CS-25 AMENDMENT 19 — CHANGE INFORMATION

EASA publishes amendments to certification specifications as consolidated documents. These documents are used for establishing the certification basis for applications made after the date of entry into force of the amendment.

Consequently, except for a note '[Amdt No: 25/19]' under the amended paragraph, the consolidated text of CS-25 does not allow readers to see the detailed changes introduced by the new amendment. To allow readers to also see these detailed changes this document has been created. The same format as for publication of Notices of Proposed Amendments (NPAs) has been used to show the changes:

(a) deleted text is marked with ~~strike through~~;

(b) new or amended text is highlighted in grey;

(c) an ellipsis (…) indicates that the remaining text is unchanged in front of or following the reflected amendment.

BOOK 1

SUBPART B — FLIGHT

Amend CS 25.147(a)(1) as follows:

CS 25.147 Directional and lateral control

(See AMC 25.147)

(a) *Directional control; general*.

(…)

(1) 'The critical engine inoperative and its propeller (if applicable) in the minimum drag position;'

(…)

SUBPART C — STRUCTURE

Amend CS 25.571 as follows:

CS 25.571 Damage tolerance and fatigue evaluation of structure

(See AMC 25.571)

(a) *General*. An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, manufacturing defects, environmental deterioration ~~corrosion~~ or accidental damage, will be avoided throughout the operational life of the aeroplane. This evaluation must be conducted in accordance with the provisions of sub-paragraph~~s~~ (b) ~~and (c)~~ of this paragraph, except as specified in subparagraph ~~(c)~~ (a)(4) of this paragraph, for each part of the structure ~~which~~ that could contribute to a catastrophic failure ~~(such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachment). (See AMC 25.571 (a) (b) and (e)~~. Additionally, a discrete source damage evaluation must be conducted in accordance with subparagraph (e) of this paragraph, and ~~For turbine engine powered aeroplanes,~~ those parts that could contribute to a catastrophic failure must also be evaluated ~~under~~ in accordance with sub-paragraph (d) of this paragraph. In addition, the following apply:

(1) ~~Each evaluation required by this paragraph~~ The evaluations of subparagraphs (b) and (c) must include: -

(i) The typical loading spectra, temperatures, and humidity expected in service;

(ii) The identification of principal structural elements and detail design points, the failure of which could ~~cause~~ contribute to a catastrophic failure of the aeroplane; and

(iii) An analysis, supported by test evidence, of the principal structural elements and detail design points identified in subparagraph (a)(1)(ii) of this paragraph.

(2) The service history of aeroplanes of similar structural design, taking due account of differences in operating conditions and procedures, may be used in the evaluations required by this paragraph.

(3) Based on the evaluations required by this paragraph, inspections or other procedures must be established, as necessary, to prevent catastrophic failure and must be included in the Airworthiness Limitations S~~s~~ection of the Instructions for Continued Airworthiness required by CS 25.1529. The limit of validity of the engineering data that supports the structural maintenance programme (hereafter referred to as LOV), stated as a number of total accumulated flight cycles or flight hours or both, established by this paragraph, must also be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness.

(4) If the results of the evaluation required by subparagraph (b) show that damage tolerance-based inspections are impractical, then an evaluation must be performed in accordance with the provisions of subparagraph (c).

If the results of the evaluation show that damage tolerance-based inspections are practical, then inspection thresholds must be established for all principal structural elements and detail design points. For the following types of structure, the threshold must be established based on analyses and/or tests, assuming the structure contains an initial flaw representative of a defect or damage of the maximum probable size that could exist as a result of manufacturing processes or manufacturing or service-induced damage:

(i) single load path structure; and

(ii) multiple load path 'fail-safe' structure and crack arrest 'fail-safe' structure, where it cannot be demonstrated that the resulting load path failure or partial failure (including arrested cracks) will be detected and repaired during normal maintenance, inspection, or operation of an aeroplane prior to failure of the remaining structure.

(5) Inspection programmes must be established to protect the structure evaluated under subparagraph (b) and (c) against the effects of environmental deterioration and service-induced accidental damage. In addition, a baseline corrosion and prevention control programme (CPCP) must be established. The Airworthiness Limitations Section of the Instructions for Continued Airworthiness must include a statement that requires the operator to include a CPCP in their maintenance programme that will control corrosion to Level 1 or better.

(b) *Fatigue and ~~D~~damage ~~T~~tolerance ~~(fail-safe)~~ evaluation.*

The evaluation must include a determination of the probable locations and modes of damage due to fatigue, environmental deterioration (e.g. corrosion), or accidental damage. ~~The determination must be by analysis~~ Repeated load and static analyses, supported by test evidence and (if available) service experience, must be incorporated in the evaluation. Damage at multiple sites due to prior fatigue exposure (including special consideration of widespread fatigue damage) must be included in the evaluation where the design is such that this type of damage ~~can be expected to~~ could occur. ~~The evaluation must incorporate repeated load and static analysis supported by test evidence.~~ An LOV must be established that corresponds to the period of time, stated as a number of total accumulated flight cycles or flight hours or both, for which it has been demonstrated by full-scale fatigue test evidence that widespread fatigue damage will not occur in the aeroplane structure.

The type certificate may be issued prior to completion of the full-scale fatigue testing provided that EASA has approved a plan for completing the required tests and analyses, and that at least one calendar year of safe operation has been substantiated at the time of type certification. In addition, the Airworthiness Limitations Section of the Instructions for Continued Airworthiness must specify an interim limitation restricting aircraft operation to not more than half the number of the flight cycles or flight hours accumulated on the fatigue test article, until such testing is completed, freedom from widespread fatigue damage has been established and the LOV is approved.

The extent of damage for residual strength evaluation at any time within the operational life of the aeroplane must be consistent with the initial detectability and subsequent growth under repeated loads.

The residual strength evaluation must show that the remaining structure is able to withstand loads (considered as static ultimate loads) corresponding to the following conditions:

(1) The limit symmetrical manoeuvring conditions specified in CS 25.3317 at all speeds up to V_C and in CS 25.345.

(2) The limit gust conditions specified in CS 25.341 at the specified speeds up to V_C and in CS 25.345.

(3) The limit rolling conditions specified in CS 25.349 and the limit unsymmetrical conditions specified in CS 25.367 and 25.427(a) through (c), at speeds up to V_C.

(4) The limit yaw manoeuvring conditions specified in CS 25.351 at the specified speeds up to V_C.

(5) For pressurised cabins, the following conditions:

(i) The normal operating differential pressure combined with the expected external aerodynamic pressures applied simultaneously with the flight loading conditions specified in subparagraphs (b)(1) to (b)(4) of this paragraph if they have a significant effect.

(ii) The maximum value of normal operating differential pressure (including the expected external aerodynamic pressures during the 1 g level flight) multiplied by a factor of 1.15 omitting other loads.

(6) For landing gear and other ~~directly~~ affected airframe structure, the limit ground loading conditions specified in CS 25.473, 25.491, and 25.493.

If significant changes in structural stiffness or geometry, or both, follow from a structural failure, or partial failure, the effect on damage tolerance must be further evaluated. ~~investigated. (See AMC 25.571 (b) and (e).) The residual strength requirements of this sub-paragraph (b) apply, where the critical damage is not readily detectable. On the other hand, in the case of damage which is readily detectable within a short period, smaller loads than those of subparagraphs (b)(1) to (b)(6) inclusive may be used by agreement with the Authority. A probability approach may be used in these latter assessments, substantiating that catastrophic failure is extremely improbable. (See AMC 25.571 (a), (b) and (e) paragraph 2.1.2.)~~

(c) *Fatigue (safe-life) evaluation*.

Compliance with the damage-tolerance requirements of sub-paragraph (b) of this paragraph is not required if the applicant establishes that their application for the particular structure is impractical. This structure must be shown by analysis, supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe-life scatter factors must be applied. Until such time as all

testing that is required for compliance with this subparagraph is completed, the replacement times provided in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness may not exceed the total accumulated flight cycles on the test article test life divided by the applicable scatter factor.

(d) *Sonic fatigue strength.*

It must be shown by analysis, supported by test evidence, or by the service history of aeroplanes of similar structural design and sonic excitation environment, that: **-**

(1) Sonic fatigue cracks are not probable in any part of the flight structure subject to sonic excitation; or

(2) Catastrophic failure caused by sonic fatigue cracks is not probable assuming that the loads prescribed in sub-paragraph (b) of this paragraph are applied to all areas affected by those cracks.

(e) ~~*Damage-tolerance (D*~~*discrete source*~~*)*~~ *damage tolerance evaluation*.

The aeroplane must be capable of successfully completing a flight during which likely structural damage occurs as a result of ~~–(1)~~ bird impact as specified in CS 25.631~~,~~

~~(2) Reserved~~

~~(3) Reserved~~

~~(4) Sudden decompression of compartments as specified in CS 25.365 (e) and (f)~~.

The damaged structure must be able to withstand the static loads (considered as ultimate loads) which are reasonably expected to occur at the time of the occurrence and during the completion of the flight. Dynamic effects on these static loads do not need ~~not~~ to be considered. Corrective action to be taken by the pilot following the incident, such as limiting manoeuvres, avoiding turbulence, and reducing speed, may be considered. If significant changes in structural stiffness or geometry, or both, follow from a structural failure or partial failure, the effect on damage tolerance must be further investigated. ~~(See AMC 25.571(a), (b) and (e), paragraph 2.7.2 and AMC 25.571 (b) and (e).)~~

SUBPART D — DESIGN AND CONSTRUCTION

Amend CS 25.603 as follows:

CS 25.603 Materials

(See AMC 25.603)

For Composite Materials, see AMC 20-29.

For use of glass in passenger cabins, see AMC 25.603(a))

(…)

Amend CS 25.785 as follows:

CS 25.785 Seats, berths, safety belts and harnesses

(…)

(h) Each seat located in the passenger compartment and designated for use during take-off and landing by a cabin crew member required by the Operating Rules must be:–

(…)

(2) To the extent possible, without compromising proximity to a required floor level emergency exit, located to provide a direct view of the cabin area for which the cabin crewmember is responsible. (See AMC 25.785(h)(2))

(…)

Create a new CS 25.788 as follows:

CS 25.788 Passenger amenities

(See AMC 25.788)

(a) *Showers*: If a shower cubicle is installed (See AMC 25.788(a) and AMC 25.1447(c)(3)):

(1) audio and visual 'Return to seat' indications, readily audible and visible to a shower-cubicle occupant, and activated at the same time as the signs required by CS 25.791(b), must be provided;

(2) audio and visual indications of the need for oxygen use, readily audible and visible to a shower-cubicle occupant, and activated in the case of cabin depressurisation or deployment of the oxygen-dispensing units in the cabin, must be provided;

(3) placards must be installed to indicate that the shower cubicle must not be used for the stowage of cargo or passenger baggage;

(4) there must be means in the cubicle to steady oneself in moderately rough air; and

(5) the shower cubicle must be designed in a way to preclude anyone from being trapped inside. If a locking mechanism is installed, it must be capable of being unlocked from the inside and the outside without the aid of any tool.

(b) *Large display panels*: Any large display panel installed in the passenger compartment must not be a source of danger to occupants when submitted to any of the following conditions (See AMC 25.788(b)):

(1) each relevant flight and ground load conditions (including the emergency landing conditions prescribed in CS 25.561);

(2) any load to be expected in service; and

(3) a cabin depressurisation.

Amend CS 25.807 as follows:

CS 25.807 Emergency exits

(…)

(e) *Uniformity.* Exits must be distributed as uniformly as practical, taking into account passenger seat distribution. (See AMC 25.807(e))

(…)

Amend CS 25.811 as follows:

CS 25.811 Emergency exit marking

(…)

(d) The location of each passenger emergency exit must be indicated by a sign visible to occupants approaching along the main passenger aisle (or aisles). There must be (See AMC 25.811(d)) ~~—~~:

(…)

(e) The location of the operating handle and instructions for opening exits from the inside of the aeroplane must be shown in the following manner: (…)

(4) All Type II and larger passenger emergency exits with a locking mechanism released by motion of a handle, must be marked so as to its operation by an ~~red~~ arrow with a shaft at least 19 mm (0.75 inches) wide, adjacent to the handle, that indicates the full extent and direction of the unlocking motion required. The word OPEN must be horizontally situated adjacent to the arrowhead and must be in ~~red~~ capital letters at least 25 mm (1 inch) high. The arrow and word OPEN must be located on a background, which provides adequate contrast. (See AMC 25.811(e)(4))

(…)

Amend CS 25.812 as follows:

CS 25.812 Emergency lighting

(…)

(b) Emergency exit signs~~—~~

(1) For aeroplanes that have a passenger-seating configuration, excluding pilot seats, of 10 seats or more must meet the following requirements:

(i) Each passenger emergency exit locator sign required by CS 25.811 (d)(1) and each passenger emergency exit marking sign required by CS 25.811(d)(2) must have red letters on an illuminated white background or a universal symbol, of adequate size (See AMC 25.812(b)(1)). These signs must be internally electrically illuminated with ~~a background~~ the brighter area having a brightness of at least 86 candela/m^2 (25 foot lamberts) and a high-to-low ~~background~~ contrast within the white background of a letter-based sign or green area of a universal symbol no greater than 3:1. These signs must also have a contrast between the brightest and darkest elements of at least 10:1.

(…)

(e) Floor proximity emergency escape path marking must provide emergency evacuation guidance for passengers when all sources of illumination more than 1.2m (4ft) above the cabin aisle floor are totally obscured. In the dark of the night, the floor proximity emergency escape path marking must enable each passenger to: ~~–~~

(1) (…); ~~and~~

(2) (…) (See AMC 25.812(e)(2)).~~;~~ and

(3) In the case of passengers seated in seats authorised for occupancy during taxiing, take-off, and landing, in a compartment that does not incorporate any part of the main cabin aisle, in lieu of CS 25.812(e)(1), egress this compartment and enter the main cabin aisle using only markings and visual features not more than 1.2 m (4 ft) above the cabin floor, and proceed to the exits using the marking system necessary to complete the actions as described in CS 25.812(e)(1) and (e)(2) above.

(…)

(l) The emergency lighting system must be designed so that after any single transverse vertical separation of the fuselage during crash landing: ~~–~~

(1) ~~Not more than 25% of all~~ The percentage of electrically illuminated emergency lights required by this paragraph which are rendered inoperative, in addition to the lights that are directly damaged by the separation~~; ~~,does not exceed the values set in the following table (See AMC 25.812(l)(1)):

Maximum approved seating capacity of the type-certified aeroplane as indicated in the aeroplane's type certificate data sheet (TCDS)	Percentage
More than 19	25 %
10 to 19	33.33 % (i.e. one third)
Less than 10	50 %

(…)

Amend CS 25.813 as follows:

CS 25.813 Emergency exit access and ease of operation

(…)

(c) (…)

(4) (…)

(i) For aeroplanes that have a passenger seating configuration of 20 or more, the projected opening of the exit provided may not be obstructed and there must be no interference in opening the exit by seats, berths, or other protrusions (including adjacent seats adjusted to their most adverse positions) for a distance from that exit not less than the width of the narrowest passenger seat installed on the aeroplane or 40 cm (15.75 inches), whichever is the least.

(…)

(e) No door may be installed between any passenger seat that is occupiable for take-off and landing and any passenger emergency exit, such that the door crosses any egress path (including aisles, cross-aisles and passageways). (See AMC 25.813(e))

(…)

Amend CS 25.854 as follows:

CS 25.854 Lavatory fire protection

(See AMC 25.854)

For aeroplanes with a passenger capacity of 20 or more, or with a cabin length of 18.29 m (60 ft) or more~~—~~ :

(…)

SUBPART F — EQUIPMENT

Amend CS 25.1309 as follows:

CS 25.1309 Equipment, systems and installations

(See AMC 25.1309)

The requirements of this paragraph, except as identified below, are applicable, in addition to specific design requirements of CS-25, to any equipment or system as installed in the aeroplane. Although this paragraph does not apply to the performance and flight characteristic requirements of Subpart B and the structural requirements of Subparts C and D, it does apply to any system on which compliance with any of those requirements is dependent. Certain single failures or jams covered by CS 25.671(c)(1) and CS 25.671(c)(3) are excepted from the requirements of CS 25.1309(b)(1)(ii). Certain single failures covered by CS 25.735(b) are excepted from the requirements of CS 25.1309(b). The failure conditions ~~effects~~ covered by CS 25.810~~(a)(1)(v)~~ and CS 25.812 are excepted from the requirements of CS 25.1309(b). The requirements of CS 25.1309(b) apply to powerplant installations as specified in CS 25.901(c).

(…)

Amend CS 25.1365 as follows:

CS 25.1365 Electrical appliances, motors and transformers

(…)

(b) The installation of galleys and cooking appliances must be such as to minimise the risk of overheat, ~~or~~ fire, burns, or spilled liquids to the aeroplane, passengers, and crew (See AMC 25.1365(b)).

(…)

Amend CS 25.1447 as follows:

CS 25.1447 Equipment standards for oxygen ~~–~~ dispensing units

(…)

(c) (…)

(3) There must be ~~at least two~~ sufficient outlets and units of dispensing equipment of a type similar to that required by sub-paragraph (c)(1) of this paragraph in all

other ~~compartments or work~~ areas that may be occupied by passengers or crew members during flight~~, i.e. toilets, washrooms, galley work areas, etc~~. (See AMC 25.1447 (c)(3))

(…)

APPENDICES

Amend Appendix H as follows:

Appendix H

Instructions for Continued Airworthiness
(See AMC to Appendix H)

H25.1 General

(a) This Appendix specifies requirements for the preparation of Instructions for Continued Airworthiness as required by CS 25.1529 and CS 25.1729.

(…)

(c) The applicant must consider the effect of ageing structures in the Instructions for Continued Airworthiness (see AMC 20-20).

(…)

H25.4 Airworthiness Limitations S~~s~~ection

(a) The Instructions for Continued Airworthiness must contain a section titled Airworthiness Limitations that is segregated and clearly distinguishable from the rest of the document. This section must set forth:

(1) Each mandatory modification time, replacement time, structural inspection interval, and related structural inspection procedure approved under CS 25.571~~.~~; and
(…)

(4) A limit of validity (LOV) of the engineering data that supports the structural maintenance programme, stated as a total number of accumulated flight cycles or flight hours or both, approved under CS 25.571. Until the full-scale fatigue testing is completed and the LOV is approved, the Airworthiness Limitations Section must specify an interim limitation restricting aircraft operation to not more than half the number of the cycles accumulated on the fatigue test article.

(b) If the Instructions for Continued Airworthiness consist of multiple documents, the section required by this paragraph must be included in the principal manual. This section must contain a legible statement in a prominent location that reads: 'The Airworthiness Limitations Section is approved and variations must also be approved'.

Create a new Appendix S as follows:

Appendix S

Airworthiness requirements for non-commercially operated aeroplanes and low-occupancy aeroplanes

(See AMC to Appendix S)

S25.1 General

(a) *Applicability*: unless otherwise specified within, the requirements of this Appendix are applicable to the passenger or crew compartments (interiors) of:

(1) non-commercially operated aeroplanes with a passenger seating configuration of:

(i) up to and including 19 passengers; or

(ii) up to and including one half of the maximum passenger seating capacity of the type-certified aeroplane as indicated in the aeroplane type certificate data sheet (TCDS), provided that:

(A) the total number of passengers approved for occupancy during taxiing, take-off or landing does not exceed 150 per deck; and

(B) the total number of passengers approved for occupancy during taxiing, take-off or landing on a deck does also not exceed one half of the maximum passenger seating capacity for that deck as indicated in the aeroplane TCDS.

(2) low-occupancy aeroplanes irrespective of the type of operations (commercial or non-commercial). A low-occupancy aeroplane is defined as an aeroplane which has a passenger seating configuration of:

(i) up to and including 19; or

(ii) up to and including one third of the maximum passenger seating capacity of the type-certified aeroplane as indicated in the aeroplane TCDS, provided that:

(A) the total number of passenger seats approved for occupancy during taxiing, take-off, or landing does not exceed 100 per deck; and

(B) the total number of passenger seats approved for occupancy during taxiing, take-off, or landing in any individual zone between pairs of emergency exits (or any dead end zone) does also not exceed one-third of the sum of the passenger seat allowances for the emergency exit pairs bounding that zone, using the passenger seat allowance for each emergency exit pair as defined by the applicable certification basis of the aeroplane. For the

purpose of determining compliance with this zonal limitation, in the case of an aeroplane which has deactivated emergency exits, it shall be assumed that all emergency exits are functional.

(b) *Aeroplane Flight Manual (AFM) Limitation:* if compliance with any part of this Appendix limits the aeroplane to non-commercial operations, this limitation must be included in the 'Limitations' Section of the AFM.

S25.10 General Cabin Arrangement

(a) *Interior Doors on Non-Commercially Operated Aeroplanes* (See AMC to Appendix S, S25.10(a)): For a non-commercially operated aeroplane, installation of doors that results in non-compliance with CS 25.813(e) is acceptable provided that it is ensured by design and procedure that:

(1) each door is open before entering any of the taxiing, take-off, and landing phases;

(2) each door remains open during taxiing, take-off, and landing, and especially during and after a crash landing; and

(3) in the case of any probable failure or jamming of a door in a position other than fully open, any occupant is able, from any compartment separated by that door, to restore in an easy and simple manner a sufficient opening to access the compartment on the other side of the door.

(b) *Interior Doors on Commercially Operated Aeroplanes* (See AMC to Appendix S, S25.10(b)): For a low-occupancy aeroplane having a passenger seating configuration of 19 or less, installation of doors that results in non-compliance with CS 25.813(e) is acceptable provided that the conditions of S25.10(a)(1), S25.10(a)(2) and S25.10(a)(3) are complied with and the following additional requirements are met for each passenger compartment created by a door or doors:

(1) Within the compartment, there is at least one emergency exit above the waterline on each side of the fuselage that meets at least the requirements of a type IV emergency exit for a compartment that has a passenger seating configuration of nine seats or less, or of a type III emergency exit otherwise; or

(2) Within the compartment, there is at least one emergency exit above the waterline on one side of the fuselage that meets at least the requirements of a type IV emergency exit for a compartment that has a passenger seating configuration of nine seats or less, or of a type III emergency exit otherwise, and:

(i) an occupant of the compartment would not need to go through more than one door to access an emergency exit above the waterline on the other side of the fuselage; and

(ii) the demonstration of compliance with the provisions S25.10(a)(1) and (2) does not rely on any passenger action, nor involve any flight crew member leaving their position in the cockpit.

(c) *Isolated Compartments*: each cabin compartment isolated from the rest of the cabin such that a fire starting in the compartment would not be directly and quickly detected by the occupants of another compartment, in an aeroplane that has a passenger seating configuration of 20 or more, or which has a cabin length of more than 18.29 m (60 ft), must be equipped with a smoke/fire detection system or equivalent which allows detection within one minute after the start of a fire and provides a visual indication in the cockpit, or a visual indication or audible warning in the passenger cabin that would be readily detected by a cabin crew member. However, if it can be demonstrated that a fire would be directly and quickly detected because the compartment is likely to be occupied for the majority of the flight time, such a system is not required (See AMC to Appendix S, S25.10(c)).

(d) *Deactivation of existing Emergency Exits*: Deactivation of one of more emergency exits that results in non-compliance with CS 25.807(e) is acceptable, provided that compliance with the following requirements is shown (See AMC to Appendix S, S25.10(d) and (e)):

(1) the number of passenger seats allowed in a zone between two remaining adjacent pairs of emergency exits is limited to one half of the combined rated capacity of the two pairs of emergency exits (rounded to the nearest whole number);

(2) the number of passenger seats allowed in a zone with only one remaining pair of emergency exits at one end (a so called dead end zone) is limited to one half of the rated capacity of the pair of emergency exits (rounded to the nearest whole number); and

(3) the distance from each passenger seat to at least one remaining emergency exit, on each side of the fuselage, remains compatible with easy egress from the aeroplane.

(e) *Distance between Emergency Exits*: deactivation of emergency exits which results in non-compliance with CS 25.807(f)(4) is acceptable on non-commercially operated aeroplanes only, provided that:

(1) compliance with S25.10(d) is shown; and

(2) a distance of more than 18.29 m (60 ft) between adjacent remaining emergency exits is created only once per side of the fuselage on each deck (See AMC to Appendix S, S25.10(d) and (e)).

S25.20 Emergency Evacuation

(a) *Flammability Requirements*

(1) Mattresses of permanent bed installations that are located in compartments isolated from the main passenger cabin by doors or equivalent means that would normally be closed during taxiing, take-off,

and landing do not need to meet the 'Oil Burner Test' requirement of Appendix F, Part II as required by CS 25.853(c) (See AMC to Appendix S, S25.20(a)(1)).

(2) On non-commercially operated aeroplanes only, compliance with CS 25.853(d) does not need to be demonstrated if it can be shown by test or a combination of test and analysis under the conditions specified in Appendix J that the maximum time for evacuation of all occupants does not exceed 45 sec.

(b) *Access to Type III and IV Emergency Exits*: low-occupancy aeroplanes that have a passenger seating configuration of 19 or less and non-commercially operated aeroplanes may have an item deployable into the region defined by CS 25.813 (c)(4)(i) or CS 25.813 (c)(1), (2) or (3) which creates an obstruction and, therefore, leads to non-compliance with one or more of the aforementioned requirements, provided that:

(1) it is ensured that the item will be safely stowed before entering any of the taxiing, take-off, approach, and landing phases, by means of a position monitoring and alerting system that, in a timely manner, notifies the flight crew and compels the passengers to stow the item if it is in a position that creates an obstruction (See AMC to Appendix S, S25.20(b)(1)). It must be substantiated that, with the item in its most adverse position(s), the remaining exit is at least as effective as a Type IV emergency exit, unless it can be shown that following any single failure, an exit at least as effective as a Type IV emergency exit can be obtained by simple and obvious means; or

(2) the approved passenger configuration is such that this number of passengers can be evacuated through the exit in question, with the obstruction in its most adverse position and under the conditions of Appendix J, at least as quickly as the maximum number of passengers allowed by CS 25.807(g) without the obstruction. It must be substantiated that, with the obstruction in place, the remaining exit is at least as effective as a Type IV emergency exit; or

(3) for aeroplanes required to have at least one cabin crew member on board, the item is intended for use only by a cabin crew member that has direct view of the deployable item and can confirm that it is correctly stowed and secured, while they are seated during taxiing, take-off, and landing.

S25.30 Circulation Inside Cabin During Flight

(a) *Width of Aisle*: for low-occupancy aeroplanes that have a passenger seating configuration of 19 or less, and for non-commercially operated aeroplanes, the design must be such that the dimensional requirements of CS 25.815 can be achieved during all flight phases, except that the width of aisle may be reduced to 0 m during in-flight operations provided that compliance with the following additional requirements is shown (See AMC to Appendix S, S25.30(a)):

(1) all areas of the cabin must be easily accessible by passengers or crew in the event of an emergency situation (e.g. in-flight fire, depressurisation);

(2) placard instructions for restoring the aisle to the taxiing, take-off, and landing configuration must be provided at the locations where the width of the cabin aisle is reduced; and

(3) procedures must be established and documented in the AFM for restoring the aisle width for taxiing, take-off, and landing.

(b) *Firm Handholds*: in lieu of the requirements of CS 25.785(j), if the seat backs do not provide a firm handhold, there must be an acceptable means to enable persons to steady themselves while using the aisles in moderately rough air (See AMC to Appendix S, S25.30(b)).

S25.40 Markings and Placards

(a) '*No Smoking' Placards and Lavatory Ashtrays*: if smoking is to be prohibited, in lieu of the requirements of CS 25.791(a) and CS 25.791(d), a reduced number of 'No smoking' placards may be provided and lavatory ashtrays do not need to be provided in accordance with the following:

(1) a 'No smoking' placard must be conspicuously located inside the passenger compartment in the immediate vicinity of each door that can be used as a passenger boarding door. Each placard must be clearly legible for passengers entering the aeroplane;

(2) compliance with CS 25.853(g) is not required; and

(3) the indication that smoking is prohibited must be the subject of a passenger briefing, and the requirement for this briefing must be part of the AFM.

(b) *Briefing Card Placard*: for non-commercially operated aeroplanes, the instructions required by CS 25.1541 for properly setting the cabin in its configuration approved for taxiing, take-off, and landing may alternatively be provided by a reduced number of placards, each one referring to a briefing card. In that case (See AMC to Appendix S, S25.40(b)):

(1) the detailed minimum instructions to be included in the briefing card must be part of the type design and referred to in the 'Limitations' section of the AFM; and

(2) the briefing card must be easily accessible from each passenger seat. A dedicated stowage must be provided to stow the briefing card within easy reach of each seated passenger with their seat belts fastened.

(c) *Seats in Excess* (See AMC to Appendix S, S25.40(c))

(1) If the total number of seats that are approved for occupancy during taxiing, take-off, and landing is greater than the approved passenger seating configuration, the difference between these two quantities is deemed to be seats in excess. If seats in excess exist, a placard indicating

the approved passenger seating configuration must be installed adjacent to each door that can be used as a passenger boarding door. This placard must be clearly legible for passengers entering the aeroplane. Additionally, a note must be included in the 'Limitations' section of the AFM stating that there are excess seats installed, and indicating the maximum number of passengers that may be transported.

(2) For each seating location available for in-flight use only (including in-flight-only seats, beds, berths, and divans), a placard indicating that the location is not to be occupied during taxiing, take-off, and landing must be installed such that the placard is legible to the seated occupant.

S25.50 Cabin Attendant Direct View

In lieu of the requirements of CS 25.785(h)(2), compliance with the following cabin attendant direct view requirements may be shown:

(a) For non-commercially operated aeroplanes, at least half of the installed cabin crew member seats must face the passenger cabin.

(b) For low-occupancy aeroplanes, cabin crew member seats must be, to the extent possible, without compromising proximity to a required floor level emergency exit, located to provide direct view of the cabin area for which the cabin crew member is responsible (See AMC to Appendix S, S25.50(b)).

BOOK 2

AMC — SUBPART B

Amend AMC 25.201(d) as follows:

AMC 25.201(d)

Stall Demonstration

(…)

2 Unless the design of the automatic flight control system of the aeroplane protects against such an event, the stalling characteristics and adequacy of stall warning, when the aeroplane is stalled under the control of the automatic flight control system, should be investigated. (See also CS 25.1329~~(f)~~(g).)

AMC — SUBPART C

Replace AMC 25.571(a), (b) and (e), and AMC 25.571(b) and (e) by a new AMC 25.571 as follows:

AMC 25.571

Damage tolerance and fatigue evaluation of structure

1. PURPOSE

This AMC provides guidance for compliance with the provisions of CS 25.571 pertaining to the damage tolerance and fatigue evaluation requirements for aeroplane metallic and non-metallic structure. It also provides rational guidelines for the evaluation of scatter factors for the determination of life limits for parts categorised as safe-life. Additional guidance material for certification of non-metallic structures that must also comply with CS 25.571 is contained in AMC 20-29.

2. (RESERVED)

3. REFERENCES

CS 25.571 Damage tolerance and fatigue evaluation of structure,

CS 25.1529 Instructions for Continued Airworthiness,

AMC 20-20 Continued Structural Integrity Programme,

AMC 20-29 Composite Structure.

4. DEFINITIONS OF TERMS USED IN THIS AMC

'Damage tolerance' is the attribute of the structure that permits it to retain its required residual strength without detrimental structural deformation for a period of use after the structure has sustained a given level of fatigue, environmental, accidental, or discrete source damage.

'Fatigue critical structure (FCS)' is structure that is susceptible to fatigue cracking that could lead to a catastrophic failure of an aircraft.

'Safe-life' of a structure is that number of events such as flights, landings, or flight hours, during which there is a low probability that the strength will degrade below its design ultimate value due to fatigue cracking.

'Design service goal (DSG)' is the period of time (in flight cycles or flight hours, or both) established at design and/or certification during which the aircraft structure is reasonably free from significant cracking.

'Principal structure element (PSE)' is an element that contributes significantly to the carrying of flight, ground, or pressurisation loads, and whose integrity is essential in maintaining the overall structural integrity of the aeroplane.

'Detail design point (DDP)' is an area of structure that contributes to the susceptibility of the structure to fatigue cracking or degradation such that the structure cannot maintain its load carrying capability, which could lead to a catastrophic failure.

In 'single load path structure' the applied loads are carried through a single structural member, the failure of which would result in the loss of the structural capability to carry the applied loads.

In 'multiple load path structure' the applied loads are distributed through redundant structural members so that the failure of a single structural member does not result in the loss of structural capability to carry the applied loads.

'Widespread fatigue damage (WFD)' in a structure is characterised by the simultaneous presence of cracks at multiple structural details that are of sufficient size and density whereby the structure will no longer meet the residual strength requirement of CS 25.571(b).

(1) 'Multiple site damage (MSD)' is a source of widespread fatigue damage characterised by the simultaneous presence of fatigue cracks in the same structural element.

(2) 'Multiple element damage (MED)' is a source of widespread fatigue damage characterised by the simultaneous presence of fatigue cracks in adjacent structural elements.

(3) 'Structural modification point (SMP)' is the point in time when a structural area must be modified to preclude WFD.

(4) 'Inspection start point (ISP)' is the point in time when special inspections of the fleet are initiated due to a specific probability of having an MSD/MED condition.

'Scatter factor' is a life reduction factor used in the interpretation of fatigue analysis and fatigue test results.

'Limit of validity' (LOV) of the engineering data that supports the structural maintenance programme is not more than the period of time, stated as a number of total accumulated flight cycles or flight hours or both, during which it is demonstrated by test evidence, analysis and, if available, service experience and teardown inspection results of high-time aeroplanes, that widespread fatigue damage will not occur in the aeroplane structure

'Normal maintenance' is understood to be those scheduled maintenance checks during minor or base maintenance inputs requiring general visual inspections and is normally associated with a zonal programme. The zonal programme is a collective term comprising selected general visual inspections and visual checks that are applied to each zone, defined by access and area, to check system and power plant installations and structure for security and general condition. A general visual inspection is a visual examination of an interior or exterior area, installation, or assembly to detect obvious damage, failure, or irregularity. This level of inspection is made from within touching distance unless otherwise specified. A mirror may be necessary to enhance visual access to all exposed surfaces in the inspection area. This level of inspection is made under normally available lighting conditions such as daylight, hangar lighting, flashlight, or droplight and may require removal or opening of access panels or doors. Stands, ladders, or platforms may be required to gain access.

'Teardown inspection' is the process of disassembling structure and using destructive inspection techniques or visual (magnified glass and dye penetrant) or other, and non-destructive inspection methods (eddy current, ultrasonic) to identify the extent of damage, within a structure, caused by fatigue, environmental and accidental damage.

'Fail-safe' is the attribute of the structure that permits it to retain its required residual strength for a period of unrepaired use after the failure or partial failure of a principal structural element.

'$WFD_{(average\ behaviour)}$' is the point in time when, without intervention, 50 % of the fleet is expected to develop WFD for a particular structure.

'Level 1 corrosion' is:

damage occurring between successive inspections that is within allowable damage limits; or

damage occurring between successive inspections that does not require structural reinforcement, replacement or new damage tolerance based inspections; or

corrosion occurring between successive inspections that exceeds allowable limits but can be attributed to an event not typical of operator usage of other aircraft in the same fleet; or

light corrosion occurring repeatedly between inspections that eventually requires structural reinforcement, replacement, or new damage-tolerance-based inspections.

5. BACKGROUND

(a) Since the early 1970s, there have been significant state-of-the-art and industry-practice developments in the area of structural fatigue and fail-safe strength evaluation of transport category aeroplanes. Recognising that these developments could warrant some revision of the existing fatigue requirements of § 25.571 and 25.573 of 14 CFR Part 25, the Federal Aviation Administration (FAA), on 18 November 1976 (41 FR 50956), gave notice of the Transport Category Aeroplane Fatigue Regulatory Review Programme and invited interested persons to submit proposals to amend those requirements. The proposals and related discussions formed the basis for the revision of the structural fatigue evaluation standards of § 25.571 and § 25.573 of 14 CFR Part 25 and the development of guidance material. To that end, § 25.571 was revised, § 25.573 was deleted (the scope of § 25.571 was expanded to cover the substance of the deleted section), and guidance material (FAA AC 25.571-1) was provided which contained compliance provisions related to the proposed changes.

(b) Since the issuance of FAA AC 25.571-1 on 28 September 1978, additional guidance material, including information regarding discrete source damage, was developed and incorporated in revision 1A on 5 March 1986. The AC was further revised on 18.2.1997 (revision 1B) to add guidance on the elements to be considered in developing safe-life scatter factors for certification. Although FAR, JAR, and CS 25.571 have, since 1978, required consideration of fatigue damage originating at multiple sites, the FAA AC was further revised on 29 April 1998 (revision 1C) to add guidance material whose objective was to preclude widespread fatigue damage (resulting from MSD or MED) from occurring within the design service goal of the aeroplane, and to aid in the determination of thresholds for fatigue inspection and/or other special fleet actions. JAR/CS 25.571 were not harmonised with the 1998 amendment of 14 CFR 25.571. Under the auspices of the Aviation Rulemaking Advisory Committee (ARAC), the General Structure Harmonization Working Group (GSHWG) drafted NPA 25C-292 proposing the Limit of Validity (LOV), greater emphasis on testing, corrosion and manufacturing, and accidental damage in the 25.571 requirements and corresponding AC material to support this. EASA AMC 20-20 'Continuing Structural Integrity Programme' introduced the LOV-concept in 2007. AC 25.571-1D, issued on 13 January 2011, provides guidance in support of 14 CFR 25 Amdt 132 which introduced the LOV requirement. Thus, AMC 25.571 has been revised to provide guidance for establishing an LOV for the structural maintenance programme as will now be required by CS 25.571. In conclusion, this AMC revision based on the GSHWG work and recently developed FAA guidance, now better harmonises with the EASA guidance, AC 25.571-1D, and industry practice.

6. INTRODUCTION

(a) General

The content of this AMC is considered by EASA in determining compliance with the requirements of CS 25.571. The objective is to prevent catastrophic structural failures caused by fatigue damage (FD) (including e.g. widespread fatigue damage (WFD)), environmental deterioration (ED) (e.g. corrosion damage), or accidental damage (AD).

Compliance involves good design practice to ensure that damage tolerance can be achieved and the establishment of maintenance actions developed in compliance with CS 25.1529. Taken together, they result in a structure where the combination of design characteristics and maintenance actions will serve to preclude any failure due to FD, ED, or AD.

CS 25.571(a)(3) requires the applicant to establish inspections or other procedures (herein also referred to as maintenance actions) as necessary to avoid catastrophic failure during the operational life of the aeroplane based on the results of the prescribed fatigue and damage tolerance evaluations.

CS 25.571(a)(5) requires development of inspections for ED and AD. CS 25.571(b) requires the applicant to establish an LOV. Furthermore, CS 25.571(b) and (c) require establishment of inspections and replacement times respectively based on the damage tolerance and fatigue characteristics of the structure. The LOV is, in effect, the operational life of the aeroplane consistent with the evaluations accomplished and maintenance actions established to prevent WFD. The LOV is established based on WFD

considerations and it is intended that all maintenance actions required to address fatigue damage, environmental deterioration (e.g. corrosion damage for metallics, moisture for composites), and accidental damage (e.g. impact, lightning), up to the LOV, are identified in the structural maintenance programme. All inspections and other procedures (e.g. modification times, replacement times) that are necessary to prevent a catastrophic failure due to fatigue, up to the LOV, must be included in the Airworthiness Limitations Section (ALS) of the Instructions for Continued Airworthiness (ICA), as required by CS 25.1529, along with the LOV.

CS 25.571(d) requires the structure to be designed such that sonic fatigue cracking is not probable or, if it arises, it will not result in a catastrophic failure. CS 25.571(e) requires the structure to be designed to withstand damage caused by specified threats such that the flight during which the damage is sustained can be completed.

(1) *CS 25.571(a)(5) — Environmental and accidental damage inspections and associated procedures*

Inspections for ED and AD must be defined. Special consideration should be given to those areas where past service experience indicates a particular susceptibility to attack by the environment or vulnerability to impact and/or abuse. It is intended that these inspections will be effective in discovering ED or AD before it interacts with fatigue related phenomena, and that the ED or AD will, therefore, be removed/repaired before it presents a significant risk. Typically these inspections are largely defined based on past service experience using a qualitative or quantitative process in combination with the Airline Transportation Association (ATA) Maintenance Steering Group (MSG)-3 process. For new structure and materials, testing may be required to evaluate likely AD and the subsequent tolerance of the design to it. For ED prevention, an effective CPCP is necessary, which will contain tasks and procedures in addition to inspections that will help prevent initiation and, when necessary, the recurrence of corrosion (see AMC 20-20). Furthermore, CS 25.571 requires that the ALS must include a statement that requires the operator to include a CPCP in their maintenance programme that will control the corrosion to Level 1 or better.

Any special inspections required for AD and ED, i.e. ones in addition to those that would be generated through the use of the MSG-3 process for AD and ED, or the baseline CPCP development, and which are necessary to prevent catastrophic failure of the aeroplane, must be included in the ALS of the ICA required by CS 25.1529. If a location is prone to accidental or environmental damage and the only means for detection is one that relies on the subsequent development of a fatigue crack from the original damage, then that inspection must be placed in the ALS of the ICA.

Note: The AD and ED inspection programme including the baseline CPCP are equally applicable to structures showing compliance with CS 25.571(b) and (c) respectively.

(2) *CS 25.571(b) and (c) — Fatigue damage inspections or replacement times*

Inspections for fatigue damage or replacement times must be established as necessary. These actions must be based on quantitative evaluations of the fatigue characteristics of the structure. In general, analysis and testing will be required to generate the information needed. The applicant should perform crack growth and residual strength testing to produce the design data needed to support crack growth and residual strength analyses. Full-scale fatigue test evidence is required to support the evaluation of structure that is susceptible to WFD. Test evidence is needed to support analysis used to establish safe-life replacement times.

(i) Inspection or replacement

Compliance with CS 25.571(b) is required unless it can be demonstrated to the satisfaction of the authority that compliance cannot be shown due to practical constraints. Under these circumstances, compliance with CS 25.571(c) is required. The only common example of structure where compliance with the requirements of CS 25.571(c), in lieu of CS 25.571(b), might be accepted, would be the landing gear and its local attachments.

(ii) ALS of the ICA

All inspections and replacement times necessary to detect or preclude fatigue cracking scenarios, before they become critical, must be included in the ALS of the ICA required by CS 25.1529.

(iii) Limit of Validity (LOV)

An LOV for the structural maintenance programme must also be determined and included in the ALS of the ICA. See section 11 of this AMC for additional guidance on the LOV.

(b) Typical loading spectrum expected in service

The loading spectrum should be based on measured statistical data of the type derived from government and industry load history studies, and where insufficient data are available on a conservative estimate of the anticipated use of the aeroplane. The development of the loading spectrum includes the definition of the expected flight plan, which involves ground manoeuvres, climb, cruise, descent, flight times, operating speeds, weights and altitudes, and the approximate time to be spent in each of the operating regimes. The principal loads that should be considered in establishing a loading spectrum are flight loads (gust and manoeuvre), ground loads (taxiing, landing impact, turning, engine run-up, braking, thrust reversing and towing), and pressurisation loads. Operations for crew training and other pertinent factors, such as the dynamic stress characteristics of any flexible structure excited by turbulence or buffeting, should also be considered. For pressurised cabins, the loading spectrum should include the repeated application of the normal operating differential pressure and the superimposed effects of flight loads and aerodynamic pressures.

(c) Areas to be evaluated

When assessing the possibility of serious fatigue failures, the design should be examined to determine probable points of failure in service. In this examination consideration should be given, as necessary, to the results of stress analyses, static tests, fatigue tests,

strain gauge surveys, tests of similar structural configurations, and service experience. Service experience has shown that special attention should be focused on the design details of important discontinuities, main attach fittings, tension joints, splices, and cut-outs such as windows, doors, and other openings. Locations prone to accidental damage (such as that due to the impact with ground servicing equipment near aeroplane doors) or to corrosion should be identified for analysis.

(d) Analyses and tests

Fatigue and damage tolerance analyses should be conducted unless it is determined that the normal operating stresses are of such a low order that crack initiation and, where applicable, significant damage growth is extremely improbable. Any method used in the analyses should be supported by test or service experience. Typical (average) values of fatigue respectively fracture mechanics material properties may be used in fatigue analysis respectively residual strength and crack growth analyses. The effects of environment on these properties should be accounted for if significant.

Generally, testing will also be necessary to support compliance with CS 25.571(b) or (c). The nature and extent of testing of complete structures or portions will depend on applicable previous design and structural tests and service experience with similar structures. Structural areas such as attachment fittings, major joints, changes in section, cut-outs, and discontinuities almost always require some level of testing in addition to analysis. When less than the complete structure is tested, care should be taken to ensure that the internal loads and boundary conditions are valid. When tests are conducted to support the identification of areas susceptible to fatigue, the duration of the test should take into account factors such as material and loading spectrum variability, together with the expected operational life. Refer to Appendix 2 for specific guidance regarding testing required to establish the LOV.

(e) Discrete source damage

It must be shown that the aeroplane is capable of successfully completing a flight during which specified incidents occur and result in immediately obvious damage. The maximum extent of the damage must be quantified and the structure must be shown to be capable of sustaining the maximum load (considered as ultimate) expected during the completion of the flight. There are no maintenance actions that result from this evaluation.

7. DAMAGE TOLERANCE EVALUATION

(a) General

The damage tolerance requirements of CS 25.571(b) are intended to ensure that, should fatigue, corrosion, or accidental damage occur within the LOV, the structure will be capable of withstanding the loading conditions specified in CS 25.571(b)(1) through (b)(6) without failure or detrimental structural deformation until the damage is detected. The evaluation should include identifying the PSEs, defining the loading conditions and conducting sufficiently representative structural tests or analyses, or both, to provide sufficient data for the establishment of the inspection programme. Although this process applies to either single or multiple load path structure, the use of

multiple load path structures should be given priority in achieving a damage-tolerant design. The principle analytical tool used for metallic materials to perform a damage tolerance evaluation is based on fracture mechanics. A discussion of this approach is presented in Appendix 1 of this guidance material. The means of establishing the LOV and maintenance actions specifically associated to WFD is addressed in detail in Section 11 of this AMC.

(b) Damage-tolerant characteristics

A damage-tolerant structure has two notable attributes:

(1) The structure can tolerate a significant amount of damage, due to fatigue, environmental or accidental deterioration without compromising the continued airworthiness of the aeroplane (residual strength and rigidity).

(2) The structure can sustain that damage long enough to be found and repaired during scheduled or unscheduled maintenance (inspectability).

(c) Design considerations

To achieve a damage-tolerant structure, criteria should be established to guide the design process so that this design objective is achieved. The design process should include a damage tolerance evaluation (test and analysis) to demonstrate that the damage-tolerant design objectives are achieved, and to identify inspections or other procedures necessary to prevent catastrophic failure. Reliance on special inspections should be minimised by designing structure with easily detectable (e.g. visual) cracking modes. Since the occurrence of WFD can complicate a damage-tolerant evaluation to the point that reliable inspections programmes cannot be developed even with extremely intensive inspection methods, it must be demonstrated, with sufficient full-scale fatigue test evidence, that adequate maintenance procedures are contained in the ALS of the ICA, such that WFD will not occur within the LOV. A discussion on several issues that an applicant might face in demonstrating freedom from WFD is contained in Appendix 2 to this AMC.

(d) Design features

Design features which should be considered in attaining a damage-tolerant structure include the following:

(1) multiple load path construction and/or the use of damage containment features to arrest fast fracture or reduce the crack growth rate, and to provide adequate residual strength;

(2) materials and stress levels that provide a slow rate of crack propagation combined with high residual strength; and

(3) arrangement of design details to ensure a sufficiently high probability that a failure in any critical structural element will be detected before the strength has been reduced below the level necessary to withstand the loading conditions specified in CS 25.571(b).

(e) Probabilistic evaluations

No guidance is provided in this AMC on probabilistic evaluation. Normally, damage tolerance assessments consist of a deterministic evaluation of design features described in paragraphs 7(d)(1), (2) and (3). Paragraphs (f) to (i) below provide guidelines for this approach.

(f) PSEs, detail design points, and locations to be evaluated

In accordance with CS 25.571(a), a damage tolerance and fatigue evaluation should be conducted for each part of the structure which could contribute to a catastrophic failure. PSEs such as wing, empennage, control surfaces and their systems, the fuselage, engine mountings, landing gears, and their related primary attachments, and all DDPs susceptible to fatigue that could contribute to a catastrophic failure should be evaluated.

In accordance with CS 25.571(a)(1)(ii), this evaluation must include the identification of PSEs and DDPs, the failure of which could contribute to catastrophic failure of the aeroplane. As defined in this AMC, a principal structural element is an element of structure that contributes significantly to the carrying of flight, ground, or pressurisation loads and whose integrity is essential in maintaining the overall structural integrity of the aeroplane. When identifying PSEs, consideration should be given to the effect caused by partial or complete loss or failure of structure with respect to continued safe flight and landing, considering all flight phases including stability, control and aeroelasticity.

A DDP is an area at higher risk of fatigue cracking than other areas, and may warrant specific actions such as special inspections or other procedures to ensure continued airworthiness.

(1) Locations requiring evaluation can be determined by analysis or by fatigue tests on complete structures or subcomponents. However, tests may be necessary when the basis for analytical prediction is not reliable, such as for complex components. If less than the complete structure is tested, care should be taken to ensure that the internal loads and boundary conditions are valid.

The selection criteria for DDPs should also include the following considerations:

(a) any evidence of cracking encountered in service on a comparable structure;

(b) any evidence of cracking found during fatigue testing on a comparable structure;

(c) available strain gauge data;

(d) locations where permanent deformation occurred on static test articles;

(e) areas analytically shown to have a relatively low crack initiation life;

(f) susceptibility to corrosion or other environmental deterioration (e.g. disbonding);

(g) potential for manufacturing anomalies (e.g. new or novel manufacturing processes where the potential for damage may not be well understood);

(h) vulnerability to in-service induced accidental damage;

(i) areas whose failure would create high stresses in the remaining structure;

(j) elements in high tension or shear;

(k) low static margin;

(l) high stress concentrations;

(m) high load transfer;

(n) materials with high crack growth rates;

(o) some DDPs may exist outside of PSEs and may also have been classified as fatigue critical structure, e.g. undercarriage door attachments (see Appendix 5 for discussion on PSEs, FCS and DDP);

(p) areas where detection of damage would be difficult;

(q) locations subject to vibrations or other mechanisms that may lead to premature wear fastener holes; and

(r) locations vulnerable to moisture ingress or retention.

(2) Examples of principal structural elements (PSEs)

Typical examples of structure which are usually considered to be PSEs are:

(i) Wing and empennage

(a) control surfaces, slats, flaps, and their mechanical systems and attachments (hinges, tracks, and fittings);

(b) primary fittings;

(c) principal splices;

(d) skin or reinforcement around cut-outs or discontinuities;

(e) skin-stringer combinations or integrally stiffened plates;

(f) spar caps;

(g) spar webs; and

(h) ribs and bulkheads.

(ii) Fuselage

(a) circumferential frames and adjacent skin;

(b) pilot window posts;

(c) pressure bulkheads;

(d) skin and any single frame or stiffener element around a cut-out;

(e) skin or skin splices, or both, under circumferential loads;

(f) skin or skin splices, or both, under fore and aft loads;

(g) skin and stiffener combinations under fore and aft loads;

(h) door skins, frames, stops and latches;

(i) window frames; and

(j) floor beams[1].

[1] Floor beams are not always critical but should be checked for criticality, particularly those located next to cut-outs or within non-circular pressurised sections.

(iii) Landing gear and their attachments

(iv) Engine mounts and struts

(v) Thrust reverser components, whose failure could result in inadvertent deployment

(3) Extent of Damage.

Each particular design should be assessed to establish appropriate damage criteria in relation to inspectability and damage-extension characteristics. In any damage determination, including those involving multiple cracks, it is possible to establish the extent of damage in terms of detectability with the inspection techniques to be used, the associated initially detectable crack size, the residual strength capabilities of the structure, and the likely damage-extension rate considering the expected stress redistribution under the repeated loads expected in service and with the expected inspection frequency. Thus, an obvious partial failure could be considered to be the extent of the damage or residual strength assessment, provided a positive determination is made that the fatigue cracks will be detectable by the available inspection techniques at a sufficiently early stage of the crack development. The following are typical examples of partial failures which should be considered in the evaluation:

(i) Detectable skin cracks emanating from the edge of structural openings or cutouts;

(ii) A detectable circumferential or longitudinal skin crack in the basic fuselage structure;

(iii) Complete severance of interior frame elements or stiffeners in addition to a detectable crack in the adjacent skin;

(iv) A detectable failure of one element where dual construction is utilised in components such as spar caps, window posts, window or door frames, and skin structure;

(v) The presence of a detectable fatigue failure in at least the tension portion of the spar web or similar element; and

(vi) The detectable failure of a primary attachment, including a control surface hinge and fitting.

(g) Inaccessible areas

Every reasonable effort should be made to ensure inspectability (reference CS 25.611) of all structural parts. In those cases where inaccessible and uninspectable blind areas exist, the damage tolerance evaluation should allow for extension of damage into detectable areas or demonstrate sufficient residual strength up to the LOV without inspection.

(h) Residual strength testing of principal structural elements

Analytical prediction of the residual strength of structures can be very complex due to non-linear behaviour, load redistribution and the potential for a multiplicity of failure modes. The nature and extent of residual strength tests will depend on previous

experience with similar structures. Simulated cracks should be as representative as possible of actual fatigue damage. Where it is not practical to produce actual fatigue cracks, damage can be simulated by cuts made with a fine saw, sharp blade, guillotine, or other suitable means. Whatever artificial means are used to simulate sharp fatigue cracks, sufficient evidence should be available from tests to indicate equivalent residual strength. If equivalency cannot be shown, every attempt should be made to apply enough cyclic loading to generate fatigue cracks from the artificial damage prior to applying residual strength loads. Special consideration should be given to the procedure for pre-cracking so that subsequent test results are representative. This can be an issue when slow stable tearing in ductile sheet or plate material is part of the failure mechanism. Inappropriate pre-cracking loads can lead to non-conservative results. In those cases where bolt failure, or its equivalent, is to be simulated as part of a possible damage configuration in joints or fittings, bolts can be removed to provide that part of the simulation.

(i) Damage tolerance analysis and tests

(1) It should be determined by analysis, supported by test evidence, that:

(i) the structure, with the extent of damage established for residual strength evaluation, can withstand the specified residual strength loads (considered as ultimate loads); and

(ii) the crack growth life under the repeated loads expected in service (between the time the damage becomes initially detectable and the time the extent of damage reaches the value for residual strength evaluation) provides a practical basis for development of the inspection programme and procedures described in Section 8 of this AMC.

(2) The repeated loads should be as defined in the loading, temperature, and humidity spectra. The loading conditions should take into account the effects of structural flexibility and rate of loading where they are significant.

(3) The damage tolerance characteristics can be shown analytically by reliable or conservative methods such as the following:

(i) By demonstrating quantitative relationships with structure already verified as damage-tolerant; or

(ii) By demonstrating that the repeated loads and residual strength load stresses do not exceed those of previously verified designs of similar configuration, materials, and inspectability.

8. INSPECTION REQUIREMENTS

(a) Damage detection

Detection and repair of damage before it becomes critical is the most important factor in ensuring that the damage tolerance characteristics of the structure are maintained. For this reason, CS 25.571 requires that the applicant establish inspections or other procedures, as necessary, to prevent catastrophic failure from accidental, environmental, or fatigue damage, and include those inspections and procedures in the

ALS of the Instructions for Continued Airworthiness required by CS 25.1529 (see also Appendix H to Part-25).

Due to the complex interactions of the many parameters that affect the damage tolerance evaluation, such as operating practices, environmental effects, load sequence effects on crack growth, and variations in inspection methods, operational experience should be taken into account in establishing inspection thresholds, repeat intervals, and inspection procedures.

(b) Environmental and accidental damage inspection programmes

The inspections developed under CS 25.571(b) are primarily for the detection of cracks developing from fatigue, accidental damage, and corrosion. As required by CS 25.571(a)(5), a separate programme needs to be implemented for the early detection of environmental and accidental damage. This is intended to minimise the risk of:

(1) interaction between corrosion and fatigue cracking;

(2) accidental damage developing into fatigue cracks; or

(3) corrosion developing due to accidental damage.

In many cases this can be accomplished through the Maintenance Review Board (MRB) activity or equivalent process agreed by EASA, for a new large aeroplane model using MSG-3 procedures. These procedures also require that a CPCP be developed.

For ED and AD programmes developed under the auspices of the MRB, the minimum ALS content associated with AD and ED may generally be limited to:

— a reference to the documents that contain the MRB report (MRBR) derived maintenance tasks for AD and ED; and

— the need to incorporate and maintain an effective CPCP in the operators' programme; and

— a statement requiring operators to control corrosion to Level 1 or better.

It is also important to explain to operators the link between the AD and ED inspection programmes and CS 25.571 and CS 25.1529 compliance.

Inspections that are designed to detect fatigue cracking resulting from AD or ED, where the originating damage cannot otherwise be demonstrated to be detected prior to the development of the fatigue cracks, must also be directly included in the ALS. For new structure where there is limited supporting data from service experience, the MRB will depend heavily on input from the analyses and test programmes conducted by the applicant during certification, and for this reason significant cooperation is required between those involved directly in certification and those participating in the MRBR development. Care should also be taken to ensure that the damage assumptions made remain conservative after entry into service. A check of the continued validity of the certification assumptions can be achieved through fleet leader programmes and robust reporting requirements. If there is any doubt about the likely performance of a completely new structure with respect to AD and ED, certain specific inspections in vulnerable areas may be better placed in the ALS.

The baseline CPCP may be established through the MRB Industry Steering Committee (ISC) using existing procedures for MRBR development or developed by the applicant and submitted directly to EASA. (Note: Provided the operator has an NAA-approved maintenance programme that controls corrosion to Level 1 or better, it does not need to follow exactly the baseline CPCP offered by the type certificate holder (TCH). However, all revisions to the TCH's programme for ED and AD must be considered by the operator for incorporation in the operators MP under the Part-M requirements.)

Reporting requirements for these programmes should extend to overhaul procedures where the condition of the part should be assessed and reported if outside of approved limits, whether or not it is to remain on the component being overhauled.

Changes and supplemental type certificates (STC) must also be provided with inspection programmes that address ED and AD.

(c) Inspection threshold for fatigue cracking

The inspection threshold is the point in time at which the first planned structural inspection is performed following entry into service. The threshold may be as low as the repeat interval, or may allow for a longer period of operation, provided certain conditions are met.

The concept of delaying an inspection threshold beyond the repeat interval is based on the premise that it will take a certain amount of time before fatigue cracks would develop to a size that would be detectable during a structural inspection. Consequently, it may be acceptable to wait some period of time before starting to inspect for fatigue cracks.

CS 25.571(a)(4) requires inspection thresholds for certain structure to be derived assuming that the structure contains an initial flaw of the maximum probable size that could exist as a result of manufacturing processes or manufacturing or service-induced damage. For metallic structure this would typically be achieved using crack growth analysis supported by tests. This approach applies to:

(1) single load path structure, and

(2) multiple load path 'fail-safe' structure and crack arrest 'fail-safe' structure, where it cannot be demonstrated that the resulting load path failure or partial failure (including arrested cracks) will be detected and repaired during normal maintenance, inspection, or operation of an aeroplane prior to failure of the remaining structure.

In this context, normal maintenance includes general visual structural inspections for accidental and environmental damage derived from processes such as the MRB application of MSG-3. Inspections should begin early enough to ensure that there is a high confidence of detecting cracks before they could lead to a catastrophic structural failure.

For the locations addressed by CS 25.571(a)(4) that are also susceptible to accidental (manufacturing or service induced) damage, the assumed initial flaw size for crack growth determination of the threshold should not be less than that which can be supported by service experience or test evidence. For example, if the type of damage

expected is well defined, e.g. it is limited to dents, then there may be data that supports a longer threshold than would be derived by the assumption of a crack that is similar in size to the dent. However, in this case, the worst case manufacturing flaw should still be considered as a crack and the most conservative resulting threshold adopted. If supporting data is not available (e.g. for a completely new design where no specific investigation of the accidental damage threats or their influence on fatigue has been made), then the fatigue cracking inspection threshold should be set equal to the repeat interval derived for a crack detectable by general visual inspection means, since the initial damage and its growth is not well defined and could occur at any time.

The remaining areas of the structure evaluated under CS 25.571(b), i.e. multiple load path 'fail-safe' structure and crack arrest 'fail-safe' structure, where it can be demonstrated that the resulting load path failure, partial failure, or crack arrest will be detected and repaired during normal maintenance, inspection, or operation of an aeroplane prior to failure of the remaining structure must also have thresholds established for fatigue cracking. For these locations, methods that do not account for worst-case damage may be used in lieu of crack growth analysis if desired. For example, fatigue SN analysis and tests with an appropriate scatter factor or slow crack growth analysis based on appropriate initial manufacturing damage, i.e. typical manufacturing flaws as opposed to the maximum probable flaw (e.g. a 0.127 mm corner crack representing a typical manufacturing flaw in a fastener hole versus a 1.27 mm crack representing the maximum probable flaw).

The means of establishing the LOV and maintenance actions (including inspections) specifically associated to WFD is addressed in detail in Section 11 of this AMC.

All inspections necessary to detect fatigue cracking that have thresholds less than the approved operating limitation (LOV or interim limitations) of the maintenance programme must be included in the ALS.

Appendix 3 provides further details on threshold determination.

(d) Inspection

The basis for setting inspection intervals is the period of time during which damage is detectable and the residual strength remains above the required levels. The reliability of the repeat inspection programme (i.e. frequency of inspections and probability of detection) should assure damage detection before the residual strength of the aircraft is compromised. Inspection intervals must be established by applying appropriate reduction factors to this period to ensure that the crack or other damage or failed load path will typically be found well before the residual strength of the structure drops below the required level. Long periods of exposure to residual strength levels only just above the load limit should be avoided. This applies in particular to crack-arrest structure. It should be borne in mind that CS 25.305 is the principle requirement for strength of the airframe, and that CS 25.571 is primarily intended to provide an inspection programme that will ensure the timely detection and repair of damage in order to restore the aircraft to the required (CS 25.305) strength capability and preserve this capability throughout the majority of the aircraft's operational life.

Detectable crack sizes and shapes assumed to determine inspection intervals should be consistent with the inspection method capabilities and the cracking characteristics of the structure being evaluated. If concurrent cracking in adjacent areas or surrounding structure is expected within the operational life of the aeroplane, then this should be accounted for in the cracking scenario assumed.

9. FATIGUE (SAFE-LIFE) EVALUATION

9.1. Reserved

9.2. Fatigue (safe-life) evaluation

9.2.1. General

The evaluation of structure under the following fatigue (safe-life) strength evaluation methods is intended to ensure that catastrophic fatigue failure, as a result of the repeated loads of variable magnitude expected in service, will be avoided throughout the structure's operational life. Under these methods the fatigue life of the structure should be determined. The evaluation should include the following:

(a) estimating or measuring the expected loading spectra of the structure;

(b) conducting a structural analysis, including consideration of the stress concentration effects;

(c) performing fatigue testing of structure which cannot be related to a test background to establish response to the typical loading spectrum expected in service;

(d) determining reliable replacement times by interpreting the loading history, variable load analyses, fatigue test data, service experience, and fatigue analysis;

(e) evaluating the possibility of fatigue initiation from sources such as corrosion, stress corrosion, disbonding, accidental damage, and manufacturing defects based on a review of the design, quality control, and past service experience; and

(f) providing necessary maintenance instructions including replacement times in the ICA in accordance with CS 25.1529.

9.2.2. Scatter factor for safe-life determination

In the interpretation of fatigue analyses and test data the effect of variability should, under CS 25.571(c), be accounted for by an appropriate scatter factor. In this process it is appropriate that the applicant justifies the scatter factor chosen for any safe-life part. The following guidance is provided (see Figure 1):

(a) The base scatter factors (BSF) applicable to test results are: BSF1 = 3.0, and BSF2 = (see paragraph 9.2.2(e) of this AMC). If the applicant can meet the requirements of 9.2.2(c) of this AMC, he/she may use BSF1 or, at his/her option, BSF2.

(b) The base scatter factor, BSF1, is associated with test results of one representative test specimen.

(c) Justification for use of BSF1. BSF1 may only be used if the following criteria are met:

(i) Understanding of load paths and failure modes

Service and test experience of similar in-service components that were designed using similar design criteria and methods should demonstrate that the load paths and potential failure modes of the components are well understood.

(ii) Control of design, material, and manufacturing process quality

The applicant should demonstrate that his/her quality system (e.g. design, process control, and material standards) ensures the scatter in fatigue properties is controlled, and that the design of the fatigue-critical areas of the part account for the material scatter.

(iii) Representativeness of the test specimen

(A) The test article should be full scale (component or subcomponent) and represent that portion of the production aircraft requiring test. All differences between the test article and the production article should be accounted for either by analysis supported by test evidence, or by testing itself.

(B) Construction details, such as bracket attachments, clips, etc., should be accounted for, even though the items themselves may be non-loadbearing.

(C) Points of load application and reaction should accurately reflect those of the aircraft, ensure correct behaviour of the test article, and guard against uncharacteristic failures.

(D) Systems used to protect the structure against environmental degradation can have a negative effect on fatigue life, and therefore, should be included as part of the test article.

(d) Adjustments to base scatter factor BSF1. Having satisfied the criteria of paragraph 9.2.2(c), justifying the use of BSF1, the base value of 3.0 should be adjusted to account for the following considerations, as necessary, where not wholly taken into account by design analysis. As a result of the adjustments, the final scatter factor may be less than, equal to, or greater than 3.0.

(i) *Material fatigue scatter.* Material properties should be investigated up to a 99 % probability of survival and a 95 % level of confidence.

(ii) *Spectrum severity.* Test load spectrum should be derived based on a spectrum sensitive analysis accounting for variations in both utilisation (i.e. aircraft weight, cg, etc.) and occurrences/size of loads. The test load spectrum applied to the structure should be demonstrated to be conservative when compared to the expected usage in-service.

(iii) *Number of representative test specimens.* Well established statistical methods should be used that associate the number of items tested with the distribution chosen to obtain an adjustment to the base scatter factor.

(e) If the applicant cannot satisfy the intent of all of paragraph 9.2.2(c) of this AMC, BSF2 should be used.

(i) The applicant should propose scatter factor BSF2 based on careful consideration of the following issues: the required level of safety, the number of representative test specimens, how representative the test is, expected fatigue scatter, type of

repeated load test, the accuracy of the test loads spectrum, spectrum severity, and the expected service environmental conditions.

(ii) In no case should the value of BSF2 be less than 3.0.

(f) Resolution of test loadings to actual loadings. The applicant may use a number of different approaches to reduce both the number of load cycles and the number of test set-ups required.

These include the following:

— spectrum blocking (i.e., a change in the spectrum load sequence to reduce the total number of test set-ups);

— high-load clipping (i.e., reduction of the highest spectrum loads to a level at which the beneficial effects of compression yield are reduced or eliminated); and

— low-load truncation (i.e., the removal of non-damaging load cycles to simplify the spectrum).

Due to the modifications to the flight-by-flight loading sequence, the applicant should propose either analytical or empirical approaches to quantify an adjustment to the number of test cycles which represents the 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.

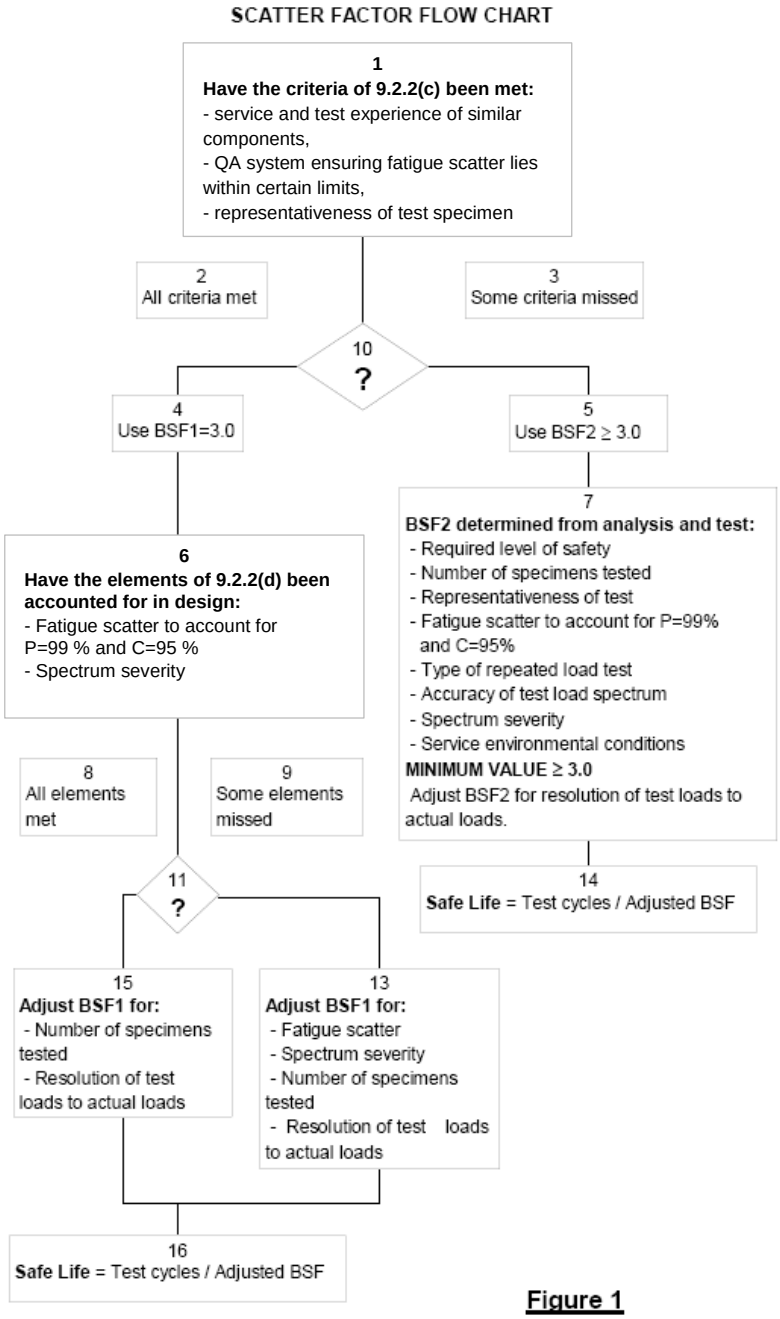

SCATTER FACTOR FLOW CHART
1
Have the criteria of 9.2.2(c) been met:
- service and test experience of similar components,
- QA system ensuring fatigue scatter lies within certain limits,
- representativeness of test specimen
2
All criteria met
3
Some criteria missed
10
?
4
Use BSF1=3.0
5
Use BSF2 ≥ 3.0
6
Have the elements of 9.2.2(d) been accounted for in design:
- Fatigue scatter to account for P=99 % and C=95 %
- Spectrum severity
7
BSF2 determined from analysis and test:
- Required level of safety
- Number of specimens tested
- Representativeness of test
- Fatigue scatter to account for P=99% and C=95%
- Type of repeated load test
- Accuracy of test load spectrum
- Spectrum severity
- Service environmental conditions
MINIMUM VALUE ≥ 3.0
Adjust BSF2 for resolution of test loads to actual loads.
8
All elements met
9
Some elements missed
11
?
14
Safe Life = Test cycles / Adjusted BSF
15
Adjust BSF1 for:
- Number of specimens tested
- Resolution of test loads to actual loads
13
Adjust BSF1 for:
- Fatigue scatter
- Spectrum severity
- Number of specimens tested
- Resolution of test loads to actual loads
16
Safe Life = Test cycles / Adjusted BSF

Figure 1

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.

Maintenance actions

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.

Design changes

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.

Appendix 1 — Crack growth analysis and tests

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 ΔK_{eff}) 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 be expected. The primary issue for these tests is the pre-cracking required to achieve a well-behaved fatigue crack before data is collected. Effective pre-cracking procedures (e.g. 'load shedding') have been established and are described in the public domain. Care must be taken to ensure that subsequent crack growth is not affected by the prior pre-cracking.

In order to minimise the test time for actual structural components and/or full-scale test articles, the test loading spectrum may be modified by eliminating small magnitude load events or by replacing them with a fewer number of larger load events that give equivalent crack growth.

Crack-growth behaviour may be obtained from actual structural components and/or full-scale test articles. However, inducing active fatigue cracks of the desired initial size and at the desired locations can be extremely difficult. Past success in obtaining useful data has been achieved on an opportunistic basis when natural fatigue cracks have developed in the course of normal cyclic testing. Naturally occurring and artificially induced fatigue cracks may be monitored and data collected for at least a portion of the overall crack-growth period to be used for setting inspection requirements. This data can be extremely useful in supplementing and validating the analytical predictions, in some cases it may be the sole basis for the establishment of inspection requirements. Where fatigue test crack growth data is used, the results should be corrected to address expected operational conditions.

Appendix 2 — Full-scale fatigue test evidence

(a) *Overview*

CS 25.571(b) requires that special consideration for widespread fatigue damage (WFD) be included where the design is such that this type of damage could occur. This Appendix focuses on the test evidence in support of establishing the LOV and applicants will also need to consider and agree with EASA the extent of testing required in support of compliance with CS 25.571 in general, in particular for validation of hot spots, areas of complex loading exhibiting crack growth, single load path components, and safe-life items. CS 25.571(b) requires the effectiveness of the provisions to preclude the possibility of widespread fatigue damage occurring within the limits of validity of the structural maintenance programme to be demonstrated with sufficient full-scale fatigue test evidence. The determination of what constitutes 'sufficient full-scale test evidence' requires a considerable amount of engineering judgment and is a matter that should be discussed and agreed to between the applicant and EASA early in the planning stage for a certification project. In general, sufficient full-scale test evidence to support an LOV consists of full-scale fatigue testing to at least two times the LOV, followed by specific inspections and analyses to determine that widespread fatigue damage has not occurred. It may be appropriate to allow for three life times of testing, especially if inspection may not be practical for areas subject to WFD and requiring SMPs to be established. The following factors should be considered in determining the sufficiency of evidence:

Factor 1: The comparability of the load spectrum between the test and the projected usage of the aeroplane.

Factor 2: The comparability of the airframe materials, design and build standards between the test article and the certified aeroplane.

Factor 3: The extent of post-test teardown inspection, residual strength testing and analysis for determining if widespread fatigue cracking has occurred.

Factor 4: The duration of the fatigue testing.

Factor 5: The size and complexity of a design or build standard change. This factor applies to design changes made to a model that has already been certified and for which full-scale fatigue test evidence for the original structure should have already been determined to be sufficient. Small, simple design changes, comparable to the original structure, or changes that are derived from the original design using the same basic design configuration and where very similar load paths and similar operating stress levels are retained could be analytically determined to be equivalent to the original structure in their propensity for WFD. In such cases, additional full-scale fatigue test evidence should not be necessary.

Factor 6: In the case of major changes and STCs, the age of an aeroplane being modified. This factor applies to aeroplanes that have already accumulated a portion of their LOV prior to being modified. An applicant should only be required to demonstrate freedom from WFD up to the LOV in place for the original aeroplane.

(b) *Elements of a full-scale fatigue test programme*

The following guidance addresses elements of a test programme that is intended to generate the data necessary to support compliance. It is generally applicable to all certification projects.

(1) *Article.* The test article should be representative of the structure of the aircraft to be certified (i.e. ideally a production standard article). The attributes of the type design that could affect MSD/MED initiation, growth and subsequent residual strength capability should be replicated as closely as possible on the test article. Critical attributes include, but are not limited to, the following:

— material types and forms;

— dimensions;

— joining methods and details;

— coating and plating;

— use of faying surface sealant;

— assembly processes and sequences; and

— influence of secondary structure (e.g. loads induced due to proximity to the structure under evaluation).

(2) *Test set-up and loading.* The test set-up and loading should result in a realistic simulation of expected operational loads.

(i) *Test set-up*. The test set-up dictates how loads are introduced into the structure and reacted. Every effort should be made to introduce and react loads as realistically as possible. When a compromise is made (e.g. wing air loading), the resulting internal loads should be evaluated (e.g. using finite element methods) to ensure that the structure is not being unrealistically underloaded or overloaded locally or globally.

(ii) *Test loading*. The test loading spectrum should include loads from all damaging sources (e.g. cabin pressurisation, manoeuvers, gusts, engine thrust, control surface deflection, and landing impact) that are significant for the structure being evaluated. Supporting rationale should be provided when a source is not represented in a sequence. Additionally, differences between the test sequence and expected operational sequence should be justified. For example, it is standard practice to eliminate low loads that are considered to be non-damaging and clip high infrequent loads that may non-conservatively bias the outcome, but care should be taken in both cases so that the test results are representative. Paragraph 9.2.2(f) provides some guidance on justifying the test loading sequence.

(3) *Test duration*. AMC 20-20 includes guidance on how to establish mandatory maintenance actions for WFD-susceptible structure needed to preclude WFD occurrence in that structure. For any WFD-susceptible area the average time in flight cycles and/or hours to develop WFD must first be determined. This is referred to as the $WFD_{\text{average behaviour}}$ for the subject area. The AMC 20-20 guidance states that the area should be modified/replaced at one third of this time unless inspection for MSD/MED is practical. If inspection is practical the guidance states that inspection should start at one third of the $WFD_{\text{average behaviour}}$ with modification/replacement at one half of that time. It is standard practice to interpret the non-factored fatigue life of one specimen as the average life. It follows that if a full-scale fatigue test article survives a test duration of X

without WFD occurrence, it can be conservatively assumed that the WFD $_{\text{average behaviour}}$ of all susceptible areas is equal to X. Based on this, and assuming that the susceptible areas are impractical to inspect for MSD/MED, the guidance of AMC 20-20 would require that replacement/modification would have to be implemented at X/3. For areas where MSD/MED inspections were practical replacement/modification could be deferred until X/2, but MSD/MED inspections would have to start at X/3. The preceding should be kept in mind when deciding what the test duration will be.

(4) *Post-test evaluation*. One of the primary objectives of the full-scale fatigue test is to generate data needed to determine the absolute WFD $_{\text{average behaviour}}$ for each susceptible area, or to establish a lower bound. Recall that the definition of WFD $_{\text{average behaviour}}$ is the average time required for MSD/MED to initiate and grow to the point that the static strength capability of the structure is reduced below the residual strength requirements of CS 25.571(b). Some work is required at the end of the test to determine the strength capability of the structure either directly or indirectly.

(i) *Residual strength tests*. One acceptable way to demonstrate freedom from WFD at the end of a full-scale fatigue test is to subject the article to the required residual strength loads specified in CS 25.571(b). If the test article sustains the loads it can be concluded that the point of WFD has yet to be reached for any areas. However, because fatigue cracks that might exist at the end of the test are not quantified it is not possible to determine how far beyond the test duration WFD would occur in any of the susceptible areas without accomplishing additional work (e.g. teardown inspection). Additionally, metallic test-articles may be non-conservatively compromised relative to their future fatigue performance if static loads in excess of representative operational loads are applied. Residual strength testing could preclude the possibility of using an article for additional fatigue testing.

(ii) *Teardown inspections*. The residual strength capability may be evaluated indirectly by performing teardown inspections to quantify the size of any MSD/MED cracks that might be present or to establish an upper bound on crack size based on inspection method capability. Once this is done the residual strength capability can be estimated analytically. Depending on the results crack-growth analyses may also be required to project backwards or forwards in time to estimate the WFD $_{\text{average behaviour}}$ for an area. As a minimum, teardown inspection methods should be capable of detecting the minimum size of MSD or MED cracking that would result in a WFD condition (i.e. residual strength degraded below the level specified in CS 25.571(b)). Ideally it is recommended that inspection methods be used that are capable of detecting MSD/MED cracking before it degrades strength below the required level. Effective teardown inspections required to demonstrate freedom from WFD typically require significant resources. They typically require disassembly (e.g. fastener removal) and destruction of the test article. All areas that are or may be susceptible to WFD should be identified and examined.

(c) *Examples of fatigue test evidence for various types of certification projects.*

The following examples offer some guidance on the types of data sets that might constitute 'sufficient evidence' for some kinds of certification projects. The scope of the test specimen and the duration of the test are considered.

(1) *New type certificates.* Normally this type of project would necessitate its own full-scale fatigue test of the complete airframe to represent the new structure and its loading environment. Nevertheless, prior to full-scale fatigue test evidence from earlier tests performed by the applicant, or others, may also be used and could supplement additional tests on the new model. Ultimately, the evidence needs to be sufficient to conclude with confidence that, within the LOV of the airframe, widespread fatigue damage will not occur. Factors 1 through 4 should be considered in determining the sufficiency of the evidence.

A test duration of a minimum of twice the LOV for the aeroplane model would normally be necessary if the loading spectrum is realistic, the design and construction for the test article principal structure is the same as for the certified aeroplane, and the post-test teardown is exhaustive. If the conformance to Factors 1 through 3 is less than ideal, a significantly longer test duration would be needed to conclude with confidence that WFD will not occur within the LOV. Moreover, no amount of fatigue testing will suffice if the conformance to Factors 1 through 3 above is not reasonable. Consideration should also be given to the possible future need for life extension or product development, such as potential weight increases, etc.

(2) *Derivative models.* The default position would be to test the entire airframe. However, it may be possible to reliably determine the occurrence of widespread fatigue damage for part or all of the derivative models from the data that the applicant generated or assembled during the original certification project. Nevertheless, the evidence needs to be sufficient to allow confidence in the calculations that show that widespread fatigue damage will not occur within the LOV of the aeroplane. Factors 1 through 5 should be considered in determining the sufficiency of the evidence for derivative models. For example, a change in the structural design concept, a change in the aerodynamic contour, or a modification of the structure that has a complex internal load distribution might well make analytical extrapolation from the existing full-scale fatigue test evidence very uncertain. Such changes might well necessitate full-scale fatigue testing of the actual derivative principal structure. On the other hand, a typical derivative often involves extending the fuselage by inserting 'fuselage plugs' that consist of a copy of the typical semi-monocoque construction for that model with slightly modified material gauges. Normally this type of project would not necessitate its own full-scale fatigue test, particularly if very similar load paths and operating stress levels are retained.

(3) *Type design changes — Service bulletins.* Normally this type of project would not necessitate the default option of a full-scale fatigue test because the applicant would have generated, or assembled, sufficient full-scale fatigue test evidence during the original certification project that could be applied to the change. Nevertheless, as cited in the previous example, the evidence needs to be sufficient to allow confidence in the calculations that show that widespread fatigue damage will not occur within the LOV of the aeroplane. In addition, Factor 5 'The size and complexity of a design change' should be considered. Therefore, unless otherwise justified, based on existing test data or a demonstration that the design change is not susceptible to WFD, the applicant should perform full-scale tests for the types of design changes listed in Appendix 4.

(4) *Supplemental type certificates (STCs)*

Unless otherwise justified according to the guidance below or based on existing test data or a demonstration that the design change is not susceptible to WFD, the applicant for an STC should perform full-scale tests for the types of design changes listed in Appendix 4.

(i) Sufficient full-scale test evidence for structure certified under an STC may necessitate additional full-scale fatigue testing, although the extent of the design change may be small enough to use Factor 5 to establish the sufficiency of the existing full-scale fatigue test evidence. The applicant for an STC may not have access to the original equipment manufacturer's full-scale fatigue test data. For aircraft types where an LOV has been published, the STC applicants may assume that the basic structure is free from WFD up to the LOV, unless:

- EASA has issued an airworthiness directive (AD), or intends to take such action (proposed AD), to alleviate a WFD condition; or
- inspections or modifications exist in the ALS relating to WFD conditions.

For the purpose of the STC applicant's demonstration, it may be assumed that the aeroplane to which the LOV is applicable has received at least two full LOV of fatigue testing under realistic loads, and has received a thorough post-test inspection that either did not detect any WFD or the ALS includes from the outset details of modifications required to address WFD that will need specific consideration by the STC applicant. With this knowledge, and considering the Factors 1 through 5, the STC applicant may be able to demonstrate that WFD will not occur on its modification (or the underlying original structure) within the LOV or a suitably revised value. If, however, the modification significantly affects the distribution of stress in the underlying structure, or significantly alters loads in other parts of the aeroplane, or significantly alters the intended mission for the aeroplane, or, if the modification is significantly different in structural concept from the certified aeroplane being modified, additional representative fatigue test evidence would be necessary.

(ii) In addition, Factor 6 'The age of the aeroplane being modified' could be considered for modifications made to older aeroplanes. The STC applicant should demonstrate freedom from WFD up to the LOV of the aeroplane being modified. For example, an applicant for an STC to an aeroplane that has reached an age equivalent to 75 % of its LOV should demonstrate that the modified aeroplane will be free from WFD for at least the remaining 25 % of the LOV. Although an applicant could attempt to demonstrate freedom from WFD for a longer period, this may not be possible unless the original equipment manufacturer cooperates by providing data for the basic structure. A short design service goal for the modification could simplify the demonstration of freedom from WFD for the STC applicant.

(5) *Repairs*. New repairs that differ from the repairs contained in the original equipment manufacturer's structural repair manual, but that are equivalent in design to such repairs, and that meet CS-25 in other respects, would not necessitate full-scale fatigue testing to support freedom from WFD up to the LOV. Concerning major repair solutions (that may be susceptible to WFD) which utilise design concepts that are different from previous approved repair data (e.g. new materials, other production processes, new design details), further testing may be required.

(d) *Use of existing full-scale fatigue test data*

In some cases, especially for derivative models and type design changes accomplished by the type certificate holder, there may be existing full-scale fatigue test data that may be used to support compliance and mitigate the need to perform additional testing.

Any physical differences between the structure originally tested and the structure being considered that could affect its fatigue behaviour must be identified and reconciled. Differences that should be addressed include, but are not limited to, differences in any of the physical attributes listed under section (b)(1) of this Appendix and differences in operational loading. Typical developments that affect the applicability of the original LOV demonstration data are:

(1) gross weight (e.g. increases);

(2) cabin pressurisation (e.g. change in maximum cabin or operating altitude); and

(3) flight segment parameters.

The older the test data, the harder it may be to demonstrate that it is sufficient. Often test articles were not conformed, nor were test plans or reports submitted to EASA as part of the compliance data package. Loading sequence rigor varied significantly over the years and from applicant to applicant. Additionally, testing philosophies and protocols were not standardised. For example, post-test evaluations, if any, varied significantly and in some cases consisted of nothing more than limited visual inspections. However, there may be acceptable data from early full-scale fatigue tests that the applicant proposes to use to support compliance. In order to use such data the configuration of the test article and loading must be verified and the issue of the residual strength capability of the article (or teardown data) at the end of the test must be addressed.

(e) *Use of in-service data.* There may be in-service data that can be used to support WFD evaluations. Examples of such data are as follows:

— Documented positive findings of MSD/MED cracks that include location, size and the time in service of the affected aircraft along with a credible record of how the aircraft had been operated since original delivery.

— Documented negative findings from in-service inspections for MSD/MED cracks on a statistically significant number of aircraft with the time in service of each aircraft and a credible record of how each aircraft had been operated since original delivery. For this data to be useful, the inspections methods used should have been capable of detecting MSD/MED crack sizes equal to or smaller than those sizes that could reduce the strength of the structure below the residual strength levels specified in CS 25.571(b).

— Documented findings from the destructive teardown inspection of structure from in-service aircraft. This might be structure (e.g. fuselage splices) removed from aircraft that were subsequently returned to service, or from retired aircraft. It would also be necessary to have a credible record of the operational loading experienced by the subject structure up to the time it was taken out of service.

— Prior to using in-service data any physical and usage/loading differences that exist between the structure of the in-service or retired aircraft and the structure being certified should be identified and reconciled as discussed above.

Appendix 3 — Methods for inspection threshold determination

Different approaches have been used to calculate inspection thresholds, although these are essentially variants of one of two methods, being:

(a) the fatigue (stress-life or strain-life) method, which uses fatigue endurance data collected under constant stress or constant strain conditions, and a linear damage accumulation model (Palmgren-Miner rule);

(b) the crack growth method, which uses crack propagation and residual strength data to calculate the growth from an assumed initial crack size to a critical crack length, according to fracture mechanics principles.

CS 25.571(a)(4) requires certain types of structure to have thresholds based upon crack growth analyses or test assuming the maximum probable flaw due to manufacturing or service-induced damage. This approach applies to:

(a) single load path structure; and

(b) multiple load path 'fail-safe' structure and crack arrest 'fail-safe' structure, where it cannot be demonstrated that the resulting load path failure or partial failure (including arrested cracks) will be safely detected and repaired during normal maintenance, inspection, or operation of an aeroplane prior to failure of the remaining structure.

Paragraph 8(c) of this AMC provides further details on identifying this structure.

In lieu of other data, an acceptable threshold for inspection for cracks emanating from the maximum probable manufacturing flaw at a fastener hole may be obtained for aluminium alloy airframe structure if an initial corner crack of radius 0.05'' (1.27 mm) is assumed and the total crack growth life is divided by 2. Whether this approach is also sufficient to conservatively address all probable forms of manufacturing and service-induced damage needs careful consideration and is highly design dependent. Where specific test or service data for service damage exists that can be used to reliably establish an appropriate threshold for all likely types of service damage then crack growth analysis may only need to consider the manufacturing flaw.

For structure susceptible to WFD specific methods for setting inspection thresholds are applicable when agreed to be practical; see Section 11 and Appendix 2 to this AMC.

Regardless of the approach used, the calculated thresholds should be supported with appropriate fatigue test evidence. The best sources of fatigue test evidence are from service experience and large component or full-scale fatigue tests. Large component and full-scale fatigue test specimens are generally constructed using the same manufacturing processes as on the actual aircraft. The results of such tests should provide sufficient information to reliably establish the typical manufacturing quality and possibly its lower bound, especially when those results are combined with service experience. Conversely, simple test specimens used to generate fatigue endurance and crack growth data, which are typically assembled under laboratory or workshop conditions, may not be representative of the actual range of manufacturing quality in the structure under consideration. Therefore, in the absence of information from the full-scale fatigue tests and service experience, consideration should be given to generating fatigue endurance and crack growth data on simple test specimens which include artificial damages that are introduced at the beginning of the test, and are representative of the lower bound of manufacturing quality.

Appendix 4 — Examples of changes that may require full-scale fatigue testing

The following are examples of types of modifications that may require full-scale fatigue testing:

(1) passenger-to-freighter conversions (including addition of cargo doors);

(2) gross weight increases (e.g. increased operating weights, increased zero-fuel weights, increased landing weights, and increased maximum take-off weights);

(3) installation of fuselage cut-outs (e.g. passenger entry doors, emergency exit doors or crew escape hatches, fuselage access doors, and cabin window relocations);

(4) complete re-engine or pylon change;

(5) engine hush kits;

(6) wing modifications (e.g. installation of winglets, changes in flight-control settings such as flap droop, and change of wing trailing-edge structure);

(7) modified or replaced skin splice;

(8) any modification that affects three or more stiffening members (e.g. wing stringers and fuselage frames);

(9) a modification that results in operational-mission change, which significantly changes the original equipment manufacturer's load/stress spectrum (e.g. extending the flight duration from 2 hours to 10 hours); and

(10) a modification that changes areas of the fuselage from being externally inspectable using visual means to being non-inspectable (e.g. installation of a large, external fuselage doubler that results in hiding details beneath it).

Appendix 5 — PSE, FCS, and WFD-susceptible structure

(a) Overview

Four key terms used when showing compliance to the damage tolerance and fatigue requirements of CS-25 and EASA guidance for the continued structural integrity of ageing aircraft in AMC 20-20 are: 'principle structural element (PSE)', 'fatigue critical structure (FCS)', 'widespread fatigue damage (WFD)-susceptible structure' and 'design detail point (DDP)'.

This Appendix provides clarification on the intended meanings of these terms and how they relate to each other.

(b) Principal structural element (PSE)

(1) The term 'principal structural element (PSE)' is defined in this AMC as follows:

'Principal structural element (PSE)' is an element that contributes significantly to the carrying of flight, ground or pressurisation loads, and whose integrity is essential in maintaining the overall structural integrity of the aeroplane.

(2) While this definition does not specifically address the fatigue susceptibility of the structure, or environmental or accidental damage, it is intended to address the majority of the structure that must be evaluated according to CS 25.571. CS 25.571(a) states the following:

'This evaluation must be conducted for each part of the structure that could contribute to a catastrophic failure'.

(3) Examples of PSEs are found in paragraph 7(f) of this AMC.

(4) The above reinforces the notion that the identification of PSEs should be based solely on the importance of the structure to assure the overall aeroplane integrity.

(5) Paragraph 7(f) of this AMC provides guidance for identifying PSEs. Many manufacturers use this list as a starting point for their list of Fatigue Critical Structure (FCS). CS 25.571(b) is intended to address all structure that could contribute to a catastrophic failure resulting from fatigue, environmental and accidental damage, and, therefore, may include some structure that is not considered FCS. Nevertheless, all PSE should be considered when developing a list of FCS.

(6) The definitions used by applicants to identify PSEs have not been consistent among applicants and, in some cases, among models produced by the same applicant. The lack of standardisation of the usage and understanding of the term 'PSE,' and the resulting diversity that exists between type design PSE lists, led authorities to introduce the new term 'Fatigue Critical Structure (FCS)' in the 'Ageing Aircraft Requirements and Guidance Material'.

(c) Fatigue Critical Structure (FCS)

(1) 'Fatigue critical structure (FCS)' is defined as aircraft structure that is susceptible to fatigue cracking, which could contribute to a catastrophic failure. Fatigue critical structure also includes structure which, if repaired or modified, could be susceptible to fatigue cracking and contribute to a catastrophic failure. Structure is most often susceptible to fatigue cracking when subjected to tension-dominated repeated loads during operation. Such structure may be part of the baseline structure or part of a modification. 'baseline structure' means structure that is designed under the original

type certificate or amended type certificate for that aircraft model (i.e. the as-delivered-aeroplane model configuration).

(2) Fatigue critical structure is generally a subset of principal structural elements, specifically those elements that are susceptible to fatigue damage. The exception may be a DDP that is susceptible to fatigue and, although not part of a PSE, could result in catastrophic failure if it were to fail (e.g. an undercarriage door hinge has been categorised by some TCHs as a DDP and FCS, when its failure would lead to loss of the door and the door could impact the aircraft with catastrophic results. In this case the door was not classified as a PSE because the TCH had not considered the door to contribute significantly to carrying flight, ground or pressurisation loads. Considering further aspects of the PSE definition now adopted, it might be claimed that the door is not essential to maintain the overall integrity of the aircraft, i.e. the aircraft may be safe without it. However, due to the need to identify all detail design points and FCS whose failure could cause catastrophic failure of the aircraft it is in any case subject to the fatigue and damage tolerance requirements.)

(d) Detail design points (DDP)

'Detail design point' is an area of structure that contributes to the susceptibility of the structure to fatigue cracking or degradation such that the structure cannot maintain its load carrying capability, which could lead to a catastrophic failure.

(e) Widespread fatigue damage (WFD)-susceptible structure

(1) 'Widespread fatigue damage (WFD)' is the simultaneous presence of cracks at multiple structural locations, which are of sufficient size and density such that the structure no longer meets the residual strength requirements of CS 25.571(b).

(2) 'Multiple site damage (MSD)' and 'Multiple element damage (MED)' are conditions that, with no intervention, can lead to WFD. The term 'WFD-susceptible structure' refers to areas of structure that, under normal circumstances, could be expected to eventually develop MSD and/or MED cracks, which could lead to WFD.

(3) Although not explicitly stated, structure susceptible to WFD cannot be inspected reliably to preclude WFD. Unless a flight cycles and/or flight hours limit is placed on an aeroplane, modifications may be needed to preclude WFD. Structure susceptible to WFD is a subset of FCS.

AMC — SUBPART D

Create a new AMC 25.603(a) as follows:

AMC 25.603(a)

Large glass items

1. General

This AMC defines acceptable minimum performance standards for large glass items used as an interior material in passenger cabin installations whereby the glass items carry no other loads than those resulting from the mass of the glass itself, rapid depressurisation or abuse loading.

Large glass items should be shown not to be a hazard during events such as an emergency landing and cabin depressurisation.

1.1. A large glass item is defined as:

(a) a glass item with a dimension that exceeds 51 cm (20 in.);

(b) a glass panel with a surface area on one side that exceeds 0.12 m² (200 in.²); or

(c) a glass item with a mass exceeding 4 kg.

In case of multiple items in close proximity, the accumulated surface area of glass as well as the total mass should be considered (i.e. effects such as tiling should be considered).

1.2. A large glass item should meet the following requirements whenever installed in compartments that may be occupied during taxiing, take-off, and landing, or may be traversed during an emergency evacuation:

(a) The glass item should be subjected to, and pass, ball impact testing (see paragraph 2 below).

(b) The glass item should be subjected to, and pass, abuse load testing (see paragraph 3 below).

(c) The glass item should meet the requirements outlined in CS 25.561(b)(3), (c) and (d). A safety factor of 2.0 should be applied to glass items to account for variability in the production of the material and for long-term degradation.

(d) Cracking of glass should not produce a condition where the material may become hazardous to the occupants (e.g. sharp edges, splinters or separated pieces). This requires destructive testing. If any of the test conditions defined below (see paragraphs 2 and 3 below) do not result in a significant failure of the glass item, testing at a higher impact energy (ball impact test) or load (abuse load test) should be performed until destruction, or until an impact energy of 80 J or double the specified abuse load is reached.

Tests should be performed for worst-case conditions (e.g. the largest glass item should be tested with the maximum engraving). Similarity justification may then be used for other items.

These tests do not need to be performed for glass items that have traditionally been installed in large aeroplanes, provided that their installation method, location, etc. are not unusual (e.g. standard lavatory mirrors, light bulbs, light tubes, galley equipment).

The instructions for continued airworthiness should reflect the fastening method used and should ensure the reliability of all methods used (e.g. life limit of adhesives, or scheduled check for security of a clamp connection). For example, inspection methods

and intervals for an adhesive-based design should be defined in accordance with adhesion data from the manufacturer of the adhesive, or actual adhesion test data, as necessary.

2. Ball Impact Tests

The test procedure(s) and pass/fail criteria of the Underwriters Laboratories standard UL 61965, *Mechanical safety for cathode ray tubes*, Edition 2, 27 July 2004, or former UL 1418, *Standard for safety cathode ray tubes*, Edition 5, 31 December 1992, or other equivalent approved method are the basis of the ball impact strength and no-hole tests described in this paragraph, combined with the impact energy in Section 5.12.2 of ANSI/SAE Z26.1, *Safety glazing materials for glazing motor vehicles and motor vehicle equipment operating on land highways — safety standard*, 1 December 1997.

The glass samples should be installed in a test fixture representative of the actual installation in the cabin.

2.1. Strength Test

The large glass item should be subjected to a single impact applied in accordance with the test conditions of paragraph 2.3 below. The impact energy should be 21 J, caused by a 51-mm diameter ball or, alternatively, by a 40-mm diameter ball, as specified in paragraph 2.3.2 below.

The test is passed if the expulsion of glass within a 1-min period after the initial impact satisfies the following criteria:

(a) there is no glass particle (a single piece of glass having a mass greater than 0.025 g) between the 0.90 and 1.50-m barriers (see paragraph 2.3.1) on either side (if appropriate);

(b) the total mass of all pieces of glass between the 0.90 and 1.50-m barriers (see paragraph 2.3.1) does not exceed 0.1 g on either side (if appropriate); and

(c) there is no glass expelled beyond the 1.50-m barrier (see paragraph 2.3.1) on either side (if appropriate).

2.2 No-Hole Test

The large glass item should be subjected to a single impact applied in accordance with the test conditions of paragraph 2.3 below. The impact energy should be 3.5 J, caused by a 51-mm diameter ball as specified in paragraph 2.3.2 below.

The test is passed if the large glass item does not develop any opening that may allow a 3 mm diameter rod to enter.

Note: If the large glass item does not develop any opening that would allow a 3 mm rod to enter when subjected to the strength test defined in paragraph 2.1 above, the no-hole test defined in this paragraph does not need to be performed.

2.3 Test Conditions

2.3.1 Test Apparatus and Setup

The large glass item should be mounted in a way representative of the aeroplane installation.

The centre of the large glass item should be 1.00 ± 0.05 m above the floor.
For the strength test (see paragraph 2.1 above), two barriers, each one made of material 10–20 mm thick, 250 mm high, and 2.00 m long, should be placed on the floor in front of the test item (or on both sides in case of a glass partition) at the specified location, measured horizontally from the front surface of the large glass item to the near surface of the barrier. The barriers may be less than 2.00 m long, provided that they extend to the walls of the test room. A non-skid surface such as a blanket or rug may be placed on the floor.
A solid, smooth, steel ball of the size specified in paragraph 2.3.2 below should be suspended by suitable means such as a fine wire or chain and allowed to fall freely as a pendulum and strike the large glass item with the specified impact energy. The large glass item should be placed in a way that its surface is vertical and in the same vertical plane as the suspension point of the pendulum. A single impact should be applied to any point on the surface of the large glass item at a distance of at least 25 mm from the edge of the surface.

2.3.2 Impact Objects
The 51-mm diameter steel ball, used as an impact object, should have a mass of approximately 0.5 kg and a minimum Scale C Rockwell Hardness of 60.
The 40-mm diameter steel ball, used as an impact object, should have a mass of approximately 0.23 kg and a minimum Scale C Rockwell Hardness of 60.

3. Abuse Loads Tests
The large glass item should withstand the abuse loads defined in paragraph 3.2 below when subjected to the test conditions defined in paragraph 3.1. The panel should remain attached to the fixture, and any failure should be shown to be non-hazardous (e.g. no sharp edges, no separation of pieces).

3.1 Test conditions
Abuse loads should be applied:

(a) at the points that would create the most critical loading conditions; and

(b) at least at the geometrical centre, and at one point located along the perimeter.

For the above-mentioned load applications, it is acceptable to use any loading pad with a shape and dimensions that fit into a 15.24-cm (6-in.) diameter circle.
For all tests, the glass item should be mounted in a test fixture representative of the actual installation in the cabin.

3.2 Loads to be applied
Abuse loads should be considered as ultimate loads, therefore, no additional factors (e.g. fitting factors, casting factors, etc.) need to be applied for abuse load analysis/testing.

Unless it is justified that one or more abuse load cases are not applicable due to the shape/size/location of the glass item making it unlikely or impossible for persons to

apply loads in the direction(s) concerned, the following abuse loads should be considered (see also Figure 1 below):

3.2.1 Pushing loads

Pushing loads are 133 daN (300 lbf) from 0–1.5 m (60 in.) above the floor, reducing linearly to 44 daN (100 lbf) at 2 m (80 in.) above the floor level (see (1) in Figure 1 below).

3.2.2 Pulling loads

One-hand pull loads (where it is not possible to grab with two hands) are 66 daN (150 lbf) from 0–1.5 m (60 in.) above the floor, reducing linearly to 22 daN (50 lbf) at 2 m (80 in.) above the floor level (see (3) in Figure 1 below).
Two-hands pull loads are 133 daN (300 lbf) from 0–1.5 m (60 in.) above the floor, reducing linearly to 44 daN (100 lbf) at 2 m (80 in.) above the floor level (see (1) in Figure 1 below).

3.2.3 Up loads

Up loads are 66 daN (150 lbf) from 0–1.5 m (60 in.) above the floor, reducing linearly to 22 daN (50 lbf) at 2 m (80 in.) above the floor level (see (2) in Figure 1 below).

3.2.4 Downloads

Downloads are 133 daN (300 lbf) from 0-1.5 m (60 in.) above the floor, reducing linearly to 44 daN (100 lbf) at 2 m (80 in.) above the floor level (see (1) in Figure 1 below).

3.2.5 Stepping, Seating loads

In the case of large glass items which may be stepped or sat on, a load of 222 daN (500 lbf) should be used. This load is to be applied at the most critical point, and on any relevant surface up to 1 m (38 in.) above the floor level (see (4) in Figure 1 below).

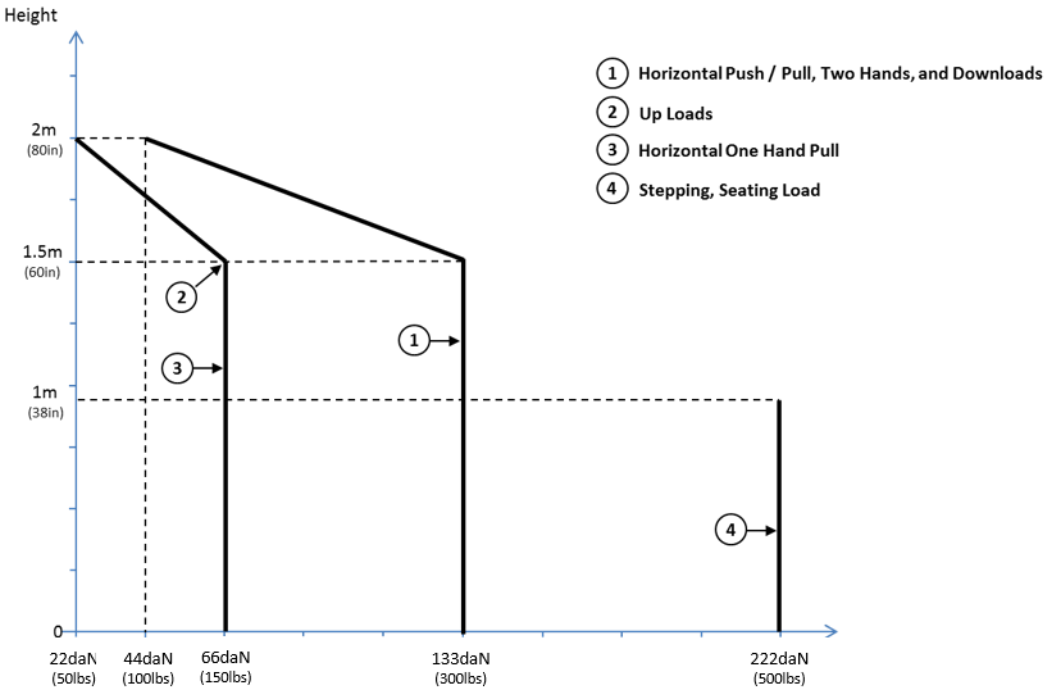


Figure 1

Amend AMC 25.785 as follows:

AMC 25.785

Seats, B~~b~~erths, S~~s~~afety B~~b~~elts, and H~~h~~arnesses

(…)

Beds, berths, or divans convertible into a bed should be equipped with a restraint device (e.g. a belt) for use by the occupant(s) when sleeping. Beds, berths, etc. that may be occupied by more than one occupant may be equipped with a single belt.

Create a new AMC 25.785(h)(2) as follows:

AMC 25.785(h)(2)

Cabin Attendant Direct View

If the total number of passenger seats approved for occupancy during taxiing, take-off, and landing is greater than the approved passenger seating configuration, the demonstration of compliance with the direct-view requirements should consider the most adverse combination of occupied seats, assuming the full passenger load on board.

Amend AMC 25.787(b) as follows:

AMC 25.787(b)

Stowage Compartments

For stowage compartments in the passenger and crew compartments it must be shown by analysis and/or tests that under the load conditions as specified in CS 25.561(b)(3), the retention items such as doors, swivels, latches etc., are still performing their retention function. In the analysis and/or tests the expected wear and deterioration should be taken into account.

Stowage Compartment Latching Mechanisms:

(1) The following areas shall be considered in a special cabin interior for the purpose of designing latching mechanisms:

- Cabin crew member areas:

  Cabin crew member areas are those areas in the passenger cabin where cabin crew members may be seated during taxiing, take-off, and landing (these are typically zones in proximity to floor level emergency exits, although other areas may exist).

  To protect flight attendants from being struck by items dislodged from galley stowage compartments, it is common practice to install additional restraint devices (dual latching) to each stowage compartment located within a longitudinal distance equal to three rows of seats fore and aft of the cabin attendant seats. However, the following additional considerations may be used:

  - A longitudinal distance of 2 metres (6.6 ft) may be used in case the 'three rows' criterion is difficult to assess due to widely spaced seating,
  - Underseat and overhead stowage bins do not need to be considered, and
  - A stowage compartment located in a closed unoccupied area during taxiing, take-off, and landing or behind a partition in the passenger cabin does not need to be considered.

- Passenger Areas:

  Passengers Areas are zones in which passenger seats designed for occupancy during taxiing, take-off, and landing are installed. In such cabin areas, if the means used to prevent the contents of the compartments from becoming a hazard by shifting is a latched door, the design should take into consideration the wear and deterioration expected in service.

- Non TTOL Areas:

  Non-TTOL areas are zones, separated from the remainder of the cabin by means of a door during taxiing, take-off, and landing (TTOL), in which no seat is installed (passenger or crew member) that may be occupied during taxiing, take-off, and landing, and which do not include any part of any possible egress route from the aeroplane (such areas may be for example lavatories, washrooms, bedrooms, closed galleys, etc.).

In such areas, a single latch mechanism for stowage compartments is acceptable, provided that the door separating this area from the rest of the cabin is shown to be capable of staying securely closed under the applicable emergency landing conditions of CS 25.561 with an additional inertia load, uniformly distributed on the door, equating to the highest placarded allowable single compartment contents mass inside that area. Such single latch mechanisms do not need to be designed to account for the wear and deterioration expected in service.

(2) The following is provided as a clarification of the considerations to be followed when designing latching mechanisms, as well as of the means by which wear and deterioration expected in service may be substantiated:

- Single latch:

  A single latch is a latching mechanism capable of retaining a load derived from the specified maximum flight, ground and emergency landing load conditions.

- Dual latch:

  A dual latch is a latching mechanism composed of two independent single latching mechanisms each of which is capable of retaining a load determined by the specified maximum flight, ground and emergency landing load conditions. It is acceptable that a single operating mechanism (e.g. handle) operates with two independent latching mechanisms at the same time.

- Latch fail indication

  Latch fail indication is any means that permits clear visual confirmation that a latch is not properly engaged. In the case of a dual latching system, a single indication may serve for the two latches if it is ensured that the failure of either latch to properly engage will result in latch fail indication. All latches, whether single or dual, should include a latch fail indication.

- Wear and Deterioration

  - Dual latching is a means of compliance to the wear and deterioration requirement. Where dual latches are installed there is no need to further demonstrate wear and tear.

  - Consideration of wear and deterioration for single latches should be substantiated by test evidence, or analysis based on test evidence, showing that latch operation as intended by the design will be maintained following a simulation of full service life, with an appropriate scatter factor. A design life of 20 000 latch cycles may be used except if EASA finds the expected use of the aeroplane justifies more endurance substantiation. Demonstration of a 20 000 cycle design life can be accomplished by submitting the latch to a 100 000 cycle test representative of operational use, and verifying after the test that the latch is still able to operate as intended and is capable of withstanding ultimate load without failure.

(3) The above considerations regarding latching mechanisms, do not apply to compartments not accessible in flight for which a special tool is needed to gain access to (e.g. maintenance panel, access panels, etc.).

Create a new AMC 25.788(a) as follows:

AMC 25.788(a)

Installation of Showers

The following should be considered in the design of a shower installation:

(a) An analysis should be performed to identify possible failures leading to water leakage, and to show that appropriate mitigation features have been included in the design.

(b) The shower cubicle should be considered as a passenger compartment in terms of the need for ventilation. The applicant should justify that adequate ventilation is provided within the shower. The cabin air itself can be considered as a 'fresh air' source for the air supply of the shower.

(c) The shower cubicle air outflow should be directed into aeroplane areas that will not be adversely affected by the high water content of this air flow.

(d) A means to steady oneself could be either (a) firm handhold(s) specifically designed and provided for the purpose or an intrinsic design feature of the cubicle. For instance, if one or more of the cubicle wall-to-wall dimensions does not exceed 1 metre (3.3 feet), it may be assumed that an occupant can steady himself/herself by placing his/her hands on opposite wall surfaces.

(e) If electrical power outlets are installed in the room or area where the shower is present, all the following requirements should be fulfilled:

(i) the shower cubicle should be enclosed up to the ceiling;

(ii) there should be no electrical power outlet inside the shower cubicle; and

(iii) no power outlet should be placed closer than 0,6m from any point on the surface of the closed shower door.

Create a new AMC 25.788(b) as follows:

AMC 25.788(b)

Large Display Panels

1. General

This AMC does not apply to flight deck display panels. A display panel should be considered large if its diagonal is greater than 51 cm (20 in.).

Any large display panel should be shown not to be a hazard during events such as emergency landing and cabin depressurisation. It should meet the following requirements:

(a) the large display panel should withstand the differential pressures caused by a worst-case cabin depressurisation event without having any adverse effect (for instance no substances should be released through cracks or openings, no sharp edges should be created);

(b) the large display panel should be subjected to, and pass, abuse load testing (see paragraph 3 below);

(c) the installation should withstand the inertia loads outlined in CS 25.561(b)(3) without any adverse effect; and

(d) if the large display panel incorporates glass, it should be subjected to, and pass, ball impact testing (see paragraph 2 below).

With the exception of the ball impact testing, large display panels incorporating any glass element should withstand the above-defined loads with no more than minor cracks (i.e. no parts released nor the surface becoming a hazard) and without becoming dislodged from their mounts. Alternatively, the installation may still be found acceptable if some means, such as a protective cover, are provided to shield the passenger cabin from the glass monitor. The installation including its protective cover should meet all the relevant criteria identified in this AMC. Furthermore, the cover should not introduce additional hazardous characteristics of its own and should comply with all pertinent aeroplane certification requirements, e.g. flammability.

Unless it has been shown that the display panel withstands all the mechanical tests in paragraphs 1.(a) to (d) above without any damage that would result in the release of chemical substances into the cabin, documentation should be provided from medical authorities which substantiates that the type and amount of chemical substances released into the cabin in case of failure would not result in adverse health effects on cabin occupants. The specific cabin volume may be considered. Alternatively, it is acceptable to show that each installed glass screen complies with A 4(1) of Directive 2002/95/EC 'on the restriction of the use of certain hazardous substances in electrical and electronic equipment' (RoHS).

2. Ball Impact Testing (only for display panels containing glass)

The test procedure and pass/fail criteria of the Underwriters Laboratories standard UL 61965, *Mechanical safety for cathode ray tubes*, Edition 2, 27 July 2004 or former UL 1418, *Standard for safety cathode ray tubes*, Edition 5, 31 December 1992, or other equivalent approved method, are the basis of the ball impact strength and no-hole tests described in this paragraph.

The large display panel should be installed in a test fixture representative of the actual installation in the cabin.

2.1. Strength Test

The large display panel should be subjected to a single impact applied in accordance with the test conditions of paragraph 2.3 below. The impact energy should be 7 J, caused by a

51-mm diameter ball or, alternatively, 5.5 J, caused by a 40-mm diameter ball, as specified in paragraph 2.3.2 below.

The test is passed if the expulsion of glass within a 1-min period after the initial impact satisfies the following criteria:

(a) there is no glass particle (a single piece of glass having a mass greater than 0.025 g) between the 0.90 and 1.50-m barriers (see paragraph 2.3.1);

(b) the total mass of all pieces of glass between the 0.90 and 1.50-m barriers (see paragraph 2.3.1) does not exceed 0.1 g; and

(c) there is no glass expelled beyond the 1.50-m barrier (see paragraph 2.3.1).

2.2 No-Hole Test

The large display panel should be subjected to a single impact applied in accordance with the test conditions of paragraph 2.3 below. The impact energy should be 3.5 J, caused by a 51-mm diameter ball as specified in P 2.3.2 below.

The test is passed if the large display panel does not develop any opening that may allow a 3-mm diameter rod to enter. Cracking of the panel is permitted.

Note: If the large display panel does not develop any opening that would allow a 3-mm rod to enter when subjected to the strength test defined in paragraph 2.1 above, the no-hole test defined in this paragraph does not need to be performed.

2.3 Test Conditions

2.3.1 Test Apparatus and Setup

The centre of the large glass item should be 1.00 ± 0.05 m above the floor.

For the strength test (see paragraph 2.1 above), two barriers, each one made of material 10–20 mm thick, 250 mm high, and 2.00 m long, should be placed on the floor in front of the test item (or on both sides in case of a glass partition) at the specified location, measured horizontally from the front surface of the large glass item to the near surface of the barrier. The barriers may be less than 2.00 m long, provided that they extend to the walls of the test room. A non-skid surface such as a blanket or rug may be placed on the floor.

A solid, smooth, steel ball of the size specified in paragraph 2.3.2 below should be suspended by suitable means such as a fine wire or chain and allowed to fall freely as a pendulum and strike the large glass item with the specified impact energy. The large glass item should be placed in a way that its surface is vertical and in the same vertical plane as the suspension point of the pendulum. A single impact should be applied to any point on the surface of the large glass item at a distance of at least 25 mm from the edge of the surface.

2.3.2 Impact Objects

The 51-mm diameter steel ball used as an impact object should have a mass of approximately 0.5 kg and a minimum Scale C Rockwell Hardness of 60.

The 40-mm diameter steel ball used as an impact object should have a mass of approximately 0.23 kg and a minimum Scale C Rockwell Hardness of 60.

3. Abuse Load Tests (all large display panels)

Large display panels should withstand a 133 daN (300 lbf) static abuse load applied, in separate tests, in 5 different locations: in the centre, at the opposite corners (two separate tests), along the perimeter, at the midpoints of the short and long sides (two separate tests), or at an equivalent set of locations acceptable to EASA (see Figure 2 below).

For all the tests to be performed, the display panels should be mounted in a test fixture representative of the actual installation in the cabin.
For the above-mentioned load applications, it is acceptable to use any loading pad with a shape and dimensions that fit into a 15.24-cm (6-in.) diameter circle.
The display panels should withstand the applied loads without any adverse effect (e.g. glass elements, if present, cracking or breaking, the unit becoming dislodged from its mounts, substances released through cracks or openings, or sharp edges created).

During the test, it is acceptable for the display to suffer minor failures, such as minor cracks, provided that no parts are detached and the surface does not become a hazard to occupants.

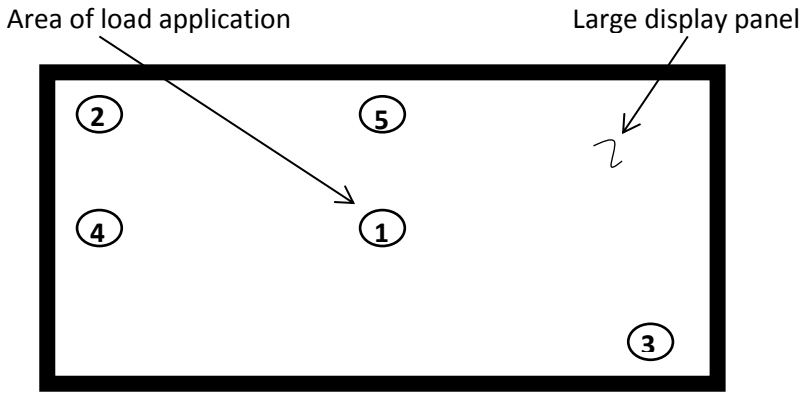


Figure 2 — Load Cases

1) centre loading;
2) corner loading;
3) opposite-corner loading;
4) short-side-midpoint perimeter loading; and
5) long-side-midpoint perimeter loading.

Amend AMC 25.807 as follows:

AMC 25.807

Emergency Exits

(…)

~~FAA Advisory Circular 25.807-1 'Uniform Distribution of Exits', dated 08/13/90 is accepted by the Agency as providing acceptable means of compliance with CS 25.807(e).~~

Create a new AMC 25.807(e) as follows:

AMC 25.807(e)

Emergency Exits Uniformity

FAA Advisory Circular 25.807-1 'Uniform Distribution of Exits', dated 08/13/90 is accepted by EASA as providing acceptable means of compliance with CS 25.807(e).

However, this Advisory Circular does not provide any guidance for those aeroplanes required to have no more than one pair of emergency exits. For those aeroplanes, ensuring that the seat-to-exit distance remains within acceptable limits as per the following criteria provides an acceptable means of compliance with CS 25.807(e).

Each passenger seat approved for use during taxiing, take-off or landing should be located such that:

(i) it is within 9.14 m (30 ft) from the nearest emergency exit on one side of the fuselage, and within 13.72 m (45 ft) from the nearest emergency exit on the other side of the fuselage; and

(ii) the occupant of that seat has the possibility to move to an emergency exit, on the left side, or the right side of the fuselage, whilst at all points along the way remaining within 9.14 m (30 ft) from an emergency exit on one side of the fuselage and within 13.72 m (45 ft) from an emergency exit on the other side of the fuselage.

When calculating the distance from a passenger seat, or from any point in the egress path of an occupant, to an emergency exit, this distance should be taken as the total longitudinal distance (i.e. as measured parallel to the aeroplane's longitudinal axis) that the escapee should cover in order to get to the emergency exit in question (i.e. the distance calculated should take into account all required changes in direction of movement but measured only longitudinally). For the distance from a passenger seat, as the starting point, the front edge of the seat bottom cushion at the centreline, with the seat in the taxiing, take off, and landing position is to be taken for seats installed at any orientation. The end point in each case is to be taken as the nearest edge of the emergency exit opening in the fuselage.

For aeroplanes with a passenger seating configuration of 19 or less, only one pair of emergency exits is required. However, such aeroplanes may have additional exits installed, which must then comply with CS 25.807(h) but not with the 18.3-m (60-feet) rule of CS 25.807(f)(4). The distance between each passenger seat and the nearest available emergency exit may be determined considering all available emergency exits, including the ones addressed by CS 25.807(h).

Create a new AMC 25.811(d) as follows:

AMC 25.811(d)

Sign Combination

The signs required by CS 25.811(d)(1), (d)(2) and (d)(3) may be combined according to the applicable parts of FAA AC 25-17A, *Transport Airplane Cabin Interiors Crashworthiness Handbook*, 18 May 2009.

Amend AMC 25.811(e)(4) as follows:

AMC 25.811(e)(4)

Emergency Exit Marking

The indicating markings for all Type II and larger passenger emergency exit unlocking handle motions should conform to the general shapes and dimensions indicated by Figures 1 and 2.

The indicating markings (arrow and word OPEN) should be consistent with the emergency exit signs chosen, i.e. red if letter emergency exit signs are installed, and green if symbolic emergency exit signs are installed.

(…)

Replace AMC 25.812(b)(1) by the following:

AMC 25.812(b)(1)

Emergency Lighting

General Requirements

Emergency exit signs should consist of a consistent type throughout the aeroplane. 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 determining the area 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 - emergency exit signs required by CS 25.811(d)(1) or (d)(3)

For each emergency exit sign required by CS 25.811(d)(1), and for each emergency exit sign required on each bulkhead or divider by CS 25.811(d)(3), at each point along any possible aeroplane egress path, the next closest required emergency exit sign visible at each point along the egress path should be sized and located such that it is no farther away from the escapee than its maximum allowable viewing distance calculated as below.

Egress paths to be assessed should be:

(1) any possible path from a passenger seat that can be occupied during taxiing, take-off, and landing to any passenger emergency exit; and

(2) any possible path from a point adjacent to any passenger emergency exit to any other passenger emergency exit.

Calculation of maximum viewing distance

For an emergency exit sign required by CS 25.811(d)(1) and for an emergency exit sign required on each bulkhead or divider by CS 25.811(d)(3), the following formulae, as modified by the notes below, apply for calculating a maximum viewing distance. The maximum allowable viewing distance for a sign is in each case the lower of the two values D_1 and D_2:

Text based signs	Symbolic signs
$D_1 = 2. Z . h_{letter}.$	$D_1 = 1.25 . Z . h_{symbol}.$
$D_2 = Z . \sqrt{(x_{sign}/2.5)}$	$D_2 = Z . \sqrt{(x_{sign}/2.5)}$

where:

1. Z is the distance factor obtained from Table 1 below;
2. h_{letter} is the overall height of each letter – which should be at least of 25 mm (1 inch) high;
3. h_{symbol} is the overall height of the white symbolic element incorporating the green 'running man' – which should be at least 40 mm (1.6 inches) high;
4. x_{sign} is the overall area of the sign; and
5. D_1, D_2, h_{letter} and h_{symbol} have the same units, and x_{sign} is in the same squared units as D_1, D_2, h_{letter} and $h_{symbol}.$

Note 1: In the case of dual-language text based emergency exit signs, only the English text is to be considered when selecting h_{letter} for use in the above formula. However, in determining the area of the sign (x_{sign}) for use in the above formula, the actual area may be used

Examples of acceptable designs of symbolic exit signs

CS 25.811(d)(1) (emergency exit locator sign)	

CS 25.811(d)(2) (emergency exit marking sign)	

CS 25.811(d)(3) (emergency exit sign on bulkhead or divider)	

Table 1: Z factor to be used for text based and symbolic emergency exit signs

Mean luminance of white contrast colour candela/m² (ft-L)	*Distance factor Z*
≥ 1.27 candela/m² (0.37 ft-L)	100
≥ 10 candela/m² (2.92 ft-L)	150
≥ 30 candela/m² (8.76 ft-L)	175
≥ 80 candela/m² (23.35 ft-L)	200
≥ 200 candela/m² (58.37 ft-L)	215
≥ 500 candela/m² (145.93 ft-L)	230

Minimum size - emergency exit signs required by CS 25.811(d)(2)

For an emergency exit sign required by CS 25.811(d)(2), any 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), with an overall area of at least 64.5 cm² (10 square inches) will be acceptable.

Supplementary directional arrows

The inclusion of an arrow or arrows in any of the signs discussed above, in order to increase the comprehension of the sign, is encouraged. The possibility to improve comprehension and the appropriate orientation of the arrows will depend on the particular installation. If arrows indicate a movement other than straight ahead, in the case of a symbolic sign, the depicted movement direction of the 'running man' (to the right/to the left) should be chosen to be compatible with the orientation of the arrow(s).
There may be other reasons to choose a particular movement direction of the 'running man', for instance where a sign required by CS 25.811(d)(2) is placed to the left or right of the emergency exit. In this case, the 'running man' should not suggest movement away from the emergency exit.
In the case of symbolic signs, the arrows should be in accordance with the style defined in European Standard (EN) ISO 7010:2012, i.e. type D of ISO 3864-3. The ratio of overall length of an arrow to the width of its tail should be not more than 7:1 nor less than 5.5:1.

Amend AMC 25.812(b)(2) as follows:

AMC 25.812(b)(2)

Emergency Lighting

~~Two acceptable methods of demonstrating compliance with the requirement of CS 25.812(b)(2) are as follows:~~

~~A locator sign, marking sign and bulkhead or divider sign should either:~~

- ~~have red letters at least 25 mm (1 inch) high on an illuminated white background at least 51 mm (2 inches) high.~~

~~or,~~

- ~~be a symbolic exit sign as derived from ISO/WD 3864-3 and ISO/CD 16069 "Safety Way Guidance System" and Draft BS 5499: Part 4 "Code of Practice for Escape Route Signing".~~

~~The symbols should be white on a green background according to ISO 3864. The lighted background-to-symbol contrast must be at least 1:10. The height of the symbols should be at least 38mm (1.5 inch).~~

For an emergency exit sign required by CS 25.811(d)(1), (2) or (3), any sign meeting the overall appearance requirements of AMC 25.812(b)(1), using English letters of at least 25 mm (1 inch) height, or a white symbolic element incorporating the 'running man' of at least 40 mm (1.6 inches), with an overall area of at least 64.5 cm^2 (10 square inches), will be acceptable.

The guidance of AMC 25.812(b)(1) regarding supplemental direction arrows is also applicable.

Amend AMC 25.812(e)(2) as follows:

AMC 25.812(e)(2)

Emergency Lighting

~~An acceptable method of demonstrating compliance with the requirement of CS 25.812(e)(2) regarding identifiers of floor level exits is to have a symbolic sign showing a white arrow on a green background as identified in the figure.~~

~~NOTE: Mixing language signs with symbolic signs is not an acceptable method of demonstrating compliance with CS 25.812(b)(1), (b)(2), and (e)(2).~~

If it is desired to identify each emergency exit by means of a symbolic sign, this sign should be white and green in compliance with European Standard (EN) ISO 7010:2012, Graphical symbols, safety colours and safety signs, registered safety signs.

Example of an acceptable design of symbolic sign to identify an exit

CS 25.812(e) (emergency exit identifier)	

The direction of the 'running man' (to the left/to the right) should not suggest movement away from the emergency exit.

The type of signs used to identify an emergency exit (letter based, symbolic) should be chosen to be consistent with the emergency exit signs throughout the cabin.

Create a new AMC 25.812(l)(1) as follows:

AMC 25.812(l)(1)

Transverse Separation of the Fuselage

Within CS 25.812(l)(1), the phrase 'in addition to the lights that are directly damaged by the separation' means that when calculating the percentage of electrically illuminated emergency lights rendered inoperative by the fuselage separation, the number of lights whose function is lost due to loss of power or loss of control input to the lights should be divided by the total number of electrically illuminated emergency lights installed. The lights that are directly damaged by the fuselage separation should not be included in the numerator of the calculation, but only those whose function is lost due to loss of power and/or control. The denominator should be the total of all electrically illuminated emergency lights installed.

Applicable parts of FAA AC 25.812-1A, *Floor proximity emergency escape path marking*, 22 May 1989 may be used.

Amend AMC 25.813(c) as follows:

AMC 25.813(c)

Emergency Exit Access and Ease of Operation

(…)

9 Minor obstructions

An item may be acceptable as meeting the intent of a minor obstruction in accordance with CS 25.813(c)(4)(ii) provided that, as soon as an occupant begins to open the emergency exit using only the required and visible operating handle, the obstruction moves such that the occupant instinctively understands how to complete removal of the obstructive item. Examples of such items are unattached (or loosely attached) soft seat back cushions on side-facing divans, provided that the cushion may be readily moved away and the emergency exit then easily fully opened. Ease of opening from the outside should also be assessed with the minor obstruction in place. Neither the emergency exit sign nor the operating handle should be obscured at any point.

Create a new AMC 25.813(e) as follows:

AMC 25.813(e)

Interior Doors

Doors separating occupiable areas of the aeroplane cabin that do not obstruct a possible passenger egress path when closed are not prohibited by CS 25 813(e).

Any such door should be openable from both sides without the use of any tool, which means without the need to use any item; it is not acceptable to require the use of even common items such as coins, credit cards, pens etc. (note: lavatory doors must comply with CS 25.820).
It is acceptable to have a door between a passenger compartment and a passenger emergency exit in contradiction with the prohibition of CS 25.813(e), provided that this door is secured in the open position by means acceptable to EASA that cannot be overridden except by a maintenance action (i.e. the necessary actions should be such that aeroplane occupants are unlikely to be equipped to perform them).

Create a new AMC 25.854 as follows:

AMC 25.854

Lavatory Fire Protection

The cabin length should be measured parallel to the aeroplane centre line from the most forward to the most aft point accessible to passengers or crew.
However, points within in-flight accessible cargo compartments, approved as meeting one of the classifications of CS 25.857, do not need to be considered.
On the flight deck, the most forward seat reference point (SRP) of the pilots' seats (with the seats adjusted to the most forward possible positions) should be used as the most forward point.

AMC — SUBPART F

Amend AMC 25.1309 as follows:

AMC 25.1309

System Design and Analysis

(…)

4. ***APPLICABILITY OF CS 25.1309*.**

Paragraph 25.1309 is intended as a general requirement that should be applied to any equipment or system as installed, in addition to specific systems requirements, except as indicated below.

(…)

d. The failure conditions ~~effects~~ covered by CS 25.810~~(a)(1)(v)~~ and CS 25.812 are excepted from the requirements of CS 25.1309(b). These Failure Conditions related to loss of function ~~Failure Conditions associated with these cabin safety equipment installations~~ are associated with varied evacuation scenarios for which the probability cannot be determined. It has

not been proven possible to define appropriate scenarios under which compliance with CS 25.1309(b) can be demonstrated. It is therefore considered more practical to require particular design features or specific reliability demonstrations as described in CS 25.810 and CS 25.812 ~~and except these items of equipment from the requirements of CS 25.1309(b)~~. Traditionally, this approach has been found to be acceptable.

(…)

5. *DEFINITIONS.*

The following definitions apply to the system design and analysis requirements of CS 25.1309 and the guidance material provided in this AMC. They should not be assumed to apply to the same or similar terms used in other regulations or AMCs. Terms for which standard dictionary definitions apply are not defined herein.

(…)

j. *Development Error*. A mistake in requirements, design, or implementation.

~~j~~k. *Error.* An omission or incorrect action by a crewmember or maintenance personnel, or a mistake in requirements, design, or implementation.

~~k~~l. *Event.* An occurrence which has its origin distinct from the aeroplane, such as atmospheric conditions (e.g. gusts, temperature variations, icing and lightning strikes), runway conditions, conditions of communication, navigation, and surveillance services, bird-strike, cabin and baggage fires. The term is not intended to cover sabotage.

~~l~~m. *Failure.* An occurrence, which affects the operation of a component, part, or element such that it can no longer function as intended, (this includes both loss of function and malfunction). Note: Errors may cause Failures, but are not considered to be Failures.

~~m~~n. *Failure Condition.* A condition having an effect on the aeroplane and/or its occupants, either direct or consequential, which is caused or contributed to by one or more failures or errors, considering flight phase and relevant adverse operational or environmental conditions, or external events.

~~n~~o. *Installation Appraisal.* This is a qualitative appraisal of the integrity and safety of the installation. Any deviations from normal, industry-accepted installation practices, such as clearances or tolerances, should be evaluated, especially when appraising modifications made after entry into service.

p. *Item*. A hardware or software element having bounded and well-defined interfaces.

~~o~~q. *Latent Failure.* A failure is latent until it is made known to the flight crew or maintenance personnel. A significant latent failure is one, which would in combination with one or more specific failures, or events result in a Hazardous or Catastrophic Failure Condition.

~~p~~r. *Qualitative.* Those analytical processes that assess system and aeroplane safety in an objective, non-numerical manner.

~~q~~s. *Quantitative.* Those analytical processes that apply mathematical methods to assess system and aeroplane safety.

~~r~~t. *Redundancy.* The presence of more than one independent means for accomplishing a given function or flight operation.

~~s~~u. *System*. A combination of components, parts, and elements, which are inter-connected to perform one or more functions.

(…)

8. *SAFETY OBJECTIVE.*

a. The objective of CS 25.1309 is to ensure an acceptable safety level for equipment and systems as installed on the aeroplane. A logical and acceptable inverse relationship must exist between the Average Probability per Flight Hour and the severity of *failure condition* effects, as shown in Figure 1, such that:

(1) Failure Conditions with No Safety Effect have no probability requirement.

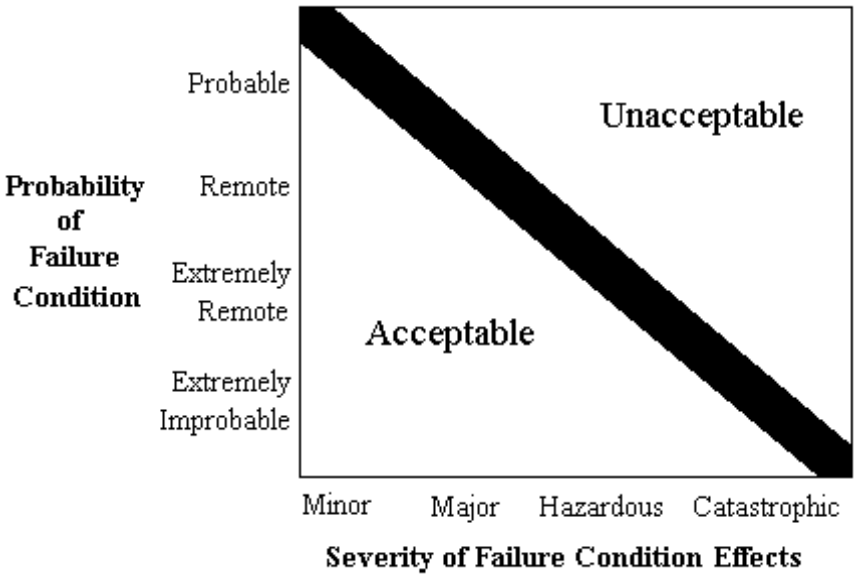
(2) Minor Failure Conditions may be Probable.

(3) Major Failure Conditions must be no more frequent than Remote.

(4) Hazardous Failure Conditions must be no more frequent than Extremely Remote.

(5) Catastrophic Failure Conditions must be Extremely Improbable.

Figure 1: Relationship between Probability and Severity of Failure Condition Effects



b. The classification of the Failure Conditions associated with the severity of their effects are described in Figure 2a.

The safety objectives associated with Failure Conditions are described in Figure 2b.

Figure 2a: Relationship Between Severity of the Effects and Classification of Failure Conditions

Severity of the Effects	Effect on Aeroplane	No effect on operational capabilities or safety	Slight reduction in functional capabilities or safety margins	Significant reduction in functional capabilities or safety margins	Large reduction in functional capabilities or safety margins	Normally with hull loss
	Effect on Occupants excluding Flight Crew	Inconvenience	Physical discomfort	Physical distress, possibly including injuries	Serious or fatal injury to a small number of passengers or cabin crew	Multiple fatalities
	Effect on Flight Crew	No effect on flight crew	Slight increase in workload	Physical discomfort or a significant increase in workload	Physical distress or excessive workload impairs ability to perform tasks	Fatalities or incapacitation
Classification of Failure Conditions		No Safety Effect	Minor	Major	Hazardous	Catastrophic

Figure 2b: Relationship Between ~~Probability and Severity of Failure Condition~~Classification of Failure Conditions and Probability

	~~Effect on Aeroplane~~	~~No effect on operational capabilities or safety~~	~~Slight reduction in functional capabilities or safety margins~~	~~Significant reduction in functional capabilities or safety margins~~	~~Large reduction in functional capabilities or safety margins~~	~~Normally with hull loss~~
	~~Effect on Occupants excluding Flight Crew~~	~~Inconvenience~~	~~Physical discomfort~~	~~Physical distress, possibly including injuries~~	~~Serious or fatal injury to a small number of passengers or cabin crew~~	~~Multiple fatalities~~
	~~Effect on Flight Crew~~	~~No effect on flight crew~~	~~Slight increase in workload~~	~~Physical discomfort or a significant increase in workload~~	~~Physical distress or excessive workload impairs ability to perform tasks~~	~~Fatalities or incapacitation~~
Classification of		No Safety	Minor	Major	Hazardous	Catastrophic

Failure Conditions	Effect				
Allowable Qualitative Probability	No Probability Requirement	<-Probable->	<--Remote-->	Extremely <----------> Remote	Extremely Improbable
Allowable Quantitative Probability: Average Probability per Flight Hour on the Order of:	No Probability Requirement	<----------> $<10^{-3}$ Note 1	<----------> $<10^{-5}$	<----------> $<10^{-7}$	$<10^{-9}$
~~Classification of Failure Conditions~~	~~No Safety Effect~~	~~< Minor >~~	~~< Major >~~	~~<Hazardous>~~	~~Catastrophic~~
Note 1: A numerical probability range is provided here as a reference. The applicant is not required to perform a quantitative analysis, nor substantiate by such an analysis, that this numerical criteria has been met for Minor Failure Conditions. Current transport category aeroplane products are regarded as meeting this standard simply by using current commonly-accepted industry practice.					

(…)

9. ***COMPLIANCE WITH CS 25.1309.***

This paragraph describes specific means of compliance for CS 25.1309. The applicant should obtain early concurrence of the certification authority on the choice of an acceptable means of compliance.

(…)

b. *Compliance with CS 25.1309(b).*

(…)

(4) *Acceptable Application of Development Assurance Methods.* Paragraph 9b(1)(iii) above requires that any analysis necessary to show compliance with CS 25.1309(b) must consider the possibility of ~~requirement, design, and implementation~~development errors. Errors made during the ~~design and~~ 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 direct techniques may still be appropriate for simple systems which perform a limited number of functions and which are not highly integrated with other aeroplane systems. For more complex or integrated systems, exhaustive testing may either be impossible because all of the system states cannot be determined or impractical because of the number of tests which must be accomplished. For these types of systems, compliance may be shown by the use of Development Assurance. The level of Development Assurance (function development assurance level (FDAL)/item development assurance level (IDAL)) should be ~~determined by~~commensurate with the severity of the Failure

Conditions the system is contributing to~~potential effects on the aeroplane in case of system malfunctions or loss of functions~~.

Guidelines, which may be used for the assignment of development assurance levels to aeroplanes and system functions (FDAL) and to items (IDAL), are described in the document referenced in 3b(2) above. Through this document, EASA recognises that credit can be taken from system architecture (e.g. functional or item development independence) for the FDAL/IDAL assignment process.

Guidelines, which may be used for providing Development Assurance, are described for ~~aircraft~~aeroplane and system~~s~~ development in the ~~D~~document referenced in ~~paragraph~~ 3b(2), and for software in the ~~D~~documents referenced in ~~paragraph~~ 3a(3) above. ~~(~~There is currently no agreed Development Assurance standard for airborne electronic hardware.~~) Because these documents were not developed simultaneously, there are differences in the guidelines and terminology that they contain. A significant difference is the guidance provided on the use of system architecture for determination of the appropriate development assurance level for hardware and software. EASA recognises that consideration of system architecture for this purpose is appropriate. If the criteria of Document referenced in paragraph 3b(2) are not satisfied by a particular development assurance process the development assurance levels may have to be increased using the guidance of Document referenced in paragraph 3a(3).~~

(…)

Create a new AMC 25.1365(b) as follows:

AMC 25.1365(b)

Installation of Cooktops

The following acceptable means of compliance are applicable to cooktops with electrically powered heating elements. Use of other types of heat sources, such as gas, is unlikely to be acceptable. If such a design is desired, EASA should be contacted for advice.

(1) Suitable means, such as conspicuous element 'on' indicators, physical barriers, or handholds, should be installed to minimise the potential of inadvertent personnel contact with hot surfaces of both the cooktop and cookware. Conditions of turbulence should also be considered.

(2) Sufficient design means should be provided to restrain cookware, including their contents, in place on the cooktop against flight loads and turbulence.

(a) Restraints should be provided to preclude hazardous movement of cookware and contents thereof. These restraints should accommodate the cookware that is approved for use with the cooktop.

(b) Restraints should be designed to be easily used and effective in service. The cookware restraint system should also be designed in a way that it may not be easily disabled, thus rendering it unusable.

(c) Appropriate placarding should be installed prohibiting the use of cookware not approved for use with the cooktop.

(3) Appropriate placarding should be installed prohibiting the use of cooktops (i.e. power on any heating surface) during taxiing, take-off, and landing.

(4) Suitable means should be provided to address the possibility of a fire starting on the cooktop or in its immediate vicinity. The following two means are acceptable:

(a) Appropriate placarding should be installed that prohibits any heating surface from being powered when the cooktop is unattended (Note: this would prohibit a single person from cooking on the cooktop and intermittently serving food to passengers while any surface is powered). A fire detector should be installed in the vicinity of the cooktop, which provides a warning audible throughout the passenger cabin; moreover, a fire extinguisher of appropriate size and extinguishing agent should be installed in the immediate vicinity of the cooktop. Access to the extinguisher should not be blocked by a possible fire on or around the cooktop. One of the fire extinguishers required by CS 25.851 may be used to satisfy this requirement if it is located in the vicinity of the cooktop and the total complement of extinguishers remains evenly distributed throughout the cabin. If this is not possible, then the extinguisher in the cooktop area should be additional to those required by CS 25.851; or

(b) An automatic (e.g. thermally activated) system should be installed to extinguish a fire at the cooktop and immediately adjacent surfaces. The agent used in the system should be an approved flooding agent suitable for use in an occupied area. The fire suppression system should have an appropriately located manual activation control. Activation of the fire suppression system (automatic or manual) should also automatically shut off power to the cooktop.

(5) The surfaces of the galley surrounding the cooktop, which would be exposed to a fire on the cooktop surface or in cookware on the cooktop, should be constructed of materials that comply with the flame penetration resistance requirements of Appendix F, Part III. During the selection of all galley materials in the vicinity of the cooktop, consideration should be given to ensure that the flammability resistance characteristics of the materials will not be adversely affected by the use of cleaning agents and utensils used to remove cooking stains.

(6) The cooktop should be ventilated with a system independent of the aeroplane cabin and cargo ventilation system. Maintenance procedures and time intervals

should be established for inspection and cleaning or replacement of ventilation system components to prevent the accumulation of flammable oils creating a fire hazard. These procedures and time intervals should be included in the instructions for continued airworthiness as required by CS 25.1529. The ventilation system ducting should be protected by a flame arrester (Note: the applicant may find additional useful information in Society of Automotive Engineers (SAE) Aerospace Recommended Practice (ARP) No 85, Revision E, ARP85E 'Air Conditioning Systems for Subsonic Airplanes' of 1 August 1991).

(7) Means should be provided to contain spilled foods or fluids in a manner that will prevent the creation of a slipping hazard to occupants as well as the loss of structural strength due to aeroplane corrosion.

(8) Cooktop installations should provide adequate space for the user to immediately escape a hazardous cooktop condition.

(9) A means to shut off power to the cooktop should be provided at the galley containing the cooktop and in the cockpit. If one (or more) dedicated switch(es) is (are) provided in the cockpit, smoke or fire emergency procedures should be provided in the AFM to cover their use.

(10) The cooktop should have either a lid that will completely enclose the cooking surface, or an appropriately located fire blanket of a size sufficient to completely cover the cooking surface should be provided. If a lid is installed, there should be a means to automatically shut off power to the cooktop when the lid is closed. The fire blanket material should be demonstrated to meet the European Standard (EN) 1869:1997, *Fire blankets*, or equivalent.

Amend AMC 25.1447(c)(1) as follows:

AMC 25.1447(c)(1)

Equipment Standards for Oxygen--Dispensing Units

(…)

6 A supplemental oxygen supply should be provided for each passenger lying on a bed or a seat that can be converted into a bed. Except for cases where the occupant's head location during sleeping is obvious, a placard indicating the correct sleeping position should be installed, unless the passenger oxygen system is designed to account for any sleeping position.

7 Sufficient illumination should be provided at all times or automatically when necessary (i.e. without the need of a crew action and without delay) at each location where supplemental oxygen is provided so that in the event of oxygen mask presentation, the user has sufficient visibility to enable quick donning.

Amend AMC 25.1447(c)(3) as follows:

AMC 25.1447(c)(3)

Equipment Standards for Oxygen--Dispensing Units

~~If~~ It is acceptable that oxygen outlets/units of dispensing equipment are not provided within ~~in a dedicated area, called here 'remote area',~~ an area where people are likely to congregate (for instance a waiting area for lavatory facilities, a bar/lounge area etc.), provided the applicant ~~should~~ demonstrates that sufficient oxygen-dispensing outlets are within five feet or five seconds reach of the ~~remote~~ area~~(s)~~ and ~~should show~~ that no visual obstruction exists between the potential oxygen users and the outlets, such as curtains or partitions, unless another method of indication (e.g. an 'oxygen in use' light) is provided in the ~~remote~~ area.

There should be at least two outlets and units of dispensing equipment in toilets, washrooms, galley work areas etc. In such areas where occupancy of more than two persons can be expected, the number of outlets (within the area or within five feet or five seconds reach) should be consistent with the expected maximum occupancy.
In the case of a shower, there should be an oxygen outlet and unit of dispensing equipment immediately available to each shower occupant without stepping outside the shower. Reaching through an opened shower cubicle door is acceptable, in which case the door should be sufficiently transparent so that the location of the mask and the required actions to access it are immediately obvious.

AMC — SUBPART G

Amend AMC 25.1541 as follows:

AMC 25.1541

Markings and Placards –— General

Markings or placards should be placed close to or on (as appropriate) the instrument or control with which they are associated. The terminology and units used should be consistent with those used in the Flight Manual. The units used for markings and placards should be those that are read on the relevant associated instrument.
Publications which are considered to provide appropriate standards for the design substantiation and certification of symbolic placards may include, but are not limited to, 'General Aviation Manufacturers Association (GAMA) Publication No. 15 — Symbolic Messages', Initial Issue, 1 March 2014.

AMC — APPENDICES

Create a new AMC to Appendix S, S25.1 as follows:

AMC to Appendix S, S25.1

Passenger seating configuration

Where this term is used in Appendix S:
'**Passenger seating configuration**' means the passenger seating capacity established during the certification process (either type certificate (TC), supplemental type certificate (STC) or change to the TC or STC, as relevant), conducted for the particular cabin interior and emergency exit arrangement of the aeroplane considered.

The passenger seating configuration is equal to, or less than, the maximum passenger seating capacity of the relevant type-certified aeroplane as indicated in the aeroplane type certificate data sheet (TCDS).

The passenger seating configuration may be less than the total number of passenger seats in the aeroplane that are approved for occupancy during taxiing, take-off, and landing, if seats in excess are installed; in such a case the requirement S25.40(c) *Seats in Excess* must be complied with.

Create a new AMC to Appendix S, S25.10(a) as follows:

AMC to Appendix S, S25.10(a)

Interior Doors on Non-Commercially Operated Aeroplanes

(1) The following provides acceptable means to ensure that a door is open before entering any of the taxiing, take-off, and landing phase, as required by S25.10(a)(1):

(a) The door should be conspicuously placarded on both sides to be in the safe (i.e. open and secured) position during taxiing, take-off, and landing;

(b) The operation of the door and the requirement that the door be secured open for taxiing, take-off, and landing must be the subject of a passenger briefing, and the requirement for this briefing must be part of the AFM; for the purpose

of this briefing, a description of the operation of the internal door should be made available to the flight crew; and

(c) There should be a means to signal to the flight crew in a timely manner if the door is not open and secured in a safe position before entering any of the taxiing, take-off, or landing phases. The indication should be triggered during the descent phase, early enough to enable the flight crew to take appropriate action before entering the approach phase, unless the aeroplane is required to have at least one cabin crew member on board. Appropriate procedures for crew action should be established.

(2) The following provides acceptable means to ensure that the door remains open during taxiing, take-off, and landing, and especially during and after a crash landing, as required by S25.10(a)(2):

(a) Dual means should be provided to secure the door in the open position for taxiing, take-off, and landing. Each of those dual means should be capable of reacting to the inertia loads specified in CS 25.561; and

(b) The indication to the flight crew mentioned in the above condition (1)(c) should be triggered without delay and remain active whenever the door is not in the safe position during any of the taxiing, take-off, and landing flight phases. Appropriate procedures for crew action should be established.

(3) Regarding the indication mentioned in the above paragraphs (1)(c) and (2)(b), if several interior doors are installed, it might not be necessary to provide a distinct indication for each door on the flight deck. Door position indication in the cockpit may be achieved by means of a single visual indication serving all interior doors installed in the aeroplane, provided that at least one of the following two conditions is met:

(a) The number and location of the interior doors is such that quick identification of the incorrectly positioned door can be made by cabin occupants. A cabin layout which may be accepted as meeting this condition may be one in which all interior doors can be easily viewed during a direct walk from the front to the rear of the cabin.

(b) There is a simultaneous indication provided to a required cabin crew member which allows easy identification of the interior door in the incorrect position. An associated procedure for coordination between the flight and cabin crew should be included in the AFM.

(4) The following provides acceptable means to comply with the requirement S25.10(a)(3):

(a) In case the door is operated (opening, closing and/or latching) manually: the door should be easily operable from both sides, and if a latch is installed to restrain the door in the closed position, the door should be capable of being unlatched from both sides without the aid of any tool and without the need of any item (it is not acceptable to require the use of even common items such as coins, credit cards, pens, etc.);

(b) In case the door is operated (opening, closing and/or latching) electrically: there should be a manual override that satisfies the above condition (4)(a), unless the electrical opening and retention in the open and secured position continues to function following complete loss of normal electrical power, and it is demonstrated that following any probable electrical failure, the door defaults to the fully open and secured position;

(c) The door should be frangible (or equivalent, e.g. it has a removable panel) in both directions. An assessment should be made of the moveable cabin features adjacent to the door in order to ensure that sufficient clearance on each side of the door, during all phases of flight, is assured by design such that the frangibility feature(s) will work as intended. Alternatively, it may be shown that, irrespective of the positioning of moveable cabin features, the overall frangibility objective is still achieved, e.g. by reaching through a reduced opening to easily move the feature before finishing the actions needed to provide the full opening intended. The frangibility should be demonstrated by test using a 5th percentile female, and the resulting aperture should be demonstrated to be large enough for a 95th percentile male to escape. The case of probable jamming in a non-fully closed position should be considered;

(d) As an alternative to the above mentioned frangibility feature, it may be demonstrated, for example with double sliding doors, that following any probable failure or jamming of the door, a sufficient opening is still ensured that allows for passing through the doorway; 'sufficient opening' would mean, in the case of a sliding door, an opening from floor to ceiling consistent with the minimum required width of aisle as prescribed by CS 25.815 for a passenger seating capacity equal to the maximum expected number of passengers that would need to evacuate through the passenger egress path crossed by the door.

(e) The pre-flight passenger briefing (as mentioned in condition (1)(b)) should contain instructions on how to restore a sufficient opening for evacuation (frangibility feature or alternative means) in case of failure or jamming of the door.

For the definition of 'probable failure or jamming of the door', refer to the definition of 'Probable Failure Conditions' in AMC 25.1309.

Create a new AMC to Appendix S, S25.10(b) as follows:

AMC to Appendix S, S25.10(b)

Interior Doors on Commercially Operated Aeroplanes

In the case of an aeroplane which is not intended to be limited to non-commercial operations, the familiarity of the occupants with the specific cabin features of the aeroplane cannot be credited in the demonstration that, in the case of any probable failure or jamming of the door in a position other than fully open, any occupant is able, from any compartment separated by that door, to restore in an easy and simple manner a sufficient opening to access the compartment on the other side of the door (compliance with the condition

S25.10(a)(3)); this means, for instance, that when the demonstration relies on the frangibility of the door, there should be a placard on each side of the door to indicate the presence and functioning of this feature.

The requirement S25.10(b)(2)(ii) states that 'the demonstration of compliance against the provision S25.10(a)(1) and (2) shall not rely on any passenger action, nor involve any flight crew member leaving their position in the cockpit'. Any of the following solutions may be employed to meet this requirement:

(1) An automatic system, for the opening of the door and retention of the door in the open and secured position.
(2) A control in the cockpit, compliant with CS 25.777, to activate remotely the opening of the door and retention of the door in the open and secured position.
(3) For aeroplanes required to have at least one cabin crew member on board, and the cabin crew is clearly tasked with ensuring that the door is open before entering any of the taxiing, take-off, and landing phases. Appropriate cabin crew procedures and cabin crew training should be established.

Create a new AMC to Appendix S, S25.10(c) as follows:

AMC to Appendix S, S25.10(c)

Isolated Compartments

(1) Cabin Compartments

(a) Compartments to be considered as isolated

Compartments in an aeroplane with an approved passenger capacity of less than 20 and a cabin length of 18.29 m (60 ft) or less do not need, in any case, to be considered as isolated. AMC 25.854 provides guidance on how to determine the cabin length.

S25.10(c) requires that a compartment in which a fire would not be directly or would not be quickly detected by occupants of another compartment must meet additional criteria in order to provide confidence that a fire will be detected. Such a compartment is described as an isolated compartment.

Any compartment that can be occupied by crew members and/or passengers during flight (other than accessible cargo/baggage compartments) should be considered as isolated for the purposes of showing compliance to S25.10(c) if it cannot be assured that fire/smoke in the compartment will be quickly detected by occupants of other occupied compartments of the aeroplane due to rapid smoke/fumes transmission enabled by the design of the aeroplane.

The assurance that fire/smoke will be quickly detected by occupants of other occupied compartments in the aeroplane may be provided by obvious smoke/fumes passage features, e.g. grills/louvres in a door, or via the aeroplane's environmental control system air recirculation characteristics. Substantiation of the effectiveness of such declared smoke/fumes transmission means, via ground and/or flight tests, may be required.

Detection of fire/smoke by occupants of another compartment only will provide the required assurance if there is confidence that this other compartment in question will be occupied, and not by sleeping persons (i.e. it is a compartment that meets the conditions set out in paragraph (1)(b)(ii) below). Thus, if smoke/fumes transmission is relied upon for compliance, the occupancy conditions of the aeroplane as a whole need to be taken into account.

(b) Isolated compartments occupied for the majority of the flight time

S25.10(c) exempts isolated compartments (as described in paragraph (a) above) that are occupied for the majority of the flight time from being equipped with a smoke/fire detection system, based on the assumption that the occupants will quickly detect the fire.

(i) However, some categories of isolated compartments will by their nature not be eligible for this approach, either because there is a risk that all occupants will be sleeping (sleeping persons will not be able to detect a fire starting in the isolated compartment), or because occupancy for the majority of the flight time cannot be realistically assessed. Examples include, but are not limited to, the following:

(A) bedrooms, (i.e. rooms containing any sleeping installations intended to provide a high level of sleeping comfort, such as beds, or berthable divans, even if they also contain seats that can be occupied during taxiing, take-off, and landing; however, passenger seats do not need to be considered as sleeping installations in this context);

(B) specialised rooms for which permanent occupation during the flight is unlikely (examples would include smoking rooms, cinema rooms, etc.);

(C) washrooms/bathrooms, although the intent of S25.10(c) will be met in any case, if they are compliant with CS 25.854; however, a shower cubicle does not need to be considered an isolated compartment;

(D) crew rest compartments; and

(E) galley compartments.

(ii) On the other hand, an isolated compartment, unless meeting one of the criteria in (i) above, will be accepted as being occupied for at least the majority of the flight time, thus providing for smoke/fire detection by the occupants, if any of the following conditions are met:

(A) it is the flight crew compartment

(B) all required cabin crew seats are located in the isolated compartment;

(C) the isolated compartment contains a crew station that due to its specialised purpose, is likely to be occupied for the majority of the flight time;

(D) the number of seats in the isolated compartment (including cabin attendant seats and seats in excess) approved for occupancy during taxiing, take-off, and landing is at least equal to the number indicated in the right hand column of the table below.

Total number of passenger seats installed on the aeroplane approved for occupancy during taxiing, take-off, and landing (including seats in excess)	An isolated compartment is accepted as being occupied for the majority of the flight time if it contains at least the following number of seats approved for occupancy during taxiing, take-off, and landing
Up to 19	2
20–23	3
24–29	4
30–36	5
37–43	6
44–49	7
50–56	8
57–63	9
64 and above	10

Note: the 'Up to 19' figure is included for the case of an aeroplane with a total cabin length in excess of 18.29 m (60 ft).

(iii) In addition, an isolated compartment featuring no seat and no stowage (e.g. a connecting corridor) might be accepted as being an isolated compartment without a smoke/fire detection system, because of the low likelihood of a fire starting in such a compartment.

(c) Minimum requirements for compartments

For all compartments, irrespective of whether or not they are required to have a smoke/fire detection system installed:

(i) For accessibility and firefighting purposes, sufficient lighting in the compartment should be provided. For compartments that could be dark during flight, a means should be provided to enable a person entering the compartment to readily gain visibility of the interior. Such means may be:

(A) a conveniently located, easy to find and use lighting control for the compartment;

(B) a flashlight within close proximity to the entrance of the compartment; or

(C) automatic illumination in the event the smoke/fire detection system in the compartment (if installed) triggers.

(ii) At least one readily accessible handheld fire extinguisher should be available for use in each compartment. Fire extinguishers required by CS 25.851(a) may be used for this purpose. On the other hand this may also lead to the need to install more fire extinguishers than the minimum required by CS 25.851(a).

(iii) Portable breathing equipment, required by CS 25.1439(a), should be located close to the handheld fire extinguisher.

(b) Smoke/fire detection in isolated compartments

For interiors with more than one isolated compartment, there should be means by which flight or cabin crew can readily identify in which compartment smoke/fire has been detected. Depending on the number of isolated compartments and the specific layout, such means might be simply moving through the cabin and checking each compartment (in the case that cabin crew are required to be on board) or might need to be a visual indication outside each compartment, or some form of annunciator panel available to an appropriate crew member. The objective in any case is that correct identification of the location of the smoke/fire should be possible without unnecessary delay.

If the isolated compartment incorporates a stowage compartment of a volume greater than 0.7 m^3 (25 ft^3), this stowage compartment should be itself equipped with a smoke detector, unless it can be demonstrated that smoke from within the stowage compartment will be detected by the detector of the isolated compartment in which the stowage compartment is located (e.g. through grilles in the stowage door), and within the time specified in the requirement S25.10(c).

If the isolated compartment incorporates a galley, or if smoking is to be allowed in the isolated compartment, nuisance triggering of the smoke/fire detection system may be minimised by a design feature that provides for temporary system deactivation by an occupant (passenger or crew member). In that case, full reactivation should be automatic after a time period of no longer than 10 minutes following the last deactivation action.

The effectiveness of the smoke/fire detection system should be demonstrated for all approved operating configurations and conditions.

For smoke detection demonstration, FAA AC 25-9A, *Smoke detection, penetration, and evacuation tests and related flight manual emergency procedures*, 6 January 1994 provides acceptable means of compliance.

During testing, it should be demonstrated that no inadvertent operation of smoke/fire detectors in any compartment would occur as a result of fire starting in any other compartment.

An assessment of the compartment design and observations during smoke/fire detection tests will be expected in order to provide a demonstration of the effectiveness of firefighting procedures. This should also include demonstrating that the compartment is provided with sufficient access in flight to enable a crew member to effectively reach any part with the contents of a handheld fire extinguisher.

Create a new AMC to Appendix S, S25.10(d) and (e) as follows:

AMC to Appendix S, S25.10(d) and (e)

Deactivation of existing Emergency Exits

(1) General

S25.10(d)(3) requires to ensure that the distance from each passenger seat to at least one non-deactivated emergency exit on each side of the fuselage remains compatible with easy egress from the aeroplane.
For the purpose of this provision, a passenger seat distribution will be considered to meet this objective, provided that each passenger seat approved for use during taxiing, take-off, or landing is located such that:

(a) It is within 9.14 m (30 ft) from the nearest emergency exit on one side of the fuselage on the same deck, and within 13.72 m (45 ft) from the nearest emergency exit on the other side of the fuselage on the same deck; and

(b) The occupant of that seat has the possibility to move to an emergency exit, on the left side, or the right side of the fuselage, whilst at all points along the way remaining within 9.14 m (30 ft) from an emergency exit on one side of the fuselage on the same deck and within 13.72 m (45 ft) from an emergency exit on the other side of the fuselage on the same deck.

When calculating the distance from a passenger seat, or from any point in the egress path of an occupant, to an emergency exit, this distance should be taken as the total longitudinal distance (i.e. as measured parallel to the aeroplane's longitudinal axis) that the escapee should cover in order to get to the emergency exit in question (i.e. the distance calculated should take into account all required changes in the direction of movement but measured only longitudinally). For the distance from a passenger seat, as starting point, the front edge of the seat bottom cushion at the seat centreline is to be taken (for forward, angled, side or aft-facing seats), and as end point, the nearest exit edge.

For aeroplanes with an approved passenger seating configuration of 19 or less, only one pair of emergency exits is required. However, such aeroplanes may have additional exits installed, which must then comply with CS 25.807(h) but not with the 18.3-m (60-feet) rule of CS 25.807(f)(4). The distance between each passenger seat and the nearest available emergency exit may be determined considering all available emergency exits, including the ones addressed by CS 25.807(h).

When deactivation of one or more emergency exits results in an emergency exit arrangement that is asymmetrical relative to the aeroplane centre line, the acceptable seating capacity for each cabin zone should be determined considering the emergency exits remaining available on each side of the fuselage separately, i.e. following a similar methodology as the one used in FAA AC 25.807-1, *Uniform distribution of exits*, 13 August 1990.

(2) Examples

The following examples illustrate the analysis method to be followed when examining the acceptability of various emergency exit deactivation schemes on an aeroplane that is originally type-certified with two pairs of Type C exits (rated at 55 passengers for each pair) at the forward and aft limits of the cabin, and a single pair of overwing Type III exits (rated at 35 passengers). In accordance with CS 25.807, this emergency exit layout will have a possible maximum approved passenger capacity of 145 (55 + 35 + 55). It is assumed that the aeroplane manufacturer has received approval for this number of passengers.

The distance between the nearest exit edges of the two pairs of Type C exits is 20 m (65.7 ft). The overwing exits pair's forward edges are 8 m (26.3 ft) from the rear edges of the forward Type C exit pair.

The figures below provide additional clarification on the methodology to be used and the resultant limitations.

A cabin area that should not include any crew or passenger seats that can be occupied during taxiing, take-off, and landing is referred to as a 'stay-out zone', coloured pink in the illustrations below. The hatched/yellow areas in the illustrations below are referred to as 'additional stay-out zones' and should also not include any crew or passenger seats that can be occupied during taxiing, take-off, and landing. Seats located within these latter zones do meet the criteria of the above paragraph (1)(a) but do not meet the criteria of the above paragraph (1)(b). In other words, although these zones are located sufficiently close to emergency exits to meet the basic emergency exit egress distance requirements on both sides of the fuselage, an occupant of one of these seats would be forced to traverse a cabin area that does not meet these requirements, i.e. a stay-out zone, in order to egress the aeroplane.

Example 1

In the first example, only the left hand (LH) overwing Type III exit is deactivated.

Identification of stay-out zones

No stay-out zone needs to be identified in the cabin, if any possible passenger seat location will be no more than 9.14 m (30 ft) from the nearest exit on one side of the

fuselage, and no more than 13.72 m (45 ft) from the nearest exit on the other side of the fuselage, i.e. in compliance with the above paragraph 1.(i).

Calculation of the basic passenger seating configuration limitations set by S25.1(a)
In the case of non-commercial operations, in accordance with S25.1(a), the passenger capacity will have an upper possible limit of 73 passengers (1/2 of 145 (55 + 35 + 55) rounded up), i.e. one half of the maximum passenger seating capacity of the type-certified aeroplane having all exits functional.

In the case of commercial operations, in accordance with S25.1(a), the passenger capacity will have an upper possible limit of 48 passengers (1/3 of 145 (55 + 35 + 55) rounded down), i.e. one third of the maximum passenger seating capacity of the type-certified aeroplane having all exits functional. Additionally, there will be an upper possible limit of 30 passengers seated forward or aft of the overwing exits (1/3 of 90 (55 + 35)), i.e. one third of the maximum passenger seating capacity for each cabin zone of the type-certified aeroplane having all exits functional.

Calculation of additional passenger seating limitations due to exit deactivation

Firstly, a zonal analysis is conducted on the right side of the fuselage in accordance with S25.10(d). Two zones are represented by the exits on this side (all original emergency exits remain functional).
The allowable number of seats between the forward Type C exit and the overwing exit is limited to one half of the sum of the ratings of the exits that bound the zone: 1/2 of 90 (55 + 35) = 45.
The same limit is valid also for the zone between the overwing exit and the rearmost Type C exit.

Secondly, a zonal analysis is conducted on the left side of the fuselage in accordance with S25.10(d). There is only one zone represented by the remaining functional exits on this side. The allowable number of passenger seats between the forward and aft Type C exits is again limited to one half of the sum of the exit ratings that bound the zone: 1/2 of 110 (55 + 55) = 55.
The passenger seating locations for taxiing, take-off, and landing should simultaneously satisfy all basic limitations set by S25.1(a) and both of the zonal analyses in accordance with S25.10(d).

In the case of non-commercial operations, this means that the passenger seating configuration is limited to 55 (i.e. in this case, the limitation resulting from the left-side fuselage zonal analysis is most constraining and defines the maximum seating capacity of the aeroplane) and a maximum of 45 passenger seats located either forward or aft of the remaining functional overwing exit may be occupied for taxiing, take-off, and landing.

However, for commercial operations, an overriding consideration applies due to the fact that there is a non-compliance with CS 25.807(f)(4) on the left side of the fuselage, and

the provisions of S25.10(d) only apply to non-commercial operations. The seating capacity of the example aeroplane in commercial operation will thus be limited to 19 seats because CS 25.807(f)(4) only applies to aeroplanes for which more than one exit pair is required. However, there will be no limitation on the passenger seating location for taxiing, take-off, and landing, as explained in AMC 25.807.

Example 2

In the second example, both left hand (LH) and right hand (RH) overwing Type III exits are deactivated. The aeroplane has thus only two pairs of remaining functional Type C exits located at either end of the cabin.

Identification of stay-out zones

A stay-out zone is identified in the middle of the cabin, where a passenger seat that can be occupied during taxiing, take-off, and landing would not be in compliance with the above paragraph 1.(i), i.e. would be further than 9.14 m (30 ft) from the nearest exit, on both sides of the fuselage. The exact limitation on the seat installation location in order to respect the stay-out zone should be calculated using the longitudinal measurement method as explained in AMC 25.807.

Calculation of the basic passenger seating configuration limitation set by S25.1(a)

In the case of non-commercial operations, in accordance with S25.1(a), the passenger capacity will have an upper possible limit of 73 passengers (1/2 of 145 (55 + 35 + 55) rounded up), i.e. one half of the maximum passenger seating capacity of the type-certified aeroplane having all exits functional.

In the case of commercial operations, in accordance with S25.1(a), the passenger capacity will have an upper possible limit of 48 passengers (1/3 of 145 (55 + 35 + 55) rounded down), i.e. one third of the maximum passenger seating capacity of the type-certified aeroplane having all exits functional. Additionally, there will be an upper possible limit of 30 passengers seated forward or aft of the overwing exits (1/3 of 90 (55+35)), i.e. one third of the maximum passenger seating capacity for each cabin zone of the type-certified aeroplane having all exits functional.

Calculation of additional passenger seating limitations due to exit deactivation

In this example, the arrangement of the remaining functional exit is symmetrical on either side of the aeroplane centre line, hence, no separate LH and RH zonal analyses are required, and only one cabin zone remains.

The zonal analysis, in accordance with S25.10(d), results in the number of seats that may be occupied during taxiing, take-off, and landing between the forward and aft Type C exits, limited to one half of the sum of the ratings of the exits that bound the zone: i.e. 1/2 of 110 (55 + 55) = 55.

The passenger seating locations for taxiing, take-off, and landing should simultaneously satisfy all basic limitations set by S25.1(a) and the zonal analysis in accordance with S25.10(d).

Therefore, for non-commercial operations, a maximum total of 55 passenger seats may be occupied during taxiing, take-off, and landing, in any combination of individual locations forward or aft of the identified stay-out zone.
For commercial operations, as in Example 1, the seating capacity of the aeroplane will be limited to 19, due to non-compliance with CS 25.807(f)(4), on both sides of the fuselage this time. However, as also explained in Example 1, the total of 19 passenger seats that can be occupied during taxiing, take-off, and landing may be in any combination of locations forward or aft of the identified stay-out zone.

Example 3

In the third example, the rearmost LH Type C exit is deactivated. The aeroplane has, thus, one pair of functional forward Type C emergency exits and one pair of functional overwing Type III emergency exits, and a functional aft Type C emergency exit on the RH side only.

Identification of stay-out zones

No stay-out zone can be identified in the cabin, i.e. any possible passenger seat location will be no more than 9.14 m (30 ft) from the nearest exit on one side of the fuselage, and no more than 13.72 m (45 ft) from the nearest exit on the other side of the fuselage.

Calculation of the basic passenger seating configuration limitations set by S25.1(a)

In the case of non-commercial operation, in accordance with S25.1(a), the passenger capacity will be limited to 73 passengers (1/2 of 145 (55+35+55) rounded up), i.e. one half the maximum passenger seating capacity of the type certified aeroplane with all exits functional.

In the case of commercial operation, in accordance with S25.1(a), the passenger capacity will have an upper possible limit of 48 passengers (1/3 of 145 (55+35+55) rounded down), i.e. one third the maximum passenger seating capacity of the type certified aeroplane with all exits functional. Additionally, there will be an upper possible limit of 30 passengers seated forward or aft of the overwing exits (1/3 of 90 (55+35)), i.e. one third of the maximum passenger seating capacity for each cabin zone of the type certified aeroplane with all exits functional.

Calculation of additional passenger seating limitations due to exit deactivation

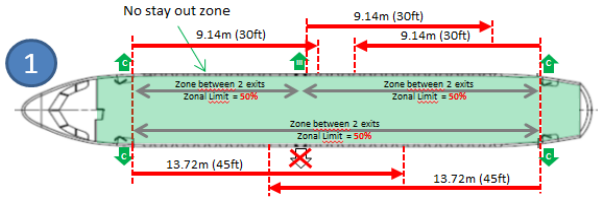
Firstly, a zonal analysis is conducted on the right side of the fuselage, in accordance with S25.10(d). Two zones are represented by the remaining functional exits on this side (all original emergency exits remain functional).
The allowable number of seats for installation between the forward Type C and the overwing exit is limited to one half of the sum of the ratings of the exits that bound the zone: 1/2 of 90 (55 + 35) = 45.
The same limit is also valid for the zone between the overwing emergency exit and the rearmost Type C exit.

Secondly, a zonal analysis is conducted on the left side of the fuselage. Again, two zones are represented by the remaining functional emergency exits on this side, but this time, one zone is a so-called dead end zone.
As for the right side, it is acceptable to install 45 seats between the forward Type C and the overwing exit: 1/2 of 90 (55 + 35) = 45.
In the dead end zone aft of the overwing exit, it is acceptable to install a maximum of 18 seats (1/2 of 35 rounded up).
The passenger seating locations for taxiing, take-off, and landing should simultaneously satisfy all basic limitations set by S25.1(a) and both of the zonal analyses in accordance with S25.10(d).
Therefore, for non-commercial operations, this results in a maximum total seating capacity of 63 when it simultaneously satisfies the upper limit for each zone, i.e. 45 for the forward zone and 18 for the aft zone.

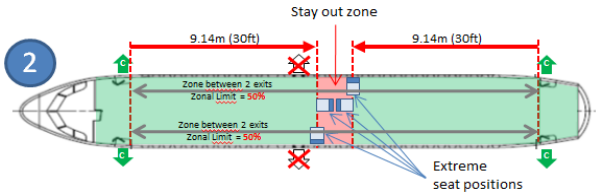
In case of commercial operations, the total capacity of the aeroplane will be limited to 48 passengers, not exceeding 30 passengers forward of and 18 aft of the overwing exits.

Further examples

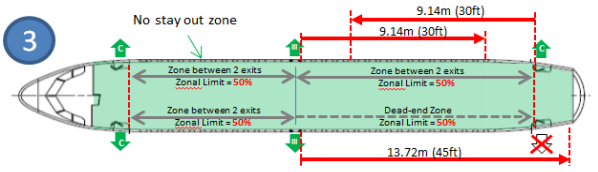
In addition to Examples 1, 2 and 3 above, further examples of exit deactivation for the same basic aeroplane are illustrated, and the resultant allowable passenger seating restrictions are summarised.
The principles evident from these examples can be used to determine zonal capacities and stay-out zones for any aeroplane.



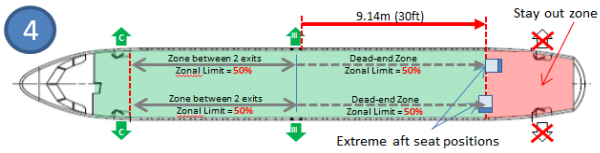
	Operations	Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	RH: 50% [90+90] = 90 LH: 50% [110] = 55	55
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Distance>60ft: No zonal analysis required	19 (Distance>60ft)
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



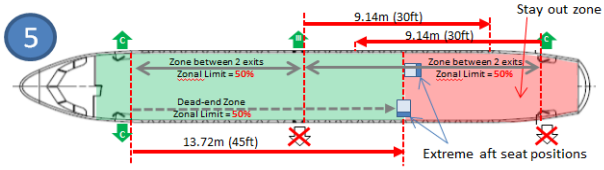
	Operations	Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	RH: 50% [110] = 55 LH: 50% [110] = 55	55
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Distance>60ft: No zonal analysis required	19 (Distance>60ft)
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



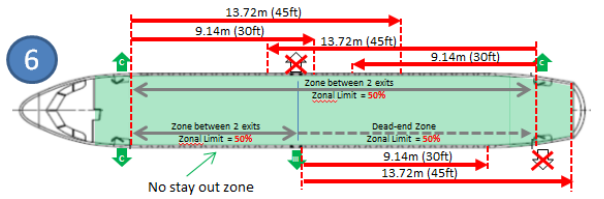
	Operations	Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	RH: 50% [90+90] = 90 LH: 50% [90+35] = 63	63
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	RH: 50% [90+90] = 90 LH: 50% [90+35] = 45+18 1/3* [55+35] = 30 / zone	48 (30+18)
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



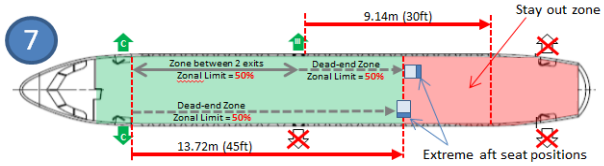
Operations		Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	RH: 50% [90+35] = **63** LH: 50% [90+35] = **63**	63
2	Commercial Low occupancy	1/3 TC = **48** 1/3 per zone	RH: 50% [90+35] = 45+**18** LH: 50% [90+35] = 45+**18** 1/3* [55+35] = **30** / zone	48 **(30+18)**
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



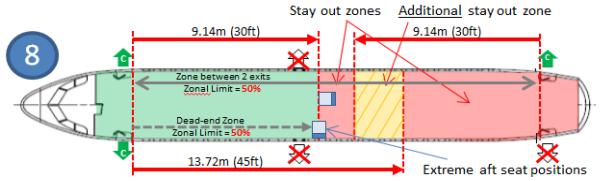
Operations		Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	Only one pair of exits: No zonal analysis required	19 (25.807 limit with only one pair)
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Only one pair of exits: No zonal analysis required	
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



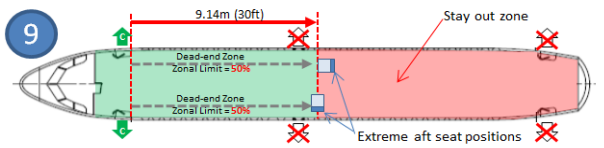
Operations		Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	RH: 50% [110] = **55** LH: 50% [90+35] = 63	55
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Distance>60ft: No zonal analysis required	19 (Distance>60ft)
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



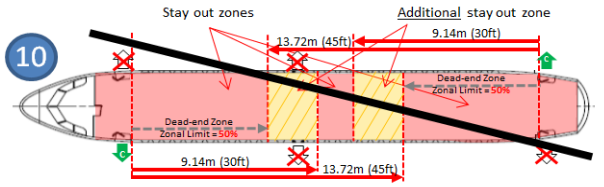
Operations		Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	Only one pair of exits: No zonal analysis required	19 (25.807 limit with only one pair)
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Only one pair of exits: No zonal analysis required	
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



Operations		Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	Only one pair of exits: No zonal analysis required	19 (25.807 limit with only one pair)
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Only one pair of exits: No zonal analysis required	
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



Operations		Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	Only one pair of exits: No zonal analysis required	19 (25.807 limit with only one pair)
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Only one pair of exits: No zonal analysis required	
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized



Operations		Eligibility limit	Zonal limits	Total Limit
1	Non-Commercial	1/2 TC = 73	Only one pair of exits: No zonal analysis required	0 Only stay out zones
2	Commercial Low occupancy	1/3 TC = 48 1/3 per zone	Only one pair of exits: No zonal analysis required	
3	Commercial (>1/3 TC + per zone)	Not Authorized	Not Authorized	Not Authorized

Create a new AMC to Appendix S, S25.20(a)(1) as follows:

AMC to Appendix S, S25.20(a)(1)

Flammability of Bed Mattresses

Mattresses of beds that are convertible to/from seats, regardless of their location in the aeroplane, and irrespective of whether or not the seat configuration is approved for occupancy during taxiing, take-off, and landing, should meet the criteria of CS-25, Appendix F, Part II.

As required by CS-25, Appendix F, mattress foam shall be tested for 12,7-mm (1/2-in.) thickness. If the mattress consists of two or more foams glued together, the foam specimen should consist of two 6.34-mm (1/4-in.) (three layers of 4.2 mm (1/6 in.), etc.) pieces glued together. Three specimens should be made for each combination of foams that are glued together in the production mattress. Any other production mattress components that are glued together should also be tested together.

If such specimens do not meet the test criteria of CS-25, Appendix F, Part I, it is acceptable to test each production mattress component separately, including a sheet of glue, using the test criteria of Appendix F, Part I.

Additionally, the Bunsen burner is then to be applied at three separate corners of the production mattress with all its components. The three-corner test does not need to be conducted if the cushion passes the tests of CS-25, Appendix F, Part II.

Create a new AMC to Appendix S, S25.20(b) as follows:

AMC to Appendix S, S25.20(b)

Exit as effective as a Type IV exit

An acceptable means of compliance with the requirement that the remaining exit resulting from an obstruction shall be as effective as a Type IV emergency exit (S25.20(b)(1) and (b)(2)), is to demonstrate that:

(1) the dimensions of the remaining exit opening are equivalent to or greater than those of a Type IV emergency exit; and

(2) the obstructing item does not protrude into the horizontally projected opening of the remaining exit.

In the assessment of the effectiveness of the remaining exit, the requirements of CS 25.807(a)(4), CS 25.809(b) and CS 25.813(c)(1) should also be considered.

Create a new AMC to Appendix S, S25.20(b)(1) as follows:

AMC to Appendix S, S25.20(b)(1)

Ensuring removal of in-flight obstructions before take-off and landing

This paragraph provides guidelines regarding the criteria under which an item, although constituting an obstruction that does not comply to CS 25.813(c), may be considered acceptable because per design and procedure, there can be high confidence that the obstruction will be removed when needed for safety (S25.20(b)(1)).

In addition to the exceptions set in Section 2 — Deployable features of AMC 25.813(c), an item which can be deployed by a crew member or passenger into the region defined by CS 25.813 (c)(4)(i) or into the passageway required by CS 25.813 (c)(1), (2) or (3), but

which, when stowed, is no longer in either of these areas, is acceptable if there is enough assurance that the item will be stowed when needed. Such assurance may be assumed when all following conditions are met:

(1) A position monitoring system is installed, which detects that the item is not properly stowed, and triggers both alerts in the passenger cabin and a visual indication to the flight crew if the item is not properly stowed before entering any of the taxiing, take-off, approach, and landing phases.

(2) The alerts in the cabin, required in paragraph (1), include an aural device which sounds continuously in all areas of the passenger cabin (it should be loud enough to clearly act as an irritant, thus assuring that occupants will stow the obstruction, but not so loud as to distract the flight crew), as well as a conspicuous electrically illuminated sign showing an appropriate text message or pictogram, in the immediate proximity of the relevant emergency exit.

(3) The alerts described in paragraph (2), are triggered without delay if the deployable item is moved away from the safe position during any of the taxiing, take-off, approach, and landing flight phases, or, if upon entering these phases, the item is not stowed in the safe position. When preparing for landing, the alerts are triggered at a point that allows ample time for a cabin occupant to re-stow the deployable item before landing. It should be considered that the cabin occupant needs to move within the cabin to reach the deployable item, therefore, the alerts should be triggered during descent, allowing enough time prior to entering the approach phase, unless the aeroplane is required to have at least one cabin crew member on board; The aural and visual alerts should both remain on until the obstacle is properly stowed.

(4) The visual indication provided to the flight crew, described in paragraph (1), is triggered without delay if the deployable item is moved away from the safe position during any of the taxiing, take-off, approach, and landing flight phases, or, if upon entering these phases, the deployable item is not stowed in the safe position. When preparing for landing, the visual indication is triggered during the descent phase, early enough to enable the crew to take appropriate action before entering the approach phase.

(5) The failure to alert in the cabin or cockpit that an item is not properly stowed is demonstrated to have an average probability per flight hour of the order of 1×10^{-3} or less.

(6) Instructions are given to the passengers and cabin crew (if any), by means of appropriate placards and a pre-flight briefing, that the obstacle should be stowed before entering any of the taxiing, take-off, approach, and landing phases. The pre-flight briefing (which could be part of a regular briefing) should describe the position monitoring and alerting system, as well as the necessary response by the passengers. The requirement for this briefing should be part of the AFM.

(7) A description of the position monitoring and alerting system is made available to the flight crew. The AFM should also include the appropriate normal procedure ensuring

that the cabin is ready (i.e. a check that no visual indication, as defined in paragraph (4), being present) prior to landing, and an instruction that the crew takes all necessary actions when the visual indication, as defined in paragraph (4), is triggered.

(8) The emergency exit provided when the obstruction in its most adverse position(s) is at least as effective as a Type IV emergency exit, unless it can be shown that following any single failure an exit at least as effective as a Type IV emergency exit can be obtained by simple and obvious means. If the obstructing item is a seat, the normal seat operating controls (e.g. track, swivel, recline etc.) may be considered as means meeting the simple and obvious requirement, provided that the controls remain visible to a person approaching the seat and are easily useable without sitting on the seat, when the seat is in any possible obstructing condition. If movement of the obstructing item to meet the above requires electrical power, it should be substantiated that the required power source(s) will remain available following an emergency landing.

Create a new AMC to Appendix S, S25.20(b)(2) as follows:

AMC to Appendix S, S25.20(b)(2)

Comparative assessment of evacuation capability

Use of the latin square method as detailed in Appendix 4 to the FAA Advisory Circular 25-17A Transport Airplane Cabin Interiors Crashworthiness Handbook, dated 05/18/09 is accepted by EASA as providing acceptable means of compliance to S25.20(b)(2).

Create a new AMC to Appendix S, S25.30(a) as follows:

AMC to Appendix S, S25.30(a)

Width of Aisle

For compliance with the 'Width of Aisle' requirement, the following applies:

(1) An obstacle in the passageway is considered easily surmountable if the aisle width reduction it creates may be negotiated by a person anywhere in the size range from 5th percentile female to a 95th percentile male.

(2) Negotiating of an obstacle may require the removal and/or movement of more than one item.

(3) If an obstacle is stepped on, it should be capable of withstanding without failure a vertical step force of 222 daN (500 lbs) applied at the most adverse stepping location, without failure to the extent that it could unsteady a person trying to surmount that obstacle.

(4) When assessing compliance, the applicant should select the most adverse in-flight configuration(s). The selection should include all possibilities regardless of subjective issues, such as the likelihood that passengers may consider the configuration advantageous. If however, an applicant feels that one or more configurations, although possible, would only result from

severely anomalous behaviour by cabin occupants, it/they may be justified for elimination from the assessment. The configuration(s) should be highlighted and their elimination justified in the assessment report, for Agency agreement. The possibility of entrapment (e.g. feet, hands etc.) during negotiating of the obstacle should be included in the assessment and selection of adverse in-flight configurations. Maintaining gaps of less than 3.5 cm (1.38 in.) is considered acceptable to eliminate the risk of entrapment. Items such as drawers or stowage doors do not need to be considered opened in the aisle. Each interior door may be considered open unless another position of the door might interact with the movement of an obstacle out of the aisle. In that case, all possible interactions between the door and the obstacle should be assessed. In general, items need only be considered in their most adverse detent or locked position.

(5) For the purpose of showing compliance, the applicant may use tests, analyses supported by test data, or, where appropriate, inspections.

(6) In principle, the total time required for a crew member to travel from the forwardmost point in the cabin to the rearmost point, with all aisle obstacles in their most adverse positions, should not exceed by more than 30 seconds the time it would take without the obstacles in place. However, the cabin may be divided into zones, provided that each zone includes the quantity and type of emergency equipment adequate for firefighting, and that it can be substantiated that at least one cabin crew member is likely to occupy that zone during the majority of the flight. It should be shown that the time required for a cabin crew member to travel from the forwardmost point to the rearmost point of each zone, with all aisle obstacles in their most adverse positions, will not exceed by more than 30 seconds the time it would take without the obstacles in place.

(7) If an unobstructed passageway exists as an alternative to the obstructed one (e.g. aeroplanes with two aisles), it may be acceptable for this alternative route to be used when showing compliance. Such acceptability will depend on a case-by-case assessment of the degree to which such an alternative route would be obvious to the crew member.

Note: interior doors are not addressed by the requirements of S25.30(a) but rather by the requirements of S25.10(a) and (b).

Create a new AMC to Appendix S, S25.30(b) as follows:

AMC to Appendix S, S25.30(b)

Firm Handholds

Where the cabin layout is similar to a standard airline layout, firm handholds as normally expected for such seating areas should be provided.

Where closely spaced firm handholds cannot be easily provided, the 'Firm Handholds' requirement can be considered as complied with, provided the following conditions are met:

(1) there should be a recommendation to passengers to remain seated with seat belts fastened, which may be a placard or a required (i.e. specified in the AFM) pre-flight briefing;

(2) there should be at least one route through each area that provides firm handholds to enable passengers to reach their designated seats; in these areas:
(a) firm handholds should be mounted at least 66 cm (26 in.) high; and
(b) the distance between firm handholds should not be greater than 2.15 m (84 in.);

(3) wherever aisles are not bordered by seats, it is acceptable that occupants may steady themselves by leaning on sidewalls or other interior components; and

(4) in any case, the applicant shall demonstrate that items used as firm handholds are structurally adequate to perform this function.

Create a new AMC to Appendix S, S25.40(b) as follows:

AMC to Appendix S, S25.40(b)

Briefing Card Placard

The instructions that may be reported on the briefing card referred to in S25.40(b) are limited to the instructions necessary to restore the configuration of the passenger cabin to that approved for taxiing, take-off, and landing. All other placards required by CS-25 are excluded from the provisions of S25.40(b).

For example, and where applicable, a briefing card may be used to deliver information related to setting seats in the upright position, stowing leg rests/armrests, repositioning 'high–low' position tables, opening/closing doors, installing crash pads, etc.

The content added to the briefing card to cover information conventionally conveyed via placarding, and the means to provide accessibility to this information will need to be approved as part of the type design. However, it may be desired to include additional safety information on the same briefing card. This may be due to operational requirements for a briefing card, or may be at the applicant's or customer's discretion. This is acceptable, and this additional information will not be subject to approval as part of the type design.

However, limitations on the presentation of this additional information on the briefing card (e.g. size, style, relative location) may need to be stated in the type design in order that both sets of information remain appropriately conspicuous to the passengers.

When design solutions are proposed using placards that make reference to a briefing card for further instructions, the following should be considered:

(1) Individual placards at each seat location may be replaced by a simplified placard referring to the briefing card. For example: 'Refer to the briefing card to configure cabin/seat/table/leg rest for taxiing, take-off, and landing'.

(2) Alternatively, one single placard stating, for example, 'Moveable items in this area should be configured in accordance with the briefing card for taxiing, take-off, and landing', and visible from each seated position of a group of seats, may be used.

(3) The briefing card should be demonstrated to be accessible from each passenger seat. A dedicated stowage (e.g. pocket) easily recognisable by a seated passenger, or when approaching the seat, shall be provided. The briefing card should be within easy reach of each passenger with their seat belt fastened, except in some cases where this may be impracticable. For instance, it may be acceptable that a passenger occupying the centre place of a three-place divan is not able to reach the briefing card with their seat belt fastened. In such a case, EASA may accept that either the left hand (LH) or right hand (RH) place of the divan will most likely be occupied, and that this passenger's access to the briefing card will provide him/her with the required awareness of necessary pre-flight and landing actions.

(4) The briefing card information should be clear and simple. It is expected that the additional space offered by the briefing card, relative to conventional placarding, will allow applicants to provide more easily understood safety instructions. The use of pictograms is encouraged.

Create a new AMC to Appendix S, S25.40(c) as follows:

AMC to Appendix S, S25.40(c)

Seats in Excess

S25.40(c) requires the installation of a placard, adjacent to each possible passenger boarding door, on aeroplanes which have a greater number of seats approved for occupancy during taxiing, take-off, and landing than the approved passenger seating configuration. It may be acceptable that the selection of which seats to occupy is at the operator's/passenger's discretion, or constraints may exist for instance due to the zonal limitations set by S25.1(a)(2), or the varying passenger seating configuration and/or direct-view limitations for an aeroplane with different, reconfigurable, cabin designs approved for private versus commercial transport operations. In such cases, the placard should indicate limitations of the allowable seating occupancy for taxiing, take-off, and landing, as appropriate, for each cabin zone, and not just for the aeroplane as a whole; moreover, different indications should be provided with reference to the different type of operations that may be performed (non-commercial/commercial).

Additionally, if it is decided to help passengers in selecting acceptable seating locations by means of markings on a seat or seats, a local placard (text or symbolic), easily readable by a passenger approaching/seated on each such seat, should be provided. The placard should be of adequate size for easy readability.

Create a new AMC to Appendix S, S25.50(b) as follows:

AMC to Appendix S, S25.50(b)

Cabin Attendant Direct View

For commercial operations, compliance with CS 25.785(h)(2) may be shown based on the criteria of FAA AC 25.785-1B, *Flight attendant seat and torso restraint system installations*, 11 May 2010, with the following deviations from Section 10 thereof:

(1) Subparagraph 10a(2) is amended to read as follows:

'(2) Each floor level emergency exit adjacent to a required crew member seat';

(2) Subparagraph 10a(3) is amended to read as follows;

'(3) At least 50 % of the total number of passenger seats authorised for occupancy during taxiing, take-off, and landing.';

(3) Subparagraph 10a(4) is amended to read as follows:

'(4) At least 25 % of the passenger seats in each visually divided zone of four or more passenger seats.'; and

(4) Subparagraph 10b(3)(a) is amended to read as follows:

'(a) A person seated in the seat is visible when they make any upper-body movement, such as moving their arm over their head or sideways, including leaning, while belted on their seat.'.