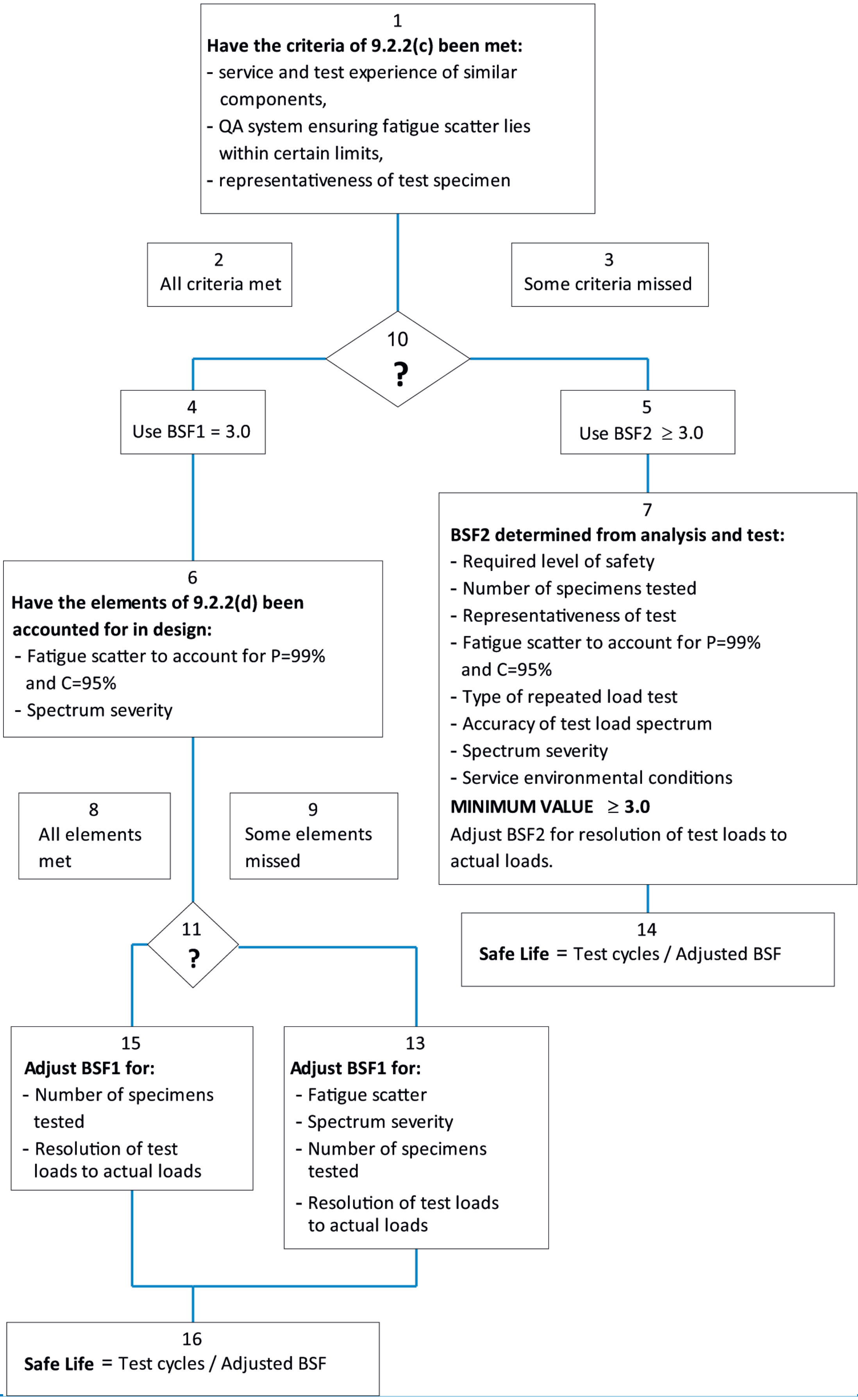
difference between the test spectrum and the assumed flight-by-flight spectrum. In addition, an adjustment to the number of test cycles may be justified by raising or lowering the test load levels as long as appropriate data supports the applicant’s position. Other effects to be considered are different failure locations, different response to fretting conditions, temperature effects, etc. The analytical approach should use well-established methods or be supported by test evidence.

SCATTER FACTOR FLOW CHART



9.3. Replacement times

Replacement times should be established for parts with established safe-lives and should, under CS 25.571(a)(3), be included in the information prepared under CS 25.1529. These replacement times can be extended if additional data indicates an extension is warranted. Important factors which should be considered for such extensions include, but are not limited to, the following:

9.3.1. Comparison of original evaluation with service experience

9.3.2. Recorded load and stress data

Recorded load and stress data entails instrumenting aeroplanes in service to obtain a representative sampling of actual loads and stresses experienced.

The data to be measured includes airspeed, altitude and load factor versus time ; or airspeed, altitude and strain ranges versus time ; or similar data. This data, obtained by instrumenting aeroplanes in service, provides a basis for correlating the estimated loading spectrum with the actual service experience.

9.3.3. Additional analyses and tests

If additional test data and analyses based on repeated load tests of additional or surviving specimens are obtained, a re-evaluation of the established safe-life can be made.

9.3.4. Tests of parts removed from service

Repeated load tests of replaced parts can be utilised to re-evaluate the established safe-life. The tests should closely simulate service loading conditions.

Repeated load testing of parts removed from service is especially useful where recorded load data obtained in service are available since the actual loading experienced by the part prior to replacement is known.

9.3.5. Repair or rework of the structure

In some cases, repair or rework of the structure can gain further life.

9.4. Type design developments and changes

For design developments, or design changes, involving structural configurations similar to those of a design already shown to comply with the applicable provisions of CS 25.571(c), it might be possible to evaluate the variations in critical portions of the structure on a comparative basis. A typical example would be redesign of the landing gear structure for increased loads. This evaluation should involve analysis of the predicted stresses of the redesigned primary structure and correlation of the analysis with the analytical and test results used in showing compliance of the original design with CS 25.571(c).

10. DISCRETE SOURCE DAMAGE

(a) General

The purpose of this section is to establish EASA guidelines for the consistent selection of load conditions for residual strength substantiation in showing compliance with CS 25.571(e) and CS 25.903(d). The intent of these guidelines is to define, with a satisfactory level of confidence, the load conditions that will not be exceeded on the flight during which the specified incident of CS 25.571(e) or CS 25.903(d) occurs. In defining these load conditions, consideration has been given to the expected damage to the aeroplane, the anticipated response of the pilot at the time of the incident, and the actions of the pilot to avoid severe load environments for the remainder of the flight consistent with his/her knowledge that the aeroplane may be in a damaged state. Under CS 25.631 continued safe flight and landing is required following the bird impact. Following the guidance of this paragraph for assessing structural damage to any part whose failure or partial failure may prevent continued safe flight and landing is an acceptable means of compliance to CS 25.631.

(b) The maximum extent of immediately obvious damage from discrete sources (CS 25.571(e)) should be determined and the remaining structure shown, with an acceptable level of confidence, to have static strength for the maximum load (considered as ultimate load) expected during completion of the flight. For uncontained rotor failure addressed under the CS 25.903(d) requirements and for applicants following AMC 20-128A, likely structural damage may be assumed to be equivalent to that obtained by using the rotor burst model and associated trajectories defined in AMC 20-128A, paragraph 9.0 'Engine and APU Failure Model'. This assessment should also include an evaluation of the controllability of the aircraft in the event of damage to the flight control system.

(c) The loads considered as ultimate should not be less than those developed from the following:

(1) At the time of the occurrence:

(i) the maximum normal operating differential pressure including the external aerodynamic pressures during 1.0 g level flight, multiplied by a 1.1 factor, combined with 1.0 g flight loads;

(ii) starting from 1.0 g level flight at speeds up to Vc, any manoeuvre or any other flight path deviation caused by the specified incident of CS 25.571(e), taking into account any likely damage to the flight controls and pilot normal corrective action.

(2) For the continuation of the flight, the maximum appropriate cabin differential pressure (including the external aerodynamic pressure), combined with:

(i) 70 % of the limit flight manoeuvre loads as specified in CS 25.571(b) and, separately;

(ii) at the maximum operational speed, taking into account any appropriate reconfiguration and flight limitations, the 1.0 g loads plus incremental loads arising from application of 40 % of the limit gust velocity and turbulence intensities as specified in CS 25.341 at Vc.

(d) At any time, the aeroplane must be shown, by analysis, to be free from flutter and other aeroelastic instabilities up to the boundary of the aeroelastic stability envelope described in CS 25.629(b)(2) with any change in structural stiffness resulting from the incident, consistent with CS 25.629(d)(8), CS 25.571(e), and CS 25.903(d).

11. ESTABLISHING THE LOV AND MAINTENANCE ACTIONS TO PREVENT WFD

(a) Structural maintenance programme

Theoretically, if an aircraft is properly maintained it could be operated indefinitely. However, it should be noted that structural maintenance tasks for an aircraft are not constant with time. Typically, tasks are added to the maintenance programme as the aircraft ages. It is reasonable to expect then that confidence in the effectiveness of the current structural maintenance tasks may not, at some future point, be sufficient for continued operation.

Maintenance tasks for a particular aircraft can only be determined based on what is known about that aircraft model at any given time: from analyses, tests, service experience, and teardown inspections. Widespread fatigue damage is of particular concern because inspection methods cannot be relied on solely to ensure the continued airworthiness of aircraft indefinitely. When inspections are focused on details in small areas and have a high probability of detection, they may be used by themselves to ensure continued airworthiness, unless or until there are in-service findings. Based on findings, these inspections may need to be modified, and it may be necessary to modify or replace the structure rather than continue with the inspection alone.

When inspections examine multiple details over large areas for relatively small cracks, they should not be used by themselves. Instead, they should be used to supplement the modification or replacement of the structure. This is because it would be difficult to achieve the probability of detection required to allow inspection to be used indefinitely as a means to ensure continued operational safety.

To prevent WFD from occurring, the structure must, therefore, occasionally be modified or replaced. Establishing all the replacements and modifications required to operate the aircraft indefinitely is an unbounded problem. This problem is solved by establishing an LOV of the engineering data that supports the structural maintenance programme. All necessary modifications and replacements are required to be established to ensure continued airworthiness up to the LOV. See paragraph 11(f) for the steps to extend the LOV.

(b) Widespread fatigue damage

Structural fatigue damage is progressive. It begins as minute cracks, and those cracks grow under the action of repeated stresses. It can be due to normal operational conditions and design attributes, or to isolated incidents such as material defects, poor fabrication quality, or corrosion pits, dings, or scratches. Fatigue damage can occur locally, in small areas or structural design details, or globally. Global fatigue damage is general degradation of large areas of structure with similar structural details and stress levels. Global damage may occur within a single structural element, such as a single rivet line of a lap splice joining two large skin panels (multiple site damage). Or it may be found in multiple elements, such as adjacent frames or stringers (multiple element damage). Multiple site damage and multiple element damage cracks are typically too small initially to be reliably detected with normal inspection methods. Without intervention these cracks will grow, and eventually compromise the structural integrity of the aircraft in a condition known as widespread fatigue damage. Widespread fatigue damage is increasingly likely as the aircraft ages, and is certain to occur if the aircraft is operated long enough without any intervention.

(c) Steps for establishing an LOV

The LOV is established as an upper limit to aeroplane operation with the inspections and other procedures provided under CS 25.1529 and Appendix H. The LOV is required by CS 25.571(a)(3) and is established because of increased uncertainties in fatigue and damage tolerance assessment and the probable development of widespread fatigue damage associated with aeroplane operation past the limit.

To support the establishment of the LOV, the applicant must demonstrate by test evidence and analysis at a minimum, and, if available, service experience and teardown inspection results of high-time aircraft, that WFD is unlikely to occur in that aircraft up to the LOV.

The process for establishing an LOV involves four steps:

— identifying a 'candidate LOV';

— identifying WFD-susceptible structure;

— performing a WFD evaluation of all susceptible structure; and

— finalising the LOV and establishing necessary maintenance actions.

Step 1 — Candidate LOV

Any LOV can be valid as long as it has been demonstrated that the aircraft model will be free from WFD up to the LOV based on the aircraft's inherent fatigue characteristics and that any required maintenance actions are in place. Early in the certification process applicants typically establish design service goals or their equivalent and set a design service objective to have structure remain relatively free from cracking, up to the design service goal. A recommended approach sets the 'candidate LOV' equal to the design service goal. The final LOV would depend on both how well that design objective was met, and the applicant's consideration of the economic impact of maintenance actions required to preclude WFD up to the final LOV.

Step 2 — Identify WFD-susceptible structure

The applicant should identify the structure that is susceptible to WFD to support post-fatigue test teardown inspections or residual strength testing necessary to demonstrate that WFD will not occur in the aircraft structure up to the LOV. Appendix 2 to AMC 20-20 provides examples and illustrations of structure where multiple site damage or multiple element damage has been documented. The list in Appendix 2 to AMC 20-20 is not meant to be inclusive of all structure that might be susceptible to WFD on any given aircraft model and it should only be used for general guidance. It should not be used to exclude any particular structure.

The applicant should do the following when developing the list of structure susceptible to WFD:

(1) Establish criteria that could be used for identifying what structure is susceptible to WFD based on the definitions of multiple site damage, multiple element damage, and WFD. For example, structural details and elements that are repeated over large areas and operate at the same stress levels are obvious candidates. The criteria should be part of the applicant's compliance data.

(2) Provide supporting rationale for including and excluding specific structural areas. This should be part of the applicant's compliance data.

(3) Identify the structure to a level of detail required to support post-test activities that the applicant will use to evaluate the residual strength capabilities of the structure. Structure is free from WFD if the residual strength meets or exceeds that required by CS 25.571(b). Therefore, post-test activities such as teardown inspections and residual strength tests must provide data that support the determination of strength.

    — For teardown inspections, specific structural details (e.g. holes, radii, fillets, cut-outs) need to be identified.

— For residual strength testing, the identification at the component or subcomponent level (e.g. longitudinal skin splices) may be sufficient.

Step 3 — Evaluation of WFD-susceptible structure

Applicants must evaluate all susceptible structure identified in Step 2. Applicants must demonstrate, by full-scale fatigue test, evidence that WFD will not occur in the aircraft structure prior to the LOV. This demonstration typically entails full-scale fatigue testing, followed by teardown inspections and a quantitative evaluation of any finding or residual strength testing, or both. Additional guidance about full-scale fatigue test evidence is included in Appendix 2 to this AMC.

Step 4 — Finalise LOV

After all susceptible structure has been evaluated, finalise the LOV. The results of the evaluations performed in Step 3 will either demonstrate that the strength at the candidate LOV meets or exceeds the levels required by CS 25.571(b) or not. If it is demonstrated that the strength is equal to or greater than that required, the final LOV could be set to the candidate LOV without further evidence. If it is demonstrated that the strength is less than the required level, at least two outcomes are possible:

(1) The final LOV may be equal with the candidate LOV. However, this would result in maintenance actions, design changes, or both, maintenance actions and design changes, to support operation of aircraft up to LOV. For MSD/MED, the applicant may use damage tolerance-based inspections to supplement the replacement or modification required to preclude WFD when those inspections have been shown to be practical and reliable.

(2) The final LOV may be less than the candidate LOV. This could reduce the need for maintenance actions or making design changes.

<u>Maintenance actions</u>

In some cases maintenance actions may be necessary for an aircraft to reach its LOV. These maintenance actions could include inspections, modifications, replacements, or any combination thereof.

— For initial certification, these actions should be specified as airworthiness limitation items and incorporated into the ALS of the ICA.

— For post-certified aircraft, these actions should be specified as service information by the TCH or included in an updated ALS and may be mandated by Airworthiness Directives.

<u>Design changes</u>

The applicant may determine that developing design changes to prevent WFD in future production aircraft is to their advantage. The applicant must substantiate the design changes according to the guidance contained in this AMC

In addition to the technical considerations, the LOV may be influenced by several other factors, including:

— maintenance considerations;

— operator’s input; and

— economics.

(d) Airworthiness Limitations Section (ALS)

In accordance with Part-21 the TCH must provide the ICA (which includes the ALS) with the aircraft. However, the TCH may or may not have completed the full-scale fatigue test programme at the time of type certification.

Under CS 25.571, EASA may issue a type certificate for an aircraft model prior to the applicant's completion of the full-scale fatigue testing, provided that EASA has agreed to the applicant's plan for completing the required tests.

Until the full-scale fatigue testing is completed and EASA has approved the LOV, the applicant must establish a limitation that is equal to not more than one half of the number of cycles accumulated on the test article supporting the WFD evaluation. Under Appendix H to CS-25, the ALS must contain the limitation preventing operation of the aircraft beyond one half of the number of cycles accumulated on the fatigue test article approved under CS 25.571. This limitation is an airworthiness limitation. No aircraft may be operated beyond this limitation until fatigue testing is completed and an LOV is approved. As additional cycles on the fatigue test article are accumulated, this limitation may be adjusted accordingly. Upon completion of the full-scale fatigue test, applicants should perform specific inspections and analyses to determine whether WFD has occurred. Additional guidance on post-test WFD evaluations is included in Appendix 2 to this AMC.

At the time of type certification, the applicant should also show that at least one calendar year of safe operation has been substantiated by the fatigue test evidence agreed to be necessary to support other elements of the damage tolerance and safe-life substantiations. Some of these tests may require application of scatter factors greater than two resulting in more restrictive operating limitations on some parts of the structure.

After the full-scale fatigue test and the WFD evaluation have been completed, the applicant must include the following in the ALS:

— Under Appendix H to CS 25, the ALS must contain the LOV stated as a number of total accumulated flight cycles or flight hours approved under CS 25.571; and

— Depending on the results of the evaluation under Step 3 above, the ALS may also include requirements to inspect, modify or replace the structure.

(e) Repairs and type design changes

Any person applying for a change to a type certificate (TC) or a supplemental type certificate (STC) must demonstrate that any affected structure is free from WFD up to the LOV. (*Note:* It is possible that the STC applicant may generate a new LOV for the aeroplanes as part of the STC limitations).

Applicants for a major repair to the original aircraft or to an aircraft modified under a major change or an STC must demonstrate that any affected structure is free from WFD up to the LOV.

The evaluation should assess the susceptibility of the structure to WFD and, if it is susceptible, demonstrate that WFD will not occur prior to the LOV. If WFD is likely to occur before LOV is reached, the applicant must either:

(1) redesign the proposed repair to preclude WFD from occurring before the aircraft reaches the LOV; or

(2) develop maintenance actions to preclude WFD from occurring before the aircraft reaches the LOV; or

(3) for significant major changes and STCs only, establish a new LOV.

For repairs, the applicant must identify and include these actions as part of the repair. For major changes and STCs, the applicant must identify and include these actions as airworthiness limitation items in the ALS of the ICA. WFD evaluation is considered part of the fatigue and damage tolerance evaluation with respect to the three-stage repair approval process.

(f) Extended LOV

To extend an LOV, an application for a major change is required.

Typically, the data necessary to extend an LOV includes additional full-scale fatigue test evidence. The primary source of this test evidence should be full-scale fatigue testing. This testing should follow the guidance contained in Appendix 2 to this AMC.

[Amdt 25/19]

Appendix 1 – Crack growth analysis and tests

ED Decision 2017/015/R

Crack growth characteristics should be determined for each detail design point identified in accordance with 7(f) above. This information, when combined with the results from the residual strength analyses and tests, will be the basis for establishing the inspection requirements as discussed in Section 8. Crack growth characteristics can be determined by analysis or test. However, due to the large number of detail design points that are typically evaluated, and the practical limitations involved with testing, analyses are generally relied on to determine crack growth at the detail design point.

(a) Analyses. In order to perform a crack-growth analysis a number of key elements are needed. These include:

(1) a load/stress spectrum applicable to the detail design point;

(2) an initial crack size and shape to be assumed;

(3) a cracking scenario to be followed;

(4) applicable stress intensity solution(s);

(5) a crack growth algorithm; and

(6) material crack growth rate properties.

A loading spectrum must be developed for each detail design point. It is derived from the overall aircraft usage spectrum that is discussed in paragraph 6(b). The spectra at each detail design point may be modified for various reasons. The most common modification for metallic structure involves the deletion of high infrequent loads that may have an unrepresentative beneficial effect on crack growth if retardation is considered. Also, local load events that are not part of the overall aircraft spectrum should be included (e.g. flutter damper loads during pre-flight control surface checks).

The initial crack size and shape and subsequent cracking scenario to be followed are problem-dependent.

Applicable stress intensity solutions may be available in the public domain or may need to be developed. Many references exist which provide technical guidance for the application and development of stress intensity solutions. Care should be taken to ensure that the reference stress used for the spectrum load and stress intensity solution are compatible.

Crack-growth algorithms used in predicting crack extension range from simple linear models to complex ones that can account for crack growth retardation and acceleration. It is generally accepted that the use of a linear model will result in conservative results. A non-linear model, on the other hand, can be conservative or non-conservative and generally requires a higher level of validation and analysis/test correlation to adequately validate the accuracy of the algorithm. Coupon testing should be performed using representative materials and spectra types (e.g. wing lower cover, pylon support lug, horizontal-stabiliser upper cover) that will be encountered in the course of the overall aircraft crack-growth evaluation.

Crack growth rate data (e.g. da/dN vs ΔK vs R, da/dN vs ΔKeff) for many common aerospace materials is available in the public domain. Additionally, testing standards (e.g. ASTM) exist for performing tests to gather this data. The generally accepted practice is to use typical or average representation of this data for performing crack growth evaluations.

(b) Tests. Crack-growth testing using coupons is typically performed to generate crack growth rate data and to validate crack growth algorithms used for analyses. Simple specimens are generally used that have well-established stress intensity solutions for the characteristic cracking that can