CERTIFICATION SPECIFICATIONS ACCEPTABLE MEANS OF COMPLIANCE AND GUIDANCE MATERIAL FOR LARGE ROTORCRAFT

CS-29 — AMENDMENT 11 — CHANGE INFORMATION

The European Union Aviation Safety Agency (EASA) issues amendments to certification specifications (CSs) as consolidated documents. These documents are used for establishing the certification basis for applications submitted after the date of entry into force of the applicable amendment.

Consequently, except for a note '[Amdt No: 29/11]' under the amended rule, the consolidated text of CS-29 Amendment 11 (Annex II to ED Decision 2023/001/R) does not highlight the changes introduced. To show the changes, this change information document was created, using the following format:

— deleted text is ~~struck through~~;

— new or amended text is highlighted in blue;

— an ellipsis '[…]' indicates that the rest of the text is unchanged.

Note to the reader

In amended, and in particular in existing (that is, unchanged) text, 'Agency' *is used interchangeably with* 'EASA'*. The interchangeable use of these two terms is more apparent in the consolidated versions. Therefore, please note that both terms refer to the* 'European Union Aviation Safety Agency (EASA)'.

SUBPART B — FLIGHT

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MISCELLANEOUS FLIGHT REQUIREMENTS

[…]

AMC1 29.251 Vibration

This AMC supplements FAA AC 29-2C, § AC 29.251 and should be used in conjunction with that AC when demonstrating compliance with CS 29.251.

The applicant should investigate each individual installation of the rotorcraft for compliance with CS 29.251. The absence of coupling with the rotors vibration frequencies should be demonstrated by a combination of analysis, vibration and flight tests.

Qualitative and quantitative flight tests should be performed depending on the extent of the change. For any installation, the failure of which or its attachment would have a catastrophic consequence, a fatigue evaluation should be performed when the vibrations are likely to affect the fatigue strength.

SUBPART C — STRENGTH

GENERAL

[…]

AMC1 29.307 Proof of structure

(a) Purpose

This AMC provides guidance and acceptable means of compliance with CS 29.307, which specifies the requirements for proof of structure.

(b) Related Certification Specifications

CS 29.303 'Factor of safety'

CS 29.305 'Strength and deformation'

(c) Definitions

(1) Detail: a structural element of a more complex structural member (e.g. gear teeth, joints, splices, stringers, stringer run-outs, lugs, or access holes).

(2) Subcomponent: a major three-dimensional structure which can provide a complete structural representation of a section of the full structure (e.g. main gearbox housing, gears, section of a blade, rotor spherical bearing, tension-torsion (TT) strap beams, or frames).

(3) Component: a major section of the airframe structure or mechanical assembly (e.g. main gearbox assembly, blade, main rotor hub assembly, cabin, tailboom, fin, horizontal stabiliser or transmission/upper deck) which can be tested as a complete unit to qualify the structure.

(4) Full scale: the dimensions of the test article are the same as design; fully representative test specimen (not necessarily complete airframe or mechanical assembly).

(5) New structure: a structure for which the behaviour is not adequately predicted by analysis supported by previous test evidence. A structure that utilises significantly different structural design concepts such as details, geometry, structural arrangements, and load paths or materials from previously tested designs.

(6) Similar new structure: a structure that utilises similar or comparable structural design concepts such as details, geometry, structural arrangements, and load path concepts and materials to an existing tested design.

(7) Derivative/similar structure: a structure that uses structural design concepts such as details, geometry, structural arrangements, and load paths, stress levels and materials that are nearly identical to those on which the analytical methods have been validated.

(8) Previous test evidence: testing of the original structure that is sufficient to verify the structural behaviour in accordance with CS 29.305.

(d) Introduction

As required by sub-paragraph (a) of CS 29.307, the structure must be shown to comply with the strength and deformation requirements of Subpart C of CS-29. This means that the structure must be able to support:

(a) limit loads without detrimental permanent deformation, and

(b) ultimate loads without failure.

This implies the need of a comprehensive assessment of the external loads (addressed by CS 29.301), the resulting internal strains and stresses, and the structural allowables.

CS 29.307 requires compliance for each critical loading condition. Compliance can be shown by analysis supported by previous test evidence, analysis supported by new test evidence or by test only. As compliance by test only is impractical in most cases, a large portion of the substantiating data will be based on analysis.

There are a number of standard engineering methods and formulas which are known to produce acceptable, often conservative, results especially for structures where load paths are well defined.

Those standard methods and formulas, applied with a good understanding of their limitations, are considered to be reliable analyses when demonstrating compliance with CS 29.307. Conservative assumptions may be considered in assessing whether or not an analysis may be accepted without test substantiation.

The application of methods such as the finite element method or engineering formulas to complex structures in modern aircraft is considered to be reliable only when validated by full-scale tests (ground and/or flight tests). Experience relevant to the product in the utilisation of such methods should be considered.

(e) Classification of structure

(a) The structure of the product should be classified into one of the following three categories:

(1) new structure

(2) similar new structure

(3) derivative/similar structure

(b) Justifications should be provided for classifications other than new structure. Elements that should be considered are:

(1) the accuracy/conservatism of the analytical methods; and

(2) the comparison of the structure under investigation with a previously tested structure.

Considerations should include but are not limited to the following:

- external loads (bending moment, shear, torque, etc.);
- internal loads (strains, stresses, etc.);
- structural design concepts such as details, geometry, structural arrangements, load paths;
- materials;
- test experience (load levels achieved, lessons learned);
- deflections;
- deformations;
- extent of extrapolation from test stress levels.

(f) Need and extent of testing

The following factors should be considered in deciding the need for and the extent of testing including the load levels to be achieved:

(a) the classification of the structure (as above);

(b) the consequence of the failure of the structure in terms of the overall integrity of the rotorcraft;

(c) the consequence of the failure of interior items of mass and the supporting structure to the safety of the occupants.

Relevant service experience may be included in this evaluation.

(g) Certification approaches

The following certification approaches may be selected:

(a) Analysis, supported by new strength testing of the structure to limit and ultimate load. This is typically the case for a new structure.

Substantiation of the strength and deformation requirements up to limit and ultimate loads normally requires testing of subcomponents, full-scale components or full-scale tests of assembled components (such as a nearly complete airframe). The entire test programme should be considered in detail to ensure that the requirements for strength and deformation can be met up to limit load levels as well as ultimate load levels.

Sufficient limit load test conditions should be performed to verify that the structure meets the deformation requirements of CS 29.305(a) and to provide validation of internal load distribution and analysis predictions for all critical loading conditions.

Because ultimate load tests often result in significant permanent deformation, choices will have to be made with respect to the load conditions applied. This is usually based on the number of test specimens available, the analytical static strength margins of safety of the structure and the range of supporting detail or subcomponent tests. An envelope approach may be taken, where a combination of different load cases is applied, each one critical for a different section of the structure.

These limit and ultimate load tests may be supported by detail and subcomponent tests that verify the design allowables (tension, shear, compression) of the structure and often provide some degree of validation for ultimate strength.

(b) Analysis validated by previous test evidence and supported with additional limited testing. This is typically the case for a similar new structure.

The extent of additional limited testing (number of specimens, load levels, etc.) will depend upon the degree of change, relative to the elements of sub-paragraphs (e)(b)(1) and (2).

For example, if the changes to an existing design and analysis necessitate extensive changes to an existing test-validated finite element model (e.g. different rib spacing), additional testing may be needed. Previous test evidence can be relied upon whenever practical.

These additional limited tests may be further supported by detail and subcomponent tests that verify the design allowables (tension, shear, compression) of the structure and often provide some degree of validation for ultimate strength.

(c) Analysis, supported by previous test evidence. This is typically the case for a derivative/similar structure.

Justification should be provided for this approach by demonstrating how the previous static test evidence validates the analysis and supports showing compliance for the structure under investigation. Elements that need to be considered are those defined in sub-paragraphs (e)(b)(1) and (2).

For example, if the changes to the existing design and test-validated analysis are evaluated to ensure that they are relatively minor, and the effects of the changes are well understood, the original tests may provide sufficient validation of the analysis and further testing may not be necessary. For example, if a weight increase results in higher loads along with a corresponding increase in some of the element thickness and fastener sizes, and materials and geometry (overall configuration, spacing of structural members, etc.) remain generally the same, the revised analysis could be considered to be reliable based on the previous validation.

(d) Test only

Sometimes no reliable analytical method exists, and testing must be used to show compliance with the strength and deformation requirements. In other cases, it may be elected to show compliance solely by tests even if there are acceptable analytical methods. In either case, testing by itself can be used to show compliance with the strength and deformation requirements of CS-29 Subpart C. In such cases, the test load conditions should be selected to ensure that all critical design loads are encompassed.

If tests only are used to show compliance with the strength and deformation requirements for a single load path structure which carries flight loads, the test article should be of the minimum acceptable material quality or alternatively the test loads should be increased to account for variability in material properties. In lieu of a rational analysis, for metallic materials, a variability factor of 1.15 applied to the limit and ultimate flight loads may be used. If the structure has multiple load paths, no material correction factor is required.

(h) Interpretation of data

The interpretation of the substantiation analysis and test data requires an extensive review of:

— the representativeness of the loading;

— the instrumentation data;

— comparisons with analytical methods;

— the representativeness of the test article(s);

— the test set-up (fixture, load introductions);

— load levels and conditions tested;

— test results.

Testing is used to validate analytical methods except when showing compliance by test only. If the test results do not correlate with the analysis, the reasons should be identified, and appropriate action taken.

This should be accomplished whether or not a test article fails below ultimate load.

Should a failure occur below ultimate load, an investigation should be conducted for the product to reveal the cause of this failure. This investigation should include a review of the test specimen and loads, analytical loads, and the structural analysis. This may lead to adjustment in analysis/modelling techniques and/or part redesign and may result in the need for additional testing. The need for additional testing to ensure that ultimate load capability depends on the degree to which the failure is understood, and the analysis can be validated by the test.

The approach described above is valid for static justification. However, a similar approach can be extended for compliance with fatigue, dynamic and crashworthiness requirements. For these applications, the criteria and the classification have to be accepted by and agreed with the authority.

AMC2 29.307 Proof of structure

FAIRING SUBSTANTIATION

This AMC supplements FAA AC 29-2C, § AC 29.307 and should be used in conjunction with that AC when demonstrating compliance with CS 29.307.

Further to CS 29.301, the specified loads must be distributed appropriately or conservatively and significant changes in the distribution of the loads, as a result of deflection, must be taken into account. FAA AC 29-2C, § AC 29.307 refers to the need for flight test measurement in the scope of the fatigue and damage tolerance demonstration. The methods used to determine load intensities and distribution should be validated by flight load measurements unless the methods used for determining those loading conditions are shown to be reliable.

Each fairing, when appropriate, should be constructed and supported so that it can resist any vibration, inertia, and air load to which it may be subjected in operation. The vibrations level, the inertia and air loads should be validated by appropriately instrumented flight measurements as recommended in FAA AC 29-2C, § AC 29.307.

For the fairings and the associated supporting structure, the loads can be shown unreliably predicted and require a measurement during flight tests.

The loads derived from flight testing should be compared with those obtained from analytical methods.

Note: AMC No.2 to CS 25.301(b) provides an acceptable means of demonstrating compliance with the provisions of CS-25 related to the validation, by flight measurements, of the methods used for determination of flight load intensities and distributions, for large aeroplanes.

The methodology presented in the CS-25 AMC material may be adapted to CS-29, to provide further guidance to this AMC.

AMC3 29.307 Proof of structure

SEAT ADAPTER PLATES

(a) Purpose

This AMC provides an acceptable means of compliance for seat adapter plates. A seat adapter plate includes any other forms of new interface structure installed between the seat and the rotorcraft floor.

(b) Related Certification Specifications

— CS 29.307 'Proof of structure'

— CS 29.561 'General'

— CS 29.562 'Emergency landing dynamic conditions'

— CS 29.785 'Seats, berths, safety belts, and harnesses'

(c) Explanation

The requirements for seats under emergency landing dynamic conditions have been developed to prevent detachment of the seat under floor deformation and for the seat to help absorb the energy developed in crash conditions. This dynamic condition has been addressed with the 10° roll and 10° pitch deformation required by CS 29.562(b)(3) to ensure that the seat and the floor attachments will be designed to accommodate deformation. This objective should be maintained when a seat adapter plate is installed between the seat and the floor.

Introducing an adapter plate can move the problems created by floor deformation from the seat-to-track interface to the adapter-to-floor interface. The same level of safety is appropriate for the occupant of the seat whether it is installed in the rotorcraft with or without an adapter plate. The floor structure itself is not subject to the dynamic requirements of CS 29.562, therefore when additional structure such as an adapter plate is introduced to fix the seat to the floor, it is very important to determine whether that structure should be considered to be part of the seat or part of the floor. The installation of any interface between the existing floor and the seat should not create a weak element between the seat and the existing airframe. This has successfully been ensured by testing the adapter with the seat according to the requirements of CS 29.562.

This AMC provides further guidance and acceptable means of compliance for classification of seat adapters, such as plinths or pallets, and supplements FAA § AC 25.562.

Plinths are subject to CS 29.562 compliance whereas pallets (traditionally defined as large adapters) are not, except for the attachment of the seat to the pallet.

FAA Policy Memo PS-ANM100-2000-00123 (which is applicable to CS-25 and can be extended to CS-29) suggests that it may also be possible to classify some smaller adapters as an integral part of the floor as follows:

'Generally speaking, adapters of the size that contain a single row of seats (whether they are individual seat places or a common assembly), and mount into seat tracks, should be treated as part of the seat for purposes of certification in accordance with § 27/29.562. Larger, or more integrally mounted adapters, should be assessed to determine whether they should be treated as part of the floor for purposes of certification in accordance with § 27/29.561.'

To treat an adapter or other new interface structure as part of the floor when it does not appear to be similar to conventional floor structure, the applicant must substantiate that the adapter plate or any other structure installed between the existing floor and the seat attachment will not constitute a weak element under emergency landing conditions. The issue is whether the critical interface is between the seat and the adapter or between the adapter and the rotorcraft. No further detailed guidance is available to assist with the assessment required to make the classification of an adapter as part of the floor.

Where the proposed floor design utilises a plate above the existing floor or otherwise significantly differs in concept from the type design's existing methods of floor construction, geometries and utilisation of load paths, it is not adequate to rely on compliance with CS 29.561 alone to determine whether the adapter plate may be considered to be part of the floor. This guidance does not intend to request a complete crash scenario evaluation, but asks for evidence that the adapter plate and associated new under floor structure will not degrade the level of protection compared to that offered by the seat if it were installed directly on the helicopter existing floor seat track and floor construction. For an adapter plate to be considered sufficiently integrated to be part of the floor, the adapter plate should be capable of accommodating floor deformation and be able to safely react and distribute the seat loads into the rotorcraft.

(d) Seat adapter plate definition and classification

(1) Definition

The definition of plinth and pallet available in AC 25.562(b) is valid.

In general, swivelling seat adapter plate systems are by definition considered to be plinths.

(2) Classification

There are three possible options for the seat-to-floor interface with corresponding means of compliance. In each case, the applicant is requested to show that any interface between the existing floor and the seat will not create a weaker element between the seat and the existing airframe than that that would exist for a CS 29.562-compliant seat attached directly to the standard floor, e.g. seat track.

Acceptable means of assessing seat installations using adapter plates:

Option 1

— The adapter is classified as a plinth following AC 25.562-1B.

— Compliance with CS 29.561 and CS 29.562 must be shown.

— The plinth must be tested as part of the seat according to CS 29.562 (b)(1) and (b)(2) unless alternative compliance is agreed as per CS 29.562(d).

— The guidance of AC 25.562-1B and AMC 29.307 may be used to reduce the number of tests based on design similarity.

Option 2

— The adapter is classified as a pallet due to its size following AC 25.562-1B.

— The seat and its attachments to the pallet only are tested according to CS 29.562 and CS 29.561.

— The pallet is justified against CS 29.561 only.

Option 3

— If neither Option 1 nor 2 clearly apply, seat-to-floor interface structure is proposed to be classified as an integral part of the floor based on one of the methods described below.

— If classification as part of the floor is agreed with the Agency, the seat and its attachments to the structure are tested according to CS 29.562, and compliance with CS 29.561 is shown for the whole installation.

Acceptable methods to be used in support of Option 3, allowing classification of the new seat-to-floor interface structure as an integral part of the floor structure:

Method 1

A design review showing the floor design for seat installation uses the same or an equivalent design principle as the current floor provided in the type design. If the pre-existing floor design used seats directly attached to seat track independently of the floor panel, then the introduction of a structural floor panel to which a seat is attached would represent a change in design philosophy, and a different method (e.g. Method 2) would need to be used to support Option 3.

Method 2

A detailed design review showing the level of integration of the plate to the floor, including the redundancy and strength of the attachments, that is acceptable to the Agency based on the experience of the applicant and the Agency with similar designs.

Any other alternative methods have to be agreed with the Agency.

Note:

When assessing the design, the following points should be considered by the applicant and the Agency, in particular for design change certification:

— The modified structure may be evaluated using AMC1 29.307 to categorise the structural elements as new, similar-new or similar. Comparison can be made with the existing type floor design (Method 1) or with designs that the applicant has previously substantiated according to Method 2.

— An adequate number of appropriately distributed attachments between the adapter plate and the rotorcraft floor structure must be provided to ensure that the additional structure behaves as an integral part of the rotorcraft floor. The appropriate number, strength and degree of redundancy of the attachments will depend on the design of the adapter plate and positioning of the seats on the plate.

— A considerable degree of engineering judgement is required when making the classification of the structure; when there is any doubt about the capability of the proposed adapter design to act as an integral part of the floor, it will be classified as a plinth under Option 1.

CS 29.309 Design limitations

The following values and limitations must be established to show compliance with the structural requirements of this Subpart:

(a) The design maximum and design minimum weights.

(b) The main rotor rpm ranges, power on and power off.

(c) The maximum forward speeds for each main rotor rpm within the ranges determined under sub-paragraph (b).

(d) The maximum rearward and sideward flight speeds.

(e) The centre of gravity limits corresponding to the limitations determined under sub-paragraphs (b), (c) and (d).

(f) The rotational speed ratios between each powerplant and each connected rotating component.

(g) The positive and negative limit manoeuvring load factors.

(h) The maximum and minimum density altitude and temperatures.

AMC1 29.337 Limit manoeuvring load factor

This AMC supplements FAA AC 29-2C, § AC 29.337 and should be used in conjunction with that AC when demonstrating compliance with CS 29.337 for determining the positive limit manoeuvring load factor.

In accordance with CS 29.337, the rotorcraft may be substantiated to a maximum positive load factor less than +3.5 (but not less than 2.0) provided that the probability of being exceeded is shown to be extremely remote. Whenever this option is selected, the maximum available rotor lift with both power on and power off rotor speed ranges throughout the entire operational density envelope should be considered.

AC 29-2C, § AC 29.337(b)(1) provides some guidance as to the necessary considerations when substantiating manoeuvre load factors less than the specified values. Further clarification should be provided in this paragraph to specify that the entire operational envelope should be considered when determining the maximum available rotor lift.

There, the guidance should be read as follows:

§ AC 29.337(b)(1) The applicant may elect to substantiate the rotorcraft for a design manoeuvring load factor less than +3.5 and more than -1.0. Whenever this option is used, an analytical study and flight demonstration are required. Maximum available rotor lift with both power on and power off throughout the entire operational density envelope should be considered when substantiating manoeuvre load factors less than the specified values.

CONTROL SURFACE AND SYSTEM LOADS

AMC1 29.395 Control system

This AMC supplements FAA AC 29-2C, § AC 29.395 and should be used in conjunction with that AC when demonstrating compliance with CS 29.395.

The design reaction loads prescribed in CS 29.395 for the flight control system should apply to the part of the control system from the pilot cockpit control sticks/pedals to the main/tail rotor servo-actuators. The remaining part of the flight control systems located between the attachment of the servo-actuators and the (main/tail) blades (i.e. rotating parts, servo-actuators and their attachments) should be substantiated to the highest of:

— maximum loads expected in service (limit loads) as per CS 29.301, CS 29.305 and CS 29.547 (nominal conditions);

— maximum loads for a single failure of the hydraulic system leading to an operating hydraulic overpressure;

— the maximum loads due to a jamming of the flight control system (rotating parts).

The maximum pilot loads from CS 29.397 to CS 29.399 should be added to these loads appropriately.

AMC1 29.427 Unsymmetrical loads

This AMC supplements FAA AC 29-2C, § AC 29.427 and should be used in conjunction with that AC when demonstrating compliance with CS 29.427.

In case of load distribution deviating from CS 29.427(b), the applicant should provide the rationale justifying that the selected load distribution conservatively addresses the limit flight load conditions of Subpart C. Dedicated flight load and/or wind tunnel measurements should be performed to confirm the suitability of the proposed criteria.

FATIGUE EVALUATION

[…]

AMC1 29.571 Fatigue tolerance evaluation of metallic structure

ROLLING CONTACT FATIGUE

This AMC supplements FAA AC 29-2C, § AC 29.571 and should be used in conjunction with that AC when demonstrating compliance with CS 29.571.

(a) Definitions

(1) Rolling contact fatigue (RCF): a form of fatigue that occurs due to the cyclic strains arising from the loading present during rolling contact between two parts of an assembly, e.g. a bearing race and a rolling element.

Note: For the purposes of this AMC, RCF also includes combinations of rolling and sliding contact phenomena.

(2) Integral race: a bearing race that is an integral part of the transmission structural component such as a gear or shaft.

(b) Explanation

Service experience has shown that RCF can initiate on the surface and below the surface in contact areas of structural elements (typically, but not limited to, bearing races and rolling elements and gear teeth) that, in some cases, can propagate to a failure with catastrophic results. It is often assumed that RCF leads first to non-critical partial failures such as micro-pitting and spalling that will be detected before more severe failure modes can develop, such as a complete crack through a part. However, experience has shown that, in some cases, critical failure modes can develop shortly after the occurrence of non-critical partial failures. In such cases, analyses and tests are necessary to demonstrate that sufficient time is available, and the performance of the detection system is adequate to ensure the timely detection to prevent a catastrophic failure.

The certification specifications in CS 29.571 require the identification and fatigue tolerance evaluation of principal structural elements (PSEs), leading to the establishment of inspection and retirement time or approved equivalent means to avoid a catastrophic failure during the operational life of the rotorcraft. In order to complete this evaluation, the impact of threats such as environmental effects, flaws and damages should be considered. Specific characteristics of parts submitted to RCF, such as:

— the difficulty to visually inspect,

— the operating nature of these elements, which can lead to mechanical degradation,

— the variability and susceptibility of the RCF mechanism in the presence of flaws or damages,

make the evaluation of the impact of RCT on fatigue tolerance evaluation challenging.

The procedures of this AMC are intended to help ensure that the effects of RCF are accounted for in the fatigue tolerance evaluations required by CS 29.571.

(c) Procedure

The fatigue tolerance evaluation of PSEs should include, when applicable, the effect of RCF considering:

— damage threats such as dents, scratches, corrosion, loss of pre-load in bearings or joints, surface and sub-surface material defects;

— residual stress coming from surface treatments and other manufacturing processes and all other applicable loading conditions.

For this purpose, steps should be taken to minimise the risk of crack initiation due to RCF on PSEs (and in particular for integrated bearing races), by minimising contact pressures, specifying high standards for surface finishes, ensuring good lubrication, guaranteeing cleanliness and maintaining lubricant quality regardless of the fatigue tolerance approach selected. The applicant should verify that the selected allowables are suitable to ensure the integrity of the affected components in the operating conditions (temperature, lubrication, cleanliness, etc.) applicable to their design. Experience has demonstrated that it can be beneficial for bearings to be designed so that the reliability of any integrated race subject to the fatigue tolerance evaluation is even higher than the less critical race of the bearing. In this way, degradation of the less critical race can lead to detection of the bearing failure before cracking initiates in the integrated race. The consequences of damage to the integrated race from the debris generated in such scenarios should be considered in the evaluation.

As it is difficult to totally preclude cracking initiated by RCF, a fail-safe approach is recommended wherever possible, such that cracking of the affected structural element(s) is detected prior to its residual strength capability falling below the required levels prescribed in CS 29.571(f). Should fatigue cracks initiate and develop into:

(1) Partial failure, such as spalling: the applicant should demonstrate that this condition will be detected at an early stage to avoid a catastrophic failure due to further fatigue failure,

or loss of integrity of the affected part or any surrounding ones. Any assumptions regarding potential surface and sub-surface cracking considering possible damages or flaws, and whether a through crack may develop and its relationship with other forms of damage including spalling should be verified.

(2) Failure, such as through-cracking of a part together with any other associated damage in the system: the applicant should demonstrate that the remaining structure will withstand service loads and design limit loads without failure until the failure is detected and damaged components are repaired or replaced to avoid a catastrophic failure. Any assumptions regarding crack path development (i.e. bifurcation, multicracks, etc.) that could affect this fail-safe demonstration should be verified.

This demonstration should be performed as appropriate using experience from similar designs, functional tests, structural tests and/or reliable analyses to substantiate that the fail-safe design objective has been achieved, including residual strength demonstration. In addition, the continued safe operation of the affected mechanical system(s) should be ensured for this period considering the potential effect of the failure or partial failure taking into account any pre-existing fatigue damage accrued prior to the failure in the affected component and/or surrounding ones on stiffness, dynamic behaviour, loads and functional performance.

The effectiveness and reliability of means of crack detection for the fail-safe approach, including indirect means of detection such as chip detection systems, and associated instructions for continued airworthiness should be evaluated to show that, if implemented as required, they will result in timely detection and repair or replacement of damaged components. Furthermore, the instructions for continued airworthiness, prescribing the maintenance actions leading up to and following detection of potential failure or partial failure should be substantiated sufficiently to ensure timely repair or replacement of damaged components. The substantiation should consider aspects such as threshold criteria on indicators of means of detection for additional investigative actions and removal from service of the damaged parts, the overall clarity and practicality of the instructions for continued airworthiness and human factors aspects.

In addition to following a fail-safe approach, inspection and retirement times may be needed in order to ensure that the assumptions supporting the fail-safety and detection of failure remain valid throughout the operational life of the component.

SUBPART D — DESIGN AND CONSTRUCTION

GENERAL

[...]

AMC1 29.607 Fasteners

This AMC supplements FAA AC 29-2C, § AC 29.607 and should be used in conjunction with that AC when demonstrating compliance with CS 29.601, CS 29.602, CS 29.603 and CS 29.607.

(a) Explanation

Designers should consistently take into account the limitations of the standards, including the applicable fastener manufacturing processes and quality controls, to ensure that when a standard part or qualified standard part is selected, its properties and associated level of reliability will meet the applicable certification requirements for the design.

The intent of this AMC is to give further guidance to the design approval holders (DAHs) and applicants for design approvals to help ensure that appropriate measures are considered for initial certification, including associated continued airworthiness aspects, to minimise the risk that the use of standard fasteners might compromise the intended level of safety.

(b) Definitions

(1) Standard fastener: a fastener that is a standard part. Fasteners (nuts and bolts) being produced according to a certain standard which is not directly approved by the Agency. They fall within the category of standard parts as defined in point 21.A.303(c) of Annex I (Part 21) to Commission Regulation (EU) No 748/2012.

(2) Qualified standard fastener: a standard fastener that requires additional verification of compliance with specification and/or control of their source, by methods defined by the DAH.

(3) Critical installation: a structural/mechanical assembly which may include fasteners the failure of which (single or multiple due to common cause) is classified as hazardous or catastrophic.

(c) Procedures

Failures of standard fasteners may have severe consequences at the aircraft level when used in critical installations.

Once demonstrated, conformance to a standard provides a certain level of reliability under known loading and environmental conditions. The reliability of a standard part or any other part specified in the design needs to be assessed and shown to be compatible with the design objectives to be met. Designers should take care to ensure that they select appropriate fasteners to meet the certification objectives for continued function and reliability, taking into account the limitations of the applicable standards including the associated manufacturing processes and applicable quality controls.

This AMC is therefore addressed to DAHs, to provide them with guidance on appropriate actions to ensure appropriate utilisation of standard fasteners in their designs, to help them to instruct production organisations and maintenance organisations as necessary to ensure continued airworthiness, and to provide means by which unsafe conditions related to the use in design of standard fasteners can be prevented.

In order to reduce the risk of critical installations failing, through the inadvertent use of defective standard fasteners or due to the inappropriate selection of standards, the Agency recommends that all applicants for type certificates and design changes perform a design review to ensure that the risk posed by the use of standard parts is mitigated by:

(1) ensuring that fasteners (nuts and bolts) used in the design will meet the certification requirements, taking into account any limitations of the selected standards, the associated fastener manufacturing processes and quality controls, and relevant service experience;

[Note: The degree to which the standard ensures relevant characteristics such as locking functions, static strength and fatigue strength should be evaluated as far as is necessary based on the criticality of the intended use and operating environment of the parts. Consideration should be given to stress levels arising from manufacture, installation requirements, external loading and temperature effects. Particular attention should be paid to standard parts that utilise high-strength alloys in combination with plating or other processes that may increase the risk of hydrogen embrittlement or deformation processes that are not closely specified.]

(2) ensuring that the design standard met and associated procedures followed for the production of the aircraft are maintained throughout the operational life of the aircraft, e.g. through the use of the ICA controlling maintenance of critical installations;

(3) creating, when standard fasteners (nuts and bolts) are selected, a list of critical installations where only qualified standard fasteners (nuts and bolts) may be used. Redundancy of fasteners alone may not negate the need to qualify the fasteners as all the fasteners on a joint could originate from a common defective batch. Similarly, required double locking functions on fasteners may also need consideration of qualified standard fasteners to ensure that the fail-safe design philosophy is maintained when common cause failure of both locking functions is possible;

(4) defining how the standard fastener is qualified wherever necessary;

(5) clearly defining any necessary additional conformity checks as part of the type design standard, specifying requirements for approved suppliers and any other criteria necessary for acceptance, storage and installation of standard fasteners that are appropriate for use in the design;

(6) ensuring through maintenance instructions that qualified standard fasteners are only replaced by other qualified standard fasteners; and

(7) considering introducing a DAH part numbering system for qualified standard fasteners, at which point they would become aviation parts. (Note: If such part numbering is implemented and further part marking is not feasible due to the part's size or for other reasons, other means such as regular appropriate batch controls should be established, and documentation provided according to point 21.A.804(b) of Part 21.)

In addition, DAHs are reminded that certain existing Certification Specifications and regulations specifically address critical parts. Typically standard parts are not appropriate for use as critical parts. All critical parts are subject to a critical parts plan that controls their critical characteristics during production and service.

AMC1 29.610 Lightning and static electricity protection

(a) Purpose

This AMC provides an acceptable means of compliance for rotorcraft components evaluation after lightning strike.

(b) Related Certification Specifications

CS 29.610 'Lightning and static electricity protection'

CS 29.571 'Fatigue tolerance evaluation of metallic structure'

CS 29.573 'Damage tolerance and fatigue evaluation of composite structures'

CS 29.1529 'Instructions for Continued Airworthiness'

(c) Explanation

CS 29.610 requires the protection of rotorcraft structural components, propulsion system, gearboxes, mechanical and hydraulic control systems from lightning damage that could result in catastrophic failures.

However, damage, failure or departure of any rotorcraft component which could endanger the rotorcraft or its occupants must be part of the evaluation.

This AMC provides detailed guidance on damage tolerance evaluation, including residual strength criteria after lightning strike to ensure continuous safe flight and landing.

Each part, the failure of which implies potential catastrophic consequences and that is exposed to damage under lightning conditions, should be subject to further evaluation which includes:

(1) the nature and extent of the lightning damage (threat assessment, damage detectability, etc.);

(2) the demonstration of the functionality of the affected part up to detection;

(3) a static residual strength capability demonstration supported by analysis and/or test;

(4) when found necessary, a fatigue evaluation of a part with lightning damage for the demonstration of the exposure time before detection.

The airworthiness instruction requested after lightning strike (flight manual and maintenance instructions, etc.) should be consistent with the functional, static and fatigue evaluation of the damage consequences (considered to be a partial failure).

A similar approach should be considered for non-metallic components (for composite, see the AMC 20-29 (11c) guidance).

The above approach is also considered to be applicable for parts departure which could preclude continued safe flight and landing.

For non-structural components (e.g. radomes, panels), only static residual strength is requested for part detachment which could preclude continued safe flight and landing.

AMC1 29.613 Material strength properties and design values

COMPOSITE SANDWICH PANEL

(a) Qualification of the manufacturing process

The conditions outlined in the guidance standard AC 21-26, 'Quality Control for the Manufacture of Composite Materials' are considered to be relevant to composite sandwich PSE structure.

The qualification is intended to demonstrate that the combination of material, tooling, equipment, procedures, and other controls, making up the process, will produce representative parts having consistent material properties that conform to design requirements.

As part of the process qualification, destructive and non-destructive inspection (NDI) should be conducted to determine conformity to specified design requirements and check the suitability of the resulting product by assessing features such as:

— uniformity of the adhesive fillets between honeycomb core cell wall and skin; in particular, the process should ensure that on both faces of the honeycomb core a regularly shaped fillet (meniscus) be established;

— absence of 'telegraphing' effects and waviness on the skins of the sandwich panel;

— distortion of the core cells — this defect could be particularly critical for highly curved panels unless suitable precautions are taken during fabrication (e.g. core thermal performing);

— presence in the adhesive of unacceptable levels of porosity or humidity;

— disbonds between core and cells; and

— weak bonds.

(b) Material strength and determination of design allowables

The strength properties of the sandwich panels should be established in order to ensure that the probability of structural failure due to material and process variability is minimised.

Because of the peculiarity of the sandwich panel construction, the material properties should be established on a specimen that is fully representative of the panel construction in terms of skin, core material and curing cycle.

Design features such as transition zones from solid laminate to core/skin should be also tested with a representative specimen for determination of strength properties.

It is expected that at least the following static allowables be established according to the statistics required in CS 29.613:

— Adhesive shear strength;

— Shear core strength (ribbon and transverse direction);

— Core compression strength;

— Flatwise strength;

— Flexural strength;

— Compressive strength; and

— Bearing strength (for a specimen representative of all the panel areas where fasteners are installed and subject to significant bearing stresses).

In determining the above properties, the effect due to humidity uptake, highest and lowest temperature expected in service, manufacturing defects up to limit of acceptability and allowable in-service damage defined in maintenance documents, if any, should be considered. For PSEs, impact damages should also be assessed in accordance with CS 29.573.

The validity of the engineering formula used to establish analytical design allowables should be always verified by dedicated experimental activity in order to assess the effects of the manufacturing process (e.g. curing pressure which is normally limited to the crush core strength) and environmental conditions on the allowables predicted by these formulas.

(c) Damage tolerance and residual strength

(1) Threat survey and damage modes

Further to good processing, and when meeting the damage tolerance and fatigue evaluation of composite rotorcraft structures requirements of CS 29.573, the applicant should clearly demonstrate that a robust structure has been produced by showing that:

— a thorough damage threat survey has been completed which identifies and defines all threats, including impacts, heat, moisture, etc. and the potential for interaction of these threats is addressed;

— all damage modes have been identified for the configuration when subject to all likely threats, paying particular attention to all likely damage modes which might not be readily detected.

For impact threats, this requires testing throughout the threat impact energy ranges up to a readily detectable damage using a range of appropriate impactor geometries, including blunt impactors up to 4 inches diameter[1], and a range of impactor stiffnesses, e.g. for hail threat damage (if appropriate), such that all competing damage modes can be identified. Representative boundary conditions should be used in the substantiating test campaigns; and

— all potentially undetectable damage modes (not only disbonds and weak bonds) have been simulated in testing (up to appropriate dimensions such that detection becomes possible, and the dimension of such damage has been quantified such that ultimate load (UL) can be maintained up to this level). The possibility of interaction between threats, e.g. impact and heat, should be considered in the simulation and substantiation process.

Note: Witness structures can be used in service, provided that a consistent and conservative correlation can be demonstrated to exist between the witness indications on the witness structure and the damage (all likely modes and extents) considered in the critical structure.

The recommendations for threat assessment and blunt impact evaluation are also addressed in AC 29.573.

[1] An alternative impactor diameter may be proposed by the applicant, based on the results of the damage threat survey.

(2) Residual strength after extensive damage or degradation

The part should be sized to sustain the required residual strength, in accordance with CS 29.573(d)(4)(ii)(B), with extensive damage or degradation of the most critical skin to core bond between available arrestment features. Such damage or degradation should be readily detectable to assure damage tolerance for bond failures which experience has shown not to be extremely improbable.

It is also expected that relevant fatigue testing at specimen level, representative of a design point (e.g. fastened joint) and typical panel configuration, be performed in order to assess the effects of the fatigue strength on:

— material/manufacturing process variability;

— environmental condition;

— allowables manufacturing defects; and

— impact damages.

(d) Instructions for Continued Airworthiness (ICA)

The ICA include clear instructions to inspect[2] (and repair), both internally and externally:

— all load paths, e.g. up to load transfer fittings, joints, and other significant changes in stiffness and section, for damage following an overload event, e.g. impact, heavy landing, excessive gust, etc.;

— all structure regularly exposed to extreme temperatures, e.g. local to engine outlets for aircraft used extensively in hot climates, etc. Although inspections intervals should have been justified according to the level of detectability and residual strength capability during certification substantiation based upon a damage threat survey, experience has indicated that the potential for interaction between heat and damage can be problematic.

(2) paying particular attention to:

— repaired structures; and

— any existing, and potentially related, ICA, e.g. existing ADs, etc.

PERSONNEL AND CARGO ACCOMODATIONS

CS 29.777 Cockpit controls

Cockpit controls must be:

(a) Located to provide convenient operation and to prevent confusion and inadvertent operation; and

(b) Located and arranged with respect to the pilot's seats so that there is full and unrestricted movement of each control without interference from the cockpit structure or the pilot's clothing when pilots from 1.57 m (5 ft 2 inches) to 1.83 m (6 ft) in height are seated.

[…]

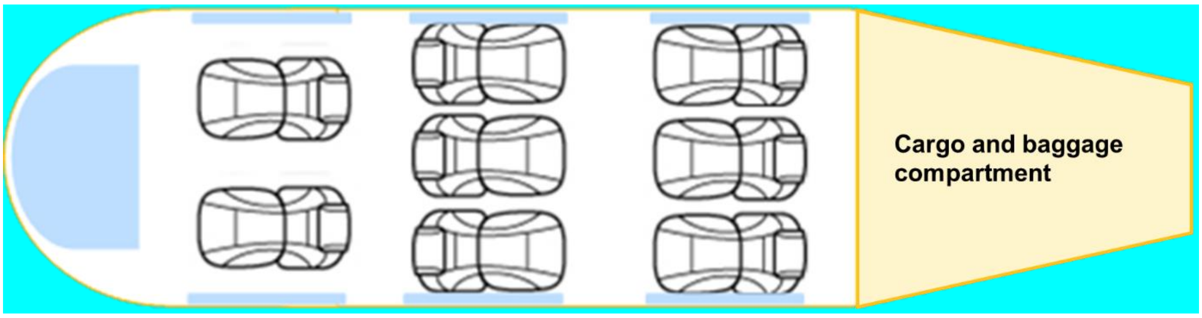
AMC1 29.787 Cargo and baggage compartments

PROTECTION OF OCCUPANTS IN THE CABIN

The CS-29 objective is to protect the occupant within the cabin from forces up to those specified in CS 29.561(b)(3).

If the cabin is forward of the cargo or baggage compartment and is separated with a structural partition, this partition should be sized to 12g forward, as per the CS 29.787 requirement, regardless of the means used to restrain the items of mass in the cargo or baggage compartment. If a structural partition is not installed, then ultimate inertial load factors specified in CS 29.561(b)(3) apply to the restrain system of the items of mass (i.e. baggage, cargo, etc.).

Conditions to be considered:



Structural Partition:
CS29.561(c)
[12g forward]

Cargo and Baggage Compartment:
No Structural Partition:
CS29.787
CS29.561(b)
[16g forward]
With Structural Partition:
CS29.787
CS29.561(c)
[12g forward]

CS 29.801 Ditching

(a) If certification with ditching provisions is requested by the applicant, the rotorcraft must meet the requirements of this CS and CS 29.563, CS 29.783(h), CS 29.803(c), CS 29.805(c), CS 29.807(d), CS 29.809(j), CS 29.811(h), CS 29.813(d), CS 29.1411, CS 29.1415, CS 29.1470, CS 29.1555(d)~~(3)~~ and CS 29.1561.

[…]

AMC1 29.801 Ditching

[…]

AMC~~1~~2 ~~to CS~~ 29.801(e) and 29.802(c) Model test method for flotation stability

This AMC should be used when showing compliance with CS 29.801(e) or CS 29.802(c) as introduced at Amendment 5.

(a) Explanation

[…]

(3) Target probability of capsizing

Target probabilities of capsizing have been derived from a risk assessment. The target probabilities to be applied are stated in CS 29.801(e) and 29.802(c), as applicable.

For ditching, the intact flotation system probability of capsizing of 3 % is derived from a historic ditching rate of 3.32 x 10-^{6} per flight hour and an A~~M~~C 29.1309 consequence of hazardous, which implies a frequency of capsizing of less than 10^{-7} per flight hour.

[…]

AMC1 29.807(d) Underwater emergency exits for passengers

[...]

CS 29.811 Emergency exit marking

[...]

(d) Each passenger emergency exit marking and each locating sign must have white letters ~~25 mm (1 inch) high~~ on a red background ~~51 mm (2 inches) high,~~ or a universal emergency exit symbol, of adequate size. These signs must be self or electrically illuminated, and have a minimum luminescence (brightness) of at least 0.51 candela/m^{2} (160 microlamberts). The colours of a text-based sign may be reversed if this will increase the emergency illumination of the passenger compartment.

[…]

(g) Exits marked as such, though in excess of the required number of exits, must meet the requirements for emergency exits of the particular type. Emergency exits need only be marked with the word 'Exit' or a universal emergency exit symbol.

[...]

AMC1 29.811(d) Emergency exit marking

EMERGENCY EXIT SIGNS

Emergency exit signs should consist of a consistent type throughout the rotorcraft. They may be letter-based or symbolic, as outlined below.

Letter-based emergency exit signs should use letters with a height to stroke width ratio of not more than 7:1 nor less than 6:1.

Symbolic emergency exit signs should be white and green in compliance with European Standard (EN) ISO 7010:2012 'Graphical symbols — Safety colours and safety signs — Registered safety signs'.

The green area of the sign should constitute at least half of the total area of the sign.

In the area determination of an emergency exit sign, no part of the sign outside of the white background (text signs) or green element (symbolic signs) — for instance, a surrounding contrasting border — should be included.

Minimum size

For each emergency exit sign required by CS 29.811(c), a sign using English letters of at least 25 mm (1 inch) height, or a white symbolic element (i.e. that part incorporating the green 'running man') of at least 40 mm (1.6 inches) height, with an overall area of at least 64.5 cm^2 (10 square inches) should be acceptable provided that the centrelines of the forward most and rearward most emergency exits are no more than 6 m (19.8 feet) apart.

Examples of acceptable designs of symbolic exit signs

Direction of running man

There may be a reason to choose a particular movement direction of the 'running man'; for instance, where a sign required by CS 29.811(c) is placed to the left or right of the emergency exit. The 'running man' should not suggest movement away from the emergency exit.

AMC2 29.811(h) Underwater emergency exit markings

[...]

FIRE PROTECTION

[...]

AMC1 29.853 Compartment interiors

CS 29.853 (a) and (b) refer directly to CS-25 flammability requirements. Furthermore, CS 29.853(d) sets a fire containment requirement for waste containers that is essentially the same as that set by CS 25.853(h).

Accordingly, the relevant guidance for complying with CS-25 flammability requirements that is found in AC 25-17A and PS-ANM-25.853-R2 may be used when showing compliance with the requirement of CS 29.853.

AMC2 29.853(c) Compartment interiors

PROHIBITION OF SMOKING

CS 29.853(c) requires that if smoking is to be prohibited, a placard so stating must be installed.

A single placard, installed such that it is clearly visible to all passengers whilst seated, is an acceptable means of compliance. Alternatively, more than one placard may be installed, in locations such that at least one placard is clearly visible to each passenger when seated.

A placard may have a text-based design, or may utilise symbols that clearly express the intent.

SUBPART E — Powerplant

ENGINES

AMC1 29.903(d)(1) Turbine engine installation

FRAGMENT CONTAINMENT

This AMC supplements FAA AC 29.901 with regard to the credit that can be taken from engine manufacturer data substantiating the capability of the engine to contain fragments.

(a) Blade containment

Single blade radial containment is a CS-E / CS-APU requirement. Full credit is given to engine certification for blade containment, and no specific certification activity is required at helicopter level for blade failure. This approach is supported by the in-service experience.

(b) Small debris containment at engine level

Some engine designs feature the capability to retain radially small debris, featuring, for instance, a reinforced casing or blade shedding capability.

The engine uncontained model features a small debris over a ±15° spread angle. Small fragments can be a collateral effect of either large or intermediate fragment release, but are released over larger spread angles, typically ±15°. Therefore, from a CS 29.903(d) point of view, no credit can be given to engine radial containment for small debris, which might however have other safety benefits.

(c) Rotor containment at engine or APU level

CS-APU has provisions to demonstrate rotor containment. For engines, while not required by CS-E, engine manufacturers might decide to design their engines featuring rotor containment systems, for all or specific rotating stages.

— For engines, the containment capability is not required by CS-E and the corresponding data is not covered by the engine type certificate; the helicopter manufacturer should propose a mechanism to ensure that the data is valid, under their DOA or by validation through the engine type certificate whereas for an APU, CS-ETSO requirements are in place, and it can be expected that the data is covered by the ETSO issuance.

— In-service experience has shown that such containment features successfully perform their intended purpose of retaining the biggest debris (large fragments). However, small debris can defeat the containment system, either by missing it or by exiting through damages caused by the large fragments. Rotor containment systems, as explained in paragraph f.(1) of AC 29.903C, still require some activity at helicopter level to ensure that the risks associated with uncontained engine or APU uncontained failure are adequately mitigated.

Note: For APUs, AMC 20.128A defines an acceptable model based upon debris exiting the containment system with a 1 % residual energy.

AMC2 29.903(e) Engines

ENGINE RESTART CAPABILITY

This AMC replaces FAA AC 29-2C, § AC 29.903B and should be used when demonstrating compliance with CS 29.903(e).

(a) Explanation

CS 29.903(e) requires that any engine must have a restart capability that has been demonstrated throughout a flight envelope to be certificated for the rotorcraft.

(b) Procedures

Compliance is usually demonstrated by conducting actual in-flight restarts during flight tests or other tests in accordance with an approved test plan. However, CS 29.903(e)(2) does not require in-flight demonstration of restart capability for single-engine rotorcraft or for all-engine shutdown of multi-engine rotorcraft. In the past, engine restart capability for single-engine rotorcraft has been demonstrated on the ground taking into account altitude effects, warm engine characteristics, depleted battery, etc. However, latest-technology engines embody electronic engine controls (EEC or FADEC) that may have sophisticated starting or restarting laws. For these designs the engine restart capability demonstrated on ground may not provide the level of representativeness required and therefore applicants are encouraged to demonstrate the capability in flight. The minimum restart envelope for category A rotorcraft is discussed in AC 29.903A. The restart capability can consider windmilling of the engine as part of this restart capability; however, most rotorcraft airspeeds and the locations of the engines do not support engine windmilling up to start speeds. Only electrical power requirements were considered for restarting; however, other factors that may affect this capability are permitted to be considered. Engine restart capability following an in-flight shutdown of the engine in single-engine rotorcraft, or all engines in a multi-engine rotorcraft, is the primary requirement, and the means of providing this capability is left to the applicant. To minimise any potential altitude loss following the failure of one or more engines, engine restart should be available at the earliest opportunity. The engine certification should be checked to ensure that the flight manual instructions for in-flight restart are consistent with any specific engine restart requirements. If the procedure was only demonstrated on ground, this should be stated in the RFM.

ROTOR DRIVE SYSTEM

AMC2 29.917 Rotor drive system design

LUBRICATION SYSTEMS

[…]

(g) Use of an auxiliary lubrication system

The use of an auxiliary lubrication system may be an acceptable means of providing extended operating time after a loss of lubrication. The auxiliary lubrication system should be designed to provide sufficient independence from the normal-use lubrication system. Since the auxiliary lubrication system is by definition integral to the same gearbox as the normal-use lubrication system, it may be impractical for it to be completely independent. Therefore, designs should be conceived such that shared components or interfaces between the normal-use and auxiliary lubrication systems are minimised and comply with the design assessment provisions of CS 29.917(b). A failure of any common feature shared by both the normal-use and auxiliary lubrication systems that could result in the failure of both systems, and would consequently reduce the maximum period of operation following loss of lubrication, should be shown to be an extremely remote lubrication failure. If compliance with CS 29.927(c) is reliant on the functioning of an auxiliary lubrication system, then:

(1) ~~in~~ for the unlikely event of a combined failure of both the normal-use lubrication system and the auxiliary lubrication system, the applicant should perform additional loss of lubrication tests simulating this condition. The aim is to substantiate additional RFM emergency procedures for this combined failure to ensure the capability of the drive system to sustain a minimum duration of safe operation. These procedures should instruct the flight crew to 'Land immediately', unless the additional tests performed ~~testing~~ representing this failure mode ~~has been performed in order to substantiate~~ demonstrate that an increased duration is justified; and

(2) a means of verifying that the auxiliary lubrication system is functioning properly should be provided during normal operation of the rotorcraft on either a periodic, pre-flight or continual basis. Following failure of the ~~normal use lube~~ normal-use lubrication system and activation of an auxiliary lubrication system, the flight crew should be alerted in the event of any system malfunction.

AMC1 29.923 Rotor drive system and control mechanism tests

(a) Introduction

This AMC supplements FAA AC 29-2C, § AC 29.923 and should be used in conjunction with that AC when demonstrating compliance with CS 29.923.

(b) 30-minute power rating

(1) Explanation

The option to establish a 30-minute power rating for turbine engines for rotorcraft has been introduced in CS-E Amendment 5 (published on 14 December 2018) with the creation of CS-E 40(b)(4). Means to demonstrate compliance with this requirement are provided in the associated AMC E 40(b)(3) and (b)(4) 30-Second OEI, 2-Minute OEI and 30-minute Power Ratings.

In particular, AMC E 40(b)(3) and (b)(4) mentions that 'The 30-Minute Power rating may be set at any level between the Maximum Continuous up to and including the take-off rating, and may be used for multiple periods of up to 30 minutes each, at any time between the take-off and landing phases in any flight.' In addition, CS-E 740(c)(2)(i) specifies additional running time for the endurance test for engines for rotorcraft for which approval with this rating is sought.

In comparison, the endurance test programme specified in CS 29.923 for rotorcraft rotor drive systems and control mechanisms:

— addresses the take-off power rating, which is 'limited in use to a continuous period of not more than 5 minutes' according to CS-Definitions, through the test runs specified in CS 29.923(b), and

— currently does not address the 30-minute power rating.

(2) Procedures

For applications including a 30-minute power rating, the applicant should consider that the approval of such rating should be supported by additional tests to be agreed with Agency, with the aim of determining that the rotor drive mechanism is safe considering the use of this specific power rating. In this context, the applicant may consider running additional test phases and/or extending the running time and/or increasing the minimum torque and speed conditions defined in CS 29.923 to include testing of this power rating.

AMC1 29.927 Additional tests

(a) Introduction

This AMC supplements FAA AC 29-2C, § AC 29.927 and should be used in conjunction with that AC when demonstrating compliance with CS 29.927.

(b) Variable rotor speed (NR)

(1) Explanation

The variable rotor speed (NR) function allows running at different NR levels to achieve, for instance, lower noise levels and better rotorcraft performance.

In addition to the endurance test prescribed in CS 29.923, additional tests may be necessary to demonstrate that rotor drive systems of rotorcraft with a variable NR are safe.

(2) Procedure

In order to substantiate an acceptable vibration and dynamic behaviour of rotor drive systems when using the available range of rotor speeds within the variable NR function, the applicant should consider performing specific test investigations, as prescribed in CS 29.927(a). The need for representative test runs at the different torque and rotor speed combinations, covering steady states and transient conditions to be encountered in operation, should be evaluated by and agreed with the Agency.

FUEL SYSTEMS

AMC1 29.959 Unusable fuel supply

This AMC supplements FAA AC 29.959.

This AMC provides clarification on the acceptability of analyses and ground testing which could be used as means of compliance if supported by actual flight test data.

FAA AC 29-2C, § AC 29.959 provides some guidance by focusing on a flight/test demonstration as being directly in line with the rule intent to validate '… any intended operations and flight manoeuvres …', but also provides for acceptability of analyses and ground testing.

In order to accept a demonstration by laboratory test with partial flight or ground test, the applicant should demonstrate the ability of the proposed substantiation method (bench testing, complemented by analysis and /or ground test) to cover the effects offered normally by the flight-testing environment.

In case the full flight-testing environment cannot be accurately simulated, it is necessary to either:

— revert to compliance demonstration based on flight test; or

— apply some conservatism factors on the unusable fuel quantity value resulting from the laboratory testing to determine the final unusable fuel value.

Any (steady or transitory) engine abnormal operation/malfunction has to be taken as an indication that the fuel in the tank is becoming unusable.

AMC1 29.965 Fuel tank tests

This AMC supplements FAA AC 29.965.

(a) Tests to be performed

CS 29.965 (a), (b) and (c) deal with the fuel tank pressure testing as follows:

— Sub-paragraph (a) prescribes general testing conditions.

— Sub-paragraph (b) prescribes testing conditions for conventional metal tanks, integral tanks and for non-metallic tanks with walls that are not supported by the rotorcraft structure.

— Sub-paragraph (c) prescribes pressure testing for non-metallic tanks with walls supported by the rotorcraft structure.

CS 29.965(d) deals with fuel tank vibration & slosh testing with large unsupported or unstiffened flat areas. A clear definition of 'large unsupported or unstiffened flat area' is provided in FAA AC 29-2C, § AC 29.965.

The intent of the tests required in sub-paragraphs (a), (b) or (c) does not cover the intent of the test required in sub-paragraph (d) and vice versa.

Therefore pressure tests, as prescribed under (a), (b) or (c), and the vibration and slosh test, as prescribed under (d), should be performed.

(b) Use of MIL-T-6396

AC 29.965 (c)(6) recognises the use of MIL-T-6396 to support the demonstration of compliance with CS 29.965. However, few clarifications are required to appropriately make use of this standard.

<u>Combined tests</u>

To be in line with the CS 29.965(d) requirement, the slosh and vibration test conditions shall be simultaneously applied to the test article.

Therefore the use of MIL-T-6396 should be restricted to paragraph 4.6.6 'Simultaneous Slosh and Vibration test'. Individual/separate performance of paragraph 4.6.7 'Vibrations test' and paragraph 4.6.8 'Slosh Test' of the referenced MIL Specification are not considered to be appropriate.

<u>Application of the slosh effect during the test as prescribed in CS 29.965(d)(5):</u>

CS 29.965(d)(5) prescribes the performance of the vibration test for 25h at 16 to 20 slosh cycles per minute (cpm).

MIL-T-6396 proposes two test durations in paragraph 4.6.6:

— Option 1: Vibrate for 25h at 16 to 20 slosh cpm, which is identical to the CS 29.965 (d)(5) requirement.

or

— Option 2: Vibrate for 25h at 10 to 16 slosh cpm with 15 hours of additional test at 10 to 16 slosh cpm.

While it is recognised that Option 2 is appropriate in terms of number of cycles to which the test article is finally submitted (extended testing duration to compensate for the reduction of rocking frequency), it potentially omits a major effect introduced by the higher rocking frequency which may induce more severe structural effects due to the fluid dynamics and subsequent shocks.

An applicant wishing to use Option 2 should demonstrate by analysis, test or a combination thereof, that the reduction of rocking frequency compared to Option 1 has no positive effect to the test results.

COOLING

CS 29.1049 Hovering cooling test procedures

[…]

For rotorcraft for which a 30-minute power rating is claimed, the hovering cooling provisions must be shown:

(a) At maximum weight or at the greatest weight at which the rotorcraft can hover (if less), at sea level, with the power required to hover but not more than 30-minute power rating, in the ground effect in still air, until:

— at least 5 minutes after the occurrence of the highest temperature recorded, or

— the continuous time limit of the 30-minute power rating if the highest temperature recorded is not stabilised before.

(b) With 30-minute power rating, maximum weight, and at the altitude resulting in zero rate of climb for this configuration, until:

— at least 5 minutes after the occurrence of the highest temperature recorded, or

— the continuous time limit of the 30-minute power rating if the highest temperature recorded is not stabilised before.

INDUCTION SYSTEM

AMC1 29.1093(b)(1)(i) Induction system icing protection

This AMC is primarily applicable to rotorcraft equipped with air intake external screens *(or any other air intake prone to the same kind of icing which may exist downstream)*, and has been developed based on in-service experience.

In icing conditions, as defined in CS-29 Appendix C, when the outside air temperature (OAT) is quite cold, typically below -5°C, the water droplets freeze at the helicopter air intake external screen that, once clogged, acts as passive protection by preventing subsequent super-cooled droplets to enter the engine duct and plenum. The air, then, enters the engine intake through screen areas where water droplets do not accrete, or through an air intake by-pass, if necessary.

For warmer temperatures, typically between -5°C and 0°C, a critical temperature can exist at which the water droplets do not freeze completely and immediately on the external screen and therefore icing conditions may exist downstream in the engine air intake ducts or engine internal screen.

Furthermore, ice accretions behind the air intake screen can then be released during an engine acceleration or a rotorcraft descent in a warmer atmosphere and thus may lead to engine damage, surge or in-flight shutdown.

In the case where the engine is also protected by its own screen, then the engine screen can then become clogged by ice. This may also lead to high pressure drop or distortion across the engine screen, resulting into engine surge, engine damage or engine shutdown.

The purpose of this AMC is to provide specific and complementary guidance for demonstrating compliance with CS 29.1093(b)(1)(i) in the determination of this critical temperature, but does not provide any other guidance to demonstrate full compliance with CS 29.1093(b)(1)(i) to cope with icing conditions as detailed in Appendix C to CS-29.

Analysis only should not be considered in the determination of the critical temperature due to the level of accuracy required for such an assessment. Its determination should be validated during combined rotorcraft (air intake / engine) icing tests in a wind tunnel or a similar test facility where the temperature can be controlled accurately showing whether icing conditions downstream the air intake screen are an issue or not. Typically, an accuracy of 0.5°C could be envisaged.

If the above-mentioned testing is done without the engine, it should be first demonstrated that the engine flow is correctly simulated, and the engine thermal impact adequately considered and validated on air intake. In a second step, the repercussion of any ice accretion should be assessed at engine level both in terms of airflow distortion and engine ingestion and duly validated by appropriate means. It has to be noted that this alternative approach without the engine may lead to difficulties in interpreting the results at engine level.

During these tests, the engine should be run at critical power in the icing conditions defined in CS-29 Appendix C depending on the claimed certification (inadvertent icing encounter or full icing certification). *The critical power could be determined following a critical point analysis (other methodologies might be acceptable) to assess the engine operability with regard to the feared events such as airflow distortion or engine ice ingestion.*

To determine the temperature at which the water does not freeze on the external screen, the test temperature may be decreased by accurate steps (typically a value of 0.5°C is suggested) from 0°C until accretion downstream the external air intake screen, if any, is maximised. If no ice is observed after 15 minutes of water injection, the test point is believed to be performed at a too warm temperature and can be stopped.

When decreasing the temperature step by step, if no ice accretion is observed downstream the helicopter external screen — typically for temperatures below -5°C the external screen catches the majority of the super-cooled droplets — it means that the above-described phenomenon does not occur.

Some other method can be proposed to reduce the test point number.

The test should demonstrate that, at the determined critical temperature, the maximum potential ice accretions downstream the rotorcraft screen do not have an adverse effect on the engine both in the full range of claimed operation and when the rotorcraft then descends in an atmosphere with a positive OAT.

As an example, the following test procedure may be considered:

— A 1st run: at the end of the test (in fact, when reaching the highest measured pressure drop in the air intake), perform three consecutive engine quick decelerations (from maximum power to idle) / accelerations (from idle to maximum power).

— A 2nd run: at the end of the test (in fact, when reaching the highest measured pressure drop in the air intake), simulate a quick descent in atmosphere with a positive OAT considering a tunnel warm-up procedure.

Quick accelerations / decelerations are to be understood as the maximum acceleration / deceleration rates that can be performed by a pilot during flight operation. The intent is to simulate a real-life engine behaviour which affects the flow/ice ingestion accordingly. For example, values close to one second from minimum to maximum power have been considered in the past for such testing.

As specified in CS 29.1093(b)(1)(i), these tests shall demonstrate that the engine operation is not adversely affected by icing conditions.

Whenever an applicant is willing to use previous icing wind tunnel tests, an analysis might be an acceptable means of compliance provided that this analysis is adequately validated and covers as a minimum the changes in configurations (air intakes, engines, engine installations, etc.), engine operability (airflow, ingestion capabilities, surge margins, etc.) and thermal environment of the air intake.

For rotorcraft certified in full icing conditions, in order to determine the rotorcraft performance in icing conditions, this test point should be used to identify the engine installation losses for flight into known icing conditions, in particular if the engine is also equipped with its own screen.

POWERPLANT CONTROLS AND ACCESSORIES

[…]

CS 29.1145 Ignition switches

(a) ~~Ignition switches must control each ignition circuit on each engine.~~ For each engine, means must be provided in the cockpit so as to:

(1) control, either directly by the crew or by the crew via a system (such as the FADEC), each ignition circuit;

(2) readily allow the crew to conduct the flight and manage both ground start and in-flight restart;

(3) check the health condition of each ignition circuit; and

(4) maintain an isolation between each engine control.

(b) There must be means to quickly shut off all ignition by the grouping of switches or by a master ignition control.

(c) Each group of ignition switches, except ignition switches for turbine engines for which continuous ignition is not required, and each master ignition control, must have a means to prevent its inadvertent operation.

AMC1 29.1145(a) Ignition switches

(a) Compliance with CS 29.1145(a) is considered to be demonstrated by providing for each engine one of the following design solutions:

(1) Independent ignition controls should be provided for each ignition circuit, or

(2) A single ignition control acting on two ignition switches should be provided to control each ignition circuit via a dual-channel FADEC.

(i) Each switch should be connected to one channel of the FADEC.

(ii) The FADEC should ensure the following functions:

(A) Ability to control automatically and independently each ignition circuit of the engine

(B) Ability to perform a health monitoring of each ignition circuit for the aircraft to meet the safety objectives of CS-29

(b) The check of the health condition of each ignition circuit could be achieved in automatic or initiated test or by procedure without any difference.

SUBPART F — EQUIPMENT

GENERAL

[…]

AMC1 29.1301 Function and installation

This AMC replaces FAA AC 29-2C, § AC 29.1301 and should be used when demonstrating compliance with CS 29.1301.

(a) Explanation

It should be emphasised that CS 29.1301 applies to each item of installed equipment which includes optional equipment as well as required equipment.

(b) Procedures

(1) Information regarding installation limitations and proper functioning is normally available from the equipment manufacturers in their installation and operation manuals. In addition, some other paragraphs in FAA AC 29-2C include criteria for evaluating proper functioning of particular systems — an example is § AC 29 MG 1 for avionics equipment.

(2) CS 29.1301 is quite specific in that it applies to each item of installed equipment. It should be emphasised, however, that even though a general rule as CS 29.1301 is relevant, a rule that gives specific functional requirements for a particular system will prevail over a general rule. Therefore, if a rule exists that defines specific system functioning requirements, its provisions should be used to evaluate the acceptability of the installed system and not the provisions of this general rule. It should also be understood that an interpretation of a general rule should not be used to lessen or increase the requirements of a specific rule. CS 29.1309 is another example of a general rule, and this discussion is appropriate when applying its provisions.

(3) If optional equipment is installed, the crew may be expected to use it. This may be the case of navigation capabilities (as, for instance, LPV capability) installed on VFR rotorcraft. Therefore, the applicant should define the optional equipment and demonstrate that it complies with CS 29.1301 for its intended function. In addition, the applicant should ensure that the optional equipment does not interfere with the other systems that are required for safe operation of the rotorcraft and that its failure modes are acceptable and do not create any hazards.

GM1 29.1302 Explanatory material

[…]

2 CS 29.1302: applicability and explanatory material

(c)(6)

[…]

(iii) CS 29.777(b) and CS 2~~5~~9.779 address the direction of motion and actuation but do not encompass new types of controls, such as cursor-control devices. These requirements also do not encompass types of control interfaces that can be incorporated into displays via menus, for example, thus affecting their accessibility;

CS 29.1305 ~~Power plant~~ Powerplant instruments

The following are the required ~~power plant~~ powerplant instruments:

(a) For each rotorcraft:

(1) A carburettor air temperature indicator for each reciprocating engine;

(2) A cylinder head temperature indicator for each air-cooled reciprocating engine, and a coolant temperature indicator for each liquid-cooled reciprocating engine;

(3) A fuel quantity indicator for each fuel tank;

(4) A low-fuel warning device for each fuel tank which feeds an engine. This device must:

(i) Provide a warning to the crew when approximately 10 minutes of usable fuel remains in the tank; and

(ii) Be independent of the normal fuel quantity indicating system or be designed and constructed so as to meet the minimum safety objectives compatible with the most severe hazard induced by the combination of any failures of the fuel quantity indicator device and the low-fuel level warning device.

(5) A means to indicate the manifold pressure ~~indicator,~~ for each reciprocating engine of the altitude type;

(6) An oil pressure indicator for each pressure-lubricated gearbox;

(7) An oil pressure warning device for each pressure-lubricated gearbox to indicate when the oil pressure falls below a safe value;

(8) An oil quantity indicator for each oil tank and each rotor drive gearbox, if lubricant is self-contained;

(9) An oil temperature indicator for each engine;

(10) An oil temperature warning device to indicate unsafe oil temperatures in each main rotor drive gearbox, including gearboxes necessary for rotor phasing;

(11) A means to indicate the gas temperature ~~indicator~~ for each turbine engine;

(12) A means to indicate the gas producer ~~rotor tachometer~~ speed for each turbine engine;

[…]

(27) For rotorcraft for which a 30-minute power rating is claimed, a means must be provided to alert the pilot when the engines are at the 30-minute power rating levels, when the event begins,

when the time interval expires and, if a cumulative limit in one flight exists, when the cumulative time in one flight is reached.

(b) For Category A rotorcraft:

(1) An individual oil pressure indicator for each engine, ~~and either an independent warning device for each engine or a master warning device for the engines with means for isolating the individual warning circuit from the master warning device~~ and either an oil pressure warning for each engine or a master warning device for all engines with means for identifying the individual circuit in case of master warning.

(2) An independent fuel pressure warning device for each engine or a master warning device for all engines with provision for isolating the individual warning device from the master warning device; ~~and~~

(3) Fire warning indicators~~.~~; and

(4) When the OEI Training Mode is prescribed, a means must be provided to indicate to the pilot the simulation of an engine failure, the annunciation of that simulation, and a representation of the OEI power being provided.

AMC1 29.1305(a)(4) Powerplant instruments

FUEL QUANTITY INDICATOR AND LOW-FUEL LEVEL WARNING

This AMC provides guidance in the case where the fuel quantity indicator and the low-fuel warning device are not fully independent.

AC 29.1305 provides guidance that supports the use of specific instruments that do not meet the principle of independence (integrated avionics, ECAS, etc.). However, it does not provide guidance regarding the independence between the fuel quantity sensor and the fuel low-level sensor.

The fuel quantity sensor and the fuel low-level sensor should be independent. However, it is considered to be acceptable to place them on the same supporting structure providing that the following design precautions are ensured:

(a) They are electrically independent. Each sensor should be connected to the aircraft systems via a dedicated connector and a dedicated harness;

(b) A test capability is provided for each sensor to preclude an associated latent failure; and

(c) It is demonstrated by tests such as equipment qualification tests, slosh and vibration tests as requested in CS 29.965, analysis (such as safety analysis, particular risk analysis, zonal safety analysis, comparison with a fully independent design), or a combination thereof that no common modes can lead to the most severe hazard determined in CS 29.1305(a)(4)(ii).

CS 29.1309 Equipment, systems, and installations

~~(a) The equipment, systems, and installations whose functioning is required by this CS–29 must be designed and installed to ensure that they perform their intended functions under any foreseeable operating condition.~~

~~(b) The rotorcraft systems and associated components, considered separately and in relation to other systems, must be designed so that –~~

~~(1) For Category B rotorcraft, the equipment, systems, and installations must be designed to prevent hazards to the rotorcraft if they malfunction or fail; or~~

~~(2) For Category A rotorcraft:~~

~~(i) The occurrence of any failure condition which would prevent the continued safe flight and landing of the rotorcraft is extremely improbable; and~~

~~(ii) The occurrence of any other failure conditions which would reduce the capability of the rotorcraft or the ability of the crew to cope with adverse operating conditions is improbable.~~

~~(c) Warning information must be provided to alert the crew to unsafe system operating conditions and to enable them to take appropriate corrective action. Systems, controls, and associated monitoring and warning means must be designed to minimise crew errors which could create additional hazards.~~

~~(d) Compliance with the requirements of subparagraph (b)(2) must be shown by analysis and, where necessary, by appropriate ground, flight, or simulator tests. The analysis must consider:~~

~~(1) Possible modes of failure, including malfunctions and damage from external sources;~~

~~(2) The probability of multiple failures and undetected failures;~~

~~(3) The resulting effects on the rotorcraft and occupants, considering the stage of flight and operating conditions; and~~

~~(4) The crew warning cues, corrective action required, and the capability of detecting faults.~~

~~(e) For Category A rotorcraft, each installation whose functioning is required by this CS–29 and which requires a power supply is an 'essential load' on the power supply. The power sources and the system must be able to supply the following power loads in probable operating combinations and for probable durations:~~

~~(1) Loads connected to the system with the system functioning normally.~~

~~(2) Essential loads, after failure of any one prime mover, power converter, or energy storage device.~~

~~(3) Essential loads, after failure of:~~

~~(i) Any one engine, on rotorcraft with two engines; and~~

~~(ii) Any two engines, on rotorcraft with three or more engines.~~

~~(f) In determining compliance with subparagraphs (e)(2) and (3), the power loads may be assumed to be reduced under a monitoring procedure consistent with safety in the kinds of operations~~

~~authorised. Loads not required for controlled flight need not be considered for the two-engineinoperative condition on rotorcraft with three or more engines.~~

~~(g) In showing compliance with subparagraphs (a) and (b) with regard to the electrical system and to equipment design and installation, critical environmental conditions must be considered. For electrical generation, distribution and utilisation equipment required by or used in complying with this CS–29, except equipment covered by European Technical Standard Orders containing environmental test procedures, the ability to provide continuous, safe service under foreseeable environmental conditions may be shown by environmental tests, design analysis, or reference to previous comparable service experience on other aircraft.~~

(a) Equipment and systems required to comply with type-certification requirements, airspace requirements or operating rules, or whose improper functioning would lead to a hazard, must be designed and installed so that they perform their intended function throughout the operating and environmental conditions for which the rotorcraft is certified.

(b) The equipment and systems covered by sub-paragraph (a), considered separately and in relation to other systems, must be designed and installed such that:

(1) each catastrophic failure condition is extremely improbable and does not result from a single failure, and for Category A rotorcraft, the occurrence of any failure condition which would prevent the continued safe flight and landing of the rotorcraft is considered as catastrophic;

(2) each hazardous failure condition is extremely remote; and

(3) each major failure condition is remote.

(c) The operation of equipment and systems not covered by sub-paragraph (a) must not cause a hazard to the rotorcraft or its occupants throughout the operating and environmental conditions for which the rotorcraft is certified.

(d) Information concerning an unsafe system operating condition must be provided in a timely manner to the flight crew member responsible for taking corrective action. The information must be clear enough to avoid likely flight crew member errors.

AMC1 29.1309 Equipment, systems, and installations

As defined in AMC 29.1, the AMC to CS-29 consists of FAA AC 29-2C Change 7, dated 4 February 2016. AMC 29.1309 provides further guidance and acceptable means of compliance to supplement FAA AC 29-2C Change 7 § AC 29.1309. As such, it should be used in conjunction with FAA AC 29-2C Change 7, but should take precedence over it, where stipulated, in the demonstration of compliance.

Single failure and common-cause considerations

According to CS 29.1309(b)(1), a catastrophic failure condition must not result from the failure of a single component, part, or element of a system. Failure containment should be provided by the system design to limit the propagation of the effects of any single failure to preclude catastrophic failure conditions. In addition, there must be no common-cause failure which could affect both the single component, part, or element, and its failure containment provisions. A single failure includes any set of failures, which cannot be shown to be independent from each other. Common-cause failures (including common-mode failures) and cascading failures should be evaluated as dependent failures from the point of the root cause or the initiator. Errors in development, manufacturing, installation, and maintenance can result in common-cause failures (including common-mode failures) and cascading failures. They should, therefore, be assessed and mitigated in the frame of the common-cause and cascading failures consideration.

Sources of common-cause and cascading failures include development, manufacturing, installation, maintenance, shared resource, event outside the system(s) concerned, etc. SAE ARP4761 describes types of common-cause analyses, which may be conducted, to ensure that independence is maintained (e.g. particular risk analyses, zonal safety analyses, common-mode analyses).

While single failures should normally be assumed to occur, experienced engineering judgement and relevant service history may show that a catastrophic failure condition by a single-failure mode is not a practical possibility. The logic and rationale used in the assessment should be straightforward and obvious that the failure mode simply would not occur unless it is associated with an unrelated failure condition that would, in itself, result in a catastrophic failure condition.

By detecting the presence of, and thereby limiting the exposure time to significant latent failures that would, in combination with one or more other specific failures or events identified by safety analysis, result in a hazardous or catastrophic failure condition, periodic maintenance or flight crew checks may be used to help demonstrate compliance with CS 29.1309(b).

Development assurance process

Any analysis necessary to demonstrate compliance with CS 29.1309 (a) and (b) should consider the possibility of development errors and should focus on minimising the likelihood of those errors.

Errors made during the development of systems have traditionally been detected and corrected by exhaustive tests conducted on the system and its components, by direct inspection, and by other direct verification methods capable of completely characterising the performance of the system.

These tests and direct verification methods may be appropriate for systems containing non-complex items (i.e. items that are fully assured by a combination of testing and analysis) that perform a limited number of functions and that are not highly integrated with other rotorcraft systems. For more complex or integrated systems, exhaustive testing may either be impossible because not all system states can be determined or impractical because of the number of tests that must be accomplished. For these types of systems, compliance may be demonstrated using development assurance.

(a) System development assurance

The applicability of system development assurance should also be considered for modifications to previously certificated aircraft.

ED-79A/ARP4754A is recognised as providing acceptable guidelines for establishing a development assurance process from aircraft and systems levels down to the level where software/airborne electronic hardware (AEH) development assurance is applied.

The extent of application of ED-79A/ARP4754A to substantiate development assurance activities depends on the complexity of the systems and on their level of interaction with other systems.

(b) Software development assurance

This AMC recognises AMC 20-115 as an accepted means of compliance with CS 29.1309 (a) and (b).

(c) AEH development assurance

This AMC recognises AMC 20-152 as an acceptable means of compliance with the requirements in CS 29.1309 (a) and (b).

(d) Open problem report management

This AMC recognises AMC 20-189 as an acceptable means of compliance for establishing an open problem report management process for the system, software and AEH domains.

Integrated Modular Avionics (IMA)

This AMC recognises AMC 20-170 as an acceptable means of compliance for development and integration of IMA.

[…]

CS 29.1310 Power source capacity and distribution

For Category A rotorcraft, each installation whose functioning is required to comply with type-certification requirements, airspace requirements or operating rules, and which requires a power supply, is an 'essential load' on the power supply. The power sources and the system must be able to supply the following power loads in probable operating combinations and for probable durations:

(a) Loads connected to the system with the system functioning normally.

(b) Essential loads, after failure of any one prime mover, or one power source.

(c) Essential loads, after failure of:

(1) any one engine, on rotorcraft with two engines; and

(2) any two engines, on rotorcraft with three or more engines.

AMC1 29.1310 Power source capacity and distribution

In determining compliance with sub-paragraphs (b) and (c) of CS 29.1310, the power loads may be assumed to be reduced under a monitoring procedure consistent with safety in the kinds of operations authorised. Loads not required for controlled flight need not be considered for the two-engine inoperative condition on rotorcraft with three or more engines.

AMC1 29.1319 Equipment, systems and network information security protection

In showing compliance with CS 29.1319, the applicant may consider AMC 20-42, which provides acceptable means, guidance and methods to perform security risk assessments and mitigation for aircraft information systems.

The term 'adverse effects on the safety of the rotorcraft' should be understood in the context of information security as catastrophic or hazardous.

The term 'mitigated as necessary' clarifies that the applicant has the discretion to establish appropriate means of mitigation against security risks.

AMC1 29.1337(b) Powerplant instruments

FUEL QUANTITY INDICATOR

This AMC supplements FAA AC 29.1337 by providing clarification regarding the susceptibility of the fuel quantity indication accuracy to water contamination.

The certification specifications in CS-29 assume different natures and levels of fuel contamination to which the fuel system shall be designed tolerant.

For water contamination, CS 29.951(c) assumes a free water presence on top of a water saturated fuel (in standard conditions).

When demonstrating compliance with CS 29.1337(b), an applicant should take into account the potential water contamination as specified in CS 29.951(c).

It is expected that the fuel quantity indication should not be affected by a water contamination as specified in CS 29.951 (c).

The fuel level sensors should be designed to prevent an accumulation of water that could lead to an erroneous indication of the fuel quantity.

AMC~~1~~2 29.1337(e) Power plant instruments

[…]

LIGHTS

[…]

AMC1 29.1413(a) Safety belts: passenger warning devices

INDICATION OF WHEN SEAT BELTS SHOULD BE FASTENED

If a means to indicate to the passengers when safety belts should be fastened is provided, it should consist of an illuminated sign or signs. At least one sign should be clearly visible to each passenger, when seated.

SUBPART G — OPERATING LIMITATIONS AND INFORMATION

OPERATING LIMITATIONS

[…]

CS 29.1505 Never-exceed speed

[…]

(c) For helicopters, a stabilised Power-OFF V_{NE} denoted as V_{NE} (Power-OFF) may be established at a speed less than V_{NE} established pursuant to sub-paragraph (a), if the following conditions are met:

(1) V_{NE} (Power-OFF) is not less than a speed midway between the Power-ON V_{NE} and the speed used in meeting the requirements of:

(i) CS 29.67(a)(3) for Category A helicopters;

(ii) CS 29.65(a) for Category B helicopters, except multi-engine helicopters meeting the requirements of CS 29.67(b); and

(iii) CS 29.67(b) for multi-engine Category B helicopters meeting the requirements of CS 29.67(b).

(2) Unless it is automatically displayed to the crew, the V_{NE} (Power-OFF) is:

(i) A constant airspeed; or

(ii) A constant amount less than Power-ON V_{NE}; or

(iii) A constant airspeed for a portion of the altitude range for which certification is requested, and a constant amount less than Power-ON V_{NE} for the remainder of the altitude range.

[...]

AMC1 29.1505 Never-exceed speed

This AMC replaces FAA AC 29-2C, § AC 29.1505 and should be used when demonstrating compliance with CS 29.1505.

(a) Explanation

(1) General

CS 29.1505 requires the never-exceed speed (V_{NE}) for both Power-ON and Power-OFF flight to be established as operating limitations. The rule specifies how to establish and substantiate these limits.

(2) Power-ON limits

(i) All engines operative (AEO)

(A) The all-engines-operating V_{NE} is established by design and substantiated by flight tests. The V_{NE} limit is the most conservative value that demonstrates compliance with the structural requirements (CS 29.309), the manoeuvrability and controllability requirements (CS 29.143), the stability requirements (CS 29.173 and CS 29.175), or the vibration requirements (CS 29.251). The Power-ON V_{NE} will normally decrease as density altitude or weight increases. A variation in rotor speed may also require a variation in the V_{NE}. The regulation restricts to two the number of variables that are used to determine the V_{NE} at any given time so that a single pilot can readily ascertain the correct V_{NE} for the flight condition with a minimum of mental effort. Helicopter manufacturers have typically presented never-exceed-speed limitation data as a function of pressure altitude and temperature. This information was placarded as well as contained in the flight manual. As the weight of some derivative models was increased, EASA and the FAA accepted altitude/temperature/ V_{NE} limitations that were categorised or contained within a weight range. Literal compliance with the regulation then required that the take-off weight be calculated and then the indicated, appropriate airspeed limitation chart or placard be used for the entire flight. However, V_{NE} charts or placards based on longitudinal centre of gravity have been found to be unacceptable, since the same chart would potentially not be used throughout the flight and the pilot would thus be dealing with more than two variables to determine the V_{NE}. Alternatively, rotorcraft that are equipped with modern avionics systems may be able to automatically calculate and display the V_{NE} in an unambiguous manner as a function of the different parameters upon which it depends. For these designs, the applicant is expected to appropriately address the criticality associated with the loss and misleading presentation of the V_{NE} when compliance of such systems with CS 29.1309 is assessed. These rotorcraft should also have a method for determining the V_{NE} that complies with the regulation for all failure conditions or combinations of failure conditions that are not extremely improbable. This method is usually more conservative than the automatic system because of the limitation in the number of parameters that can be varied. A placard may be used or appropriate RFM instructions.

(B) To ensure compliance with the structural requirements (CS 29.309), vibration requirements (CS 29.251), and flutter requirements (CS 29.629), the all-engines-operating V_{NE} should be restricted so that the maximum demonstrated main rotor tip Mach number will not be exceeded at 1.11 V_{NE} for any approved combination of altitude and ambient temperature. Previous rotorcraft cold weather tests have shown that the rotor system may exhibit several undesirable and possibly hazardous characteristics due to compressibility effects at high advancing blade tip Mach numbers. As the centre of pressure of the advancing rotor blade moves aft near the blade tip due to the formation of localised upper surface shock waves, rotor system loads may increase, the rotor system may exhibit an aerodynamic instability such as rotor weave, rotorcraft vibration may increase substantially, and rotorcraft static or dynamic stability may be adversely affected. Which, if any, of these adverse characteristics are exhibited at high rotor tip Mach numbers is dependent on the design of each particular rotor system. EASA and the FAA experience has shown that some adverse characteristics exist for all the types of rotor systems (articulated, semirigid, rigid, etc.) and the various rotor blade designs evaluated at high advancing blade tip Mach numbers during past certification programmes. Therefore, it has been EASA and the FAA policy to establish V_{NE} so that it is not more than 0.9 times the maximum speed substantiated for advancing blade tip Mach number effects for the critical combination of altitude, approved Power-ON rotor speed, and ambient temperature conditions. This policy was incorporated as a specific regulatory requirement with Amendment 29-24 to § 29.1505. High main rotor tip Mach numbers obtained power off at higher-than-normal main rotor rotational speeds should not be used to establish the maximum Power-ON tip Mach number V_{NE} limit. In addition, since the onset of adverse conditions associated with high tip Mach numbers can occur with little or no warning and amplify very rapidly, no extrapolation of the maximum demonstrated main rotor tip Mach number V_{NE} limitation should be allowed.

(C) A maximum speed for use of power in excess of maximum continuous power (MCP) should be established unless structural requirements have been substantiated for the use of take-off power (TOP) at the maximum approved V_{NE} airspeed. TOP is intended for use during take-off and climb for not more than 5 minutes at relatively low airspeeds. However, EASA and the FAA experience has shown that pilots will not hesitate to use TOP at much higher than best-rate-of-climb airspeeds unless a specific limitation against TOP use above a specified airspeed is included in the RFM. Structural and fatigue substantiations have not normally included loads associated with the use of TOP at V_{NE}. Thus, a TOP airspeed limitation should be established from the structural substantiation data to preclude the accumulation of damaging rotor system and control mechanism loads through intentional use of the TOP rating at high airspeeds.

(ii) One engine inoperative (OEI)

An OEI V_{NE} is generally established through flight test and is usually near the OEI V_H of the rotorcraft. It is the highest speed at which the failure of the remaining engine must be demonstrated. For rotorcraft with more than two engines, the appropriate designation would be 'one-engine-operating' V_{NE} and would be that speed at which the last remaining engine could be failed with satisfactory handling qualities. It is possible that a rotorcraft with more than two engines could have different V_{NE} speeds depending upon the number of engines still operating. It is recommended that the OEI V_{NE} not be significantly lower than the OEI best range airspeed. For the last remaining engine failure case, a multiengine rotorcraft may require an OEI V_{NE} if the handling qualities are not satisfactory, if the rotor speed decays below the Power-OFF transient limits, or if any other unacceptable characteristic is found at speeds below the all-engine-operating V_{NE}.

(3) Power-OFF limits

(i) A Power-OFF V_{NE} may be established either by design or flight test and should be substantiated by flight tests. A Power-OFF V_{NE} that is less than the maximum Power-ON V_{NE} is generally required if the handling qualities or stability characteristics at high speed in autorotation are not acceptable. A limitation of the Power-OFF V_{NE} may also be used if the rotorcraft has undesirable or objectionable flying qualities, such as large lateral-directional oscillations, at high autorotational airspeeds. The Power-OFF V_{NE} must meet the same criteria for control margins as the Power-ON V_{NE}. The regulation requires that the Power-OFF V_{NE} be no less than the speed midway between the Power-ON V_{NE} and the speed used to comply with the rate of climb requirements for the rotorcraft. When the regulation was written, rotorcraft V_{NE} speeds were significantly lower than those of recently certificated rotorcraft. The high V_{NE} speeds of current rotorcraft result in relatively high values for the Power-OFF V_{NE}. Speeds lower than those specified in the regulation have been found acceptable through a finding of equivalent safety if the selected Power-OFF V_{NE} is equal to or greater than the Power-OFF speed for best range. In any case, the Power-OFF V_{NE} must be a high enough speed to be practical. A demonstration is required of the deceleration from the Power-ON V_{NE} for Category B rotorcraft, or OEI V_{NE} for transport rotorcraft with Category A engine isolation, to the Power-OFF V_{NE}. The transition must be made in a controlled manner with normal pilot reaction and skill.

(ii) In addition to the minimum speed requirements for Power-OFF V_{NE}, the rule restricts the manner in which Power-OFF V_{NE} can be specified when it is not automatically calculated and displayed to the crew. To reduce the crew workload, in all the cases where the Power-OFF V_{NE} is not automatically calculated, Power-OFF V_{NE} may be a constant airspeed which is less than Power-ON V_{NE} for all approved ambient conditions/gross weight combinations; a series of airspeeds varying with altitude, temperature or gross weight that is always a constant amount less than the Power-ON V_{NE} for the same ambient condition/gross weight combination; or some combination of a constant airspeed for a portion of the approved altitude range and a constant amount less than Power-ON V_{NE} for the remainder of the approved altitude range.

(b) Procedures

The tests to substantiate the different V_{NE} speeds are ordinarily conducted during the flight characteristics flight tests. The flight test procedures are discussed for the various limiting areas in earlier sub-paragraphs of this document. The controllability test techniques are covered in § AC 29.143, static stability test techniques in § AC 29.175, and the vibration test techniques in § AC 29.251.

AMC1 29.1521 Powerplant limitations

(a) Introduction

This AMC supplements FAA AC 29-2C, § AC 29.1521 and should be used in conjunction with that AC when demonstrating compliance with CS 29.1521.

(b) 30-minute power rating

(1) Explanation

The 30-minute power rating may be set at any level between the maximum continuous up to and including the take-off rating, and may be used for multiple periods of up to 30 minutes each, at any time between the take-off and landing phases in any flight.

This rating is associated with some limitations which should be adequately established and declared.

(2) Procedure

CS 29.1521(a) refers to the limits for which the engines are type certificated. This should include the 30-minute power rating usage and:

— the associated usage limit:

- maximum duration in one single shot up to 30 minutes;
- cumulative limit, if any, in one flight; and

— any other limits associated with the usage of the 30-minute power rating declared in the installation and/or operating manual of the engine.

AMC1 29.1529 Instructions for Continued Airworthiness

(a) Introduction

This AMC supplements FAA AC 29-2C, § AC 29.1529 and should be used in conjunction with that AC when demonstrating compliance with CS 29.1529.

(b) Abnormal events

The ICA should include instructions that ensure that operators conduct appropriate inspections or other actions following abnormal events in operation, maintenance or during transportation of components.

Abnormal events that should be considered include hard landings, severe gust encounters, lightning strike, exposure to high winds when parked and dropping components during maintenance or transport.

The instructions should consider the nature of the components, including but not limited to critical parts, and in particular the possibility of damage that can occur during impact or overload events that may not be detectable but could subsequently lead to premature failure in operation. In such cases, scrapping the component or parts of it may be the only appropriate action to take.

(c) Time between overhaul (TBO) development

(1) Explanation

The purpose of this AMC is to provide guidance for establishing a TBO for rotorcraft drive system gearboxes at type certificate approval and to increase it during the service life of the product.

A rotorcraft rotor drive system gearbox is usually a complex assembly composed of many parts of which a significant proportion can be critical parts. Many are rotating parts which are subject to high torque and fatigue loads, such as bearings, shafts, gears, and free wheels with the primary function of transmitting power from the engine to the rotors. Non-rotating components have other functions such as support, lubrication, load transfer or condition monitoring.

Most gearbox components are enclosed inside the housings, which prevents the possibility of detailed maintenance inspections without disassembly. As a result, to ensure that the internal gearbox components remain in serviceable condition, periodic overhauls of the assembly are typically scheduled. Overhaul allows an in-depth and periodic inspection of gearbox components, controlling and limiting the development of degradation and build-up of debris, as well as checking for cracks and other damages that may be developing. In addition, the inspection findings can determine whether parts are sufficiently protected and whether they remain in serviceable condition. In summary, the overhaul of the gearbox is intended to verify the condition of its elements, restore them to a serviceable condition or replace them where needed, and ensure that the gearbox will be safe for operation until the following overhaul. The TBO is the periodic interval between two overhauls and is traditionally defined in flight hours and calendar time.

During the type-certification process, rotorcraft drive system gearbox components are subject to various forms of analyses and tests, which assess their criticality, integrity and reliability. These assessments rely on a number of assumptions regarding the condition of the components during their service life and have an impact on aspects such as contact conditions between elements, fretting, wear, loads and environmental deterioration. The applicant should consider that the continued validity of these assumptions is typically linked to an appropriate TBO. As a result, the validation of these assumptions and the development of the TBO are processes that should be progressed in parallel after entry into service (EIS).

The final and mature TBO should normally be based on the results of investigations from in-service aircraft, overhauled gearboxes and data acquired during development, certification, and maturity tests substantiating the reliability of the parts and their capability to operate safely. However, until this data becomes available, the applicant should maintain a conservative TBO, extending it throughout the life of the product as positive supporting data from service becomes available.

(2) Guidance

For drive system gearboxes that are essential to drive the rotors, EASA considers that the initial TBO at EIS and the plan to increase it in service should be justified. For this purpose, the following should be considered by the applicant:

— Initial TBO (applicable at EIS)

At EIS, the available data supporting the justification of the TBO of a rotor drive system gearbox is typically limited. The applicant should, therefore, propose a conservative initial TBO supported by the data coming from:

- the endurance test,
- flight tests,
- other relevant tests, and
- experience on similar design having the same characteristics.

The applicant should take into account that, in general, only limited experience of the real operating environment and conditions for a new gearbox is available at EIS.

This initial TBO should ensure enough opportunities to verify the condition of internal gearbox components in order to validate the assumptions made at the time of certification, preventing that any compromised assumption may lead to an in-service catastrophic or hazardous failure.

— TBO step increase

The increase of a gearbox TBO in service should be accomplished in steps providing confidence progressively in the validity of the certification assumptions. Each TBO step increase should:

- only be proposed when the current TBO is supported by a sufficient number of gearbox overhaul inspection results;
- be based on a sufficient number of gearboxes from the fleet to be inspected, and take into account the representativeness of operational and environmental aspects of the selected samples to represent the full spectrum of gearbox usage;
- be based on technical justifications from overhauled gearboxes (e.g. condition of inspected parts, evidence from similar designs, etc.), maturity testing and in-service feedback (incidents, health and usage monitoring system (HUMS) data, etc.); and
- be completed prior to formally increasing the TBO to verify acceptable behaviour and condition of the gearbox components prior to starting a new increase phase.

— Management of TBO steps

The process for managing the evolution of the TBO of drive system gearboxes should be documented in a TBO maturity plan. This should include:

- planned increase steps and target TBO, technical criteria for the validation of the steps planned and justification of the proposed plan (see note 1);
- definition of the number of gearboxes and selection criteria considering operation and environment (see note 1);
- definition of responsible parties for performing the TBO step increase validation inspections, activities involved and information to be reported;
- proposed analysis process of the inspection results, responsible parties and methods of analysis; and
- the TBO step increase validation process and associated deliverables (see note 2).

Any findings arising from the TBO development process which might bring into question the suitability of the current TBO or impair the capability of the gearbox to reach the planned increase in TBO should be reported to the Agency.

Finally, if a major change is introduced to or affecting a drive system gearbox, the applicant should evaluate the need to revise the TBO and incorporate additional steps in the gearbox TBO maturity plan.

Note 1: The TBO maturity plan and the associated TBO increase validation criteria should be defined by the applicant and provided to the Agency during the certification process. The results of the process of validation of each step might lead to revisions of the maturity plan.

Note 2: The acceptance of each individual step as well as the closure of the maturity plan should be formally endorsed by the applicant and duly documented.

MARKINGS AND PLACARDS

[…]

CS 29.1549 Powerplant instruments

For each required powerplant instrument, as appropriate to the type of instruments:

(a) Each maximum and, if applicable, minimum safe operating limit must be marked with ~~a red radial or~~ a red line;

(b) Each normal operating range must be depicted ~~must be marked with a green arc or green line, not extending beyond the maximum and minimum safe limits~~ as a green or unmarked range;

(c) Each take-off and precautionary range must be marked with a yellow ~~arc~~ range or yellow line;

(d) Each engine or propeller range that is restricted because of excessive vibration stresses must be marked with red ~~arcs~~ ranges or red lines.

(e) Each OEI limit or approved operating range must be marked to be clearly differentiated from the markings of sub-paragraphs (a) to (d) except that no marking is normally required for the 30-second OEI limit.

CS 29.1555 Control markings

(a) Each cockpit control, other than primary flight controls or controls whose function is obvious, must be plainly marked as to its function and method of operation.

(b) For powerplant fuel controls:

(1) Each fuel tank selector valve control must be marked to indicate the position corresponding to each tank and to each existing cross feed position;

(2) If safe operation requires the use of any tanks in a specific sequence, that sequence must be marked on, or adjacent to, the selector for those tanks; and

(3) Each valve control for any engine of a multi-engine rotorcraft must be marked to indicate the position corresponding to each engine controlled.

(c) Usable fuel capacity must be marked as follows:

(1) For fuel systems having no selector controls, the usable fuel capacity of the system must be indicated at the fuel quantity indicator~~.~~ unless it is:

(i) provided by another system or item of equipment readily accessible to the pilot; and

(ii) contained in the limitations section of the rotorcraft flight manual.

(2) For fuel systems having selector controls, the usable fuel capacity available at each selector control position must be indicated near the selector control.

AMC1 29.1555 Control markings

This AMC supplements FAA AC 29.1555.

[…]

AMC2 29.1555 Control markings

CLARIFICATION OF TERMS

This AMC supplements FAA AC 29.1555.

The fuel quantity should be understood as the actual amount of usable fuel at a given time contained within a tank of constant fuel capacity.

The usable fuel capacity of a tank is the maximum amount of usable fuel that the tank can have. It was historically used to define the fuel quantity for flight planning when the fuel quantity indicator displayed only levels (such as full, half, etc.) of the total capacity. The pilot had to calculate the fuel quantity in an appropriate unit based on the usable fuel capacity of the tank and the level shown on the fuel quantity indicator.

The design and accuracy in all operating and environmental conditions of modern fuel quantity indication systems decreases the crew workload by displaying directly the fuel quantity in the appropriate unit. This data can be used for compliance demonstration.