SOME RESULTS OF GARTEUR ACTION GROUP HC-AG 19 ON METHODS FOR IMPROVEMENT OF STRUCTURAL DYNAMIC FINITE ELEMENT MODELS

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Abstract

The issue of vibration in helicopters is of major concern to operators. This requires close attention to the vehicle dynamics. The ability to faithfully simulate and optimise vehicle response, structural modifications, vehicle updates, the addition of stores and equipment is the key to producing a low vibration helicopter. GARTEUR Action Group, HC-AG14, concluded that helicopter dynamic models are still deficient in their capability to predict airframe vibration. The AG looked at the methods for improving the model correlation with modal test data along with the suitability of existing shake test methods. The helicopter structure tested in AG14 was suspended in the laboratory. However, this is not the operational environment where there are very significant mass, inertia and gyroscopic effects from the rotor systems. Nowadays, modal analysis consists of two principal approaches: experimental modal analysis (EMA) and operational modal analysis (OMA). The EMA evaluates the modal parameters by considering that the excitation and the response of the system are both measurable. The OMA evaluates the modal parameters using only the measured response. The lack of knowledge of the input is replaced by the assumption that the input is a distributed stochastic load, constant in a broad frequency band, e.g. white noise, and uncorrelated in space. This hypothesis, nevertheless, is restrictive in rotorcraft applications, because in these cases the load is characterized by harmonic components, i.e. deterministic signals, originating from the rotating parts. A new action group HC-AG19 was formed to study the benefit of using in-flight dynamic data for improving finite element models. Methodologies were assessed to evaluate vibration measurements from flight tests. The objective is to extract modal parameters and demonstrate that the dynamic model can be updated using this data. This paper presents one of the approaches developed by the University of Rome “La Sapienza".

1. INTRODUCTION

A recent GARTEUR Action Group, HC/AG-14, concluded that helicopter dynamic models are still deficient in their capability to predict airframe vibration. The AG looked at the methods for improving the model correlation with modal test data along with the suitability of existing shake test methods.

Among others, the following recommendations were made for continued research:

- Study effects of configuration changes in the structure. How significant are these effects? How can uncertainties be handled in the context of an FE model. What is the influence of flight loads?
- The helicopter structure tested in HC/AG-14 was suspended in the laboratory. However, this is not the operational environment where there are very significant mass, inertia and gyroscopic effects from the rotor systems. Could in-flight measurements be made? What are the benefits?

The main purpose the follow-on action group HC/AG-19 is to address these two issues.

Presented at 44th European Rotorcraft Forum, Delft, The Netherlands, 19-20 September, 2018

2. THE GARTEUR ORGANISATION

The group for aeronautical research and technology in Europe (GARTEUR) is an organisation for research collaboration in Europe in the field of aeronautics. It is based on a memorandum of understanding between governments of 7 European nations with major research and test capabilities in aeronautics. It is an effective alliance with proven potential for integration of European aeronautics research and technology. Garteur was formed in 1973 by three nations. Today seven are involved in Garteur: France, Germany, Italy, the Netherlands, Spain, Sweden and the United Kingdom. There is a formal procedure for participation by organisations from non-member countries. LMS, a Belgian company, has joined hc/ag-19 on this basis.

GARTEUR focuses on collaborative research topics mainly aimed at longer-term research and technology that is essential to assure sustained European Aeronautics Industry competitiveness. The subjects of interest within the GARTEUR programme are not restricted by application, whether defence, dual use or civil. It is a unique forum of aeronautical experts from Academia, research establishments and industry that offers an opportunity for networking. GARTEUR interacts with other fora, such as EU, EREA, ASD and EDA. An overall balance of benefits between the member countries is pursued.

3. ACTION GROUP HC-AG19

The main purpose of the follow-on action group HC/AG-19 was to explore methods and procedures for improving finite element models through the use of in-flight dynamic data. For the foreseeable future it is expected that validated finite element models will be the major tool for improving the dynamic characteristics of the helicopter structural design. It is therefore of great importance to all participants that the procedure of validating and updating helicopter finite element models with such in-flight data is robust, rigorous and effective in delivering the best finite element model. The group assessed the methodology with respect to evaluating vibration measurements from flight tests where effects of aerodynamics and rotating machinery affect the vehicle response. The objective is to extract modal parameters from in-flight measured data. Advantages and disadvantages of the approaches should be given and possible future developments of the procedures presented.

4. OMA TECHNIQUES FOR HELICOPTERS

Experimental techniques based on the analysis of response-only data recorded from an operating system (OMA) have been developed [1]. Among the OMA techniques available in literature, it is worth recalling the well-known time domain technique called the Balance Realization Method (BR) [2], whereas the frequency domain techniques are Frequency Domain Decomposition (FDD) [3] and Stochastic Subspace Identification (SSI).[4]. Another technique in frequency domain, is based on the Hilbert Transform [5] and is developed by the Department of Mechanical and Aerospace Engineering at the University of Rome “La Sapienza". It is called the Hilbert Transform Method (HTM) [6].The objective of this paper is to present one of the innovative methodologies considered in HC/AG-19: the identification of modal parameters by means of the HTM applied to flight data of a helicopter in order to use them for numerical model validation. The approach relies on the use of HTM because it provides the biased FRFs (i.e. Frequency Response Function with rotor harmonics contamination). The modal parameters can be obtained from a residue/poles curve fitting method. Possible corruptions of random excitation loading due to harmonic excitations are detectable using the Entropy Index.[7].

The proposed methodology has been assessed using flight data of the AH-64D Apache Helicopter, recorded at different speed conditions. Because the structure under investigation is a helicopter, the responses contain both structural and harmonic components whose frequencies are integer multiples of the operational frequencies characterizing the main and the tail rotors. These frequencies depend on the number of blades of each rotor.[8].

The robustness of the Entropy Index to identify harmonic contributions blended in the random excitation has been investigated through a comparison with the information achievable from the application of the Kurtosis Index.[9]. Once such harmonic contributions have been recognized, the modal parameters of the structure can be separated from the operational frequencies and corresponding detection shapes. This estimate has been carried out by using OMA methods implemented into a single numerical platform called Natural Input Modal Analysis -NIMA, developed using Matlab at the Department of Mechanical and Aerospace Engineering of the University of Rome “La Sapienza". The proposed method based on HTM has been assessed using Frequency Domain Decomposition (FDD) concerning the modal parameter estimate. Once the experimental modal parameters are obtained from the actual operational conditions, a correlation with the numerical prediction is carried out using FEMtools software.[10].

Presented at 44th European Rotorcraft Forum, Delft, The Netherlands, 19-20 September, 2018

5. THE INNOVATIVE PROCEDURE

In this paper, a procedure to investigate the modal behaviour of a rotorcraft-like structure, is introduced and assessed. It is based on the usage of HTM, to find the biased FRF, coupled with the Entropy Index, to identify the operational contribution due to rotor input, and the correlation process with the numerical model. The proposed methodology can be divided in three main steps:

- The first step is focused on the recognition of the deterministic loads in the measured output signals in order to neglect them in the modal characterization on the structure. This step is the one that deals with the main weak point of OMA for rotorcraft applications, in which the harmonic components could be treated as structural ones. This is done by using the Entropy Index.
- The second step is the core of the procedure, because it has been used to estimate the modal parameters in terms of natural frequencies, damping ratios and mode shapes, in operational conditions. This is achieved using HTM.
- Finally, the third step consists of the validation of the numerical model by means of matching between numerical and experimental modal properties.

Thanks to this innovative procedure, it is possible to get an accurate estimate of the modal characteristics of helicopter in flight conditions, as shown in section 6.

6. MEASUREMENT SETUP

The capabilities of the proposed approach to identify the dynamic properties of a rotating structure have been investigated through flight tests of an AH-64D Apache Helicopter from the Royal Netherlands Air Force (RNLAF) shown in figure 1. The tests were conducted in cooperation with the Netherlands Aerospace Centre, NLR, located in Amsterdam, the Netherlands. The flight tests were conducted to validate the installation of new pods at the tips of the stub wings. See Figure 1.

The measuring instruments, that were used in order to record the deformations of the structure, are accelerometers. These were placed on the pods. The pods are circled in Figure 1, at the stub wing tip interfaces. The exact number and the position of these sensors are illustrated in Figure 2.

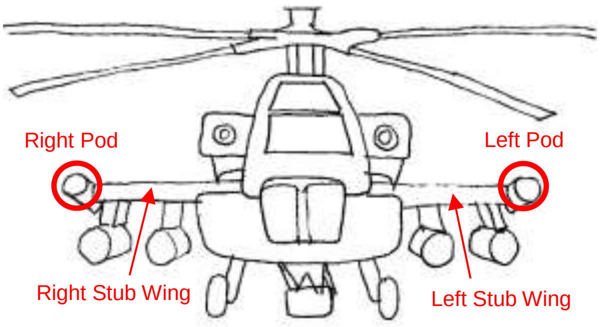

Figure 1 Apache AH-64D Helicopter with the Pods at the Stub Wing Tips.

Only four accelerometers (shown in figure 2 with red circles) are used for the analyses. The other accelerometers had very noisy signals. This was caused in most cases by local vibrations of the mounting point structural details. Accelerometers on the joints between the stub wings and the pods (3, 8 and 8A) measure only in z direction. The accelerometers which are under the seat of the co-pilot, near the centre of gravity (COG) in figure 2, measure in x, y and z directions.

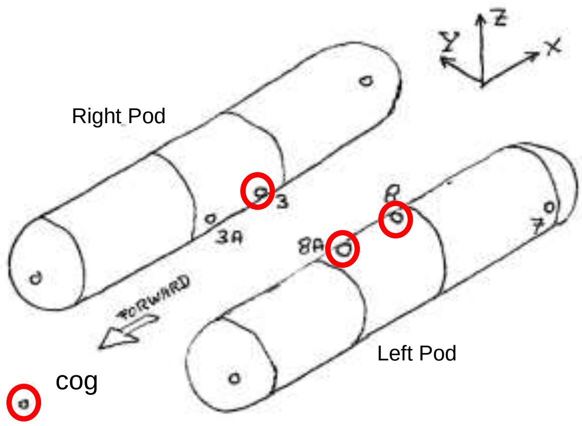

Figure 2 Accelerometer Positions on the Pods and fuselage near the COG

Note that this limited sensor set and the absence of accelerometers on the fuselage and stores severely limits the observability of global airframe mode shapes. The data was not collected with modal analysis as an objective.

Eight separate flights have been conducted in different rotorcraft configurations. However, only one configuration for the pods without pylon stores under the stub wings has been analysed.

Data is collected continuously during the whole flight. However, for each specific flight condition of interest a recording that lasts a few seconds is extracted. Besides the accelerations, several flight parameters are recorded. For example: air speed, altitude, the manoeuvres performed (e.g. climb, cruise, auto-rotation, turns) and the commands

Presented at 44th European Rotorcraft Forum, Delft, The Netherlands, 19-20 September, 2018

that are given to the control surfaces. In this paper, two different flight segments have been chosen. Both have been recorded during cruise flight at two different speeds, as shown in table 1.

Condition #	Recording #	Event
1	5	V=20 m/s; level forward
2	7	V=32 m/s; level forward

Table 1 Flight segment definition

7. IDENTIFICATION OF HARMONIC CONTRIBUTIONS

The first step of the methodology, explained in section 5, requires the identification of possible harmonic contributions. Therefore, a non-Gaussianity test of the output response signals has been performed. A narrow-band filter has been introduced in the time domain in order to find those frequencies to which the signals have an operational nature, and could not be considered as typical Gaussian responses. In particular, the methodology for the evaluation of the Entropy function can be divided in two steps: the first is the definition of a set of band-pass filters centred at each frequency available in the analysis, which is a running filter, the second is the statistical characterization of the filtered time responses. Therefore, for all the frequencies, a Butterworth filtering[a] has been carried out and the Entropy Index is calculated for each frequency. The presence of harmonic excitations in the output signals is clearly exploited from the several minima reported in figures 3 and 4, for the first and the second flight conditions, respectively. As shown in figures 3 and 4, the Entropy Index is plotted as a function of the frequency considering all responses from available channels, in order to recognize uniquely the minima representing the harmonic contributions. Indeed, only local minima which are identified from all output signals are taken into account.

As shown in figures 3 and 4 by dashed vertical lines, the frequencies, at which the Entropy Index presents the same local minima for both flight conditions, correspond to the frequencies of the main and tail rotors.[8]

[a] The Butterworth filter is a type of signal processing filter designed to have as flat a frequency response as possible in the passband. It is also referred to as a maximally flat magnitude filter.

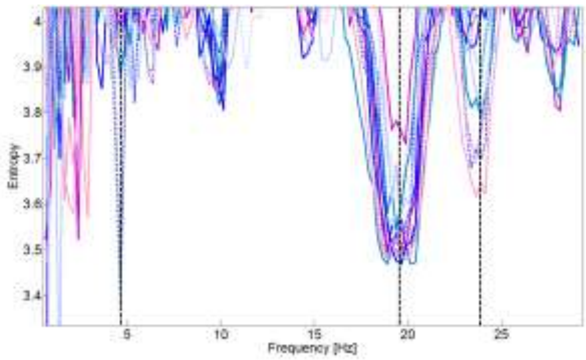

Figure 3 Effects of the 4th order of the Butterworth Filter on Entropy Index with frequency width equal to 1.25 Hz; Flight Condition 1

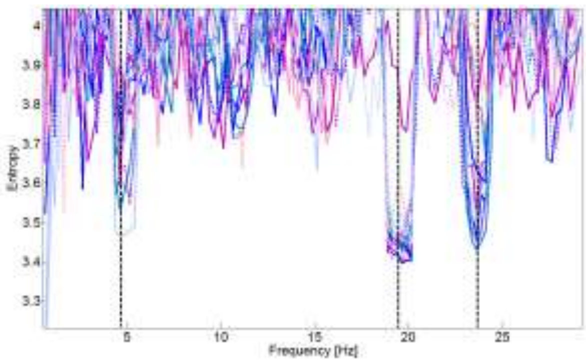

Figure 4 Effects of the 12th order of the Butterworth Filter on Entropy Index with frequency width equal to 1.25 Hz; Flight Condition 2.

The exact values of these frequencies are shown in table 2, where n is the number of main rotor blades and the blade passage frequency is the number of revolutions that the rotor makes per second, multiplied with the number of blades of the rotor. For the AH-64D Apache, n = 4.

rotor	Frequency name	expression	Value [Hz]
main	fundamental	$F_{1,m}$	4.86
main	Blade passage	$4 \cdot F_{1,m}$	19.44
tail	fundamental	$F_{1,t}$	23.60

Table 2 Rotor Frequencies[8]

The assessment of the use of the Entropy Index with respect to the use of Kurtosis has been carried out. Thus, the Kurtosis Index has been calculated on the same output signals the obtained results are shown in figures 5 and 6.

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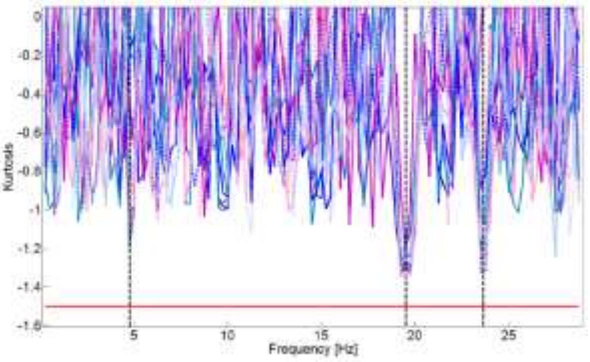


Figure 5 Effects of the 4th order of the Butterworth Filter on Kurtosis Index with frequency width equal to 0.5 Hz; Flight Condition 1.

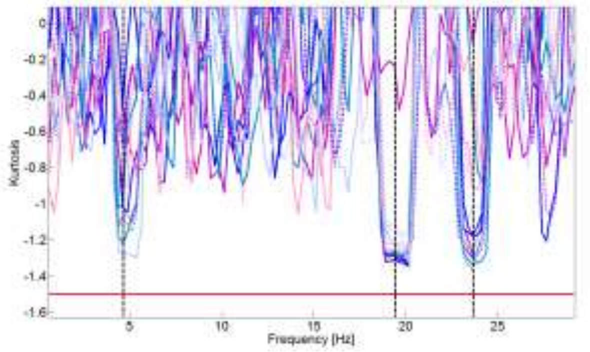


Figure 6 Effects of the 8th order of the Butterworth Filter on Kurtosis Index with frequency width equal to 1.25 Hz; Flight Condition 2.

Note that the behaviour of this harmonic identification is similar to the entropy index, even if the local minima that recognize the harmonic contributions are not clearly visible. Indeed, no local minima reach the formally required value of -1.5. Only a minimum condition is reached, when approaching the frequency of a harmonic component. So, for this reason, the Entropy Index is more robust because it does not require a prescribed value for the harmonic identification, as in the case of the Kurtosis Index. This result is very important, because it is the first time that Entropy Index is validated by comparison with Kurtosis Index by using flight data of a real helicopter. Moreover, this is a confirmation of outcomes already found and shown in [7].

8. MODAL PARAMETERS ESTIMATE

Once the frequencies of the harmonic excitations have been recognized, the next step of the methodology is devoted to the estimate of the modal parameters. Thus, firstly the Hilbert Transform Method has been applied, in order to estimate the biased frequency response functions of the vibrating structure, skewed by the presence of the harmonic loading. In this way, it is possible to identify the structural poles by means of the stabilization diagram, excluding the operational poles found by using the Entropy Index, as shown in figures 7 and 8 for the first and the second flight conditions, respectively. The structural poles are indicated by Bordeaux straight lines and the operational ones are encircled with a blue line.

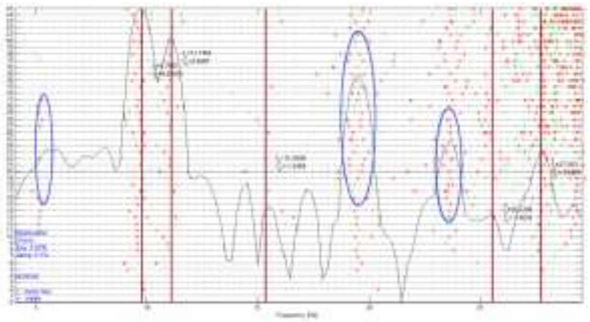


Figure 7 Stabilization Diagram of the Systems Poles for Flight Condition 1.

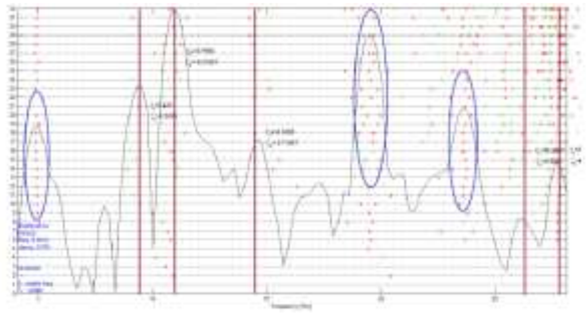


Figure 8 Stabilization Diagram of the Systems Poles for Flight Condition 2.

From the figures 7 and 8, it is possible to note that without using the Entropy Index the recognition of the the harmonic poles among the structural ones is quite quite hard, because the stabilization does not give information on the nature of the poles. In fact, the structural and harmonic poles are comparable in terms terms of shape and the amplitude. The obtained modal modal parameters using the -NIMA- platform, i.e. natural frequencies, damping ratios and mode shapes, shapes, are shown in note: modes 1 and 2 differ in fuselage deformation. See Figure 16.

Table 3 and Table 4 and in Figure 9.

Mode #	f_n [Hz]	ζ_n [%]	description
1	9.77	0.82	1st Bending of Stub Wings
2	11.12	2.05	1st Bending of Stub Wings
3	15.34	1.35	2nd Bending of Stub Wings
4	25.53	1.48	Torsion of left Stub Wing
5	27.71	1.24	Torsion of right Stub Wing

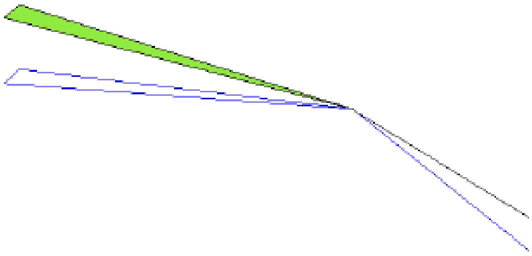
note: modes 1 and 2 differ in fuselage deformation. See Figure 16.

Presented at 44th European Rotorcraft Forum, Delft, The Netherlands, 19-20 September, 2018

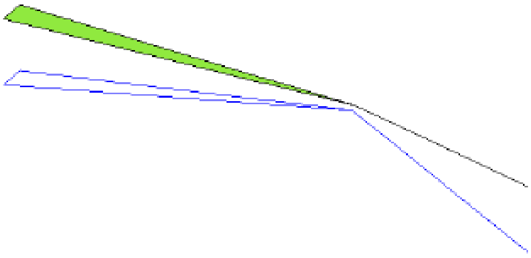
Table 3 Natural Frequencies, Damping Ratios and Mode Shapes for Flight Condition 1 using HTM.

Mode #	f_n [Hz]	ζ_n [%]	description
1	9.42	4.24	1st Bending of Stub Wings
2	11.00	0.22	1st Bending of Stub Wings
3	14.51	0.71	2nd Bending of Stub Wings
4	26.30	0.60	Torsion of left Stub Wing
5	27.83	0.97	Torsion of right Stub Wing

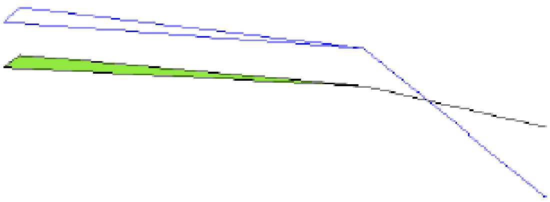
Table 4 Natural Frequencies, Damping Ratios and Mode Shapes for Flight Condition 2 using HTM.



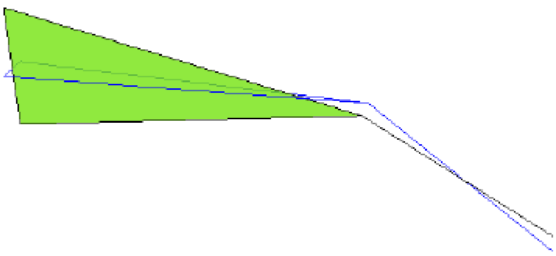
(a) Mode Shape 1: 1st Bending of Stub Wings.



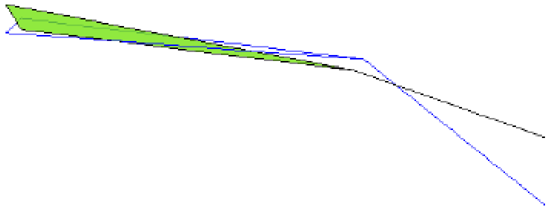
(b) Mode Shape 2: 1st Bending of Stub Wings.



(c) Mode Shape 3: 2nd Bending of Stub Wings.



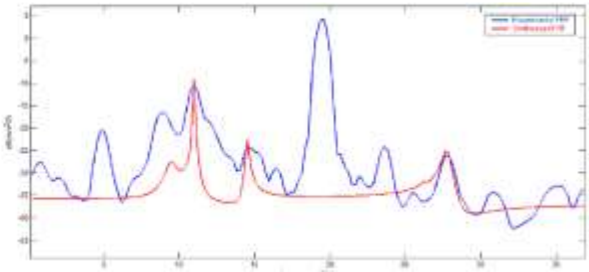
(d) Mode Shape 4: Torsion of Left Stub Wing.



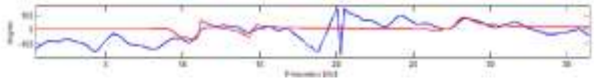
(e) Mode Shape 5: Torsion of Right Stub Wing.

Figure 9 Mode Shapes of the Stub Wings.

Moreover, by using HTM there is the advantage that the frequency response functions are resynthesized excluding the harmonic components, as shown in Figure 10.



(a) modulus



(b) phase

(Red = Synthesised, Blue = Experimental)

Figure 10 Synthesis of the FRF with the residual estimate process for Flight Condition 2

Finally, in order to demonstrate the efficiency of the proposed method, the same flight data are analysed by using another OMA method, FDD - Frequency Domain Decomposition as explained in section 3. In this case, the structural modes are estimated by selecting the peaks of the average of the normalized singular value of the PSD matrix, neglecting the peaks recognized by Entropy Index as an operational mode, as shown in figures 11 and 12. The chosen peaks are shown figures 11 and 12 and the estimated modal parameters in tables 5 and 6 for the first and second flight conditions, respectively. Note that the structural peaks in figures 11 and 12 are identified by black straight lines, whereas the operational ones are encircled with a burgundy line.

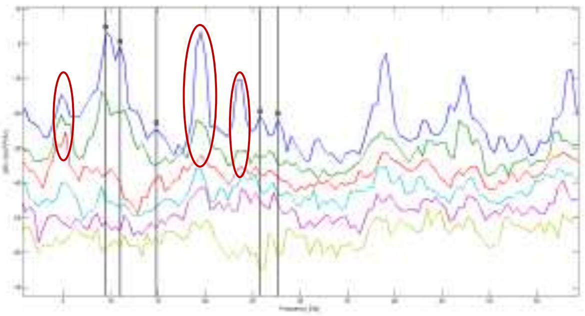


Figure 11 Average of the normalized singular value of PSD matrix for Flight Condition 1.

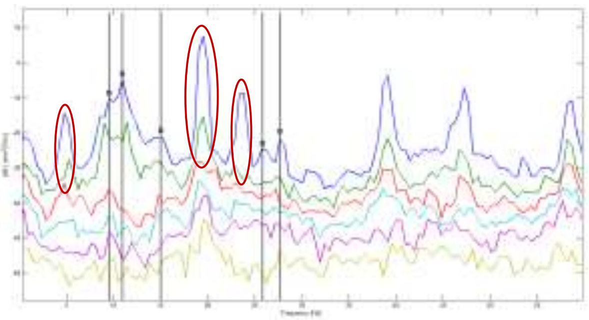


Figure 12 Average of the normalized singular value of PSD matrix for Flight Condition 2.

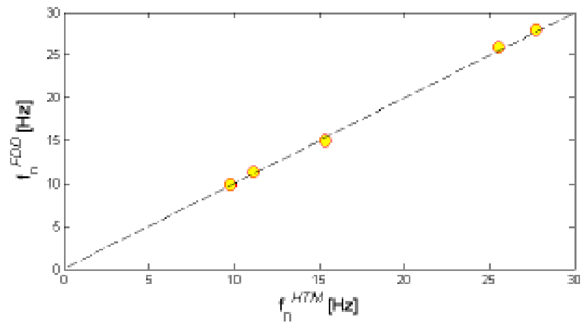
Mode #	f_n [Hz]	ζ_n [%]	description
1	9.92	3.32	1st Bending of Stub Wings
2	11.27	2.42	1st Bending of Stub Wings
3	15.00	1.36	2nd Bending of Stub Wings
4	25.94	0.78	Torsion of left Stub Wing
5	27.97	1.03	Torsion of right Stub Wing

Table 5 Natural Frequencies, Damping Ratios and Mode Shapes for Flight Condition 1 using FDD.

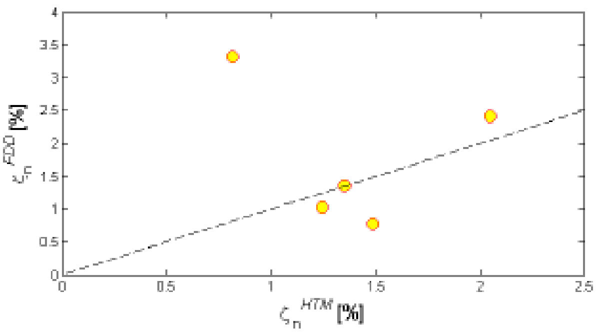
Mode #	f_n [Hz]	ζ_n [%]	description
1	9.97	3.06	1st Bending of Stub Wings
2	11.18	2.90	1st Bending of Stub Wings
3	15.47	1.31	2nd Bending of Stub Wings
4	25.94	0.78	Torsion of left Stub Wing
5	27.84	0.73	Torsion of right Stub Wing

Table 6 Natural Frequencies, Damping Ratios and Mode Shapes for Flight Condition 2 using FDD

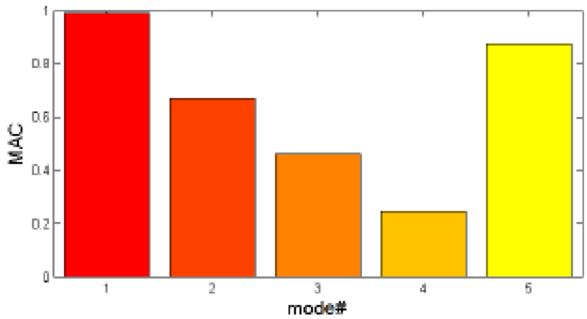
A comparison of obtained data, using the two different methods, can be performed by means of NIMA for frequencies and damping ratios with a straight 45° - slope line plot and the MAC value for mode shapes. The results are shown in Figure 13 and Figure 14.



(a) Frequency comparison

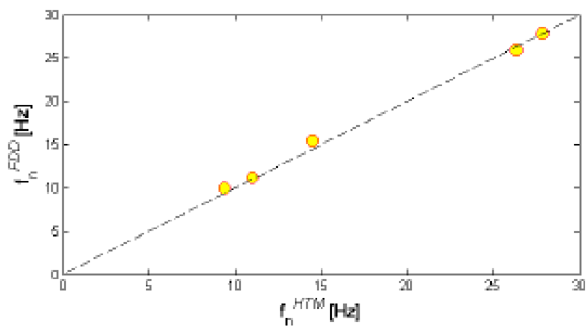


(b) Damping comparison



(c) Mode shape comparison

Figure 13 Comparison between the results obtained from FDD and HTM for Flight Condition 1.



(a) Frequency comparison

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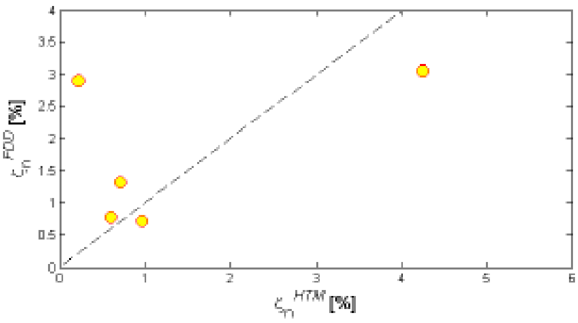
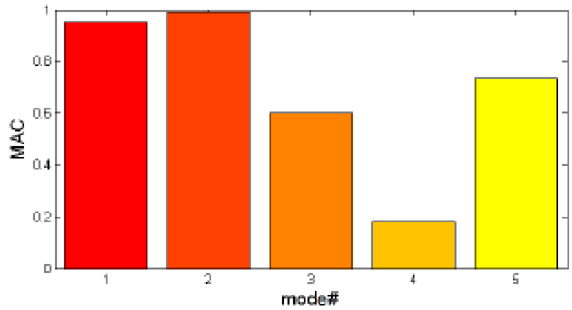


(b) Damping comparison



(c) Mode shape comparison

Figure 14 Comparison between the results obtained from FDD and HTM for Flight Condition 2.

As shown in Figure 13(a) and Figure 14(a), the frequencies are almost exactly on a straight 45° - slope line. This means that the values found by the two methods are practically the same.

The points in figures Figure 13(b) and Figure 14(b) are not on a straight 45° - slope line. The damping values do not differ a lot, except for the first mode, which is represented by the points far from the diagonal line for both flight conditions.

Note that the damping of the third mode obtained by means of HTM is almost equal to that obtained by means of FDD. This result can be explained, looking at Figure 11 and Figure 12. Recall that the method relies on the main hypothesis that system is composed by a series of independent SDOF harmonic oscillators. So, in order to achieve a good estimate of the modal parameters is necessary that the peaks are sufficiently separated from each other. The first peak is very close to the second one, whereas the third is isolated. Thus, the hypothesis of SDOF behaviour is not met, resulting in a poor estimate of the damping for the first mode. Nevertheless, the results are satisfactory, because these are typical uncertainties characterizing the damping ratio.

The calculated MAC values using mode shapes obtained from the two methods are shown in Figure 13(c) and Figure 14(c). The first, the second and the last mode shapes are well identified by the two different methods, while the third and fourth have a low value of MAC. It is possible to explain this result by the fact that the peaks corresponding to these modes have low amplitude in dB, as shown in the stabilization diagram in Figure 7 and Figure 8. As a result, their estimate is not robust.

In conclusion, it is possible to confirm that these methods give practically the same results from the accuracy point of view. However, it seems the HTM is more efficient from the numerical side.

Note that the difference in damping for the two flight conditions is small, hence aero elastic damping effects do not play a significant role (no dependency of frequency or damping on airspeed). For this reason, it is possible to use the structural finite element model, in which aerodynamic effects are not included, to perform the correlation with experimental data.

9. NUMERICAL MODAL PARAMETERS

A finite element model of the helicopter has been developed for MSC.NASTRAN by NLR. The airframe is modelled using bar (cbar), beam elements (cbeam), quadrilateral plate elements with membrane behaviour (cquad4), triangular plate elements (ctria3), and concentrated masses rigidly connected to local structural elements. See Figure 15.

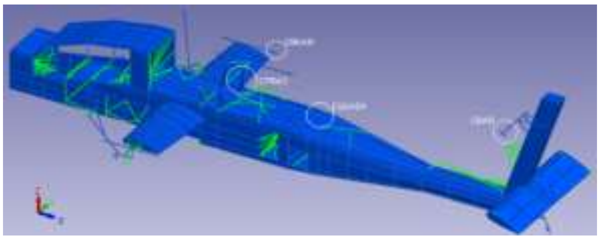


Figure 15 Finite Element Model of the Apache AH-64D

The finite element model used is the latest available version, i.e. it has been previously updated using data from ground vibration tests. The pods are modelled with a one-dimensional element (CBEAM) and joined to the stub wings by means of rigid connections. The value of concentrated masses represents the aircraft configuration and the fuel level in the fuel tanks, according to data reported by the pilot. Corresponding modal parameters are presented in Table 7.

Mode #	f_n [Hz]	description
1	9.01	1st Bending of Stub Wings
2	9.98	1st Bending of Stub Wings
3	13.80	2nd Bending of Stub Wings

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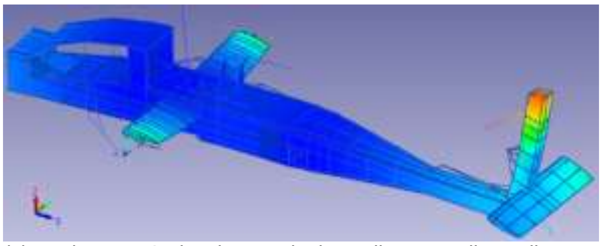
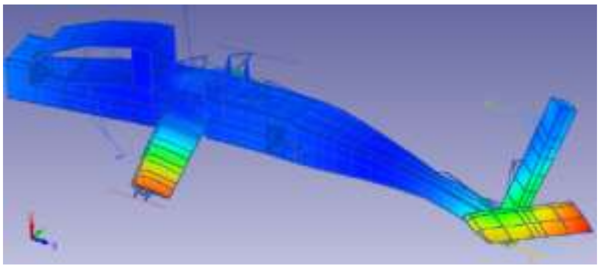
4	29.38	Torsion of left Stub Wing
5	30.87	Torsion of right Stub Wing

Table 7 Natural Frequencies for the FE model

Note that the first and the second modes are both characterized by an equal stub wing vertical bending and by a different tail deformation, as illustrated in Figure 16. Bending and torsion of the tail occur in the first and second modes, respectively.
This result is not observable from the mode shapes obtained from the operational modal analysis, as shown in Figure 16(a) and Figure 16(b), because no accelerometers are positioned on the tail.



(a) Mode 1: 1st Stub Wing Vertical Bending, 1st Tail Bending.



(b) Mode 2: 1st Stub Wing Vertical Bending, 1st Tail Torsion

Figure 16 Numerical Modes of Apache AH-64D.

10. CORRELATION BETWEEN NUMERICAL AND EXPERIMENTAL MODELS

Once the modal parameters have been estimated from the flight data and the simulation has been performed, the correlation between the numerical and experimental models can be carried out. This operation is important because it is needed in order to validate the numerical model. In this paper two correlation criteria have been used: the error percentage ε between the frequencies and the modal assurance criterion (MAC), which verifies the consistency of obtained mode shapes. The results of the correlation, using the experimental modal parameters obtained from HTM, are shown in Table 8 and Table 9. The 3D representation of the MAC, in which it is possible to check if the mode shapes are coupled, is shown in Figure 17 and Figure 18.

Mode #	fea	freq [Hz]	oma	freq Hz	ε [%]	MAC [%]
1	1	9.01	1	9.77	-7.75	89.2
2	2	9.98	2	11.12	-10.23	83.7
3	4	13.80	3	15.34	-10.00	32.4
4	12	29.38	4	25.53	15.07	16.0
5	14	30.87	5	27.71	11.42	82.1

Table 8 Frequency and MAC by using HTM; Flight Condition 1

Mode #	fea	freq [Hz]	oma	freq Hz	ε [%]	MAC [%]
1	1	9.01	1	9.42	-4.24	84.0
2	2	9.98	2	11.00	-9.27	98.5
3	4	13.80	3	14.51	-4.86	66.2
4	12	29.38	4	26.30	11.73	11.4
5	14	30.87	5	27.83	10.94	65.7

Table 9 Frequency and MAC by using HTM; Flight Condition 2

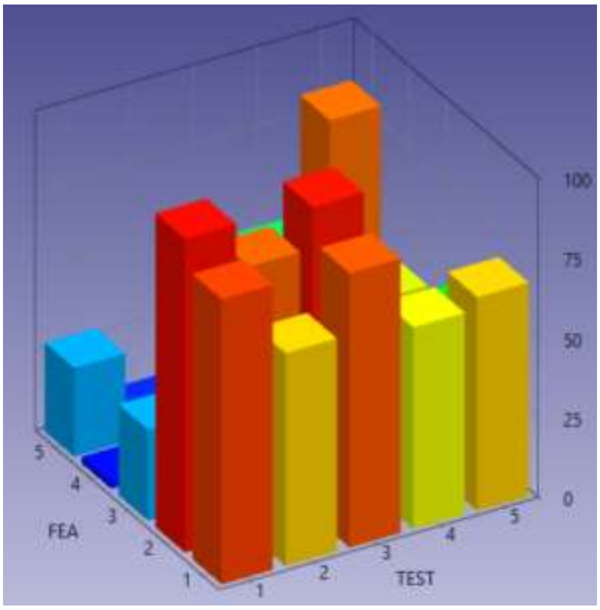


Figure 17 MAC for Flight Condition 1 using HTM

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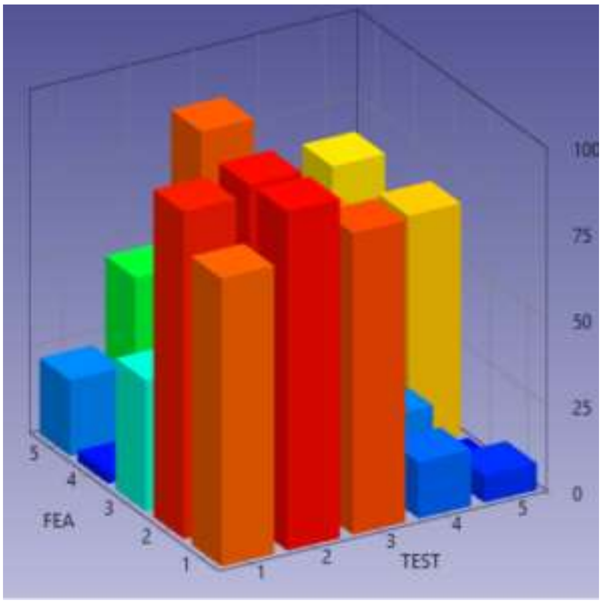

Figure 18 MAC for Flight Condition 2 using HTM

The error between the natural frequencies is remarkable, especially for the correlation done using the data which belong to the first flight condition, as shown in Table 8. With regard to the MAC values, shown in Table 8 and Table 9, note that the values of the first and the second mode are very high. The MAC values of the fourth mode are very low. This result is due to the estimate of the modal parameters. Indeed, as already explained in subsection 8, the peaks of first two modes are well defined, whereas the peak of the fourth mode is just noticeable. Moreover, observe that the modes are strongly coupled, as shown in Figure 17. This is due to the fact that 4 measurement points are insufficient to describe the deformation of the stub wings adequately. The correlation using the experimental modal parameters obtained from FDD has been evaluated too. The results are shown in Table 10 and Table 11 and in Figure 19 and Figure 20.

Mode #	fea	freq [Hz]	oma	freq Hz	ε [%]	MAC [%]
1	1	9.01	1	9.77	-7.75	89.2
2	2	9.98	2	11.12	-10.23	83.7
3	4	13.80	3	15.34	-10.00	32.4
4	12	29.38	4	25.53	15.07	16.0
5	14	30.87	5	27.71	11.42	82.1

Table 10 Frequency and MAC by using FDD; Flight Condition 1

Mode #	fea	**freq [Hz]**	oma	freq Hz	ε [%]	MAC [%]
1	1	9.01	1	9.42	-4.24	84.0
2	2	9.98	2	11.00	-9.27	98.5
3	4	13.80	3	14.51	-4.86	66.2
4	12	29.38	4	26.30	11.73	11.4
5	14	30.87	5	27.83	10.94	65.7

Table 11 Frequency and MAC by using FDD; Flight Condition 2

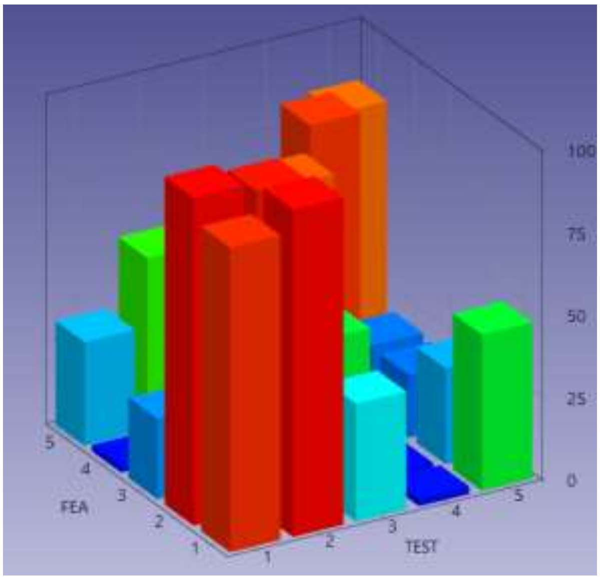

Figure 19 MAC for Flight Condition 1 using FDD.

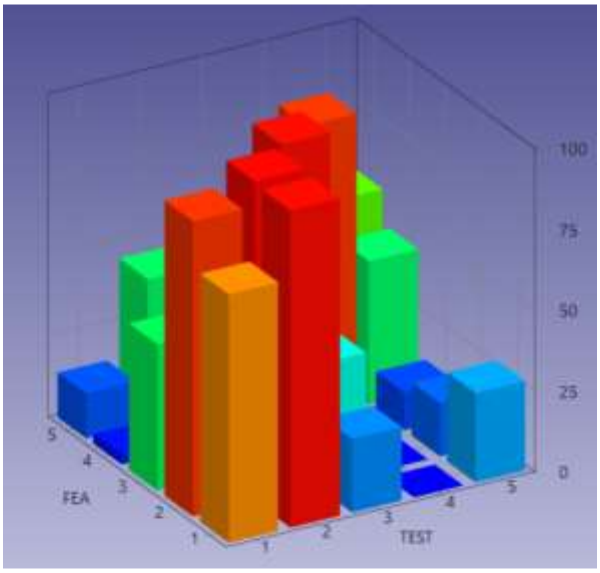

Figure 20 MAC for Flight Condition 2 using FDD

Also, in this case, as shown in Table 10 and Table 11, the error between the frequencies is high, whereas the MAC values are very high for all modes, except for the last one. This result is due

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to fact that the last mode is a right stub wing torsion. Because one measurement point is not enough to represent a torsion in a correct manner and there is only one accelerometer on the right pod, this mode is not estimated accurately. Moreover, note that compared to the previous results, the modes are less coupled, as shown in Figure 19 and Figure 20. As a general consideration, the numerical model could be considered well correlated with the experimental findings although no aeroelastic effects have been modeled. Finally, the proposed approach is found to be accurate enough to follow changes in dynamic properties of flying helicopters regardless the harmonic excitation corruption of the response time histories.

11. CONCLUSION

In this paper an approach capable to deal with the problem of estimating modal parameters of a flying structure characterized by strong harmonic excitation has been assessed.

The Entropy statistical index is used to identify possible harmonic loading contributions in the dynamic response. The Hilbert Transform Method (HTM) is used to estimate the biased Frequency Response Functions (i.e. FRFs including operational harmonic contributions) from flight test response accelerations. The modal parameters of the flying structure are estimated using any residue / pole - based estimating method applied to the “cleaned” FRFs. The identified harmonic loading contributions can be removed from the re-synthesised FRFs.

This approach has been applied to flight data of the Apache AH-64D Helicopter. Two flight conditions have been considered.

The entropy statistical index prediction of the harmonic contributions has been compared with those obtained by applying the Kurtosis Index. Results showed that the behaviour of both indices is similar. Nevertheless, it has been assessed that the Entropy Index is more robust and numerically efficient, with respect to the Kurtosis, because it does not require a prescribed value of the index for the identification of the harmonic presence, as is the case for the Kurtosis Index.

Secondly, modal parameters have been estimated using a residue/pole curve fitting technique applied to the previously identified biased FRFs from the HTM method. The estimating approach has been assessed by comparing the resulting modal parameters with those obtained by applying the FDD approach.

From the obtained results, HTM and FDD methods prove to be equivalent, but HTM has the big advantage of reconstructing the biased FRFs.

Finally, the achieved modal parameters have been used to assess the helicopter finite element dynamic model through correlation. High values of the modal correlation are reported whereas acceptable eigenfrequency shifts characterized such initial correlation, confirming the quality of the dynamic numerical model of the helicopter.

The proposed approach thus represents a promising tool in estimating modal behaviour of the structure in operational conditions with strong harmonic excitation to be used for a structural updating process.

12. ACKNOWLEDGMENTS

This paper has been developed thanks to the projects “GARTEUR AG-19”” “LR 13/2008” with the title: “Laboratori teorico-sperimentali a supporto delle applicazioni spaziali delle industrie laziali” and “Sapienza Ricerca 2016”, entitled “Identification of the dynamic properties of time-varying mechanical systems with application to launch vehicles” of Univerisity of Rome “La Sapienza".

The Dutch Ministry of Defence, Defence Material Organisation has supported this project by funding all NLR activities and test flights.

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Presented at 44th European Rotorcraft Forum, Delft, The Netherlands, 19-20 September, 2018