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RESEARCH AND TECHNOLOGY ORGANIZATION

BP 25, 7 RUE ANCELLE, F-92201 NEUILLY-SUR-SEINE CEDEX, FRANCE




**RTO LECTURE SERIES 218 bis**

# Aging Aircraft Fleets: Structural and Other Subsystem Aspects

(le Vieillissement des flottes d'avions militaires : aspects structures et autres sous-systèmes)

*The material in this publication was assembled to support a Lecture Series under the sponsorship of the Applied Vehicle Technology Panel (AVT) and the Consultant and Exchange Programme of RTO presented 13-16 November 2000 in Sofia, Bulgaria.*


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# The Research and Technology Organization (RTO) of NATO

RTO is the single focus in NATO for Defence Research and Technology activities. Its mission is to conduct and promote cooperative research and information exchange. The objective is to support the development and effective use of national defence research and technology and to meet the military needs of the Alliance, to maintain a technological lead, and to provide advice to NATO and national decision makers. The RTO performs its mission with the support of an extensive network of national experts. It also ensures effective coordination with other NATO bodies involved in R&T activities.

RTO reports both to the Military Committee of NATO and to the Conference of National Armament Directors. It comprises a Research and Technology Board (RTB) as the highest level of national representation and the Research and Technology Agency (RTA), a dedicated staff with its headquarters in Neuilly, near Paris, France. In order to facilitate contacts with the military users and other NATO activities, a small part of the RTA staff is located in NATO Headquarters in Brussels. The Brussels staff also coordinates RTO's cooperation with nations in Middle and Eastern Europe, to which RTO attaches particular importance especially as working together in the field of research is one of the more promising areas of initial cooperation.

The total spectrum of R&T activities is covered by the following 7 bodies:

- AVT Applied Vehicle Technology Panel
- HFM Human Factors and Medicine Panel
- IST Information Systems Technology Panel
- NMSG NATO Modelling and Simulation Group
- SAS Studies, Analysis and Simulation Panel
- SCI Systems Concepts and Integration Panel
- SET Sensors and Electronics Technology Panel

These bodies are made up of national representatives as well as generally recognised 'world class' scientists. They also provide a communication link to military users and other NATO bodies. RTO's scientific and technological work is carried out by Technical Teams, created for specific activities and with a specific duration. Such Technical Teams can organise workshops, symposia, field trials, lecture series and training courses. An important function of these Technical Teams is to ensure the continuity of the expert networks.

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# Aging Aircraft Fleets: Structural and Other Subsystem Aspects

## (RTO EN-015 / AVT-053)

# Executive Summary


Aging aircraft concerns have dramatically escalated in the military community during the past decade. The percentage of aircraft, operated beyond their original design life both in terms of flight hours and/or calendar years is steadily increasing. Some models, which have already been in service for more than 30 years, will need to be retained for another two decades or longer, often serving in roles and in theaters very different from what was envisioned when they were originally designed.

Aging Aircraft has several connotations. Among them: (a) technological obsolescence, (b) the need for system upgrading, (c) changing mission requirements, (d) the specter of runaway maintenance costs, (e) concern about safety, (f) impairment of fleet readiness and (g) possible unavailability of home depot facilities. However, if there is one thread that runs through the above list, it is the adverse impact on sustainment of the fleet.

There are other considerations when dealing with the Aging Aircraft issue; for example, availability of spare parts, processes and tooling may no longer be available, logistic procedures may have changed and suppliers may be out of the business. Budgetary limitations and higher fleet utilization will increase the demand to cope with aging aspects for the structure and major subsystems like engines and avionics. Awareness in the user community about typical challenges and technical solutions can ameliorate some of the concerns. New technologies are now available for dealing with many of the aging aircraft concerns. They relate to inspection, repair and corrosion-resistant materials, structural modeling and more sophisticated maintenance scheduling. Thus a Lecture Series (LS) under the auspices of the NATO Partnership for Peace (PfP), is proposed, the main emphasis of which will be an in-depth discussion of these new technologies and methods. The LS will cover aspects of systems upgrades and structural airworthiness linked to fixed wing and helicopter fleets with emphasis on life enhancement strategies used by NATO nations.

The material in this publication was assembled to support Lecture Series 218 bis under the sponsorship of the Applied Vehicle Technology Panel (AVT) and the Consultant and Exchange Programme of RTO presented 13-16 November 2000 in Sofia, Bulgaria.

# le Vieillissement des flottes d’avions militaires : aspects structures et autres sous-systèmes

## (RTO EN-015 / AVT-053)

# Synthèse


Le problème du vieillissement des aéronefs militaires s’est considérablement amplifié au cours de la dernière décennie. Le pourcentage d’aéronefs en exploitation au-delà de leur durée de vie théorique, tant du point de vue d’heures de vol que d’années de service, augmente régulièrement. Certains modèles, déjà en service depuis plus de 30 ans, devront être maintenus pendant encore deux décennies au moins, souvent pour des missions et des théâtres très différents de ceux qui étaient envisagés à l’origine.

Le terme “aéronefs vieillissants” a plusieurs connotations différentes, parmi lesquelles l’on peut distinguer : (a) l’obsolescence technologique, (b) la nécessité de procéder à la mise à niveau d’un système, (c) l’évolution de la mission, (d) le spectre des coûts de maintenance incontrôlés, (e) des considérations de sécurité, (f) l’atténuation de l’état de préparation de la flotte et (g) la non-disponibilité des dépôts de base. Mais tous ces aspects ont un facteur commun : l’impact négatif sur le maintien de la flotte.

Il y a aussi d’autres considérations à prendre en compte; par exemple la disponibilité de pièces de rechange, de processus et d’outillage, les procédures logistiques qui peuvent avoir changé et les fournisseurs qui peuvent avoir fait faillite. Les limitations budgétaires et l’utilisation accrue des flottes aériennes nécessiteront de porter plus d’attention aux aspects de vieillissement de la structure et des sous-systèmes principaux des aéronefs, tels que les moteurs et l’avionique. Une meilleure sensibilisation des utilisateurs aux défis et aux solutions techniques typiques pourrait pallier certains de ces problèmes. De nouvelles technologies, qui permettront de résoudre bon nombre de ces questions, sont désormais disponibles. Elles concernent l’inspection, la réparation, les matériaux résistants à la corrosion, la modélisation structurale et l’amélioration de la programmation de la maintenance.

Par conséquent, il est proposé d’organiser un Cycle de Conférences (LS) sous l’égide du programme OTAN de Partenariat pour la paix (PfP), dont l’objectif principal sera de permettre une discussion approfondie de ces nouvelles technologies et méthodes. Le Cycle de Conférences couvrira tous les aspects de la modernisation des systèmes et de l’aptitude au vol du point de vue structural des flottes d’avions à voilure fixe et d’hélicoptères, l’accent étant mis sur les stratégies d’extension de la durée de vie adoptées par les pays membres de l’OTAN.

Cette publication a été rédigée pour servir de support de cours pour le Cycle de conférences 218 bis, organisé par la Commission de AVT dans le cadre du programme des consultants et des échanges de la RTO du 13-16 novembre 2000 à Sofia, Bulgarie.

Click inside the blue boxes or on the titles to view the corresponding section

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# LOADS MONITORING and HUMS

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**SUMMARY**
The fatigue life of aircraft's in service is different from the design life for many weapon systems not only due to the extended need for the airframe as a platform for new/upgraded systems (life extension), but also due to different usage compared to the initial design spectrum. Monitoring of the life consumption is therefore essential to assess practicability and cost effectiveness of planned upgrades and modifications. Methods and concepts to establish the "used life" are described for two different types of fixed wing aircraft's and the influence of aircraft missions and -equipment as well as structural weight increase over time are discussed.
New integrated health monitoring systems with intelligent data processing and software capable comparing actual events or accumulated damage / wear with predefined limits, evaluate their criticality and provide information to other systems are presented.

## 0. BACKGROUND
The effectiveness of military force depends in part on the operational readiness of aircraft which itself is largely dependent on the condition of the airframe structure. This condition again is affected by a number of factors among those the physical loads in various forms together with the used life of the airframe are important. With increased and extended usage of airframes in all airforce inventories and the requirement for various role changes the subject of airframe loads-monitoring becomes more important, not only for flight safety but also and with an increasing tendency for economic reasons.

## 1. LOADS MONITORING AND "FATIGUE LIFE" OF AIRFRAMES

### 1.1 Historical Overview
Fatigue management requirements and techniques have evolved over a period of more than 40 years, originating from simple cg-acceleration-counters to multi-channel systems with on-board processing capabilities. Originally a driving factor for load measurements was the generation of databases for design purposes, especially the wing loads and the wing to fuselage interface was of interest for subsonic and aerodynamically stable A/C- configurations. Combining the data with parameters, easy to retrieve like speed, altitude, weight and time this transformed later into the bases for a first set of "fatigue meters", used as a tool to record repeated service loads on the airframe.
During 1960 and 1970 the fact that loads on many parts of the structure could not be related in any way to c.g.- acceleration and the simplified approach of the fatigue meters led to improved methods of fatigue monitoring. The first approaches to monitor on a fleetwide basis evolved and the philosophy of monitoring local fatigue sensitive areas, using mechanical strain recorders, see Fig. 1.1-1.

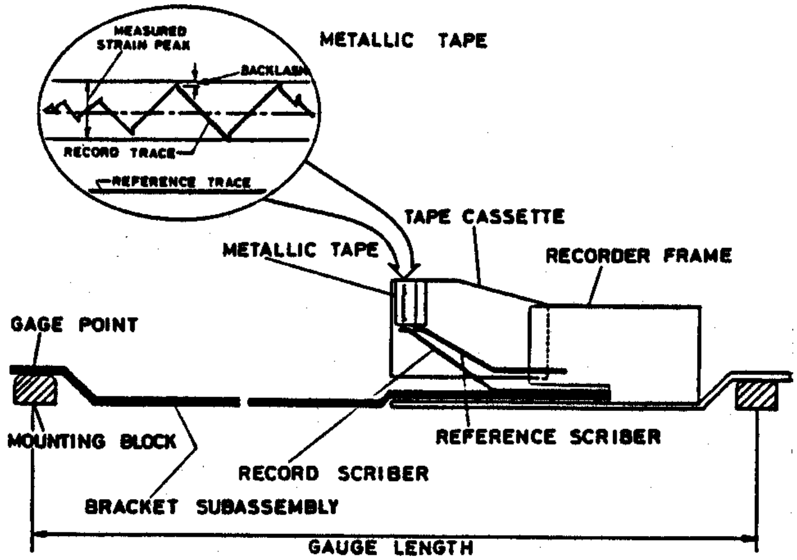


**Fig. 1.1-1 Princip of Mechanical Strain Recorder**

*Paper presented at the RTO AVT Lecture Series on "Aging Aircraft Fleets: Structural and Other Subsystem Aspects", held in Sofia, Bulgaria, 13-16 November 2000, and published in RTO EN-015.*

Later calibrated strain gages on the structure were introduced to record strain histories and calculate fatigue damage, either locally on so-called “hot spots” or for the overall component via load calibration processes.
In 1968 the NATO Military Committee required an AGARD-SMP-Study on "Fatigue Load Monitoring of Tactical Aircraft" which subsequently presented agreed conclusions and recommendations for efforts to:

* Establish statistical relationships between movement parameters and structural loads
* Develop simple strain recording techniques
* Establish fatigue life monitoring techniques for all NATO countries

Within the last two decades a number of concepts for aircraft loads monitoring with either fleetwide data recording, supplemented by additional data from limited number of aircraft representative for squadron usage or individual aircraft tracking methods have been developed (1).

### 1.2 Loads Monitoring and Damage Rate Assessment

Monitoring of the airframe loading scenarios and technologies to assess the "Used Life" or "Damage Rate" of airframe structures are key elements to the management of an ageing aircraft fleet. The term Ageing Aircraft can be defined in many different ways, among them are flight hours (or equivalent flight hours) approaching the designed service life; number of flights reaching the projected number of ground-air-ground cycles; or even pure age in the form of calendar years.
From a structures point of view the governing factor for ageing airframes is the degradation of strength and rigidity of structural components with time and usage, applied to the aircraft as damage of different nature, the most obvious ones being fatigue cracks and corrosion.This degradation will continue, increase and finally form a threat to safety of flight without appropriate actions in the form of prevention, detection and repair through scheduled maintenance efforts.
Therefore terms like "Damage Rate", “Fatigue Life Expended” or “Fatigue Index” have been identified as an indicator for the structural status of an aircraft, where a rate of 100% or 1.0 identifies the end of the designed fatigue life of a component or the limit for economic repair and usage of the aircraft.

#### 1.2.1 The Object of Fatigue Monitoring Programs

In service individual aircraft’s are subject to different operational loading causing different damage rates in their fatigue prone areas. Dependent on how an aircraft is used, it may have an expended life significantly different from what is predicted at the time of service entry.
The simple fact is that aircraft are often not used the way they were intended to be used during design and aircraft are used differently even when flown for similar missions.
Fig. 1.2.1-1 shows an example for consumed fatigue life of TORNADO lower wing skins for aircraft with comparable missions, Fig. 1.2.1-2 the wing root bending life-consumption for Canadian CF-18`s from one squadron. Factors of up to 5 for the damage rate have been identified between the most and least severe flown aircraft. If no fatigue monitoring program for individual aircraft is carried out, maintenance actions, modifications and finally retirement of the equipment is based on the number of flight hours which the most severe flown aircraft is allowed to accumulate.

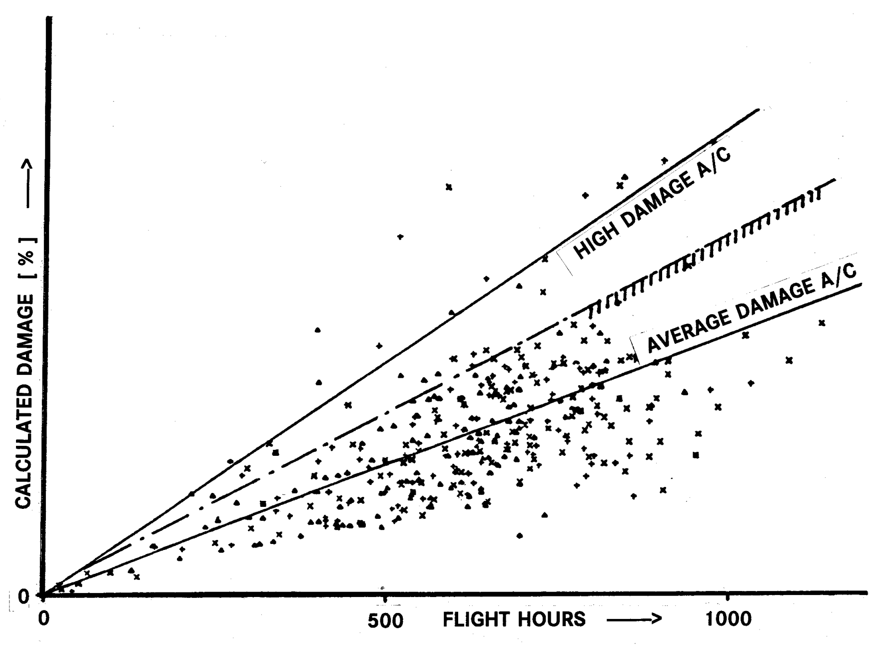


**Fig. 1.2.1-1 Lower Wing Skin Life Consumption for Similar Missions, TOR**

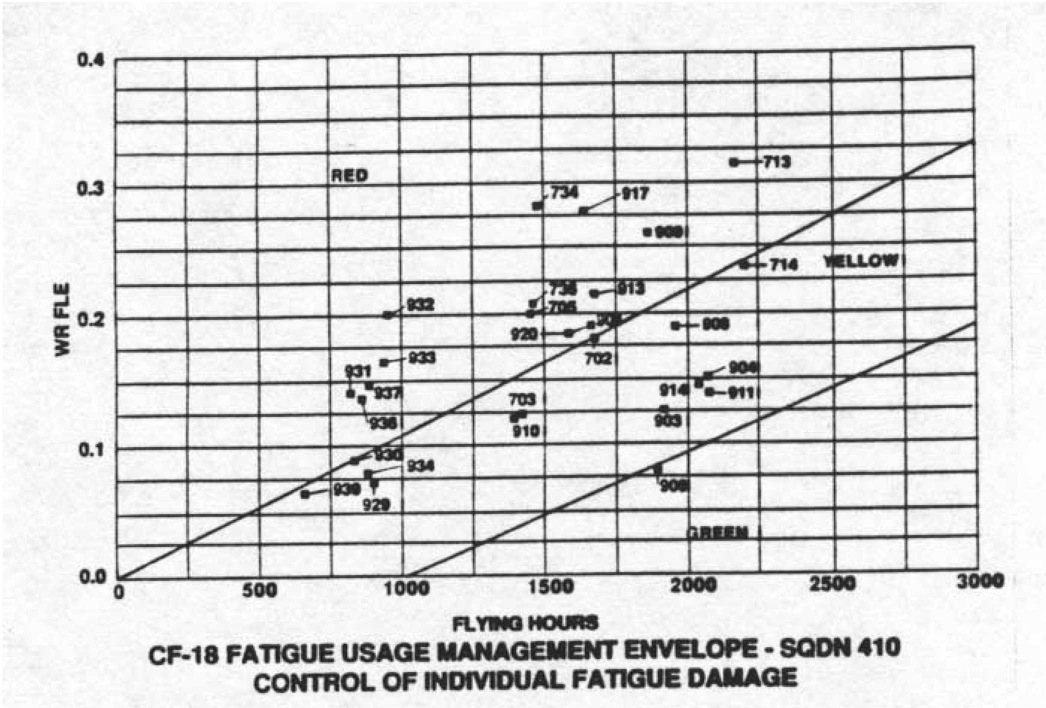


**Fig. 1.2.1-2 Wing Root Bending Life Consumption, CF-18**

Hence, a sound and comprehensive operational loads data acquisition and evaluation will be an effective tool for cost savings during the operational life of an aircraft.
With consideration of the life already consumed and with predictions about further usage the remaining service life of components can be determined and actions to adopt fatigue enhancement policies can be initiated at least for loads initiated damage, i.e. aircraft's with high damage rates can be allocated to fly less severe missions/configurations or structural modifications can be introduced before fatigue damage occurs.

Any monitoring and fatigue assessment program is therefore set up to answer the question:

*"What is the fatigue life ratio of the operational stress spectrum rated against the design/test spectrum on the different airframe locations?"*
or:

*"How many operational flight hours are equivalent to a simulated flight hour during fatigue testing ?"*

### 1.2.2 Structural Monitoring Concepts and Systems

The main activities during a structural monitoring concept to determine the consumed life of each individual airframe are shown in Fig. 1.2.2-1.

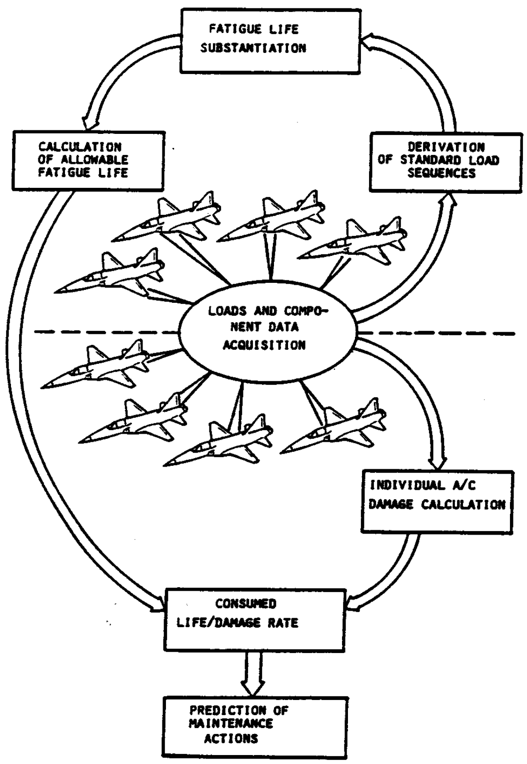


**Fig. 1.2.2-1 Structural Monitoring Activities**

The initial step of *Loads and Component Data Aquisition* is performed using flight data recorders for overall aircraft load parameters and local sensors for fatigue critical areas together with aircraft identification information ("Tail-No.-Tracking") or component information for exchangeable items (i.e. horizontal stabilators).
Special post-processing is needed to separate, correct or replace faulty data.
The *Damage Calculation* is performed with respect to the design philosophy of the aircraft:

* For Safe Life - structures the calculation is based on S/N-curves and Miners rule to determine the accumulated damage.
* For Damage Tolerant designed structures initial flaws are assumed and crack growth analysis is performed for each fatigue critical part of the structure, ensuring that the initial flaw of a given size (i.e. 0.005 in or 0.127mm) will not grow to a functional impairment size within a given lifetime.Inspections, replacements or repair actions are scheduled by durability analysis using the flight loads data in the form of cycle by cycle stress histories coupled by probability of detection (POD) data.

From the registered loads data, a Derivation of Standard Load Sequences or Spectra (SLS) is extracted to create specific parameter or load histories. They should fulfil the following criteria:

* The mean damage of the registered load sequence of individual A/C should be equal to the mean damage of the SLS
* The distribution of actual missions, configurations and other relevant operational parameters should be characteristic for the A/C operational usage. In some cases different SLS or spectra have to be generated for one A/C, i.e. Training-, Air-to-Air or Air-to-Ground dominated usage.

The *Fatigue Life Substantiation* is demonstrated through fatigue analysis and a qualification process including component and full scale fatigue tests in the development phase, validation of loads within flight envelope tests as well as operational experience during A/C-usage.

Since the tests are usually carried out within or in direct sequence with the design phase and based on the loads and structural configuration status of this time, deviations during the operational usage phase are normally scaled to the fatigue test, determining the so-called "Usage Factor".
Assessment of the allowable fatigue life depends on the results of the fatigue life substantiation (in most cases the full scale test) and the design philosophy. Demonstrated fatigue test hours divided by the scatter factor and linked to the standard load-spectrum are the limit for a safe life designed structure, whereas for damage tolerant structures the test hours leading to cracks that impairs function of the structure divided by a factor are considered for the *Calculation of Fatigue Life.*
The *Consumed Life or Damage Rate* for each component is the relation of the actual individual A/C damage calculation and the allowable life and is used to schedule inspections, replacements or repair actions in order to ensure structural integrity.

### 1.2.3 Aircraft Fatigue Tracking Systems for the GAF-TORNADO

The TORNADO Multi Role Combat Aircraft was designed in the early ´70 and followed the safe life design principal for durability with a scatter factor of 4, used on the design life of 4000 FH. The fatigue tracking concept of the A/C is divided into three sectors with different numbers of aircraft's from the fleet involved and different amount of data (parametric and strain gages) gathered, as shown in Fig. 1.2.3-1.

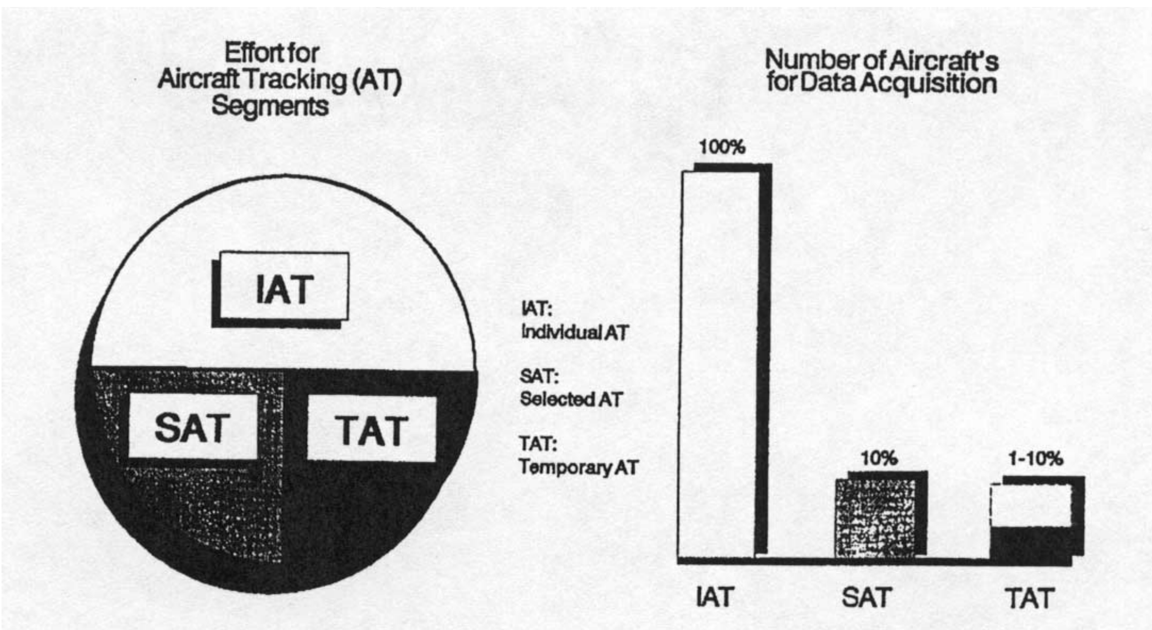


**Fig. 1.2.3-1 Aircraft Tracking Segments, TOR**

Monitoring is based essentially on flight parameters, which are available through the existing flight recorder unit and defined as Recorder Parameter Set (RPS).
An extended Full Parameter Set (FPS) is generated through differentiation's and conversions of existing data. The flight recorders are distributed on a statistically representative basis throughout the squadrons and register the spectrum of selected aircraft. Additionally, strain gages in various fatigue critical areas of the structure are monitored on a limited number of aircraft, the results are evaluated by regression techniques to produce a realistic correlation between operational strain on the structure and the flight parameters causing it.

A reduced Pilot Parameter Set (PPS) is collected from each individual aircraft through the Nz-counter plus aircraft weight and configuration data, see Fig. 1.2.3-2 on a flight by flight bases.

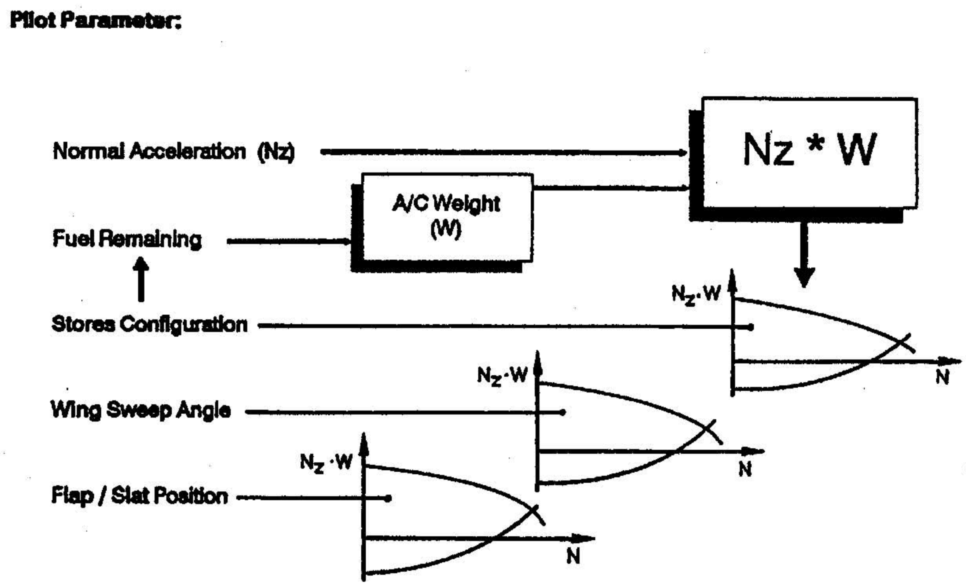


**Fig. 1.2.3-2 Reduced Parameter Set (PPS) for IAT**

Thus, a "multi-level" tracking is performed:

* Individual Aircraft Tracking with Pilot Parameter Set
* Temporary Aircraft Tracking with Recorder Parameter Set + Strain gages
* Selected Aircraft Tracking with Full Parameter Set

Fig. 1.2.3-3 lists the recorder parameter set and strain gage sampling rates for the Temporary Aircraft Tracking level.

| No. | Parameter | Sampling Rate / s | No. | Parameter | Sampling Rate / s |
|---|---|---|---|---|---|
| 1 | Pressure Altitude | 0.5 | 11 | Inboard Spoiler STBD | 1.0 |
| 2 | Calibrated Airspeed | 0.5 | 12 | Rudder Position | 2.0 |
| 3 | Normal Acceleration | 16.0 | 13 | Wing Sweep Angle | 0.5 |
| 4 | True Angle Of Attack | 2.0 | 14 | Primary Strain Gauge | 16.0 |
| 5 | Roll Rate | 8.0 | 15 | Secondary Strain Gauge | 4.0 |
| 6 | Pitch Rate | 4.0 | 16 | Flap Position | 1.0 |
| 7 | Yaw Rate | 2.0 | 17 | Slat Position | 1.0 |
| 8 | Taileron Pos. PT | 4.0 | 18 | Fuel Remaining | 1.0 |
| 9 | Taileron Pos. STBD | 4.0 | 19 | Stores Configuration | 4.0 |
| 10 | Outboard Spoiler PT | 1.0 | 20 | Oleo Switch | 0.5 |
| | | | 21 | Identification Data | 1.0 |

**Fig. 1.2.3-3 Recorder Parameter Set Data and Sampling Rates**

From a conception point of view, the individual aircraft tracking permits optimum utilisation of the structural life of a fleet. This naturally requires appropriate sensors existing in the individual aircraft for the acquisition of local stress history data.
Since not all of the TORNADO aircraft's are equipped with strain gages, PPS acquired by IAT are converted via the regression table from TAT-A/C into stress spectra for the fatigue critical areas.
Monitoring of the TORNADO´s fatigue critical areas uses the local strain concept, too. For this, a suitable local strain measurement location was established for every area during the Full Scale Fatigue Tests. Fig. 1.2.3-4 shows an example for a critical area in the engine duct , where "reference" strain gages are located at the wingbox shearlink to the fuselage for on-board monitoring.

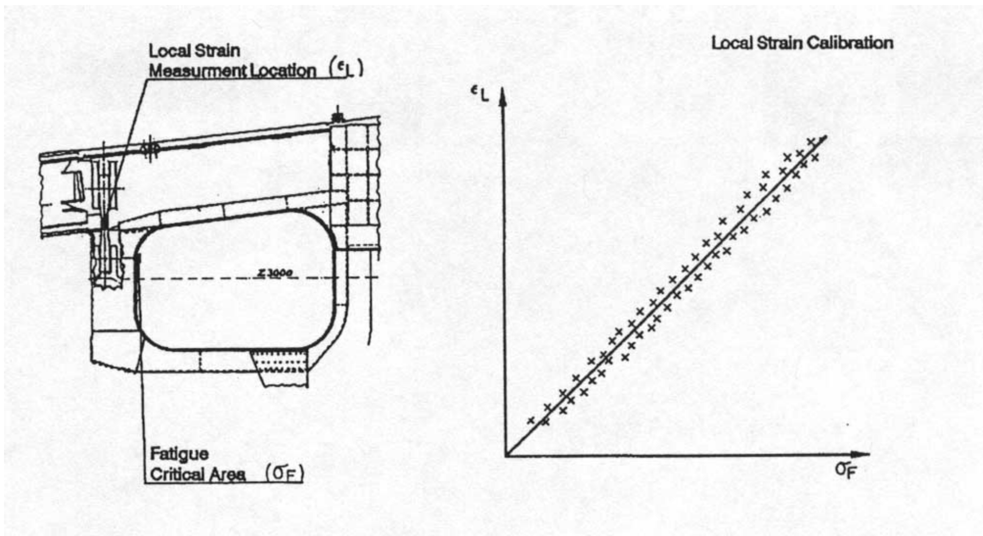


**Fig. 1.2.3.-4 Reference Strain Gage on Wing Attachment**

The damage in the duct location is traced to the wing bending moment. By applying the transfer functions for inner wing shear force and bending moment to the recorder parameter set and the correlation equation for the reference gage from fatigue test, the stress history for this area is generated.

### 1.2.4 On-Board Loads Monitoring System of Canadian Forces CF-18 Aircraft (2)

Usage characterisation of the CF-18 fleet is also a key element of fatigue life management of the CAF F-18 fleet. In contrary to the TORNADO, all of the CF-18 aircraft are equipped with strain gage sensors at different locations during production, see Fig. 1.2.4.-1.

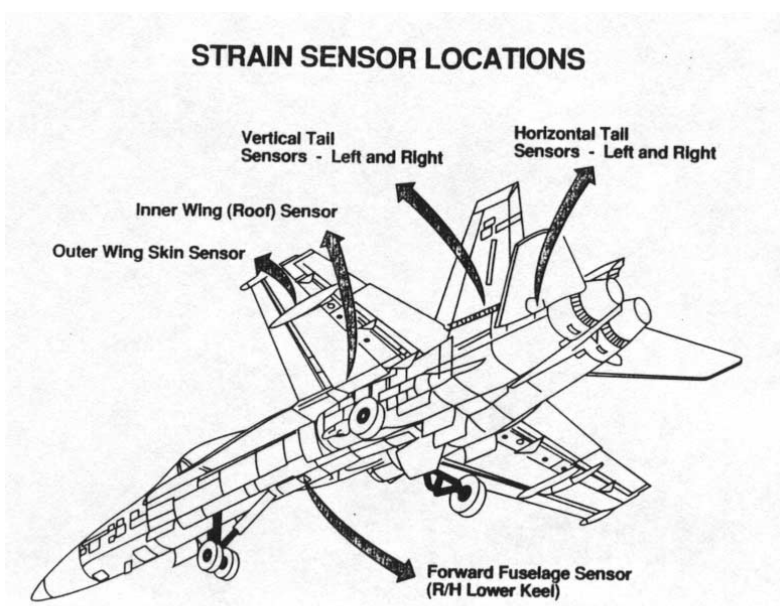


**Fig. 1.2.4-1 CF-18 Strain Gage Locations**

Flight parameters are recorded together with the strain gage signals on a flight by flight bases within the Maintenance Signal Data Recorder (MSDRS) and allow individual aircraft tracking throughout the service life of every aircraft.
Location of the strain gages were selected by the manufacturer based on criticality of the structure, its accessibility and the degree of protection from accidental damage. Prime and spare gages are applied for redundancy. Use of the direct strain measurements inherently accounts for parameters like airspeed, altitude, weight, store configuration and cg-variations during flight. However, the accuracy of the fatigue calculation is dependent upon the reliability and proper installation of the sensor.
Data are stored on magnetic tape and downloaded to a ground station. Different level of data reduction and reporting can be generated from limited fatigue analysis codes at operating bases to assess severity of individual flights or mission profiles to annual reports for long-term trend analysis.
Since the F-18 was also designed to a safe life philosophy, fatigue consumption is calculated in terms of Fatigue Life Expended (FLE) against the 6000 FH life of the design usage spectrum. This linear relationship was established using the information collected during full scale fatigue test conducted by the manufacturer and is scaled for CF in-service usage and structural configuration changes between test article and fleet.
For the purpose of fatigue calculations, crack initiation was defined as formation of a crack of 0.25 mm or 0.01inches. Cracks usually originate at locations of tensile stress concentrations, where material strength is exceeded when high load magnitudes are frequently encountered in-service.
From the in-flight MSDRS recorded strain peaks and valleys, a representative loading spectrum is generated, and by using the individual material stress-strain relationship of the components, the corresponding stress spectrum is obtained.

From this spectrum the amount of damage per cycle and afterwards the crack initiation life can be calculated by using material dependent S/N-curves, Fig. 1.2.4-2.

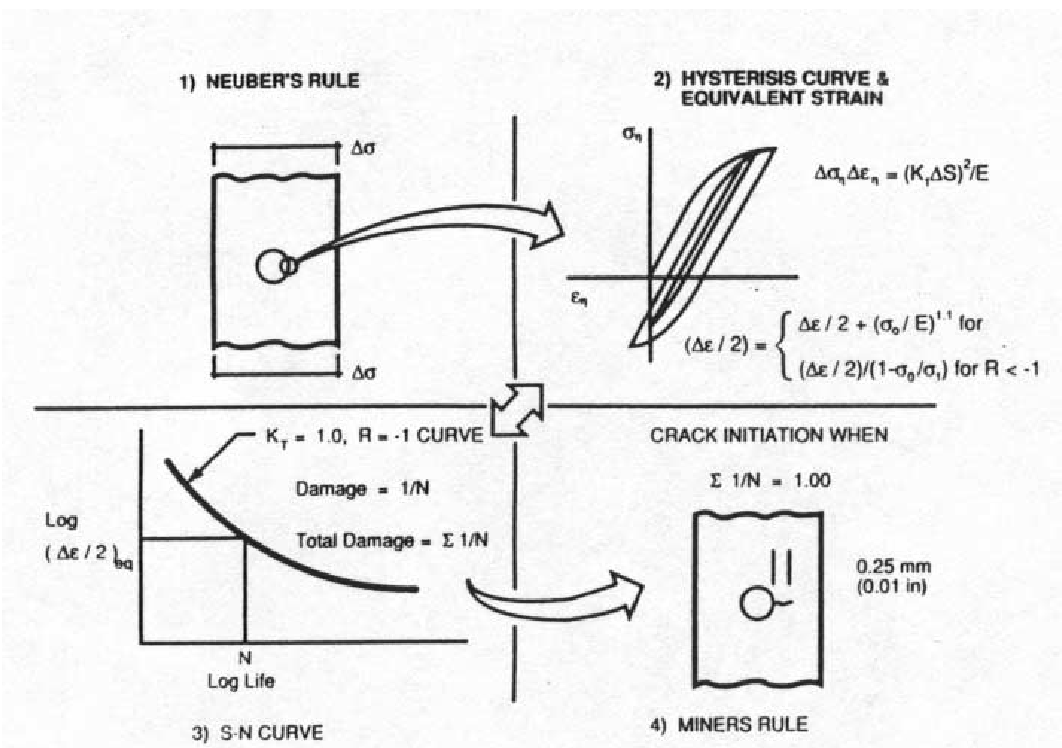


**Fig. 1.2.4-2 Crack Initiation Concept**

The FLE is then expressed as the total damage accumulation to date divided by the total structural fatigue damage required to initiate a 0.25 mm crack under the design loading spectrum.
After initiation, remaining life of the component is used by crack growth up to the critical crack length. Currently, the fatigue analysis program does not contain a crack growth prediction model.

Together with fatigue awareness and control programs, reducing configuration severity for missions, within 2 years of implementation, the CF was able to improve fleet attrition trends already by approx. one year of service, Fig. 1.2.4-3

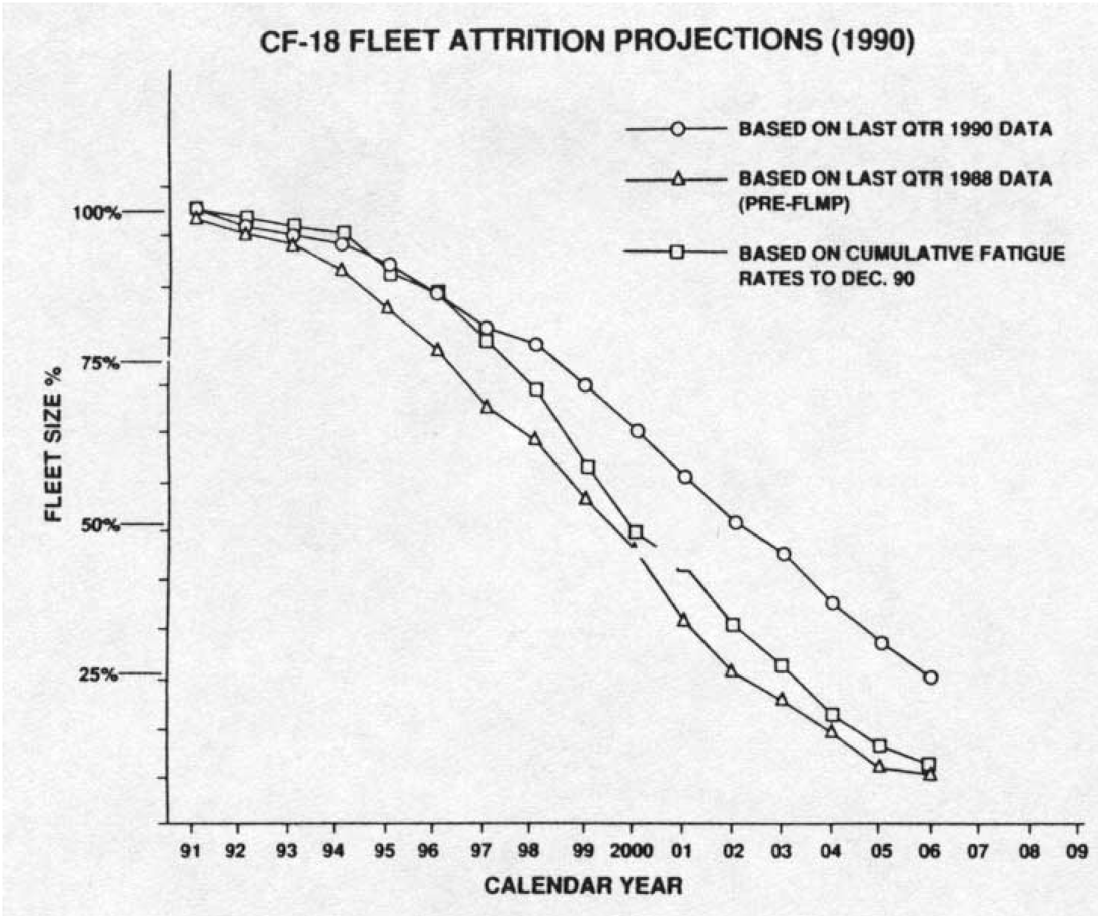


**Fig. 1.2.4-3 Life Improvement of CF-18 Fleet**

Some of the experiences with the system of individual aircraft tracking through strain gage sensors are:

* Fatigue damage calculations are improved by direct strain measurements due to elimination of A/C flight parameters from the equations
* Accuracy of the measurements are vital and gage drift over time is a concern
* In flight-calibration of gages through reference manoeuvres during maintenance test flights can be a solution to gage drift
* Reliability of the strain sensor is vital, since drop-outs must be replaced with conservative "fill-in"-algorithm, leading to artificially higher FLE data.
* Timely reporting schedules are essential for feedback of damage accumulation and on the effects of role changes/aircraft usage to the operational squadron as well as to the fleet manager.

## 2. INFLUENCE OF THE STRUCTURAL CONFIGURATION STATUS

An aircraft in service or produced over an extended period of time will change its structural and system configuration in many areas due to structural modifications, additional systems installed, improved engine performances etc.

While major structural modifications are usually covered by either extensive analysis, accompanied by component testing and sometimes even full scale tests, the smaller modifications and "updated" system installations are well documented in production configuration control files, but mostly "neglected" for internal loads influence for some time.

Fig. 2.0-1 shows the increase of the TORNADO structural mass aft of the rear transport joint, including vertical and horizontal tail components for the different batches within a production period of 14 years together with the design weight used in the unified analysis in 1976.

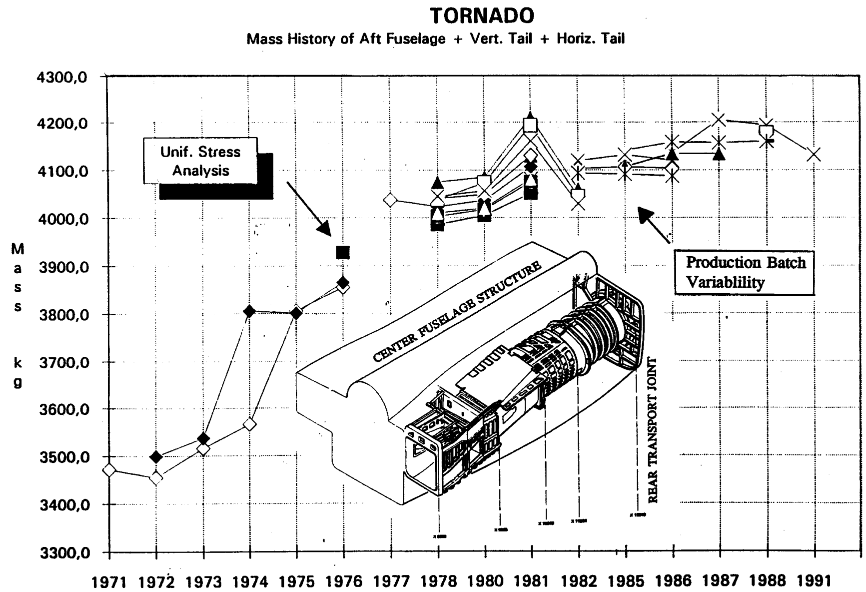


**Fig. 2.0-1 Historic Structural Mass Increase of TOR Aft Fuselage**

The "immediate solve" for weight increase of reducing internal fuel and keeping the Nz-level (Nz x m = constant) will obviously not work for this problem, based on the fuel sequence the wet wing mass definition is no longer valid and leads to higher wing loads. The same effect is also valid for the front fuselage, as explained in the previous paper “AIRCRAFT LOADS” .
At the same time engine thrust has been raised also by 16%, although only a fraction of it is used during peacetime operations, the heavier engine contributes to the mass increase. More important, in contradiction to a special role equipment, which may be cleared with restrictions like "Not for peacetime training missions", this mass increase influences the fatigue life consumption permanently during every flight hour and every manoeuvre.

The influence of the higher loads can be clearly seen on the structural transport joint loading leading to vertical shear load increase of approx. 20 kN or 4500 klbs and vertical bending of 30 KNm or 265000 inlbs respectively an additional 6.5 % based on the design limit loads, Fig. 2.0-2.

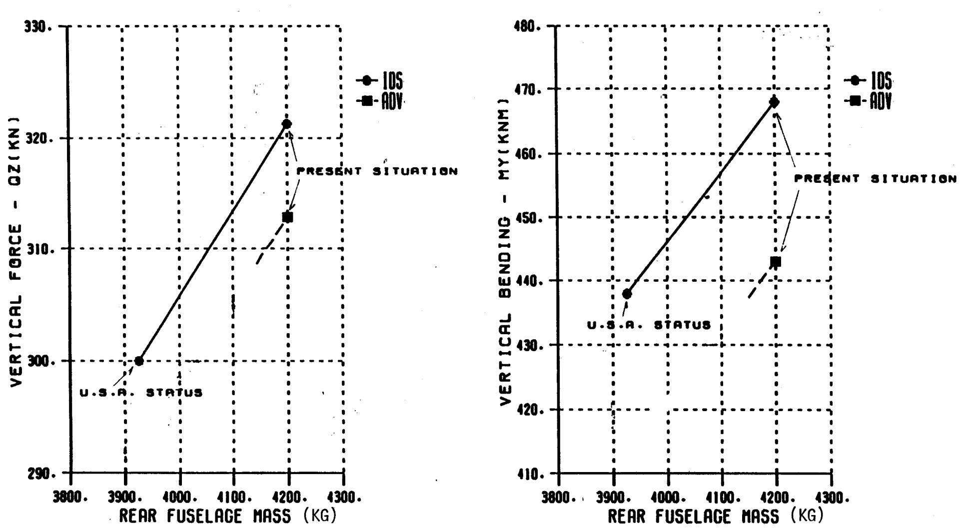


**Fig. 2.0-2 Interface Load Increase at Rear Transport Joint**

A regular check of the present inertia loads status after modifications and system upgrades is therefore mandatory to make loads monitoring concepts, based on parametric data, work.

## 3. HEALTH AND USAGE MONITORING SYSTEMS (HUMS) FOR AGING AIRCRAFTS

The major research in the area of smart vehicle technologies including integrated health and usage monitoring systems for inherent or onboard diagnostic of the structural status is directed towards future aircraft to improve performance, reliability and survivability or reduce pilot loads. Some of this technology will also be applicable to existing fleets of fixed and rotary wing aircraft's and help to improve flight safety and reduce maintenance cost.

While onboard computing devices already offer means to process strain gage readings and flight parameter data during flight or at the end of every mission, the subsequent analysis of this ever increasing data base require careful consideration for fleet management and maintenance planning. The need for automation of the data reduction including diagnostic software to support the decision making process is vital for the future.

At the same time care needs to be taken in defining analysis and handling techniques for the enormous amount of data that is generated and becomes the basis for decisions, affecting flight safety and maintenance procedures, thus becoming a certification item itself.

### 3.1 The HUMS Procedure

The key elements of any HUMS are the real time diagnostic of the structural status of the aircraft using a sensor, linked to a processor and display unit and an intelligent software to compare actual events or accumulation of damage / wear with predefined limits, evaluate the criticality and provide information to other systems like pilot alert or maintenance recording units for later retrieval.

Sensors used must have the capability to detect the type, extent and location of the damage within the component without being disturbed by the in flight environment (noise, vibration, temperatures etc.) and should have the robustness to endure the airframes life, not creating an additional / critical maintenance issue.

Processors obtain, verify and process the sensor data through software routines and perform the health assessment for the component. The output is either stored for subsequent usage within a maintenance data recorder unit or displayed onboard during flight for event alert.

Software includes data collection, analysis algorithm and expert systems to initiate the "decision making process". In some cases Neural Network technology has been promoted to link loads and fatigue data to flight parameters, especially for rotorcraft where direct measurement of local data through strain gages are difficult or inappropriate (i.e. on rotating elements for vibration loads). However, these Neural Networks require training and validation (especially when HUMS is used within the certification process) which again can only be measured using direct techniques.

Fig. 3.1-1 gives a schematic overview of a HUMS architecture for structural applications.

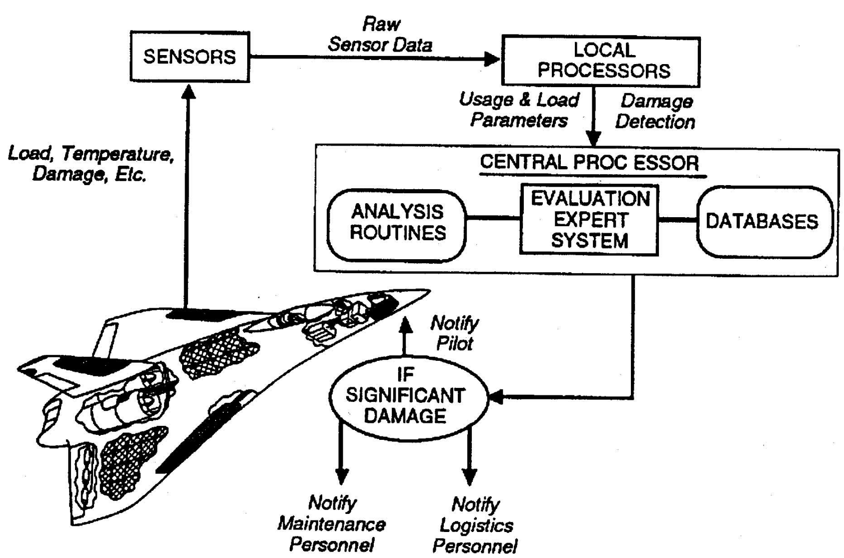


**Fig. 3.1-1 Schematic overview of HUMS architecture**

### 3.2 Sensors

The following table gives an overview of sensors commonly evaluated in HUMS programs:

| **Sensor Type** | **Structural Application** |
|---|---|
| Acoustic Emission | Damage Detection, Cracks, Delaminations, Impacts |
| Acousto Ultrasonics | Damage Detection, Cracks, Delaminations, Impacts |
| Modal Analysis | Vibration modes, Damage Detection |
| Strain Gage | Strain Measurement |
| Fibre Optic | Strain Temperature Pressure |
| Crack Gage | Crack Growth |
| Accelerometer | C.G. or Local Acceleration, Vibration, Buffet |
| Pressure Transducer | Pressure |
| Displacement Transducer | Structural Deformation |
| Electro Chemical | Corrosion, Environment |
| Thermocouple | Temperature |

While strain gages, accelerometers and thermocouples are well known sensors used in existing fatigue monitoring programs, fibre optics and acoustic emission sensors have found recent application in research programs for health monitoring of structures. While isolated sensor function and data collection on coupon level is well understood, the sensor array, the distribution architecture and the method to collect and analyse data on complex structures is still being developed.

Since for metallic structures the dominant mechanical damage are fatigue cracks, the sensor must be able to identify damage as small as 2.5 mm in areas like sharp radii, around fasteners or in build-up structure without knowing the precise location up-front.

### 3.3 Structural Application of HUMS

Application of HUMS to detect and monitor fatigue damage in metallic structure has been successfully demonstrated during ground testing on coupons, complex sub-elements and full scale structures. Fig 3.3-1 shows the application of acoustic emission sensors located in the web of a typ. machined bulkhead in an array around the critical location of the hole. During the monitoring phase, the major tasks of the systems is to identify and “filter” structural noise from damage events, identify crack initiation and monitor crack growth.

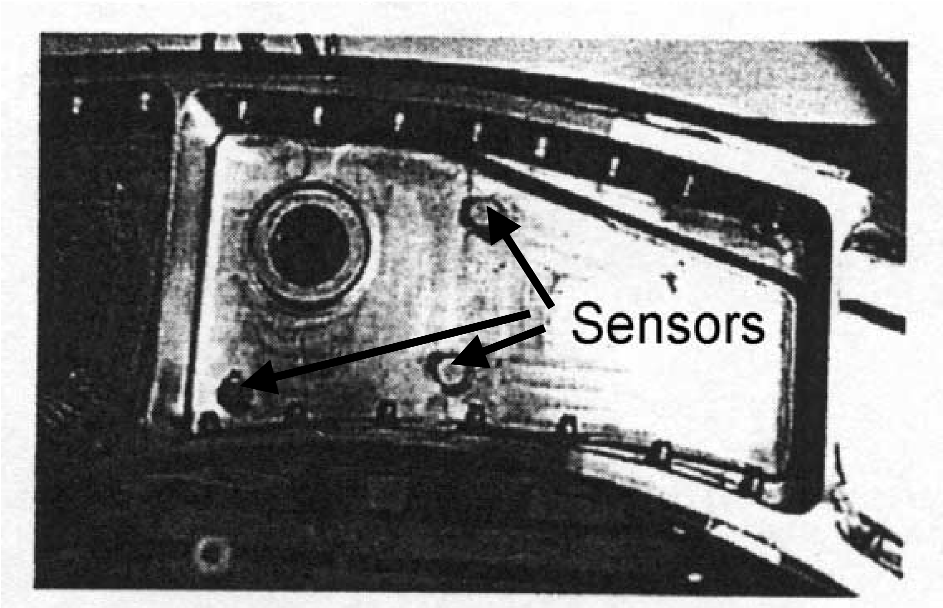


**Fig. 3.3-1 Acoustic emission sensors in web area of frame**

While in simple structures the distance from crack location to sensor to detect events can be as far as 450 mm, a more complex structure with joints or geometric discontinuities requires the sensors much closer to the expected failure location to obtain reliable results.

Fig. 3.3-2 shows monitoring locations on a full scale testarticle, were “hot spots” were monitored during a 9000 spectrum flight hour fatigue test. Failure occurred in Zone No. 4 just prior to the 9000 h inspection and the system was able to discriminate signals due to crack growth from background noise, starting at app. 7000 spectrum flight hours.

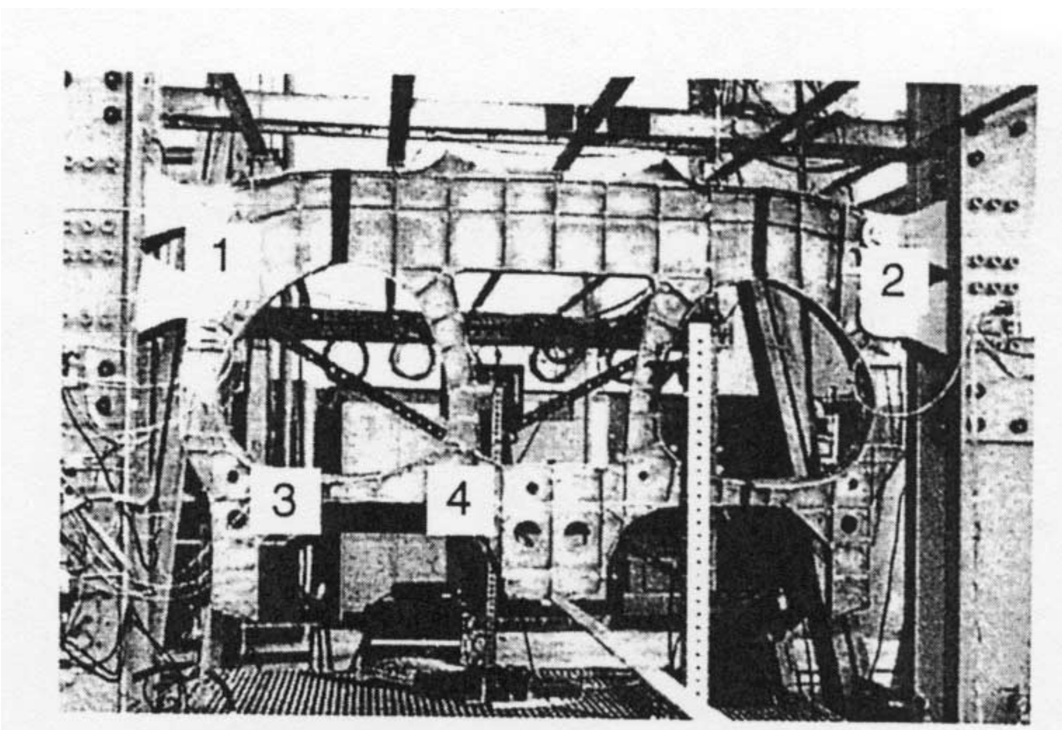


**Fig. 3.3.-2 Full scale test article with monitoring loactions**

Fig. 3.3-3 shows sensor location in Zone No.4, the signal versus time and frequency band for both, background noise and the crack growth event.

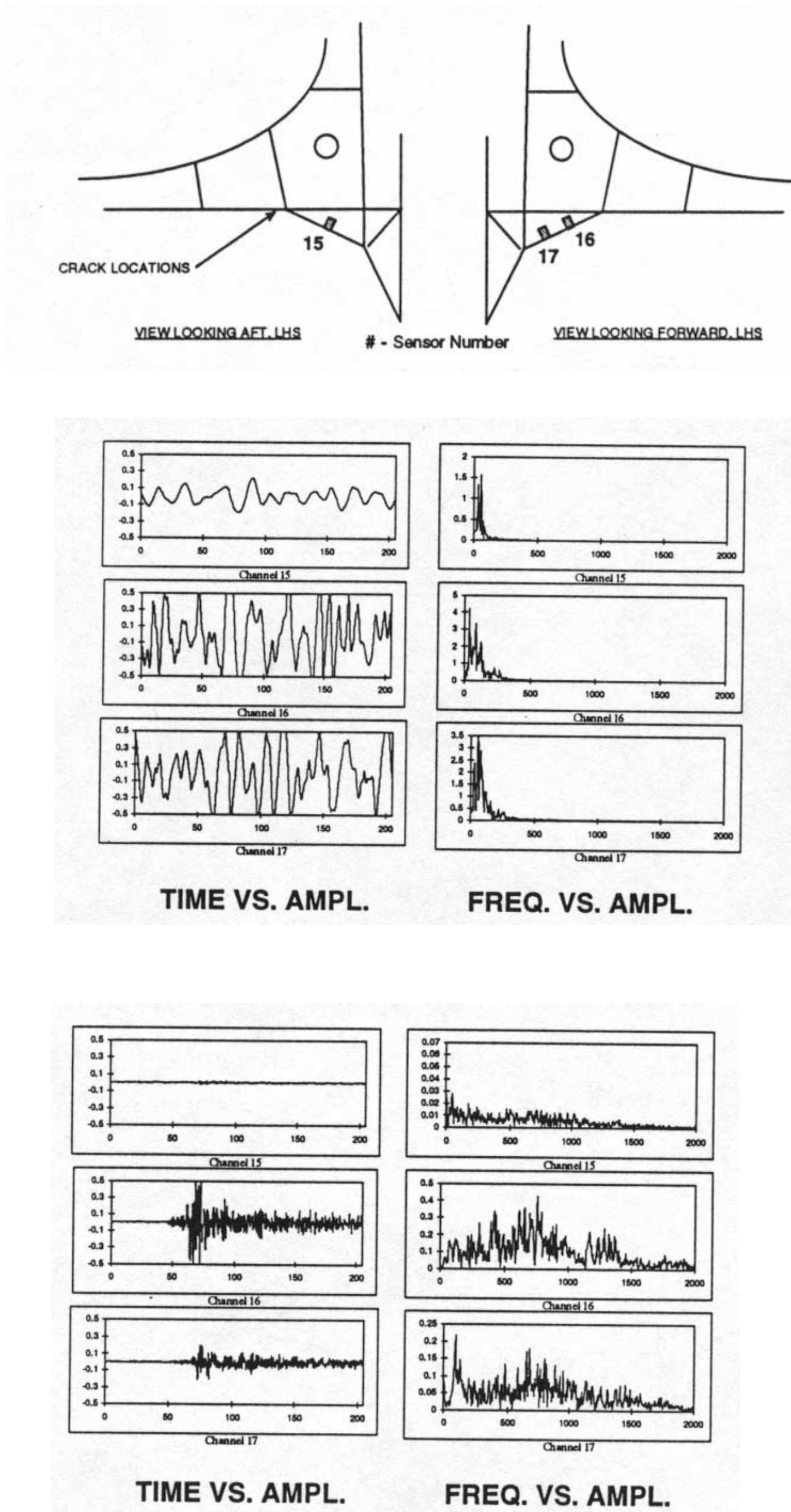


**Fig. 3.3-3 Zone 4 sensor location and results**

A different method of monitoring structural health is shown in Fig. 3.3-4, a fibre optic array embedded in the composite structure during manufacturing of the part. This technology has been mainly applied to advanced composites on research and test bench level. Issues like the effect of the fibre on the basis material, robustness and long term stability of the fibre and the sensor interface, repairability, sensitivity of the sensor and degradation with damage occurring are a few areas for continuos research.

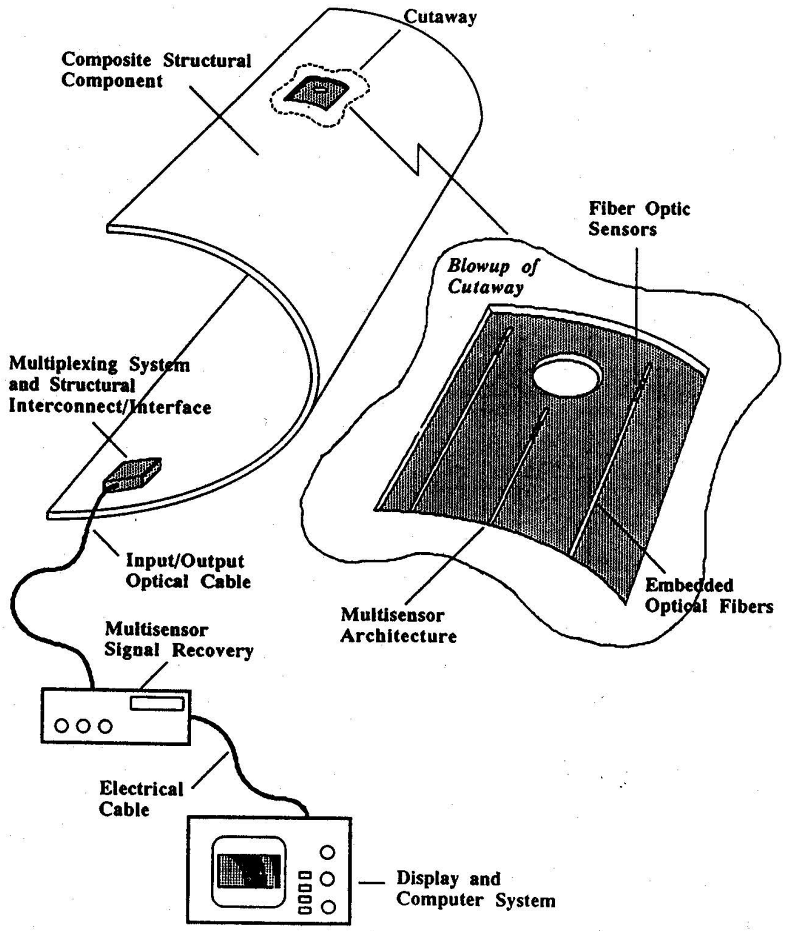


**Fig. 3.3-4 Fibre Optic monitoring array embedded in structure**

The two major tasks of structural health monitoring:

- Identification of events / damages
- Continuous monitoring of loads within the structure could be achieved within one system and using one sensor only, if the system is designed accordingly.

The fibre would have adequate sensitivity to measure strain levels and detect anomalies that might indicate the development of structural weakness through fatigue and/ or local damage, while impact damage above a predefined level would lead to a radical signal response change and in-flight or post mission actions would be triggered.

While today’s existing and ageing fleets of fixed wing aircraft and helicopters still rely on direct monitoring methods and these technologies need to be refined for future applications, the fully integrated HUMS on individual component level will lead to higher exploitation of structural life for existing structures, an option for on condition maintenance if cost effective and the reduction of some conservatism in the design process of new weapon systems.

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# Aging Systems and Sustainment Technology

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**Summary**

Fatigue is a failure mode in aircraft that emerged in the fifties and sixties as a significant threat to their structural integrity. Since that time, the research community has extensively studied the phenomenon and has developed the technology to describe the propagation of fatigue cracks in a structure. This paper describes an approach that, when followed, will virtually eliminate catastrophic failures from this mechanism.

**Introduction**

Most engineering structures, particularly metallic components, when subjected to cyclic loading have the potential for failing below their pristine strength. Such a failure is referred to as a fatigue failure. There is a progressive degradation of the strength from cracks emanating from manufacturing damage, in-service induced damage, or intrinsic defects in the material. Constant amplitude testing is used to characterize the residual strength of a structural member after it has been subjected to a specified number of loading cycles. This paper examines the approaches that researchers have used to prevent catastrophic structural failure resulting from cyclic loading.

Except for those failures resulting from exceeding the operational envelope of the aircraft, structural failures prior to the mid-forties were rare. One reason for this is that before the mid-forties, aircraft rarely accumulated sufficient flight time on their aircraft to suffer from fatigue failures. Further, the ductile materials and conservative methods used for analysis tended to preclude failures. Experience has shown that early aircraft manufactured with ductile materials and designed based on static strength only are typically safe from failure caused by fatigue for at least 1000 flight hours. It was rare for a combat aircraft in World War II, for example, to remain operational for more than 1000 hours. The demand for improvements in performance in the late forties; however, introduced new materials with high strength, but few other virtues. Further, the demand for performance improvements reduced analytical conservatism and introduced designs that were to operate at high altitudes. The design community appeared oblivious to the consequences of their actions. Even before the time of the first flight by the Wright Brothers, fatigue was a major issue in many industries. In the railway industry alone, fatigue failures of wheels caused numerous deaths. These failures seemed to make no impression on aircraft designers. The success they had stemming from the days of the Wright Brothers appeared to continue without interruption although fatigue failures in aircraft can be traced back to the late twenties.

The reality of the consequences of aging came sharply into view for the United States Air Force (USAF) on March 13, 1958 [1] when they lost two B-47 aircraft because of fatigue cracking in the wing. It was on this day that the aging aircraft research effort started for the USAF. The USAF did not specify a service life for the B-47. Consequently, they based the design on the assumption that failure from overload was the only threat to its structural integrity. This was common practice for aircraft designed in the late forties such as the B-47. Review of the then current literature on structural design provided no

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hint fatigue was a serious consideration. However, the USAF intended to maintain the aircraft in service until 1965. The technical basis for maintaining the aircraft in service for that length of time did not exist.

The 1958 failures motivated the USAF to establish the USAF Aircraft Structural Integrity Program (ASIP). The USAF designed this program for use with new weapon systems acquisition for their inventory. The program as originally conceived defined a sequence of tasks that progressively reduced the risk. These tasks, composed of analyses and tests, included all efforts for the qualification of USAF aircraft. This concept is just as valid today as it was in 1958. The approach, although sound in its concept, had a fatal defect. The original program incorporated a reliability concept called safe life to qualify the structure for the loads environment expected in operational service. The USAF determined the safe operational life from the results of a full-scale fatigue test of the structure. They conducted this test in a laboratory environment. They divided the number of successfully tested flight hours by a factor called the “scatter factor.” The scatter factor (usually in the interval from two to four) supposedly accounted for material property and fabrication variations in the population of aircraft. The trouble with this approach was that it did not preclude the use of low ductility materials operating at high stress. Unfortunately, it was at this time that aluminum companies were introducing high strength alloys in response to the insatiable desire for improved aerodynamic performance. Consequently, the “safe life” concept did not eliminate the in-service failures the USAF designed it to prevent. The “safe life” approach adopted by the USAF in 1958 proved to be ineffective in eliminating fatigue cracking as evidenced by the failures in operational aircraft.

Probably the most significant in-service event since 1958 that changed the original version of the ASIP was the failure of an F-111 in December1969. F-111A number 94 (SN 67-049) failed on 22 December 1969 as a result of a wing failure in the lower plate of the left wing pivot fitting. At the time of failure, the aircraft had approximately 100 hours of flight time. Catastrophic loss of this F-111 demonstrated the fatal defect in the "safe life" method. That is, the safe life method did not preclude designs that were intolerant to manufacturing and service-induced defects. Other losses (e.g., F-5, B-52, and T-38) and incidents of serious cracking (e.g., KC-135) during this period confirmed this shortcoming. These failures lead to a new approach for the protection of USAF aircraft safety, a damage tolerance approach. The approach selected by the USAF was damage tolerance. The concept of damage tolerance is discussed in detail in Section 2. The basis for the process is to assume the structure has a flaw, a sharp crack, that is the least upper bound of the expected flaw distribution. The operator makes inspections such that the crack does not reach the point of rapid propagation before it is detectable. The damage tolerance approach is in a state of continual improvement because research and development has lead to better methods in fracture mechanics methods and stress analysis over the last thirty years. The introduction of damage tolerance principles by the USAF in their structural inspection program in the early seventies virtually eliminated fatigue as a safety issue in their aircraft.

The USAF incorporated the damage tolerance approach in the ASIP, and in 1975, they published the process. This program, for a new acquisition, provides a series of time related tasks that will provide progressive risk reduction in the progression of the engineering and manufacturing development phase of procurement. The current version of the ASIP includes five separate tasks that cover all aspects of the development and support of an aircraft structure. For any given program, if the USAF does not plan to include a specific element, then they must establish the rationale and potential impact on the structural integrity of the weapon system for the exclusion. The main tasks of ASIP [2] are as follows:

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

The original goals of ASIP were to (1) control structural failure in operational aircraft, (2) devise methods of accurately predicting service life, and (3) provide design and test approach that will avoid structural fatigue problems in future weapon systems.

The ASIP is also the standard by which the USAF can evaluate aging aircraft issues for structural components. For this purpose, the USAF normally emphasizes a subset of the elements of ASIP. For example, they extracted the appropriate elements of this program to perform the damage tolerance assessments (DTAs) during the seventies and eighties. The Air Force invested approximately one million man-hours in that effort to provide an inspection and modification program that greatly enhanced the safety of aging aircraft. Aging aircraft for many years have had a significant influence on the USAF research and development programs and have been a major driving influence on the elements of the ASIP.

Two of the main products of the ASIP process are development of the report on strength and operating restrictions and the development of the Force Structural Maintenance Plan (FSMP). If there is a need to change either of these documents because of flight beyond design usage that could introduce new critical areas, corrosion, WFD, or repairs, then the aircraft is said to be in a state of aging.

Experience with operational aircraft has shown they rarely fly according to their design spectrum of loads. Data from flight load recorders have typically shown there are considerable differences in usage severity among aircraft with the same designation. The USAF often finds the average aircraft usage is more severe than originally perceived early in the design process. This finding is made more significant by the fact the damage tolerance analysis may have not identified an area that would be a concern for aircraft with usage more severe than that assumed for design. Experience has shown the mass of an aircraft increases because of additional equipment or modification after an aircraft enters operational service. In addition, there are differences because there are changes in pilot techniques as they become more familiar with the aircraft, and mission changes because of new weapons and tactics. The aircraft-to-aircraft variability comes from several sources such as base to base variations in distance to test ranges and training. These experiences tend to degrade the capability of the full-scale durability test that consisted of two lifetimes of average usage to identify all the areas of the aircraft that could potentially cause a loss of safety. In most cases, an update of the DTA can account for any change needed in the inspection or modification program.

For the past forty years, the United States Air Force has used the USAF Aircraft Structural Integrity Program (ASIP) to maintain safe and economical operation of aging aircraft. This program has been supported over the years by USAF laboratory programs in the areas of fracture mechanics, corrosion prevention, flight loads, nondestructive evaluation, human factors, and maintenance and repair. These efforts provided the Air Force with the technology required to support the operational aircraft maintenance programs based on damage tolerance.

As indicated above, the USAF significantly changed this program because of the failure of an F-111 in 1969. This event ushered in the era of damage tolerance in the USAF [3]. The first assessments performed on the C-5A and the B-1A in 1971 and 1972 help derive the original DTA requirements for the USAF. These requirements were derived for monolithic (i.e., slow crack growth) structures. The failure of an F-4 wing on 23 January 1973 in a structural location the USAF considered fail-safe demonstrated to them that a structure could not be fail-safe without an inspection program. This failure strongly influenced the damage tolerance requirements as initially established first in MIL-A-83444 and subsequently in AFGS-87221A. The technology for the analysis of fail-safe designs has evolved slowly, primarily because of the need for extensive finite element programs supported by expensive test programs. The change to a damage tolerance approach prompted considerable research and development in the area of fracture mechanics. The then Air Force Flight Dynamics Laboratory was the focal point for much of this research. In the sixties and seventies, they developed much of the fracture technology that is still in use today. In addition, since the damage tolerance approach forced the engineer to better understand the stresses in the structure, finite element techniques emerged as the method of choice for the stress analysis. These capabilities permitted the USAF to perform a DTA of all the major weapon systems in the inventory in the seventies and eighties. This effort required over one million man-hours to complete and every major manufacturer was involved with this activity. Because of this activity, industry was able to develop the technology required for this type of analysis. This technology is also suitable for application to new aircraft developments. Consequently, the USAF was able to include damage tolerance requirements in the specification for new aircraft procurement.

After completion of the DTA on every major weapon system [4], the USAF laboratories continued research in other areas associated with aging aircraft. One of these was to make a better determination of the durability of aircraft. For this purpose, they sponsored research in the determination of initial crack distributions in aircraft structures. Much of this effort was concentrated on the interpretation of the cracks found in the teardown inspection of the F-16 wing after completion of the durability test. Another effort related to aging aircraft was the development of the procedure for the evaluating the probability of failure for a population of aging aircraft.

The need for nondestructive inspection technology to enable the damage tolerance driven inspections has been a major thrust of the Air Force for many years. Among these technology programs was a major effort to determine the probability of crack detection in an operational environment. Both the USAF and the FAA recognize the need for continuing the effort to quantify the capability of inspection techniques since this capability is critical to flight safety.

There are significant research and development efforts currently underway in the area of nondestructive evaluation of aging aircraft. NASA LaRC and several academic institutions including Iowa State University and Johns Hopkins University are doing much of this work. The USAF is working with these institutions and the FAA Technical Center to ensure these efforts meet the their requirements.

The USAF research and development program for aging aircraft has provided the technology base for safe and economic operation of military aircraft through the ASIP. As an indicator of this success, the failure rate for all systems designed to and/or maintained to the current policy is one aircraft lost due to structural reasons in more than ten million flight hours. This is significantly less than the overall rate of aircraft losses from all causes by two orders of magnitude. It has also, at times, given program managers a false sense about the remoteness of structural failures. This success, however, should not be used to indicate there is no need for continued research on the structure of aging aircraft. The return on the investment in this research is reduced cost and downtime with inevitable structural problems.

As indicated above, the materials in many aircraft were the result of the desire for improved performance with little attention given to the potential for corrosion and stress corrosion cracking damage. Further, at the time of manufacture of many of these aircraft, the focus on corrosion protection was not what it is today. Many of these early corrosion protection systems have broken down. In the open areas, the operator can readily renew them. There is, however, no easy way to renew the corrosion protection system in the numerous joints. Experience with modification and repair of aging aircraft has revealed that joints without proper protection experience significant damage that results in costly part replacements.

The corrosion concern is now becoming more acute in that the environmental protection laws have eliminated the use of some of the standard corrosion inhibitors. Another issue is that the nondestructive evaluation techniques are marginal. The standards for corrosion damage are so poorly defined that it is difficult to properly characterize the damage found. This deficiency creates a real problem in the future years cost projection for structural maintenance.

**The Damage Tolerance Assessment (DTA) Process**

The definition of damage tolerance is the following:

Damage tolerance is the attribute of a structure that permits it to retain it required residual strength for a period of unrepaired usage. It must be able to do this after it has sustained specified levels of fatigue, corrosion, accidental, or discrete source damage. Examples of such damage are (a) unstable propagation of fatigue cracks, (b) unstable propagation of initial or service induced damage, and/or (c) impact damage from a discrete source.

Figure 1 shows the steps in the DTA process. This description applies primarily to the process used by the USAF. The procedure used by commercial operators is quite similar.. The DTA is an integral part of the aging aircraft program for both military and commercial aircraft. The concept is simple. The flight time to the first inspection is based on the time required for the largest defect expected in a fleet of aircraft from manufacturing or in-service damage to grow to critical crack length. Subsequent inspections are based on flight time for the NDI detectable defect size to grow to critical crack length. A crack growth function illustrating this process is shown in Figure 2.

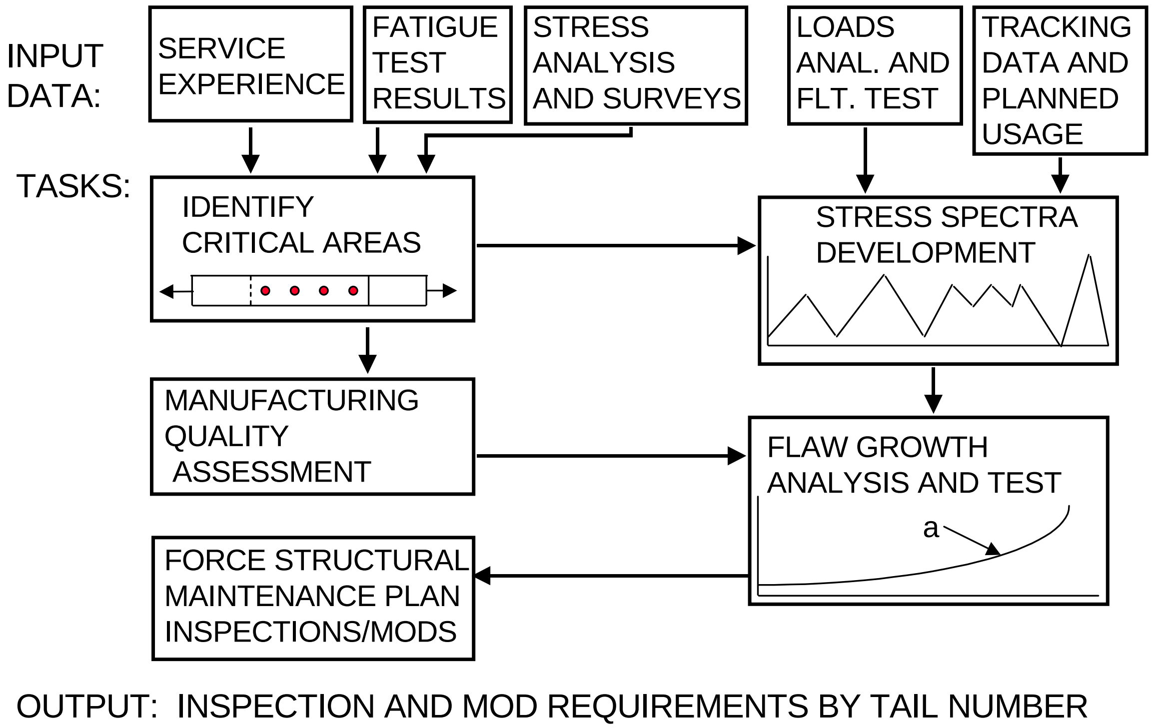


**Figure 1 The Damage Tolerance Process**

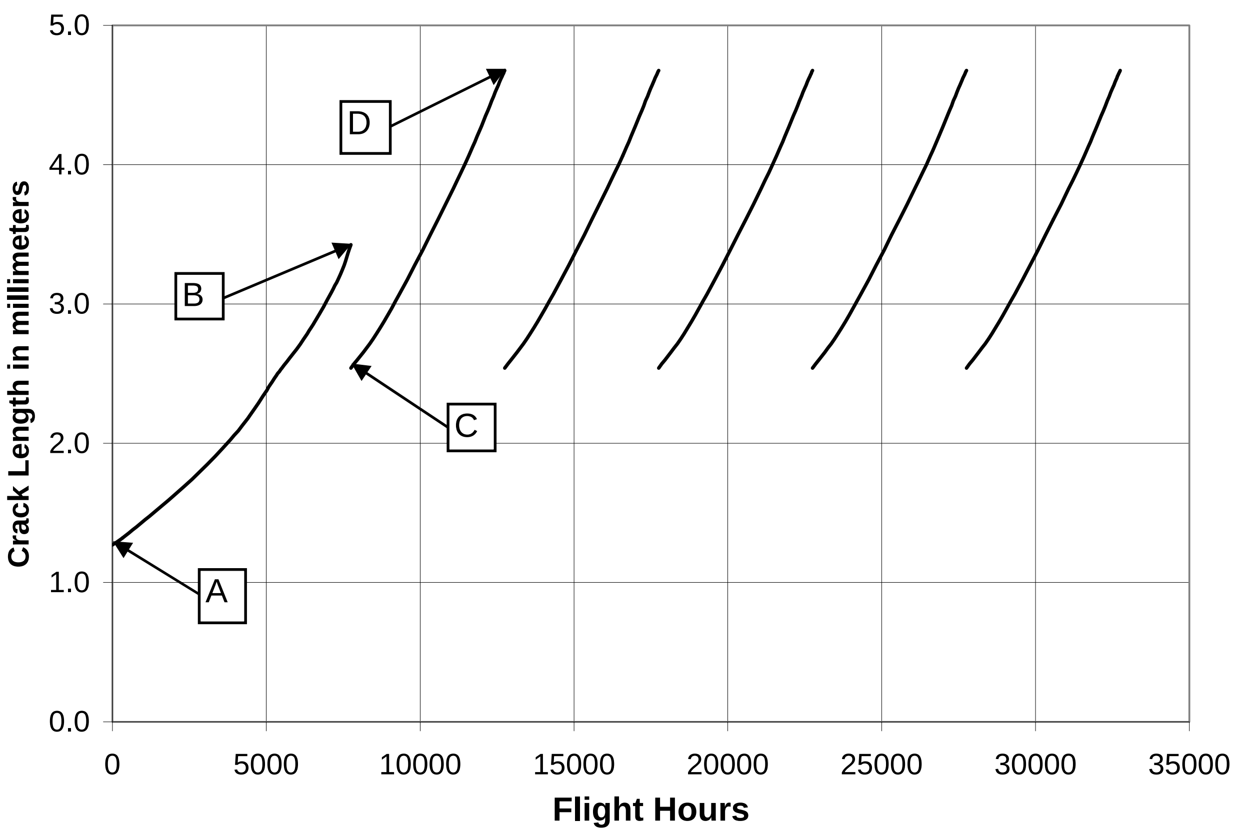


**Figure 2 Damage Tolerance Crack Growth Function**

The ordinate of the point A is the initial flaw assumed for the analysis. The abscissa of the point B is half the time needed for the initial flaw to grow to critical. The ordinate of the point C is the NDI detectable crack length. This crack is then grown to the point D whose abscissa is half the time required for crack to grow from B to critical crack length. The process is repeated until the inspections reveal an actual crack or the structure needs to modification for WFD.

The process evolved over a period of several years after the USAF applied it initially to the B-1A and the C-5A. Its successes include the F-4, an aircraft that did not have a requirement for life when the U.S. Navy procured it. The USAF purchased this aircraft in large quantities, and it became an essential ingredient of their fighter fleet. After a crash at Nellis Air Force Base in 1973 caused by fatigue, the USAF found themselves in a difficult situation. They initiated a recovery program that included a DTA and fatigue test conducted in their laboratory at Wright-Patterson Air Force Base. Because of this effort, the F-4 remained in operational service until the nineties without further incident. During the seventies and eighties, the USAF performed a DTA of every major weapon system in their inventory [17]. These successes motivated the USAF to apply this technology to engine structures with similar results. The discussion below describes the method used for damage tolerance with examples on how the USAF applied it.

The first task of the DTA is the identification of critical areas of the structure. A critical area is a location or part of the structure that could affect flight safety and may need maintenance in the form of an inspection or modification during the life of the aircraft. There are several techniques for identification of these areas. Actual cracking experience through service operations or durability testing is usually the most important consideration. Areas that have high predicted or measured stress and details that make them prone to cracking are, of course, prime candidates for the assessment. Another consideration in the selection of critical areas is its ease of inspection. In general, the analyst gives higher priority for selection on critical areas difficult to inspect. It has been extremely helpful to use the accrued knowledge of the original aircraft contractor in identifying potentially critical areas. On some of the assessments, preliminary estimates were made of the flaw growth in the candidate critical areas. When this of inspection. In general, the analyst gives higher priority for selection for areas that are difficult to inspect. It has been extremely helpful to use the accrued knowledge of the original information was available, it was much easier to make a decision on which of the candidate areas the analyst should subject to a final analysis. For small aircraft, the number of candidate areas generally was of the order of 40 to 70. The analyst would normally be able to screen these down to 10 to 30 for final analysis. For larger aircraft, the number of candidate areas generally was of the order of 60 to 150. The analyst would screen these down to 30 to 60 for final analysis.

The second task of the DTA is the development of the stress spectrum for each area identified for a final analysis. This is one of the more demanding aspects of the DTA process. The reason is there are significant changes in the rate of flaw growth due to relatively small changes in the cyclic stresses. To perform this task properly, generally three data items must be available to the analyst. First, he must have operational experience available in a usable form. This operational data must provide a basis for establishing a flight-by-flight sequence of points in the sky (i.e., altitude, weight, and aircraft motion parameters). This was usually available from multi-channel data on fighter or attack aircraft. For transport category aircraft, the USAF usually derived the sequence from flight log information supported by multi-channel data to define the maneuver and gust environment. In all cases except one, there was a sufficient database to derive the sequence. This exception was the A-7D, which was equipped with counting accelerometers only. Consequently, as a part of the A-7D DTA, an operational data base was derived from collecting 1,250 hours of multi-channel data from aircraft located at two bases. The second data item necessary for the derivation of the stress spectra is the set of equations needed to determine the external loads (i.e., shear, bending moment, and torsion) for a given point in the sky. For USAF aircraft, the manufacturer usually determined the external loads through analyses, wind tunnel testing and in-flight strain surveys. For all aircraft except the F-4, there was sufficient confidence in the existing data to perform this task. The USAF elected to perform a flight loads survey on this aircraft during the course of the DTA. This turned out to be very beneficial because the pre-existing data would have produced a pessimistic view of the maintenance burden for this aircraft. The final data item needed for the generation of the stress spectra is the external loads to stress transformation. For all of the aircraft studied, there were at least some experimentally derived stresses from previous static and durability tests. However, in all cases it was necessary to conduct additional stress analyses. The contractor performed these additional analyses typically using the finite element method. The scope of this finite element effort ranged from evaluating stresses at local details to finite element models of the complete airframe. The finite element effort varied significantly from aircraft to aircraft because of differences in the test database and the complexity of the critical details. Simplification of the stress spectra effort would have been possible if direct strain measurements had been available. In general, these data were not available. In a few cases, such as the C-5, this kind of information was available and was invaluable for determining the environment from maneuver, turbulence, and aerial refueling.

The techniques used in deriving the stress spectra for the assessments varied quite widely from aircraft to aircraft. Part of the reason for this difference was due to available database. For example, for the F-4, the data collected from the VGH recorder provided the number of occurrences of combinations of Mach number, load factor, and altitude in predetermined bands. Consequently, the assessment of areas of the aircraft sensitive to asymmetrical loading required data from other aircraft or from pilot interviews. For the F-15, however, the Signal Data Recorder provided a time history of both symmetrical and unsymmetrical parameters for use in developing the stress spectrum. The F-15 database more accurately accounted for the unsymmetrical loading. Moreover, it permitted a more realistic assessment of the minimum stress excursion that followed a maximum stress excursion. For the F-4, the conservative assumption had to be made that after a maximum stress there followed either a stress corresponding to one-g flight or a stress corresponding to less than one-g flight. The database on the F-15 enabled the analyst to remove this conservatism.

The USAF performed all of the fighter and attack aircraft assessments by reconstituting the individual flights from the databases except for the F-111. For this aircraft, the multi-channel recorder data was used directly to randomly generate a "block" of flights of approximately 500 hours. This is a very effective approach if one can be sure the selected flights are representative of the aircraft usage. For the F-111, they used the counter data for load factor as a guide for this selection. There was no attempt made to maintain the original order of the individual flights since previous studies for the F-4 and other aircraft showed sequence effects were insignificant if the flights were randomly selected.

As indicated earlier, the VGH recorder was the basis for the F-4 usage database. A sampling technique based on VGH recorder data provided the approach for the derivation of the stress exceedance function. In this method, the analyst computed the stress for a representative set of Mach number, load factor, weight, and altitude combinations. The surface derived from the representative points provided the means to determine the stress at the flight-measured points. Thus, the recorder data determined the stress exceedance function accounting for "all points in sky." The USAF used a modification of this approach in the development of the stress exceedance function for the A-7D DTA. For the A-7D, the approach involved a regression equation to interpolate based on the stresses computed for a representative set of aircraft flight conditions.

The environmental data that augmented the flight log data for the large aircraft were extremely important. The USAF refers to these data, used in the ASIP, as the loads/environmental spectral survey (L/ESS) data. It provides the means to quantify the three dimensional nature of wing gust loads, the phasing of shear and bending moment, and the aerial refueling loads on the C-5. These data were also very helpful for evaluating the low-level turbulence on the B-52, C-141, and C-130. In many cases, such as the fire-fighting mission for the C-130, special mission maneuver data needed quantification. It is the intent in the derivation of the stress spectra to determine the "baseline usage" as an average usage for the force. For aircraft where there were significant usage changes during their life or there were possible changes in their future usage, the baseline usage reflected these changes. For some aircraft, such as the F-111, with different Mission Design Series (MDS), the USAF derived a separate baseline usage for each MDS. In addition to the baseline usage, there is a need to derive stress spectra that represent potential variations from the baseline. The testing of these variations develops confidence the procedure for tail number tracking by fracture mechanics methods is valid. For the older aircraft assessments, the usual procedure was to define a spectrum more severe than the baseline and a spectrum less severe than the baseline. Changing the baseline mission mix generally accomplished this.

For the larger tanker, transport, and bomber aircraft, the main source of data was the flight logs. In general, these logs had sufficient detail such that engineers could divide the usage among a relatively few missions (of the order of ten). Typically, the assessment had to include two or more distinct usage changes. For example, for the B-52Gs there were differences in usage prior to, during, and after their Southeast Asia operations. In addition, the USAF anticipated the usage of the aircraft in the future to be different from all the previous usage.

The third task of the DTA is to establish the initial flaw size for the fracture analysis. Because of their inherent stress concentration, fastener holes were predominant as candidates for critical areas of the airframe. The USAF noted there had not been a structural failure in the number of flights it takes for a 1.27 mm corner flaw in a fastener hole to grow to critical crack length. By 1975, they believed there was sufficient data to make the judgment that this size was sufficient to ensure aircraft safety. They derived this belief partially from teardown inspections of full-scale fatigue test aircraft, but primarily from observing operational aircraft such as the F-4, C-5A, and the KC-135. There has never been a rigorous substantiation of this belief. However, experience in subsequent years supports use of this size defect as being adequate to protect flight safety. The remaining task then was to determine the flaw size for holes that were cold worked or filled with an interference fit pin. The USAF determined this flaw size on ad hoc basis. In some cases, where there was a question of the adequacy of the installation of the interference fit pins, there was no reduction allowed. In other cases, where there was confidence the installation was proper, the USAF reduced the initial flaw size to 0.127 mm. The primary considerations in making this judgment were durability test performance and manufacturing procedures. For example, the C-141 durability test showed that the tapered fasteners did extend the life of that aircraft. However, there was some concern about the quality of the hand held drilling operations in some areas of the wing. Consequently, the DTA did not account for the benefit of the tapered pins. However, machines with controlled feed and speed drilled many of the wing fastener holes. For these holes, the USAF made a decision to reduce the initial flaw by approximately a factor of two.

The identification of the stress spectra for each critical area and the initial flaw permits the initiation of the task of establishing operational limits. This combined analysis and test effort uses the disciplines of fracture mechanics to find the safety limit for each critical area. The fracture mechanics technology has improved significantly from the early 1970 period. However, even with the analytical capability available today, the process would be meaningless without test substantiation.

There are two main reasons for fracture testing. First is analysis verification. The aim is to accomplish this with the least specimen complexity possible in order to isolate the local detail (e.g., a fastener hole) and evaluate the spectrum retardation. The specimens used for this purpose were generally dog bone specimens with the proper material, thickness, size of fastener, and load transfer. The second reason for testing is to establish high and low side truncation levels. The low side truncation is primarily an economic consideration. The object is to eliminate as many cycles as possible with a small stress range without significant change in the crack growth. The removal of the high stresses (or clipping) eliminates those cycles that make a crack grow slower and retain those which make a crack grow faster.

In the early 1970 period, there was a belief that crack growth was quite sensitive to the loading sequence within a flight. Therefore, as part of the DTA, the USAF required tests to evaluate these effects. It was learned that if they simulated loading on a flight-by-flight basis, then the ordering of loads within a flight was of secondary importance. Consequently, they discontinued this type testing. For almost all of the aircraft, the fact that the critical crack sizes were sufficiently small such that there was little if any redistribution of stresses during crack growth simplified the fracture analysis. Further, the primary structural issue was crack growth from a fastener hole or an open hole. Therefore, the analyst needed to concern himself with the part through flaw in mode I cracking from both filled and unfilled holes, load transfer on the fasteners, and retardation effects. For simulating the retardation effects, the analysts generally used the Wheeler model, the Willenborg, the modified Willenborg, or some form of a contact stress model. The T-38 analysis used the Vroman model for part of its DTA. This model was not used for any other assessment.

Analysts learned that proper counting of the stress cycles in the spectrum was essential for obtaining accuracy. The so-called rain flow procedure is now commonly accepted as an adequate procedure for counting the stress cycles in the spectrum [30].

After the verification of the crack growth analytical model through coupon and in some cases, component testing, the effects of the chemical environment entered the analysis process. For the early assessments, the tendency was to take a conservative view of the environment. That is, the USAF required the selection of an environment more aggressive with respect to crack growth than actually expected. This position was relaxed in the late 1970s and they placed emphasis on selecting a realistic environment. Constant amplitude crack growth tests performed in the desired environment provide the basis for the quantification of these effects. The crack growth analysis includes the environmental effect in the data used for the crack growth rates. The procedure is subject to criticism because it may not accurately account for the effects of cyclic frequency and load interaction effects with the environment. There is no indication from the inspections performed on operational aircraft the error is significant.

The crack growth analysis plays a dominant role in damage tolerance approach. The tool must be usable for different chemical as well as loading environments. In other words, it is the mechanism for tracking the crack growth on each tail number in the force and thereby ensuring aircraft safety. Therefore, it is extremely important to validate the analysis for the expected range of service operations. After the analyst establishes the safety limits for all the critical areas of the structure, the development of the Force Structural Maintenance Plan (FSMP) can proceed. The FSMP provides the how, when, and where for structural inspections or the when and where for modifications. In many cases, it was found, based on either economic or safety considerations, that modifications were preferable to continued inspections. This situation existed for the C-5, T-38, F-4, and KC-135, for example. Of course, the DTA process should include the modifications.

One of the more important tasks in the damage tolerance approach was to establish the NDI capability. This was done with the help of the NDI experts from the now Air Force Research Laboratory (AFRL) Materials Directorate, the appropriate Air Logistics Center (ALC), and the contractor. In some cases, such as the EF-111, they conducted an NDI reliability program to determine the flaw size corresponding to 90 percent probability of detection with 95 percent confidence. However, these cases were in the minority and, consequently, the USAF based most of the NDI detection capability on judgment. When possible, they avoided inspections that involved removal of fasteners. In addition, the USAF rejected the concept of sampling inspections rather than inspecting 100 percent of the force.

The FSMP covered the period of the planned operational usage of the aircraft. Thus, the FSMP permitted the ASIP manager at the ALC to determine the out years maintenance cost. The accuracy of these costs was suitable for budgetary estimates. The accuracy for any given tail number is; of course, dependent on how closely that aircraft flies to the baseline.

USAF structural engineers have long recognized the need for tail number tracking of aircraft. This is evident from the emphasis given to it in the 1959 version of ASIP. The only significant change from the original version is the tracking process is for crack growth rather than fatigue damage. The USAF developed the first tracking program based on fracture for the F-4 during its DTA. Now all aircraft that have had a DTA have a tracking program based on fracture mechanics. For many of these aircraft, the ASIP manager has the computer programs to provide an immediate view of the maintenance status of his aircraft. This provides him with both near and far term planning and decision making capability. It provides him with the capability to determine the consequences of a mission change. There is also a need for commercial aircraft to have a periodic reassessment of their usage. The availability today of excellent digital recording devices has made this task considerably more manageable than in the past.

The damage tolerance approach has led to a greatly improved understanding of aircraft structures and their performance. It is the foundation for maintaining flight safety in aging aircraft. It has also led to a greater recognition that additional research and development in the areas of materials, structures, and nondestructive evaluation were not only needed, but could further increase the reliability of systems. Consequently, over the last several years, many programs have focused themselves on increasing the knowledge base available to enable longer lives and more reliability from airframes and engines. Overall, the damage tolerance experience has been good. The criticism, which is rare, has come from people who believe the approach is too conservative when they perform an inspection, and find no cracks. On the other hand, the DTA process has correctly directed inspections to areas that full-scale testing did not indicate they were critical. Figure 3 shows the DTAs performed by the USAF during the seventies and eighties.

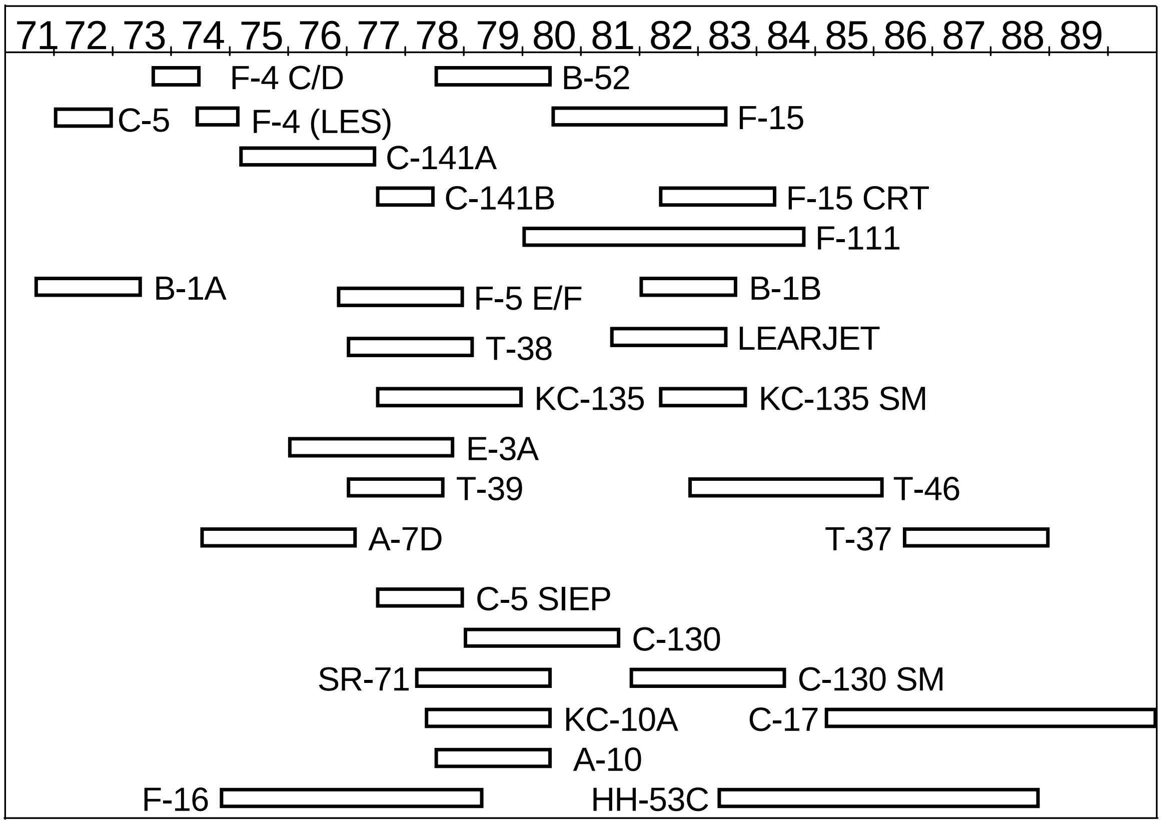


**Figure 3  Damage Tolerance Experience in the USAF**

## Conclusions

Operational aircraft failures from fatigue in the fifties and sixties motivated a fundamental change in the approach for ensuring safety of flight for aircraft.  In the seventies, many certification authorities endorsed the damage tolerance process for design and maintenance of safety critical structure.  The process uses stress analyses, loads analyses, and fracture mechanics to determine inspection intervals or modification times to the in-service maintenance program.  This disciplined process has proven to be successful in preventing structural failures from fatigue.

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# Snecma ATAR Engines Cost Effective Maintenance in a 1960-2020 Life time

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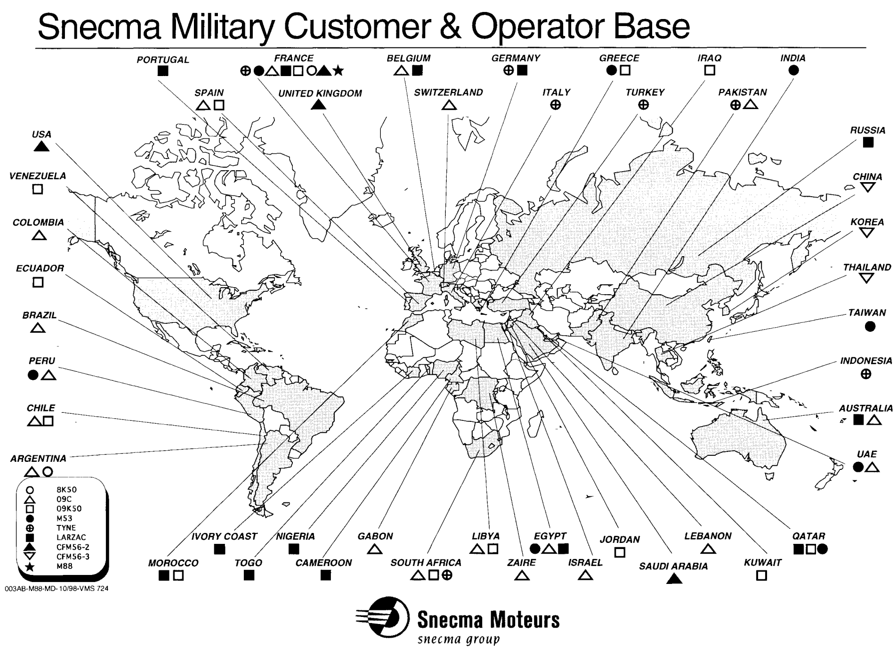
## 1. **Introduction** :

Today, 47 airforces are operating more than 6000 engines sold by Snecma or by CFMI, the joint company (50/50) of Snecma (France) and GE (USA) – (fig1).




Among those engines, some have been operated for more than 30 years (fig.2).

**SNECMA Military Engine Experience** (as of December 31st, 1999)

| ENGINE | AIRCRAFT | ENGINES IN SERVICE | OPERATORS | SERVICE EXPERIENCE |
|---|---|---|---|---|
| • Atar 08/09K50 | Super Etendard, Mirage F1, 50, Cheetah, Pantera | 842 | 14 | 1,870,680 h |
| • Other Atar (08C/09B/09C/09K) | Etendard, Mirage III, IV, V | 768 | 14 | 4,126,240 h |
| • Tyne | Transall, Atlantic 1, Atlantique 2 | 838 | 9 | 5,798,510 h |
| • Larzac | Alphajet, MiG-AT | 1,129 | 12 | 2,665,675 h |
| • M53 | Mirage 2000 | 620 | 8 | 781,616 h |
| • CFM56-2A/-2B/-2C | E3, KE3, E6, C135FR, KC135R, DC8-72 | 2,018 | 7 | 7,480,000 h |
| • CFM56-3 | B737-300 *(12-31-1998) | 13* | 3* | 78,000 h* |
| TOTAL | **22** | **6,228** | **47** | **22,800,721 h** |

**EVERY MINUTE,**
**A SNECMA MILITARY ENGINE TAKES OFF**

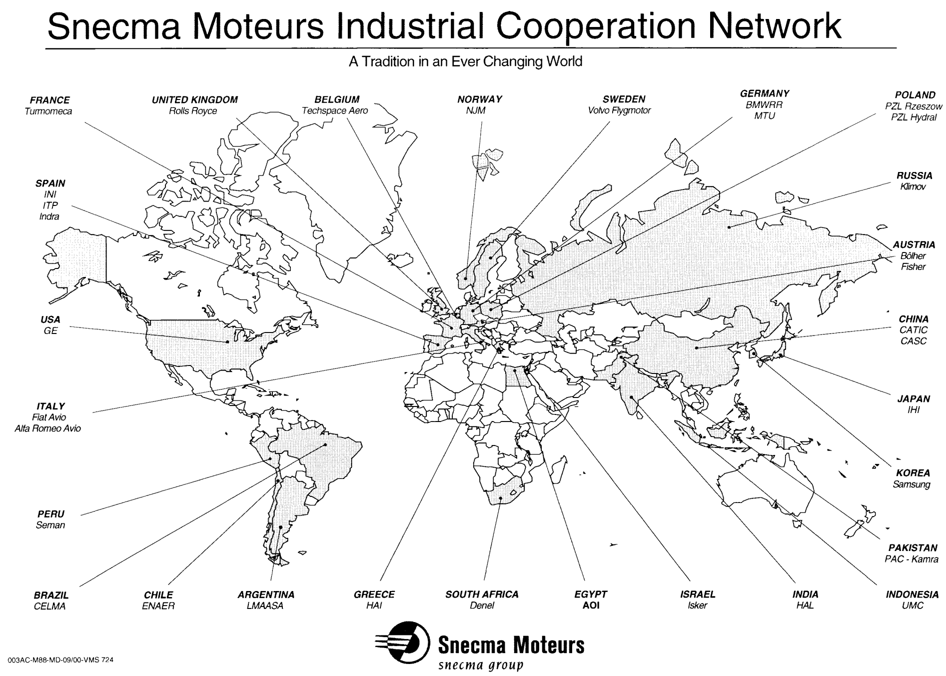
006-SNECMA-06/00-VMP 1197

Snecma Moteurs
groupe snecma

*Paper presented at the RTO AVT Lecture Series on "Aging Aircraft Fleets: Structural and Other Subsystem Aspects", held in Sofia, Bulgaria, 13-16 November 2000, and published in RTO EN-015.*

Snecma's target is to have all our customers satisfied. Therefore, we have developed a philosophy of product and service continuous improvement, covering in in particular:

- Life extension and maintenance cost reduction programs.
- Modification proposals triggered by mission profile evolution.
- Better of involvement of customers national industry (fig 3).



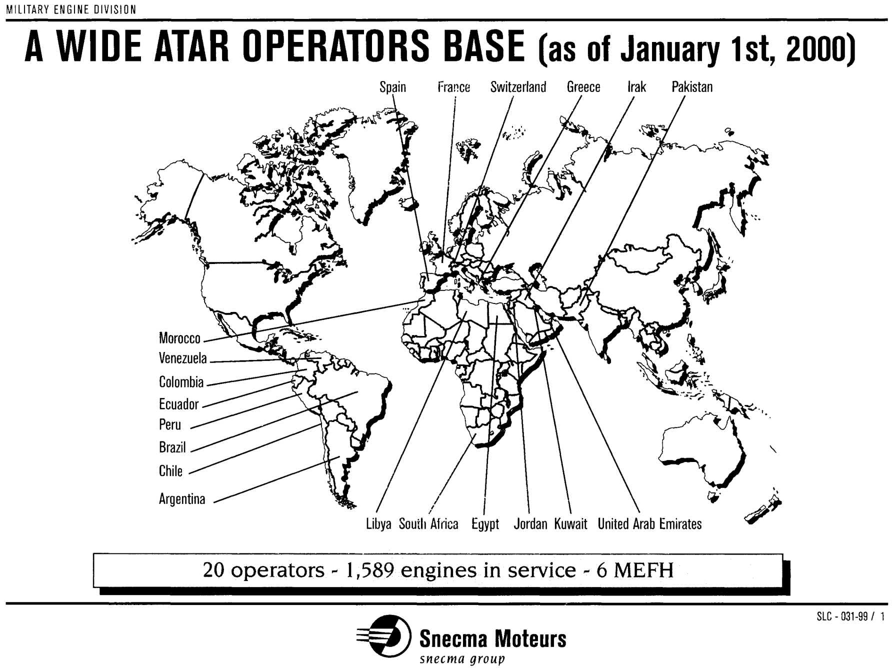

The following paragraphs of this paper will explicit how those principles have been implemented on the ATAR engine program between Snecma and the operators.

## 2. **The Snecma commitment**.

- ATAR 9C engines are installed on Mirage 3s and Mirage 5s
- ATAR 9K50 are installed on Mirage F1 and Mirage 50.

The present operators base (fig 4) includes a large number of operators with limited resources and who have planned to operate the engines up to 2020.

***Snecma is committed to support the ATAR customers until the end of service of the ATAR engines.***




The traditional way to maintain ATAR engines is to send the complete engines to the Depot Level maintenance shop once the operating life limit has been reached and then perform the DLM overhaul and repair.

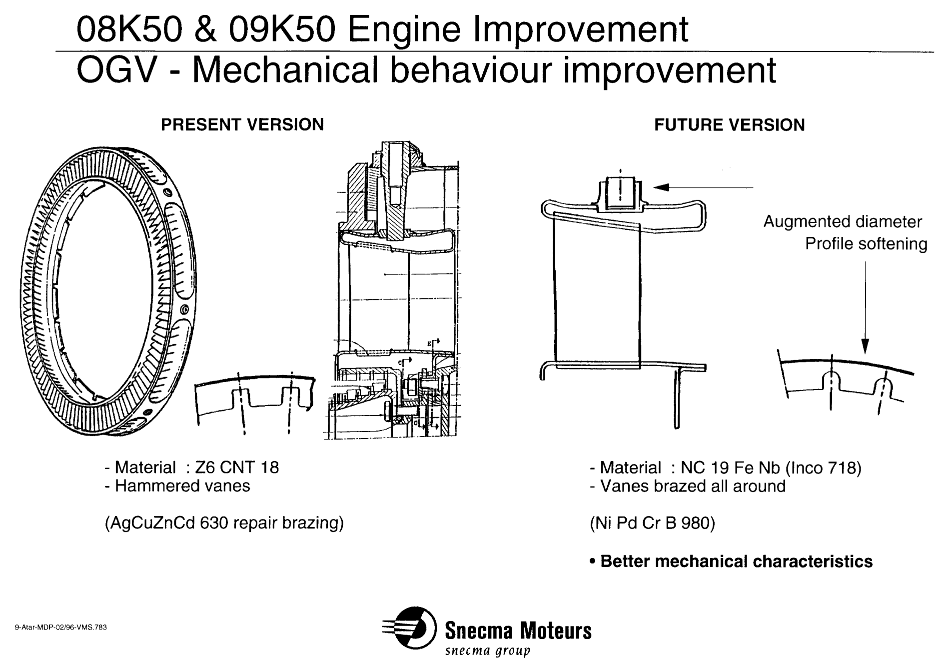
This scheme fits well the needs of large fleet operators. However, it was found a little too expensive by some operators with smaller fleets and more limited resources. This is the reason why Snecma has developed the following tools:

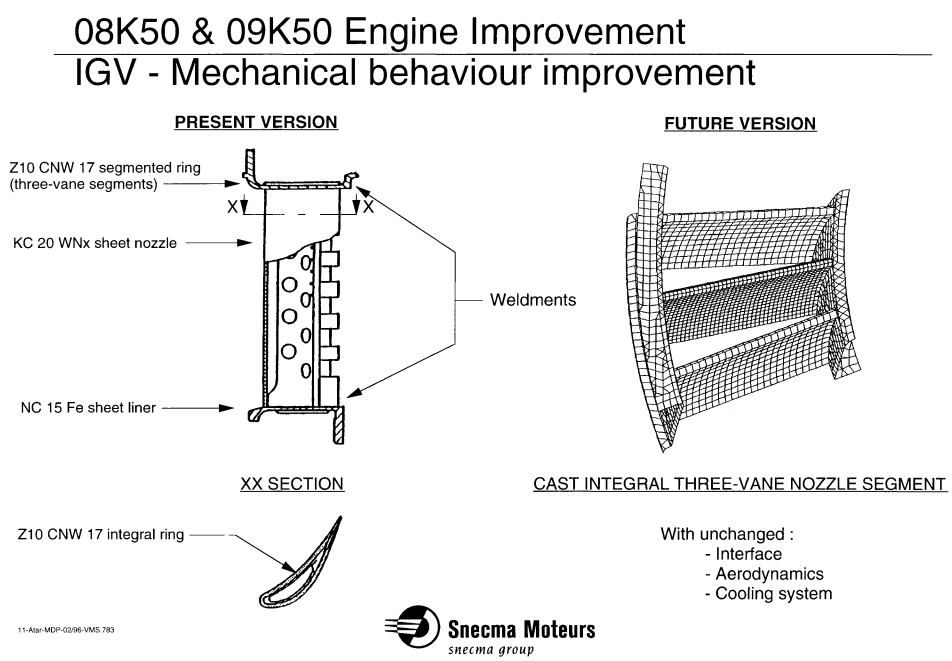
- Long term support contracts
- Modular maintenance
- Second hand hardware availability.
- Standard exchange instead or repairs.

## 3. <u>The ATAR Plus program</u>

France, South Africa and Spain have jointly determined that one of the ways to limit ATAR 9K50 maintenance costs was to introduce a series of modifications known as the “ATAR Plus” program, including

- Compressor OGV upgrade (fig.5)
- HP turbine NGV upgrade (fig.6)




The ATAR Plus program has been launched by a consortium of three companies:

- Snecma (France)
- Industria de Turbopropulsores (Spain)
- Denel Aviation (South Africa)

and is now entering production.

## 4. **Long term support Contracts Initiative**

Cost limitation implies cost control.
Snecma has identified the need, voiced by some customers, to have a complete support contractual package based on the following principles:

- The operator performs maintenance operation only at the airfield location.
- Snecma Moteurs performs all the rest of the engine maintenance and support on a design to cost basis.

This leads to contracts between Snecma and the customers with the following typical features:

- Time of the contractual package : 5 years
- Fixed yearly price for general support (engine, test cell, GSE) and Technical assistance - training - documentation.
- Snecma commitment to maximize the involvement of the customer's national industry.

## 5. **Modular maintenance**.

ATAR engines family was designed between 1946 and 1960s at a time where performance was found more important than cost and particularly maintenance cost.

The engine maintenance could be split into modules only at the Depot Level.

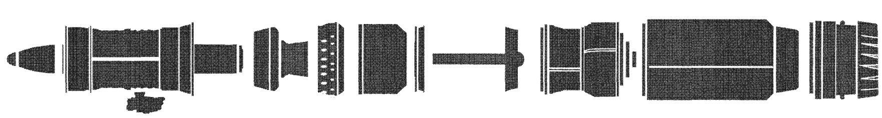
Some years ago, Snecma has proposed to some customers to split the engines into modules (fig.7) at the airfield level in order to:

- Improve drastically the engine availability in the fleet
- Reduce the overall maintenance cost by about 30 %.

### ATAR 9K50 / Modular Maintenance

- ❑ The ATAR 09K50 is broken down into Overhaulable Sub-Assemblies (OSA) which are interchangeable as far as their dimensions and operation are concerned
- ❑ There are 23 structural sub-assemblies, 4 sub-assemblies for equipment parts and 93 accessories included in the sub-assemblies but which may be replaced individually

***Main Sub-Assemblies***



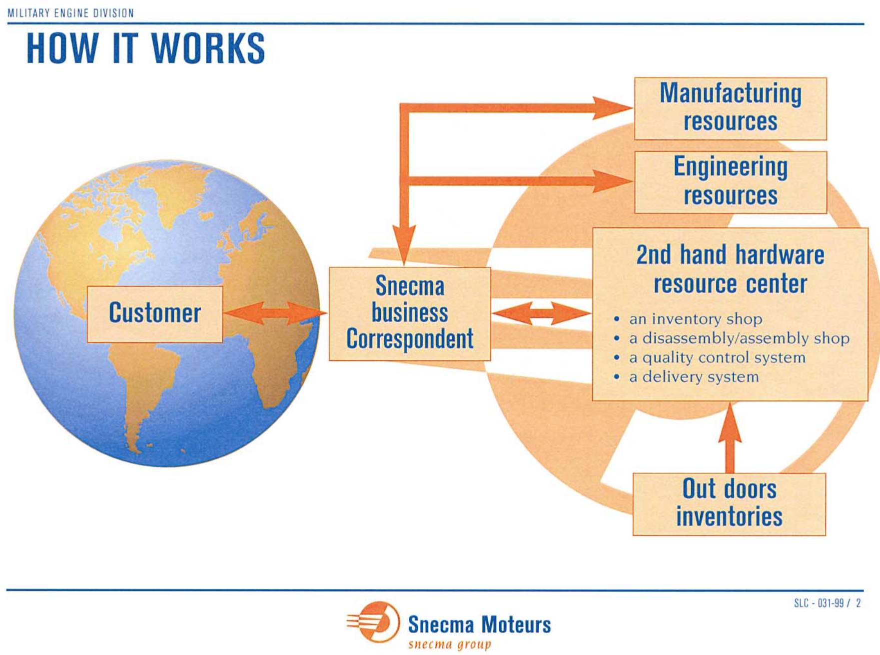
36A-9K50-McD-06/00-VMS-750

Snecma Moteurs
snecma group

## 6. Second hand hardware Resources

French Air Force has progressively retired its Mirage 3 and ATAR 9C in the 1980s time period.
A large number of modules and parts, either stored but not used, or operated for a certain time, but still with an interesting life time have become available.

Snecma has installed recently a "ATAR Second Hand Resource Center" (fig.8).




Snecma regularly publishes the list of the available second hand parts and modules and when a request for proposal is received at Snecma, the response mostly includes a mix of new parts and second hand parts.

This approach allows the customer to optimize its ownerships cost of the parts, while maintaining a high level of quality on its engines since

***Snecma grants any second hand hardware the same level of quality and guarantee than the one granted for new parts***

## 7. Standard exchange vs repair approach

The availability of low cost engine modules with an interesting remaining life time (paragraph 6) allows Snecma and customers to consider module standard exchange, at a cost substantially lower than a module repair.

This standard exchange can be done either at the airfield level or at the Depot Level, according to the customer's choice.

In any case, Snecma supports the customer and is assisting him in the implementation of his decision.

## 8. **Critical parts management**

Second hand hardware allows to reduce maintenance costs, but all the customer's needs are not fulfilled by to second hand parts.

Some ATAR parts still need to be manufactured by Snecma and subcontractors.

As the manufacturing quantities are declining, manufacturing prices are rising.

Because Snecma wants to have ATAR operators satisfied until the end of their operation, we have initiated a Critical Parts Management dialogue with our customers as follows:

Step 1 : The operators have provided Snecma their critical parts needs evaluation covering operations until the end of their ATAR operation.

Step 2 : On this basis, Snecma has issued a preliminary possible critical parts production plan explaining what parts are likely to stay on the production line.

Step 3 : Snecma will issue last batch production offers including prices, schedule and launching conditions.

Step 4 : For the parts where launching conditions are met, Snecma will launch <u>critical parts last batches</u> manufacturing and subsequent deliveries.

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# REPAIR OPTIONS FOR AIRFRAMES

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## 1. INTRODUCTION

Maintaining the airworthiness of in-service aircraft and at the same time keeping the maintenance cost low is of prime concern to the operators and regulatory authorities. In order to keep maintenance cost low, right decisions need to be made regarding replacing or repairing the in-service damaged components. The choice between replacing or repairing a structural component is governed by a number of factors such as the availability of spares, duration a structural component is expected to be in service, feasibility of repair, repair meeting structural integrity requirements, and inspection requirements for the repair. If it is economical to repair the component then the optimum repair design needs to be selected.

This paper discusses structural life enhancement techniques along with the state-of-practice methods of repairing metallic and composite structures. Applications of advanced repair methods such as composite patch repair of cracked metallic structures are discussed. Available computer codes for designing repairs are briefly described.

## 2. STRUCTURAL LIFE ENHANCEMENT OPTIONS

Stress levels, load spectrum, environment, structural details and the material of the structural component, govern the life of an aircraft structure. Under certain loading and environmental conditions a crack may initiate and propagate in a metallic structural component or environmental conditions may cause severe corrosion in the component. Depending on the structural details, the crack or corrosion damage may result in a catastrophic failure or costly repairs. A logical preventive method is to retard the initiation and growth of the cracks by pre-stressing so that the cracks do not result in catastrophic failure before the useful life of the structure. In certain cases this may not be feasible and a structure may have to be repaired to meet the useful life requirements. In addition, the in-service damage due to foreign objects in both metallic and composite structures frequently requires repairs so that the structure is able to carry the required load. Two commonly used techniques of structural life enhancement (Reference 1) by prestressing and repairs are summarized in Figure 1.

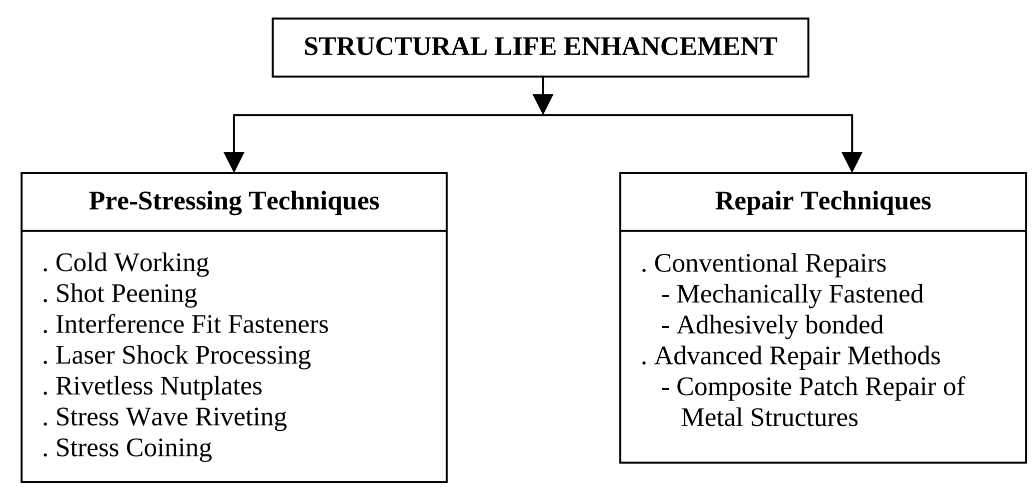



**Figure 1. Life Enhancement Techniques**

*Paper presented at the RTO AVT Lecture Series on "Aging Aircraft Fleets: Structural and Other Subsystem Aspects", held in Sofia, Bulgaria, 13-16 November 2000, and published in RTO EN-015.*

Prestressing techniques to enhance structural life are generally used before a problem has occurred. In the design and analyses process, if a component or some parts/areas of a component are not able to meet design life requirements, prestressing process may be used for these locations to meet service life requirements. In case of in-service aircraft, if fleet data indicates cracking problems in certain areas, these areas may be subjected to prestressing process to enhance life before cracks initiate.

**Life Enhancement Through Pre-stressing Techniques**

In this technique a residual compressive stress field is created at highly stressed locations such as holes where cracks are likely to initiate. Subsequent inflight loads have to overcome the compressive stresses in order for the cracks to initiate and propagate. Some prestressing techniques have been fully developed while others are still in the development stage and have shown good promise to enhance structural life. The applications of these techniques to in-service aircraft are shown in Figure 2. The figure also shows the locations where these techniques are applied (e.g. whether the technique can be used at the manufacturing line, depot or field). The analysis methodology that can be used for life predictions is also shown in the figure. The level of verification testing required for successfully implementing the technique is also given in the figure. The extent of life enhancement achieved through these techniques is discussed in Reference 1.

| **PRE-STRESSING TECHNIQUE** | **IN-SERVICE APPLICATIONS** | **LOCATION WHERE PERFORMED** | **ANALYSES METHODS** | **REQUIRED TESTING** |
|---|---|---|---|---|
| COLD WORKING | T-38, F-5, F-16, JSTARS F-18, F-111, C-141, 747 | MANUFACTURING LINE, DEPOT AND FIELD | EQUIVALENT INITIAL FLAW(EIF), FATIGUE LIFE FACTOR(FLF) | MINIMUM |
| SHOT PEENING | T-38, F-5, F-18, F-14, 737,747,C-130,B-1 | MANUFACTURING LINE, DEPOT AND FIELD | EIF, FLF | MINIMUM |
| INTERFERENCE FIT FASTENERS | T-38, F-5, F-18, 747 | MANUFACTURING LINE, DEPOT AND FIELD | EIF, FLF | MEDIUM |
| LASER SHOCK PROCESSING | NONE KNOWN | MANUFACTURING LINE | DEVELOPMENT REQUIRED | SUBSTANT- IAL |
| RIVETLESS NUTPLATES | F-22, T-38 | MANUFACTURING LINE, DEPOT AND FIELD | EIF, FLF | MEDIUM |
| STRESS WAVE RIVETING | F-14, A6E | MANUFACTURING LINE AND DEPOT | EMPIRICAL | MEDIUM |
| STRESS COINING | F-18, DC-8, DC-9, DC-10 | MANUFACTURING LINE AND DEPOT | EMPIRICAL | MEDIUM |

**Figure 2. Prestressing Life Enhancement Techniques Applications**

**Life Enhancement Through Repairs**

Structural life enhancement techniques through repairs for in-service fatigue, corrosion and foreign object damage (FOD) have been well established for metallic aircraft. With the increasing use of composites for improved structural efficiency, these methods have been developed for composite materials. However, there are basic differences between the damage types and their behavior in composite and metallic materials (Ref. 2-4). The basic differences between the behavior of metals and composites need to be understood so as to design proper repairs for metallic and composite structures. Figure 3 shows a comparison of typical metal and composite fatigue behavior under fighter aircraft wing spectrum loading. The data are plotted for each material’s most sensitive fatigue loading mode, which is tension-dominated (lower wing skin) for metals and compression-dominated (upper wing skin) for composites. The figure shows that composite fatigue properties are far superior to those of metal.

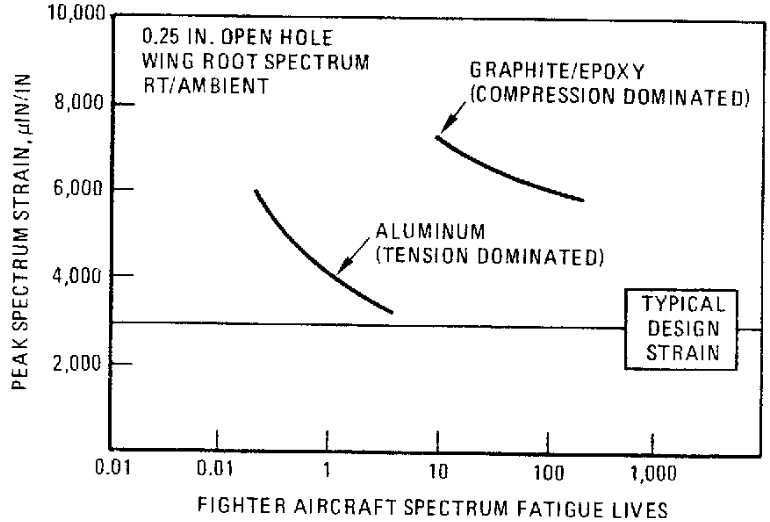



**Figure 3. Comparison of Fatigue Behavior of Metallic and Composite Materials**

A major consideration in the design of composite structures is the in-service impact damage. Impact damage occurs during ground handling, take-off and landing, and in-flight due to foreign objects. Hard objects (e.g. tool drops and runway debris) may cause impact damage and soft objects (e.g. bird impacts that occur at low altitude during take-off and landing). The impact damage caused by tool drops, etc. is termed as low velocity damage. Considerable reduction in compression strength may occur due to low velocity damage that is not visually detectable on the impacted or other external surfaces. The non-visual damage may cause internal damage in the form of delaminations between plies, matrix cracking, and fiber breakage. The longitudinal cross-section of an impact-damaged panel is shown in Figure 4. The damage due to impact is influenced by the factors such as laminate material properties, size of the laminate, support conditions, substructure, impactor size and shape, impactor velocity, impactor mass, impact location, and environment (Reference 5).

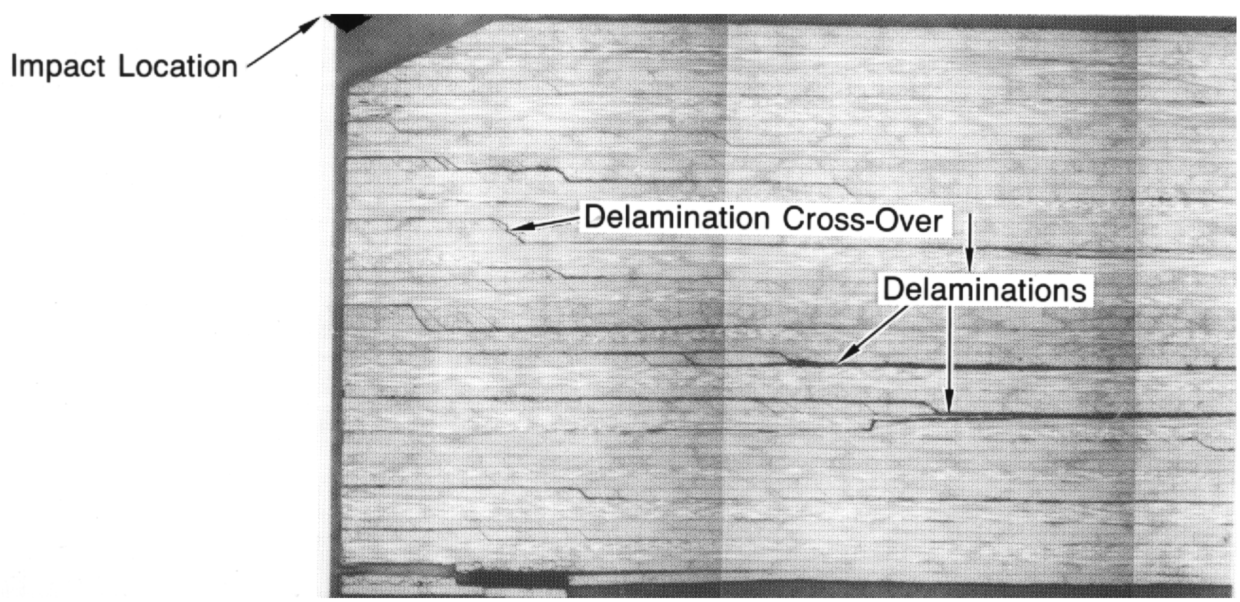



**Figure 4. Impact Damage in Composites**

Experimental data have shown (Figure 5) that impact damage can cause significant loss in strength. The degradation in compression strength is more severe than tension strength due to the delaminations between the plies caused by the impact damage (Reference 4).

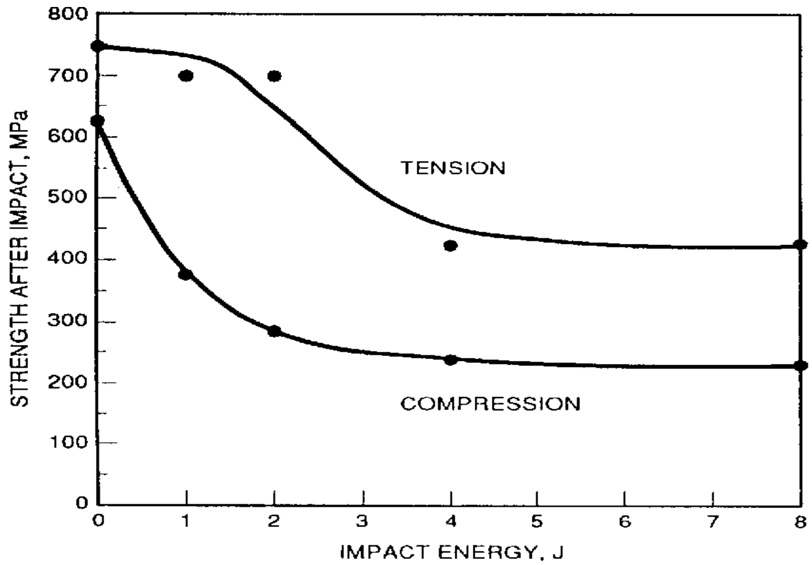



**Figure 5. Strength Degradation Caused by Impact Damage**

## 3. DAMAGE EVALUATION AND REPAIR CONCEPT SELECTION

The first step in designing any repairs is to evaluate the extent and nature of damage. Commonly occurring in-service damages in metallic and composite structures are shown in Figure 6. The overall process involved in damage evaluation and making repair decisions for a metallic and composite structure is outlined in Figure 7. Once the nature and extent of damage is found it is important to determine the effect of damage on structural integrity. If in a metallic structure, the damage found is a small crack that is much smaller than critical crack length, the repair may be performed by enlarging the hole to remove the crack and using an oversize fastener. In such cases, a revised damage tolerance analysis needs to be performed and new inspection requirements imposed for that location.

| **Metallic Structures** | **Composite Structures** |
|---|---|
| Fatigue Cracks | Delaminations |
| Corrosion | Impact Damage |
| Stress Corrosion | Foreign Object Damage |
| Foreign Object Damage | |

**Figure 6. In-service Damage Types in Metallic and Composite Structures**

The type of repair to be performed will be determined by the following factors-
1. Type of structural material to be repaired (metal, composite, sandwich construction)
2. Type of structural component to be repaired (skin, spar, rib, longeron, etc.)
3. Type and extent of damage (e.g. fatigue cracks, corrosion, impact damage, etc.)
4. Load levels and fatigue spectrum experienced by the structure
5. Material thickness to be repaired
6. Skill of the available labor
7. Availability of repair materials
8. Repair facility

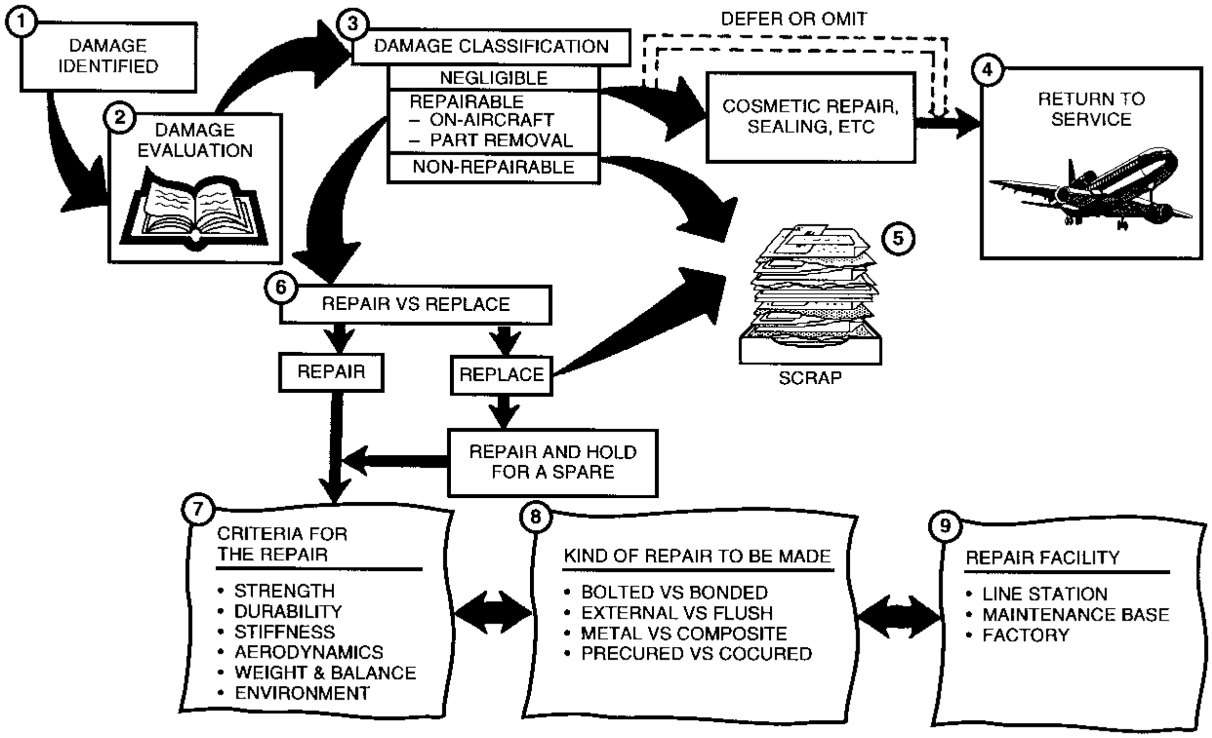



**Figure 7. Damage Evaluation and Selection of Repair Methods**

## 4. REPAIR OPTIONS

### 4.1 Repair of Composite Materials

Repairs of composite materials are similar to those for metallic materials if mechanically fastened repairs are to be used. However, the repairs of composite materials are different from those of metals if the repairs are to be bonded. The damage must be evaluated and classified. If the damage is repairable, a decision has to be made whether to repair or replace a part. If the structure is to be repaired, additional decisions have to be made regarding maintenance level, where work will be done, kind of repair materials, and repair configuration. The first step in the repair of composite materials is to remove the damage area including the delaminated area in the impacted region. The next step is to clean the surface to be repaired and apply a bolted or bonded patch. These repair concepts are discussed in the following paragraphs.

### BOLTED REPAIRS

Bolted repairs for composite structures are similar to those for metallic structures. The major differences between the repairs for composites and metals are:

a) Different tools are used for drilling fastener hole in composites.

b) Special care is needed in drilling holes in composites to prevent splintering on the exit side of the hole. A back support is desirable.

c) Matrix in composite is brittle compared to metal, hence the fasteners that expand to fill the hole (e.g. driven rivets) are not suitable for composites.

d) Sharing of loads in different fasteners in composites is not uniform because composite materials do not yield as metals where the load distribution tends to be more uniform.

Three commonly used bolted repair concepts are shown in Figure 8 and are discussed here.

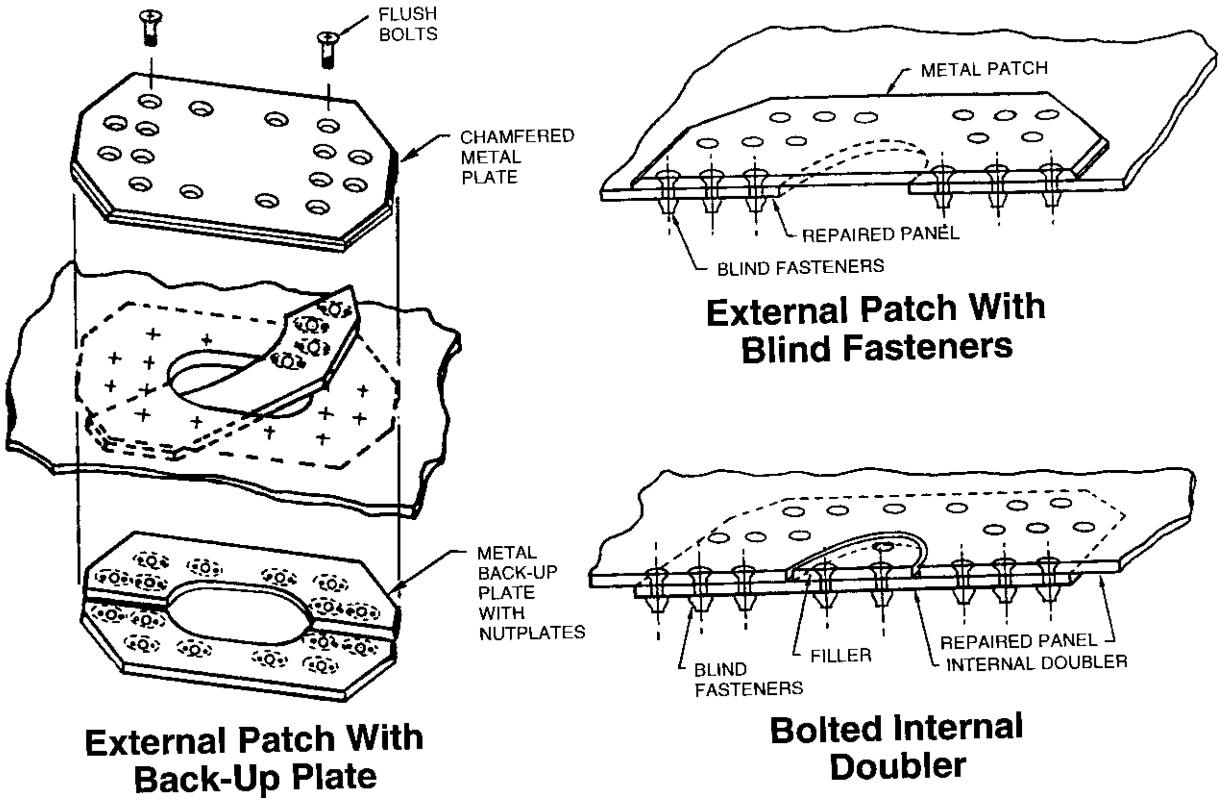



**Figure 8. Bolted Repair Concepts**

**External Patch with Backup Plate-** This concept uses an external chamfered metal patch bolted to the panel being repaired as shown in Figure 8. The bolts thread into nut plates mounted on metal backup plates that are on the side of the repaired panel. The backup plate can be split into two or more pieces and slipped through the opening as shown in the figure.

**External Patch with Blind Fasteners-**This concept is similar to the previous one, except that the backup plates are not used as shown in Figure 8. Blind fasteners are not as strong as bolts and nutplates, but if acceptable strength can be restored, this concept is easier to use.

**Bolted Internal Doubler-**This concept has been used as a standard repair for metal structures. Access to the backside is required to install the doubler as shown in Figure 8. The doubler cannot be installed through the hole as a separate piece because the doubler has to be continuous to carry loads in all directions. Filler is used to provide a flush outer surface, and is not designed to carry loads.

## BONDED REPAIR CONCEPTS

Bonded repair concepts can restore greater strength to a damaged composite structure as compared to bolted repairs. External repair patches are suitable for thin skins, however, for thick skins the eccentricity of the external patch reduces its strength. Flush patches are preferred for thick structures, heavily loaded structures, or where aerodynamic smoothness is required. Commonly used repair concepts are step-lap and scarf repairs.

**Step-Lap Repair-** This repair concept is shown in Figure 9. The steps allow the load to be transferred between specific plies of the patch and parent material. This advantage tends to increase the strength of the joint; however, it is offset by the peaks that exist in the adhesive shear stress at the end of each step.

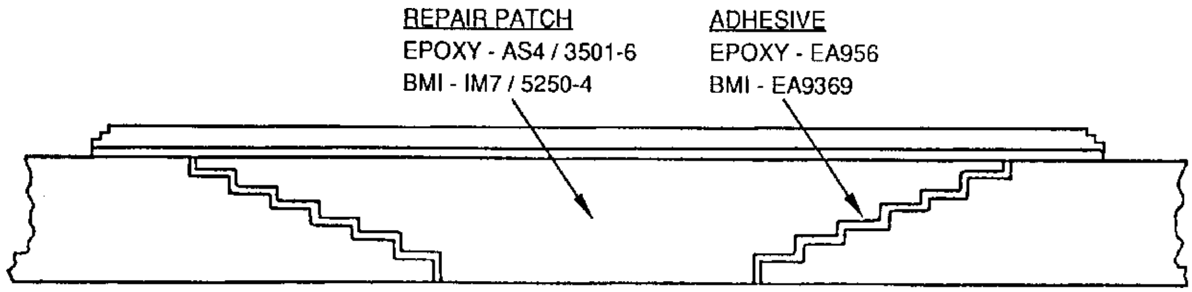



**Figure 9. Step-Lap Repair**

**Scarf Repair-** This repair concept is shown in Figure 10. The patch material is within the thickness to be repaired, with additional external plies added for strength. This configuration can restore more strength than an external patch as it avoids the eccentricity of the load path and provides smooth load transfer through gradually sloping scarf joint. A properly designed scarf joint can usually develop the full strength of an undamaged panel. The patch material is usually cured in place, and therefore must be supported during cure. While the patch material can be cured and then later bonded in place, it is generally difficult to get a good fit between the precured patch and the machined opening. In practice, well-made step-lap and scarf joints have approximately the same strength. A disadvantage of step-lap joints is the difficulty in machining the step to the depth of the exact ply that is desired on the surface of the step.

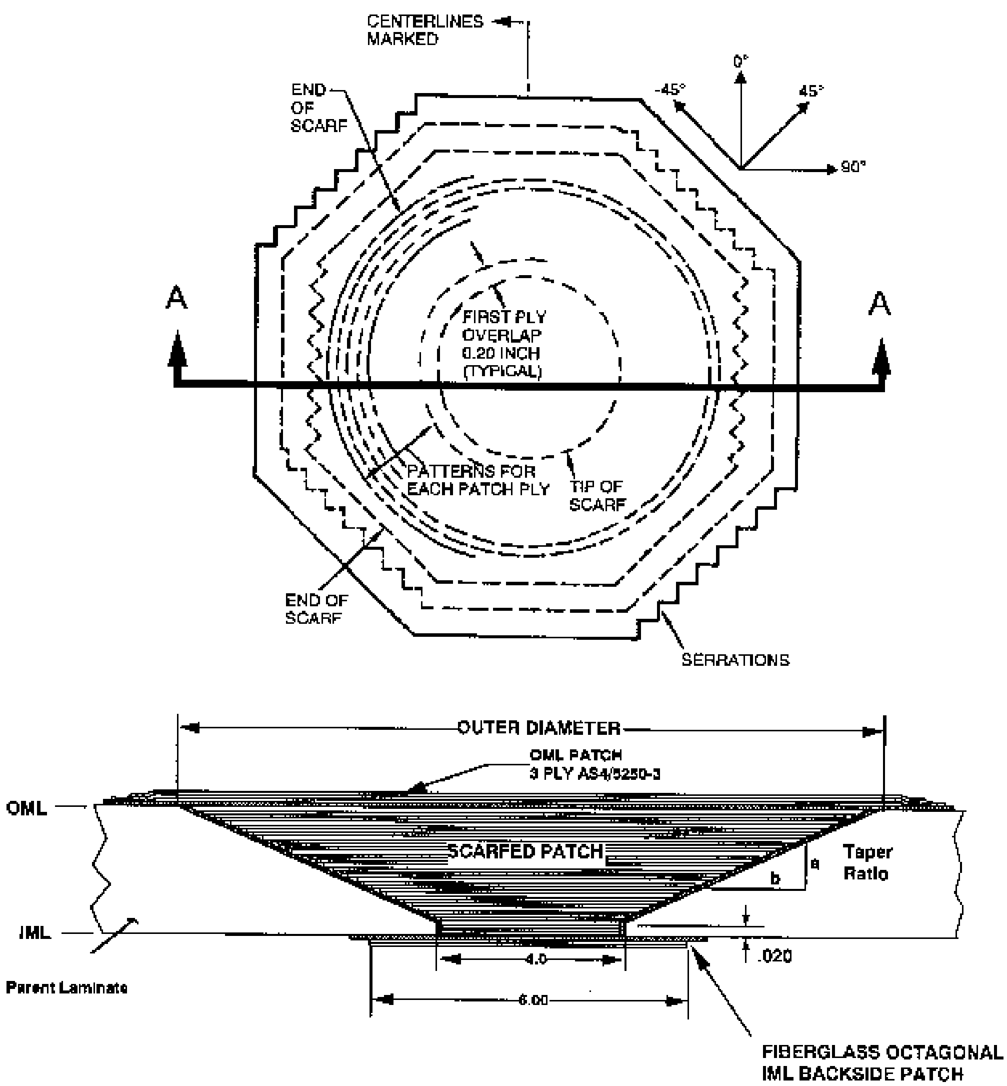



**Figure 10. Scarfed Repair**

**4.2 Repair of Sandwich Structures**

A typical in-service damage to a sandwich structure with composite face sheets is shown in Figure 11. The damage to composite face sheets is visible damage with surface indentation. Delaminations are seen in the composite face sheets as well as disbonding between the face sheets and honeycomb core. In addition, core buckling is seen.

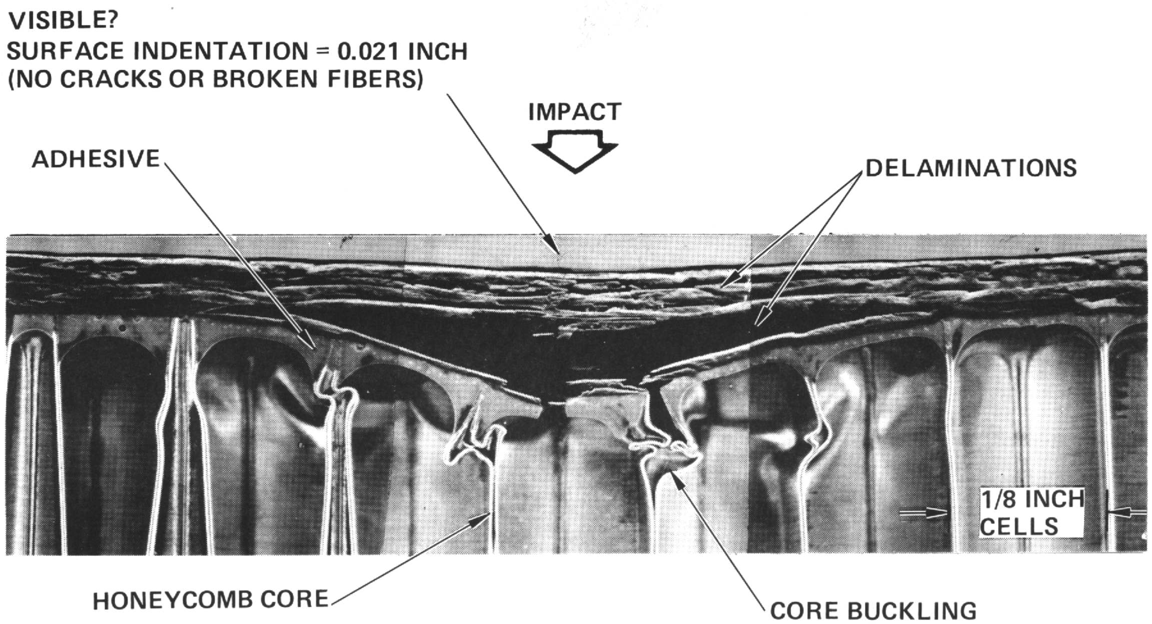



**Figure 11. Typical Impact Damage in Sandwich Structure with Composite Face Sheets**

The repair of a sandwich structure will depend on the extent of the core damage. Full depth and partial through the depth repair concepts are shown in Figure 12. The core damage has to be machined out and a plug prepared before performing the repairs. Various steps involved in the repair are illustrated in the figure.

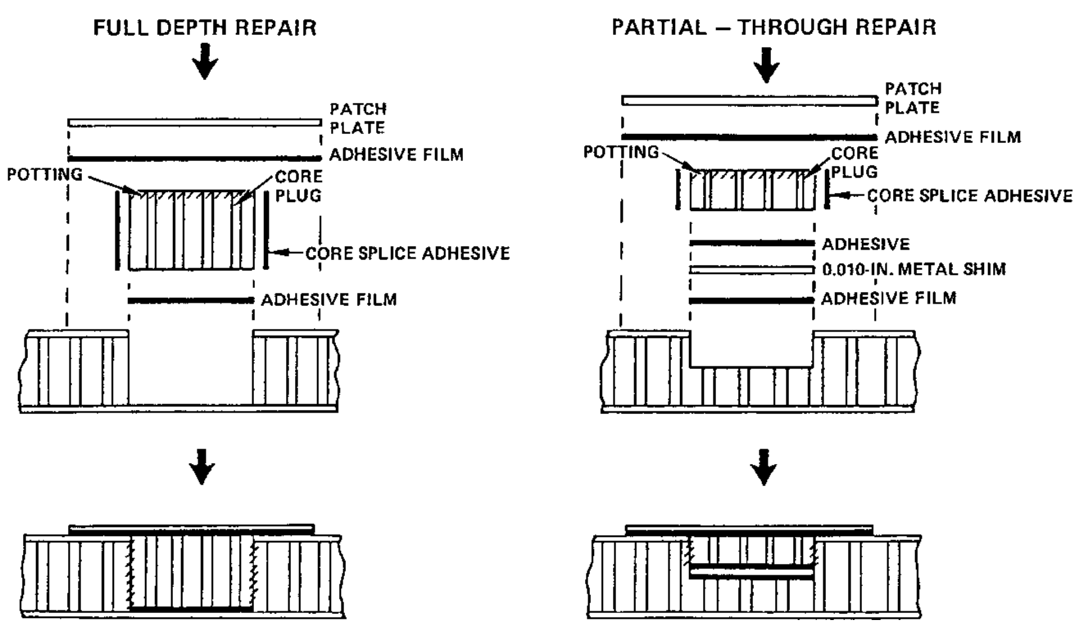



**Figure 12. Repair Concepts for Sandwich Structure with Composite Face Sheets**

**4.3 Repair of Metallic Structures**

**4.3.1 MECHANICALLY FASTENED REPAIRS OF METALLIC STRUCTURES**

Repair concepts for metallic structures are well established. The bolted repair concepts, discussed earlier for composites are applicable to metallic repairs. Standard repairs are generally given in repair manuals. However, in many cases in-service inspections show damages that are not covered by standard repair manuals and special repairs have to be designed. For such cases detailed static and damage tolerance analyses have to be carried out. An example of cracked frame in a transport aircraft (Figure 13) is shown in Figure 14. The flange and the web of the frame are cracked as shown in Figure 15a. Standard repair manuals generally do not cover a repair for the damage shown in Figure 14. The cross-sections of the flange and web repairs are shown in Figure 15b. The details of the frame repair are shown in Figure 16.

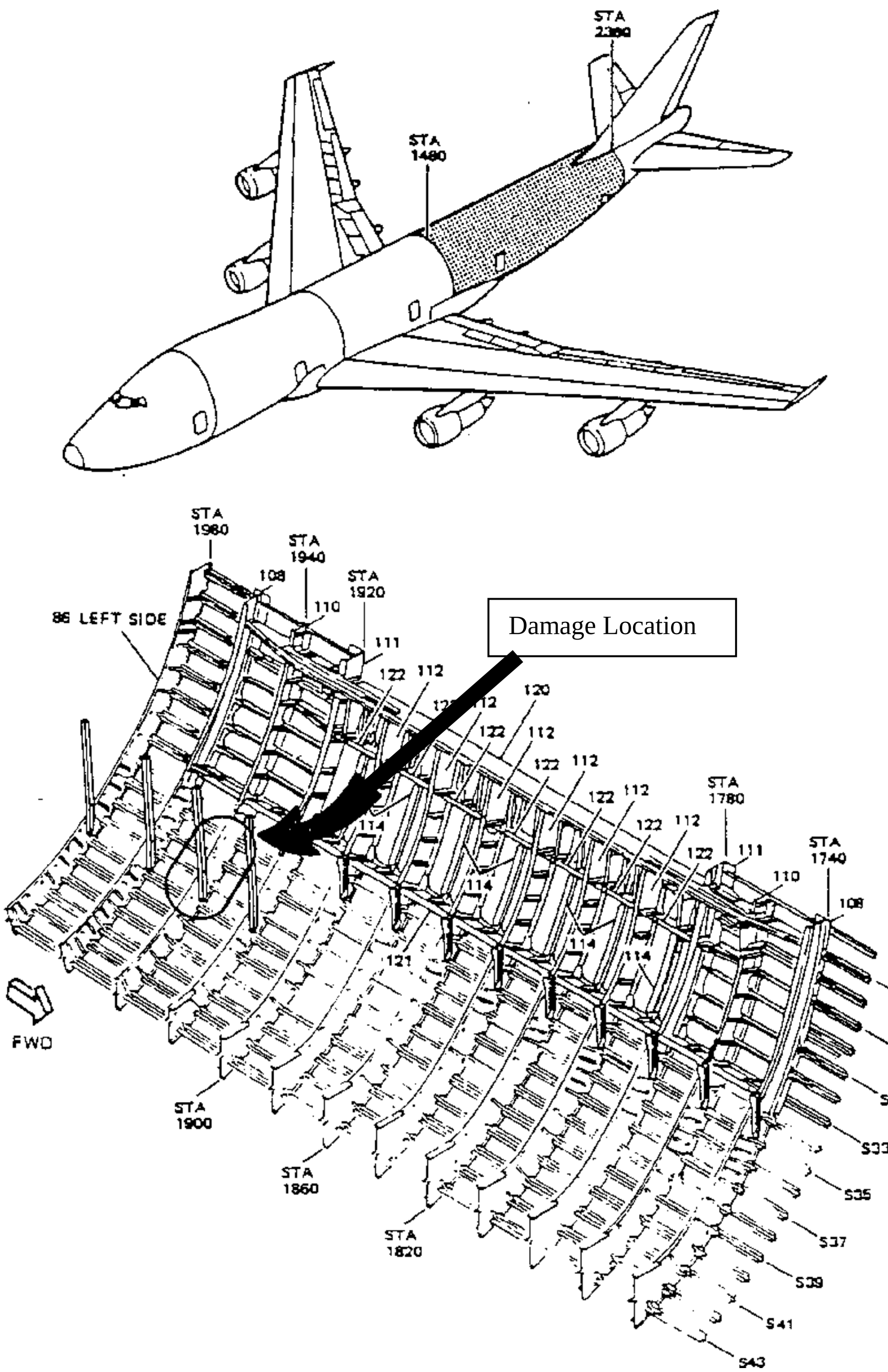



**Figure 13. Cracking Location in Transport Aircraft Fuselage**

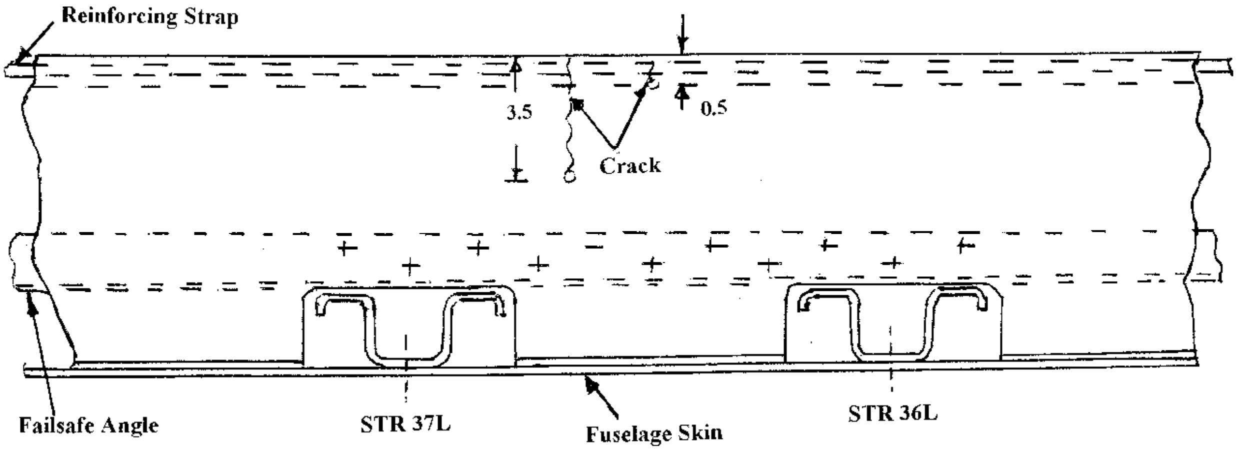
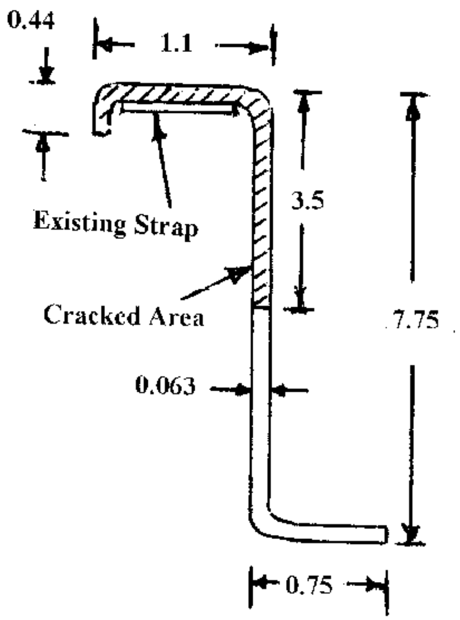



**Figure 14. Cracked Frame**



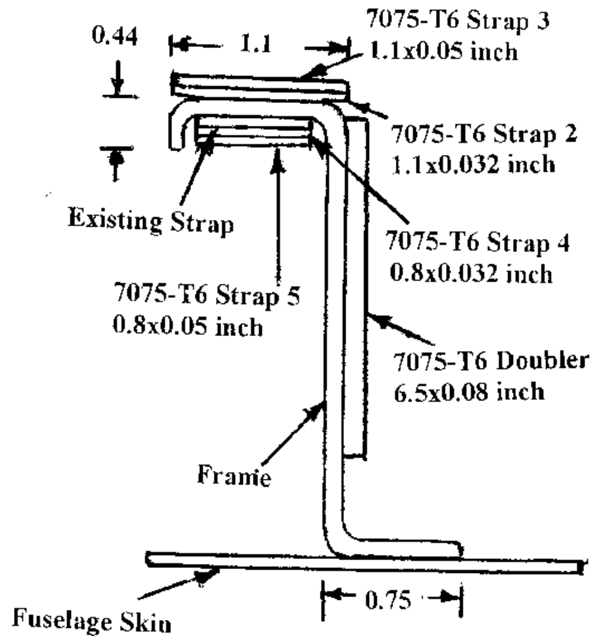




**Figure 15a. Cross-section of Cracked Frame** **Figure 15b. Cross-section Showing Flange and Web Repair**

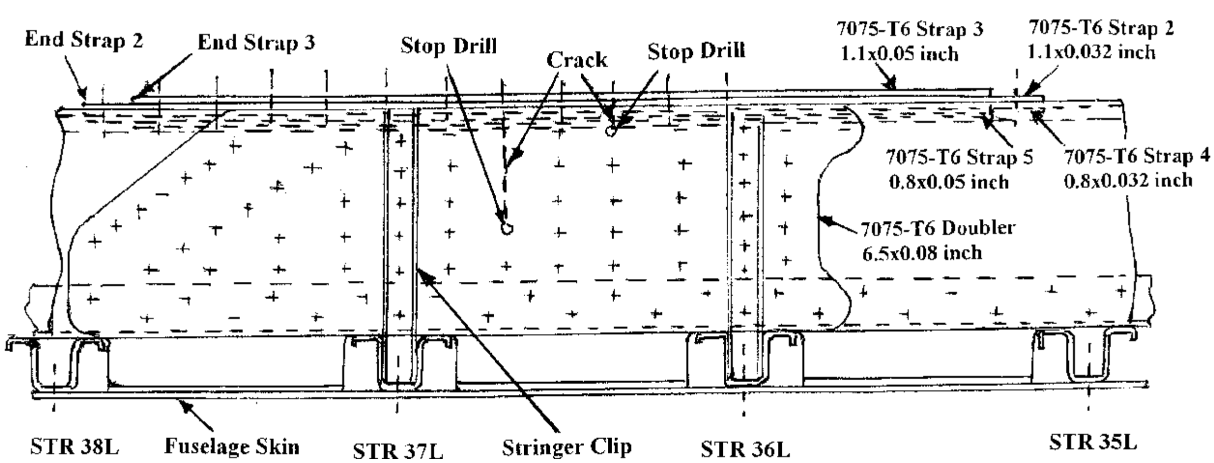



**Figure 16. Details of Frame Repair**

### 4.3.2 BONDED REPAIRS OF METALLIC STRUCTURES

The conventional mechanically fastened repair concept has disadvantages primarily due to the drilling of holes for additional fasteners that affect the structural integrity of the structure. In many cases the parts have to be scrapped due to the repaired structure not meeting the fail safety requirements. In most cases if the thinning due to corrosion is more than 10% of part thickness the parts are replaced. The development of bonded composite repair concept has provided excellent opportunities to design more efficient repairs (References 4, 6-13) and in many cases has made it possible to repair damaged structures which could not be repaired with the conventional mechanical fastening and were scrapped. Composite patch repairs also result in reduced inspection requirements compared to mechanically fastened repairs. In fact, in many cases the composite patch repairs can be designed such that the cracks in metallic structures underneath the repairs will not grow thereby eliminating inspection requirements, except those imposed by Integrated Logistics Supports (ILS) plan.

In bonded composite repair concept a composite patch is bonded to the damaged metallic part instead of a conventional mechanically fastened patch. Bonded composite repair has many advantages over conventional mechanically fastened repair, namely: 1) More efficient load transfer from a cracked part to the composite patch due to the load transfer through the entire bonded area instead of discrete points as in the case of mechanically fastened repairs, 2) No additional stress concentrations and crack initiation sites due to drilling of holes as in the case of mechanically fastened repairs, 3) High durability under cyclic loading, 4) High directional stiffness in loading direction resulting in thinner patches, and 5) Curved surfaces and complex geometries easily repairable by curing patches in place or prestaging patches. The cross-section of a typical 16-ply graphite/epoxy patch bonded to an aluminum sheet is shown in Figure 17.

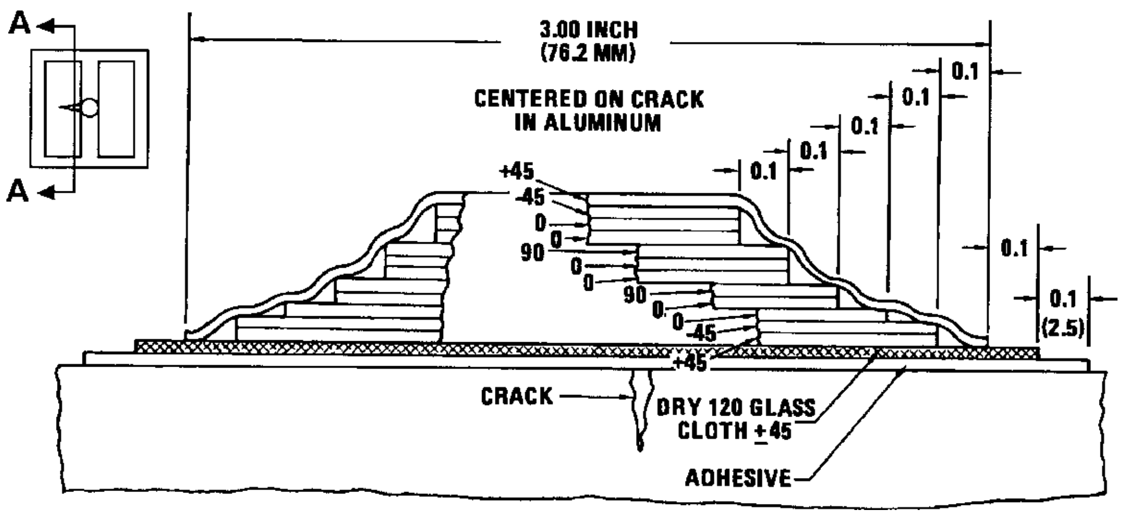



**Figure 17. Cross-section of a Typical Composite Repair Patch**

The critical parameters for this type of repair are 1) Surface preparation, 2) Adhesive material selection, 3) Composite repair material selection, and 4) Bonding operation.

#### Surface Preparation

Proper surface preparation is one of the most important considerations in bonded structures. The surface preparation process consists of paint removal, anodizing and priming. Liquid chemical paint strippers are not recommended, as they may become entrapped in cracked areas and faying surfaces of adjoining structures, thereby causing a corrosion problem. Aluminum oxide abrasive cloth has been found to be suitable for small repair areas.

Both silane and phosphoric acid non-tank anodize (PANTA) have been found to be suitable. The silane process has the advantage of being non-acid process. However, from the point of view of long term durability of repairs, the PANTA process may be desirable, as sufficient test data is available on this process.

Primer is applied to the aluminum surface after anodizing with PANTA to prevent contamination and improve long-term durability. BR-127 primer has been found to be suitable for FM-73 adhesive.

**Adhesive Material Selection**

Room temperature cure adhesives are not considered suitable due to service temperature requirements of 180F (82C) in the majority of aircraft repair applications. Also, room temperature cure adhesives are paste adhesives and generally do not result in uniform bond line thickness in the repair. Thus, affecting the load transfer to composite patch. Hence, high temperature film adhesives are preferred. Also, long term durability of room temperature adhesives is not well characterized. A 350F (177C) cure film adhesive is not considered desirable, as the curing at such a high temperature is likely to cause undesirable high thermal stresses. Also, an aluminum structure exposed to a 350F (177C) temperature will undergo degradation in mechanical properties. A 250F (121C) cure adhesive system is considered suitable for the composite patch repair of aluminum structure. Ductile adhesives such as FM-73 are preferred over brittle adhesives such as FM-400 due to the tendency of the brittle adhesives to disbond around the damage area, thereby reducing the load transfer to the repair patch.

**Composite Repair Material Selection**

Both boron/epoxy and graphite/epoxy composites are suitable for the repairs. The choice between boron or graphite fibers should be based on availability, handling, processing and the thickness of the material to be repaired. Boron has higher modulus than graphite and would result in thin repair patches. Thin patches are more efficient in taking load from damaged parts as compared to thick patches. For repairing relatively thick parts, boron may be preferred over graphite. It is considered desirable to use highly orthotropic patches, having high stiffness in the direction normal to the crack, but with some fibers in directions at 45 and 90 degrees to the primary direction to prevent matrix cracking under biaxial loading and inplane shear loads which exist for typical applications. This patch configuration can be best obtained with unidirectional tape. Woven material has greater formability and could also be used, although it would not make a very efficient patch.

The composite patches may be precured, prestaged or cured in place. For locations where vacuum bagging represents a problem, a precured patch may be prepared in an autoclave and then secondary bonded to the repair area. For relatively minor contours, a prestaged patch may be used. For curved surfaces the patch may be cured in place during the bonding operation.

**Bonding Operation**

Bonding of repair patches requires a proper temperature control within +10F and -5F in the repair area. Thermal blankets are available to provide temperature in excess of 1000F (538C). A proper temperature control within tolerances is necessary for bondline to achieve desirable strength. A large aircraft structure compared to a small repair area may act as a heat sink and jeopardize maintaining desired temperature control for the required duration. Proper heat blankets for surrounding areas may be required for such cases.

**Crack Growth Life Enhancement with Bonded Composite Repairs**

The crack growth data obtained from a repaired center-crack panel (7075-T6 aluminum, 0.063-inch (1.6-mm) thickness) are shown in Figure 18. It is seen that starting with the same initial crack length, the panel without a repair patch fails after about 870 missions (0.92 lifetime) at a crack length of 1.36-inch (34.6-mm). The panel with the repair patch did not fail even after 2350 missions (2.5 life times) at a crack length of 1.93 inches (49 mm). Thus, a considerable extension in life was obtained with the composite repair patch.

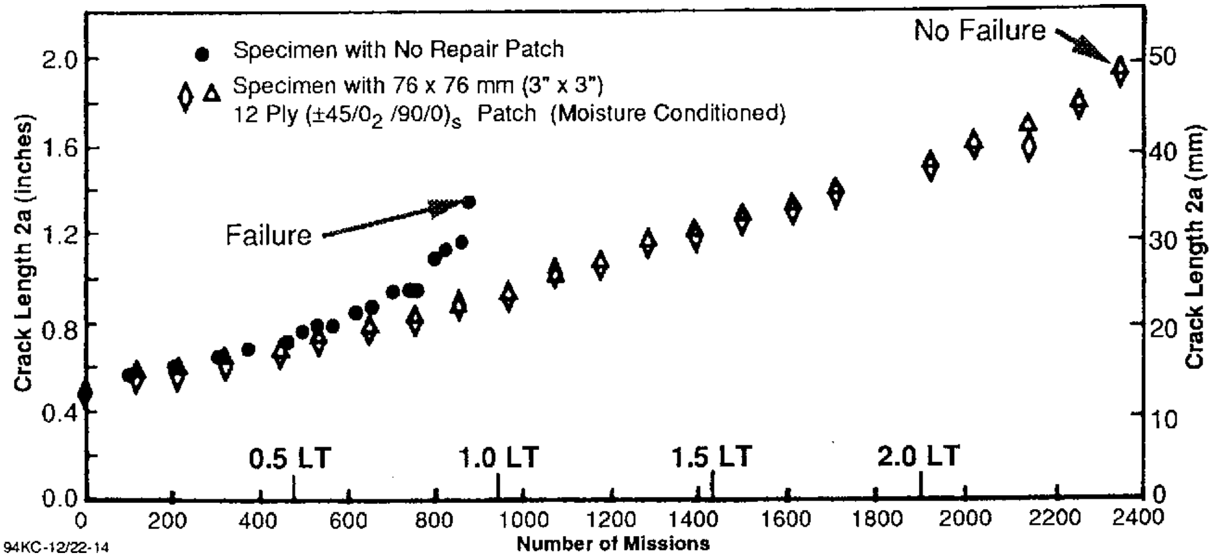



**Figure 18. Comparison of Crack Growth in Specimen With and Without Repair Patch**

**Comparison of Analytical and Experimental Results**

The crack growth behavior of the cracked panel with a composite patch was predicted using analytical stress intensity factors (Ref. 14-15) for the patched structure and the crack growth data, obtained on an unpatched center crack specimen. Comparison of observed and predicted fatigue crack growth behavior in a 7075-T6 aluminum 0.063 inch (1.6 mm) thickness repaired with a 3 inch (76 mm) square 12 ply graphite/epoxy patch, moisture conditioned to one percent moisture, is shown in Figure 19. It is seen that the correlation between predicted and observed crack growth is excellent. The specimen did not fail even after two life times of spectrum loading.

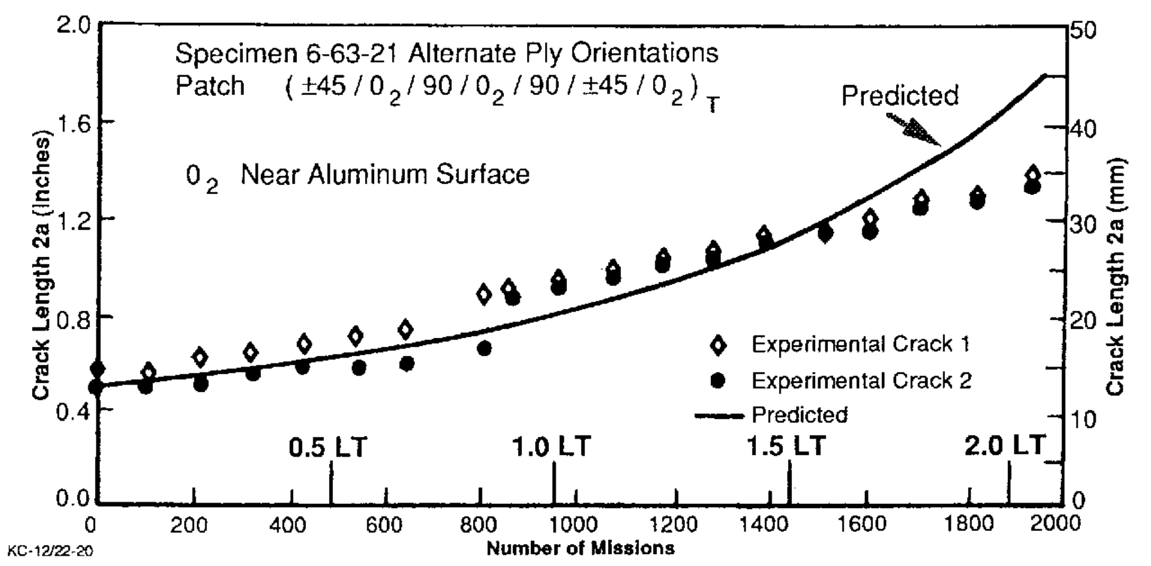



**Figure 19. Comparison of Observed and Predicted Crack Growth**

**Repair Design for No Damage Growth**

It is possible to design composite repair patches so that the damage in the repaired structure will not grow. Of course, the feasibility of such a design depends on the stress level, the type of material to be repaired, material thickness, the crack length to be repaired, and spectrum. In the majority of transport aircraft where design stress levels are relatively low, it is possible to design repairs such that the damage does not grow. This is particularly true for fuselage structures where material is predominantly 2024-T3 aluminum and gauge thicknesses are small. Crack growth behavior in 2024-T3 material 0.032-inch (0.8-mm) thick specimen, repaired with 12-ply Gr/Ep patch is shown in Figure 20. No crack growth in two lifetimes of spectrum loading is seen. Thus, the repairs can be designed for no damage growth and there by eliminating inspection requirements.

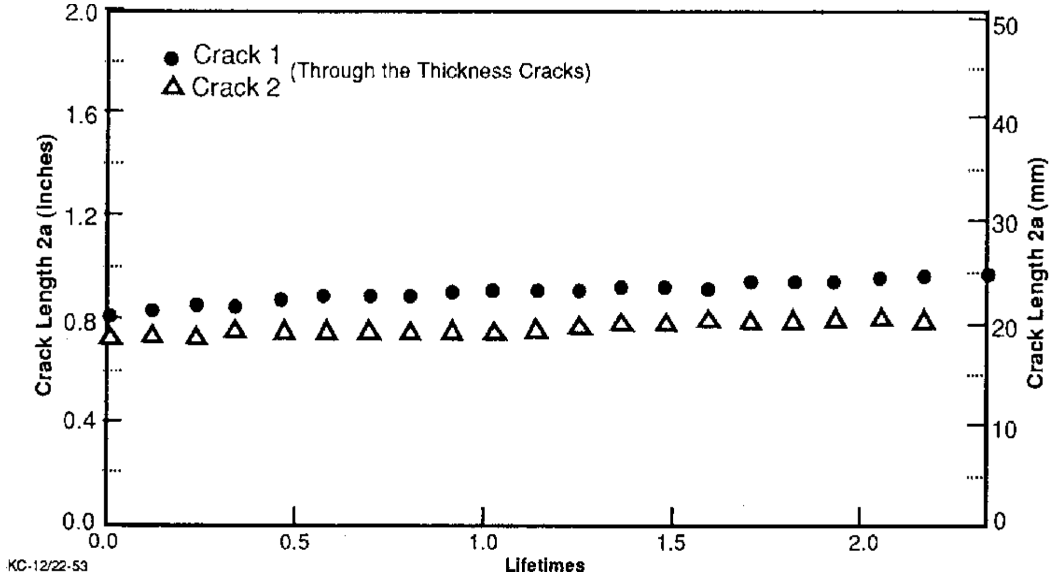



**Figure 20. Crack Growth in 2024-T3 Aluminum, 0.032 inch (0.8 mm) Thick With 12-Ply Gr/Ep Patch**

### 4.3.3 IN-SERVICE APPLICATIONS OF COMPOSITE PATCH REPAIRS

Applications of composite patch repair to in-service aircraft are found in T-38 lower wing skin (References 16-19), C-141 weep holes (Reference 20) and F-16 fuel access hole (Reference 21). T-38 lower wing skin has developed in-service cracking problems at "D" panel attachment holes and at machined pockets between 39% and 44 % spars and 33% and 39% spars as shown in Figure 21. Composite patch repair concepts were developed for these locations.

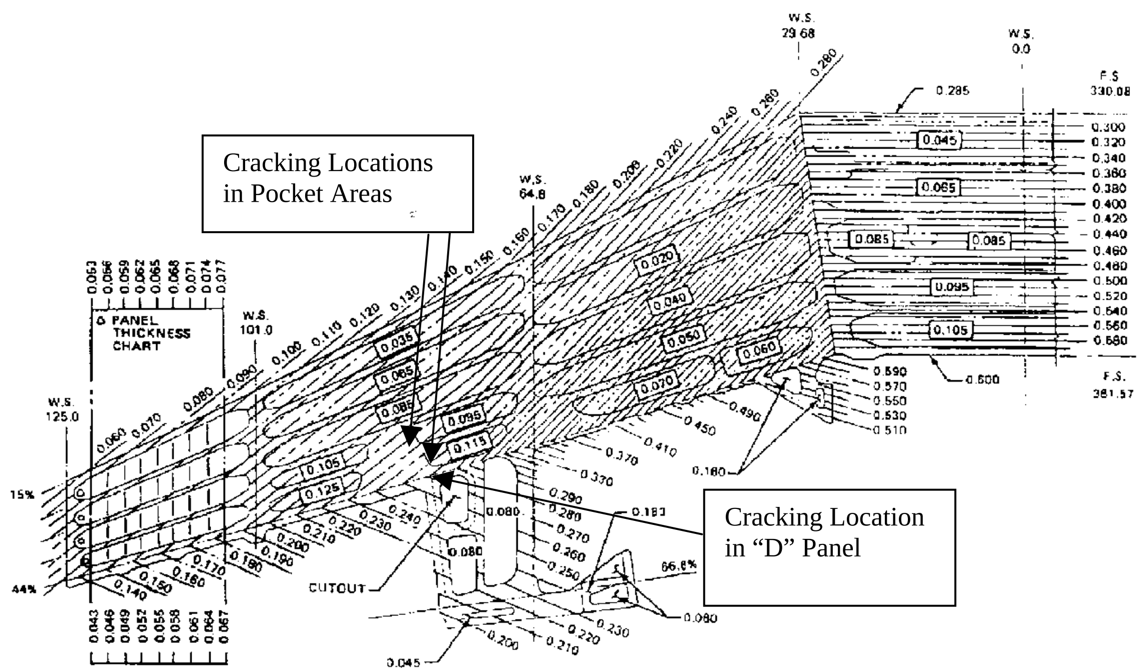



**Figure 21. Cracking Location in T-38 Lower Wing Skin**

Conventional mechanically fastened repair concepts at the location of “D” panel are not possible due to the limited space available for drilling the fastener holes. Bonding of an aluminum doubler will provide only limited doubler stiffness and will not result in an efficient repair. A bonded boron repair is ideal for this location. An external boron patch could not be applied as the door has to fit in the area and has to be flush with the outer mold line. Hence, an internal repair patch was designed as shown in Figure 22. A pre-cured boron repair patch was secondary bonded through the ‘D’ panel door.

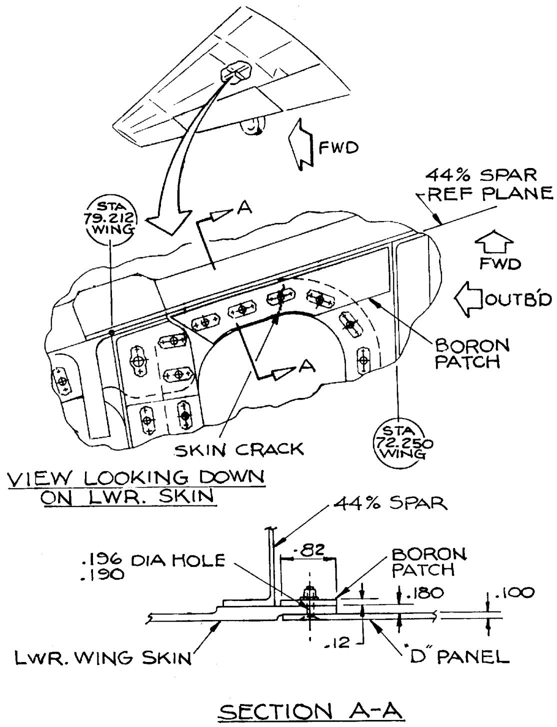



**Figure 22. T-38 Lower Wing Skin Composite Patch Repair**

Lower wing skin pockets in T-38 aircraft between the 39% and 44% spars and 33% and 39% spars at Wing Station (WS) 78 have shown a propensity for crack initiation and propagation during service. The cracks have initiated at the pocket radius in the inner moldline of the wing skin. This cracking has been occurring primarily under Lead-in-Fighter (LIF) spectrum loading. These areas are ideal for composite reinforcement to reduce stress levels and enhance fatigue life. As there is no access for bonding reinforcement on the inner moldline, a one sided reinforcement bonded onto the outer moldline of the wing skin was selected. Due to the complexity of the structure in the area, it was considered necessary to verify the reinforcement design by structural testing. The test program was devised in two parts. In the first part of the test program, testing was performed on specimens that simulate the configuration and load environment in the pocket areas of the wing. The results of this study are reported in Reference 18. The second part of the test program involved bonding of the reinforcement to a T-38 wing (Figure 23) subjected to durability testing at Wright Patterson Air Force Base (WPAFB), Ohio, as a part of Air Force Contract F33615-90-C-3201, entitled “Advanced Technology Redesign of Highly Loaded Structures (ATROHS)”. The wing with composite reinforcement has undergone 3,500 hours of testing under LIF spectrum loading (Reference 17).

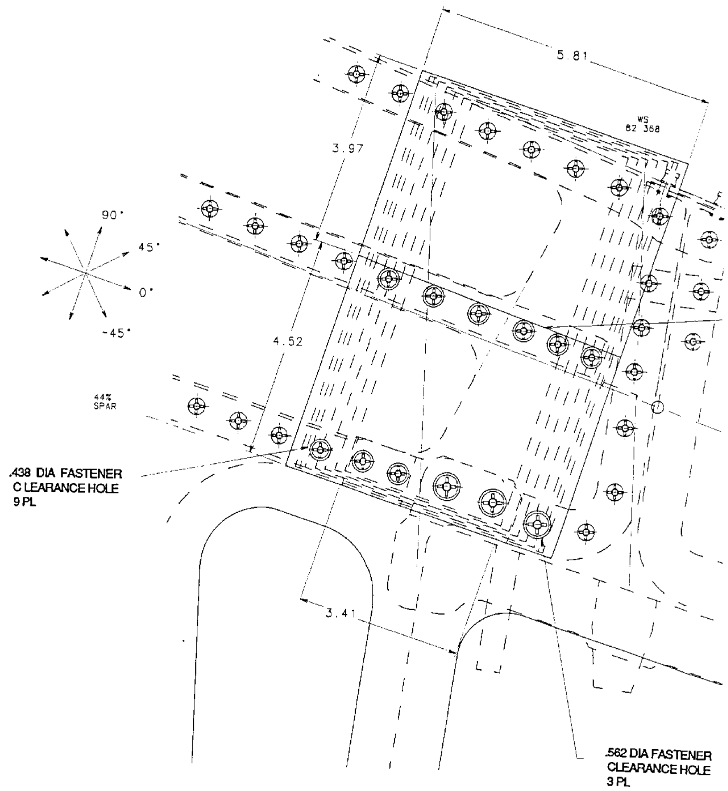



**Figure 23. Composite Reinforcement in Lower Wing Skin Pocket Areas**

Vacuum-bagged composite reinforcement assembly on T-38 test wing is shown in Figure 24 and bonded reinforcement assembly is shown in Figure 25.

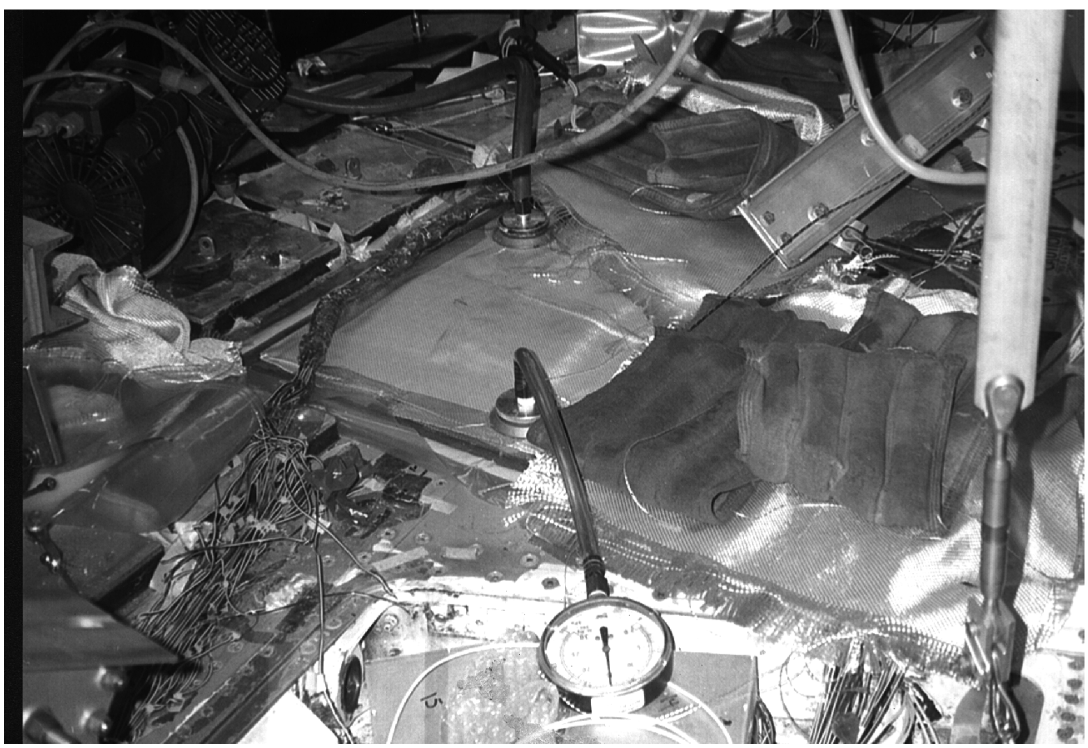


**Figure 24. Vacuum Bagged Reinforcement Assembly on T-38 Test Wing**

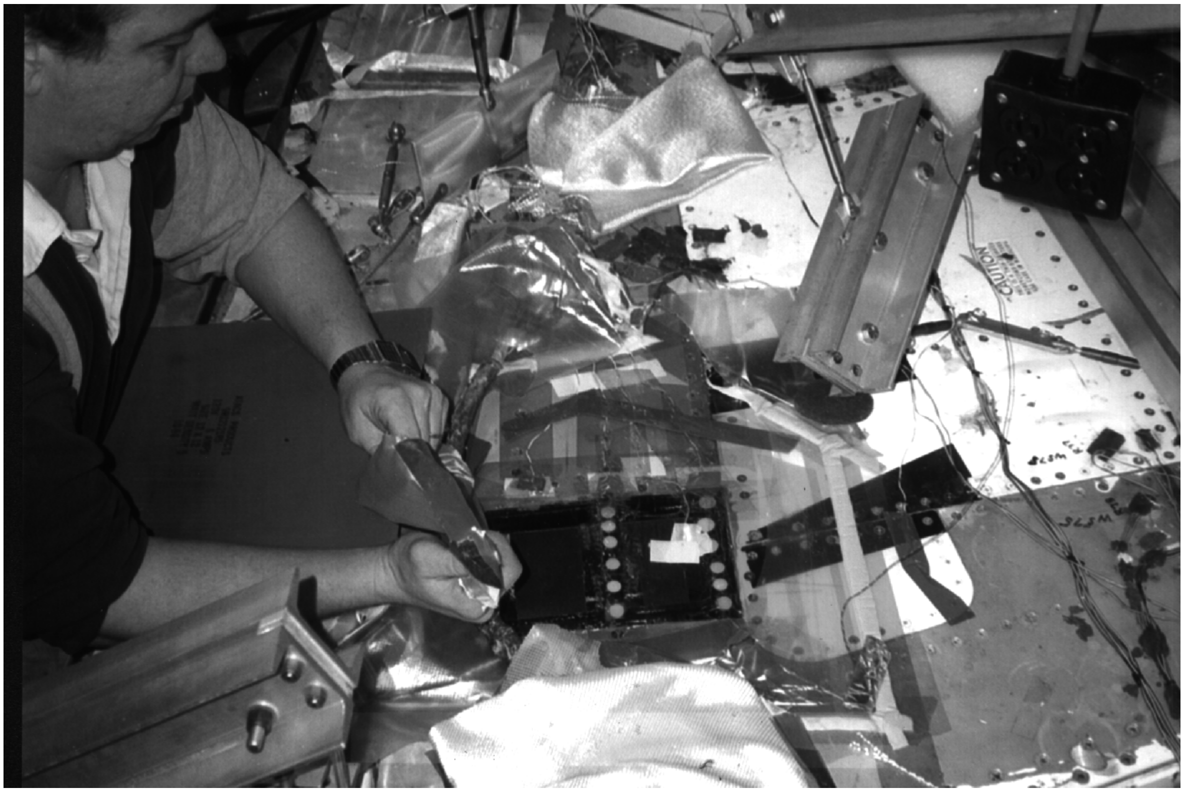


**Figure 25. Bonded Reinforcement Assembly**

Composite patch repair application to C-141 lower wing skin at weep holes is shown in Figure 26. Figure 27 shows composite reinforcement application to lower wing skin splice area.

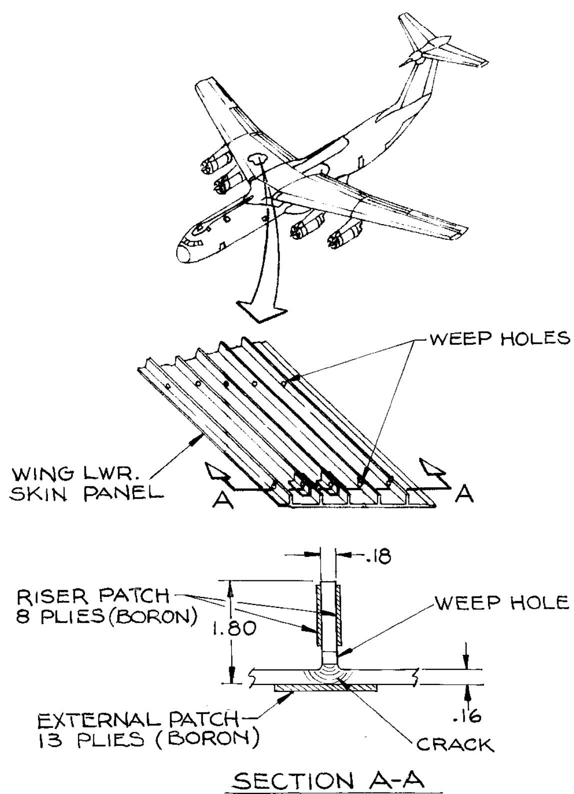



**Figure 26. C-141 Composite Patch Repair at Weep Holes**

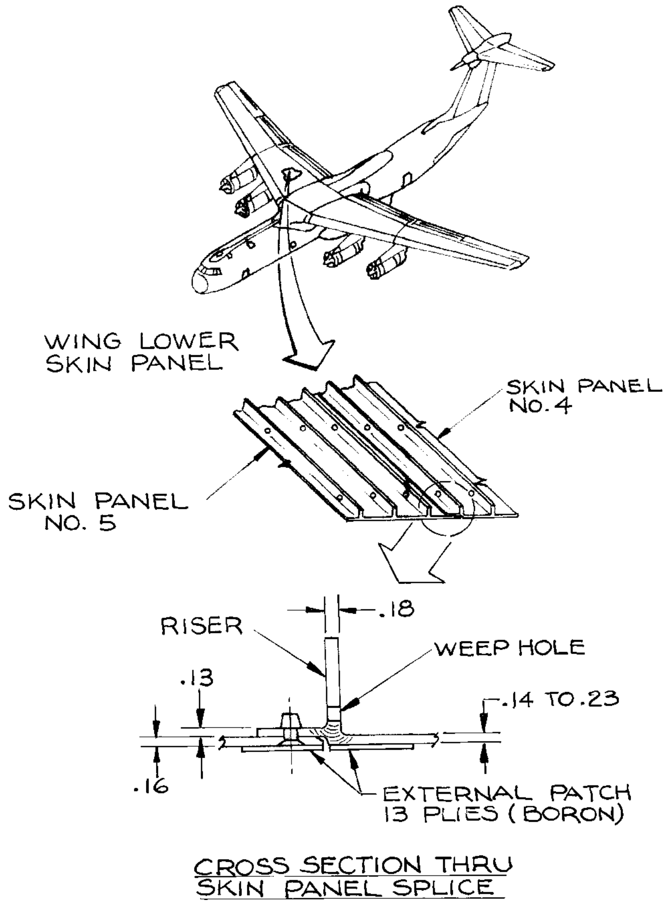



**Figure 27. Composite Patch Repair at C-141 Lower Wing Skin Splice**

## 5. SOFTWARE FOR REPAIR DESIGN AND ANALYSES

A number of software programs have been developed for designing repairs for aircraft structures. Some of these programs are briefly described here.

1. RAPID- This program has been developed under FAA and US Air Force sponsorship and is primarily for mechanically fastened repairs of transport aircraft. The program has capability to perform analysis under spectrum loading.
2. RAPIDC- This program has been developed under FAA sponsorship and is primarily for mechanically fastened repairs of commuter aircraft.
3. AFGROW- This is US Air Force developed code for durability and damage tolerance analyses of aircraft structures under spectrum loading. This code has capability to design composite patch repairs.
4. CalcuRep- This code has been developed by Dr. Rob Fredell during his stay at US Air Force Academy in Colorado. This code is for designing bonded repairs, using GLARE, for fuselage type of structures.
5. FRANC2D- This is a finite element code and can be used for damage tolerance analysis and composite patch repair design under constant amplitude loading.
6. COMPACT3D- This is a finite element code for designing composite patch repairs under constant amplitude loading.
7. NASGRO- This program has been developed by NASA Johnson Space Center and is available in public domain. The program is primarily for damage tolerance analyses.

## 6.0 CONCLUDING REMARKS

The life enhancement technologies have provided excellent opportunities to fulfill aging aircraft needs such as:
1) Reduced life cycle costs
2) Reduced/eliminated repairs
3) Reduced/eliminated inspections
4) Simplified maintenance

5) Reduced support requirements
6) Fulfilled severe usage requirements
7) Extended airframe life
8) Improved payload

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# Risk Assessments of Aging Aircraft

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## Summary

The USAF believes the damage tolerance approach incorporated in ASIP process in the seventies is still the cornerstone for protecting the safety of our aging aircraft. This process is primarily deterministic in that the calculations do not quantify the reliability of the process. As indicated above, however, the reliability achieved is consistent with the new aircraft guidance identified in USAF structural specification. The USAF derives the Force Structural Maintenance Plan (FSMP) from the damage tolerance assessment (DTA). The FSMP prescribes for the maintainer how, when, and where to perform inspections to maintain safety of flight. There are cases, however, where probabilistic methods need to be used. It is the purpose of this paper to illustrate the use of probabilistic methods to ensure structural integrity.

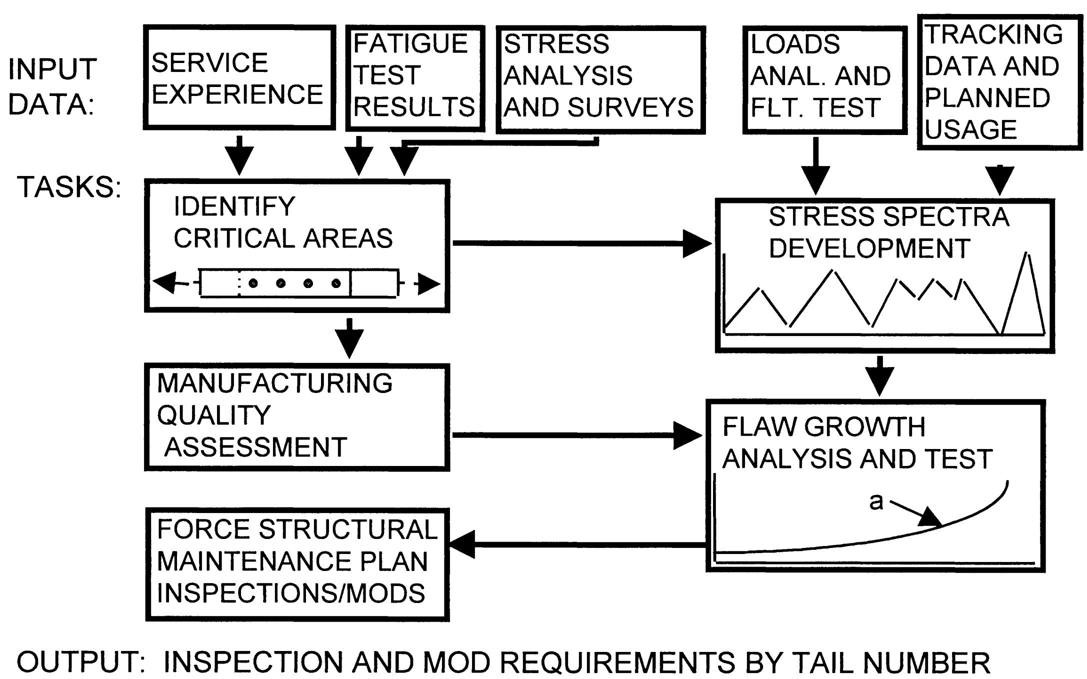


**Figure 1 Damage tolerance approach**

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**Role of Probabilistic Methods**

In the seventies and eighties there was considerable activity associated with the performance of DTAs on older aircraft. The USAF sometimes found the DTA revealed critical locations that were over the safety limit. In these cases, the USAF policy is to ground (or severely restrict) these airplanes until they made an inspection. In some of these cases the inspection was so onerous they grounded the aircraft for a long time thus hindering training operations. Such a case occurred on the F-5 dorsal longeron. The inspection required approximately 1350 work hours on each aircraft. The USAF decided to modify the structure to eliminate this inspection burden. To determine the feasibility of continued use of the aircraft before the modification the USAF performed a risk assessment based on the method described in [3]. This method considers the crack length distribution and the stress distribution as random number sets. The procedure further assumes the crack growth and the residual stress functions are deterministic.

Another opportunity for a risk assessment arose when the USAF needed to keep the T-37 in operational service after the cancellation of the T-46 program. The USAF subjected the T-37 to a damage tolerance assessment. They found several areas, in particular, the wing to fuselage attachment area, where the flight hours on the aircraft exceeded the safety limit. The USAF performed an extensive risk assessment to allow these airplanes to continue in their training role until they could modify them.

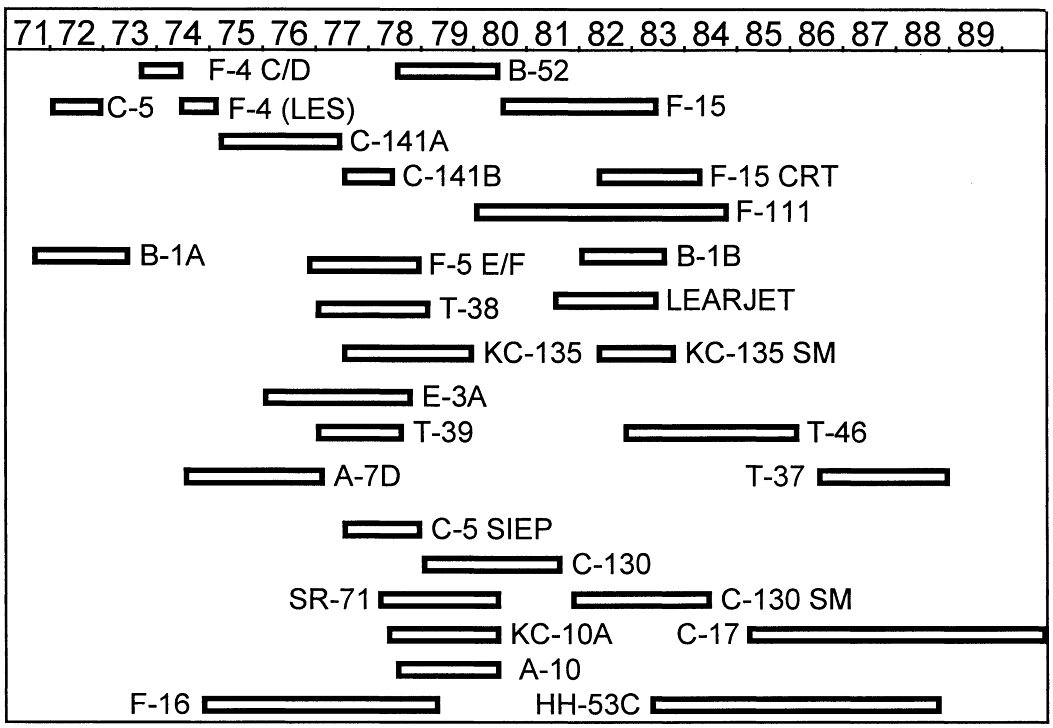


**Figure 2 Damage tolerance experience - aircraft**

In all of these cases such as those cited above, the risk assessment did not include the possibility of a “rogue defect” (as assumed in the damage tolerance assessment). Rather, they derived the flaw distribution from extrapolation of defects found in typical structural details. Therefore, in these cases the structural engineers made it clear to the aircraft operators they had not accounted for the rogue defect.

Another problem where the risk assessment is valuable is in the case where the structure is in a state of generalized cracking. In this situation the inspection intervals derived as indicated above from the deterministic DTA may be unconservative.

The USAF had an opportunity to address such a situation for the wings of the T-38 aircraft operating in the Air Training Command. In the mid-seventies, the USAF performed a damage tolerance assessment for the trainer discussed above in Air Training Command usage [3]. This study concluded they should inspect the wing center section at intervals of 1350 flight hours. They based this on an inspection capability for a corner crack of 2.54 mm and an inspection at one half of the safety limit. This was the time required to grow a crack of 2.54 mm to a critical size crack length of 5.5 mm. In the late seventies, a usage change took place that made the loading environment more severe. The USAF made a damage tolerance reassessment for this new usage. They found under the same ground rules the recurring inspection interval should be 430 hours.

To provide an evaluation of the necessity of performing inspections at an increased rate, they performed a risk assessment for the new usage, but old inspection schedule. The assessment based on an inspection interval of 1350 flight hours showed the risk was unacceptable. When they reduced the inspection interval to 300 hours, they found an acceptably low probability of failure. Therefore, they concluded they had to improve the inspection reliability or decrease the inspection interval from that derived from the deterministic DTA.

There are other cases where probabilistic methods can complement the DTA. These cases typically involve difficulty with the performance of the DTA. One can find examples of this in the assessment of mechanical subsystems. Many of these parts are not tolerant to the size defects assumed for airframe structure. Also, the loading environments are difficult to simulate analytically. One finds another example in the high strength steel structures such as gears. In these cases some of the classical reliability approaches may be useful. As indicated in [4], W. Weibull from Sweden performed a number of fatigue experiments in the middle of the fifties. He found the results of these experiments conformed to a probability distribution, known today as the Weibull distribution.

Figure (3) shows Weibull distributions that cover the range normally found in the fatigue of aircraft structural components. One notes the coefficients of variation (the standard deviation divided by the mean) of these distributions are typically much higher than found for static strength. Figure (4) shows the reliability with 95 percent confidence as dependent on the number of test lifetimes. The results shown are for several Weibull shape numbers. This calculation assumed there were no more than two like features in the aircraft. One sees a high reliability structure is difficult to achieve when the Weibull shape number is of the order of two or three.

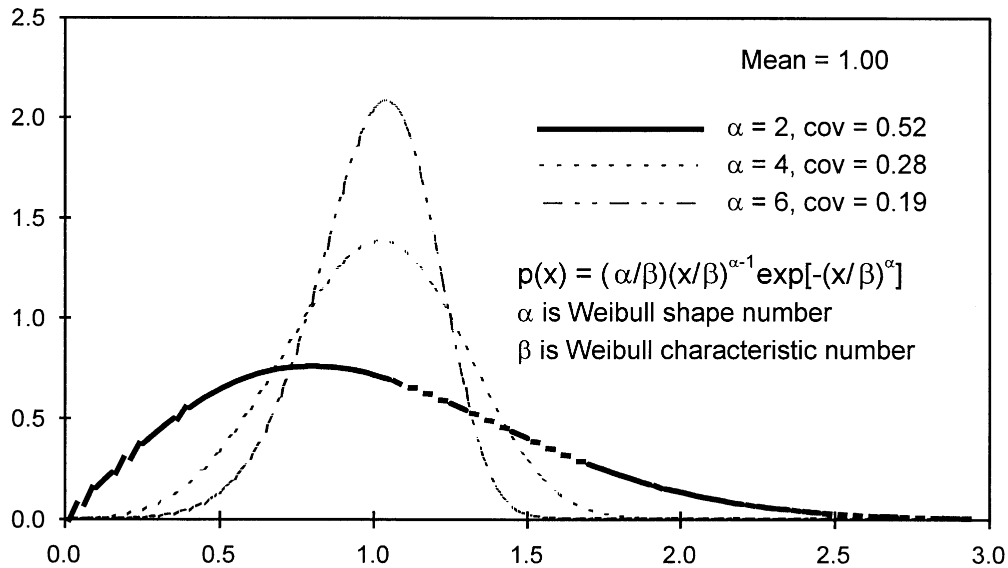


**Figure 3 Weibull probability density functions**

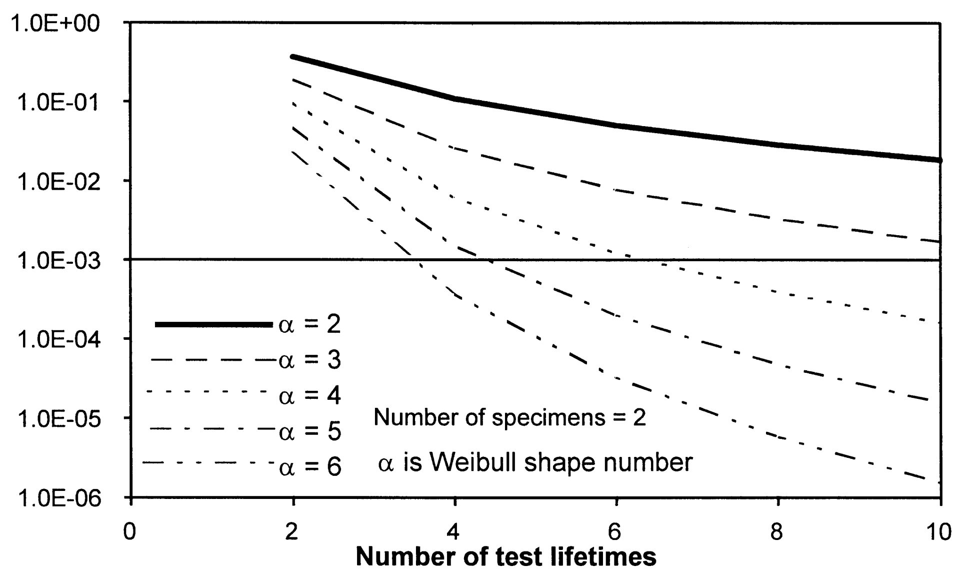


**Figure 4 (1 - reliability) with 95% confidence**

One of the problems associated with the early applications of the safe life approach was that it did not account for the fatigue characteristics of the individual materials in the structure. Therefore, the USAF used the same scatter factor independent of the structural material or the stress spectra. The structural analyst knows today there are considerable differences between the Weibull scale numbers depending on material and spectra.

The currently acceptable structural reliability as reflected in [2] is for a single flight of an aircraft from a given population the probably of failure should be no greater than $10^{-7}$. This means the desired reliability of the structure should be of the order of 0.999.

Typically, one determines the Weibull shape number through testing of multiple similar parts. An analytical example serves to illustrate how accurately one could determine the

Weibull shape number. For this purpose, one may sample a population with a known Weibull shape number. In the first case under consideration the Weibull shape number is two and the analyst selected ten random samples to simulate the testing of ten specimens. A simple transformation permits plotting these ten sample points on a graph where the Weibull distribution is a straight line. Further, on this graph the negative of the slope of that line is the Weibull shape number. Figure (5) shows the comparison of the original distribution and the sampled distribution. Figure (6) shows these distributions in the usual manner. One may use the same process to sample a distribution where the Weibull shape number is four. Figure (7) and Figure (8) show these results. One sees for small samples such as used here, the potential for error in the assessment of the Weibull shape number may be significant. In these cases the judgment on the reliability of the structural component could be in considerable error. However, if one adequately interrogates the population the results are useful. Because of the difficulties cited, the USAF recommends the application be limited to structures that are fail-safe.

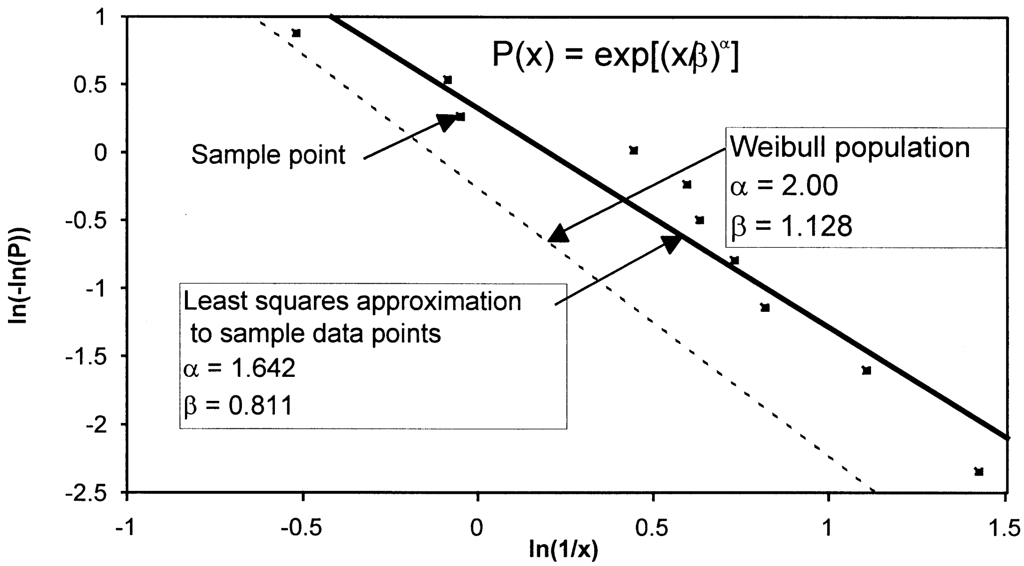


**Figure 5 Weibull approximation for α = 2 and 10 samples**

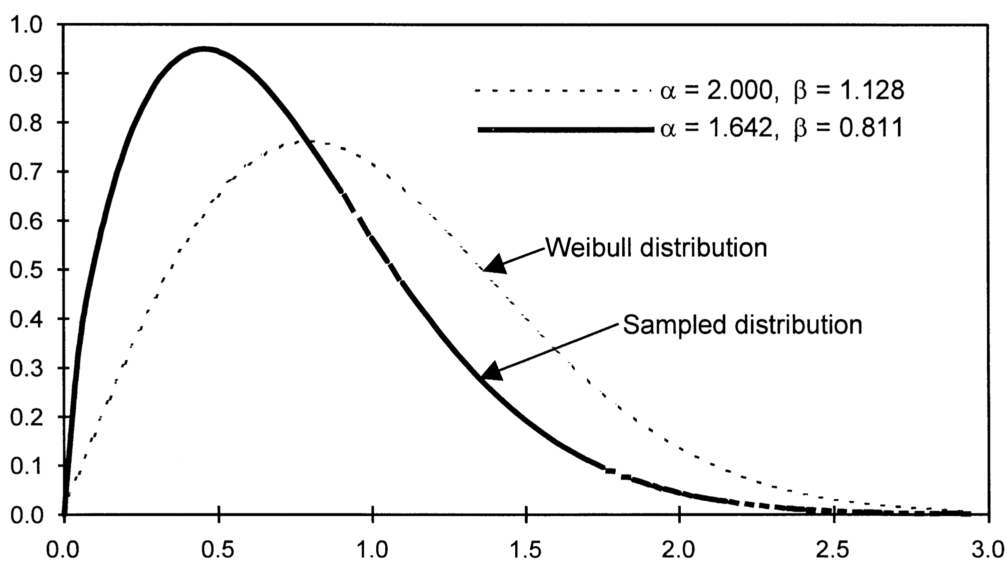


**Figure 6 Weibull sample comparison for α = 2**

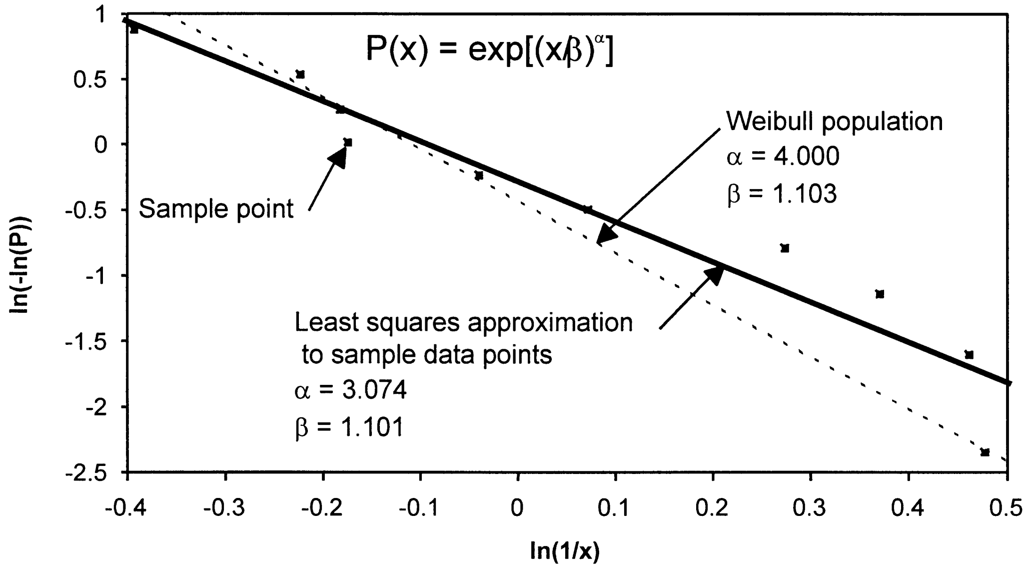


**Figure 7  Weibull approximation for α = 4 and 10 samples**

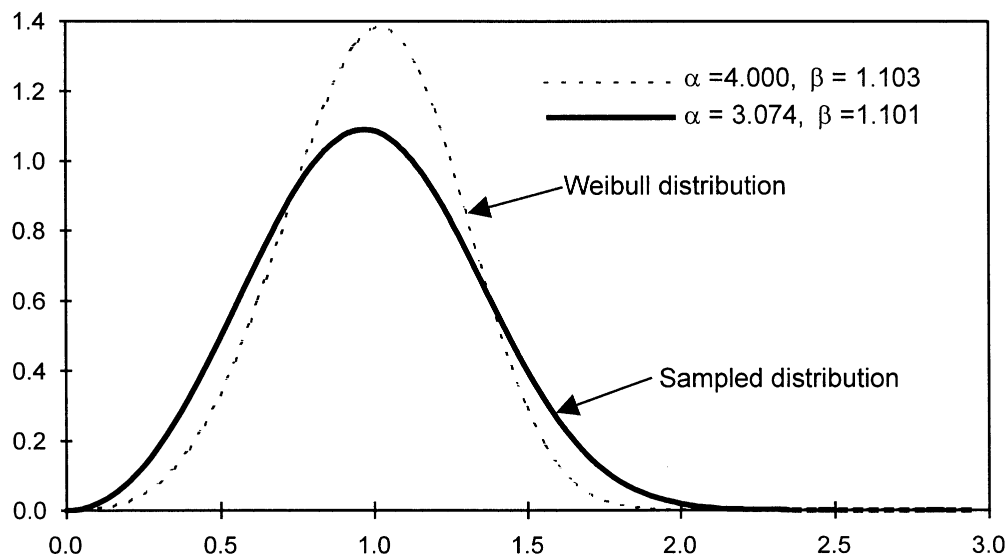


**Figure 8  Weibull sample comparison for α = 4**

## Widespread Fatigue Damage

A phenomenon occurring more frequently than generalized cracking is widespread fatigue damage (WFD). WFD is a major concern in aircraft that rely on fail-safety for structural integrity. The USAF has learned WFD can degrade the fail-safe capability of a structure with cracking that is of the order of one to two millimeters [5].

A deterministic definition of WFD is the following: The onset of WFD in a structure is characterized by the simultaneous presence of cracks at multiple structural details which are of sufficient size and density whereby the structure will no longer meet its damage tolerance requirement (that is, maintaining required residual strength after partial structural failure).

In many cases this definition is difficult to apply because of the complex cracking scenarios. Further, this definition may lead to an excessively conservative determination of the time of WFD onset. An alternate definition that removes these problems is the following: The onset of widespread fatigue cracking is that point in the operational life of an aircraft when the damage tolerance or fail-safe capability of a structure has been degraded such that after partial structural failure the probability of failure of the structure falls below the thresholds specified by the procuring (or certification) agency.

For the USAF, the threshold single flight probability of failure for the intact structure is $10^{-7}$. The USAF has determined the threshold for the acceptable conditional single flight probability of failure through their perception of the discrete source damage threat. In the case of the C-5A they assumed the probability of discrete source damage was $10^{-3}$ [6]. For the case of the 707 they assumed it was $10^{-4}$ [7].

One of the primary inputs to the risk assessment approach to determine the onset of the time to WFD is the distribution of cracks in the structure. The USAF has determined this distribution through teardown inspections of full-scale fatigue test articles or operational aircraft. They believe this is the best method currently available to obtain the data required to derive the probability distribution function for equivalent initial cracks in the critical areas of the structure. The word "critical" here refers to an area that could significantly contribute to the probability of failure.

The probabilistic approach also requires that the analyst determine the stress density function for each critical. The USAF derives this function from the available usage information generated by their individual aircraft tracking programs. The desired stress density function is the one for a single flight of an aircraft selected at random. The structural analyst can easily derive this function from the stress exceedance function developed as a part of the deterministic damage tolerance analysis. One can then compute the joint probability distribution of cracks and stress and integrate this function over the point set where the crack size has reached critical length. The result of this calculation is the single flight probability of failure. The time at which the probability of failure is unacceptable is the onset of WFD.

Therefore, the USAF considers the cracks in the structure and the stresses at the critical locations as random number sets. The crack growth function and the residual strength function are also treated as random functions because of the intrinsic variability of the material properties. Unfortunately, for a given population of aircraft these random number sets are not easily quantifiable. Fortunately, the variability of these functions does not appear to have a major impact on the risk of failure. Therefore, the analyst uses his best estimate of the mean of these functions in the risk assessment.

The damage scenarios in an airplane that could constitute WFD differ depending on location in the aircraft. However, typically, they fall into two categories. The first of these is multiple-site damage - characterized by cracks in multiple details in the same structural element. The second is multiple-element damage where there are cracks in multiple structural elements.

Previous efforts have shown the analyst can readily apply this type of analysis to the structures where the concern is multiple-element damage. This was the case, for example, for the KC-135 and the C-5A. The application of the risk assessment technology to the case of multiple-site damage is very much the same as it is for the case of multiple-element damage. In the case of multiple-site damage there will typically be a "boundary" that will determine if the cracking has the potential to become catastrophic. For example in the case of the fuselage lap splice, the boundary would be the crack stopper built into the structure at the frame or between the frames and its surrounding structure. This crack arrest feature protects the integrity of the structure. The condition of the crack stopper and its surrounding structure (that is, the boundary) will determine if the damage could propagate to catastrophic failure. Therefore, the interest is primarily in the degradation of the boundary with time and not the growth of the holes in the lap splice to link-up. When one thinks of the problem in this manner, then it may be solvable in a manner similar to that used for the multiple-element damage problem. Lockheed [8] demonstrated an example this of this in their risk assessment on of the inner to outer wing joint of the C-141 aircraft.

There must be emphasis placed on the detection, through nondestructive evaluation, of cracks that could be significant for determination of the onset of WFD. As indicated above, there is a need to make an estimate of this onset based on probabilistic assessment of cracking data derived from the teardown inspection of fatigue test articles or operational aircraft. One must recognize, however, that this is only an estimate. It is not realistic to expect the analyst could determine this time with great accuracy even with the most sophisticated fracture mechanics programs. The actual time may be either somewhat earlier or later than this estimate. It is important, therefore to be able to validate this prediction with nondestructive evaluation. This is difficult because the size of defect the inspector must find is quite small. The experimental evidence to date indicates cracks of the order of two millimeters can significantly lower the fail-safety capability of certain structural configurations.

## Weapon System Risk Assessments

### C-5A Risk Assessment

One of the early technical challenges for this program was how long to leave this aircraft in service with the original wing design. By the mid-seventies, the USAF established the damage tolerance initial flaw size for slow crack growth structure for fastener holes as 1.27 millimeters [9]. On the basis of this flaw size the safety limit was 7,000 flight hours of the so-called 14 mission flight profiles. In this case the time for the 1.27 millimeter flaw to grow to the critical crack length was the safety limit. Since the wing was not inspectable, this was also the life limit for the wing. The USAF made a final validation of the life of the wing through a teardown inspection. They took this wing from service when it had accumulated 7,000 hours equivalent to the 14 mission flight profiles. In the teardown inspection, Lockheed examined 44,641 fastener holes in detail for cracking. They did this work in the late seventies. From the population of cracks found in this teardown inspection the USAF performed an assessment to determine the probability of catastrophic failure and the time the wing lost its fail-safety. The USAF found that at 7,000 hours the wing had initially exceeded the acceptable $10^{-7}$ single flight failure probability. Further, they found the wing had lost fail-safety based on a conditional single flight failure probability of $10^{-4}$. This effort confirmed the USAF should take this structure out of service no later than 7,000 flight hours of equivalent 14 mission profile usage. They decided to allow the aircraft to fly to 7,000 hours with fail-safety compromised at 4,500 hours. The replacement wing box will easily meet the original life requirement of 30,000 flight hours.

### C-141 Risk Assessment

The USAF found a major WFD problem in the wing at Wing Station 405 joint [8]. The USAF observed first cracking on an operational aircraft in late 1984. In early 1989, they found an aircraft with a severed beam (or spar) cap. The USAF recommended that Lockheed perform a risk assessment based on operational aircraft cracking data to assess the likelihood of catastrophic failure of the aircraft. The risk assessment, as expected, indicated the joint was extremely critical.. The USAF had found numerous cracks in the area of the rear beam on many airplanes. In addition, they found a number of spar cap failures. Also, there have been multiple cracks discovered in the area of the forward beam on many airplanes. The risk assessment performed by Lockheed showed although inspections were somewhat effective in reducing the risk, the best alternative was to perform a modification on the joint. The USAF initiated aircraft restrictions, an inspection program, and an accelerated modification program to alleviate this problem. The action to remove WFD by a modification is similar to the earlier actions taken on the KC-135 and C-5A [6]. In the case of the KC-135 and C-5A the emphasis was on the elimination of the WFD problem rather than trying to manage it through an inspection program.

The USAF found another major WFD problem in wing lower surface fuel transfer holes (weep holes). There are more than 1500 such weep holes in each wing (both sides). The cracking experiences with the weep holes dates back to the original fatigue test. After 90,000 hours of block testing on the test article, Lockheed found cracking in many of the weep holes. Lockheed cold expanded these holes before they resumed testing with flight-by-flight loading. The additional 28,468 hours of testing showed the cold expansion was effective in controlling the weep hole cracking. Lockheed made a recommendation to WR-ALC in September of 1983 to perform the cold expansion on C-141 aircraft with 30,000 hours. In January 1993 the USAF Scientific Advisory Board (SAB) reviewed the potential for a service life extension of the C-141. They found the USAF had cold expanded weep holes on only six operational aircraft. They also found the weep hole inspection results were difficult to understand. One aircraft the USAF had found ninety-nine weep hole cracks, the longest of which was approximately 12 millimeters. They found other aircraft relatively free of cracks. However, there had been several cases where the weep hole cracks had progressed through the skin and had caused in-flight evident fuel loss. To understand this apparent anomaly, the SAB recommended a teardown inspection of an aircraft. The USAF tore down aircraft number 66-0186, in which the USAF had found ninety-nine cracks. It had 23,824 flight hours of relatively high damage usage, which converted to 44,539 damage hours (that is, hours of equivalent SLA-IIB spectrum usage) on the lower inner wing skin. The teardown inspection on aircraft number 66-0186 has revealed numerous holes with poor quality and a total of 255 cracked holes. Subsequently, WR-ALC performed an additional inspection and a limited teardown inspection on 66-9410, which had 45,202 equivalent damage hours on the inner wing lower surface. The results of the additional inspection have shown there was extensive cracking in the weep holes of this aircraft. Consequently, the USAF concluded the cracking observed in these two is representative of the aircraft with that number of equivalent damage hours. They concluded the early inspection results were unreliable. They changed the inspection procedure and validated it on a teardown inspection aircraft. The size of the cracks found led them to the conclusion there was severely degraded fail-safe capability in the wing. Also as indicated by the distribution of cracks, the cracks tend to line up which contributes to the loss of fail-safety. These airplanes were in a state of WFD. Therefore, the USAF placed the airplanes on restrictions and an inspection program. They developed an inspection program designed to preclude the cracks from reaching critical length and failing a wing panel. In addition, they developed a modification program to eliminate this problem. The modification program consisted of three parts. They found they could remove, or nearly remove, most of the cracks by reaming them. They elected to cold expand these holes. In many airplanes there were only approximately ten locations where cold expansion was not an alternative because the cracks were too large. Fortunately, at this time, the Wright Laboratory was completing a major program that would give the USAF the technology for patching metallic structures with composites. This appeared to be a more attractive alternative than the conventional metallic patches that required additional fastener holes in the lower surface of the wing. Therefore, the modification for those airplanes with a small number of large cracks would be composite patching. For aircraft that had a large number of large cracks the only alternative was replacement of wing panels. Lockheed performed a risk assessment to better understand the severity of the weep hole cracking problem. After reviewing the results of this assessment, the USAF made recommendations for subsequent actions. They decided not to fly any aircraft that had in excess of 40,000 damage hours on the lower inner wing surface until they performed a weep hole inspection. They would inspect the remainder of the aircraft and modify them based on a one year schedule. They found weep hole cracking in practically all of the aircraft. The nondestructive inspection program revealed a total of 11,000 cracks in the weep holes in the entire fleet. The USAF found no cracks in the weep holes that had been cold expanded. WR-ALC, with the support of the Wright Laboratory [9], accomplished the tremendous task of restoring these airplanes to flight status. They repaired the wings carefully with composite patching to ensure they had not degraded structural integrity of the aircraft. They returned these airplanes to unrestricted usage when they placed them back into service.

The USAF believed that WFD of the inner wing spanwise splices was a significant factor in the C-141 continued airworthiness. They had learned this from the loss of fail-safety in the C-5A wing. In 1990 the USAF [11] estimated they could expect WFD in the spanwise splices in inner wing lower skin at about 45,000 SLA-II equivalent flight hours. They based this estimate the teardown inspection of the C-141 fatigue test article (Specimen A).. The size of cracks that could cause loss of fail-safety in the C-141 inner lower wing is in the order of 1.5 millimeters. Lockheed performed an additional assessment of the risk based on teardown inspections of wing panels taken from operational aircraft. They found significant degradation of fail-safety at 37,000 hours. The USAF made the decision to manage the safety of those airplanes above 37,000 hours by slow crack growth. This decision resulted in a very difficult inspection program [7].

707 Risk Assessment

The USAF elected in the eighties to use the 707 aircraft for Joint Stars (Joint Surveillance Target Attack Radar System). When Northrop Grumman, the contractor for Joint Stars, selected the aircraft, the configuration was the primary concern - not the age. Many of the airplanes selected were close to (or above) the original life goals of sixty thousand flight hours and twenty thousand flights established by Boeing for the 707.

The largest concern about the structure of this aircraft was the potential for the degradation of fail-safety because of WFD in the wing. Boeing performed a teardown inspection on a relatively high time aircraft in the mid seventies. The inspection performed by Boeing, completed in 1976, revealed numerous cracks in the aircraft. The cracks that caused the most concern were in the lower wing splicing stringers and the large stringers around the lower wing inspection holes adjacent to the splicing stringers. Boeing published several Service Bulletins as a result of these wing crack findings. These Service Bulletins called for either a high frequency eddy current inspection inside of the wing or an external low frequency eddy current inspection. These inspections have revealed major damage including a severed stringer and skin cracks in excess of 44 millimeters. The Boeing database, however, was not definitive enough to be usable in an assessment of the risk of failure. Consequently, the USAF contracted with Boeing to examine higher time aircraft parts taken from retired aircraft at Davis Monthan Air Force Base to quantify the risk associated with WFD [7].

Boeing performed a teardown inspection on a 707-300 wing from an aircraft at Davis Monthan Air Force Base. This aircraft, representative of the Joint Stars aircraft, had experienced 57,382 flight hours and 22,533 flights. They performed the teardown inspection on the wing lower surface and the wing stringers. Stringers and skins where Boeing used steel fasteners contained most of the cracks found. This was typically in the area of the wing skin splices and the large adjacent stringers. The beneficial effects of the aluminum rivets attaching the other stringers to the wing skins apparently reduced the amount of cracking there. There was, however, some cracking found in these locations.

Boeing found that cracking in the aircraft in the area of the steel fasteners was quite extensive. They found a total 1915 cracks found in five sections removed from the aircraft. Most of the cracks found were small. However, they found a significant population cracked to the point of considerable concern. They found that increasing the size of the holes in the splicing stringer and the large adjacent stringer would not remove all of the cracks. About twenty percent of these holes would still have stringer cracks. Further, they found significant cracking outboard of the Wing Station 360 production joint. Therefore, the problem involved most of the wing. Typically, the large adjacent stringers had more large cracks than the splicing stringers. The largest crack found in a stringer was approximately 38 millimeters in length. It was near the point of rapid fracture. There were, however, many cracks found that would have gone to failure in the planned life span of the Joint Stars aircraft. There was a concern about cracking that would degrade the capability of the structure to sustain discrete source damage. There was also a concern about the fatigue failure of the stringers and subsequent catastrophic loss of the aircraft after a skin failure.

Boeing calculated the stress intensity of each of the cracks found. They then determined for each of them the size of the corner crack with the same stress intensity. From these cracks, the USAF derived the crack distribution function. They used a population taken from the largest of them to approximate the crack distribution with a two parameter Weibull distribution function. It is typical that a single Weibull distribution function will not approximate the longer cracks as well as the shorter cracks. This is not a problem since only the longer cracks will have a significant effect on the risk of failure.

The USAF needed two stress distribution functions for the assessment. The first is the stress distribution function for the intact structure. Boeing derived this in the usual manner from the intended usage of the aircraft, the external load analysis, and the stress analysis of the wing. Second, for the cases where discrete source damage was present they determined the local stress increase from the damage. In many cases the local stress increased to the point where there was significant plastic deformation of the structure. When this occurs it is essential the plastic deformation be included in the analysis. A linear analysis in these cases would likely lead to serious errors in the determination of risk.

For the cases of discrete source damage the maximum single flight failure probability allowed was $10^{-3}$ and for the intact structure case the maximum single flight failure probability allowed was $10^{-7}$. For the stated criteria for discrete source damage, the USAF found significant degradation of fail-safety beyond 50,000 flight hours of commercial usage. Therefore, for some aircraft, there will be unacceptable fail-safety degradation before the end of the planned 20,000 hours of Joint Stars usage. This will occur for Joint Stars aircraft with more than 36,000 commercial usage hours. Further, for the case of no discrete source damage, there will be safety degradation beyond 58,000 hours of commercial usage. Therefore, aircraft with initially more than 44,000 hours of commercial usage will have a high probability of failure before operationally flying 20,000 hours.

There are two possible approaches for solution of this problem. The first is to remove the steel fasteners in the area of concern in the lower wing surface and perform an eddy current inspection. If the inspector finds no indication of a crack or if increasing the size of the hole would remove the indication, then this hole would be cold expanded. For cracks that are too large for this remedy, the USAF could utilize a repair such as composite patching. This approach appears to be viable for aircraft with less than 45,000 commercial usage flight hours. It also may be viable for aircraft in the 45,000 to 55,000 flight hour range. A second alternative would be to replace the wing panels and stringers in the area of concern. This may be the only alternative for aircraft with more than 55,000 commercial usage flight hours.

**Widespread Fatigue Damage Example Risk Assessment**

The following example illustrates some of the essential features of the risk analysis process. The example determines the risk of catastrophic failure for both the intact and partially failed structure of a hypothetical aircraft designed for a 30,000 hour life. The aircraft is to fly only one mission that is two hours in length. The aircraft has one critical area with 500 fastener holes. The initial crack distribution is the crack distribution function derived from a teardown inspection. Figure (9) shows the corresponding crack density function. Figure (10) shows the corresponding crack distribution function. For the intact structure, Figure (11) shows the stress exceedance function for each of these holes. Figure (12) shows the corresponding stress probability distribution function, derived from the exceedance function. Figure (13) shows the stress density function. The threat of discrete source damage is $10^{-3}$. For the partially failed structure, only ten of the 500 fastener holes have their stress increased to 1.5 times the stress for the intact structure. Figure (14) shows the residual stress function. The crack growth function modifies the initial crack distribution function so the crack probability distribution has the correct time dependence. Figure (15) shows the crack growth function. Figure (16) shows the final function needed for the calculation of risk. This is the inspection probability of detection function.

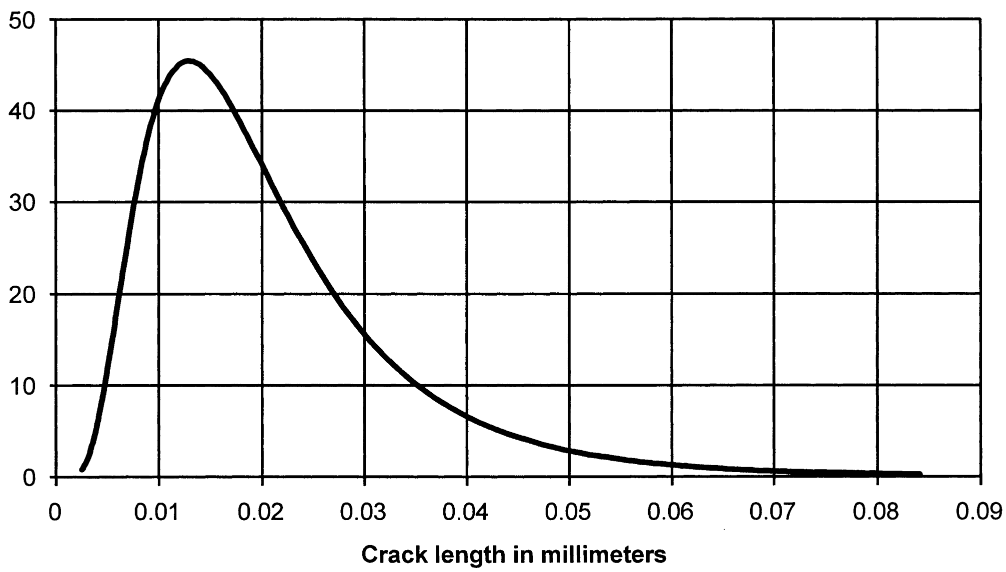


**Figure 9 Crack density function from the A-7D**

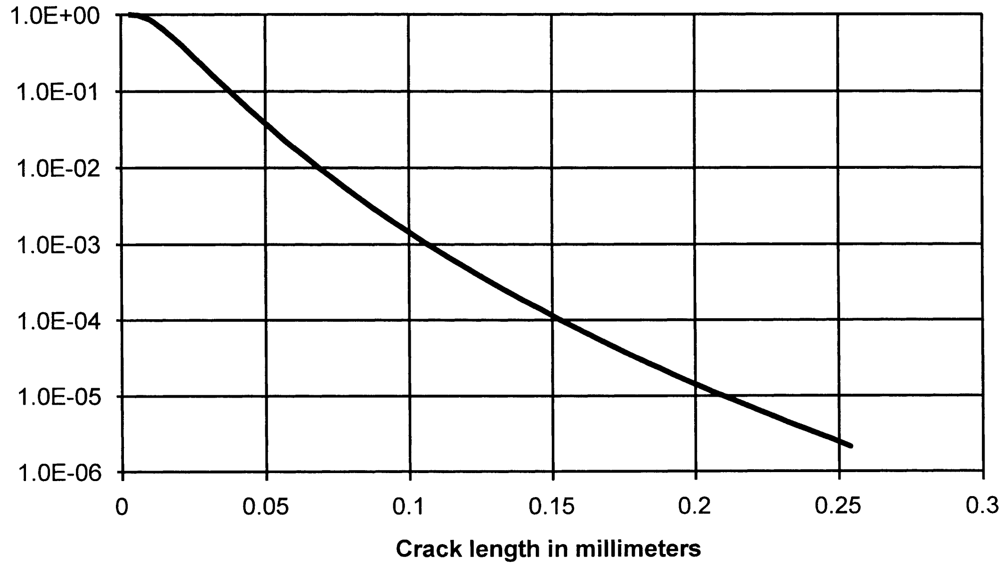


**Figure 10 Crack distribution function from the A-7D**

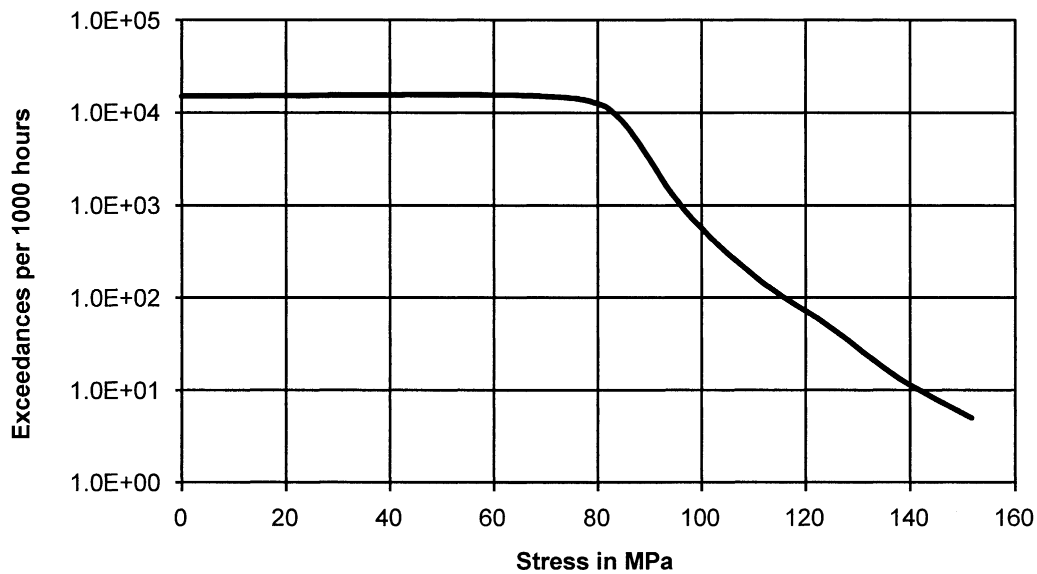


**Figure 11 Stress exceedance function**

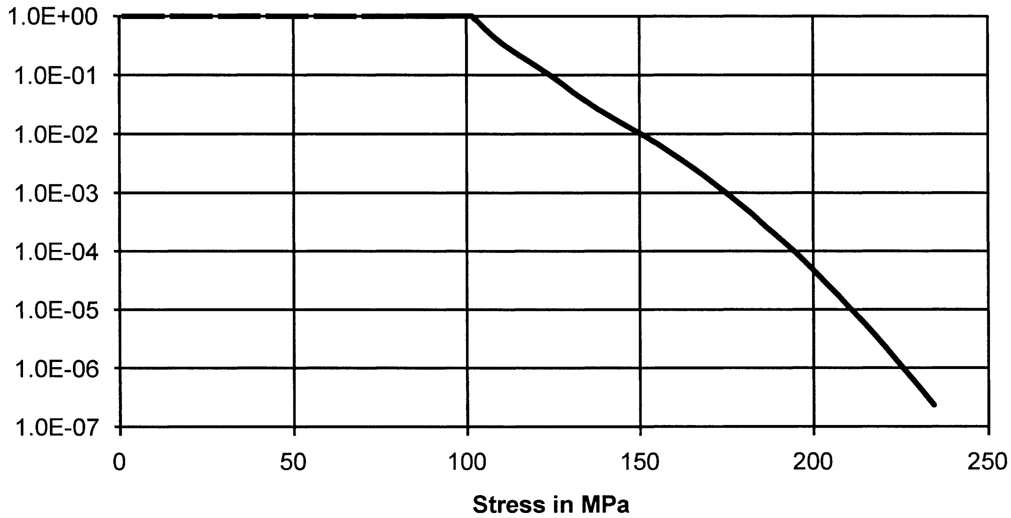


**Figure 12 Stress probability distribution function**

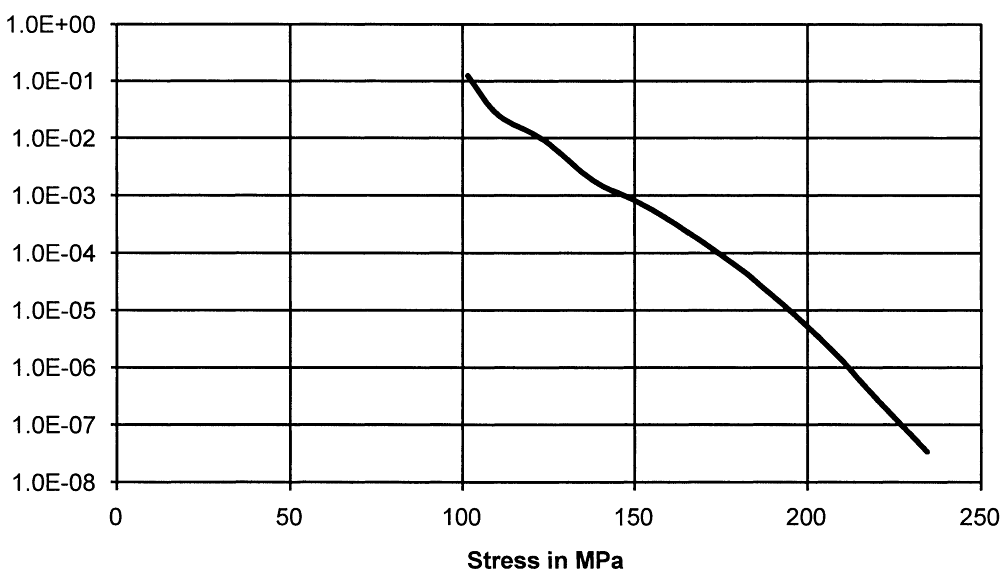


**Figure 13 Stress probability density function**

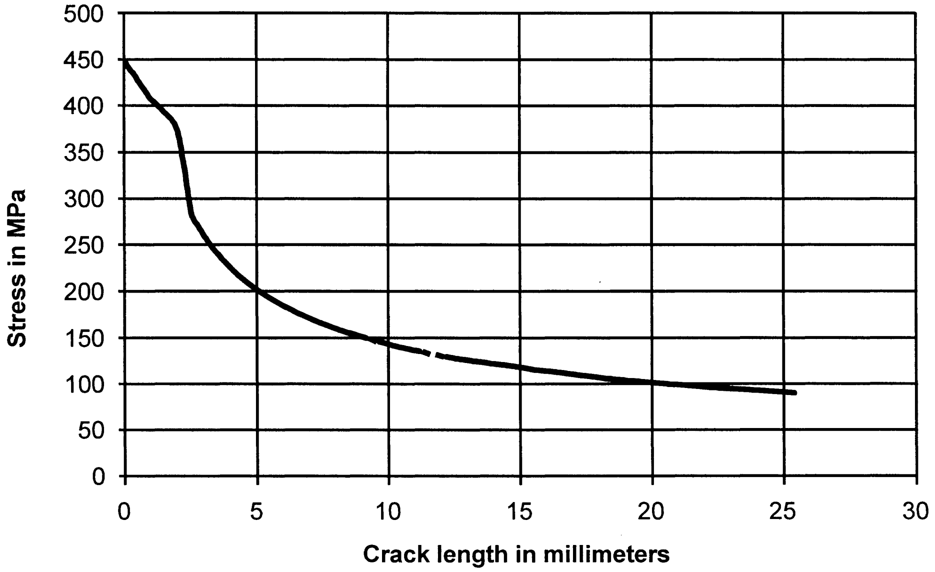


**Figure 14 Residual stress function**

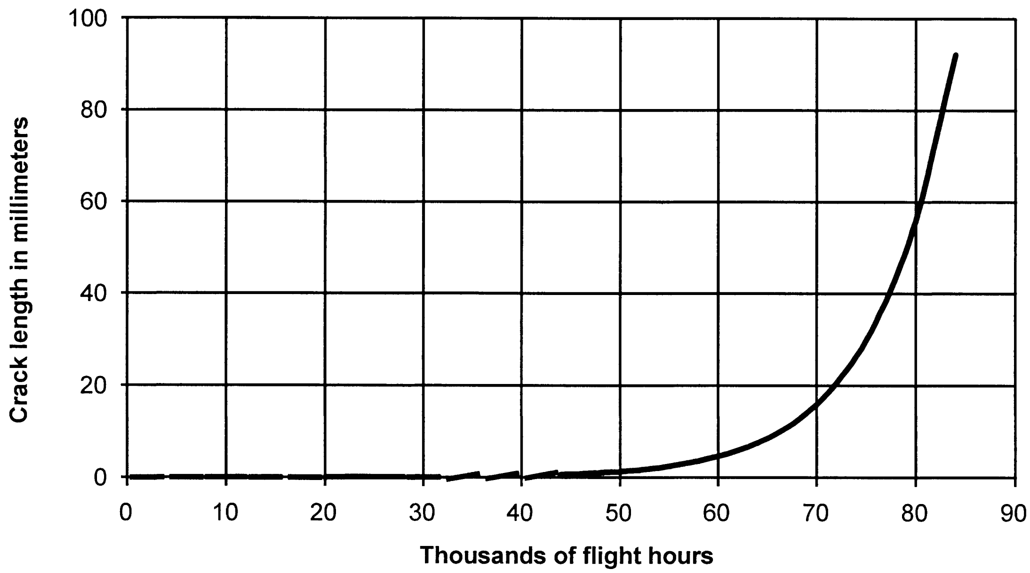


**Figure 15 Crack growth function**

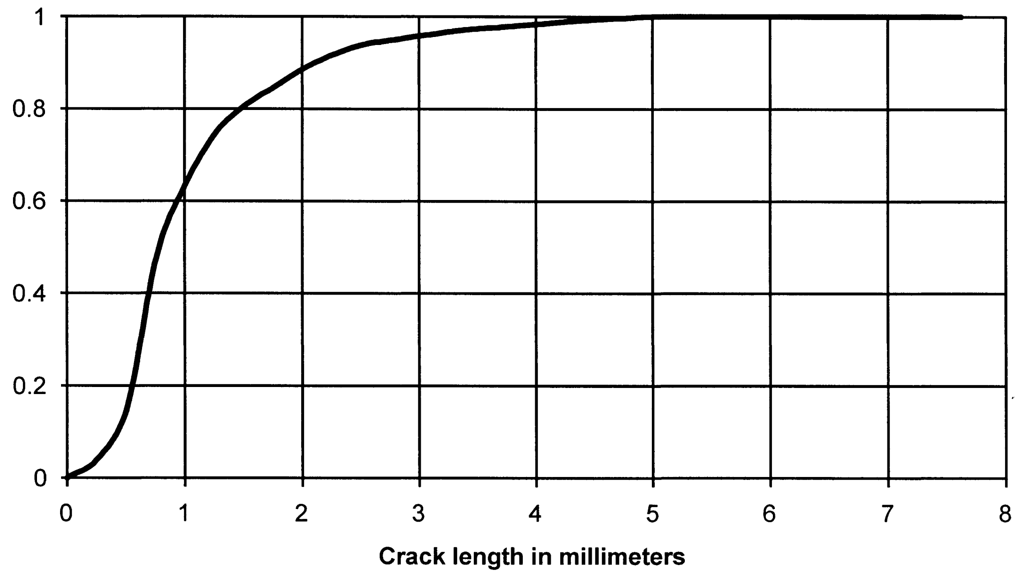


**Figure 16 Probability of detection function**

Figure (17) shows the single flight probability of failure for the intact structure without inspections. From this figure, one sees the risk exceeds the $10^{-7}$ threshold of acceptability at about 22,000 flight hours. From Figures (14), (15), and (16) the analyst can determine the damage tolerance inspections. The first inspection is at 7600 flight hours and the inspection interval following the first inspection is 5000 hours. Figure (18) shows the single flight probability of failure for the intact structure with inspections. One sees these inspections are quite effective in reducing the risk of failure and containing the risk within acceptable limits to 30,000 flight hours. It is clear from this figure that on the basis of the inspection capability assumed and the inspection interval derived from the damage tolerance methodology the risk is increasing significantly. Therefore, one must make a reduction in the inspection period if one intends to fly the aircraft significantly beyond its original life of 30,000 flight hours.

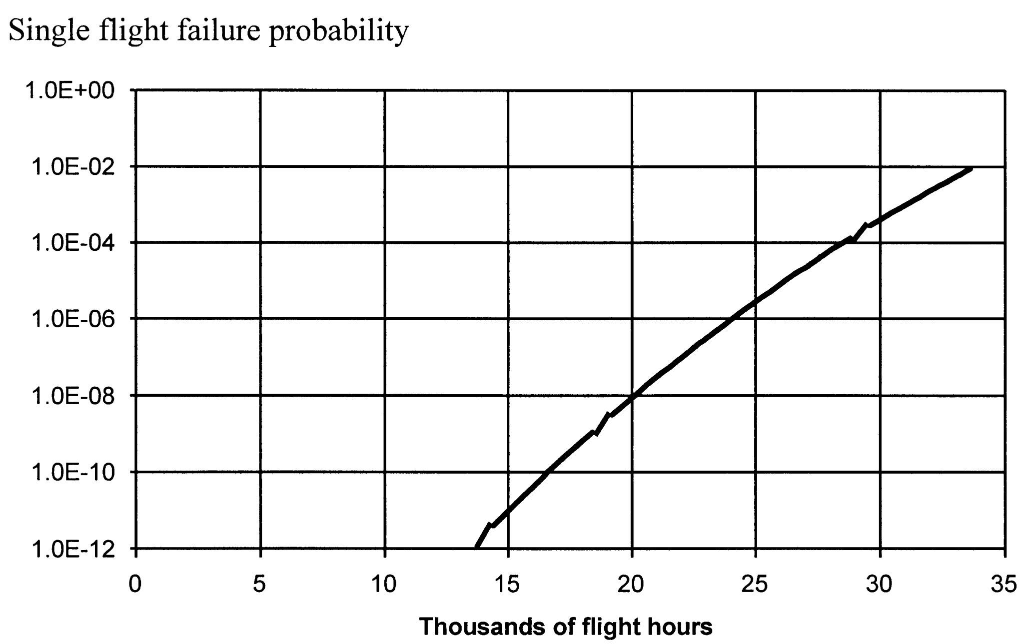


**Figure 17 Intact structure with no inspections**

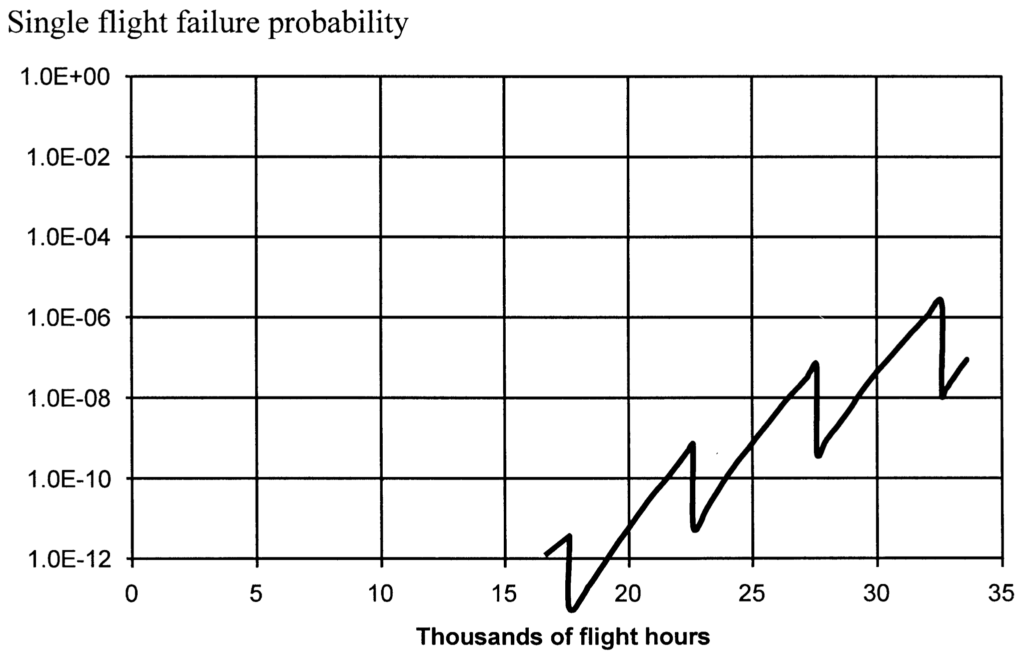


**Figure 18 Intact structure with inspections**

Figure (19) shows the single flight failure probability for the partially failed structure without the effect of inspections. This is the conditional probability for the structure damaged from an external source. One sees the risk crosses the threshold of acceptability for this case (that is, $10^{-4}$) at approximately 16,000 flight hours. The aircraft has degraded fail-safe capability long before the time the intact structure has reached the unacceptable risk threshold. Figure (20) shows the influence of inspections on the probability of failure. One sees the inspections are essentially ineffective in reducing the risk for this case. This example clearly illustrates the damage tolerance derived inspection program may not adequately protect the fail-safety of an aircraft in the presence of widespread fatigue cracking.

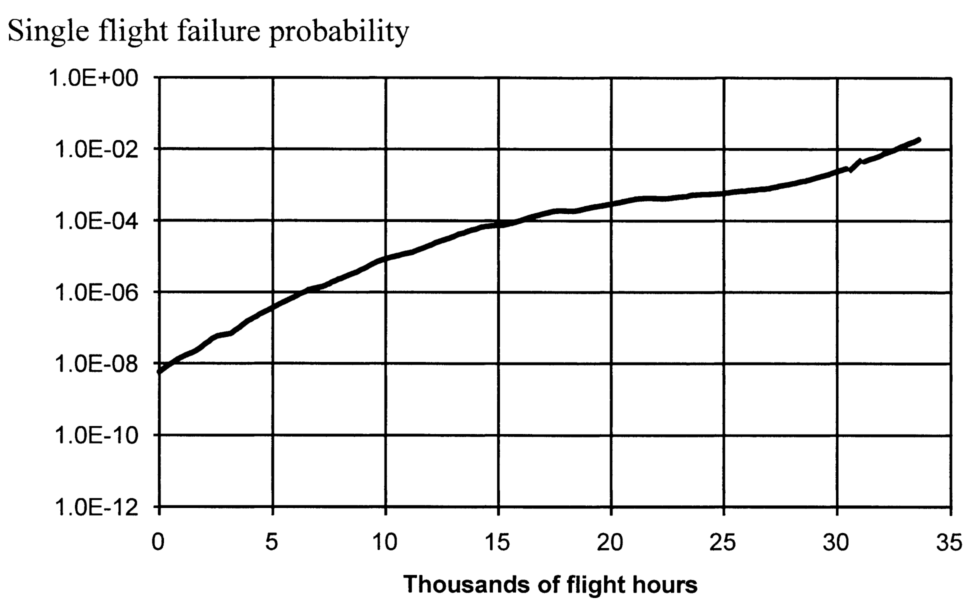


**Figure 19 Damaged structure with no inspections**

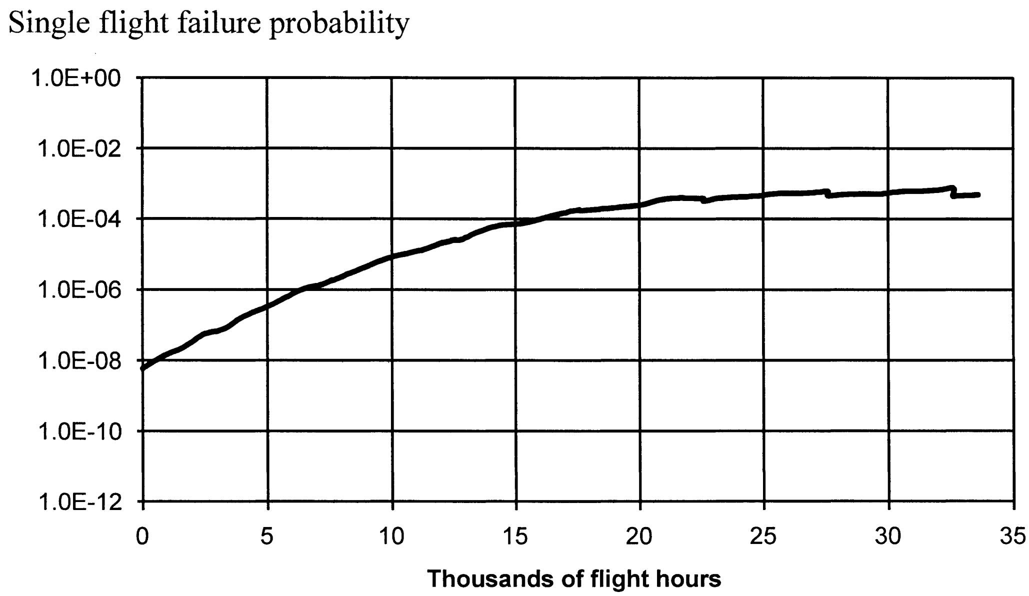


**Figure 20 Damaged structure with inspections**

## Conclusions

As indicated above, the cornerstone for protecting the safety of USAF aircraft is damage tolerance. There are some cases, however, where probabilistic methods find an important use. One approach that appears to be attractive especially for mechanical subsystems is the use of reliability analyses based on testing. In some cases these methods can provide satisfactory solutions where a damage tolerance assessment may be impractical. The USAF believes the process may apply to mechanical subsystems since they are typically fail-safe by design.

A major problem in aging aircraft is WFD. It is essential to establish an estimate of the time of onset of this problem. The USAF does this through the analysis of data derived from teardown inspections of fatigue test articles and of operational aircraft. They will need to corroborate these estimates through the use of detailed inspections of suspect structural elements. In some cases the nondestructive inspection capability does not exist to economically find WFD size cracks. The USAF must continue their effort to attain this capability. Once the aircraft operator determines the aircraft has reached the time of onset of WFD, he needs to make modifications of the structure to remove this problem.

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# Occurrence of Corrosion in Airframes

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## 1. SUMMARY

Degradation of the mechanical properties of a material interacting with the environment is probably the best and widest definition for corrosion. In particular, as mechanical properties are the driving forces in the design of military aircraft, corrosion in airframes must be considered as a major problem because it directly affects safety, economic and logistic issues.
Considering the variety of materials, environments and mechanical stresses involved in the aeronautical field, it represents one of the areas where the largest spectrum of corrosion types is observed.
Many classification can be used to categorize aircraft corrosion phenomena: wet or dry corrosion depending on the environment, time dependent or time independent phenomena, mechanically or not mechanically assisted corrosion failures, etc.; all of them are useful to understand the main cause of the observed corrosion case and consequently to apply the most adequate corrective actions.
The purpose of this lecture is to provide an overview on the most common forms of corrosion experienced in the past, in order to present a wide range of severity arising from cosmetic to catastrophic failures.
Particular attention will be given to the corrosion aspects related with aging aircraft issues.

## 2. INTRODUCTION

Although aircraft corrosion is an old matter and many advances have already been done in corrosion prevention and materials selection science, nevertheless it seems far to be solved.
For instance, corrosion matter, that is a serious problem for every high engineered system, in airframes became more and more important in this last decade when aging aircraft subject was promoted by many different factors, most of them afferent to economic constraints[1].
Recently, corrosion contribution to the aging aircraft related costs has been estimated up to 80%.
On the other hand corrosion problems also have an heavy impact on safety and about 45% of the observed component failure can be ascribed to corrosion, when both direct and initiation effects are considered.

Corrosion in airframes is mainly an electrochemical matter, where an electrically conducting solution assists the transfer of metal ions, dry corrosion being almost always limited to engine components.
In spite of this limitation, a lot of different forms can be observed and one of the most useful theory that can be used to categorize them is the Structural-Electrochemical one[2].
In agreement with this theory, the driving force of an electrochemical corrosion process must be considered the presence of heterogeneity on the metal surface.
Depending on the nature and the dimension of this nonuniformity three different categories of corrosion must be experienced:
**Uniform corrosion**, in presence of sub-microstructural heterogeneity from 1 to 1000 Å, comparable to the cristallographic lattice dimensions (i.e. differences in the position of atoms, thermal fluctuations of metal ions in solution, etc.).
**Selective corrosion**, in presence of microscopic inhomogeneities from 0,1 m to 1 mm, comparable to the size of the cristallographic structure of the metal (i.e. grain boundaries, second phases in alloys, etc.).
**Localized corrosion**, in presence of macroscopic inhomogeneities greater than 1 mm, comparable to the size of the component (i.e. galvanic coupling, differential aeration, etc.).

## 3. UNIFORM CORROSION

Here, the inhomogeneities on the metal surface interacting with an aggressive environment, are so small in dimension and potential that the same area will change, playing continuously a different anodic and cathodic role.

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The total effect is an attack on the whole surface leading to a uniform or quasi-uniform lost in depth of the metal.
Although this is a very common mechanism in many corroded systems, it is not so often observed on airframes because the chosen aeronautical materials are always less prone to it.
Uniform corrosion is usual for non-stainless steel and iron where it can be easily recognized by red rust.
Being easily detected and forecasted uniform corrosion can't be considered as a very dangerous form of corrosion.
Usually general attack occurs on parts where the original protective coating has failed for any reason. The most typical case is certainly observed on cadmium plated steels after the anodic coating has been totally sacrificed (Fig. 1).

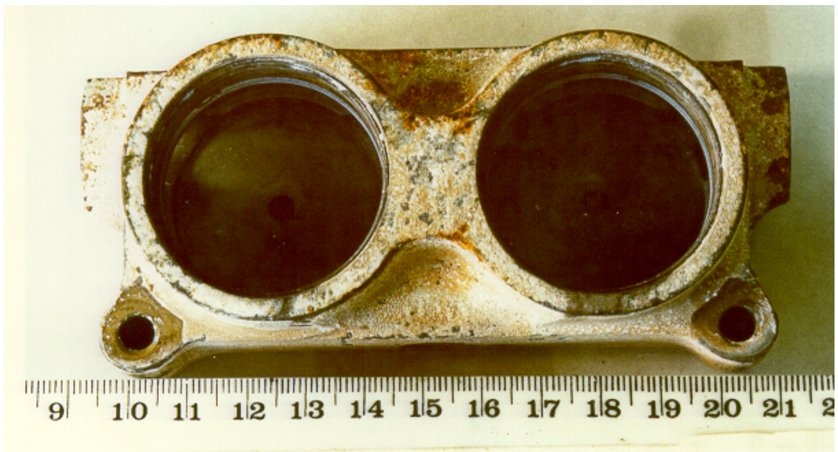


**Fig. 1 – Uniform Corrosion on a cadmium plated AM-X Air Combustion Chamber**

Erosion, caused by the action of a fast moving fluid, can also lead to a uniform or quasi-uniform attack. This specific mechanism, called erosion-corrosion, becomes more severe in aircraft operating in hot desert climates, where an high humidity content, especially in night time, is associated with sand: the solid particle content, furthermore rich in salt, acts as an extremely abrasive media, removing paint, surface finish and corrosion products, offering continuously new metal surface.
Aging aircraft issues exacerbate uniform corrosion problems on electrical and avionics equipment where, in order to obtain the requested performance, materials are often inferior in terms of corrosion resistance.

## 4. SELECTIVE CORROSION

In this category are included all the phenomena depending on the presence of heterogeneities in chemical composition. In this sense we can also talk about this electrochemical attack as caused by an intrinsic heterogeneity of the material.

### 4.1 Intergranular Corrosion

On airframes, intergranular corrosion (Fig. 2) is the more often observed mechanism of this class because it is characteristic for the aluminum alloys, both the Al-Cu (2xxx) and the Al-Zn (7xxx) alloys, where the driving force for the electrochemical process is the difference in potential between the second phases (richer in copper - more noble -, or richer in zinc – less noble -) and the aluminum matrix.

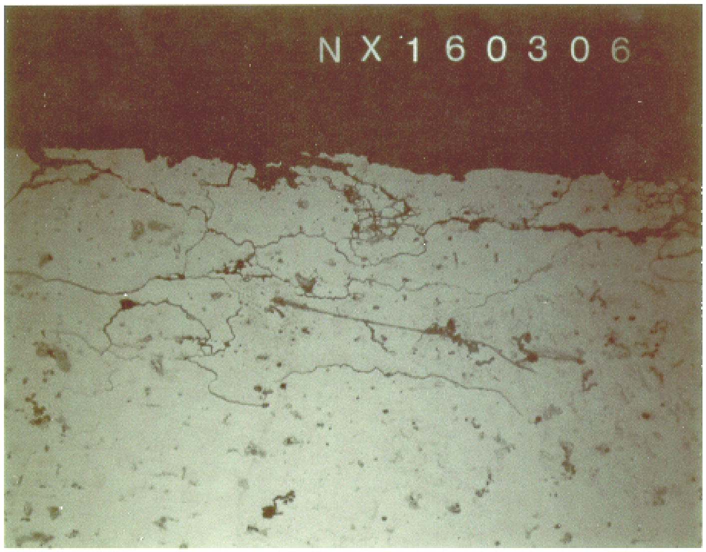


**Fig. 2 – Intergranular Corrosion on AA2024 (160x)**

In this case corrosion profile follows the shape of grain boundaries (Fig. 3), where second phases are precipitated, and must be considered very dangerous because, in spite of a minimum material lost, mechanical properties fall dramatically down[3]. Furthermore, intergranular corrosion is frequently hard to detect also by means of NDE.

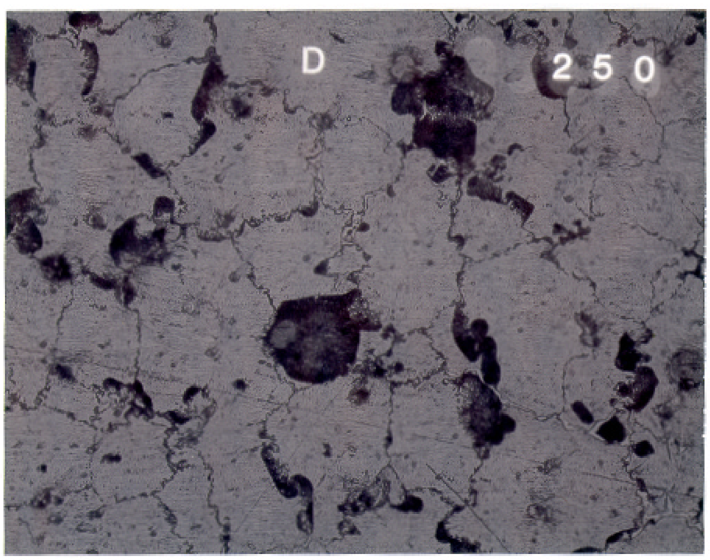


**Fig. 3 – Intergranular Corrosion on Mg Alloy AZ-91C (250x)**

When intergranular corrosion occurs on heavily rolled or extruded parts having elongated grains in the direction of working, the produced phenomenon has the very characteristic aspect of an exfoliation (Fig. 4).

**Fig. 4 – Exfoliation on a Breguet Atlantic Br.1150 AA2024 spar**

Intergranular attack can also be observed on austenitic stainless steel. On these materials an incorrect cooling procedure after an heat treatment can lead to a sensitization of the part, caused by the grain boundary precipitation of chromium carbide ($Cr_{23}C_6$) and according to this the strong depletion in chromium content of the contiguous areas. This can be the case of wrong welding procedures (Fig. 5).

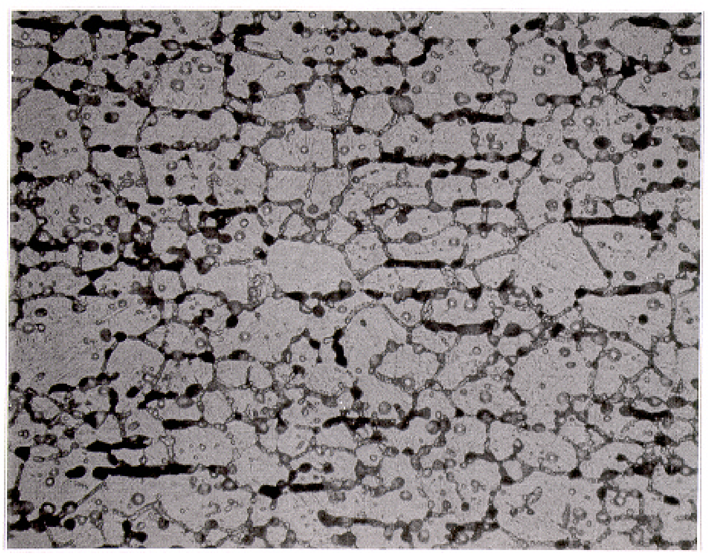

**Fig. 5 – Low Temperature Sensitization on a PH 17-7 Stainless Steel**

4.2 Crystallographic Corrosion
Although much less common on airframes, another kind of selective attack to be mentioned is the crystallographic corrosion which can be generated when whole grains or volumes are each other electrochemically different enough.
This is the case of some brasses where parts richer in zinc leave the metal leading to a spongy structure.

## 5. LOCALIZED CORROSION

This is certainly the class where the widest number of corrosion mechanisms are observed.
The common factor among the different forms of corrosion in the case of a localized attack is the presence of stable and clearly separate cathodic and anodic areas.

### 5.1 Pitting Corrosion

Pitting corrosion is a dangerous attack which occurs on passive materials when the protective oxide layer breaks.
It is often observed on stainless steel and aluminum alloys that spontaneously form a protective film: as a result of small damages on the passive layer, the damaged areas will work as anodes immersed in a very large cathodic area and will suffer in consequence of this a very localized attack which leads to the formation of deep and narrow cavities.
Pitting corrosion is particularly common on aircraft structures operating in marine environments, since the chloride ions and halide ions in general promote the local dissolution of protective oxide films.
Here following (Fig. 6 and 7) some cases occurred in the recent past are shown.

**Fig. 6 – Pitting Corrosion on a HH-3F Compressor Blade**

**Fig. 7 – Pitting Corrosion on MB-326 Balance Tabs**

Some authors[4] include in pitting corrosion mechanism also those corrosion phenomena that take place on active metals, previously protected by a suitable external coating, when the protection is locally damaged.

In any case pitting must be considered very insidious since it tends to accelerate its corrosion rate because of the increasing acidity and chloride content inside the cavity; furthermore, in highly loaded structures, the stress concentration at the base of a pit is often sufficient to promote fatigue or stress corrosion cracking.

5.2 Crevice Corrosion

This form of attack (Fig. 8) is originated by the difference in the concentration of dissolved oxidant agent (usually oxygen) inside and outside a crevice. In this case the area inside the crevice will act as anodic and there a pit will develop.

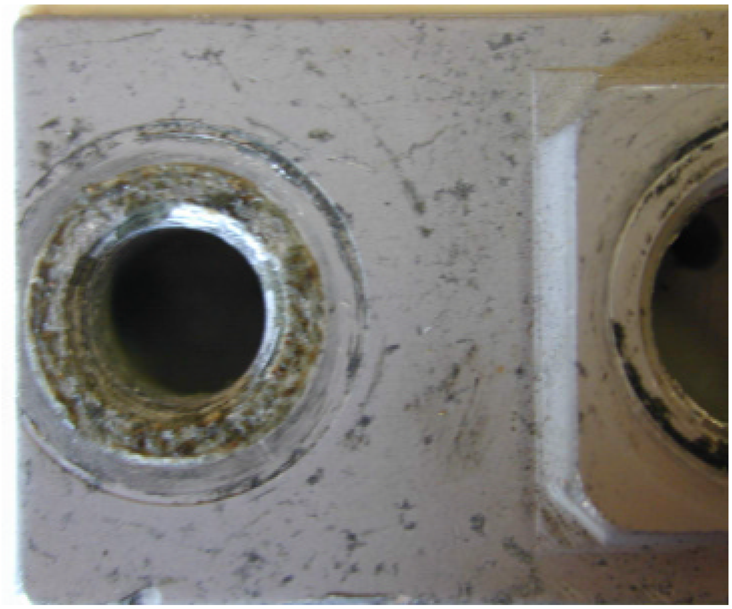

**Fig. 8 – Crevice Corrosion on Tornado**

In airframes, corrosion crevice is frequently observed on lap joints or under surface deposits in presence of stagnant solution. It is usually associated with a poor performance of the sealant or sometimes can be caused by a defect of design (i.e. poor drainage conditions).

Its nature makes it dangerous because often occur on unexpected areas and can't be detected by visual inspection if not disassembling.

5.3 Galvanic Corrosion

Galvanic corrosion is the most evident form of localized attack, where anodic and cathodic areas are very clearly identified.

It occurs when two metals of different electrochemical potential are in contact in a corrosive medium and the resulting damage to the less noble metal will be more severe than if it was exposed alone to the same medium.

The extension of the corroded area on the anode as far as the corrosion rate will depend on the difference in the electrochemical potential between the metals and the conductivity of the aggressive medium. Anyway, the corrosion attack will be more concentrated in the part of the anodic metal closest to the cathode.

In aircraft structures is often necessary to use different metals and galvanic corrosion can't be completely avoided. In this case is important to take care about the ratio between the cathode and the anode: increasing the ratio the corrosion will tend to be superficial.

This is the typical example occurring at fastener holes in aluminum alloy skin when steel bolts or rivets are used.

Looking at the galvanic series it's easy to realize that magnesium alloys are very susceptible to suffer galvanic attack when used in junction with any other metal (Fig. 9-11).


**Fig. 9 – Galvanic Corrosion on a Mg Alloy Spacer, coupled with a steel beam in the MB-326 Central Section**

**Fig. 10 – Galvanic Corrosion on the MB-339 between Mg Alloy Trim and Aluminum rivets**

**Fig. 11 – Galvanic Corrosion on the MB-339 Attach Fitting**

5.4 Filiform Corrosion

Filiform corrosion (Fig.12) can be found under organic coatings such as paints, due to penetration of moisture through the coated surface under specific temperature (T ≥ 30 °C) and humidity conditions (Hr ≥ 85%).

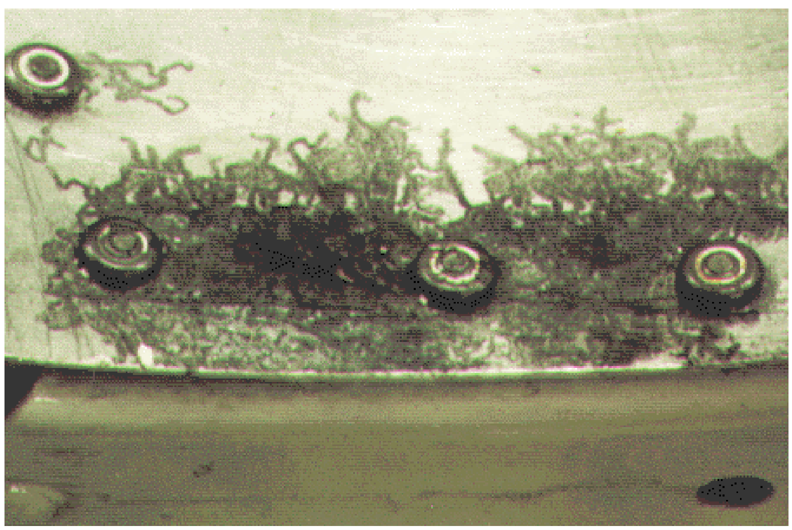

**Fig. 12 – Intergranular Corrosion on Aluminum Alloy**

This mechanism is not particularly dangerous on itself since it propagates creating blistering "wires" of corrosion products on the surface of the metal, actives just on the tip of each wire, but can degenerates in more serious attacks if not detected and removed in an early stage.

5.5 Stress Corrosion Cracking (SCC) and Corrosion-Fatigue

These two dangerous localized corrosion mechanisms are often unfortunately observed on airframes.

Both produce cracks, different in shapes and patterns, whom growth is caused by the synergetic action of a moderate corrosive environment and a mechanical stress: a static load (lower than the material's yield tensile stress) in the case of SCC (Fig. 13), or a cyclic load in the case of corrosion-fatigue (lower than the material's fatigue limit).

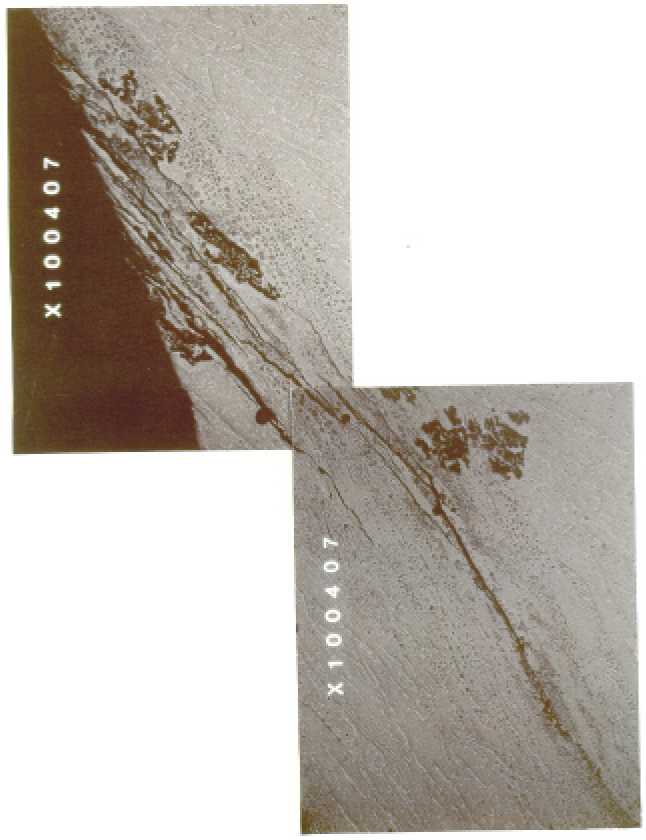

**Fig. 13 – Stress Corrosion Cracking on a Br.1150 bomb bay guide rail**

Many models have been proposed[5] to explain the crack growth process for these attacks, all of them coinciding that just the crack tip is anodic while the rest of the metal (including the walls of the crack) act as cathodic.

Once the crack has formed it will continue to grow, stopping only when the static (SCC) or the cyclic (corrosion-fatigue) load has fallen below the critical value, or alternatively excluding the corrosive environment.

These forms of corrosion must be considered as a major problem in aging airframe related issues, particularly corrosion-fatigue at low frequency cyclic stresses, where the time dependent corrosion process has the opportunity to explicate its action.

In effect Multiple Side Damage, a phenomenon under intensive investigation since the last ten years, can often be seen as an extension of the corrosion-fatigue mechanism.

5.6 Hydrogen Embrittlement

Hydrogen embrittlement is often considered as a special case of the more general SCC mechanism[6].

Its effect is to lower the ductility in metals when penetrated into the material[7] by means of a natural corrosion reaction or, more often, during a plating or a pickling process.

High strength steel and austenitic stainless steel are the most commonly affected aerospace materials, their susceptibility also depending on the metal composition[8].

Parts more often failed for hydrogen embrittlement are bolts and main landing gear items.

5.7 Fretting

Fretting is an insidious form of corrosion that occurs when the environmental action is assisted by material wear under low vibratory relative mote of parts.

The abrasion of the surface finishing, and after that of the corrosion products, continuously offer new metal surface to the environmental aggressive attack: because of the abrasive nature of the corrosion products, this mechanism is rate increasing, usually leading to hemispheric pits where fatigue marks are often found on their bottom (fig. 14).

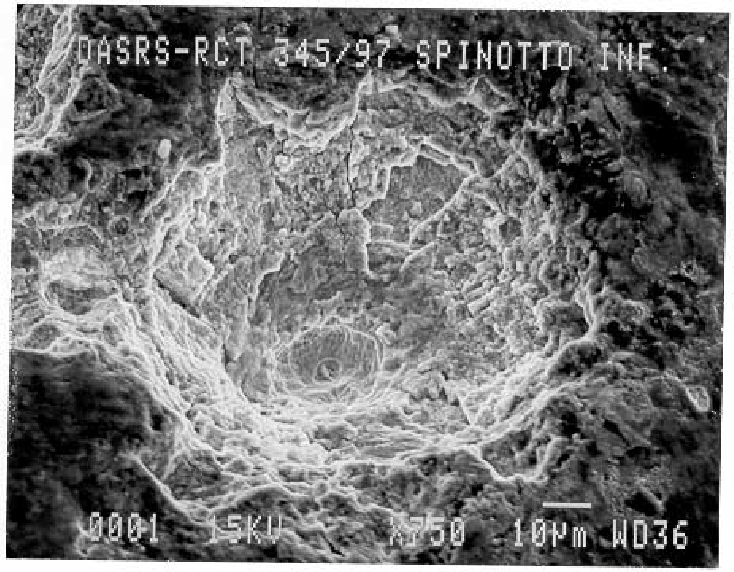


**Fig. 14 – Fretting Corrosion on the MB.339 Landing Gear spine**

## 6. CONCLUSIONS

This lecture has given a compendium of the deterioration phenomena induced by corrosion most frequently observed on airframes.

The scheme followed in the presentation of the corrosion forms was derived from the Structural-Electrochemical theory, in order to clarify some aspects common to different corrosion mechanisms.

An always increasing knowledge of the corrosion problems, based on the past experiences and on a multidisciplinary approach comprehensive of design philosophy, condition based maintenance and NDE development, is then essential to win the new economic and safety challenges offered by the aging concerns.

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# Human Factors in Aircraft Inspection

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**Abstract:** Inspection of both airframes and engines is a key activity in maintaining continuing airworthiness. Unless structural defects are detected at the appropriate time, structural failure may result. The reliability of the inspection system must be known in order to schedule safe inspection intervals. However, inspection reliability necessarily includes human inspector reliability so that knowledge of human inspection performance is vital to safety. This paper describes models of the major functions of the human inspector, and applies these within a framework of inspection reliability. From these models, and field experiments on inspectors a set of factors known to affect inspection reliability is derived. These can be used to define good practices necessary to continuously improve inspection performance.

**1. Introduction:** Inspection plays a critical role in airworthiness assurance. It is used as the detection system for required maintenance procedures and as a final check that the maintenance has been performed correctly. Inspection failure at either stage can compromise public safety. A critical defect may remain undetected and thus unrepaired, or on aircraft with a procedural error (e.g. a missing lock-wire) may be released for service.

These issues have been demonstrated in dramatic fashion in aircraft accidents. In 1988 an Aloha Airlines B-737 aircraft suffered fuselage failure from undetected multi-site damage. In addition to aircraft structures, inspection errors have caused engine failures, for example the JT8-D failure on takeoff on a Delta flight from Pensacola in 1998. In both instances the inspection technique was technically capable of detecting the defect (a crack) but the overall system of technology-plus-human inspector failed. These incidents focused attention on the role of the human inspector in the technology-plus-inspector system.

For many years (see Swain, 1990) human factors engineers had been quantifying human reliability using techniques derived from system safety. Fault tree analysis (FTA) and Failures Modes and Effects Analysis (FMEA) had been employed to determine how failures in the human components of a system affected overall system reliability. This set of techniques was first applied to aircraft inspection by Lock and Strutt (1985), who used their detailed task description of inspection to derive potential systems improvements.

Two parallel lines of research also impact on improving human reliability in inspection. First, for many years it has been traditional to measure inspection system reliability in terms of the probability of detecting defects with specified characteristics under carefully controlled conditions. This set of techniques is used to define the inspection system capability, particularly for non-destructive inspection. The second research thread has been the on-going study of human factors in industrial and medical inspection. Early realization that industrial inspectors were not perfectly reliable led to many hundreds of studies aimed at modeling and improving inspection performance.

This paper covers the modeling and improvement of aviation inspection performance, treating human factors as an explicit aspect of inspection capability. Parts of the text that follow are modified from a recent report on one inspection technique, Fluorescent Penetrant Inspection (FPI), published in Drury (1999).

**2. NonDestructive Inspection (NDI) Reliability:** Over the past two decades there have been many studies of human reliability in aircraft structural inspection. All of these to date have examined the reliability of Nondestructive Inspection (NDI) techniques, such as eddy current or ultrasonic technologies.

From NDI reliability studies have come human/machine system detection performance data, typically expressed as a Probability of Detection (PoD) curve, e.g. (Rummel, 1998). This curve expresses the reliability

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of the detection process (PoD) as a function of a variable of structural interest, usually crack length, providing in effect a psychophysical curve as a function of a single parameter. Sophisticated statistical methods (e.g. Hovey and Berens, 1988) have been developed to derive usable PoD curves from relatively sparse data. Because NDI techniques are designed specifically for a single fault type (usually cracks), much of the variance in PoD can be described by just crack length so that the PoD is a realistic reliability measure. It also provides the planning and life management processes with exactly the data required, as structural integrity is largely a function of crack length.

A typical PoD curve has low values for small cracks, a steeply rising section around the crack detection threshold, and level section with a PoD value close to 1.0 at large crack sizes. It is often maintained (e.g. Panhuise, 1989) that the ideal detection system would have a step-function PoD: zero detection below threshold and perfect detection above. In practice, the PoD is a smooth curve, with the 50% detection value representing mean performance and the slope of the curve inversely related to detection variability. The aim is, of course, for a low mean and low variability. In fact, a traditional measure of inspection reliability is the "90/95" point. This is the crack size that will be detected 90% of the time with 95% confidence, and thus is sensitive to both the mean and variability of the PoD curve.

In NDI reliability assessment the model of detecting a signal in noise is one very useful model. Other models of the process exist (Drury, 1992) and have been used in particular circumstances. The signal and noise model assumes that the probability distribution of the detector's response can be modeled as two similar distributions, one for signal-plus-noise (usually referred to as the signal distribution), and one for noise alone. (This "Signal Detection Theory" has also been used as a model of the human inspector, see Section 3.1). For given signal and noise characteristics, the difficulty of detection will depend upon the amount of overlap between these distributions. If there is no overlap at all, a detector response level can be chosen which completely separates signal from noise. If the actual detector response is less than the criterion or "signal" and if it exceeds criterion, this "criterion" level is used by the inspector to respond "no signal." For non-overlapping distributions, perfect performance is possible, i.e. all signals receive the response "signal" for 100% defect detection, and all noise signals receive the response "no signal" for 0% false alarms. More typically, the noise and signal distributions overlap, leading to less than perfect performance, i.e. both missed signals and false alarms.

The distance between the two distributions divided by their (assumed equal) standard deviation gives the signal detection theory measure of discriminability. A discriminability of 0 to 2 gives relatively poor reliability while discriminabilities beyond 3 are considered good. The criterion choice determines the balance between misses and false alarms. Setting a low criterion gives very few misses but large numbers of false alarms. A high criterion gives the opposite effect. In fact, a plot of hits (1 – misses) against false alarms gives a curve known as the Relative Operating Characteristic (or ROC) curve which traces the effect of criterion changes for a given discriminability (see Rummell, Hardy and Cooper, 1989).

The NDE Capabilities Data Book (1997) defines inspection outcomes as:

| | | **Flaw Presence** | |
|---|---|---|---|
| | | **Positive** | **Negative** |
| **NDE Signal** | **Positive** | True Positive<br>No Error | False Positive<br>Type 2 Error |
| | **Negative** | False Negative<br>Type 1 Error | True Negative<br>No Error |

And defines

$$\text{PoD = Probability of Detection} = \frac{TruePositives}{TruePositives + FalseNegatives}$$

$$\text{PoFA = Probability of False Alarm} = \frac{FalsePositives}{TrueNegatives + FalsePositives}$$

The ROC curve traditionally plots PoD against (1 – PoFA). Note that in most inspection tasks, and particularly for engine rotating components, the outcomes have very unequal consequences. A failure to detect (1 – PoD) can lead to engine failure, while a false alarm can lead only to increased costs of needless repeated inspection or needless removal from service.

This background can be applied to any inspection process, and provides the basis of standardized process testing. It is also used as the basis for inspection policy setting throughout aviation. The size of crack reliably detected (e.g. 90/95 criterion), the initial flaw size distribution at manufacture and crack growth rate over time can be combined to determine an interval between inspections which achieves a known balance between inspection cost and probability of component failure.

The PoD and ROC curves differ between different techniques of NDI (including visual inspection) so that the technique specified has a large effect on probability of component failure. The techniques of ROC and PoD analysis can also be applied to changing the inspection configuration, for example the quantitative study of multiple FPI of engine disks by Yang and Donath (1983).

Probability of detection is not just a function of crack size, or even of NDI technique. Early work by Rummel, Rathke, Todd and Mullen (1975) demonstrated that FPI of weld cracks was sensitive to metal treatment after manufacture. The detectable crack size was smaller following a surface etch and smaller still following proof loading of the specimen. This points to the requirement to examine closely <u>all</u> of the steps necessary to inspect an item, and not just those involving the inspector.

**3. Human Factor in Inspection:** Human factors studies of industrial inspection go back to the 1950's when psychologists attempted to understand and improve this notoriously error-prone activity. From this activity came literature of increasing depth focusing an analysis and modeling of inspection performance, which complemented the quality control literature by showing how defect detection could be improved. Two early books brought much of this accumulated knowledge to practitioners: Harris and Chaney (1969) and Drury and Fox (1975). Much of the practical focus at that time was on enhanced inspection techniques or job aids, while the scientific focus was on application of psychological constructs, such as vigilance and signal detection theory, to modeling of the inspection task.

As a way of providing a relevant context, we use the generic functions which comprise all inspection tasks whether manual, automated or hybrid (Drury, 1992). Table 1 shows these functions, with an example from fluorescent penetrant inspection. We can go further by taking each function and listing its correct outcome, from which we can logically derive the possible errors (Table 2).

**Table 1. Generic Task Description of Inspection Applied to Fluorescent Penetrant Inspection.**

| Function | Description |
|---|---|
| 1. Initiate | All processes up to visual examination of component in reading booth. Get and read workcard. Check part number and serial number. Prepare inspection tools. Check booth lighting. Wait for eyes to adapt to low light level. |
| 2. Access | Position component for inspection. Reposition as needed throughout inspection. |
| 3. Search | Visually scan component to check cleaning adequacy. Carefully scan component using a good strategy. Stop search if an indication is found. |
| 4. Decision | Compare indication to standards for crack. Use re-bleed process to differentiate cracks from other features. Confirm with white light and magnifying loupe. |
| 5. Response | If cleaning is below standard, then return to cleaning. If indication confirmed, then mark extent on component. Complete paperwork procedures and remove component from booth. |

Humans can operate at several different levels in each function depending upon the requirements. Thus, in Search, the operator functions as a low-level detector of indications, but also as a high-level cognitive component when choosing and modifying a search pattern. It is this ability which makes humans uniquely useful as self-reprogramming devices, but equally it leads to more error possibilities. As a framework for examining inspection functions at different levels the skills/rules/knowledge classification of Rasmussen

(1983) will be used. Within this system, decisions are made at the lowest possible level, with progression to higher levels only being invoked when no decision is possible at the lower level.

For most of the functions, operation at all levels is possible. Presenting an item for inspection is an almost purely mechanical function, so that only skill-based behavior is appropriate. The response function is also typically skill-based, unless complex diagnosis of the defect is required beyond mere detection and reporting.

**Table 2. Generic Function, Outcome, and Error Analysis of Test Inspection.**

| Function | Outcome | Logical Errors |
|---|---|---|
| Initiate | Inspection system functional, correctly calibrated and capable. | 1.1 Incorrect equipment<br>1.2 Non-working equipment<br>1.3 Incorrect calibration<br>1.4 Incorrect or inadequate system knowledge |
| Access | Item (or process) presented to inspection system | 2.1 Wrong item presented<br>2.2 Item mis-presented<br>2.3 Item damaged by presentation |
| Search | Individuals of all possible non-conformities detected, located | 3.1 Indication missed<br>3.2 False indication detected<br>3.3 Indication mis-located<br>3.4. Indication forgotten before decision |
| Decision | All individuals located by Search, correctly measured and classified, correct outcome decision reacted | 4.1 Indication incorrectly measured/confirmed<br>4.2 Indication incorrectly classified<br>4.3 Wrong outcome decision<br>4.4 Indication not processed |
| Response | Action specified by outcome decision taken correctly | 5.1 Non-conforming action taken on conforming item<br>5.2 Conforming action taken on non-conforming item |

*3.1 Critical Functions: search and decision:* The functions of search and decision are the most error-prone in general, although for much of NDI, setup can cause its own unique errors. Search and decision have been the subjects of considerable mathematical modeling in the human factors community, with direct relevance to airframe and engine inspection.

In FPI, visual inspection and X-ray inspection, the inspector must move his/her eyes around the item to be inspected to ensure that any defect will eventually appear within an area around the line of sight in which it is possible to have detection. This area, called the visual lobe, varies in size depending upon target and background characteristics, illumination and the individual inspector's peripheral visual acuity. As successive fixations of the visual lobe on different points occur at about three per second, it is possible to determine how many fixations are required for complete coverage of the area to be searched.

Eye movement studies of inspectors show that they do not follow a simple pattern in searching an object. Some tasks have very random appearing search patterns (e.g., circuit boards), whereas others show some systematic search components in addition to this random pattern (e.g., knitwear). However, all who have studied eye movements agree that performance, measured by the probability of detecting an imperfection in a given time, is predictable assuming a random search model. The equation relating probability ($p_1$) of detection of an imperfection in a time (t) to that time is

$$\mathrm{p}_{\mathrm{t}} = 1 - \exp\left(-\frac{t}{\bar{t}}\right)$$

where $\bar{t}$ is the mean search time. Further, it can be shown that this mean search time can be expressed as

$$\bar{t} = \frac{t_o A}{apn}$$

where

$t_o$ = average time for one fixation
A = area of object searched
a = area of the visual lobe
p = probability that an imperfection will be detected if it is fixated. (This depends on how the lobe (a) is defined. It is often defined such that p = ½. This is an area with a 50% chance of detecting an imperfection.
n = number of imperfections on the object.

From these equations we can deduce that there is speed/accuracy tradeoff (SATO) in visual search, so that if insufficient time is spent in search, defects may be missed. We can also determine what factors affect search performance, and modify them accordingly. Thus the area to be searched (A) is a direct driver of mean search time. Anything we can do to reduce this area, e.g. by instructions about which parts of an object not to search, will help performance. Visual lobe area needs to be maximized to reduce mean search time, or alternatively to increase detection for a given search time. Visual lobe size can be increased by enhancing target background contrast (e.g. using the correct developer in FPI) and by decreasing background clutter (e.g. by more careful cleaning before FPI). It can also be increased by choosing operators with higher peripheral visual acuity (Eriksen,1990) and by training operators specifically in visual search or lobe size improvement (Drury, Prabhu and Gramopadhye,1990). Research has shown that there is little to be gained by reducing the time for each fixation, $t_o$ , as it is not a valid selection criterion, and cannot easily be trained.

The equation given for search performance assumed random search, which is always less efficient than systematic search. Human search strategy has proven to be quite difficult to train, but recently Wang, Lin and Drury (1997) showed that people can be trained to perform more systematic visual search. Also, Gramopadhye, Prabhu and Sharit (1997) showed that particular forms of feedback can make search more systematic.

Decision-making is the second key function in inspection. An inspection decision can have four outcomes, as shown in Table 3. These outcomes have associated probabilities, for example the probability of detection is the fraction of all nonconforming items which are rejected by the inspector shown as $p_2$ in Table 3.

**Table 3. Attributes Inspection Outcomes and Probabilities.**

| | **True State of Item** | |
|---|---|---|
| **Decision of Inspector** | **Conforming** | **Nonconforming** |
| Accept | Correct accept, $p_1$ | Miss, (1 - $p_2$) |
| Reject | False alarm, (1 - $p_1$) | Hit, $p_2$ |

Just as the four outcomes of a decision-making inspection can have probabilities associated with them, they can have costs and rewards also: costs for errors and rewards for correct decisions. Table 4 shows a general cost and reward structure, usually called a "payoff matrix," in which rewards are positive and costs negative. A rational economic maximizer would multiply the probabilities of Table 3 by the corresponding payoffs in Table 4 and sum them over the four outcomes to obtain the expected payoff. He or she would then adjust those factors under his or her control. Basically, SDT states that $p_1$ and $p_2$ vary in two ways. First, if the inspector and task are kept constant, then as $p_1$ increases, $p_2$ decreases, with the balance between $p_1$ and $p_2$ together by changing the discriminability for the inspector between acceptable and rejectable objects. $p_1$ and $p_2$ can be changed by the inspector. The most often tested set of assumptions comes from a body of knowledge known as the theory of signal detection, or SDT (McNichol, 1972). This theory has been used for numerous studies of inspection, for example, sheet glass, electrical components, and ceramic gas igniters, and has been found to be a useful way of measuring and predicting performance. It can be used in a rather general nonparametric form (preferable) but is often seen in a more restrictive parametric form in earlier papers (Drury and Addison, 1963). McNichol (1972) is a good source for details of both forms.

**Table 4. Payoff Matrix for Attributes Inspection.**

| | **True State of Item** | |
|---|---|---|
| **Decision of Inspector** | **Conforming** | **Nonconforming** |
| Accept | A | -b |
| Reject | -c | D |

The objective in improving decision making is to reduce decision errors. There can arise directly from forgetting imperfections or standards in complex inspection tasks or indirectly from making an incorrect judgement about an imperfection's severity with respect to a standard. Ideally, the search process should be designed so as to improve the conspicuity of rejectable imperfections (nonconformities) only, but often the measures taken to improve conspicuity apply equally to nonrejectable imperfections. Reducing decision errors usually reduces to improving the discriminability between imperfection and a standard.

Decision performance can be improved by providing job aids and training which increase the size of the apparent difference between the imperfections and the standard (i.e. increasing discriminability). One example is the provision of limit standards well integrated into the inspector's view of the item inspected. Limit standards change the decision-making task from one of absolute judgement to the more accurate one of comparative judgement. Harris and Chaney (1969) showed that limit standards for solder joints gave a 100% performance improvement in inspector consistency for near-borderline cases.

One area of human decision-making that has received much attention is the vigilance phenomenon. It has been known for half a century that as time on task increases, then the probability of detecting perceptually difficult events decreases. This has been called the vigilance decrement and is a robust phenomenon to demonstrate in the laboratory. Detection performance decreases rapidly over the first 20-30 minutes of a vigilance task, and remains at a lower level as time or task increases. Note that there is not a period of good performance followed by a sudden drop: performance gradually worsens until it reaches a steady low level. Vigilance decrements are worse for rare events, for difficult detection tasks, when no feedback of performance is given, and where the person is in social isolation. All of these factors are present to some extent in FPI, so that prolonged vigilance is potentially important here.

A difficulty arises when this body of knowledge is applied to inspection tasks in practice. There is no guarantee that vigilance tasks are good models of inspection tasks, so that the validity of drawing conclusions about vigilance decrements in inspection must be empirically tested. Unfortunately, the evidence for inspection decrements is largely negative. A few studies, e.g. for chicken carcass inspection (Chapman and Sinclair, 1975) report positive results but most, e.g. eddy current NDI (Spencer and Schurman, 1995; Murgatroyd, Worrall and Waites, 1994) find no vigilance decrement.

It should be noted that inspection is not merely the decision function. The use of models such as signal detection theory to apply to the whole inspection process is misleading in that it ignores the search function. For example, if the search is poor, then many defects will not be located. At the overall level of the inspection task, this means that PoD decreases, but this decrease has nothing to do with setting the wrong decision criteria. Even such devices as ROC curves should only be applied to the decision function of inspection, not to the overall process unless search failure can be ruled out on logical grounds.

**4. NDI/Human Factors Links:** As noted earlier, human factors has been considered for some time in NDI reliability. This often takes the form of measures of inter-inspector variability (e.g. Herr and Marsh, 1978), or discussion of personnel training and certification (Herr and Marsh, 1978). There have been more systematic applications, such as Lock and Strutt's (1990) classic study from a human reliability perspective, or the initial work on the FAA/Office of Aviation Medicine (AAM) project reported by Drury, Prabhu and Gramopadhye (1990). A logical task breakdown of NDI was used by Webster (1988) to apply human factors data such as vigilance research to NDI reliability. He was able to derive errors at each stage of the process of ultrasonic inspection and thus propose some control strategies.

A more recent example from visual inspection is the Sandia National Laboratories (SNL/AANC) experiment on defect detection on their B-737 test bed (Spencer, Drury and Schurman, 1996). The study used twelve experienced inspectors from major airlines, who were given the task of visually inspecting ten different areas. Nine areas were on AANC's Boeing 737 test bed and one was on the set of simulated fuselage panels containing cracks which had been used for the earlier eddy-current study.

In a final example an analysis was made of inspection errors into search and decision errors (Table 5), using a technique first applied to turbine engine bearing inspection in a manufacturing plant. This analysis enables us to attribute errors to either a search failure (inspector never saw the indication) or decision failure (inspector saw the indication but came to the wrong decision). With such an analysis, a choice of interventions can be made between measures to improve search or (usually different) measures to improve decision. Such an analysis was applied to the eleven inspectors for whom usable tapes were available from the cracked fuselage panels inspection task.

**Table 5. Observed NDI errors from classified by their function and cause (Murgatroyd et al, 1994).**

| Function | Error Type | Etiology/Causes | Miss | False Alarm |
|---|---|---|---|---|
| 3. Search | 3.1 Motor failure in probe movement | Not clamping straight edge<br>Mis-clamping straight edge<br>Speed/accuracy tradeoff | X<br>X<br>X | X |
| | 3.2 Fail to search sub-area | Stopped, then restarted at wrong point | X | |
| | 3.3 Fail to observe display | Distracted by outside event<br>Distracted by own secondary task | X<br>X | |
| | 3.4 Fail to perceive signal | Low-amplitude signal | X | |
| 4. Decision | 4.1 Fail to re-check area | Does not go back far enough in cluster, missing first defect | | |
| | 4.2 Fail to interpret signal correctly | .Marks nonsignals with ?<br>Notes signals but interprets it as noise<br>Mis-classifies signal | <br><br>X | X<br>X<br>X |
| 5. Response | 5.2 Mark wrong rivet | Marks between 2 fasteners | X | |

The results of this analysis are shown in Table 6. Note the relatively consistent, although poor, search performance of the inspectors on these relatively small cracks. In contrast, note the wide variability in decision performance shown in the final two columns. Some inspectors (e.g. B) made many misses and few false alarms. Others (e.g. F) made few or no misses but many or even all false alarms. Two inspectors made perfect decisions (E and G). These results suggest that the search skills of <u>all</u> inspectors need improvement, whereas specific individual inspectors need specific training to improve the two decision measures.

**Table 6. Search and decision failure probabilities on simulated fuselage panel inspection (derived from Spencer, Drury and Schurman, 1996).**

| Inspector | Probability of Search Failure | Probability of Decision Failure (miss) | Probability of Decision Failure (false alarm) |
|---|---|---|---|
| A | 0.31 | 0.27 | 0.14 |
| B | 0.51 | 0.66 | 0.11 |
| C | 0.47 | 0.31 | 0.26 |
| D | 0.44 | 0.07 | 0.42 |
| E | 0.52 | 0.00 | 0.00 |
| F | 0.40 | 0.00 | 1.00 |
| G | 0.47 | 0.00 | 0.00 |
| H | 0.66 | 0.03 | 0.84 |
| I | 0.64 | 0.23 | 0.80 |
| J | 0.64 | 0.07 | 0.17 |
| K | 0.64 | 0.17 | 0.22 |

With linkages between NDI reliability and human factors such as these given above, it is now possible to derive a more detailed methodology for this project.

**5. Practical Issues in Inspection Human Factors:** As can be seen from the review of human factors in inspection, a number of interventions is derivable from models and field data. Human factors recognizes that any system comprises several components that must work together harmoniously to ensure system performance and human well being. There have been several proposed taxonomies of system components, including ICAO's SHELL model, but here we will use the TOMES model for simplicity: Task/ Operator/ Machine/ Environment/ Social. For detailed reference on each see, for example Drury (1992).

***5.1 Task Interventions***. The task comprises all of the steps necessary to perform the inspection reliability. Task factors affecting performance include:

- Time available for task completion. Because search is resource-limited, overall probability of detection is very sensitive to time limitations. In particular, external pacing of inspection tasks increases errors.

- Nature of defect. Some detects are inherently more difficult to find than others. In addition, defect size is a major driver of probability of detection. This makes early detection of progressive defects such as cracks and corrosion difficult.

- Mix of defects. If the inspector must search simultaneously for several defects, performance on detecting any particular defect decreases.

- Probability of a defect. As noted under decision models, inspectors have a higher probability of detection where a defect is more likely. Conversely, rare defects are very difficult to detect, providing an ultimate limit to human inspection performance.

***5.2 Operator Interventions***. The operator here is usually the inspector, although others involved for example in set-up or part cleaning may also be operators.

- Selection and Placement. Historically there has been a continuing interest in providing tools to select a "good" inspector. However, such efforts have been largely unsucessful when "good" is defined in terms of detection probability. A primary reason has been that performance of inspectors is task-dependent, with no guarantee that an inspector who performs well on inspection task A will also perform well on task B.

- Training. Human factors engineers have had considerable success in using the generic inspection functions (Tables 1, 2) as the basis for improved training. Both in manufacturing industry (Kleiner and Drury, 1993) and in aviation maintenance (Gramopadhye, Drury and Sharit, 1997) such training must cover search strategy as well as decision criteria if it is to be effective.

*5.3 Machine Interventions*. Hardware and software aspects of the task inspection.

- Inspection object handling. If the component inspected is difficult to reach or has poor visual access, inspection performance will suffer to some extent. Access is limited by aircraft and engine design factors, but steps can be taken for improvement. Examples include customized access stands for airframe inspection, easily maneuverable hangers for engine components and improved mirrors/loupes.
- Software aspects of inspection cover the design of documentation such as workcards, manuals and service bulletins. Poor wording and layout of these documents, or their computer equivalents, can have a major effect on error rates (Patel, Prabhu, and Drury, 1992).

***5.4 Environment Interventions:*** All inspection takes place in a physical environment (this section) and a social environment (following section).

- Visual environment. Obviously, enough light must be available for inspection, but performance typically depends more on the quality of the visual environment than the intensity of illumination. Lighting must be developed to maximize the probability of defect detection.
- Other environmental factors. Human performance decreases in adverse noise and thermal environments. For inspectors, such adverse conditions are common, both in line inspection and within the maintenance hangar.

*5.5 Social Interventions*. Inspection is part of a socio-technical system of aircraft maintenance, so that relationships between the inspector and others will influence inspection performance.

- Management interactions. If inspectors' decis-ions are contradicted by management, then the inspectors are likely to change their decision criteria for reporting defects (see Section 3.1). Most inspectors are fiercely independent, and their departmental managers respect this. But external pressures for hurried work will have obvious effects on inspection reliability.
- Peer interactions. Inspectors hand off work whenever a shift changes or an interruption occurs. The handover procedures have been implicated in incident and accident reports so that good practices need to be followed whenever ownership of a job changes.
- Working hours. Inspection demands continuous vigilance, which is a cognitively demanding activity. People do not perform well during long hours of work. Nor do they perform well when sleep patterns are disrupted. Yet much inspection is carried out on night shifts, and large amounts of overtime are common during initial inspection. Neither practice helps inspection reliability.

**6. Conclusions:** Airframe and engine inspection is a complex activity dependent upon its human and hardware components alike for its reliability. Human factors engineers have developed useful models of the generic tasks in inspection. Such models can be used both to guide field investigation of inspection tasks and to predict those factors having the greatest impact on inspection reliability. Using this approach it is possible to derive good practices to improve inspection performance. For one unique inspection task, Fluorescent Penetrant Inspection, a set of good practices has been derived and is available at www.hfskyway.com.

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# Extension of the Usable Engine Life by Modelling and Monitoring

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## Summary

After providing some commonly used definitions of fracture critical parts, the influence of improved calculation methods on the design of such components is shown. Various approaches to the determination of usable fatigue life are discussed, particularly the classical safe life philosophy and approaches exploiting the damage tolerance of components. Within this general framework there exist various possible lifing policies, that have to be discussed and agreed between the engine manufacturer, the users and the regulatory agencies. The methods for life usage management may be adapted to changing environments, taking into account the experience gained during operational usage. The introduction of recording or monitoring systems makes it possible to calculate the actual life usage of individual components or at least to determine the scatter of usage within an aircraft fleet. These results enable a specific exploitation of the life potential of the parts without giving rise to an increased risk. The use of the life potential beyond the safe crack initiation life requires experimental and computational methods to gain insight into the fracture mechanical processes governing crack propagation. The corresponding results can also be used to determine inspection intervals that ensure a detection of cracks before those cracks start uncontrolled growth. Results from an on-board life usage monitoring system used by the German air force are presented. An outline of the tasks of usage monitoring is given. Finally some remarks on fleet management are presented.

## Introduction

A considerable percentage of the military aircraft and engines now in use have experienced operation over usage times not foreseen when those engines were designed. In Germany we have the F-4F (Phantom II), designed around 1965, that entered service at the German air force in 1974 with its J79 engine. The same engine also powers the F-104S-ASU (Starfighter) still in use at the Italian air force. Both types will continue to fly until being replaced by the Eurofighter. The French air force continues to use the Alphajet and the Mirage F1, both with engines designed in the early 70's. Although many of the older MiG and Su types have been withdrawn from service in the last years, there remains a large amount of aircraft that have seen more than a quarter of a century of operation in the air forces of Eastern Europe. The vast majority of aircraft operated by the NATO nations is more than 15 years old. As engines typically contribute 30% of the life cycle cost of an aircraft, methods aiming at an extension of their usable life attract a widespread attention. There have been various conferences and working groups on this topic initiated by the RTO [RTOMP17, RTOTR28]. All topics of this presentation are addressed in great detail in unclassified sources, some of which are put together in the references, and I strongly recommend to retrieve at least some of the available material from the world wide web.

## Definition of fracture critical parts

A typical definition used by regulatory agencies in civil aviation is the one given in [JAR-E]: "Where the failure analysis shows that a part must achieve and maintain a particularly high level of integrity if hazardous effects are not to occur at a rate in excess of Extremely Remote then such a part shall be identified as a Critical Part". "Extremely

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**Fig. 1**: Largest fragment of fan hub after burst (Pensacola accident)

**Fig. 2:** Disk driven to burst during overspeed spin test

Remote" probability means [JAR1] "unlikely to occur when considering the total operational life of a number of aircraft of the type in which the engine is installed, but nevertheless, has to be regarded as being possible ($10^{-7}$ - $10^{-9}$ per hour of flight)". In the glossary of [RTOTR28] the following definition is given: "A part which will physically break, causing catastrophic damage, after experiencing a statistically described number and mix of missions. Such components are identified at design time, and removed from service before failure occurs."

The engine parts most likely to cause severe damage to the aircraft are the components of the rotors, the most massive ones being the compressor and turbine disks, but also including spacer rings or rotating air seals, that may sometimes also penetrate the engine casing when a failure occurs. Engine design usually is required to ensure containment of broken single blades, but also numerous incidents with uncontained fan or turbine blades have been reported (e.g. [WB96, JSSG2007]). Although the focus will be on disks, many aspects of the following presentation are applicable to blades as well. Unfortunately, actual disk failures are not limited to experiments performed in the test beds of engine manufacturers (see example in Fig. 2), but they also happen in the engines of commercial passenger jets with hundreds of passengers aboard. Fig. 1 shows a fragment of the fan hub, whose failure killed two passengers aboard a MD-88 in Pensacola, Florida, in 1996 [NTSB98]. There is a not too short list of other uncontained disk failures in the engines of civil aircraft. Many of them occurred during the run-up to takeoff power of the engines on ground, thus limiting somewhat the possible consequences, but aircraft have been destroyed by subsequent fires, as in the Valuejet accident in 1995 [NTSB96], and there was one catastrophic in-flight failure of a fan hub, claiming the lives of 111 persons. The direct cost of that crash of a DC-10 in Sioux City in 1989 totaled over 300 million US$ [Hall97].

A recent (June 2000) uncontained failure of the HP compressor spool of a GE CF6-80, that happened during takeoff of a Boeing 767 in Sao Paulo, Brazil, led to the recommendation to remove certain engines from service to perform inspections to detect possible cracking.

The military flying community is affected by failing fracture critical parts as well. The following description, which gives a typical example for the consequences of an uncontained disk failure during flight is cited from the Flying Safety Magazine of the United States Air Force [Woo96]:

“The one T-38 engine-related Class A mishap was from another known problem, compressor disk corrosion. A crack propagated from a corrosion pit in the No. 1 engine's eighth stage compressor disk. When the disk eventually failed, it penetrated the case, severed several fuel and oil lines, and caused an in-flight fire. The shrapnel and fire affected the mishap aircraft's flight controls, forcing the crew to eject. The aircraft crashed in an apartment complex, killing two and injuring several other civilians. The source of the corrosion is still unknown. Oddly enough, no other users of the J85 engine have reported corrosion, including the Navy. Regardless, life limit reductions are being implemented to reduce the risk. Corrosion-resistant coatings and materials are also being explored.” This short report bears nearly all ingredients of what may happen and what consequences are typically deduced. It also highlights the problem of corrosion, that may invalidate the results of sophisticated life extension schemes.

Disregarding disk failures due to overspeed which might occur after a total malfunction of the control or fuel system or due to a broken shaft in the engine, disk failures usually are the final consequence of underestimated and undetected material fatigue. Even in initially defect-free parts cracks may start to grow at highly loaded areas of the rotor structure. If cracks remain undetected and operation of the part is continued, even normal cyclic loading will eventually lead to an unstable crack growth. The final burst, that results from insufficient residual strength of the heavily cracked disk under high load will produce a few (typically 3 - see Fig. 2 and [DK99]) high energy fragments, that will inevitably penetrate the engine casing, with a high chance for mission abort, air vehicle loss, and fatalities.

## Evolution of the design process

The tools available for the design of aero engine components have seen a dramatic evolution during the past 30 years. The finite element (FE) method is now used to determine (and avoid) in advance the locations of high stress concentrations. For most of the old engines, that were designed before 1970, such computer based tools were only available to a very limited extent. At that time the design of rotors was largely based on empirical methods, supported by experiments (e.g. photoelastic strain analysis). The limited accuracy of the available computation methods for temperatures and stresses had to be compensated by the selection of larger safety margins. Only since the middle of the 70's FE programs were used for the stress calculation of rotor components. It became possible to design disks which were stressed more evenly, than it was possible with the classical empirical procedures. Modern design methods try to minimize weight and to fully exploit the available strength of materials. “Old” rotor components often have only a few, clearly identified critical life-limiting areas, whereas a larger number of potentially life-limiting areas has to be taken into consideration for newly designed “fully stressed” components.

The evolution of computational methods within the last 20 years is illustrated in Figs. 3 and 4. Fig. 3 shows a finite element mesh that was used for the axisymmetric stress analysis of the IP compressor rotor of the RB199 engine during rotor design in the year 1979. The same component has been reassessed in 1999 to check for possible life extensions. The refined mesh shown in Fig. 4 partially removes the necessity to apply empirical stress concentration factors to take into account geometric details not resolved by the coarse mesh. In 1979 stress engineers had to wait days for the computation results. With modern workstations a time dependent stress analysis, including thermal stresses and a life assessment for the whole design mission can be run within a few minutes. Fig. 5 shows an example

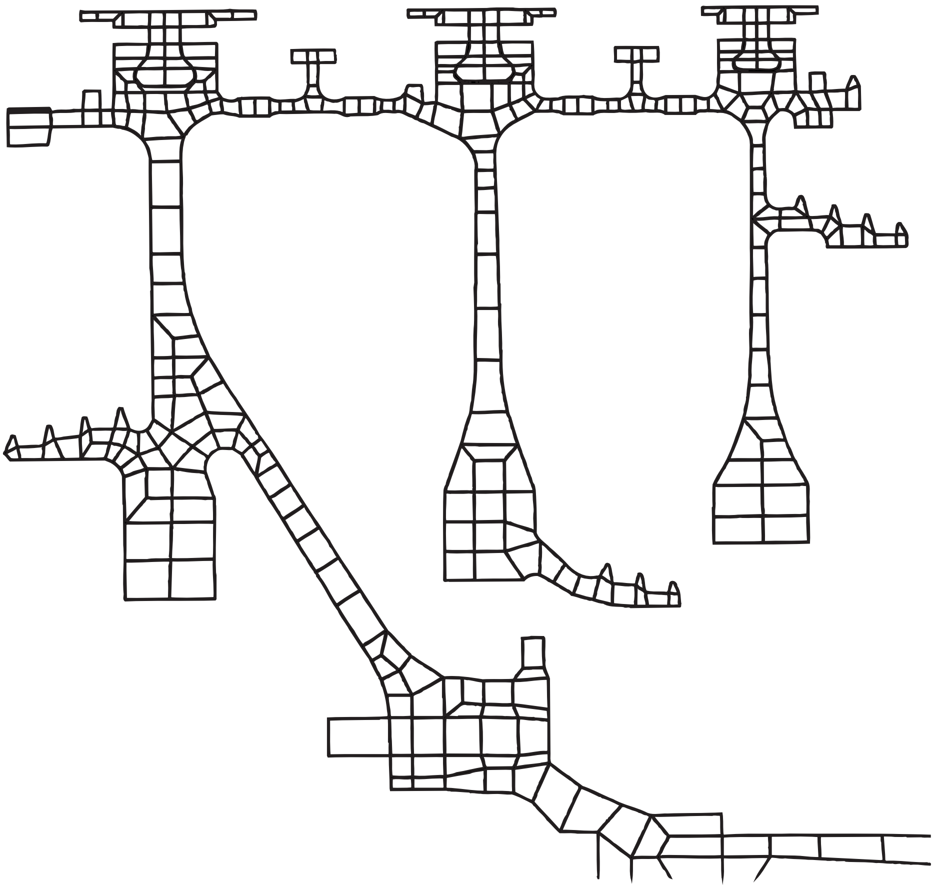


**Fig. 3:** Compressor rotor, finite element mesh for axi-symmetric stress calculation in 1979

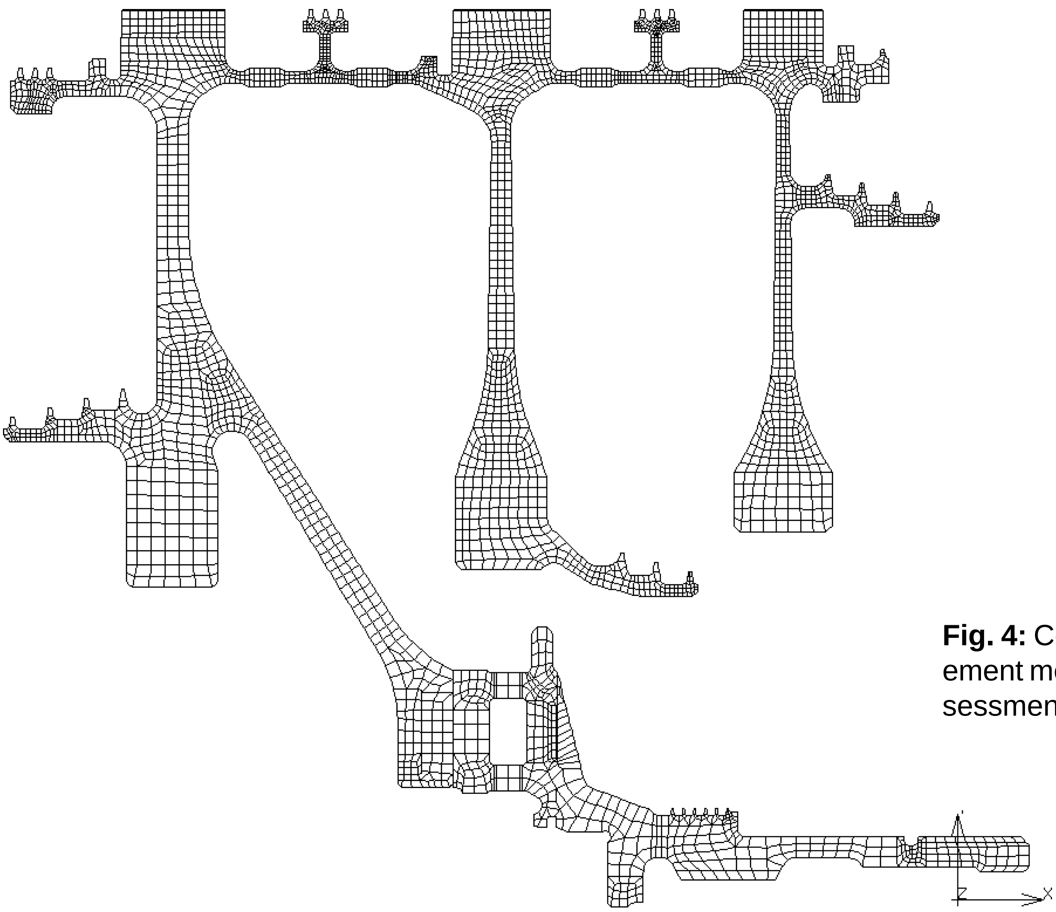


**Fig. 4:** Compressor rotor, finite element mesh used for recent re-assessment of stresses

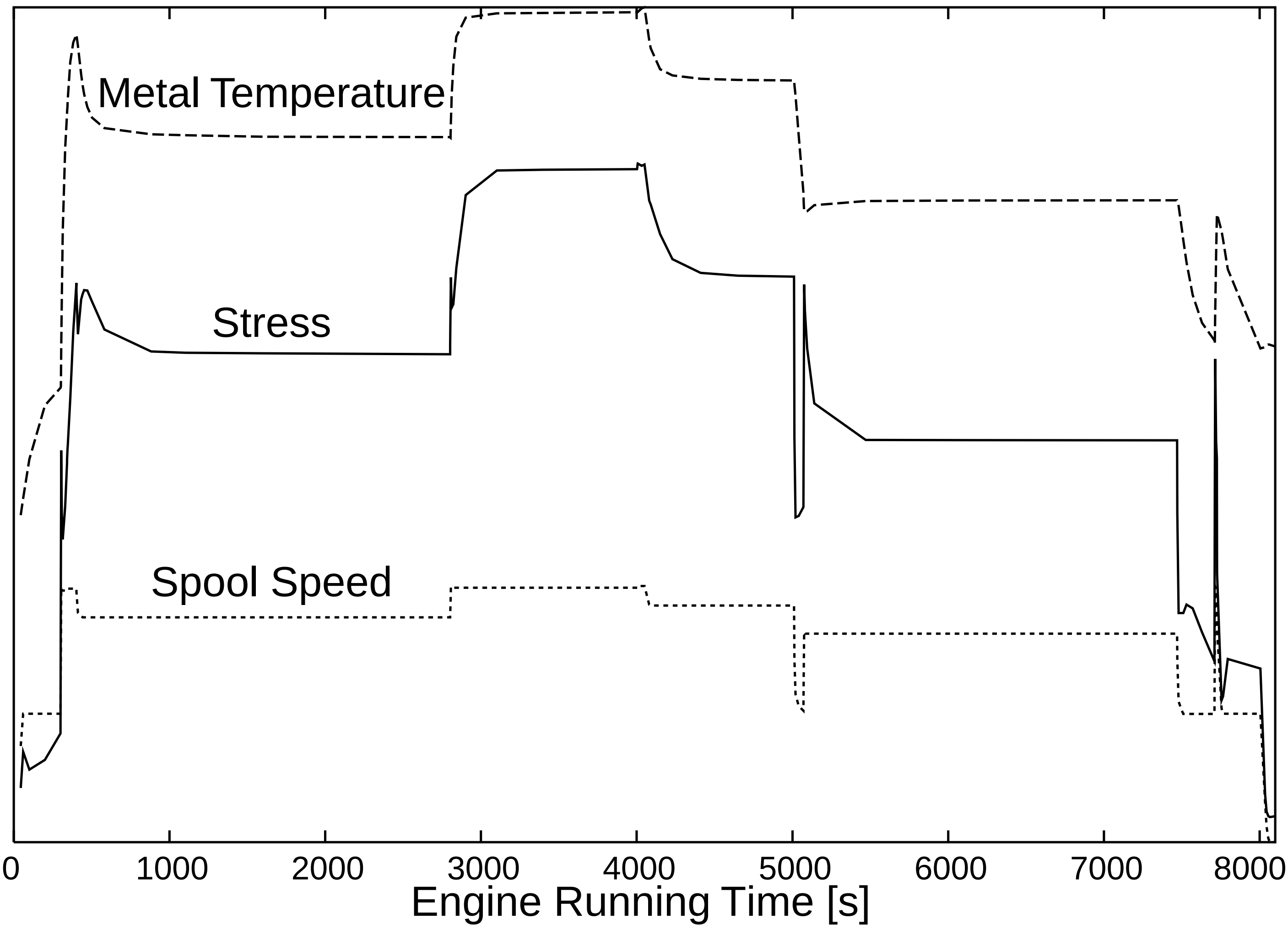



**Fig. 5:** Design mission: Results of finite element calculation for one critical area

of the calculated temperature and stress at a critical area, together with the spool speed for the design mission. Per definition, the largest stress cycle of this mission is used as reference for LCF life counting for a certain critical area. The life usage of this cycle is set to a value of 1.0. Life releases are expressed as multiples of this cycle.

## Safe life versus damage tolerance design

Currently the most widely used lifing policy is that of “Safe Crack Initiation Life”. This is the classical method for lifing in the low cycle fatigue regime. The idea of the concept is as follows: It is assumed, that a new part is free of defects. Under operational loading a defect (e.g. a fatigue crack) is generated. When the defect has been generated, the part’s life is expired. The life end criterion is a certain predefined crack depth (e.g. 0.4 mm). The usable life is defined as the life of the weakest individuum of a population of parts. As a result of experiments and experience, a lognormal distribution of lives to first crack (LTFC) with a ±3sigma scatter factor of 6 is assumed for conventional disk materials. The method is illustrated in Fig. 6, showing schematically the scatter band of the S/N curves of a large number of similar parts. Due to the requirement for an acceptably low statistical probability (e.g. 1/750) for the existence of a crack with the predefined depth, only a fixed portion of the average life is available for operational use. There is no check for the presence of the life limiting crack, when parts are retired. Details of the method are discussed in [BLH98]. One shortcoming of this method is its inability to predict a failure margin. The method tacitly assumes, that a 0.4 mm crack is sufficiently far away from growing in an uncontrolled manner. This is the starting point for the so-called damage tolerance concepts. It is assumed, that even a new part may have an initial defect, which behaves like a crack

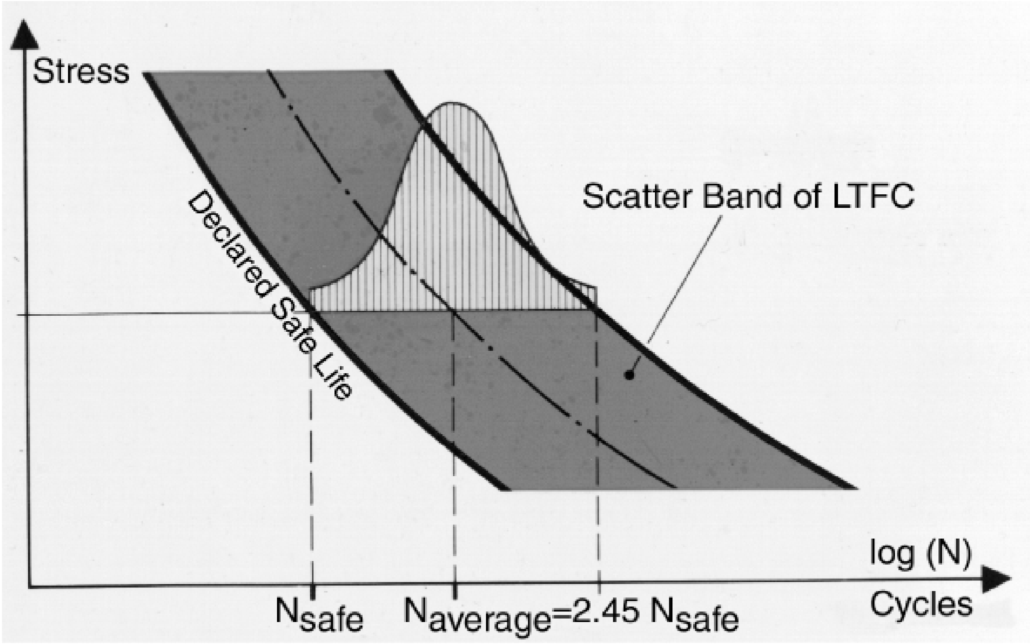



**Fig. 6:** Distribution of cycles to crack initiation

of a certain depth. This crack propagates under operational loading. When the crack enters a phase of unstable growth, the part's life is expired. The application of a damage tolerant lifing policy requires an understanding of the crack growth process. Experiments and fracture mechanical methods have to be combined to determine the number of load cycles needed to propagate cracks at the critical areas from the assumed initial size to a size implying the risk of disk burst (dysfunction). There are various criteria for dysfunction of a part [BB98]: Unstable crack growth under basic operational loading, onset of continuous crack propagation due to superimposed vibratory stresses, loss of overspeed capability (insufficient residual strength), unacceptable out-of-balance conditions.

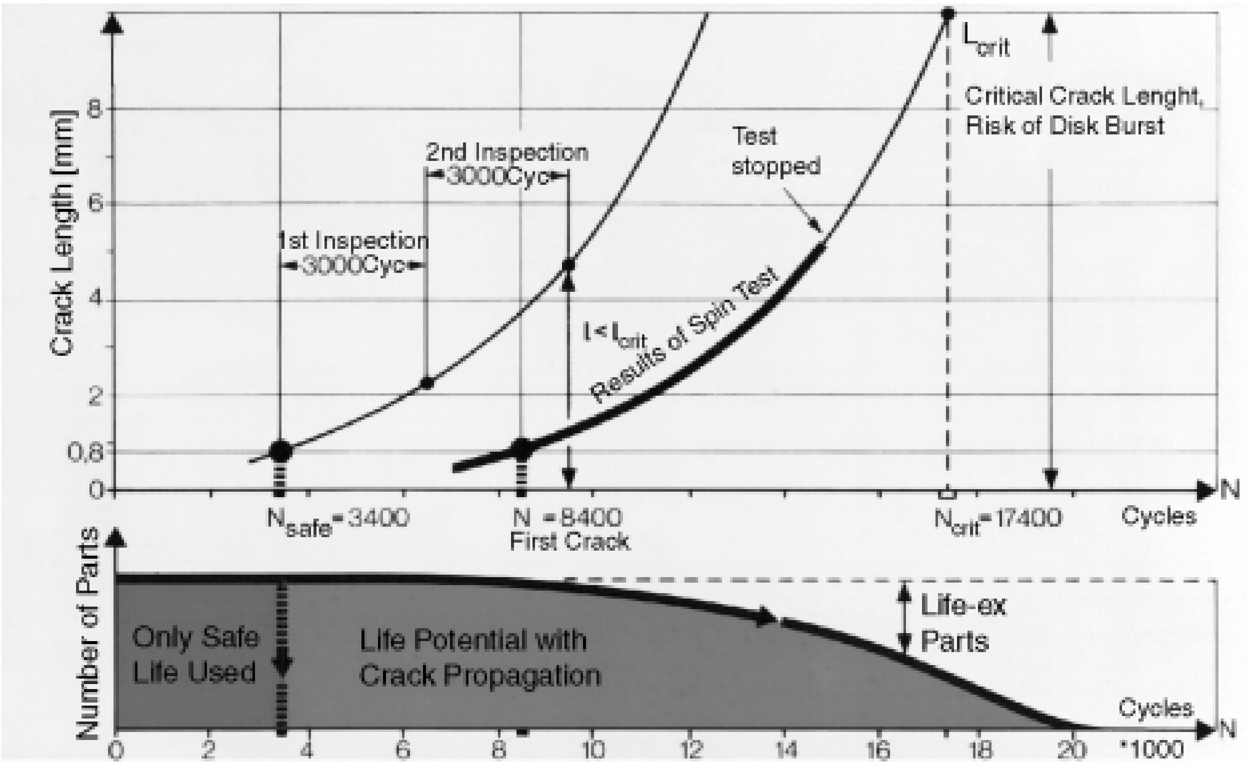



**Fig. 7:** Life extension into the crack propagation regime

The information about crack growth can be used in different manners. The first method is an extension of the “Safe Initiation Life” concept, called the “Safe Crack Propagation Life”. This method is described in [BB98]. The “Initiation Life” criterion (e.g. occurrence of a 0.4 mm crack) is replaced by the “2/3 Dysfunction Life” criterion. For crack tolerant components having a long crack propagation life, significant life extensions compared to “Initiation Life” are possible. On the other hand the application of the “2/3 dysfunction life criterion” to components with low damage tolerance may even require a reduction of the life figures derived from the “Initiation Life” criterion. This is necessary to ensure a consistent safety margin.

All of the methods described so far do not exploit information from inspections. Parts are scrapped when they have reached their released lives irrespective of the actual presence of cracks. If reliable nondestructive inspection (NDI) methods are available, that are able to guarantee defect sizes below prescribed limits, then the method illustrated in Fig. 7 is applicable. A part can be returned into service, if it is found defect-free or the defects are so small, that the expected crack propagation period is longer than the planned inspection interval.

# Lifing policies

The lifing policy that will be applied to a new engine is usually discussed and agreed by the contractors. As already mentioned, the most commonly used lifing policy in Europe is the "Safe Life" approach. In this method only a chosen percentage (e.g. 50%) of the calculated expected life is released at entry into service of a new engine. This applies also to the introduction of engine or component modifications, that are judged to significantly influence the life of the affected components. Evidence has to be provided by performing spin pit tests with full scale components, by which the component is subjected to a series of cyclic load changes. The load levels are chosen to exceed the expected operational loads by a chosen, usually moderate percentage. This overload provides some safety margin against uncertainties in the stress calculations and it also serves to shorten test times, that are a substantial cost factor during component qualification. Spin tests are continued until cracks start to grow at critical areas of the disks.

If the need arises, the test may also be continued into the crack propagation regime, however requiring some extensions of the experimental planning and evaluation (e.g. application of marker loads, determination of crack geometry). This is necessary to produce the data required for a prospective inclusion of the parts crack growth potential into an extended life release.

Dependent on the number of tested disks and on the achieved number of test cycles, at first only a certain percentage of the life demonstrated in the spin test is released. Safety factors have to be included taking into account the very small (typically not more than 2 or 3) sample size. Spin tests are continued in parallel to the operation of the engines in the users' fleets. Based on an extrapolation of usage data, that may either consist of cycles accumulated by engine monitoring systems or simply based on the number of flights or accumulated flight time, required schedules for life releases are determined. When the first components have reached the life released so far, at least one of those components is removed from service and it is then tested for remaining life in a cyclic spin test. The new results are used to release a further portion of the life. This process is continued, until 100% of life can be released.

The formal life release usually is authorized by a regulatory agency based on available evidence. In contrast to commercial aviation, where international standards (e.g. [JAR-E]) are applied, it is quite common in military aviation, that different users of the same engine types use different methods of life releases or even different lifing policies. Lifing philosophies may be different within one country: In the USA, the US Air Force practices the damage tolerance philosophy, whereas the US Navy practices the safe life approach.

There is a number of reasons for an enhancement or modification of the existing methods. Possible extension are the inclusion of test results from other sources than spin-pit tests, e.g. specimen tests, an improved statistical approach [BLH98] to take into account so-called non-finite test results (i.e. no crack has occurred at some intermediate stage of the cyclic spin pit tests).

Experience from a number of projects shows, that it is not always possible to take into account all life limiting areas in the initial assessment of a new or redesigned component. If cracks occur at unexpected locations (either during the spin-pit tests or during operational use), those areas have to be introduced into the defined lifing process or they have to be treated by a different method, e.g. scheduled inspections.

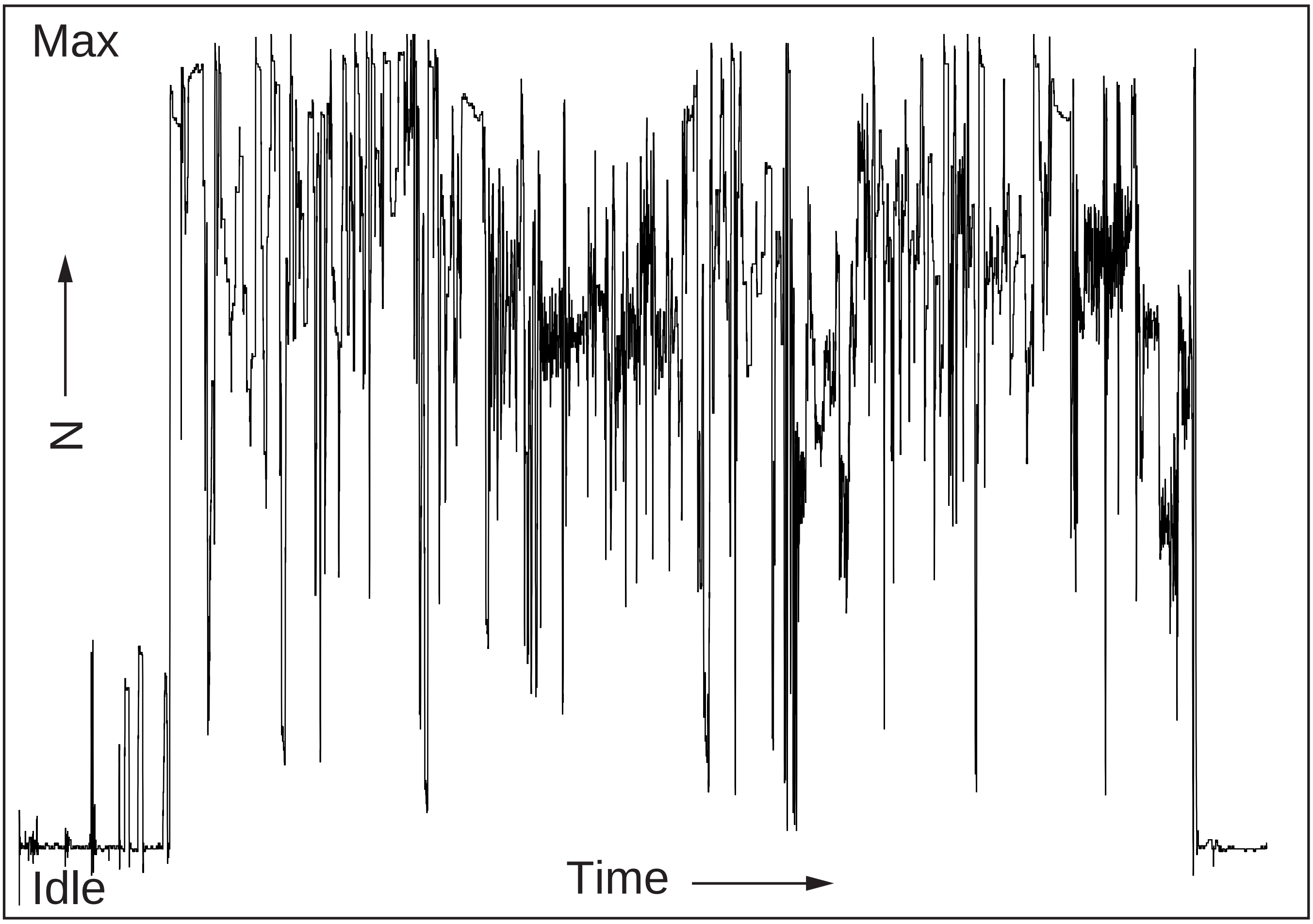



**Fig. 8:** Spool speed of  real mission

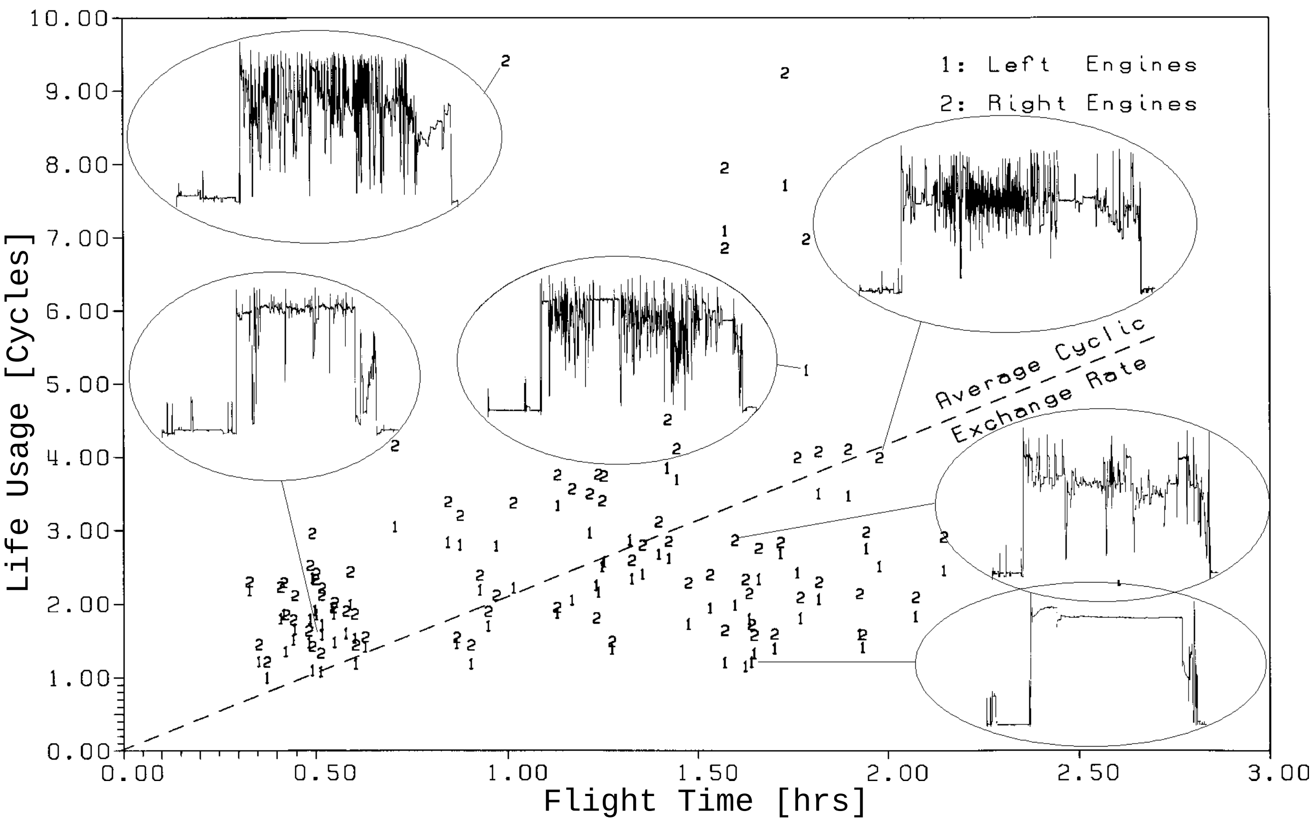



**Fig. 9:** Distribution of single flight life usage

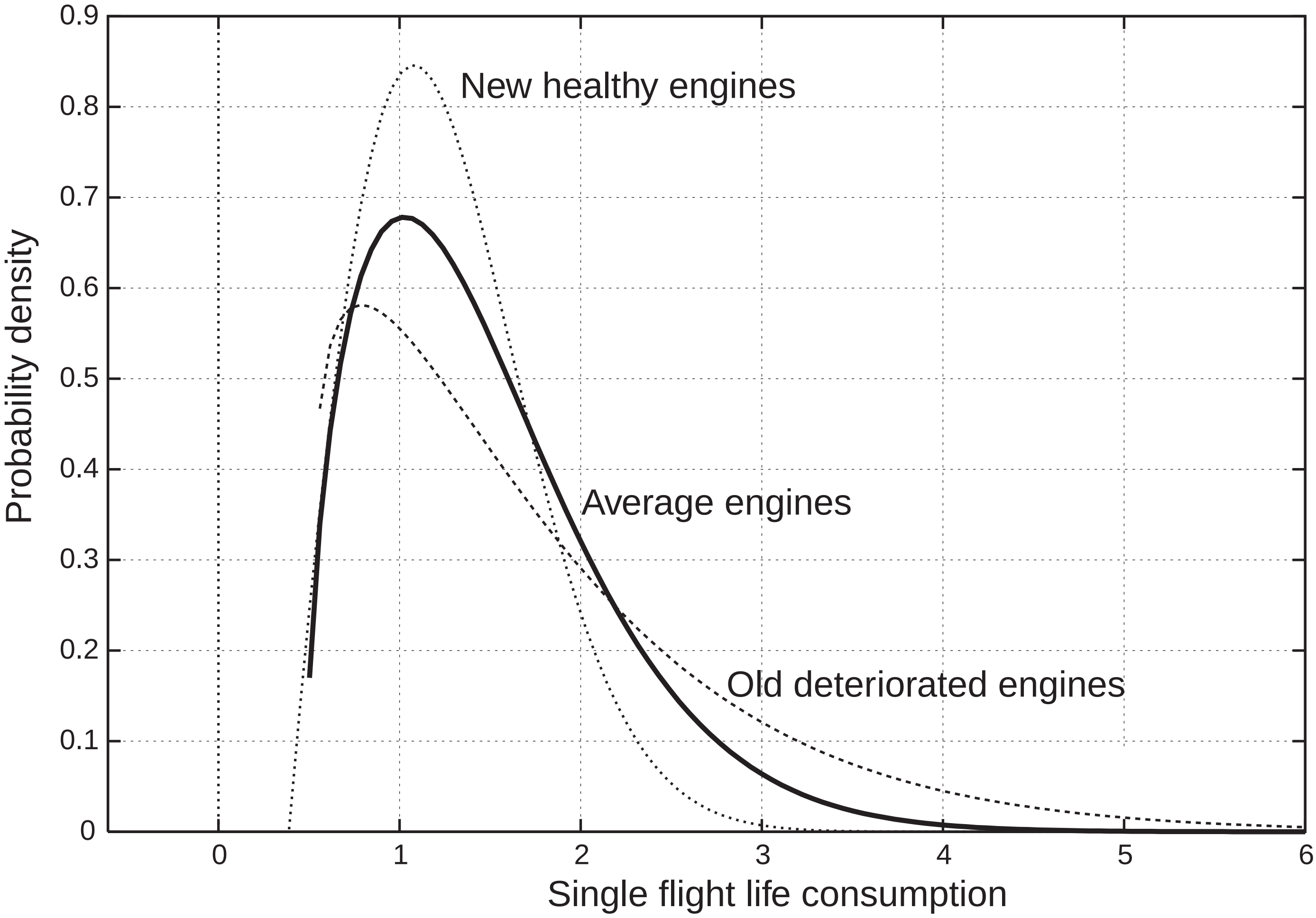



**Fig. 10:** Assumed shift of life consumption distribution for aging engines

For the RB199 engine the “Safe Initiation Life” policy was initially chosen. During the first years of operation, cracks were found at some unexpected areas of the rotors. If treated with the original method, the affected components would have to be retired long before reaching their expected life limits. To recover some of the life potential, the damage tolerance at the newly detected critical areas was assessed. The application of the “2/3 dysfunction life criterion” may restore the originally expected usage times, if the crack propagation life extends over a sufficiently large number of load cycles. The application of the “Safe Crack Propagation Life” to the IP compressor and the IP turbine of the RB199 is described in [BB98].

If cracks occur at locations with lacking damage tolerance, it is also possible to integrate inspections into the lifing process. In some circumstances inspections may also be required, if only a limited number of parts behaves different from the rest of the population. If deviations in the production process have occurred, whose influence on life was not known at production time, the information required to decide on possible life reductions may only accessible by an inspection.

## Adaptation of the lifing process to in-service experience

The first experience in nearly every military engine project is the realization, that there exist non-negligible differences between the design missions and the actual usage. Although design missions have become more realistic for newer projects (see e.g. [JSSG2007]), it is nearly impossible to cast usage patterns similar to those of Fig. 8 into manageable specifications. Data recordings taken during the first time of in service usage may be used to determine the scatter of life usage caused by different missions. An example (Fig.9) from [BP97] shows, that the assumption of a life usage proportional to flight time does not hold for single flights. A better approximation is to determine probability distributions describing the life usage per flight (Fig. 10). Due to thrust requirements for take off, life usage per flight is always greater than some minimum value for most of the critical areas.

The requirement for maintaining some prescribed thrust level is also the reason, that we have to assume some shift of the life usage distributions for aging or deteriorated engines (Fig. 10). If the control system tries to maintain the thrust level by increasing engine temperature and speed, usage will be more severe due to increased thermal and mechanical stresses and also due to lower life potential of the materials at higher temperatures, even if no change in mission types occurs.

There are numerous parameters, by which the use of a component can be described. Flight time, engine running time, number of flights, number of engine runs, engine running time above certain spool speeds, time at certain temperatures. More appropriate for a description of the usually life limiting processes are parameters approximating the cyclic properties of engine operation. The best known method is the counting of so-called TACs (total accumulated cycles) mainly in use at the USAF.

Because of the somewhat arbitrary definition of the power ratings this procedure can be refined by admitting arbitrary spool speed values in the assessment of the contribution of a spool speed cycle. The method outlined below is an extension of counting TACs. If it is implemented in an on-board monitoring system or in a ground-based system for the assessment of recorded engine data, the results are directly comparable with specification values using TACs as a measure for cyclic engine or component usage. The method for "continuous TAC cycle counting" consists of the following steps:

1) Calculation of non-dimensional spool speeds $N = N_{phys} / N_{ref}$, where $N_{ref}$ is the 100% spool speed, equivalent to the “intermediate rated power” (IRP) used in the definition of the “Type I, III, IV” cycles [JSSG2007].

2) $(N_{min}, N_{max})$ cycle extraction for the selected spool speed signal with a rainflow method, using one of the available very efficient methods for real-time cycle extraction. The rainflow method replaces the simple gate based counting used in the original TAC definition.

3) Computation of hypothetical stresses for the extreme values of the cycle, i.e. $S_{min} = N_{min}^2$, $S_{max} = N_{max}^2$.

4) Assumption of a maximum (e.g. burst) speed $N_{lim}$ and a corresponding hypothetical stress $S_{lim} = N_{lim}^2$ and of a hypothetical "endurance limit" $S_{cut} = S_{lim} \cdot FCUT$, where $FCUT$ is a chosen percentage of $S_{lim}$. Cycles $(0,S_{max})$ with $S_{max} < S_{cut}$ contribute zero usage. The endurance limit is defined as the maximum applied cyclic stress amplitude for an ‘infinite’ fatigue life. Generally ‘infinite’ life means more than 10 million cycles to failure.

5) Conversion of the $(S_{min}, S_{max})$ cycle into an equivalent $(0,S_{eq})$ cycle with the Goodman formula

$S_{eq} = S_{lim} \cdot ( S_{max} - S_{min} ) / ( S_{lim} - S_{min} )$

6) Calculation of an auxiliary stress $S_{aux} = S_{eq} / S_{cut} - FCUT$

7) Calculation of the hypothetical damage of the found spool speed cycle $D = ( S_{aux} / ( 1 / S_{cut} - FCUT ) )^{ESN}$, where $ESN$ is an assumed slope of a hypothetical S/N curve.

The parameters in the formulas above can be chosen to closely match the definition of TACs provided in [JSSG2007] or in the appendix of [RTOTR28]: $TAC = LCF + FTC/4 + CIC/40$,

where $LCF$ = "Engine Start to IRP to Engine Stop Excursion", $FTC$ = "Idle to IRP to Idle Excursion",

$CIC$ = "Cruise to IRP to Cruise Excursion".

The following parameter settings match the TAC definition assuming the spool speeds of the HP spool of the RB199 engine (Idle=65%, Cruise (assumed)=81%): $N_{lim}$=120%, $FCUT$=0.55, $ESN$=3.5. With this definition a (65%,100%) cycle will produce a usage of 0.25, a (81%,100%) cycle will give usage 0.025.

For engines with 2 or three spools, it is advisable to use separate definitions for the different spool speeds, since the percentage values of idle and cruise will be significantly different for HP and LP spools. The definition $N_{lim}$=130%, $FCUT$=0.4, $ESN$=2.0 is a choice giving higher weight to “sub-cycles”. The results for the two selected parameter settings are shown in Fig. 11 for $(0,N_{max})$ cycles and in Figs. 12 and 13 for arbitrary cycles.

A cycle counting system using the above mentioned or similar parameters would produce consistent usage figures, that allow the recognition of changing usage due to new tactics, operational procedures, pilot training etc. However, it must be emphasized that TAC cycle counting provides only a gross measure of usage and cannot fully replace monitoring functions specifically tailored to the thermomechanical behaviour of the critical parts.

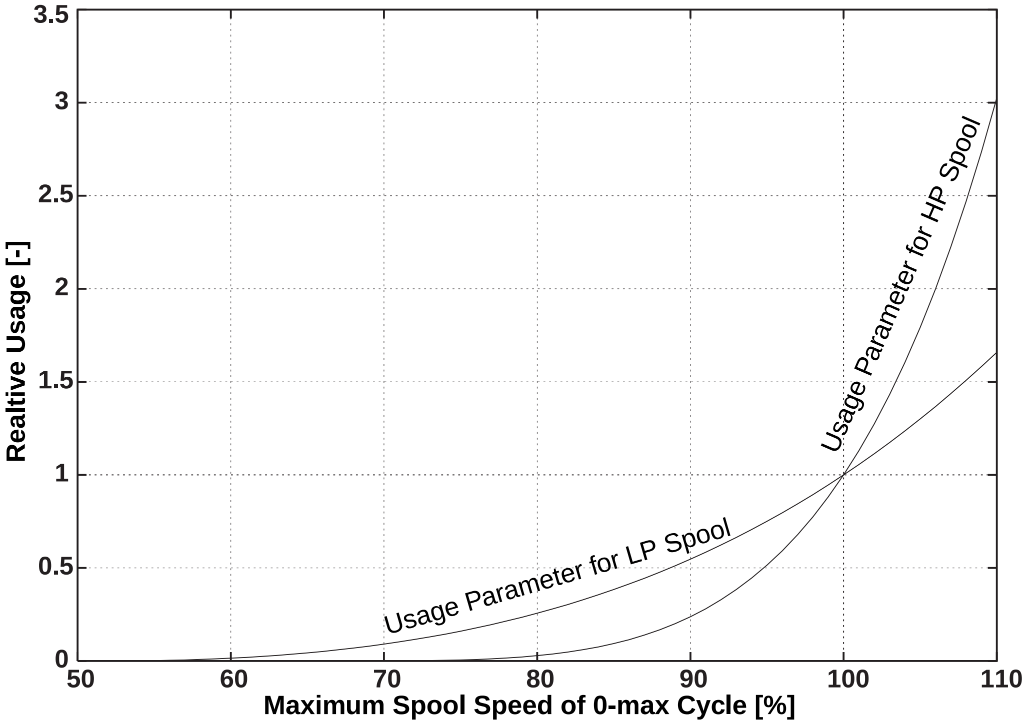



**Fig. 11:** TAC equivalent of arbitrary 0-max spool speed cycle

## The impact of recording and monitoring systems

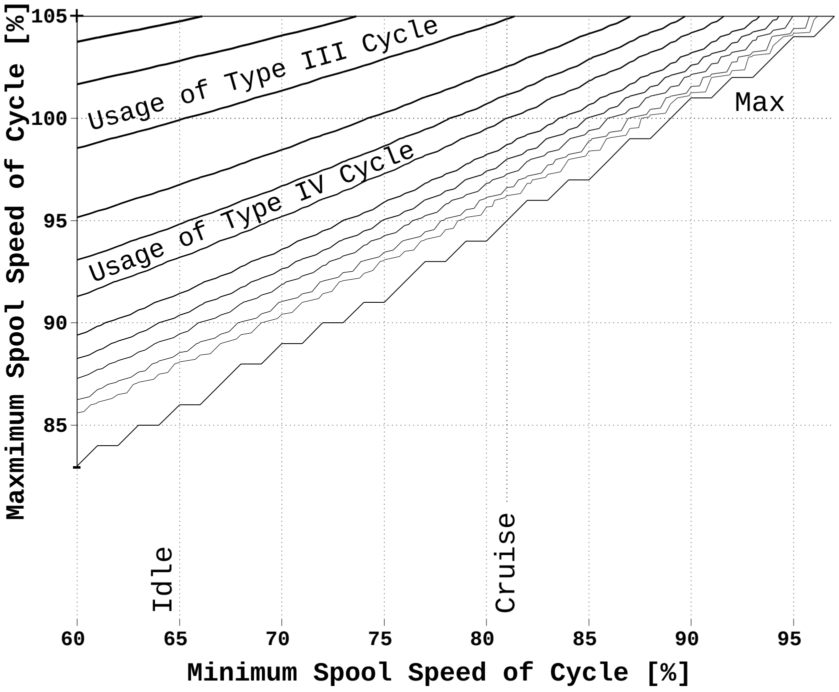
Without information on the usage of an engine in service, very conservative assumptions have to be made with regard to the life usage of critical parts. If a new or modified engine is introduced into service, a commonly used method is either to assume a certain mission mix composed of the specified design missions, or, somewhat better, to use recordings from flight tests to substitute the missing information on usage in service. From those assumed or recorded data, the life consumption at all critical areas of the engine rotors has to be computed. However this requires mathematical models of the thermal and mechanical behaviour of the rotors suitable to calculate the life consumption for arbitrary input data. Such simplified models have to be derived from the complex finite element models used by the manufacturer during component design.

If no life usage monitoring system is used, the common practice is the assignment of conservative life usage figures to all critical areas. Each area then has a so-called ß-factor (average cyclic exchange rate) describing the life usage per flight hour or some other easily available usage figure (e.g. engine running time, number of flights). The fatigue life consumption at a critical area is then computed by a multiplication of the ß-factor with accumulated flight time. The accumulated cycles have to be compared with the released lives. A part is removed from service, if either the life limit is reached or if the part is accessible during maintenance and the low remaining life makes the reuse of the part uneconomical.

A more accurate determination of cyclic exchange rates can only be obtained from a sufficiently large number of flight and engine data recordings, with some side conditions concerning data quality, availability of configuration information and statistical significance (e.g. data from different engines, air bases, mission types etc.). With those data, it is possible to derive statistically meaningful data on the usage scatter within the fleet for each critical area of all critical parts.

## Risk mitigation techniques

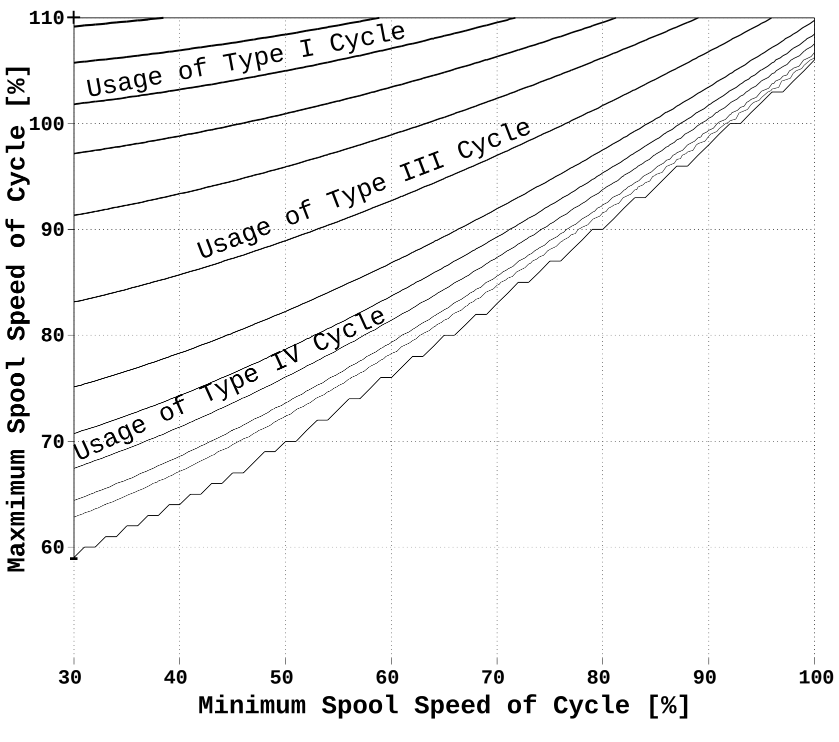
The most popular and probably most costly risk mitigation technique is a regular inspection of all candidate locations for fatigue cracks, using NDI methods. This is currently performed in the commercial aviation world for certain components of older engines known to be at higher risk level due to deviations either in the material properties, in the




**Lines of Equal Usage**

5.000E-04 1.000E-03 2.500E-03 5.000E-03 1.000E-02
2.500E-02 5.000E-02 1.000E-01 2.500E-01 5.000E-01
7.500E-01

**Fig. 12:** TAC equivalent of arbitrary HP spool speed cycle




**Lines of Equal Usage**

5.000E-03 1.000E-02 2.500E-02 5.000E-02 1.000E-01
2.500E-01 5.000E-01 7.500E-01 1.000E+00 1.250E+00
1.500E+00

**Fig. 13:** TAC equivalent of arbitrary LP spool speed cycle

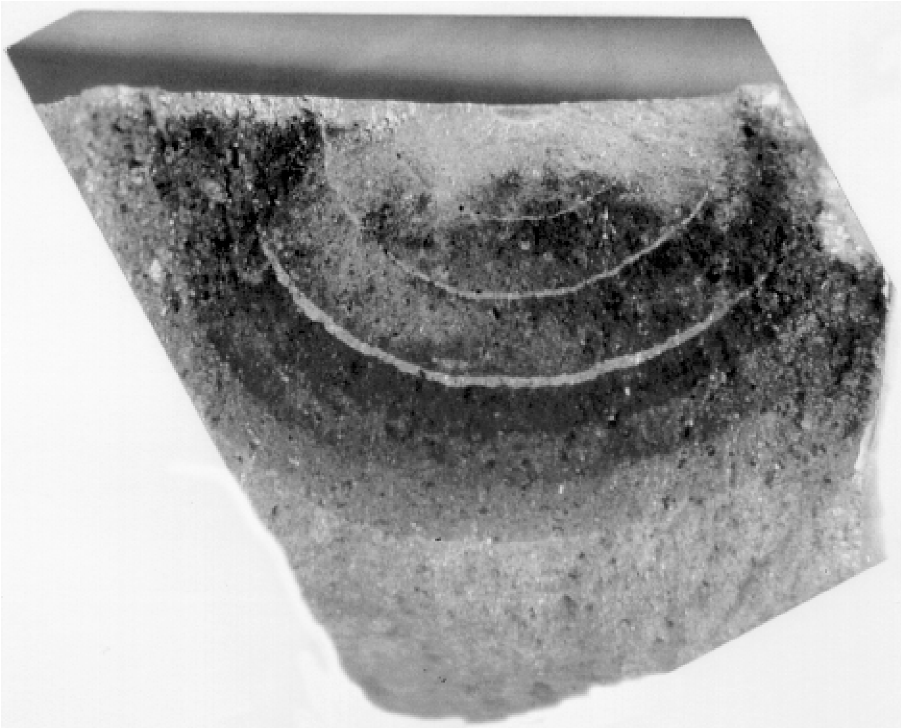


**Fig 14:** Measured crack growth history for rim area

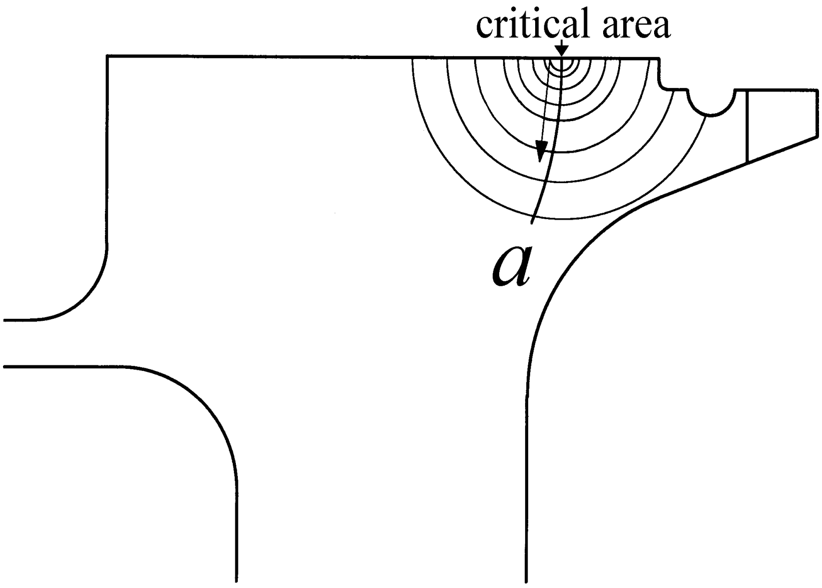



**Fig. 15:** Propagation of crack front with crack length

manufacturing process or in the application of certain repair methods causing a reduction of the life potential. If shortages of parts occur, it is sometimes inevitable to "Inspect-In-Safety" as a risk management tool. Knowledge gained by recording or monitoring can be used to mitigate the risks. Naturally even the most sophisticated monitoring system or crack propagation prediction cannot really recover life. However a risk evaluation is simplified, if sound statistical information on usage spectra is available.

It may sound surprising, but sometimes it is easier to make statistic statements about the use of engines in a fleet, if the missions are randomly assigned to the available aircraft, as if the missions are assigned aimed at individual aircraft. Sometimes the allocation of certain mission types to individual aircraft cannot be prevented, if e.g. these aircraft are equipped with special electronics, armament etc. On a long-term basis it should be tried however not to always equip these aircraft with the same engines since the underlying assumptions of a risk evaluation otherwise possibly can be

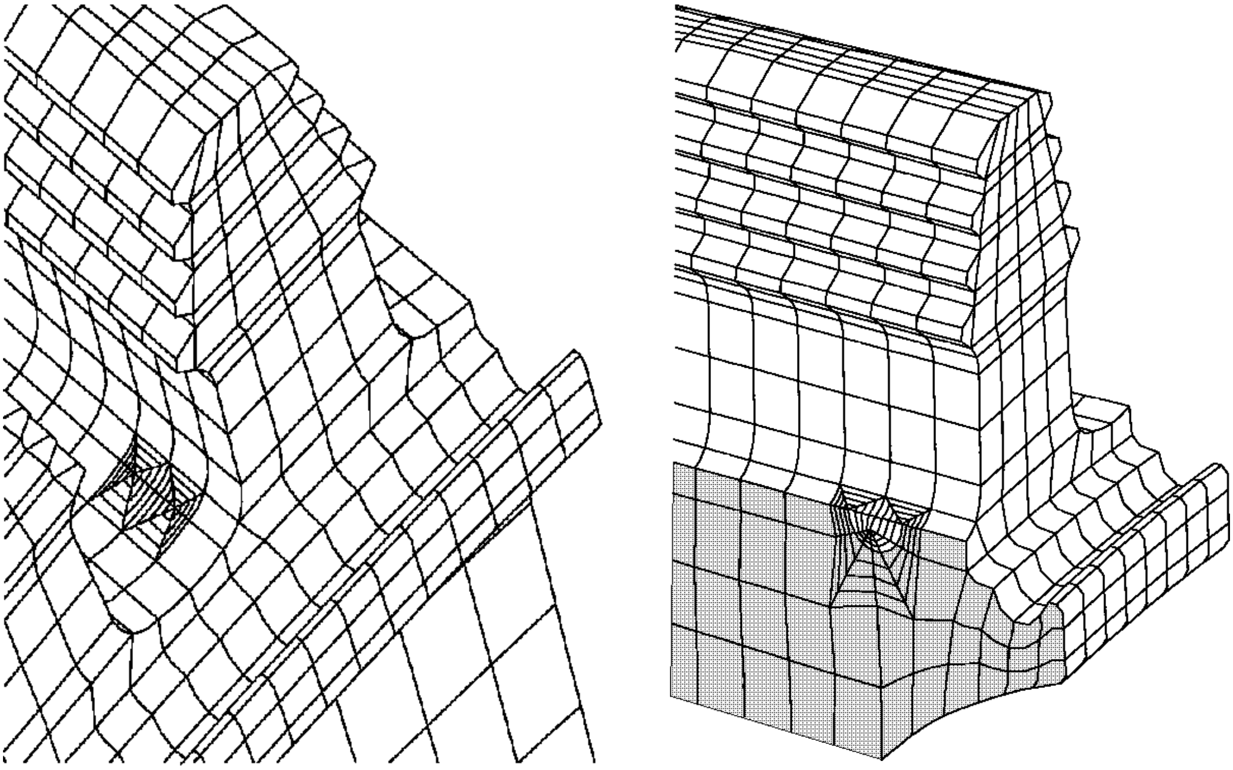


**Fig. 16:** Locally refined FE mesh for crack growth calculation

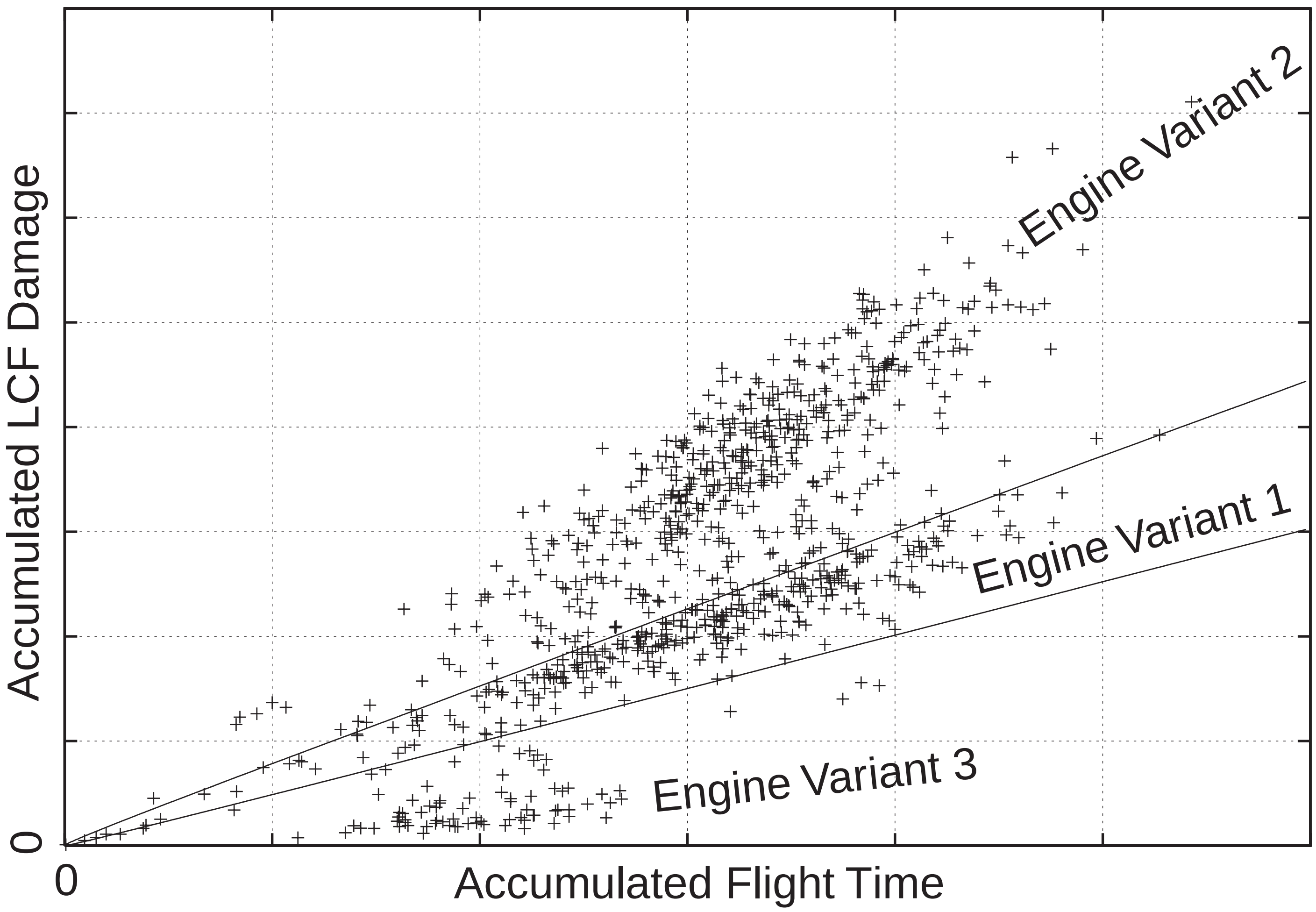



**Fig. 17:** Accumulated LCF damage for turbine, determined by fleet-wide monitoring

hurt, e.g. if a strongly damaging mission type for the engine is continuously flown. Plans for parts replacement or inspection schedules assuming average usage (e.g. relying on usage distributions like those shown in Fig. 10) may then considerably underestimate the real risk.

## Application of fracture mechanical methods to determine safe inspection intervals

Cited from [Suk00]: "The relevance and importance of the computation of fracture parameters and the simulation of three-dimensional crack growth stems from the widespread use of numerical fracture mechanics in fatigue life predictions of safety critical components such as aircraft fuselages, pressure vessels etc. Fatigue failure usually occurs due to the initiation and propagation of surface or near-surface cracks, which are often assumed to be elliptical or semi-elliptical for numerical modelling. Closed-form solutions for the stress intensity factors (SIFs) are available for simple crack geometries in three dimensions; however, for arbitrary-shaped cracks in finite specimens, numerical methods are the only recourse to modelling three-dimensional fatigue crack growth."

Although final knowledge can only be obtained by performing expensive tests, the application of finite element methods to the cracking of components is now within reach. To determine how a crack will propagate from an initial flaw at a critical area of the component, the traditional FE methods have to be enhanced by re-meshing techniques, which adapt the mesh to the crack geometry [Dho98]. Such methods have already been used to predict the crack growth for components, whose life would have been expired, if the classical safe life criteria were applied. One such component was the IP turbine disk of the RB199 engine, for which cracks were found in the rim area. Fig. 14 shows the results of a cyclic spin-pit test, including the application of experimental techniques (marker loads) for a visualization of the crack front. To understand the crack propagation process and to obtain verified data that can be used for a life extension into the crack propagation regime, the crack growth process (Fig. 15) was studied in a 3-dimensional FE calculation. Fig. 16 shows details of the FE mesh used to calculate the crack growth at the bottom of the firtree area of the IP turbine disk. The method is described in more detail in [BK99]. The results of the simulation were compared with the experimental data. A simplified model was developed and implemented in the OLMOS system, thus recovering a considerable amount of usable life.

Recently also methods have become available, that try to avoid the explicit meshing of the crack surface by adding "enrichment functions" to the FE approximation in the vicinity of the crack-tip [Suk00]. The fnite element calculations have to rely on suitable material data, e.g. crack-growth-rate curves as a function of stress intensity factor ranges. There is an urgent need for an extension of the comparatively small database of available crack growth data for engine materials, which is still a field of intensive worldwide research.

## Experience from operative engine monitoring systems

Starting around 1980, various monitoring systems have been developed. One of the most comprehensive systems is OLMOS (On-board Life usage Monitoring System), which is now in use for more than 13 years. This system monitors the fatigue life usage of the engines, the airframe structure and performs a variety of other monitoring tasks. There have been several publications on this system, its architecture and its results, e.g. [BP97]. One of the most important findings was, that such a system is not a static one. The method for tracking the life of critical parts is not necessarily to be held constant during the whole life of an engine. There exist a lot of external and internal factors requiring a continuous adaptation of the life usage monitoring functions [PR95]. The OLMOS is installed in all Tornado aircraft of the German air force. Fig. 17 shows an example of the computed life usage for one critical area on a turbine disk, including all flying engines an also spare parts. The two solid lines show the results of a statistical fleet simulation performed for the engine variant "1", using distribution functions similar to those shown in Fig. 10, that have been derived from recorded flight data. As the turbine disk under consideration is exchangeable between the engine variants, the 3 distinct clouds will continue to diffuse, partially also caused by deliberate decisions of the fleet managers to change disks between the engine variants.

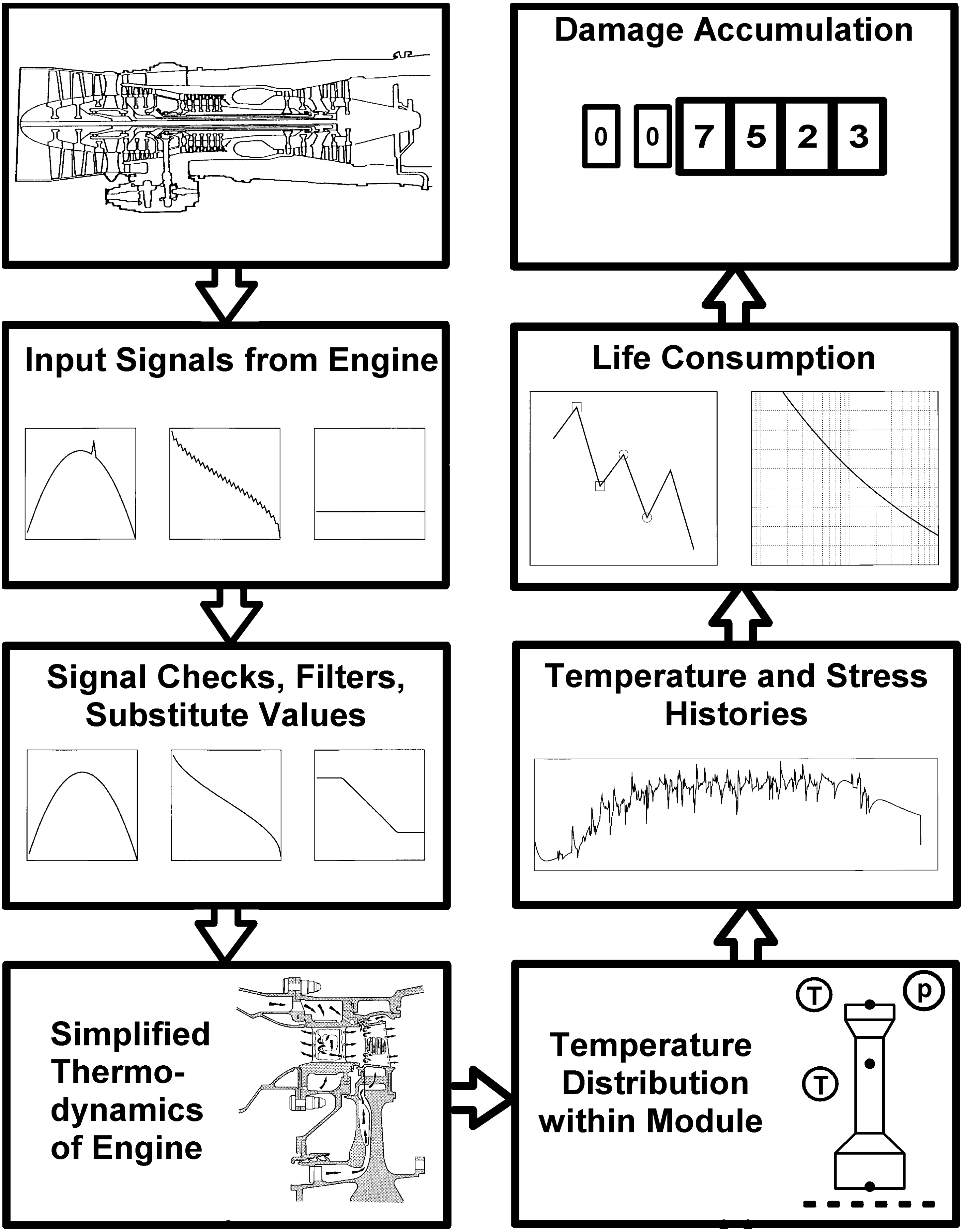



**Fig. 18:** Overview of life usage monitoring

## Requirements for engine monitoring systems

If a new system is defined, there is a wide range of possible architectures. Requirements for an engine monitoring system have to be a balance of the selected benefits and the available capabilities. [ARP1587] provides an extensive list of possible design options. Improved life management would need reliable usage monitoring systems to have realistic stress and temperature cycles [LI98]. Fig. 18 gives an overview of the data acquisition and calculation procedure to be implemented in an engine life usage monitoring system. A comprehensive treatment of all aspects of engine life consumption monitoring is given in [RTOTR28]. The calculation of the usage parameters need not necessarily be performed on-board. It is also possible to use recorded flight and engine data, that are collected on-board and then downloaded by some suitable means (ranging from magnetic tapes to satellite communication links), and do all the processing in a ground station or even at a centralized facility or at industry. We are currently investigating systems, that use highly compressed data storage in the aircraft, to remove the need for frequent downloads. Monitoring systems based on data recording have a potential to remove some of the problems found in the existing on-board systems, e.g. their inability to quickly react on changes in critical areas or the high cost of updates after engine modifications.

## Recommendations for fleet usage management

Although carefully planned inspections and the evaluation of usage data are the basic building blocks for a minimum risk extension of aircraft and engine life, the allocation of material can significantly contribute to an acceptable availability of an aircraft fleet. A fleet manager will usually try to avoid that engines have to be removed from an aircraft only because a single fracture critical part has reached its life limit. He will also try to avoid foreseeable engine changes due to parts becoming life-ex, if the aircraft is at some remote base without proper maintenance support. A centralized logistical database containing the life usage data of all flying and spare parts can be used to direct the necessary parts to the right locations at the right time.
For components with a high variability in flight to flight usage there will also be a larger scatter in accumulated life usage for a given range of engine running or flight time. Systematic differences (e.g. those resulting from different thermomechanical environments if the parts are used in engines of different build standards) can lead to distinct clusters of parts in the cycles versus hours plot. Fig. 17 shows an example from the GAF database for one critical area on a turbine disk of the RB199 engine. This area experiences systematically different life usage dependant on the engine variant. The reason is the introduction of engine modifications that had an influence on the spool speed relations between the HP, IP and LP spools. The engine standard present at entry into service corresponds with variant 1. Some years later modifications have been introduced, that led to somewhat higher LP spool speeds. This standard, indicated by “variant 2” is currently the most frequent one. The data marked with "engine variant 3" are from redesigned engines with a new larger fan with a significantly reduced rotational speed of the LP spool. Those engines were introduced 7 years after delivery of the first engine variant. Life usage distributions similar to the one of Fig. 17 have some advantages for fleet life management. The large differences can be taken into account in a parts allocation strategy, which exploits the life potential by clever changes between different engine variants.
If certain critical parts need frequent inspections due to deficiencies in their design, manufacture, material, corrosion resistance or fatigue life, those inspections may become a decisive cost factor. In [WB97] an example is given for the F100-PW-100 engine. For this engine failures of the third-stage fan disk lug resulted in uncontained fan blade liberations. The frequent ultrasonic inspections of the disk lug, that had to be performed to keep the risk acceptable were reported to have become the No.1 maintenance man-hour driver in the F100. The solution was to incorporate new zero-time disks to gain some inspection-free time until redesigned disks and blades became available.

The management of parts has to take into account long-term plans for fleet size. Many air forces are now considering to reduce fleet sizes far below the figures planned and acquired at the end of the cold war era. A balance has to be found between affordable peace time operation and high availability during a potential crisis. It is not always economical to use the full life potential of all parts. It is noted in [TH98] that it may be sometimes advantageous to scrap parts a considerable time before their life is expired, if the cost of engine disassembly, downtime etc. is considered. This will also reduce cost due to handling, depot operation and administration.

## Acknowledgments

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# AIRCRAFT LOADS

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**SUMMARY**
The life of a weapon system is influenced to a high degree by the structural integrity of the airframe. Numerous programs to ensure this have been established within NATO´s Air Forces. Structural loads, leading to fatigue as well as corrosion, depending on the usage environment, are the major reason for degradation of structures. The many different classes of loads, the generation of loading conditions during the design phase, as defined in the weapons systems specification, consideration of static and fatigue loads for structural lay-out and validation concepts are presented.
The procedure of converting overall aircraft loads ("external loads") into individual component loads is shown in principal .

## 0. BACKGROUND

The effectiveness of military force depends in part on the operational readiness of aircraft which itself is largely dependent on the condition of the airframe structure. This condition again is affected by a number of factors among those the physical loads in various forms together with the used life of the airframe are important. With increased and extented usage of airframes in all airforce inventories and the requirement for various role changes the subject of airframe loads assessment, -qualification and aircraft loads-monitoring becomes more important, not only for flight safety but also and with an increasing tendency for economic reasons.
A general understanding of the various types of airframe loads, their generation and application during the design process, the transfer processes from "external loads" into "structural loads", loads qualification during ground and flight testing is therefore of equal importance to the process of usage monitoring and derivation of usage factors from the different fatigue tests or the set-up of structural inspection programs.
When life of aircrafts are discussed, often the flight hours or number of flights are still considered the governing factor, sometimes adapted with factors on "damage hours" or "usage", while from a structural engineering viewpoint the operational stress spectrum and therefore the life on the different aircraft components are not only a matter of flight hours and spectrum ratio but also driven by modification status, structural weight status and role equipment.
This paper describes loads- analysis and verification activities during the major phases of the life of an airframe, where structural loads and their influences on the airframe condition are vital to the structural integrity and the economic usage of the weapon system:

* The structural loads during design and Qualification of A/C structures
* Loads monitoring during usage
* Impacts due to aircraft modification and role changes.

Trends with respect to the increased usage of theoretical modelling are also discussed.

## 1. STRUCTURAL LOADS DURING THE DESIGN AND QUALIFICATION OF AIRCRAFT STRUCTURES

Loads are accompanying an aircraft's life from "the cradle to the grave". Although the overall type and magnitude of major load sets remain the same, there is no "fixed" loadset that is be applied to one aircraft model throughout the life and often identical airframes serving different roles within a fleet over time will be subjected to very different loads.
To include as much as possible (or specified) of these loading scenarios in the early process of designing a new type of aircraft is the responsibility of the loads engineering department, while ensuring that these loads can be safely endured throughout the specified life is the task of the design and stress engineers. "New" loadsets, developed later during usage of the aircraft are common tasks and handled similar as the "initial design loads" by the design authority with the constriction, that now the airframe is already build and deployed and the focus is on minimising changes though structural modifications to qualify the structure for its new environment either through analysis and / or test.
In short, every major change in the aircraft's role, payloads or usage in principle influences the loads acting on the airframe or at least some components. Fig. 1-1 gives an idea how loads are initially generated and how they are used throughout the design-, qualification- and usage process.

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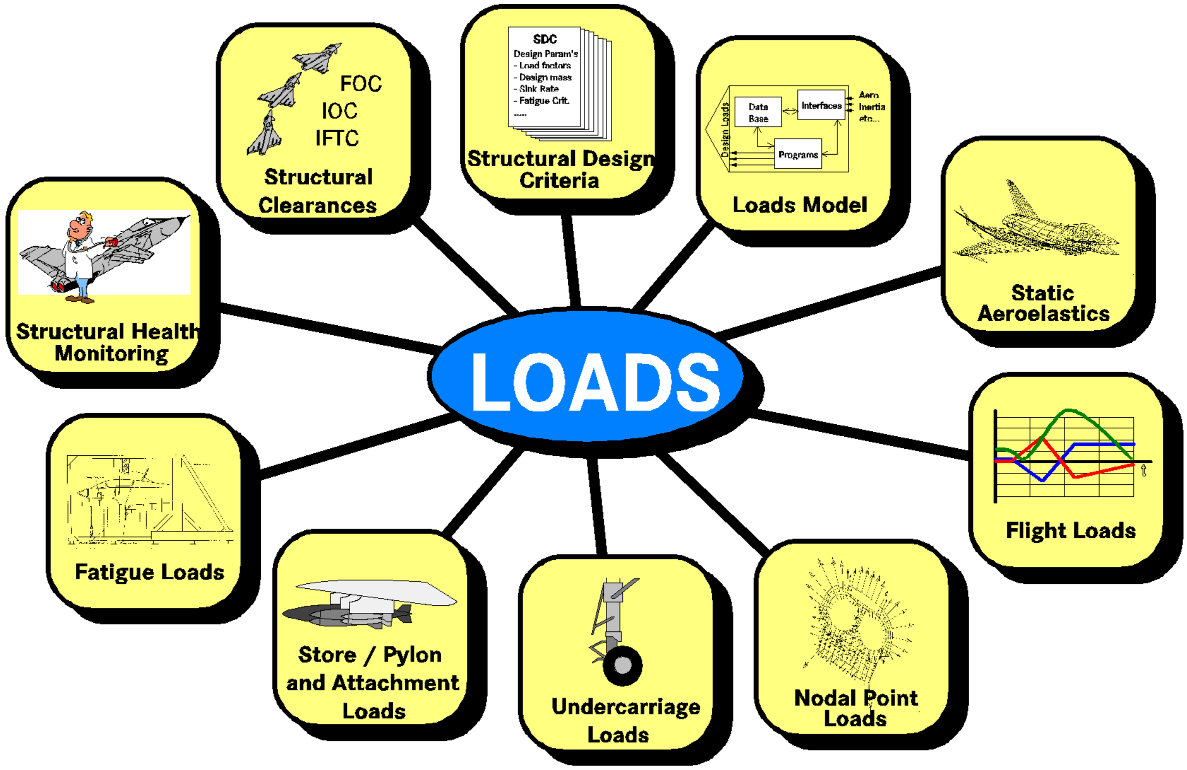



**Fig. 1-1 Loads Main Tasks**

### 1.1 Loads and Fatigue

The determination of loads together with the qualification for static strength and fatigue by calculation and test for all important structural components is a main prerequisite for successful design and safe operation of any aircraft.

Whereas for transport aircraft with their rather limited range of operational manoeuvres and high number of flight hours / cycles fatigue is the main design driver for the airframe, fighter aircraft are predominantly designed to (static) limit load cases for the “corners” of the envisaged flight envelope, which in general cover a lot of strength required for fatigue of their comparatively short life.

But this is only true as long as fighter life does not exceed the originally planned lifetime and the roles, missions etc. are compatible with the design criteria at the beginning.

Aging aircraft in both cases does not only mean that an aircraft is getting older in terms of flight hours and flight cycles, it also means that some of the reference data for the basic design criteria have changed during time, i.e.:

- airframe and equipment mass growth
- enhancement of systems performance, especially engine thrust
- new configurations (stores)
- update of flight control systems (FCS) (electronically or hardware changes like added slats or enlarged ailerons)
- mission profiles and additional/changed roles
- actual usage spectrum

Most of these changes have an immediate impact on aircraft load scenarios, others will not change load levels but may change underlying statistic, e.g. fatigue spectra. Assessment of external loads is therefore a basic task throughout the life of a fleet.

Admittedly in many cases there is no simple one to one relationship between “external” loads and local internal stresses, which after all are the basis for the assessment of “life consumption” or “remaining life” of structural components. But providing loads are known for a special structural interface or component, reliable conclusions can be drawn regarding local stresses relating to the manifold of load cases from experience, measurement and detailed FE analysis during design, qualification and test phases in many cases.

In addition the comparison of load spectra alone may already be suitable for drawing conclusions without recourse to detail stress calculations of specific locations for components with limited loadcase variations i.e. landing gears.

**1.2 The Determination of Design Loads**

Design loads, better "Initial Design Loads" are the first step in the loads history of an airframe that influences the detail design of a component (i.e. wing or fuselage structure) or, at a later stage in the design process, a part (i.e. wing spar cap or fuselage skin panel) in many details. Since not every load is determining these design tasks, establishment and identification of the "design loadcases" is important. The following is a summary on the methods how design loadcases are determined, with special attention to points where an immediate context with fatigue calculations exists.

Fig. 1.2-1 shows a typical "loads loop" which usually is repeated several times in the different phases of the aircraft design. First of all the Structural Design Criteria (SDC*)* are prepared as a basis for design, specifying the basic performance and flight parameters, then a Loads Model (LM) is built, based on the SDC's, the aerodynamic, flight mechanic and weight and balance data of the aircraft.

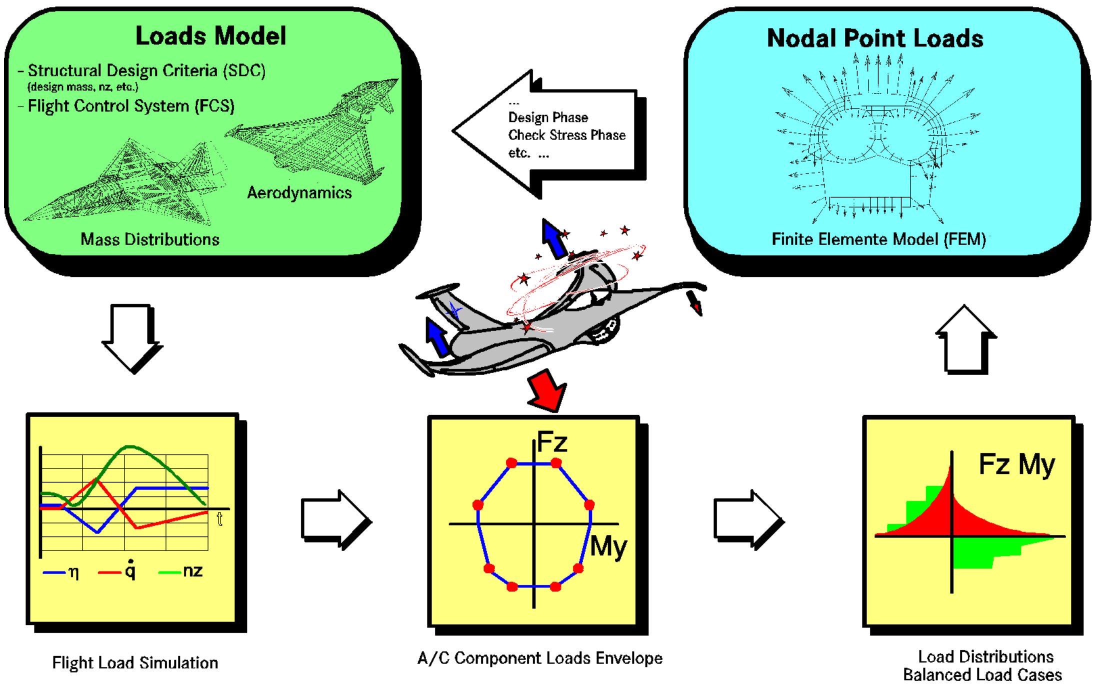



**Fig. 1.2-1 Loads Loop**

The loads module ensures that loadcases selected for design are analysed for an overall balanced aircraft (mass, inertia and aerodynamic forces) for all manoeuvres and the loads analysis is performed in a time history sequence, thus providing load information on structural interfaces for every timestep of the chosen manoeuvre. Results of the loads module is either continues external load distribution for any component (i.e. bending, torque and shear force distribution along fuselage stations for all loadcases or a) distribution of loads on the Finite Element (FE) grid nodal points for subsequent "global FE-Analysis".

Thus, starting with the SDC the load loop ends with the preparation of external loads for stress analysis of components.

Usually an improved or changed data basis results in an update of the LM and consequently in more accurate and more detailed design load cases. Typical improvements are a better aerodynamic data basis (i.e. via extensive windtunnel testing) or a refined FE-model because of an advanced design status. Modifications in the mass and balance status, control laws etc. may also result in substantial changes of the loads model, especially in advanced computer controlled flight vehicles.

The importance of the link between knowledge of external loads and structural stress distributions for the assessment of fatigue life cannot be underestimated. Whereas in the past the available computer resource was rather poor and strong software tools were scarce goods, leading to a strong selection of loadcases to be analysed in detail, today there are virtually no limits, from this side. Computers power do play an important part with respect to better and refined results in the assessment of loads, however the correct selection of the critical manoeuvres for the fatigue spectrum and their loads analysis still influences the fatigue performance of a structure during the design phase.

Most of today's ageing aircraft fleets of the NATO airforces were designed and flight tested by the end of the sixties or the beginning seventies, like the Tornado, Harrier, F-16, F-18, Mirage 2000 etc. An aircraft like the F-4 Phantom even dates back to the fifties and is still in service in some air forces of the alliance.

When comparing design environments of the a.m. models it should be pointed out that in the meantime the circumstances and requirements for aircraft design and analysis have changed in many ways, in detail:

- much better tools, soft- and hardware, and with that a very intensive investigation to calculate and control limit and fatigue loads (including a substantial increase in the number of component load monitoring stations)

| | Tornado IDS | Future Europ.Fighter |
|---|---|---|
| Basic Loads Cases (BLC) Flight and Ground Handling Loads | 33 | 105 |
| Unit Loads Cases (ULC) Hammershock, Engine Thrust, Airbrake etc. | 12 | 16 |
| Combined Load Cases Superposition of scaled ULCs to BLCs | ~ 100 | 590 |

- more accurate loads databases in terms of
  - advances in "Carefree Handling"- Flight Control Systems (FCS)
  - aircraft mass distributions predictions
  - aircraft aerodynamics calculated with mature CFD (Computational Fluid Dynamics) methods and verified earlier and more reliable in wind tunnel tests.
  - coupling of structural models and aerodynamic models for aeroelastic effects available
  - Finite Element modelling of the structure with interfaces to the Loads Model
- extensive flight testing, especially dedicated flight load surveys
- extensive structural ground tests

Basically this means that the static design of "old" aircraft usually is rather conservative and on the safe side.
With respect to fatigue the situation is often less satisfying, i.e. without powerful tools like a balanced Loads Model, one procedure was balancing loads over the aircraft artificially in those days, and design loadcases therefore were generated for parts of the structure like aft or forward fuselage or tailplane only, the effect of these loads on other areas of the structure remained unknown and components, not immediately under survey were not analysed for this loadcase, therefore the effect of changes to these loadcases later remained also unknown.

### 1.2.1 Structural Design Criteria (SDC)

Aircraft loads are determined according to requirements and regulations collected in a systems specification document called Structural Design Criteria, the major reference for loads and structural analysis engineers during the design phase.
Many of the SDC requirements come from the customer, others are prepared in co-operation between customer and original equipment manufacturer (OEM), usually the principal design contractor. The SDC are also subject to revisions during the design process.
Some of the more important items regarding loads and structures are:

*Design masses* are defined for different flight conditions to cover the whole mass and center of gravity (C.G.) range, i.e.:
- basic flight design mass
- landing design mass
- maximum take off mass

Total mass and mass distribution not only affect loads on wing as is sometimes believed but loads on most parts of the aircraft's structure. Design mass is one of the most important criteria for structural design. For example the basic flight design mass is coupled to the max/min allowed vertical load factor Nz, for increased masses through the rule: Nz·Weight = const. to avoid overloads or assessing the effects of over-g's.

*V-n Diagrams* define the regime of speeds in combination with max/min allowable load factor Nz including gust conditions, see Fig. 1.2.1-1. For low speed regimes the attainable limit Nz depends on the maximum lift and dynamic pressure for the wing whereas for higher speed Nz is limited by the structural strength of the aircraft. The v-n diagram is referenced to a specific mass and store configuration, i.e. clean wing and design mass.

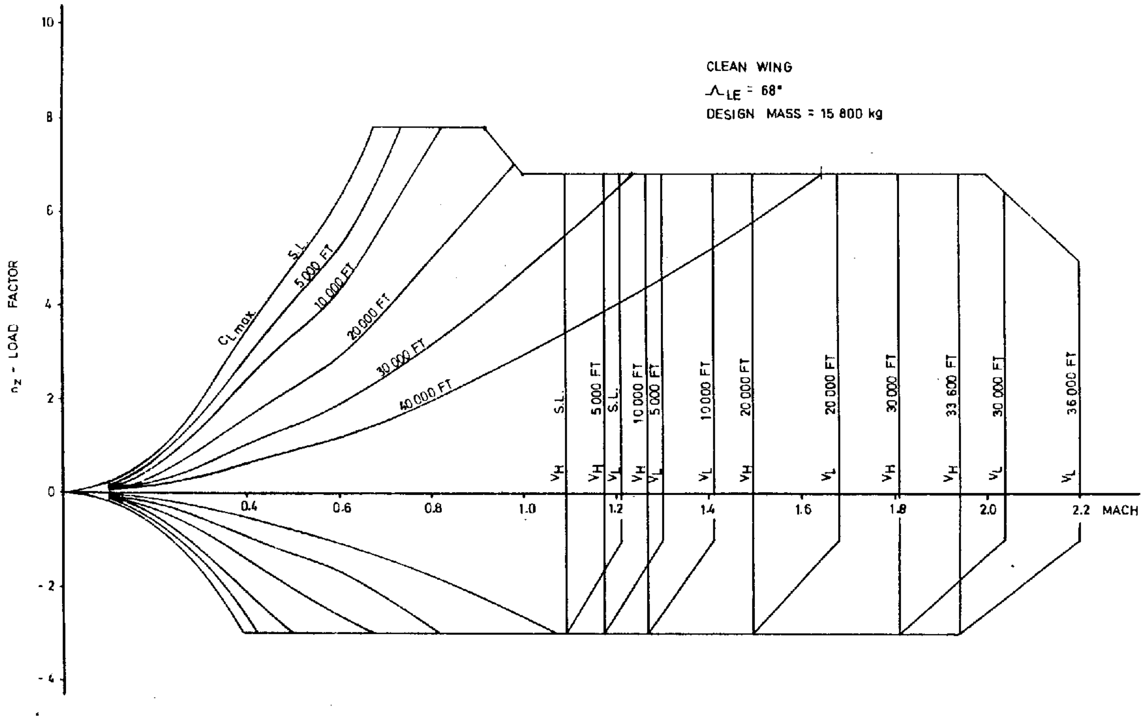



**Fig. 1.2.1-1 Ma-n Diagram in Altitude**

*Flight Envelope(s)* define the operating range with respect to Mach-Altitude regime, for which the aircraft is designed. Limits are determined by attainable Nz, temperature etc. Fig. 1.2.1-2 shows a typical flight envelope for the Tornado aircraft.

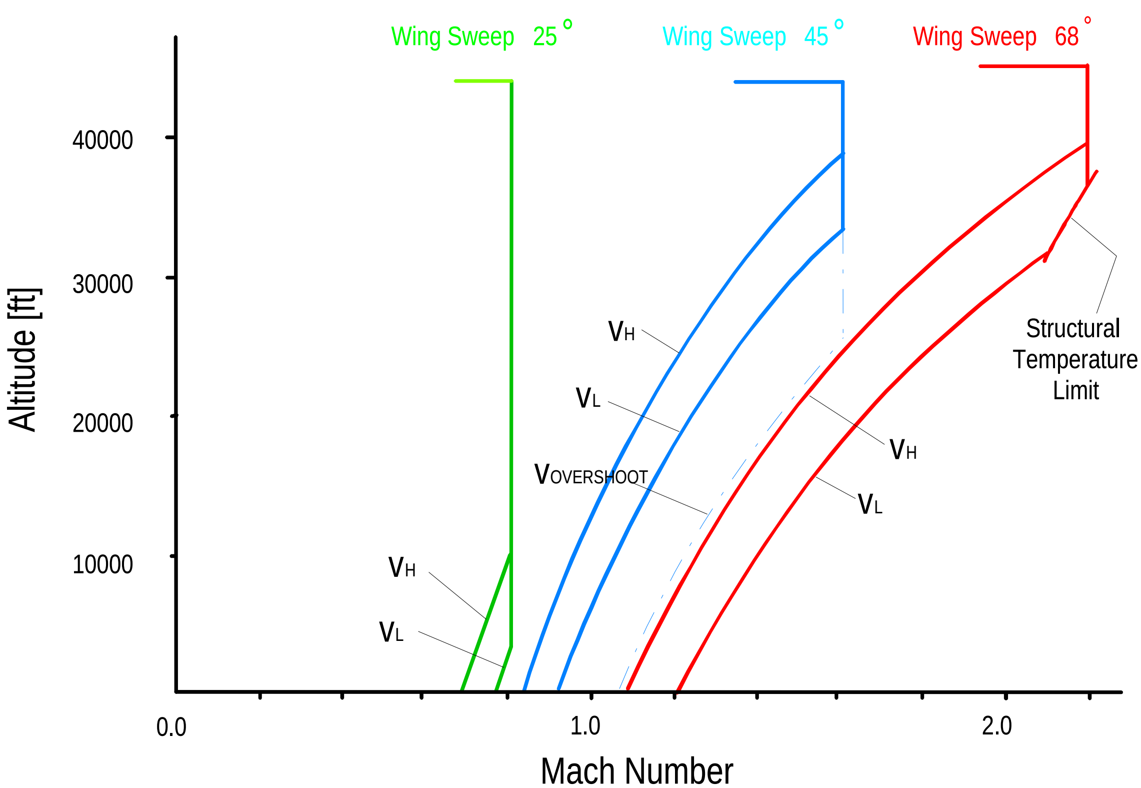



**Fig. 1.2.1-2 Altitude – Mach Number Envelopes**

For an fixed wing aircraft usually only one flight envelope diagram has to be defined, but the Tornado, like other swept wing designs, presents an additional complication as each (fixed) sweep position has to be considered as a different aircraft. This is clearly seen by the different flight envelopes for the shown sweep positions of the wing.
Fig. 1.2.1-3 indicates what part of the flight envelope is of importance for the investigation of loads and shows points in the Mach-Altitude range for which loads are calculated according to the scheme explained later. The points are selected to cover all essential effects due to high Nz, incidence, roll rate, gust, Mach effects etc. Traditionally the analysed manoeuvres could be found following the low pressure altitude and high mach number boundary, but non-linear aerodynamic effects of flexible structure and the modern flight control layouts are the reason for many “interior” points in the Mach-Altitude range (“points in the sky”) of importance for today’s loads analysis.

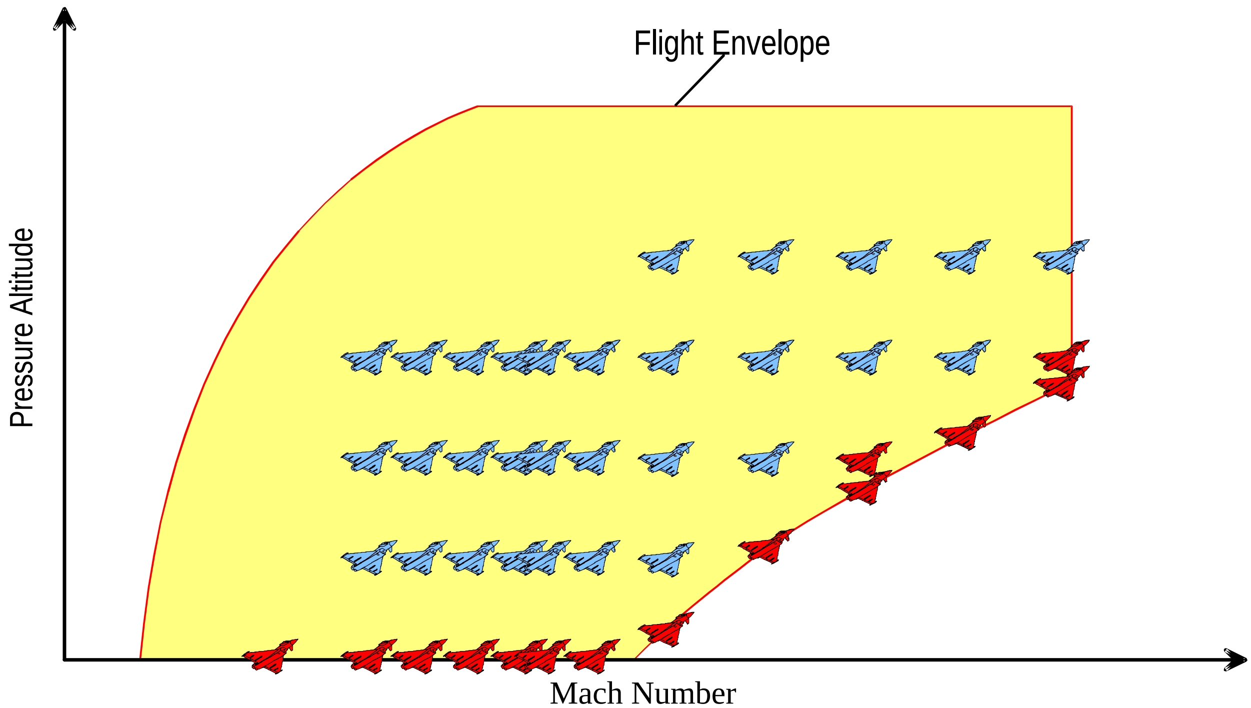



**Fig. 1.2.1-3  Mach - Altitude Points of Loads Model (flex. Aerodynamics)**

*Environmental Conditions* also define or influence tructural loads and include

- System pressures
- Cabin and fuselage bay pressures
- Temperatures and noise levels
- Local accelerations for qualification of equipment
- Vibration levels

*Performance Requirements* with respect to steady state manoeuvres, transition response, flight and ground handling qualities are to be fulfilled.

Example: Due to aeroelastic deformation under load the effectiveness of a control surface may be reduced substantially, for differential tail design’s even roll reversal may occur. Therefore a typical specification would be the max. allowable degradation in control efficiency under such circumstances. This means that an optimisation of the flap structure, its control devices and the attached structure must be carried out to ensure a required roll rate for a given control input.

*Configuration specification* with respect to external stores, and control surface schedules like high lift devices , airbrakes etc. Store configuration definitions can have great impact on fatigue spectra due to either load alleviation or increments by inertia effects (stores on wing  versus on fuselage ). See also Chapter 1.3 for a discussion and example of component load changes due to store configurations.

*Fatigue Load Spectra* are defined based on expected usage and mission schedules for the aircraft and based on the customer weapon systems specification. Together with the applied scatterfactor it defines the loading scenario for qualification of the structural design through analysis and ground tests. A more detailed discussion of this point can be found in the second paper "LOADS MONITORING AND HUMS" of this Lecture Series.

### 1.2.2 Aircraft Loads

The characteristics of loads acting on aircraft are of different kind. Although non-exhaustive, the following grouping shall give an idea of the "classes" of loads to be considered in parallel during design:
<u>Quasi-static loads</u>**:**

Flight Loads:
- Symmetric manoeuvres
- Asymmetric manoeuvres
- Deep and flat spin
- Gust loads

Ground Handling:
- Take off
- Landing
- Repaired runway
- Taxiing (asymmetric braking, turning etc.)
- Towing, Pivoting etc.

Local and Internal Loads:
- Max./min. aerodynamic pressures (outer surfaces)
- Local accelerations
- System pressures
- Bay pressures (pressurised areas)
- Hydrostatic pressures ( fuel tanks)
- Intake duct pressures (steady state)
- Engine thrust

<u>Dynamic Loads</u>**:**

- Buffet ( Outer wing, vertical fin buffet etc.)
- Dynamic Gust
- Vibrations
- Acoustic Noise
- Limit cycle oscillation
- Shimmy (Undercarriage)
- Engine hammershock conditions (Duct)

<u>Fatigue Loads</u>**:**

Fatigue load cases are derived from the a.m. quasi-static and dynamic load conditions if the frequency of the respective load cycle is sufficiently high during the assumed usage. Fatigue loads are always a combination of loads from the a.m. list, especially flight loads combined with local and internal loads or acoustic noise. Other loads, occurring only during failure situations are excluded from the fatigue load sets ( i.e. engine hammershock will certainly not be a fatigue case), Dynamic buffet, although difficult to predict, needs to be included die to its high cycle characteristic and therefore high damage potential.
Flight measured buffet on a vertical fin is shown in Fig. 1.2.2-1 for a symmetric, no side slip pitch-up manoeuvre to 50° AOA, indicating bending moments Mx and torque Mz at the fin root with R=-1, picking up around 35° AOA and increasing to the max angle of attack flown during this manoeuvre.

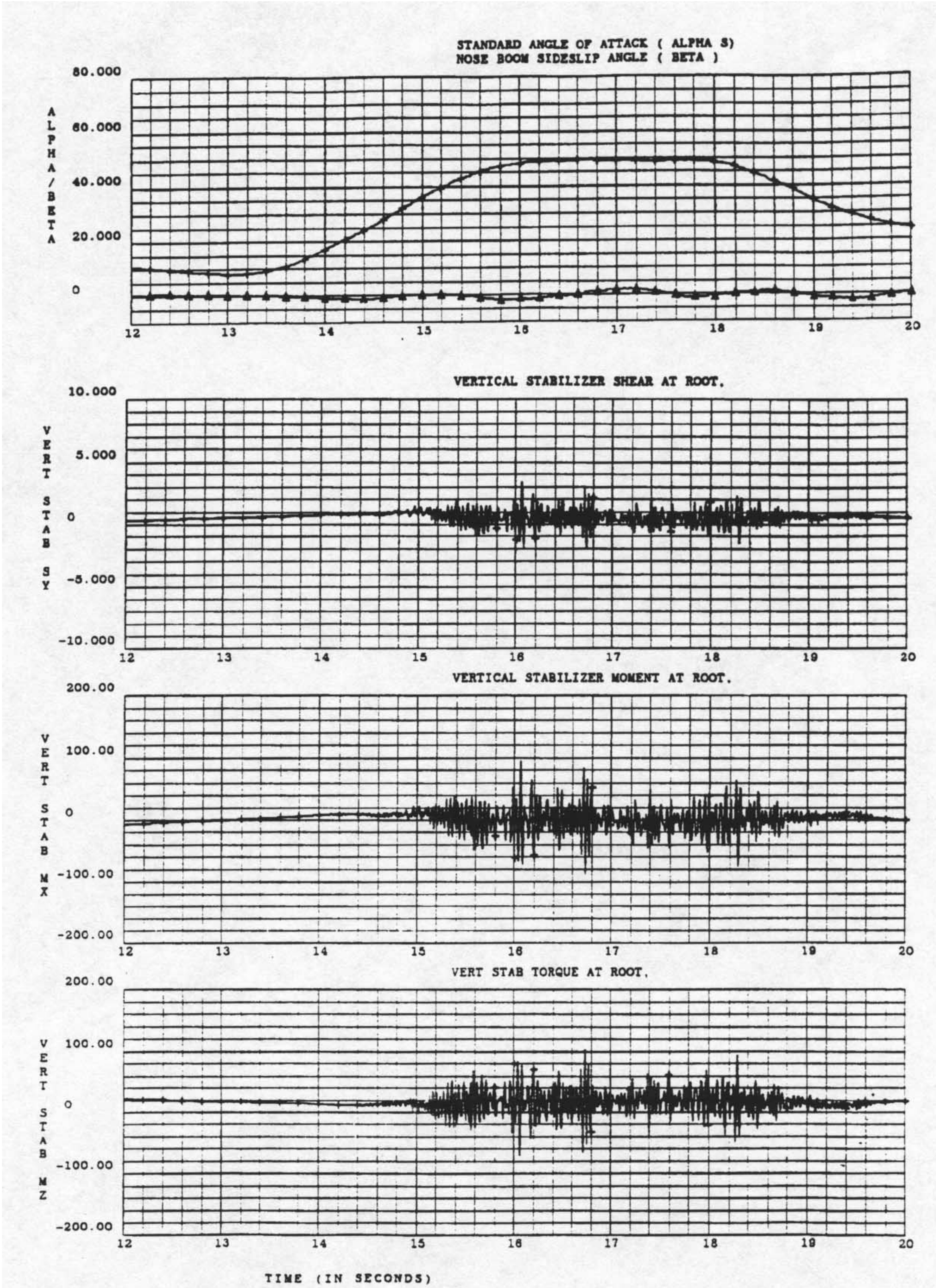



**Fig. 1.2.2-1 Fin Buffet at High Angle-of-Attack (Flight Test Results)**

The above static, dynamic and fatigue loads have to be combined with the corresponding structural temperatures, for the worst environmental conditions (i.e. cold / hot day) and also moisture conditions if material properties like for composites are effected.

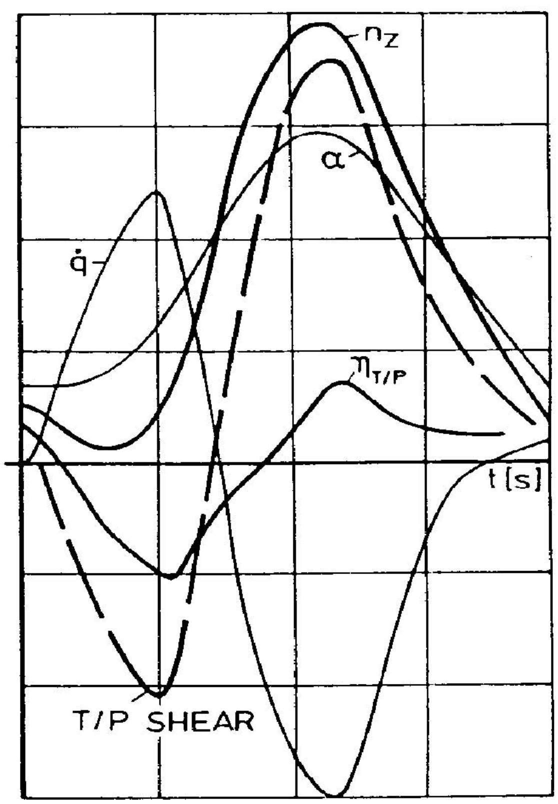
### 1.2.3 Flight Parameter Envelopes

Loads are not a function of Nz alone but depend on many other flight parameters, the most important are:

Incidence or angle of attack (AOA)
- Sideslip (for design the significant factor is • ·Q, the product of sideslip and dynamic pressure)
- Control surface deflection angles (aileron, rudder, tailplane etc.)
- Lateral load factor Ny
- Vertical load factor Nz
- Roll rate / Roll acceleration
- Pitch acceleration
- Yaw acceleration

Usually less important for load derivation:
- Longitudinal load factor Nx
- Pitch rate
- Yaw rate

Adequate combinations of those parameters - as occurring during real flight manoeuvres - can yield high loads on different parts of the aircraft structure, even for rather moderate vertical load factors. In order to illustrate this context, Fig. 1.2.3-1 shows flight parameters during a typical MIL-Std. pitch manoeuvre versus time and indicates the delay between command input (tailplane deflection angle), change in AoA for the aircraft and the increase in loadfactor and the force on the tailplane (=T/P SHEAR), the value for the loads envelope for this component.




| | |
|---|---|
| nz | vertical load factor |
| $\alpha$ | angle-of-attack |
| $\dot{q}$ | pitch acceleration |
| $\eta_{T/P}$ | tailplane deflection angle |
| T/P SHEAR | tailplane vertical shear force |

**Fig. 1.2.3-1 MIL-SPEC Pitch Manoeuvre**

Therefore it is the engineers skill to find all the critical combinations for the different aircraft configurations and the possible manoeuvres within the whole flight regime. Regulations like Mil-Spec for fighter aircraft or FAR for other A/C provide a good guide to determine the critical combinations of flight parameters for design, at least in the case of stable aircraft and conventional FCS. Very often it is desirable to determine flight parameter values from response calculations, using an aircraft response and loads simulation program.

However, in the early and intermediate stages of modern fighter aircraft design a reliable model of an FCS usually is unavailable, therefore agreement between specialists of different disciplines (aerodynamics, flight mechanics, loads etc.) on flight parameter limits in the form of envelopes is the adequate way ahead. Fig. 1.2.3-2 shows typical envelopes as used in the early design phases with the envelope corners design critical regions for different aircraft components.

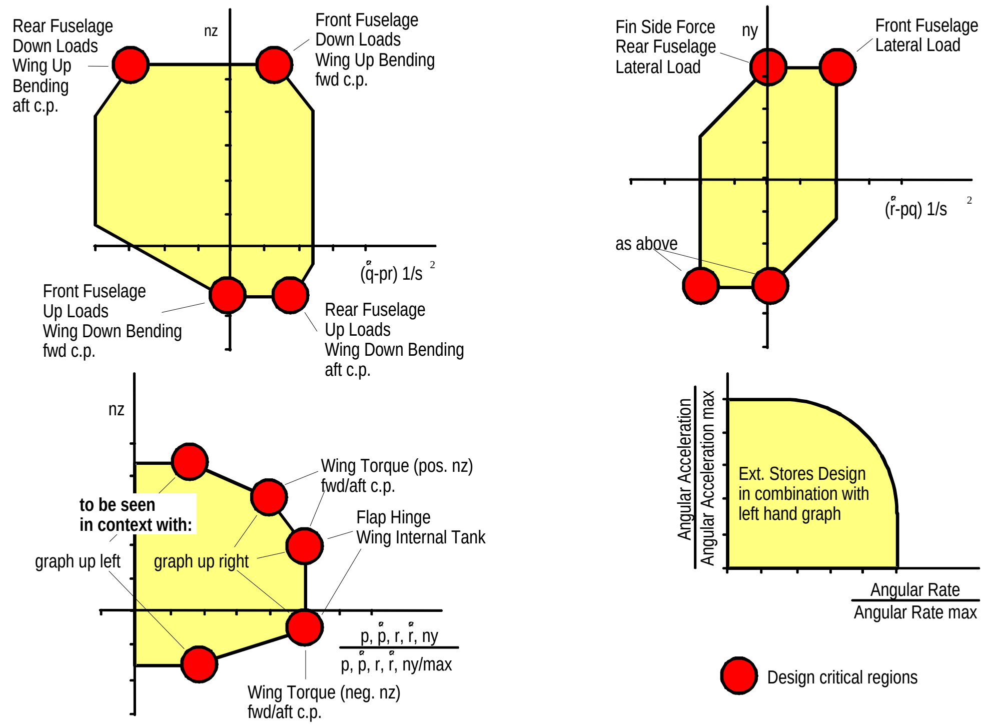



**Fig. 1.2.3-2 Flight Parameter Envelopes for Structural Design**

### 1.2.4 The Loads Model

The Loads Model is the central tool for running the "loads-business". It presents a model (on computer) of the total aircraft, integrating the physics of motion, the aerodynamic dataset, structural design criteria etc. and has interfaces to other disciplines, in detail:

A collection of all input data relevant for the calculation of (static) loads like
- Wind Tunnel and flight test aerodynamic data
- FEM-grid including stiffness matrix
- structural, systems and role equipment masses and mass distributions
- FCS program module (for simulation of flight load specific manoeuvres and landing cases)
- Aerodynamic surface grid

provides a computer program to determine loads and load-specific data like:
- Pressure distributions as a function of Mach number, incidence, control deflections on all surfaces
- Calculation of aeroelastic effects from the coupling of structural flexibility and loads (aerodynamic and inertia)
- Aerodynamic derivatives for total aircraft (used to simulate A/C motion) and aircraft component aerodynamics, harmonised with respect to flight test and wind tunnel data
- Manoeuvre response simulation and interface loads (at component monitor stations), calculation for preparation of component loads envelopes
- Landing gear model and landing simulations (flexible aircraft) with structural loads calculations
- Generation of external loads distributions along structure components axis
- Distribution of design loads on nodal points of the subsequent FEM for stress analysis

and makes available a data base of
- Flexible aerodynamics (components and total aircraft) for the complete Mach/Altitude regime
- Manoeuvre response and -load cases
- Nodal point distributions for design load cases

One of the focal points realised by the Loads Model is the fact, that all (design) load cases are calculated as balanced load cases, i.e. all conditions with respect to aerodynamics, mass distribution and flight manoeuvre match and provide the correct loads for each structure item for any load case. In other words, the sum of net[1)] forces and net moments at all monitoring sections of the structure must be zero:

$$\Sigma_{x,y,z}\ F(x,y,z) \equiv 0 \quad \text{and} \quad \Sigma_{x,y,z}\ M(x,y,z) \equiv 0$$

As mentioned above, such a complete Loads Model was not available for aircraft's developed in the '60 and '70.

**1.2.5 Aircraft Component Loads and -Design Cases**

Loads may be calculated in 3 degrees of refinement:

- Interface or component loads
- Load distributions, e.g. bending moment along wing span, usually one dimensional
- Nodal point loads for Finite Element Analysis

The latter two are suitable to stress analysis and sizing of parts and are usually only applied to design load cases. Component loads, however, are used to find the design load cases, which usually are different for individual structure locations. Therefore the A/C structure is divided in components, with the boundaries representing main constructive items like interfaces, bulkheads, system attachments etc.
An example can be seen in Fig. 1.2.5-1, showing the aircraft components

- Wing
- Wing spoiler
- Front fuselage transport joint
- Fwd front fuselage
- Radom
- Rear fuselage transport joint
- Taileron
- Fin
- Rudder
- Airbrake

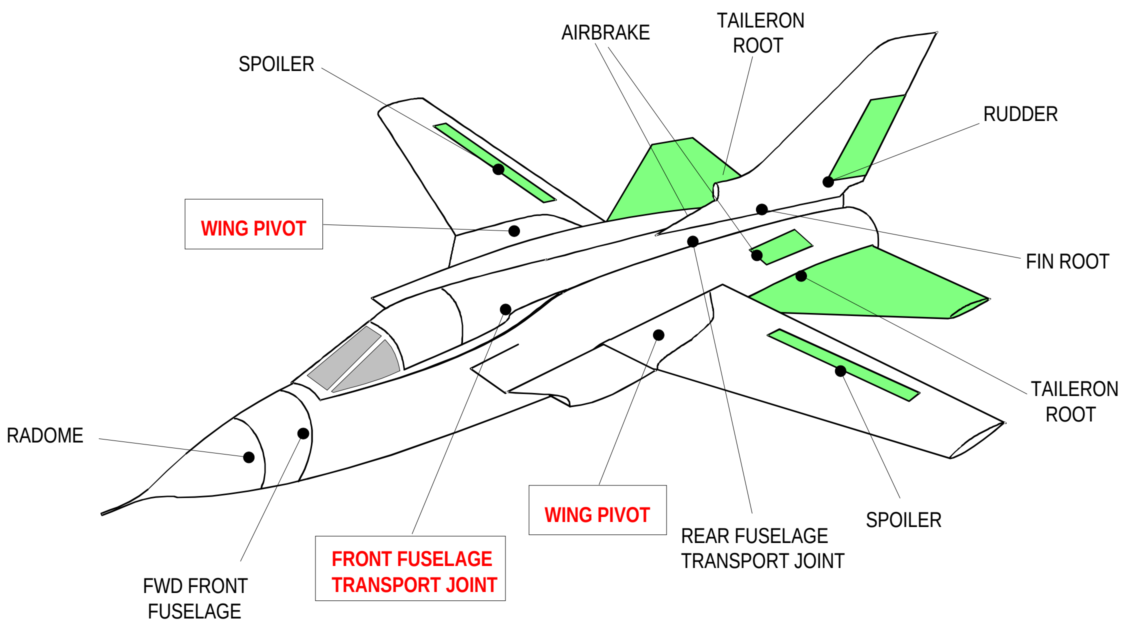



**Fig. 1.2.5-1 Load Monitoring Stations**

-------------------------------
1) net forces / loads = aerodynamic load + inertia load

The respective load monitoring stations are also shown in the figure, where probably the maximum loads are acting. For these stations the forces and moments are calculated for the whole variety of possibly critical manoeuvres (flight/landing conditions, aircraft configuration and mass etc. as parameters) resulting in at least one loads envelope for each monitor station.
Fig. 1.2.5-2 illustrates the concept of load envelopes for the front fuselage and the wing root. Indicated at the corner points of the envelope are the essential conditions, which lead to the design loadcases.

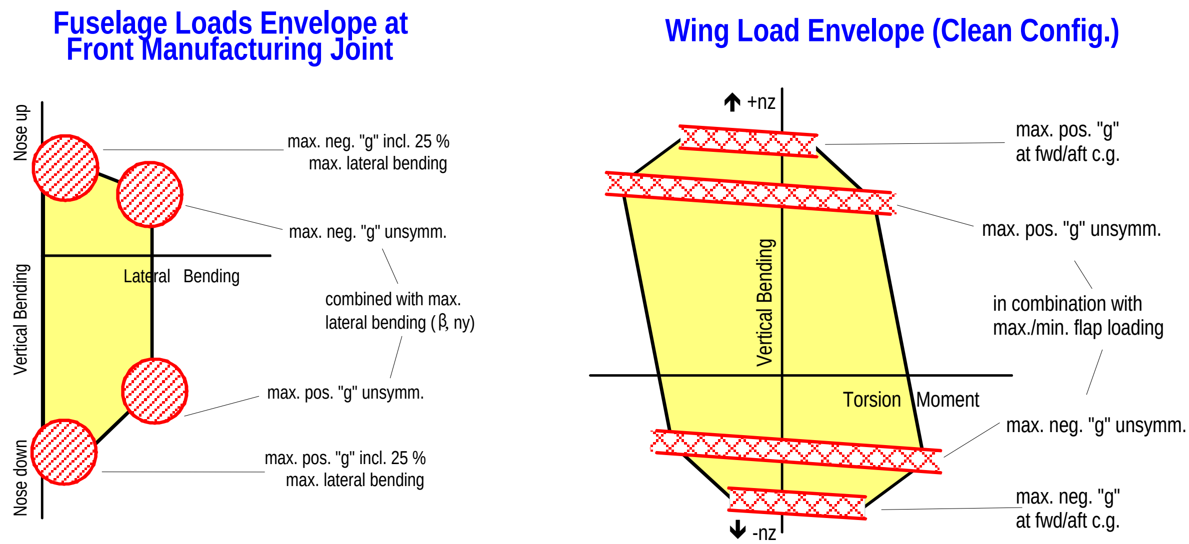



**Fig. 1.2.5-2 Major Aircraft Component Loads Envelopes**

As a first and in many cases correct approximation the design cases can be selected from the corner points of the different loads envelopes.
Usually there is a rather unique relation between corner points of a loads envelope and the flight parameters involved. Therefore considering modifications in the aircraft's role or changes in equipment, mass or performance it is often straightforward to draw conclusions with respect to component load changes and therefore to stress/fatigue implications. This aspect is discussed in chapter 2.

To illustrate the practical sequence of steps to be carried out in order to calculate a flight load at a certain structural component a typical procedure could be as follows, see also Fig. 1.2-1:

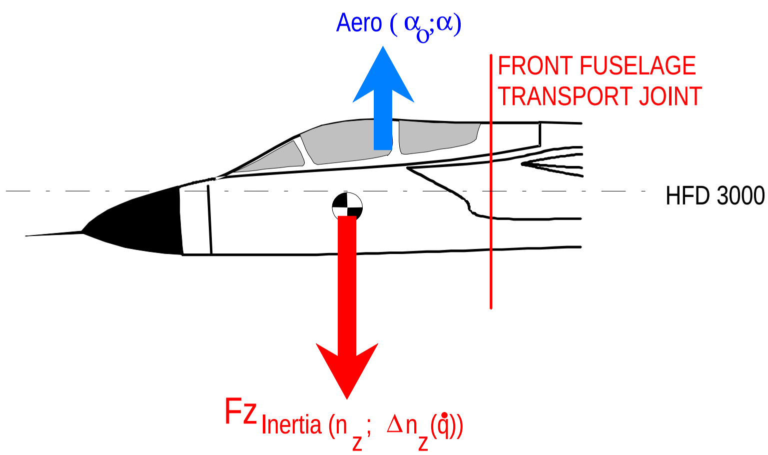
1 Define mass and c.g.
2 Define point in Mach-Altitude range
3 Define sort of manoeuvre (symmetric, roll man., combined man. etc.)
4 Simulate manoeuvre and calculate response parameters
5 Calculate external net loads (forces & moments) on component from aerodynamic pressures, inertia forces etc.
6 Convert external load distribution to nodal point loads on FE grid
7 Analyse structure and determine local stresses (e.g. NASTRAN)

### 1.3 Impact of Changes (Mass, Role, etc.) on Component Loads

Forces acting on an A/C caused by various effects:

| Load | Dependant on (list not Complete) |
|---|---|
| Aerodynamic loads | Incidence, sideslip, control angles, Mach, Altitude etc. |
| Inertia loads | Nx, Ny, Nz, angular ratesand accelerations etc. |
| Engine thrust | Mach, Alt. Combat thrust, idle etc. |
| Internal loads e.g. cabin pressure | Specs, local accelerations |
| Actuator forces for Control surfaces | Hinge moment = f (Mach, Alt.) |
| Hydrostatic pressure | Local accelerations |

The different kind of forces and moments contribute to the loads on the monitor stations in a different manner. The front fuselage up bending is clearly dominated by inertia loads, therefore an increase in the front fuselage mass will result in a higher front fuselage load, see Fig. 1.3-1




*Max. down bending design loads*

| | Air | Inertia | Net | |
|---|---|---|---|---|
| Vertical Shear Force | 100 | -310 | -210 | % |
| **Vertical Bending Moment** | **100** | **-480** | **-380** | **%** |

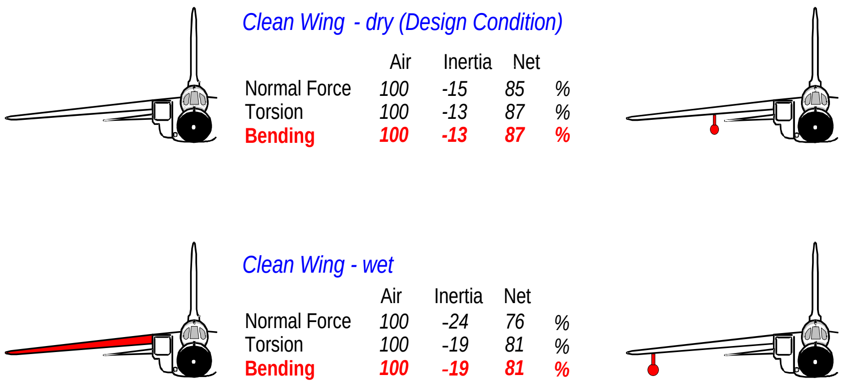
***Conclusion:*** **An increasing Front Fuselage mass will lead to higher Front Fuselage loading.**

**Fig. 1.3.-1 Front Fuselage Transport Joint Critical Load Conditions**

This is not an fictitious case, Tornado front fuselage mass has increased over the years and so the current critical load is definitely higher (max 15 %) than calculated during design.

In a similar manner it can be seen that the rear fuselage monitor station is dominated by inertia loads for the vertical bending, but aerodynamic loading (mainly from the horizontal tail) increases the total load, in contrast to the front fuselage case.
Torque, which is neglectable for the front fuselage design, plays an important part for the rear fuselage and is almost entirely dominated by aerodynamic forces from the taileron (differential tail) and the fin (sideslip and rudder, horizontal gust), which may result in high loads during rapid roll manoeuvres.

Looking at the wing, it is clear that the wing bending is dominated by aerodynamic forces - the wing has mainly to carry the aircrafts weight - but substantial relief come from inertia forces as shown in Fig. 1.3-2.



**Clean Wing - dry (Design Condition)**

| | Air | Inertia | Net | |
|---|---|---|---|---|
| Normal Force | 100 | -15 | 85 | % |
| Torsion | 100 | -13 | 87 | % |
| **Bending** | **100** | **-13** | **87** | **%** |

**Clean Wing - dry + i/b store (350 kg)**

| | Air | Inertia | Net | |
|---|---|---|---|---|
| Normal Force | 100 | -22 | 78 | % |
| Torsion | 100 | -17 | 83 | % |
| **Bending** | **100** | **-17** | **83** | **%** |

**Clean Wing - wet**

| | Air | Inertia | Net | |
|---|---|---|---|---|
| Normal Force | 100 | -24 | 76 | % |
| Torsion | 100 | -19 | 81 | % |
| **Bending** | **100** | **-19** | **81** | **%** |

**Clean Wing - dry + o/b store (350 kg)**

| | Air | Inertia | Net | |
|---|---|---|---|---|
| Normal Force | 100 | -22 | 78 | % |
| Torsion | 100 | -23 | 77 | % |
| **Bending** | **100** | **-24** | **76** | **%** |

***Conclusion:*** **Adding mass to the wing (e.g. carriage of stores) leads to reduced wing loads.**

**Fig. 1.3.-2 Influence of Wing Loading Contitions on Wing Loads**

As indicated, for the Tornado the wing root bending moment is 11% less carrying outboard stores than for the clean wing without stores.
If the assumption for fatigue design includes the majority of missions, flown **with stores** on the outboard wing station, this does not correspond to reality and although the overall aircraft mass might be lower, a severe reduction in lifetime can be the result. This example highlight's, how changes in the usage and configuration affect lifetime and how this can be assessed by rather simple considerations.

The following case of the Tornado undercarriage also shows impact of how design loads were calculated and how usage assumed during design may be completely different from real life usage later:
When it became apparent that the number of starts and landings for a certain squadron was much higher than projected the conclusion was that the nominal lifetime of the squadron's aircraft was exhausted, at least with respect to the landing gear and the support structure. The question arose, whether lifetime could be prolonged and an investigation came to the following conclusions:

- Design of the landing gear was based on the assumption of dry runway conditions. Dry runway landing yields higher loads because of an high friction coefficient. But in reality dry runway landings occurred much less than expected, lifetime could be extended.
- At the same time takeoff and landing mass had increased relative to the design landing weight, causing a lifetime reduction.
- Assumptions during design that approximately 50% of all landings would be 3-point landings were completely unrealistic for this squadron. As only about 10% of all landings were identified to be 3-point landings, the nose landing gear could be expected to have a far longer lifetime than projected.
- Overall methods (e.g. MIL) often result in safe but unrealistic loads. A detailed analysis of landing simulations led to more accurate loads and therefore to a far better assessment of landing gear lifetime.

Considering all the a.m. points together sufficient life for projected usage of airframes for this squadron could be guaranteed.

### 1.4 Qualification of Loads, Static and Dynamic Tests

Static and dynamic loads critical for the structure are checked not only during the early stages of aircraft operational flight test but previously through ground tests as required by the certification procedures for the individual aircraft type.

The major milestones for ground testing are the ground resonance Test (GRT) to check dynamic structural response and confirm flutter margins established analytically to prevent flutter during initial flight tests, the "Major Airframe Static Test" (MAST) and the "Major Airframe Fatigue Test" (MAFT) for critical loadcases identified during structural analysis. The loads for both tests coincide with the loadset used during the development phase, a requirement critical for validation of analytical results.

One possibility to prove the correctness of loads itself can be done by wind tunnel measurements (pressure plotting wind tunnel model or component balances) and/or modern flight load survey. Flight load survey provides information from exact in-flight pressure measurements which, together with wind tunnel data, is fed back to the aerodynamic model of the aircraft and leads to an update of the Loads Model, including other reference data (masses etc.). Then critical load cases are recalculated and thereby confirm/update design load calculations. A typical layout of pressure measurement locations for flight test is shown on Fig. 1.4-1.

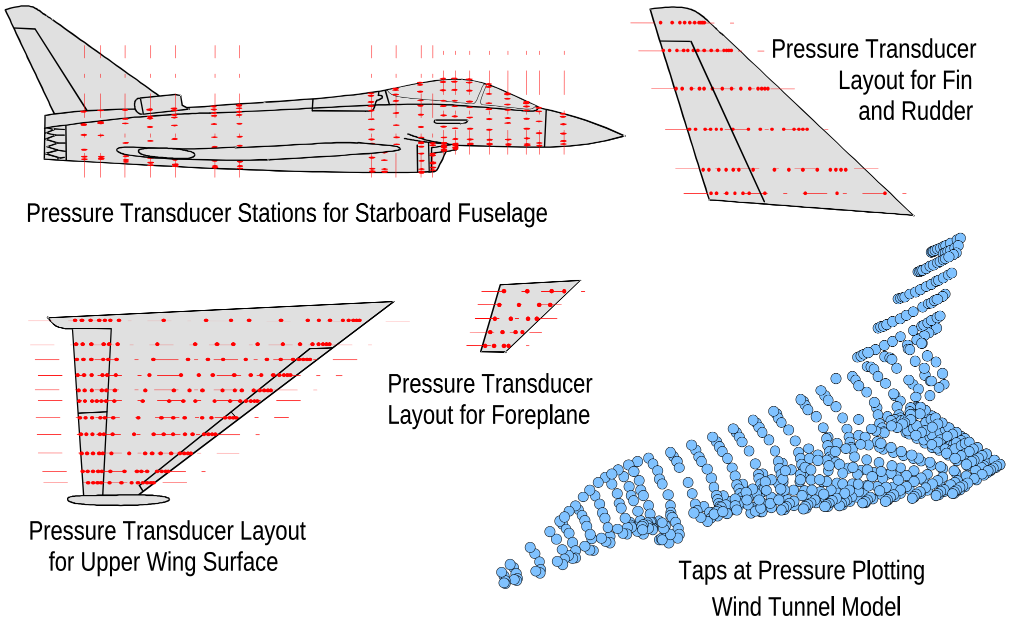



**Fig. 1.4- 1 Prototype Pressure Plotting for Flight Load Survey**

A further procedure to gather flight loads data is by measuring net loads with calibrated strain gauges on test aircraft's.

## 2. AIRCRAFT ANALYSIS USING STATIC LOADS AND FATIGUE LOADS SPECTRA

### 2.1 Static load conditions and fatigue spectrum generation

Safety of flight for any aircraft rely on the recognition that the structure must withstand maximum static loads as well as repeated loads in addition to a certain amount of manufacturing defects and in-service damage throughout the service life without detrimental degradation of the structure leading to catastrophic failure of components. The two major tools for achieving this are the engineering analysis in accordance with the Structural Design Requirements (SDR) and fleet inspection programs.

The SDR documented in the aircraft weapon systems specification are the background for the set of loadcases to be addressed during the sizing of the different aircraft components.

In general these loadsets can be divided into the following groups:

* Limit loadcases
  (relevant for fatigue design requirements)
* Ultimate loadcases
  (relevant for static strength requirements)
* Special loadcases
  (i.e. birdstrike, crash, weapon release, buffet, etc.)

The defined set of missions for the aircraft configuration is the base for the generation of static and fatigue loadcases, which the structure should withstand throughout its intended service usage under defined environmental conditions, demonstrated through engineering analysis in the development phase and proofed via full scale testing (static ultimate and fatigue) later. Typical static loads criteria for a "care free handling"-flight control system equipped aircraft are shown in Fig. 2.1-1.

**STATIC LOADS DESIGN CRITERIA**

| | | |
|---|---|---|
| - **Two Load Levels:** | **Design Limit Load (DLL) =** | **Max. Operational Load in Service** |
| | **Design Ultimate Load (DLL) =** | **Failure Load of Structural Components** |
| - **Ultimate Load:** | **1.4 x Limit Load**<br>**for all Loadcases controlled by FCS** | |
| - **Ultimate Load:** | **1.5 x Limit Load**<br>**for all loadcases not controlled by FCS**<br>**e.g. undercarriage cases, actuator loads, store attachments etc.** | |
| - **Requirements:** | **No structural failure at DUL**<br>**No permanent deformation at DLL**<br>**Buckling of panels must remain elastic at DLL**<br>**No buckling at DUL for items where structural integrity is affected by stability**<br>**No buckling up to 110% DLL for items where operational function is affected by stability** | |

**Fig. 2.1- 1 Static Loads Design Criteria for Airframes**

The results of the calculations are documented in "Static Strength Reports" for each part and form the input during the flight envelope expansion phase from the structural side, the so-called "Strength Envelope".
Durability or fatigue criteria are extracted from the planned/defined mission profile and combined with the overall life requirements in term of flight hours (FH) and/or flights within a defined timeframe of service years. If several aircraft roles are defined in the specification, overall life is split into Flights/Mission, appropriate representation of fatigue critical conditions within the fatigue spectrum is essential.
Manoeuvre loads are covered by an "overall g-spectrum" for the prime aircraft missions, i.e. Air-to-Air or Air-to-Ground as "Points in the Sky" for a given Mach/Altitude level and A/C-Weight/Store-configuration. Excedance curves are then generated as shown in Fig. 2.1-2 for combat aircraft.

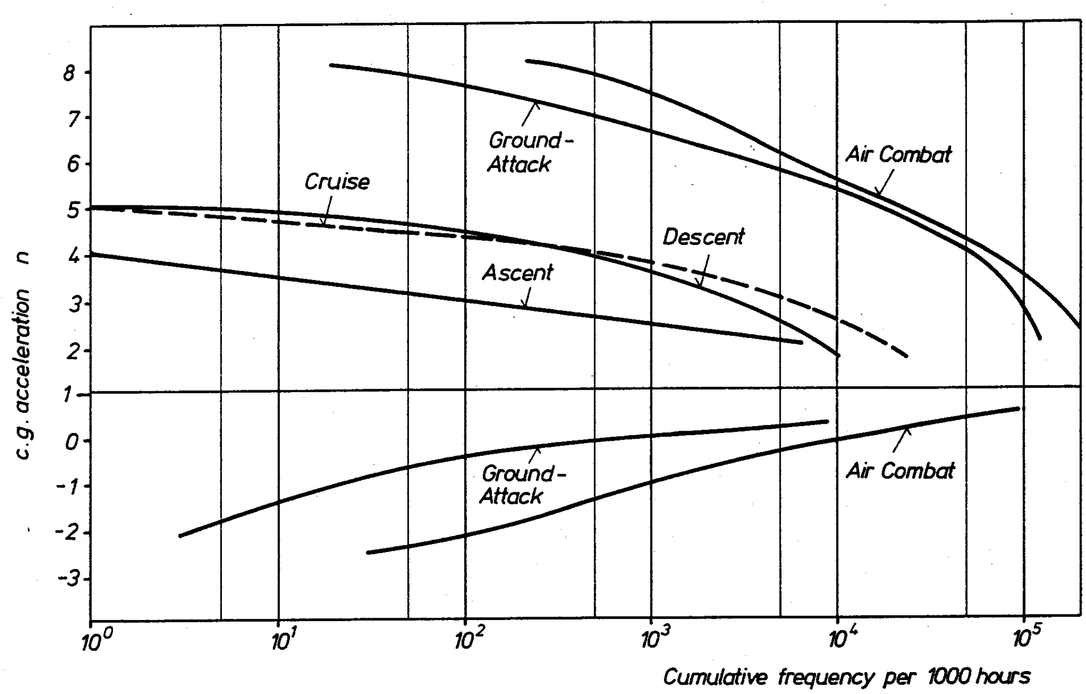



**Fig. 2.1- 2 Typical Excedance Curves for Combat Aircraft**

Special load spectra are needed for components like control surfaces, airbrakes, engine mounts, stores or landing gear.
For transport A/C cabin pressure cycles are an important factor for fuselage durability together with gust spectra.
The various loading spectra form the basis for the fatigue or fracture mechanics analysis depending on the design concept - *Safe Life* or *Damage Tolerance*- adopted.

### 2.2 Conversion of "external loads" into structural airframe loads

For the static and dynamic analysis of airframe structures a mathematical model of the aircraft is build using the Finite Element Analysis (FEA) -technique, representing the geometry and structural stiffness of the major items and providing the bases for generation of "internal" structural forces in components like bulkheads, longerons, skins, spars and ribs etc. as well as other important information like maximum deformation of parts under loads. The detailing of these FE-models depend on the different phases within the iterative process and has improved dramatically with computer performance and modern Pre- and Post-processing capabilities in recent years. "Global" coarse mesh models are used to analyse load paths in the overall structure of aircraft or large components. "Local" models in general are more detailed and they do simulate the special stiffness distribution like thickness changes, cut-outs etc. Structural trade-off studies with this techniques in all phases of airframe development are standard procedures for some years, computer based optimisation of major elements like skin thicknesses are used today in early design stages. A decrease of computer cost and processing time, and in parallel the improvement of model generation, linking the design software (i.e. CATIA) with the loads model output of FEA-nodal forces and the finite element solver through pre-processors, will continue this trend towards more detailed models, better (and more) pre/post-processing information but also increased number of loadcases and refined component loads as discussed in chapter 1.2.
Fig. 2.2-1 shows a typical "coarse mesh"-finite element model of a wing structure with wing box and flaps, where 40-50 "design loadcases" were identified from the loads database of 500 load conditions and used for subsequent strength analysis.

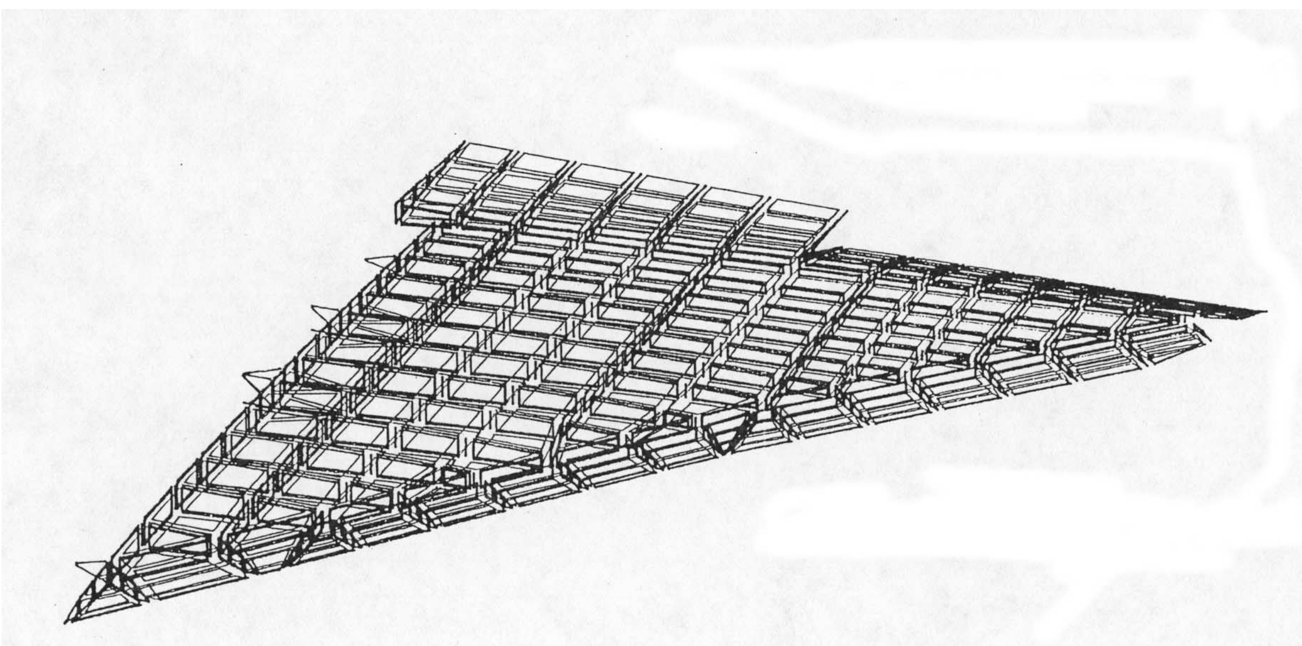


**Fig. 2.2-1 Coarse Mesh FE-Model of Wing Structure**

Fig. 2.2-2 shows a similar model of a center fuselage for a fighter aircraft, cut at Y0-station for symetrie.

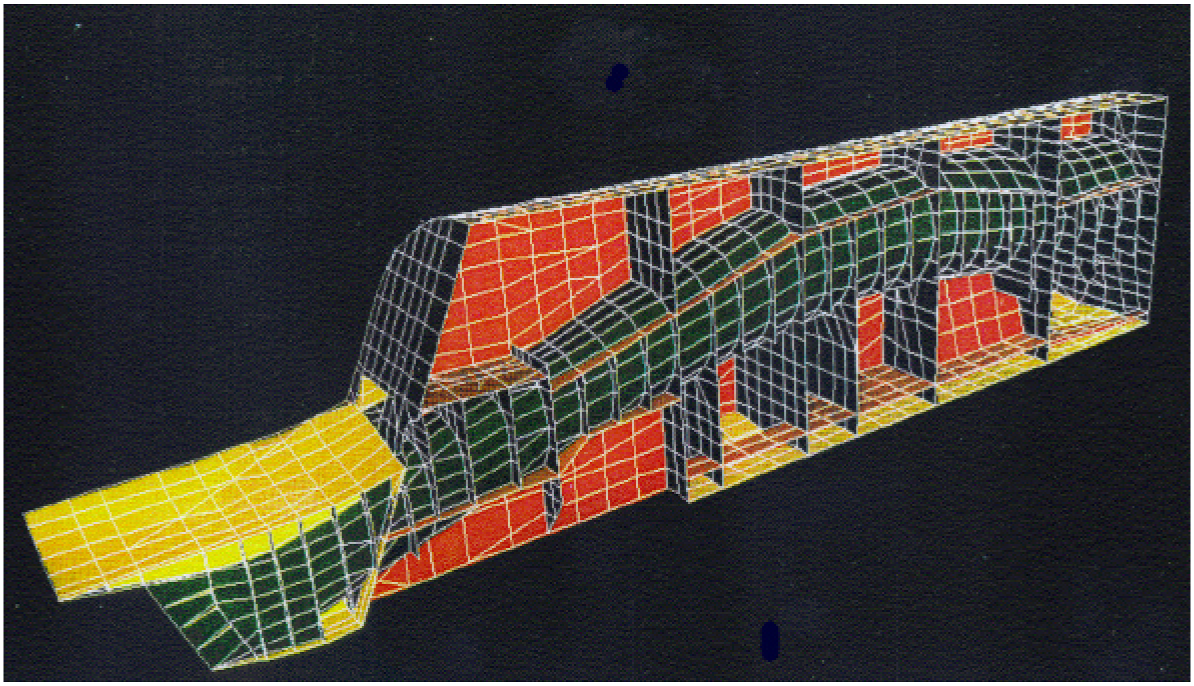


**Fig. 2.2-2 FE-Half-Model of Center Fuselage Structure**

The general trend in international programs towards development and production-workshare is mirrored in the global finite element model as well as through superelement techniques requiring detailed data transfer checks and- protocol requirements. The Eurofighter global model shown in Fig. 2.2-3 was generated by 5 European aircraft companies on different computer hardware and operating systems, therefore model compatibility and -quality checks were essential during the so-called "Check Stress Full A/C- Finite Element Model Static and Dynamic Assembly". The overall model size is about 35000 elements and more than 580 loadcases after superposition. After the unified analysis the results were transferred back to each company for further processing and structural analysis.

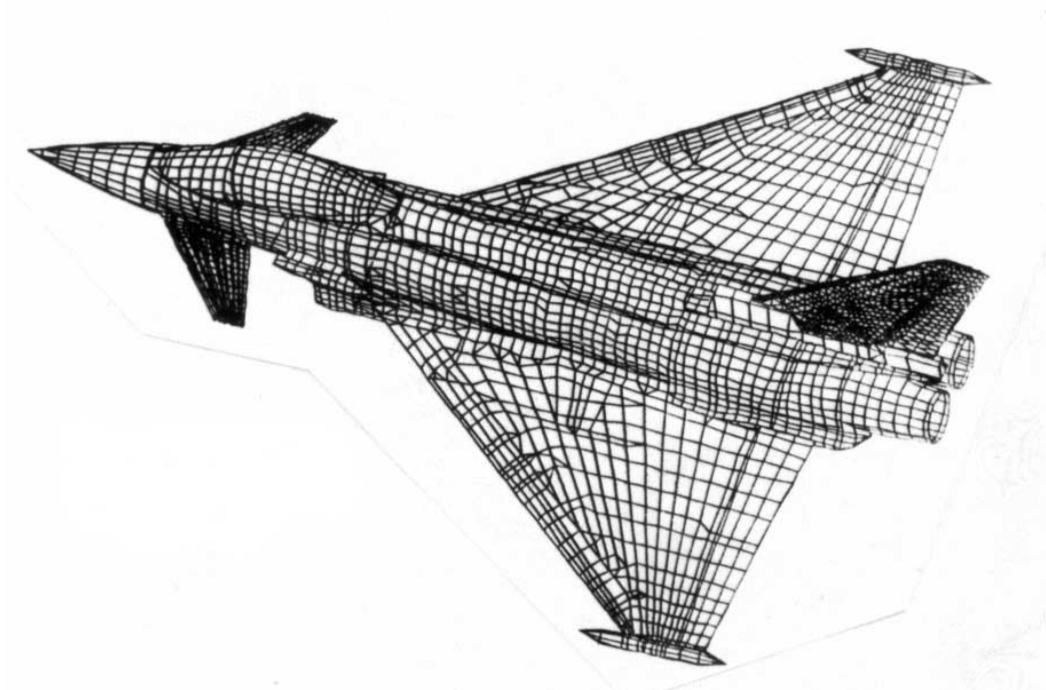


**Fig. 2.2-3 EF2000 Global Model for Unified Analysis**

To further detail the loads in components and individual parts for actual sizing of the structural members, a "cut-out" of the global model with the exact bondary conditions applied to the “edges” of the component of interest from the results of the global model is possible and often used for detail investigations like effects of local cut-outs, reinforcements, stability checks, etc.
Fig. 2.2-4A and 2.2-4B shows an example of this technique for a center fuselage bulkhead.

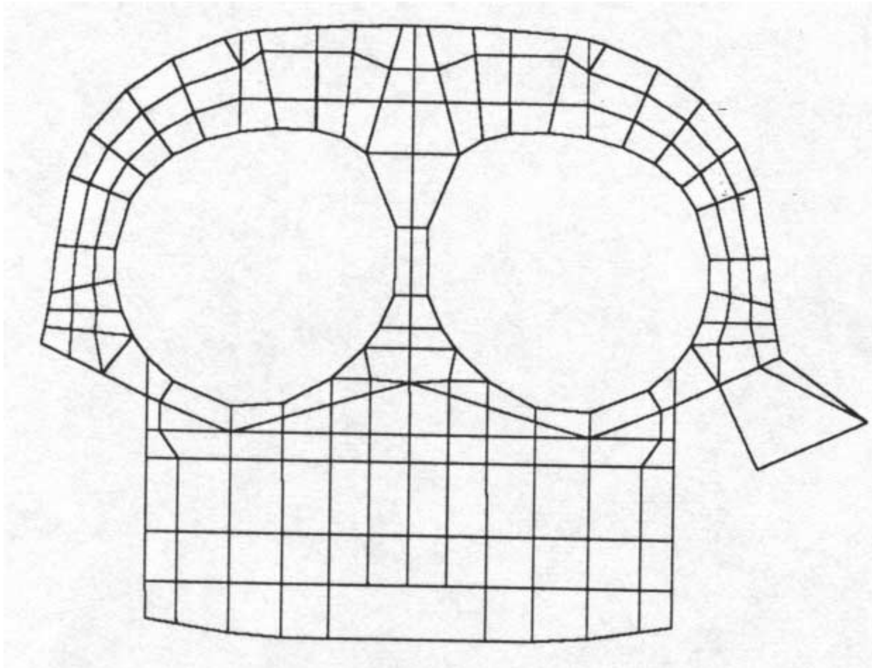


**Fig. 2.2-4A Coarse Mesh FE-Model of Cemter Fuselage Frame**

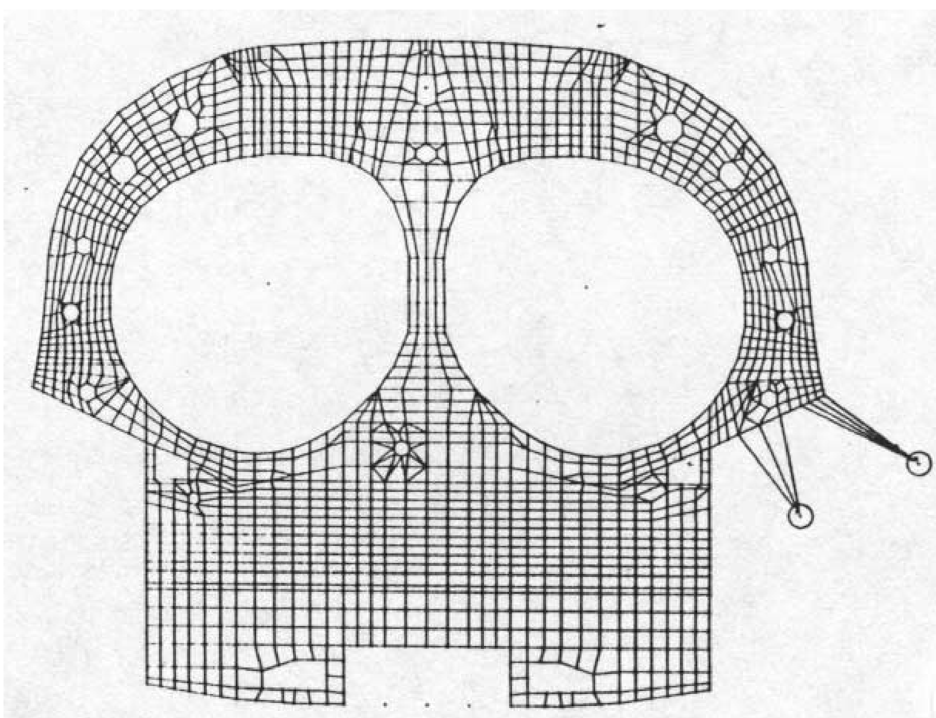


**Fig. 2.2-4B Fine Mesh FE-Model for Detail Analysis**

The results of these detailed model technique provide the background for strength analysis of static ultimate loads as well as fatigue loadcases in accordance with the allowable for the materials used and the geometric effects in the design.

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# Prevention and Control in Corrosion

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**Abstract**

Aircraft corrosion is a never-ending challenge where prevention and control play the fundamental role of ensuring the airworthiness requested.
Corrosion prevention moves from design optimization and proper material selection, but it includes much more following phases like a correct finish specification and plans for effective maintenance, inspection and repair.
Corrosion control, in this meaning including prediction and diagnostics, is complementary to prevention and it is actually the field where more efforts are provided, because early corrosion detection is the easiest way to avoid costly aircraft damage or failures.
In effect, considering that corrosion can account for 60% of all maintenance and repair costs, economic factors must be considered as the most important constraint affecting both prevention and control.
In this lecture the attention will be focused on the different corrosion prevention and control strategies adopted and their actual modifications in accordance with the exacerbation of the aging aircraft issues.

## 1. INTRODUCTION

Control and prevention are both issues used to describe the procedures necessary to provide an effective corrosion maintenance on aircraft.
In effect they must be considered as complementary because corrosion and prevention can have a synergistic effect when each one explicates its specific action. However, it is important to remember that corrosion control includes:

- Corrosion detection
- Corrosion removal
- Renewing the protective systems

On the other side, corrosion prevention is devoted to:

- Material design
- Surface treatments, finishes and coatings
- Corrosion inhibitors compounds and sealants
- Preservation techniques

The entire process including all these phases has been recently called corrosion surveillance[1], indicating the increasing interest from aircraft operators in this matter, largely due to the growing number of aging fleets.
For many years "find and fix" has been the maintenance philosophy all over adopted but now that aircraft are being flown beyond their design life, this practice will not allow a safe and cost effective management of the fleets[2].
So corrosion control and prevention both improved in many aspects in the last decade where environmental constraints also played a very important role.

## 2. CORROSION CONTROL

Many significant advances have been done in this field and probably more are expected in the near future.
In the past, control procedures were just related to scheduled maintenance, non-destructive evaluation and repair but now early diagnostic, condition based maintenance and paint removal technologies are some of the most interesting areas where impressive improvements are continuously carried out.

*Paper presented at the RTO AVT Lecture Series on "Aging Aircraft Fleets: Structural and Other Subsystem Aspects", held in Sofia, Bulgaria, 13-16 November 2000, and published in RTO EN-015.*

2.1 Corrosion Detection and Monitoring

Several NDT were used since many years to detect corrosion, the most commons of them being:

- Visual inspection
- Magnetic particle flaw detection
- X-ray
- Ultrasonic inspection
- Eddy current
- Dye penetrants

However, the increasing corrosion costs recently introduced the need to obtain an early detection and, at the same time, to reduce the unnecessary inspections.
Monitoring during service became the key of this new approach and as a consequence of that different strategies were investigated.

Corrosion data collection and analysis[3] carried out in order to evaluate the areas most affected, estimate the costs and plan the priority of intervention, should be considered as the first stage, followed by the development of in-situ monitoring systems.
Thin film Au-Cd galvanic sensors[4] were developed and successfully installed on military aircraft for monitoring hidden corrosion or corrosivity in aircraft interiors, sealants and coatings[5]. These bimetallic sensors are kept isolated until moisture from the environment bridges the two electrodes: when it occurs the sensors will develop a galvanic current directly proportional to the corrosivity of the trapped moisture. In harsh environments Ni-Au sensors are recommended to provide a long term life.
Promising investigations are being actually carried out to incorporate fluorescence based sensors into paint coatings to provide an easy and economic means to detect corrosion[6].

At the same time, new technologies are more and more used to reduce the time consuming corrosion control activity, and in this area the Thermal Wave NDI[7] that uses an IR video camera to image the surface of the aircraft after the application of a short pulse oh heat seems very interesting as far as the Double Pass retroreflection Aircraft Inspection System (DAIS)[8].
An user friendly probe with a high degree of accuracy and sensitivity, based on Electrochemical impedance (EI) measurements[9] has also been developed.

2.2 Paint and Corrosion Removal

Corrosion control on aircraft often need paint removal but today chemical stripping is no longer the only way to achieve it: diffusion of composite materials, environmental regulations and health and safety considerations are eroding such a  monopoly.
New technologies have been investigated, some of them are widely used, first of all Plastic Media Stripping (Fig. 1), a method that involves subjecting the paint surface to a high pressure stream of acrylic particles, or its closest variation that uses natural products (wheat starch) as the stripping medium.

**Fig. 1 – IAF Tornado stripped by means of Plastic Media**

However, these techniques need special care and are strongly dependent on the operator ability: wrong swell times or stream pressure could remove the clad on aluminum parts or produce damage on composite materials. Researches are in progress to evaluate safe and cost effective alternative solutions: at the moment the two more attractive options seem :

- Flashjet
(a Xenon flashlamp with carbon dioxide pellet)
- Hand held laser

Though at an early stage of development, interesting, at least for components, seems to be a photochemical process that uses only waterborne stripping media with no organic solvents[10].

Once detected, corrosion must be removed by means a pickling operation also necessary as a surface preparation for the following treatments.
Even in this field, environmental compliance needs to substitute the traditional solpho-chromic pickling with a chromate-free alternative.
In this sense a hot sulfuric-ferric acid mix[11] showed at the moment the best performance.

## 3. CORROSION PROTECTION

Many factors have to be taken into account in order to carry out an effective corrosion prevention, most of them are being strongly correlated.
Of course, the starting point must be the materials design that will depend not just on its corrosion behavior but, often more than this, on its mechanical properties.
Once chosen the material, its corrosion behavior will not be fixed unless surface treatments, finishes, coatings and operating environment are not clearly identified.
Aging aircraft and environmentally acceptability have deeply modified old concepts and rules, making of all this matter a big deal of research and development of technologies[12].

### 3.1 Materials

With regard to the materials, if it is true that in the design of new aircraft there exists a trend towards plastics, nevertheless, aging fleet requires in many cases the substitution of alloys with equivalent strength but with higher corrosion resistance in order to extend maintenance schedules and decrease down time.
Particular attention is given to some of the most dangerous forms of corrosion as Stress Corrosion Cracking (SCC) and Exfoliation.
On aluminum alloys, the most interesting performances have been achieved by means of the new tempers (in particular the T77) that allows to have a better control of the size, the spatial distribution and the copper content of the strengthening precipitates[13][14].
The 7055-T77, provides an high resistance to intergranular corrosion, exfoliation and SCC, attributed to its high ratio of Zn:Mg and Cu:Mg and, as a consequence of that, an optimum microstructure at and near grain boundaries[15].
Chemical composition improvements, finalized at a lower Fe and Si content, have brought reduction to pitting initiation on aluminum alloys series 2xxx. 2024-T3 suffers in effect pitting corrosion attack, and this phenomenon is strongly dependent on the Fe and Si bearing second-phase constituent particles[16]: the reduction of their density and size achieved on the derivative alloy 2524-T3, resulted in a reduction of the pits nucleation that can act as potential initiation of fatigue cracks.

### 3.2 Surface Treatments, Finishes and Coatings

Chromate based pre-treatments and chromate pigmented primers are extensively used in the corrosion protection of aluminum alloys because of their excellent performance.
However, many investigations about chromate-free protection schemes have been undertaken since about ten years and some of them (cerium salts, nickel metavanadate[17], Phosphoric Sulfuric Acid anodizing[18], etc.) have already given promising results.
A non-toxic trivalent chromium conversion coating formed applying dilute solutions of basic chromic sulfate plus hexafluorozirconate has been already successfully proposed[19]; it appears at the same time also promising for applications to cadmium and zinc-nickel coated steels.

Anyway, just cadmium plating process, able to provide an effective corrosion sacrifical protection and high lubricity on high strength steels, will be no longer allowed even in military and aerospace applications; its main disadvantages is the toxicity of the cyanide baths.
Many studies have been started and investigations are still in progress to evaluate the best alternative process (zinc-nickel or zinc-cobalt-iron electrodeposition, metallic-ceramic consisting of aluminum particles in an organic matrix spray, etc.)[20]

3.3 Corrosion Preventive Compounds and Sealants

Corrosion preventive compounds (CPC) are able to explicate a really effective corrosion protection, and their use is considered essential to procrastinate the corrosion initiation, extend the scheduled maintenance and reduce costs.
They explicate a combined effect: isolating the metal surface from the environment (barrier effect) by means of a water displacing action carried out by the wax base, and modifying the local environment to make it less aggressive (active effect) by means of the inhibitors included in their formulation.
CPC will be consumed and as a consequence of that they must be renewed with a frequency dependent on the environmental aggressiveness they will be exposed, usually every two years.
Here the environmental compliance forces the R&D to look at new products reducing the VOC content.

Sealants and jointing compounds on the other hand are necessary to avoid both galvanic coupling between dissimilar metals and crevice corrosion that could act as nucleation points for fatigue crack propagation.

It's important to remember that effectiveness of the protective measures both by sealants and CPC depends on a good preparation and proper application; it means that specialists training is a decisive step in corrosion prevention.

3.4 Preservation techniques

Preservation is a really wide area that includes many different actions. The most common preservation technique is washing and rinsing the aircraft after each mission, mostly when it was a low-high mission on the sea: to eliminate chloride and salts from the metal surface in this case is considered a must.
Usually preservation is conceived in agreement with three different strategies, depending on the preservation time:

- short term (0-90 days) preservation
- medium term (up to 1 year) preservation
- long term (beyond 1 year) preservation

When long term preservation is required, dehumidification is necessary.
In any case, more is the preservation time and less will be the manhours spent on maintenance.
Sometimes can be necessary to protect the aircraft or part of it for a long time from contamination and the effects of high relative humidity by means of a barrier material. This is the case of the Nitrogen Purging Packaging (NPP) System[21], that uses a flexible barrier to form a cocoon around the object to be protected and the inner atmosphere is modified to achieve the desired level of relative humidity.

## 4. SUMMARY

Corrosion prevention and control have been separately described in this presentation in order to deal with the most interesting concerns in their respective matters, although they represent a continuous that can be summarized as corrosion surveillance.

They cover many different areas and represent a really multidisciplinary subject strongly related to airworthiness.

This paper contains a selection of the numerous studies and investigations that have been undertaken in the recent past, many of them being still in progress, to ensure an effective corrosion protection and control under the aging aircraft and the environmental constraints.

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# SAFETY AND SERVICE DIFFICULTY REPORTING

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Today, safety is considered to be of highest importance in most societies. In the context of the military, safety is essential to averting loss of life and damage to a high-value asset. While safety may take second place to winning a war, its importance is further accentuated because of its connotation to battlefield readiness. There have been numerous instances to illustrate this last point. To wit:

- Widespread Fatigue Damage (WFD) was discovered in "weep holes" of fuel tanks of some C-141 military transport airplanes. Because of the loss of minimum residual strength, with the attendant risk of catastrophic fracture posed by WFD, the entire fleet had to be grounded and an expensive refurbishment program had to be undertaken before the fleet was deemed to be airworthy. In this instance, the unsafe condition was detected and corrected quickly, so no lives were lost nor did any of the airplanes in the fleet suffer catastrophic damage. However, the grounded aircraft were certainly not battle-ready for a certain length of time. Had they been sent into battle, they would have had to be operated under severe flight restrictions and, thus, their utility to serve the purpose of the deployed forces would have been very restricted. Had they been deployed without any restrictions, in all probability they would have been unable to complete their missions and the Air Force could have lost valuable aircraft assets. Also, the necessary logistic support to properly carry out tactical operations in the battlefield would not have been available.

- WFD was the primary cause of a highly publicized air accident involving a commercial aircraft. The wide publicity given to that single accident, abetted by on-site video tape recording of the condition of the aircraft after it had landed, shook the confidence of the public in the safety of commercial aviation. As a result, inspection and refurbishment of 3000 jet transport airplanes among a fleet of about 5000 was mandated by the authorities, to be undertaken on an urgent basis. The economic impact of this mandate on the airlines, the aircraft manufacturer and the flying public was high and resulted in numerous complaints to the regulatory authorities. It must be noted that since that time more than twelve years have elapsed without a single accident attributable to WFD.

These instances explain my motivation for including the subject of safety during this Lecture Series. However, the subject is extensive and so many books have appeared that address some aspect or the other that my remarks are meant to complement the existing literature. Much of what I intend to share with you today is not something I have developed on my own, rather it has been influenced by my comrades and peers when I was in the civil aviation community.

**Scope of the Lecture - Analysis and Data Requirements for Assessment of Operational Safety**:

An aircraft is an assemblage of complex and highly integrated sub-systems - the structure, the power-plant, the electrical, the mechanical, and hydraulic systems, the avionics suite, the human-in-the-loop to name a few. To eliminate the risk of the sub-systems to fail, individually or in concert, safety analyses are routinely performed by aircraft manufacturers. The manufacturer also conducts analyses to ascertain the consequence of a failed part to assure that it does not in any way threaten the safety of the entire system.

Before an aircraft model enters service, whether for military or civil use, the design has to satisfy a rigorous set of requirements, which are governed by regulations. These requirements include an analysis of the probability of failure of each component and the hazard caused by the failure. This subject, termed as "Systemic Safety [1]," will be beyond the scope of this lecture. Rather, the remarks will concentrate on the operational phase of the aircraft's life. That is the phase subsequent to the aircraft put into operational use for the first time.

However, keep in mind that before the aircraft enters the fleet, there are numerous design reviews, ground and flight tests, and production approvals that are required to assure that the aircraft is safe and able to perform as intended in the operating environment. At times, the origin of problems that are encountered in service may be inherent in the design or the manufacturing stage or due to construction methods. For instance, an element in the chain that led to the failure of the commercial aircraft mentioned earlier was a failed bond. The failed bond resulted from an inadequate bonding process. It created stress risers at the rivets, which were designed to merely serve as secondary conduits for transferring

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load. The resulting fatigue cracks were aggravated by loss of material due to corrosion, resulting in intrusion of moisture from condensation and precipitation. Such problems that are encountered in service must be quickly corrected in order to prevent accidents and to maintain battle-readiness of the fleet. An essential requirement for quick resolution of these type of problems is a technical team that is familiar with not only the design features of the aircraft model and any subsequent modifications that had been effected previously but also the original design philosophy that guided the design. Often, it is beneficial to retain some members of the original design team to serve in the maintenance group in order to maintain the necessary know-how.

**Measurement of Safety**

In order to assess safety of a system after it enters service one must define safety and establish a set of metrics (measurement standards) for safety. A metric may be the number of failures per one thousand operations, or it may be an incident rate or an accident rate. Such gross metrics are normally refined by dividing the accidents into categories by causal relationships. Furthermore, metrics are often normalized in terms of usage. In any event, the establishment of safety metrics has been subjective, to say the least, and a bit disorganized from the standpoint of relating the accident cause, the events leading up to the accident, and the design fix. The problem is best illustrated through Figure 1, and 2. Both figures have been extracted from publicly released Boeing Airplane Company documents [2, 3]. They depict the relative risk of an accident as a function of the phase of flight, based on historical data. Clearly, if miles flown is chosen as the normalizing factor for a safety metric, the metric chosen ignores the fact that risks between destinations involving multiple flight legs and the risk involved for a single leg, for the same distance traveled, are unequal - hence, the metric would be inappropriate. Similarly, in the assessment of military aircraft, the hours of operation is usually chosen as the normalizing factor but such a choice ignores the fact that the mission profiles could be vastly different, even for the same aircraft model but used in different squadrons. Thus, the establishment of multiple metrics for risk using the same database increases the opportunity for establishing a correlation between data and risk, thereby making the safety management system more robust.

**Figure 1.**

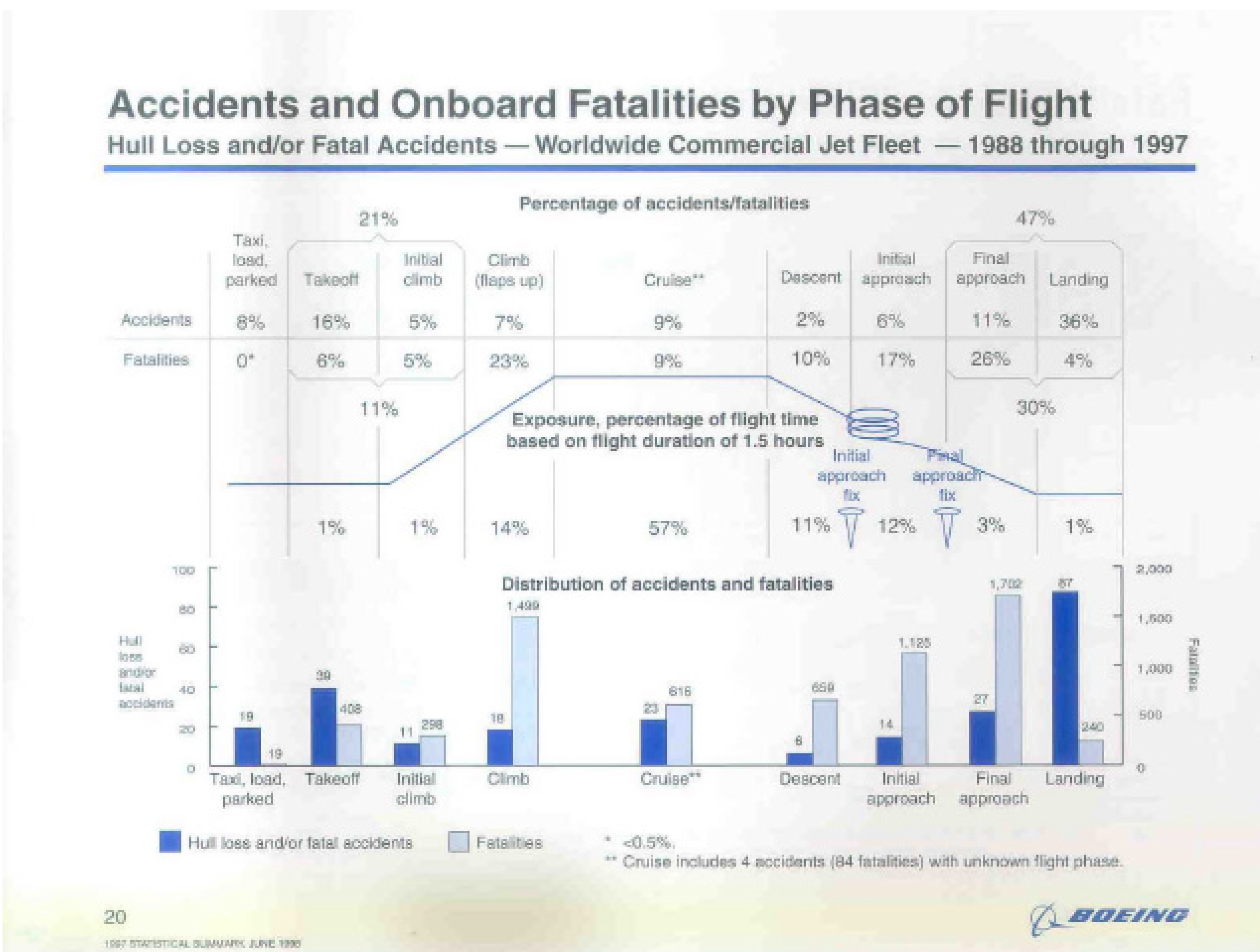
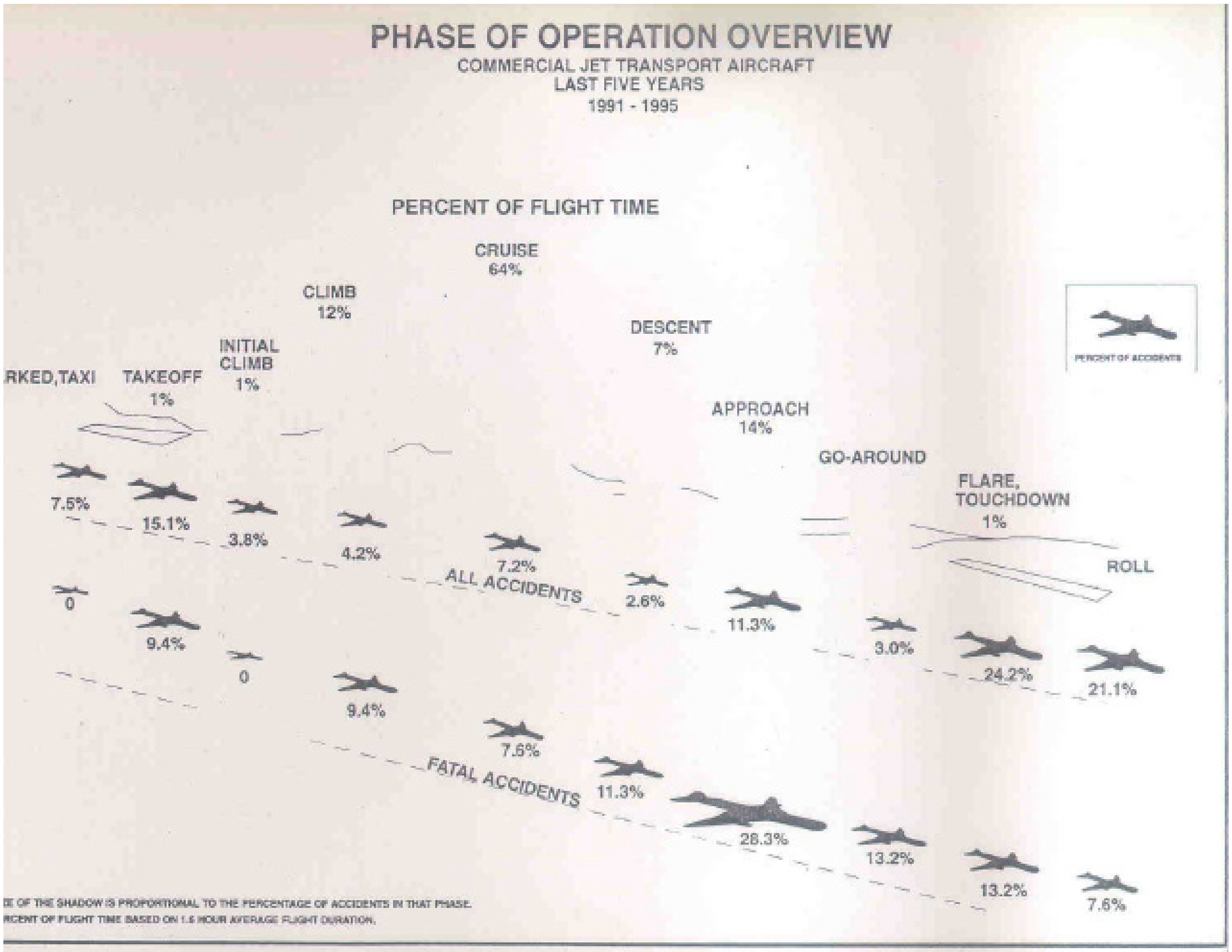


**Figure 2.**




## Accidents and the Role of Precursors:

It is generally agreed that there exist certain precursors to each accident and incident. If one of these precursors is not recognized and the underlying condition that has caused it is not corrected in time, then it can graduate into an incident or even an accident. Aircraft are highly engineered systems, endowed with redundancies and fail-safe features. They are "noisy" systems. That is, they can give so many indications, of which only a few are precursors, that one can easily be lulled into complacency. Fail safety embraces two concepts. One is the concept that the first failure does not impair functionality of the system. The second is that the first failure must be obvious to the extent that it will, in all likelihood, be detected well before the onset of subsequent failures, which may endanger the safety of the system. Thus, the first occurrence of a service difficulty associated with a sub-system in an aircraft is a prospective precursor of progressive failures that could result in an incident or accident. Furthermore, multiple occurrences of service difficulties, especially after corrective actions have been attempted, are indicators that the risk of an incident or accident is rising. To take full advantage of being given such warnings, the organization responsible for safe operation of the aircraft must systematically collect reports of service difficulties. Just as importantly, this same organization must systematically and expeditiously analyze the reports being collected to establish their root cause of the difficulty or difficulties and its potential for a resulting accident or incident. The analysis must be accomplished early in order to allow sufficient lead-time for corrective action to be taken. Even with a service difficulty collection and analysis system in place, the organization will be unable to use it to reduce or eliminate incidents and accidents unless higher management in the organization recognizes their value and directs development and implementation of corrective action. Clearly, improved safety will result if attention is more focused on precursors.

## Detection of Service Difficulty

A Service Difficulty is symptomatically manifested by one of the following:

Visual, such as cracks, warning lights, observation of smoke, etc.

Aural, such as alarms, abnormal sounds, etc.

Tactile, such as excessive vibration, electrical shock, stick response, etc.

Olfactory, such as fumes from electrical systems or oil or rubber, etc.

Response to transducer devices such as those used for nondestructive inspection of structural components.

Service difficulties can manifest themselves during airworthiness inspections and other maintenance related activities. One example is the detection of a structural fatigue crack in an area adjacent to the area being inspected. The maintenance program had no instructions for inspecting this cracked area. Had the service difficulty report not been filed on this crack, and had a single observant authority representative not discovered this difficulty report and investigated it, further crack growth in this area and other aircraft might have occurred and graduated into something serious.

It would be erroneous, however, to draw a correlation between the number of service difficulty reports generated and risk. A large number of reports may mean that the operational and maintenance personnel are alert and diligent in reporting discrepancies, not necessarily that the risk of failure is rising. In this case, it may simply be a tribute to the robustness of the inspection and maintenance program. Only systematic analysis performed by trained and knowledgeable analysts can correlate the risk level to the number of service difficulty reporting rates.

## Analysis and Data Requirements

There is a symbiotic relationship between: (a) the purpose of safety analysis, (b) the methodology to be used for evaluating safety (or risk), (c) the data required to perform safety analysis, (d) the confidence to be reposed in the results, (e) and the burden of the data collection effort. All five aspects will have to be considered in concert to devise a robust system that balances system costs (figure 3).

**Figure 3. Elements Associated With Service Difficulty Related Activities**

**Define Requirements**
↗ ↘
**Validate Requirements** **Collect Data**
↑ ↓
**Analyze Data** **Store & Retrieve Data**
↖ ↙
**Ensure Data Quality Standards**

Safety analysis may be required for a variety of purposes. For instance, to gage the general health or safety of the fleet would require a different methodology and could be accomplished with an abbreviated set of data elements than what might be needed for a forensic analysis of an accident or incident. Thus, the circulation of a questionnaire among the various groups involved in maintaining safety to establish the connections between analysis methodologies that are being used or desired, and the respective data requirements is advocated.

Aircraft systems are becoming more and more complex, placing more sophisticated demands on data collection and analysis methods. Also, the increased attention being given to safety and the accompanying demand for data driven safety programs, makes the data elements that would have been considered adequate in the past appear as lacking in precision and detail. Thus, the number of data elements, the extent of detail to be included in any gathering effort, and the configuration of the database itself should be designed to allow for some growth in data requirements. It is imperative that an organization designing a service difficulty reporting system that mandates the collection of certain data elements simultaneously considers the analysis to be conducted of the collected data. Many existing databases, such as the Service Difficulty Reports being maintained by the Federal Aviation Administration have come in for criticism [4, 5]. These databases collect many pieces of data that are not used or are redundant. Such databases are primarily designed to facilitate the collection of data but with little or no attention being paid to the needs of the analyst to correlate the data with the airworthiness of the individual aircraft or the fleet. Hence, it is advocated that a safety program - any safety program - be revisited, perhaps re-tuned, every five years, both from the viewpoint of currency and adequacy.

Avionics-related malfunctions may have serious implications in terms of safety of new generation aircraft. These systems are being given more authority over primary flight control of the aircraft. Thus, the reporting of associated malfunctions, defects, and failures become more critical to proactive safety analysis. Their failures during any phase of operation may have safety implications. In any event, data should be collected to support explicit program requirements. Terminology such as “abnormal or emergency actions” and “endanger the safe operation” in regulations will not provide consistent reporting without further definition and guidance.

The distinction between reliability and safety is much debated in the context of data requirements. It has been argued the data needed for performing safety analysis is not as extensive as that for maintaining reliability. However, with the emergence of the nearly synonymous philosophies of Reliability-Based Maintenance and Condition-Based Maintenance, which takes the risk of failure(s) into account, the distinction is blurring.

Hand-held electronic devices have eliminated much of the paperwork in data gathering. Such devices make possible the gathering of voluminous data without making the data gathering effort either burdensome or time consuming. In fact, the development of software that can readily depict on a hand-held device the geometrical layout of components as well as the inter-connectivity of the functional units would make facilitate acquisition of data that capture more details about a malfunction or a failure than is now the case. Electronic entry of data has another great advantage, viz., it avoids data corruption due to transcription errors and expedites the addition of more data elements to the database.

## Data Standards

The term "data quality" can at once mean different things, such as erroneous data, inconsistencies in the data, insufficient detail that has been captured in the data, completeness of the data sets, etc. Each of the meanings has a bearing on safety. For instance, there is a wealth of data about instances of cracking in airframe structures but they are not very useful because of lack of precision and standardization. From the standpoint of systematic analysis of large quantities of data, the most important attribute of a safety related database is consistent reporting. The adoption of a common terminology is one aspect of consistency. Clarity of terminology is a related aspect. A critical need for data that is stored in relational databases is that fields should be assigned in each data record (report) to allow for supplementary comments by the mechanic. The FAA maintains one of the largest safety database in the world, the Service Difficulty Reporting (SDR) System. However, because the SDR is a relational database, no provision has been made for supplementary notes. For instance, the database does not allow the mechanic to record the specific location of a crack, even if one is found in a principal structural element. As a result, many users rely on the SDR system only to confirm critical problems that have already been found or suspected - not to give precursory evidence of potential incidents or accidents.

. Table 1 exemplifies a form for data recording, which would make possible supplementary notes to be made by the mechanic or inspector. The form for reporting incidents was drafted by an internal FAA team, of which the author was a member. The data requirements for reporting service difficulty can be developed in an analogous fashion.

**TABLE 1: EXAMPLE OF A FORM FOR RECORDING AN INCIDENT**

BATCH # ______________________________ I.D. #__________________________________

| REV. | DATE | ANALYST | REVIEWER |
|---|---|---|---|
| 0 | ___/___/___ | ________________________ | ________________________ |
| 1 | ___/___/___ | ________________________ | ________________________ |
| 2 | ___/___/___ | ________________________ | ________________________ |
| 3 | ___/___/___ | ________________________ | ________________________ |

EVENT ID NUMBER:
___/___/___/____
YY MM DD SE

TIME OF EVENT: (SELECT ONE)
UNKNOWN ___________
UT ___________
LOCAL TIME ___________

EVENT CLASSIFICATION:
HAZARDOUS__________________
MAJOR ______________________
MINOR ______________________
DAMAGE ____________________

LOCATION:
DEPARTURE AIRPORT_______________________
DESTINATION AIRPORT _____________________
EVENT LOC. (CITY) __________________________
COUNTRY (EVENT) __________________________
LAT/LONG____________________________________
UNKNOWN ___________________________________

AIRCRAFT:
TYPE-SERIES___________________
A/C MAKE _____________________
FUSELAGE NO._________________
DATE MANUFACTURED ________
TAIL NUMBER ________________
SERIAL NUMBER_______________
ENGINE MAKE_________________
ENGINE MODEL(S) _____________
ENGINE SERIAL NO(S). _________
FLIGHT NUMBER ______________

TYPE OF MISSION: (SELECT UP TO 2)

| | |
|---|---|
| SCHEDULED PAX | CARGO |
| UNSCHEDULED PAX | FERRY |
| FLIGHT TEST | TRAINING |
| UNKNOWN | |
| MAINT | |

AIRLINE/OPERATOR:
OPERATOR NAME__________________________
OPERATOR OAG CODE ______________________

METEOROLOGICAL/ENVIRONMENT CONDITIONS:

IMC/VMC_______________________
CLOUD CEILING FT OR M_______
LIGHT CONDITIONS
DAY/NIGHT/DUSK/DAWN_______
VISIBILITY FT, M, MI ___________
WIND: DIRECTION _____________
VELOCITY IN KTS______________
TEMPERATURE F OR C _________
MICROBURST__________________
CAT___________________________
WINDSHEAR___________________

VERTICAL TURBULENCE___________________
HAZE ________________________________________
HAIL_________________________________________
BIRDS________________________________________
SNOW/SLUSH_________________________________
SAND/ASH____________________________________
THUN STRMS_________________________________
LIGHTNING __________________________________
OTHER WEATHER ___________________________
ICE/RAIN/FOG/GUSTS _______________________

PHASE OF OPERATION

| | | |
|---|---|---|
| BOARDING | DESCENT | DEBOARDING |
| CARGO LOADING | APPROACH | PARKED |
| ENGINE START | INITIAL | REFUELING |
| TAXI | FINAL | INSPECTION |
| TAKE OFF | LANDING | TOWED |
| ROLL | FLARE & TOUCHDOWN | SERVICING |
| ROTATION | ROLL | UNKNOWN |
| INIT CLIMB | TOUCH AND GO | CLIMB TO CRUIS |
| GO AROUND | CRUISE | TAXI |
| DURING DIVERT | | |

<u>HARDWARE INVOLVED IN INCIDENT</u>: <u>ATA CODE</u> ___/__/___

NAME ____________________________________
MODEL ____________________________________
MAKE ____________________________________
LOCATION____________________________________
PART NUMBER________________________________
TOTAL TIME_________________________________
TIME SINCE O/H_____________________________
CYCLES SINCE O/H___________________________
TOTAL CYCLES_______________________________

__ __ __ __ __ __ __ __ __ __ __ __ __ __ __
__ __ __ __ __ __ __ __ __ __ __ __ __ __ __
__ __ __ __ __ __ __ __ __ __ __ __ __ __ __
__ __ __ __ __ __ __ __ __ __ __ __ __ __ __
__ __ __ __ __ __ __ __ __ __ __ __ __ __ __

<u>TYPE OF HUMAN MACHINE INTERFACE ERROR</u>
Suggest that a coded list be developed that is similar to ATA codes

<u>NAT. AVIATION SYSTEM (NAS)</u>: TBD

<u>FLIGHT CREW EXPERIENCE</u>:
CAPTAIN ______________________
TIME IN TYPE ACFT ____________
FIRST OFFICER
TIME IN TYPE ACFT ____________
SECOND OFFICER
TIME IN TYPE ACFT ____________

PILOT IN COMMAND ______________________
TOTAL FLYING TIME ______________________
TOTAL FLYING TIME ______________________
TOTAL FLYING TIME ______________________

FLIGHT CREW__________________
MAINTENANCE_________________
OPERATOR____________________
MANUFACTURER_______________
NTSB____________________________
WAAS___________________________

<u>DATA SOURCES</u>:
ATC_______________________________________
CAA ______________________________________
FLT INT ___________________________________
FLIGHT SAFETY FOUNDATION______________
NEWS_____________________________________
AIRCLAIMS ________________________________
OTHER____________________________________

<u>BRIEF DESCRIPTION</u>: ____________________________________________________________

Describe the event/situation. Keeping in mind the following topics, discuss those which you feel are relevant and anything else you think is important. Include what you believe really caused the problem, and what can be done to prevent a recurrence, or correct the situation. (USE ADDITIONAL PAGES IF NECESSARY)

1. <u>CHAIN OF EVENTS</u>
   How the problem arose
   Contributing factors
   How was it discovered
   Corrective actions taken
   System configurations and operating modes
   What procedures were used
   How did you decide what to do
   What stopped the incident from becoming an accident
   Failure in Cockpit Resource
   Management Fatigue

2. <u>HUMAN PERFORMANCE CONSIDERATIONS</u>
   Perceptions, judgements, decisions
   Factors affecting the quality of human performance
   Actions or inactions
   Lack of positional awareness
   Lack of awareness of circumstances of flight
   Incorrect selection on instrument/navaid
   Action on wrong control/instrument
   Slow/delayed action
   Omission of action/inappropriate action
   Fatigue
   State of mind
   Lack of qualification/training/experience
   Incapacitation/medical or other factors reducing crew performance
   Deliberate non-adherence to procedures

<u>FULL NARRATIVE</u>: ____________________________________________________________

<u>ANALYST COMMENTS</u>: __________________________________________________________
Factors Relevant to Incident
(Each incident usually has more than one factor)

| Group | Factor | No. acc. |
|---|---|---|
| A. *Causal factors* | | |
| A.1 Aircraft systems | 1.1 System failure – affecting controllability<br>1.2 System failure – flight deck information<br>1.3 System failure - other | |
| A.2 ATC/Ground aids | 2.1 Incorrect or inadequate instruction/advice<br>2.2 Misunderstood/missed communication<br>2.3 Failure to provide separation - air<br>2.4 Failure to provide separation - ground<br>2.5 Ground aid malfunction or unavailability | |
| A.3 Environmental | 3.1 Structural overload<br>3.2 Wind shear/upset/turbulence<br>3.3 Icing<br>3.4 Wake turbulence - aircraft spacing<br>3.5 Volcanic ash/sand/precipitation etc.<br>3.6 Birds<br>3.7 Lightning<br>3.8 Runway condition unknown to crew | |
| A.4 Crew | 4.1 Lack of positional awareness - in air<br>4.2 Lack of positional awareness - on ground<br>4.3 Lack of awareness of circumstances in flight<br>4.4 Incorrect selection on instrument/navaid<br>4.5 Action on wrong control/instrument<br>4.6 Slow/delayed action<br>4.7 Omission of action/inappropriate action<br>4.8 "Press-on-Us"<br>4.-9 Failure in CRM (cross-check/co-ordinate)<br>4.10 Poor professional judgments/airmanship<br>4.11 Disorientation<br>4.12 Fatigue<br>4.13 State of mind<br>4.14 Interaction with automation<br>4.15 Fast and/or high on approach<br>4.16 Slow and/or low on approach<br>4.17 Loading incorrect<br>4.18 Flight handling<br>4.19 Lack of qualification/training/experience<br>4.20 Incapacitation/medical or other factors reducing crew performance<br>4.21 Failure in look-out<br>4.22 Deliberate non-adherence to procedures | |
| A.5 Engine | 5.1 Engine failure<br>5.2 Propeller failure<br>5.3 Damage due to non-containment<br>5.4 Fuel contamination<br>5.5 Engine failure simulated | |
| A.6 Fire | 6.1 Engine fire or overheat<br>6.2 Fire due to aircraft systems<br>6.3 Fire - other cause<br>6.4 Post crash fire | |
| A.7 Maintenance/ ground handling | 7.1 Failure to complete due maintenance<br>7.2 Maintenance or repair error/oversight/inadequacy<br>7.3 Ground staff struck by aircraft<br>7.4 Loading error<br>7.5 SUPS - Suspected Unapproved Parts<br>7.6 Unapproved Parts | |

| Group | Factor | No. acc. |
|---|---|---|
| A *Causal factors* | | |
| A.8 Structure | 8.1 Corrosion/fatigue<br>8.2 Overload failure<br>8.3 Flutter | |
| A.9 Infrastructure | 9.1 Incorrect, inadequate or misleading information to crew<br>9.2 Inadequate airport support | |
| A.10 Design | 10.1 Design shortcomings<br>10.2 Unapproved modification<br>10.3 Manufacturing defect | |
| A.11 Performance | 11.1 Unable to maintain speed/height<br>11.2 Aircraft becomes uncontrollable | |
| A.12 Other | 12.1 Caused by other aircraft<br>12.2 Non-adherence to cabin safety procedures | |

B *Circumstantial factors*

| | | |
|---|---|---|
| B.1 Aircraft systems | 1.1 Non-fitment of presently available safety equipment (GPWS, TCAS, windshear warning, etc.)<br>1.2 Failure/inadequacy of safety equipment | |
| B.2 ATC/ground aids | 2.1 Lack of ATC<br>2.2 Lack of ground aids | |
| B.3 Environmental | 3.1 Poor visibility<br>3.2 Other weather<br>3.3 Runaway condition (ice, slippery, standing water, etc.) | |
| B.4 Training | 4.1 Training inadequate<br>4.2 Presented with situation beyond training<br>4.3 Failure in CRM (cross-check/co-ordinate) | |
| B.5 Infrastructure | 5.1 Incorrect/inadequate procedures<br>5.2 Company management failure<br>5.3 Inadequate regulation<br>5.4 Inadequate regulatory oversight | |

C *Consequences*

| | |
|---|---|
| C.1 Controlled flight Into Terrain (CFIT)<br>C.2 Collision with terrain/water/obstacle<br>C.3 Mid-air collision<br>C.4 Ground collision with other aircraft<br>C.5 Ground collision with object/obstacle<br>C.6 Loss of control in flight<br>C.7 Fuel exhaustion<br>C.8 Overrun<br>C.9 Undershoot<br>C.10 Structural failure<br>C.11 Post crash fire<br>C.12 Fire/smoke during operation<br>C.13 Emergency evacuation difficulties<br>C.14 Forced landing - land or water<br>C.15 Other cause of fatality | |

D *Unknown*

*Level of confidence* _____ *High* ______ *Medium* ______ *Low* ______*Insufficient Information*

Note: Acts of terrorism and sabotage, test and military-type operations, and fatalities to third parties not caused by the aircraft or its operation are excluded.

In the military context, harmonization of data standards with our NATO allies will inevitably result in more robust safety systems for all concerned. Also, since the occurrences of many types of malfunctions are rare, harmonization will allow data to be shared between nations that operate similar aircraft systems and increase the data pool, thereby decreasing uncertainty inherent in statistics-based analysis schemes.

Completeness of data, whether the entry relates to deviation, malfunction, or wear is nearly as important. The need to report and record every deviation from the norm, even though the vast majority of cases are benign, cannot be over-emphasized. It is also essential for the analyst (or analysis group) to promptly acknowledge receipt of each report and, once the analysis of a report is complete, to communicate the results to the maintenance group. Otherwise, the latter group may lose faith in the system.

## Data Archival and Retrieval

An efficient database storage system has to take into account several factors. Simultaneous access to multiple users may be one requirement. Inclusion of pictures, and documents in the database may be another. There are several ways to store and present data and several types of database management systems (DBMS) have been devised and are commercially available. In choosing the right type of DBMS it is important to consider the capability of a typical user and the purpose underlying the use of the data. For safety analyses purposes, the DBMS should be capable of storing and manipulating complex objects and data types efficiently. The most suitable type and currently available DBMS are the ones known as object-oriented DBMS. Such relational databases allow for computer-aided searches and sorts that are simple to implement , allowing the user to concentrate on deriving the information he or she is seeking rather than focusing on the design of the database extraction tool. On the other hand, if one is willing to invest in more complex search engines, the database may need to be less structured and therefore contain much more information. An explanation of the various types of DBMS can be found in reference [6]. Even object-oriented DBMS have their drawbacks and, thus, the entire subject deserves research attention.

## Analysis Methods

Service difficulty data can be used for a variety of purposes and in a variety of ways. The common thread that runs through all of them, however, is risk mitigation. Obviously, the criticality of the component associated with the data, the number of incidences of failure, the consequences of failure, the method(s) used for analysis, the confidence band inherent in the analysis results, and the statistical character of the occurrence are inextricably related.

Accidents and, to a lesser extent, incidents and malfunctions typically involve a chain of events. The chain may simultaneously involve a design deficiency, a defect induced during the manufacturing process, improper maintenance or other human factors. Some aspects that are frequently involved are given in Table 2.

**TABLE 2: ASPECTS THAT CAN ADVERSELY AFFECT SAFETY**

| | | |
|---|---|---|
| Design | Manufacturing | Maintenance |
| New Technology | Technological Obsolescence | Human Factors |
| Repair | Unwitting Exceedances | Configuration Management |
| Flight Operations | Air Traffic Control | Adverse Environment |
| Software | Training | Records |
| Regulations | Environmental Rules | Unapproved Parts |
| Hazardous Cargo/Stores | | |

It has been argued that, since many factors are involved in causing an incident or accident, the safety management system should be highly centralized. The author would argue in favor of the opposite, mainly because the safety system would be redundant and, hence, more robust. The responsibility for safety should be divided into sub-groups, whose prime responsibilities are related to maintenance or air traffic control or some other factor identified in the table. Each group should be persuaded to believe that they are ultimately responsible for safety and each group should be allowed to devise their own system for monitoring risk. Of course, each such group will be much better versed in their own specialty and might tend to give greater attention to it. On the other hand, it can be argued that they will tend to take less for granted in other specialty areas and therefore subject them to greater scrutiny.

If the aforementioned view is accepted, it would follow that each group will have different data requirements. The latter can be fulfilled with relative ease by customizing data, but which is drawn from the same master data pool.

One example of an extensive and well-disciplined service difficulty reporting and collection system, as has been previously mentioned, is that being maintained by the FAA. Unfortunately, the FAA does not have the means to systematically analyze the data reported, which purportedly is not all-inclusive. Instead, it does so in an ad-hoc manner. That is, it researches the database to seek service difficulties that indicate the pervasiveness of a fault in the aircraft fleet. Such searches are carried out after the problem has been brought to the attention of the authority through other means, such as an incident or an accident. However, the efforts of the FAA are a valuable adjunct to the safety analysis efforts by industry. Moreover, the SDR database is accessible to other users, such as aircraft manufacturers and operators, who, because of their focus tend to be more systematic in the analysis of the data.

## Causal Analysis

Causal analysis of an accident or incident seeks to establish those factors that were judged to be directly responsible in causing the event (primary causal factors) and those that contributed to the event (secondary causal factors) by deconstructing the accident. For these causal factors, a causal chain can usually be established for each accident or incident [7]. The advantage of causal chain analysis is that in the case of multiple causes and multiple accidents or incidents, the common events or elements in the chain can be identified and subjected to greatest attention. Thus, the safety system can concentrate on those common events and maximize its responsiveness and effectiveness in for cutting down-times, and reducing or eliminating accidents. The perceived disadvantage of this approach is that it is reactive rather than proactive. That is, the regulating authority and the industry (or the military operators) seek to eliminate the causal factor after the accident in order to prevent accidents due to the same cause from happening again.

Causal analysis does have an advantage over simulation and technical conjecture in that it is based on factual data rather than models that mimic a hypothetical event or engineering judgement, which relies on the knowledge base and experience of the technical team. Moreover, as has already been mentioned, in today's aviation industry, it is difficult to retain an engineering team that is intimately familiar with the continuous changes in the aircraft design after production begins.

The causal analysis approach, however, also suffers from the disadvantage that the analysis has a good measure of subjectivity, both in regard to the list of factors and their relative contributions. Also, due to the inter-dependencies of the various factors, such as those listed in Table 2, that are frequently encountered, the relative weights ascribed to the various causal factors can vary a great deal, as a function of the analyst. Thus, an intimate knowledge of the aircraft system is a prerequisite for someone engaging in causal analysis. The challenge of managing aircraft safety is identify and focus on truly hazardous conditions, so they can be eliminated before a potential accident becomes a reality.

## Trend Analysis

One simple and effective method is used in the Aviation Safety for Accident Prevention (ASAP) program that is used by the FAA's Rotorcraft Directorate in Ft. Worth, Texas. The program selects components that fail by part numbers. For each part, it reviews the service history for 3, 6, 12 or 24 months periods. Based on the counts of service difficulty reports involving the part number, it predicts trends.

A risk level is assigned to each report. ASAP has the ability to quickly research whether an accident had a service difficulty history. For example, responding to a fatal accident involving the tail rotor driveshaft, the analyst was able to track part numbers, and identify five service difficulty reports that had found the part to have been worn beyond limits, and contained cracks or corrosion. Two of the reports described the results of inspection to be a sheared tail rotor driveshaft. Based on the accident and the supporting trend indicated by the service history, the Authority issued an Airworthiness Directive (AD). A year after the issuance of the AD there were no more service difficulty reports, citing that particular part was reported. But, more importantly, the incidence of sheared rotor drive shafts has been drastically reduced. However, ASAP has one drawback: usage of ASAP is not yet proactive in that the analyst must be prompted by an event, such as an accident or incident to conduct trend analysis on a given part or component.

## Monitoring of Safety Through Performance Indicators

The FAA's Flight Standards Service has developed a heuristic-based system called Safety Performance Analysis System (SPAS), primarily for the benefit of their corps of safety inspectors. They started building the system by getting teams of highly experienced and proficient inspectors together, with each inspector identifying the parameters that he or she uses during surveillance of an operator or a repair station facility. Each team discussed each of the identified parameters and developed a consensus about the relative importance of the parameters that must be scrutinized. Next, the parameters were weighted according to their perceived importance and aggregated into groups, with each group being termed as an "indicator." The advantage of a system that is based on indicators is that pools the knowledge and experience of the "gray beards" or the more experienced inspectors in the regulating Authority for use by the younger, less-experienced inspectors. Hence, it focuses attention on what is a warning rather than on events that are merely "noises." The disadvantage is that a rational derivation of threshold values, which signal caution or even danger, is not possible.

A variation of the idea of performance indicators as measures of safety is proposed by the author. It is based on "wiring diagrams" of sub-systems being used in conjunction with the concept of indicators. In the pristine condition, every cell in the wiring diagram would be colored white. When a failure of a certain part occurs, the analyst assesses the criticality of the part to flight safety and assigns a hue to that part (cell) in the wiring diagram. A deeper hue or color would signify that the part has a relatively high criticality. The wiring diagram is constantly updated by adding more color to the particular part to reflect arrival of new service difficulty reports. Two events will attract the attention of the analyst. The first is the depth of the hue of a certain cell and the second is the contiguity of cells (the ones that are sequentially tied or represent the redundant feature), in terms of their function, that are hued. The idea is based on the recognition of the fact that in both cases the risk of sub-system failure is increasing, and that the wiring diagram pictorially represents the rise. In fact, it would be relatively easy to convert the logic into a computer code that automatically raises a flag in either case, which cannot escape the attention of the analyst. Also, different colored flags may be set up to indicate the level of alert. The scheme will also need to take into account replacement or re-design of the part, or the sub-assembly itself. That is also easily done by washing out the color in the particular cell representing the part or in the block of cells if the sub-assembly has been redesigned or refurbished

## Probabilistic Risk Analysis

Several probabilistic approaches to safety have been proposed [8}. However, such approaches are not looked upon with enthusiasm because no one wants to look upon safety management in a manner that resembles a game of chance. However, there are at least two major advantages of a probabilistic approach. First, it takes into account the variability in the data as well as the trends in the number of occurrences. It also provides for considering the relationship between seemingly unrelated occurrences. The analyst must examine the estimated probability of an accident, given a high probability of the occurrence of service events, and determine if intervention is required. A unique advantage of the probabilistic approach over a deterministic approach is that it enables the Authority or the Safety Office in the military to focus on the most likely causes of hypothetical, future accidents, and prevent them. By far the most important advantage is that it enables the Authority, and the operators, to get ahead of the power curve - that is, to correct the condition before the first accident occurs.

## Concluding Remarks

As new technology is inducted, aircraft systems will inevitably become more complex. New technology generally means better performance and lower costs but there might be safety-related challenges as well. Also, increased usage and operating missions beyond what was envisaged in the design stage will magnify the accident rate as well as the fatalities, injuries, or losses of high-value assets. Safety systems will need to be more sophisticated and better methods of analysis will need to be employed. Authorities, and in the case of the military - themselves, will need to focus more on preventing accidents due to service related events rather than using service data to confirm the analysis of accidents that have already happened.

Concomitantly, more extensive data require-ments and data archival systems will need to be engineered. Thus, the cost of maintaining a high level of safety is bound to rise but the cost due to not having an effective system will be many times greater. Safety of highly engineered systems, like aircraft, has a high price tag but the alternative will prove to be much, much more expensive.

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# TUTORIAL ON REPAIR SOFTWARE

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## 1.0 INTRODUCTION

Throughout the world military and commercial aircraft fleet are being used beyond their original design life. This is primarily due to the reduction in the budget for procurement of new systems and ever increasing cost of acquiring new aircraft. This has resulted in paying more attention to enhancing life of aircraft structures and at the same time maintaining the safety of flight. Improved life enhancement techniques and repair concept are being developed to keep maintenance cost low, reduce down time of aircraft for repairs, reduce inspection requirements without jeopardizing the safety of aircraft structures.

To reduce the down time of aircraft for repairs and perform more reliable durability and damage tolerance analyses, a number of software programs have been developed. These software programs are user friendly and a user does not have to be an expert in the repair technology or durability and damage tolerance analyses. For most of these programs basic knowledge of stress analyses, fatigue and fracture mechanics is required. This tutorial discusses some of these programs, and steps involved in the analyses of repairs to assure safety of flight.

## 2.0 SOFTWARE PROGRAMS FOR REPAIR DESIGN, AND DURABILITY AND DAMAGE TOLERANCE ANALYSES

A number of software programs have been developed for designing repairs for aircraft structures and performing durability and damage tolerance analyses. These programs are operational on a personal computer (PC). Some of these programs are briefly described here.

### 1) AFGROW

This code has been developed by US Air Force Wright Patterson Air Force Base (WPAFB), Ohio, for durability and damage tolerance analyses of aircraft structures under constant amplitude and spectrum loading. Code has capability to design composite patch repairs for metallic structures. Crack growth life predictions can be made in an aggressive environment accounting for the corrosive effects on crack growth. The code is user friendly with an excellent graphical user interface. The code has a good database of material properties needed for damage tolerance analysis. A user has an option to input own material properties. The code has a built in library of stress intensity factors for a number of crack configurations and structural geometries. The user has an option to input own stress intensity factors. For crack growth predictions under spectrum loading, the user has to input loads spectrum. The code takes spectrum in a certain format. Majority of airframe manufacturers have own ways of generating spectrum. Hence, a translator is required to convert input spectrum in the format used by the code. The translator varies with the input spectrum format. Translators for a number of spectra formats have been included in the code. If a user's spectrum input is not in a format used in AFGROW code, Mr. Jim Harter at WPAFB, Ohio, may be contacted for assistance in developing a translator.

For crack growth predictions under spectrum loading, a number of retardation models have been included in the code. Retardation models included are-

a) Willenborg
b) Wheeler
c) Crack Closure.

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The retardation parameters needed for these retardation models are included in the code for some of the materials. A user has option to input own retardation parameters.

The code has capability for predicting fatigue life under spectrum loading. Strain-life approach has been used for fatigue life predictions. The strain life approach requires cyclic stress-strain and strain-life data for the structural material of interest. The cyclic stress-strain and strain-life data for some materials have been included in the code.

This code has capability to design composite patch repairs. A knowledge-based system to design repairs has been developed in the code. The code recommends the most suitable material for composite patch design based on the following considerations-

a) Thickness to be repaired.
b) Loads spectrum experienced by the aircraft component.
c) Stress level in the spectrum

Repair material choices available are- boron/epoxy, graphite/epoxy, and GLARE. The properties of these materials are included in the code. The code recommends ply orientations and thickness for the composite patch. The code uses damage tolerance approach for designing repair patches. The design of repair patches is based on using ductile adhesive FM-73 for bonding process.

**2) NASGRO**

This user-friendly program has been developed by NASA Johnson Space Center and is available in public domain. The program is primarily for damage tolerance analyses of structures. The program has an excellent database for material properties varying from aluminum to steel, plate, sheet, forging, etc.

The program uses boundary element technique to compute stress intensity factors. Crack growth predictions can be made under constant amplitude and spectrum loading. The program does not have capability to design repairs. However, it is a very useful tool for the damage tolerance analysis of structures and repairs.

**3) RAPID**

This program has been developed under FAA sponsorship with support from US air Force and is primarily for mechanically fastened repairs of transport aircraft (Reference 1). The program has an excellent Graphical User Interface (GUI). The program is available in public domain. The program is not suitable for damage tolerance analysis of aircraft structures. The program has capability to perform repair analysis under constant amplitude as well as spectrum loading.

The program has an excellent database for material and fastener properties. The program is suitable for designing three different types of repairs-

1) One external and one internal doubler (Figure 1).
2) Two external doublers (Figure 2).
3) One external doubler (Figure 3)

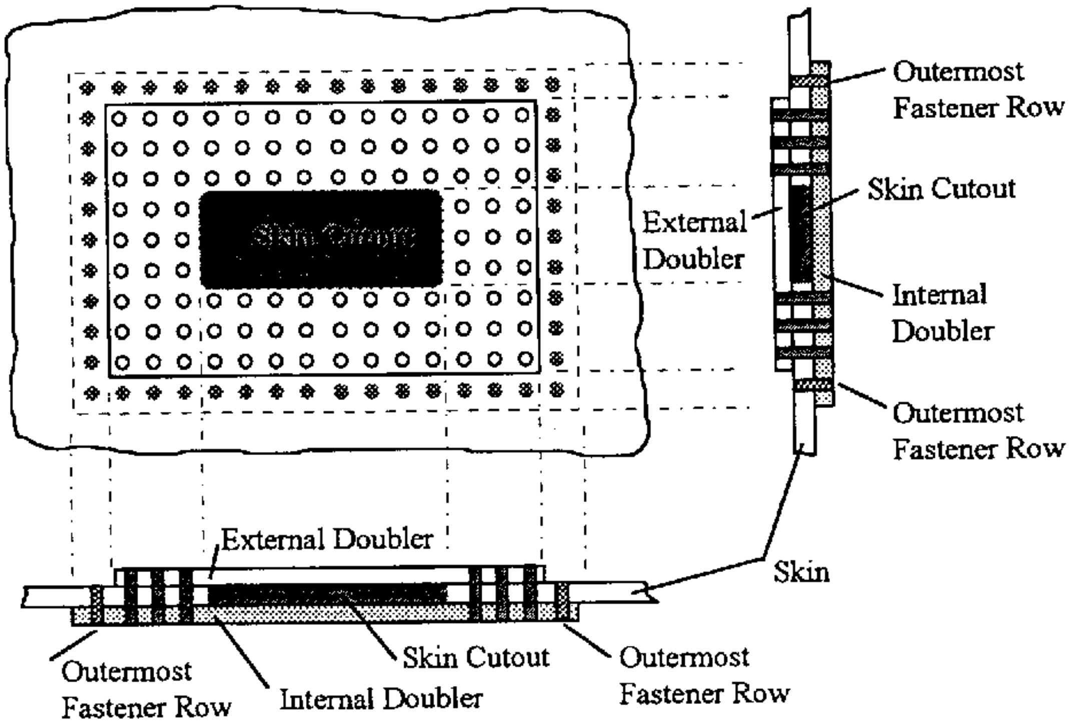



**Figure 1. RAPID Software- One Internal and One External Doubler Repair**

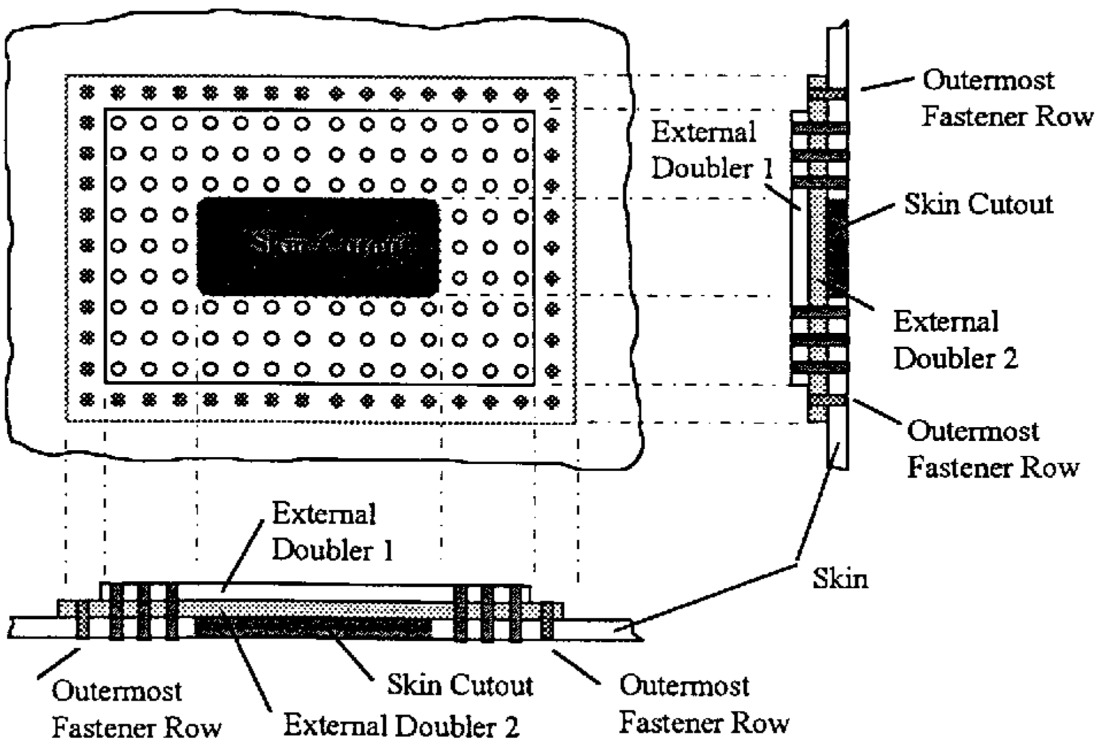



**Figure 2. RAPID Software- Two External Doublers Repair**

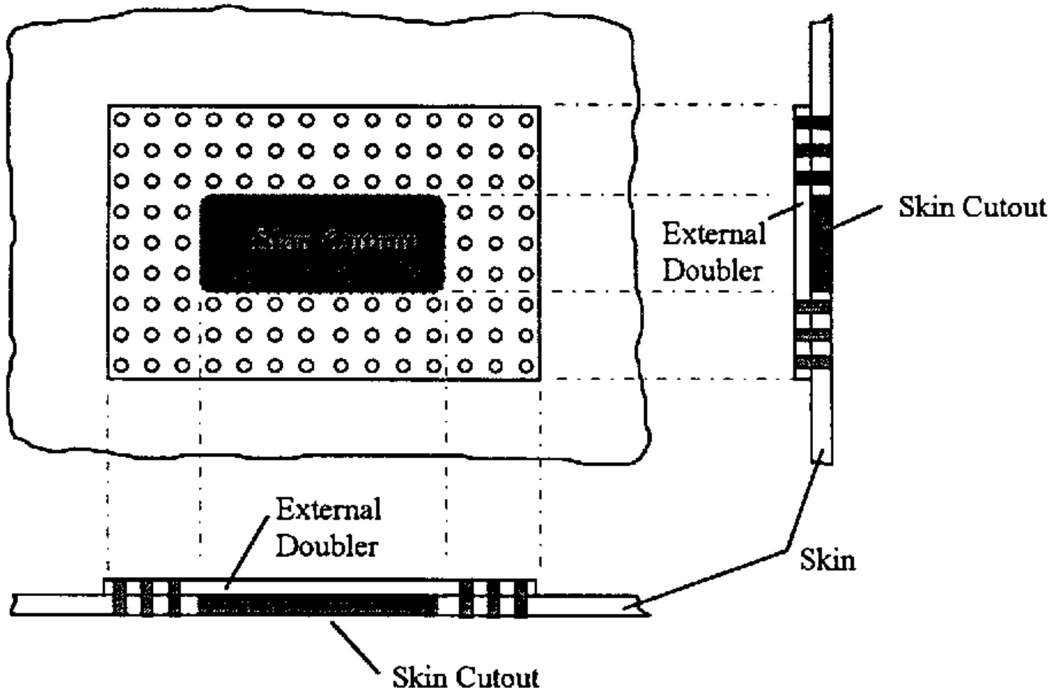



**Figure 3. RAPID Software- One External Doubler Repair**

**4) RAPIDC**

RAPIDC is a derivative of RAPID program developed under FAA sponsorship. The program has been developed primarily for mechanically fastened repairs of commuter aircraft. The program is still being beta tested. The program is available in public domain.

**5) CalcuRep**

This code was developed by Dr. Rob Fredell during his stay at US Air Force Academy in Colorado (Reference 2). This code is for designing bonded repairs for fuselage type of structures subjected to internal pressure loads. The code is available in public domain and is user friendly. The code is primarily useful for designing repairs using Glare material.

**6) FRANC2D**

This is a finite element code and can be used for crack growth analysis of metallic structures (Reference 3) under constant amplitude loading. The code is available in public domain. Finite element analysis of the structural configuration with crack is carried out and stresses intensity factors determined. The code uses these stress intensity factors for crack growth predictions. A metallic structure with composite patch can be modeled with the code as 3 layers (metal, adhesive and composite patch) and stress intensity factors obtained. These stress intensity factors are used to make crack growth predictions in repaired structure under constant amplitude loading.

## 3.0 SAMPLE PROBLEMS

### 3.1 Fuselage Frame Repair

Standard repairs are generally given in repair manuals. However, in many cases in-service inspections show damage that is not covered by standard repair manuals and a special repair has to be designed. For such cases detailed static and damage tolerance analyses have to be carried out. An example of cracked frame in a transport aircraft (Figure 4) is shown in Figure 5. The flange and the web of the frame are cracked as shown in Figure 6a. Standard repair manuals generally do not cover a repair for the damage shown in Figure 5. The cross-sections of the flange and web repairs are shown in Figure 6b. The details of the frame repair are shown in Figure 7.

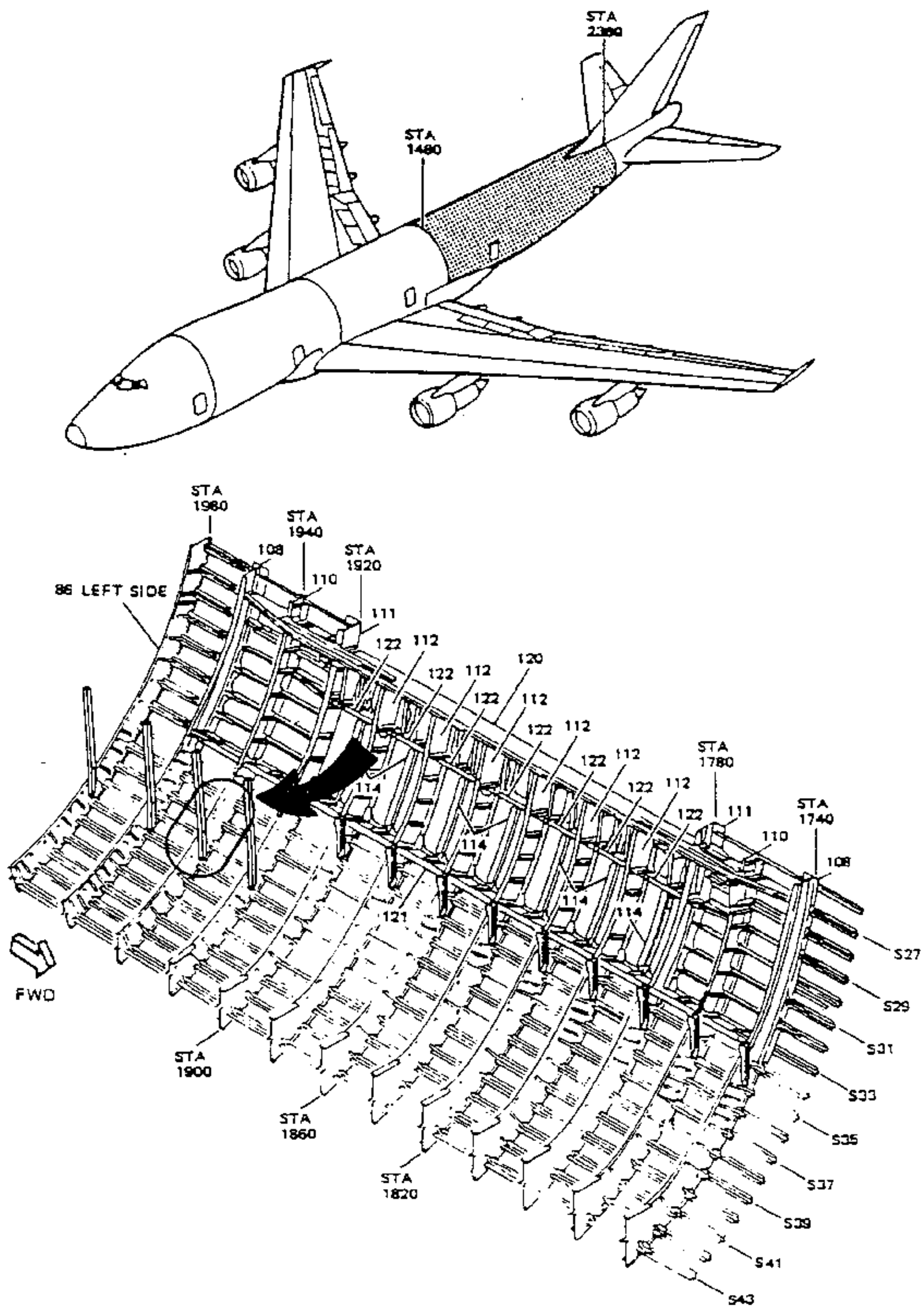



**Figure 4. Fuselage Frame Cracking Location in Transport Aircraft**

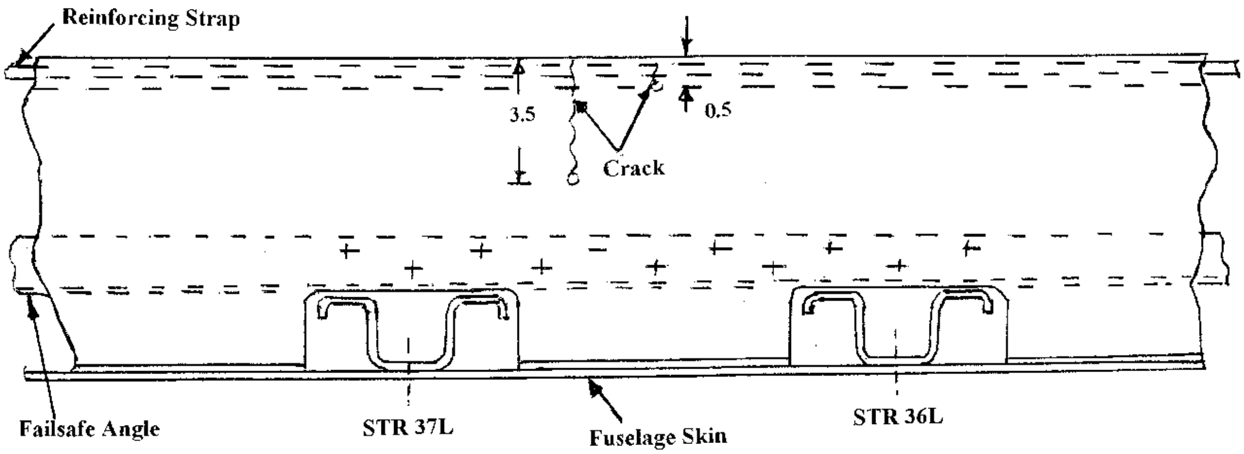
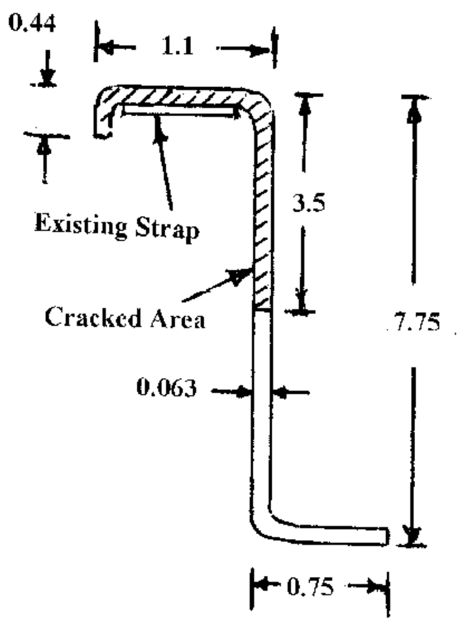



**Figure 5. Cracked fuselage Frame**



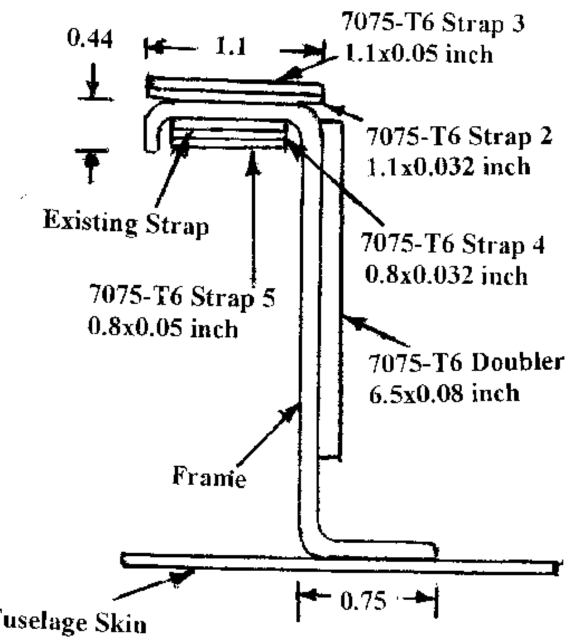




**Figure 6a. Section Showing Cracked Frame** **Figure 6b. Section Showing Flange and Web Repairs**

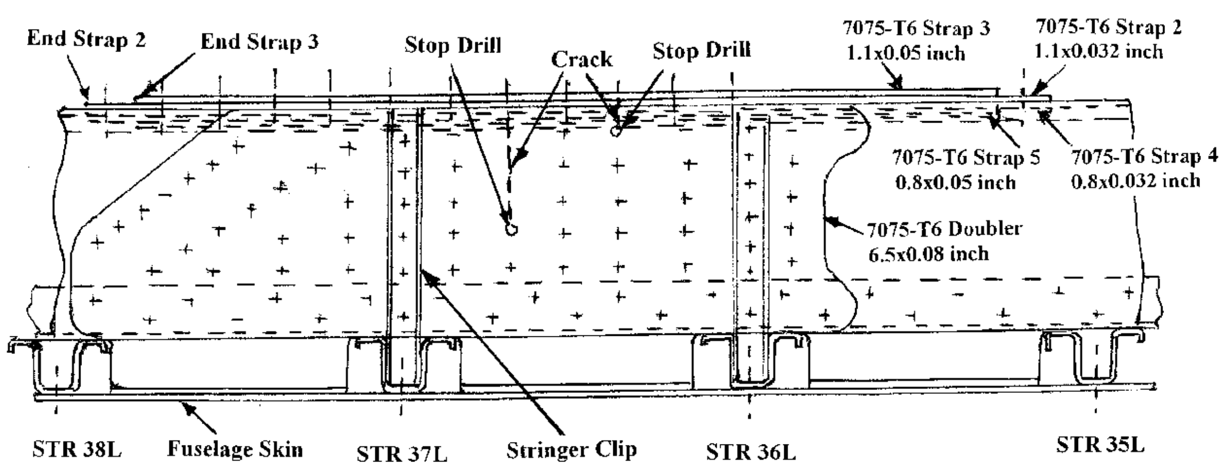



**Figure 7. Details of Frame Repair**

## STATIC STRENGTH DESIGN

Ultimate strength of flange material 7075-T6 aluminum is assumed to be 75 ksi (517.1 MPa).

### Load Capacity Lost Due to Cracking

1. Flange area lost due to cracking = (0.44+1.1) x 0.063 = 0.097 $in^2$ (62.6 $mm^2$).
Load capacity lost = 0.097 x 75 = 7.275 kips or 7,275 Lb.
2. Web area lost = 3.5 x 0.063 = 0.221 $in^2$ (142.6 $mm^2$)
Load capacity lost = 0.221 x 75 = 16.575 kips or 16,575 Lb.
Total Area Lost = 0.097+0.221 = 0.318 $in^2$ (205.2 $mm^2$)
Total Load Capacity Lost = 7.275 + 16.575 = 23.850 kips or 23,850 Lb.

### Repair Analysis

Repair area required in flange = 1.25 x Area lost = 1.25 x 0.097 = 0.121 $in^2$ (78.1 $mm^2$).
Repair area added in flange (Figure 6b) = (0.032 + 0.05) x 1.1 + (0.032 + 0.05) x 0.8
= 0.09 + 0.067 = 0.157 $in^2$ (101.3 $mm^2$).

Repair area required in web = 1.25 x 0.221 = 0.276 $in^2$ (178.1 $mm^2$).
Repair area added in web = 0.08 x 3.5 = 0.28 $in^2$ (180.6 $mm^2$).
Total area added in flange and web = 0.157 + 0.28 = 0.437 $in^2$ (281.9 $mm^2$).

Margin of Safety = (Area added / Area lost) – 1.
= (0.437/0.318) - 1. = 1.37 –1. = 0.37

### Fastener Requirements

Using HL18-6 HI-Lok in 0.063, 7075-T6 sheet. Allowable loads are given by-
$P_S$ (Shear) = 2,694 Lb.
$P_B$ (Bearing) = 1,197 Lb.
Number of Fasteners Required in Flange = 7,275 / 1,197 = 6 .07.
Use 6 Fasteners.
Number of Fasteners Required in Web = 16,575 /1,197 = 13.8
Use 15 Fasteners.
Total Fastener Load Capacity = 21 x 1,197 = 25,137 Lb.

## 3.2Composite Reinforcement of T-38 Lower Skin in Machined Pockets

Lower wing skin pockets in T-38 aircraft between the 39% and 44% spars and 33% and 39% spars at Wing Station (WS) 78 have shown a propensity for crack initiation and propagation during service. The cracks have initiated at the pocket radius in the inner moldline of the wing skin. This cracking has been occurring primarily under Lead-in-Fighter (LIF) spectrum loading. These areas are ideal for composite reinforcement to reduce stress levels and enhance fatigue life. As there is no access for bonding reinforcement on the inner moldline, a one sided reinforcement bonded onto the outer moldline of the wing skin was selected (Reference 4). Composite reinforcement bonded to the wing is shown in Figure 8.

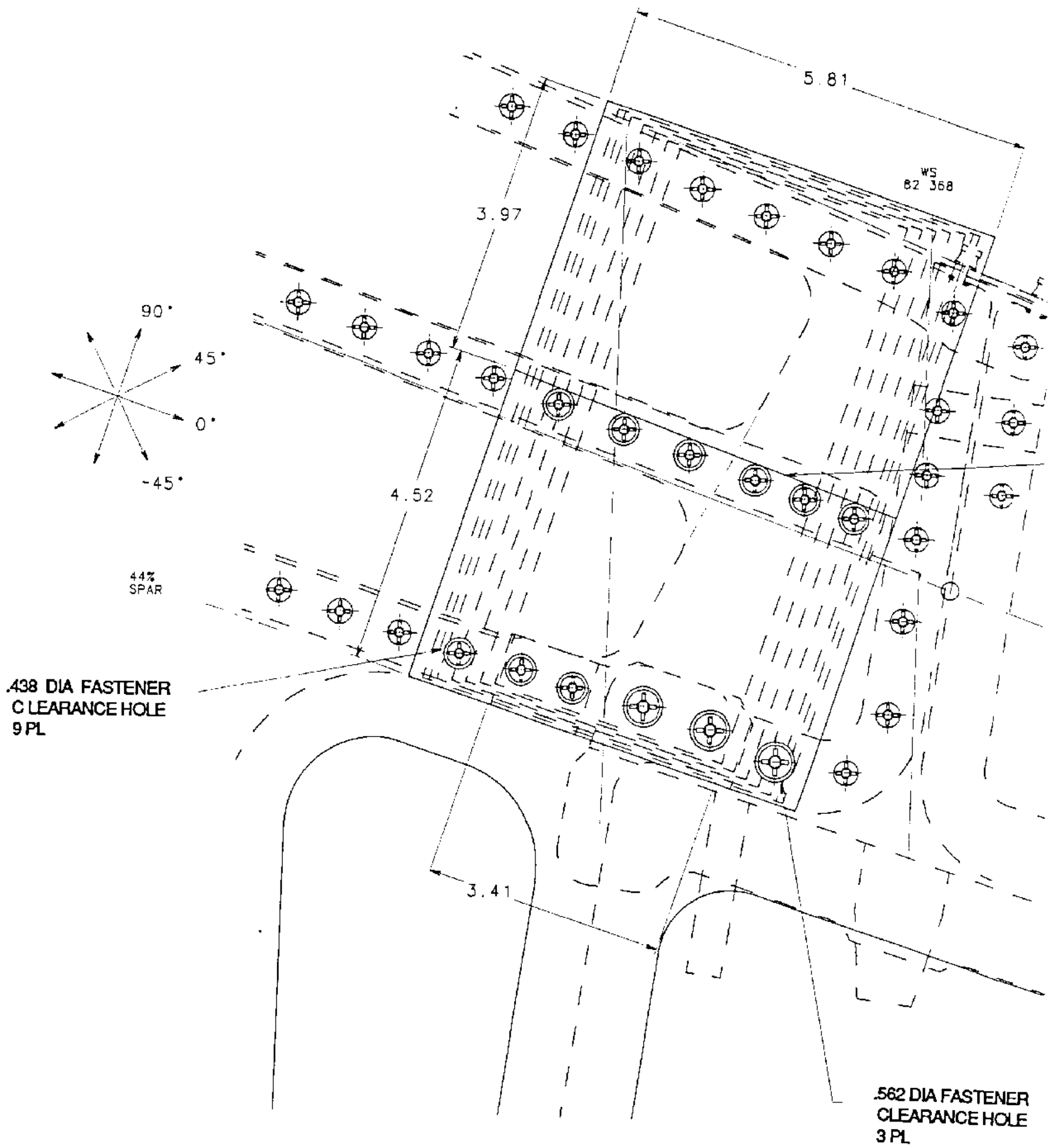



**Figure 8. Location of Composite Reinforcement on Lower Wing skin**

A detailed finite element analysis of the local area with and without composite reinforcement was carried out (Reference 4). The finite element model of the structure is shown in Figure 9. Typical output of the outer moldline stresses is shown in Figure 10. Using NASTRAN stresses, a detailed damage tolerance analysis of the pocket area was carried out using AFGROW computer code.

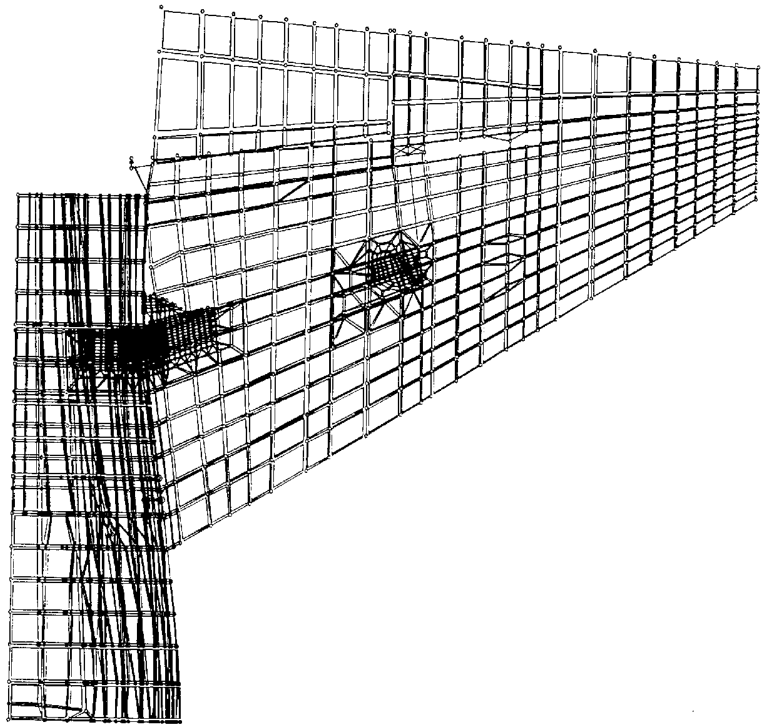


**Figure 9. Finite Element Model**

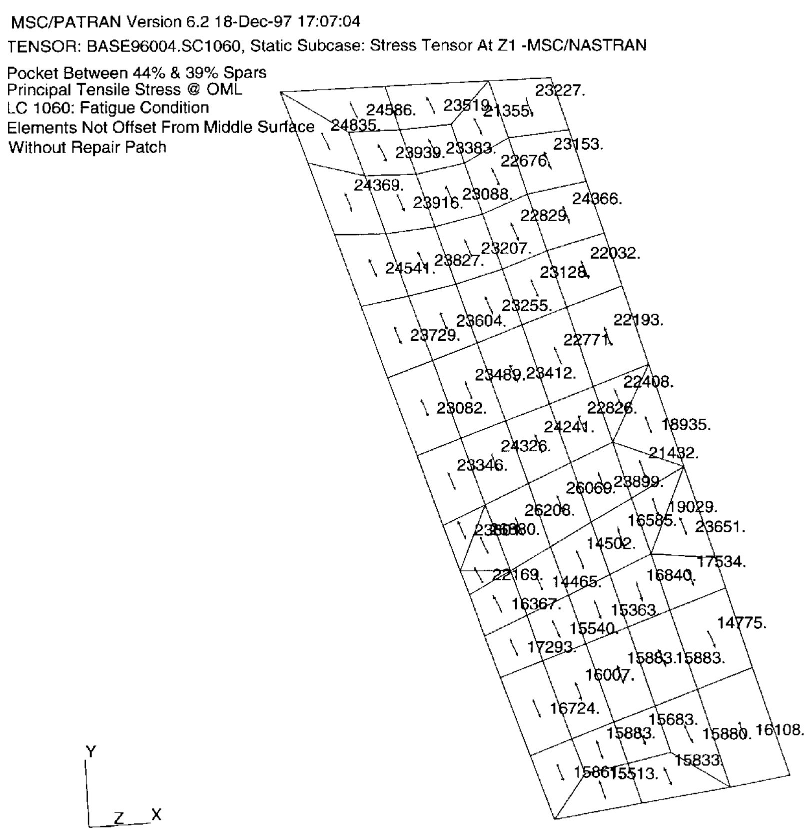



**Figure 10. Typical NASTRAN Output Showing Element Stresses**

Structural testing of the T-38 wing at Wright Patterson Air Force Base has been carried out under LIF spectrum loading (Reference 4). Prior to the fatigue testing of the wing, strain gages were applied to the lower wing skin pocket area. The strain surveys of the pocket areas were performed prior and subsequent to the bonding of the reinforcement.

The location of the strain gages on the wing is shown in Figure 11. The strain survey was carried out for the two most critical load conditions (namely S0985 and 10556) to 50% of limit load. The stress analysis of the pocket area was performed using coarse mesh and fine mesh finite element models. The comparison of test strains and analytical strains for the pocket between 39% and 44% spars obtained from both models is shown in Figure 12. The figure shows that for condition S0985, both finite element models show a good correlation with the test results. However, for condition 10556, only the fine mesh results correlate well. The coarse mesh results are shown to be slightly higher.

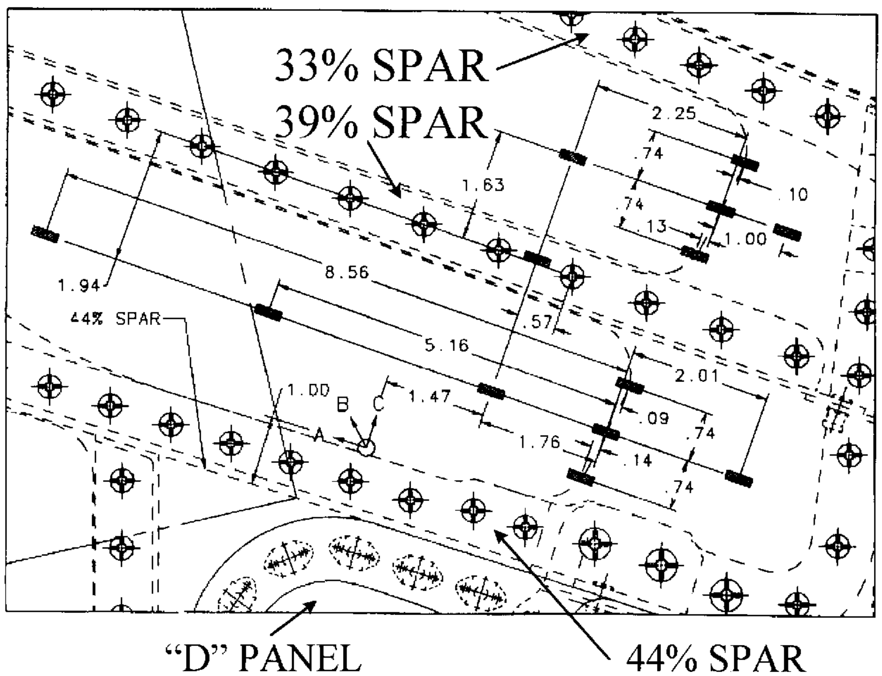



**Figure 11. Strain Gage Location in the pocket Area of T-38 Wing Skin**

A good comparison between analysis and full scale test indicates that finite element analysis can reliably predict structural behavior with proper modeling techniques and applied boundary conditions. The stresses from finite element analysis along with appropriate loads spectrum can be used in any of the damage tolerance codes to make crack growth and residual strength predictions.

## 4.0 CONCLUDING REMARKS

A number of codes are available to perform damage tolerance analysis and design repairs for aging aircraft fleet. Most of these codes are available in public domain at no cost to the users. These codes have a good user interface and are user friendly. The operation of these codes does not require expert knowledge. With little training and some knowledge of fracture mechanics, these codes can be very effectively used to make life predictions of aircraft structures. These codes can assist in selecting right life enhancement techniques, designing repairs, and identifying inspection requirements. The application of these codes will reduce down time of an aircraft for repairs.

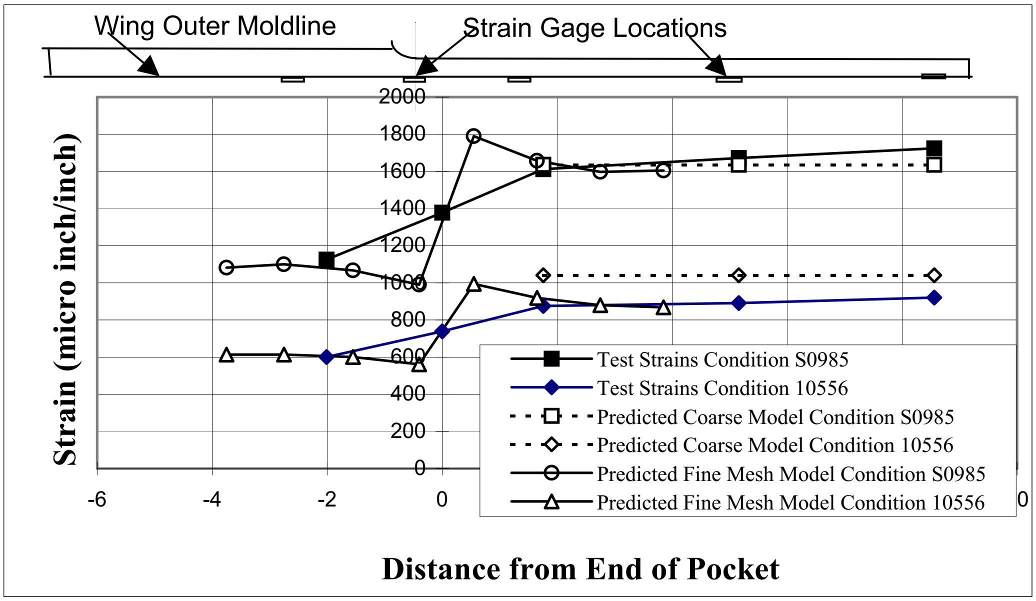



**Figure 12. Comparison of Observed and Predicted Strains in Pocket Between 39% and 44% Spars (No Composite Reinforcement)**

## 5.0 REFERENCES


1. Repair Assessment Procedures and Integrated Design (RAPID) User's Manual, 1996.
2. CalcuRep for Windows- User's Manual, Prepared by United States Air Force Academy, may 1995.
3. FRANC2D: A Two-Dimensional Crack Propagation Simulator- User's Manual, NASA Contractor Report 4572, march 1994.
4. Helbling J, Grover R and Ratwani M.M "Analysis and Structural Test of Composite Reinforcement to Extend the Life of T-38 Lower Wing Skin", Proceedings Aircraft Structural Integrity Conference, San Antonio, 1998.

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# INSPECTION TECHNOLOGIES

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## 1.0 INTRODUCTION

Regular maintenance of airframe is an important aspect of assuring flight safety of aircraft structures. One technology area, which plays an important role in proper maintenance and assuring the flight safety of aircraft, is the inspection at regular intervals. Reliable visual and nondestructive inspection (NDI) methods are needed to assure the airworthiness of these aircraft and at the same time keeping maintenance costs to a minimum. Commercial aircraft maintenance programs are shown in Figure 1. For military aircraft the inspection requirements are generally defined by Integrated Logistic Support (ILS) organization for non-critical components. For critical components, the inspections are defined by damage tolerance analysis.

1. **Corrosion Inspection Program**

   Level 1- Local corrosion that can be removed within allowable limits.
   Level 2- Local corrosion that exceeds allowable limit. This requires repair or partial/complete replacement.
   Level 3- Potential urgent airworthiness concern requiring expeditious action.

**Note- Airline's existing maintenance programs must control all primary structures to level 1 or better.**

2. **Periodic Maintenance Inspections**

   A Check- Visual inspection of interior and exterior every 65 to 75 hours.
   B Check- Access panels removed for inspection and engine servicing every 30 days.
   C Check- Heavy structural and maintenance check after every 5,500 flying hours.
   D Check- Interior stripped to fuselage walls every 20,000 to 25,000 flying hours.

**Figure 1. Commercial Aircraft Maintenance Programs**

For in-service military aircraft, the inspection requirements may be defined by the usage (e.g lead-in-fighter, dissimilar air combat, air training command, etc.). Using the stress analysis and loads data, it is possible to predict the life of a structural component with durability and damage tolerance analyses techniques. From the crack growth analysis of a critical area of a structural component under actual spectrum, experienced by a structural component, it is possible to identify initial inspection and subsequent inspection requirements as shown in Figure 2. The crack growth curve for a critical location is obtained from assumed initial flaw $a_0$, based on damage tolerance requirements, to critical size $a_c$ at which the flaw grows to be catastrophic at $N_f$ flight hours. If the inspection capability of the Non-Destructive Inspection (NDI) equipment to be used in field or depot is $a_i$ , then the cycles to grow the crack from $a_0$ to $a_i$ are determined to be $N_i$. The initial inspection requirement is given by $N_i/2$ and subsequent inspection requirements are given by $(N_f - N_i)/2$.

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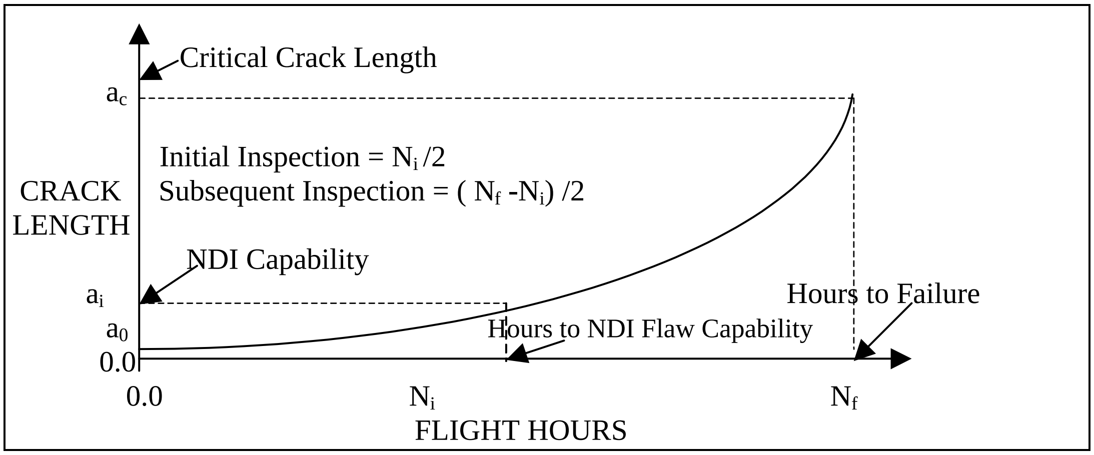
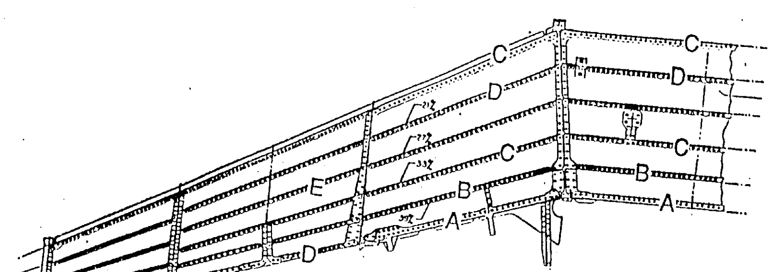



**Figure 2. Initial and Subsequent Inspection Requirements from Crack Growth Life**

The procedure outlined in Figure 2 is used to zone an aircraft structure for inspections depending on the severity of loads and structural details such as thickness, presence of substructure, fastener diameter and type, etc. Typical zoning of a wing structure for Air Training Command (ATC) usage is shown in Figure 3 (Reference 1). The wing in the figure has been divided in 5 zones, namely A, B, C, D, and E. The fasteners in each zone have different inspection requirements depending on the structural details and stress levels. The fasteners in zones D and E are in an area where the stresses are rather small and crack growth life is very large. The zoning and inspection requirements depend on the usage of an aircraft, as the load spectrum will change with the usage. For usage other than ATC the inspection requirements will be different from those shown in Figure 3, however, the inspection zones may still be the same. Analytical techniques provide tools to define inspection requirements based on usage and structural details to reduce inspection cost.



INSPECTION INTERVALS

| ZONE | ATC |
|---|---|
| A | 2450/1200 |
| B | 8500/5500 |
| C | 11500/6000 |
| D | >20000/>12500 |
| E | >20000/>12500 |

**Figure 3. Zoning of Military Aircraft Structure for Inspection**

This paper discusses currently available techniques for detecting damage in structures and their limitations. Inspection of cracks in substructure and hidden corrosion has always presented a nightmare for NDI engineers. Some recent advances made in the NDI technology to solve these problems are discussed.

## 2.0 COMPARISON OF NONDESTRUCTIVE INSPECTION (NDI) METHODS

A number of visual and nondestructive inspection methods are available for inspection. However, their application to detect flaws depends on the type of structure, access, desired degree of accuracy, and inspection time. The comparison of conventional NDI methods is shown in Figure 4.

| NDI Method | Ultrasonic | Eddy Current | Radiography | Penetrants | Magnetic Particle |
|---|---|---|---|---|---|
| Flaw Type | All | Cracks, Corrosion | All Except Small Cracks | All | All |
| Sub-surface | All | Shallow | All | Surface only | Shallow |
| Area of Scan | Small | Small | Large | Large | Medium |
| Flaw Sizing | Fair | Poor | Good | Very Good | Good |
| Test Time | Slow | Slow | Very Slow | Varies | Fast |

**Figure 4. Comparison of NDI Methods**

The advantages and disadvantages of various NDI methods (References 2-3) are shown in Figure 5 along with their applications. Some of these techniques are discussed in the following paragraphs.

| NDI Method | Detection Application | Advantages | Disadvantages |
|---|---|---|---|
| Visual | Large Surface Defects or Damage in all Materials | Simple to use | Reliability depends on experience of user |
| Optical | Surface defects/structural damage in all materials | Rapid large area inspection Good for bonded and cored structures | Accessibility required for direct visibility |
| Penetrant | Surface cracks in metals | Simple to use, accurate, fast, easy to interpret | Surface defects only, access required, defect may be covered |
| High Frequency Eddy Current | Surface defects, cracks, intergranular corrosion, pits, heat treat | Useful for detecting cracks at holes not detectable by visual or penetrant, fast, sensitive, portable | Trained operators, special probes for each application, reference standards required |
| Low Frequency Eddy Current | Subsurface defects, corrosion thinning | Useful for detecting cracks under fasteners or substructure without disassembly | Trained operator, time consuming, special probe for each application |
| Sonic | Delaminations, debonds, voids, and crushed core in composites, honeycombs | One side access, does not require paint removal or surface preparation | Difficult to interpret results, loses sensitivity with increasing thickness |
| X-Ray | Internal flaws and defects, corrosion, inclusions and thickness variations | Eliminates disassembly requirements, permanent record, high sensitivity | Radiation hazard, trained operators, crack plane must be parallel to x-ray beam, special equipment |
| Magnetic Particle | Surface and sub-surface defects in ferromagnetic materials | Simple, portable, easy to use, fast | Trained operator, parts to be cleaned before and demagnetized after check Magnetic flux must be normal to defect plane |
| Ultrasonic | Surface and sub-surface defects, cracks, disbonds in metals and composites | Fast, easy to operate, accurate, portable | Trained operator, test standards required, electrical source needed |

**Figure 5. Relative Advantages and Disadvantages of NDI Techniques**

## 3.0 PROBABILITY OF DETECTION (POD)

Probability of detection (POD) is a statistically based quantitative measure of inspection capability. The POD is different for different inspection equipment and even for the same NDI equipment is affected by a number of factors such as: material properties, structural details, defect shape, inspection conditions, etc. Another parameter generally associated with POD is the confidence level with which a flaw can be detected. A 95% confidence level is considered acceptable for flaw detection. An NDI equipment capability is generally designated as 90% probability of detecting a flaw with 95% confidence level. The POD of various NDI equipment for through the thickness damage (Reference 4) is shown in Figure 6. Figure 7 shows POD for sub-surface and internal defects. These figures indicate that the probability of detection varies significantly with each NDI equipment.

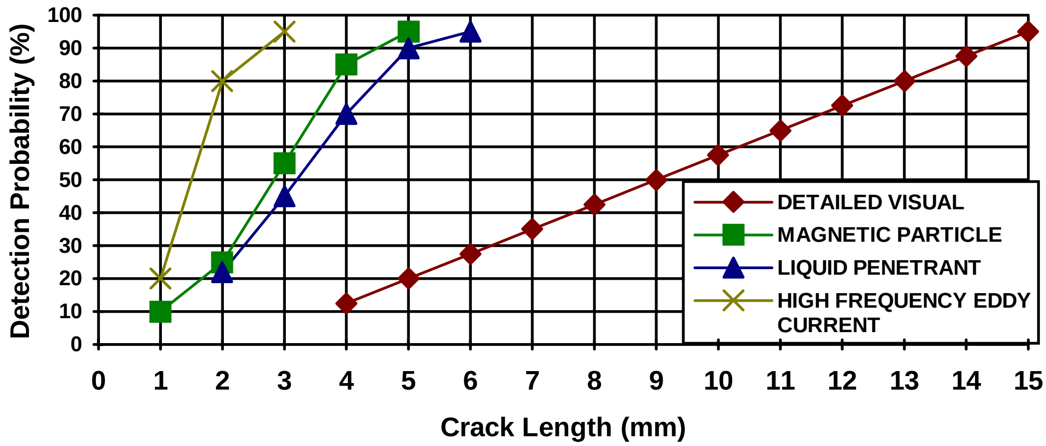



**Figure 6. Probability of Detection for Through the Thickness Defects**

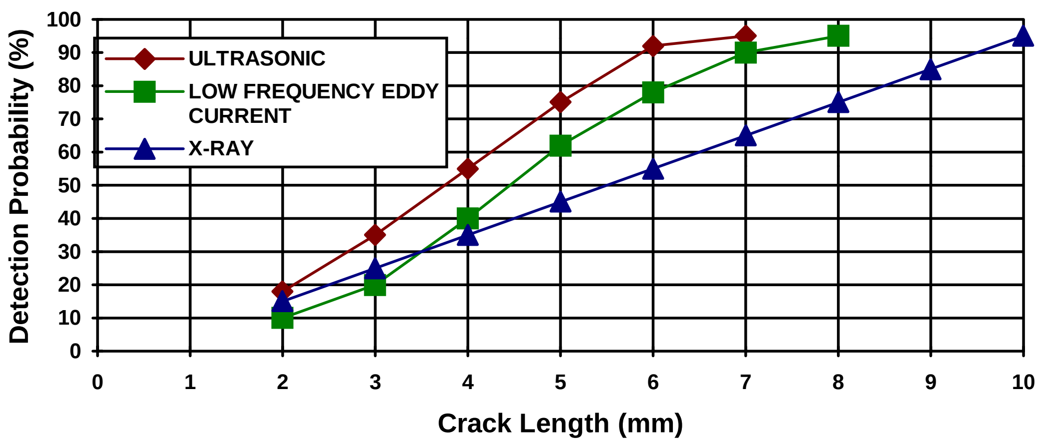



**Figure 7. Probability of Detection for Sub-Surface and Internal Defects**

## 4.0 VISUAL INSPECTION

Visual inspection is a sensing mechanism in which eye alone or in conjunction with other aids is used to judge the condition of a component being inspected. Visual inspection is an integral part of airplane maintenance and is considered as a component of NDI. Over 80 % of the inspections on large transport aircraft are visual inspections. On small aircraft the percentage of visual inspection is even higher. Typical defects found by visual inspection are cracks, corrosion and disbonding. Detection of disbonding due to corrosion is generally difficult; however, disbonding may be accompanied by local bulging due to corrosion or entrapped moisture and may be easily detectable.

Visual inspection is perhaps the simplest, most economical and most efficient method of assessing the condition of an aircraft. A large number of defects are generally found by visual inspection and the operators depend highly on the visual inspections to ensure the airworthiness of an aircraft. Hence, visual inspection plays an important role in the safe operation of an aircraft. The details of visual inspection are given in References 5-6.

### 4.1 Factors Affecting Visual Inspection

The manufacturer or regulatory authorities in the maintenance or overhaul manuals generally specify visual inspection procedures. A number of factors affect the results of visual inspection. Some of the important factors are:

1) Qualifications and Training of Inspection Personnel- Inspection should be done by qualified personnel or under the supervision of qualified personnel. These personnel should have knowledge of the structural details being inspected, types of defects which are commonly found and the causes of these defects.
2) Inspection Area Access- Proper access to the inspection area is an important factor in the reliability of visual inspection. An easy access to the component to be inspected will assist in the decision making process and ability to interpret results.
3) Lighting**-** Proper light without glare is essential for a quality visual inspection. Poor lighting can mask the defects and cause fatigue to the inspectors there by affecting their judgment.
4) Pre-cleaning- The part to be inspected should be free from dirt, contamination, and any foreign material that will obscure the detection of defects.
5) Working Environment- A proper working environment is necessary for the visual inspectors. Presence of excessive temperature, wind, rain or any other adverse condition can influence the interpretation capability of operators and increase the potential for errors.

### 4.2 Levels of Visual Inspection

Visual inspection is divided in four categories (Reference 5), namely: 1) Walkaround Inspection, 2) General Visual Inspection, 3) Detailed Inspection, and 4) Special Detailed Inspection.

**Walkaround Inspection-**The purpose of a walk around inspection is to serve as a quick check to detect any obvious discrepancies that would affect the performance of an aircraft. Most maintenance manuals specify a walkaround inspection on a periodic basis. Flight or maintenance personnel may do this inspection from the ground. This inspection includes: fuselage, left and right wings, leading edges, control surfaces, propeller or fan blades, exhaust areas, pylons and gear well. The walkaround is done twice to make sure that nothing was missed the first time. The inspector looks for any major dents in the skin, missing fasteners, corrosion, leaks etc.

**General Inspection-**A general inspection of an exterior is carried out with open hatches and openings of interior to detect obvious damage. A general inspection is carried out when a problem is suspected or routinely when panels are open for normal inspection. The tools required for this inspection include: flashlight, mirror, droplight, rolling stool, ladder, stand and tools for removing panels.

**Detailed Inspection-** A detailed inspection is required when a specific problem is suspected or general inspection has identified some problems. This inspection is an intensive examination of a specific area, system, or assembly to detect any damage, failure or discrepancy. Surface preparation and special access may be required for this type of inspection along with special aids in addition to the tools required for general inspection.

**Special Detailed Inspection-** A special detailed inspection is a thorough examination of a specific component, installation or assembly to detect damage, failure or any discrepancy. Disassembly of sub-components and cleaning may be required for this type of inspection. Tools required for this type of inspection may include flashlight, mirror, borescope, image enhancement and recording devices, rolling stools etc.

## 4.3 Visual Inspection Equipment

Various aids are used for visual inspection. One of the most important aids in visual inspection is the proper lighting and illumination. Reference 5 describes the ideal lighting and illumination required for proper visual inspection. The reference describes various portable lighting aids. The other inspection equipment required include: mirrors, magnifiers and equipment to obtain images from inaccessible places being inspected.

**Inspection Mirrors-** These are used to look at the areas which are not in the normal line of sight. A number of different mirrors are available to inspect hidden areas (Reference 5).

**Magnifying Devices-**These are used in the visual inspection to expand the area being inspected for detecting damage and other anomalies. These devices include: simple magnifying glass, microscope and illuminated magnifiers.

**Photographic and Video Systems-** A photographic image of the area being inspected enhances the decision-making capability of an inspector to interpret what he sees. Photographic and video systems are available which can be attached to borescope, fiberscopes or any other visual equipment for documentation and interpretation of visual inspection images. The photographic images can be stored as permanent records for later viewing. A number of systems are available in the market.

**Borescopes-** A borescope is a tubular precision optical instrument with built-in illumination to allow remote visual inspection of internal surfaces. Borescope tubes may be rigid or flexible and are available in a wide variety of lengths and diameters. These are available in a number of designs and manufacturers can supply custom made borescopes to serve customer needs. The selection of a borescope depends on a particular application and is governed by factors such as- resolution, illumination, magnification, field of view, working length, direction of view, etc.

Borescopes are used in aircraft structures and engine maintenance programs to inspect the areas which are difficult to reach and there by reduce/eliminate costly teardown inspections. These can be used to inspect the interiors of pipes, hydraulic cylinders, turbine blades and valves. They are also used to locate foreign object damage and verify the proper placement and fit of seals, bonds and gaskets.

## 4.4 Visual Inspection of Composite Structures

The in-service damage in composite structures is quite different from conventional metallic structures. In metallic structures detection of cracks and corrosion is of prime concern to the operators whereas in composite structures this kind of damage does not occur. The most common damage occurring in composites is impact damage which may result in internal matrix cracking, fiber breakage and delamination between plies without any appearance of external damage known as non-visible impact damage. Fortunately, all composite structures are designed for non-visible impact damage.

Any serious in-service damage that may affect the integrity of a structure has to penetrate, chip away or abrade the paint finish of the composite structure. Any damage caused by hailstorm, lightning or paint strippers will be easily visible on the surface and can be detected. Once the damage has been detected, the affected area needs to be inspected by other NDI methods for assessing the effect of the damage on structural integrity.

## 5.0 NONDESTRUCTIVE INSPECTION METHODS

As mentioned earlier a number of NDI methods are available and the use of a specific method depends on the type of structure being inspected, available access and the desired degree of accuracy in the inspection. Significant advancements have taken place in NDI methods recently. The methods and recent advancements are discussed in the following paragraphs.

### 5.1 Eddy Current

Eddy current is generally used to detect cracks and corrosion near the surface of metallic structures or in thin structures. Eddy current is also used for verifying and separating alloys by differences in their electrical conductivity. This technique has been gradually replacing x-ray. Hand-scanned eddy current probe coils can detect small cracks at fastener holes, however, the method is time consuming and tedious. As most conventional eddy current instruments display variations in the complex impedance, corrected for lift-off as seen by the probe coil, the flaw indications may be sometimes ambiguous. This generally requires trained and experienced operators to interpret the results. Also, the lift-off variations produced by surface roughness or paint thickness can result in false calls. The paint removal may be required prior to inspection with conventional eddy current equipment. Recent trends in eddy current technology have been towards the computerization, automation, improving capabilities to detect small flaws and flaws in multi-layer structures. Two NDI techniques which show significant promise in detection of corrosion and subsurface cracks without disassembly are Magneto-Optic/Eddy Current Imager (MOI) (References 7-10) and Low Frequency Eddy Current Array (LFECA) (References 11-14).

**Magneto-Optic/Eddy Current Imager (MOI)-** The MOI technique makes it possible to do faster, simpler and more reliable detection of cracks and corrosion in structures. This real-time imaging technology is based on a combination of magneto-optic sensing and eddy current induction. The images of holes, cracks or other defects are formed as the presence of these discontinuities in a material diverts the otherwise uniform flow of current near the surface of a structure as shown in Figure 8 (Reference 8). At eddy current frequencies of 25.6-102.4 kHz most through-the thickness fatigue cracks in aluminum are easily detected and imaged, whereas at lower frequencies (e.g. 6.4 kHz) hidden multi-layer cracks, corrosion and substructure (Reference 7) can be imaged. Figure 9 shows POD of sliding probe and MOI, indicating superior performance of MOI. Figure 10 shows typical cracks detected by MOI and Figure 11 shows corrosion detected by MOI.

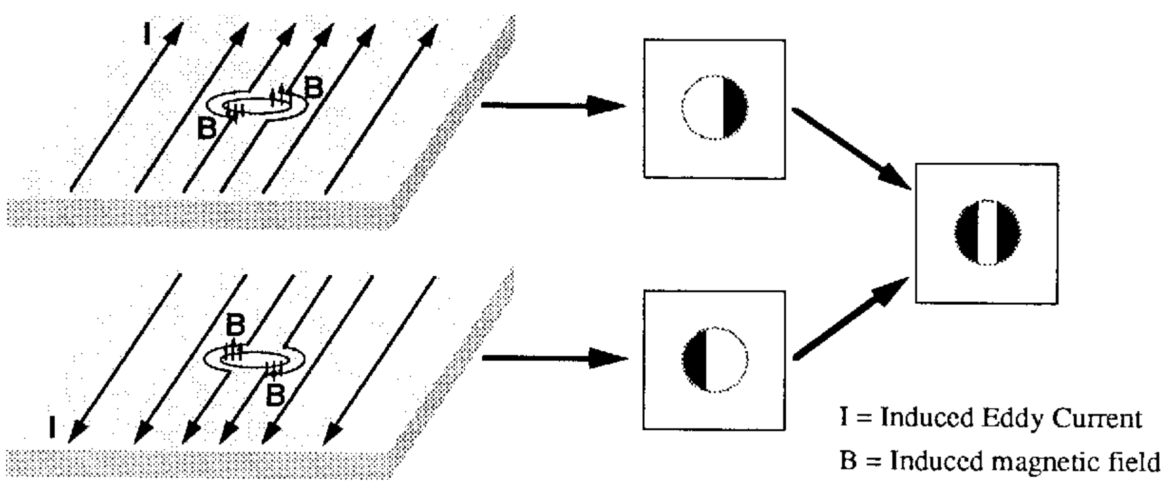
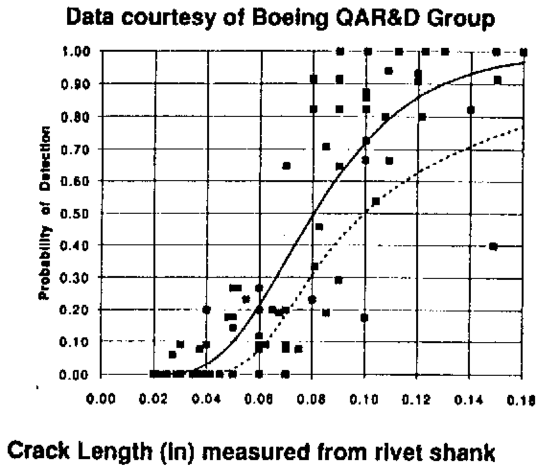
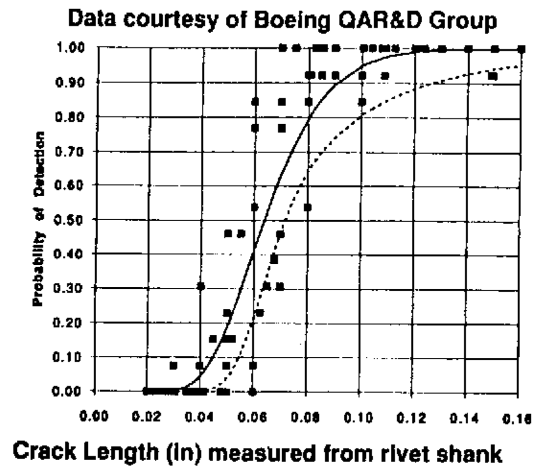



**Figure 8. Formation of Images with Magneto-Optic/Eddy Current Imaging**

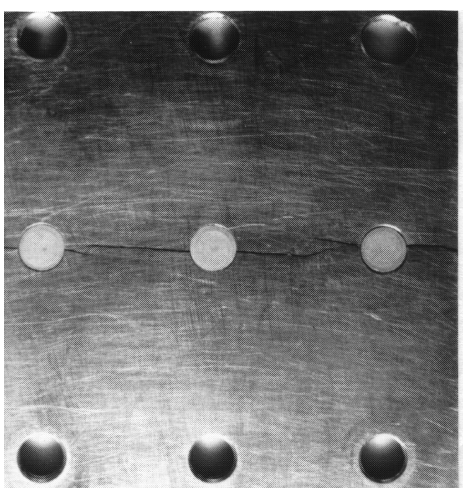



SLIDING PROBE

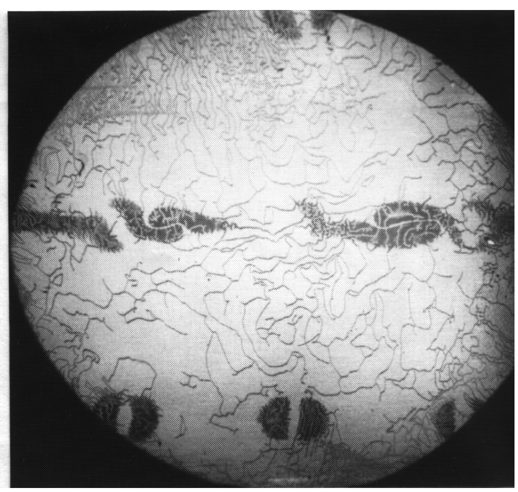



MOI

**Figure 9. Detection POD for Sliding Probe and MOI**

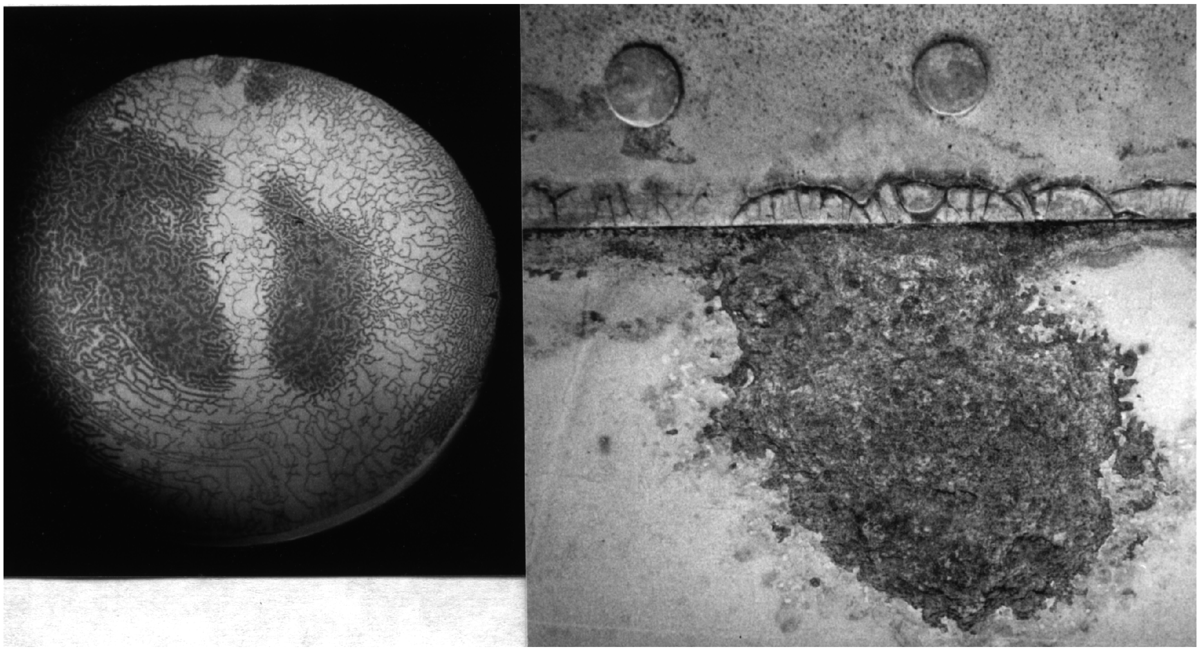


Specimen with Cracks at Fastener Holes



MOI Image

**Figure 10. MOI Image of Cracks at Fastener Holes**



**MOI Image** **Specimen with Corrosion**

**Figure 11. MOI Image of Corrosion**

The key advantages of MOI are (Reference 7): 1) Speed of operation 5 to 10 times faster than conventional eddy current, 2) Easy to interpret image formation, 3) No false calls, 4) Elimination of paint or decal for inspection, 5) Easy documentation of results on video or film, and 6) No operator fatigue.

**Low Frequency Eddy Current Array (LFECA)-** The LFECA system, developed by the Northrop Grumman corporation, is a portable eddy current inspection equipment to detect subsurface cracks under installed fasteners in multi-layer aircraft structures (References 11-14). The inspections can be performed in near real time without the removal of fasteners. The LFECA system can detect cracks, determine crack length and also give crack depth and orientation. The system consists of a LFECA probe for inspection, shown in Figure 12, three printed circuit boards, a cable and software all assembled in a portable personal computer. The LFECA probe consists of a cylindrical core made from ferrite material with a drive coil located on the center post of this core to generate an eddy current distribution that encircles the fastener being inspected. An array of 16 sense elements, spaced evenly around the outer rim of the core, measures the spatial distribution of these eddy currents. The presence of a crack causes a disruption in the eddy current distribution and is measured by the sense element array. The outer drive coil is used to measure the response due to the adjacent structural features independent of the features at the structural hole. A typical response obtained from the LFECA system is shown in Figure 13 (Reference 11) for various crack sizes along with the probability of detection. The horizontal tick marks in the figures indicate the 16 angular positions around the fastener hole such that going from left to right will indicate going around the fastener hole once. The horizontal location in the response indicates the orientation of the crack and the magnitude of the peak indicates the crack length.

The probability of detection of cracks with the LFECA system was obtained at Federal Aviation Administration (FAA) NDI validation center at Sandia National Laboratories in Albuquerque, New Mexico, USA (References 11-13). The POD process consists of a blind test of eddy current equipment to inspect a lap joint typical of a commercial airline fuselage shown in Figure 14. The process involves inspection of 43 specimens with each specimen containing 20 fastener holes.

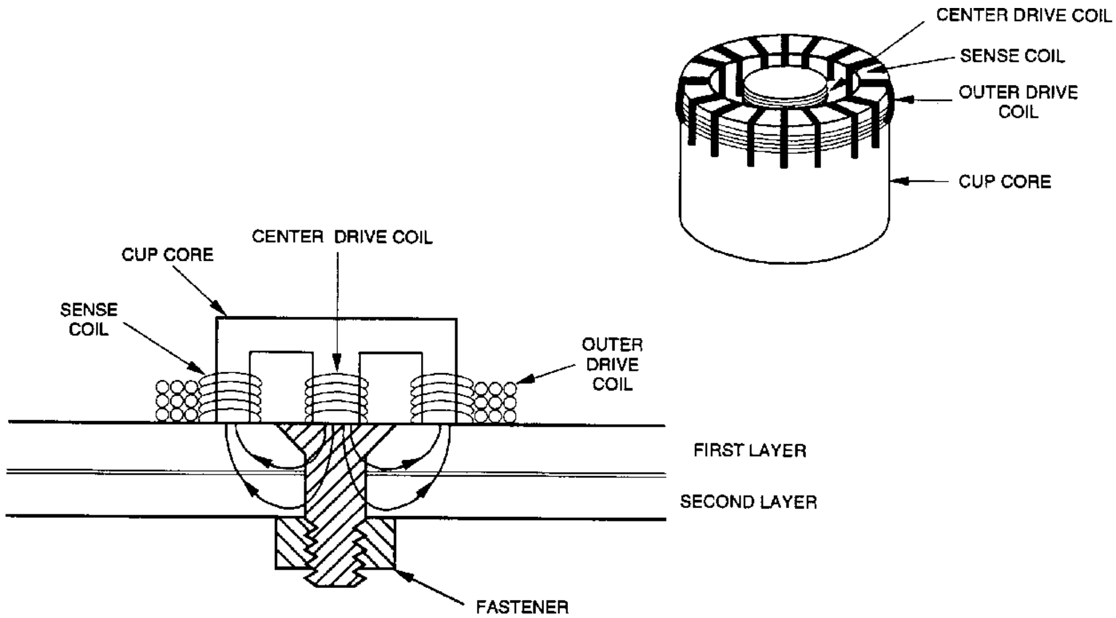



**Figure 12. Low Frequency Eddy Current Array Probe**

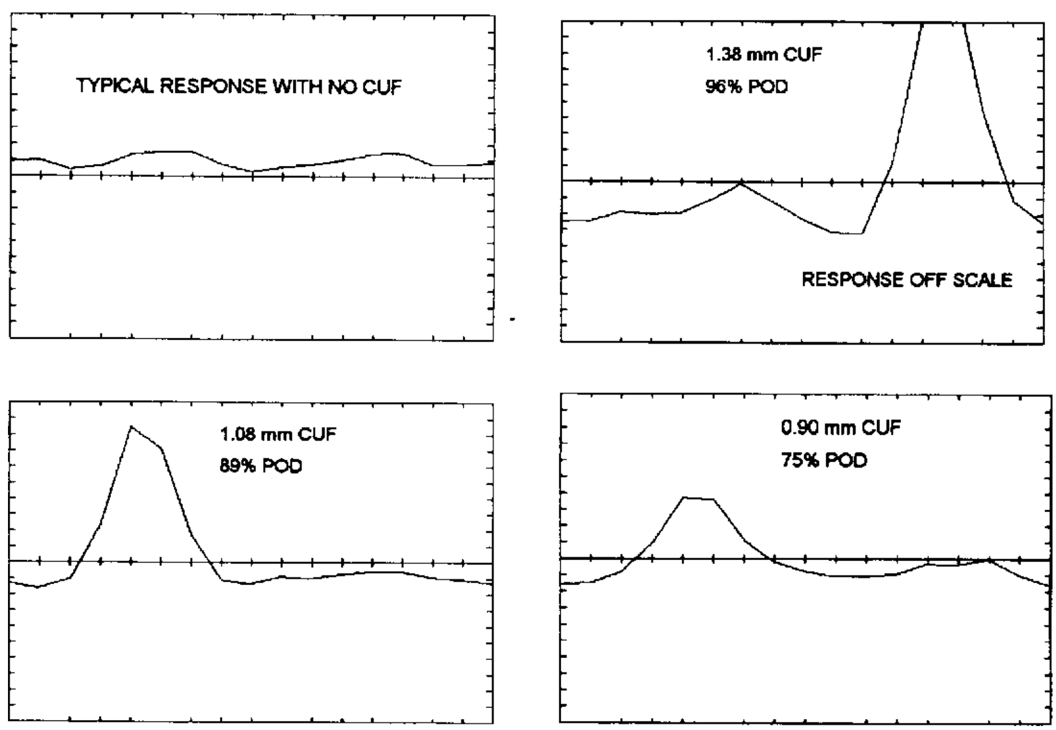



**Figure 13. LFECA Response for Cracks of Various Lengths Under Fasteners**

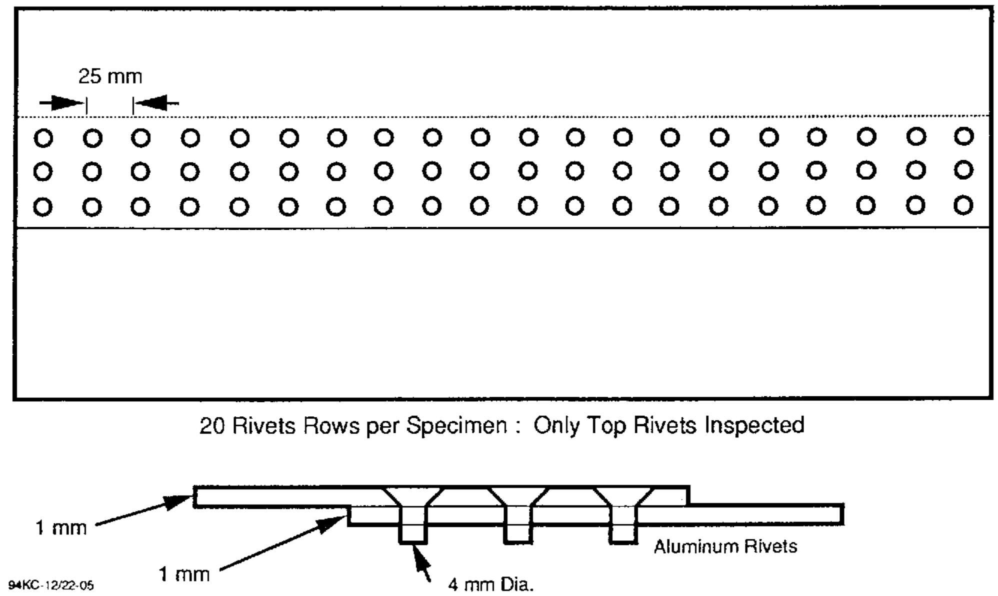



**Figure 14. Boeing 737 Lap Splice Specimen Configuration**

The specimen were constructed using 1 mm thick 2024-T3 aluminum sheets which were fastened together with three rows of 4 mm diameter aluminum flush head rivets. Fatigue cracks were grown in the first layer of selected holes prior to riveting the panels. A range of crack sizes from 0.3 to 25 mm (a hole to hole crack) were grown within +/- 22 degree orientation (0 degrees being the direction from hole to hole). Holes with cracks on one and both sides were present. Specimens contained either none, a low, a medium or a high number of cracks. A total of 860 holes were inspected with 708 being unflawed holes. The validation exercise contained only the first layer cracks under installed fasteners. Figure 15 shows the POD for the LFECA system and conventional eddy current techniques. It is seen that POD obtained with the LFECA system far exceeds that obtained with the conventional system.

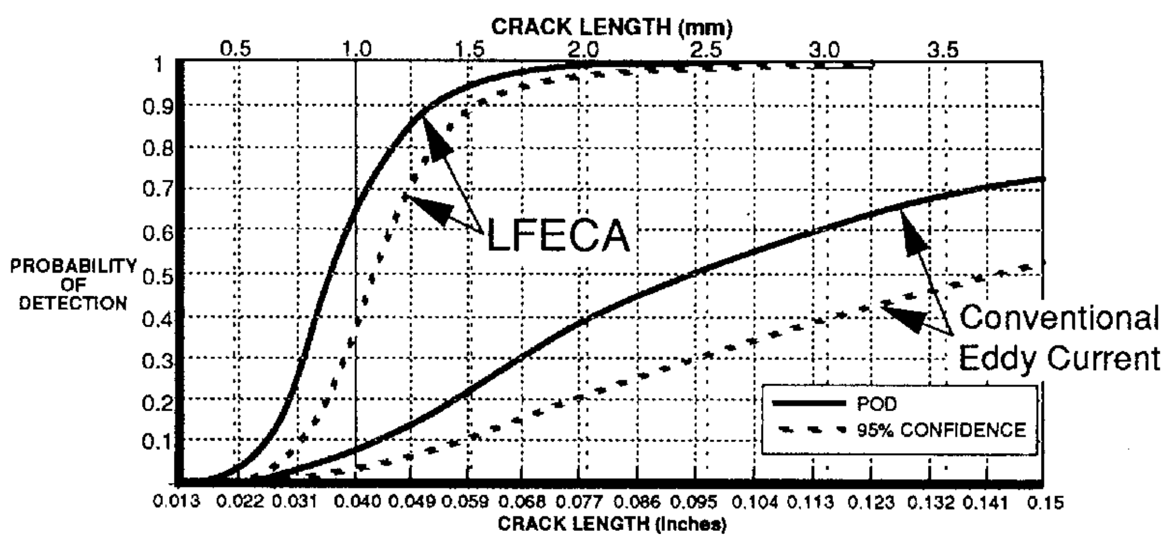



**Figure 15. Probability of Detection with Low Frequency Eddy Current Array and Conventional Eddy Current NDI System**

### 5.2 Ultrasonic Methods

Ultrasonic inspection techniques are widely used for quick and relatively inexpensive evaluation of flaws in composite structures. Portable inspection devices are used for on-site inspection of areas with suspected damage. Two methods, namely pulse-echo and through-transmission, are used. In the pulse-echo method, a transducer transmits the ultrasound and the same transducer receives the reflected signal after the signal has been reflected from the back surface of the composite part being inspected. The attenuation of the reflected pulse is influenced by the presence of the internal defects, and the time delay of the reflected pulse is related to the depth location of the defect. This method is generally used in contact mode of testing and only one side access is required. Inspection of honeycomb structures will require access from both sides for inspection of both face sheets. Ultrasonic inspection using through transmission method is generally conducted with water as a couplant by two methods- 1) Immersion, and 2) Squirting. In the immersion method the part and transducer are immersed in water whereas the squirting method employs dynamic water column that is squirted and the transducer and the part are suspended. In both methods water acts as the medium that transmits the ultrasound into and out of the part. The images of the defects may be recorded as B-scan, C-scan or 3-D scan. Scans for typical impact damage in a composite part are shown in Figure 16.

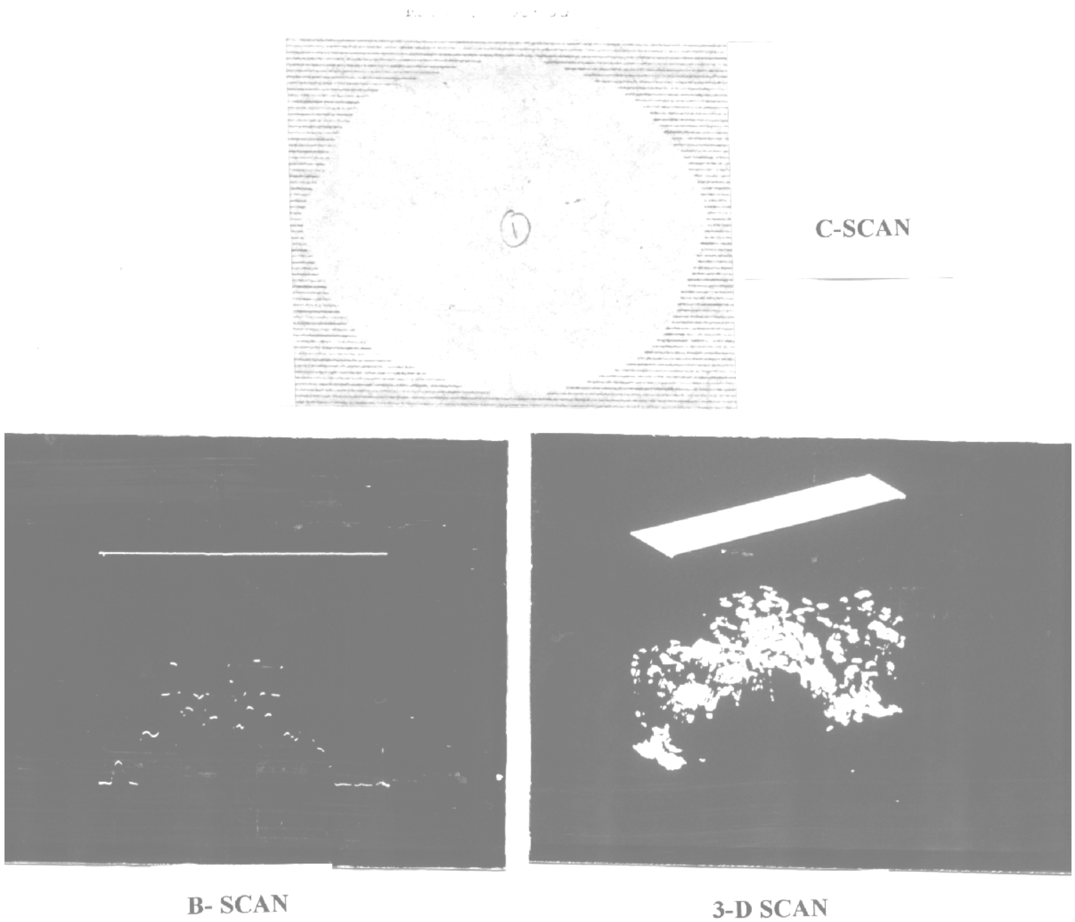



**Figure 16. B, C and 3-D Scans of Typical Impact Damage in Composite Laminate**

An ultrasonic technique to detect corrosion in a wing box has been developed in Reference 15. The technique has been successfully used to detect corrosion in DC-9 wing box substructure. The current method of inspection is to enter the wet wing box for corrosion inspection. The technique of Reference 15 eliminates entry in the wing box for the inspection and will result in significant savings in the inspection costs.

### 5.3 Radiographic Methods

The present trend seems to be getting away from using radiographic methods due to safety, cost and maintenance logistics. However, these methods are still being used to detect internal cracks and corrosion in aging aircraft structures. An advanced system known as COMSCAN, developed by Phillips, allows to form images of underlying structure and requires access to one side of the part only. It is currently being used to find corrosion in bulkheads under thin skins, and sonar dome inspections. The system is limited to finding defects near the surface and has the same detection capability as conventional x-ray. Digiray makes a system that has better resolution and better image quality than the conventional systems. The system is basically the reverse of a conventional digital x-ray imaging system as shown in Figure 17. The x-ray source is formed by a large scanned screen like a TV screen and the detector is a single point sensor as shown in the figure.

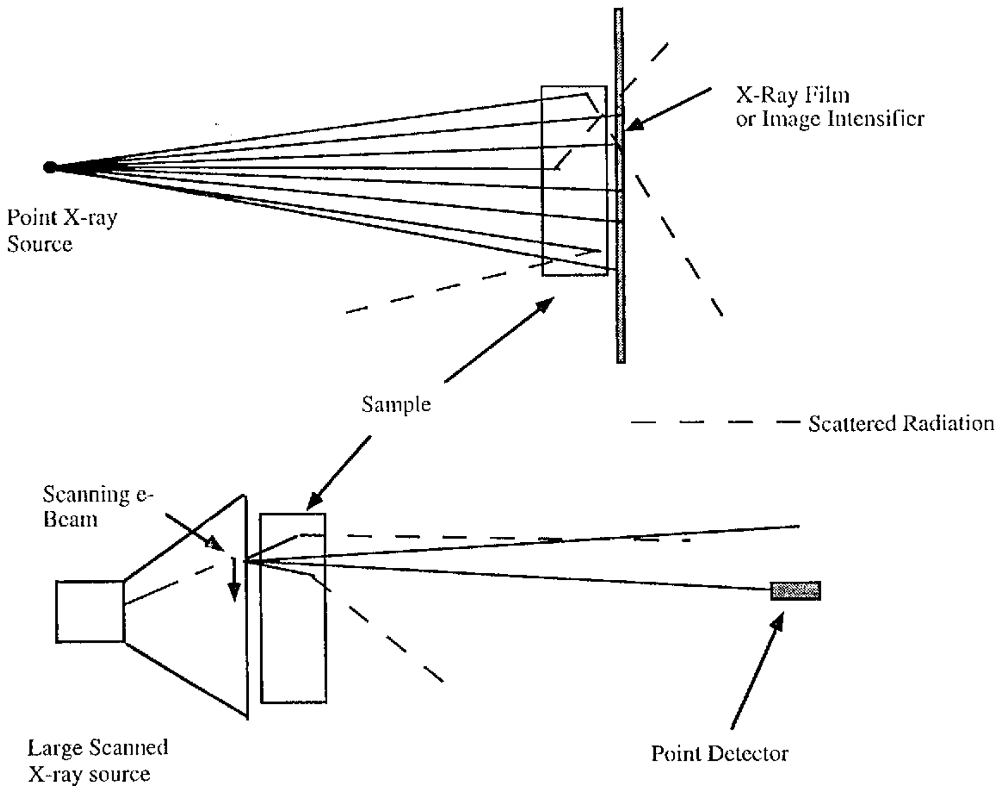



**Figure 17. Conventional and Reverse Geometry X-Ray Radiography**

### 5.4 Acoustic Emission

The acoustic emission (AE) technique is used to identify the flaw characteristics by change in acoustic emission signal. Acoustic emissions are transient waves that are generated by the rapid release of energy within a material when it undergoes deformation or fracture. This technique has been used to detect damage in composite materials and cracks in metallic structures. Various types of damages in composites such as matrix cracks, fiber/matrix debonding, fiber fracture and delaminations produce acoustic emissions that vary in magnitude, duration and frequency. Various damages in composite materials can be identified by the acoustic emission characteristics. Cracks in aircraft wing were located during ground test with AE technique in Reference 16 using AE sensors 20 inch (51 mm) apart. However, the source location of flaws could not be precisely predicted.

### 5.5 Optical Methods

Significant advancements have taken place in optical methods to detect damage in aircraft structures. Some of the techniques being- shearography, DIAS system and thermography.

**Shearography-** This is a field inspection technique which images internal defects as concentration of surface strain due to an applied stress. A reference image is stored electronically using the shearography video laser interferometer, then a uniform stress is applied in the form of vibration, pressure or thermal, and the subsequent images of the test part are compared with the reference image which will indicate flaws on video monitor (References 17-18). This is a cost-effective method for inspection of honeycomb and composite structures. Most of the other NDI techniques do point by point inspections whereas shearography provides a full field video image of flaws in real time. Defects such as disbonds, delaminations and impact damage can be detected with this technique.

**D Sight Aircraft Inspection System (DAIS)-** This is a fast and sensitive enhanced visual inspection system for detecting surface irregularities such as pillowing caused by corrosion (References 19-20). In Reference 19, DAIS system was used in the laboratory as well as in the field to detect corrosion in fuselage lap splices. The results of this reference showed that corrosion pillowing indicative of thickness loss as low as 2% is detectable. A typical D sight optical set-up is shown in Figure 18.

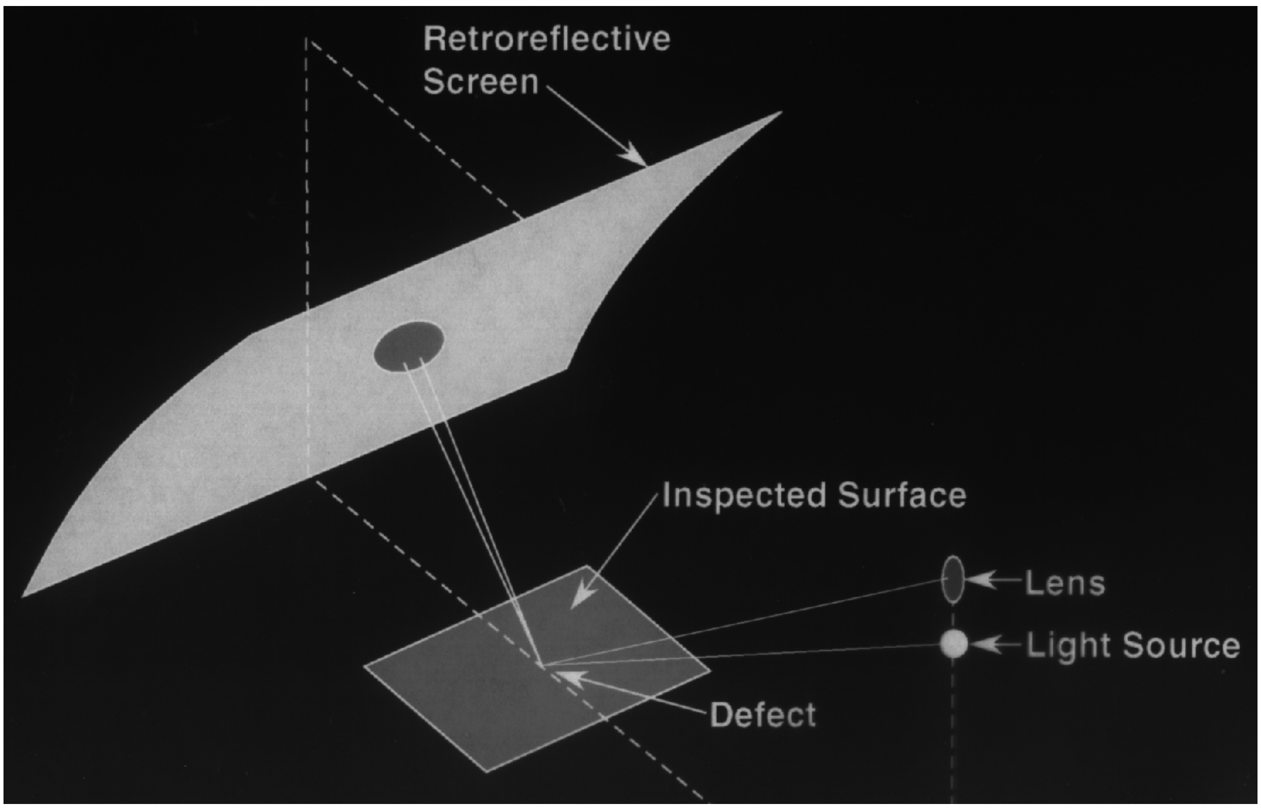
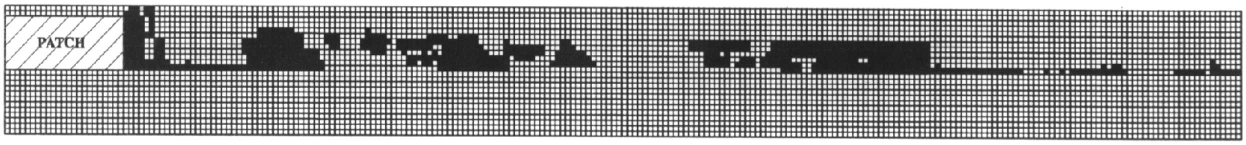
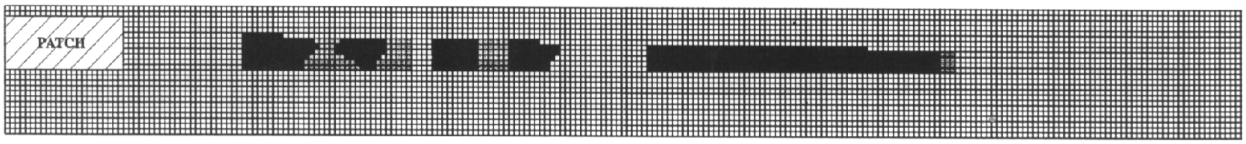



**Figure 18. D Sight Optical Set-up**

A comparison of fuselage joint corrosion detected by X-ray and D Sight is shown in Figure 19. The figure shows a very good correlation between the two techniques.




Boeing 727 fuselage lap joint corrosion, analyzed using X-Rays




Boeing 727 fuselage lap joint corrosion, analyzed using D Sight

**Figure 19. Comparison of Fuselage Joint Corrosion Detected by X-Ray and D Sight Techniques**

**Thermography-** This technique uses differential in the thermal conductivity of a defect free part and a part with defects as a basis for locating defects in a structure. A heat source is used to elevate the temperature of the structure being inspected and surface heating effects are observed through a radiometer. For example bonded areas conduct more heat than unbonded areas, the amount of heat either absorbed or reflected indicates the quality of the bond line.

A new technology known as “Thermal Wave Imaging” uses pulses of heat to examine the subsurface in solid objects (Reference 21). The pulses propagate in the structure being examined as thermal waves and are reflected from any defects, present in the structure, as surface “echoes”. These echoes are detected by the use of infrared video cameras, coupled to appropriate hardware and software. The patterns of the echoes on the surface of the structure are used to image subsurface corrosion and disbonds in aircraft structures. Thermographic inspection technique for detection of water ingress in sandwich structures is discussed in Reference 22. It is shown in the reference that this technique can be reliably used to detect water in sandwich structures.

## 6.0 NONDESTRUCTIVE INSPECTION OF METALLIC STRUCTURES REPAIRED WITH COMPOSITES

Nondestructive inspection of composite patch repair of metal structure involves two inspection issues- 1) inspection of bondline for disbonds, and 2) inspection of cracks underneath the repair patch. Bondline inspection has been reliably carried with Kraut Kramer Branson bond tester. Other bondtester such as Fokker bondtester have also been used.

Application of eddy current procedure to detect cracks underneath a composite repair has been investigated in Reference 23. A comparison of measured crack length using eddy current and anticipated crack length is shown in Figure 20. The figure shows the actual crack length when the crack was visible outside the patch and dotted line represent the anticipated crack length when the crack was not visible. A comparison between NDI measured crack length and anticipated length is good.

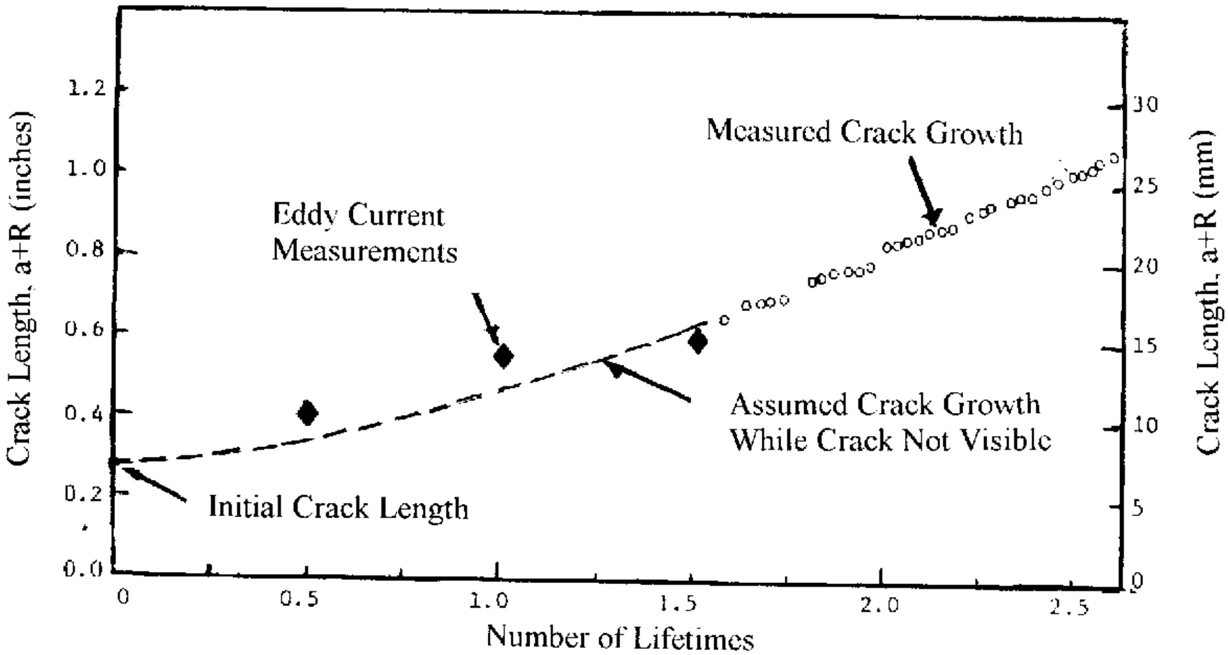



**Figure 20. Comparison of Measured, Using Eddy Current, and Anticipated Crack Lengths**

Conventional eddy current seems to be effective in detecting crack lengths of 0.25-inch or larger. However, for smaller crack lengths Low Frequency Eddy Current Array (LFECA) system, discussed earlier, has shown promise.

## 7.0 CONCLUDING REMARKS

Significant advancements have been achieved in NDI technology in the recent past. Some of the advancements are discussed in this paper. The use of a particular NDI method is highly dependent on the type of structure being inspected, structural material, desired accuracy, the size of the flaw to be inspected, type of damage, time available, and the labor skill. NDI and structural engineers have to make proper choices to assure the reliable detection of the damage with desired accuracy. Structural engineer can work together with NDI engineers to identify the requirements. Reliable inspection techniques are available for detection of damage in metallic structures underneath composite repair patches.

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# Human Factors in Aircraft Maintenance

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**Abstract:** Human error is cited as a major causal factor in most aviation mishaps, including the 15% - 20% that involve maintenance error. Errors can be described as active failures that lead directly to the incident, and latent failures whose presence provokes the active failure. Typical aviation maintenance errors are presented as examples and two approaches to human error reduction given: incident based and task analysis based. Each approach provides data on performance shaping factors, i.e. situation variables that affect the probability of error occurrences. Examples are given of interventions derived from analysis of incidents and from task analysis.

**1. The Need for Human Factors in Maintenance:** A sound aircraft inspection and maintenance system is important in order to provide the public with a continuing safe, reliable air transportation system (FAA, 1993). This system is a complex one with many interrelated human and machine components. Its linchpin, however, is the human. While research and development related to human factors in aviation has typically focused on the pilot and the cockpit working environment, there have been maintenance initiatives. Under the auspices of the National Plan for Aviation Human Factors, the FAA has recognized the importance of the role of the human in aircraft safety, focusing research on the aircraft inspector and the aircraft maintenance technician (AMT) (FAA, 1991, 1993). The classic term, "pilot error" or "human error", is attributed to accidents or incidents over 75% of the time; however, a recent study in the United States found that 18% of all accidents indicate maintenance factors as a contributing agent (Phillips, 1994).

Two incidents help clarify the issues involved and demonstrate that even though humans in the system were trying to do a good job, systems problems combined with errors to allow a serious event.

> *Case 1: Lockheed L-1011*. An in-flight turn-around was caused by all three engines failing on a flight from the USA to the Caribbean when the oil leaked out of each. The oil leak was caused by missing "O" rings on the magnetic chip detectors. They were missing because the mechanic had not notices that the new chip detectors were not fitted with "O" rings in the usual way. All work was performed outside in darkness, where a black "O" ring was difficult to see. Until that night, chip detectors had always come with "O" rings attached, even though the mechanic had to sign for both components. The new packaging still said they were ready for use.

> *Case 2: BAC-111.* During industrial action at the airline, a maintenance manager changed a windshield himself. He had not performed this task for two years, but checked the Maintenance Manual and it looked straightforward. He replaced 80 of the 84 bolts. The correct bolts were A211-8D, although A211-7D were on the old windshield. He matched the old bolts to new ones in a stores bin, but chose A211-8C, which was the correct length but the wrong thread. They engaged in the holes, but he used the wrong torque in setting them. Also because of the awkward posture required he could not see the bolts tighten. On the first flight, the windshield blew out, severely injuring the pilot and forcing an in-flight turn-around.

As a result of such incidents, the public has become more aware of the importance of aircraft maintenance as a safety issue, and both the civil aviation industry and its regulatory bodies have responded with programs to increase safety. Such programs have included hardware-based initiatives, such as the FAA's Aging Aircraft

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Program, and human factors initiatives by the FAA and many international bodies, for example by Transport Canada and the European JAA.

Over the last decade various human factors studies in maintenance-related issues have been initiated by agencies such as the FAA and NASA, by manufacturers, and by the aircraft maintenance industry. Examples of these initiatives are the National Aging Aircraft Research Plan (NAARP), the "Safer Skies" initiative, the White House Panel on Aviation Safety, and NASA's aircraft maintenance program. The objective of all these has been to identify research issues and to promote and conduct both basic and applied research related to human factors in aircraft maintenance. The human factors approach in maintenance research considers the human as the center of the system. Not only can human factors research have a significant effect on the design of new systems but it can also mitigate problems found in the sub-optimal designs of current systems.

**2. Human Factors Approaches to Maintenance:** Clearly, the main issue in aviation from a safety viewpoint is errors, or alternatively reliability. Where humans are part of the system, errors cannot be separated from the other two aspects of humans at work: performance speed and human well-being. Performance, typically measured by both reliability and speed, is the major concern of employees. Human well-being, e.g. health and safety of the workforce, is also an employer concern but is vital to continuing human work within a system. To some extent, there are tradeoffs between speed, reliability and well-being, but any human factors changes we make to improve the human/system fit can be expected to have a beneficial impact on all three measures. For this reason, our main consideration in this paper will be error reduction, or its equivalent: reliability improvement.

Perhaps the most widely accepted error models arise from systems reliability analysis (e.g. Embrey, 1984) but in the human error field more cognitive models have gained wide acceptance. These were originally developed for tasks such as aircraft piloting, industrial process control or air traffic control (Nagel, 1988). The ideas of Reason (1990) concerning error in complex systems have been particularly influential.

Reason differentiates between the proximal cause of an accident, the Active Failure, and more hidden causes that make the accident sequence more likely, the Latent Failures or Resident Pathogens. Active failures are at the "sharp end" of the incident, for example the pilot who fails to prevent an aircraft impacting a mountain, and are thus usually discovered easily. Resident pathogens, in contrast, can lie dormant in a system for considerable periods before they become manifest. In the L-1011 example, the active failure was that "O" rings were not installed, but a number of latent failures ensured that the changed parts led to an incident:

1. The mechanic's habituation to signing for both components even when they came as an assembled unit.
2. The dark outside environment that prevented ready detection of the error.
3. The assignment of the same mechanic to service all three engines, a practice no longer tolerated, e.g. for ETOPS certification.
4. The unchanged packaging and lack of alerting of the mechanic to the change.

In Reason's model, if we can reduce the latent failures, then the active failure will be prevented. We try to reduce the impact of latent failures by attempting to prevent error propagation through a system, usually by providing barriers (or error traps or recovery mechanisms). Thus, in the above example, training the mechanic to check each component before signing would be a barrier to error propagation. It would be a relatively poor barrier as training people to perform unnatural acts is not particularly reliable. We characterize such a barrier as being porous. Clearly, if enough of the barriers are porous, a triggering event such as the changed packaging can propagate through several barriers to impact public safety. In aviation the aim is to provide barriers which prevent such propagation. In fact, a system that has only a single barrier is considered unsafe and thus prevents an aircraft being certified as airworthy. At times it may not be apparent from initial analyses that there is only a single barrier. The recent crash of Concorde in Paris gives such an example.

There are two complementary ways to locate resident pathogens in any system:

1. *Incident-based.* If incidents have occurred, then detailed analyses of them will list resident pathogens as well as active failures.
2. *Task-Analysis-based.* Whether or not incidents have occurred, comparison of task demands with human capabilities will locate task elements where errors are likely, i.e. resident pathogens.

Both of these approaches have been used successfully in aviation maintenance. The next two sections cover products of the FAA's Human Factors in Aviation Maintenance and Inspection Program over the past decade and provide instances of usable findings to help achieve non-porous error barriers. More comprehensive accounts can be found in the special issue of *International Journal of Industrial Ergonomics*, Volume 26(2000), 125-240, and in the *Proceedings of the IEA 2000/HFES 2000 Congress*, Volume 3, pages 766-798. Information from the FAA's program is available on-line at hfskyway.faa.gov.

**3. Incident-Based Approaches:** In Section 1, two incidents were presented to illustrate human factors in maintenance accidents—an incident-based approach. Similar approaches have been the analysis of all maintenance-caused accidents investigated by the National Transportation Safety Board (available on hfskyway.faa.gov) and the development of a set of prototypical unsafe behaviors, known as the "Dirty Dozen" (Dupont, 1997). The Aviation Safety Reporting System (ASRS) in the USA can also be searched for maintenance-related incidents to provide additional examples of latent as well as active failures.

Based on such listings of maintenance-related latent failures, several incident analysis schemes have been proposed. These focus at least initially on single incidents, whether the severity of the incident is an aviation catastrophe (very rare), an operational incident such as an aircraft diversion (more common) or even an error discovered before it had propagated (quite common). There is good evidence that the same latent failures are found in the path leading towards severe incidents as in paths towards incidents of less consequence (Schmidt et al, 2000). Typical incident investigation systems include the Human Factors Analysis and Classification System – HFACS (Schmidt et al, 2000), the Proactive Error Reduction System – PERS (Drury, Wenner and Murthy, 1997) and the Maintenance Error Decision Aid – MEDA (Rankin, 2000). The last of these is typical in that it was derived from applying the human reliability analysis tradition to maintenance incidents. Typical errors were used to derive Performance Shaping Factors (PSF's) which describe situational variables that affect error likelihood, such as poor training or adverse weather conditions. Hierarchical lists of such factors were developed and used to provide a 4-page checklist covering:

*Error Types*

- improper installation improper
- servicing, improper/incomplete repair, improper fault
- isolation/inspection/testing, actions causing foreign
- object damage, actions causing surrounding equipment
- damage, and actions causing personal injury

*Contributing Factors (PSFs)*

1. Information-written or computerized source information used by maintenance technicians to do their job, e.g., maintenance manuals, service bulletins, and maintenance tips
2. Equipment, tools, and parts
3. Airplane design and configuration
4. Job and task
5. Technical knowledge and skills
6. Factors affecting individual performance-e.g., physical health, fatigue, time constraints, and personal events
7. Environment and facilities
8. Organizational environment issues-e.g., quality of support from other Maintenance and Engineering organizations, company policies and processes, and work force stability

9. Leadership and supervision-e.g., planning, organizing, prioritizing, and delegating work
10. Communication-e.g., written and verbal communication between people and between organizations.

*Error Prevention Strategies*: existing procedures, processes, and policies in the maintenance organization that were intended to prevent the error, but did not.

Even though analysis of each investigated incident produces error prevention strategies specific to that incident, more use can be made of the accumulated data from many incidents to guide broader policies. Wenner and Drury (2000) analyzed data from 130 incidents that had resulted in ground damage to civil aircraft. These incidents, Ground Damage Incidents (GDIs), had been investigated and reported to the Technical Operations Department of an airline from 1992-1995.

Initially, each GDI report was reviewed to determine the specific action that caused the ground damage. The reports could be sorted into twelve distinct patterns covering almost all of the GDI reports, termed here as Hazard Patterns after Drury and Brill (1978).

Next, each GDI report was analyzed to determine the specific active failures, latent failures, and local triggers that contributed to the incident. A scenario was then developed for each hazard pattern, illustrating the common factors between all of the incidents. Each of these was also summarized as an event tree illustrating how each of the latent failures contributed to the final damage event. This form of analysis, which has much in common with Fault Tree Analysis, was originally developed by CNRS in France (Monteau, 1977). The scenarios developed for each hazard pattern are given in Table 1 with their frequencies.

**Table 1. GDI Hazard Patterns**

| Hazard Pattern | Number of Incidents | | | % of Total |
|---|---|---|---|---|
| 1. Aircraft is Parked at the Hangar/Gate/Tarmac | 81 | | | **62** |
| 1.1 Equipment Strikes Aircraft | | 51 | | 39 |
| 1.1.1 Tools/Materials Contact Aircraft | | | 4 | 3 |
| 1.1.2 Workstand Contacts Aircraft | | | 23 | 18 |
| 1.1.3 Ground Equipment is Driven into Aircraft | | | 13 | 10 |
| 1.1.4 Unmanned Equipment Rolls into Aircraft | | | 6 | 4 |
| 1.1.5 Hangar Doors Closed Onto Aircraft | | | 5 | 4 |
| 1.2 Aircraft (or Aircraft Part) Moves to Contact Object | | 30 | | 23 |
| 1.2.1 Position of Aircraft Components Changes | | | 15 | 12 |
| 1.2.2 Center of Gravity Shifts | | | 9 | 7 |
| 1.2.3 Aircraft Rolls Forward/Backward | | | 6 | 4 |
| 2. Aircraft is Being Towed/Taxied | 49 | | | **38** |
| 2.1 Towing Vehicle Strikes Aircraft | | 5 | | 4 |
| 2.2 Aircraft is Not Properly Configured for Towing | | 2 | | 2 |
| 2.3 Aircraft Contacts Fixed Object/Equipment | | 42 | | 32 |
| 2.3.1 Aircraft Contacts Fixed Object/Equipment | | | 13 | 10 |
| 2.3.2 Aircraft Contacts Moveable Object/Equipment | | | 29 | 22 |
| **Totals** | 130 | 130 | 130 | 100% |

In a similar way, the latent failures were categorized into a second hierarchy in Table 2.

**Table 2. Incidence of Latent Failures**

| SHELL Model Category | | Latent Failure | Number of Incidents | | | % of Total |
|---|---|---|---|---|---|---|
| **Hardware** | | | | | **72** | **27** |
| | **H1** | **Poor Equipment** | | **72** | | **27** |
| | | H1.1 Poor Equipment: Inappropriate for Task | 39 | | | **15** |
| | | H1.2 Poor Equipment: Mechanical Problem | 33 | | | **12** |
| **Environment** | | | | | **51** | **19** |
| | **E1** | **Inadequate Space** | | **30** | | **11** |
| | | E1.1 Inadequate Space: Congested Area | 22 | | | **8** |
| | | E1.2 Inadequate Space: Ill-suited for Task | 8 | | | **3** |
| | **E2** | **Problems with Painted Guidelines** | | | | **8** |
| | | E2.1 Guidelines: Do Not Exist | 7 | | | **3** |
| | | E2.2 Guidelines: Do Not Extend Out of Hangar | 4 | **21** | | **1** |
| | | E2.3 Guidelines: Not Suitable for Aircraft | 10 | | | **4** |
| **Liveware (Individual)** | | | | | **34** | **13** |
| | **LI** | **Lack of Awareness of Risks/Hazards** | | **34** | | **13** |
| **Liveware - Liveware** | | | | | **108** | **41** |
| | **LL1** | **Poor Communication** | | **29** | | **11** |
| | | LL1.1 Poor Communication: Between Crew | 24 | | | **9** |
| | | LL1.2 Poor Communication: Between Shifts | | | | |
| | | **Personnel Unaware of Concurrent Work** | 5 | | | **2** |
| | | **Correct Number of People Not Used** | | | | |
| | **LL2** | **Pressures to Maintain On-Time Departures** | | **8** | | **3** |
| | **LL3** | **Pushback Policies Not Enforced** | | **36** | | **14** |
| | **LL4** | | | **19** | | **7** |
| | **LL5** | | | **16** | | **6** |
| | | **Total** | | | **265** | **100%** |

Note: Totals exceed the number of incidents due to multiple latent failures per incident.

After consistent latent failures were identified, a logical structure was imposed using ICAO's SHELL Model (ICAO, 1989). For the tasks leading to ground damage, no software failures (e.g. documentation design) were found. Note that there are typically multiple latent failures for each hazard pattern, so that their total is 215 rather than 130.

Next, a cross-tabulation was made of the hazard patterns and overall latent failures to give the results in Table 3.

**Table 3. Chi-Square Analysis of the Hazard Patterns/Latent Failure Relationship**
*** Indicates a frequency larger than expected**

| | HP 1: Aircraft Parked | HP 1.1: Equipment Strikes Aircraft | HP 1.2: Aircraft (or Component) Moves to Contact Object | HP 2: Aircraft Being Towed/Taxied | HP 2.3: Aircraft Contacts Equipment |
|---|---|---|---|---|---|
| Hardware | 53* | 47* | 6 | 19 | 11 |
| Environment | 20 | 18 | 2 | 31* | 31* |
| Liveware (Individual) | 22 | 10 | 12* | 12 | 9 |
| Liveware-Liveware | 62 | 4 | 31* | 46 | 40 |

This data table was tested using a Chi-Square test and found a significant relationship ($X^2_3 = 15.2$, $p < 0.001$). Further analysis using standardized residents gave the over-represented cells (denoted by * in Table 3). These show, for example, that if the aircraft is parked, hardware latent failures are over-represented (HP1) and that most of this relationship comes from equipment striking aircraft (HP1.1) rather than HP1.2. In fact, this cause was usually poorly-maintained ground equipment striking the aircraft. Similarly, for HP1.2, where the aircraft or a component moved to contact another object, most of the incidents involve people failures (liveware, and liveware-liveware interaction). These were in fact lack of awareness of on-going activities (L-L) or failure to perceive hazards (L).

In this way, a listing could be made of those latent failures associated with each hazard pattern. These in turn defined intervention strategies likely to be successful in prevention of future incidents. Thus, for equipment striking the aircraft (HP1.1) concentrating on maintenance of ground equipment would be a more successful intervention strategy then, for example, improving individual motivation or training.

**4. Task Analysis-Based Approaches:** The earliest approach to analysis of human factors in aviation maintenance was task-analysis-based (Lock and Strutt, 1985). These authors followed classical human reliability analysis techniques (e.g. Swain, 1990) to break down an aircraft inspection task into successively smaller units of human behavior. Each behavior was then considered for its error potential so that performance shaping factors and interventions could be developed.

In general, task analysis proceeds by progressive redescription of the whole task into successively smaller units. Thus, the overall task may be "check tire pressures and condition." If the aircraft has six tires, then the pressure and condition of each must be checked. For the left nosewheel (for example), the outer surface and bead must be examined for a series of defects (tread wear, de-lamination, cuts, etc.), and so on. When each step is described in sufficient detail, the task description is complete and task analysis can begin (e.g. Drury, Paramore, Van Cott, Grey, and Corlett, 1987). In task analysis, task demands are compared with expected human capabilities to determine potential human/ system mismatches. These in turn define error-prone steps so that countermeasures can be developed. This technique was used as part of the structuring of the FAA's Human Factors in Aviation Maintenance and Inspection Program, beginning with analysis of many inspection tasks (Drury, Prabhu and Gramopadhye, 1990). From these came a set of generic functions (logically-related groups of tasks) for inspection, that were later expanded to cover both inspection and maintenance (Drury, Shepherd and Johnson, 1997). These tasks are defined in Table 4.

**Table 4. Generic Function and Task Descriptions of Maintenance and Inspection**

| Maintenance | | Inspection | |
|---|---|---|---|
| **Function** | **Tasks** | **Function** | **Tasks** |
| **Initiate** | Read and understand workcard. Prepare equipment, collect and inspect supplies. | **Initiate** | Read and understand workcard. Select and calibrate equipment. |
| **Site Access** | Locate and move to worksite with equipment, parts and supplies. | **Site Access** | Locate and move to worksite. |
| **Part Access** | Remove items to access parts, inspect and store items. | | |
| | | **Search** | Move eyes or probe across area, stop if any indication. |
| **Diagnosis** | Follow diagnostic procedures. Determine parts to replace/ repair and collect/ inspect needed parts/ supplies. | **Decision** | Re-examine area of indication. Evaluate indication against standards to decide if defective. |
| **Replace/ Repair** | Remove parts to be replaced/ repaired, repair and replace. | | |
| **Reset Systems** | Add supplies/ fluids. Adjust systems to specifications, inspect and buyback if needed. | | |
| **Close Access** | Refit and adjust items removed for access. Remove equipment parts and unused supplies. | | |
| **Respond** | Write up documentation on repair. | **Respond** | Write up documentation for repair. Mark defect for repair. Return to search. |
| **BuyBack** | (performed by inspector) | **BuyBack** | Examine repair against standards and sign off. |

The FAA/AAM program has used FAA researchers, human factors practitioners and airline partners to develop and test systems, procedures, job aids and computer tools designed to ease the utilization of human factors/ ergonomics knowledge within aviation maintenance and inspection. We have developed an application methodology in which each individual project teams a researcher with an airline partner. Most of the major air carriers in the USA have now taken part in these projects. Projects have included design of computer-based training programs, design of enhanced maintenance documentation, development of human factors audit programs, and applications of concepts such as team training and group situation awareness to maintenance. The tools and job aids developed on these projects have been made available to the aviation industry, initially on CD-ROM but more recently on the World Wide Web.

A good overview of the program is given in Latorella and Prabhu (2000). Interventions have been proposed, researched and developed for many of the functions in Table 4. A classification of these (up to 1997) is given in Table 5 from Drury, Shepherd and Johnson (1997). Note that these are also system-level actions such as the development of Crew Resource Management (CRM) training for maintenance, now characterized as Maintenance Resource Management (MRM) (see Taylor and Christensen, 1998). Currently, training of maintenance personnel in human factors concepts is the most popular intervention for maintenance human error. Indeed, such training is now mandated by the International Air Transport Association (IATA) and by the regulatory authorities in many countries.

**Table 5. Classification of Interventions for Human Factors in Maintenance and Inspection**

| System Level Actions | |
|---|---|
| Development of Human Factors Audit Programs | Characterization of Visual Inspection and NDI |
| CRM Analysis of Maintenance and Inspection | Error Analysis and Reporting Systems |
| CRM Training for Maintenance and Inspection | PENS System for Audit |
| Hangar-Floor Ergonomics Programs | Human Factors Guide |

**Function-Specific Interventions**

| Function | Personnel Subsystem | Hardware/SoftwareSubsystem |
|---|---|---|
| Initiate Inspection | | Workcard redesign |
| Inspection Access | | Restricted space changes |
| Search | Visual search training | Task lighting design |
| Decision | Feedback for decision training<br>Individual differences in NDI | |
| Inspection Response | | Computer-based workcards |
| Initiate Maintenance | | |
| Maintenance Site Access | | |
| Diagnosis | Diagnostic training | ITS computer-based job aid |
| Maintenance Part Access | | |
| Replace/Repair | International differences | |
| Reset System | | |
| Inspection Buy-back | International differences | |
| Close Access | | |
| Maintenance Response | | |

Here we will explore one intervention in a little more depth, that of improved communication through better document design.

The work documents themselves serve a number of different purposes. First, they are part of a work control system that assures that all tasks are completed according to a time schedule. Assigning a work card to a mechanic, and receiving it back with the stamps and data completed, allows the scheduling system to function reliably. Second, the workcard is part of a quality audit trail which allows management to analyze tasks at a later time if an error was detected. For both reasons, the interactions of mechanics with the work card must proceed reliably for the system to function correctly. Finally, the work card is a job aid in the Human Factors sense of a tool used to help reliable performance of a procedure. Any human/work card mismatches in this final sense will compromise the overall system reliability. Thus, it is vitally important that the work card be designed to meet user needs, i.e. it must fit the mechanic or inspector.

From previous research, we have found that well-designed documents have a significant impact on performance reliability. Our work has focused on layout rather than content, but has included studies of Simplified English, order of steps, typography, and computer-presented documents (Patel et al 1994). We have also shown how the process for writing and changing documents using user teams can lead to improvements. Two examples are worth mentioning.

First, an existing airline procedure that had caused operational problems was analyzed (Drury, 1998). This document required 9 inspector responses, and had yielded an error rate of 1.5% of responses, meaning that 21% of all documents contained at least one error. When we compared the errors to Human Factors guidelines for good document design, we found that the error rate was 2.5% where these guidelines were violated, and 0.0% where they were met. That is, ALL of the errors could have been eliminated by following good Human Factors practices.

Second, a study of repair station errors (Drury, Kritkauski and Wenner, 1998) compared comprehension of work documents used by two different airlines with those designed using our Documentation Design Aid (DDA). Significant differences in comprehension errors were found between the three formats on two different tasks:

| | **Cable Workcard** | **Wing Workcard** |
|---|---|---|
| | Errors | Errors |
| Airline A Format | 52% | 36% |
| Airline B Format | 27% | 20% |
| Documentation Design Aid | 4% | 17% |

Thus, the design of work documents DOES impact reliability. It is not just a matter of designing to please operators, or meeting arbitrary style rules, but of using evaluated data to improve the probability that a document will be used correctly in practice. This issue of improved design is relatively simple to incorporate into procedure work cards.

These documentation design rules have been incorporated into a Documentation Design Aid (Drury, 2000) that provides both rule-based advice and knowledge-based reasons for formatting and wording improvements to documents. The DDA program for Windows is available for downloading from hfskyway.faa.gov.

**5. Conclusions:** This paper provides only a sampling of the research performed and products made available from human factors in aviation maintenance and inspection research programs. The first issue in developing a human factors program in an airline or other maintenance organization is to recognize that human error will not be eliminated by blame, motivation or even most training. True system interventions require an integrated approach of all the elements in the SHELL system: software, hardware, environment, liveware and liveware/ liveware interaction. Organizations should now have sufficient data and incentive to undertake human factors programs. This paper gives a logical approach to error reduction, combining error investigation and task analysis-based approaches.

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# Material and Process Technology Transition to Aging Aircraft

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## Summary

A method is described that may be used to help ensure that a structural material or process will be successful when transitioned from the laboratory for replacement of existing materials and processes in an aging aircraft. Experience with laboratory and aircraft development programs has shown that five factors are essential for success in the technology transition process. An example is shown where the transition using this process was successful.

## Introduction

Most aircraft operated commercially or in the military reach a state referred to as aging at sometime after entering operational service. Aging of an aircraft is not the same as it becoming obsolete. An aircraft may be obsolete before it reaches the aging state or, more typically, it reaches the state of aging before it is obsolete. A commercial aircraft is obsolete when it is no longer economically viable to keep it operational. A military aircraft is obsolete when its capabilities are no longer competitive with potential adversaries. The time when an aircraft reaches the aging state is usually much more difficult to determine. It is important to distinguish between the characteristics of the structure of a young aircraft and an aging aircraft. A young aircraft is one that continues to be airworthy with the maintenance program prescribed at the time of manufacture. The primary concern with a young aircraft is the potential for design errors that introduce unintentional high stresses in the structure that could lead to premature fatigue cracking incidents. When these are discovered the structure is modified to eliminate the problem. An aging aircraft may be characterized as one where the effects of corrosion and cracking from fatigue require modification of the maintenance program to retain adequate structural integrity. The word adequate here means that the expected number of failures would be less than one in a given fleet of aircraft. As an aircraft accumulates calendar time and flight time the effects of corrosion and cracking from fatigue, as well as accidental damage, leads to repairs on the aircraft. Cracking from fatigue can be so widespread that it degrades the integrity of the structure. When this occurs, the structure is said to be in a state of widespread fatigue damage or WFD and must undergo modifications to remove this problem. In addition, as an aircraft accumulates flight time, it may exceed its design life goal. Therefore, the maintenance program will require modification to include additional structural inspections. If the initial maintenance program requires modification from any of these events, then the aircraft may be considered to be in a state of aging.

No one should be surprised there are aircraft all over the world today in a state of aging. Economic considerations demand that aircraft be operated long beyond originally identified retirement times. One reason for keeping aircraft in the inventory is that technological advances allow currently designed aircraft to effectively perform their mission for much longer than previously possible. An aircraft, even when sold by one airline, sees extended life in another airline's operations. In the commercial sector, new aircraft tend to be evolutionary in their designs. Consequently, they are maintained in service until they are not economically viable to operate. The cost of new aircraft, particularly for the military, is enormous. Each new military aircraft is a revolutionary

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change from the previous model since the services must maintain combat effectiveness in an environment of ever-changing threats. Therefore, military aircraft stay in the inventory until they are operationally obsolete or they are no longer economically viable to operate.

Sustainment of an aircraft is the act of keeping it operational (i.e., airworthy). Maintenance of an aircraft (that is, the work done by mechanics in keeping it airworthy) is one aspect of sustainment. However, sustainment also includes the engineering analyses and tests needed to determine an adequate maintenance plan for the aircraft. Sustainment is life management. One task of sustainment is the determination of structural inspections based on damage tolerance principles. These inspections protect against failure from defects that could be in the structure because of manufacturing or from operational service. The approach for developing a damage tolerance derived inspection program is well documented. Another task of sustainment is the determination of the time of onset of fatigue cracking in the structure so widespread that the structural integrity of the aircraft could be compromised. Experimental evidence shows that fatigue cracks smaller than those that could be easily detected by current inspection methods could constitute WFD.

Today, the primary concern with aging aircraft is the cost of their sustainment. The commercial operators buy new aircraft when it becomes economically viable for them to do so. The aging aircraft problem, however, has often made itself known to both the commercial and military operators through failures of in-service aircraft. Both operators found the maintenance programs did not adequately protect aircraft as they progressed through their service lives. The failures in both commercial and military aircraft have been the primary factor that has changed rules and specifications that are the basis of their design. In many cases, the failures have identified threats to structural integrity that were not previously identified by the certification authorities. In many cases, the commercial failures have influenced the military specifications and the military failures have influenced the commercial aircraft rules. The new rules and specifications have lead to better maintenance programs that help alleviate many of the threats to failure that previously existed. However, the economic demand to fly these aircraft longer and longer has emphasized the need to re-examine these aircraft for the possibility of WFD, corrosion damage, and loss of damage tolerance capability through repairs.

It is difficult to determine the exact moment in time when an aircraft has reached the state of aging. However, all would agree that the costs associated with repairs or modifications from corrosion or cracking from fatigue would be an indicator of this condition. When these costs rise significantly, then the aircraft are certainly in that state. Chronological age is not always a good indicator of aging. However, since corrosion and fatigue are somewhat related to time in service, it does give some insight for the potential of this problem.

The time of development an aircraft is an important factor. Many development programs for materials in the fifties and sixties responded to aircraft performance needs. At this time, the material suppliers introduced high strength low ductility alloys in an attempt to satisfy the demand for less structural mass. Integrity programs such as USAF Aircraft Structural Integrity Program (ASIP) were either nonexistent or immature. Consequently, there was a lack of knowledge of threats to structural integrity

The structural engineer is fortunate to have many new technologies that could be used to help in meeting the challenges of aging aircraft. Some of these new technologies have been the basis for the sustainment of the aging fleets. There are several life enhancement techniques. An example of this is the cold expansion of fastener holes to improve their fatigue resistance. An outgrowth of this technology for the bushing of lugs has proved to be quite useful. The use of shot peening and laser peening have been shown to be useful for both crack and corrosion resistance improvement. Another technology is the use of boron composite repair of cracks and corrosion. This technology is in use extensively in the C-130 and C-141 aircraft to extend their useful lives. Protection approaches such as improved coatings and corrosion prevention compounds have found use in both military and commercial aircraft. Many aircraft have used replacement of existing materials to reduce the cost burden of maintaining structural integrity. One example of this is the replacement of the lower wing skins on the KC-135 aircraft. Other successful applications include:

Replacement of the F-16 479 bulkhead with 2097 aluminum lithium to replace existing 2024-T851

Replacement of the 7178-T6 KC-135 wing upper surface with 7075-T73

Replacement of the 7079-T6 C-5A fuselage skins with 7475-T761.

There are many other examples of successful transitions of technology from the laboratory to engineering and manufacturing development. Many of these successes were derived from a well-conceived plan or "road map" that formed the basis or criteria for technology transition. In general, these road maps have included programs directed at several levels of technology maturity. These levels are referred to as basic research, exploratory development, advanced development and manufacturing technology development. Most of the advanced development and manufacturing technology development program effort is directed towards the demonstration of the technology by means of the manufacture and testing of a specific piece of hardware.

A study of those successful road maps for transition of technology to engineering and manufacturing development reveals that that they had certain factors in common. These factors may be combined to form a criterion for the transition process to be successful. The importance of such a criterion for incorporating new structural technologies into an aircraft can be judged from transition experiences that were not successful. There are many examples that could be used to illustrate this. One example that has resulted in tremendous cost in money and productivity is the use of the high strength, low toughness 7XXX-T6 aluminum alloys in the fifties and sixties. The attractiveness of the weight reductions realized through their high strength caused the structural design engineers to utilize these alloys in many product forms and locations in the airframe. Therefore, in many cases they were improperly incorporated in engineering and manufacturing development. This was done without serious consideration of their susceptibility to failure from corrosion, stress corrosion cracking, and manufacturing and service induced defects. An example of use of this material based on strength considerations only is the skin of the lower wing of the KC-135, which was originally manufactured from 7178-T6. The inability to maintain the integrity of this structure in operational use because of fatigue cracking resulted in the replacement of the lower wing skin on more than 700 of these aircraft. The corrosion problems experienced in the 7075-T6 used in the center section of the C-141 wing resulted from inadequate sealing of the structure from the environment. This has caused and will continue to cause significant cost problems with that weapon system. The KC-135 and the C-141 are two of many that have experienced dramatic increases in their maintenance burden because these alloys were used in applications that were not compatible with their characteristics. The material problems are not restricted to aluminum. Improper use of magnesium, titanium, and steel has also resulted in high maintenance costs. In addition, to materials, processes have also experienced deficiencies in the transition criteria from the laboratory environment to engineering and manufacturing development as evidenced by the early manufacturing problems with the F-16 horizontal tail composite parts.

There are also many examples of faulty transition criteria led to serious problems in engines. As with the airframe, there are numerous examples where this has caused problems. One of these is the use of low ductility titanium in the F100 fan blades and fan disks. This usage led to extremely small critical flaw sizes and consequently a costly maintenance burden. This situation was corrected by a redesign. Another example was the use of powdered Rene 95 in the F-101 engine. The database for use of this material was generated from small coupons. However, when the process was scaled up to full size parts, the associated contamination degraded the properties to the point that a substitute material had to be used.

Another reason for establishing a criterion for transition is that there is a need for understanding the limitations on the technology. The laboratory demonstration program is seldom of sufficient scope such that provide confidence that all aspects of a given technology are suitable for operational use. Therefore, there is a danger that the engineer responsible for technology development may use it improperly. There is also the danger of an acceptable application.

It is the intent of this paper to describe the essential features of a criterion for successful transition of a structural technology from the laboratory to engineering and manufacturing development of a production aircraft. It has been demonstrated by service experience that the USAF Aircraft Structural Integrity Program (ASIP) [1] has the elements necessary to ensure that production aircraft developed from these requirements will be safe and economical to operate. The ASIP was initially developed in 1958 [2] to preclude the reoccurrence of some catastrophic failures that took place at that time. It was updated in the early seventies to include the currently used damage tolerance philosophy for structural design. All aircraft used or developed by the Air Force are subject to the requirements of the ASIP.

The ASIP provides the guidance for the engineering and manufacturing development phase of an aircraft. Consequently, its importance for this paper is that a structural technology may be judged against the elements of the ASIP to help determine if it is suitable for transition to this phase. Although the ASIP was conceived before the development of composites, the process may be easily tailored to these structures [3].

**Technology Transition Criteria**

From a study of the successful transitions of structural technologies from the laboratory to an operational aircraft, it was found that five factors constituted a common thread among these successes. In addition, it was found that these five factors were essential to the successful completion of the tasks of the ASIP. These five factors are:

1. Stabilized material and/or material processes

2. Producibility

3. Characterized mechanical properties

4. Predictability of structural performance

5. Supportability

In the listing, there was no attempt to establish a ranking of importance of these factors. A deficiency in any one of the factors could constitute a fatal defect.

It is readily seen that stabilized material and/or material processes are essential. With the time constraint on establishing the final allowables, any significant change in the material or processing could be disastrous. However, it is not expected that all of material and processing specifications be completed at the start of material or process substitution process. It is adequate to have preliminary documentation of the following:

Material qualification and acceptance specifications

Processing specification and acceptance standards

Manufacturing instructions

This factor must address the issue of corrosion. It is expected at the start of the material and substitution process that the corrosion resistance of the material be characterized. In addition, the analyst should establish the requirements for cladding, anodizing, priming and top coating.

Several producibility considerations must be addressed. First, the material supplier must be capable of supplying the material in appropriate quantity and forms. Experience has shown that the time required for scaling up material sizes or changing product forms is generally not compatible with time available to implement the desired change. Even in the case of large, but state of the art, forgings care must be exercised because the properties are extremely configuration dependent.

Another producibility requirement is to use the technology to fabricate detail parts and assemblies. For this purpose, generally full-scale parts are required. This fabrication must cover the range of forming parameters and material heat treat conditions that are appropriate. In rare cases, fabrication of subscale parts could be acceptable if the behavior of the full-scale parts can be confidently predicted.

Inspectability through the manufacturing process is another essential issue in the producibility factor. If conventional methods are appropriate then they must be identified and approximate capability established for the critical locations. If new methods are required then they must have progressed beyond the laboratory stage and have a demonstrated capability to perform the intended inspections.

It must be demonstrated that the manufacturing process is compatible with the shop environment. The government environmental standards that are imposed on the manufacturing facility may have a severe impact on the allowed processes. This problem is aggravated by the fact that these standards are rapidly changing and becoming more severe.

Of all the factors in the transition criteria, characterized mechanical properties is the most difficult to identify the specific requirements. There are three guiding principles that should be used to establish these requirements. One of these is that the final characterization should be complete before the design process is complete. Another is that at the start of the design process the properties should be established with sufficient accuracy to determine the weight of the structure. The final guideline is that property investigation should be extensive enough in scope to preclude the possibility that the technology will fail to reach its intended purpose. In some cases this may require an evaluation of the material or process characteristics when exposed to nuclear and/or non-nuclear threats. The initial emphasis should be on the assessment of many characteristics of the technology rather than an in-depth assessment of any single one of them.

As a minimum, the following mechanical properties must be evaluated in the presence of environmental effects:

Strength - Coupon tests are required for tension, compression, shear, and bearing for yield and ultimate as appropriate

Modulus - Coupon tests as appropriate are required to derive the stress strain relationship

Elongation - Coupon tests as appropriate are required

Fracture Toughness - For metals, $K_{Ic}$, $K_c$ data and R curve data are required as needed for establishing design stresses

Crack Growth Rate - For metals, data required for range of values of R expected in operational aircraft

Dimensional stability - includes data for potential creep effects to be expected in operational aircraft and thermal coefficient of expansion data

Stress Corrosion Cracking - Either $K_{Iscc}$ data or stress

Threshold data required for thickness and product form combinations expected to be found in the operational aircraft

For many technologies, it is not adequate to base the success of a technology on coupon data. For example, in many composite structures the interlaminar stresses produce critical failure modes that can not be confidently interrogated in the coupon level tests. It is expected that appropriate element and subcomponent tests (i.e., building block tests) be performed. Generally, strength, durability, and damage tolerance testing will be performed in these building block tests.

The three guidelines stated above are believed to be adequate to determine the quantity of data needed. However, some additional guidance may be useful. For the coupon strength tests, a minimum of four tests from each of four material lots would satisfy the requirement for transition. This database would not be adequate for the generation of allowables as recommended in MIL-HDBK-5. However, the database should be adequate for establishing the Weibull parameters and making usable preliminary estimates of the allowables. For fracture toughness, the contractor should supply metal materials with fracture toughness with a minimum guaranteed. Therefore, the tests should demonstrate that there is a balance between structural performance and economics of procurement. For rates of crack growth in metal structures (i.e., da/dn data) average properties are acceptable. However, multiple tests are needed to estimate the variance in this data. If the analyst judges these variances as excessive, then there should be a reassessment of the material and process specifications to try to reduce the scatter in this property.

Prediction of structural performance is an important factor because the full-scale static, durability, and damage tolerance tests are very expensive. Therefore, there should be reasonable expectation that the analyses and design development tests preceding these tests would lead to success. Further, and even more important, is that a structure developed under the ASIP will provide safe and economic operation of production aircraft. Generally, the predictability of structural performance can be validated with a subcomponent test. That is, a section of a wing or fuselage etc. would be adequate. These tests must be carefully designed to ensure that all potential critical load paths are assessed.

Usually the key to satisfying the predictability factor for structural performance is sound analytical procedures. This is the hallmark of currently used metallic materials. For these materials, critical strength failure modes in tension, compression, shear, and bearing are predictable. Further, the availability of crack growth laws for metals permits the damage tolerance capability to be established. For composite structures, the analytical procedures are still emerging. Difficulties persist in the prediction of interlaminar stresses and the growth of delaminations derived from impact damage. However, these problems can be handled with empirical methods. When empirical processes can be used confidently then this factor can be satisfied.

The damage tolerance era, which started in the early seventies, has helped to reduce some of the supportability requirements for currently designed aircraft. However, the supportability factor is still a major consideration because of accidental and battle damage. Any new structural concept forces the maintainers and the operators to prepare to make repairs. This places demands on supply lines and training of personnel. The establishment of facilities with trained personnel normally lags the development of the technology significantly. For example, the composite repair capability for in-service aircraft is only now maturing but the composite technology has been transitioned to operational aircraft for approximately two decades. Further, the acceptance of a new technology by the maintainers and operators may place a large economic burden on them. This could be difficult to justify. Consequently as a prerequisite for transition, the repairs associated with the technology must be at least conceptually developed. That is, the repair must be viable in the using command environment and must be manageable from economic considerations.

In addition to the repair aspect of supportability, inspectability must be addressed. This includes both manufacturing and in-service inspections. If the inspections required are not viable with existing equipment then there must be reasonable assurance that these inspections can be made when they are needed.

**Example of Successful Transition of Technology**

Previous and current inspection programs have revealed that the vertical tail attachment pads in the F-16 Fuselage Station 479 bulkhead are significantly cracked in the majority of early aircraft. Figure 1 shows the configuration of the bulkhead and Figure 2 shows the area where cracking has been experienced. The depot responsible for F-16 maintenance has considerable experience in the eddy current inspection for these cracks and have established appropriate guidelines for the disposition of cracks found in operational aircraft. The root cause of this cracking is high stress concentration inherent in the detail design compounded by increased usage severity precipitated by weight increases in the aircraft without compensating structural modifications. The stresses at the point of crack initiation are extremely high although they do decrease with increasing crack depth. These high stresses cause early cracking in the structure.

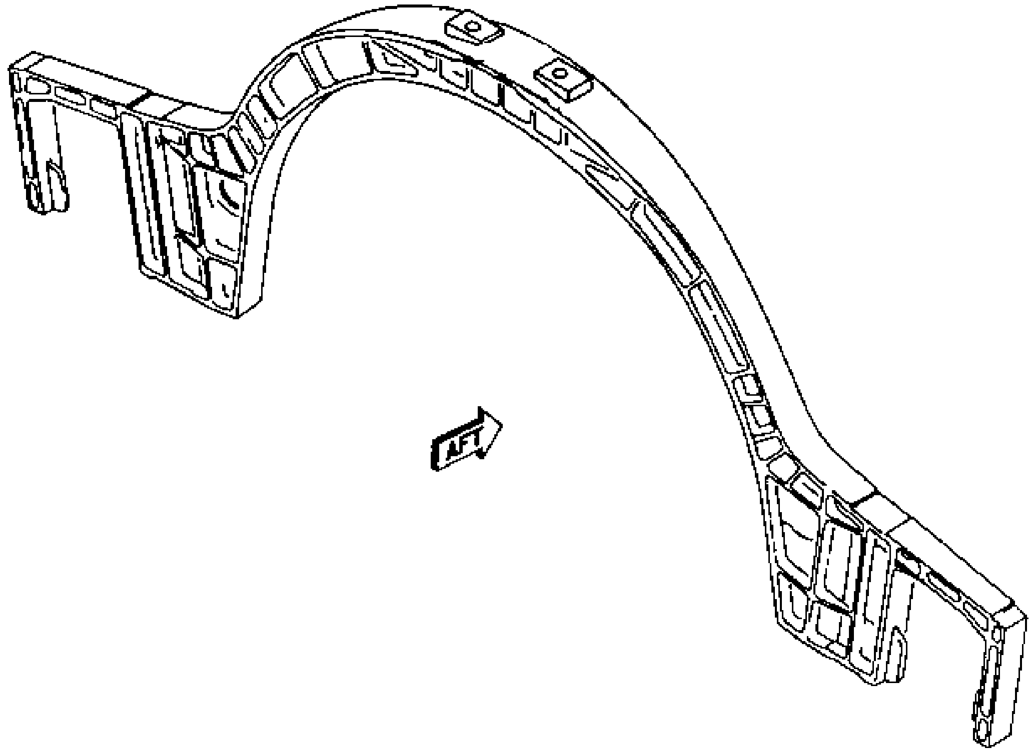

**Figure 1 The Fuselage Station 479 Vertical Tail Attachment Bulkhead**

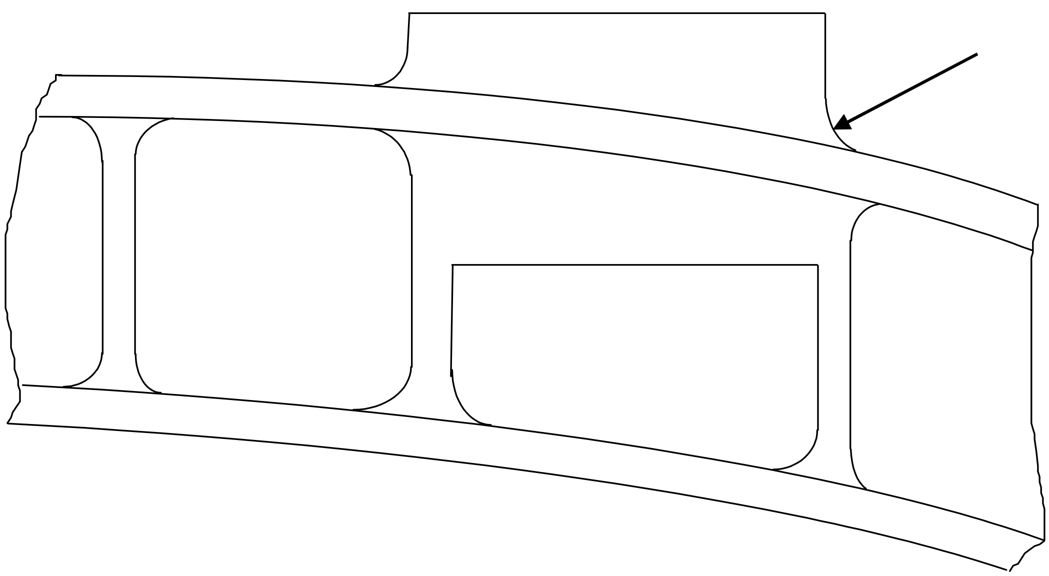

**Figure 2  The Fuselage Station 479 Bulkhead Area of Cracking**

Normally, the USAF would not fly these aircraft with known cracking.  In this situation the situation is relieved because the vertical tail structure has been analytically shown to have adequate fail-safety in the event of a failure of the F.S. 479 bulkhead.

Preliminary studies showed that the fracture toughness of aluminum lithium was high enough such that a crack would grow through stable propagation through the high stress concentration region. This feature enabled the bulkhead to remain in service considerably longer than the baseline bulkhead. However, it would be expected to have crack indications early in its life because of the inherent high stress at the radius. Its structural integrity would be ensured through fail-safety.

Consequently, these studies indicated that the use of a new bulkhead material identified as 2097 aluminum lithium would provide considerable improvement in the structural integrity in this location. The original material was 2024-T851 aluminum. The USAF requested the contractor to develop a program around the five elements of technology transition for a new material or process. These five elements are; (1) stabilized material and processes, (2) producibility, (3) characterized mechanical properties, (4), prediction of structural performance, and (5) supportability.

The 2097 material satisfies the first element - stabilized material and material processes. The material specification has been finalized and significant amounts of the material have been produced to this specification. All of the replacement components will be procured to this specification. The boundaries of the processing parameters have been examined and no significant degradation of properties has been observed.

The second element - producibility has been satisfied in that full-scale parts have been fabricated with the same tools that were used to fabricate the conventional aluminum parts. There have been no toxicity issues found for the handling and machining of the forged part. Also, the scrap disposal is feasible without contamination of the conventional alloys.

The third element - characterized mechanical properties was completed. Since the final strength properties had not been characterized to the MIL-HDBK-5 standards, the design team used S-values for strength characterization. The strength properties that are significant to this application for the 2097 material exceed the strength of the baseline conventional aluminum. The fracture properties are well characterized for the "long crack" analyses that would be used for this bulkhead. The corrosion resistance has been successfully demonstrated through the EXCO test procedure. Also, the stress corrosion cracking thresholds are in excess of the baseline material, which has performed well with respect to this issue.

The fourth element - predictability of structural performance had to be satisfied by a combined analysis and test program. Conventional linear fracture mechanics approaches were not able to completely characterize the growth of small cracks. The problem was that the very high stress concentration in the area of cracking increased the strains beyond the yield point. Therefore, the contractor used component testing to augment the analytical results to ensure that the new material was significantly better than the baseline material.

The fifth element - supportability was found to be adequate. Aluminum lithium has the characteristic that the fracture face is quite rough compared with the baseline material. However, this should not be an inhibitor to making an inspection for cracks in operational aircraft. This was demonstrated through component testing.

Therefore, the team recommends that the aluminum lithium material be used for the F.S. 479 bulkhead replacement.

**Conclusions**

This paper has discussed the five factors believed to be essential for successful transition of a technology from the laboratory to successful application on operational aircraft.

Experience has shown that deficiencies in these factors will result in a less than satisfactory operational performance. It is evident that there is considerable judgment involved in addressing each of these factors. However, it is believed that the effort required in making this judgment is justified when the consequences of using a deficient technology are considered.

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| Loads (forces) | Aircraft maintenance | Inspection | Aging aircraft |
| Aging (metallurgy) | Service life | Reliability | Transitions |
| Airframes | Equipment health monitoring | Airworthiness | Process technology |


**14. Abstract**

Aging Aircraft concerns have dramatically escalated in the military community and commercial aviation during the past decade. Some models, which have already been in service for more than 40 years, will need to be retained for another two decades or longer, often serving in roles and in theatres very different from what was envisioned when they were originally designed. Aging Aircraft has several connotations. To name a few: technological obsolescence, the spectre of runaway maintenance costs, and safety. Moreover, spare parts, processes and tooling may no longer be available, logistic procedures may have changed and suppliers may be out of the business. Budgetary limitations and higher fleet utilisation will increase the demand to cope with aging structures and major subsystems like engines and avionics.

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