Load Monitoring and Structural Health Monitoring (SHM) within the Royal Netherlands Air Force

Customer

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EXECUTIVE SUMMARY

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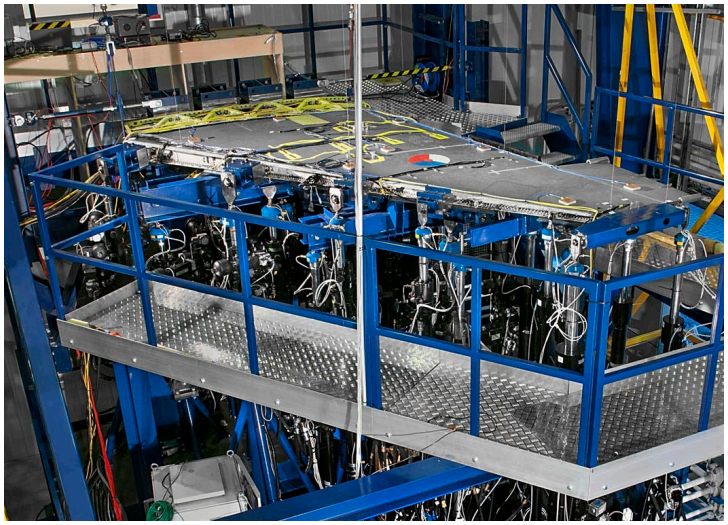

Description of work

In-flight aircraft structural load monitoring and off-board structural health monitoring (SHM) are integral parts of the Royal Netherlands Air Force (RNLAF) fleet life management process. So far the RNLAF has relied on the more traditional technologies like electrical resistance strain gauges for hot-spot monitoring and non-destructive inspection techniques such as eddy current sensors for structural damage detection. Recently however, following the introduction of the latest generation of aircraft, the focus has started to shift to new and emerging technologies that enable on-board SHM and advanced load monitoring. Within this context the RNLAF has tasked NLR to develop and evaluate novel technologies to further improve safety and availability of the fleet.

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Currently, enhanced load monitoring by means of optical fibers in the aircraft structure is being explored. For this purpose use is made of the F-16 Block 15 wing full-scale durability test that is currently being conducted at NLR. Furthermore, technologies which can detect and localize damage in a (composite or metal) structure are evaluated. This paper summarizes recent developments within NLR on these topics.

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Summary

In-flight aircraft structural load monitoring and off-board structural health monitoring (SHM) are integral parts of the Royal Netherlands Air Force (RNLAF) fleet life management process. So far the RNLAF has relied on the more traditional technologies like electrical resistance strain gauges for hot-spot monitoring and non-destructive inspection techniques such as eddy current sensors for structural damage detection. Recently however, following the introduction of the latest generation of aircraft, the focus has started to shift to new and emerging technologies that enable on-board SHM and advanced load monitoring. Within this context the RNLAF has tasked NLR to develop and evaluate novel technologies to further improve safety and availability of the fleet. Currently, enhanced load monitoring by means of optical fibers in the aircraft structure is being explored. For this purpose use is made of the F-16 Block 15 wing full-scale durability test that is currently being conducted at NLR. Furthermore, technologies which can detect and localize damage in a (composite or metal) structure are evaluated. This paper summarizes recent developments within NLR on these topics. Additionally, an affordable Fleet Life Management approach, called the stethoscope method, is discussed. The storage of measured flight and loads data on a day-to-day basis will allow detailed analyses that are beyond the scope of traditional loads & usage monitoring programs.

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Content

Abbreviations

Acronym	Description
ADD	Agency for Defense Development (Rep. of Korea)
ANN	Artificial Neural Network
CBM	Condition Based Maintenance
CCD	Charge-Coupled Device
cgFEM	coarse grid Finite Element Model
CVM	Comparative Vacuum Monitoring
DADTA	Durability and Damage Tolerance Assessment
DI	Damage Indicator
DNW	German-Dutch Wind tunnels
FACH	Chilean Air Force
FBG	Fiber Bragg Grating
FDR	Flight Data Recorder
FE	Finite Element
FLM	Fleet Life Management
FSW	Friction Stir Welded
FTC	Fatigue Test Campaign
LHS	Left-Hand Side
LM	Lockheed Martin
MSE	Modal Strain Energy
NDI	Non-Destructive Inspection
NLR	National Aerospace Laboratory NLR
OEM	Original Equipment Manufacturer
OLM	Operational Loads Monitoring
POD	Probability of Detection
R&D	Research and Development
RNLAF	Royal Netherlands Air Force
SDR	Structural Data Recorder
SHM	Structural Health Monitoring
SNR	Signal-to-Noise Ratio
SLED	Super-luminescent Light Emitting Diode
TAPAS	Thermoplastic Affordable Primary Aircraft Structure

1 Introduction

Aircraft require regular costly inspections to guarantee their safety, which currently relies mainly on manual Non-Destructive Inspection (NDI) methods. During the last decade, much research has been dedicated to automated Structural Health Monitoring (SHM) systems. SHM technology aims to achieve long-term and in-service monitoring of the condition and damage state of vehicle systems in operation using advanced sensors that are permanently attached to the structure. In-flight aircraft structural load monitoring and off-board SHM are an integral part of the Royal Netherlands Air Force (RNLAF) fleet life management process, and NLR envisions that SHM will complement current NDI techniques in the mid-term future. In the long-term future, SHM techniques are expected to replace classic NDI. Furthermore, SHM, complemented with advanced load monitoring techniques, is a key element in introducing Condition Based Maintenance (CBM), which will bring along a reduced cost of ownership and an improved system operational availability while maintaining current safety levels. Additionally, SHM techniques can solve problems of poor accessibility and remove the human factor of inspection.

The following sections provide examples of SHM and advanced load monitoring research projects that have recently been performed or are being performed at NLR. Chapter 2 describes the current activities with regard to in-flight damage detection and localization. These activities encompass the development of a SHM system design tool and the evaluation of a specific system that is based on the comparative vacuum monitoring technique. Chapter 3 describes the current activities with regard to advanced flight loads monitoring. They include the demonstration of an optical fiber network on a full-scale fatigue test of a F-16 wing. In addition the so-called “stethoscope method” is described; it is the key to an affordable fleet life management approach.

2 Damage detection and localization

Typically, an SHM system consists of a network of sensors to detect changes in the physical and/or geometric properties of a structure from data gathered at two different states, an initial reference state, considered as the undamaged state, and the current damaged state. Changes can be caused by damage present in the structure. Especially in composite structures damage such as debonding of stringers or impact damage may easily go undetected by visual inspection. SHM techniques can be operated on-line

during the flight or off-line on the ground and can be focused on the global inspection of large areas or on the local inspection of highly critical areas (hot spots).

The main objectives of SHM are to reduce the cost of ownership and to improve the system operational availability. For this, various types of sensors exist to monitor the condition of the structure and to timely signal that damage is present. Examples are acoustic and ultrasonic sensors (mid to high frequency range) and optical fibres with Fibre Bragg Gratings (FBG, low frequency range). The use of optical fibres with FBGs (instead of using conventional electrical resistance strain gauges) offers a number of appealing advantages for application in aircraft structures, such as light weight, tolerance for harsh environments, long term stability, complete passivity and no interference with other signals. Section 2.1 presents a design tool for such a system with which the required number and position of the FBGs can be determined to enable the global detection of damage in a generic composite structure. It includes a model assisted approach to determine the Probability of Detection (POD) curve, which is demonstrated for a stiffened composite panel. Additionally, section 2.2 presents the evaluation of a simple hot spot SHM technology called Comparative Vacuum Monitoring (CVM).

2.1 SHM System Design Tool

SHM systems can reduce the cost of ownership and improve the operational availability of a structural system. However, it remains a difficult task to determine the number and position of sensors to be able to detect damage. Therefore, a SHM system design tool has been developed with which the number and position of strain sensors can be determined for a general composite or metallic structure to enable the detection of damage. The network of sensors is used to detect changes in the response of a structure from data gathered at two different states, an initial reference state and the current damaged state. The damage detection algorithm applied is based on a modal approach and is able to detect the presence and location of the damage. The SHM design tool is highly automated and allows for automatic damage insertion in the Abaqus finite element model, which is a requirement for a fast design. Damage indicator plots are automatically generated [1].

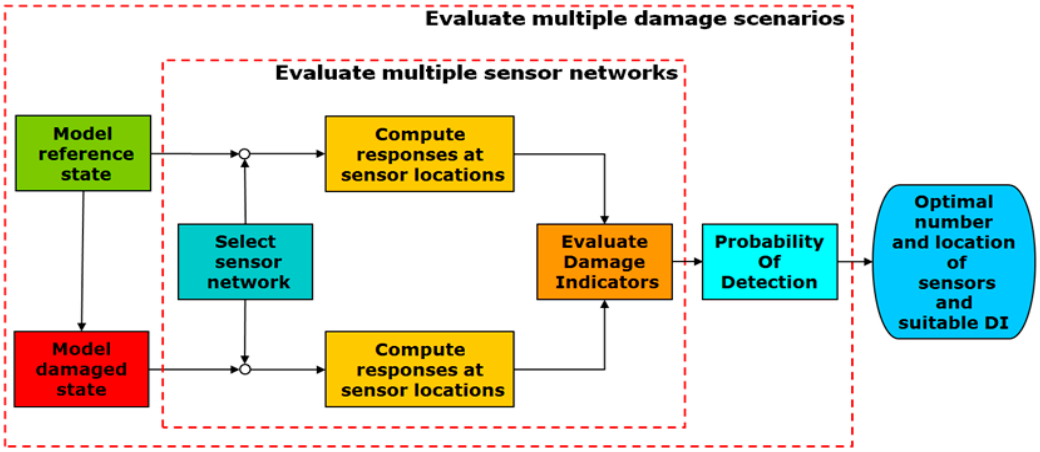


Figure 1: Flow diagram of the SHM design tool.

A flow diagram of the damage detection algorithm is depicted in Figure 1. Since the FBG sensor measures the strain in the direction in which it is oriented, only limited information about the modal behaviour of the structure is available in reality. Based on the strain response measured by the sensor network in the initial and current state, a number of damage indicators (DI) can be determined. Examples are:

- Changes in natural frequencies
- Changes in mode shapes
- Changes in modal assurance criterion
- Changes in Modal Strain Energy (MSE)

Other research already pointed to the suitability of the MSE criterion for SHM. Stubbs [2] has formulated the MSE damage indicator and successfully applied it to a steel bridge using a beam formulation. Cornwell et. al. [3] has expanded the applicability of the MSE function to a plate structure. Grooteman [4] demonstrated that the MSE criterion is sensitive to small changes in the stiffness of the structure, making it a suitable indicator to detect small damages. Furthermore, the damage location is directly provided as well. This criterion has therefore been adopted as a DI in the SHM design tool.

The SHM design tool has been demonstrated on the composite fiber reinforced thermoplastic panel depicted in Figure 2 (see [5]). This panel has been developed by Fokker Aerostructures in the TAPAS (Thermoplastic Affordable Primary Aircraft Structure) project. It consisted of a tailored skin with thickness varying from 8.14 mm (59 layers) to 5.24 mm (38 layers). Sensor networks have been analyzed consisting of three, four and five optical fibers times eight FBG sensors per fiber, different FBG orientations (0º, 90º, ply direction), surface mounted or embedded at different thickness fractions or plies. Furthermore, eight different damage scenarios have been considered, such as stringer

debonding and skin impact damages of different sizes (barely visible to visible). Especially disbonds are hard to detect with conventional non-destructive inspection methods.

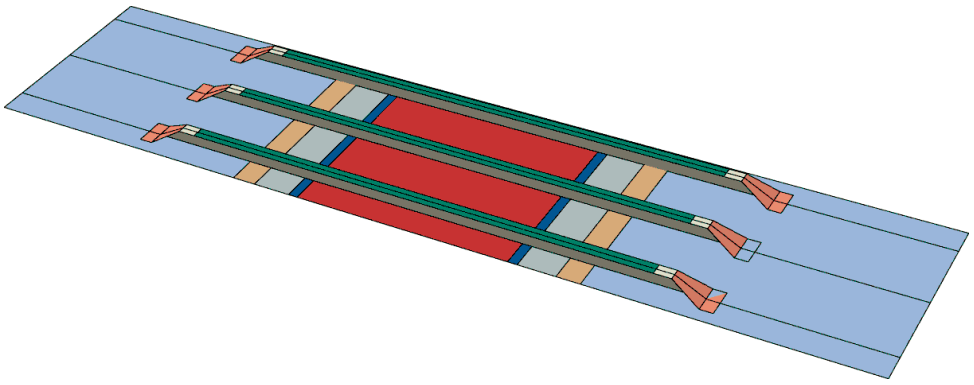


Figure 2: FE model of three-stringer thermoplastic composite TAPAS panel. Different colors indicate different layups.

Figure 3 shows the automatically generated output results for a FBG sensor network consisting of three optical fibers with eight FBGs, each running along the stringers; the MSE-based DI is shown for a number of different damage scenarios. The stringers have been left out for clarity. The dark grey area indicates the debonded area and the blue colored elements denote the location of the sensors inside the element. In all cases the SHM system was capable to correctly detect and localize the damage. Some other nearby sensor may show some response (green color) as well. These lower insignificant values (e.g. z < 2) can be filtered out of the plots to prevent a wrong interpretation by the operator.

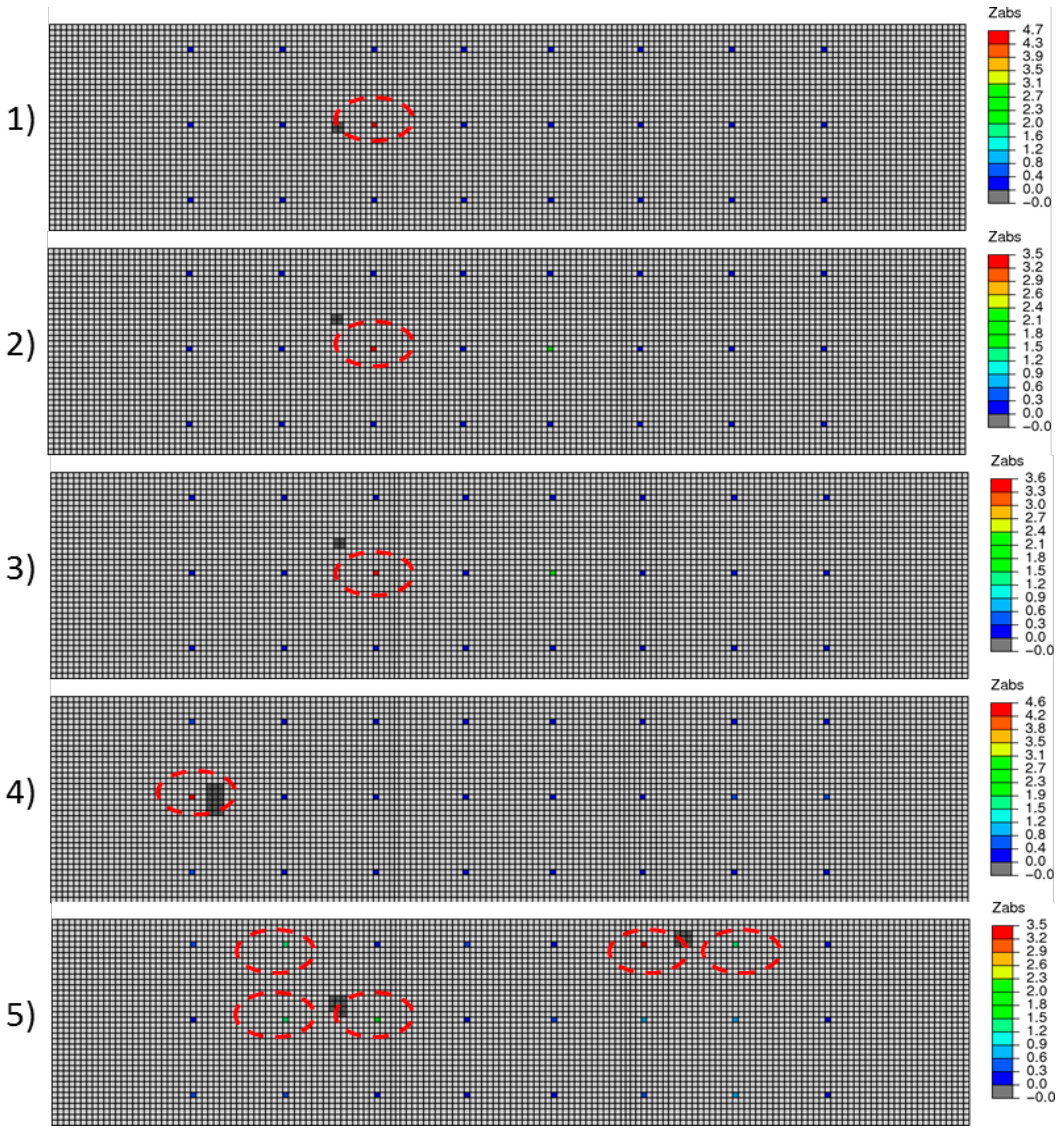


Figure 3: MSE DI for 1) stringer debond, 2) single impact, 3) single impact after initial stringer runout debond, 4) stringer runout after initial single impact, 5) double impact.

Although the numerical results showed that even small damages can be detected with a relatively coarse sensor network, it is clear that in a realistic application noise will be present in the measurement signal, resulting in errors. A noise component can be added to the computed responses to simulate a Signal-to-Noise Ratio (SNR) present in a realistic application. This in combination with randomly generated damages, in location and size, allows the computation of the detection capability of the SHM system, expressed in terms of the Probability of Detection (POD). The POD reflects the chance to detect a damage of a certain size within the structure, which is an important quantity for the certification of the SHM system.

In manual NDI methods the operator plays an important role in the accuracy of the system, i.e. in the chance of finding a damage of certain size. This depends on the training level, experience and alertness of the inspector. Damages can be found or missed; based on a population of damage sizes found and missed, a POD distribution function can be determined. Constructing such a statistical data set is expensive. In case of a SHM system the same principle can be applied. However, the human factor is no longer present. Missed cracks are only due to the capability of the system in finding a damage of certain size at a certain location. For instance, a damage located away from a sensor will in general be harder to detect than a nearby damage. A SHM system consists of a network of sensors and signal processing capability designed for a specific structure. To determine the detection capability, the complete structure now has to be manufactured instead of a representative part in case of a manual NDI system. Furthermore, for each damage location a new structure has to be manufactured in which the damage size can be gradually increased to generate different size data. Experimental validation of the detection capability of a SHM system is therefore very expensive. On the other hand in the absence of the uncertain human factor, which is hard to model, the detectability can to a large extent be computed. Such a model assisted approach, in which the damage detection is simulated, can alleviate the costs significantly and only requires a limited amount of experimental data for validation of the numerical analyses.

A probabilistic framework has been set-up to determine the POD, as indicated in Figure 1. For a given sensor network, damages of random size and location are generated throughout the structure. For the latter a predefined area can be specified where damages are expected to occur. A noise component is added to the computed responses ε_{sim} to simulate a SNR present in a realistic application according to:

$$\varepsilon = \varepsilon_{sim}\left(1 + random.Gauss(0., \frac{1}{SNR})\right) \qquad (1)$$

The noise component is assumed to have a standard normal distribution. Other sources of uncertainty can be easily added as well. The result of the probabilistic simulation is a similar data set of found and missed damage sizes as in the case of an NDI system. Based on this hit-miss set a POD distribution can be fitted. The POD curve can be used in the SHM design to optimize the number and position of the sensors. As an example, a simulation was run for 150 randomly generated impact damages. A SNR of 100 was assumed in the analysis. The resulting hit-miss data was subsequently used to determine a lognormal fit by means of the maximum likelihood estimation, depicted as the blue-line in Figure 4. The red-lines represent the 95% lower and upper confidence bound, which will narrow for increasing number of simulations. Figure 5 shows an overview of the

randomly simulated impact damages. The true POD strongly depends on the SNR. In general, the more sensitive the damage indicator the more sensitive it is for noise as well.

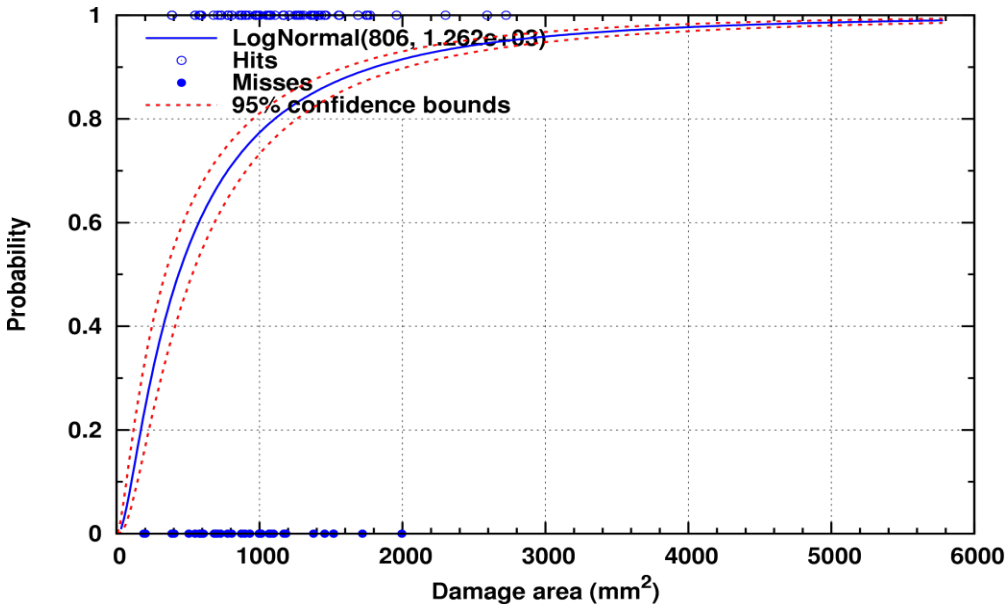


Figure 4: Computed POD distribution for 5x8 sensor network.

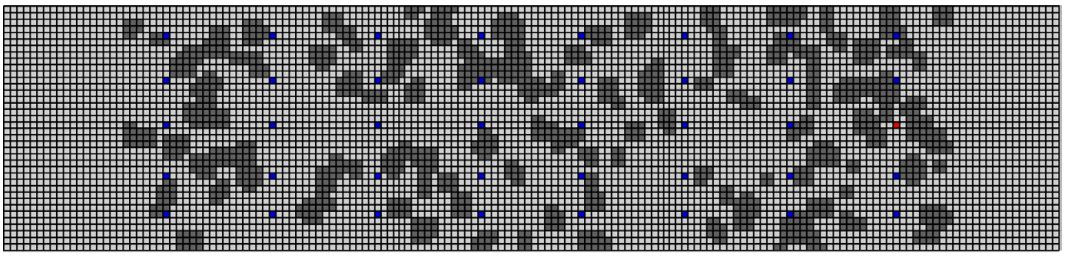


Figure 5: Randomly simulated impact damages.

An experimental program was performed on the TAPAS panel using the Technobis FBG interrogator [5] for damage detection from modal strain response measured with FBGs – see Figure 6. The optical fibers were surface mounted, but can be embedded in the composite structure as well. Measurements were performed on the undamaged panel and two damaged configurations. The measured and numerically determined eigenfrequencies and mode shapes correlated well for the first eleven modes. Hence, it can be concluded that the finite element model provides a good representation of the real TAPAS panel. For most sensors only low strain levels of about ±15 microstrain were obtained in the modal tests, in the order of the noise level at a high sampling rate of 20 kHz. Irrespective of the very low signal-to-noise levels obtained for the FBGs, modal responses could be obtained up to 300 Hz, from which the damage indicator was computed. Promising results were obtained for the random excitation, for which the

damages in both panels were detected, although not by the nearest sensor but one farther away [5].

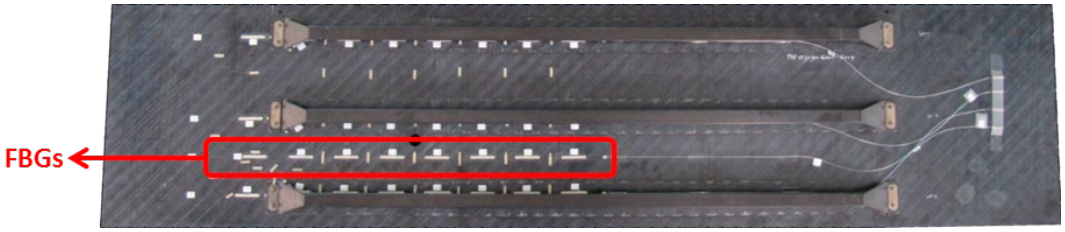


Figure 6: Three-stringer thermoplastic composite TAPAS panel with installed FBGs.

The current results can be improved by increasing the accuracy of the measured modal strain amplitudes. This can be achieved by reducing the noise in the interrogator system, increasing the SNR. Plan is to validate this SHM design tool with more complex aircraft structure.

2.2 Comparative Vacuum Monitoring Technique

In 2009 an evaluation of the Comparative Vacuum Monitoring (CVM) technique for fatigue crack detection has been performed. The CVM technique is based on the principle that a small volume maintained at a low vacuum (~ 0.7 atm.) is extremely sensitive to any ingress of air, e.g. via a crack. This is explained in Figure 7.

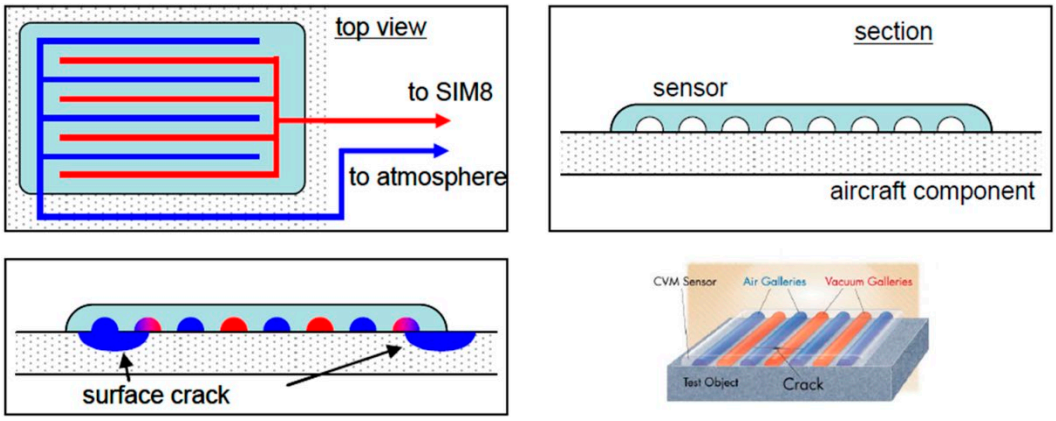


Figure 7: Comparative vacuum monitoring (CVM) technique. The red lines represent the vacuum galleries, the blue lines the atmospheric galleries of the sensor.

A self-adhesive, elastomeric sensor with fine channels is adhered to the surface of a component where damage is expected to occur. The sensor forms a seal with the surface so that the channels together with the structure form a manifold of air tight channels. These channels form closed “galleries” to which alternately a low vacuum and

atmospheric pressure is applied. One set of the galleries (red) is connected to a vacuum source through an accurate flow meter; the other galleries (blue) are left to contain ambient, atmospheric pressure. If there is no damage on the component, then the vacuum in the sensor will remain at a stable low-level of about 20 kPa below atmospheric pressure. If a crack develops in the sensor area, a leakage path occurs across the different galleries and a measurable change in the vacuum level will be produced. The reduction in the vacuum level is detected by a flow meter. The CVM-system measures in fact the pressure drop of the leakage flow and 'detects' a crack if the leakage flow exceeds a threshold level.

The evaluation has been performed using a CVM laboratory system – see Figure 8. Basic features of the inspection technique were described and a literature study on CVM applications was made, followed by measurements on Friction Stir Welded (FSW) stiffened aluminium panels under fatigue and static loading [6]. In general it was found that the sensors were easily applied and attached well to the specimen surfaces. Fatigue testing showed that the CVM laboratory system is easy to use and has a high sensitivity for crack detection, which occurs when the crack tip reaches the first vacuum channel of the CVM sensor. No false calls were obtained during the complete testing period.

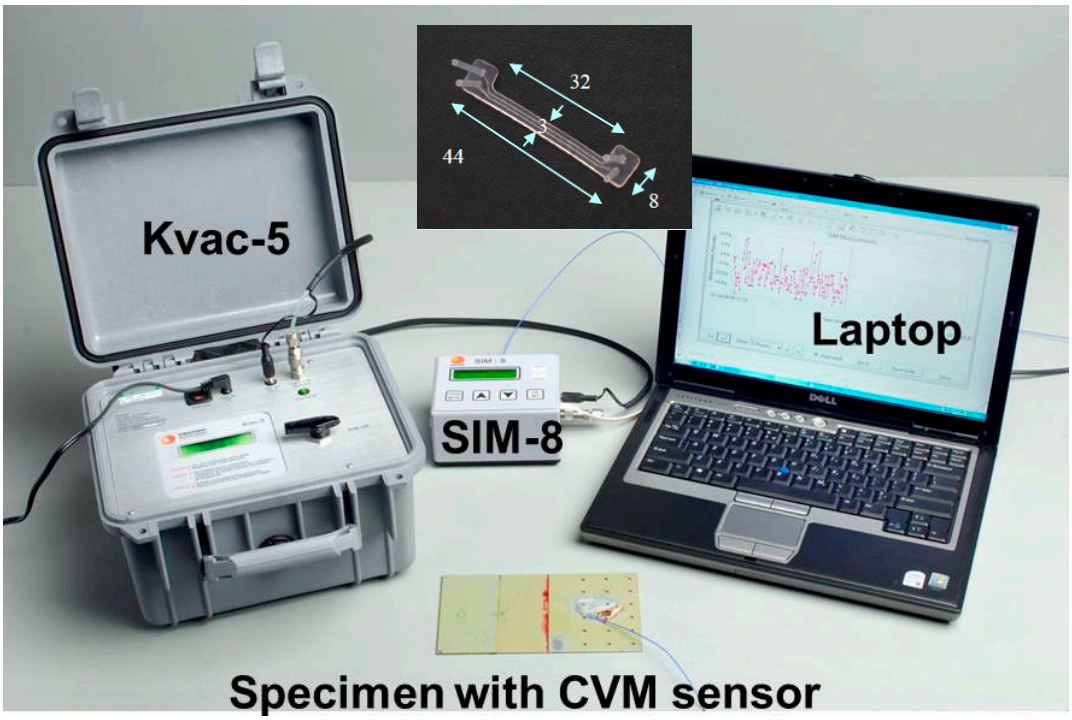


Figure 8: Main components of the CVM laboratory system: a stable reference vacuum source (Kvac-5), a highly sensitive flow meter (SIM-8), a laptop for data logging and a specimen with a CVM sensor.

Under static loading conditions the CVM measurements were less successful, most probably because of residual compressive stresses in the FSW panels (up to 40 MPa). It appeared that the CVM system can only properly detect cracks when they are open. For closed cracks the detectability is poor. For the time being, CVM application on RNLAF aircraft structures is intended to be off-line on the ground (periodic inspection at predetermined intervals) for structural parts known to be under tension loading.

More recently a new and portable version of CVM equipment (PM200), suitable for in-service use on aircraft, has also been evaluated – see Figure 9. The PM200 is claimed to be more accurate and less sensitive to environmental influences and to crack closure.

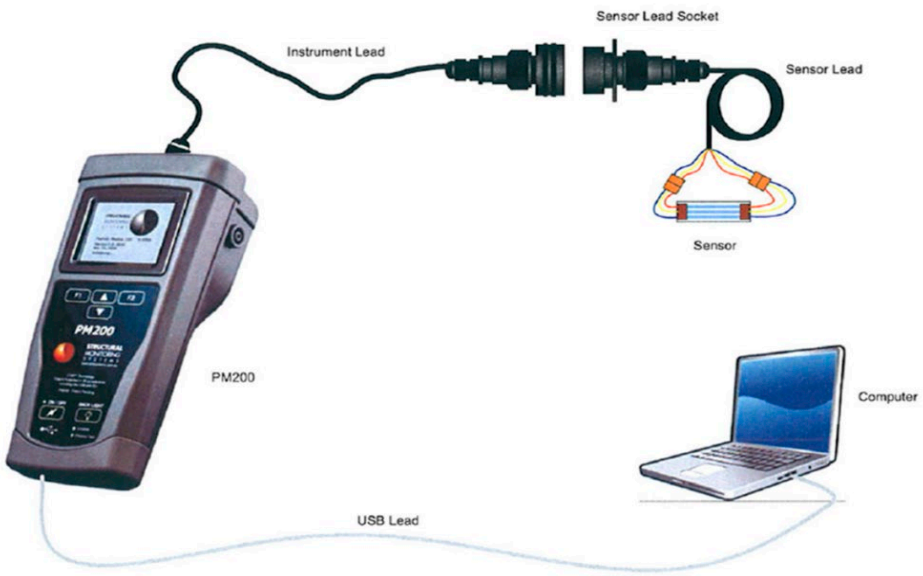


Figure 9: The PM200 system, with integrated vacuum source and flow meter.

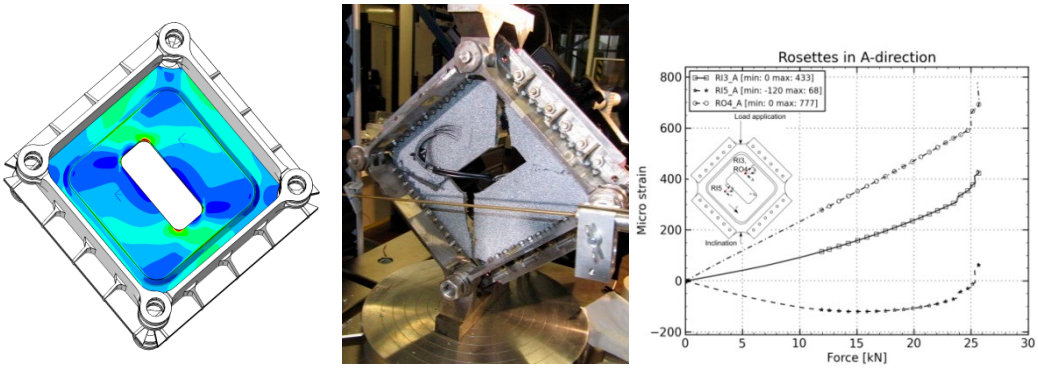
The PM200 system has been tested in fatigue crack growth tests using standard aluminium 2014-T651 M(T) coupons [7]. The measurements were taken under stationary conditions (simulating on-ground measurements on an actual aircraft), both under compressive and tensile loading. To enable a fair comparison, the tested coupons were also monitored with the CVM laboratory system. The main conclusions were:

- The PM200 system is more complex and requires a more detailed knowledge as compared to the CVM laboratory system. It has many options, possibilities and settings that can be adjusted. Measurement preparation requires more time and a trained operator. The PM200 instrument must be configured, sensor lead sockets must be configured and preparation tests are recommended to tune the system.

- PM200 test execution is straightforward and can be done by minimally trained personnel. Test execution takes considerably more time as compared to the CVM laboratory system, however. Installation of the sensor is more delicate. Extra care must be paid to surface preparation and air blocking at the sensor edges.
- The PM200 system is more sensitive and is able to detect cracks under compressive loading. However, cracks have to be rather large to be detected conclusively. Smaller cracks are not always detected under compressive loading. The PM200 system is less sensitive to crack closure effects but crack tip conditions and the degree of matching of the mating crack surfaces might be of influence.
- PM200's crack detection capability in static loading conditions has significantly improved as compared to the CVM laboratory system.

The CVM system is also suitable for use on fiber reinforced composite materials, to detect damages due to fatigue or static loads. This has been verified in an extensive RNLAF funded research program that aimed to develop a better understanding of in-service composite materials behavior and to develop damage indicators that can be used for SHM of composite structures. One of the reasons that damage indicators have not yet been developed for composite aircraft structures is the common belief that composite materials do not suffer from fatigue. This is partly true as most of the composite aerospace structures are designed to a maximum allowable strain under normal operation conditions that is in the order of 3000 micro-strain. In general, this strain level is below the level for which fatigue becomes an issue. However, this is only true for the in-plane properties. In cases where out-of-plane loadings are significant (e.g. ply-drop-offs, eccentrically loaded sandwich facings and local attachment points), fatigue resistance should be taken into account.

To demonstrate the performance and applicability of the developed damage models, a representative demonstrator shear panel with a rectangular cut-out was designed, analyzed and tested both statically and in fatigue, under R=-1 loading (Figure 10, see [8]). The sandwich panel was made of carbon fiber fabric reinforced composite sheets with a 5 mm thick honeycomb core. No impact damages were inflicted.



a) Pre-test analysis *b) Test setup* *c) Strain result during static testing*

Figure 10: Static damage development in a shear loaded carbon fiber fabric reinforced composite sandwich panel.

The fatigue test results were in line with reported results in literature on coupon scale. The fatigue failures occurred only at relatively high load levels. These load levels are still realistic, however, since it is expected that with the increasing pressure on weight savings and with increasing knowledge, the allowable design stresses in composite structures will increase. The scatter found in the fatigue test program was surprisingly low, as one order of magnitude of difference is often encountered for composites. A power law fit was made which covered the data well see Figure 11. Note that the applied force can be assumed to be (nearly) linearly related with stress as long as bending remains small.

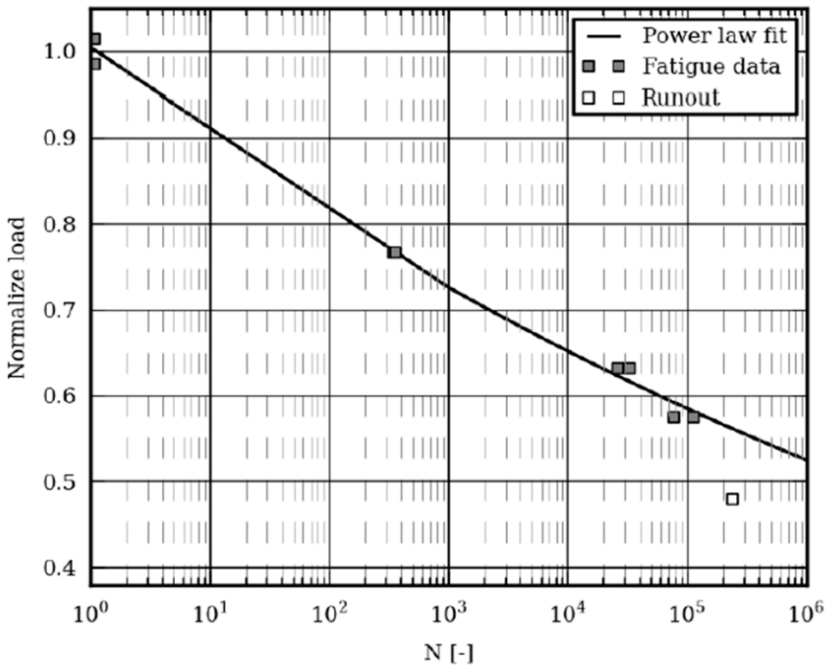


Figure 11: Fatigue life diagram of the shear panels for R=-1 loading. A power law fit has been made through the fatigue data.

In the test program the CVM laboratory system has been applied to monitor the onset of static and fatigue cracking. For this purpose CVM patches were installed at two of the four corners of the rectangular panel cut-out. A typical example of the use of the CVM system is given in Figure 12 where a crack initiated at the outer face of the panel and was captured by the CVM system. Although a few false alarms were given by the CVM system during the test program, it generally turned out to be a reliable system which detects the onset of fatigue cracking very early and accurately. The false alarms were probably due to a weak bonding between the rough composite layer and CVM patch.

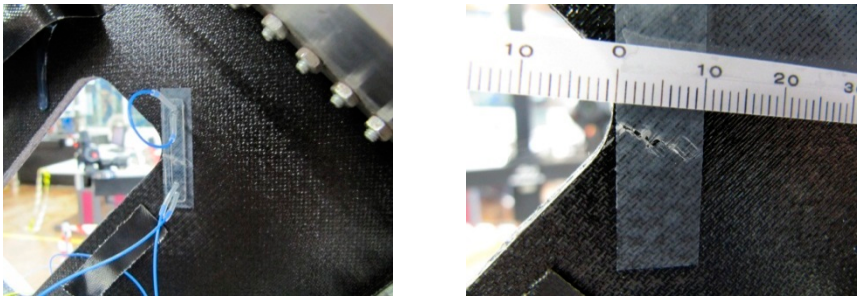


Figure 12: Crack initiation on outer face of the tested panel, captured by CVM patch with CVM patch still installed (left) and removed (right).

This evaluation of the CVM system on fiber reinforced composite material is published in ref. [8].

3 Advanced Load Monitoring

The design usage spectrum assumed by the OEM of an aircraft is usually very different from the actual usage spectrum that is experienced in service. This is especially the case for military aircraft, due to their versatile mission profiles. The consequence of this is that the actual service life of an aircraft can significantly deviate from its certified life. For this reason the RNLAF monitors the operational loads and usage for most of its fleets, including the F-16 fighter fleet [9] and the CH-47D/F transport helicopter fleet [15]. In these legacy aircraft the internal loads are measured using conventional electrical resistance strain gauges. This technology is a widely accepted and proven method to perform load monitoring in the aerospace sector. However, there are a number of drawbacks, such as heavy cabling and sensitivity to Electro Magnetic Interference (EMI), that have led to the development of new generations of sensing systems over the last few decades. Of these new technologies, optical sensing systems have shown the greatest potential [10], [11]. For this reason, NLR has conducted various studies to evaluate the

potential of using optical fibres for SHM purposes, both as an add-on system [12] and as an embedded system in composite material [13]. Section 3.1 provides a brief description of NLR's work on utilizing the Fiber Bragg Grating (FBG) technology for load monitoring. Section 3.2 provides a vision of how to affordably utilize such a system in a Fleet Life Management concept.

3.1 Optical Fiber Network on the F-16 Wing

To evaluation the potential of using optical fibers for in-flight load monitoring, use has been made of the full-scale fatigue test campaign (FTC) on a F-16 wing that is currently being conducted at NLR. This so-called damage enhancement test on the LHS wing of a decommissioned F-16 Block 15 is being performed under a contract from the Royal Netherlands Air Force (RNLAF). The results of the test program will be relevant for both the RNLAF and the Chilean Air Force (FACH), who will share the program results with the RNLAF on a government to government basis. An impression of the test setup is provided by Figure 13.

The damage enhancement test aims to grow in-service cracks of sub-detectable size to a size where they may readily be detected. This significantly enhances the teardown inspection program that is conducted in parallel on the RHS wing of the same aircraft. The main objective of the test is to determine if the ex-service wing contains damage not accounted for in the early durability test program that was performed in the late 1970s or in the current durability and damage tolerance analysis of Lockheed Martin (LM).

Figure 13: Impression of the F-16 Block 15 wing test setup.

In order not to miss any potentially critical cracks because of insufficient loading conditions, the test setup is fairly complex and involves the use of 23 load actuators and the application of a representative fatigue load spectrum that is based on an extensive loads survey that was taken of the RNLAF fleet in 2004. The same spectrum has been used by LM in the development of the current structural maintenance plan. The added advantage is that the observed crack growth rates in the tested wing will be representative of those in service. This will enable a translation of the test results to service conditions and provide an experimental validation of the theoretical crack growth curves provided by LM in the DADTA. It must be realized that the results are not meant for certification purposes, however. The available budget and time only allows a relatively simple approach which, for instance, precludes the presence of the leading edge flap and flaperon (the calculated interface loads will be applied, however) and the pressurization of the fuel tank. It is noted in this respect that the durability test involves a simplification by considering the wing only configuration; the design of a representative wing root support has been dealt with using calculations with the F-16 Block 15 coarse grid Finite Element Model (cgFEM) of LM that is available at the NLR. The support structure has been equipped with calibrated load sensors to monitor the distribution of the reaction forces over the four wing attachment fitting stations and the shear ties.
No artificial damages have been applied to the test article. Only naturally existing damages, caused by in-service fatigue loading, corrosion, etch pits, tool marks, etc. are considered. The durability test will cover two design lifetimes or less (in case of untimely failure). No residual strength test will be conducted at the conclusion of the test.

The durability test has started in May 2013 and is expected to be completed by the end of October 2013, after which an extensive teardown and inspection will be conducted.

The wing has been instrumented with 75 conventional bonded resistance strain gauges (single gauges, full bridges & rosettes) that are continuously monitored and three optical fibers on the upper wing skin with a number of Fibre Bragg Gratings that act as strain gauges. An impression of the sensors on the upper wing skin is provided in Figure 14, including optical fibers (yellow lines), acoustic emission sensors and strain gauges.

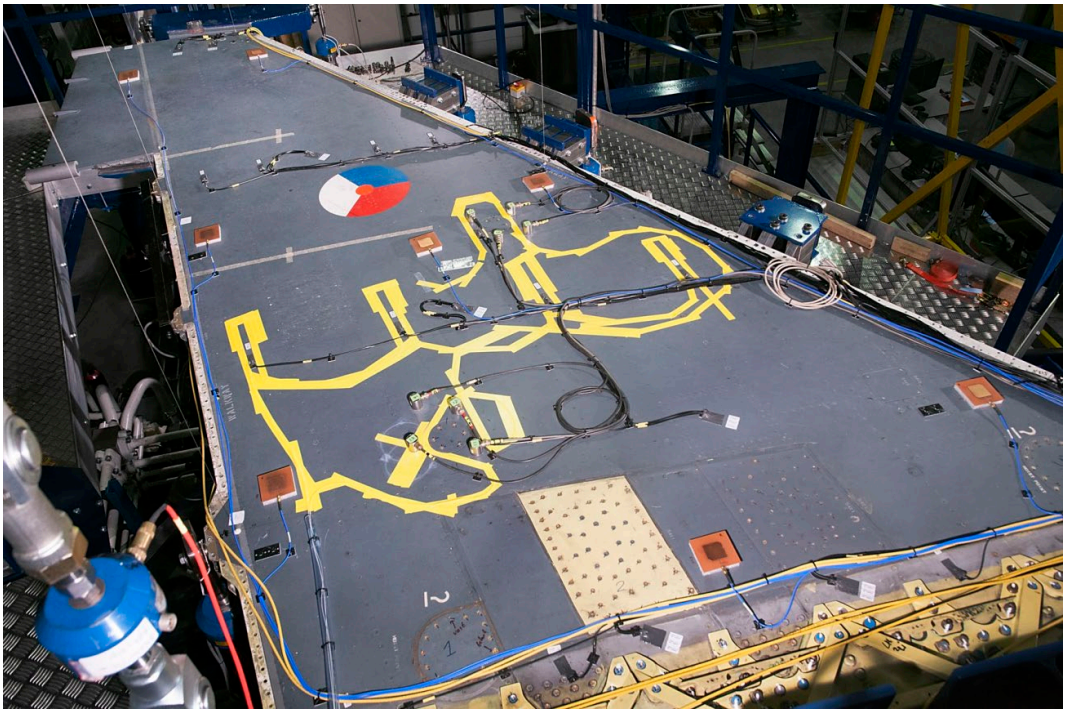


Figure 14: Evaluation of a number of SHM and load monitoring techniques on the F-16 Block 15 wing full-scale durability test at NLR. The yellow tape serves to protect the optical fibers with the FBGs.

A total of nineteen FBGs are installed on the upper wing skin, divided over three fibres. Details are provided in Table 1. Twelve FBGs are positioned alongside conventional strain gauges, six FBGs are placed in an array along two access holes, and one FBG is used for temperature compensation. In this way, the performance of the FBGs can be assessed by comparing the strain measurements from the FBGs with the readings from the adjacent strain gauges. Figure 15 gives a schematic of the placement of the FBG sensors on the upper wing skin.

Fiber	Spectral range	# FBGs	Purpose	Interrogator
A	1529 – 1576 nm	7	Wing-skin cut-out monitoring, temperature measurement	IFIS-100
B	1522 – 1578 nm	8	Comparison with strain gauges	IFIS-100
C	839 – 860 nm	4	Comparison with strain gauges	Deminsys Ultra

Table 1: List of optical fibers used during the F-16 FTC.

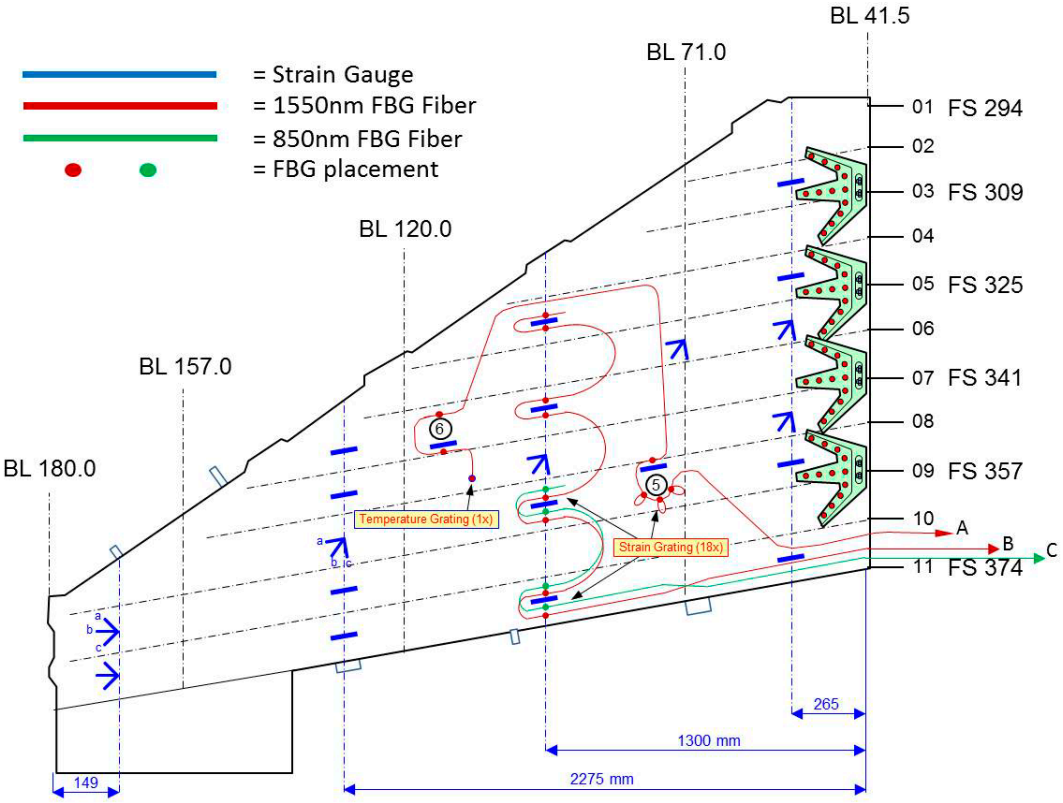


Figure 15: Optical fiber alignment on the F-16 wing upper surface. Red and green dots depict the position of the 1550nm and 950nm FBGs respectively.

Two FBG interrogators are used: a Deminsys Ultra system from the Dutch company Technobis Fibre Technologies and an IFIS-100 system from the Korean company Fiberpro. Both systems use a Super-luminescent Light Emitting Diode (SLED) light source and a Charge-Coupled Device (CCD) based method. However, the light sources of the systems accommodate different broadband light: the IFIS-100 works with a 1532 ~ 1558 nm light band source, while the Deminsys Ultra operates around the 850nm-band.

Prior to the FTC, a number of complementary coupon tests were conducted to assess the variables which could influence the strain transfer behavior from test specimen to the fiber core. This assessment has shown that the combination of cyanoacrylate-based bond M-200 and a polyamide coating allows a linear strain transfer when the material of the test specimen is aluminium 2024-T3, which is comparable to that of the F-16 upper wing skin. A similar result is reported in Harold et. al. [14].

At this moment, the FTC is still in progress. The required two life times of 16,000 simulated flight hours are expected to be reached by the end of October 2013, after which the evaluation of the optical fiber systems will be performed.

3.2 Outlook

The routine storage of high-resolution operational strain and loads data (measured with either classical bonded resistance strain gauges or optical fibers) in a relational database provides a flexibility and a growth potential that will turn out to be invaluable for any fleet life management program. One option in this respect is the development of so-called virtual strain gauges, in which the measured strain at a particular point in the airframe is correlated to the flight and engine parameters that are routinely collected with a digital databus. This can be done by means of an Artificial Neural Network (ANN) or by a deterministic regression technique – see for instance [15]. Once such a correlation has been established for all possible conditions in the flight and ground envelope of the aircraft, the gauge that has been used to measure the strains at this particular location is no longer needed.

In combination with a relational database[1], virtual strain gauges are the key to the rational management of the life of any structural component of any aircraft in the inventory of an operator. Based on this technique, a new fleet life management (FLM) concept has been developed called the "Stethoscope Method", which to a large extent is already operational for the CH 47D helicopter fleet of the RNLAF [15]. This method is outlined in Figure 16. It involves the fleet wide collection and storage of all relevant flight, engine and control parameters that are available from the digital data bus, plus the simultaneous collection of loads data in one or more dedicated OLM aircraft[2]. For this purpose all aircraft in the fleet need to be equipped with a digital Flight Data Recorder (FDR). The OLM aircraft will need an additional Structural Data Recorder (SDR), for instance the ACRA KAM 500 unit that is used in the Chinook of the RNLAF, or, alternatively, an additional FDR loads monitoring functionality.

The loads data from the SDR in the OLM aircraft can be used to continually train and improve the ANN-based virtual strain gauge models. By moving the strain gauges in the OLM aircraft around on a regular basis, models will be obtained for more and more points in the airframe that are relevant from a fatigue point of view. This will allow the establishment of the safe life consumed so far and the assessment of the severity of in-service developed fatigue cracks for each critical point. In this respect it is essential to start collecting the digital bus data from the very first moment that an aircraft is commissioned into service. For the virtual strain gauge models this is less crucial; when

[1] NLR has developed a dedicated database for the storage of loads & usage data, called HELIUM.
[2] It is noted that the OLM aircraft are part of the operational fleet; no dedicated test flights are needed apart from those needed to commission the SDR.

developed at a later stage these models can use the historic bus data that are stored in a relational database to 'roll back' to day one.

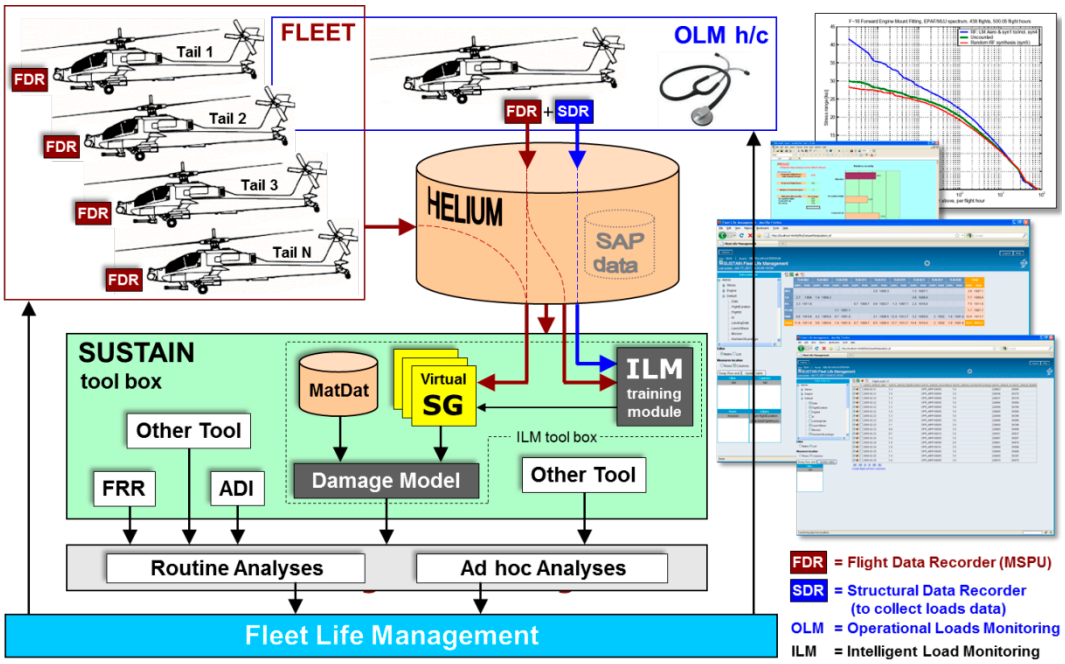


Figure 16: The stethoscope method for fleet life management.

4 Conclusion

New technologies are emerging that enable on-board Structural Health Monitoring and advanced load monitoring in fixed-wing aircraft and helicopters. Once mature, these technologies can be employed to enhance flight safety and to reduce maintenance efforts and, thus, downtime and costs. The introduction of these new technologies on flying platforms entails a tremendous amount of certification effort. NLR, sponsored by the RNLAF, aims to contribute to this effort by evaluating new technologies and identifying the potential benefits and problems. This paper gives an overview of some of the recent work performed within NLR in this respect. Additionally an affordable Fleet Life Management approach, called the stethoscope method, is discussed. The storage of measured flight and loads data on a day-to-day basis will allow detailed analyses that are beyond the scope of traditional loads & usage monitoring programs.

5 Acknowledgement

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References

[1] Grooteman, F.P. (2012). “Damage Detection and Probability of Detection for a SHM System Based on Optical Fibres Applied to a Stiffened Composite Panel”, ISMA2012, Leuven, 17-19 Sept. 2012.

[2] Stubbs, N., Kim, J.T. and Topole, K. (1992). “An Efficient and Robust Algorithm for Damage Localization ins Offshore Platforms”, Proceedings of the ASCE, 10th Structures Congress, p. 543-546.

[3] Cornwell, P., Doebling, S.W. and Farrar, C.R. (1999). “Application of the strain energy damage detection method to plate like structures”, Journal of Sound and Vibration, vol. 224-2, p. 359-374.

[4] Grooteman, F.P. and Touzé, R.S.P. (2011). “Vibration-Based Damage Indicator Evaluation for Structural Health Monitoring with Optical Fibres”, NLR report, nr. CR-2011-026.

[5] Grooteman, F.P. (2012). “Proof of Concept Structural Health Monitoring System for Damage Detection using Optical Fibres”, NLR report report, nr. CR-2012-330.

[6] Zee, M.G. van der. and Heida, J.H. (2007). “Comparative Vacuum Monitoring (CVM) System Evaluation”, NLR report, nr. CR-2007-290.

[7] Dominicus, J.A.J.A. (2010). “Comparative Vacuum Monitoring (CVM); CVM laboratory and PM200 system behaviour compared under crack closure conditions”, NLR report, nr. CR-2010-013.

[8] Vrie, G. van de et. al. (2013). On the static and fatigue behaviour of a shear loaded fabric CFRP sandwich panel”, publ. in the proceedings of the 27th symposium of the International Committee on Aeronautical Fatigue and Structural Integrity, 5-7 June 2013, Jerusalem, Israel.

[9] Winkel, F.C. et. al. (2002). “RNLAF/F-16 Loads and Usage Monitoring/Management Program”, NLR report, nr. TP-2002-309.

[10] Guo, H.L. et. al. (2011). “Fiber Optic Sensors for Structural Health Monitoring of Air Platforms”, Sensors 2011, vol. 11, p. 3687-3705.

[11] Yari, T. et. al. (2008). “Monitoring Aircraft Structural Health Using Optical Fiber Sensors”, Mitsubishi Heavy Industries Technical Review, Vol 45-4, Dec 2008.

[12] Rabelo Faria, J. (2013). ”The Application of Fibre Bragg Grating Sensors for Load Monitoring of Aircraft Components”, NLR Memorandum nr. AVGS-2013-019.

[13] Beukema, R.P. (2012). “Embedding Technologies of FBG Sensors in Composites: Technologies, Applications and Practical Use”, NLR report nr. TP-2012-501.

[14] Harold, D.A. et. al. (2012). “The Effects of Adhesive Geometry and Type on Strain Transfer for Optical Fiber Strain Sensing”, Material Evaluation, December 2012, p. 1411-1419.

[15] Oldersma, A. and Bos, M.J. (2011). “Airframe loads & usage monitoring of the CH-47D Chinook helicopter of the Royal Netherlands Air Force”, publ. in the proceedings of the 26th symposium of the International Committee on Aeronautical Fatigue and Structural Integrity, 1-3 June 2011, Montreal, Canada.

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