Structural Health Monitoring (SHM) System for Rotary and Fixed-Wing Aircraft

Background

This paper describes an innovative method for Structural Health Monitoring (SHM) for helicopters based on a proven technique called, "Strain Energy Mode Shape (SEMS)." ***Strain energy*** (a type of potential energy) is stored within an elastic solid when the solid is deformed under load. In the absence of energy loss from forces like friction, damping or yielding -- *strain energy* represents the work done on the solid by external loads. Thus, the difference in strain energy, from a damaged structure compared to an undamaged structure, can be used as an indicator of damage [1]. By applying this SEMS-based methodology to the job of detecting early damage in structures using nondestructive testing techniques - a reliable, real-time, and cost effective SHM system can be created.

Introduction

Structural Health Monitoring (SHM) is an advanced, sensor based, nondestructive inspection method (NDI) conducted on all types of structures, including aeronautical components. By selecting and accurately placing sensors, and then collecting the damage-sensitive sensor data and employing advanced data analysis techniques, SHM can be used to identify micro-damage before it becomes an issue. SHM is used effectively on a variety of structures, including rotary and fixed-wing aircrafts [8]. The most important features of a SHM system are to identify the correct location for sensor placement and employ the most useful set of data sampling and data analysis techniques.

The primary objectives of SHM are to evaluate the following:

- Is the component/system damaged?
- Where is the damage located?
- What type of damage is present?
- What is the extent of damage?
- What can be learned about the overall structure being monitored?
- What is the remaining useful life or prognosis of the structure?

This paper discusses SHM in relation to rotary and fixed-wing aircraft. Topics include forces on aircraft structures, composite materials, failure modes and issues with maintenance. In addition, this paper provides an overview of Erallo's novel SEMS-based SHM system and the associated technology aspects.

SHM Considerations: Rotary and Fixed-Wing Aircraft

Although the basic aeronautical principles of flight between fixed and rotary wing aircraft are universal, the mechanical manner in which lift is generated differs – resulting in a differing profile of dynamic and static forces acting upon the various aircraft structures. Thus, when an aircraft SHM system is being considered, the sensor type, sensor location, energy harvesting selection and data analysis methodology will be unique for the type of aircraft being analyzed.

An SHM system for rotary aircraft should focus on critical components like the main gear, vertical stabilizer, rotor head and landing gear; and should include multiple sensor types, like environmental sensors, vibration sensors, strain gauges and acoustic emission sensors. Data collected from the sensors needs to be time synchronized to achieve efficient multi sensor data fusion (MSDF). The determination of the most appropriate sensor location is also of extreme importance. For example, for the rotor

Fig. 1: SHM can be used on Rotor Mechanisms

head component, SHM sensors should be located on the areas of highest stress.

In the realm of helicopters; the main rotor, tail boom and gear box components are subjected to high levels of force [8]. Due to the high amount of dynamic force on helicopter rotor blades, they are a prime example of an aircraft component that is highly susceptible to structural damage (see Fig. 1). In addition to dynamic forces; structural aging, environmental conditions, and system overload can also impact the life and reliability of rotor blades. Since rotor blades are crucial elements for the safe and proper operation of a rotary aircraft, their structural health needs to be monitored on a regular basis to prevent catastrophic failure.

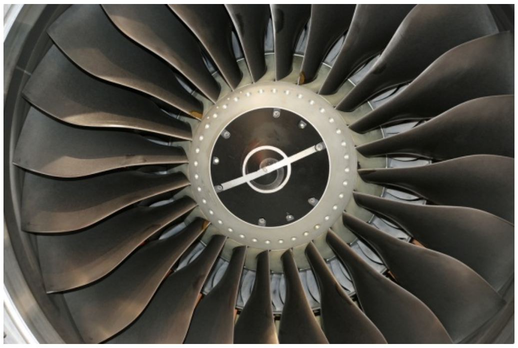


Fig 2: The leading edges on first stage rotor blades often have damage from foreign objects that is difficult to visualize

Fig 3: Damage in a compressor case [5]

Many critical aircraft components are designed using a method called Safe-Life [8, 9]. The Safe-Life method assigns extremely conservative lifespan estimates to component in order to reduce the chance of component failure. This method, although useful for reducing failure, has the downside of requiring the replacement of expensive aircraft components (such as those of the main rotor of a rotary aircraft) long before their actual, structural life-time has been reached. A well designed SHM system -- that provides real-time, continuous data and SEMS related data analysis -- could enhance or replace the Safe-Life method. Using a SEMS-based SHM system could maximize the working life of critical aircraft components and save cost, as well as increasing safety by providing specific knowledge on the structural state of given components.

Online and Ground-Based SHM

There are two broad approaches for employing a SHM aircraft system: "on-line" and "ground-based." Ground-based SHM involves using nondestructive testing methods such as modal (vibrational) analysis, ultrasound and acoustic emission. The main issue with ground-based SHM is that the helicopter must be "grounded," resulting in a more time consuming maintenance process.

On-line health monitoring involves monitoring selected in-flight system response parameters (such as vibration and blade response) and then comparing these responses to baseline reference values. The system response parameters, as compared to the baseline reference values, can be used to identify and monitor damage to a given system. An in-flight system can employ a variety of sensor types, including strain sensors that can monitor strain on blades and hub assemblies.

Fig. 4 shows the Southwest Airlines airplane where in 2011, a hole occurred in-flight on the roof of the aircraft while cruising at 35,000 feet. Federal investigators and National Transportation Safety Board (NTSB) determined metal fatigue as the cause: a crack developed in an area where two sheets of aluminum skin were bonded together.

Studies using mathematical models to analyze the effects of rotor blade damage on the overall rotor system during forward flight have been performed (Azzam and Andrew, 1992 and Ganguli et al. 1996).

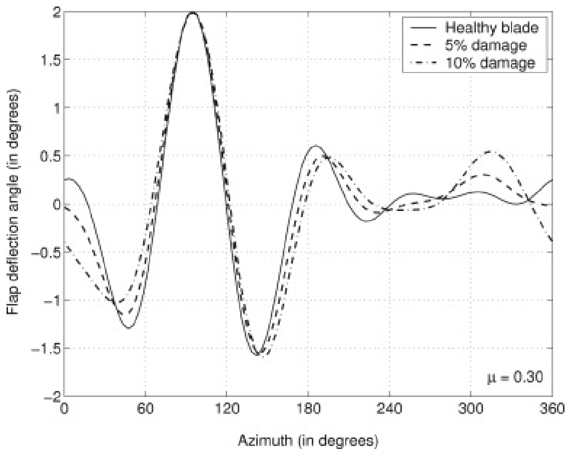


Fig. 5: Optimal Flap Control Input for a Healthy versu a Damaged Rotor Blade, μ=0.30 [6]

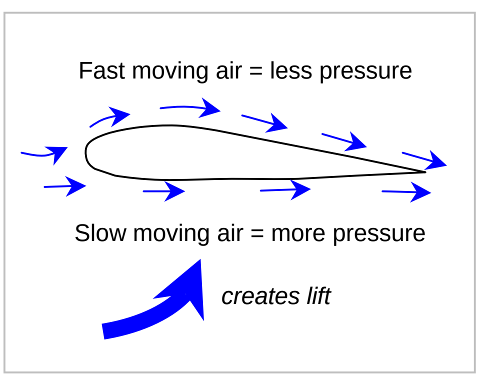
Blade damage was modeled using changes in blade stiffness and inertial and damping properties. The findings of these studies showed that there were measurable parameter changes within the rotor system between the damaged and undamaged blade models. In addition it was found that an undamaged rotor only transmits vibrations to the rotor hub, while a rotor with one damaged blade transmits many other harmonics.

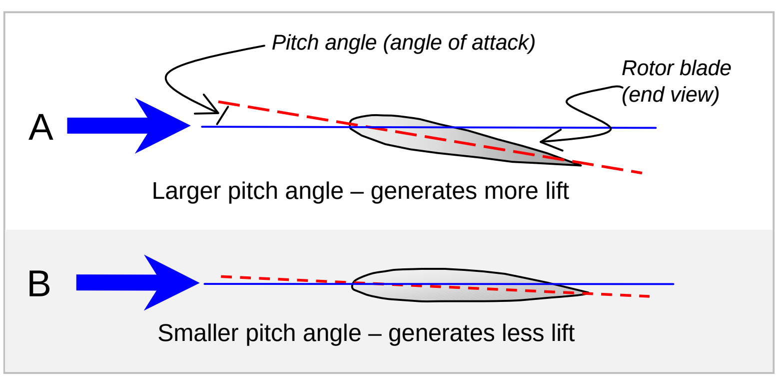
In addition, work done by Viswamurthy S.R. and Ganguli, R. in 2008, showed performance sensitivity in healthy blades versus damaged blades at various percent damage levels (see Fig. 5.)

Online SHM of helicopter rotors, using a model-based approach, can be complicated due to the difficulties in accurately predicting rotor system behavior. For example, helicopter vibration is very difficult to predict accurately even when using sophisticated aerodynamic and dynamic models, as found in a study comparing various analysis predictions with flight test data (Hansford and Vorwald, 1998).

SHM Considerations: Rotary and Fixed-Wing Aircrafts

Construction and Lift: The most obvious difference in the construction of a rotary versus a fixed-wing aircraft is the use of a rotor instead of a wing. The weight of a fixed-wing aircraft is primarily carried by the wings, whereas that of a rotary aircraft is carried by the shaft of the main rotor. With regard to lift, a fixed-wing aircraft depends on its forward motion, and the unique, non-symmetrical, cross section shape of the wing (to create a higher pressure underneath the wing, than above it). A rotary aircraft, on the other hand, has symmetrically shaped blades (with two or more blades off the main rotor) that depend on the rotation speed to create lift. Rotary lift can be controlled by adjusting the angle of attack, or pitch, of the rotor blades. When the rotor is turning and the blades are at zero angle (flat pitch), no lift is developed. Since helicopter lift and control are independent of forward speed, no runway is required for a helicopter to take off or land.





Torque Reaction: Another major difference between rotary and fixed-wing aircraft is the torque reaction. As the helicopter's main rotor turns in one direction, the fuselage of the helicopter tends to rotate in the opposite direction (Newton's third law of motion – equal and opposite forces) causing a *torque reaction*. In a single, main rotor helicopter, the usual way of minimizing the torque reaction is by utilizing a tail rotor. This rotor is mounted vertically on the outer portion of the helicopter's tail section. The tail rotor produces thrust in the opposite direction of the torque reaction (see Fig. 8).

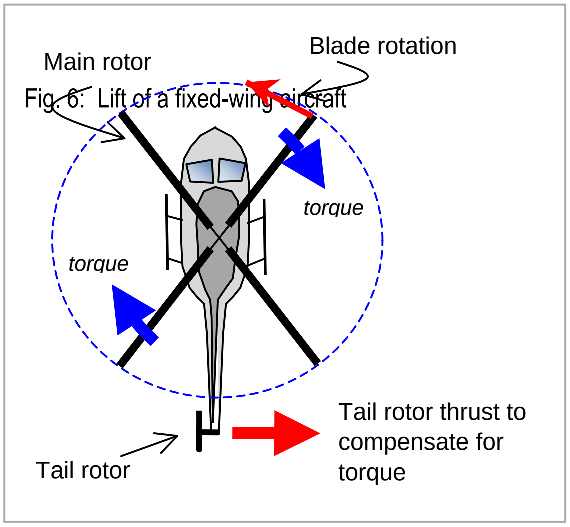


Fig. 8: Tail rotor compensates for torque of blades

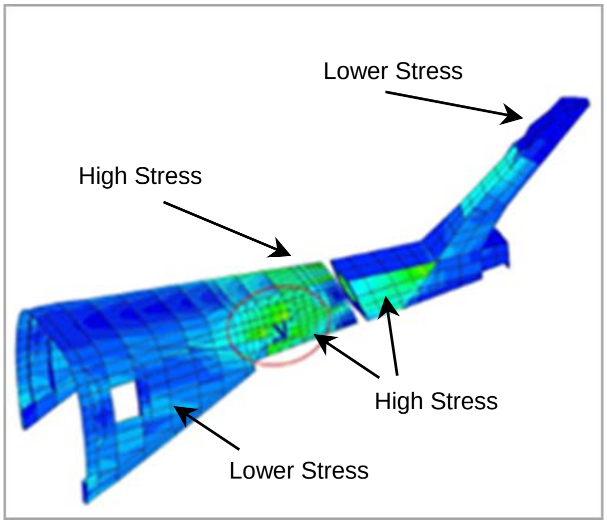


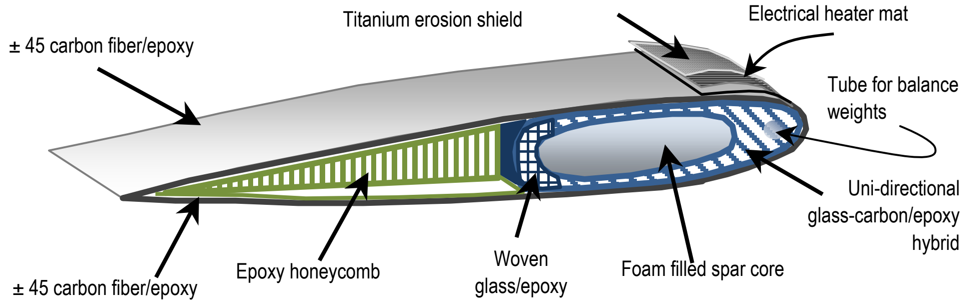
Fig. 9: FE Stress model of a tail section; hot spot analysis

The tail section of a helicopter is subject to continuous and excessive forces due to the torque generated by the main rotor and the counter torque generated by the tail rotor. This results in areas of highly concentrated stress (or hot spots) as depicted in the FE stress model diagram (Fig. 9). Areas of green indicate high stress concentrations.

SHM Challenges: Composite Materials and Micro-Cracks

The use of composite materials in the manufacturing of aircraft structures has a number of advantages, including low weight, high static and fatigue strength and the ability to manufacture large integral shell structures. Although composites have many benefits, it is known that composite materials, such as

Fig 10: Section of a typical composite helicopter rotor blade



carbon fiber/epoxy composites are inherently brittle and often exhibit a linear elastic response up to the point of failure -- with little or no plasticity. This quality makes composite structures particularly vulnerable to impact damage and the development of micro-cracks. For example, composite aircraft shells (stiffened with ribs) have been found to develop micro-cracks during routine maintenance from dropping a hammer.

Current Maintenance and Repair Processes: Current maintenance practices for composites, such as rotor system components, rely on conservative Safe-Life assumptions to ensure the highest degree of airworthiness; however, this also results in higher associated maintenance costs [9]. In addition, due to the lack of a scientifically-based fatigue and damage evolution analysis, maintenance routines for composites must rely heavily on pilot reports of flight hours and overloads, and on frequent technician visual inspections. This manual based maintenance process is prone to errors in accuracy, omissions and potential errors in judgment. Furthermore, micro-cracking cannot be detected by typical visual inspections. Impact damage to composite structures is often much more difficult for a repair technician to detect than that of metallic structures. Furthermore, the quality of a composite repair is highly dependent on technician skill and the repair procedure and is thus, more susceptible to human error than metal repairs.

Need for SHM Techniques for Composites: Due to the micro-cracking nature of aircraft composites and lack of a standardized and reliable maintenance process, there is a definite need to develop a remote SHM system capable of detecting, locating and characterizing damage. An in-flight monitoring system with the capability of detecting damage and stress/strain levels *prior to failure*, as well as estimating the Remaining Useful Life (RUL) of a given critical component, would be of immense importance. Substantial increases in passenger safety, component lifetime before replacement, and cost savings due to decreased downtime would result.

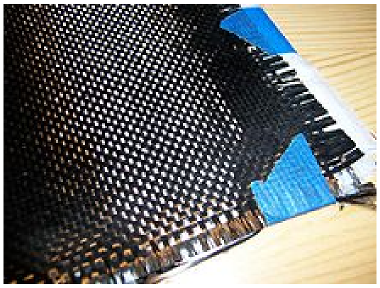


Fig 11: Carbon Fiber Cloth – commonly used in composite construction

Issues with SHM Development for Composites: At the present there are no universally accepted material laws for crash and impact simulations with composites. One reason for the lack of accepted standards is due to the orthotropic nature of composite structures. Since orthotropic materials have mechanical properties that depend on the direction in which measurements are taken; hence, a dynamic SHM system requires complex algorithms and modeling. Other factors that contribute to the complexity of composite materials include fiber type, fabric weave, the rate of dependence of polymer resin on failure modes, and non–standardized manufacturing processes.

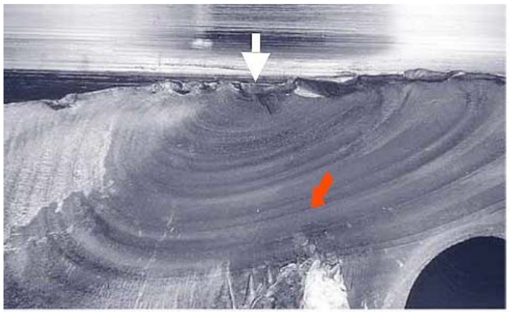


Fig 12: Fractured aircraft engine crankshaft. Beech Marks are a sign that a crack progressed across the part and failure was due to metal fatigue (red arrow). The white arrow shows the crack initiation point. [7]

Metallic Materials: In comparison to composites, the methods to determine structural health in metal components are much better understood. Extensive information exists in literature on the dynamic material properties of metal components at large strains and high strain rates. Equations to model metal behavior under impact have been implemented; for example, conventional metallic structures absorb impacts and crash energy through plastic deformation and mechanisms, like geometric folding. Still, there is a need for SHM with regard to metals, for example, bonded metals areas are often susceptible to fatigue damage.

Vibration Based Damage Detection and SHM

Vibration-based damage detection is based on the premise that damage will result in changes in the measured modal (vibration) parameters, such as energy frequencies, mode shapes and damping ratios. Therefore, vibration-based damage detection methods utilize measured vibration parameters to assess structural health (Kumar, Shenoi and Cox, 2009). Many such methods have been developed for civil engineering structures (like bridges, dams, tunnels) to enhance (or eventually replace) current visual inspection techniques. Nevertheless, there are several challenges still remaining to optimally design a robust vibration-based SHM system. One such challenge is determining the most effective placement of sensors on the structure to capture the vibration signals (Li, 2010).

Optimum Sensor Placement: Optimum sensor placement deals with both the number of sensors needed for functional and economic implementation, and the best location for the sensors. The sensor configuration should be based on the following:

- vibrational modes of interest,
- the characteristics of excitation (including the source, type and frequency range), and
- the likelihood of damage in various regions of the structure.

Although existing vibration-based SHM systems (in field use) have generated much useful information over the years, there is still debate on how the problem of test and analysis mismatch – in terms of degrees of freedom – can be tackled (Yang, 2009). Frequency sensitivity studies between damaged and undamaged structural areas (on a bridge, for example) have demonstrated that the change in frequency can be minute. For instance, even though the local stiffness drop at a damage site may be high, the global stiffness drop results in relatively small frequency changes. Therefore, it is imperative that the sensors be placed optimally and that very precise sensing systems be employed.

SHM Application: Aging Helicopters

Loose fasteners and cracks are significant issues to the serviceability of the UH-60L (Blackhawk) helicopters. The US Army has identified five structural subsystems of the UH-60L that require in-depth evaluations for cracking and loosening of fasteners. As per the UH-60 Airframe Condition Evaluation (ACE) Technical Review published on 2/17/2009, loose fasteners occur frequently in the forward beam with cracks developing in the vicinity of the transmission mounting points. It is apparent that cracks in transmission support beams were induced in the vicinities of high stresses and high stress concentration zones, primarily due to bending and/or torsion. However, axial forces under large deformations may be playing a role in these failures as well. It can be further stated that loose fasteners are highly concentrated in the cross beams near the mid-span. This further implies that the crossbeams are vibrating close to their resonance.

Fig 13: UH-60 Blackhawk Helicopter

Fatigue Issues on UH-60 Parts: In the UH-60 fuselage part (FS308) frame cracks were found at high stress concentration zones, likely induced by bending and/or by torsion. Loose fasteners were also typically found at places of high amplitude, which may have been due to inadequate bolt torque. The beaded panels had many cracks, but with fewer loose fasteners. Approximately 75% of the total defects were found on one side of the beaded panel. And cracks were particularly concentrated at the aft edge of the side panel from WL 230 to WL 250.

The UH-60 frame part, FS 485 is a closed, near-rectangular frame with twice the thickness in the bottom of the frame, as compared to the three other sides. The majority of the cracks on this part were found to be located at the top left corner and grew at a relatively constant rate. This observed cracking implies that certain design issues exist under torsional vibration.

Current Non-Destructive Testing (NDT): Current NDT techniques for detecting cracks are most effective at detecting damage *only after failure has begun*. These NDT techniques also typically require disassembly of the system, resulting in substantial downtime of the aircraft. To ensure the on-going integrity of the UH-60L airframe and to detect issues *before* failure, a new method of NDT is needed that can determine the condition and detect and identify the yield point of structural components -- without requiring the laborious disassembly and downtime of current NDT processes [8].

A smart, remote sensor-based SHM that uses validated structural analysis models and algorithms could significantly extend the safe operational life of UH-60s. This system could be used in the future to identify design issues and enhance the overall design process of an aircraft.

SEMS-based Structural Health Monitoring

As a result of an SBIR Army Grant for SHM of Army Bridges, Erallo has teamed up with the WVU-CFC (West Virginia University Constructed Facilities Center) to develop a novel structural health monitoring technique based on “Strain Energy Mode Shapes” (the methodology first developed by Dr. Hota GangaRao and his team at WVU-CFC) [1] and that integrates wireless technology with high-sensitivity stress/strain gauges, programmable custom controllers, and innovative structural analysis models and algorithms. This system can be a powerful and effective part of a conditions-based maintenance (CBM) program providing “predictive failure”, while increasing reliability and decreasing maintenance and operating costs.

As stated earlier in this paper, ***strain energy*** is a type of potential energy that is stored within an elastic structure when it is deformed under load. When there is no loss of energy from forces like friction, damping or yielding -- *strain energy* can represent the work done on the solid by external loads. Thus, the difference in strain energy between a damaged structure and an undamaged structure can be used as an indicator of the damage [2]. This method is built on the Fourier analyses of strain energy distribution. Damage peaks are analyzed through the separation of damage information in the frequency domain instead of the time domain [3, 4]. The dynamic strain energy response of a structure then clearly identifies damage due to fatigue. This SEMS-based methodology can be used to detect early damage (micro-cracking) in structures – resulting in a reliable, nondestructive testing technique that is also real-time and cost effective.

Approach to Detecting Structural Damage or Cracks: The SEMS-based SHM system focuses on identifying structural cracks due to aging, environmental loads, and stress concentrations. A damage tolerance design includes detailed failure analysis under a wide range of dynamic load conditions (bending, torsion, shear and axial) and is essential for the safe operation of an aircraft. It is assumed that some degree of crack propagation is tolerable (i.e. within the plastic zone or yield plateau) during a component’s service life. Crack propagation may be due to excessive loading, intrinsic material flaws, manufacturing errors or physical aging. There is usually a “maximum tolerable initial crack length and width” for each part of a structural component or subsystem. Conversely, the “initial tolerable crack size” is typically the minimum crack size that can be normally observed in practice before replacing the part. The “critical crack size” is the maximum tolerance of the component or subsystem. In maintenance, it is important to identify and monitor the crack size and determine the point where the remaining strength will drop below tolerable limits. This monitoring can be accomplished using commercially available, high-sensitivity foil strain gauges, including uniaxial gauges and rosettes. Based on this assessment, the operational safety of a helicopter can be granted despite the presence of cracks, as long as proper maintenance procedures are undertaken. Once the crack exceeds the tolerable threshold, the component should either be replaced or repaired to ensure safe operation of the aircraft.

Strain Gauge Measurement Capabilities*:* Typically, strain gauge data is only reliable under the following circumstances - before material yielding, soon after yielding and prior to strain hardening. Beyond the strain hardening point, the gauge itself begins to yield and the data is no longer reliable. Gauges can reliably measure strain response in the range of +/- 1500 microstrain up to a million cycles, though some gauges are capable of measuring up to +/- 2500 microstrain for a million cycles. Higher strain ranges and fatigue cycles decrease the lifespan of the gauge.

Fig. 14: Example of strain gauge

***Smart, Condition-based Sampling**:* When metal is not in stress, strain gauges cannot “capture” the micro-cracks. However, when the metal is under stress, the gauge signature will reveal any micro-cracks that are present [3]. For this reason, real-time, condition-based sampling is a key feature of the SEMS-based SHM system. Sensor sampling is a task which requires specifying parameters, such as when to start or stop sampling, and at what rate to sample. It is a critical aspect of SHM sensor based monitoring.

Sampling Parameters: The SEMS-based SHM system design allows the user to pre-define the sampling parameters, such as sampling rate and sampling duration. The default mode for the Erallo SHM system is a “*continuous sampling”* mode. The continuous sampling mode is set-up such that the sensors will begin sampling as soon as the aircraft leaves the ground, and will stop sampling as soon as the aircraft lands.

The second option is a “*conditions-based sampling”* mode which is based on starting and stopping the sampling parameters base on a pre-defined given condition. For example, the SEMS-based SHM system includes an accelerometer to monitor the G forces on the aircraft. The user may be interested in sampling the sensors only when the G forces exceed a certain value. The user interface enables the user to specify the G force value on which to start and stop sampling. Sampling will be triggered only when the specified G force value is met. Thus, the user is able to customize the sampling parameters and collect the data that is of interest.

The system can also be configured to employ wireless sensor networks and web-based services. This capability conveniently provides data access to technicians using hand-held devices (such as smart phones, tablets, notebooks and laptops) during service in maintenance facility. Real-time alarms can also be configured to alert pilots to dangerous conditions.

***Fault Detection Techniques**:* Alarm conditions can be triggered using two different methods: direct and indirect. For direct measurement techniques, such as strain gauge and displacement based measurements, the alarm will occur when the strain gauge readings exceed a certain value, such as 1/2 of the yield strain, or a deflection greater than L/1500 (with L being the span length). In the past, accelerometers or laser vibrometers have been used to “indirectly” determine the frequency response of the structure by detecting the frequency response shifts in the structure, and then computing a displacement curve through double integration, and further converting the displacement to a curvature. All sensor technologies can utilize strain energy mode shapes (SEMS) after converting the response to strain versus time.

SEMS Based Methodology: Our analyses show that the amplitude spectra are band limited to the first few harmonics and the amplitude spectrum of a crack/distress is significantly wider [1,3]. Thus, the damage in a structure creates additional harmonics in the amplitude spectrum that were not present in the undamaged structure [2]. These additional harmonics can be extracted by filtering to produce an enhanced damage peak. The resulting dynamic strain energy response of a structure clearly identifies damage due to fatigue. This SEMS-based SHM system is capable of identifying the difference between the strain energy of an undamaged and damaged structure, without the need for comparing it to baseline values, or pre-established datum. The location of damage can then be identified by peaks in the strain energy graph, as seen in the graph in Fig. 15.

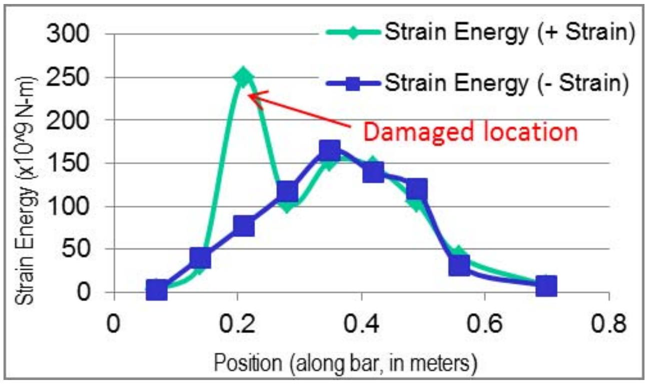


Fig. 15: Graph of strain-energy response; location of damage can be identified by the peaks [1]

Conclusion

Structural Health Monitoring (SHM) and Non Destructive Testing (NDT) techniques are of significant interest in the avionics industry and are core components of Condition-Based Maintenance (CBM) methods. Most rotary-wing aircraft in the U.S. Army's inventory are several decades old and will continue to be in service much longer than they were originally designed. Thus, a primary issue for aging military helicopters is to ensure structural integrity. In order to maintain the structural integrity of these helicopters, it is imperative that innovative and properly designed SHM systems be implemented. The implementation of these SHM systems will not only aid in the safety of structural integrity, but will reduce maintenance and operational costs (which currently attributes to approximately 25% of the direct operating cost of an aircraft).

This paper proposes a novel SEMS-based SHM system for damage detection of aging aircraft. The method is based on a modified SEMS methodology that is built on the Fourier analyses of strain energy distribution. Damage peaks are analyzed through the separation of damage information in the frequency domain instead of the time domain. The dynamic strain energy response of a structure clearly identifies damage due to fatigue. The SEMS-based SHM system is capable of identifying the difference between the strain energy of an undamaged and damaged structure – without the need for comparing it to baseline values, or pre-established datum. We believe this novel SHM system and its underlying SEMS technology will be of great importance in maintaining the structural health of all types of aircraft, as well as civil structures, like bridges, dams, tunnels, turbines, and cell towers.

References

1. Sazonov, E. S., Klinkhachorn, P., Halabe, U. B., and GangaRao, H. V. S., "Non-Baseline Detection of Small Damages from Changes in Strain Energy Mode Shapes," technical paper, Nondestructive Testing and Evaluation, Vol. 18, No. 3-4, 2002a, pp. 91-107.
2. Sazonov, E. S., Klinkhachorn, P., GangaRao, H. V. S., and Halabe, U. B., "Fuzzy Logic Expert System for Automated Damage Detection from Changes in Strain Energy Mode Shapes," technical paper, Nondestructive Testing and Evaluation, Vol. 18, No. 1, March, 2002b, pp. 1-20.
3. Napolitano K., "Damage Detection using Reduced Measurements: Analytical Investigation", 2002, retrieved from http://casl.ucsd.edu/casl/abstracts/abstract_damage_detect_ai.htm
4. Shi Z. Y., Law S. S., Zhang L. M. (1998). "Structural Damage Localization from Modal Strain Energy Change," Journal of Sound and Vibration, Vol. 218, No. 5, pp. 825-844.
5. Transportation Safety Board of Canada. Aviation Investigation Report A04P0206 (amended report); "Engine Power Loss Quantum Helicopters, LTD." MD Helicopter (Hughes) 369D C-GWPQ, Bob Quinn Airstrip, British Columbia, June 11, 2004. (http://www.tsb.gc.ca/eng/rapports-reports/aviation/2004/a04p0206/a04p0206.asp)
6. Viswamurthy S.R., Ganguli, R. (2008). "Performance Sensitivity of Helicopter Global and Local Optimal Harmonic Vibration Controller," Computers & Mathematics with Applications, Vol. 56, No. 10, November 2008, Pages 2468–2480.
7. http://mechanicsupport.blogspot.com/2011/11/metal-fatigue-cracks-and-turbo-mallards.html
8. Maley S., Plets J., Phan N.D. (2007). "US Navy Roadmap to Structural Health and Usage Monitoring – The Present and Future," Presented at the American Helicopter Society 63rd Annual Forum, Virginia Beach, VA, May 1-3, 2007. Copyright © 2007 by the American Helicopter Society International, Inc.
9. Kim Y., Sheehy S. and Lenhardt D., (2006). "The U.S. Air Force's Aging Fleets Require an Improved Structural Integrity Program," RAND Project AIR FORCE *A Survey of Aircraft Structural-Life Management Programs in the U.S. Navy, the Canadian Forces, and the U.S. Air Force*, MG-370-AF, 2006, 144 pp., ISBN: 0-8330-3862-1