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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.

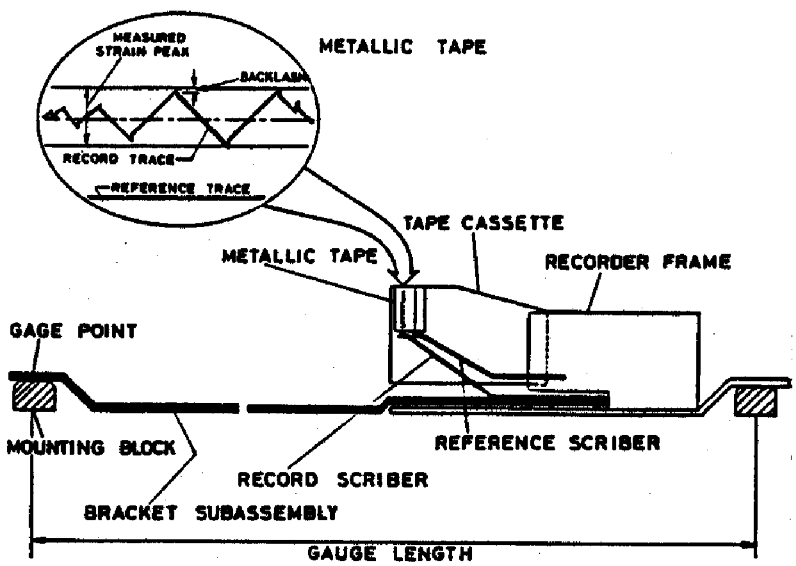


Fig. 1.1-1 Princip of Mechanical Strain Recorder

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.

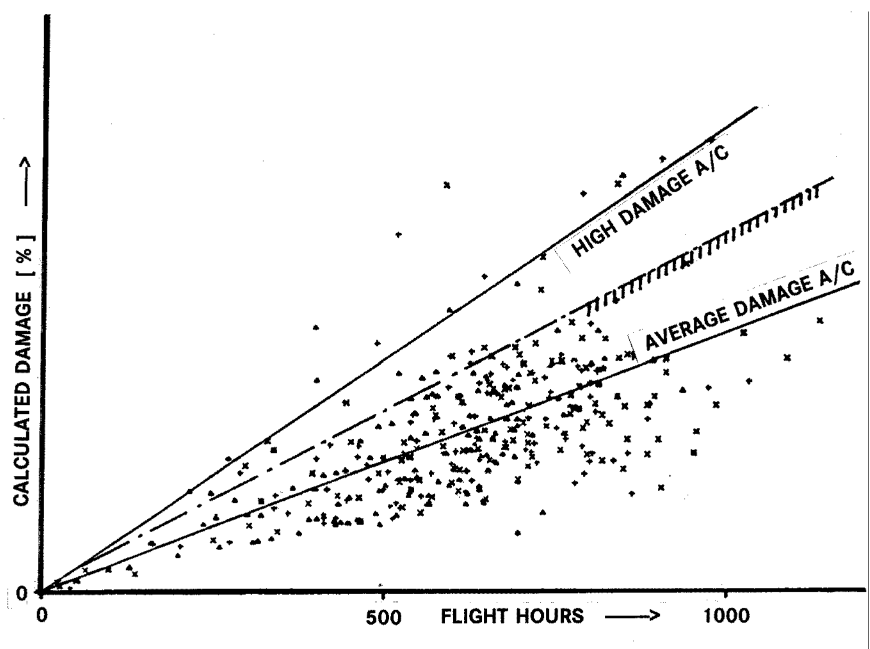


Fig. 1.2.1-1 Lower Wing Skin Life Consumption for Similar Missions, TOR

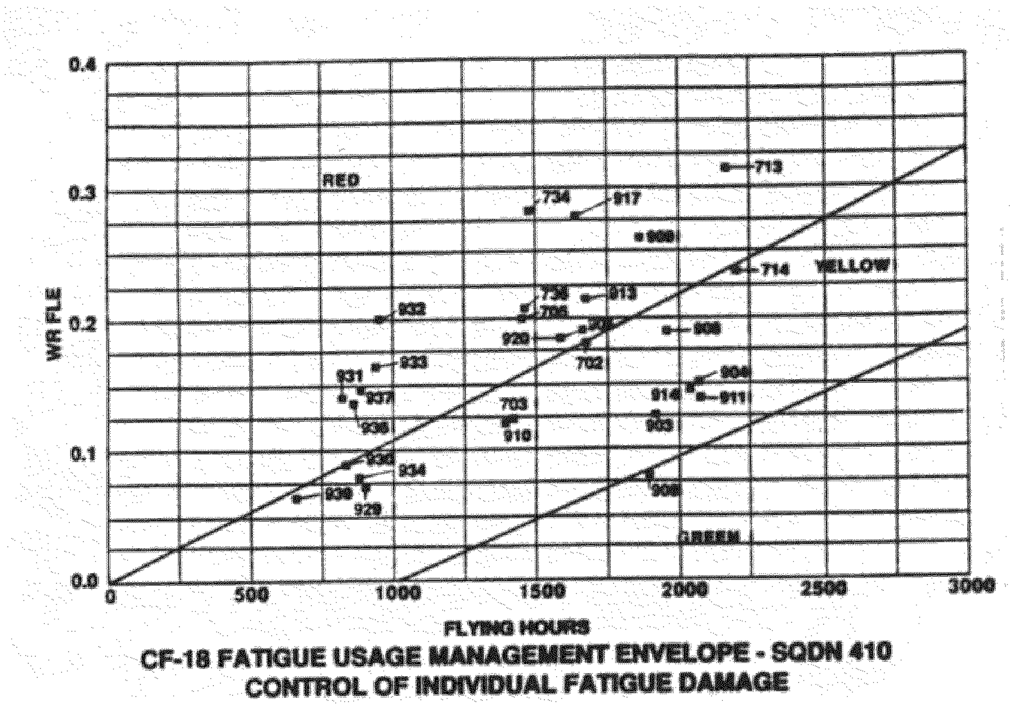


Fig. 1.2.1-2 Wing Root Bending Life Consumption, CF-18

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.

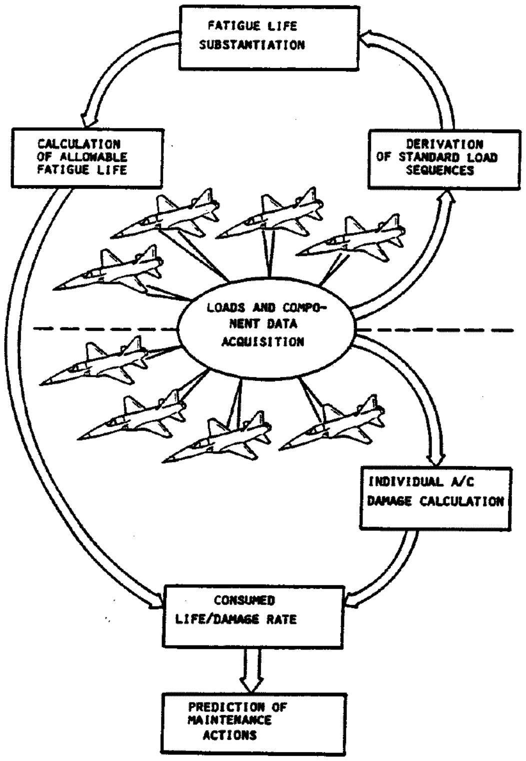


Fig. 1.2.2-1 Structural Monitoring Activities

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.

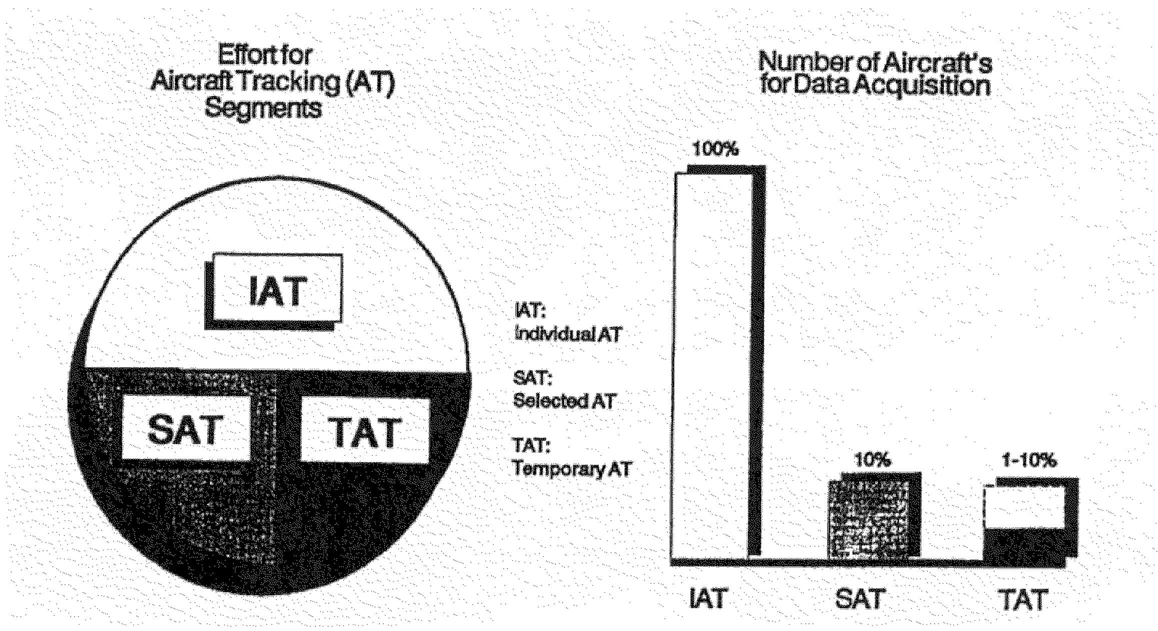


Fig. 1.2.3-1 Aircraft Tracking Segments, TOR

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.

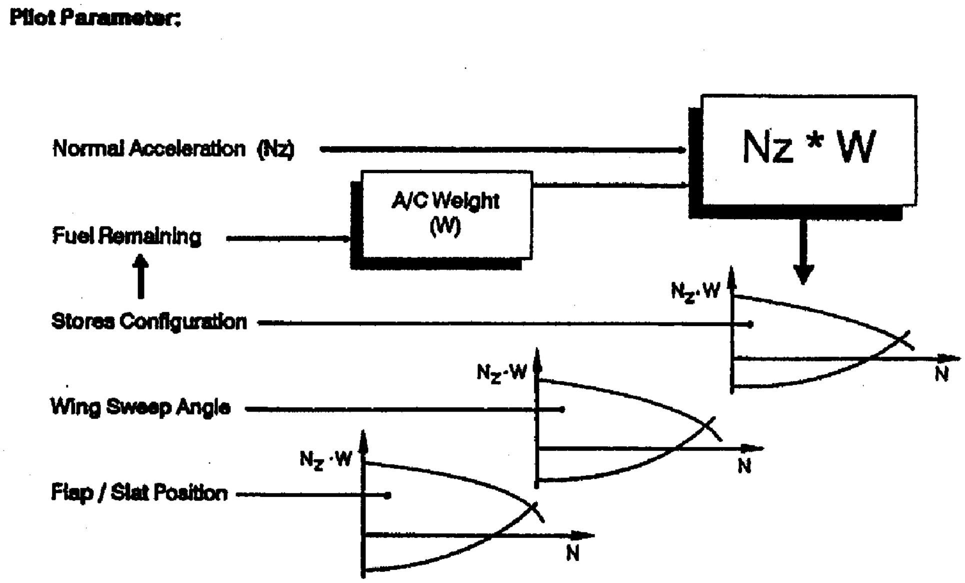


Fig. 1.2.3-2 Reduced Parameter Set (PPS) for IAT

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.

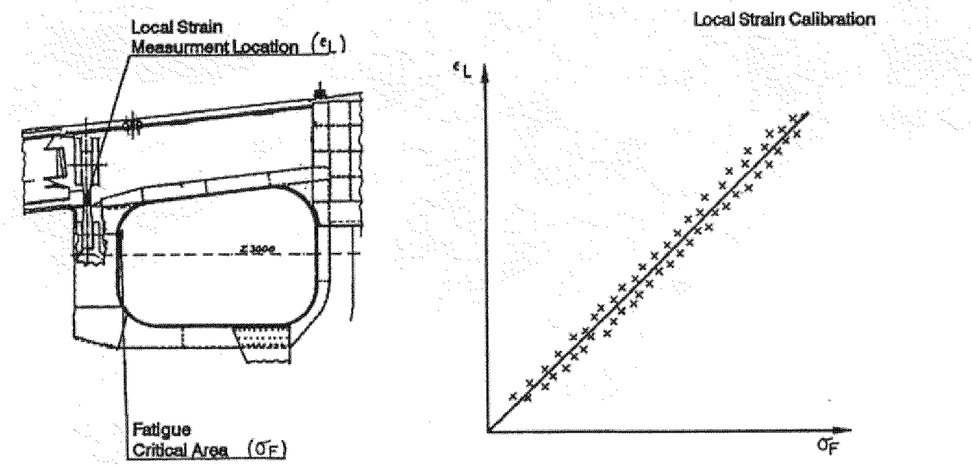


Fig. 1.2.3.-4 Reference Strain Gage on Wing Attachment

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.

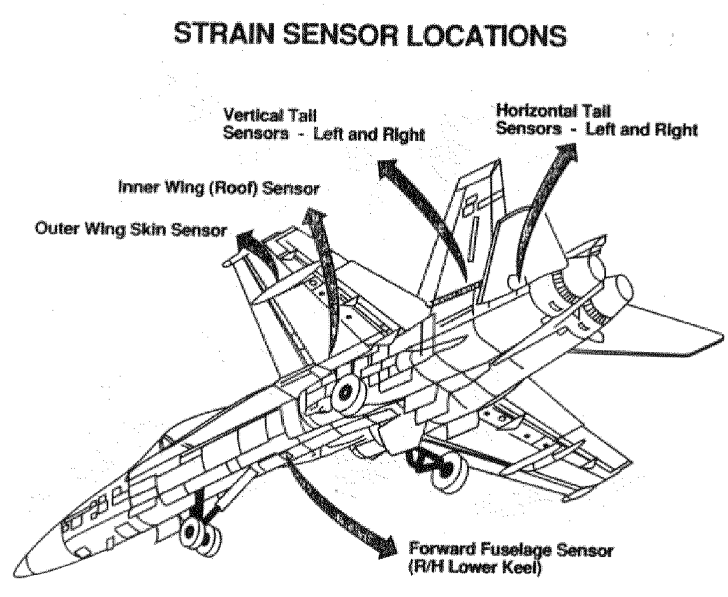


Fig. 1.2.4-1 CF-18 Strain Gage Locations

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.

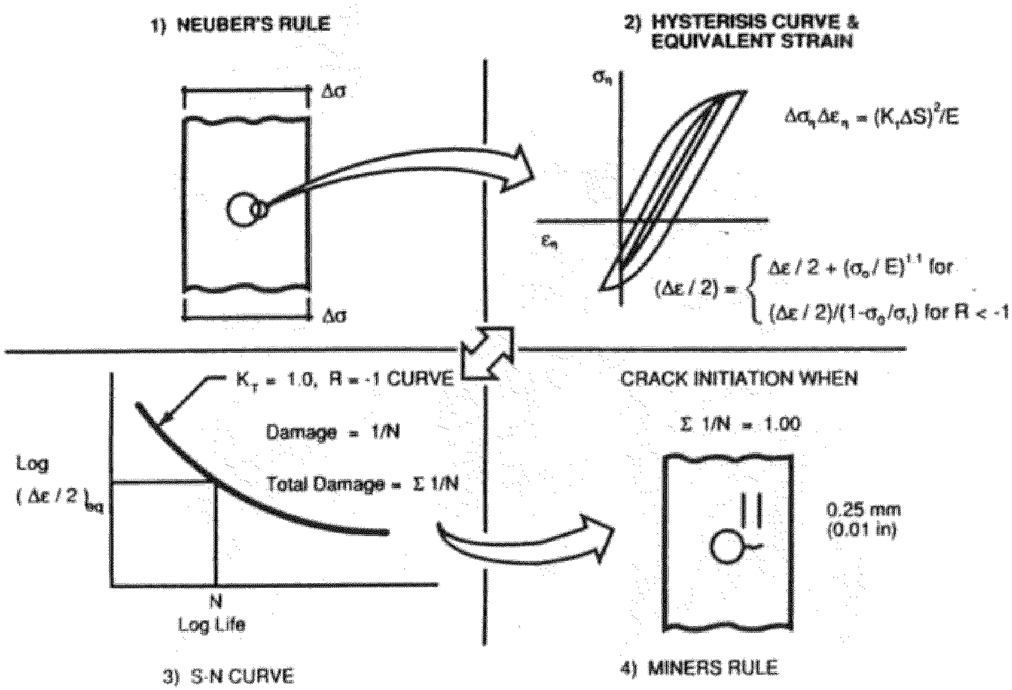


Fig. 1.2.4-2 Crack Initiation Concept

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

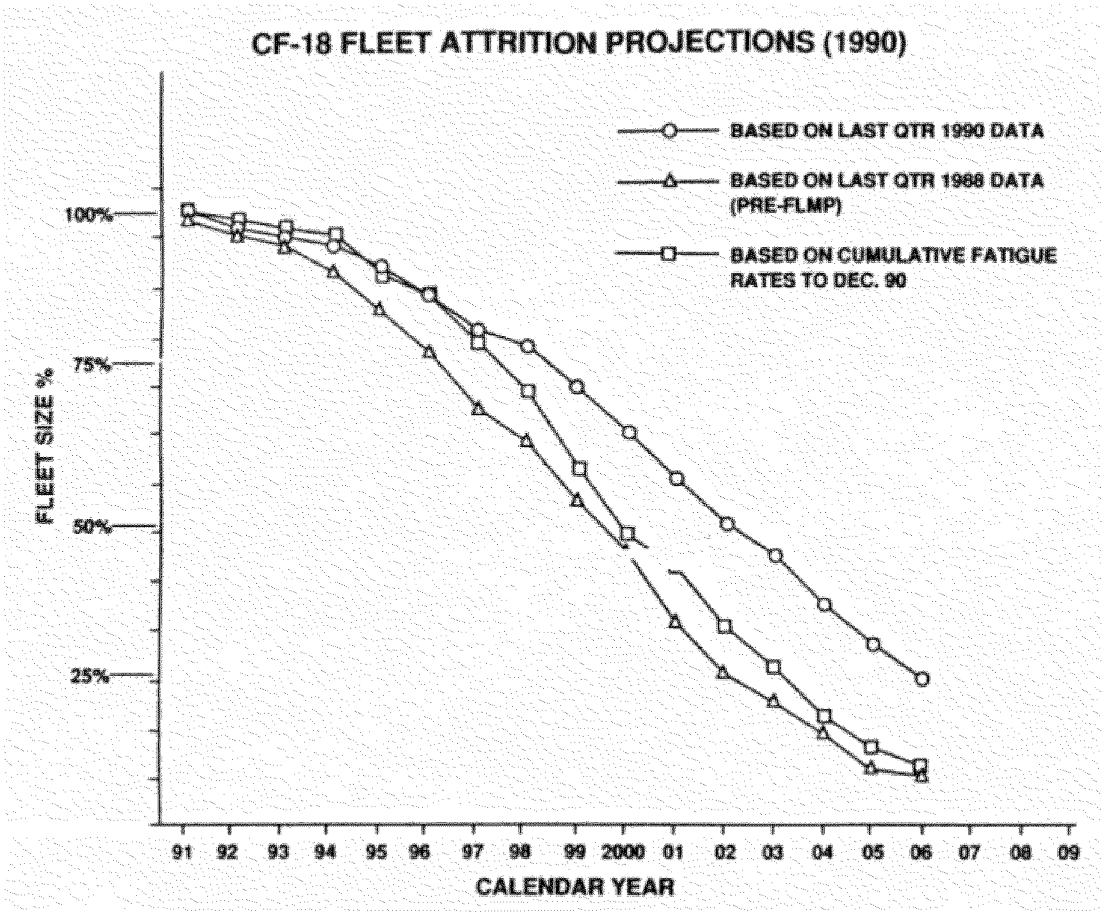


Fig. 1.2.4-3 Life Improvement of CF-18 Fleet

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.

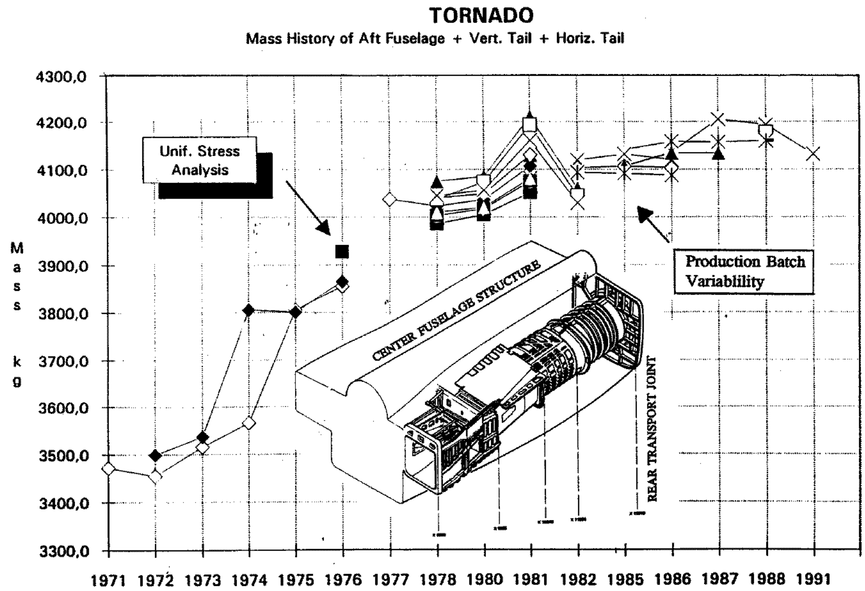


Fig. 2.0-1 Historic Structural Mass Increase of TOR Aft Fuselage

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.

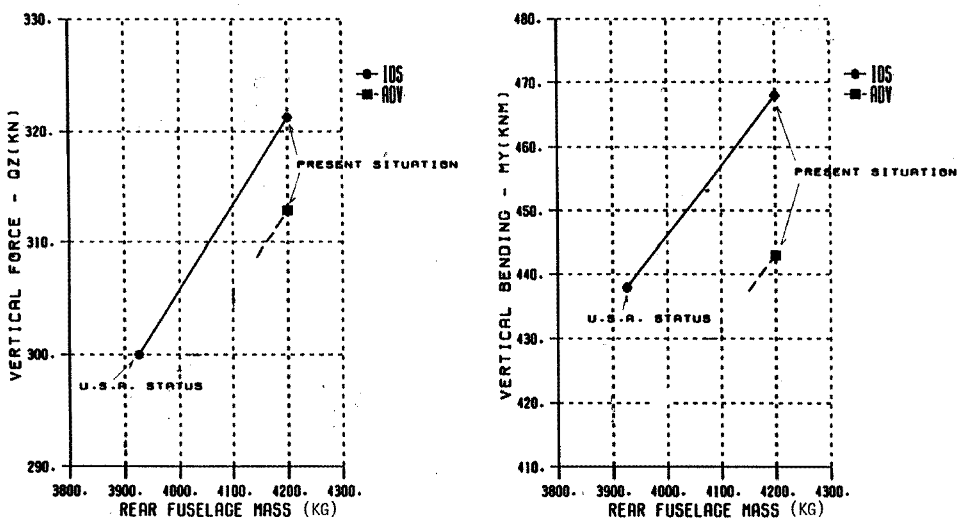


Fig. 2.0-2 Interface Load Increase at Rear Transport Joint

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.

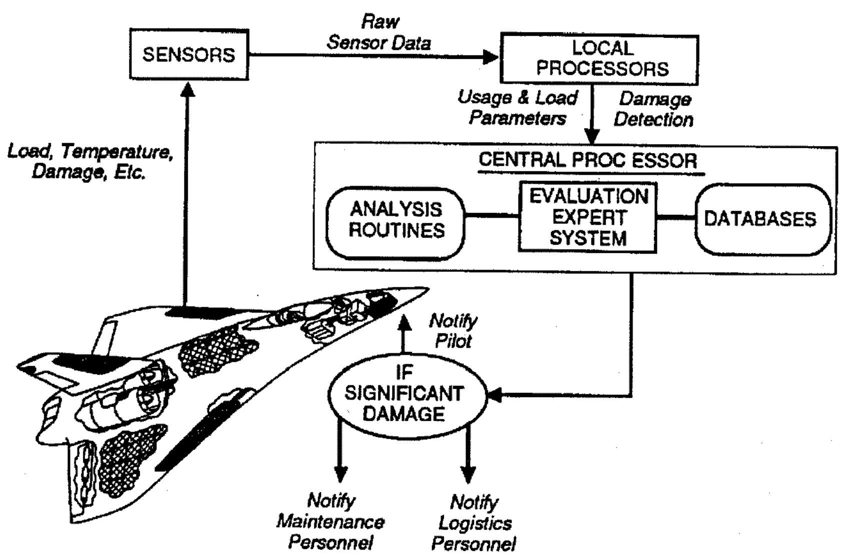


Fig. 3.1-1 Schematic overview of HUMS architecture

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.

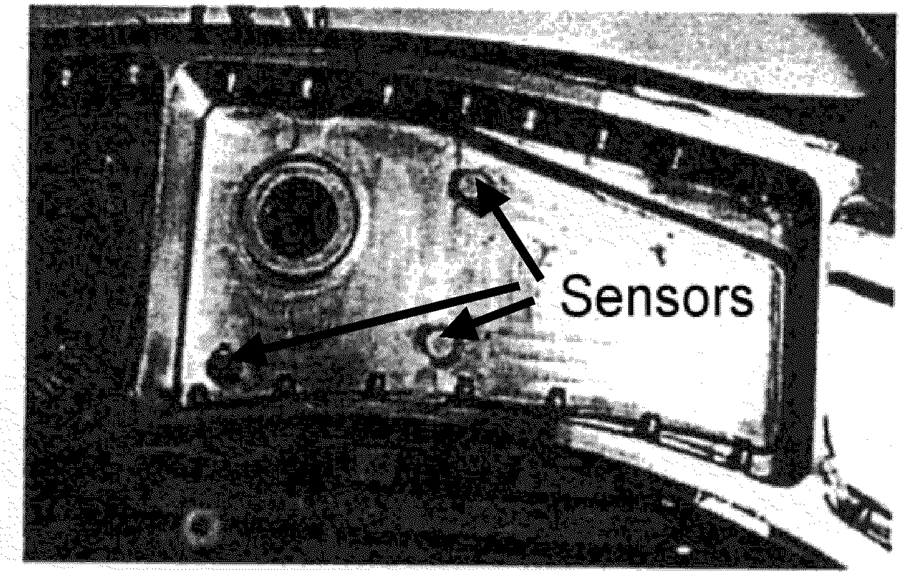


Fig. 3.3-1 Acoustic emission sensors in web area of frame

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.

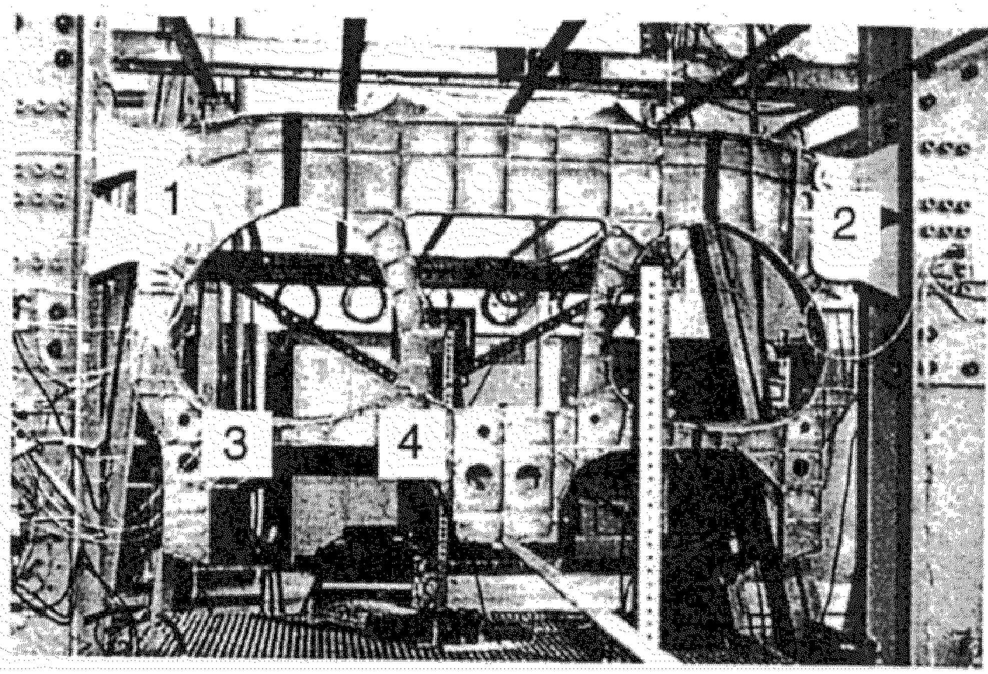
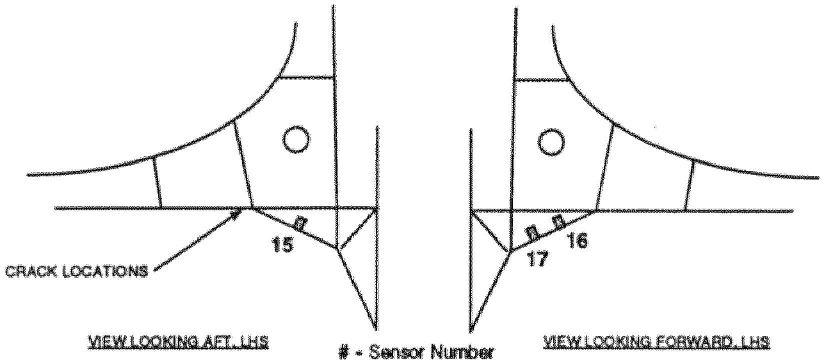
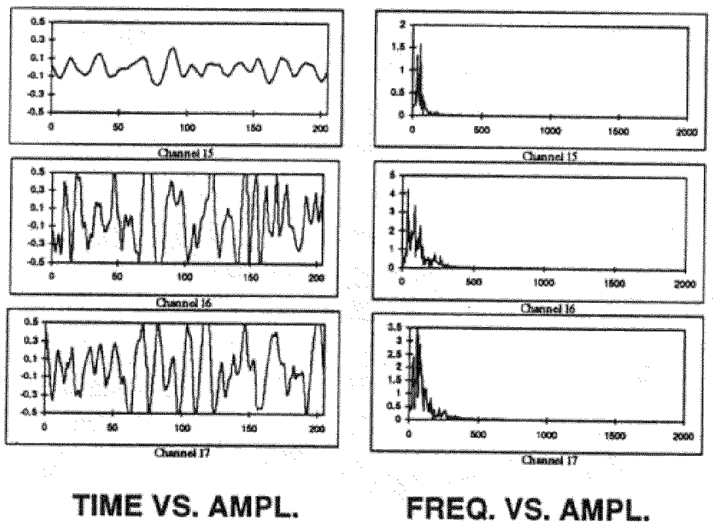


Fig. 3.3.-2 Full scale test article with monitoring loactions

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.





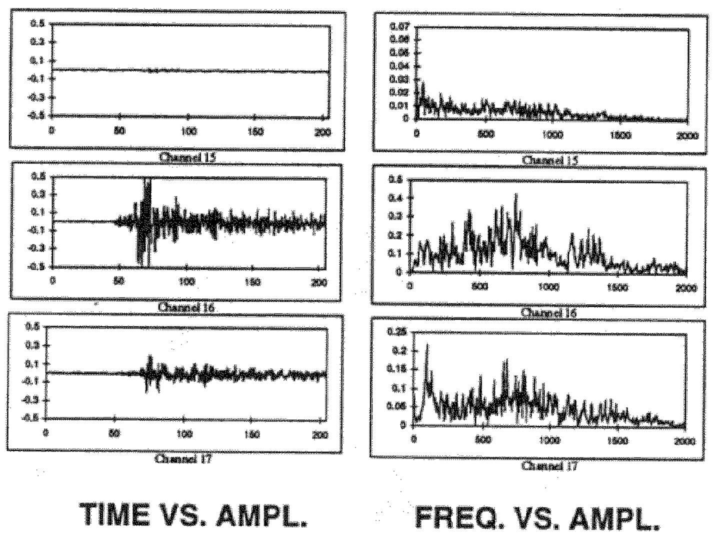


Fig. 3.3-3 Zone 4 sensor location and results

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.

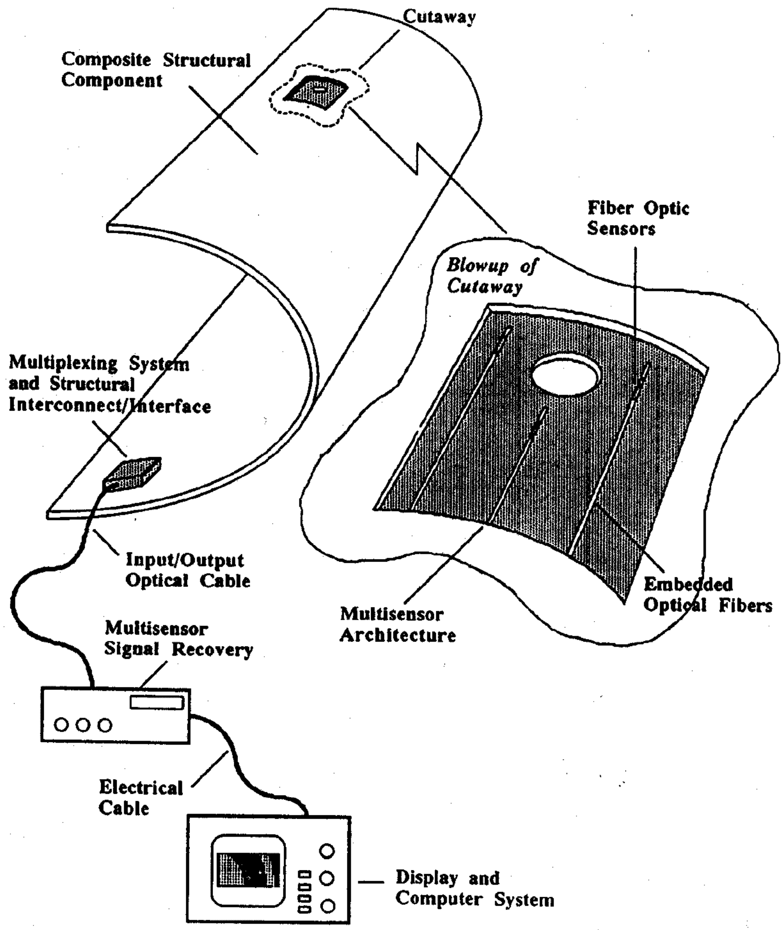


Fig. 3.3-4 Fibre Optic monitoring array embedded in structure

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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