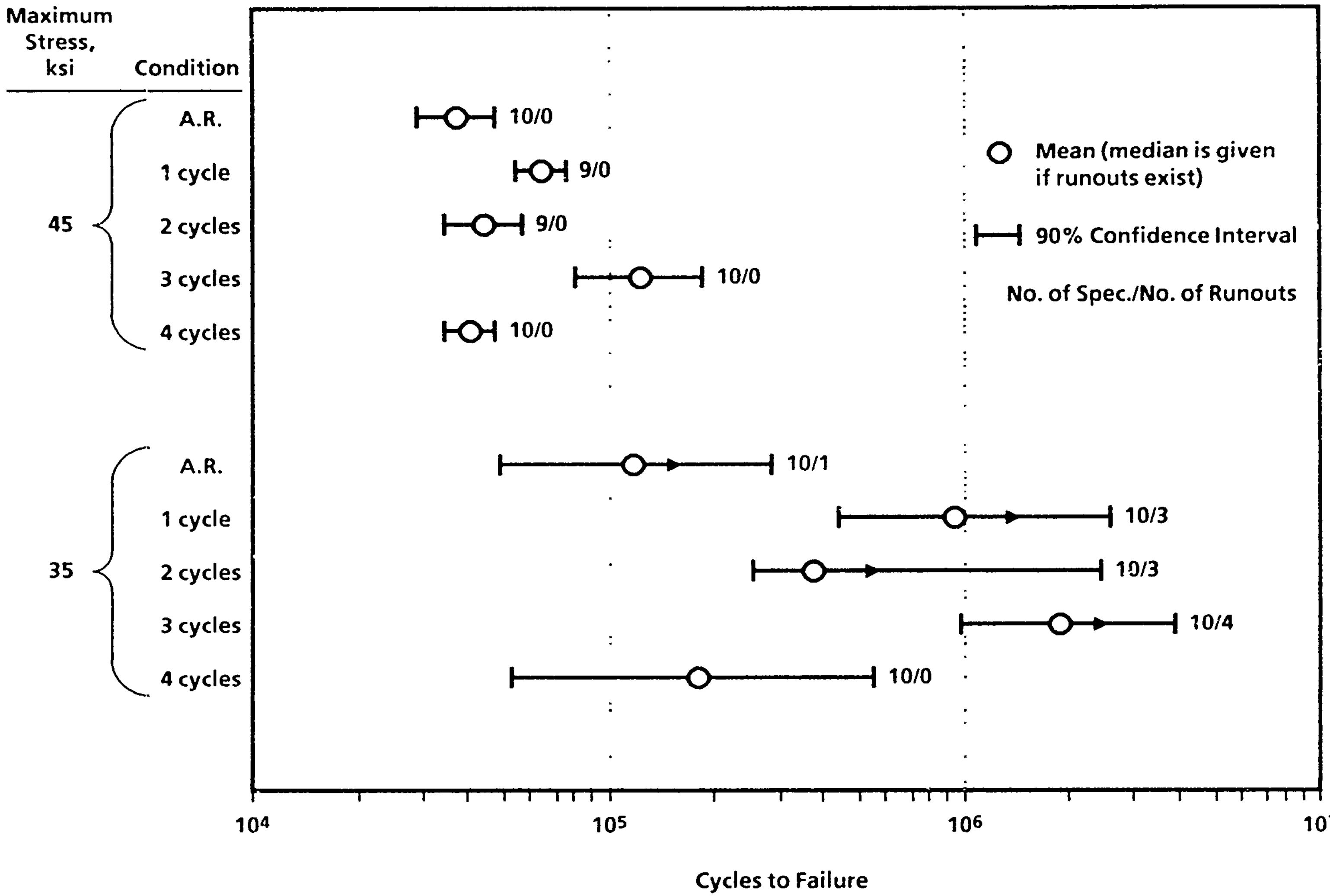


FIGURE 9. 7077-T6 BARE ALUMINUM FATIGUE RESULTS, t = 0.071 in., R = +0.1

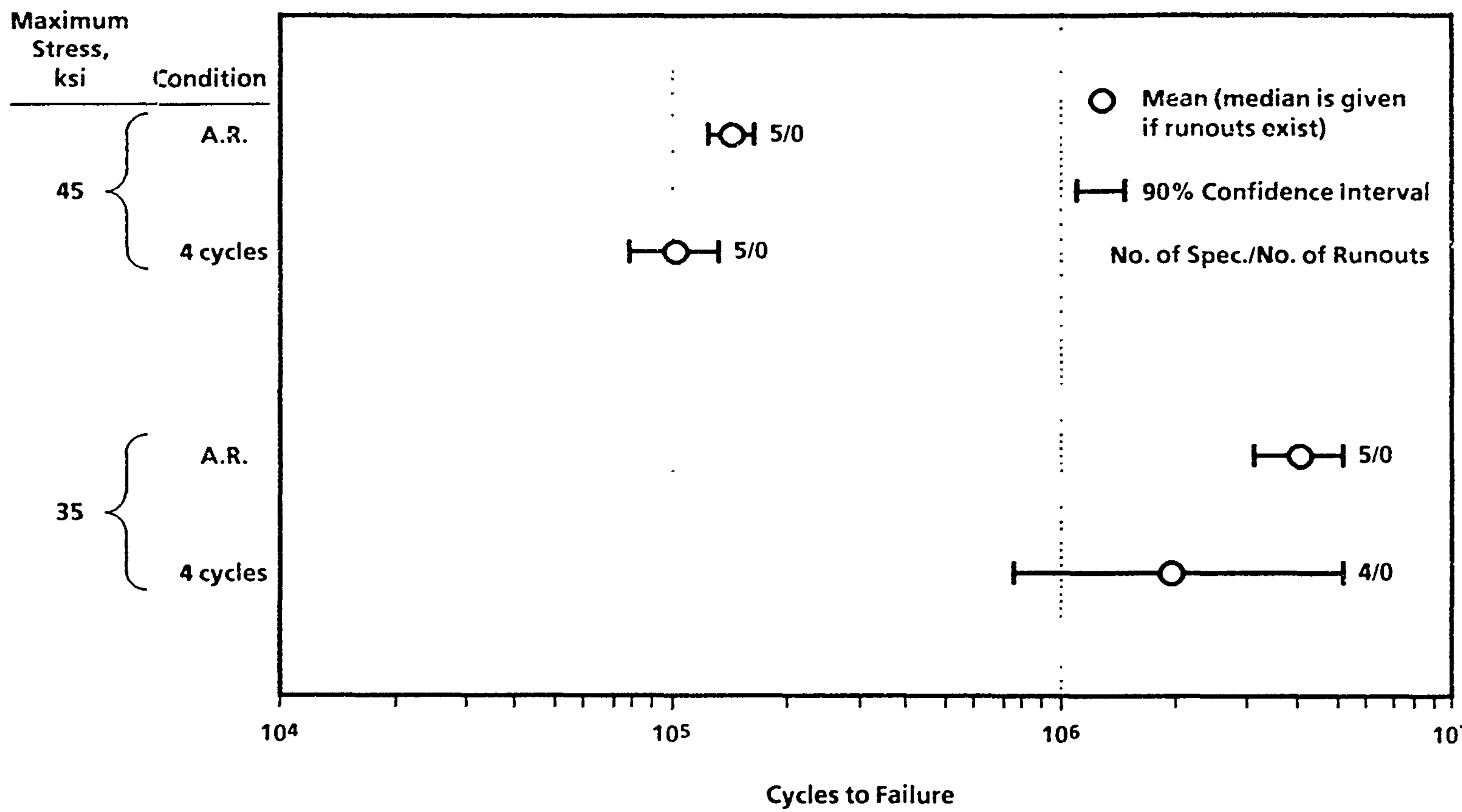


FIGURE 10. 7075-T6 BARE ALUMINUM FATIGUE RESULTS, t = 0.190 in., R = +0.1

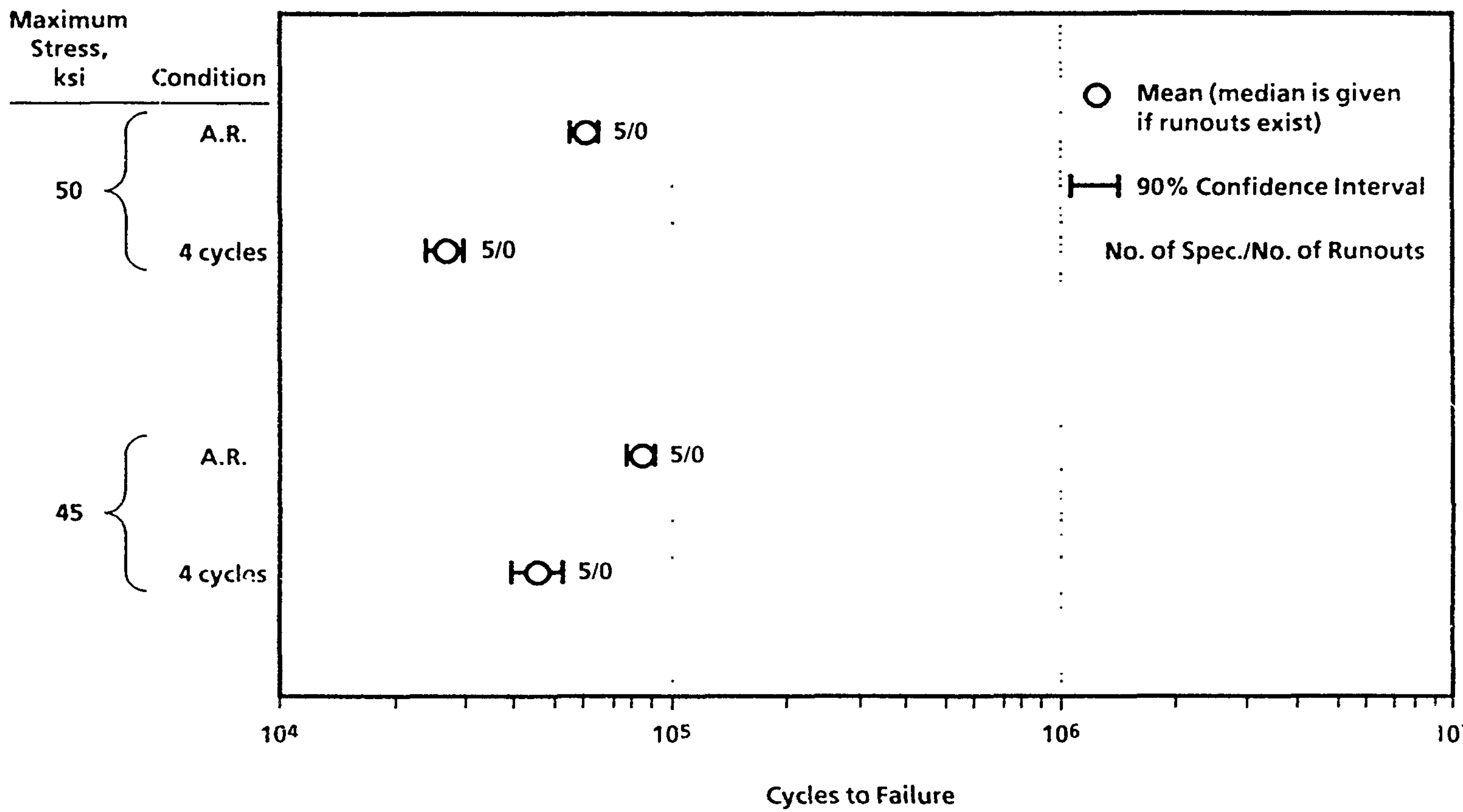


FIGURE 11. 7075-T6 CLAD ALUMINUM FATIGUE RESULTS, t = 0.016 in., R = +0.3

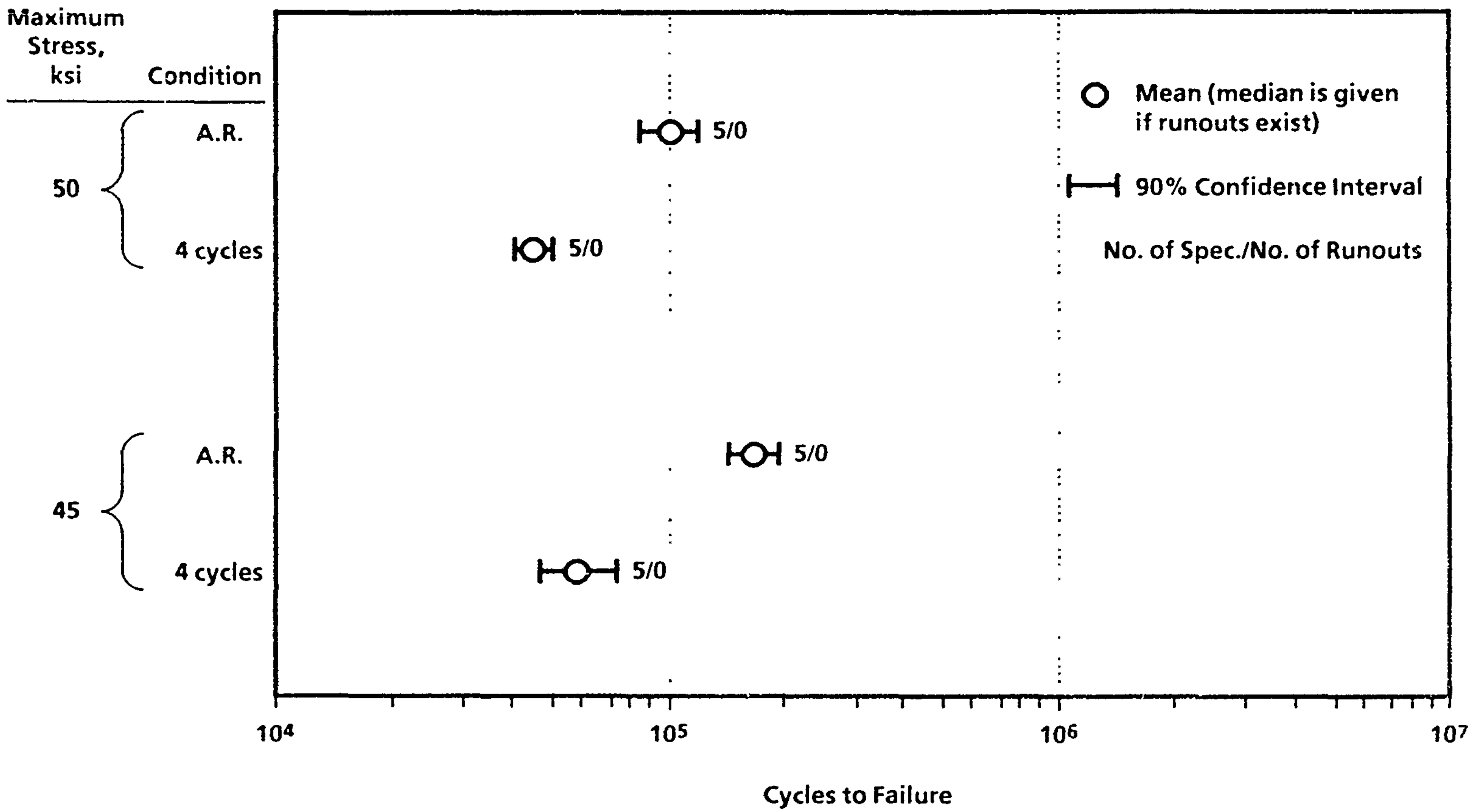


FIGURE 12. 7075-T6 CLAD ALUMINUM FATIGUE RESULTS, t = 0.32 in., R = +0.3

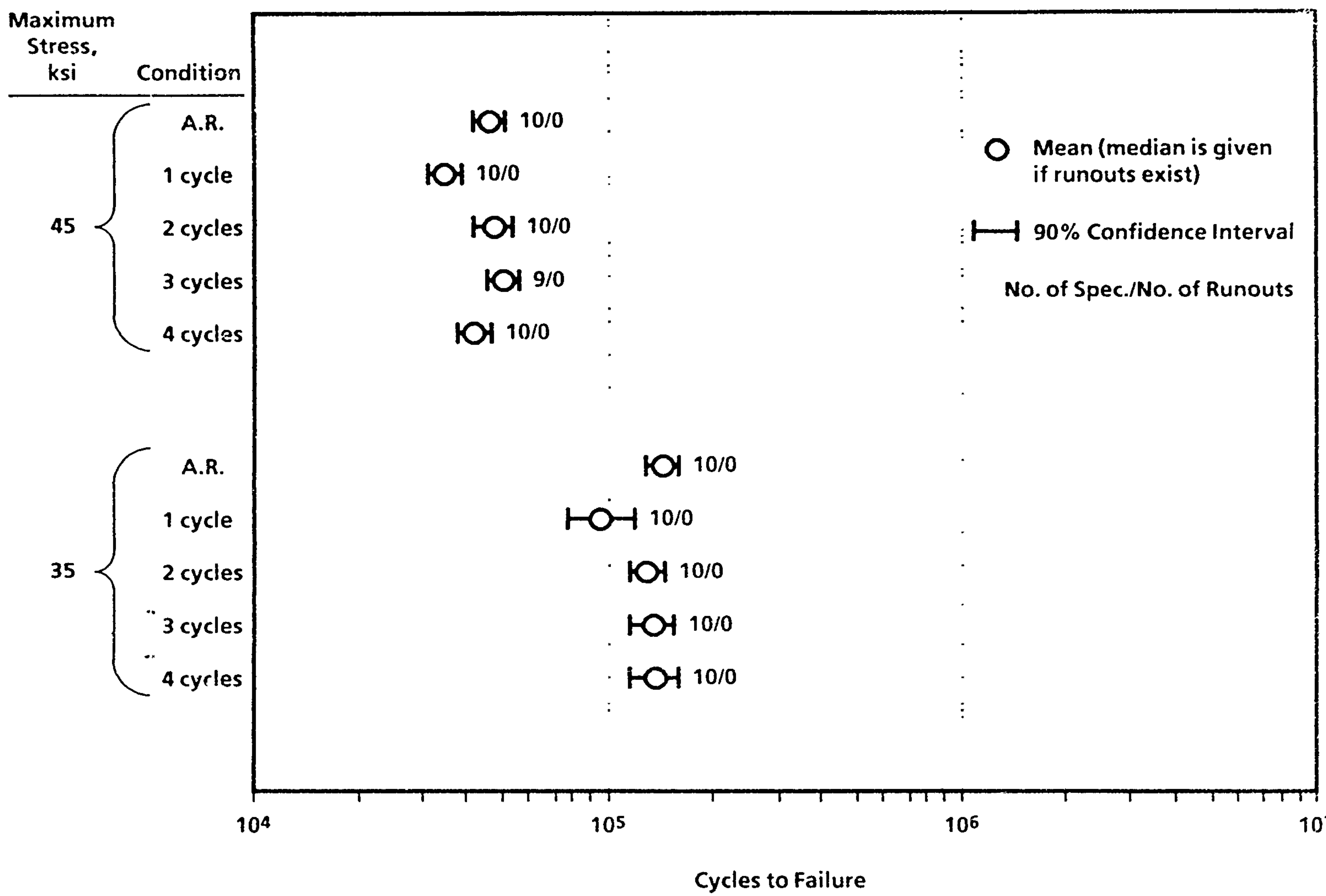


FIGURE 13. 7075-T6 CLAD ALUMINUM FATIGUE RESULTS, t = 0.071 in., R = + 0.1

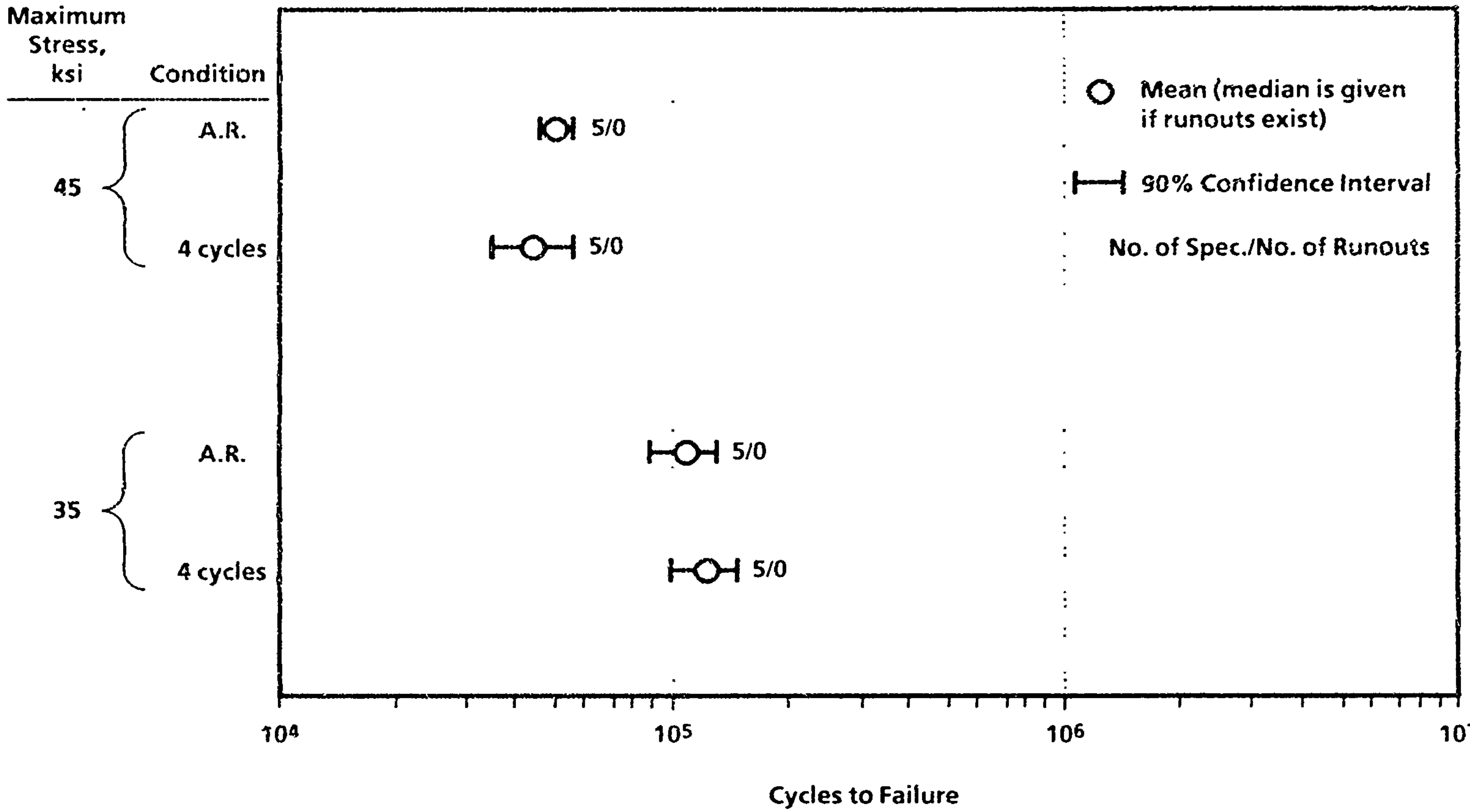


FIGURE 14. 7075-T6 CLAD ALUMINUM FATIGUE RESULTS, $t = 0.160$ in., $R = +0.1$

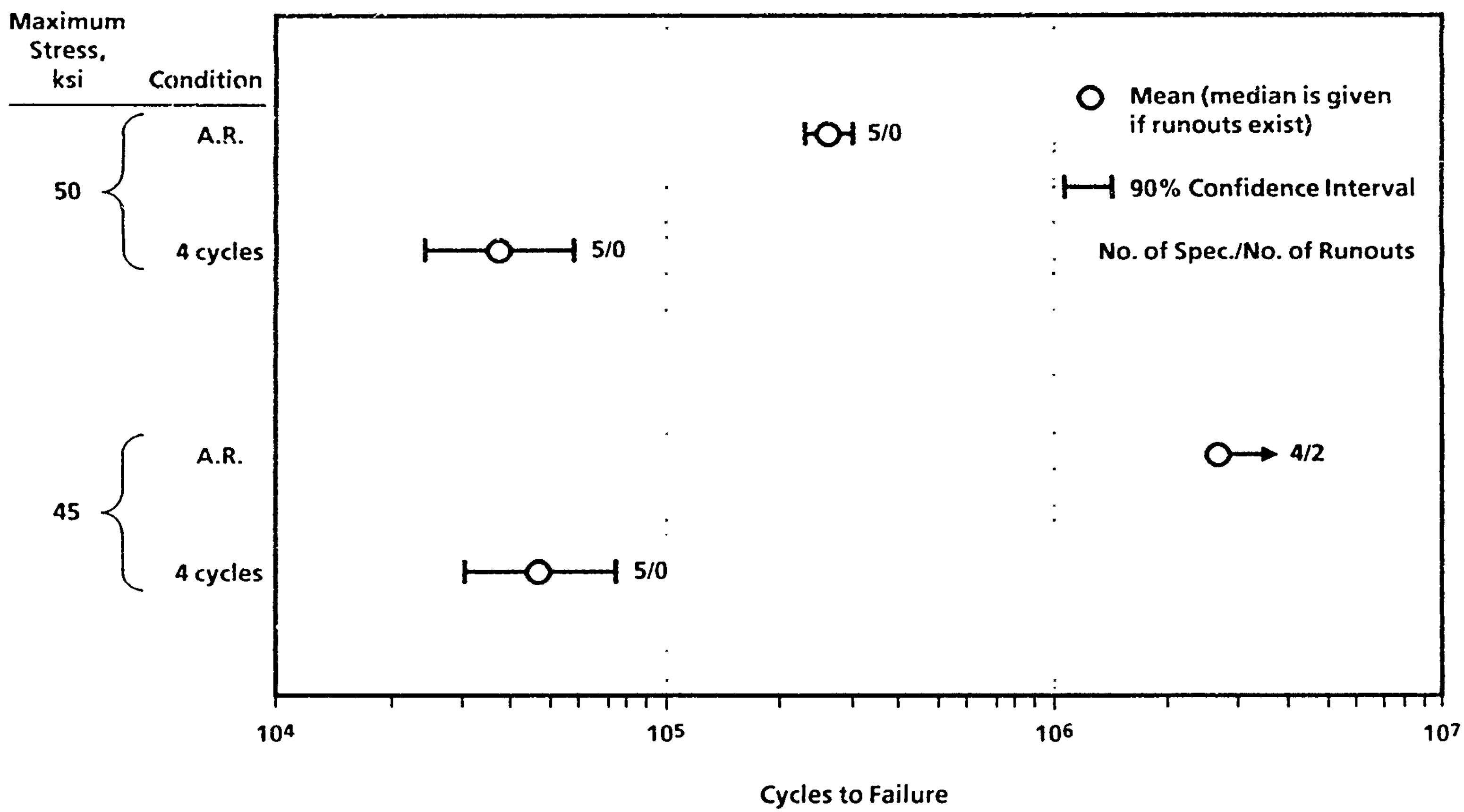


FIGURE 15. 2024-T3 BARE ALUMINUM FATIGUE RESULTS, t = 0.016 in., R = +0.3

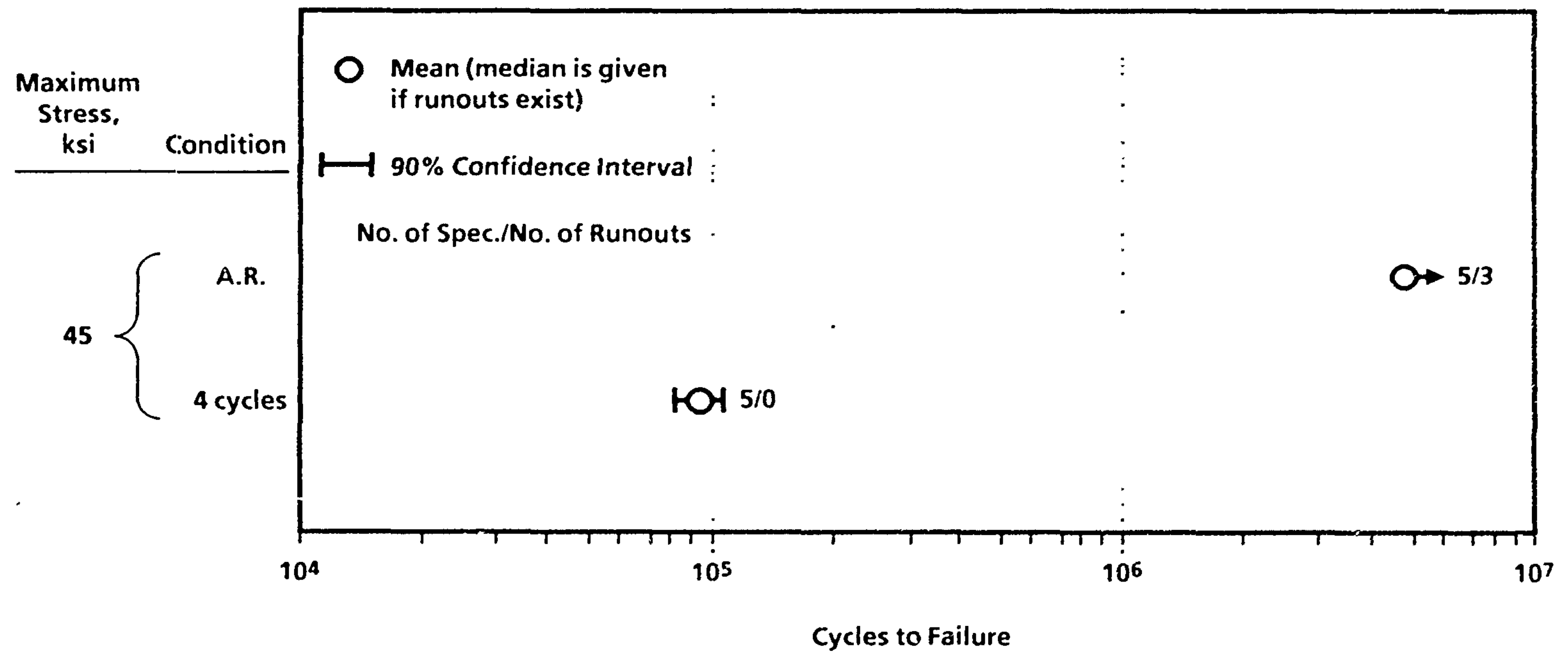


FIGURE 16. 2024-T3 BARE ALUMINUM FATIGUE RESULTS, $t = 0.032$ in., $R = +0.3$

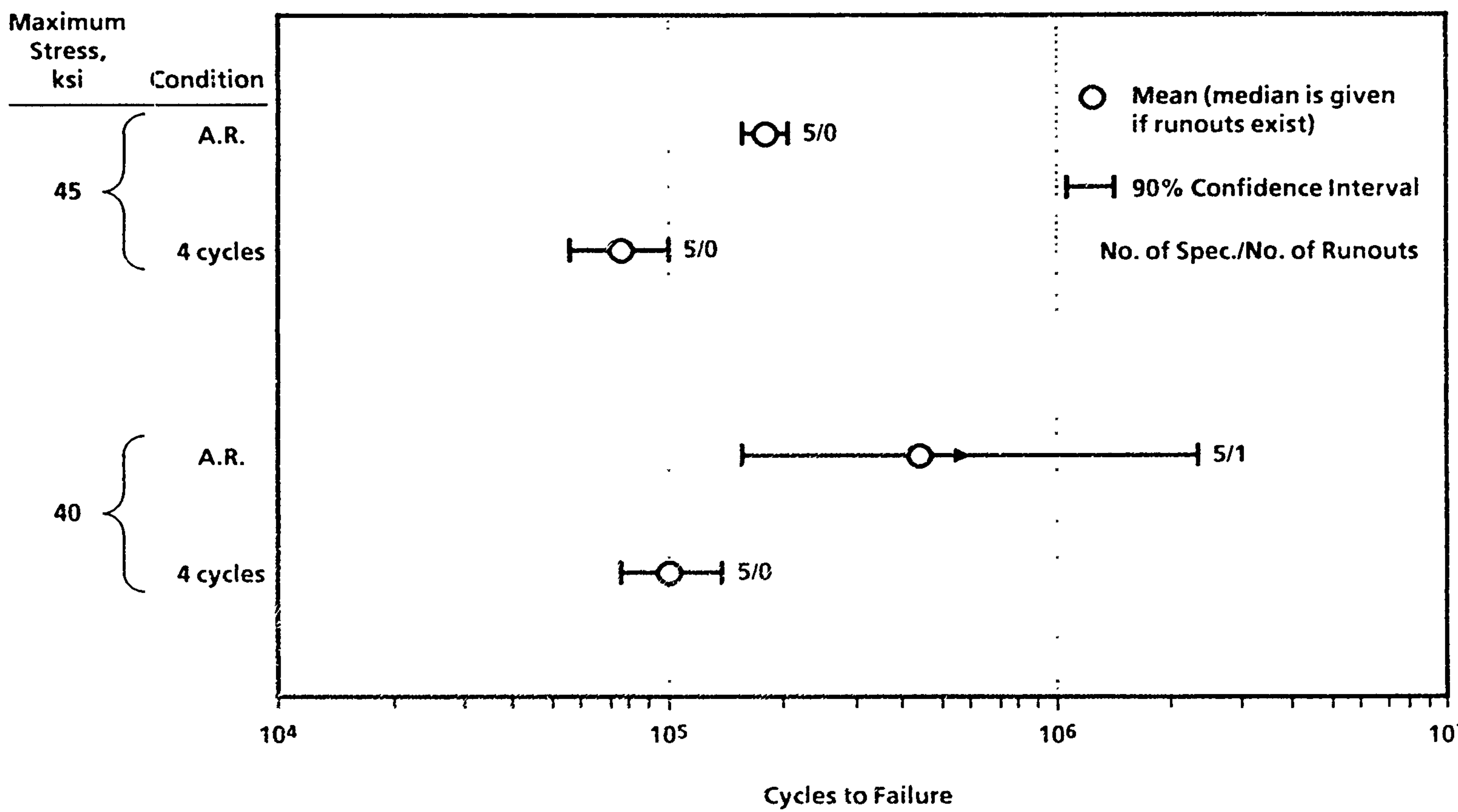


FIGURE 17. 2024-T3 BARE ALUMINUM FATIGUE RESULTS, t = 0.063 in., R = +0.1

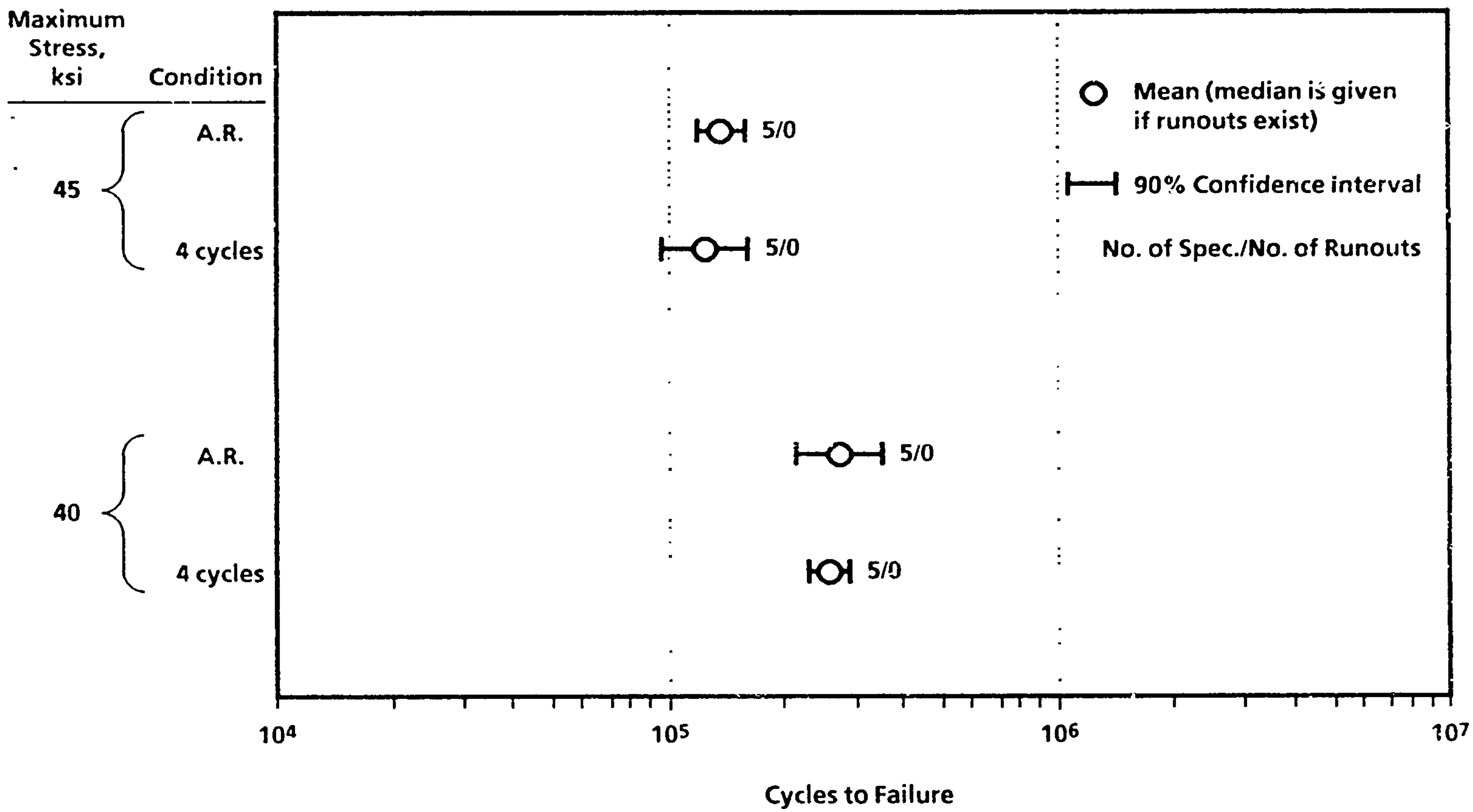


FIGURE 18. 2024-T3 BARE ALUMINUM FATIGUE RESULTS, $t = 0.190$ in., $R = +0.1$

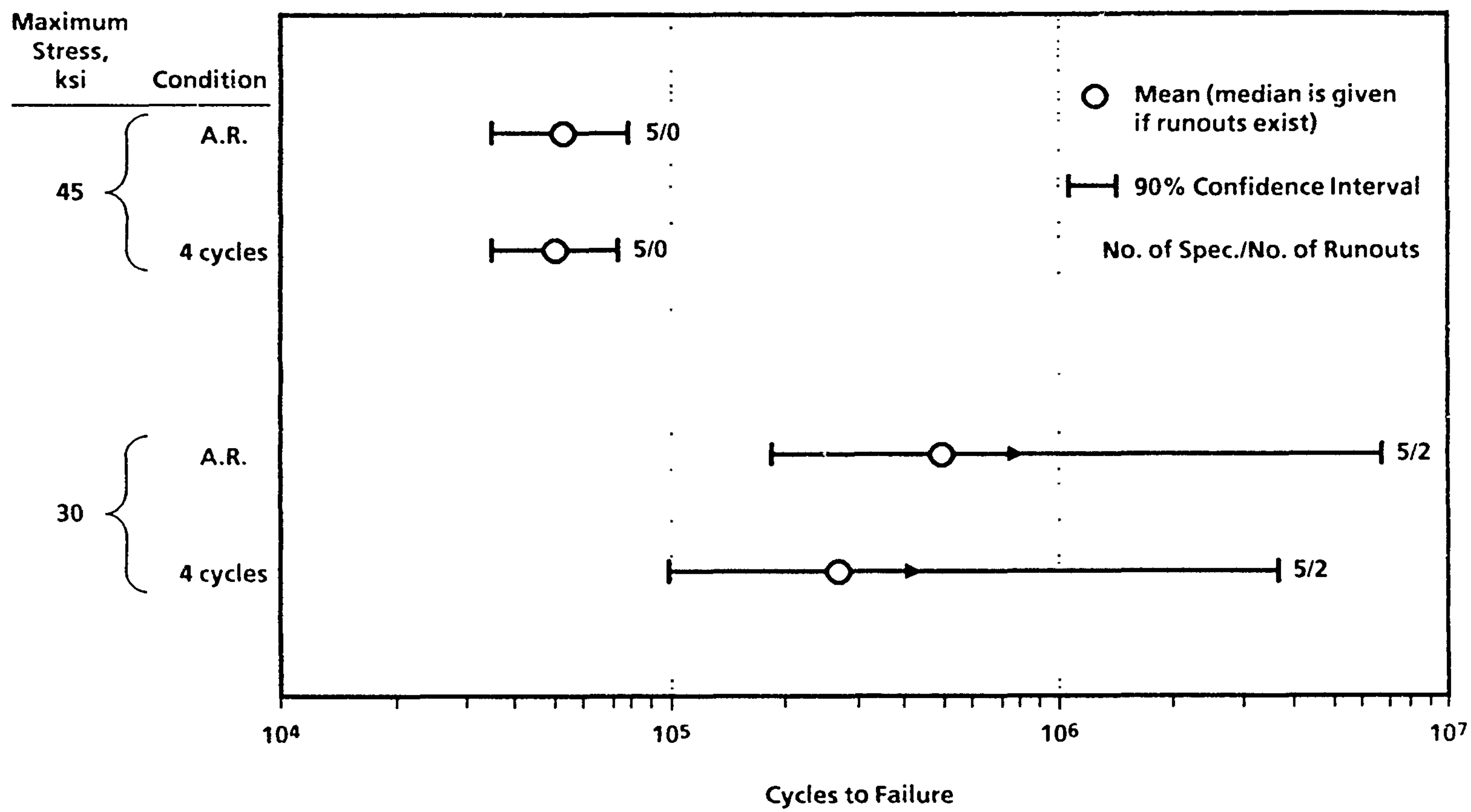


FIGURE 19. 2024-T81 BARE ALUMINUM FATIGUE RESULTS, t = 0.080 in., R = +0.1

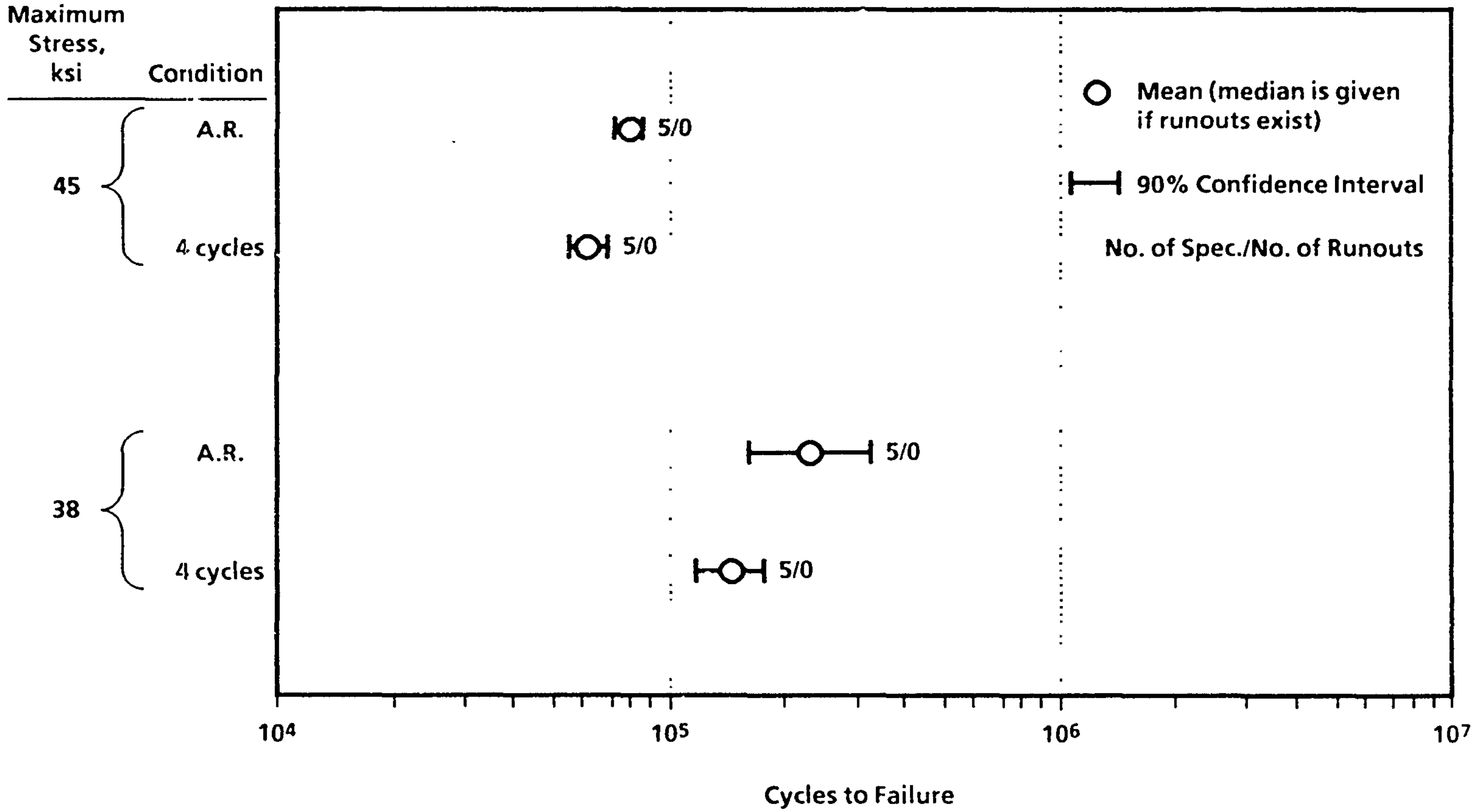


FIGURE 20. 2219-T81 BARE ALUMINUM FATIGUE RESULTS, t = 0.063 in., R = +0.1

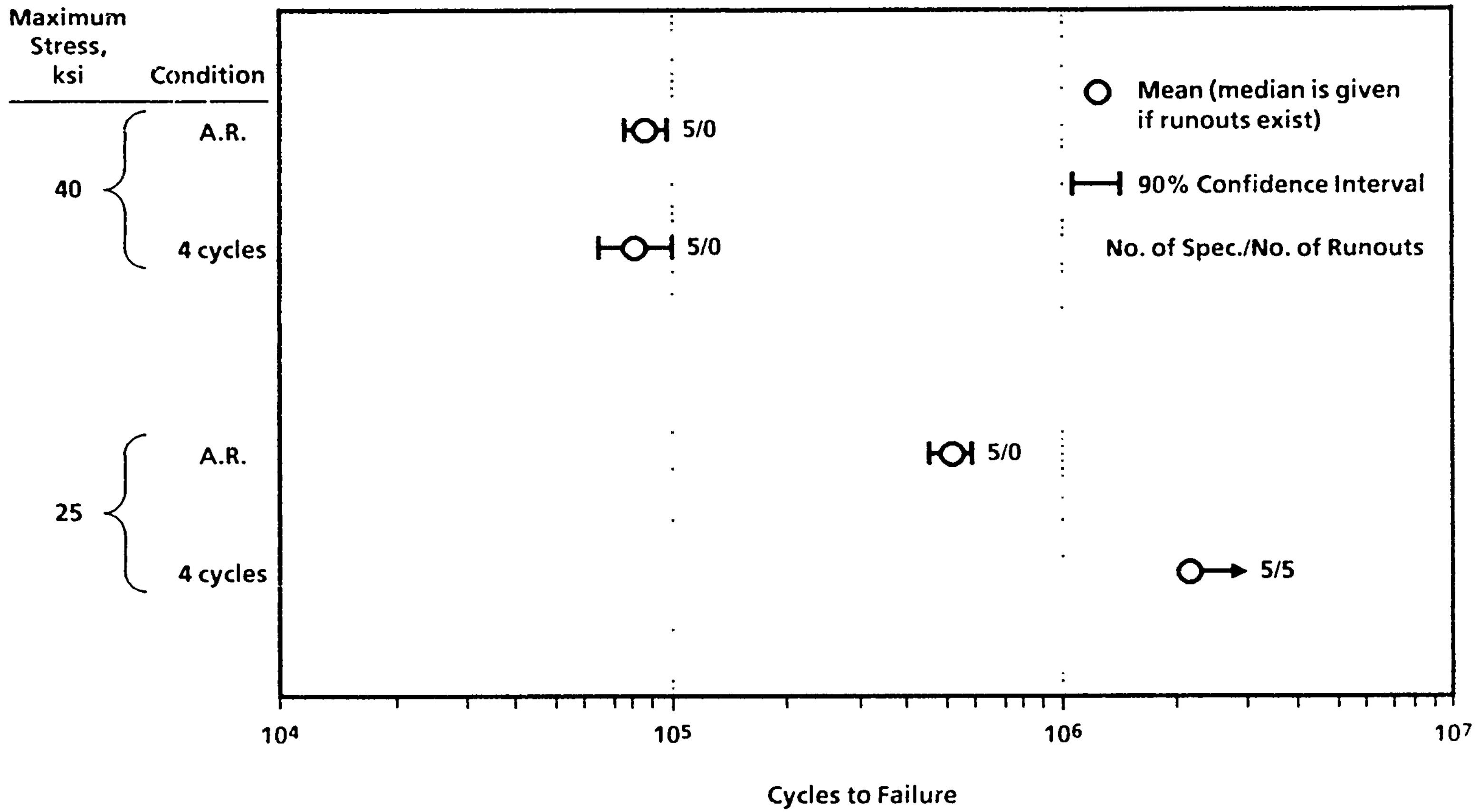


FIGURE 21. 7475-T761 ONE SIDE CLAD ALUMINUM FATIGUE RESULTS, $t = 0.071$ in., $R = +0.1$

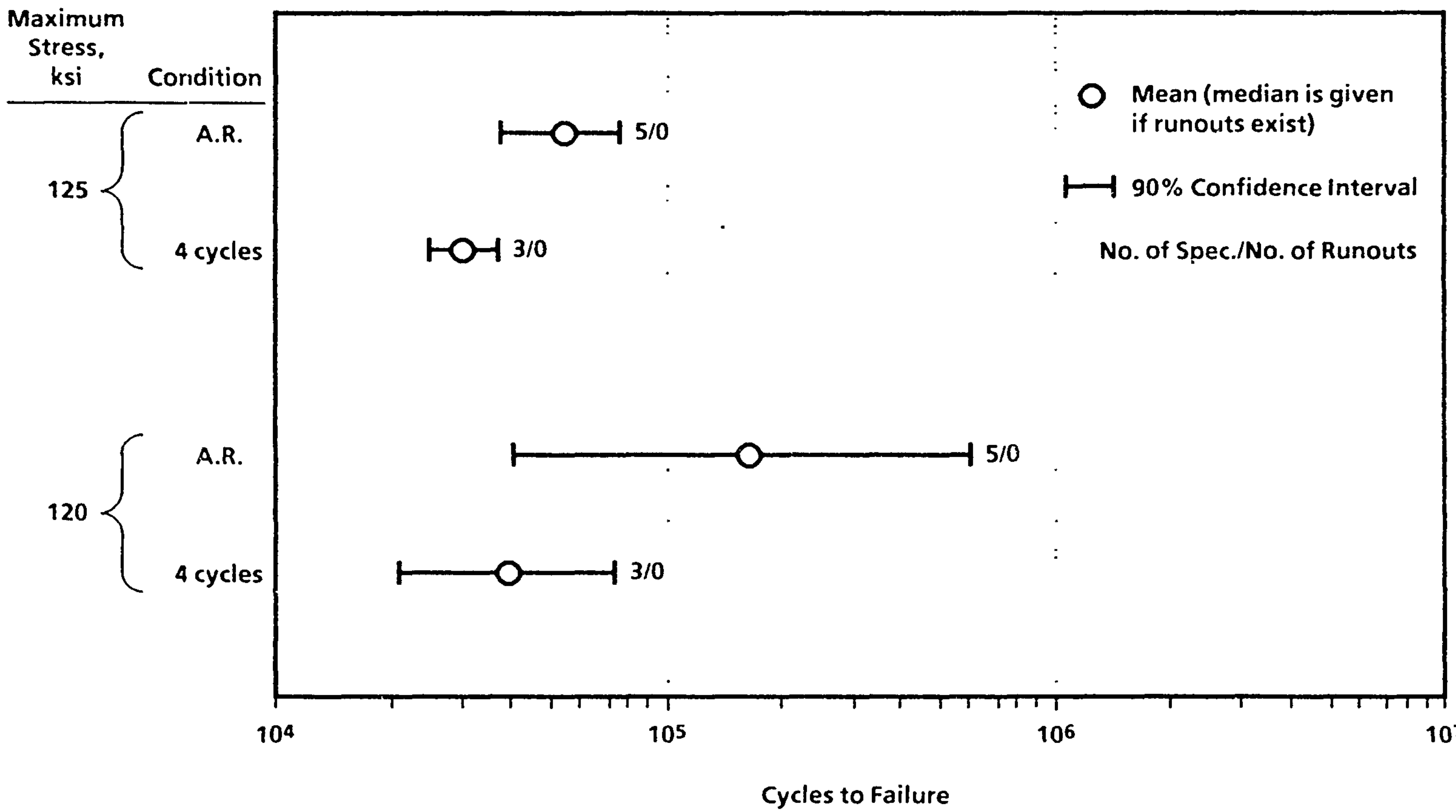
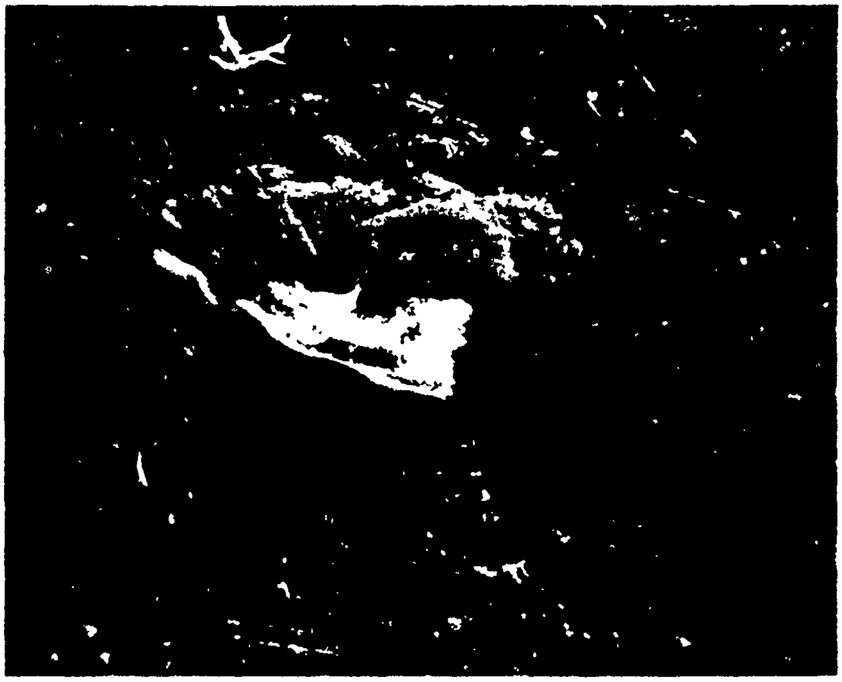
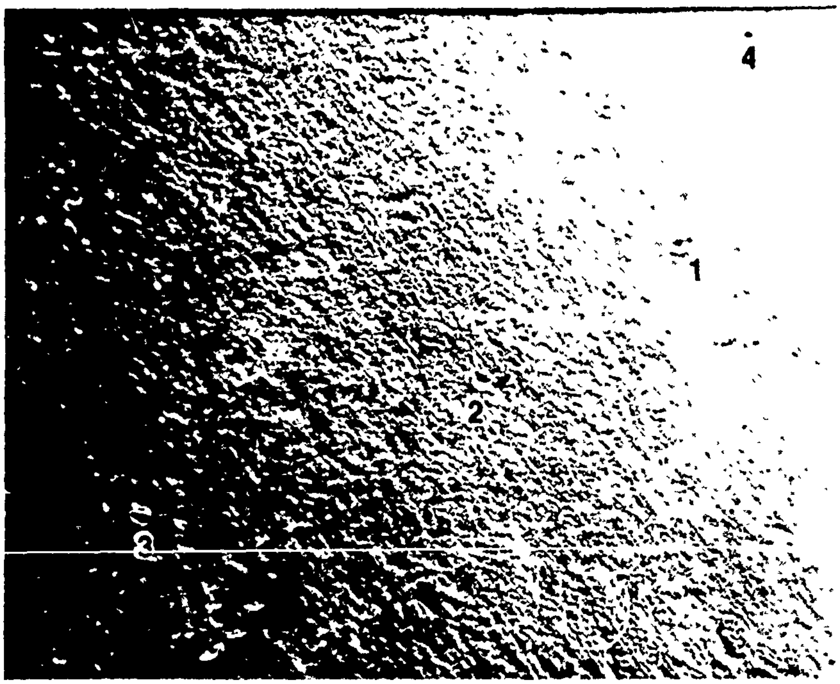


FIGURE 22. Ti-6A1-4V FATIGUE RESULTS, t = 0.063 in., R = +0.1



400X

Figure 23 - Typical Surface Defect at a 400X Magnification
(Site Number 3 in Figure 24)



15X

Figure 24 - Typical Surface Defects After Ogden ALC PBB

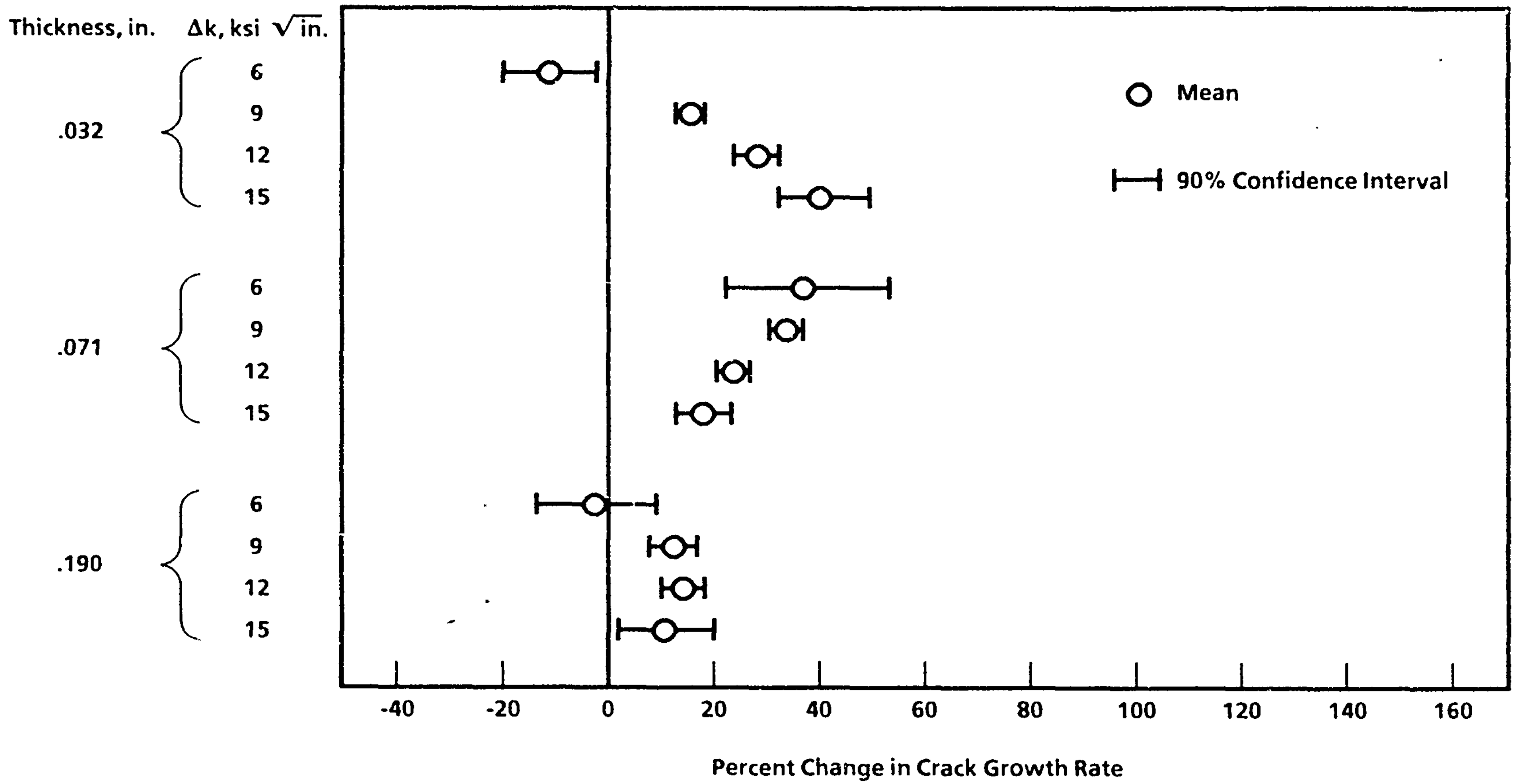


FIGURE 25. 7075-T6 BARE ALUMINUM PERCENT CHANGE IN CRACK GROWTH RATE DUE TO FOUR BEAD BLAST CYCLES

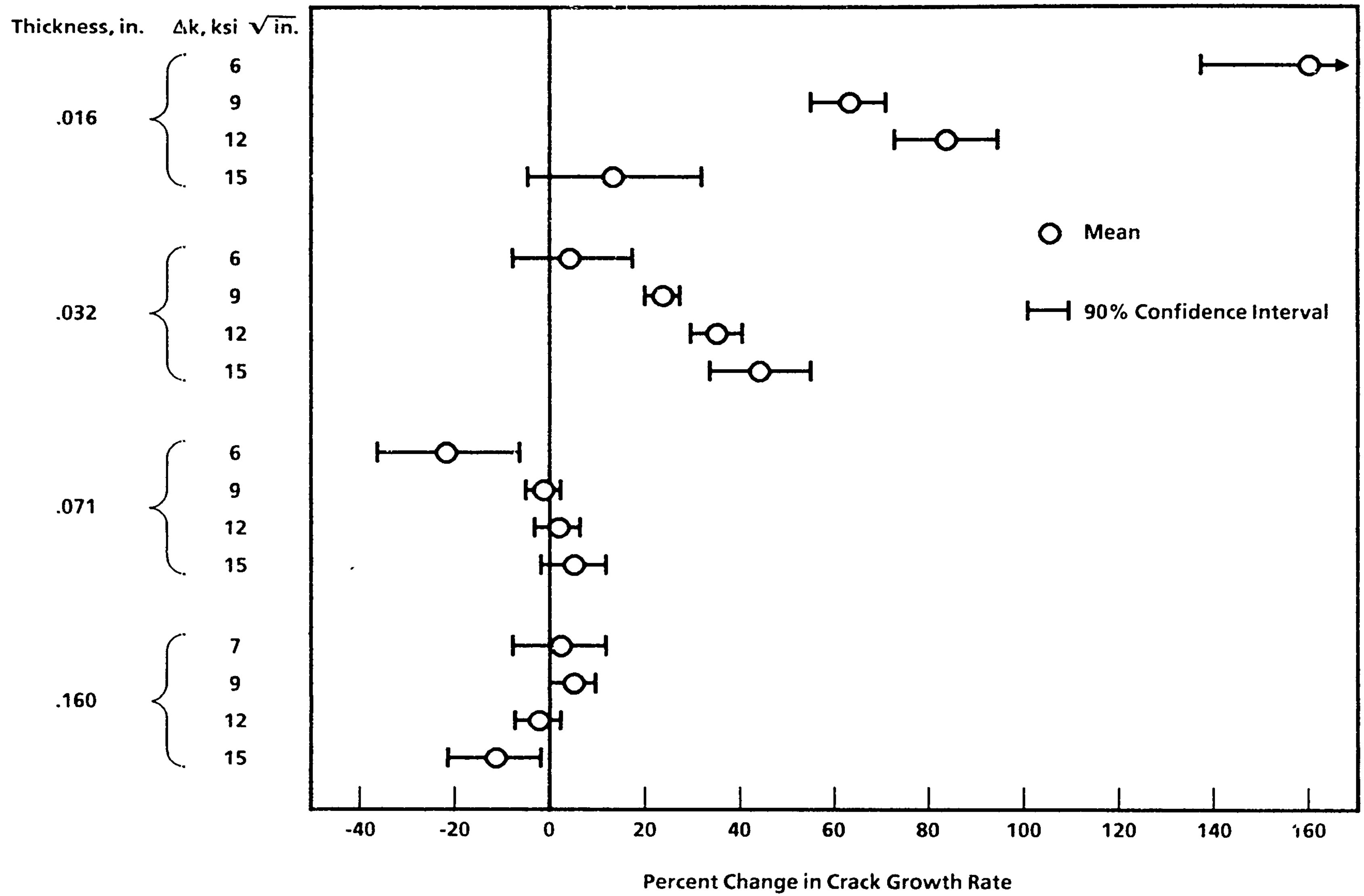


FIGURE 26. 7075-T6 CLAD ALUMINUM PERCENT CHANGE IN CRACK GROWTH RATE DUE TO FOUR BEAD BLAST CYCLES

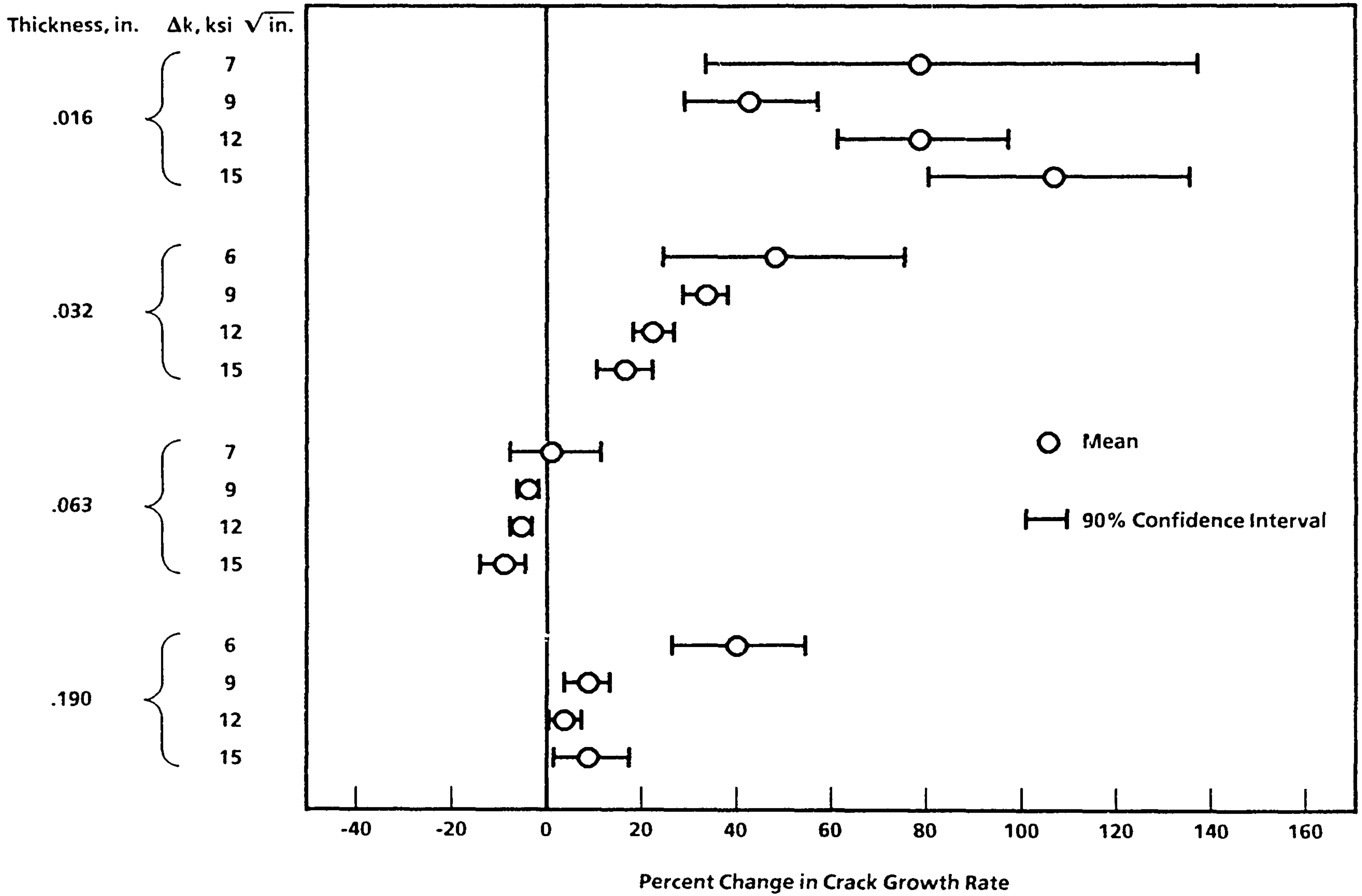


FIGURE 27. 2024-T3 BARE ALUMINUM PERCENT CHANGE IN CRACK GROWTH RATE DUE TO FOUR BEAD BLAST CYCLES

In-Service Inpsection Of Composite Aircraft Components

By

F. H. Chang, J. R. Bell
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IN-SERVICE INSPECTION OF COMPOSITE AIRCRAFT COMPONENTS*#

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General Dynamics/Fort Worth Division

ABSTRACT

Composite aircraft components are increasingly being deployed in Department of Defense aircraft inventory. The demand for in-service inspection of these components precipated the need for the development of computerized ultrasonic inspection systems specifically designed for advanced composite materials. The In-Service Inspection System (ISIS) is such a system to serve as a prototype. The microprocessor-based semi-automated ultrasonic system provides C-scan recordings of the inspection results for manual inspection. System hardware was designed in a modular format. Inspection schemes and algorithm were developed for the inspection of composite laminates, adhesively bonded structures, and honeycomb-core structures. System implementation and inspection algorithm were evaluated in several field applications. Applications to repaired honeycomb-core structures were also developed as an extension of the system specifications.

The modular architecture of ISIS is considered an important feature for an in-service system. The portability versus transportability issue remains a questionable item. System maintainability and reliability are pre-requisite for test equipment of field and depot usage such as ISIS. Inspection algorithms for composites require a degree of sophistication for these complex material systems such that a delicate balance must be reached between inspection efficiency and ease in system application. Data acquisition and analysis aspects of the system must retain a link among production inspection data, in-service inspection records, and repair information/post repair inspection results. Integration of these needs in the inspection system will fulfill the requirements of field/depot service inspection.

*Work supported by the Manufacturing Division of the Materials Laboratory of Air Force Wright Aeronautical Laboratory under Contract No. F33615-78-C-5152.

Manufacturing Technology For Nondestructive Evaluation (NDE) System To Implement Retirement For Cause (RFC) Procedures For Gas Turbine Engine Components

By

Systems Research Laboratories, Inc.

MANUFACTURING TECHNOLOGY FOR NONDESTRUCTIVE EVALUATION (NDE) SYSTEM TO IMPLEMENT RETIREMENT FOR CAUSE (RFC) PROCEDURES FOR GAS TURBINE ENGINE COMPONENTS

USAF Contract No. F33615-81-C-5002

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INTRODUCTION

Systems Research Laboratories, Inc., with a team of subcontractors, has developed an automated NDE inspection system to detect 0.005 x 0.010 inch surface flaws and 0.015-inch inclusions in jet engine rotary parts. The system implements the USAF Retirement for Cause philosophy in which good, used engine parts are returned to service, and flawed parts are "retired for cause." The system also provides current capability for the USAF Engine Structural Integrity Program (ENSIP), based on anticipated ENSIP inspection requirements. Emphasis has been placed on improving flaw detection and characterization by using computer algorithms and removing the human decision-making process.

A typical RFC/NDE Inspection System consists of an operator console, a system computer, and eddy current and ultrasonic inspection stations. The operator console is used to monitor the system's operational status, track inspection status at each NDE station, and generate inspection data reports. The system computer performs advanced data processing, system-wide communication and sophisticated, high-speed mathematical and scientific data analyses critical to the inspection process. The NDE inspection stations perform the automated part inspections, flaw detection and signal preprocessing activities.

Phase I Acceptance Tests were successfully conducted on the core inspection system under simulated production conditions at SRL in October 1985. The system demonstrated reliable detection of 5 x 10 mil flaws and exceeded overall operational test objectives by demonstrating a >90% production "uptime." Variability tests showed the system to be virtually unaffected by operator changes, probe changes, loading procedure and flaw orientation. The core system was shipped to the Kelly Air Force Base Engine Overhaul Facility on November 27, 1985 and was operational within a week after delivery.

Phase II Acceptance Tests examined flaw detection capability and reliability under production conditions at Kelly AFB. Tests included automatic scans of engine disks and a statistically significant number of representative fatigue-cracked test specimens. Rivet hole inspection data showed a 90/95% level of confidence at the 4-mil crack depth range, which exceeded Phase I results. Bolt hole and flat surface data indicated reliable detection in the desired 5- to 10-mil depth range. A strong correlation between apparent-versus-actual flaw depth data was seen in all test data. Ultrasonic inspection data were similarly encouraging.

The RFC/NDE Inspection System began production inspections on selected F100 engine disks in October 1986, and has been in daily operation since then.

SYSTEM DESCRIPTION

The RFC/NDE Inspection System was designed per structured analysis methodology. A typical system (Figure 1) includes an Operator Console, a System Computer, Eddy Current Inspection Stations, and Ultrasonic Inspection Stations.

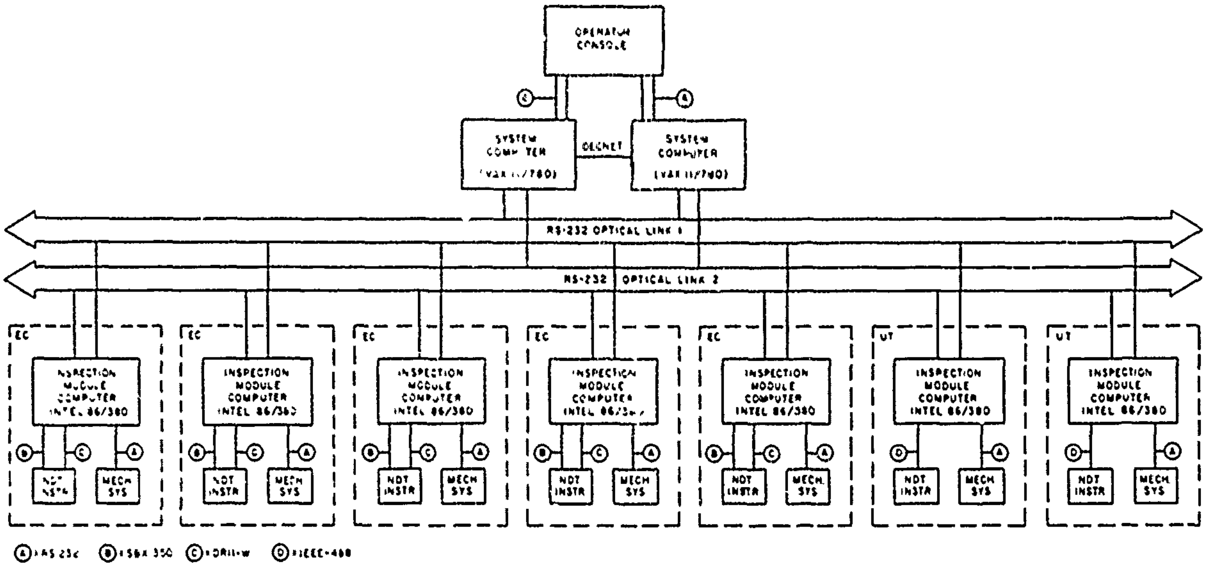

Figure 1. Typical RFC/NDE Inspection System

The Operator Console (Figure 2) is a passive station used to monitor overall system operational status, track the individual part inspections at each NDE Station, and generate inspection data reports.

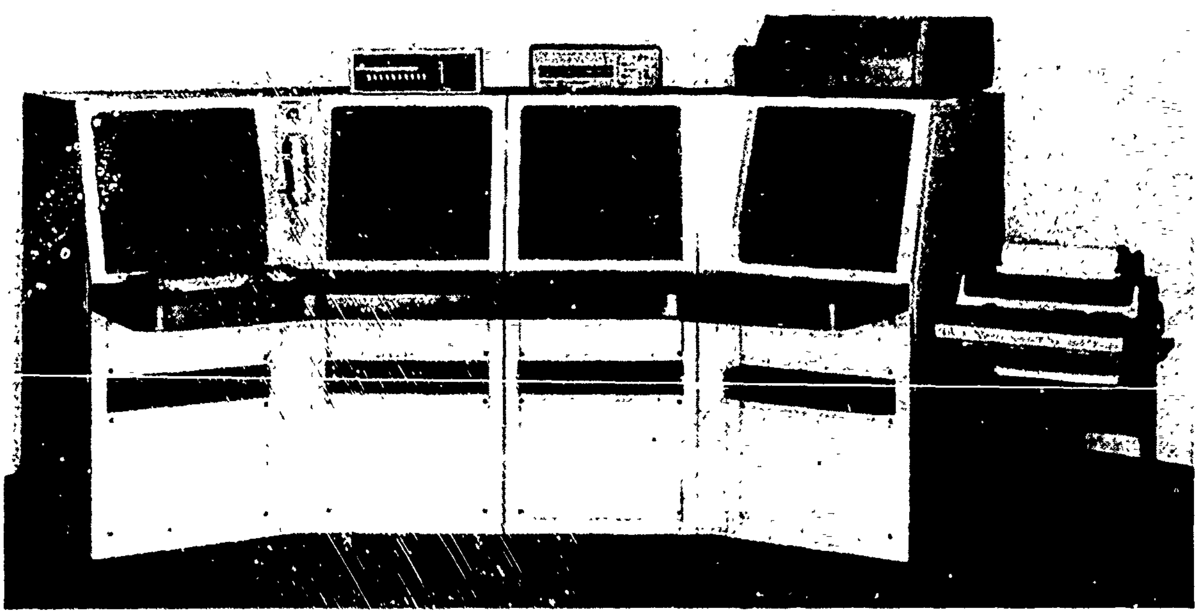

Figure 2. Operator Console.

The console has 4 color CRTs functionally dedicated as follows: The Main Menu is a command-driven display used to generate inspection data reports and graphic displays from previous inspections; the Defect Graphics Display provides color images of inspection results as they are generated at the NDE Inspection Stations. This display contains top and side views of the engine part with color-coded flaw indications, and specific summary data such as relative flaw sizes and locations, inspection start/stop times, the cumulative number of part inspections, date of inspection, accept/reject status, engine cycle number, and scan plan version. The Diagnostic Display provides a continuously updated operational status display of the inspection system, including the communication networks. The Part Tracking Display shows the current status of the part inspections at each NDE Inspection Station.

The Operator Console also has an intercom system for communication with the NDE Inspection Stations; a color printer for hardcopy graphic printouts; and a letter printer for inspection data reports. The Operator Console's software resides on the System Computer. All communication between the System Computer and the Operator Console is via an RS232 interface.

Typically, a dual VAX 11/780 configuration functions as the System Computer. This dual configuration (Figure 3) provides the central intelligence for the entire RFC/NDE Inspection System, and meets redundancy and back-up requirements. The System Computer performs advanced data processing, system-wide communications, and sophisticated, high-speed mathematical and scientific data analysis critical to the inspection process. The primary tasks of the System Computer are part tracking, Operator Console/NDE Inspection Station interface and communication, NDE data cross-correlation, archival data base storage, system diagnostics, advanced signal processing, RFC proprietary data analysis, and graphics processing.

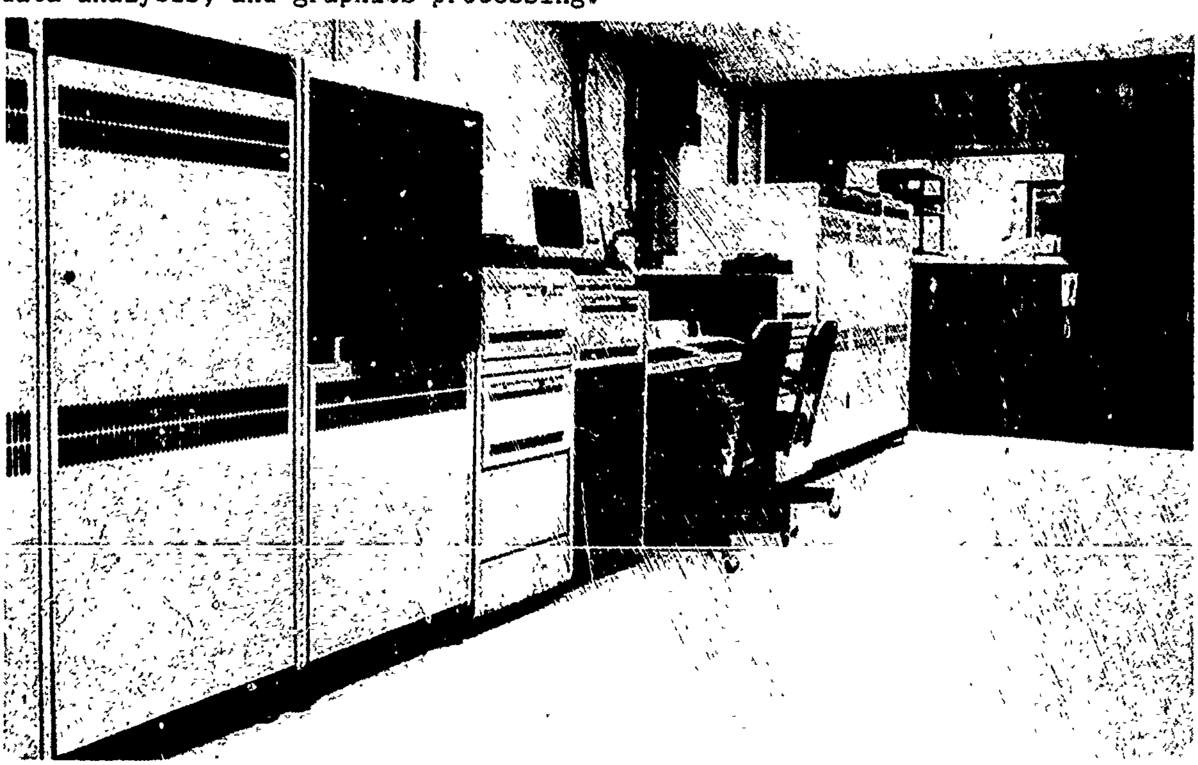

Figure 3. Dual VAX 11/780 System Computer Configuration.

Each VAX combines a 32-bit architecture, efficient memory management, and a virtual memory operating system to provide essentially unlimited program space. The VAX/VMS virtual memory operating system provides the multiuser, multiprogramming environment critical to the RFC System's application. In addition, the VAX floating-point instructions and accelerators, efficient scheduler, and FORTRAN-77 programming language are ideal for the System's real-time and scientific computational environments.

The combined disk space for the dual-VAX configuration totals 1,446 MB. The permanent on-line storage space is dedicated to storing all archival data, RFC-application software, and engine manufacturer proprietary data, thus ensuring ready access and increased software security. The removable disks are allocated for part-specific scan plan software storage thus providing easy software updates and file expansion.

Dual RS232 optical links provide communications between the System Computer and the NDE Inspection Stations. This dual structure provides redundancy, high-speed data transfer, flexibility, and the capability to map functional elements around failed components. The two VAX computers are directly interfaced with DECNET.

The RFC/NDE Inspection System employs both eddy current and ultrasonic inspection techniques. Ultrasonic "squirter" technology is used to detect volumetric flaws and voids. Eddy current is used for surface flaw detection. Each eddy current station (Figure 4) and ultrasonic station (Figure 5) consists of a mechanical assembly and an instrumentation cabinet.

Figure 4. Eddy Current Inspection Station.

Figure 5. Ultrasonic Inspection Station.

The eddy current and ultrasonic mechanical assemblies consist of an X-Y-Z axes mechanical manipulator (manufacturerd by M&M Precision Systems, Inc.), and subassemblies that provide up to seven axes of motion and automated operation.

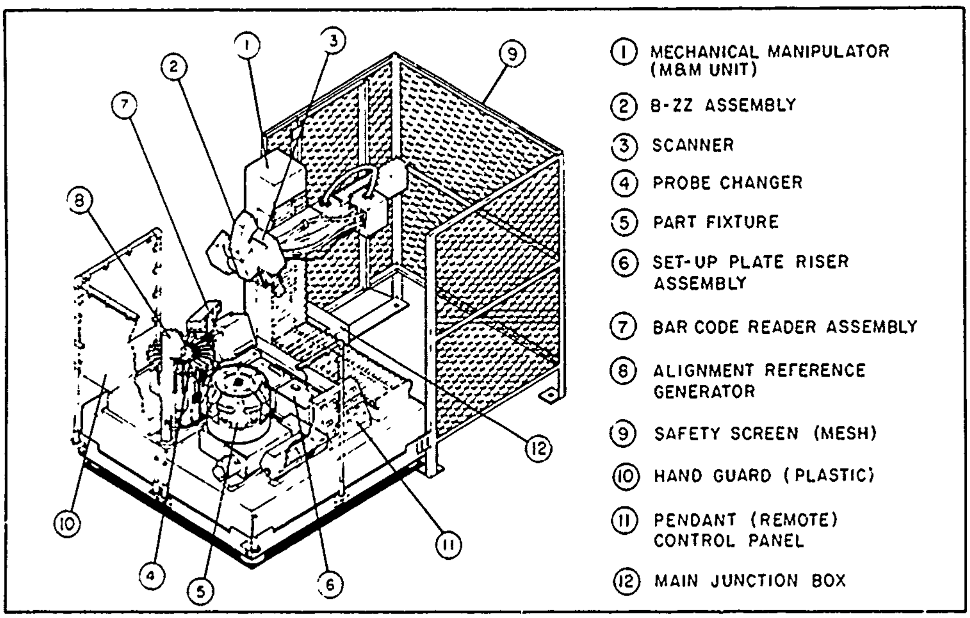

Figure 6. Eddy Current Mechanical Assembly.

The eddy current mechanical assembly (Figure 6) has seven primary subassemblies mounted directly to the base manipulator. A rotary scanner physically couples the probe to the mechanical manipulator and electronically couples the flaw signals to the eddy current instrument. The scanner has a drive mechanism (CC Axis) for rotating RECHII probes up to 1500 rpm; an air plenum to supply air to the air-bearing surface probes; and rotary transformers and specialized printed circuit cards for signal transfer and enhancement.

The calibration plate assembly consists of a metal riser upon which calibration plates with known flaws are mounted. The calibration plates are used to ensure that the eddy current instrument and probes are operating properly, before and after each geometry inspection.

The part fixture assembly holds and rotates an engine part during an inspection. The rotary table operates in either a continuous rotation mode (at 0 to 20 rpm) or an index mode. The pneumatically activated part fixture automatically clamps the part (on the outer or inner diameter, as required) to the rotary table.

The probe changer is a carousel assembly that holds up to 24 eddy current probes on its outer diameter. The carousel automatically indexes the correct probe into position for retrieval by the mechanical manipulator. The bar code reader scans each probe in the probe changer prior to the part inspection to ensure correct probe placement by the operator. A pneumatically activated rod pushes the probe into the scanner collet.

The alignment reference generator is a low-power laser that generates a thin reference beam across the engine part after the part has been placed on the part fixture. The operator uses the beam as a reference to properly align the part on the fixture.

The B-ZZ assembly provides the robotic wrist action for the eddy current mechanical manipulator. The B axis rotates in a sweep pattern parallel to the X axis and the ZZ axis provides linear thrust on the B radial.

The ultrasonic mechanical assembly (Figure 7) has six major subassemblies, two of which are identical to eddy current units (the part fixture assembly and the alignment reference generator). The other four subassemblies are unique to the

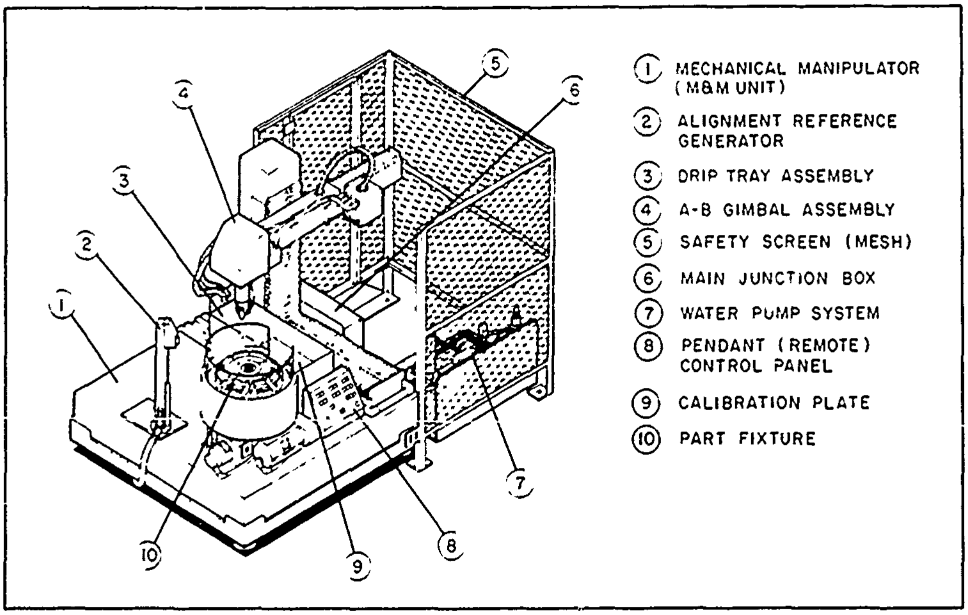

Figure 7. Ultrasonic Mechanical Assembly.

ultrasonic station: the calibration plate assembly, the water pump/filtration assembly, the splash guard/drip tray assembly, and the A-B axes gimbal assembly. The ultrasonic calibration plate assembly is mounted inside the splash pan assembly and is used for squirter/NDT instrument calibration prior to and after each part inspection. The block contains 4 interior voids of a known size and depth.

The water pump/filtration assembly circulates and cleans the water during the squirter inspection process. The splash guard/drip tray assembly fits around and under the rotary table and catches the water used during the part inspection and returns it to pump/filtration assembly.

The A-B axes gimbal assembly physically couples the squirter to the mechanical manipulator and provides the mechanical robotic wrist action for the ultrasonic stations. The A axis has a 60° range of motion and the B axis has 130°.

The instrumentation cabinets (Figure 8) are the inspection station operator's control units. Each cabinet assembly contains an inspection module computer, a visual alarm, a color CRT display, a pushbutton control panel and auxiliary keyboard, an intercom link to the Operator Console, the mechanical manipulator controller, and an NDE instrument.

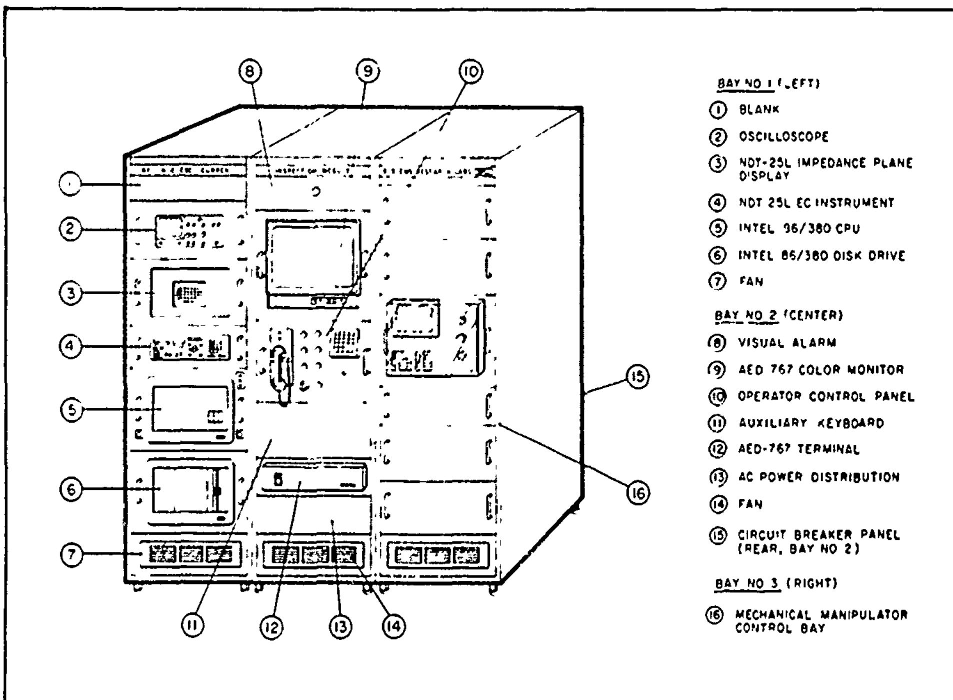

Figure 8. Eddy Current Instrumentation Cabinet.

The RFC System's eddy current and ultrasonic NDE instruments represent the most advanced computer-controlled instrumentation available. The eddy current instrument is the NORTEC NDT-25 with dual channel digital sampler and a frequency range of 10 KHz to 6 MHz. The SRL Model 1712A Computerized Ultrasonic Instrument (CUI) is computer interfaceable and contains a microprocessor-controlled square-wave pulser, high-speed digital sampler (55 MHz), and multibus-compatible receiver boards.

The Inspection Module Computers provide local intelligence and operator communications for the inspection stations. The computer is an Intel 86/380 microprocessor which contains multiple Intel microcomputer boards for individual processing functions (e.g., I/O, mechanical system scan control, and instrument control) and specialized instrument boards. The Intel 86/380 System features a 32 MB Winchester Disk Drive, 1 MB floppy disk, iRMX operating system, and 12 multibus card slots for specialized user functions. The Intel 86/380 features delineated functions, modularity, and component board plug-in capability to enhance modification, diagnosis, and repair of the Inspection Module Computer.

The RFC System's data acquisition module includes all probes and transducers used for flaw detection, adaptive positioning, dimensioning, and scanning. Advanced technology incorporated into this module includes Southwest Research Institute's air bearing probes and ultrasonic squirter, NORTEC's RECHII probes, and SRL's ultrasonic squirter.

CONCLUSION

The RFC/NDE contract was awarded to SRL in October 1981. During the first two years of the program, major effort was directed toward organizing and coordinating subcontractor program activities, establishing the core in-house project team, defining the system's performance criteria and specifications, and designing and fabricating a prototype inspection system.

In March 1984, the prototype system (operator console, one eddy current station, one ultrasonic station, one system computer) was successfully demonstrated to the Air Force and subcontractor participants. During 1984 and 1985, the RFC/NDE team performed a planned in-depth system evaluation and upgrade to meet production inspection performance specifications.

Phase I Qualification and Acceptance Tests were successfully conducted at SRL in the fall of 1985. The tests examined system reliability and NDE flaw detection capability/reliability under simulated production conditions. NDE tests included automatic scans of engine disks and enough representative fatigue-cracked specimens to yield a statistically significant prediction of detection reliability. The data indicated a 90/95% confidence level for detecting surface flaws in the 5-mil to 10-mil depth range in bolt holes and on flat surfaces. An encouraging aspect of the data was a strong correlation between the apparent-vs.-actual flaw depth data. Of particular interest were the variability test results, which showed the system to be virtually unaffected by operator changes, probe changes, part loading, flaw orientation, and repeated scans.

Phase II tests were conducted in the production environment of the Kelly Air Force Base Engine Overhaul Facility in July-August 1986. Preliminary Phase II test results were extremely encouraging. Data indicated a 90/95% confidence level for detecting surface flaws in the 4-mil depth range for rivet holes. Bolt hole and flat surface data again showed reliable detection in the desired 5-mil to 10-mil depth range. Based on these results, the RFC/NDE Inspection System began production inspections on selected F100 engine components in mid-October 1986, with no major implementation difficulties being experienced. Figure 9 shows the production RFC/NDE Inspection System at Kelly AFB.

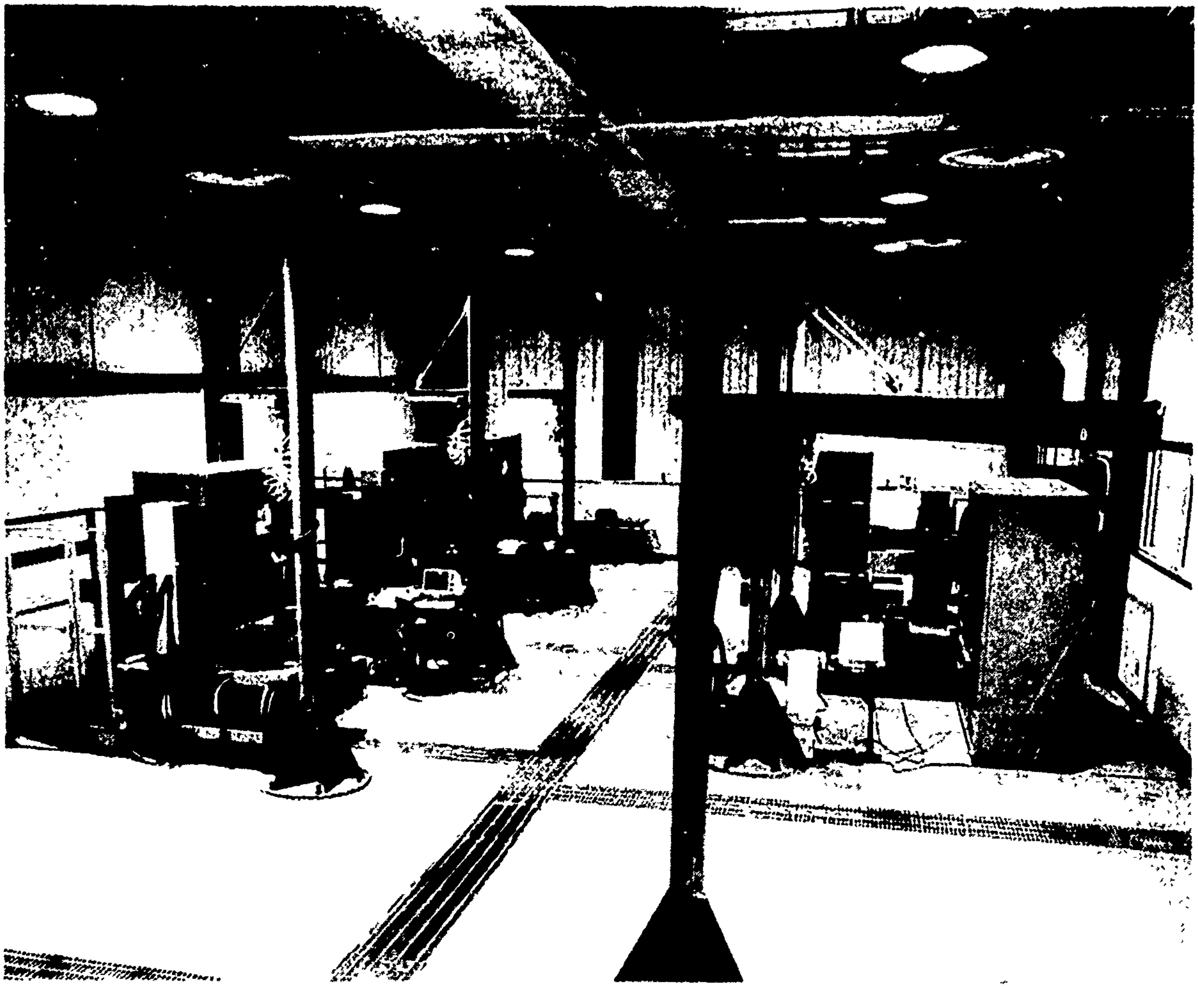

Figure 9. Production RFC/NDE System Installation.

FOURTH SESSION
ENSIP/ANALYSIS

Chairman

John W. Lincoln

ASD/ENFS

Certification Of Composites For Aircraft

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CERTIFICATION OF COMPOSITES FOR AIRCRAFT

John W. Lincoln*

The process for certification of composite structures for USAF aircraft has been evolving for approximately fifteen years. The technology for the certification process has been developing and is now mature enough to support the process. This paper describes how the Aircraft Structural Integrity Program (ASIP) is tailored to the specific requirements of composite structures.

INTRODUCTION

The establishment of the requirements for structural integrity of composite structure for an aircraft has long been a challenge for the certification authorities. This challenge is much greater when the aircraft is operated in an environment where heating of the structure is a factor. However, even for structures where heating does not appreciably affect the structural capability, there are some major considerations. One of these is the scatter in strength and fatigue data. This scatter, which is larger than observed in metals, is not a deficiency in composites but a fact that must be accounted for in the certification process. Another consideration is the difficulty in establishing the growth characteristics of manufacturing or service induced defects due to load application. This difficulty is due to the mathematical problems in simulating this growth and to the apparent inconsistent empirical results from presumably identical damage conditions. Still another consideration is the effect of low energy impact on thin laminates. This is a durability issue that should be considered in establishing requirements for composite structures.

There have been several efforts that have been aimed at addressing the issues related to composite certification. One of the contributions in this area was made by a TTCP (The Technical Cooperation Program) HAG-5 panel in 1983. This panel brought the major issues into focus and described some alternative approaches that could be used. Another contribution was a Navy sponsored effort by Northrop (Reference (1)). This work concentrated on approaches relating to reliability and made recommendations on probability distribution parameters that could be used for both strength and durability certification. In addition, the Air Force Wright Aeronautical Laboratories have sponsored numerous programs that have contributed to the understanding of composite behavior. Some of which that should be mentioned are the Fatigue Sensitivity program performed by Northrop, the Environmental Sensitivity of Composites program performed by Grumman, the Wing/Fuselage

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Critical Components program performed by Northrop, and the Damage Tolerance of Composites program performed by Boeing and Northrop.

PROCEDURES OF OTHER AGENCIES

The Federal Aviation Administration (FAA) has published their certification procedure for composite structures in The Advisory Circular 20-107A, 25 April 1984. They are currently placing the following interpretation on this document. For single load path structures they require the use of A basis allowables and for redundant load path structures they permit the use of B basis allowables. They accept the property data published in MIL-HDBK-17 for these allowables. Otherwise, it is incumbent on the manufacturer to develop the required data with the methods described in MIL-HDBK-17. They require the structure to survive a static test to ultimate after the structure has been impacted to the point of barely undetectable damage from a realistic threat. The effects of repeated loading and environmental exposure which may result in material property degration should be addressed in the evaluation of the static test results. They are allowing the static test to be performed at room temperature and dry when there is comparable test data for establishing a knockdown factor from room temperature dry components and from environmentally conditioned components. Also, when the contractor can compute the change in strain due to the environment by a finite element analysis supported by test data he may use this computation to interpret the results of a room temperature static test. Otherwise an environmentally conditioned static test is required. The only other major test required is a damage tolerance test of the structure.The movable control surfaces are included when their failure could result in catastrophic loss of the aircraft or result in a flutter condition. The structure is subjected to damage that is detectable by appropriate non destructive inspection techniques. The test article is cycled with this damage using the load amplification and/or life enhancement approach as described in Reference (1) to demonstrate a B basis life capability. The effects of humidity are accounted for in the cyclic portion of the test and the ensuing residual strength test results must account for both temperature and humidity. Care is taken to ensure that the test frequencies are not high enough to bias the results. Also, no truncation of high loads is allowed. During the cyclic portion of this test no significant damage growth is allowed. These damage tolerance rules apply to both Part 23 and Part 25 aircraft certification.

The U.S. Navy has completed the certification of composites in the F-18 and the AV-8B aircraft and are in the process of certifying the composites in the V-22 and the A-6 wing replacement. They have adapted the lower of a B basis allowable or a 85% of the mean for a strength allowable to be used in design. They require a component test program that includes environmentally conditioned static and fatigue test specimens. The fatigue test components as well as the full-scale fatigue test article is loaded with a severe (critical point in the sky) spectrum. The full-scale static test article and the fatigue test articles are not environmentally conditioned. They are currently requiring an environmental knockdown on the full scale static test results. The full-scale fatigue test article is cycled for two lifetimes of the severe usage. In recent procurements, the Navy has added requirements for damage tolerance. They require that the structure after being damaged to the point of being readily detectable with an external visual inspection have ultimate strength capability fully compensated for the knockdown described above. No growth of this damage is allowed from cyclic loading.

The U.S. Army is planning to use for design a B basis allowable that includes the effect of moisture and temperature. They are planning for a development test program with durability and damage tolerance testing for the helicopter airframe to be completed at the component level. They are also considering a full scale helicopter airframe static test to ultimate with the test article not environmentally conditioned. The helicopter rotor components are planned to be qualified with a durability test only.

USAF COMPOSITES CERTIFICATION

The USAF has qualified several composite structures for flight. Among these are the F-111 horizontal tail, the F-4 rudders, the A-7 outer wing, the F-16 empennage and the B-1 horizontal tail. Each of these structures was qualified for flight on an ad hoc basis. Consequently, there was little commonality in the qualification processes. For example, the A-7 outer panel was subjected to an environmentally conditioned durability test and the F-16 horizontal tail was subjected to a proof test to ensure its structural integrity.

In 1976 a paper was written (Reference (2)) by members of the Structures Division of the Aeronautical Systems Division (ASD/ENFS) that reflected the status of the certification process in the Air Force at that time. This paper painted a rather bleak picture mainly because the technology base for composites had not matured. The value of this paper was that it looked at the certification of composites in the light of the Aircraft Structural Integrity Program and it cited the need for the technology development required for certification of future aircraft.

By 1981 there had been sufficient progress in technology development that ASD/ENFS felt there should be an update of the position taken in the 1976 paper. This paper (Reference (3)) was significantly influenced by the Fatigue Sensitivity program and the Environmental Sensitivity of Advanced Composites program. In this paper a position was taken on the primary aspects of composites certification. However, this position was taken without the benefit of the results of the Wing/Fuselage Critical Components program and the Damage Tolerance of Composites program. Consequently, there are certain aspects of the 1981 version of the certification process that are inadequately defined. Now that these programs are essentially complete amd since the Air Force anticipates near term development of aircraft with extensive use of composites, it is believed timely to update the 1981 position.

Past experience in the development of the certification process has shown that the Aircraft Structural Integrity Program (ASIP), as defined in MIL-STD-1530A and required by Air Force Regulation 80-13, is flexible enough to be used for composite structures. The major differences between the applications for metal aircraft components and composite aircraft components is a change of emphasis in several of the elements of ASIP. The five major tasks that comprise ASIP are:

I. Design Information
II. Design Analyses and Development Tests
III. Full-Scale Testing
IV. Force Management Data Package
V. Force Management

In each of these major tasks, there are elements that are appropriate to the task heading.

Within Task I there are several aspects relating to the composites design that must be addressed. In the area of structural design criteria the requirements for strength, damage tolerance, durability, flutter, vibration, sonic fatigue and weapons effects must be defined for both the metal and composite structural elements. For composites, particular emphasis must be placed on the issue of battle damage from weapons since the containment of this damage may well dictate the design configuration. In addition to a composite design that can contain weapons damage, the design must also be repairable from that damage to maintain operation readiness. The composite structure must also be designed to be easily repairable for expected in-service damage. Another Task I effort that must considered carefully is the selection of the design usage. The design missions must be adequately defined such that the potentially damaging high load cases are properly represented.

In Task II of ASIP, there is an element for establishing material allowables that needs to be discussed. The MIL-A-87221 requirement for metallic structure is that components other than multiple load path structure which are subjected to structural tests shall use A basis or S basis allowables. Those tested multiple load path structures may use B basis allowables. B basis allowables will be accepted for the design of all structurally tested composite strucures. However, these allowables must be established including the effects of temperature and moisture. The temperatures will be derived from the design operational envelope of the aircraft and the moisture conditions ranging from dry to the end of lifetime condition expected from a basing scenario that is representative of the worst expected moisture exposure.

The allowable for a given flight condition will be based on the temperature appropriate for that flight condition combined with the most critical of the range of possible moisture conditions. The factor of safety to be used in the application of the allowables derived above is 1.5. Since the strength of a composite structure is inherently dependent on the lay up of the laminate, geometry and type of loading, the B basis allowable must include these factors. However, the cost of a test program involving the number of complex components necessary to determine the B basis allowable could be prohibitive. An acceptable approach would be to determine a B basis allowable from coupon data generally representative of lay up and loading.

This B basis allowable divided by the mean strength of the coupons used for the B basis allowable calculation would be the fraction of the strength allowed when interpreting the results of single complex component tests. It was shown by Kan in Reference (1) that the scatter in strength of composite structural is greater than that exhibited by metal structure. This was quantified by a Weibull shape parameter for composite strength of approximately 20. The Weibull shape parameter for aluminum structure is somewhat larger indicating less scatter. It appears that a shape parameter of approximately 25 is representative of the aluminum materials. The question of the impact on safety from the composite strength variability may be addressed as follows. Suppose that the once per lifetime loading on an aircraft is defined as limit load. This is the most unconservative assumption possible based on the requirements of MIL-A-87221. Further, suppose that the stress cumulative probability is characterized by a Weibull shape of six. This appears to be reasonable since it is generally representative of measured stress exceedance data. Further, from an examination of measured flight data the exceedance of ultimate load is approximately 1000 times less likely than the exceedances of limit load. Therefore, the cumulative probability of exceeding a given load in one lifetime was defined by a Weibull distribution function with a shape parameter of six and a 0.001 probability of exceeding ultimate load. For the case of undamaged structure, the joint probility density funtion for stress can be combined with the probability density function for strength (see Figure 1) to determine the probability of failure for a single aircraft in its lifetime. This is accomplished by forming the joint probability density function for load and strength from the product of the probability density function for load and the probability density function for strength. The probility of failure is the volume under this joint density function that is over the region where the load exceeds the strength. This computation yields a failure probability of 1.5×10^{-3}. The same type of computation for the aluminum structure yields a failure probability of 1.0×10^{-3}. Although this calculation shows that the aluminum may be less of a risk, the absolute risk levels in both cases is more than two orders of magnitude less than the failure rate from all sources and therefore is judged to be acceptable.

The AFWAL programs alluded to earlier have demonstrated that composite structures are relatively insensitive to low cycle fatigue loading for the low stress cycles, but much more damaged by the high stress cycles. Unfortunately, the data base from which the high stress cycles for a new aircraft are derived is somewhat meager. Consequently, there will have to be extreme care used in defining the design usage element of Task I.

As for metal structures, the strength, durability and damage tolerance analyses elements in Task II for composites are inexorably linked to the design development tests element also in Task II. For support of all three of these analyses it is envisioned that the design development testing will consist of "building blocks" ranging from coupons to elements, to subcomponents and finally components. These building block tests must include room temperature dry laminates. Also, if the effects of the environment are significant, then environmentally conditioned tests must be performed at each level in the building block process. The test articles are to be strain gaged adequately to obtain data on potentially critical locations and for correlation with the full scale static test, and in addition, the test program is to be performed so that environmentally induced failure modes (if any) are discovered. The design development tests are complete when the failure modes have been identified, the critical failure modes in the component tests are judged to be not significantly affected by the non representative portion of the test structure and the structural sizing is judged to be adequate to meet the design requirements. For static test components, this judgement is based on adjusting the failure loads to the B basis environmentally conditioned allowable.

For durability test components, the success criteria is somewhat more complicated by the relatively large scatter in fatigue test results and the potential of fatigue damage from large spectrum loads. It has been demonstrated, however, that the durability performance of composites is generally excellent when the structure is adequate to meet its strength requirements. Therefore, the thrust of the durability test must be to locate detrimental stress concentration areas that were not found in the static tests. An acceptable way to achieve this goal is to test the durability components to two lifetimes with a spectrum that is expected to be the upper bound of loading for the aircraft. One possibility of acquiring this spectrum is to use the "worst point in the sky" approach that has been used extensively by the U.S. Navy. When the effects are judged to be significant, the durability tests for design development tests will be moisture conditioned. In addition to the testing performed to the baseline design usage spectrum, testing will be performed to determine sensitivity to potential usage changes. Also, it is evident from the approach described above that separate tests may be required for the metallic and mixed metallic and composite structural parts.

Composite structures as well as metal structures must be designed to minimize the economic burden of repairing damage from low energy impacts such as tool drops, etc. To accomplish this goal, the structure is to be divided into two types of regions. The first type is one where there is a relatively high likelihood of damage from maintenance or other sources. The second type of region is one where there is a relatively low probability of the structure being damaged in service. The specific requirements for these two areas are given in Table I. There are two other threats to the structure that may cause an economic burden. These threats are hail damage to parked aircraft and runway debris damage to aircraft from ground operations. The hailstone size for which the structure must be hardened was chosen such that this size or smaller was representative of 90 percent of the hailstorms. The runway debris size was also chosen to be include most of the objects potentially damaging objects found in ground operations. The velocity of these objects is dependent on the weapon system. The details of the hail and runway debris requirements are shown in Table II. The loading spectrum and environmental conditioning for the testing associated with the Table I and Table II requirements will be the same as that described above for the durability tests.

In addition to the threats described above, the safety of flight structure must be designed to meet other damage threats. These threats are those associated with manufacturing and in-service damage from normal usage and battle damage. The non-battle damage sources are described in Table III for manufacturing initial flaws and in-service damage. These flaws are discussed more fully in the draft damage tolerance requirements discussed in Reference (4). The design development tests to demonstrate that the structure can tolerate these defects for its design life without in-service inspections will utilize the upper bound spectrum loading and the environmental conditioning developed for the durability tests. These two lifetime tests will need to show with high confidence that the flawed structure will meet the residual strength requirements in Table IV. These residual strength requirements are the same for the metallic structures. To obtain the desired high confidence in the composite components it may be necessary to show that the growth of the initial flaws is insignificant. As for the durability tests there will be a program to assess the sensitivity to changes in the baseline design usage spectrum.

For many composite structures, the damage tolerance requirements will determine the allowable strain. However, the battle damage requirements are likely to influence the composite structure arrangement. For example, the need to contain battle damage to prevent catastrophic loss of the aircraft may well dictate the use of fastener systems and/or softening strips. The battle damage threat must be examined in the initial phase of the design. A fall out capability for battle damage based on configurations that meet all other requirements may not be adequate.

Task III of ASIP is composed of the full-scale testing elements. There will normally be a full-scale durability and damage tolerance test in the development of a weapon system, however, these tests will generally be for the verification of the metal structure. In the cases where the metallic structure durability and damage tolerance capability can be confidently established in the design development tests then the full-scale durability and damage tolerance tests may not be required. For example, a structure that is primarily composite, but contains a limited number of metallic joints may fall in this category. Normally, the durability and damage tolerance capability of the composite structure can be verified by the design development tests of Task II. The full-scale static test, however, is essential for the verification of the composite structure. This test is, of course, also essential for the verification of metallic structure. This test to ultimate may be performed without environmental conditioning only if the design development tests demonstrate that a critical failure mode is not introduced by the environmental conditioning. To provide assurance that the component static tests were representative of the component tests, these articles must be extensively strain gaged. A test of this structure to failure will be a program option. If the failure mode criterion above cannot be met, then the static test article must be environmentally conditioned.

With one exception it is expected that the tracking program requirements in Task IV of ASIP will change very little for the composite portion of the aircraft. Since the composites may be critical for the severe loading cases then care must be exercised that these high level occurrences are properly recorded.

SUMMARY

The Aircraft Structural Integrity Program may be easily tailored to provide the essentials of a certification program for composite structures. It is believed that the program described will provide a structure that is safe and economical to operate. Further, it is believed that the effort involved in the certification process for composites is approximately the same as a metals certification program. The specific features of the USAF program have been strongly motivated by the AFWAL programs and experiences from military and commercial applications of this technology. However, it must be pointed out that no certification program can be successful without the skill of engineering, quality control and manufacturing organizations in industry. The cost effective use of the composites technology ultimately depends on their capability.

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FIGURE 1 - WEIBULL DISTRIBUTION FUNCTIONS FOR LOAD AND STRENGTH

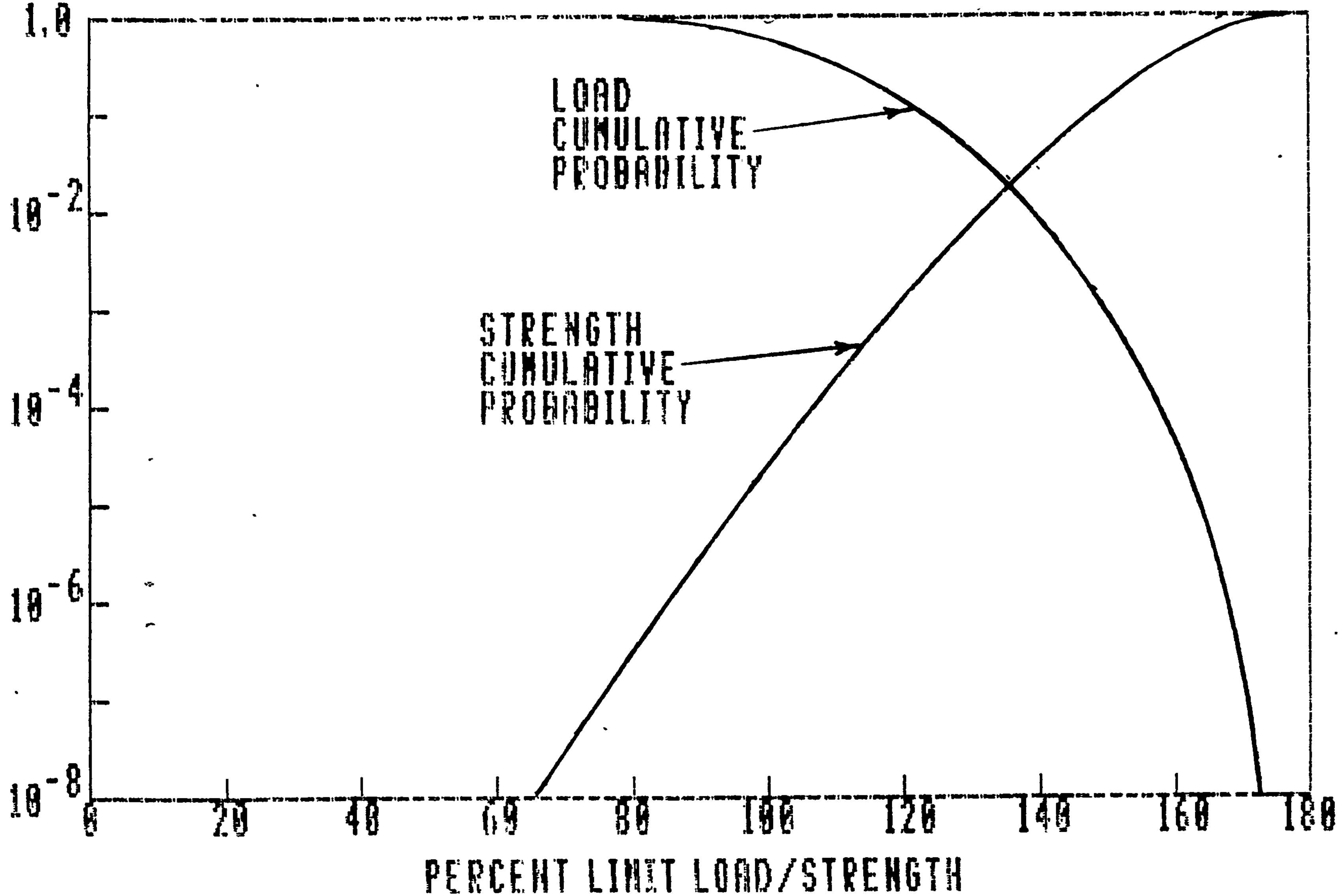

TABLE I - LOW ENERGY IMPACT (TOOL DROP)

ZONE	DAMAGE SOURCE	DAMAGE LEVEL	REQUIREMENTS
1 HIGH PROBABILITY OF IMPACT	* 0.5 IN. DIA. SOLID IMPACTOR * LOW VELOCITY * NORMAL TO SURFACE.	IMPACT ENERGY SMALLER OF 6 FT-LBS OR VISIBLE DAMAGE (0.1 IN. DEEP) WITH MIN. OF 4 FT-LB.	* NO FUNCTIONAL IMPAIRMENT OR STRUCTURAL REPAIR REQUIRED FOR TWO DESIGN LIFETIMES AND NO WATER INTRUSION * NO VISIBLE DAMAGE FROM A SINGLE 4 FT-LB IMPACT
2 LOW PROBABILITY OF IMPACT	SAME AS ZONE 1	IMPACT ENERGY SMALLER OF 6 FT-LBS OR VISIBLE DAMAGE (0.1 IN. DEEP)	* NO FUNCTIONAL IMPAIRMENT AFTER TWO DESIGN LIFETIMES AND NO WATER INTRUSION - AFTER FIELD REPAIR IF DAMAGE IS VISIBLE

TABLE II - LOW ENERGY IMPACT (HAIL AND RUNWAY DEBRIS)

ZONE	DAMAGE SOURCE	DENSITY	REQUIREMENTS
ALL VERTICAL AND UPWARD FACING HORIZONTAL SURFACES	HAIL * 0.8 IN. DIA. * SP. GR. = 0.9 *90 FT/SEC * NORMAL TO HORIZONTAL SURFACES * 45 DEG ANGLE TO VERTICAL SURFACES	UNIFORM DENSITY 0.8 IN. ON CENTER	* NO FUNCTIONAL IMPAIRMENT OR STRUCTURAL REPAIR REQUIRED FOR TWO DESIGN LIFETIMES * NO VISIBLE DAMAGE
STRUCTURE IN PATH OF DEBRIS	RUNWAY DEBRIS * 0.5 IN. DIA. * SP.GR. = 3.0 * VELOCITY APPROPRIATE TO SYSTEM		* NO FUNCTIONAL IMPAIRMENT FOR TWO DESIGN LIFETIMES AND NO WATER INTRUSION - AFTER FIELD REPAIR IF DAMAGE IS VISIBLE

TABLE III - INITIAL FLAW/DAMAGE ASSUMPTIONS

FLAW/DAMAGE TYPE	FLAW/DAMAGE SIZE
SCRATCHES	SURFACE SCRATCH 4.0 INCHES IN LENGTH AND 0.02 INCHES DEEP
DELAMINATION	INTERPLY DELAMINATION EQUIVALENT TO A 2.0 INCH DIAMETER CIRCLE WITH DIMENSIONS MOST CRITICAL TO ITS LOCATION
IMPACT DAMAGE	DAMAGE FROM A 1.0 INCH DIAMETER HEMISPHERICAL IMPACTOR WITH 100 FT-LBS OF KINETIC ENERGY OR WITH THAT KINETIC ENERGY REQUIRED TO CAUSE A DENT 0.10 INCHES DEEP, WHICHEVER IS LESS

TABLE IV - RESIDUAL STRENGTH REQUIREMENTS

P_{XX}*	DEGREE OF INSPECTABILITY	TYPICAL INSPECTION INTERVAL	MAGNIFICATION FACTOR
P_{FE}	IN-FLIGHT EVIDENT	ONE FLIGHT**	100
P_{GE}	GROUND EVIDENT	ONE DAY (TWO FLIGHTS)**	100
P_{WV}	WALK-AROUND VISUAL	TEN FLIGHTS**	100
P_{SV}	SPECIAL VISUAL	ONE YEAR	50
P_{DM}	DEPOT OR BASE LEVEL	1/4 LIFETIME	20
P_{LT}	NON-INSPECTABLE	ONE LIFETIME	20

*P_{XX} = MAXIMUM AVERAGE INTERNAL MEMBER LOAD THAT WILL OCCUR ONCE IN M TIMES THE INSPECTION INTERVAL. WHEN P_{DM} OR P_{LT} IS DETERMINED TO BE LESS THAN THE DESIGN LIMIT LOAD, THE DESIGN LIMIT LOAD SHOULD BE THE REQUIRED RESIDUAL STRENGTH LOAD LEVEL. P_{XX} NEED NOT BE GREATER THAN 1.2 TIMES THE MAXIMUM LOAD IN ONE LIFETIME, IF P_{XX} IS GREATER THAN DESIGN LIMIT LOAD.

**MOST DAMAGING DESIGN MISSION.

United States Air Force Engine Damage Tolerance Requirements

By

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UNITED STATES AIR FORCE ENGINE DAMAGE TOLERANCE REQUIREMENTS

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Abstract

This paper summarizes the damage tolerance requirements that are applied to Air Force engine development and acquisition programs. The importance of the damage tolerance control plan and the use of reliable inspection methods during development, production, and life management is emphasized. It is highlighted that development efforts in the last five years have identified process improvements for Fluorescent Penetrant Inspection (FPI). These improvements must be implemented within industry and Air Force depots to improve reliability to detect small flaws. Case examples of damage tolerance design are presented for an improvement to an existing engine (F100-PW-220 Increased Life Core) and for a new engine (PW5000). The Increased Life Core (ILC) information shows that damage tolerance requirements were met with small or modest increases in component weight, with cost savings and reduced inspection requirements and with lower life cycle costs (LCC). The importance of early trade studies to define the optimum life (Low Cycle Fatigue-LCF) and inspection interval requirements for a new engine design is shown for the PW5000. Data presented for both of these examples clearly illustrate that damage tolerance is achieved without adverse impact on conventional measures of merit (i.e., weight and cost).

Background

The Deputy for Engineering, Aeronautical Systems Division, has developed an aggressive program for engine mechanical and structural integrity which is applied to each development program. This program is commonly referred to as the Engine Structural Integrity Program (ENSIP) and is contained in MIL-STD-1783 (USAF) dated 30 November 1984. The ENSIP tasks are shown in Figure 1. A new and critical element now contained in ENSIP is the damage tolerance requirement which is highlighted in Figure 1. The damage tolerance requirement has been incorporated into USAF aircraft and engine development programs to prevent safety of flight structural failures that have been caused by material defects, manufacturing defects or fatigue induced cracks. Many of these incidents could have been avoided by proper material selection; control of stress levels; use of fracture resistant design concepts, manufacturing and processing controls; and use of reliable inspection methods during production and periodic in-service maintenance.

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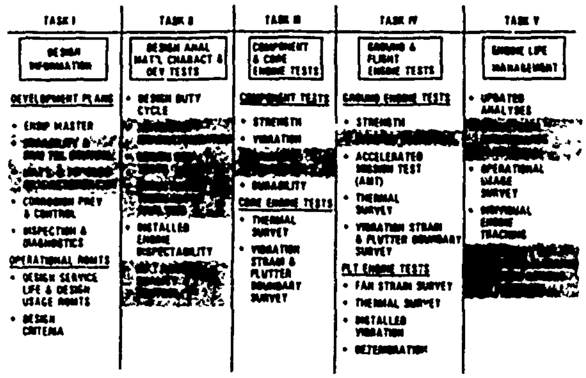


Fig. 1 The ENSIP tasks.

The primary need for damage tolerance requirements has occurred due to the ever present drive to minimize engine weight through development of materials with increased strength and resistance to crack initiation. An undesirable but attendant feature of these newer materials has been decreased resistance to crack growth. Some trends are shown in Figure 2. The conventional

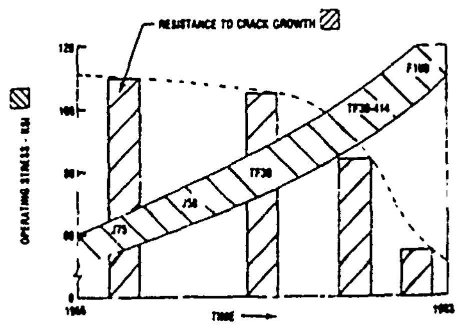


Fig. 2 Some material trends - resistance to crack growth.

approach to life management of cycle limited components is shown in Figure 3. Components are designed so that the low cycle fatigue (LCF) limit exceeds the required usage interval in terms of engine flight hours or cycles. The LCF limit is based on the lower bound (-3 sigma or 1/1000) of the distribution of crack initiation times. The lower bound limit has been chosen to prevent occurrence of cracking and resultant failure and the need for repair (economics). The main concern with the conventional approach is that no recognition or provision exists regarding the impact initial defects can have on total component life (i.e., component failure can and has occurred prior to reaching the LCF limit). An additional concern with the conventional approach is that the majority of components will be discarded prior to reaching their crack initiation time if the LCF limit is less than the full life requirement. Once again, this situation has occurred frequently in the past due to several factors such as the usage being more severe than assumed in design, system life extension beyond the original requirement, deficient detail design, etc. In summary, our concerns with the conventional life management approach involve both safety and economics.

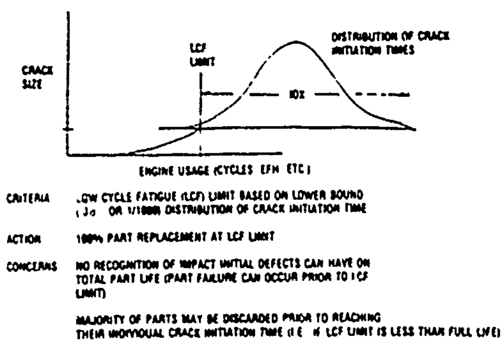


Fig. 3 Conventional approach to life management of cyclic limited engine components.

Damage Tolerance Requirements

General

Damage tolerance is defined as the ability of the engine to resist failure due to the presence of flaws, cracks or other damage for a specified period of unrepaired usage. The damage tolerance approach to life management of cycle limited engine components is shown in Figure 4. Components are designed so that the safety limit exceeds two times the required inspection interval. The safety limit or residual life is the time for assumed initial flaws to grow and cause failure. Since the requirement is to inspect at one-half the safety limit, the design goal for the safety limit is two times the required design life (i.e., no inspections). The minimum design requirement for the safety limit is two times the planned depot visit interval. The basis and assumptions for initial flaws are covered elsewhere in this paper.

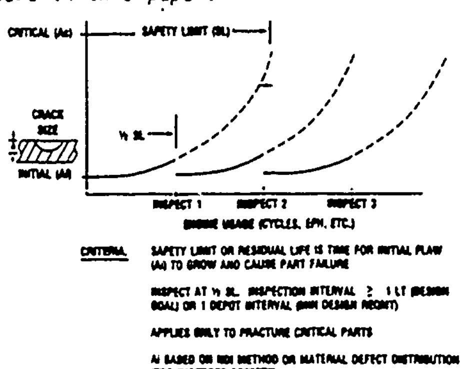


Fig. 4 Damage tolerance approach to life management of cyclic limited engine components.

It is required that the "cost" as a function of the "requirement" be defined via trade studies since the magnitude of the safety limit impacts overall engine configuration variables such as weight, costs, material selection, inspection methods, detail design. Trade study results are used to set the damage tolerance requirement and associated variables. Examples of redesign of an existing "off the shelf" engine and a new engine design are provided later. Additional discussion of the damage tolerance requirement and associated variables follows.

Fracture Critical Components

Fracture critical components are defined as those components whose failure will result in probable loss of the aircraft as a result of noncontainment or, for single engine aircraft, power loss preventing sustained flight due to direct part failure or by causing other progressive part failures. Damage tolerance requirements are applied only to fracture critical components and not, in general, to durability critical components. As can be expected, component classification is affected by aircraft/engine configuration; i.e., single engine or multi-engine. Component classification can, in some instances, be rather subjective and historical records and experience gained during development tests should be used to aid in classification. Component classification is established early and is identified in the contract specifications.

Initial Flaw Size

Initial flaws are assumed to exist in fracture critical components. Aircraft and engine experience reveals that premature cracking (i.e., crack initiation prior to the LCF limit) occurs at high stressed areas and initial conditions have included both material and manufacturing related quality variations (i.e., voids, inclusions, machining marks, scratches, sharp cracks, etc.). The damage tolerance requirement uses a sharp crack initial flaw assumption to characterize these abnormal initial conditions. Assumed initial imbedded flaw sizes are based on the intrinsic material defect distribution or the Nondestructive Inspection (NDI) methods to be used during manufacture. Assumed initial surface flaw sizes are based on the NDI methods to be used during manufacture. An inspection reliability of 90 per cent Probability of Detection (POD) at the lower bound 95 per cent Confidence Level (CL) is required for the assumed initial flaw sizes.

An initial flaw size not less than 0.030 inch length (surface) or 0.015 inch by 0.015 inch length (corners) for nonconcentrated stress areas (bores, webs, etc) is required. The basis for this requirement is twofold: (1) to establish an initial flaw size that can be screened by use of fluorescent penetrant inspection (FPI) as the standard NDI method and (2) to provide capability for application of upgraded NDI methods at a few locations when required. Initial flaw sizes for other surface locations (holes, fillets, scallops, etc) will be consistent with the demonstrated capability (90% POD/95% CL) of the inspection system proposed for use.

The assumed initial flaw sizes for use in design are summarized in Figure 5.

- 0.030 INCH SURFACE LENGTH WHERE NDI METHOD IS FLUORESCENT PENETRANT
- 0.010 INCH SURFACE LENGTH WHERE NDI METHOD IS EDDY CURRENT OR ULTRASONICS
- 0.002 SQUARE INCH AREA FOR IMBEDDED DEFECTS UTILIZING ULTRASONICS
- 0.200 INCH SURFACE LENGTH AND IMBEDDED SPHERE = 0.2×t FOR WELDMENTS
- WHEN INITIAL FLAW SIZES ARE BASED ON MATERIAL DEFECT DISTRIBUTION, SELECTED SIZE SHALL ENCOMPASS 99.99% OF THE DISTRIBUTION
- DEMONSTRATION THAT ASSUMED FLAW SIZES CAN BE RELIABLY DETECTED AT 90/95 IS REQUIRED

Fig. 5 Assumed initial flaw sizes used in design.

Residual Strength

Residual strength is defined as the load carrying capability of a component at any time during the service exposure period considering the presence of damage and accounting for the growth of damage as a function of exposure time. The requirement is to provide limit load residual strength capability throughout the service life of the component. Expressed in another way, the minimum residual strength for each component (and location) must be equal to the maximum stress that occurs within the applicable stress spectra based on the design duty cycle. Normal or expected overspeed due to control system tolerance and engine deterioration is included in the residual strength requirement but fail safe conditions such as burst margin are excluded. A pictorial presentation of the residual strength requirement is shown in Figure 6.

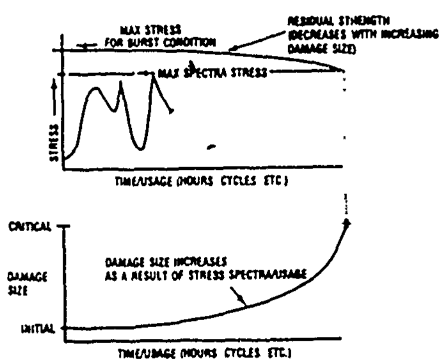


Fig. 6 The residual strength requirement.

Inspection Intervals

It is highly desirable to have no damage tolerance inspections required during the design lifetime of the engine. This inservice non-inspectable classification requires that components be designed such that the safety limit be twice the design life. Designing components as inservice non-inspectable is a requirement for those components or locations which cannot be inspected during the depot maintenance cycle (i.e., imbedded defect considerations and other non-inspectable areas).

It is recognized though that the weight penalty incurred to achieve a safety limit twice the design life may be prohibited on some components/locations. Therefore, inservice inspections will be allowed on some components subject to justification. The basis for the justification is characterization of the cost as a function of the requirement as established by trade studies. Cost is usually expressed in terms of weight or life cycle cost (LCC) and the requirement in terms of safety limit or flaw growth interval.

The depot or base level inspection interval for damage tolerance considerations must be compatible with the overall engine maintenance plan. Once again, it is highly desirable that the inspection interval be equal to the hot part design service life as this is the expected minimum depot or maintenance interval for the engine or module. It is required that the damage tolerance inspection interval be contained in the contract specification.

Flaw Growth

Flaw growth interval, safety limit and residual life are used interchangeably in this paper. It is required that the assumed initial flaw sizes will not grow to critical size and cause failure of a component due to the application of the required residual strength load in two times (2X) the inspection interval. The flaw growth interval is set equal to 2X the inspection interval to provide margin for the variability that exists in the total process (i.e., inspection reliability, material properities, usage, stress predictions, etc). Factors other than two should be used when individual assessments of the variables that affect crack growth can be made (e.g., to account for observed scatter in crack growth during testing).

It is very important that the effects of vibratory stress on unstable crack growth be accounted for in establishing the safety limit. Experience shows that the threshold crack size can be significantly less than the critical crack size associated with the material fracture toughness depending on the material, major stress cycle and the vibratory stress. Indeed, as shown by Figure 7, the conventional Goodman Diagram may not disclose the true sensitivity of initial defects to vibratory stresses.

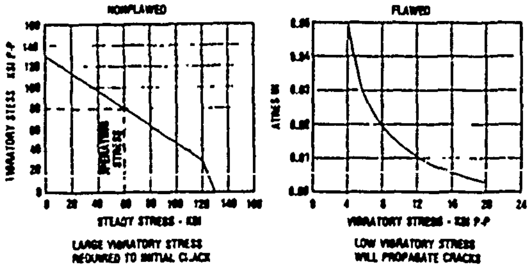


Fig. 7 Interaction of vibratory stress and and initial flaws - example.

Verification

Verification that damage tolerance criteria is met is assured via development and implementation of a damage tolerance control plan, by analysis and test, and by implementation of reliable inspection methods during manufacture and field/depot maintenance.

Damage Tolerance Control Plan. This plan identifies and schedules each of the tasks and interfaces in the functional areas of design, materials selection, test, manufacturing control and inspection. Specific tasks that are addressed in a control plan are shown in Figure 8. A typical schematic for the several functional areas is shown in Figure 9. Most of the tasks to be contained in the damage tolerance control plan have been accomplished by engine manufacturers in past development and production programs. However, the damage tolerance requirement now established by the Air Force imposes the need for new tasks as well as tighter controls and more involvement between the functional areas. Experience indicates that the development and implementation of a damage tolerance control plan is very difficult but experience also shows very strongly that development of a plan results in an improved understanding of what must be done.

1. TRADE STUDIES - DESIGN CONCEPTS/MATERIAL/WEIGHT/ PERFORMANCE/COST
2. ANALYSIS
3. DEVELOPMENT AND QUALIFICATION TESTS
4. FRACTURE CRITICAL PARTS LIST
5. ZONING OF DRAWINGS
6. BASIC MATERIALS FRACTURE DATA
7. MATERIAL PROPERTIES CONTROLS
8. TRACEABILITY
9. NONDESTRUCTIVE INSPECTION (NDI) REQUIREMENTS

Fig. 8 Tasks - damage tolerance control plan.

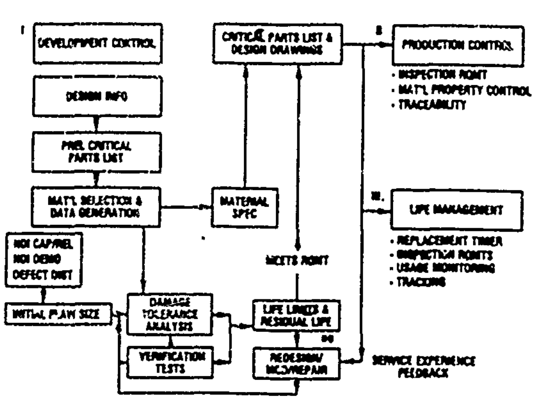


Fig. 9 Typical flow diagram - control plan.

A particularly important part of the damage tolerance control plan is the requirement for early trade studies for design concepts/ material/weight/ performance/cost. These trade studies are critical to defining cost versus requirement (e.g., weight impact versus inspection interval). A later section of this paper illustrates how trade studies are used to set the design for a new engine.

Analysis and Test Requirements. A summary of the analysis and test objectives for damage tolerance is shown in Figures 10 and 11. Particular emphasis is placed on establishing correlation between analytical predictions and test measurements for growth of cracks in critical areas. Refined analysis models that predict the stress state at and away from the surface as well as multiple cyclic tests of coupons, subcomponents and full scale components in the presence of initial damage are required. Test requirements are summarized in Figure 12. Engine test with components that are pre-flawed or cracked in critical locations is important to determine the effects of the "real" environment (temperature and gradients, vibration, etc). Such tests must be closely controlled and monitored using the inspection requirements planned for service engines to assure safety of the test engine. A summary of the damage tolerance test conducted on the F100 engine is shown in Figure 13. Test requirements for damage tolerance as conducted on a recent development program are discussed in a later section of this paper.

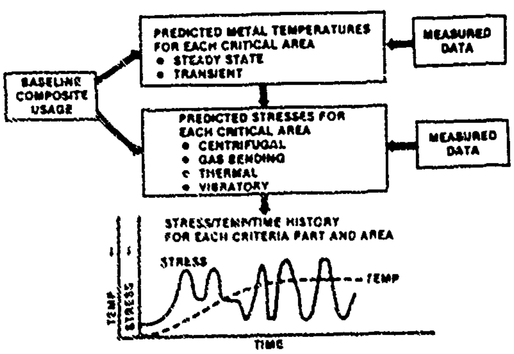


Fig. 10 Development of stress spectra.

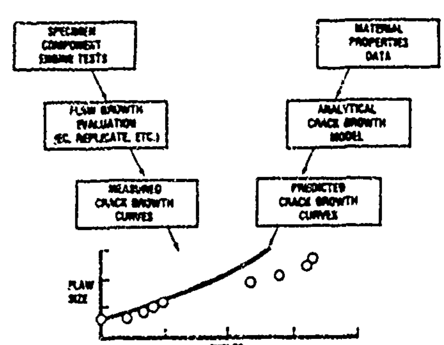


Fig. 11 Residual life analysis and test procedures.

SPECIMEN/COUPON
- BASIC MATERIAL PROPERTIES (DA/DN VS ΔK Kc)
  - STANDARDIZED CONFIGURATION (I.E., ASTM)
  - CONSTANT AMPLITUDE STRESS CYCLES
  - MATERIAL, FREQUENCY, STRESS RATIO, TEMPERATURE TEST MATRIX
- VERIFICATION OF CRACK GROWTH MODEL (a VS N)
  - REPRESENTATIVE STRUCTURAL DETAILS (BOLT HOLES, SCALLOPS, ETC.)
  - VARIABLE AMPLITUDE STRESS SPECTRUM
  - MATERIAL, TEMPERATURE, STRESS SPECTRUM TEST MATRIX

COMPONENT
  - VERIFICATION OF CRACK GROWTH MODEL
  - FULL-SCALE ROTOR/STRUCTURAL DETAILS
  - MULTIPLE PRE-CRACKS IN CRITICAL STRUCTURAL DETAILS

ENGINE
  - VERIFICATION OF CRACK GROWTH MODEL
  - "REAL" ENVIRONMENT (TEMPERATURE, GRADIENTS, VIBRATION ETC.)
  - REPRESENTATIVE MISSION STRESS SPECTRUM (I.E., AMT)
  - MULTIPLE PRE-CRACKS IN CRITICAL STRUCTURAL DETAILS

Fig. 12 Damage tolerance test requirements.

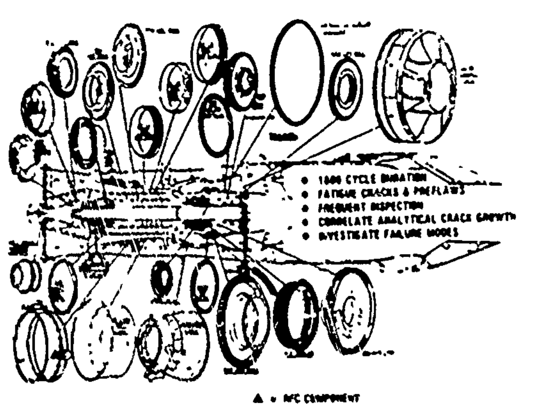


Fig. 13 F100 residual life engine test.

Non Destructive Inspection (NDI). NDI requirements are implemented on fracture critical parts during manufacture and during field/depot inspection to protect safety. Specific inspection requirements are derived via design analysis trade-offs between initial flaw size assumption, stress level, and material properties for a given usage (stress environment spectrum). As discussed in the section on inital flaw size, a flaw size assumption less than 0.030 inch surface length requires implementation of enhanced NDI (i.e., eddy current). Primary emphasis on use of eddy current is for stress concentration areas where a small flaw size assumption is required to achieve the required residual life without excessive weight penalty. Typical probability of detection data for eddy current is shown in Figure 14. A summary of enhanced inspection requirements for Air Force engines is shown in Figure 15.

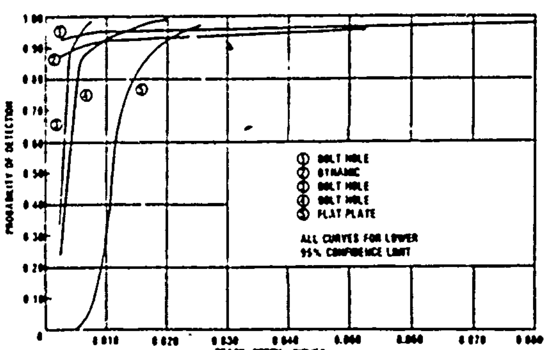


Fig. 14 Probability of detection - eddy current.

F100 ENGINE	
EDDY CURRENT @ DEPOT	3/80
EDDY CURRENT @ PRODUCTION	11/82
FAN DISK CRYOGENIC PROOF TEST	1/84
TF34 ENGINE	
EDDY CURRENT @ DEPOT	1/84-12/84
F101 ENGINE	IMPLEMENTATION
EDDY CURRENT @ PRODUCTION & DEPOT	PLAN DEVELOPED
FUTURE IMPLEMENTATIONS	BEING DEFINED
F110	
F109	
ATFE	

Fig. 15 Status- implementation of upgraded NDI requirements.

In general, flourescent penetrant inspection (FPI) is specified for areas requiring 0.030 inch surface length or larger to achieve the required residual life. The ability of current FPI processes in common use today to reliably detect .030 inch flaws is not clear. Therefore, in some instances eddy current may be specified for large surface areas or for additional locations if susceptibility to inservice damage exists, if the area is not readily accessible or if reliability data indicates the need. Data generated on numerous demonstration programs clearly indicates that the FPI process can be significantly improved over that in general use today via implementation of upgraded training, equipment and procedures (e.g., proper cleaning including etch, hydrophilic emulsifier, wet developer) and by doing redundant (multiple) inspections. These demonstration programs have been conducted on several Air Force engine development programs (F100, F101, F110) and laboratory technology programs (Air Force Wright Aeronautical Laboratories). Detection improvements that have been demonstrated for improved surface preparation including etch are shown in Figure 16. A critical near term need is to implement the best

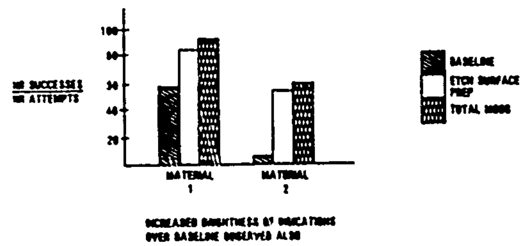


Fig. 16 FPI flaw detection improvement - etch surface preparation.

FPI process within industry and within the Air Force Logistics Centers since FPI will remain to be the most widely used method to inspect large areas for crack-like damage. Another critical need for whole field inspection is development of advanced methods for longer term applications. The most promising approaches appear to be refinement of existing methods with emerging technologies such as laser scanning with FPI and high speed eddy current or ultrasonics. Finally, we need to quantify the POD of ultrasonics to detect imbedded defects in bulk volumes and to develop inspection methods for finished shapes. Very limited data indicates that reliable detection limits may be as large as 2000 square mils (i.e., approximately equal to a planar disc of 3/64 inch diameter). The goal is to develop and implement ultrasonic inspection methods such that a residual life equal to 2X the required life or inspection interval can be achieved assuming the largest undetectable flaw size without excessive impact on weight.

F100-PW-220 Increased Life Core (ILC)

The Increased Life Core (ILC) of the F100-PW-220 engine provides an excellent case example of the application of ENSIP based damage tolerance requirements to a significant evolutionary change to an existing engine. The ILC contains major design improvements to the "heart" of the engine giving the F100-PW-220 the capability of extended inspection intervals and part lives thereby reducing costs and increasing readiness. The current F100 core provides a firm baseline against which cost and weight impacts to meet ENSIP can be measured as well as benefits in maintenance intervals and life extension.

The F100-PW-220 is a fully qualified new engine model. However it provides many of the benefits of an off-the-shelf engine because of its high degree of commonality with current -100 and -200 production models. This commonality was achieved in part through the requirement that interchangeability of design improvements be maintained with current production parts, sets, and modules. Although this constraint on interchangeability existed for the ILC design, major improvements were nevertheless possible in the areas of durability and damage tolerance. These improvements resulted in the following benefits.

- 4000 TACs* minimum inspection interval
- 4000 TACs hot section part life
- 8000 TACs cold section part life

These advances in part life were made possible through the use of advances in materials and special attention to design details to enhance damage tolerance.

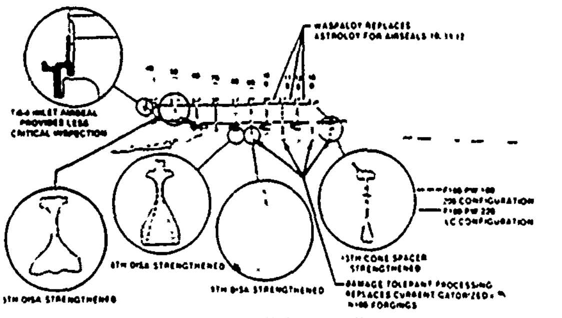


Fig. 17 F100 ILC compressor rotor design.

Compressor Rotor Design

Basic construction of the compressor rotor features a combination of integral arm and flat disks as shown in Figure 17. Tiebolts at the flat disk locations connect the disks to form the basic rotor backbone. The first three disks are made of titanium alloys whereas the latter stages are nickel-base alloys. Spacers with integrally machined coated wedge shaped seals are assembled between adjacent disk rims to limit rim vibratory stresses and deflections. Most of the high strength alloys and integral parts that have been proven in the F100 are retained in the F100-PW-220 HPC rotor.

However, a number of changes were made to enhance damage tolerance capability consistent with ENSIP criteria and are also shown in Figure 17.

Material selection was a key factor in obtaining the improvements in life and damage tolerance to achieve the ILC goals without major redesigning of structural components in the compressor. As an example, selection of a substitute material for Astroloy rim spacers illustrates the potential benefits possible through material choice. Substitution of Waspaloy for Astroloy in the aft (10th, 11th and 12th) compressor spacers was possible by very slight thickness changes which added less than .25 pounds. The resulting benefits include both improvements to LCF life and damage tolerance. The improvement in damage tolerance provides an increase of 100% in residual life at operating stresses and temperatures typical of mission usage as shown in Figure 18.

The importance of design detail on damage tolerance can be seen through the example of the fifth stage disk which is made of high strength titanium. The web of the fifth stage was strengthened to resist the loading of the rim spacers which heat up and thermally expand faster than the bolted structural backbone of the compressor. Modest recontouring of the disk web and its transition to the disk bore resulted in .25 pounds increase in weight. Bending stresses were reduced 50 percent and overall combined (effective) stress was reduced 40 percent. Improvement in damage tolerance is reflected in an increase in the flaw size required to attain an inspection interval equal to the depot

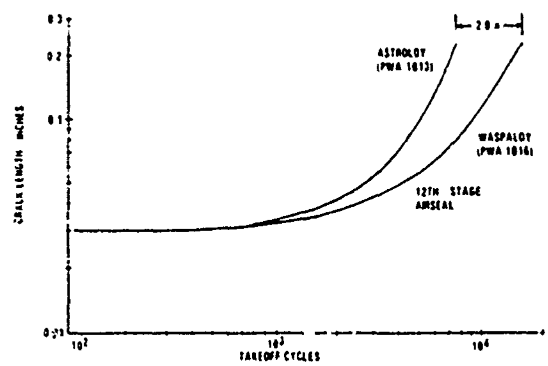


Fig. 18 Effect of material choice on component life.

interval of 4000 TACs. Eddy current inspection at production is not necessary with the larger flaw size at the reduced stress level. Figure 19 shows that a threefold increase in permissible flaw size was obtained. Cost of inspection was thereby reduced with no increase in cost of the disk.

Overall weight increase to the ILC high pressure compressor to meet damage tolerance requirements total a modest four pounds.

Turbine Rotor Design

The ILC turbine rotor system shown in Figure 20 is a new and advanced configuration designed for damage tolerance. It is unique in that there are no life limiting cooling air or attachment holes in either the rim, web, or bore area of either disk. Since holes in these areas were

*TACs = Total Accumulated Cycles = LCF cycles + .25 FTC
where LCF cycles = Cold start - Intermediate and Above - Shutdown
and Full Throttle Cycles (FTC) cycles = Idle - Intermediate and Above - Idle

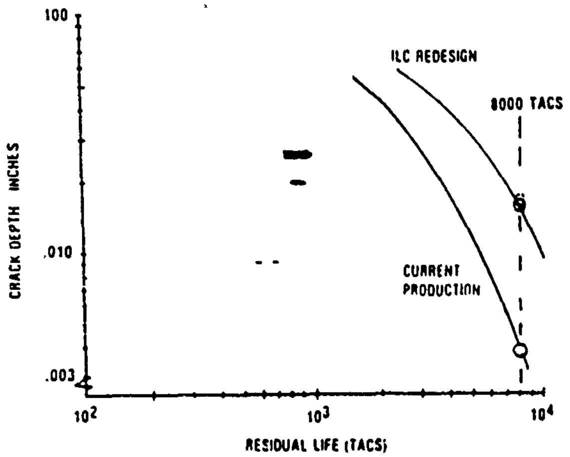


Fig. 19 F100 HPC 5th stage disk web residual life.

found to severely limit the fracture life of the disk, a configuration was identified that did not require life limiting holes in the disk body. Eliminating these holes reduces stesses to a much lower level than those of the current F100 production rotor. The two-stage configuration has been maintained to ensure interchangeability and commonality with the F100, minimize overall weapons systems weight, and provide maximum performance with moderate physical speeds.

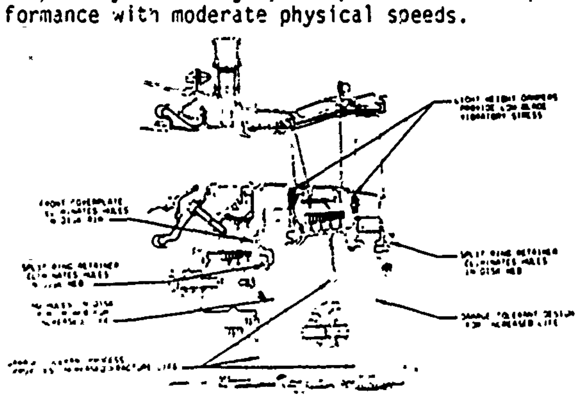


Fig. 20 F100 ILC turbine rotor design.

Temperatures and pressures used for analysis of the HPT rotor were verified in an experimental core engine. Low stresses of the ILC HPT rotor have been verified in two room temperature spin tests. Figure 21 shows a comparison of predicted stresses converted from measured strains for the spin test condition.

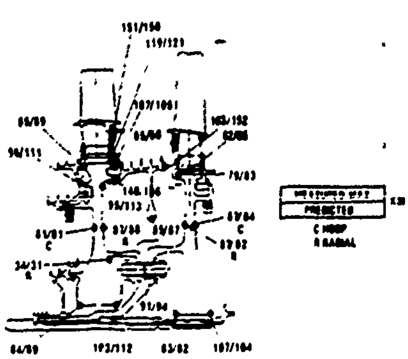


Fig. 21 Correlation of measured and predicted stresses.

Improved crack growth capability of the ILC HPT rotor design has been verified by a hot cyclic spin test of a preflawed rotor (Figure 22). Initially a rotor constructed of current production processed PWA1074 (Gatorized TM IN100) was used to verify the design improvements in crack growth relative to the current production rotor. After 12,000 cycles a damage tolerance processed 1st stage disk was substituted for the PWA1074 disk and testing continued to 17,000 cycles. Testing was conducted at 850°F. Rotor speed was cycled from 1000 to 14,500 rpm. Maximum engine speed is 13,500 rpm. The higher test speed was used to accelerate testing. A total of 109 preflaws were introduced by EDM (Electrical Discharge Machining) and Tack Welding in critical stress locations. The preflaws were slow to propagate as detectable cracks. By comparison, the current production HPT rotor has been tested in four similar tests with 107 preflaws from 1,000 to only 13,500 rpm at 850°F. Propagation to detectable cracks and subsequent crack growth in these tests agreed with predictions and were much faster than in the ILC rotor test.

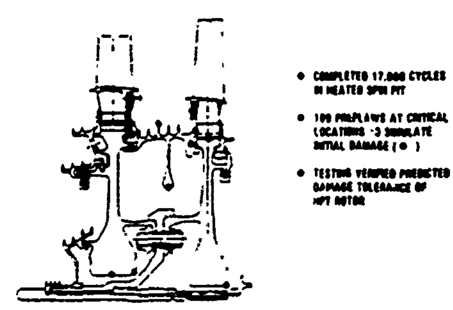


Fig. 22 F100 Cyclic spin test - F100 ILC turbine rotor.

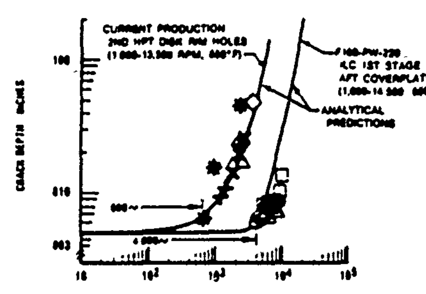


Fig. 23 Actual and predicted flaw growth.

A comparison of the fastest growing cracks of the ILC and current production rotors is shown in Figure 23. The ILC coverplate was chosen for comparison because it was the first component to exhibit cracks. Even at the higher rotor test speed, the ILC rotor has demonstrated improved crack growth life and improved damage tolerance. The earliest ILC preflaws became detectable cracks at 4,000 spin cycles while the current production rotor exhibited cracks as early as 600 spin cycles although rotor test speed was only 13,500 rpm.

The ILC HPT rotor was designed to be capable of meeting typical crack growth of 3000 TAC's from inspectable flaw sizes. To achieve this capability it was recognized that extensive eddy current inspections, approaching whole field capability, would be required if current material was retained. Damage tolerant processing which was shown under an IR&D program to have promise of increasing crack growth by at least a factor of two, and for some stress/flaw inspection sizes up to four times, was placed under a full scale characterization program to allow its use in the ILC. A comparison of crack growth of PWA1074 and DTP Gatorized TM IN100 is shown in Figure 24. Through the use of DTP it was possible to maintain the use of Flourescent Penetrant Inspection (FPI) for the bulk of inspections performed on production parts. Eddy current use is limited to three locations. The total specialized inspection requirements are significantly reduced when compared to the current production rotor even though its inspection interval has increased from 1800 TACs to 4000 TACs. Comparison of requirements is shown in Table 1. Significant cost avoidance has resulted by the use of DTP to meet the inspection criteria. The cost of inspection for the HPT would be approximately two times higher if conventionally processed material were used.

Because the tensile and yield strength of DTP is reduced somewhat compared to PWA1074 and the rotor was designed for lower stresses to maximize damage tolerance, a weight increase of 25 pounds was required relative to the earlier production HPT.

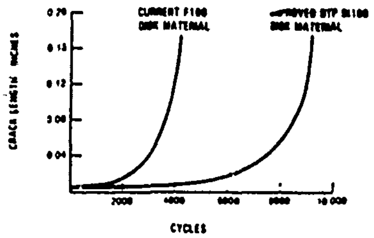


Fig. 24 Flaw growth life comparison.

Assessment of Damage Tolerance Cost

Increased part costs occur due to the increase in input weight and due to slightly higher machining costs associated with DTP material. These costs for both the HPC and HPT in the ILC have been assessed and are shown in Table 2.

Table 1 Comparison of eddy current inspection requirements

Number of:	Current F100 at 1800 TACs	ILC at 4000 TACs
Locations	10	3
Individual Inspections	332	70

Table 2 Damage tolerance design cost increase (approx)

Increased Inspection Costs	$ 800/engine
- Focused Zyglo Inspections	
- Eddy Current Inspections	
Increased Fabrication/Machining Costs	$ 500/engine
Increased Materials Costs	$2300/engine
Total Increased Cost	$3600/engine

Another significant cost factor is associated with increases in design and development costs. In the design arena a very large investment in finite element analysis is required to define internal stress and strain states for crack growth analyses. Experience in the ILC has shown a 100% increase in stress analysis effort during the design phase of new components. The crack growth life computation for all critical areas is also a significant factor because of the complexity of typical fighter missions in which time dependent relationships of pressures, temperatures and rotor speeds interact. Added costs for damage tolerance analysis were assessed to be $2 million.

In the area of development similar cost impacts are felt. Material characterization, especially crack growth testing for da/dn over the wide range of temperatures and R ratios required for mission analysis, adds significant costs to the already expensive costs of new material characterization such as the DTP Gatorized TM IN100. In addition, verification tests for damage tolerance along with expensive prototype test parts must be accounted for.

These additional development costs of material characterization and verification testing totalled approximately $2 million. Total additional design and development costs therefore approximated $4 million. Although this represents a substantial increase in total costs, on a per engine basis, it is a modest investment. For example, on a one thousand engine fleet, cost is $4000 per engine.

In contrast to a moderate initial cost increase per engine, very considerable benefits will be realized in reduced maintenance costs and increased utilization of parts. Increased maintenance interval (i.e., time between inspections) from 1300-2000 TACs to 4000 TACs is projected to save $120,000 per engine over the 8000 TAC design life by eliminating two shop visits. Even greater savings are realized when the potential for extended utilization is examined. It was found that design for damage tolerance usually resulted in minimum LCF life margins of 50% to 200%. Thus parts can be used for another one-half life resulting in parts replacement costs savings of $140,000 per engine. This coupled with another $60,000 per engine savings to eliminate an additional shop visit results in additional savings of $200,000. Total cost savings per engine over 12,000 TAC usage is thus $320,000 per engine.

The bottom line in cost based on a hypothetical one thousand engine fleet is then a benefit to cost ratio of 40 to 1.

$$\frac{\text{Benefits/engine}}{\text{Costs/engine}} = \frac{\$320,000}{7,600} = 40$$

This extremely advantageous benefit to cost ratio is promised for a very small rotor weight increase, less than 1% in engine weight. Damage tolerant design concepts have been accepted on the basis of increased safety. Clearly the LCC example also provides a compelling argument for damage tolerant design from a cost viewpoint and justifies its use as prime criteria in new engine programs.

PW5000

To meet potential wartime threats in the 1990s, an advanced aircraft known as the Advanced Tactical Fighter (ATF) is being developed. The power plant to be used to power this aircraft has been designated as the Advanced Tactical Fighter Engine (ATFE). Pratt & Whitney's engine for this application is the PW5000. This engine is an advanced, twin spool, augmented turbofan with thrust vectoring and reversing capability. The engine cycle has been structured to provide a balance between supersonic dry power capability and transonic maneuverability requirements. For both flight and ground operation, thrust vectoring and reversing capabilities are provided to improve aircraft maneuverability and reduce landing distance.

Improved durability and reduced maintenance requirements over current day engines are required for the ATFE. At the same time, the aircraft/engine must be maneuverable (light weight) and affordable (minimum Life Cycle Cost - LCC) to the Air Force. Therefore, a study was needed in the early days of the ATFE program to determine the optimum life and inspection interval goals for the PW5000.

Based on initial durability goals provided by the procuring activity, a preliminary engine configuration was defined for the PW5000. As part of the configuration definition, several features were incorporated into the design to improve durability. This was accomplished by two methods. First, stress risers are minimized by eliminating such things as bolt holes, oil drain holes, etc. Secondly, materials are utilized which have improved Low Cycle Fatigue (LCF) and damage tolerance capabilities. Upon completion of this effort, the study was initiated to define "optimum" life and inspection interval.

PW500 Study Overview

The study objective was to define the optimum life and inspection interval for the ATFE. This was accomplished by defining weight impacts and the resulting LCC associated with various life goals and inspection intervals. The following is a brief discussion of the ground rules defined for the study.

Life Limits. Weight and LCC impacts were defined for assumed LCF life values of 8000 TACs (current engine life), 12,000 TACs and 16,000 TACs (future engine desirable goals).

Damage Tolerance. Weight and LCC impacts were defined for inspection intervals of 3000, 4000, and 5000 TACs. A margin of safety of 2.0 was used on all crack growth calculations to determine the inspection interval; i.e., the inspection interval was equal to one-half of the fracture mechanics safety limit. Deterministic fracture mechanics techniques were used for all crack growth calculations.

Mission Usage. To reduce the amount of analysis required, a single mission was defined which is representative of the anticipated flight usage of the ATF.

Thermal Analysis. Detailed component temperatures were defined throughout the mission for the parts studied.

Materials. Projected improved LCF and crack growth properties were utilized The fan rotor component materials were assumed to be titanium. The remaining components were assumed to be nickel alloys.

Flaw Sizes. Flaw sizes were based on projected inspection techniques and meet the requirements of Figure 6 for flourescent penetrant and eddy current. Philosophically, the number of locations requiring eddy current (EC) were minimized. However, to minimize weight impacts, improved inspection techniques (i.e., smaller flaw size associated with EC) were incorporated at disk limiting locations prior to increasing the component weight. This resulted in an increased number of EC inspections, but minimized the weight and LCC increases.

Structural Limits. Short time limits such as disk burst margin, local yield and plastic growth (deflections) were adhered to for all parts studied.

Study Results - LCF Life

The results of the study for the ATFE showed that variations in inspection intervals had considerably more impact on engine weight and LCC than changes in LCF. Therefore, only a brief discussion of the life goal study is presented herein.

Three disks were selected for detailed stress and life analysis to represent all other disks. To insure that the study was representative of the entire rotor system, "typical" disks were chosen to accomplish the detailed analysis. This prevents a bias from occurring when determining rotor system weights and LCC. Also, the inspection interval was kept constant relative to the LCF design goal. For example, the inspection interval was 3000 TACs for a life goal of 12,000 TACs(1/4 inspection interval). Similarly, 4000 TACs inspection interval was used when the life goal was 16,000 TACs.

The life goal study showed that the optimum rotor part life is equal to the planned operational usage; i.e., equal to the aircraft design life. This is illustrated in Figure 25 where the operational usage was 12,000 TACs and results in the minimum LCC for the system. Increasing the design life above the planned usage causes unacceptable increases in disk weight which means a heavier aircraft and more fuel used. Conversely, designing the rotor to a life goal less than the planned usage results in a significant increase in the number of replacement parts. This increase in the number of replacement parts does not compensate, from a LCC standpoint, for the resultant rotor weight savings.

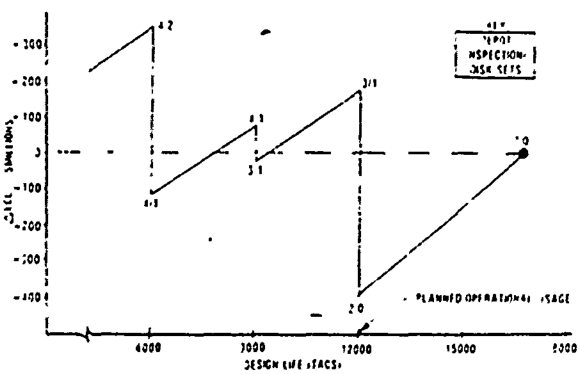


Fig.25 Life cycle cost impact vs design life (PW5000).

Study Results - Inspection Interval

As in the LCF life portion of the study, three "typical" disks were chosen to determine the optimum inspection interval. A disk was selected from each of the Fan, HPC and HPT rotor sections. Weight and LCC impacts were then estimated for the remaining stages.

Inspection goals of 3000, 4000 and 6000 TACs were selected for the study. For all inspection intervals, the LCF life was kept constant at 12,000 TACs. Sizing of the disks was accomplished by determining allowable stress levels for all critical features in each of the components studied. The allowable stress levels were then input into a disk synthesis computer program which produced an estimated geometric shape. Figure 26 is an example of two synthesized disks. The smallest disk was sized for 3000 cycles and the larger disk was sized for 6000 cycles. Upon completion of the disk synthesis, all critical features were analyzed in detail to determine the safety limits and resulting LCF lives.

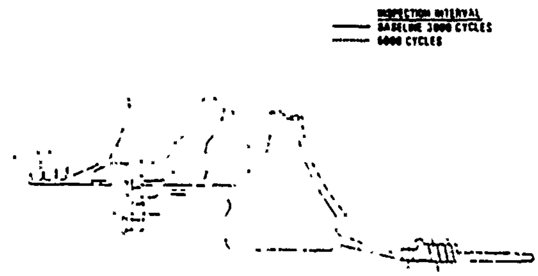


Fig. 26 Impact of inspection interval on configuration.

To determine the safety limit and subsequent inspection intervals, deterministic fracture mechanics techniques were utilized. As mentioned earlier, to minimize weight increase, additional EC inspections were assumed prior to changing the disk size to meet the respective inspection intervals of 3000, 4000 or 6000 TACs. If incorporation of the added EC inspections resulted in an inadequate safety limit, then weight was added to the disk to meet the required inspection interval.

In addition to weight changes, it is extremely important to define "fundamental" configuration impacts. Configuration changes may result in a significant impact to the basic design concept resulting in additional weight penalties other than just the disks being studied. In the case of the engine section shown in Figure 26, additional design changes were required. The added disk weight caused a rotor critical speed problem. In order to correct this critical speed concern, the rotor backbone had to be increased in diameter resulting in an additional weight penalty.

For the purposes of the study the 3000 TAC inspection interval was assumed to be the baseline. All weights, LCC numbers and increases in the number of EC inspections were done relative to this baseline. Detailed results of this study are shown in the following paragraphs:

EC Inspection Impact. Table 3 shows how the number of EC inspections increases with an increase in desired inspection interval. With a 3000 TAC inspection interval as the base, a significant increase in the number of features needing EC is required for 4000 and 6000 TAC inspection intervals, respectively.

Weight Impact. Table 4 shows the weight increases as a function of inspection interval. To increase the inspection interval from 3000 to

4000 TACs only requires a 20 pound weight penalty. However, a 6000 TAC inspection interval would require a weight increase of 190 pounds when compared to a 3000 TAC interval. This magnitude of weight increase (190 lbs) would not be acceptable for a high thrust/weight weapon system like the ATF. However, a 20 pound weight penalty would not be considered excessive.

LCC Impact. Figure 27 illustrates how LCC changes with inspection interval. As in the weight summary, a 4000 TAC inspection interval would increase the weapon system LCC by a modest $44 million. However, going to a 6000 TAC inspection interval would result in a significant and unacceptable LCC increase of $1,170 million.

Table 3
Features requiring eddy current inspection

	3000 CYCLE INSP.	4000 CYCLE INSP.	6000 CYCLE INSP.
FAN	BASE	+ 4	+ 10
HPC	BASE	+ 6	+ 10
HPT	BASE	+ 2	+ 5
LPT	BASE	+ 1	+ 2
TOTAL:	BASE	+ 13	+ 27

Table 4
Weight impact vs inspection interval

	3000 CYCLE INSP	4000 CYCLE INSP.	6000 CYCLE INSP.
FAN	BASE	+ 10 LBS	+ 50 LBS
HPC	BASE	+ 10 LBS	+ 35 LBS
HPT	BASE	0 LBS	+ 75 LBS
LPT	BASE	0 LBS	+ 30 LBS
TOTAL:	BASE	+ 20 LBS	+ 190 LBS

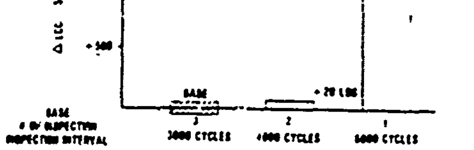


Fig. 27 Life cycle costs impact vs inspection interval.

Study Summary

To accomplish a new engine design and get the engine to test in a reasonable period of time, specific structural requirements are needed at the initiation of the program. This is especially true for component life and inspection interval requirements. If these requirements are not defined early in the program, then there is a high risk that either significant redesigns are required "down the road" to meet the life and inspection requirements, or the engine may be overly penalized in weight and LCC. Therefore, trade studies like the one performed on the ATFE engine are needed early in the design phase of a new engine program so that the weapon system can be optimized for weight and LCC.

In the case of the PW5000 engine, the following conclusions can be made:

- The optimum LCF life goal should be equal to the weapon system life. Significant weight and/or LCC penalties are incurred when the engine is designed to a life different than that of the weapon system.

- The damage tolerance portion of the study showed that a 3000 TAC inspection interval is the lowest weight and LCC for the weapon system. However, for modest weight increase, the inspection interval can be increased 33%. Conversely, an inspection interval increased to 6000 cycles results in unacceptable weight and LCC increases as well as 500% increase in the number of features requiring EC inspection.

Summary

Damage tolerance requirements that are applied to Air Force engine development and acquisition programs have been summarized. These requirements are an integral part of the Engine Structural Integrity Program (ENSIP) contained in MIL-STD-1783 (USAF) dated 30 November 1984. Recent applications experience clearly demonstrates that damage tolerance is achieved without adverse impact on conventional measures of merit (i.e., weight and cost). Case examples for the F100-PW-220 ILC and the PW5000 have been presented.

Some ENSIP Application Perspective
(The General Electric View)

By

Dr. L. Beitch
General Electric Company

HISTORICAL PERSPECTIVE

- ASIP PHILOSOPHY/FORMAT BASIS OF INITIAL ENSIP STRUCTURE
- PROVIDES USEFUL FRAMEWORK ... BUT SOME SIGNIFICANT DIFFERENCES EXIST:
    - AIRFRAME DESIGN PRACTICES TEND TO BE PUBLIC DOMAIN ... COMMON DATA BASES, ETC.
    - REDUNDANT STRUCTURES NOT PRACTICAL FOR TURBOMACHINERY
    - AIRFRAME A STRUCTURE ... ENGINE AN AERODYNAMIC MACHINE
    - HIGHER TEMPERATURES AND STRESSES IN ENGINE CRITICAL STRUCTURES

HISTORICAL PERSPECTIVES (CONT.)

- ADVANCED HIGH STRENGTH ENGINE ALLOYS LESS DAMAGE TOLERANT

- ENSIP NDE REQUIREMENTS GENERALLY MORE DEMANDING ... LEAD TO STATE-OF-THE-ART INSPECTION TECHNIQUES

$\downarrow$

ENSIP DAMAGE TOLERANCE HAS MAJOR DESIGN IMPACT

OVERVIEW

- FROM A BROAD PERSPECTIVE, ENSIP BRINGS TO AIRCRAFT ENGINES AN ORGANIZED, DISCIPLINED APPROACH TO ...

  - STRUCTURAL DESIGN
  - ANALYSIS
  - DEVELOPMENT
  - PRODUCTION
  - PARTS LIFE TRACKING AND MANAGEMENT

OVERVIEW

THE OBJECTIVES OF ENSIP INCLUDE...

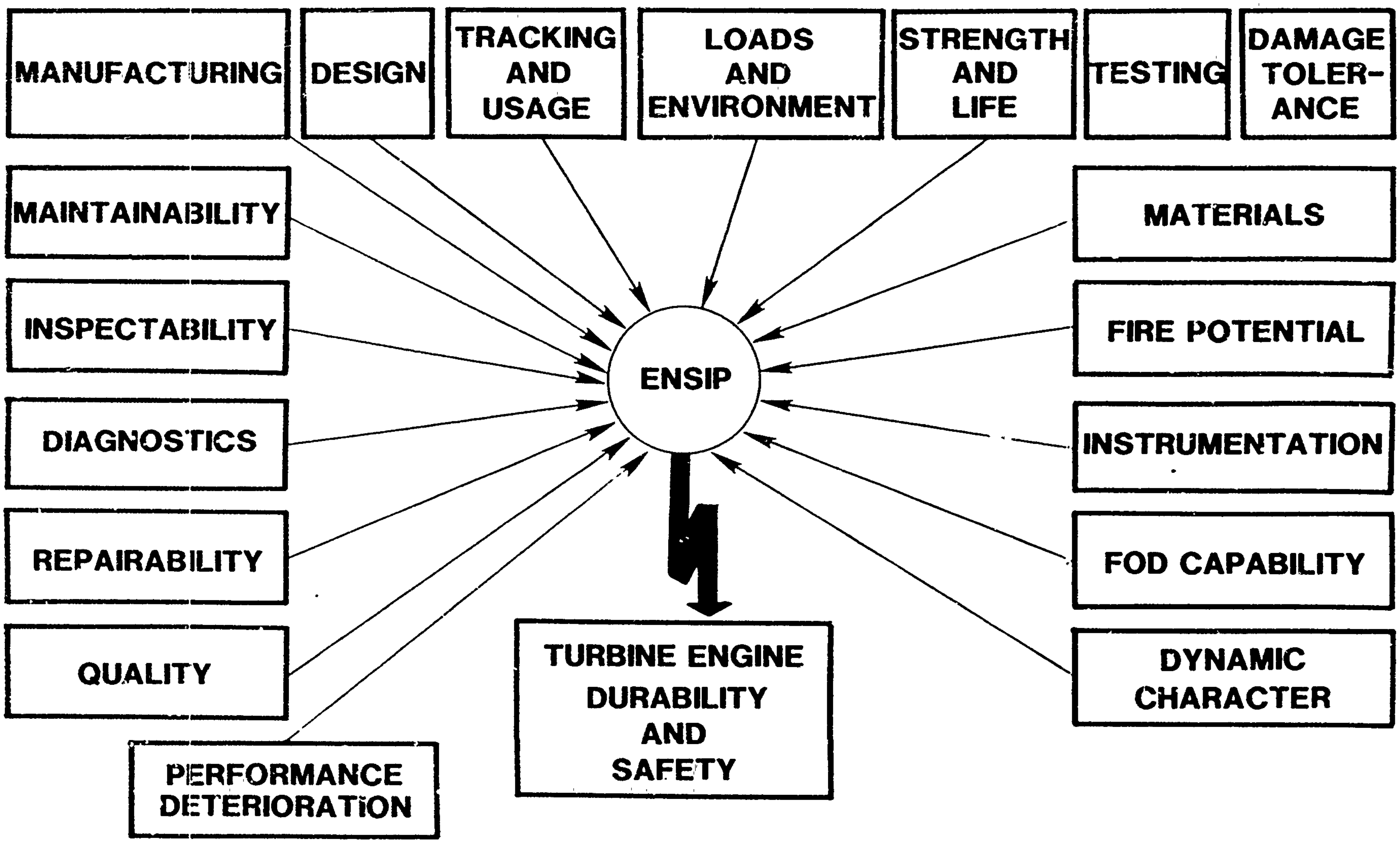
- SAFETY
- DURABILITY/DAMAGE TOLERANCE
- INCREASED READINESS
- REDUCED LIFE CYCLE COSTS

OVERVIEW

- ENSIP ACTIVITIES ARE WIDE RANGING...
  - PRE-QUALIFICATION
    - DESIGN
    - ANALYSIS
    - DEVELOPMENT
  - POST-QUALIFICATION
    - PRODUCTION
    - FIELD TRACKING
    - LIFE MANAGEMENT

THE SCOPE

(MANY SPECIFIC DISCIPLINES)



THE EVOLUTION OF ENSIP AT GENERAL ELECTRIC

THE ENSIP PHILOSOPHY IS CLEAR TO US ALL ...

NOW, EXAMINE THE IMPLEMENTATION OF ENSIP
IN A GROWING MILITARY BUSINESS

THE EVOLUTION OF ENSIP AT GENERAL ELECTRIC

IN GENERAL TERMS IT HAS MEANT ...

- THE DEVELOPMENT OF A STRONG HIGH LEVEL COMMITMENT TO ENSIP
- A LEARNING EXPERIENCE ... STILL ONGOING
- A SUBSTANTIAL ADJUSTMENT IN ORGANIZATION, RESOURCES, AND FACILITIES

.
.
.

- IT'S A CHALLENGE ... THAT WE ARE MEETING!

ENSIP IMPLEMENTATION

EDUCATION HAS BEEN KEY AT GE ...

- ENSIP AND DAMAGE TOLERANCE NEW TO MANY ... AND REQUIRED MODIFICATION OF EXISTING APPROACHES AND THINKING

  - NEW APPROACHES AND CHANGE OFTEN GENERATE OPPOSITION

- IMPACT ON PRODUCTION AND LABORATORY FACILITIES EXTENSIVE

- ACCEPTANCE OF FRACTURE MECHANICS ON FLAWED MATERIALS, BUT PAST GE FIELD EXPERIENCE DOES NOT TOTALLY SUPPORT NEED FOR DAMAGE TOLERANCE ON "NON-DEFECT SENSITIVE" MATERIALS

  ... ACCEPTANCE GAINED THROUGH EDUCATION

THE EVOLUTION OF ENSIP AT GENERAL ELECTRIC

LET'S EXAMINE PERSPECTIVES IN SEVERAL AREAS ...

- ENSIP ACCEPTED BY GE ... PARALLELS GE DESIGN PRACTICES
    - ONLY DAMAGE TOLERANCE A VARIANCE
- FROM "ASSESSMENT" TO FORMAL REQUIREMENTS
- PRODUCTION IMPLEMENTATION
- DEPOT IMPLEMENTATION
- PARTS LIFE TRACKING AND MANAGEMENT
- FIELD MISSION DATA EVALUATION

DAMAGE TOLERANCE AT GENERAL ELECTRIC

BEGINNING WITH ASSESSMENTS ...

- FOR ENGINES ALREADY DESIGNED AND IN SOME STAGE OF PRODUCTION/USAGE
- U.S.A.F. FUNDS ENGINE MANUFACTURER TO PROVIDE...
  - ANALYTICAL STATUS OF CRITICAL PARTS WITH RESPECT TO DURABILITY AND DAMAGE TOLERANCE
  - COST PROPOSAL TO IMPLEMENT DAMAGE TOLERANCE INSPECTIONS
- RESULTS IN NEGOTIATIONS AND CONTRACT MODIFICATION TO IMPLEMENT

GE TF34 ASSESSMENT EXAMPLE

- ASSESSMENT OF TF34 (FOR A10) - 1980
  - USAF/GE ENGINEERING TEAM ESTABLISHED
  - STEERING COMMITTEE ALSO FORMED
- ENGINE ALREADY IN SERVICE
- RESULTED IN ...
  - DAMAGE TOLERANCE EVALUATION FOR TYPICAL FIELD MISSION
  - DEVELOPMENT OF FIELD INSPECTION PLAN/NO PRODUCTION INSPECTIONS

F101 ASSESSMENT EXAMPLE

- MORE FORMAL ASSESSMENT OF F101 (FOR B1 BOMBER) - BEGUN IN 1980
- ENGINE DESIGNED ... AND IN EARLY PRODUCTION
- ASSESSMENT BASED ON 3000 ENGINE FLIGHT HOUR GOAL FOR ANTICIPATED FIELD MISSION

F101 ASSESSMEMT EXTENDED TO F110

- F101 ASSESSMENT RESULTED IN ...
  - INSPECTIONS AT PRODUCTION AND DEPOT
    - MULTI-PHASED IMPLEMENTATION PLAN
  - "BEST EFFORT" MATCHING OF INSPECTION METHOD AND ITS P.O.D. WITH FEATURE CRACK PROPAGATION CAPABILITY VS. 3000 EFH GOAL
  - GE FUNDING AND USAF CIP PROGRAMS TO GROW NDE CAPABILITIES
- SIMILAR ASSESSMENT AND IMPLEMENTATION FOR F110 (FOR F15/F16)

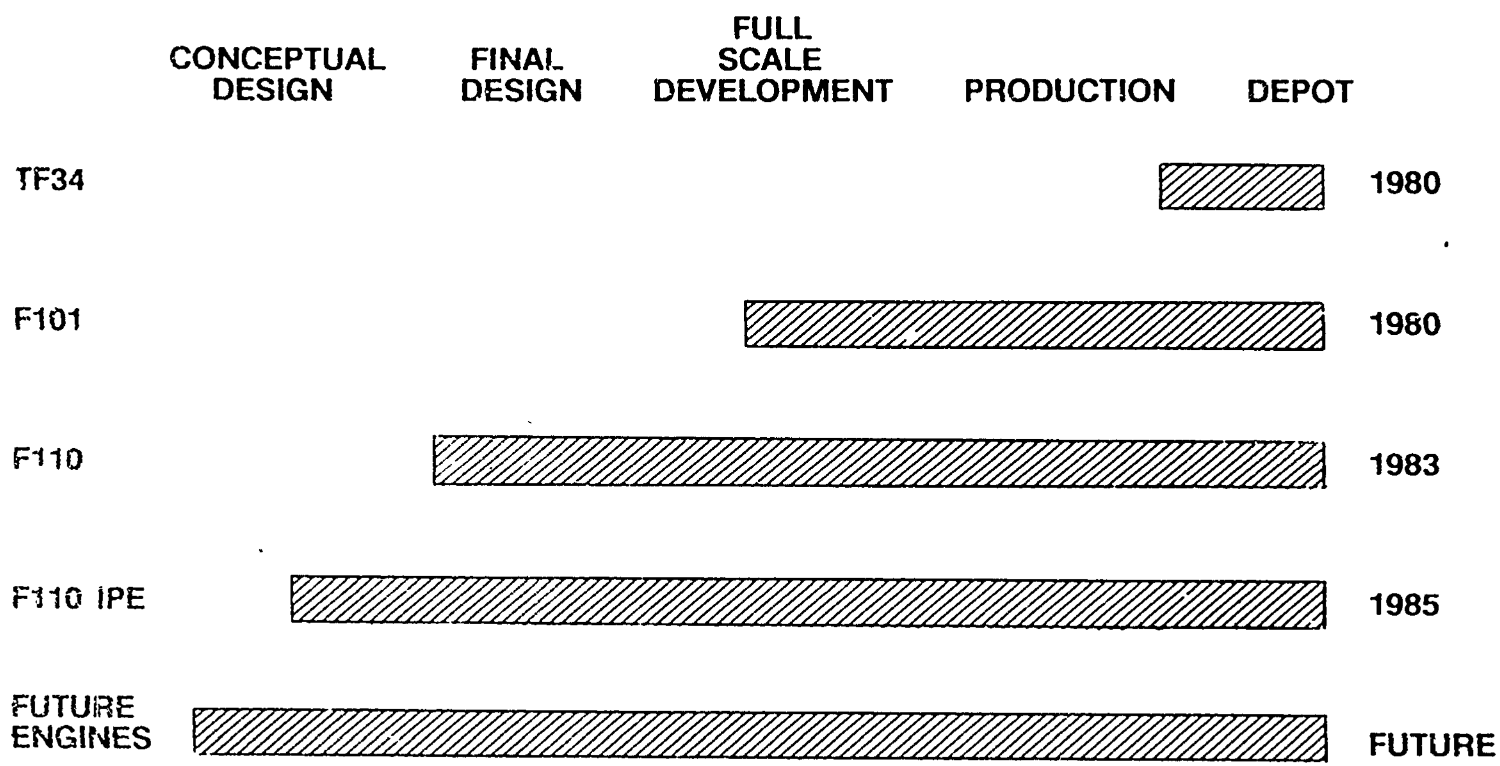
DAMAGE TOLERANCE EVOLUTION - THE FUTURE

FOR FUTURE ENGINES, REQUIREMENTS ARE MORE SPECIFIC AND DEMANDING ...

- INSPECTION INTERVALS HAVE GONE FROM GOALS TO REQUIREMENTS
- NDE RELIABILITY NOW SPECIFIED ... P.O.D./CONFIDENCE LEVELS
- MINIMUM FLAW SIZE NOW SPECIFIED ... MUST BE DEMONSTRATED
- MAXIMUM TIME REQUIRED TO PERFORM INSPECTIONS ALSO SPECIFIED

DAMAGE TOLERANCE EVOLUTION - A SUMMARY

THE EVOLUTION OF DAMAGE TOLERANCE HAS RESULTED IN REQUIREMENT BEING IN PLACE THROUGHOUT TOTAL ENGINE LIFE . . .



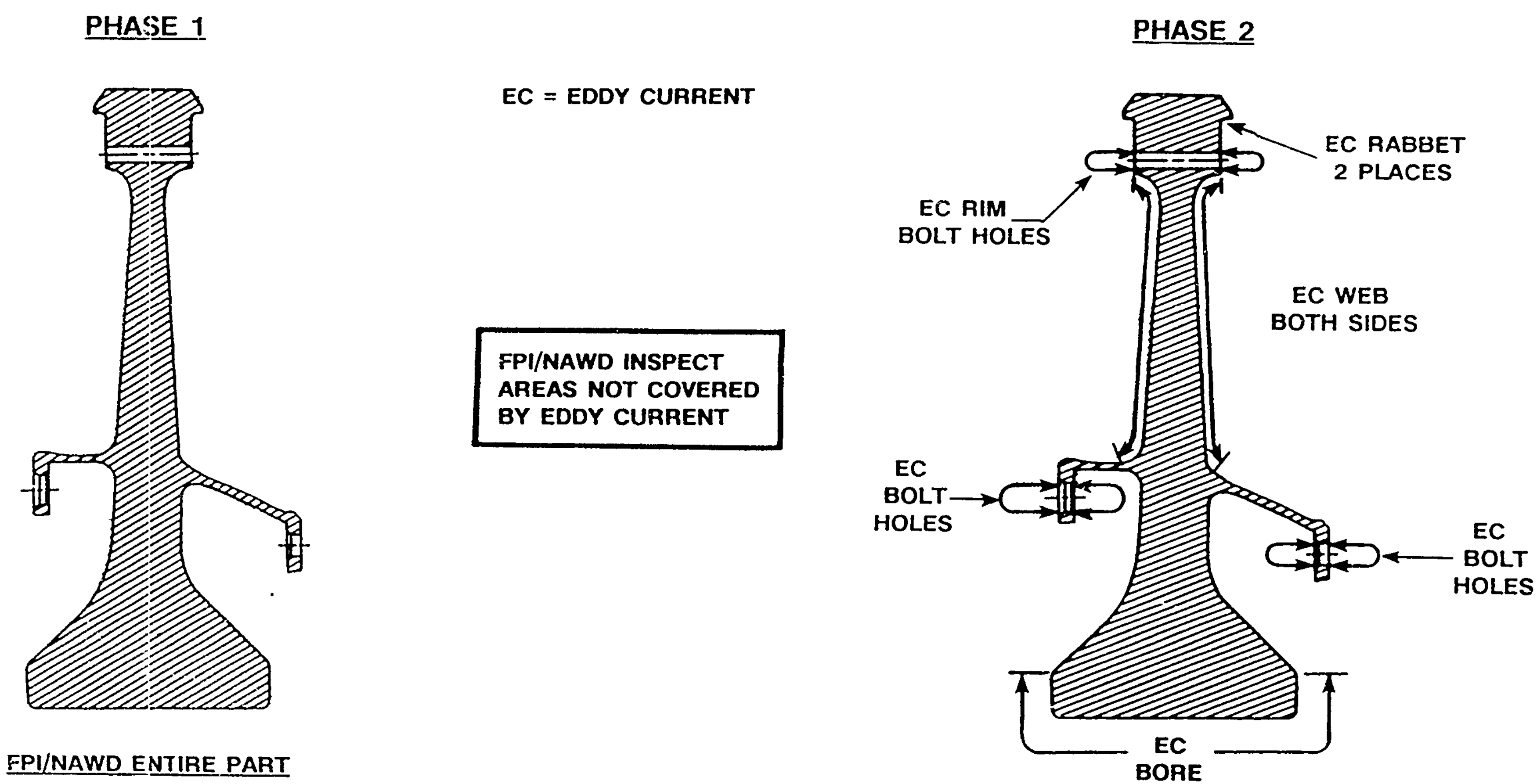
PRODUCTION IMPLEMENTATION OF DAMAGE TOLERANCE

F101/F110 PROGRAMS REQUIRED PHASED IMPLEMENTATION OF DAMAGE TOLERANCE INSPECTIONS TO ALLOW DEVELOPMENT OF ...

- PRODUCTION INSPECTION CAPACITY
- NDE CAPABILITIES FOR SELECTED FEATURES
- MANUFACTURING PROCESS CHANGES
- CUSTOMER INTERACTIONS ... TECHNICAL/CONTRACTUAL

EXAMPLE OF PHASED NDE IMPLEMENTATION

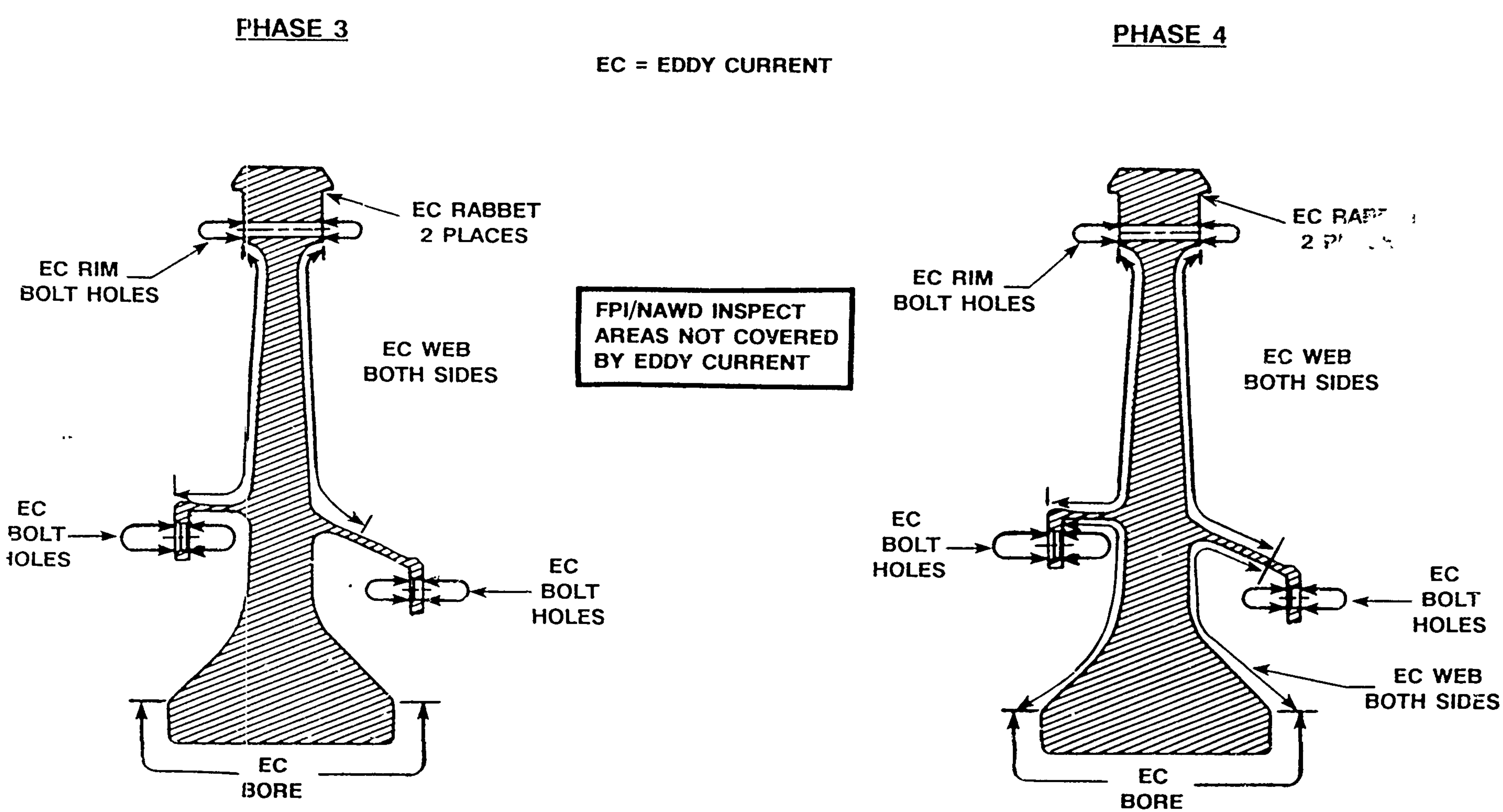
HPT DISK



EXAMPLE OF PHASED NDE IMPLEMENTATION

HPT DISK

EC = EDDY CURRENT



EXAMPLE OF PHASED NDE IMPLEMENTATION

F101 HPT DISK

- TOTAL DISK INSPECTION TIME AT PRODUCTION 50+ MAN-HOURS ... TOTAL ENGINE ~200 MAN-HOURS

- PROCESS YIELD ON DISK IMPROVING ...

  - INITIALLY, NDE WAS REJECTING 100% ... AT LEAST ONE BOLT HOLE OR OTHER FEATURE ON EVERY DISK REQUIRED REWORK

  - SIGNIFICANT IMPROVEMENTS ACHIEVED

<u>DEPOT IMPLEMENTATION ... MUCH WORK TO BE DONE</u>

ENSIP INSPECTIONS <u>WILL</u> <u>BE</u> <u>A</u> <u>CHALLENGE</u> <u>AT</u> <u>DEPOT</u> ....

- MANY NEW ENGINES BEING PROCURED ...
  - COULD HAVE 1800+ F101/F110 ENGINES IN SERVICE BY 1990'S
- WILL REQUIRE EDDY CURRENT AND ULTRASONIC INSPECTIONS
  - DEPOT NEEDS EXPERIENCE WITH THESE METHODS

DEPOT IMPLEMENTATION

MANY FUNDAMENTAL LOGISTICS ISSUES MUST BE RESOLVED BY GE AND USAF...

- INSPECTION INTERVAL GOALS ARE ...
  - FLOATING TARGETS - WITH ONGOING CONTRACT NEGOTIATIONS
  - DIFFERENT FOR F101 AND F110 AND FOR EARLY AND LATER PARTS
  - NOT LIKELY TO BE MET FOR SOME EXISTING DESIGNS WITH CURRENT DAMAGE TOLERANCE APPROACH
- GE AND USAF NOW DEVELOPING OVERALL INTEGRATED INSPECTION PLAN

DEPOT IMPLEMENTATION

ONCE OVERALL PLAN IS IN PLACE, FACILITIZATION AND TRAINING WILL BECOME KEY ISSUES ...

- PROCUREMENT OF NECESSARY FACILITIES AND EQUIPMENT
  - NEEDS MUST BE IDENTIFIED
  - LONG LEAD TIMES AN ISSUE
- TRAINING WILL BE CRITICAL ...
  - WILL BE "STEP CHANGE" AT DEPOT
  - TRAINING MANUALS, NDE PLANS MUST BE DEVELOPED

DEPOT IMPLEMENTATION

FINALLY, MANY TECHNICAL ISSUES MUST BE RESOLVED ...

- DEPOT DEMONSTRATION OF INSPECTION CAPABILITY
  - STATE-OF-THE-ART TECHNIQUES
  - FIELD RUN PARTS ... CLEANLINESS, SHOTPEENING ... AFFECT P.O.D.
  - MUST HANDLE PARTS INSPECTED AT PRODUCTION BEFORE FABRICATION
- DEVELOPMENT OF PROCEDURES TO INTERPRET INSPECTION RESULTS
- REWORK/REPAIR PROCEDURES NEEDED FOR PARTS NOT PASSING INSPECTIONS

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- WE WILL, WITH USAF, OVERCOME THESE PROBLEMS!

PARTS LIFE TRACKING

FOR BOTH F101 AND F110, SYSTEMS ARE BEING DEVELOPED TO TRACK LIFE USAGE FOR EACH LIFE LIMITED PART ...

- <u>BASIC</u> PARAMETERS BEING RECORDED ON EACH AIRCRAFT
  - PROVIDES BASIC TIME/CYCLIC HISTORY OF EACH PART
- DAMAGE ALGORITHMS BEING DEVELOPED FOR CRACK INITIATION, CRACK PROPAGATION, AND CREEP/RUPTURE ...
  - PROCESSING RAW DATA THROUGH ALGORITHMS PROVIDES PARTS LIFE USAGE

PARTS LIFE TRACKING

PARTS LIFE TRACKING WILL ALLOW USAF TO MANAGE THEIR FORCE ...

- OVERALL TRENDING ANALYSIS
- BASE TO BASE COMPARISONS
- LOGISTICS PLANNING AND DEPOT INSPECTION LOADING
- PROVIDES PROBLEM INVESTIGATION DATA BASE

PARTS LIFE TRACKING

MUCH WORK TO DO TO PREPARE FOR 16,000+ FLIGHTS PER YEAR DATA LOAD OF 1990'S ...

- DEVELOPING USAF/AIRFRAMER/GE INTERFACES ... A JOINT EFFORT
  - PERSONNEL
  - COMPUTER DATA SYSTEMS
- CONSTRUCTING DAMAGE ALGORITHMS ... K-FACTORS
- ONGOING SOFTWARE DEVELOPMENT EFFORT

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<u>MUST WORK CLOSELY WITH CUSTOMER TO BE READY!</u>

FIELD MISSION ANALYSIS

BEYOND BASIC PARTS LIFE TRACKING, FIELD USAGE MONITORING ...

- ALLOWS DEVELOPMENT OF ACTUAL FIELD USAGE MISSION PROFILES

- PROVIDES BASIS FOR MORE ACCURATE STRESS AND LIFE ANALYSIS WHICH MAY RESULT IN ...

    - MODIFIED STRUCTURAL MAINTENANCE PLAN INSPECTION INTERVALS
    - MORE ACCURATE DAMAGE ALGORITHMS
    - POTENTIAL IMPACT ON LIFE CYCLES COSTS
    - DETAILED DATA BASE FOR PROBLEM INVESTIGATION

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<u>MUST CONTINUE TO HAVE CLOSE GE/USAF/AIRFRAMER COOPERATION TO MAKE THIS WORK!</u>

ENSIP - THE NEXT STEPS

- REVIEW/DEVELOP CRITERIA APPLICABLE TO CONTROLS AND ACCESSORIES (CASIP)

- QUANTIFY RISKS INVOLVED IN ENSIP LIFE MANAGEMENT (SAFETY INSPECTIONS) PROCEDURE. POTENTIAL UTILITY OF RISK ANALYSIS APPROACHES USING PROBABILISTIC FRACTURE MECHANICS METHODOLOGY.

- STRUCTURE ENSIP PHILOSOPHY FOR FAR-TERM ENGINES WITH METAL MATRIX - COMPOSITES/CERAMIC FRACTURE CRITICAL COMPONENTS

Garrett Turbine Engine Company F109 Engine Damage Tolerance Verification Program

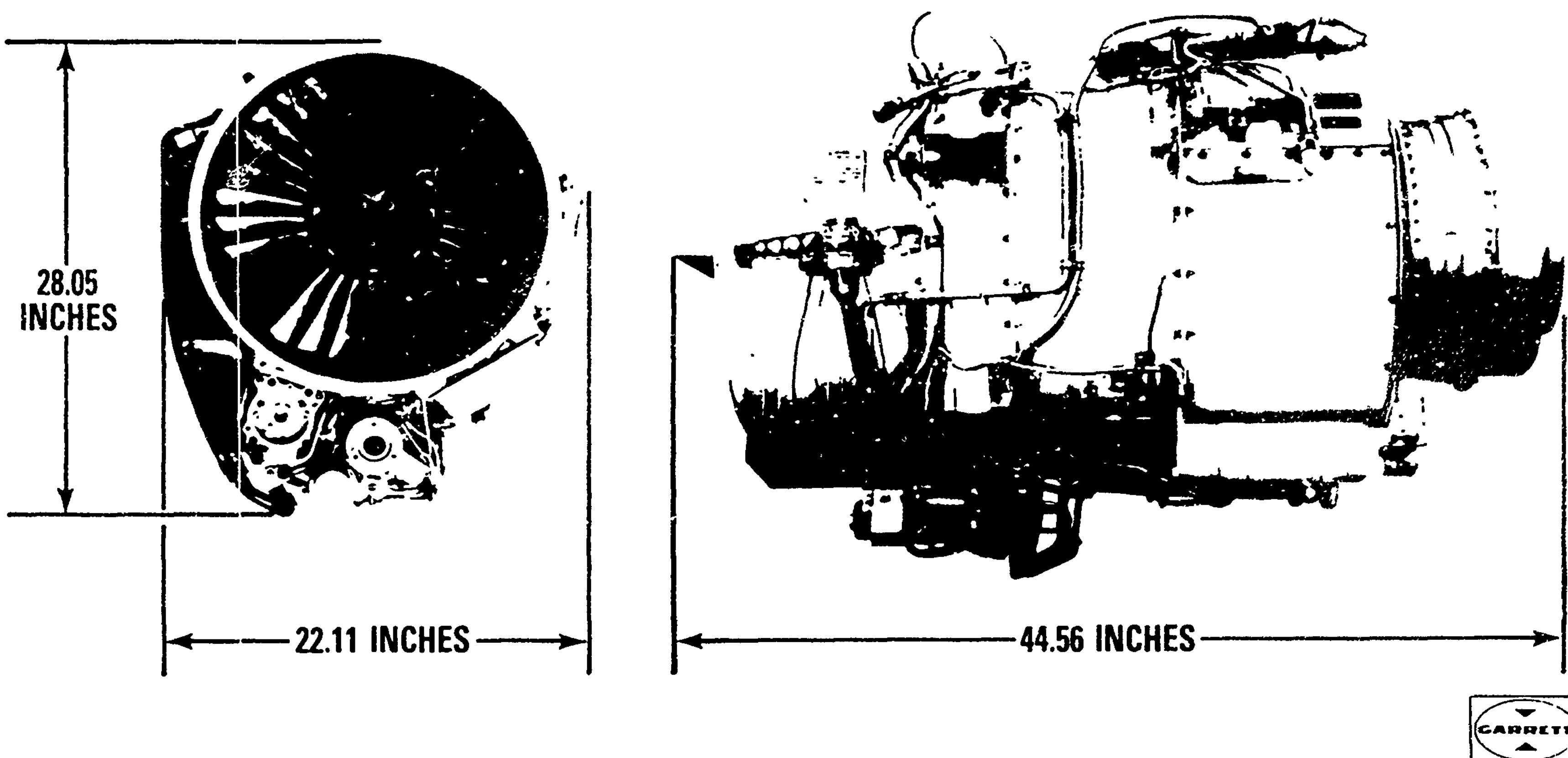
By
Hans Maertins
F109 Project

- F109 ENGINE DESCRIPTION/DAMAGE TOLERANCE CONSIDERATIONS
- SPECIMEN CCGR TESTING
- WHIRLPIT CCGR TESTING
- ENGINE CCGR TESTING
- CONCLUSIONS

G6-273-2

F109 TURBOFAN ENGINE

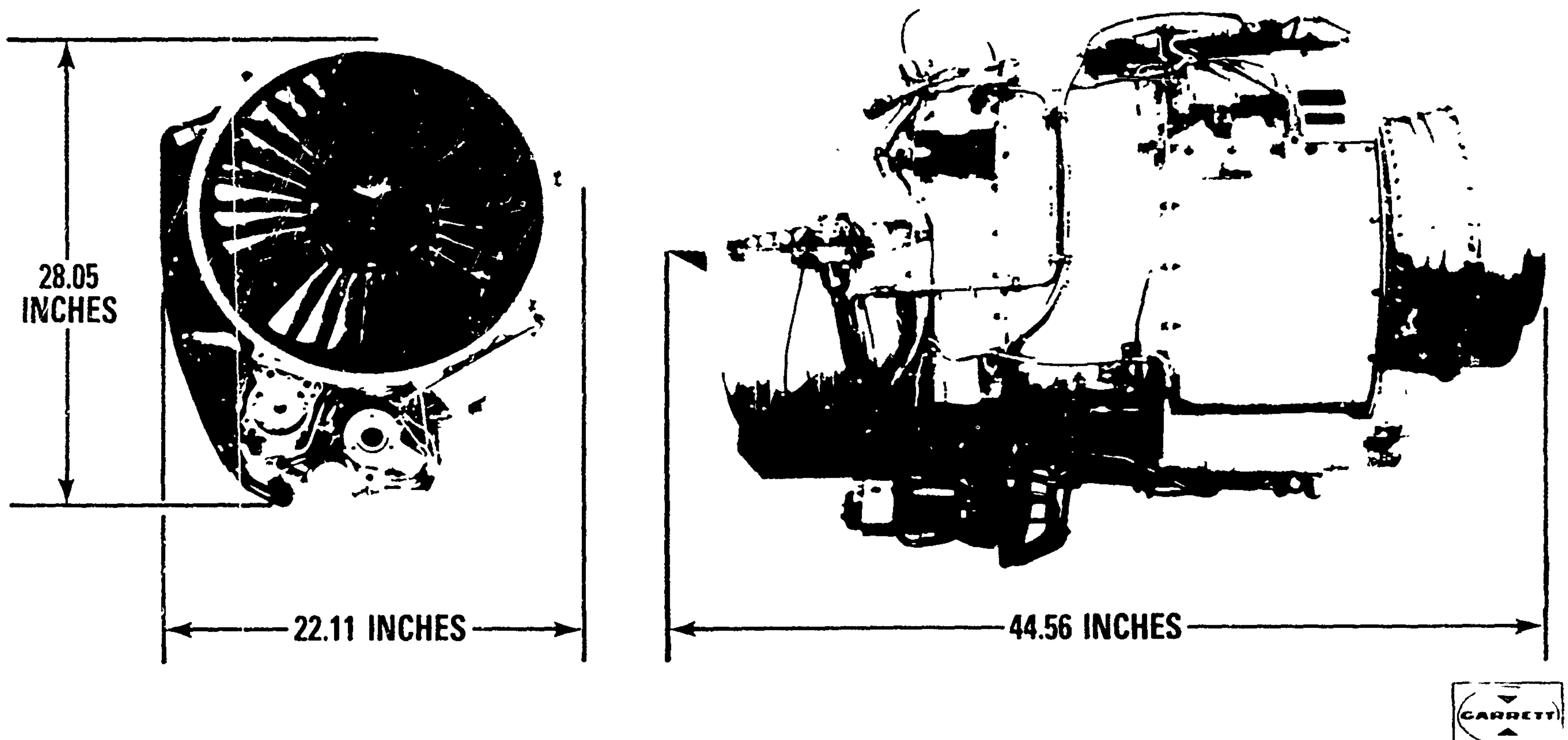
1330 LBS THRUST SLS TO STD DAY



G6-273-3

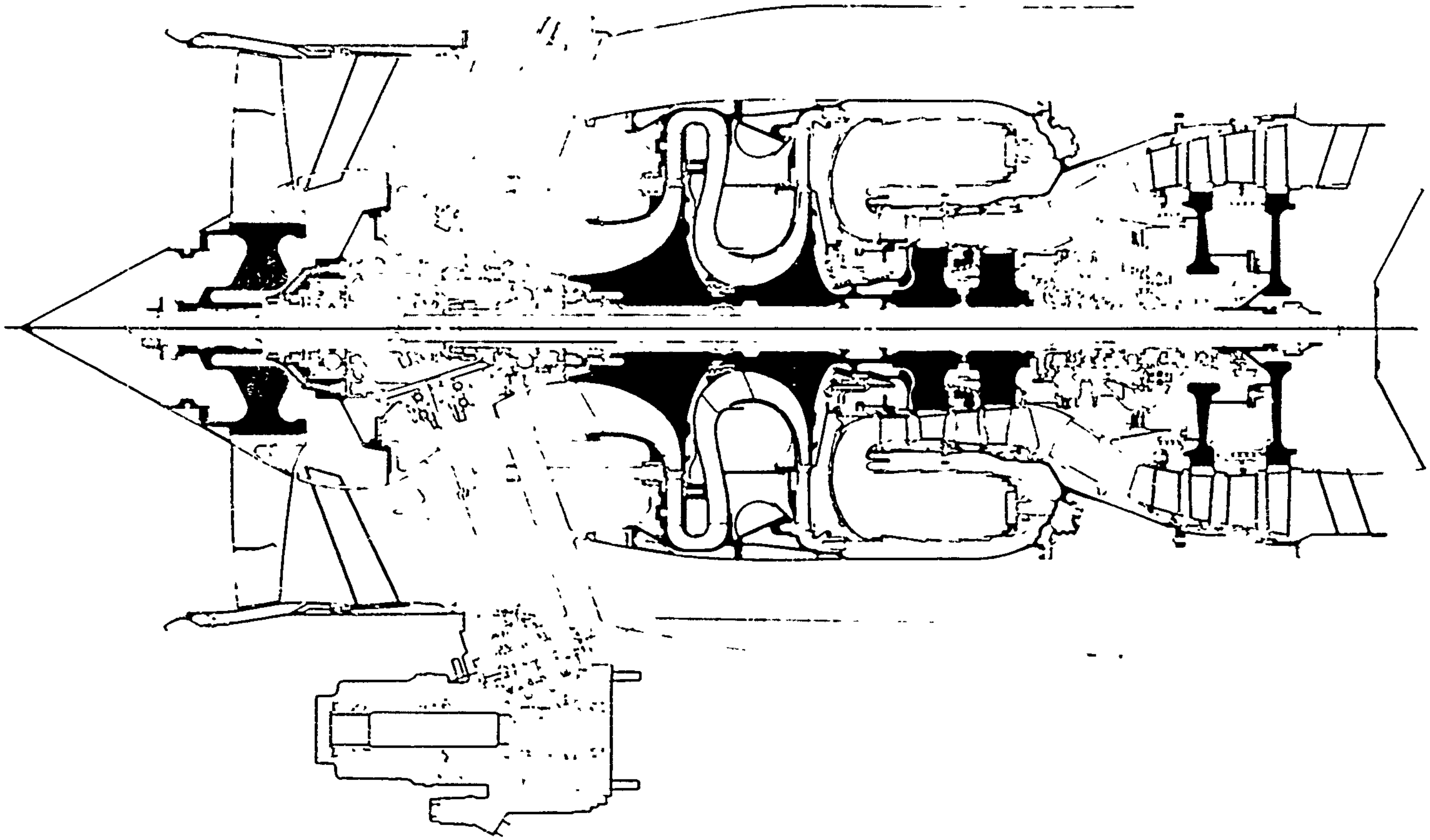
F109 TURBOFAN ENGINE

1330 LBS THRUST SLS TO STD DAY



G6-273-3

F109 ENGINE CONFIGURATION



G6-273-4

ENGINE DESIGNED AND DEVELOPED TO ENSIP CRITERIA

ENSIP OFFERS SIGNIFICANT BENEFITS

- RIGOROUS DESIGN VALIDATION AND VERIFICATION
- HIGHER MATURITY AT PRODUCTION
- IMPROVED RELIABILITY/DURABILITY/AVAILABILITY
- UNPARALLELED SAFETY (DAMAGE TOLERANCE)
- LOWER OPERATING COST

G6-273-5

GARRETT

DAMAGE TOLERANT DESIGN CONSIDERATIONS DEFINED

- FRACTURE CRITICAL PARTS IDENTIFIED AS THOSE WHICH IMPACT FLIGHT SAFETY
  - NINE ROTATING PARTS
  - THREE STATIC PARTS
- DESIGN ANALYSIS FOR FRACTURE CRITICAL PARTS CONDUCTED ASSUMING INITIAL MATERIAL FLAWS OF 0.015 IN. x 0.030 IN.
- TWO INSPECTION INTERVALS REQUIRED WITH MINIMUM DETECTABLE SURFACE FLAW
- VERIFICATION OF CCGR LIFE ANALYSIS REQUIRED

GARRETT

G6-273-6

COMPREHENSIVE TEST PROGRAM NEARING COMPLETION

- MATERIALS CHARACTERIZATION

- STRAIN GAGE TEST
  - PERFORMED ON CORE, RIG, WHIRL PIT
  - ANALYTICAL STRESS PREDICTION TECHNIQUES VERIFIED

DAMAGE TOLERANCE - ANALYTICAL CORRELATION AND VERIFICATION

- ENGINE THERMAL SURVEY
  - MEASURES COMPONENT TEMPS AND BOUNDARY CONDITIONS

- CRACK GROWTH VERIFICATION SPECIMEN TEST
  - CRITICAL LOCATIONS OF CRITICAL ROTATING COMPONENTS REPRESENTED
  - MISSION LOADINGS SIMULATED
  - STRESS FIELDS DUPLICATED
  - MANUFACTURING FEATURES DUPLICATED

- DAMAGE TOLERANT ENGINE TEST TO BE COMPLETED IN ISR WILL BE A FINAL CHECK

- DAMAGE TOLERANCE WHIRL PIT TEST
  - FAN DISK - Ti-6-4
  - 2ND STG HPC IMPELLER - Ti-6-2-4-2
  - 1ST STG HPT DISK - WASPALOY B
  - ROOM TEMP SPIN PIT

G6-273-7

GARRETT

VERIFICATION OF CCGR CALCULATIONS REQUIRED TO ENSURE ADEQUATE ENSIP INSPECTION INTERVAL

- CRACK GROWTH SPECIMEN TESTS
- STATIC STRUCTURE DAMAGE TOLERANCE TESTS
- ROTATING COMPONENT DAMAGE TOLERANCE TESTS
- ISR ENGINE DAMAGE TOLERANCE TEST (2/3 COMPLETED)

ANALYTICAL CORRELATION AND VERIFICATION

GARRETT

G6-273-8

- F109 ENGINE DESCRIPTION/DAMAGE TOLERANCE CONSIDERATIONS
- SPECIMEN CCGR TESTING
- WHIRLPIT CCGR TESTING
- ENGINE CCGR TESTING
- CONCLUSIONS

G6-273-9

GARRETT

SPECIMEN PROGRAM ALLOWS FLEXIBILITY TO EVALUATE EFFECTS SEPARATELY AND IN COMBINATION

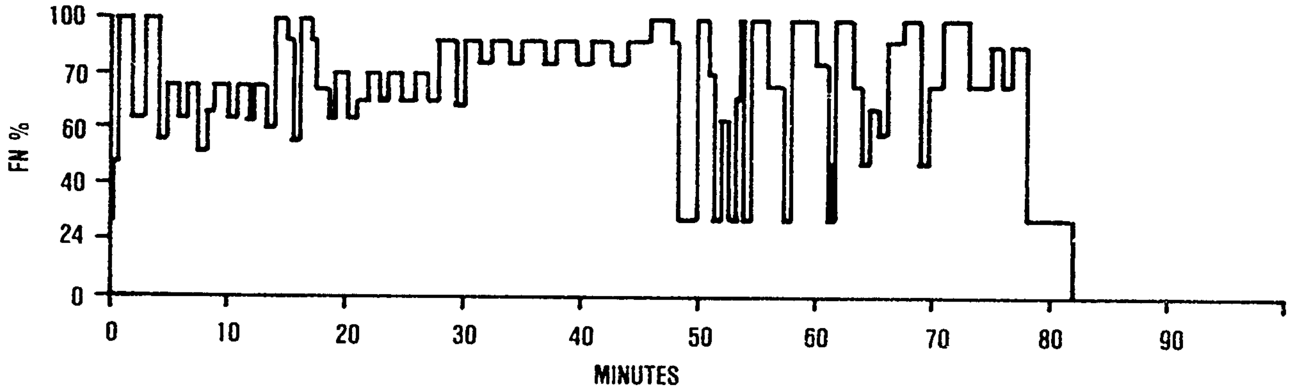
FACTORS CHARACTERIZED FOR EACH COMPONENT

- MISSION LOADING
- GEOMETRIC FEATURES
- STRESS FIELDS/GRADIENTS
- RESIDUAL STRESSES (OVERSPEED)
- TEMPERATURE EFFECTS

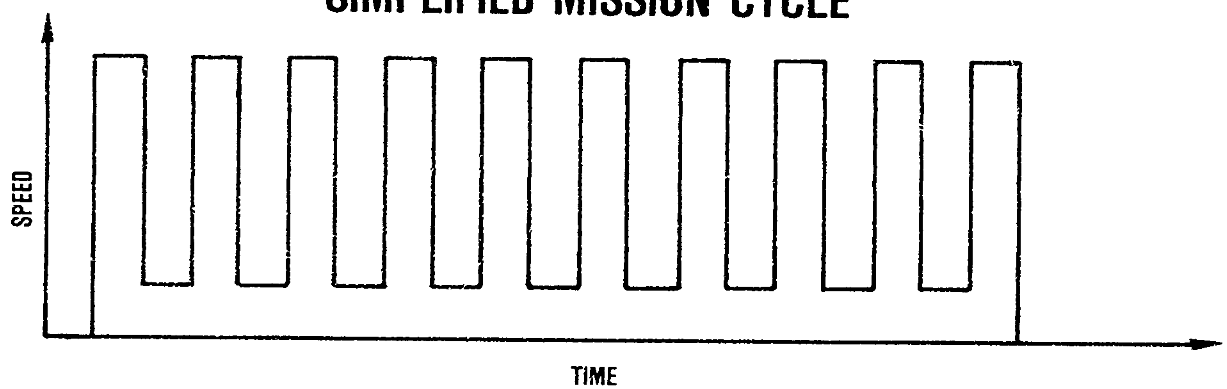
G6-273-10

MISSION PREDICTION PARALLELS AMT STRATEGY

F109 ENGINE MISSION CYCLE

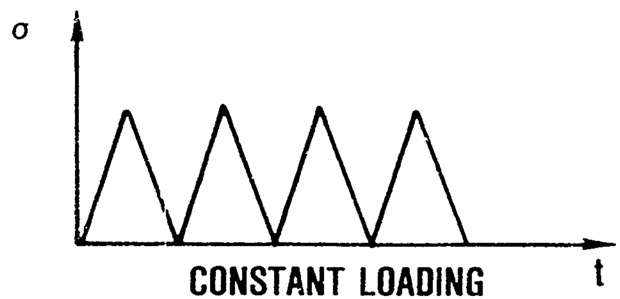


SIMPLIFIED MISSION CYCLE

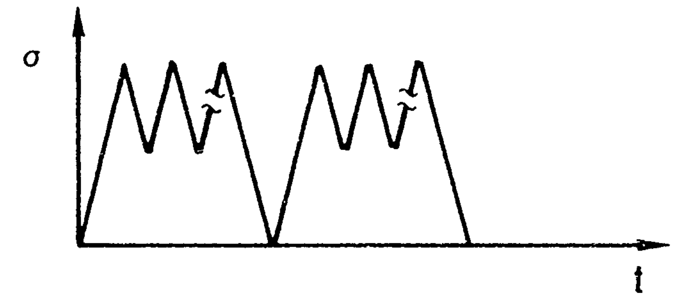


G6-273-11

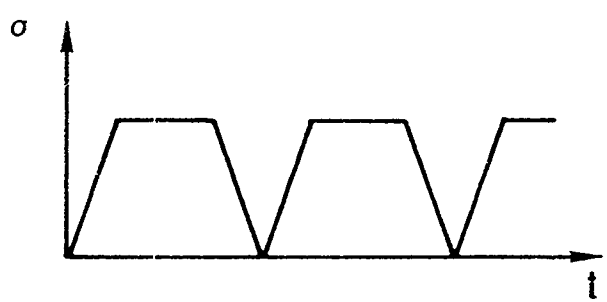
LOAD SPECTRA INCLUDES DWELL PERIODS AND MINOR CYCLES



CONSTANT LOADING



MISSION LOADING

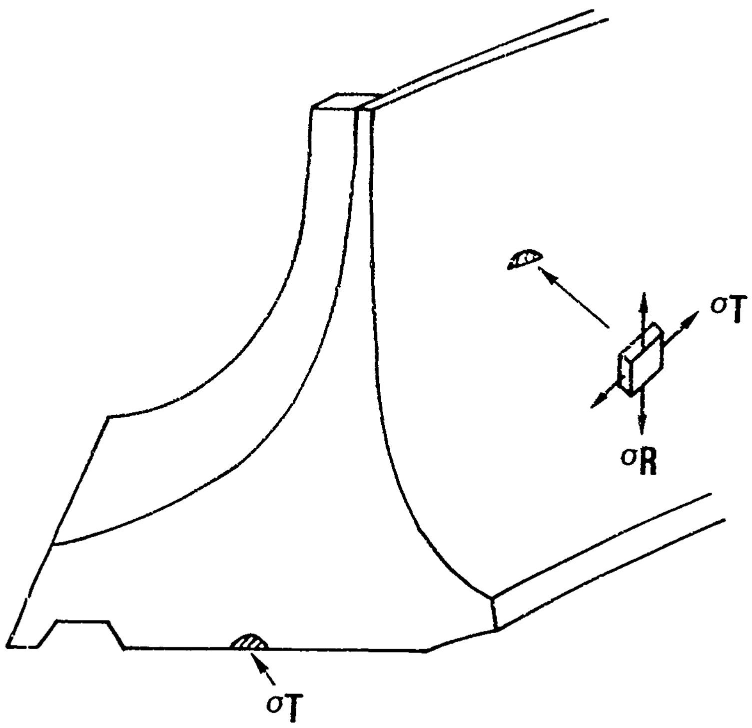


CONSTANT LOADING WITH HOLD TIME

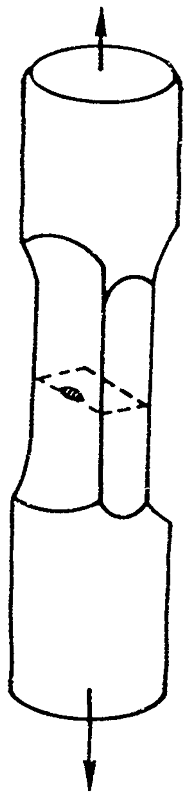
G6 273-12

K_b BAR SIMULATES 2ND STAGE BORE AND BACKFACE GEOMETRY/STRESS

BORE AND BACK FACE



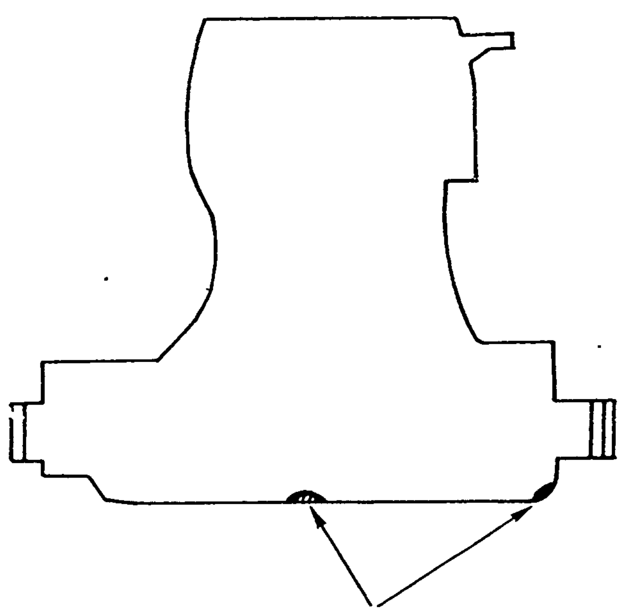
K_B-BAR SPECIMEN



G6-273-13

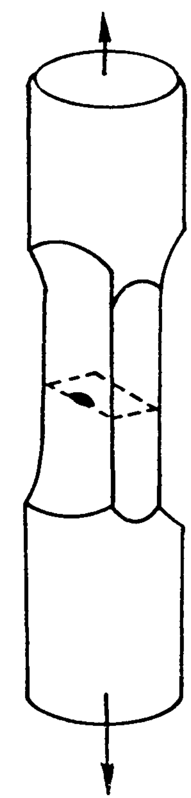
K_b BAR SIMULATES HP TURBINE BORE GEOMETRIES/STRESSES

HPT-1 DISK BORE



σ_T = TANGENTIAL STRESSES

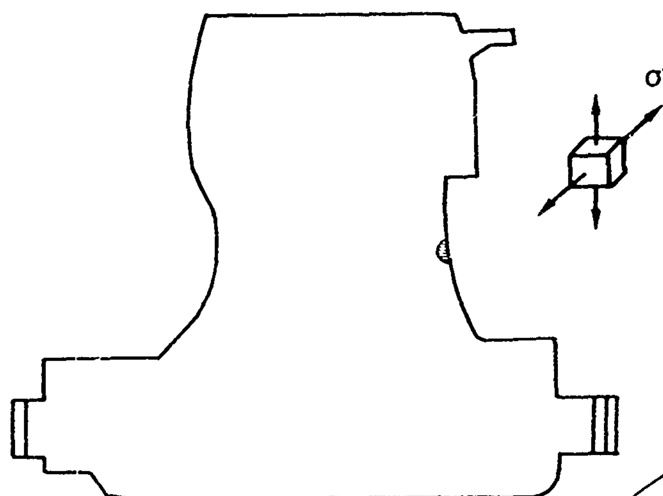
K_b-BAR SPECIMEN



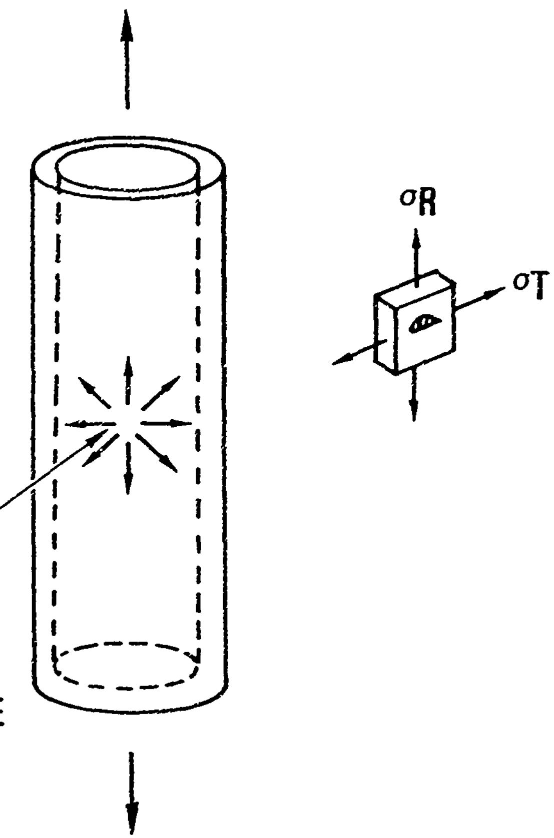
G6-273-14

SPECIAL TESTS REQUIRED TO SIMULATE HP TURBINE WEB STRESSES

HPT-1 DISK WEB

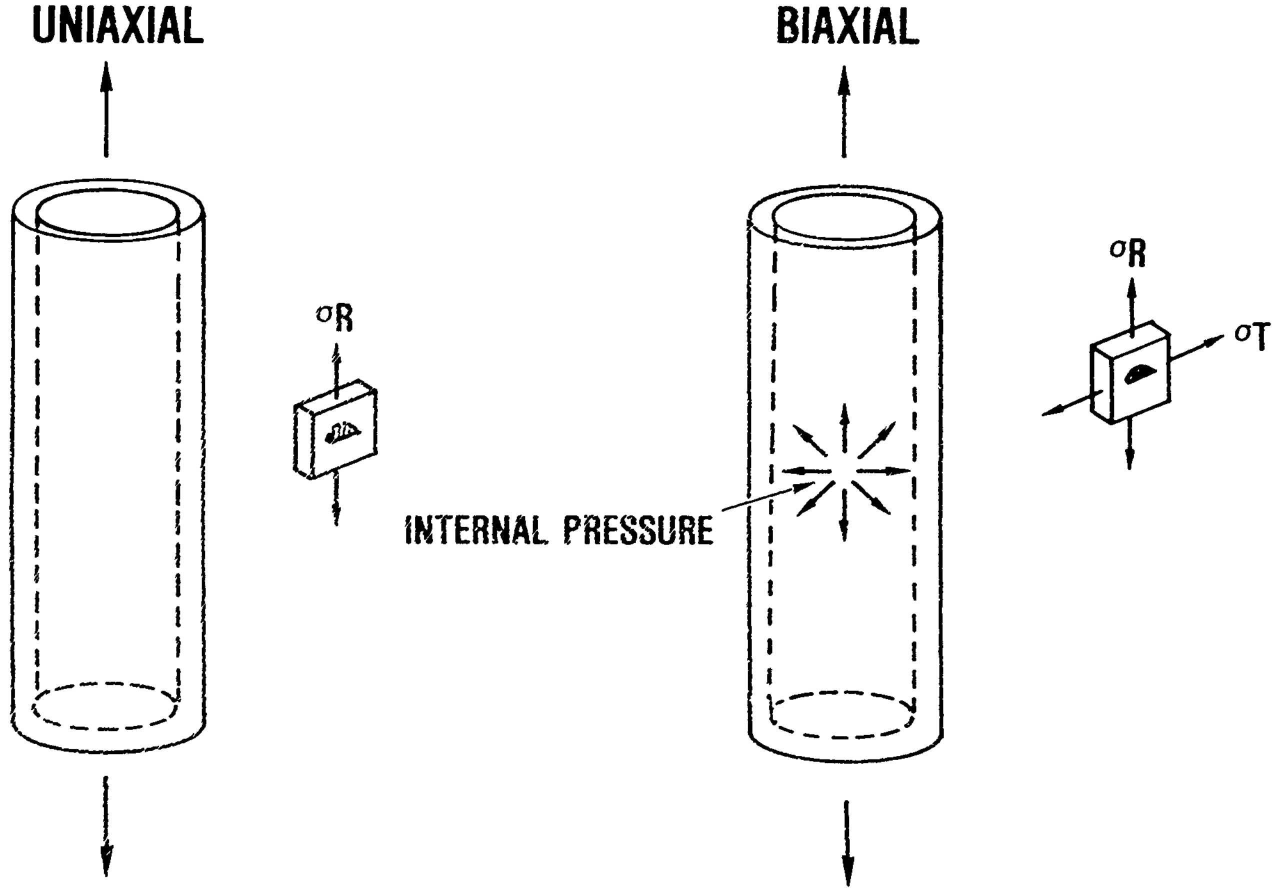


THIN-WALLED TUBULAR SPECIMEN



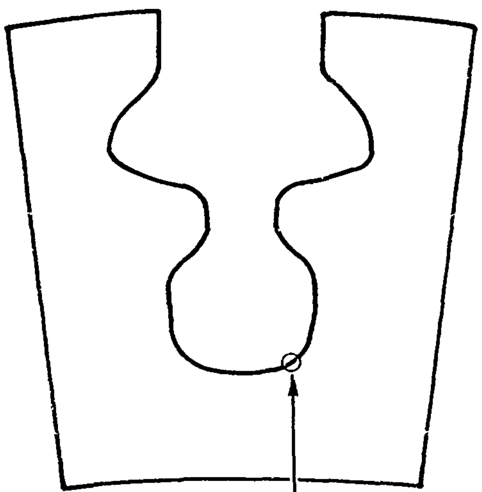
G6-273-15

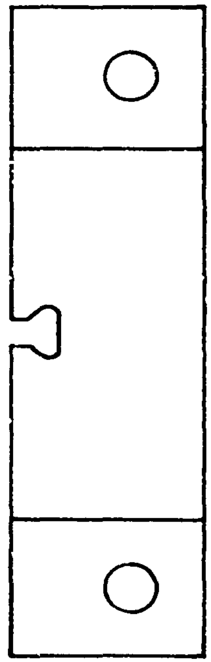
BIAXIAL TESTING IDENTIFIES EFFECT ON CRACK GROWTH



G6-273-16

SLOT SPECIMEN SIMULATES HP TURBINE ATTACHMENT GEOMETRIES/STRESSES



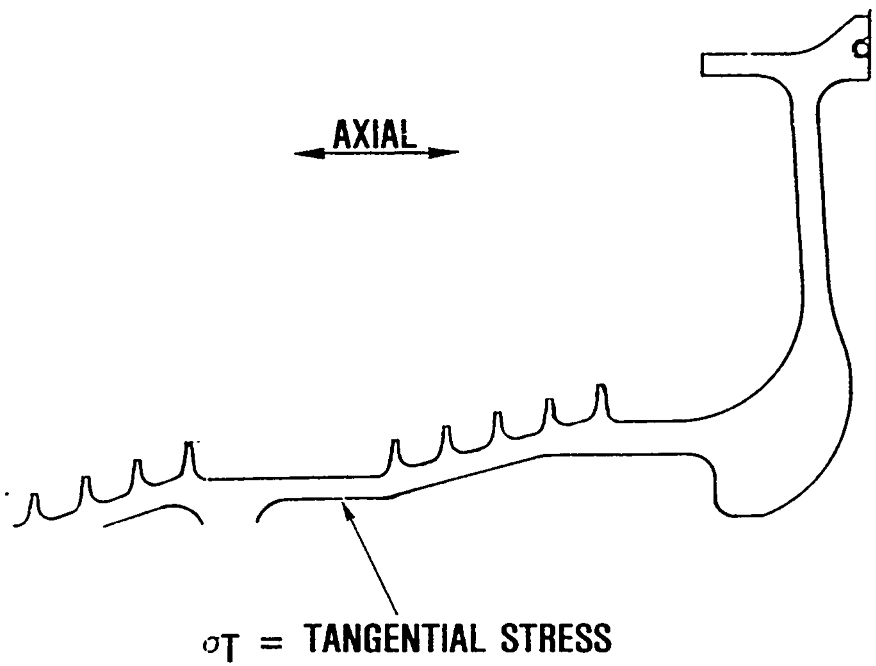


σP = MAX PRINCIPAL STRESS

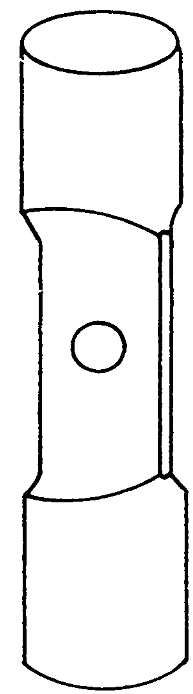
G6-273-17

SEAL PLATE SIMULATED WITH CENTER NOTCH

BONE AND FLOW HOLE



CENTER NOTCHED SPECIMEN FOR FLOW HOLE



G6-273-18

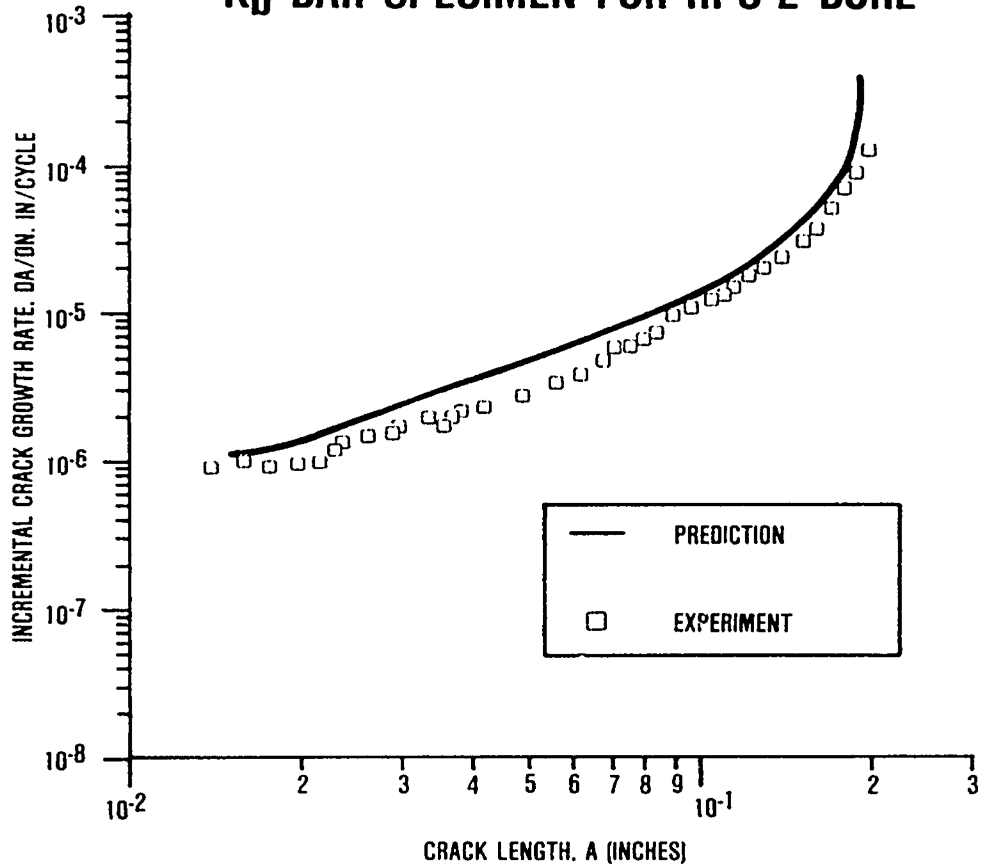
SIMULATION SPECIMEN TEST RESULTS

- GEOMETRIC FACTOR
- GEOMETRIC FACTOR + MISSION CYCLE
- GEOMETRIC FACTOR + MISSION CYCLE + STRESS GRADIENT

G6-273-19

COMPARISON UNDER CONSTANT STRESS AND CONSTANT CYCLE

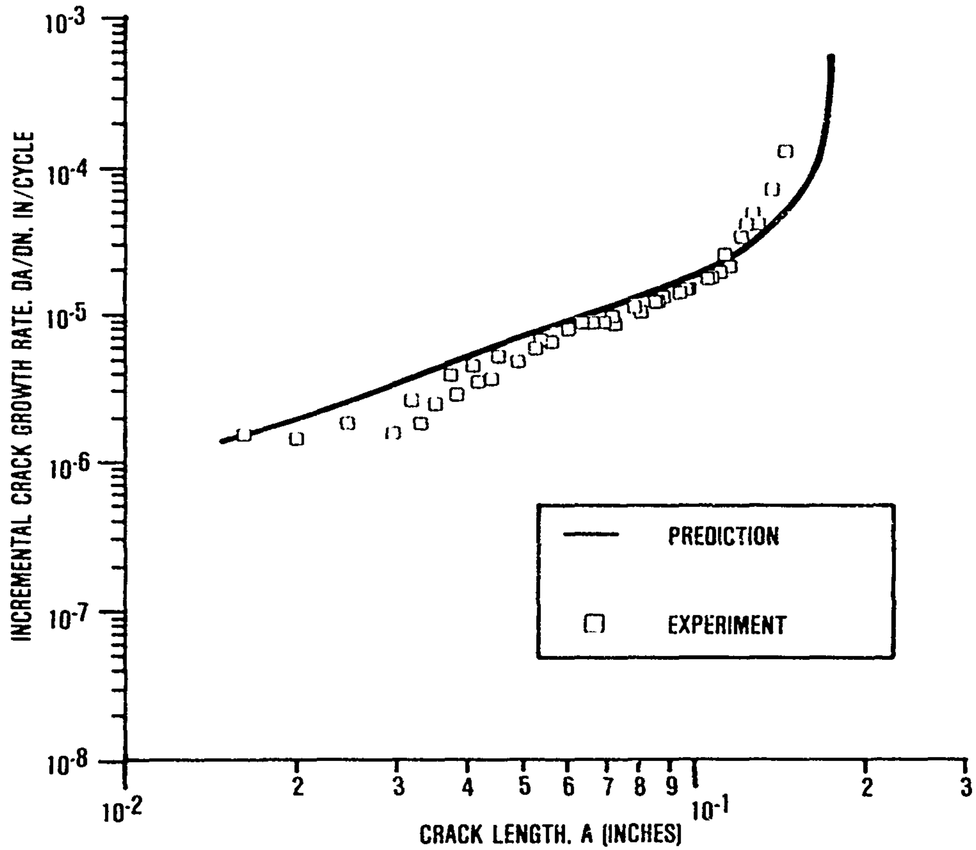
K_b BAR SPECIMEN FOR HPC-2 BORE



G6-273-20

COMPARISON UNDER CONSTANT STRESS AND CONSTANT CYCLE

K_b BAR SPECIMEN FOR HPT-1 DISK BORE



G6-273-21

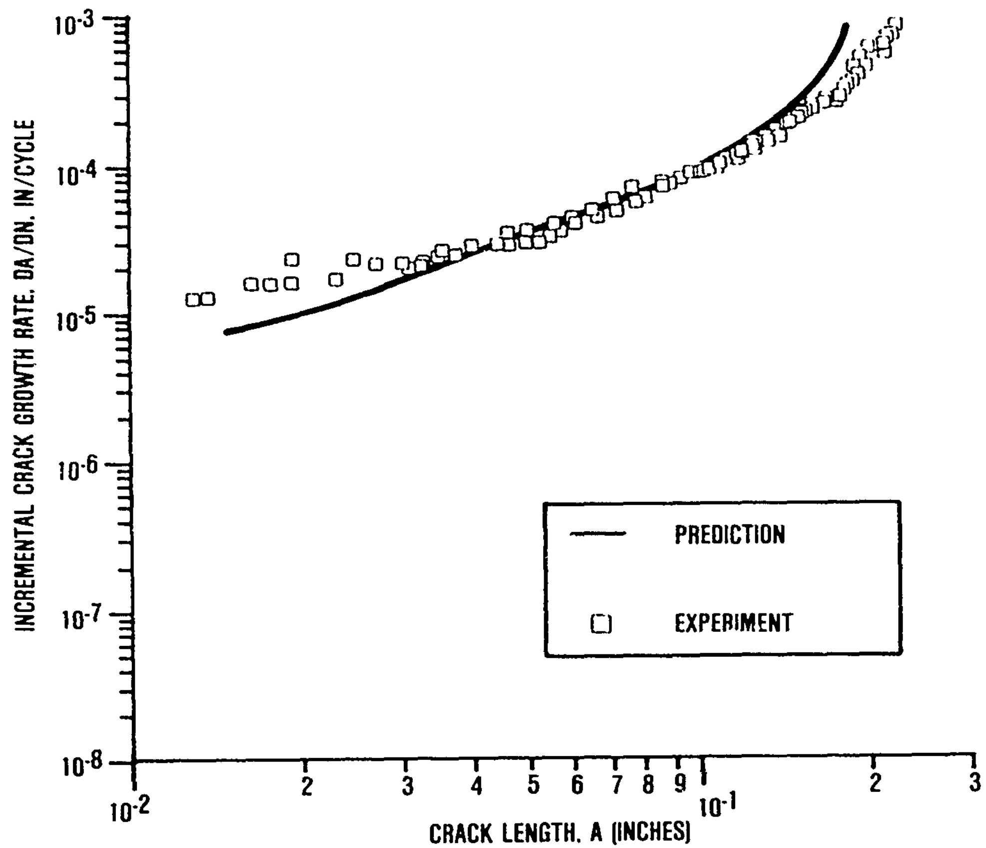
SIMULATION SPECIMEN TEST RESULTS

- GEOMETRIC FACTOR
- GEOMETRIC FACTOR + MISSION CYCLE
- GEOMETRIC FACTOR + MISSION CYCLE + STRESS GRADIENT

G6-273-22

COMPARISON UNDER CONSTANT STRESS AND MISSION CYCLE

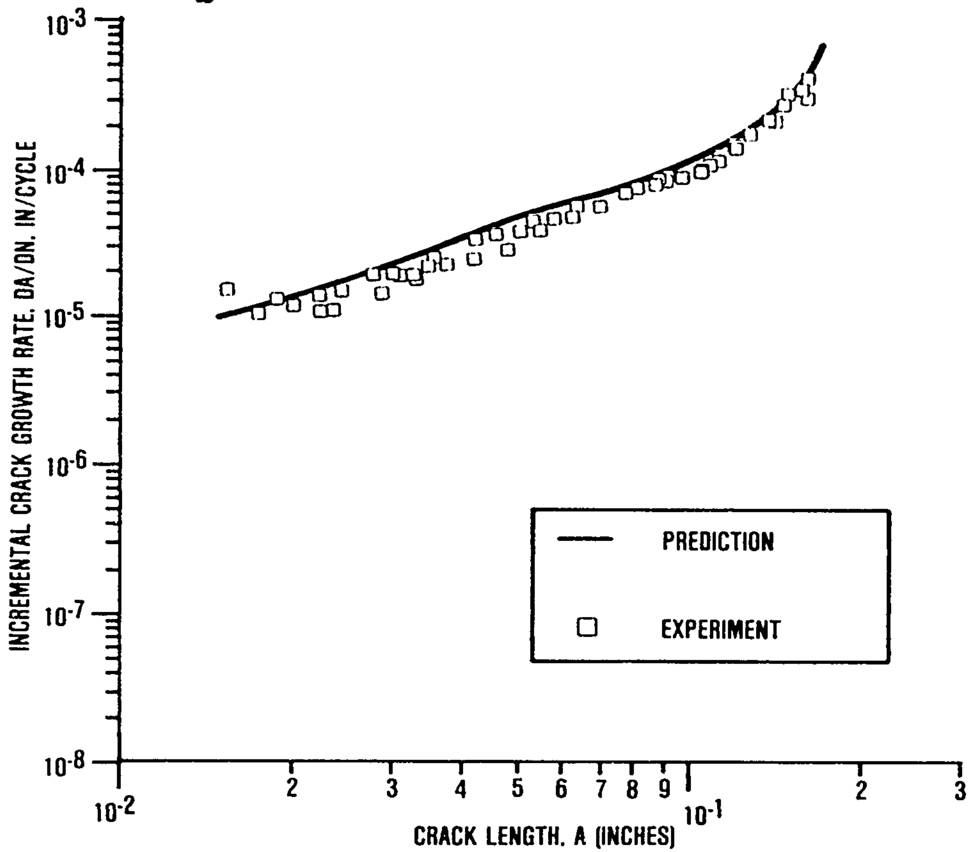
K_b BAR SPECIMEN FOR HPC-2 BORE



G6-273-23

COMPARISON UNDER CONSTANT STRESS AND MISSION CYCLE

K_b BAR SPECIMEN FOR HPT-1 BORE



G6-273-24

SIMULATION SPECIMEN TEST RESULTS

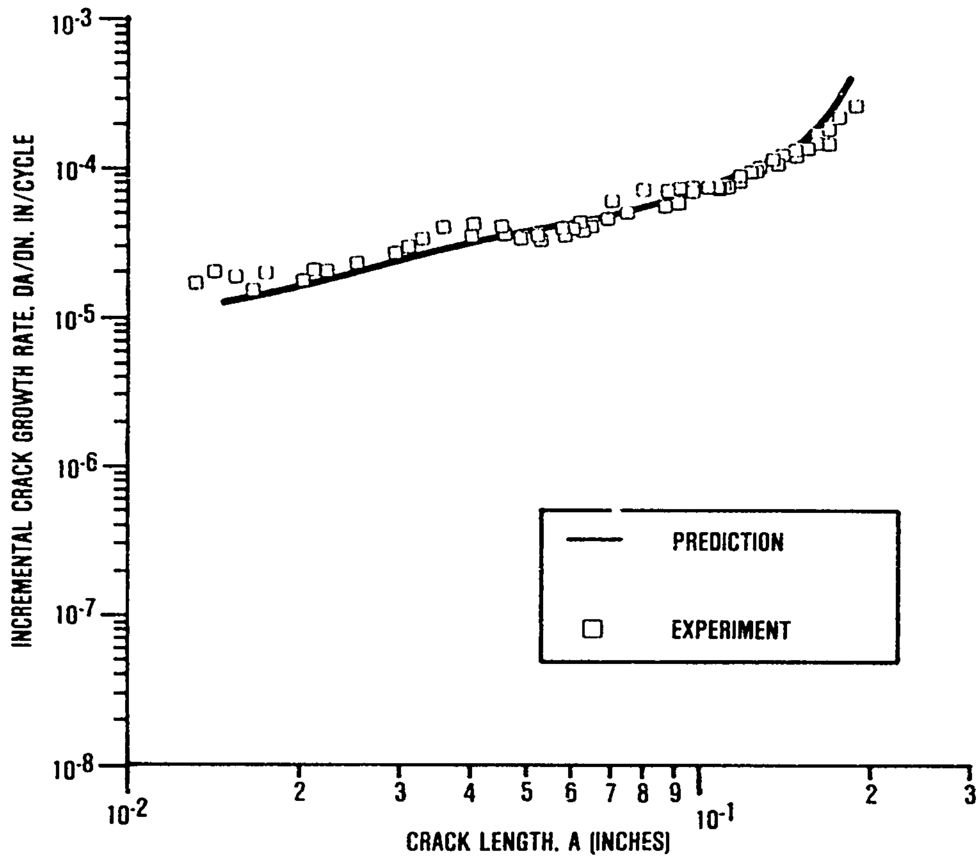
- GEOMETRIC FACTOR
- GEOMETRIC FACTOR + MISSION CYCLE
- GEOMETRIC FACTOR + MISSION CYCLE + STRESS GRADIENT

G6-273-25

GARRETT

COMPARISON UNDER STRESS GRADIENT AND MISSION CYCLE

SIMULATION SLOT SPECIMEN FOR HPT-1 DISK SLOT



G6-273-26

SPECIMEN TESTING SUCCESSFULLY COMPLETED

- 122 TOTAL SPECIMENS WERE TESTED
- LIFE PREDICTION IS EITHER CONSISTENT WITH THE RESULTS OF SIMULATION SPECIMEN TESTS OR IS SLIGHTLY CONSERVATIVE

G6-273-27

- F109 ENGINE DESCRIPTION/DAMAGE TOLERANCE CONSIDERATIONS
- SPECIMEN CCGR TESTING
- WHIRLPIT CCGR TESTING
- ENGINE CCGR TESTING
- CONCLUSIONS

G6-273-28

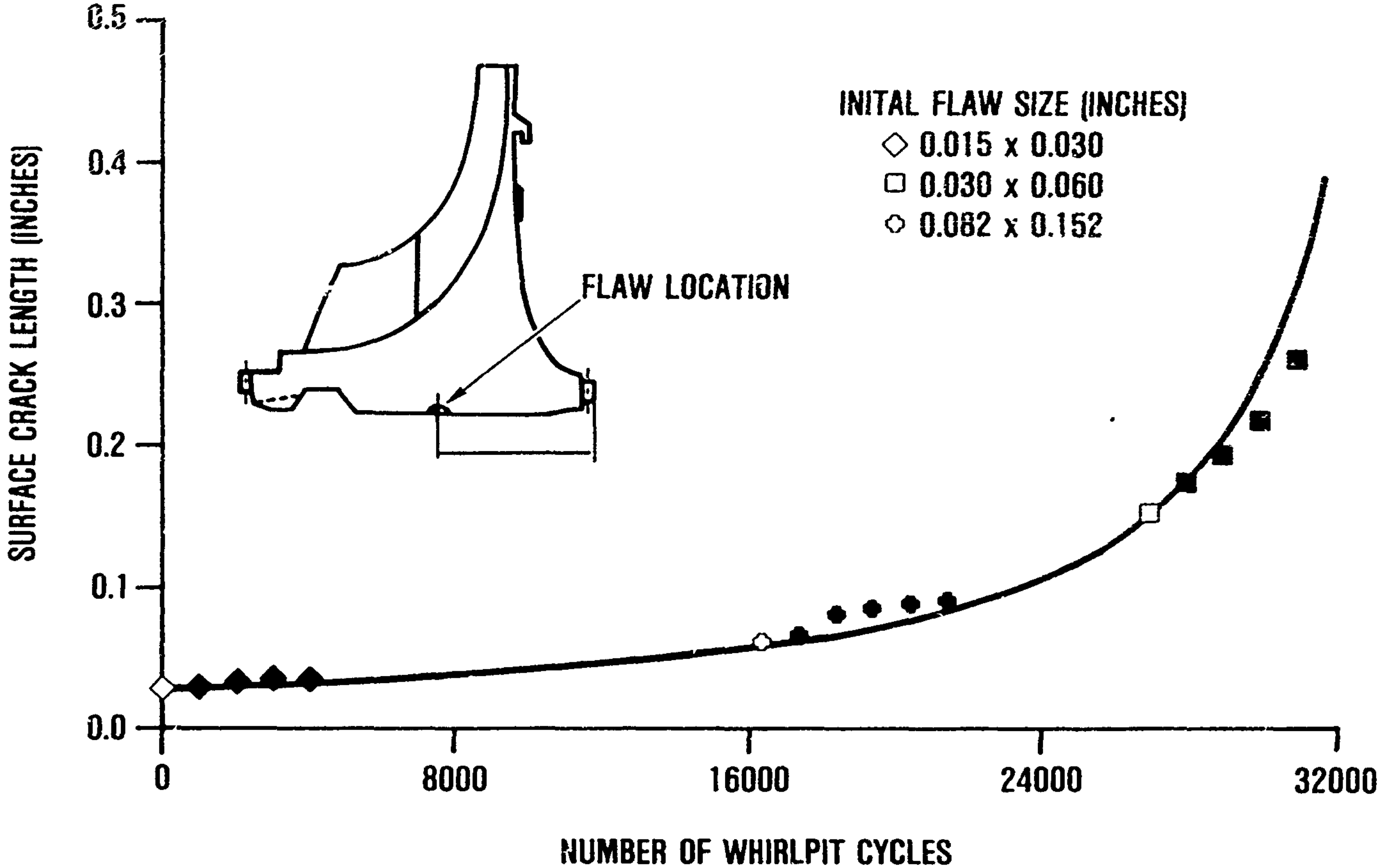
WHIRLPIT TEST OBJECTIVE WAS TO DETERMINE EFFECT OF ACTUAL PART GEOMETRY ON CRACK GROWTH RATE

- MACHINE FLAWS IN CRITICAL LOCATIONS
- PRE-TEST INSPECT:
  - FLUORESCENT PENETRANT
  - FLAW REPLICATION
- CYCLE ROTORS WITH IN-PROCESS INSPECTIONS
- POST-TEST INSPECT:
  - FINAL CRACK REPLICATION
  - HEAT TINT
  - DETERMINE CRACK DEPTH
  - VERIFY CRACK LENGTH

G6-273-29

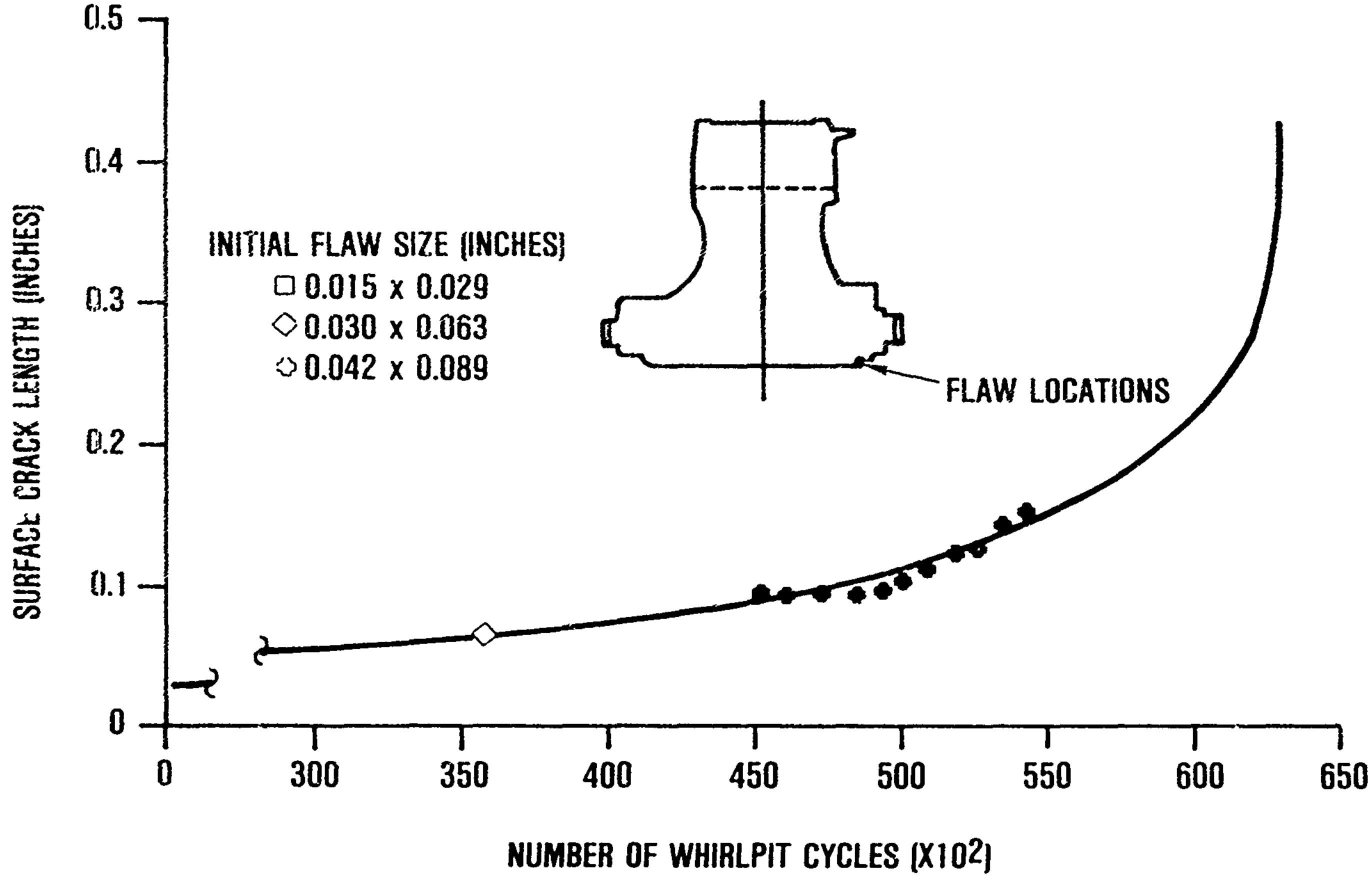
GARRETT

SECOND STAGE HP COMPRESSOR IMPELLER WHIRLPIT TEST RESULTS



G6-273-30

FIRST STAGE HP TURBINE DISK WHIRL PIT TEST RESULTS



G6-273-31

WHIRLPIT DAMAGE TOLERANCE TEST SUCCESSFULLY COMPLETED

- 3-D STRESS EFFECTS VERIFIED
- CRACK GROWTH CALCULATIONS IN GOOD AGREEMENT WITH TEST DATA
- CRACK REPLICATION RESULTS IN GOOD AGREEMENT WITH MEASUREMENTS

G6-273-32

- F109 ENGINE DESCRIPTION/DAMAGE TOLERANCE CONSIDERATIONS
- SPECIMEN CCGR TESTING
- WHIRLPIT CCGR TESTING
- ENGINE CCGR TESTING
- CONCLUSIONS

G6-273-33

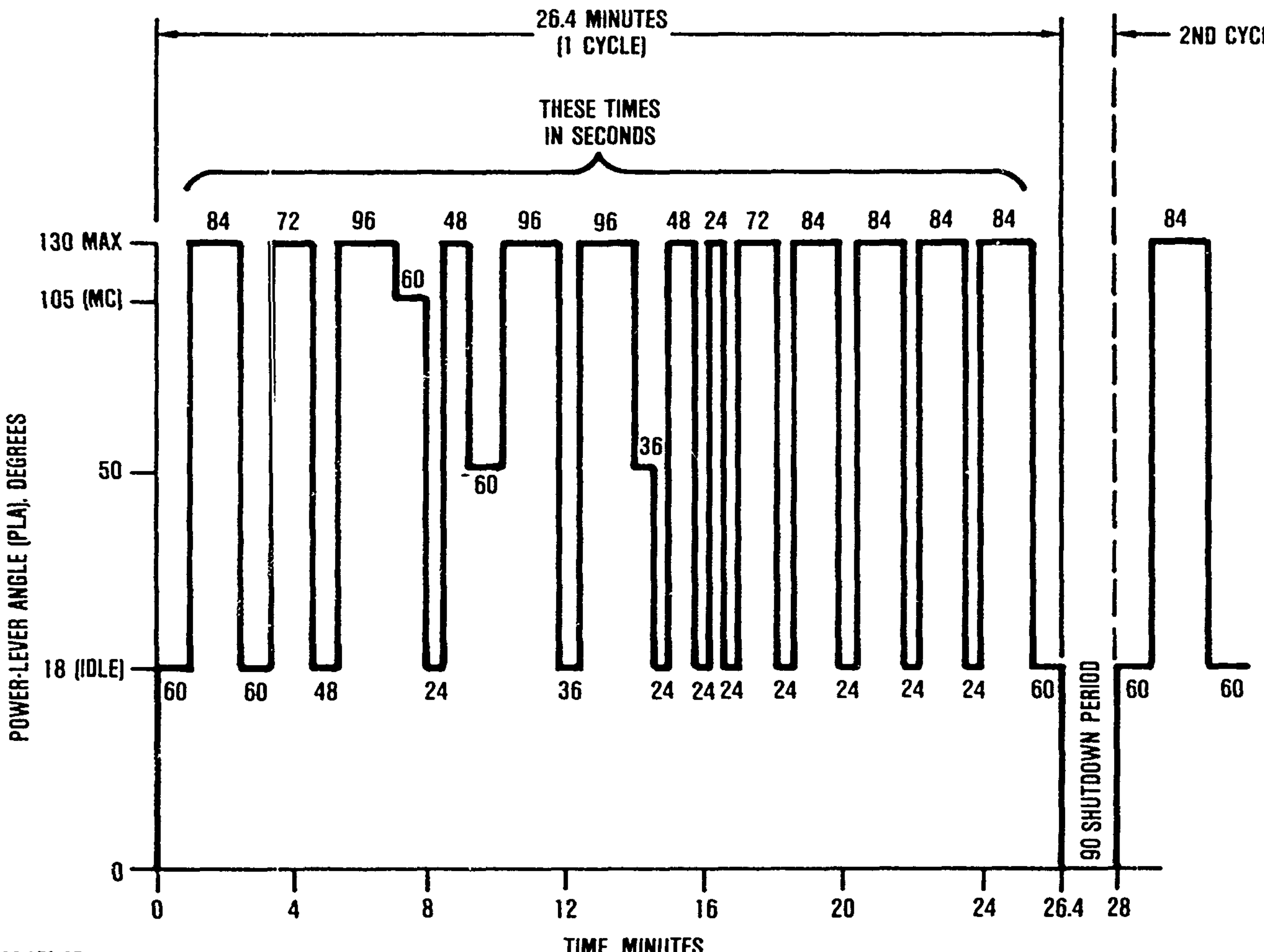
ENGINE OPERATIONAL EFFECTS ON CRACK GROWTH RATES DETERMINED IN ENGINE AMT TEST

- MACHINE FLAWS IN CRITICAL LOCATIONS
- CYCLE IN WHIRLPIT UNTIL CRACK INITIATES FROM FLAW
- PRE-TEST INSPECT
- ENGINE TEST USING ACCELERATED MISSION CYCLE
- POST-TEST INSPECT

GARRETT

G6-273-34

ACCELERATED MISSION TEST DEFINITION

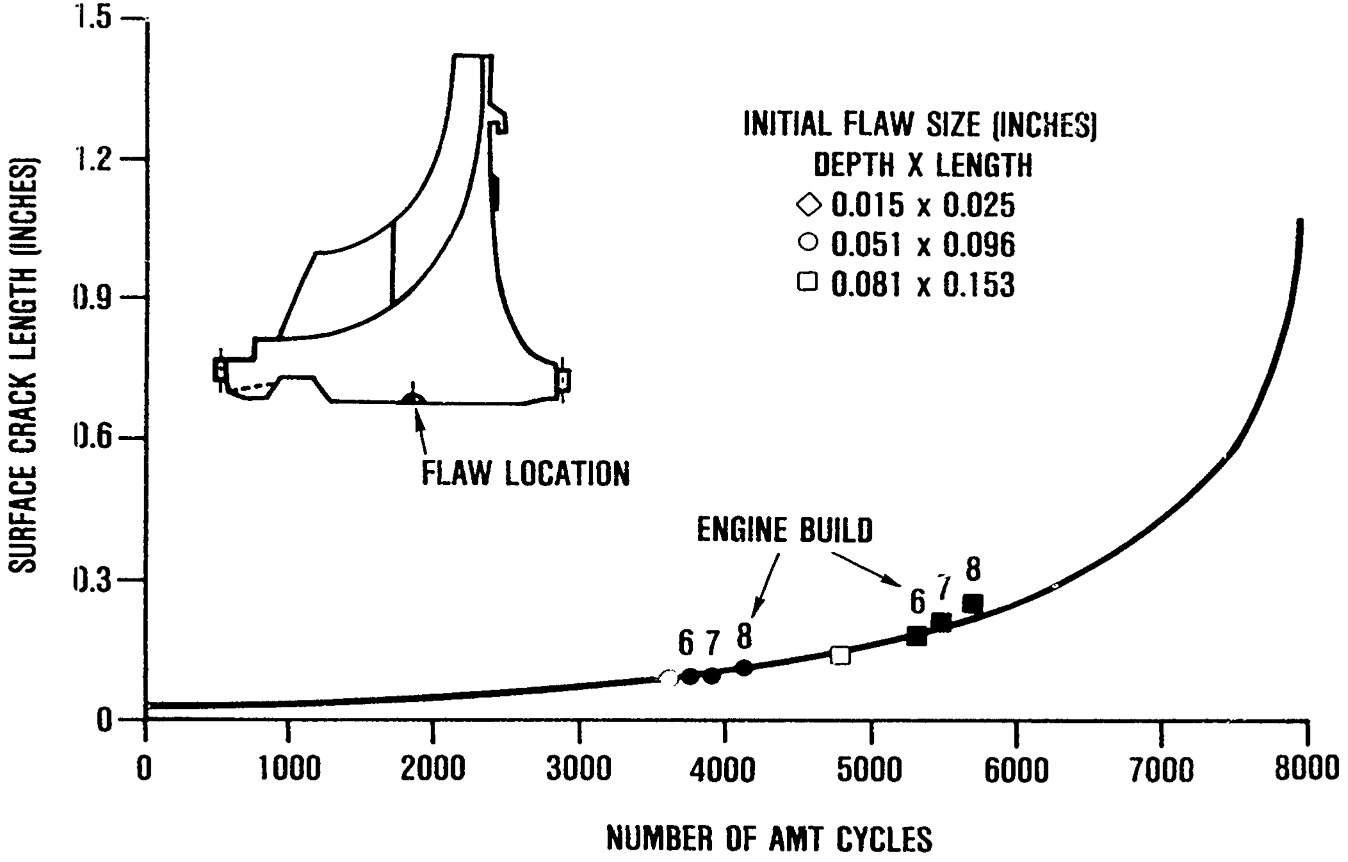


LEGEND

TOTAL TIME	26.4 MIN
TIME MAX	16.2 MIN
TIME MAX. CONT. (MC)	1.0 MIN
1-MAX-1	12 CYCLES
0-MAX-0	1 CYCLE

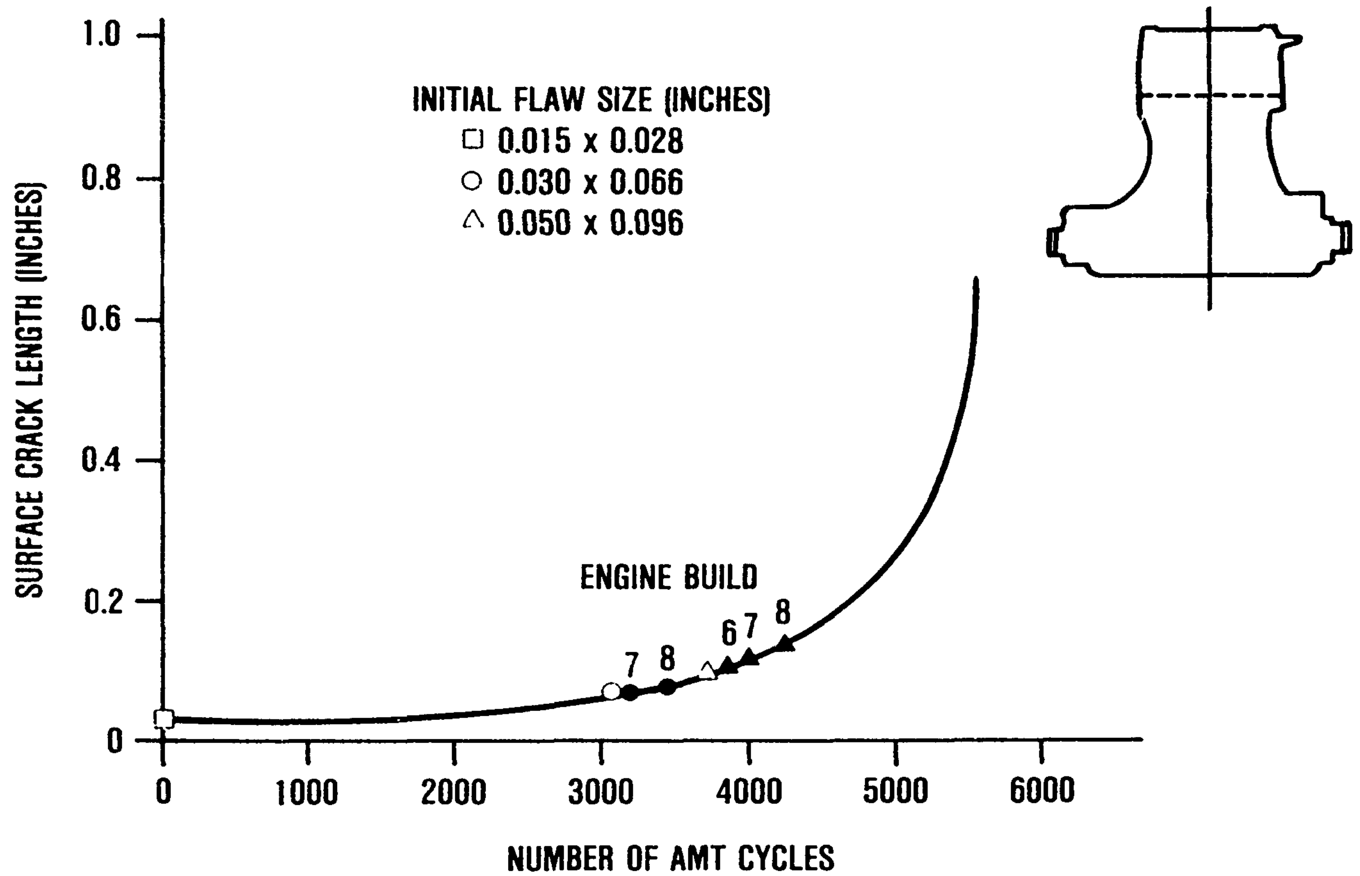
G6-273-35

SECOND STAGE HP COMPRESSOR IMPELLER AMT ENGINE TEST RESULTS



G6-273-36

FIRST STAGE HP TURBINE DISK AMT ENGINE TEST RESULTS



G6-273-37

AMT ENGINE DAMAGE TOLERANCE TEST NEARING COMPLETION

- ENGINE OPERATIONAL EFFECTS VERIFIED
- RESULTS TO DATE IN EXCELLENT AGREEMENT WITH ANALYTICAL PREDICTIONS

G6-273-38

- F109 ENGINE DESCRIPTION/DAMAGE TOLERANCE CONSIDERATIONS
- SPECIMEN CCGR TESTING
- WHIRLPIT CCGR TESTING
- ENGINE CCGR TESTING
- **CONCLUSIONS**

G6-273-39

GARRETT

DAMAGE TOLERANCE PROGRAM TEST RESULTS ARE IN EXCELLENT AGREEMENT WITH ANALYTICAL PREDICTIONS

- BUILDING BLOCK APPROACH PROVIDES NEEDED KNOWLEDGE OF ALL PERTINENT FACTORS FOR DAMAGE TOLERANCE VERIFICATION
- SPECIMEN TESTING PROVIDES EXCELLENT TOOL TO EVALUATE EFFECTS SEPARATELY AND IN COMBINATION
- WHIRLPIT TEST RESULTS VERIFY EFFECT OF ACTUAL COMPONENT GEOMETRY
- ENGINE OPERATIONAL EFFECTS VERIFIED BY ENGINE AMT TEST

DAMAGE TOLERANCE PROGRAM HAS VERIFIED CRACK GROWTH RATE CALCULATION PROCEDURE

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GARRETT

MSC Nastran Composite Laminate Analysis

By

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DATE

STRUCTURES & DYNAMICS TECHNICAL REPORT

"MSC NASTRAN COMPOSITE LAMINATE ANALYSIS"

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1.0 INTRODUCTION

1.1 With the increasing use of composite materials in aircraft structures, greater use of computer-based analysis techniques will be required to ensure structural integrity. A major item of software with a composite laminate analysis capability is the MSC NASTRAN finite-element code that is widely used in both government and commercial applications.

1.2 The intent of this presentation is to give a brief overview of the capability of MSC NASTRAN and give an example of its use for analysis of the F-15 radome.

2.0 MSC NASTRAN COMPOSITE ANALYSIS CAPABILITY

2.1 Details of the MSC NASTRAN composite analysis capability can be found in the "MSC NASTRAN Application Manual" [1]. MSC NASTRAN uses classical lamination theory to perform static, dynamic, eigenvalue and buckling analyses on four thin shell element types: QUAD4, QUAD8, TRIA3, and TRIA6. The elements are viewed as a stack of orthotropic materials for which data on ply thicknesses, material properties, and relative orientations are supplied by the user. Matrices of elastic moduli are calculated by MSC NASTRAN for internal calculation from the input data. For static analysis, which can include thermal and gravitational loads, stresses, and failure indices for individual plies, and interlaminar shear stresses bonding failure indices may be output. Three failure theories; Hill, Hoffman, and Tsai-Wu; may be used by NASTRAN for calculation of failure indices.

2.2 MSC NASTRAN accepts material and element properties for composite laminates by reading MAT8 and PCOMP cards respectively. The MAT8 card supplies the elastic properties of a single ply as in Figure 1. The PCOMP card defines the actual layup of the composite, including the thickness, orientation, and material identification of each ply; as well as defining the output data format. An example of a PCOMP card is shown in Figure 2.

2.3 For a composite eleme t, MSC NASTRAN produces output in two forms, both of which are shown in Figure 3. Data available for each ply of the layup are stresses in the fiber and matrix directions, interlaminar stresses, principal stresses, failure indices for each ply, and interlaminar bonding stresses. Interlaminar bonding stresses are calculated by an approximate technique which does not take account of edge effects, so these results must be used with care.

2.4 When MSC NASTRAN is given element and material properties on MAT8 and PCOMP cards, it calculates equivalent MAT2 and PSHELL cards. Each PCOMP card is replaced by a single PSHELL card which references up to four MAT2 cards. The MAT2 cards describe the

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"smeared" elastic properties of the whole layup, in matrix form, for membrane loads, bending loads, transverse shear and membrane-bending coupling. An example of the MAT2 and PSHELL cards created by MSC NASTRAN is in Figure 4. In this example, only three MAT2 cards have been created as the transverse shear properties of the material were not entered on the MAT8 cards.

2.5 The PSHELL and MAT2 created by NASTRAN can be reformatted for re-input (as they are in Figure 4) to treat the composite layup as a single layer orthotropic material. The advantage of this technique is that these cards and the output obtained when they are used is compatible with the SDRC SUPERTAB pre- and post- processing in use at WR-ALC. This allows easy identification of the stress distribution in the model and enables model error checking. The stress output if these cards are used takes the form shown in Figure 5. In this case, the stresses expressed along the element co-ordinate axis, the principal stresses and the Von-Mises stresses are output for both the top and bottom surfaces of the element.

3.0 F-15 RADOME FINITE ELEMENT MODEL

3.1 As an example of the MSC NASTRAN capability, an analysis of the F-15 radome is presented here. The analysis has been carried out as a step in determining structural repair limits for the radome. Figure 6 shows a radome with typical repairs and Figure 7 shows the geometry of these repairs which remove all of the original radome material. MSC NASTRAN cannot easily be used to analyze these repairs, but stress output obtained from the unrepaired structure can be used as input to other analysis software such as the Grumman CADAS programs [2].

3.2 The radome is composed of a composite shell that is fixed to the aircraft structure through an aluminum support ring. The composite shell is a five layer, filament-wound fiberglass-epoxy structure that has a [90°/0°/90°/0°/90°] layup as shown in Figure 8. The 0° plies lie along the aircraft longitudinal axis. The shell tapers in thickness from the nose to the support ring; the individual plies are varied in thickness by the winding process. The geometry of the support ring is shown in Figure 9. The ring is attached to the composite shell with two rows of mechanical fasteners, and is attached to the fuselage forward bulkhead at four points.

3.3 The finite element model was constructed using the SUPERTAB pre- processing software which makes use of high resolution computer graphics to display modelling entities as they are generated. SUPERTAB does not support the composite modelling features of MSC NASTRAN, so it was used only to generate the geometry of the model. Details of the finite element model are shown in Figure 10.

3.4 The forward two-thirds of the radome is a solid of revolution about an axis 5.4 off the aircraft centerline. This portion was easily generated by SUPERTAB by creating one row of nodes and copying them circumferentially at 10 intervals. The remaining portion of the radome was created by entering loft data directly, and interpolating using SUPERTAB's spline and surface generators.

3.5 The composite shell was modelled using QUAD4 and TRIA3 thin shell elements. These elements support bending, membrane, and transverse shear loads, and are compatible with MSC NASTRAN's composite analysis capability. Because these are flat elements, included angles had to be kept less than 10° to minimize excess flexibility errors. The shell consists of twenty-nine circumferential rings of elements, referring to twenty separate property tables to account for the longitudinal change in thickness. In all, the composite shell was composed of 1008 QUAD4 elements and 36 TRIA3 elements.

3.6 The support ring was modelled as a ring of CBEAM elements with moments of inertia equivalent to the built-up section. The beam elements support bending in two directions, torsion, and axial loads. The beam elements were attached to the composite shell at existing nodes using the appropriate offset vectors to locate the axis.

3.7 Rigid bar elements model the stiff structure which attaches the support ring to the forward fuselage bulkhead at the four attach points. These four points were restrained for all six degrees of freedom. The most aft ring of nodes around the lower half of the radome were restrained in the longitudinal direction only, to simulate contact between the radome and the forward fuselage bulkhead under design load.

3.8 Material data used for the analysis so far have been typical properties for fiberglass-epoxy obtained from composite material textbooks [3,4]. These data are shown in Figure 11. Material data for the support ring were readily obtainable from the "Military Standardization Handbook - Metallic Materials and Elements for Aerospace Vehicle Structures" [5].

4.0 LOADS

4.1 Two different loads systems have been applied - a uniform gravitational load, and element pressure loads corresponding to the aerodynamic loads for the design flight condition.

4.2 The gravitational loads are input to NASTRAN using the GRAV card which defines the magnitude and direction of the field. To use this card, the element mass must be entered on the property cards. The low density of the composite shell resulted in very low stresses for gravitational fields up to 12g. It was concluded from this that for the purposes of this analysis, stresses due to inertial loads were negligable and could be ignored.

4.3 The critical flight condition for the radome is a -3g symmetrical pushdown maneuver, resulting in a positive pressure on the upper surfaces of the radome, and a negative pressure on the lower aft surfaces (Reference 6). Circumferential pressure distributions at four radome stations were taken from Reference 6 and interpolated to produce a smooth, continuous pressure over the entire surface. Plots of pressure versus degrees from vertical are shown in Figures 12A and 12B. Each curve on these figures corresponds to a particular ring of elements. Due to the symmetry of the flight condition, only half of the pressures were calculated. These pressures were then applied to the model using PLOAD4 cards which describe the magnitude of the normal pressure to the element surface. As the pressure distribution varies continuously over the surface, each of the pressures had to be input by hand.

5.0 ANALYSIS RESULTS

5.1 Figure 13 shows a plot of the Von Mises stress distribution in the outer surface of the radome as produced by SUPERTAB from the NASTRAN run using PSHELL and MAT2 cards.

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This plot allowed identification of the most highly stressed elements in the radome. Using the Tsai-Wu failure theory, a maximum failure index of 0.06 was calculated, indicating that the radome structure is generally not highly stressed and should be able to accept large repairs without detriment to its mechanical characteristics.

6.0 SUMMARY

6.1 MSC NASTRAN's composite laminate analysis capability is widely available and will be used more often as composite materials become more prevalent in aircraft structures. The use of MSC NASTRAN for the analysis of the F-15 radome has proven the value of this tool for the solution of aircraft structural problems.

7.0 FIGURES

8.0 REFERENCES

1. The MacNeal-Schwendler Corporation, "MSC/NASTRAN Application Manual (Version 65 Edition)", February 1986.

2. Grumman Aerospace Corporation, "User's Guide for Basic Advanced Composites Design/Analysis System", September 1985.

3. Brian C. Hoskin and Alan A. Baker, "Composite Materials for Aircraft Structures", American Institute of Aeronautics and Astronautics, 1986.

4. Bhagwan D. Agarwal and Lawrence J. Broutman, "Analysis and Performance of Fiber Composites", Wiley, 1976.

5. Department of Defense, "Military Standardization Handbook - Metallic Materials and Elements for Aerospace Structures", MIL-HDBK-5D, June 1983.

6. McDonnell Douglas Corporation, Report No. MDCA0833, Volume 5.

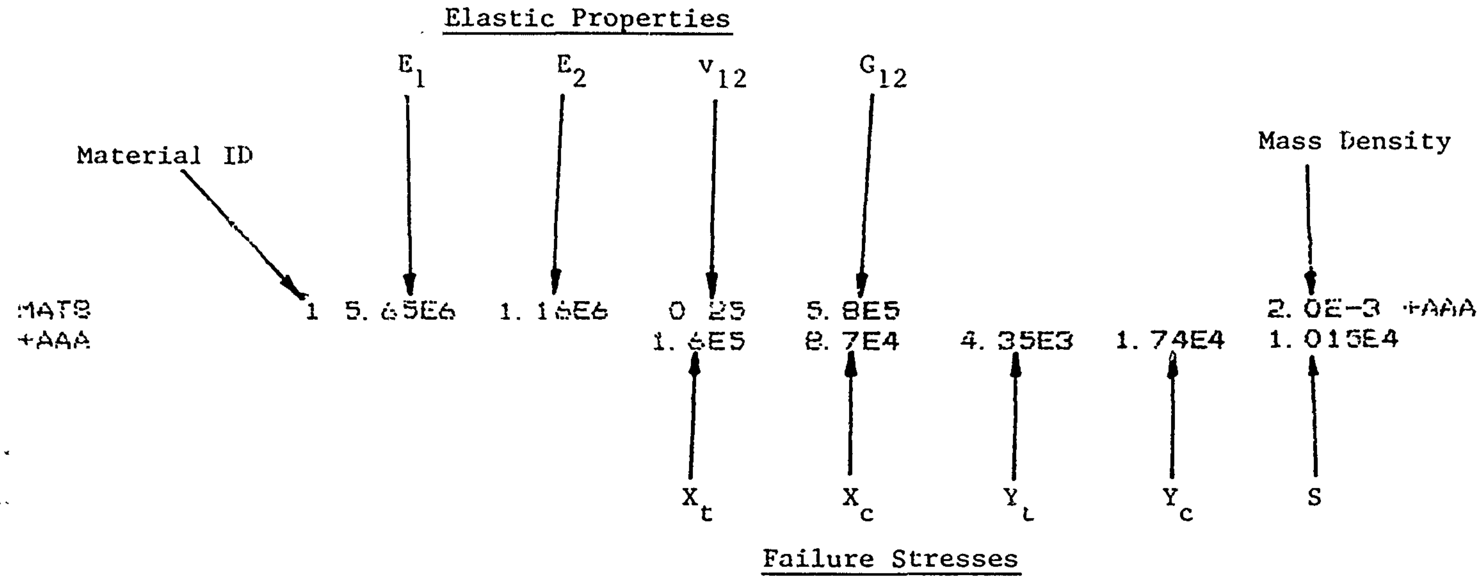


Figure 1: MAT8 Card

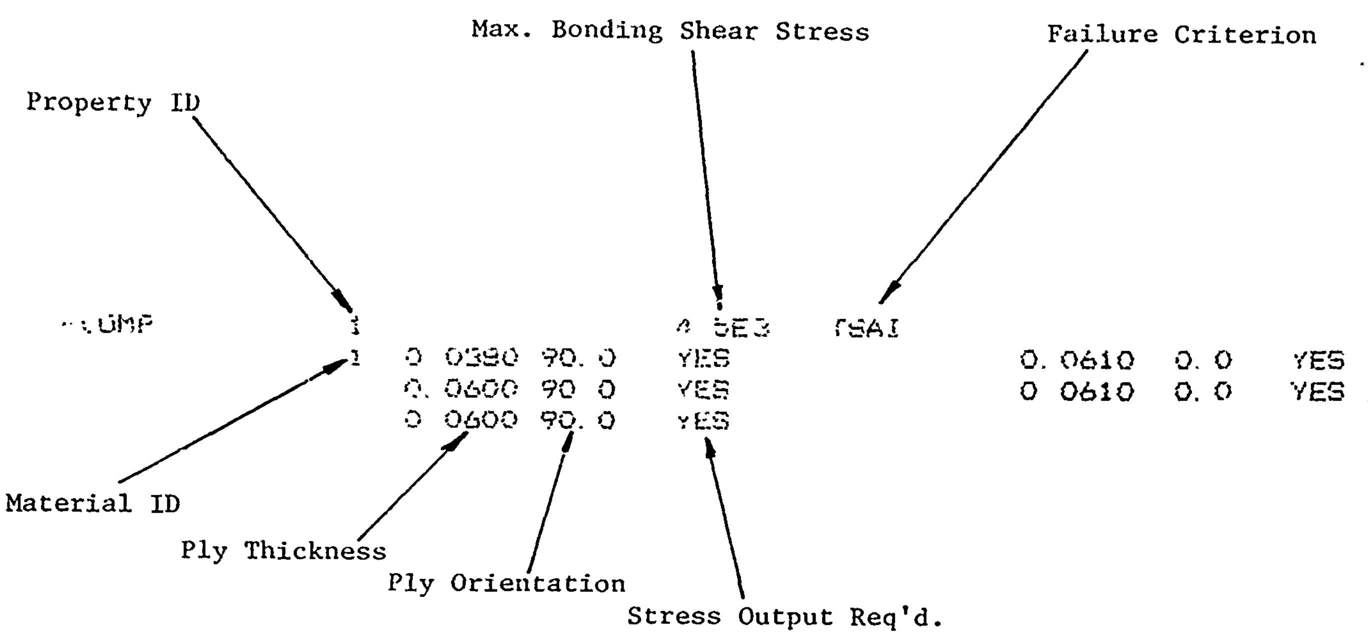


Figure 2: PCOMP Card

S T R E S S E S   I N   L A Y E R E D   C O M P O S I T E   E L E M E N T S   (Q U A D 4)

ELEMENT ID	PLY ID	STRESSES IN FIBRE AND MATRIX DIRECTIONS NORMAL-1	NORMAL-2	SHEAR-12	INTER-LAMINAR STRESSES SHEAR-1Z	SHEAR-2Z	PRINCIPAL STRESSES (ZERO SHEAR) ANGLE	MAJOR	MINOR	MAX SHEAR
942	1	2. 31252E+02	-3. 19707E-02	-4. 87399E+01	-4. 57866E-01	1. 67344E-01	-4. 61	2. 35180E+02	-3. 23635E+02	3. 04407E+02
942	2	-1. 63620E+03	8. 34881E+00	4. 29453E+01	-1. 38642E+00	1. 80838E-01	88. 51	9. 46951E+00	-1. 63732E+03	3. 23396E+02
942	3	4. 59666E+02	-3. 14981E+02	-3. 71403E+01	-1. 37693E+00	1. 68034E-01	-2. 74	4. 61442E+02	-3. 16757E+02	3. 39100E+02
942	4	-1. 64886E+03	4. 41194E+01	3. 13352E+01	-3. 70129E-01	1. 50088E-01	88. 94	4. 46992E+01	-1. 64944E+03	8. 47070E+02
942	5	6. 29215E+02	-3. 10489E+02	-2. 61169E+01	-2. 17952E-09	-7. 35013E-10	-1. 59	6. 29940E+02	-3. 11214E+02	4. 70577E+02

F A I L U R E   I N D I C E S   F O R   L A Y E R E D   C O M P O S I T E   E L E M E N T S   (Q U A D 4)

ELEMENT ID	FAILURE THEORY	PLY ID	FP=FAILURE INDEX FOR PLY (DIRECT STRESSES)	FB=FAILURE INDEX FOR BONDING (INTER-LAMINAR STRESSES)	FAILURE INDEX FOR ELEMENT MAX OF FP,FB FOR ALL PLIES	FLAG
942	TSAI-WU	1	-0. 0552			
				0. 0001		
		2	0. 0102			
				0. 0003		
		3	-0. 0554			
				0. 0003		
		4	0. 0165			
				0. 0001		
		5	-0. 0555			
					0. 0165	

Figure 3: MSC NASTRAN Composite Laminate Output

```
PSHELL*                1             101     0. 2800E+00        201*P101
*P101         0. 1000E+01               0     0. 1000E+01   0. 0000E+00*P102
*P102        -0. 1400E+00     0. 1400E+00            301

MAT2*                101     0. [illegible]E+07     0. 2938E+06  -0. 6874E-02*M101
*M101         0. 3742E+07    -0. [illegible]E+00     0. 5800E+06   0. 2000E-02*M102
*M102         0. 0000E+00     0. 0000E+00     0. 0000E+00   0. 0000E+00*M103
*M103         0. 0000E+00     0. 0000E+00     0. 0000E+00   0. 0000E+00*M104
*M104                  0

MAT2*                201     0. 241[illegible]E+07     0. 2938E+06  -0. 8858E-02*M105
*M105         0. 4482E+07    -0. [illegible]E+00     0. 5800E+06   0. 2000E-02*M106
*M106         0. 0000E+00     0. 0000E+00     0. 0000E+00   0. 0000E+00*M107
*M107         0. 0000E+00     0. 0000E+00     0. 0000E+00   0. 0000E+00*M108
*M108                  0

MAT2*                301     0. 7[illegible]E+05     0. 1954E-02   0. 2085E-0[illegible]*M109
*M109        -0. 7786E+05     0. 7175E-02     0. 3858E-02   0. 2000E-02*M110
*M110         0. 0000E+00     0. 0000E+00     0. 0000E+00   0. 0000E+00*M111
*M111         0. 0000E+00     0. 0000E+00     0. 0000E+00   0. 0000E+00*M112
*M112                  0
```

Figure 4: MSC NASTRAN Created MAT2 and PSHELL Cards

S T R E S S E S   I N   Q U A D R I L A T E R A L   E L E M E N T S   (Q U A D 4)

ELEMENT ID	FIBRE DISTANCE	STRESSES IN ELEMENT COORD SYSTEM NORMAL-X	NORMAL-Y	SHEAR-XY	PRINCIPAL STRESSES (ZERO SHEAR) ANGLE	MAJOR	MINOR	VON MISES
942	-1 351000E-01	-7 484731E+02	1. 189055E+02	9 109200E+01	84 0690	1. 283688E+02	-7 579363E+02	8 296031E+02
	1 351000E-01	-8 691321E+02	4 455564E+02	8 278340E+01	86 4111	4 507486E+02	-8 743243E+02	1. 166927E+03

Figure 5: Output from MAT2 and PSHELL Cards

Figure 6: Radome with Typical Repairs

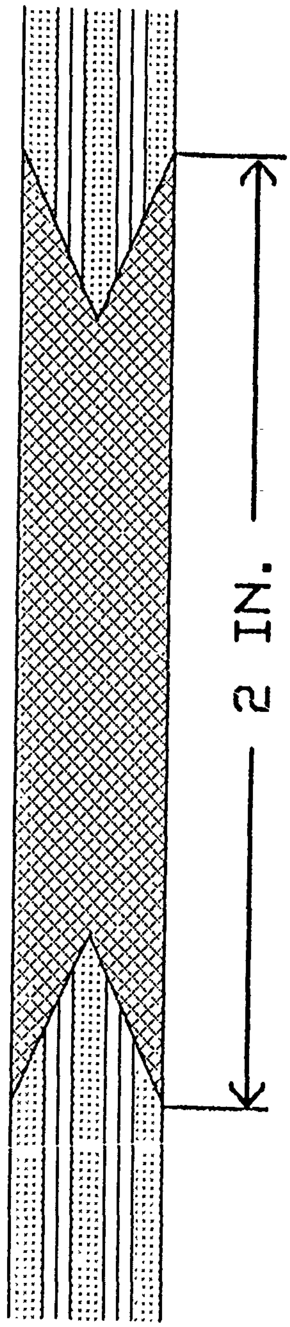


Figure 7: Typical Repair Geometry

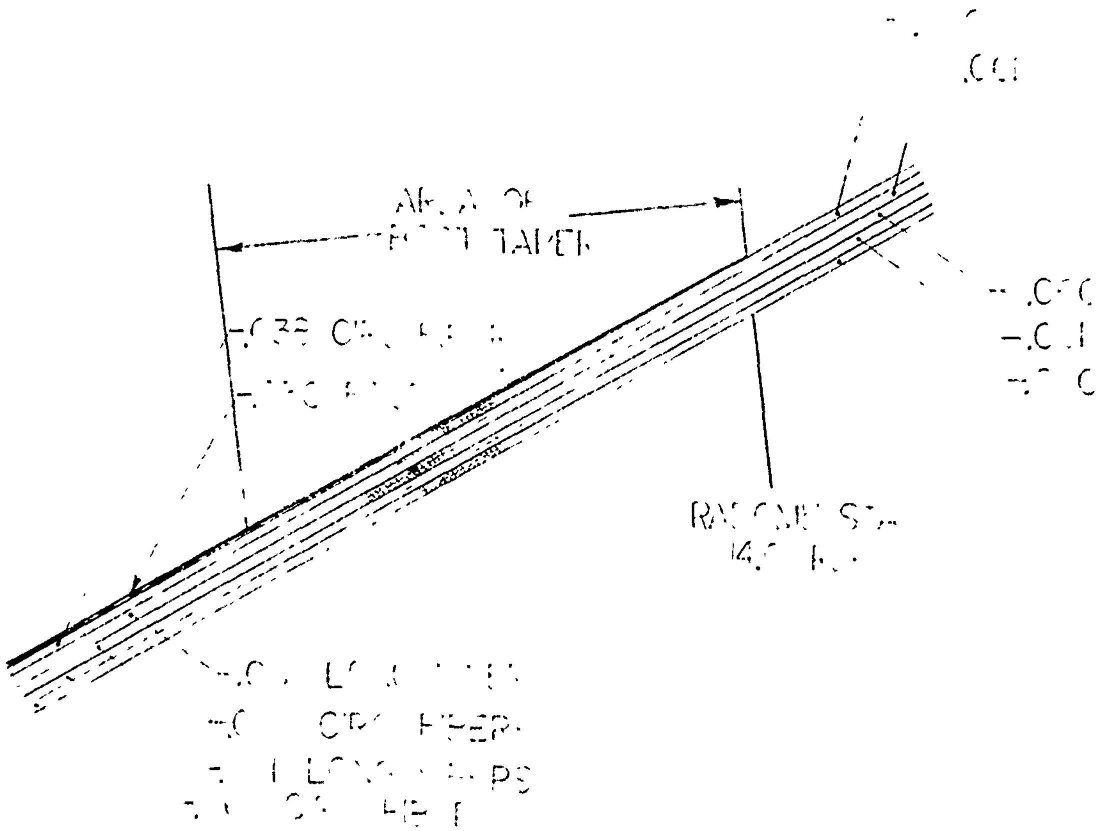


Figure 8: Radome Composite Shell Layup

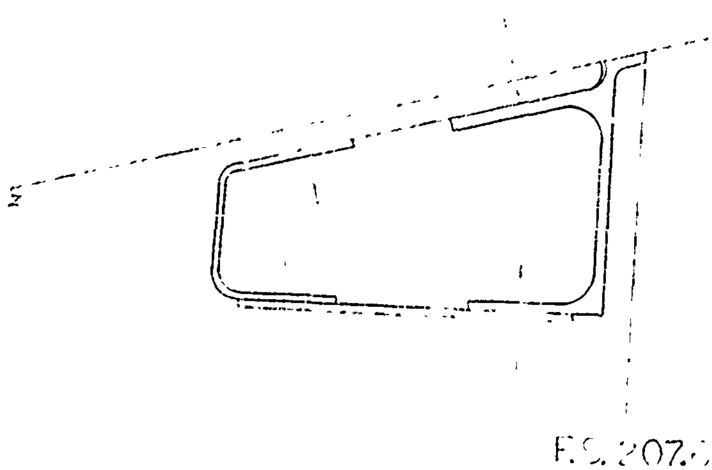


Figure 9: Radome Support Ring Geometry

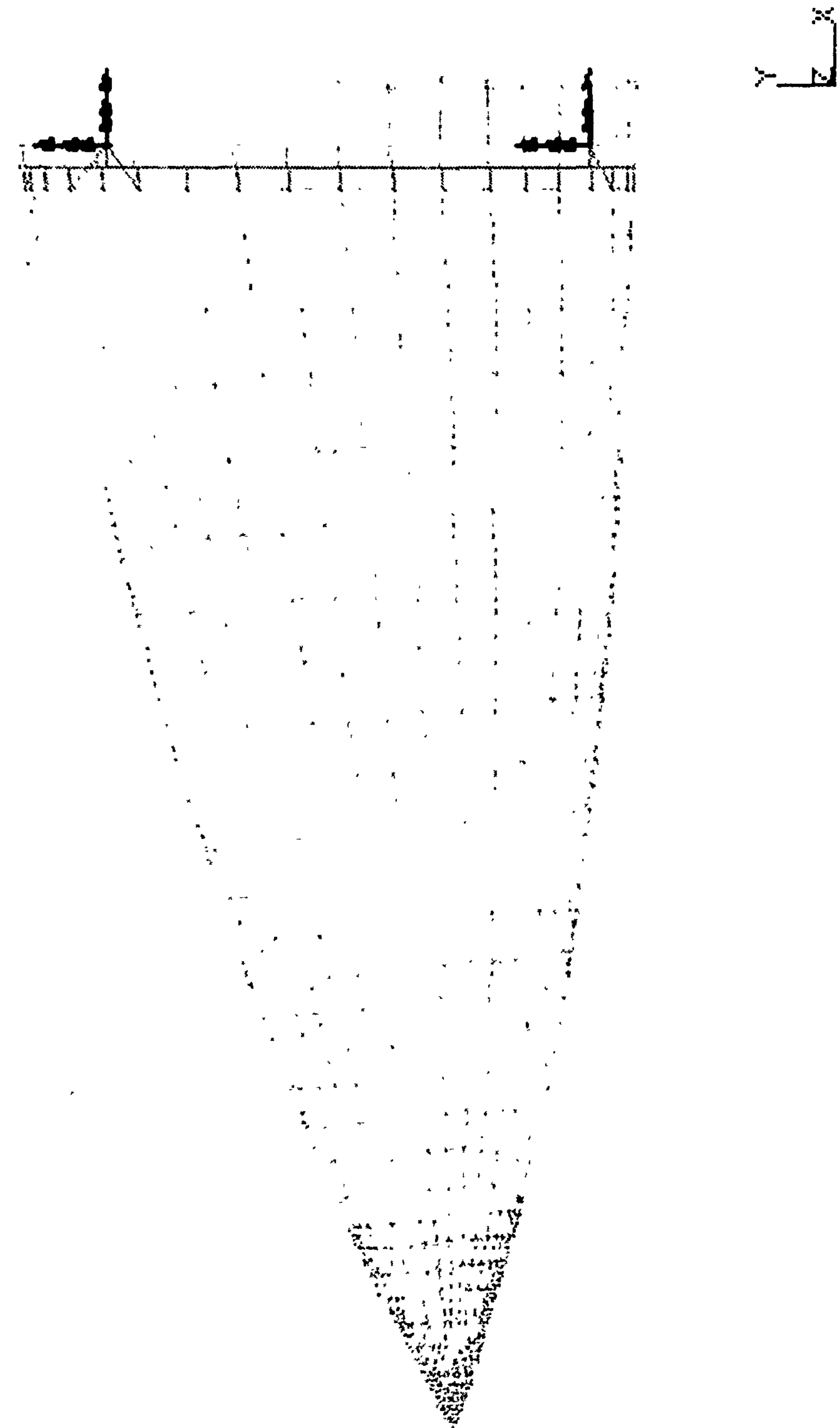
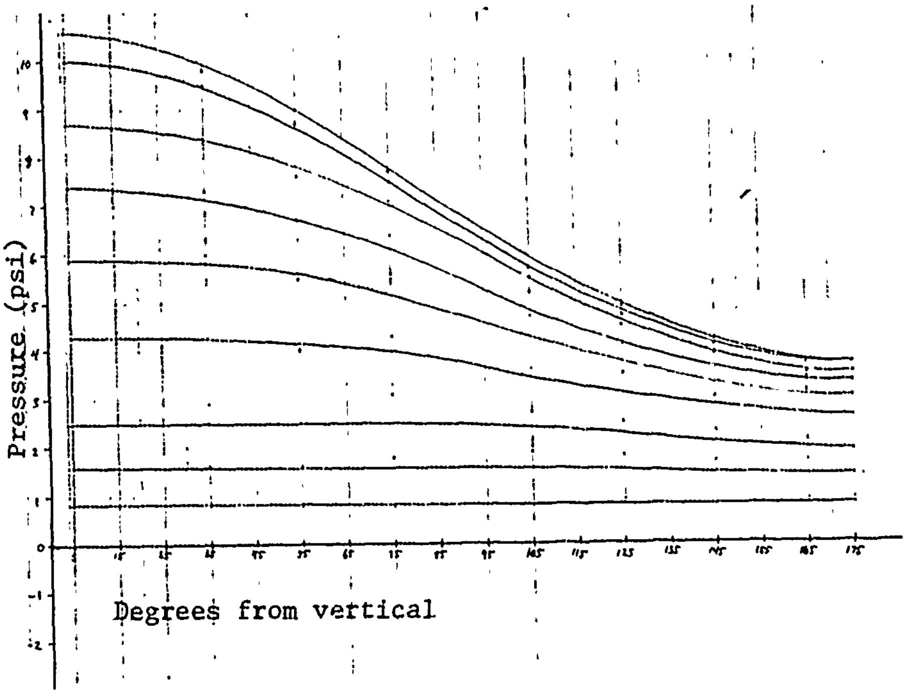


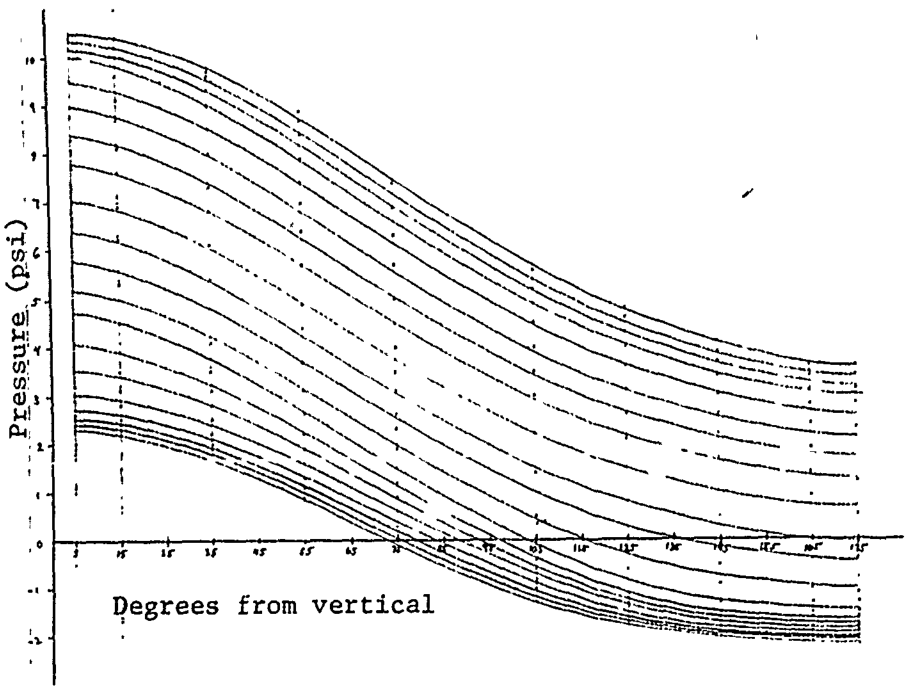
Figure 10: Radome Finite Element Model

PROPERTY	VALUE
Tensile Strength, 0°	160 ksi
Tensile Modulus, 0°	5656 ksi
Tensile Strength, 90°	4.35 ksi
Tensile Modulus, 90°	1160 ksi
Compression Strength, 90°	17.4 ksi
Compression Strength, 0°	87 ksi
In-Plane Shear Strength	10.15 ksi
In-Plane Shear Modulus	580 ksi
Longitudinal Poisson's Ratio	0.25
Bonding Shear Strength	4.5 ksi

Figure 11: Typical Fiberglass-Epoxy Properties [3,4]



Figures 12A: Applied Pressure Load Distribution (Rings 1-9)



Figures 12B: Applied Pressure Load Distribution (Rings 10-29)

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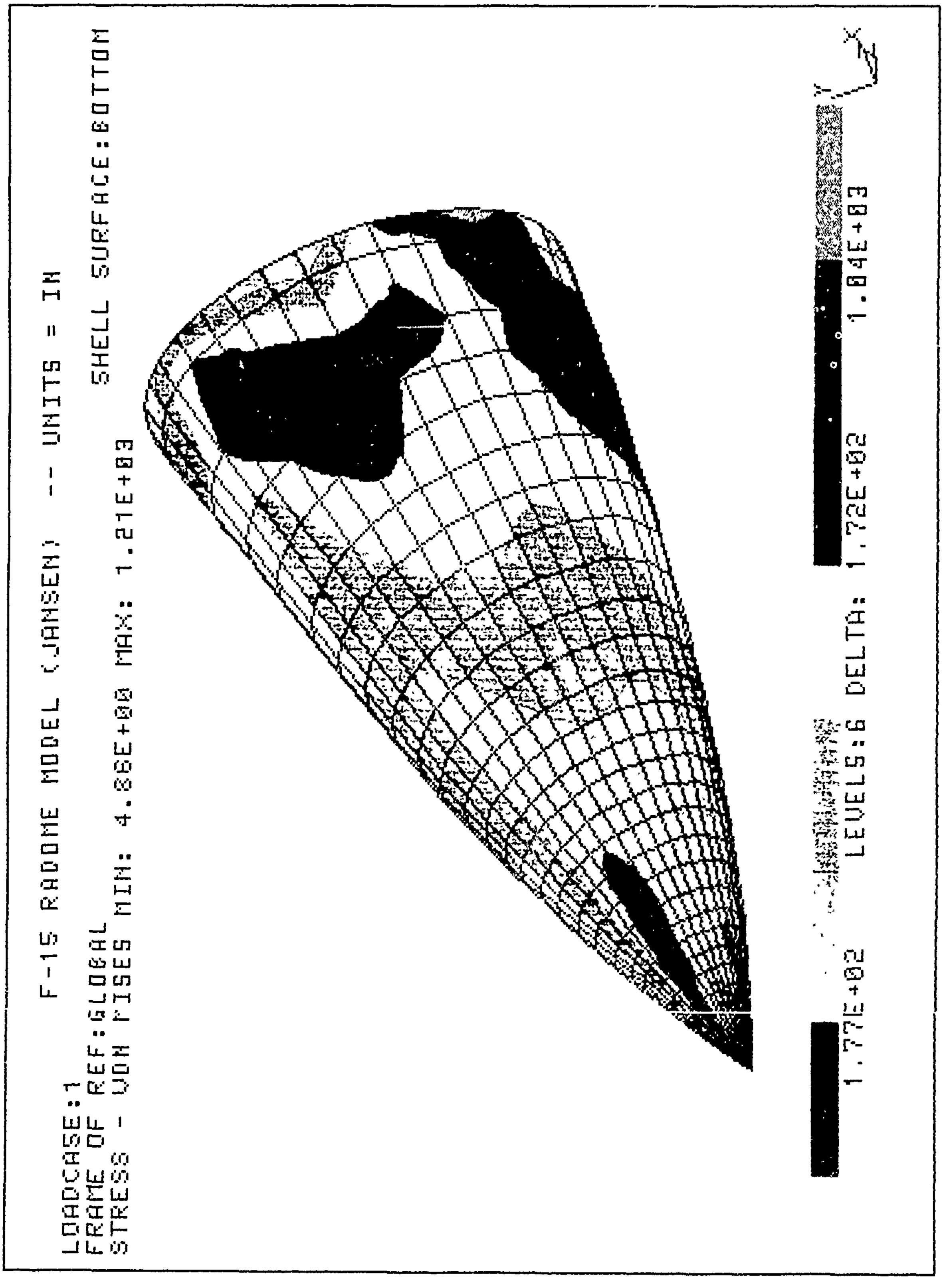


Figure 13: Outer Surface Von Mises Stress Distribution

FIFTH SESSION
COMPOSITES

Chairman

Joseph Birilli

SM-ALC/MMSR

Damage Tolerant Design Concepts In Composites

By

Edvins Demuts
AFWAL/FIBAC

ABSTRACT
of a technical paper
by Edvins Demuts
to be presented at
The 1986 USAF Aircraft Structural Integrity Program (ASIP) Conference
2-4 December 1986, Capitol Plaza Holiday Inn, Sacramento, California
sponsored by the Air Force Logistics Command

This paper defines the general aircraft structures damage tolerance concept, places it prespectively in the USAF Aircraft Structural Integrity Program (ASIP), regardless of airframe material, and briefly reviews the recent history of military aircraft standards and specifications, their role in and application to current systems with respect to structural damage tolerance. The problems of and needs for design and qualification criteria of damage tolerant composites airframes are listed and currently developed supporting experimental data are presented.

The paper discusses the current understanding of damage tolerant design concepts in fiber-reinforced matrix composites structures and lists the associated problems. THe experimental data of deliberately damaged composite specimens illustrate: the relative severity levels for each of the various types of flaws/damages examined; the characteristics of cyclic loading effects and the static behavior; damage tolerance sensitivities to design configurations and specimen complexity levels and some currently employed means to achieve a damage tolerant composites structural design as specified by the owner/user. Solutions to the various problems identified are proposed.

SUMMARY

of a technical paper on

DAMAGE TOLERANT DESIGN CONCEPTS IN COMPOSITES

by Edvins Demuts

to be presented at

The 1986 USAF Aircraft Structural Integrity Program (ASIP) Conference
2-4 December 1986, Capitol Plaza Holiday Inn, Sacramento, California
sponsored by the Air Force Logistics Command

CURRENT UNDERSTANDING

The requirement for airframes to be damage tolerant stems from safety considerations and is to prevent catastrophic failures due to unsuspected or undetected damage. This is achieved by designing a structure with an assumed initial damage as prescribed in a user's specification. Assuming that a visible damage will be detected and repaired, the logical initial damage for design purposes would not have to exceed the maximum invisible damage possible. In composites, this is the maximum internal damage possible without any external evidence or signs of the cause on the surface that receives the damage. Such damage visibility thresholds have not been established in a uniform, rigorous and comprehensive fashion for different laminate thicknesses and various materials.

Past and current research of graphite epoxy composites have provided the following evidence. Impact damage is the most severe among the many types of possible flaws and damage. The fatigue S-N curves for common laminates are relatively flat and if the ratio of maximum spectrum load to static damaged strength is below 0.6 the fatigue life exceeds 10^6 cycles. Hence static damaged strength may govern the design. Damage tolerant designs based on coupon data tend to be conservative as real structures with greater complexity and number of load paths provide more damage resistance than a flat and relatively small coupon could. Damage tolerance of composite structures is quite sensitive to configuration. The above evidence is based primarily on experimental data.

The following list is a sample of the numerous damage tolerant design concepts that have been developed and some of which have applied with varying degrees of success in actual aircraft. Among skin concepts: 100% of ±45°, 10%/80%/10% of 0°/±45°/90°; stitching, fastener rows, buffer strips, softening strips, planks, tear straps, hybrid materials, fabric, braided reinforcement, 3-d reinforcement, stitched woven fabrics, adhesive interleafs, localized adhesive strip interleaving. Other concepts: blade, I, hat stiffeners, isogrid stiffening; multiple load paths; hybrid structural box; tougher materials. The damage tolerance aspects in these concepts are more empirical than analytical and reflect the experience and ingenuity of the designer who knows that the concept works but may not be able to explain why.

CURRENT AND FUTURE PROBLEM AREAS

1. Lack of adequate analysis to predict the following: damage resulting from an impact, structural behavior of a damaged structure, damage growth characteristics including growth rate, fatigue life and residual strength of a damaged structure. Current analyses are relatively primitive and unable to properly model the very complex damage state in composites. Analytic deficiencies must be substituted by much more expensive testing.

2. Nonuniformity and absence of standards in experimental characterization of damage pertain to: size, shape, layup, configuration of test specimens; specimen support, prestress level, impactor shape and size, impact energy level; fixtures and procedures of post impact test. A great deal of the abundantly available damage tolerance data, unfortunately, cannot be compared for lack of common standard basis. This prevents data base expansion and statistical allowables determination without penalty for a low number of specimens.

3. Inadequate quantification of penalty and benefits for providing a degree of structural damage tolerance.

4. Identification and prioritization of the most influential configuration parameters in damage tolerant designs. Determination of the type and degree of their influence.

5. As new composite materials become available, the need arises for uniform criteria in comparing and assessing their contribution to the structure's damage resistance early in the design. Current ability of material selection may not reflect optimum satisfaction of all needs.

6. Lack of uniform definition of damaged area in a laminate that is also physically meaningful in an analysis: cumulative area as seen in plan form by NDI versus either just one found between any two plies or the sum of all damaged areas through the thickness. Use of improper damage area may lead to incorrect analytic predictions.

7. inadequate investigation of multiple damage involving the number, proximity and intensity of damage.

8. NDI and damage repair techniques are not totally adequate for more complex structural configurations. This may hamper the applications of these design concepts.

METHODS FOR SOLVING PROBLEMS

1. Expand efforts on the development and validation of damage tolerance analysis. Improvement to the prediction of static strength, damage growth, fatigue life and residual strength of damaged structure will reduce otherwise needed testing and associated costs.

2. Attempt to convince structure users to realize the need for standardization in damage tolerance characterization and assessment. A uniform approach must be developed and implemented. The goal - ability to use all available data on a standard basis. This will reduce design development costs.

3. Develop methods for quantifying penalties and benefits associated with damage tolerant designs - weight, performance, reliability, acquisition and life cycle costs. This will aid in achieving optimum designs.

4. Determine the effect of various design parameters on damage tolerance, preferably by using improved analytic methods and experimental verification. Results of such assessment will considerably reduce trade studies and design efforts.

5. Develop a criterion for assessing a material's contribution to the structural damage tolerance - as a tool for the designer to select a material system.

6. Examine the effect on analytical predictions of using damaged areas due to various definitions. Recommend the use of the most meaningful definition. A break through in analysis development may be expected.

7. Investigate the effects of multiple damage and bound the problem by determining the most influential combinations of the number, proximity and intensity of impacts. Examine the effects of these combinations on the structural design, determine the critical one and develop corresponding design criterion and qualification method.

DAMAGE TOLERANCE

DAMAGE TOLERANCE

- STRUCTURAL INTEGRITY

  = STAT. STR. + STIFFNESS + DURAB. + SAFETY
- SAFETY = DAMAGE TOLERANCE
- F-111 PROBLEMS - STRUCT. COMPON. NOT DAM. TOLERANT
- SEP 72 MIL-STD-1530 ; ASIP
- JUL 74 MIL-A-83444 D/T REQUIREMENTS
- DEC 75 MIL-STD-1530A
- F-16 FIRST DAMAGE TOLERANT AIRFRAME
- BUT NO D/T RQMTS/SPEC FOR COMPOSITE STRUCTURES

DAMAGE TOLERANCE

- D/T – ABILITY TO WITHSTAND GIVEN LOADS IN THE PRESENCE OF DAMAGE W/O CATASTROPHIC FAILURE – RET'N TO BASE
- D/T – INHERENT IN ANY OBJECT (GLASS, RUBBER)
- DEGREE OF D/T IS ARBITRARY – UP TO THE USER
- QUANTIFYING D/T BENEFITS AND "PRICE" FOR A DEGREE OF D/T IS DIFFICULT
- FACTORS AFFECTING D/T:
  IMPACTOR SHAPE, WEIGHT, VELOCITY
  SPECIMEN GEOMETRY, MATERIAL, SUPPORT

DAMAGE TOLERANCE

- VELOCITY - LOW, MEDIUM, HIGH
- CURRENT PROGRAM w. BOEING/NORTHROP
  - -LOW VELOCITY IMPACT
  - -NO BATTLE DAMAGE
  - -GRAPHITE EPOXY (TS) - MAINLY
  - -GRAPHITE THERMOPLASTIC (TP) - SOME
  - -MULTI-SPAR WING BOX (FIGHTER)
  - -MULTI-RIB WING BOX (CARGO)
  - -D/T RQMTS FOR COMPOSITE STRUCT. (GR/EP)

DAMAGE TOLERANCE

- TYPES OF DAMAGE:
  - MFG FLAWS, MFG IMPACT DAMAGE
  - IN-SERVICE LOW VELOCITY IMPACT DAMAGE
- ASSUMPTIONS:
  - VISIBLE DAMAGE CAN BE EXAMINED, REPAIRED
  - INVISIBLE DAMAGE MAY BE FLOWN AWAY
- D/T DESIGN ASSUMES INITIAL DAMAGE AND PROVIDES ENOUGH MATERIAL FOR SAFE OPERATION IN PRESENCE OF ASSUMED DAMAGE
- INITIAL DAMAGE SIZE AND LOCATION MUST BE SPECIFIED BY USER

DAMAGE TOLERANT DESIGN CONCEPTS

DAMAGE TOLERANT DESIGN CONCEPTS

CURRENT UNDERSTANDING

- STATIC STRENGTH + STIFFNESS + DURABILITY + D/T = STRUCTURAL INTEGRITY (ASIP 1972/75)
- DAMAGE TOLERANCE IS SAFETY (MIL-A-83444, 1974)
- F-16 THE FIRST MIL A/C TO MEET D/T RQMTS
- D/T PREVENTS CATASTROPHIC FAILURE DUE TO UNDETECTED DAMAGE
- INITIAL DAMAGE ASSUMPTION - SPEC, ARBITRARY
- DVT - MAX. INVISIBLE DAMAGE NOT WELL ESTABLISHED

(CONT'D)

DAMAGE TOLERANT DESIGN CONCEPTS

CURRENT UNDERSTANDING (CONT'D)

- IMPACT DAMAGE - MOST SEVERE
- FLAT S-N CURVES; 60% THRESHOLD
- STATIC STRENGTH DESIGN - MAY GOVERN
- COUPON DATA - CONSERVATIVE
- D/T VERY SENSITIVE TO CONFIGURATION

(CONT'D)

DAMAGE TOLERANT DESIGN CONCEPTS

CURRENT UNDERSTANDING (CONCLUDED)

- ± 45 ; 10/80/10; STITCHING; FASTENERS
- BUFFER, SOFTENING STRIPS; PLANKS; TEAR STRAPS
- FABRIC – PLAIN, STITCHED; BRAIDED, 3-D REINFORCING
- ADHESIVE INTERLEAVED – ALL; LOCAL
- BLADE, I-, HAT STIFFENERS; ISOGRID
- MULTIPLE LOAD PATH
- HYBRID BOX – SKIN, SUBSTRUCTURE
- TOUGHER MATERIALS – TP'S

EMPIRICAL vs ANALYTIC
EXPERIENCE, INGENUITY
IT WORKS, BUT WHY?

DAMAGE TOLERANT DESIGN CONCEPTS

PROBLEMS

1. INADEQUATE ANALYSIS, PREDICTION CAPABILITY
   - RESULTING DAMAGE, ITS BEHAVIOR, GROWTH
   - FATIGUE LIFE, RESIDUAL STRENGTH

   MUST SUBSTITUTE EXPENSIVE TESTING
2. NO STANDARDS FOR D/T CHARACTERIZATION
   - SPECIMEN GEOMETRY, LAYUP, CONFIGURATION, SUPPORT
   - IMPACTOR SHAPE, SIZE, ENERGY LEVEL
   - POST IMPACT TEST FIXTURE, PROCEDURE
3. PENALTIES & BENEFITS OF D/T – NOT QUANTIFIED
4. CONFIGURATION PARAMETER EFFECTS ON D/T
   - NOT DETERMINED OR PRIORITIZED

(CONT'D)

DAMAGE TOLERANT DESIGN CONCEPTS

PROBLEMS (CONCLUDED)

5. MATERIAL TOUGHNESS CONTRIBUTION
   - NO UNIFORM ASSESSMENT
6. DAMAGED AREA – MOST MEANINGFUL DEFINITION FOR ANALYSIS NOT EXAMINED
7. MULTIPLE DAMAGE
   - INADEQUATE INVESTIGATION, CRITERIA
8. NDI and REPAIRS
   - NOT FULLY ADEQUATE FOR COMPLEX CONFIGURATIONS
   - MAY SLOW DOWN APPLICATIONS

DAMAGE TOLERANT DESIGN CONCEPTS

PROPOSED SOLUTIONS

1. IMPROVE ANALYSES TO PREDICT
   - STATIC STRENGTH, DAMAGE GROWTH
   - FATIGUE LIFE, RESIDUAL STRENGTH

   WILL REDUCE COSTS OF OTHERWISE NEEDED TESTS

2. DEVELOP AND IMPLEMENT UNIFORM APPROACH FOR CHARACTERIZING DAMAGE TOLERANCE

   WILL REDUCE DESIGN DEVELOPMENT COSTS

3. DEVELOP METHODS TO QUANTIFY PENALTIES & BENEFITS OF DAMAGE TOLERANT DESIGNS
   - WEIGHT, PERFORMANCE, RELIABILITY
   - ACQUISITION AND LIFE CYCLE COSTS

4. DETERMINE CONFIGURATION PARAMETER EFFECTS
   - USE IMPROVED ANALYSIS
   - VERIFY EXPERIMENTALLY

(CONT'D)

DAMAGE TOLERANT DESIGN CONCEPTS

PROPOSED SOLUTIONS (CONCLUDED)

5. DEVELOP ASSESSMENT OF MATERIAL TOUGHNESS CONTRIBUTION TO STRUCTURAL D/T
   WILL HELP DESIGNER SELECT MATERIAL
6. EXAMINE VARIOUS DAMAGED AREA EFFECTS ON PREDICTIONS; RECOMMEND MOST MEANINGFUL
7. INVESTIGATE MULTIPLE DAMAGE EFFECTS
   DEVELOP DESIGN CRITERION, QUALIFICATION METHOD

ARTS
State of the Art Force Management Tool

By

Leonard Wright
Boeing Military Airplane Co.

ARTS
AUTOMATED RELATIONAL TRACKING SYSTEM

- INDIVIDUAL AIRPLANE AND FLEET TRACKING
- RELATIONAL INFORMATION MANAGEMENT SYSTEM
- USER FRIENDLY-MENU DRIVEN SYSTEM
- "QUERY" - DATA RETRIEVAL SYSTEM
- "WHAT IF" - IMPACT OF PROPOSED CHANGES

INDIVIDUAL AIRPLANE AND FLEET TRACKING

The forerunner of ARTS was the Individual Airplane Tracking which goes back to the 1960's with basic Fatigue Damage tracking. Then in the late 1970's we got into the crack growth era with DADTA's performed on both the B-52 and KC-135 aircraft. The DADTA identified potentially critical structural details. Individual Airplane Tracking Programs were developed to track potential crack growth from an assumed initial flaw.

The crack growth calculations made in the IATP provide data on individual aircraft usage and an estimate of remaining time to Economic Repair limit, to critical crack length, and to inspections at selected locations on the airplane.

A large amount of historical data pertaining to ASIP modifications, base changes, PDM, etc. is also included

These data are used to produce an extensive set of fleet management reports to show the status of the fleet and define inspection requirements for PDM for each airplane based on actual usage.

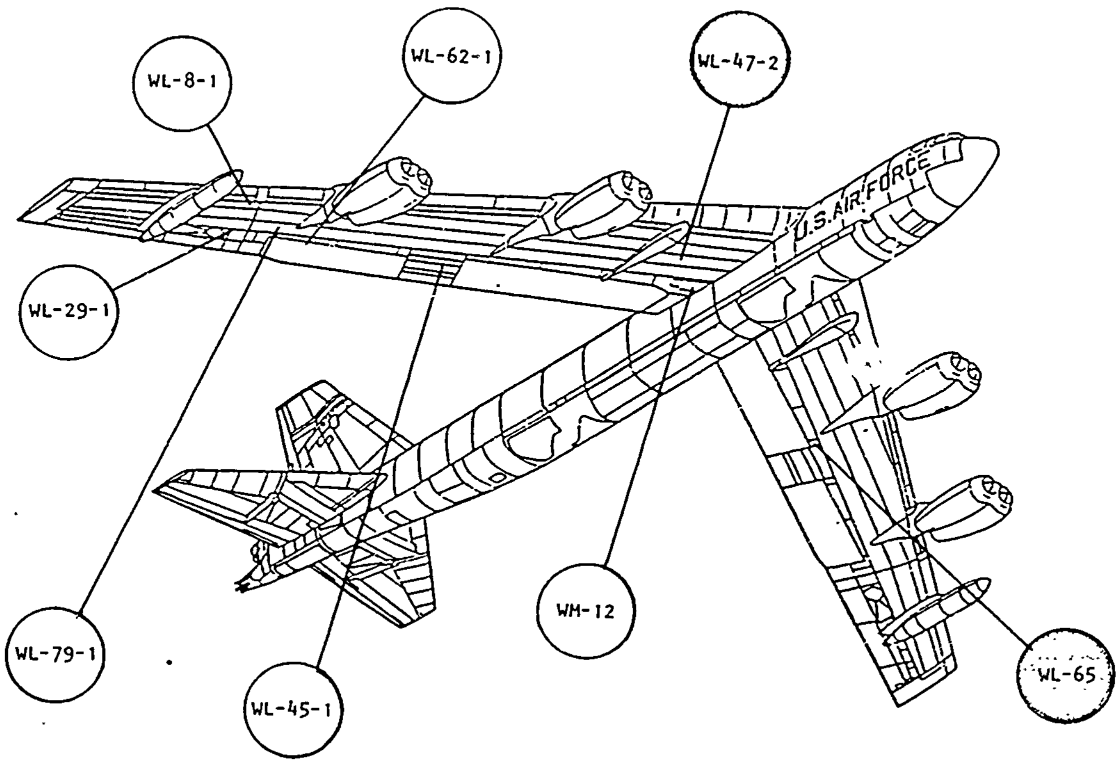
INDIVIDUAL AIRPLANE AND FLEET TRACKING

- MONITORED LOCATIONS
- CRACK GROWTH TRACKING METHODOLOGY
- HISTORICAL INFORMATION
- FLEET MANAGEMENT REPORTS

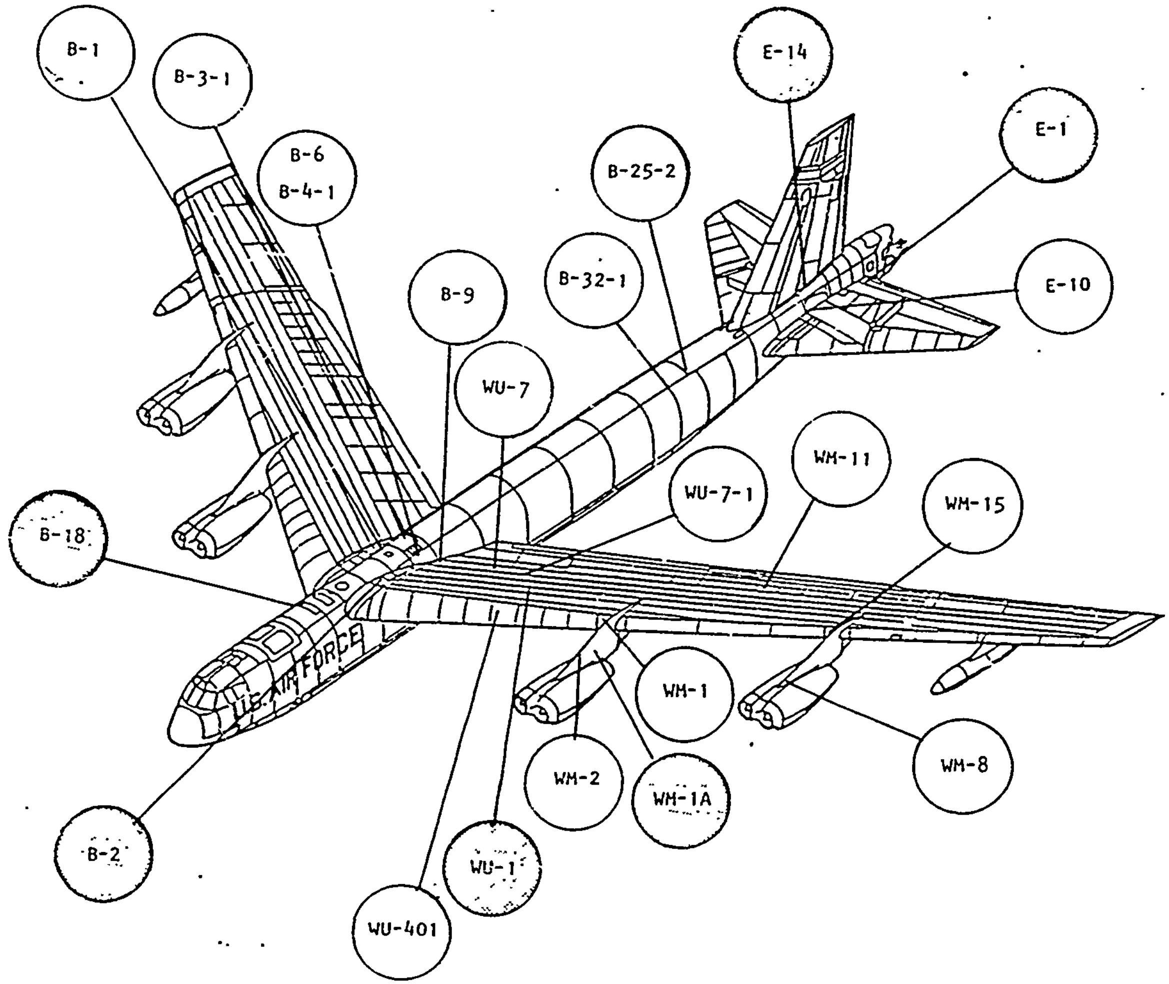
MONITORED LOCATIONS

There are thirty locations monitored on the B-52 G/H fleet. These represent the potentially critical details identified in the DADTA. Some of the details do not require inspections, however, are monitored to be representative of particular areas on the airplane. Crack growth is calculated for each of ten control points. Crack lengths at the twenty other locations, referred to as related locations, are based on calculations made at the control points and crack growth relationships developed from the DADTA.

MONITORED LOCATIONS



MONITORED LOCATIONS



CRACK GROWTH METHODOLOGY

Tracking potential crack growth for each individual airplane of the B-52 or C/KC-135 fleet is dependent on the mission flight log form (AFTO Form 16). These forms, completed by the flight crew, provide the definition of the mission flown, segment by segment.

The crack growth for each flight is calculated for each control point using a factor dependent on the stress spectrum referred to as a Growth Factor (GF), a crack size parameter or Geometry Factor (GEF) for each detail based on a geometry dependent parameter Y as a function of flaw size, and a Retardation Factor (RF) based on mission severity.

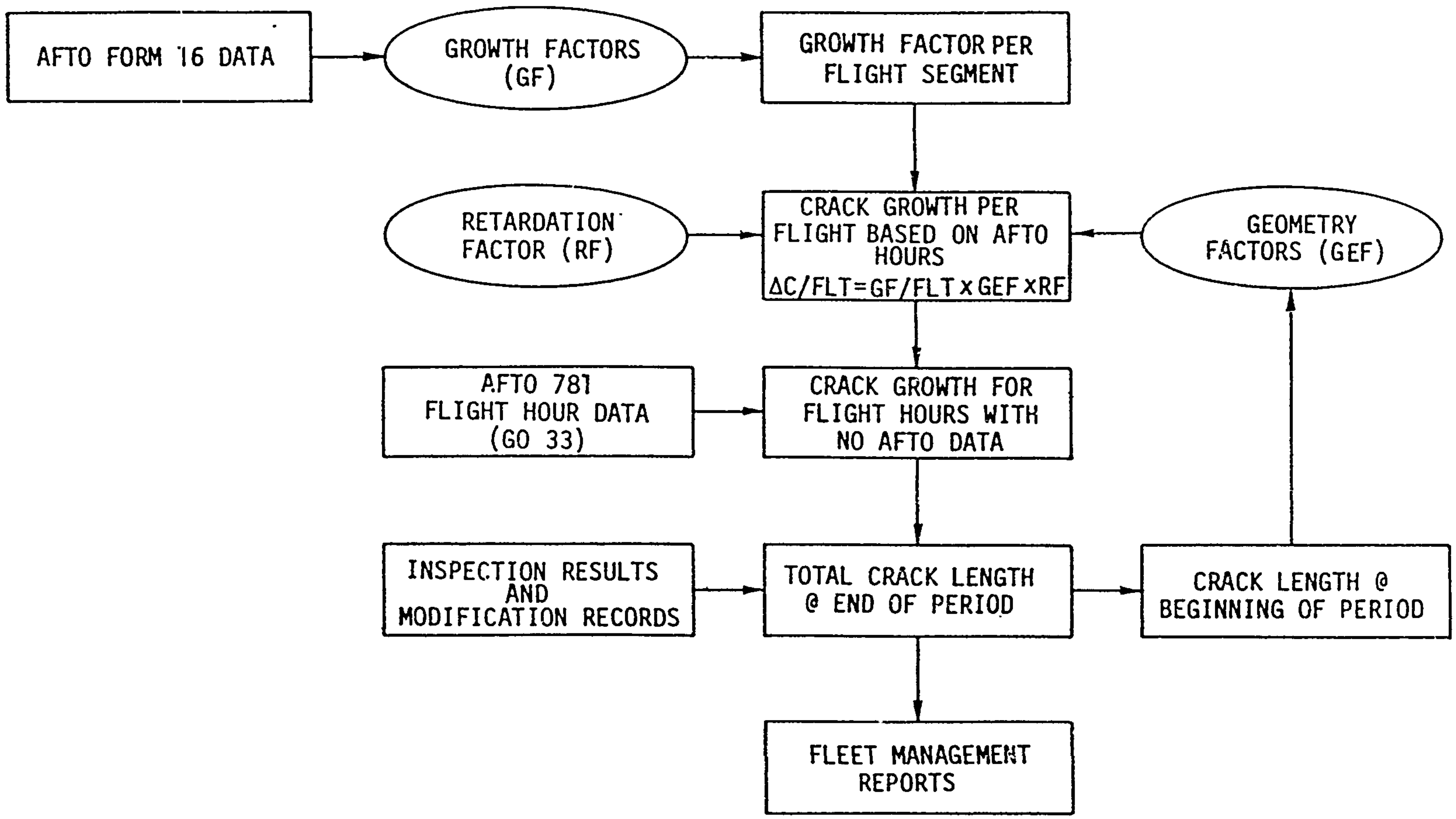
The IATP provides tables of GF's that were determined from parametric analyses covering the operational envelope of the fleet for the B-52G and H with and without major modifications that significantly affect the stress spectrum. A GF can thus be determined for each flight and/or ground segment defined on the AFTO Forms 16 and accumulated to obtain the GF for that particular flight.

AFTO 781 Flight hour data is used to assure all the hours flown for the period are represented by AFTO 16 data. If not the crack is grown for the hours with no AFTO 16 data.

Crack growth for the related location is determined from the calculations made for the control points using crack growth relationships described from baseline crack growth data.

If an inspection or modification was accomplished, the crack size is reset so the total crack length at the end of the quarter is determined for input into the next quarter run and the fleet management reports. The tracking methodology as defined allows for tracking any mission type as long as Growth Factors are defined.

CRACK GROWTH TRACKING METHODOLOGY



HISTORICAL INFORMATION

The IATP has a list of attrited airplanes with information on each as to reason, hours, date. Past base locations for each airplane from 1975 are currently in the program. Plans are to include best base information possible from delivery to aid in future corrosion prediction capabilities.

The flight hours and date of PDM's along with inspections are included. In addition, there is information pertaining to ASIP related modifications. This historical information is shown in the Status-3 report.

HISTORICAL INFORMATION

- ATTRITED AIRPLANES
- BASE LOCATIONS
- PDM ACCOMPLISHMENTS
- ASIP RELATED MODIFICATIONS

FLEET MANAGEMENT REPORTS

There are 24 basic types of fleet management reports produced showing a variety of information with options. Each report includes a complete word description of what is contained in the report. There are reports providing current airplane status. Status-1 provides a description of all structural details in the IATP. A Status-3 report, for which an example is shown, is produced for each airplane in the fleet. An example of one of the inspect reports (Inspect-4) is also included.

FLEET MANAGEMENT REPORTS

STATUS-1	DETAIL DESCRIPTION SUMMARY
STATUS-2	TCTO SUMMARY REPORT
STATUS-3	INDIVIDUAL A/P CRACK GROWTH STATUS REPORT
COMPARE-1	GFR SUMMARY REPORT
FLIGHT-1	CURRENT FLIGHT HOURS REPORT: (ORDERED BY A/P S/N)
FLIGHT-2	CURRENT FLIGHT HOURS REPORT: (ORDERED BY TOTAL FH)
FLIGHT-3	BARCHART OF FLIGHT HOUR DISTRIBUTION
FLIGHT-4	BARCHART OF AFTO CAPTURE RATE
FLIGHT-5	BASE/MDS AVERAGE FLIGHT HOURS REPORT
FLIGHT-6	BASE/MDS COMPARISON REPORT: (DELTA/AFTO/PLANNED HOURS)
FLIGHT-7	BASE COMPARISON REPORT
FLIGHT-8	ATTRITED AIRPLANE REPORT
INSPECT-1	DETAIL DESCRIPTION SUMMARY
INSPECT-2	INSPECTION REQUIREMENTS SUMMARY
INSPECT-3	INSPECTION SUMMARY CHART (20 YEARS)
INSPECT-4	INDIVIDUAL A/P DETAIL INSPECTION CHART (20 YEARS)
INSPECT-5	PDM REQUIREMENTS FOR SPECIFIED TIME PERIOD
CRIT-1	CRITICAL DETAIL REPORT
CRIT-2	FLIGHT HOURS PAST CRITICAL REPORT
CRIT-3	EQUIVALENT BASELINE HOURS PAST CRITICAL REPORT
CRIT-4	FLIGHT HOURS TO CRITICAL REPORT
CRIT-5	EQUIVALENT BASELINE HOURS TO CRITICAL REPORT
USAGE-1	AIRPLANE USAGE SUMMARY
USAGE-2	AIRPLANE USAGE SUMMARY BY MISSION PARAMETERS

STATUS - 3 INDIVIDUAL A/P STATUS REPORT

Status - 3 is produced for each airplane in the tracking system. It provides a wealth of current and historical information pertaining to that particular airplane.

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STATUS-3 INDIVIDUAL A/P CRACK GROWTH STATUS REPORT: DATE 11/12/84 QUARTER 8502 PAGE 20

A/P SUMMARY FOR 5706490
FLT HRS 12671
EST FLTS 1258

PDM DATE 8608

	A/P-AVG	BASE-AVG
DELTA HRS	627	432
PROJ HRS	13298	13103

QUARTER	8403	8404	8501	8502	YEAR	SINCE 75
BASE CODE	105	105	105	105		
DELTA HRS	121	65	127	225	537	4203
AFTO HRS	121	65	126	195	508	4024
LOGGED MSN	14	8	13	22	57	541
LOGGED TAG	4	11	1	35	51	771
LOGGED FS	1	2	1	4	8	68
FLT HRS	12255	12328	12447	12671		

BASE SUMMARY
BASE CODE 105
BASE GRIFFISS
ICAO KRME
NO. A/P 16
PLANNED HRS/YR 370.

PAST BASES: MATHER A 11362
GRIFFISS 9475

INSPECTION/MODIFICATION SUMMARY

ICTO	DATE	HRS	ECP	DESCRIPTION	ICTO	DATE	HRS	ECP	DESCRIPTION
1431	6203	1970	1050	WING LIFE EXTENSION	2012	7402	8162	1422	ELCT-OPTICAL VIEW SY
1666	6502	3466	1124/2	FUS/EMP SUPPORT STRG	2037	7402	8162	1551	PHASE 6 ECM RIVET AC
1741	6701	4323	1185	FUSELAGE LIFE EXTENS	2252	8204	11362	1744	CRUISE MISSILE INTEG
1855	7003	5503	1195	STAB AUGMENTATION SY	2253	8204	11362	1746	OFFENSIVE AVIONICS S
1890	7402	8162	1369	AGM 69					

	DETAIL NO	DETAIL CODE	GROWTH FACTOR RATE AIRPLANE	GROWTH FACTOR RATE FLEET	SAFETY CRITERIA DELTA LENGTH	SAFETY CRITERIA CURR LENGTH	SAFETY CRITERIA CRIT LENGTH	SAFETY CRITERIA HR-TO CRIT	ECONOMIC CRITERIA CURR LENGTH	ECONOMIC CRITERIA ERL	ECONOMIC CRITERIA HR-TO ERL	MOD NO.	MOD CHFG	INSP NO.	LAST ICTO	LAST RSLT	NEXT INSP DATE	NEXT INSP ICTO	NEXT INSP HR-TO	HR-TO SUBS
CP	1	WL-47-2	1.84091	1.83541	0.0013	0.0568	1.2000	22339	0.0055	0.0300	99999	0	1	0	1431	1	9904	0	5819	***
	2	WM-12			0.0016	0.1006	1.8600	31712	0.0111	0.0300	46655	0	1	0	1431	1	9904	0	10505	***
CP	3	WL-65	1.70591	3.44728	0.0048	0.1062	0.8000	8782	0.0063	0.0300	76174	0	1	0	1431	1	****	0	-960	2734
	4	WL-8-1			0.0000	0.6302	1.3627	20890	0.0050	0.0300	99999	0	1	0	1431	1	9902	0	5094	***
	5	WL-29-1			0.0004	0.0405	4.0000	32029	0.0050	0.0300	33955	0	1	0	1431	1	9904	0	10664	***
	6	WL-45-1			0.0005	0.0457	2.4720	24998	0.0050	0.0300	75508	0	1	0	1431	1	9904	0	7148	***
	7	WL-62-1			0.0004	0.0407	4.0000	25866	0.0050	0.0300	67939	0	1	0	1431	1	9904	0	7582	***
	8	WL-79-1			0.0004	0.0415	4.0000	3135[illegible]	0.0050	0.0300	99999	0	1	0	1431	1	9904	0	10575	***
CP	9	WU-1	2.35711	2.83156	0.0434	0.9277	1.6800	1998	0.0166	0.0300	4421	0	1	0	1431	1	****	0	-4352	4374
CP	10	WU-7	2.77832	3.28572	0.0000	0.8240	0.8240	-2935	0.0142	0.0300	17753	0	1	0	1431	1	****	0	-6818	2818
	11	WU-7-1			0.0000	0.6400	0.6400	-2329	0.0081	0.0300	24001	0	1	0	1431	1	****	0	-6515	1223
	12	WU-401			0.0028	0.1454	0.5330	5792	0.0300	0.0300	-1633	0	1	0	1431	1	****	0	-2655	1283
CP	13	E-14	0.05562	0.04620	0.0000	0.0537	1.0200	99999	0.0110	0.0300	99999	0	1	0	0	0	9904	0	99999	***
	14	B-9			0.0005	0.2992	1.1400	37350	0.0144	0.0300	17282	0	1	0	0	0	9904	0	12339	24409
CP	15	WM-1A	30.11229	25.06296	0.0000	1.5000	1.5000	-10288	0.0300	0.0300	-10408	0	1	0	0	0	****	0	-11480	465
	16	WM-1			0.0000	0.6768	0.6768	-3961	0.0050	0.0300	99999	0	1	0	0	0	****	0	-8316	1684
	17	WM-2			0.0000	1.7022	1.7022	-6988	0.0300	0.0300	-9318	0	1	0	0	0	****	0	-9830	1165
	18	WM-15			0.0000	0.5280	0.5280	-10054	0.0056	0.0300	25523	0	1	0	0	0	****	0	-11363	416
CP	19	E-1	1.24223	1.08502	0.0033	0.1407	1.8300	51400	0.0139	0.0300	12319	0	1	0	0	0	9904	0	19364	23255
	20	E-10			0.0067	0.1603	0.9714	9184	0.0131	0.0300	15599	0	1	0	0	0	****	0	-1744	6421
CP	21	B-2	3.97711	6.95937	0.0188	0.3449	5.9250	27245	0.0224	0.0300	3195	0	1	0	1431	1	9904	0	8272	11670
	22	B-1			0.0016	0.0965	1.7160	60343	0.0087	0.0300	17185	0	1	0	1741	1	9904	0	15998	6113
	23	B-3-1			0.0051	0.2137	3.3000	24001	0.0300	0.0300	-225	0	1	0	0	0	9904	0	5665	7638
	24	B-4-1			0.0026	0.1032	3.9740	56105	0.0136	0.0300	7569	0	1	0	1741	1	9904	0	23878	***
	25	B-6			0.0023	0.1100	1.0625	31052	0.0192	0.0300	5034	0	1	0	1741	1	9904	0	11352	***
CP	26	B-18	5.19421	5.82627	0.0472	1.8441	1.5000	1498	0.0300	0.0300	-9993	0	1	0	0	0	****	0	-5587	3978
	27	WM-8																		
	28	WM-11			0.0016	0.1047	3.0000	23367	0.0191	0.0300	5740	0	1	0	0	0	9904	0	5348	11966
CP	29	B-25-2	1.72000	1.57803	0.0024	0.1101	2.6270	31883	0.0134	0.0300	18429	0	1	0	0	0	9904	0	9606	12178
	30	B-32-1			0.0011	0.0946	3.0770	45389	0.0118	0.0300	30581	0	1	0	1741	1	9904	0	18480	***

HOURS REMAINING TO ECONOMIC LIFE
WING LOWER 60721 WING UPPER 26141 FIN 324827 STABILIZER 39232 BODY 38922

*** NO SAFETY INSPECTION REQUIRED NOR DEFINED BY DADTA

INSPECT-4 INDIVIDUAL A/P DETAIL INSPECTION CHART (20 YEARS)

Inspect-4 is a schedule of when inspections are required for each of the details in the IATP. The scheduled PDM is shown. This provides a graphic illustration of past due inspections and those details requiring frequent inspections. The nacelle details show frequent inspections required. An ECP is being installed to modify these detail to preclude more frequent inspection than PDM. The inspections are grouped for scheduling those required at a given PDM.

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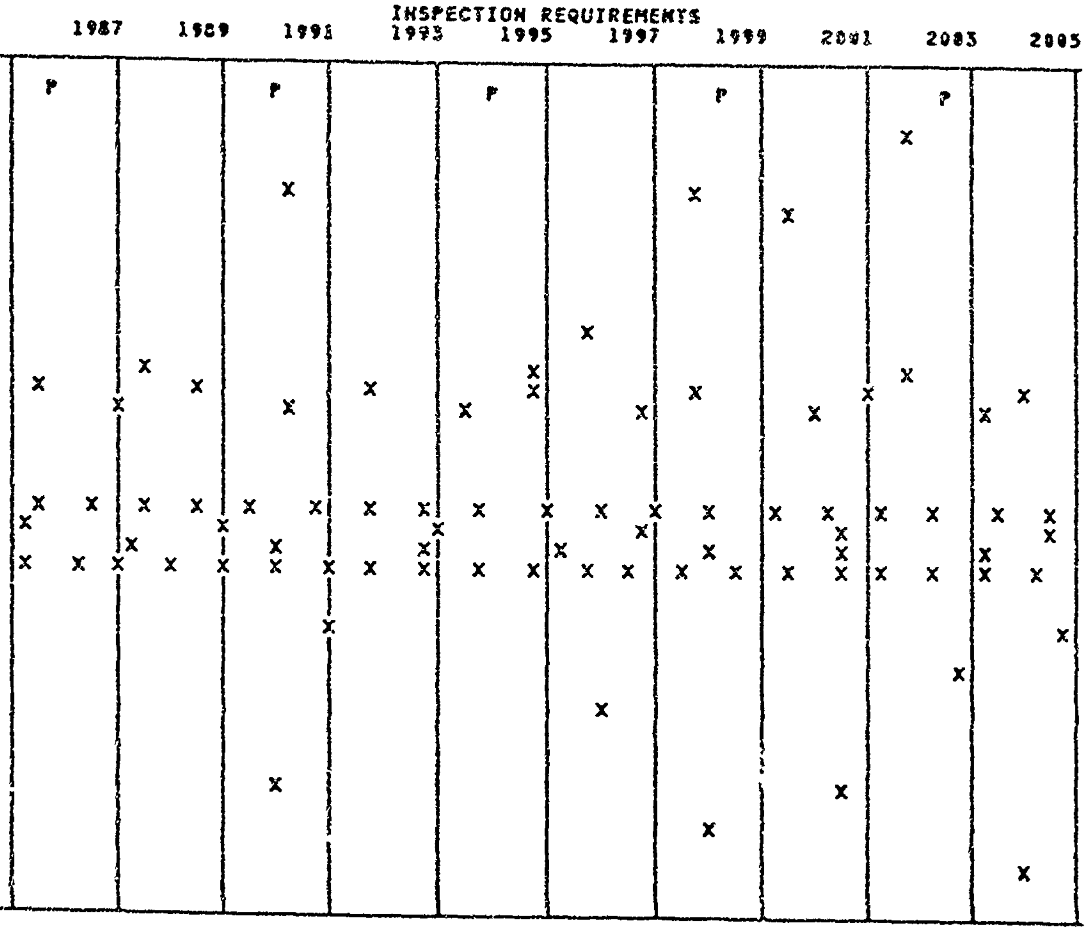
INSPECT-6 INDIVIDUAL A/P DETAIL INSPECTION CHART (20 YEARS): MODEL G DATE 11/18/86 QUARTER 8502 PAGE 1

S/N = 5706690
BASE CODE = 105
FLIGHT HOURS = 12671

	DETAIL NUMBER		CODE
			PDM
	CP	1	WL-47-2
		2	WM-12
$	CP	3	WL-65
		4	WL-8-1
		5	WL-29-1
		6	WL-45-1
		7	WL-62-1
		8	WL-79-1
$	CP	9	WU-1
$	CP	10	WU-7
$		11	WU-7-1
$		12	WU-401
	CP	13	E-14
		14	B-9
$	CP	15	WM-1A
$		16	WM-1
$		17	WM-2
$		18	WM-15
	CP	19	E-1
$		20	E-10
	CP	21	B-2
		22	B-1
		23	B-3-1
		24	B-4-1
		25	B-6
$	CP	26	B-18
		27	WM-8
		28	WM-11
	CP	29	B-25-2
		30	B-32-1



P - SCHEDULED PDM
X - INSPECTION REQUIREMENT
$ - DELINQUENT PDM OR INSPECTIONS EXIST

ARTS
AUTOMATED RELATIONAL TRACKING SYSTEM

- INDIVIDUAL AIRPLANE AND FLEET TRACKING
- RELATIONAL INFORMATION MANAGEMENT SYSTEM
- USER FRIENDLY-MENU DRIVEN SYSTEM
- "QUERY" – DATA RETRIEVAL SYSTEM
- "WHAT IF" – IMPACT OF PROPOSED CHANGES

RIM DATA BASE MAMAGEMENT SYSTEM

ARTS uses a modern relational information management system called RIM to manage the enormous amount of data it needs to access, to allow easy retrieval of data and to take advantage of RIM's built-in sorting, searching and data loading capabilities.

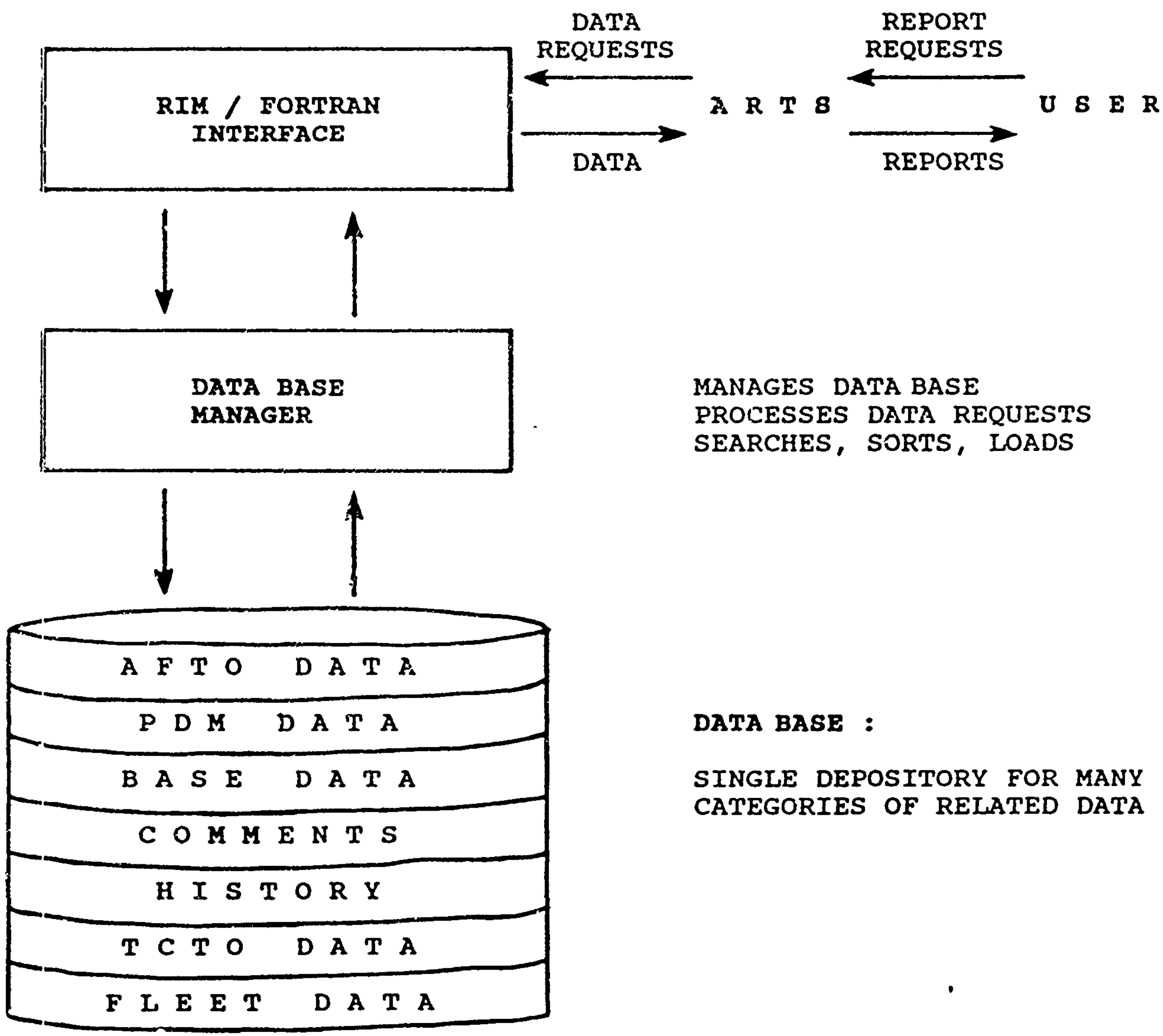
The RIM data base management system has three main components:

Data Base - The actual data file. It serves to put all of the data needed by ARTS in a single location.

Data Base Manager - The built-in component of the system which manages the data, processes data requests, searches the data base and sorts the data as required.

RIM / FORTRAN Interface - This component is not part of all data base management systems. In RIM, this interface allows a FORTRAN program such as ARTS to directly request data from the data base. Thus a program such as ARTS can present user-friendly menus to a user and then take care of technicalities of data access and transfer without the user needing to know how to use the data base system.

RELATIONAL INFORMATION MANAGEMENT SYSTEM
(RIM)



MANAGES DATA BASE
PROCESSES DATA REQUESTS
SEARCHES, SORTS, LOADS

DATA BASE :

SINGLE DEPOSITORY FOR MANY CATEGORIES OF RELATED DATA

ARTS
AUTOMATED RELATIONAL TRACKING SYSTEM

- INDIVIDUAL AIRPLANE AND FLEET TRACKING
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USER FRIENDLY - MENU DRIVEN SYSTEM

ARTS is a very user friendly system, Menus and panels are displayed requesting the user to select from various options.

Help screens are available from everywhere in the system. Data entries are checked for validity wherever possible. Terminal keys are programmed to display data tables.

USER FRIENDLY–MENU DRIVEN SYSTEM

- INITIAL SCREEN
- MAIN MENU
- SECONDARY MENUS
- HELP SCREENS
- DIAGNOSTIC MESSAGES
- PROGRAM FUNCTION (PF) KEYS

INITIAL SCREEN

After logon to the system and selection of B-52 or KC-135, the initial screen is displayed. The user must enter an access password - part of the security system devised to protect the integrity of the main data base.

The year and quarter of the most current data base is also displayed. The user is requested to input the date desired. Since quarterly data are stored in the data base for the current and three previous quarters, the user is given the option of querying earlier quarters.

INITIAL SCREEN

```
ENTER ACCESS PASSWORD ======>                                   PF1 FOR HELP
                                                                PF3 TO EXIT

    AAAAAA          RRRRRRR            TTTTTTTT          SSSSSS
   AA    AA         RR     RR             TT           SS     SS
   AA    AA         RR     RR             TT           SS
   AAAAAAAA         RRRRRRR               TT             SSSSSS
   AA    AA         RR   RR               TT                  SS
   AA    AA         RR    RR              TT           SS     SS
   AA    AA         RR     RR             TT             SSSSSS

   AUTOMATED        RELATIONAL         TRACKING          SYSTEM

 ENTER THE QUARTER DESIRED (YYQQ)======> 8502
(DATABASE IS COMPLETE THROUGH 8502)

LAST RUN DATES OF:
          FLIGHT HOURS LOAD     DELTA     DAMAGE     DAMAGE STATUS
                8502            8502       8502          8502
```

MAIN MENU

The main menu, shown here, contains all the options available to the user. As a part of the data base security, the access password, entered on the previous screen, determines the options shown on the screen. The ASIMIS (MMO) personnel responsible for loading data and making quarterly IATP runs have all the options. System managers might have only options 15 and 16 although, as will be seen later, all options are available to them when operating in a sub-data base environment.

Each option number has the same function in both the B-52 and KC-135 systems.

The ARTS system is used with terminals which have special keys called Program Function or "PF" keys. These keys can be used to communicate with ARTS via a single keystroke. There is a complete word description of the ARTS system that can be accessed by PF1 for help.

A typical selection might be Option 7 to accomplish a TCTO. The user would enter **7** in the **ENTER SELECTION** field and press **ENTER**.

MAIN MENU

------------------------ARTS MAIN MENU------------------------

ENTER SELECTION ===> 7

QUARTER 8502 DATA BASE B52B

1) CREATE NEW FLIGHT HOURS TABLE
3) EDIT NEW FLIGHT HOURS TABLE
5) ADD/UPDATE BASE INFORMATION
6) ADD/UPDATE COMMAND/SOURCE OF REPAIR INFORMATION
7) ADD/UPDATE/ACCOMPLISH TCTO
8) UPDATE PDM DATA
10) PRODUCE DELTA REPORTS
11) RUN DAMAGE PROGRAM
12) RUN DAMAGE STATUS PROGRAM
13) RUN SYSTEM (DEFAULT REPORT SELECTIONS)
14) IATP REPORT SELECTION
15) QUERY MAIN DATA BASE
16) WHAT IF GAMES (ENTER SUBDATA BASE NAME) ==> ASNDC
19) RUN USAGE 1 AND 2 PROGRAMS
22) CREATE G033 CURRENT DATA TABLE
23) ADD/UPDATE COMMENT INFORMATION
25) VIEW JOB OUTPUT
26) CHECK LOG FILES ==> _ EXLOG
28) DELETE SUB-DB FILES
30) VIEW SPACE/SUB-DB FILE NAMES

PF1 HELP
PF3 EXIT
PF14 FLEET DESC
PF16 TCTO
PF17 G033 DATA
PF18 DETAIL
PF22 BASE
PF23 SOR
PF24 CMD

SECONDARY MENU (SELECTION 7)

The secondary menu, shown here, provides several options to the user. Six different maintenance actions are available. The PF16 key will provide a table of the TCTO's currently in the data base. For more information on how to use the screen and enter data, the user may use the PF1 key for the tutorial.

SECONDARY MENU (SELECTION 7)

```
QUARTER 8502-------------ADD/UPDATE/ACCOMPLISH TCTO MENU----------------PF1 FOR HELP
DATA BASE B52B                                                          PF3 TO EXIT
                                                                        PF16 TCTO

ENTER SELECTION ====>

        1) ADD A NEW TCTO TO THE DATA BASE
        2) ACCOMPLISH A TCTO ON AN AIRPLANE
        3) UPDATE GENERAL DATA ABOUT A TCTO NOT ACCOMPLISHED ON AN AIRPLANE
        4) DISPLAY DATA ABOUT AN EXISTING TCTO
        5) UPDATE KIT COSTS AND THE YEAR OF THOSE COSTS ON A TCTO
        6) UPDATE AIRPLANE TCTO DATA ENTERED FOR A FUTURE QUARTER

     ENTER TCTO NUMBER                          ====>
     ENTER SERIAL NUMBER (FOR SELECTION 2,6)    ====>
```

HELP SCREEN (PF1)

This is a typical example of a help screen. A more detailed explanation of each option may be displayed in sequence by pressing **ENTER**. For information about a specific option, the number is entered in the **NEXT SELECTION** field. The PF3 key is used to return to the menu.

Help screens provide immediate access to information from any screen in the system. The complete tutorial may be accessed from the main menu.

HELP SCREEN (PF1)

TUTORIAL--------------------UPDATE/CORRECT TCTO INFORMATION------------------------TUTORIAL
NEXT SELECTION ====>   PF3 TO EXIT

BY SELECTING OPTION 7, ADD/UPDATE/ACCOMPLISH TCTO, THE USER IS GIVEN THE OPPORTUNITY OF ACCOMPLISHING ONE OF THE FOLLOWING:

1) ADD/UPDATE A NEW TCTO TO THE DATA BASE
   INFORMATION REQUIRED: TCTO NUMBER AND DATATO BE ENTERED
2) ACCOMPLISH A TCTO ON AN AIRPLANE
   INFORMAITON REQUIRED: A/P TAIL NUMBER AND TCTO APPLICATION DATA
3) DISPLAY INFORMATION ABOUT A TCTO
4) UPDATE KIT COSTS AND THE YEAR OF THOSE COSTS ON THE TCTO
5) UPDATE AIRPLANE TCTO DATA ENTERED FOR A FUTURE QUARTER

THE TCTO NUMBER TO BE EITHER INPUT OR ACCOMPLISHED ON AN AIRPLANE MUST BE ENTERED AND MAY BE UP TO 5 NUMERIC CHARACTERS. FOR OPTIONS 2 AND 5 ABOVE, A SEVEN CHARACTER NUMERIC SERIAL NUMBER MUST BE ENTERED.

ADDITIONAL INFORMATION ON THE ABOVE OPTIONS IS PRESENTED IN SEQUENCE, OR MAY BE SELECTED BY ENTERING THE APPROPRIATE CODE IN THE "NEXT SELECTION" FIELD.

DIAGNOSTIC MESSAGE

As data are entered into the system, internal checks are made for validity. If discrepancies are found, a message is displayed on the sceen. The PF1 key will cause a more detailed message to appear. Pressing the PF1 key again will bring up the tutorial.

On the example, both messages are displayed. The user attempted to update maintenance data for a TCTO installed in a previous quarter.

DIAGNOSTIC MESSAGE

```
QUARTER 8502-----------------ADD/UPDATE/ACCOMPLISH TCTO MENU----------------------CANNOT UPDATE
DATA BASE B52B                                                                   PF3 TO EXIT
THIS ROW OF DATA CANNOT BE UPDATED BECAUSE IT IS NOT IN THE FUTURE               PF16 TCTO

ENTER SELECTION ====> 6

        1) ADD A NEW TCTO TO THE DATA BASE
        2) ACCOMPLISH A TCTO ON AN AIRPLANE
        3) UPDATE GENERAL DATA ABOUT A TCTO NOT ACCOMPLISHED ON AN AIRPLANE
        4) DISPLAY DATA ABOUT AN EXISTING TCTO
        5) UPDATE KIT COSTS AND THE YEAR OF THOSE COSTS ON A TCTO
        6) UPDATE AIRPLANE TCTO DATA ENTERED FOR A FUTURE QUARTER

    ENTER TCTO NUMBER                           ====> 2253
    ENTER SERIAL NUMBER (FOR SELECTION 2, 6)    ====> 5800217
```

PROGRAM FUNCTION KEY (PF17)

The PF keys are utilized in the system to accomplish certain functions and to display basic data stored in the data base. PF1 (help) and PF3 (exit) have already have discussed. PF7 and PF8 are used to page forward and back in tables displayed for viewing. PF14, PF16, PF17, PF18 and PF21 through PF24 are used to display tables. They are active from any screen in the system and the keys are consistent in both systems, B-52 and KC-135. The data are only displayed and cannot be changed by the user.

PF17 was utilized to display the most current flight hour data available. Due to the time required to process flight log data, the quarterly IATP runs are usually a few months behind the current date. The G033 data are available and loaded into the data base monthly.

PROGRAM FUNCTION KEY (PF17)

```
= = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = =  ROW 1 OF 263
                          LISTING OF CURRENT G033 DATA                            PF1 FOR HELP
COMMAND = = >                  FROM:  86-04-01   TO:  86-04-30                    PF3 TOEXIT
S = > CSR                                                                         PF7 PAGE BACK
= = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = =  PF8 PAGE FWD
```

TAIL NUMBER	FLDR	CUM. FLT HRS.	****** MONTH LAST FLOWN ****** DATE	FLTS	HOURS	LDGS	BASE CODE	ORG	TYPE MISS
5706468		12214.7	86-04-30	001	5.2	001	NRCH	0042	T3M
5706469		15330.9	86-03-31	006	37.4	016	PLXL	0320	T3M
5706470		10925.8	86-04-30	007	51.0	025	JREZ	0416	T3M
5706471		11056.9	86-03-31	008	54.6	016	ZJXD	0379	T3M
5706572		11054.2	86-04-30	001	7.8	008	DESR	0093	T20
5706473		10619.4	86-04-30	004	35.3	010	NRCH	0042	T3M
5706474		9141.7	86-04-30	005	37.2	013	AWUB	0002	T3M
5706475	R	10754.5	86-04-30	005	34.1	012	AWUB	0002	T3M
5706476		11307.7	86-04-30	007	62.0	013	NRCH	0042	T36
5706477	R	13877.7	86-04-30	012	90.6	083	DESR	0093	T20
5706478		11888.3	86-04-30	004	24.5	012	NRCH	0042	T3M
5706480		13899.6	86-03-31	001	9.3	003	JFSD	0319	T3H
5706483		13554.5	86-02-28	006	36.6	022	AWUB	0002	T3M
5706484		11143.3	84-12-31	002	15.1	007	NRCH	0042	T30
5706485		12952.3	86-02-28	002	11.3	005	BWKR	0097	T3M
5706486	R	13312.7	86-04-30	008	63.9	070	DESR	0093	T20

PROGRAM FUNCTION KEY (PF22)

Another example of a table displayed by the use of a PF key is shown here. All the bases currently in the data base are shown with code number, ICAO and AF acronyms and description. This table is automatically updated by the system as new bases are entered into the data base using Option 5.

PROGRAM FUNCTION KEY (PF22)

```
= = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = =ROW 1 OF 28
                              LISTING OF BASES                                PF1 FOR HELP
COMMAND = = >                                                                 PF3 TO EXIT
S = > CSR                                                                     PF7 PAGE BACK
= = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = = =PF8 PAGE FWD
```

BASE CODE	BASE ACRONYM (ICAO)	BASE DESCRIPTION	BASE ACRONYM (AIR FORCE)
1	KBAD	BARKSDALE AFB LA.	AWUB
2	KBYH	BLYTHEVILLE AFB ARK.	BWKR
3	KMER	CASTLE AFB CALIF.	DESR
4	KSKA	FAIRCHILD AFB WASH.	GJKZ
5	KRME	GRIFFISS AFB N.Y.	JREZ
6	KLIZ	LORING AFB MAINE	NRCH
7	KMHR	MATHER AFB CALIF.	PLXL
8	KWRB	ROBINS AFB GEORGIA	UHHZ
9	KGSB	SEYMOUR-JOHNSON N.C.	VKAG
10	KOSC	WURTSMITH AFB MICH.	ZJXD
11	KEDW	EDWARDS AFB CALIF.	FSPM
12	KRDR	GRANDFORKS N.D.	JFSD
14	PGUA	ANDERSEN AFB GUAM	AJJY
21	KMER	CASTLE AFB CALIF.	DESR
22	KRCA	ELLSWORTH AFB S.D.	FXBM
23	KRDR	GRAND FORKS N.D.	JFSD
24	KSAW	KI SAWYER AFB MICH.	LWRC

"QUERY" - DATA RETRIEVAL SYSTEM

One of the most valuable tools provided by the ARTS system is the capability to query the data base. This is interactive and provides instant access to data stored in the data base.

ARTS
AUTOMATED RELATIONAL TRACKING SYSTEM

- INDIVIDUAL AIRPLANE AND FLEET TRACKING
- RELATIONAL INFORMATION MANAGEMENT SYSTEM
- USER FRIENDLY-MENU DRIVEN SYSTEM
- "QUERY" - DATA RETRIEVAL SYSTEM
- "WHAT IF" - IMPACT OF PROPOSED CHANGES

"QUERY"
AN INTERACTIVE DATA BASE QUERY SYSTEM

Query provides access to the data stored in the data base, both the quarterly data which is replaced every quarter and the historical data added on every quarter.

Sub-data bases are created under this option. The user also has the power to select data, view them, rearrange them and write them to a hard copy report.

Query is entered by selection of Option 15 on the Main Menu.

"QUERY"

AN INTERACTIVE DATA BASE QUERY SYSTEM

- ACCESS TO CURRENT AND HISTORICAL DATA
- CREATION OF SUB-DATA BASES
- USER DESIGNED REPORTS

ARTS MAIN MENU

The main menu is shown with Option 15 entered in the selection field. Note the date shown on the menu. This is the date entered by the user on the initial screen and, for quarterly data, will be the date associated with the data displayed in Query.

ARTS MAIN MENU

```
---------------------------------------------ARTS MAIN MENU---------------------------------------------
ENTER SELECTION = = = > 15

              QUARTER 8502                              DATA BASE B52B

    1)  CREATE NEW FLIGHT HOURS TABLE                          PF1   HELP
    3)  EDIT NEW FLIGHT HOURS TABLE                            PF3   EXIT
    5)  ADD/UPDATE BASE INFORMATION                            PF14  FLEET DESC
    6)  ADD/UPDATE COMMAND/SOURCE OF REPAIR INFORMATION        PF16  TCTO
    7)  ADD/UPDATE/ACCOMPLISH TCTO                             PF17  G033 DATA
    8)  UPDATE PDM DATA                                        PF18  DETAIL
   10)  PRODUCE DELTA REPORTS                                  PF22  BASE
   11)  RUN DAMAGE PROGRAM                                     PF23  SOR
   12)  RUN DAMAGE STATUS PROGRAM                              PF24  CMD
   13)  RUN SYSTEM (DEFAULT REPORT SELECTIONS)
   14)  IATP REPORT SELECTION
   15)  QUERY MAIN DATA BASE
   16)  WHAT IF GAMES (ENTER SUBDATA BASE NAME) = = > ASNDC
   19)  RUN USAGE 1 AND 2 PROGRAMS
   22)  CREATE G033 CURRENT DATA TABLE
   23)  ADD/UPDATE COMMENT INFORMATION
   25)  VIEW JOB OUTPUT
   26)  CHECK LOG FILES = = > __ EXLOG
   28)  DELETE SUB-DB FILES
   30)  VIEW SPACE/SUB-DB FILE NAMES
```

QUERY SELECTION

The seven data relations are displayed for the user to select the desired one. Airplane, base and fleet data relations contain quarterly data generated every quarter. The other relations expand every quarter and some, such as PDM history and TCTO, may contain data for future quarters, not currently entered into the IATP system, but available for display in Query.

Quarterly data are stored in the data base for the current and three previous quarters.

QUERY SELECTION

```
QUARTER 8502----------------------QUERY DATA SELECTION----------------------PF1 FOR HELP
DATA BASE B52B                                                               PF3 TO EXIT
ENTER SELECTION = = = >

                         1) AIRPLANE DATA
                         2) BASE DATA
                         3) BASE HISTORY
                         4) COMMENT DATA
                         5) FLEET DATA
                         6) PDM HISTORY DATA
                         7) TCTO DATA
```

QUERY AIRPLANE DATA

The airplane data screen is displayed with all the variables stored in that relation. The PF1 help screen will display definitions of these. The first four columns are single attributes; the last two are detail variables and represent arrays of values for each structural detail monitored

QUERY AIRPLANE DATA

QUARTER 8502----------INITIAL SELECTION----------PF1 FOR HELP
DATA BASE B52B PF3 TO EXIT
ENTER SELECTION "D" - NEXT TO THE ATTRIBUTES TO BE DISPLAYED
"W" - NEXT TO THE ATTRIBUTES TO BE USED FOR CONSTRAINTS

*********	AIRPLANE	DATA	*********		
		HR DATA	S75 DATA	DET DATA	
S/N	LOG HRS	REF HRS	DHR S75	CRACK L	DET CONF
MODEL #	TG/AP	DEL HRS	LHR S75	AP GFR	#MODS
BASE #	FS/AP	ACT HRS	MSN S75	HR-CRIT	#INSP
CUR DAT	GND ALT	PAT HRS	#TG S75	HR-INSP	RHRS SB
OWN CMD	#MSN	LLC HRS	#FS S75	2ND-INS	RHRS EB
USR CMD	#EMSN	LLTA HR		HR-ERL	EQB HRS

QUERY AIRPLANE DATA

Variables to be displayed are selected by placing a **D** to the left of the variable name. A maximum of 13 may be selected. 'Where' constraints (variables to be searched on) are selected by placing a **W** to the left of the variable name. A maximum of 8 may be selected. **DW** will cause a variable to be displayed as well as used as a constraint. **W** alone will use it as a constraint but not display it. If multiple lines are needed to enter constraints (e.g., twenty tail numbers at six per line - 56 characters per line), **DW***n may be entered. In the example, the user would enter **DW***4.

QUERY AIRPLANE DATA

QUARTER 8502--------------------------------INITIAL SELECTION--------------------------------PF1 FOR HELP
DATA BASE B52B PF3 TO EXIT
ENTER SELECTION "D" - NEXT TO THE ATTRIBUTES TO BE DISPLAYED
"W" - NEXT TO THE ATTRIBUTES TO BE USED FOR CONSTRAINTS

********* AIRPLANE DATA *********

					HR DATA	S75 DATA	DET DATA	
D	S/N	D	LOG HRS		REF HRS	DHR S75	CRACK L	DET CONF
D	MODEL #	D	TG/AP		DEL HRS	LHR S75	AP GFR	#MODS
DW	BASE #	D	FS/AP	D	ACT HRS	MSN S75	HR-CRIT	#INSP
	CUR DAT		GND ALT		PAT HRS	#TG S75	HR-INSP	RHRS SB
	OWN CMD		#MSN		LLC HRS	#FS S75	2ND-INS	RHRS EB
	USR CMD		#EMSN	D	LLTA HR		HR-ERL	EQB HRS

QUERY AIRPLANE DATA

After entering the desired **D**'s and **W**'s, the user presses **ENTER**. This will cause the variables selected with a **W** to be displayed on the screen requesting the constraints to be entered.

On the example, base code is used as a constraint. Base code numbers plus 100 are used to indicate aircraft with the OAS modification installed. Here both pre and post-OAS aircraft at the Castle AFB were selected by entering **EQ 3, 103** on the screen. PF22 key will cause a table of all base codes to be displayed.

If multiple **W**'s had been entered, a line for each constraint would be displayed. The standard FORTRAN syntax (EQ, LT, LE, GT, GE, AND and OR) is used to enter constraints. Again PF1 will provide help.

QUERY AIRPLANE DATA

```
QUARTER 8502-------------------------------INITIAL SELECTION--------------------------------PF1 FOR HELP
DATA BASE B52B                                                                              PF3 TO EXIT
ENTER SELECTION "D"  - NEXT TO THE ATTRIBUTES TO BE DISPLAYED
                "W"  - NEXT TO THE ATTRIBUTES TO BE USED FOR CONSTRAINTS

     *********      AIRPLANE              DATA        *********
                                          HR DATA     S75 DATA     DET DATA

D    S/N          D  LOG HRS              REF HRS     DHR S75      CRACK L      DET CONF
D    MODEL #      D  TG/AP                DEL HRS     LHR S75      AP GFR       #MODS
DW   BASE #       D  FS/AP           D    ACT HRS     MSN S75      HR-CRIT      #INSP
     CUR DAT         GND ALT              PAT HRS     #TG S75      HR-INSP      RHRS SB
     OWN CMD         #MSN                 LLC HRS     #FS S75      2ND-INS      RHRS EB
     USR CMD         #EMSN           D    LLTA HR                  HR-ERL       EQB HRS

ENTER WHERE CONSTRAINTS
BASE #           EQ 3, 103
```

QUERY AIRPLANE DATA

After entering all the desired constraints and pressing **ENTER**, a count of the number of rows that meet the constraints is displayed. The example shows there are 26 airplanes at Castle.

QUERY AIRPLANE DATA

```
QUARTER 8502--------------------------------INITIAL SELECTION--------------------------------PF1  FOR  HELP
DATA BASE B52B                                                                                PF3  TO  EXIT
ENTER SELECTION  "D"   - NEXT TO THE ATTRIBUTES TO BE DISPLAYED
                 "W"   - NEXT TO THE ATTRIBUTES TO BE USED FOR CONSTRAINTS

     *********         AIRPLANE          DATA           *********
                                         HR DATA        S75 DATA      DET DATA

D    S/N          D    LOG HRS           REF HRS        DHR S75       CRACK L       DET CONF
D    MODEL #      D    TG/AP             DEL HRS        LHR S75       AP GFR        #MODS
DW   BASE #       D    FS/AP        D    ACT HRS        MSN S75       HR-CRIT       #INSP
     CUR DAT           GND ALT           PAT HRS        #TG S75       HR-INSP       RHRS SB
     OWN CMD           #MSN              LLC HRS        #FS S75       2ND-INS       RHRS EB
     USR CMD           #EMSN        D    LLTA HR                      HR-ERL        EQB HRS

NUMBER OF ROWS SELECTED = 26
BASE #           EQ 3, 103
```

DISPLAY OF SELECTED AIRPLANE DATA
(TYPE I)

This shows a typical display of airplane single attributes. Each row represents a single airplane. Only eight variables were selected; the maximum is thirteen. The user may page **RIGHT** or **LEFT** and **UP** or **DOWN** to view the display. The **D**'s followed by numbers in the upper left corner indicate the order variables are displayed. By reordering the numbers and entering **DISPLAY** in the **ENTER COMMAND** field, the user may rearrange the columns displayed. Thus the user can design the reports to suit requirements. If **WRITE** is entered in the command field, the variables displayed on the screen will be written to a hard copy report in the same format.

If the user plans to use the What-If capability and wishes to create a sub-data base of this set of airplanes, **CREATE** is entered in the command field. A screen is then brought up for the user to enter data to launch a batch job to create a sub-data base.

DISPLAY OF SELECTED AIRPLANE DATA
(TYPE 1)

QUARTER 8502----------------------------------DISPLAY SELECTIONS----------------------------------PF1 FOR HELP
DATA BASE B52B PF3 TO EXIT

DATA LOADED FOR THE FOLLOWING ATTRIBUTES
"D"N - NEXT TO THE ATTRIBUTES TO BE DISPLAYED WHERE N = COLUMN NO.

D1	S/N	D5	BASE#
D2	LOG HRS	D6	FS/AP
D3	MODEL #	D7	ACT HRS
D4	TG/AP	D8	LLTA HR

ENTER COMMAND = = = > DOWN

S/N	LOG HRS	MODEL #	TG/AP	BASE#	FS/AP	ACT HRS	LLTA HR
5706472	181.	1	94	103	16	10472.	9.
5706486	203.	1	97	103	14	12658.	6.
5706487	130.	1	49	103	12	15855.	7.
5706492	157.	1	85	103	16	11943.	6.
5706515	156.	1	83	103	10	11112.	1.
5800159	179.	1	86	103	16	11477.	7.
5800164	145.	1	79	103	10	12412.	2.
5800191	168.	1	94	103	12	12619.	5.
5800193	193.	1	119	103	14	11276.	6.
5800195	0.	1	0	3	0	10279.	0.
5800203	85.	1	51	3	9	10177.	3.
5800206	78.	1	29	3	6	10625.	6.

DISPLAY OF SELECTED AIRPLANE DATA (TYPE 2)

This screen shows a second type of data display which includes four single value airplane attributes plus five detail arrays. Here, there is a row displayed for each detail selected. Again, the order of the columns may be changed by renumbering the D's. Again, **UP**, **DOWN**, **RIGHT** and **LEFT** commands enable the user to view the display.

DISPLAY OF SELECTED AIRPLANE DATA

(TYPE 2)

QUARTER 8502-----------------------------DISPLAY SELECTIONS--------------------------------PF1 FOR HELP
DATA BASE B52B PF3 TO EXIT

DATA LOADED FOR THE FOLLOWING ATTRIBUTES
"D"N - NEXT TO THE ATTRIBUTES TO BE DISPLAYED WHERE N = COLUMN NO.

D1 S/N	D1	CRACK L	D5 HR-INSP
D2 LOG HRS	D2	DET CONF	
D3 BASE #	D3	AP GFR	
D4 ACT HRS	D4	HR-CRIT	

ENTER COMMAND = = = > DOWN

S/N	LOG HRS	BASE #	ACT HRS		
5706472	181.	103	10472.		
DETAIL	CRACK LG	DET CONF	AP GFR	HR-CRIT	HR-INSP
WL-47-2	0.04980	1	1.75347	24533.	8092.
WM-12	0.09113	1		34282.	12967.
WL-65	0.08199	1	3.22237	10602.	1126.
WL-8-1	0.0319	1		23511.	7581.
WL-29-1	0.03864	1		35387.	13519.
WL-45-1	0.04282	1		27891.	9771.
WL-62-1	0.03881	1		28817.	10234.
WL-79-1	0.03947	1		35198.	13424.
WU-1	0.57207	1	2.99573	4016.	-2167.
WU-7	0.10845	1	3.61387	-322.	-4336.

DISPLAY OF SELECTED AIRPLANE DATA
(TYPE 3)

The third type of data display is shown. Two detail attributes for three details plus one airplane attribute are shown on each row. The criteria for this display is that the total number of variables requested for display is less than or equal to eight.

DISPLAY OF SELECTED AIRPLANE DATA
(TYPE 3)

QUARTER 8502--------------------------------DISPLAY SELECTIONS---------------------------------PF1 FOR HELP
DATA BASE B52B PF3 TO EXIT

DATA LOADED FOR THE FOLLOWING ATTRIBUTES
"D"N - NEXT TO THE ATTRIBUTES TO BE DISPLAYED WHERE N = COLUMN NO.

D1 S/N
D2 ACT HRS
D1 CRACK L
D2 HR-CRIT

ENTER COMMAND = = = > DOWN

S/N	ACT HRS	CRACK L WL-47-2	HR-CRIT WL-47-2	CRACK L WM-12	HR-CRIT WM-12	CRACK L WL-65	HR-CRIT WL-65
5706472	10472.	0.04980	24533.	0.09113	34282.	0.08199	10602.
5706486	12658.	0.05962	22810.	0.10548	32559.	0.11536	8948.
5706487	15855.	0.06605	21823.	0.11673	31573.	0.14045	8008.
5706492	11943.	0.05538	23459.	0.09882	32208.	0.10260	9530.
5706515	11112.	0.05344	23758.	0.09669	33507.	0.09441	9916.
5800159	11477.	0.05582	23392.	0.09930	33142.	0.10158	9578.
5800164	12412.	0.05772	23100.	0.10217	32849.	0.10867	9245.
5800191	12619.	0.05631	23317.	0.09984	33066.	0.10515	9410.
5800193	11276.	0.05467	23569.	0.09804	33318.	0.09946	9678.
5800195	10279.	0.05048	25466.	0.09253	35667.	0.08488	10600.
5800203	10177.	0.04883	25967.	0.08910	36168.	0.07935	10953.

DISPLAY OF COMMENT DATA

After entry of a "4" on the Query Data Selection screen, the comment selection screen is shown. After the display selections "D" have been made and a constraint "W" entered, the entire relation will be displayed.

The comment relation provides a place to store special data about a particular aircraft. This information is stored by tail number and date. The maintenance relation includes a comment flag. If something special occurred during the inspection or modification of a particular aircraft, the comment flag can be entered while the TCTO is being accomplished. If the maintenance data for a particular aircraft are being scanned, the flag will indicate there is more information in the comment relation.

DISPLAY OF COMMENT DATA

```
QUARTER 8502--------------------------DISPLAY SELECTIONS---------------------------PF1 FOR HELP
DATA BASE B52B                                                                      PF3 TO EXIT
                        DATA LOADED FOR THE FOLLOWING ATTRIBUTES
                        "D"N - NEXT TO THE ATTRIBUTES TO BE DISPLAYED WHERE N = COLUMN NO.

D1  S/N
D2  EFF DATE
D3  COMMENT

ENTER COMMAND = = = > DOWN

 S/N        EFF DATE    COMMENT

5706484     840615      MATHER AIRPLANE.  ECP 1175-216 WING REPAIR
5800178     860527      LORING AIRPLANE.  ECP 1175-221 FWD BODY REPAIR
5800245     810610      THIS A/P WAS THE OAS FLIGHT TEST A/P WITH FSD
                        HARDWARE INSTALLED.  PRODUCTION OAS HARDWARE
                        WAS INSTALLED IN 1984 DURING PDM.
6000020     841001      THIS A/P WAS THE FIRST PRODUCTION H CMI A/P
6000050     850104      THIS A/P WAS THE H CMI KIT PROOF AIRPLANE
```

"WHAT-IF" - IMPACT OF PROPOSED CHANGES

Potentially the most valuable contribution of ARTS to the system manager is **WHAT-IF** capability. The ability to create a sub-data base containing the airplanes of interest, make changes to variables such as planned hours per year (used in all future projections), next PDM date or PDM interval and then make a run through the system programs provides a valuable tool. Since the data base contains growth factor data covering the operating range of the aircraft, almost any set of proposed missions can be analyzed and the effect upon future maintenance actions evaluated.

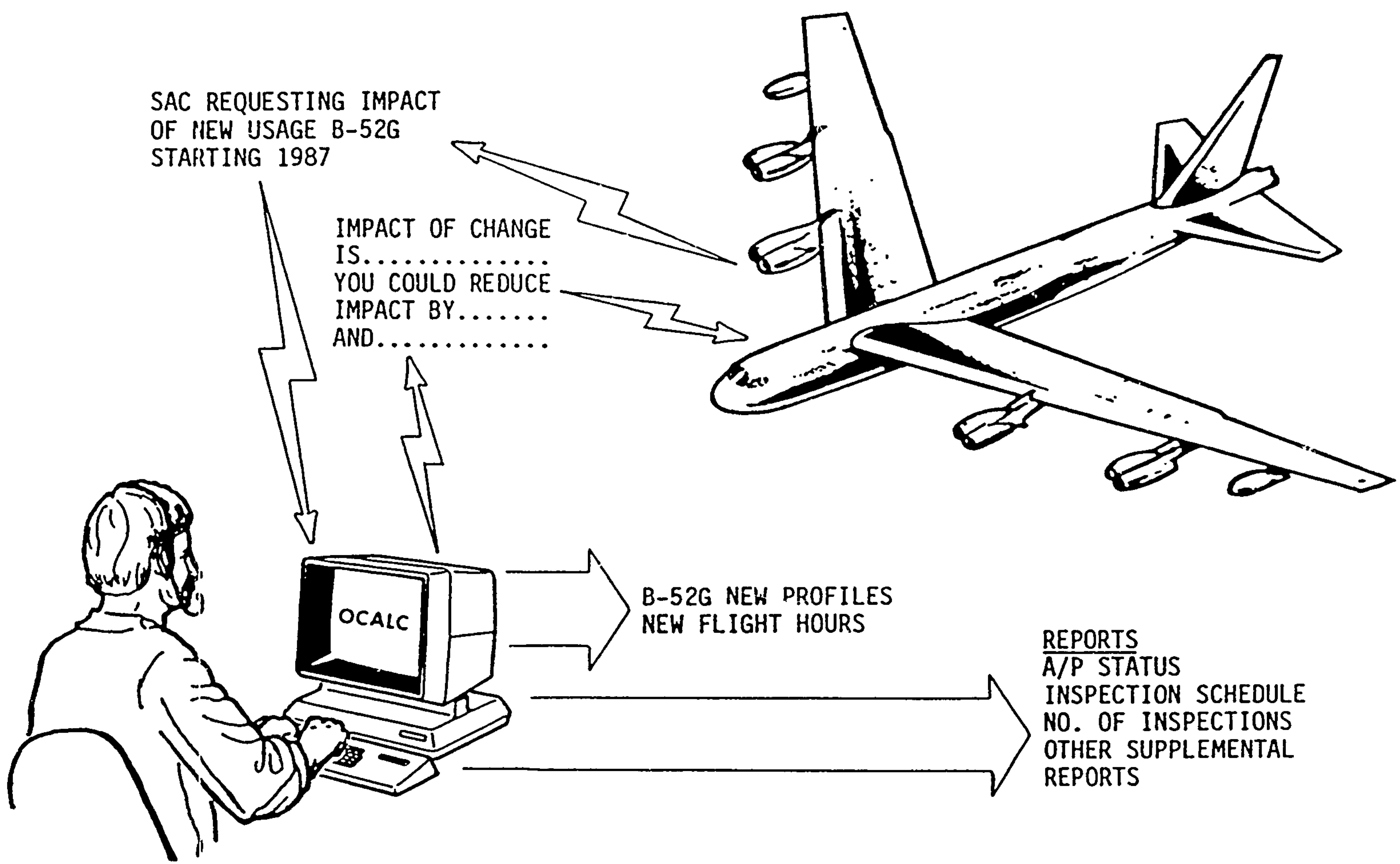
ARTS
AUTOMATED RELATIONAL TRACKING SYSTEM

- INDIVIDUAL AIRPLANE AND FLEET TRACKING
- RELATIONAL INFORMATION MANAGEMENT SYSTEM
- USER FRIENDLY-MENU DRIVEN SYSTEM
- "QUERY" - DATA RETRIEVAL SYSTEM
- "WHAT IF" - IMPACT OF PROPOSED CHANGES

WHAT IF?

A typical example of What-if games is shown here with SAC requesting what the impact of proposed B-52 usage would have on maintenance schedules. The system manager at Oklahoma City can evaluate the usage and respond in a relatively short period of time.

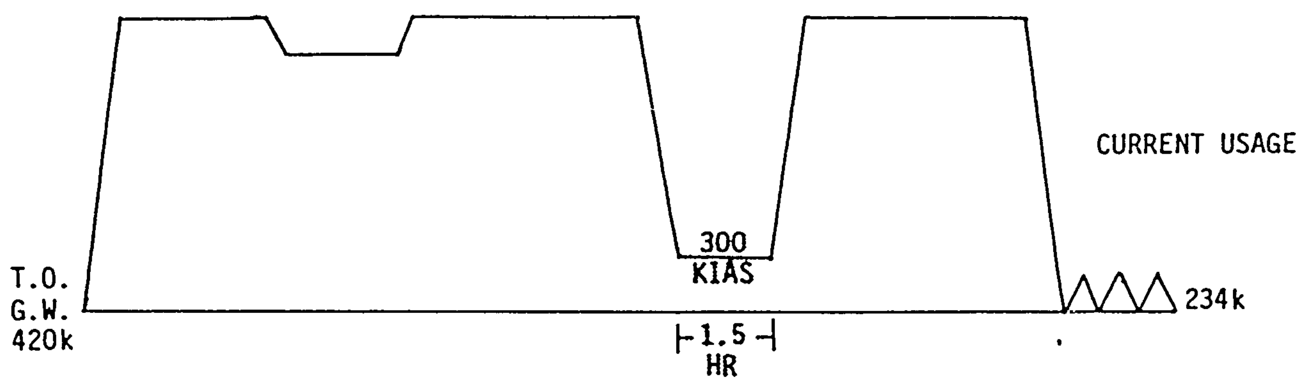
"WHAT IF?"

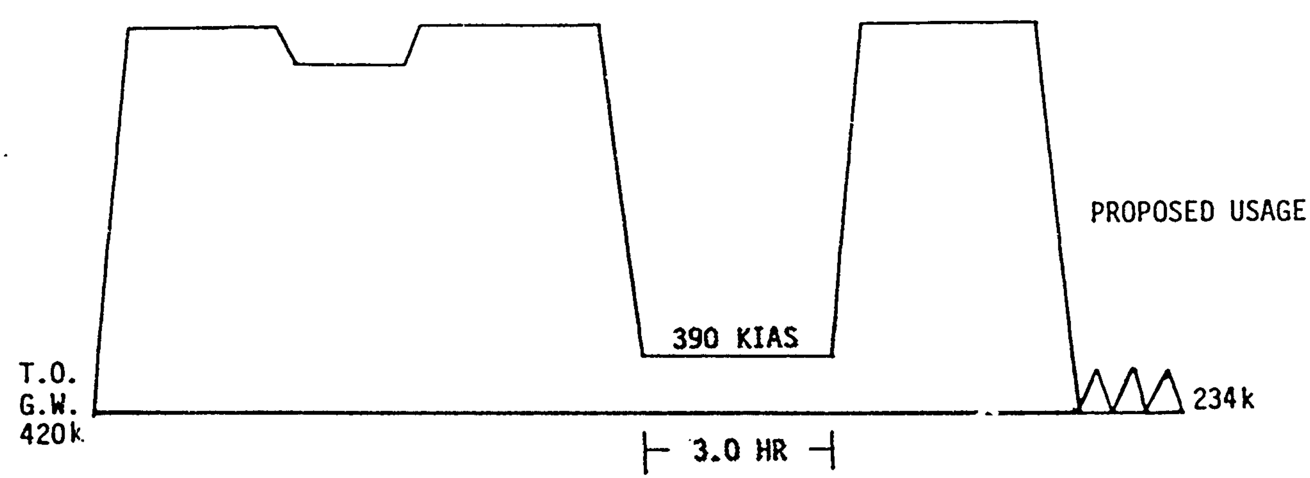

SAC REQUESTING IMPACT
OF NEW USAGE B-52G
STARTING 1987
IMPACT OF CHANGE
IS..............
YOU COULD REDUCE
IMPACT BY.......
AND.............
OCALC
B-52G NEW PROFILES
NEW FLIGHT HOURS
REPORTS
A/P STATUS
INSPECTION SCHEDULE
NO. OF INSPECTIONS
OTHER SUPPLEMENTAL
REPORTS

"WHAT-IF" SCENARIO

The current usage includes missions with several hours of low level at 300 KIAS. SAC proposes to double the low level time and increase the airspeed to 390 KIAS, however retaining the same takeoff and landing weights and mission lengths. The mission shown has 1.5 low level hours which will be increased to 3.0 hours.

"WHAT IF" SCENARIO





ARTS MAIN MENU

A sub-data base is created for one or several airplanes with the desired flight hours (base code, configuration . . .). The user logs onto the system and selects option 16, entering the name of the sub-data base.

ARTS MAIN MENU

```
---------------------------ARTS MAIN MENU---------------------------
ENTER SELECTION ===> 16

          QUARTER 8502                        DATA BASE B52B

  1)   CREATE NEW FLIGHT HOURS TABLE                          PF1   HELP
  3)   EDIT NEW FLIGHT HOURS TABLE                            PF3   EXIT
  5)   ADD/UPDATE BASE INFORMATION                            PF14  FLEET DESC
  6)   ADD/UPDATE COMMAND/SOURCE OF REPAIR INFORMATION        PF16  TCTO
  7)   ADD/UPDATE/ACCOMPLISH TCTO                             PF17  G033 DATA
  8)   UPDATE PDM DATA                                        PF18  DETAIL
 10)   PRODUCE DELTA REPORTS                                  PF22  BASE
 11)   RUN DAMAGE PROGRAM                                     PF23  SOR
 12)   RUN DAMAGE STATUS PROGRAM                              PF24  CMD
 13)   RUN SYSTEM (DEFAULT REPORT SELECTIONS)
 14)   IATP REPORT SELECTION
 15)   QUERY MAIN DATA BASE
 16)   WHAT IF GAMES (ENTER SUBDATA BASE NAME) ==> TOXLL
 19)   RUN USAGE 1 AND 2 PROGRAMS
 22)   CREATE G033 CURRENT DATA TABLE
 23)   ADD/UPDATE COMMENT INFORMATION
 25)   VIEW JOB OUTPUT
 26)   CHECK LOG FILES ==> __ EXLOG
 28)   DELETE SUB-DB FILES
 30)   VIEW SPACE/SUB-DB FILE NAMES
```

SUB-DATA BASE MAIN MENU

The first step to evaluate the proposed mission usage is to create a sub-data base. Airplane selection is based upon user criteria: flight hours, base, configuration, average airplane . . . The data base is entered from the main menu in Query, the airplane or airplanes selected and the sub-data base created. After the sub-data base batch job has completed, the user logs on and selects option 16, entering the name of the sub-data base. The sub-data base main menu is displayed. Note that all options are available to the user in a sub-data base. Option 1 is first used to initialize the system and load AFTO data into the sub-data base. The user may have a data set of analysis missions available or AFTO data for those airplanes in the sub-data base can be retrieved from the quarterly tape. These can then be edited, repeated or deleted as desired using Option 20.

SUB–DATA BASE MAIN MENU

```
----------------------------------------SUB-DATA BASE MAIN MENU----------------------------------------
ENTER SELECTION ===> 20

               QUARTER 8502                          DATA BASE TOXLL

   1)  CREATE NEW FLIGHT HOURS TABLE                          PF1   HELP
   3)  EDIT NEW FLIGHT HOURS TABLE                            PF3   EXIT
   5)  ADD/UPDATE BASE INFORMATION                            PF14  FLEET DESC
   6)  ADD/UPDATE COMMAND/SOURCE OF REPAIR INFORMATION        PF16  TCTO
   7)  ADD/UPDATE/ACCOMPLISH TCTO                             PF17  G033 DATA
   8)  UPDATE PDM DATA                                        PF18  DETAIL
  10)  PRODUCE DELTA REPORTS                                  PF22  BASE
  11)  RUN DAMAGE PROGRAM                                     PF23  SOR
  12)  RUN DAMAGE STATUS PROGRAM                              PF24  CMD
  13)  RUN SYSTEM (DEFAULT REPORT SELECTIONS)
  14)  IATP REPORT SELECTION
  15)  QUERY/CHANGE SUB-DATA BASE
  16)  RUN USAGE 1 AND 2 PROGRAMS                                RUN DATES   SUB DB
  20)  AFTO UPDATE (SUB-DATA BASE)                               FLIGHT HRS  8502
  21)  WHAT-IF TOOLS                                             DELTA       8502
  25)  VIEW JOB OUTPUT                                           DAMAGE      8502
  26)  CHECK LOG FILES ==> __ EXLOG                              STATUS      8502
```

CURRENT USAGE

In this scenario, five missions typical of B-52G usage were loaded into the sub-data base. The AFTO update screen is displayed with the first mission for the first airplane. The PFI key may be used to get variable definitions or event codes. Changes are made by overtyping the displayed values. The command field (COM) for each flight segment is used to insert, copy or delete segments as a part of the edit process. **DOWN M**, **DOWN S/N**, **FIND xxxxxxx** and **UP** are commands available to scroll through the missions and serial numbers. **REPEAT** and **DELETE** commands repeat and delete missions. **COUNT** gives a tally of the number of missions for a particular airplane.

CURRENT USAGE

QUARTER 8502------------------------------------AFTO UPDATE------------------------------------PF1 FOR HELP
DATA BASE TOXLL PF3 TO EXIT

ENTER COMMAND = = = > DOWN

MSN NO: 1

S/N	FLT HR	BASE	DATE	TO GW	TO FW	INW	EXW	I T	E T	#	T#	MDUR	LD GW	W D	G A	M
5706490	11766.0	KRME	850625	420000	230000	0	0	0	0	0	10	820	234000	1	0	1

COM	EV	SALT	SGW	IAS	IWD	EWD	#	TG	FS	SDUR	EALT	EGW
	1	910	420000	268	0	0	1	0	0	39	29200	400000
	2	29200	400000	275	0	0	2	0	0	100	29000	381000
	4	29500	381000	274	0	0	3	0	0	73	28800	371000
	2	29500	371000	273	0	0	4	0	0	122	27300	347000
	3	27300	347000	272	0	0	5	0	0	29	1000	341000
	7	1000	341000	300	0	0	6	0	0	150	1000	303000
	1	1000	303000	275	0	0	7	0	0	27	30300	296000
	2	30300	296000	266	0	0	8	0	0	134	28900	271000
	3	29200	271000	255	0	0	9	0	0	31	2000	264000
	8	2000	264000	166	0	0	10	3	1	115	0	234000

PROPOSED USAGE

The edited mission is displayed. Gross weights and segment times have been adjusted to account for the increased low level time. Similar changes were made to the other missions. Returning to the sub-data base main menu, the user submitted system runs, Damage (Option 11), Status (Option 12) and Inspect (Option 14). Screens for submittal of these runs on a sub-data base allow the user to enter the frequency of each mission per year and stipulate that the composite usage rate calculated be used in Status to project inspection requirements.

PROPOSED USAGE

QUARTER 8502------------------------------------AFTO UPDATE------------------------------------PF1 FOR HELP
DATA BASE TOXLL PF3 TO EXIT

ENTER COMMAND * * * > DOWN

MSN NO: 1

S/N	FLT HR	BASE	DATE	TO GW	TO FW	INW	EXW	I T	E T	#	T#	MDUR	LD GW	W D	G A	M
5706490	11766.0	KRME	850625	420000	230000	0	0	0	0	0	10	820	234000	1	0	1

COM	EV	SALT	SGW	IAS	IWD	EWD	#	TG	FS	SDUR	EALT	EGW
	1	910	420000	268	0	0	1	0	0	39	29200	400000
	2	29200	400000	275	0	0	2	0	0	75	29000	388000
	4	29500	388000	274	0	0	3	0	0	48	28800	378000
	2	29500	378000	273	0	0	4	0	0	72	27300	363000
	3	27300	363000	272	0	0	5	0	0	29	1000	356000
	7	1000	356000	390	0	0	6	0	0	300	1000	285000
	1	1000	285000	275	0	0	7	0	0	27	30300	278000
	2	30300	278000	266	0	0	8	0	0	84	28900	271000
	3	29200	271000	255	0	0	9	0	0	31	2000	264000
	8	2000	264000	166	0	0	10	3	1	115	0	234000

USAGE COMPARISON

The Inspect-4 report for the airplane used for the evaluation is shown for the current IATP usage as of the second quarter of 1985. It is scheduled for PDM in the third quarter of 1986.

INSPECT-4 INDIVIDUAL A/P DETAIL INSPECTION CHART (20 YEARS): MODEL G DATE 11/18/86 QUARTER 8502 PAGE 1

S/N = 5706490
BASE CODE = 105
FLIGHT HOURS = 12671

	DETAIL NUMBER	CODE	1987	1989	1991	1993	1995	1997	1999	2001	2003	2005
		PDM	P		P		P		P		P	
	CP 1	WL-47-2										
	2	WM-12									X	
#	CP 3	WL-65			X				X			
	4	WL-8-1								X		
	5	WL-29-1										
	6	WL-45-1										
	7	WL-62-1										
	8	WL-79-1										
#	CP 9	WU-1						X				
#	CP 10	WU-7		X			X				X	
#	11	WU-7-1	X	X		X	X		X	X		X
#	12	WU-401	X		X		X	X		X		X
	CP 13	E-14										
	14	B-9										
#	CP 15	WM-1A	X X	X :	X X	X X	X X	X X	X	X X	X X	X X
#	16	WM-1	X	X		X		X		X		X
#	17	WM-2		X	X	X		X	X	X		X
#	18	WM-15	X X X	X X	X X	X X	X X	X X	X X	X X	X X	X X
	CP 19	E-1										
#	20	E-10			X							
	CP 21	B-2										X
	22	B-1									X	
	23	B-3-1						X				
	24	B-4-1										
	25	B-6										
#	CP 26	B-18			X					X		
	27	WM-8										
	28	WM-11							X			
	CP 29	B-25-2										
	30	B-32-1										X

P - SCHEDULED PDM
X - INSPECTION REQUIREMENT
- DELINQUENT PDM OR INSPECTIONS EXIST

USAGE COMPARISON

The inspection requirements with the proposed for the same airplane usage are shown h re. Note that many of the inspections are required earlier and more often. Some should be accomplished more than once in a PDM interval.

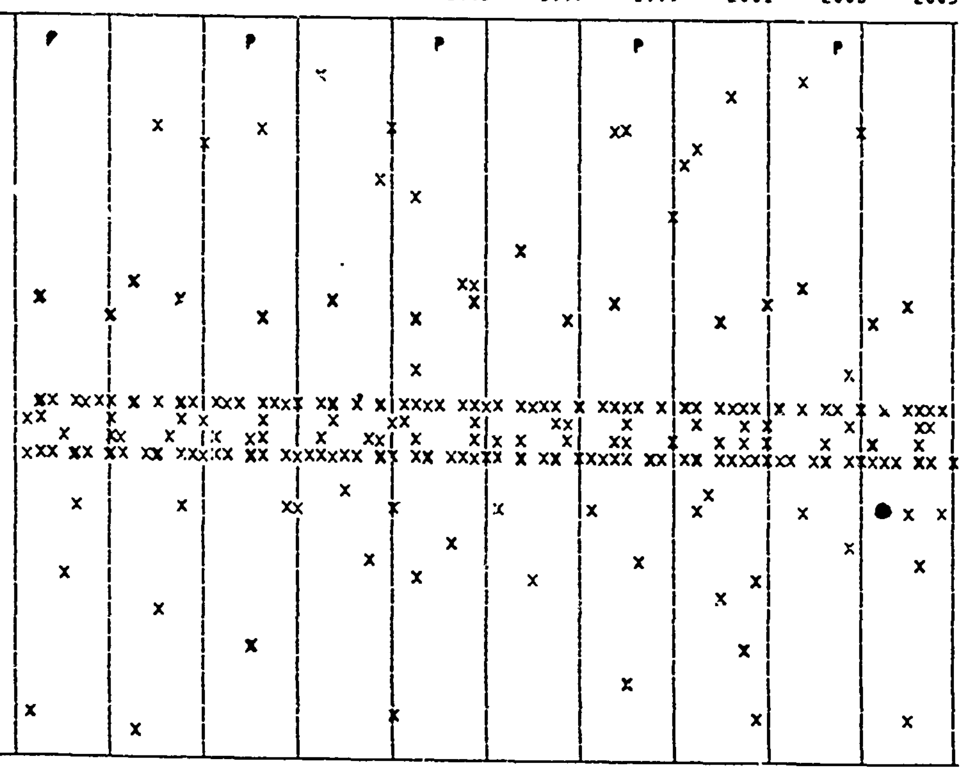
BOEING MILITARY AIRPLANE COMPANY

INSPECT-4 INDIVIDUAL A/P DETAIL INSPECTION CHART (20 YEARS): MODEL G   DATE 11/18/86   QUARTER 8502   PAGE 1

S/N = 5706490
BASE CODE = 105
FLIGHT HOURS = 12671

INSPECTION REQUIREMENTS: 1987 1989 1991 1993 1995 1997 1999 2001 2003 2005

	DETAIL NUMBER	CODE
		PDM
	CP 1	WL-47-2
	2	WM-12
●	CP 3	WL-65
	4	WL-8-1
	5	WL-29-1
	6	WL-45-1
	7	WL-62-1
	8	WL-79-1
●	CP 9	WU-1
●	CP 10	WU-7
●	11	WU-7-1
●	12	WU-401
	CP 13	E-14
	14	B-9
●	CP 15	WM-1A
●	16	WM-1
●	17	WM-2
●	18	WM-15
	CP 19	E-1
●	20	E-10
	CP 21	B-2
	22	B-1
	23	B-3-1
	24	B-4-1
	25	B-6
●	CP 26	B-18
	27	WM-8
	28	WM-11
	CP 29	B-25-2
	30	B-32-1



P - SCHEDULED PDM
X - INSPECTION REQUIREMENT
● - DELINQUENT PDM OR INSPECTIONS EXIST

ARTS
AUTOMATED RELATIONAL TRACKING SYSTEM
STATE OF THE ART FORCE MANAGEMENT TOOL

We felt that the IATP systems developed for Oklahoma City ALC to track the C/KC-135 and B-52 fleets were outstanding because of their ability both to cover the operational range of the aircraft on a flight-by-flight basis and to provide fleet management reports that included a wide variety of information about the past, current and future status for the aircraft.

With the advent of ARTS, the capabilities of the system managers have been expanded dramatically. Query allows them instant access to a wealth of information. If there is a report of an airplane in trouble or one with cracks or corrosion found during a PDM inspection, the manager can logon to the system and secure information about that airplane's past history and current status and search for comments about any peculiarities of the aircraft.

An even greater capability is provided with What-If. In this presentation, we touched on only one type of scenario, the evaluation of changes to missions in a planned usage. Effects of changes to planned hours per year, PDM intervals or other such variables could be evaluated. Because the system utilizes growth factors for a wide range of weights, altitudes, airspeeds and types of operation, changes in usage can be evaluated more readily than in a system using canned missions.

The possibilities for expansion for the ARTS systems are unlimited. A complete maintenance data base could be developed to interface with the IATP data base. Similar systems could be developed for other fleets of aircraft. Additionally, there are many capabilities available in RIM which have not yet been utilized in ARTS.

ARTS
AUTOMATED RELATIONAL TRACKING SYSTEM
STATE OF THE ART FORCE MANAGEMENT TOOL

- INDIVIDUAL AIRPLANE AND FLEET TRACKING
- RELATIONAL INFORMATION MANAGEMENT SYSTEM
- USER FRIENDLY-MENU DRIVEN SYSTEM
- "QUERY" - DATA RETRIEVAL SYSTEM
- "WHAT IF" - IMPACT OF PROPOSED CHANGES

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