Non-invasive dynamic measurement of helicopter blades

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Abstract. This paper presents the development and the application on helicopter blades of a measurement system based on FBG strain gauges. Here, the main goal is the structural characterization of the main rotor blades, with the aim of showing the potentialities of such a system in blades quality check applications, as well as in the development of structural health monitoring and rotor state feedback devices. The device has been used in both non-rotating and rotating tests, and does not require the presence of slip rings or optical joint since it is completely allocated in the rotating system. It has been successfully applied to characterize the frequency response of blades lead-lag, flap and torsion deformations, up to 250 Hz.

1. Introduction

Aircraft structures integrity is subject to several threats due to normal and abnormal operating conditions. Moreover, the production phase is not always sufficiently monitored, especially when considering processes with low quality control standards. Structural quality and integrity checks and structural health monitoring (SHM) are then areas of paramount importance in aircraft industry to reduce failures and performance degradation of aircraft structures. Nowadays, structural monitoring is mainly based on the traditional damage tolerance philosophy [1], and it is performed through the combination of visual checks and off-line non-destructive testings (NDTs). They are based on different techniques, including ultrasonic, liquid penetrant, magnetic particle, electromagnetic, laser, radiography [2]. Although this methodologies are reliable and consolidated, being offline, they rely on periodical control protocols and then present some obvious disadvantages: (i) time granularity: between one check and another, there is a non monitored period; (ii) high maintenance cost, due to the necessity of having limited amount of time between checks; (iii) decreasing of time-on-wing. Moreover, they require dedicated and usually costly equipment and dedicated spaces. This is especially true in helicopter applications, where helicopter blades also require periodic track and balance to keep low vibration levels, generated by differences of blade characteristics due to the fabrication process and use.

In-flight systems for structural monitoring are receiving more and more attention as they may increase flight safety reducing, at the same time, maintenance cost [3, 4]. This could be achieved thanks to the continuous observation of many critical aircraft components. The sensors to be used must have specific characteristic in terms of low intrusiveness and weight and the potential candidates include, among the others, micro electromechanical systems (MEMS)
and FBG strain gauges. Moreover, their applications to helicopter main rotor blades, present additional constraints due to the relative motion between rotating and non-rotating frames. The transmission of signal and power between the two frames via slip-ring or optical joints is particularly critical, making a system entirely housed in the rotating frame preferable. A drawback of such a system lies in the fact that it requires an autonomous power source directly located on the rotating system. To alleviate this problem a low-power system processing and exchanging few data is desirable. In terms of measurement capability, the in-flight health monitoring system must be able to deal with moderate lead-lag, flap and torsion deflections, typical of helicopter blades. Moreover, the monitoring system has to be able to measure and resist signal up to 50 Hz, caused by the periodic loads acting on the rotor system and by low damping values of its aeroelastic modes. Finally, for inertial (centrifugal forces) and aerodynamic (to produce small interferences) issues, optimal positioning of the sensors would be the inner part of the blade, near the root section. This makes the strain-based measurements preferable with respect to acceleration-based measurements, as in this region the acceleration due to the deformation is small, whereas the strain is maximum there.

In this paper, a quality check system based on the use of a limited number of Fiber Bragg grating (FBG) strain gauges entirely housed in the rotating frame is proposed and tested. The system is used for the structural characterization of ultra-lightweight helicopter blades, through experimental tests in non-rotating and rotating conditions. The work done in this paper has to be seen as a first step toward the development of a fully sensorized blade, allowing online Structural Health Monitoring, Rotor-State Feedback and Track and Balance.

2. Measurement setup

The measurement setup consists of: (i) a battery-powered interrogator, located at the top of rotor mast; (ii) six FBG measurement points along a single fiber, arranged following the scheme depicted in figure 1.

The sensors are fixed on the blade surface through cyanoacrylic glue, and subsequently protected along with the fiber between them with aluminum tape (figure 2). Only five sensors out of six (1,2,3,5,6) are actually used for strain measurement, whereas sensor 4 works as temperature probe to compensate thermal expansion effects on strain. Among the five sensors dedicated to strain measurements, sensors 1, 3 and 5 are parallel to the blade span, whereas sensors 2 and 6 are angled ±45° with respect to it. Sensors are used in pair (i.e., evaluating the difference between the signal from two of them): (i) sensors 1 and 5 are used for flap banding measurement; (ii) sensors 1 and 3 are used for lead-lag banding measurement; (iii) sensors 2 and 6 are used for
torsion measurement. In order to improve the selectivity of individual pairs (i.e., the capability of each pair to identify the natural frequencies pertaining the deformation they are dedicated to), a weighted difference between sensor signals may be used, on the basis of the knowledge of the sensors position with respect to the blade elastic axis [5].

3. Non-rotating tests
To perform non-rotating tests a suited flange to constraint the blade root has been designed and manufactured (figure 3). This flange has been especially designed to allow the blade rotation around its feathering axis, in order to have the possibility of analyzing blades at different pitch angles.

First of all, the proposed technique has been used to verify the accuracy of the structural properties of rotor blades identified through the experimental characterization procedure described in [6, 7]. This has been performed comparing the blade non-rotating natural frequencies, numerically predicted by using the identified experimental structural properties [8, 9], with those experimentally determined from the blade transient response to an initial excitation. Specifically, three different excitation techniques have been employed, namely: step relaxation, hammering and shaking.

Note that, since the goal of these tests was the analysis of the free blade response, an accurate measure of the excitation signal was not necessary, thus avoiding the need of expensive tools.

Figure 2. Final setup for flight testing.

Figure 3. Flange for non-rotating tests.

Figure 4. Output of two tests in response to step relaxation excitation.

Figure 5. Output of two tests in response to hammering excitation.
specifically dedicated to this purpose (like instrumented hammer); it was sufficient to analyze
the signals once the transient response was over. It is worth noting that the above experimental
approach may be easily used for rotor blades quality check.

Figures 4 to 6 show the time histories of the FBG output due to two consecutive tests per
each excitation procedures. It is possible to notice that the time histories are very similar to
each other when the same input signal is used, whereas they exhibit different harmonic contents
when different excitation methodologies are applied, essentially due to the different natural
modes excited by each of them. The step relaxation, obtained releasing a weight tied to the
blade, mostly excites low frequency modes. The hammering excitation yields a richest response
in terms of frequency content. The shaking excitation gives a spectrum strongly dependent
on the spectrum of the input, as expected. These considerations are confirmed by figure 7,
which depicts the spectra of the difference between the sensors signals from the three different
excitation techniques. Note that the flap spectrum is here also evaluated by using the sum of
signals from sensors 5 and 3 (top-left figure), with similar results with respect to the difference
between sensors 5 and 1. This is expected, since most of the deformation energy is associated to
flap modes, due to their relatively small stiffness. However, although the high frequency modes
are captured only through the hammering, the three excitation techniques give equivalent results
on the low-to-mid-frequency range.

In figures 8 and 9 the effect of moving the application point of the excitations on the PSD
of the output signals is presented, for both step relaxation and hammering. Generally, moving
the excitation point from the inner to the outer portion of the blade gives an increase of low-
frequency content of the spectrum. Moreover, the higher frequency content is only weakly
influenced by the excitation location, making the identification of high frequency modes a very
hard task. Contrary to what expected, this is the case even when the excitation input is applied
at the inner part of the blade: this behavior is probably due to the low level of deformation
energy given to the blade in this test condition. Nevertheless, performing several tests with
different excitation points is useful to reduce the possibility of having hidden modes, due to the
eventual positioning of the excitation device in one of their nodes.

Next, figure 10 shows the natural frequencies of the blades, obtained through a statistical
analysis on several set of data obtained from several tests. Note that, for graphical reasons,
frequencies are associated to a set of curves that are specific of a single test; in this sense, this
picture has to be considered only as an illustrative one. It is worth noting that, independently
from the aforementioned problems in dealing with high frequency modes, it has been possible
to identify modes up to 250 Hz. The outcome of the statistical analysis of the test data is
reported in Table 1. Results from step relaxation and hammering excitation seem to be quite
similar, except for the third torsion frequencies which differ of about 4%, whereas the shaking
Figure 7. Comparison of the spectra for different excitation techniques.

Figure 8. Effect of step relaxation excitation location on identified spectra.

tests completely miss the high-frequency modes. Moreover, since the third flap frequency has been identified by the shaking only in one test, its standard deviation is not shown. The results repeatability is quite good, with the only exception pertaining the torsional modes, for which a slightly deterioration has been detected. Finally, it is worth noting note that, due to the significant difference between flap and lead-lag bending stiffnesses, it was impossible to identify higher lead-lag modes, essentially for the practical difficulty to transmit to them a significant amount of deformation energy.

The comparison with the numerical eigenfrequencies evaluated by using the structural properties from the characterization process described in [7] is shown in table 2, in terms of their relative error. The torsion modes are missing, in that it was impossible to determine the sectional moment of inertia during the characterization process. The other modes are in quite good agreement with those experimentally identified.
Finally, using the moving block technique [10, 7], it has been also possible to identify the structural damping of the low frequency modes (see table 3), which is an important parameter for structural stability. Moreover, its change may be an indicator of some kind of damage [11], making it a possible candidate in detecting blade damages (i.e., a possible index in the definition of health monitoring criteria).

4. Rotating tests
After the non-rotating tests, the measuring system has been set up on the main rotor of a T-22 helicopter, in order to assess its in-flight reliability and performance. The flight test has been a
Figure 10. Identified non-rotating eigenfrequencies.

Table 2. Relative error on measured and predicted blade eigenfrequencies.

| Mode     | Relative error |
|----------|----------------|
| 1\textsuperscript{st} Flap | < 0.5%          |
| 1\textsuperscript{st} Lag  | < 0.5%          |
| 2\textsuperscript{nd} Flap | 6.6%            |
| 3\textsuperscript{rd} Flap | 6.8%            |

Table 3. Mean value and standard deviation of identified damping.

| Mode     | Step Relaxation | ζ Hammering | ζ Shaking |
|----------|-----------------|-------------|-----------|
| 1\textsuperscript{st} flap | Mean 0.022 | 0.016 | 0.023 |
|         | Std Dev 0.008 | 0.004 | 0.004 |
| 2\textsuperscript{nd} flap | Mean 0.047 | 0.051 | 0.068 |
|         | Std Dev 0.003 | 0.007 | 0.002 |
| 3\textsuperscript{rd} flap | Mean 0.226 | 0.224 | —       |
|         | Std Dev 0.011 | 0.013 | —       |
| 1\textsuperscript{st} lag | Mean 0.057 | 0.058 | 0.061 |
|         | Std Dev 0.005 | 0.003 | 0.001 |

low level hovering of about eight minutes from ignition to shutdown. Figure 11 shows the time history of the output of sensors 1,2,3,5 and 6 during the whole test. In order to excite blade modes, the pilot performed simple maneuvers, slightly modifying the trim controls. Figure 12 depicts a detail of the sensors signal time history after the application of a collective double-step input. Note that, from Figure 11 it is possible to clearly recognize all the flight phases, as well
as the response to the variation of the control inputs.

To highlight the harmonic content of the signal, it has been analyzed through a Fast Fourier Transform. First, three 8-seconds chunks of the signal, 1638 s - 1646 s, 1646 s - 1654 s and 1662 s - 1670 s, have been processed. The first chunk (green line in figure 13) concerns the double-step collective variation phase, the second chunk (blue line in figure 13) is immediately subsequent to this phase and the third one (red line in figure 13) is representative of the stationary hovering phase. As expected, the spectra are significantly richer than those in the static test. Several peaks, corresponding to rotating modes frequencies and to the periodic response are present. Note that, the latter are caused by the non perfect hovering flight condition, and by the aerodynamic interference between the main rotor blades and the fuselage. The spectra are almost coincident for $\omega \geq 10$ while significant discrepancies arise in the low-frequency range,
which is dominated by the rigid body motion induced by collective pitch actuation. A better insight of the spectra may be obtained from the analysis of figures 14 and 15. The former shows the output of a sensor in terms of the adimensional frequency $\nu = \omega / \Omega$, where $\Omega$ is the rotor angular speed. Several peaks are evident at integer values of $\nu$ (i.e., at frequencies multiple of $\Omega$), due to the steady periodic elastic response. The latter shows a detail of the low frequency spectrum of the signal from the flapping sensors. In addition to the peaks at frequencies $\Omega$ and $2\Omega$, other two peaks are present: one associated to the first elastic flap, and the other one to the rigid flap motion. The latter is only visible during the collective variation phase, and it is probably hidden by $\Omega$ peak during the other two flight conditions. Both flapping frequencies are coherent with the rotating ones numerically evaluated (error below 1%). The rigid flap peak is very difficult to identify in steady hovering conditions, in that it is barely visible in some cases and completely hidden in the other ones. This is particularly evident in figure 16, which shows the output of the flapping sensors regarding four phases of steady hovering flight.
Finally, with the aim of assessing the possibility of recognizing maneuvered flight phases from the strain gauges signal, an analysis on intervals of signal longer than the previous ones has been performed. Specifically, Figure 17 shows the comparison between the spectrum from a sampling period longer than 8 seconds (1638 s - 1670 s) with two 8-seconds-long periods (1638 s - 1646 s and 1646 s - 1654 s), previously analyzed. As expected, the spectra are very rich as the related periods contains frequencies associated to both control input and vehicle response. As noted before, in the range of frequencies characteristic of aeroelastic phenomena (≈ 10 Hz - 100 Hz), the spectra are significantly similar. Moreover, the increase of the sampling period (black line) reveals a low frequency peak associated to the maneuver. This fact is of great interest in the development of a structural health monitoring system. Indeed, these are based on the detection of the discrepancies between actual signals and nominal ones. Hence, one of the main challenges in the definition of reliable structural health monitoring criteria concerns the difficulty of distinguishing between blade response changes due to a damage and those caused by the voluntary input of the pilot.

Figure 15. Fast Fourier Transform of flap signals during flight, detail. --- signal during collective excitation; --- immediately after collective excitation; --- steady hovering.

Figure 16. Fast Fourier Transform of flap signals during steady hovering flights.
Figure 17. Fast Fourier Transform of flap signals during flight, detail. 

At the end of this section, it is worth noting that another important output of the flight test campaign has been the fact that the pilot did not report anomalous rotorcraft behavior due to the presence of the measuring system (interrogator and fiber optic cable) on the main rotor.

5. Conclusions and future work

The measurement system has been installed on a helicopter rotor without the need of slip rings or optical joint, which would significantly increase the complexity of the system and compromise its reliability. During the flight test no problems have been reported by the pilot related to the presence of the system. In this work the measuring system has been used for the dynamic characterization of helicopter blades both in rotating and non-rotating conditions, showing its potentialities as blades quality check system. Flight tests demonstrated that the system is able to give information on blade lead-lag, flap and torsion deformations with good accuracy up to 250 Hz, as well as the possibility of distinguishing signals associated to the helicopter rigid body motion from those pertaining the blades aeroelastic behavior. These characteristics suggest this measuring device as starting point for the development of structural health monitoring and rotor state feedback systems, which are the most interesting applications envisaged by the authors for the next future. From the technological point of view, the development of an autonomous power source not based on batteries (e.g., a small electrical generator coaxial with the mast) is mandatory for real-life health monitoring applications, as well as wireless transmission capabilities to integrate the output of the monitoring system into the avionics.

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