Utilisation des Spectres de Dommage par Fatigue (SDF) pour la qualification de composants soumis à des vibrations mesurées sur hélicoptère

Using the Fatigue Damage Spectrum (FDS) to determine flight qualification of vibrating components on helicopters

Dr. Andrew Halfpenny, Chief Technologist, HBM-nCode Products
Mr. T. C. Walton, Principal Dynamicist, AgustaWestland

1 HBM-nCode Products. AMP – Tech Centre, Brunel Way, Catcliffe, Rotherham. S60 5WG, UK
www.ncode.com

2 AgustaWestland, Yeovil, Lysander road, Somerset, BA20 2YB. UK
www.agustawestland.com

Résumé
Cette publication présente les types de vibrations rencontrées dans un hélicoptère et décrit les spécifications d’essais de qualification forfaitaires utilisées pour s’assurer de la bonne tenue des équipements. On décrira comment l’environnement vibratoire peut être représenté sous la forme de Spectres de Dommage par Fatigue (SDF) et Spectres de Réponse aux Chocs (SRC) à partir de mesures en vol. Enfin, une nouvelle technique visant à obtenir ces spectres à partir des spécifications forfaitaires est présentée. Des cas d’applications complètent ce document. Un premier exemple illustre comment ces techniques sont mises en œuvre pour comparer les sévérités d’essais établies pour différents types d’hélicoptères. Ceci permet de réduire le nombre d’essais nécessaires et donc les coûts associés mais permet aussi le déploiement rapide d’équipements lors de missions critiques. Le second exemple applique ces techniques pour obtenir une nouvelle spécification d’essais pour la qualification de composants de sécurité sur hélicoptère.

Abstract
This paper reviews the source of vibration loading on a helicopter and describes standard qualification tests that are used to ensure the safe life of equipment on the aircraft. It describes how the vibration environment can be measured in terms of a Fatigue Damage Spectrum (FDS) and Shock Response Spectrum (SRS), and describes how these are obtained from measured flight load data. It then describes a new technique to determine the same spectra directly from a description of the test profiles. Case studies are presented to illustrate how the new techniques were used to compare existing flight qualification evidence obtained for one aircraft, with requirements for another. This can eliminate needless and expensive re-testing of equipment and enables the rapid deployment of mission-critical equipment in-theatre. The case study shows how the technique was used to provide quantitative evidence to support limited flight approval, so non-compliant equipment could be deployed in-theatre without delay. Finally, a case study is presented to show how the method was used to tailor a new vibration qualification test for flight safety-critical components on an aircraft.

1 Introduction
All aircraft vibrate and all components are designed, tested and certified to survive these vibration levels over their entire service life. Helicopters are particularly challenging with regards to vibration, and this paper reports on the latest techniques for quantifying vibration-induced fatigue damage on components and structure.

Techniques discussed in this paper provide a means of comparing damage severity across different vibration tests and different aircraft platforms. This enables us to use test and service evidence obtained on one aircraft platform to qualify equipment on a new platform. This ‘read-across’ evidence has been successfully used to qualify equipment without the need for any additional vibration testing. It offers considerable cost savings and also enables a rapid path for the deployment of mission-critical equipment in military operations.

Techniques which derive tailored vibration tests based on measured flight load data are also discussed. Accelerometers record the vibration levels at a number of positions on the aircraft whilst it flies a prescribed sequence of manoeuvres. The fatigue damage dosage for each manoeuvre is calculated using a FDS (Fatigue Damage Spectrum), which effectively plots damage vs. frequency. The damage from each manoeuvre is summed over the usage profile of the aircraft to determine the whole-life damage dosage. From this profile we determine a statistically representative vibration test which contains at least the same damage content as the whole-life, but over a much shorter test period. This facilitates the provision for ‘Test Tailoring’, as specified in GAM-EG-13 (1). The approach is also facilitated by Annex A of US defence standard MIL-STD-810F (2) and also the RTCA DO-160E (3) design standard.
In this paper we review the source of vibration on a helicopter and present a mathematical technique to quantify damage using a FDS. Techniques are documented for calculating the FDS from measured accelerometer data, as well as a newly developed method for deriving the FDS directly for sine-on-random test profiles and the classical swept-sine and dwell test. Case studies are presented to describe how the analysis is used for test tailoring applications and for quantifying ‘read-across’ evidence to support flight clearance of equipment for new aircraft platforms. Studies also describe how cases for limited type approval (i.e. restricted flight envelope or service life), or experimental flight approval are assessed quantitatively using these techniques. The paper concludes by illustrating how the technique can be used to provide quantitative evidence to support an extended service life, where actual aircraft vibration levels are sufficiently lower than the original qualification test.

2 Review of background theory – Fatigue Damage Spectrum (FDS) and Shock Response Spectrum (SRS)

The basis of the theory used in this paper originates from the work of American engineer Biot in 1934. Extensive development on this basic approach was conducted by Lalanne and the French Ministry of Defence in preparation of the military design standard GAM EG-13 (1) in the 1980’s. In this section we introduce the two principal components of the approach: the Shock Response Spectrum (SRS) and the Fatigue Damage Spectrum (FDS).

The SRS is used to determine the maximum peak amplitude of loading which typically results from extreme shock events such as severe landings, impact, weapons discharge or nearby explosions. These extreme events can give rise to catastrophic failure as component stresses exceed the design strength. The FDS, on the other hand, is used to accumulate the damage caused by long term exposure to fatigue damaging vibrations which, although modest in amplitude, give rise to microscopic cracks that steadily propagate over time and lead to eventual fatigue failure.

2.1 The Shock Response Spectrum (SRS)

The SRS is used to determine the peak amplitude of loading seen during a flight event or a vibration test. The safety margin of the test can be determined by comparing the test SRS with the flight SRS. It is insufficient to simply record the highest static acceleration level because this does not account for the frequency of the vibration. Structural failure is attributable to excessive strain, and strain is proportional to displacement rather than acceleration. Therefore the damaging effect of acceleration is seen to reduce with the square of the frequency. High frequencies become less damaging than low frequencies. However, dynamic systems will be more sensitive to certain resonant frequencies and so it is always important to maintain both amplitude and frequency information. The SRS essentially represents a plot of the peak amplitude vs. frequency. A typical SRS plot of helicopter flight data compared with a typical vibration qualification test is illustrated in Figure 1. In this case the qualification test exceeds the peak in-flight levels by at least a factor of 2.

Why is the SRS preferred over more traditional methods? A peak-hold FFT analysis offers an obvious approach to calculating a frequency spectrum of peak acceleration; however, this method is somewhat restricted. To obtain a sufficiently high frequency resolution will necessitate using a large buffer of digitised values; however, this will also result in significant amplitude averaging which leads to an under-estimate of the peak. Techniques are available to overcome the problem of Joint Time-Frequency resolution and these are documented by Qian and Chen (4). The SRS technique offers a simple and effective approach which has been used extensively for many decades within the engineering industry.
Utilisation des Spectres de Dommage par Fatigue (SDF) pour la qualification de composants soumis à des vibrations mesurées sur hélicoptère

Figure 1 Comparison between in-flight shock exposure and a typical vibration test profile

The SRS was developed by American engineer Biot in 1932 (5) and predates the FFT algorithm by 30 years. To compute Biot’s Shock Spectrum the measured acceleration signal is first of all filtered by a Single Degree of Freedom (SDOF) transfer function centred on a specified natural frequency as illustrated in Figure 2. The maximum value of the filtered response is then calculated and this represents a single point in the SRS plot. This calculation is repeated over a whole range of natural frequencies to create the entire SRS. In 1934, Biot (6) published a paper on earthquake analysis and used the term ‘Shock Spectrum’ for the first time.

Biot used the SDOF response function as a frequency filter because of its ability to select a specific frequency in a manner consistent with the physical response of a structural system. It is also mathematically stable and is ideally suited to rapid time-domain convolution. Other spectra have been documented which use different filter characteristics. Rupp et al (7) describe an analogous approach based on a band-pass filter which is used by some automotive companies in Europe. However, the SDOF approach is most commonly used.

Figure 2 Schematic flowchart illustrating the SRS and FDS calculation process
The SDOF response function is dominated by a single spike located at the natural frequency ‘f_n’. At frequencies below the natural frequency, the component behaves quasi-statically, while at frequencies exceeding the natural frequency, the response is significantly attenuated. Around the natural frequency the component will respond dynamically and will become greatly amplified with its maximum response being limited only by the damping in the system.

The ratio of the maximum dynamic response to the static response is known as the ‘Dynamic Amplification’ (Q) factor. For typical 5% structural damping, this has the value of Q = 10 as illustrated in Figure 2. It is possible to vary the amplification factor Q; however, established procedure assumes a value of Q=10 for comparative analysis. This assumes that we use the same Q value when calculating the SRS in-flight and the SRS from the qualification test.

The Shock Response Spectrum (SRS) can be expressed in terms of acceleration or displacement response depending on the frequency response function used. For fatigue purposes, we are mostly interested in the displacement response. Fatigue cracks initiate and grow through the cyclic release of strain energy and, therefore, the displacement response provides a proportional relationship with the energy driving the failure. Acceleration might be the origin of the load but it is the resulting strain (displacement) that drives the structural failure. The SRS of displacement can therefore be used to quantify the damaging effect of the input acceleration for any SDOF system over a range of natural frequencies.

Biot proposed using the SDOF assumption for all components under excitation regardless of their actual frequency response. Over the past years many have contested the conservatism of this assumption when applied to components with a multi-modal response. Lalanne (8) documents a number of these studies which all conclude that the SDOF response, used in conjunction with a frequency sweep, is a suitably conservative assumption for all practical cases.

The arrival of digital computers has made it possible to calculate the SRS for long time signals very rapidly. Using the Z-transform, Irvine (9) derives the equations for a very efficient Infinite Impulse Response (IIR) filter.

### 2.2 The Fatigue Damage Spectrum (FDS)

Lalanne (10), working on the hypothesis of the Shock Response Spectrum (SRS), proposed an analogous Fatigue Damage Spectrum (FDS). This provides a relationship between fatigue damage and frequency. The FDS is calculated in the same way as the SRS but rather than simply finding the maximum displacement response, the filtered displacement response is now rainflow cycle counted and the fatigue damage obtained using a Wöhler calculation. The approach is illustrated in Figure 2. An explanation of fatigue theory and rainflow analysis is beyond the scope of this paper: for more details consult Halfpenny (11) and Downing et al. (12) respectively. Figure 3 shows a comparison between in-flight damage exposure and a typical qualification test profile.
Fatigue damage varies exponentially with respect to strain amplitude as shown in Equation 1. For comparative analysis the Basquin coefficient ‘C’ is usually taken as unity; however, the Basquin exponent ‘b’ is significant to the FDS analysis. For traditional fatigue analysis ‘b’ is obtained from fatigue tests on the material and is then modified to account for geometrical stress concentrations, etc. For FDS type analysis we are principally interested in the first failure site and this usually coincides with a geometrical stress concentration or the location of bolted, riveted, welded or soldered joints. In these cases the value of ‘b’ tends to lie in the range 3<b<6. MIL-STD-810F recommends using a value of b=4 where the loading profile is mainly broadband random, and b=6 where the loading profile is mainly sinusoidal. In practice a value of b=4 leads to a more conservative test but the analysis is always confirmed using a value of b=6 as well for prudence.

\[
D = \frac{1}{N_f} = \frac{S^b}{C}
\]

Where; D = damage, \(N_f\) = number of cycles to failure, S = cyclic stress amplitude, b = Basquin exponent and C = Basquin coefficient

2.3 Calculating the FDS and SRS\(^1\) of a sine-on-random test profile

In the previous section we looked at the basis of the SRS and FDS calculation and described how these spectra can be calculated from a time signal of measured acceleration. The SRS and FDS of the sine-on-random test profile could be calculated using the time domain technique described in Figure 2. This approach involves an inversion of the random PSD (Power Spectral Density function) to produce a representative time signal, followed by the addition of the sinusoidal tones. The process is quite straightforward but does require a very long time signal at a very high sampling frequency. The computational requirements of this approach has lead the author to develop a direct approach based on a description of the sine-on-random test profile in terms of a PSD of random vibration along with the amplitude and frequency of the superimposed sinusoidal tones. The presentation of this technique is beyond the scope of this paper; however a comparison of the time inversion approach with the direct approach is illustrated in Figure 4.

Figure 4 shows excellent correlation between the FDS using both the time inversion approach and the author’s approach. This level of correlation requires a time inversion sampling rate of at least 20 times the maximum frequency (i.e. 10kHz in this case) and a statistically representative duration of time signal. The duration of the inverted time signal was slowly increased until a convergence between successive FDS results was observed. This example required a time signal in excess of 1 million points before convergence was noted. In all cases the converged time inversion results coincide with the author’s direct method.

A comparison of the SRS shows excellent correlation in peak amplitude at the principal harmonic frequencies; however, the author’s approach tends to overestimate the levels in the intermediate range by as much as 20%. This is mainly attributable to an inability to calculate a statistically long enough time inversion and the error here is consistent with the methods proposed in GAM-EG-13 for similar estimates of the SRS from a Power Spectral Density (PSD) function.

\(^1\) Note: the term ‘Shock Response Spectrum (SRS)’ is usually replaced with the term ‘Extreme Response Spectrum (ERS)’ when it has been derived using statistical means from a PSD. Both plots are directly analogous but the terminology reflects on the origin of the original data. In this paper we use the term SRS loosely to encompass both SRS and ERS to avoid unnecessary complication.
The new direct approach offers a consistent high quality analysis capability that is both efficient in computation time and robust for the user. The user does not have to worry about errors that are due to convergence issues inherent with the time inversion approach. The direct method is fully supported by the ‘GlyphWorks Accelerated Testing’ package from HBM-nCode (13). This package also supports the direct analysis of swept sine and dwell profiles which are presented in GAM-EG-13 (1).

3 Vibration Environment on a Helicopter

3.1 Sources of helicopter vibration

The vibration spectrum of a helicopter can be described as a series of sinusoidal tones superimposed on a background of random noise. An example recorded in the fuselage of a helicopter is shown in Figure 5. The main source of these sinusoidal tones is attributable to harmonics of the main rotor. The main rotor frequency of a helicopter is relatively low (typically 3-8 Hz) and inaccuracies in the rotor track, balance or blade pitch will result in sinusoidal tones at this frequency. The main rotor frequency is often denoted by the term 1R whilst the tail rotor frequency is denoted by the term 1T. The tail rotor frequency of a helicopter is typically within the range 15-50 Hz.
When the helicopter is in flight the pitch of each blade varies cyclically along with its azimuth angle (i.e. angle of the blade relative to the fore/aft axis of the aircraft). Furthermore, the blades will slice through many turbulent eddies which arise from turbulence, aerodynamic effects of the aircraft, ground effects, blade-induced wake effects, etc. These cyclically periodic effects give rise to peaks at harmonics of the blade passing frequency as shown in Figure 5. The blade passing frequency is denoted by the term ‘nR’ where ‘n’ is the number of blades in the rotor. Most helicopters have between 2 and 6 blades in the main and tail rotors. The principal harmonics are denoted as nR, 2nR, 3nR, etc. The effect becomes less obvious for the higher order harmonics in excess of 3nR as these amplitudes are typically lower than the background random noise. The helicopter in Figure 5 uses 4 blades in both the main and tail rotors.

All components will witness significant vibration from the main rotor and this dominates the low frequency spectrum for positions throughout the aircraft. Components sited towards the tail of the aircraft will also witness principal harmonics of the tail rotor. Components that are sited adjacent to the engines and gearboxes will see additional harmonics of the engine, shaft, and gearbox meshing frequencies. It is usual practice to segregate the aircraft into regions and assume that the vibration amplitudes are similar for all equipment positioned in that region. The most commonly defined regions are:

- **Fuselage** – vibration is dominated by harmonics of the blade passing frequency of the main rotor
- **Avionics bay** (similar to fuselage but vibration isolated mounts are designed to reduce rotor-induced vibration amplitudes)
- **On or near engines** – additional sinusoidal harmonics induced through engine and gearbox harmonics and meshing frequencies
- **On or near tail rotor** – additional sinusoidal harmonics induced through tail rotor and gearbox harmonics
- **External stores and sponsons** – additional aerodynamic loads induced by downwash from the main rotor and aerodynamics of the aircraft

In most cases the vertical and lateral accelerations dominate the loading environment and the fore/aft axis is relatively benign.
3.2 Types of vibration qualification test

Vibration qualification tests are typically performed in accordance with the aircraft manufacturer’s specification or to one of the commonly used military design standards such as; US Department of Defense standard MIL-STD-810F (2), and RTCA DO-160E(3). The qualification test is performed in the following stages:

1. Initial resonance search – swept sine test to determine the resonant frequencies of the component. Ideally, resonant frequencies should not coincide with any of the principal harmonics of the aircraft. A component will usually fail qualification if low-damped resonances are encountered within avoid bands of a principal harmonic unless the supplier can prove adequate durability and the aircraft OEM can prove that the resonant issues will not adversely affect the durability of the airframe or mounting structure.
2. Endurance test – consisting of either a multiple sine-on-random vibration test or a swept sine and dwell test (as discussed in the next paragraph)
3. Final resonance search – swept sine test as per step 1 to ensure no changes in resonant frequencies which could indicate the presence of an emerging fatigue crack

Most modern vibration tests are performed using uniaxial electro-dynamic vibration rigs. The endurance portion of the test commonly uses a multiple sine-on-random vibration profile. A test duration of 16 hours per axis (repeated over x, y and z axes sequentially) is typically equivalent to 10,000 hours of operational exposure.

An alternative approach is to specify a swept sine and dwell vibration profile. This approach involves an extended swept sine test (typically 1 hour) followed by a sequence of static sinusoidal tests designed to excite the principal harmonics of the aircraft (typically 1 million cycles at each harmonic). The test is repeated for all axes sequentially.

The swept sine and dwell test profile is less efficient than the sine-on-random because each principal harmonic (sine tone) must be tested separately for 1 million cycles and this leads to a very lengthy and expensive test. The sine-on-random test profile excites all harmonics simultaneously which is more representative of the actual aircraft loading profile. Sine-on-random tests have largely superseded the swept sine and dwell test. The techniques discussed in this paper have been successfully employed as a means of converting existing swept sine and dwell test profiles to the more representative and efficient sine-on-random.

A final ‘impact (hammer) test’ is performed on the equipment as mounted on the aircraft to ensure that any additionally flexibility in the mounting does not give rise to resonances within the avoid bands of the principal harmonics.

3.3 Estimation of acceleration levels used in the vibration test

While the aircraft is at the design stage we can obtain estimates of test acceleration levels for use in preliminary qualification. Suitable estimates are provided by both MIL-STD-810F and RTCA DO-160E. Acceleration levels are provided in the form of equations where the vibration amplitude is given as a function of the principal harmonic frequency. Different equations are provided for each position on the helicopter to account for variation in vibration severity.

As measured flight data becomes available then these preliminary design estimates should be reviewed against measured data. The test aircraft is instrumented with accelerometers which record the vibration levels at several positions whilst it flies a prescribed sequence of manoeuvres. Manoeuvres are flown under various weight conditions so we can obtain a series of measured flight events that are representative of the real conditions seen in-service. The fatigue damage dosage for each flight event is calculated using a FDS. The damage from each event is summed over the usage profile of the aircraft to determine the whole-life damage dosage. An equivalent FDS can be calculated for the proposed qualification test and the test specification is iterated so the test FDS exceeds the flight FDS by an acceptable safety margin. The approach is illustrated in Figure 6.
The objective of test tailoring is to derive a qualification test that contains at least the same fatigue damage content as the real aircraft environment but in a shorter test time. As the damage is fixed then the vibration amplitude used in the test must vary with the duration of the test. A shorter test will require greater vibration amplitudes in order to achieve the same degree of damage in a shorter period. The Shock Response Spectrum (SRS) is used to compare the worst amplitude used in the test against the worst amplitude seen during flight. In most cases the worst shock load seen in flight will only occur for very short periods of time at infrequent intervals. Most of the fatigue damage will be attributed to long periods of flight at very modest vibration levels. Test tailoring uses this effect to derive the optimum test duration. The optimum test duration is achieved when the SRS of the test coincides with the SRS obtained for the worst flight condition. This allows the test to operate at the optimum acceleration level so damage is accumulated at the maximum rate without exceeding the worst loads seen in flight.

Most traditional helicopter tests are ‘over-accelerated’. This means that the loading amplitude exceeds the worst flight loads by a significant margin. This approach is justified on account of the high safety margins implicit in the design of aircraft components. However, care is required when over-accelerating a vibration test to ensure that the excessive loads do not introduce plasticity into the component which could alter the load paths and change the failure mode.

4 Case Studies

4.1 Case Study 1: Vibration qualification based on read-across evidence from other aircraft

Equipment was urgently required for deployment on a military helicopter. No vibration qualification had been performed for this aircraft; however, previous clearance had been awarded for a different helicopter type. The objective of this analysis is to compare the damage content of the original aircraft test with that required for the new helicopter and assess whether the existing qualification evidence is sufficient for flight approval on the new helicopter type.

The original sine-on-random qualification test was performed in accordance with MIL-STD-810E for equipment mounted to the fuselage. The principal rotor harmonics are different on this helicopter and the vibration levels are lower. The manufacturer’s vibration requirements for the new helicopter are expressed in terms of a swept sine and...
Utilisation des Spectres de Dommage par Fatigue (SDF) pour la qualification de composants soumis à des vibrations mesurées sur hélicoptère

dwell test. A direct comparison between the two tests was performed using the SRS/FDS approach and the results are shown in Figure 7.

Figure 7 Comparison of available qualification evidence with aircraft requirement specification

A comparison of the SRS for both tests is shown in Figure 7a. The SRS required by the new helicopter specification significantly exceeds that provided in the existing qualification evidence. However, a considerable overload is acknowledged in the new helicopter specification in order to achieve the desired test time. A comparison was therefore made against measured flight load data and this clearly shows an acceptable safety margin.

Figure 7b shows a comparison of the FDS for both tests. The frequencies of the principal harmonics are seen to vary and cumulative damage offered by the existing qualification evidence is considerable less than that required by the new helicopter specification. There is insufficient evidence to consider full type approval of the equipment at this stage.

Figure 7c shows the affect of reducing the safe operational life from 10,000 hours to 100 hours. Due to the urgent requirement of this equipment, limited flight approval was awarded for 100 operational hours and the equipment was fitted to service aircraft. During the first year of operation the equipment was re-tested to the new specification and was eventually awarded full type approval. However, it had been deployed straight away and used successfully in the field over this entire period.

Limited flight approval is also possible through a restriction of the flight envelope; i.e. by restricting some flight conditions and manoeuvres. For this type of analysis it is preferable to use measured flight load data directly in the comparison rather than using the manufacturer specification. This approach is called 'Test Tailoring' and is covered in Case Study 2.
Utilisation des Spectres de Dommage par Fatigue (SDF) pour la qualification de composants soumis à des vibrations mesurées sur hélicoptère

In many cases the existing qualification evidence proves sufficient for the new aircraft and full type approval can be awarded without recourse to additional testing. Vibration tests are often very expensive on account of the direct testing costs and the cost of the test component which is life-expired at the end of the test.

This approach to qualification has also proved useful for assessing experimental flight approval for new equipment. Qualification evidence based on fixed wing installations or transport-induced damage tests is often sufficient to allow very limited flight clearance for experimental purposes.

4.2 Case Study 2: Test tailoring of control rod vibration test

New yaw control rods and mountings were required on a helicopter. The control rods run through the main fuselage and tail cone and are subjected to additional vibration from the tail rotor, the tail rotor gearbox and the intermediate gearbox. These components are flight safety-critical and the general vibration specification was considered inadequate in this case. An alternative swept sine and dwell test had been proposed but this was unacceptably long and expensive (98 hours per axis), and the safety margin was uncertain. A test tailoring exercise was therefore authorised to determine a more appropriate sine-on-random qualification test along with a complete assessment of the inherent safety margin.

Acceleration measurements were taken over a number of flight events using triaxial accelerometers located at several positions on the helicopter. The SRS and FDS were calculated for each accelerometer and an envelope taken to represent the worst loading condition. The FDS was scaled over the aircraft usage profile as described previously to determine the whole-life damage. The SRS and FDS were calculated over a range 5–2000Hz. A comparison of the measured spectra with the standard test specification is illustrated in Figure 8.

From Figure 8, the in-flight shock response is well represented by the existing test specification over the first few rotor harmonics; however, it does not address the high frequency gearbox-induced peak. The in-flight damage response is also well represented over the first few rotor harmonics but has a negligible safety margin. The existing test specification does not address any of the high frequency gearbox-induced vibrations or the peaks at 1R and 2R which are significant on this aircraft.

4.2.1 Test tailoring process

The new test is based on a sine-on-random profile comprising the following steps:

1. Initial resonance search at a sweep rate not exceeding 1 octave/min in accordance with manufacturers existing specifications
2. 16 hour sine-on-random test in accordance with Figure 9
3. Final resonance search as per step 1
4. Repeat all above steps for each axis (x, y, z)
Utilisation des Spectres de Dommage par Fatigue (SDF) pour la qualification de composants soumis à des vibrations mesurées sur hélicoptère

This test is designed to offer clearance for up to 10,000 operational hours and test tailoring was performed using the ‘GlyphWorks Accelerated Testing’ package from HBM-nCode (13). A comparison of the SRS and FDS are illustrated in Figure 10.

### PSDrandom

<table>
<thead>
<tr>
<th>Freq.</th>
<th>PSD g²/Hz</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>0.01</td>
</tr>
<tr>
<td>300</td>
<td>0.01</td>
</tr>
<tr>
<td>2000</td>
<td>0.001</td>
</tr>
</tbody>
</table>

### Sine tones

<table>
<thead>
<tr>
<th>Freq.</th>
<th>Amp. g</th>
</tr>
</thead>
<tbody>
<tr>
<td>2R = 11Hz</td>
<td>1.1g</td>
</tr>
<tr>
<td>4R = 22Hz</td>
<td>2.2g</td>
</tr>
<tr>
<td>8R = 44Hz</td>
<td>2.2g</td>
</tr>
<tr>
<td>4T = 210Hz</td>
<td>1.0g</td>
</tr>
</tbody>
</table>

From Figure 10 we see that the new test specification offers an acceptable safety margin on both peak shock and fatigue damage. The sine-on-random test takes only 16 hours per axis as apposed to 98 hours for the previous swept sine and dwell test and this offers a significant cost savings.

These techniques provide a tailored test which accounts for the real vibration environment and avoids potential under-design issues by allowing direct control of the safety margin. In other situations test tailoring has been used to relax the original test specification where the measured usage profile is less damaging. This can reduce the inherent cost implications of over-testing, and the inherent weight implications of over-design. In some cases it has been used successfully to qualify important equipment that was previously considered inadequate.

Figure 9 Tailored vibration test based on MIL-STD-810F

Figure 10 Comparison between flight vibration exposure and tailored test

From Figure 10 we see that the new test specification offers an acceptable safety margin on both peak shock and fatigue damage. The sine-on-random test takes only 16 hours per axis as apposed to 98 hours for the previous swept sine and dwell test and this offers a significant cost saving.

These techniques provide a tailored test which accounts for the real vibration environment and avoids potential under-design issues by allowing direct control of the safety margin. In other situations test tailoring has been used to relax the original test specification where the measured usage profile is less damaging. This can reduce the inherent cost implications of over-testing, and the inherent weight implications of over-design. In some cases it has been used successfully to qualify important equipment that was previously considered inadequate.
5 Conclusion

This paper has reviewed the source of vibration on a helicopter and described standard tests that are used to ensure the safe life of equipment on the aircraft. It has described how the vibration environment can be measured in terms of a FDS and SRS, and shown how these are obtained from measured flight load data. It then described a new technique to determine the same spectra directly from a description of the sine-on-random test profile in an efficient and robust manner.

Case studies have been made to illustrate how the new techniques were used to compare existing flight qualification evidence obtained from one aircraft, with the requirements for another. This can eliminate needless and expensive re-testing of equipment and enables the rapid deployment of mission-critical equipment in-theatre. The case study shows how the technique was used to provide quantitative evidence to support limited flight approval so non-compliant equipment could be deployed in-theatre without delay. Finally, a case study was presented to illustrate how the method was used to tailor a new vibration qualification test for flight safety-critical components on an aircraft. The new qualification test was based on measured flight load data recorded under real operational conditions.

6 Bibliography


7 Definitions, Acronyms, Abbreviations

EFA: Experimental Flight Approval
FDS: Fatigue Damage Spectrum
FFT: Fast Fourier Transform
FTA: Full Type [flight] Approval
IIR filter: Infinite Impulse Response filter
LTA: Limited Type [flight] Approval
RMS : Root Mean Square
SDOF: Single Degree of Freedom System
SRS: Shock Response Spectrum