Fielding a Structural Health Monitoring system on legacy military aircraft

A business perspective

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EXECUTIVE SUMMARY

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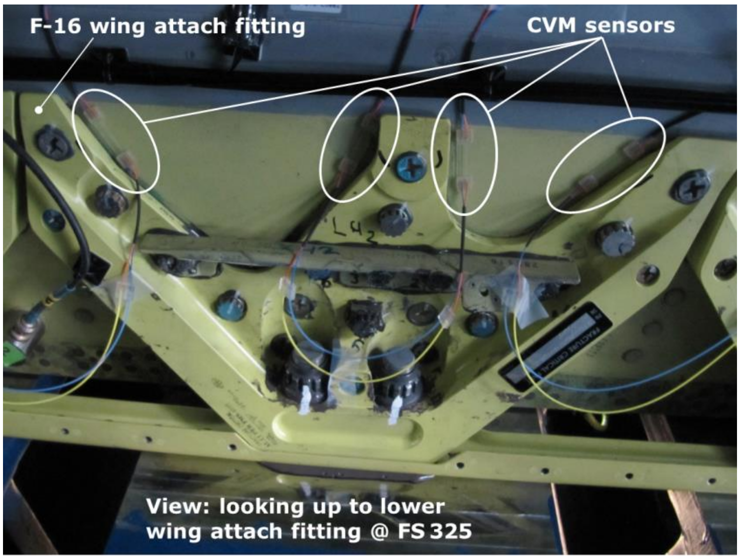

Example of a Structural Health Monitoring system: Comparative Vacuum Monitoring as applied during a full-scale fatigue test on an F-16 wing.

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Executive summary

The sustainment costs of military aircraft constitute a substantial part of the total life cycle costs, which implies that the application of innovative methods and technologies in the sustainment process may lead to large cost savings. An important trend in this respect is the transition from preventative maintenance to Condition Based Maintenance (CBM). For CBM it is essential that the actual system condition can be measured and that the measured condition can be reliably extrapolated to a convenient moment in the future in order to facilitate the planning process while maintaining flight safety. Much research effort is currently being put in the development of technologies that enable CBM,

This report is based on a presentation held at the 2nd International Conference on Advances in Structural Health Management and Composite Structures in Jeonju, South Korea, August 27-29, 2014.

among which Structural Health Monitoring (SHM) systems. Good progress has already been booked when it comes to sensors, sensor networks, data acquisition, models and algorithms, data fusion/mining techniques, etc. However, the transition of these technologies into service is very slow. Reasons are that business cases are difficult to define and that certification of SHM systems is very challenging.

This paper describes a possibility for fielding a SHM system on legacy military aircraft with a minimum amount of certification issues and with a good prospect of a positive return on investment. For appropriate areas in the airframe the application of SHM will reconcile the fail-safety and slow crack growth damage tolerance approaches that can be used for safeguarding the continuing airworthiness of these areas, combining the benefits of both approaches and removing the drawbacks

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Summary

The transition of Structural Health Monitoring (SHM) technologies into service is very slow. Reasons are that business cases are difficult to define and that certification of SHM systems is very challenging.

This paper describes a possibility for fielding a SHM system on legacy military aircraft with a minimum amount of certification issues and with a good prospect of a positive return on investment.

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Content

Abbreviations

Acronym	Description
a_c	Critical fatigue crack size
a_d	Detectable fatigue crack size
a_i	Initial fatigue crack size
ASIP	Aircraft Structural Integrity Program
CBM	Condition Based Maintenance
CVM	Comparative Vacuum Monitoring
DLL	Design Limit Load
FS	Fuselage Station
NDI	Non-Destructive Inspection
NLR	National Aerospace Laboratory
POD	Probability Of Detection
SHM	Structural Health Monitoring
SOF	Safety Of Flight
T_c	Time of failure
T_i	Time of initial inspection
TRL	Technology Readiness Level
USAF	United States Air Force
ΔT_r	Recurring inspection interval

1 Introduction

Military operators worldwide are looking for ways and means to maintain or even improve the availability and continuing airworthiness of their fleets of aircraft while decreasing the cost of ownership. The sustainment costs of military aircraft constitute a substantial part of the total life cycle costs (typically in the order of two-thirds), which implies that the application of innovative methods and technologies in the sustainment process may lead to large cost savings. An important trend in this respect, especially within the USAF, US Navy and US Army but also within the aerospace industry, is the transition from preventative maintenance – based on calendar time, flight hours or flight cycles – to Condition Based Maintenance (CBM). Maintenance is then only performed when needed, which is expected to lead to a significant cost reduction. For CBM it is essential that the actual system condition can be measured (diagnostics) and that the measured condition can be reliably extrapolated (prognostics) to a convenient moment in the future in order to facilitate the planning process while maintaining flight safety. Much research effort is currently being put in the development of technologies that enable CBM, among which Structural Health Monitoring (SHM) systems. Good progress has already been booked when it comes to sensors, sensor networks, data acquisition, models and algorithms, data fusion/mining techniques, etc. However, the transition of these technologies into service is very slow. There are two reasons for this:

- ***Business Cases*** are difficult to define since CBM represents a disruptive technology that produces a paradigm shift for maintenance support [1];
- ***Certification*** is difficult as the validation of the SHM system's capability to reliably and accurately detect impending in-service failures is extremely challenging; in addition, the procedures for obtaining maintenance credits are still being developed.

One option to validate the performance of a particular SHM system is to use a seeded fault test. This requires a hi-fidelity and expensive test bench and a good a priori knowledge of the location and the nature of the failure modes that are to be detected. An alternative is to field the SHM system in a sufficient number of aircraft and evaluate its performance after a sufficient number of flight cycles. 'Sufficient' in this respect is indeterminate and may cover a significant part of the service life in order to be able to collect relevant data. This, of course, is undesirable. Fortunately there are some special cases where certification of a SHM system for use in military aircraft is much easier. This paper describes such a case. It forms an opportunity to field a SHM system on legacy military aircraft such as the F-16 with a minimum amount of certification issues and with a good prospect of a positive return on investment. Seizing it would be an evolutionary step towards more challenging applications.

2 Structural integrity concepts of military aircraft

The formation and growth of fatigue cracks is still considered to be the major threat to the structural safety and continuing airworthiness of military combat and transport aircraft [2]. To guard against the detrimental effects of structural fatigue, a number of design and maintenance concepts have been evolved over the years. Two philosophies are currently in use, viz. the *safe life* concept, which precludes the presence of fatigue cracks, and the *damage tolerance* concept, in which fatigue cracks and other flaws that are assumed to be present from day one should not grow to a critical size within a reasonable period (e.g. lifetime or inspection interval), in order to allow for timely detection and repair. The initial USAF damage tolerance requirements were introduced in 1974, in MIL-A-83444 [3], and the F-16 is the first fighter aircraft that has been designed and certified to this specification. MIL-A-83444 allowed the use of either fail-safe or slow crack growth design concepts. The focus for the F-16 and other contemporary fighters was on slow crack growth however, since most combat aircraft were designed with many single load path structures and in its original form the MIL-A-83444 requirements tended to discourage the application of fail-safety [4]. With the slow crack growth concept it is mandatory that material, manufacturing and/or service induced defects not be allowed to grow to their critical crack sizes before they are detected and repaired. The slow crack growth damage tolerance concept therefore only provides safety if it incorporates a rigorous inspection program. Conservative initial crack sizes were specified in MIL-A-83444 – and later in the Joint Services Specification Guide, JSSG-2006 [5], and Structures Bulletin EN-SB-08-002 [6] – for use in design and in establishing inspection requirements. A typical value is 1.27 mm (or 0.05”) for a corner crack that is to represent a flaw (i.e. manufacturing defect, material defect, corrosion pit, maintenance induced damage, etc.) that is assumed to be present at the most critical location (e.g. a fastener hole) in a flight critical structural item. The required time T_i for the initial inspection is then determined by dividing the time that it takes for a fatigue crack to grow from its initial size a_i to its critical size a_c by a safety or scatter factor of two, where a_c is the crack size at which design limit load (DLL) will lead to unstable fracture. This is schematically shown in Figure 1. This figure also shows how the recurring inspection interval ΔT_r is determined. The recurring inspection interval is generally shorter than the time to initial inspection since it is based on the safe crack growth life of an in-service detectable flaw with size a_d, which depends on the inspection method that is used (visual, eddy current, etc.), the location in the aircraft (easy access or not, lighting conditions), the presence of fastener heads that block the view on the crack, etc. The minimum detectable flaw sizes used in the establishment of the recurring inspection intervals should be based on experimentally determined probability of detection (POD) curves that are relevant for the selected inspection method and the material and geometry of the structural area that is to be inspected. Guidelines are provided in USAF Structures Bulletin EN-SB-08-012 [7].

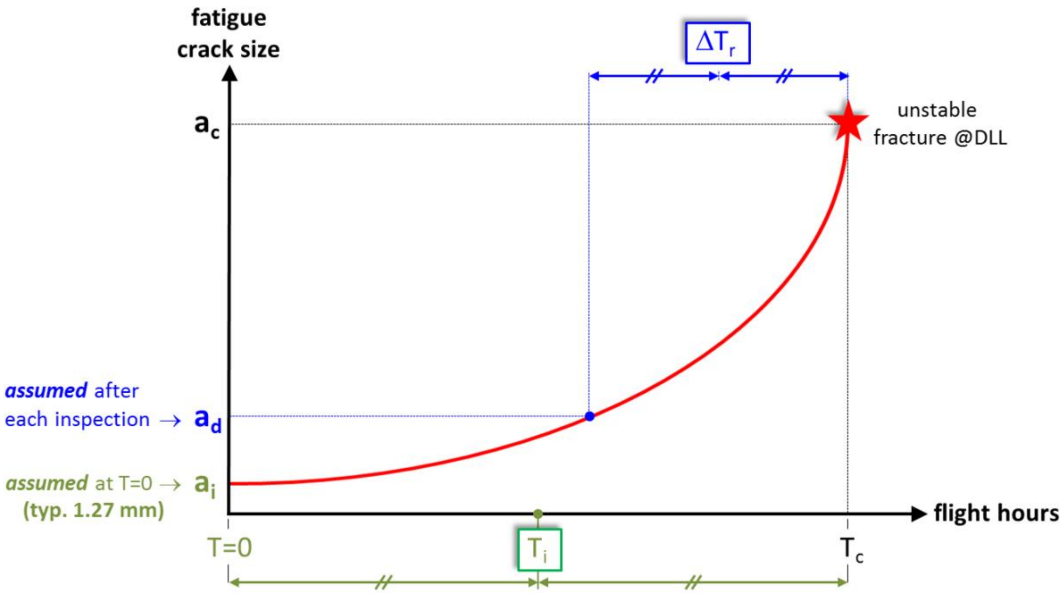

Figure 1: Determination of the initial inspection time T_i (in green) and the recurring inspection interval ΔT_r (in blue) in the slow crack growth damage tolerance concept.

Fatigue crack growth curves are determined with fracture mechanics models that are calibrated against the results from fatigue tests on coupons, components and/or full-scale structures.

It should be realized that the initial flaw with size a_i that is assumed to exist at T=0 is entirely fictitious. This conservative approach provides safety against a multitude of potential threats such as material imperfections, manufacturing problems, maintenance induced damage, etc. Actual cracks are therefore rarely found during the inspections, especially during the ones scheduled early in the service life of the aircraft, as illustrated in Figure 2. This notion has led to the relatively recent development of using risk-based methods to establish maintenance requirements. The advantage of using probabilistic methods is that they can be fed with service findings and that any factor that affects safety of flight can be included in the analysis, such as the probability of missing an inspection, the increasing probability of the formation of fatigue cracks with time and the variability of material parameters, initial flaw sizes, service loads, usage, etc. A description of these methods is beyond the scope of the present paper, however, but an introduction can be found in ref. [8].

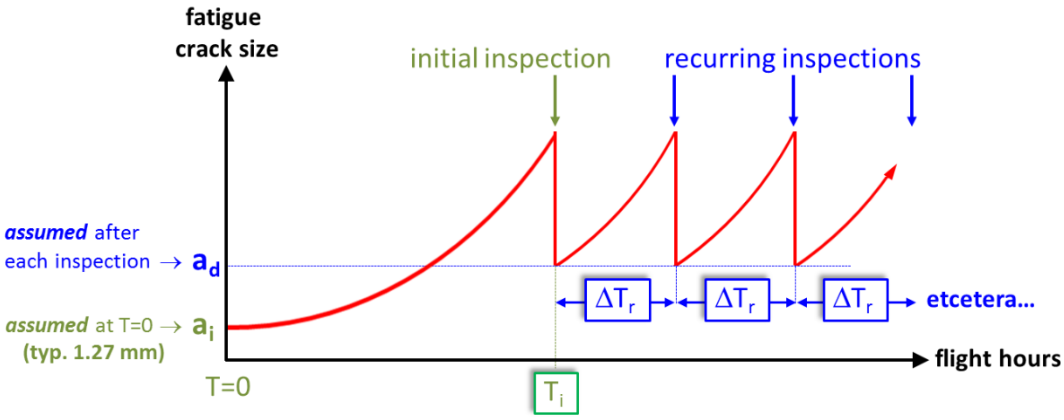

Figure 2: In-service inspections for fatigue cracks. After each inspection the size of the assumed analytical crack is reset to its in-service detectable size a_d.

Be that as it may, the USAF have recently revised the original MIL-A-83444 fail-safe requirements in an attempt to encourage fail-safe design and certification of future military aircraft as well as provide the basis for fail-safe assessments of legacy aircraft. Although no military aircraft has been designed and certified to the MIL-A-83444 fail-safe requirements, most of these aircraft do feature some fail-safety through the use of multiple redundant load paths. In a fail-safe structure a primary component is allowed to completely or partially fail, provided that the residual strength of the adjacent secondary structural elements is sufficient to sustain critical design limit load conditions and that the failure of the primary load path is either readily detectable during a scheduled visual inspection (so not NDI) or malfunction evident, meaning a failure would result in the malfunction of other systems (e.g. fuel leakage or pressure loss) that would alert the flight or ground personnel to the existence of the failure. “Readily detectable” could also mean that the failure is apparent from in-flight or post flight visual observations. The new fail-safe requirements are laid down in Structures Bulletin EN-SB-08-001 [9]. Some of the MIL-A-83444 requirements that discouraged the application of fail-safety, such as the stipulation of dependent load paths [4], have been removed and also the definition of the fail-safety life limit has been revised and in general the life limit is now longer than the one defined in MIL-A-83444.

There were a number of reasons to promote fail-safety and revise the criteria:

- Fail-safety provides protection against all forms of damage an aircraft may encounter in its lifetime (incl. battle damage and discrete source damage due to bird strike, uncontained engine disk failures, etc.) instead of fatigue damage only.
- The minimum in-service detectable flaw sizes as specified in USAF Structures Bulletin EN-SB-08-012 issued in 2013 are generally larger than what was assumed previously. For legacy aircraft such as the F-16, that has many structural areas with small critical crack

sizes, this has led to revised recurring inspection intervals for slow crack growth damage tolerant structure which, in some cases, were unacceptably short or even zero.

- Fail-safe damage tolerance structure only needs to be inspected visually, which entails a very limited maintenance burden. Identification of those safety-of-flight (SOF) locations which have inherent fail-safe capability, and classifying these locations as such, will allow relaxation of the current inspection burden by focusing special non-destructive inspections (NDI) on only those SOF locations which are not fail-safe. This will entail significant cost reductions and it will lead to an increase of aircraft safety, availability and readiness, especially for aging fleets.
- NDI often requires the removal of sealant and/or fasteners. By doing so damage may inadvertently be inflicted to the SOF locations in question. Scratches and dents are often the precursors to fatigue cracks. Fail-safe damage tolerance structure only needs to be inspected visually, with less associated risk of inflicting damage to critical structure.

Fail-safety can also assist when areas are inaccessible or not practical to inspect regularly [10]. An example is provided in Figure 3, which shows the F-16 fuel shelf of which the joint bolt holes in the upper wing carry-through bulkhead at fuselage station FS 341 are fracture critical.

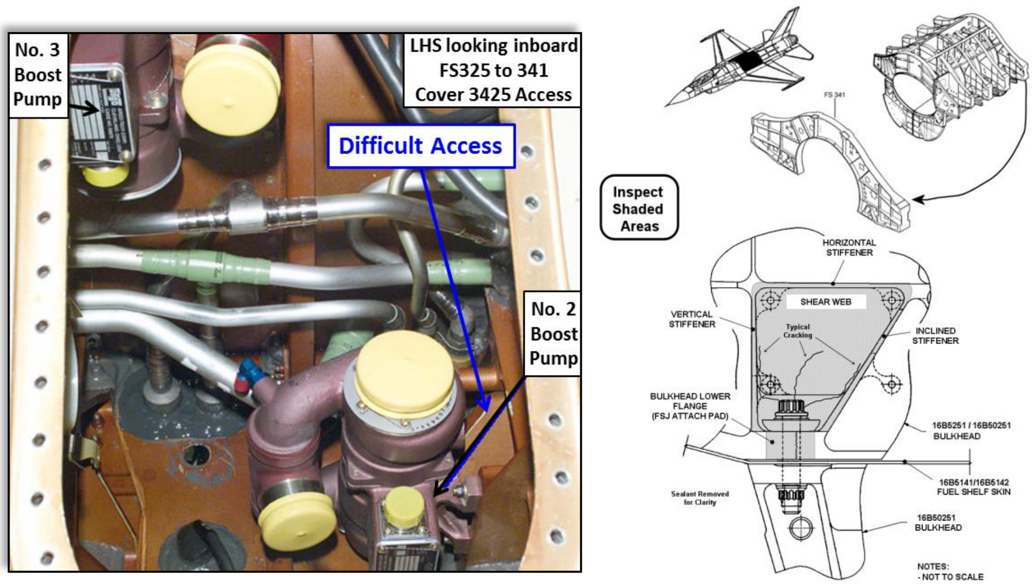

Figure 3: Difficult access to the fracture critical F-16 fuel shelf joint bolt holes at FS341 (left). Visual inspection for large cracks in flanges and web of the upper bulkhead is much easier (right) [10].

When managed using slow crack growth damage tolerance, the joint bolts need to be removed during depot level maintenance to enable the bolt hole eddy current inspection that is required for the detection of small cracks. This is very difficult and often damage is induced. Managing this area using fail-safety and visual inspection for fuel leaks or for large cracks in the flanges or webs of the upper bulkhead is much easier and does not require specialized technicians and tools.

For this particular case it has been shown by the aircraft manufacturer that if the lower flange and web at the fuel shelf joint bolt hole of the upper FS 341 bulkhead fail, limit load can indeed be carried by the adjacent bulkheads and wing attachment fittings [11], which is a prerequisite for fail-safety. Another requirement for fail-safety is that wide-spread fatigue in the form of multi-element damage should be precluded. This means that there is a fail-safety life limit. This limit is determined by the fatigue or durability life of the secondary structural elements, which is the life of a very small fatigue crack – representative of normal production quality or 'fatigue quality', typically sized at 0.25 mm [9] – to failure. This is explained in Figure 4. Once the fail-safety life limit is reached, the probability of secondary structural elements failing in fatigue becomes very high and fail-safety can no longer be guaranteed. Inspections should then again be based on slow crack growth damage tolerance criteria.

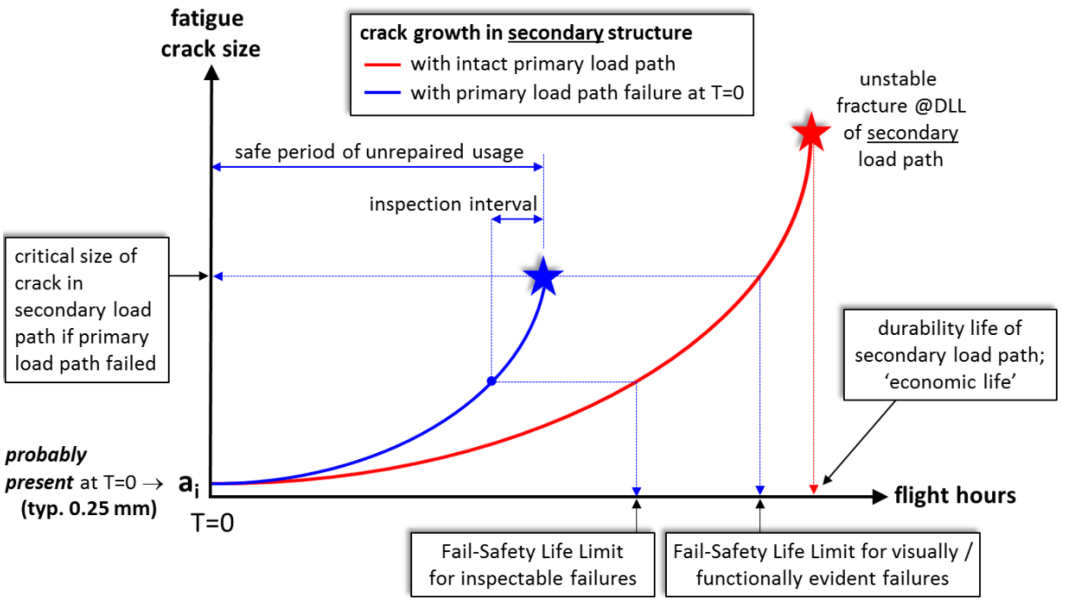

Figure 4: Determination of the fail-safety life limit from the crack growth curve for adjacent structure [9]. The visual inspection interval usually is aligned with phased maintenance. The initial inspection is not prescribed in ref. [9], but is usually taken as half the safe period of unrepaired usage.

It should be noted that damage tolerance, irrespective whether it is based on slow crack growth or fail-safety, provides safety against incidental cracks that may occur during the service life. When the fatigue life of the structure expires, the formation of widespread fatigue damage is to be expected. In this condition, damage tolerant design concepts become ineffective and the structure should be retired.

3 Potential SHM business case

Managing the continuing airworthiness of a fracture critical structural item with fail-safety damage tolerance can be an effective means of mitigating the inspection burden and, as explained in the previous section, has the potential of saving money, decreasing aircraft downtime and increasing safety and fleet readiness. In legacy aircraft such as the F-16 many structural areas could probably be re-classified as fail-safe structure due to their inherent redundancy in load paths. Therefore, by implementing the new fail-safe criteria, this structure would no longer require a special NDI per slow crack growth damage tolerance criteria, but only require visual inspections for large failures. There is one significant disadvantage to this, however: upon detection of cracks their sizes will be such that simple repairs will not be possible anymore. Small fatigue damage at fastener holes can be repaired by reaming the hole and installing a bushing or oversize fastener. Other cases of small fatigue or corrosion damage can often be blended away or cut out and reinforced with a strap or angle. Completely failed load paths, however, usually entail a costly and lengthy repair and may even involve the replacement of an entire wing spar, bulkhead, skin or longeron. This is why many F-16 and other military aircraft operators are hesitant about switching from the NDI-based slow crack growth maintenance approach (with the potential of detecting small repairable cracks) to fail-safety that relies on frequent visual inspections.

This dilemma of having to choose between slow crack growth, to avoid the risk of expensive repairs, and fail-safety, to avoid cumbersome inspections, can be resolved by the application of SHM technology. Normally it would require an extensive and very challenging validation and certification process to replace a mandated and well-established NDI inspection by an automated inspection with a SHM system. This is an important barrier for implementing SHM technology in an operational fleet of aircraft. However, in the case of fail-safety managed airframe structure, it is conceivable to install SHM sensors at the primary structural load path without relying on them for safety. The SHM system is then used for economic reasons only, to detect cracks in the primary structural area while they are still small and easy to repair. In case the SHM system fails to do so, safety is not jeopardized since the continuing airworthiness of the aircraft is still managed by means of fail-safety with visual inspections for large cracks. This means that certification of the SHM system will not be much of an issue, whereas the business case of potentially avoiding large and expensive repairs without the need for cumbersome NDI may be sufficiently worthwhile to justify the upfront investments in the development and installation of a suitable SHM system.

This approach can also be taken to increase the Technology Readiness Level (TRL) of the currently available SHM technology, by testing it on flying aircraft (instead of in a laboratory environment only) without compromising the safety or disrupting the maintenance process of the fleet. The financial side of the business case is less important then and the outlook of over-the-horizon benefits could justify the investments and convince a military aircraft operator to participate in such a development program. What needs to be done is finding suitable structural aircraft elements that can be classified as fail-safe structure due to their inherent redundancy in load paths, and develop appropriate SHM solutions for monitoring these items. An example described in ref. [10] is the F-16 wing root rib, which contains a number of manifold holes that are fatigue critical. Eddy current inspection requires the removal of the wing, which is very labour intensive. Visual inspection for fuel leaks is much easier but will only permit the detection of large difficult-to-repair cracks. Another example was already provided in Figure 3 (which pertains to a difficult-to-repair wing carry-through bulkhead) but, at least for the F-16, there are a number of other airframe components that would qualify for this purpose.

Examples of potentially suitable SHM technology for the detection of small cracks are the comparative vacuum monitoring (CVM) system from SMS plc [12,13] or the permanently-mounted conformable eddy-current sensors such as those developed by Jentek [14] or DSTO [15]. This is not further elaborated here, as the present paper mainly serves to point out the possibility of demonstrating or even qualifying the capability of a SHM system on an operational fleet of aircraft without the need for a rigorous certification process but with a tangible benefit.

4 Conclusion

The application of SHM technology will potentially reduce the sustainment costs of new and existing military aircraft. The transition of the currently available technologies into service is very slow, however. This is mainly caused by the very challenging process to validate any SHM system's capability to reliably and accurately detect impending in-service failures, and the difficulty in defining attractive business cases. The present paper describes the possibility to field a SHM system on legacy military aircraft such as the F-16 with a minimum amount of certification issues and with a good prospect of a positive return on investment. For appropriate areas in the airframe the application of SHM will reconcile the fail-safety and slow crack growth damage tolerance approaches that can be used for safeguarding the continuing airworthiness of these areas, combining the benefits of both approaches and removing the drawbacks.

The SHM business case can be summarized as:

- fly it...
- ...but do not have to rely on it (safety)...
- ...while still benefiting from it ($$$)!

Demonstrating SHM technology on flying aircraft will increase the TRL of the demonstrated technology and the confidence in its reliability and use needed for any military aircraft operator to accept it. Seizing this opportunity would be an evolutionary step towards more challenging applications. The author hopes that the present paper will give an impetus to the SHM community to do so.

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