A Viable Opportunity for Fielding an Aircraft Structural Health Monitoring System

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Abstract

An important trend in the sustainment of aircraft is the transition from preventative maintenance to Condition Based Maintenance. 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 order to facilitate the planning process while maintaining flight safety. Much effort is being put in the development of technologies that enable CBM, including Structural Health Monitoring. The transition of innovative SHM technologies into service is very slow, however. Reasons are that business cases are difficult to define and that the certification of SHM systems is very challenging. This paper describes a possibility for fielding an aircraft SHM system with a minimum amount of certification issues and with a good prospect of a positive return on investment. For appropriate areas in the airframe of an aircraft, 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.

Keywords: Structural Health Monitoring, Condition Based Maintenance, Aircraft Structural Integrity, Damage Tolerance, Certification, Business Case.

Introduction

Aircraft operators worldwide are looking for ways and means to maintain or even improve the availability and airworthiness of their fleets of aircraft while decreasing the cost of ownership. The sustainment costs of aircraft generally constitute a substantial part of the total life cycle costs (for military aircraft typically in the order of two thirds or more), 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 – based on calendar time, flight hours or flight cycles – to Condition Based Maintenance (CBM), 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 CBM enabling technologies, among which Structural Health Monitoring (SHM) systems. Good progress has already been made 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 a few reasons for this:

- Business cases are difficult to define since CBM represents a disruptive technology that produces a paradigm shift for maintenance support and requires the allowance to fly with known defects in order to exploit their full potential [1];

- Would a viable business case exist, incorporating all the above-mentioned "hard knowledge items" into the business enterprise (e.g. adequate planning, logistics and information, reporting and maintenance procedures) is still a complex task, often overseen or neglected;
- Certification is difficult as the validation of an SHM system's capability to reliably and accurately detect impending in-service failures is extremely challenging, regulatory guidance is still lacking [1], and the procedures for obtaining maintenance credits are still being developed[1]. Also, to fully exploit the benefits of an SHM system, its required reliability should be considered in relation to the reliability of all other systems in the aircraft, including the structure. A comprehensive system engineering approach is thus needed in the design phase.

One option to validate the performance of a particular SHM system is to use a seeded fault test. This requires a high 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 one or more 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 for which the certification of an SHM system for use in aircraft is much easier. This paper describes such a case. It involves the fielding of an SHM system 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.

In order to be able to fully appreciate this opportunity, it is necessary to have some understanding of aircraft structural integrity concepts. First a basic explanation is therefore given of the damage tolerance concept that is used for ensuring the initial and continuing airworthiness of metallic aircraft structures.

Aircraft structural integrity concepts

The formation and growth of fatigue cracks in metallic structure is still considered to be the major threat to the structural safety and continuing airworthiness of military combat and transport aircraft [2]. This is also true for commercial aviation and, despite a good safety record, rules and associated regulatory guidance material are still being issued with regard to the fatigue management of metallic structure, a notable one being the introduction by the Federal Aviation Administration (FAA) of the so-called Limit Of Validity (LOV) of the engineering data that supports the structural maintenance program, in order to ensure that an airplane remains free from wide-spread fatigue damage [3].

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 (i.e. once fatigue cracks are expected to have formed, the structural component must be retired), 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 civil

[1] It is noted that the Aerospace Industry Steering Committee on Structural Health Monitoring, AISC-SHM, which operates under the auspices of SAE International, has formulated a joint aerospace industry SHM viewpoint to the FAA for commercial fixed-wing aircraft [1].

aviation airworthiness codes in the US and Europe, and also the US Air Force (USAF) standards, require aircraft structures to be damage tolerant unless this is shown to be impractical, e.g. for landing gear structure. The approaches that are taken to substantiate damage tolerance for civil and military aviation differ somewhat, however, even though the objectives (i.e. ensure safety of flight) are the same. The USAF philosophy is more prescriptive and easier to explain. This approach is therefore considered in the following.

The initial USAF damage tolerance requirements were published by the US Department of Defense in MIL-A-83444 [4], 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 damage tolerance 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 damage tolerance [5]. 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 JSSG-2006 [6] and Structures Bulletin EN-SB-08-002 [7] – 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 Fig. 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, ultrasonic, 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.

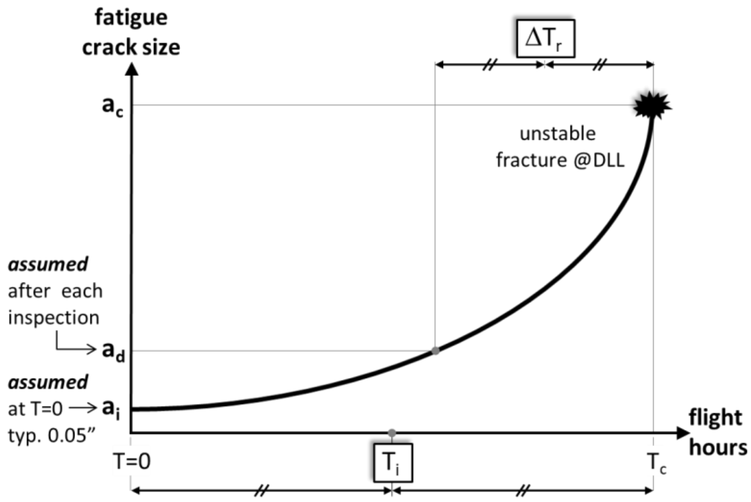

Fig. 1: Determination of inspection intervals in the slow crack growth damage tolerance concept

The minimum detectable flaw sizes used in the establishment of the recurring inspection intervals should be based on experimentally determined probability of detection 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 [8]. 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 Fig. 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 article, however.

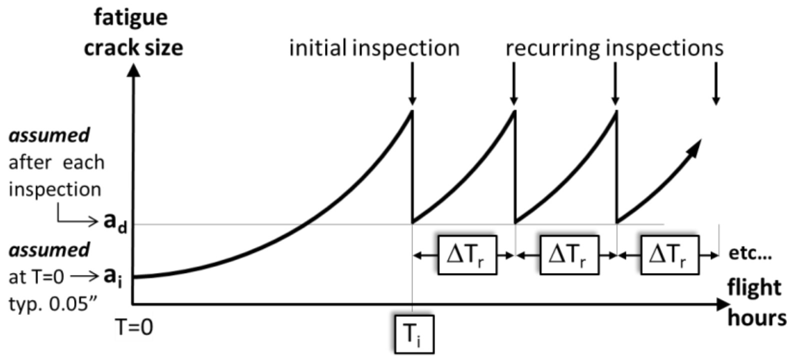

Fig. 2: In-service inspections for fatigue cracks; after each inspection the size of the assumed crack is reset to its in-service detectable size

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 to provide the basis for fail-safe assessments of legacy aircraft. In fact, the latest release of MIL-STD-1530 – which implements the USAF policies, procedures, and responsibilities that ensure aircraft structural integrity – stipulates that fail-safety damage tolerance is the preferred design concept [9]. 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 or malfunction evident, meaning that 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 [10]. Some of the

MIL-A-83444 requirements that discouraged the application of fail-safety, such as the stipulation of dependent load paths [5], 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 that 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 EN-SB-08-012 are generally larger than what was assumed previously. For legacy aircraft such as the F-16, which 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 [11]. An example is provided in Fig. 3 (left), which shows the F-16 fuel shelf of which the joint bolt holes in the upper wing carry-through bulkhead at a particular fuselage station are fracture critical. In order to be able to perform the mandated bolt hole eddy current inspection it is necessary to remove the bolts, which is very difficult. Visual inspection for large cracks in the flanges and the web of the upper bulkhead is much easier and is therefore preferred, see Fig. 3 (right). 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 bulkhead fail, limit load can indeed be carried by the adjacent bulkheads and wing attachment fittings [12], 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 a somewhat conservative value of 0.25 mm [10] – to failure. 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.

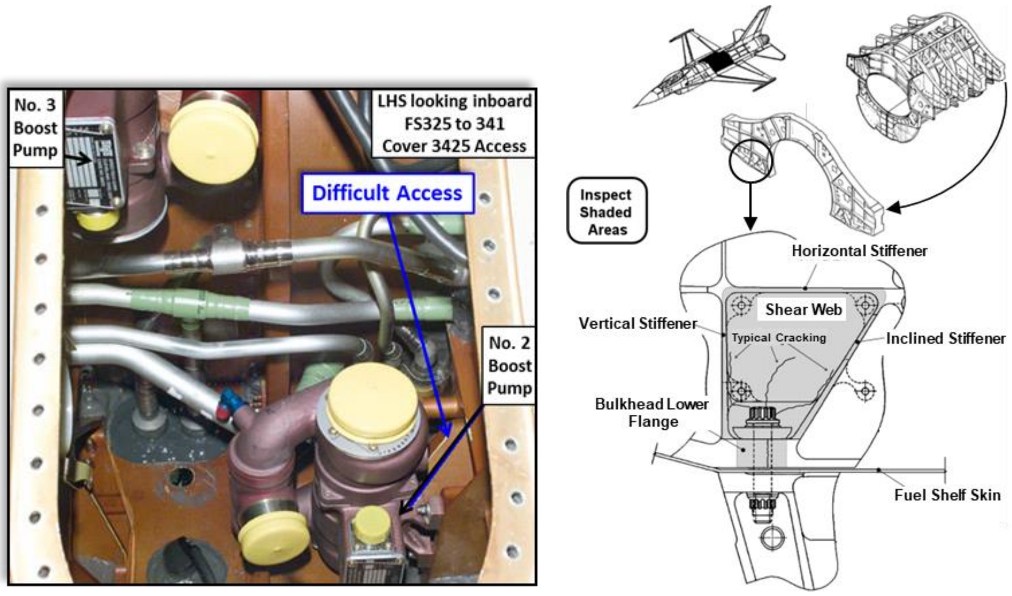

Fig. 3: Difficult access to the fracture critical F-16 fuel shelf joint bolt holes (left), easy visual inspection for large cracks in flanges and web of the upper bulkhead (right) [11]

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.

The advantages and disadvantages of the two USAF damage tolerance approaches are summarized in Table 1.

Table 1: Pros and cons of the two USAF damage tolerance approaches

Slow Crack Growth	Fail-Safety
√ well-established concept, easy to analyze	√ easy non-intrusive inspections
√ affordable repair of detected damage	√ also provides safety against discrete source damage
☹ cumbersome inspection program	☹ expensive & lengthy repair of detected damage
☹ only provides safety against fatigue	☹ complex analysis
☹ inspections may induce damage	☹ only possible for multiple redundant loadpath structure
☹ latest spec. of in-service detectable flaw sizes leads to short inspection intervals	☹ not possible throughout complete life of aircraft (Fail-Safety Life Limit)

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, 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 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 an 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 an 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. 11 is the F-16 wing root rib, which contains a number of manifold holes that are fatigue critical. NDI 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 Fig. 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. For commercial airliners, it is conceivable to install SHM sensors at stringers in aluminium fuselage structure. The current damage tolerance requirements call for an uninspectable broken stringer scenario and flight safety is ensured by the timely visual detection of large cracks in the skin. Repairs will then be relatively expensive and installing SHM sensors in the stringers may alleviate this.

Examples of potentially suitable SHM technology for the detection of small cracks are the Comparative Vacuum Monitoring (CVM) system from SMS [13], which is already used on the 737 NG [1], or the permanently-mounted conformable eddy current sensors such as those developed by Jentek [14] or the DST Group in Australia [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 an SHM system on operational aircraft without the need for a rigorous certification process but with a tangible benefit.

Conclusion

The application of SHM technology will potentially reduce the sustainment costs of new and existing 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 and implementing attractive business cases. The present paper describes the possibility to field an aircraft SHM system 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...
- ...without having 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 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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