CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 1. CERTIFICATION PROCEDURE FOR ROTORCRAFT AVIONICS EQUIPMENT.

a. Pre-Test Requirements.

(1) General. This test guideline has been prepared as an aid in the evaluation of rotorcraft avionics (aviation electronics) equipment installations. The criteria presented are not to be considered exclusive, but are offered as one method of evaluating design practice and performance. The testing and qualification of an electronic installation should be considered as consisting of three phases: preinstallation, ground, and flight. The amount of testing necessary during each phase will vary with the amount of testing performed on previous phases. For example, if a system is TSO'd, the preinstallation performance is probably substantiated and therefore the ground and flight testing can be reduced accordingly. Also, a thorough ground testing program should result in reduction in necessary flight testing. When the operating or airworthiness regulations require a system to perform its intended function, the use of TSO'd equipment or the submission of data substantiating the equipment performance is strongly recommended.

(2) Regulatory References. Sections 29.1301, 29.1309, 29.1329, 29.1331, 29.1333, 29.1355, 29.1431, 29.1457.

(3) System Design. Systems or equipment presented for installation approval, when not qualified by TSO or other approval means, should be accompanied by sufficient data to substantiate their design acceptability.

(i) Operation of Controls. The operation of controls intended for use during flight, in all possible position combinations and sequences, should not result in a condition that would be detrimental to the continued safe performance of the system.

(ii) Electrical Shock. Systems should be designed so that under all probable conditions the risk of dangerous electrical shock is minimized.

(iii) Fire Hazard. The design of the system should be such that all components meet the applicable fire and smoke protection requirements of §§ 29.853 and 29.863. Cables and equipment to be installed in designated fire zones, that are used during emergency procedures should be at least fire resistant.

(iv) Plugs and Cables. Connector pins for sensitive signal circuits should not be adjacent to pins used for AC power circuits. When redundant wiring is used to

comply with the systems independence regulations such as §§ 29.1331, 29.1333, or 29.1355, the wires should be routed through separate plugs and/or cables with as much physical separation as practicable. The system should be designed so that incorrect mating of plugs is not possible. Cable grounding and shielding techniques should be used to minimize electromagnetic interference.

(4) System Performance. Where the operating or airworthiness regulations require a system to perform its intended function, and when the equipment is not qualified by TSO or other approval means, performance data furnished to the FAA/AUTHORITY can reduce the installed performance testing. The appropriate TSO minimum performance standard may be used as a guide.

(i) Environment. An appropriate means for environmental testing is set forth in Radio Technical Commission for Aeronautics (RTCA) document DO-160. The applicant should submit test reports showing that the laboratory tested categories such as temperature, vibration, altitude, etc., are compatible with the environmental demands to be placed on the rotorcraft.

(ii) Failure Analysis. Procedures are contained in paragraph AC 29.1309, section c of this advisory circular.

(5) Installation Design

(i) Mechanical Installation. Installations should be made to (1) ensure compliance with the airworthiness regulations and (2) comply with the equipment manufacturer's recommendations. The designer should observe good engineering practices in specifying material type, thickness, fastener type, edge distance, and attachment to the equipment rack. By analysis or static tests the mounted equipment should be shown to withstand the inertia forces of §§ 29.561(b)(3) and 29.337. Refer to AC 43.13-2a for static test procedures.

(ii) Arrangement and Visibility. The mounting position of all instruments, switches, position labels, and control heads should make them plainly visible to the pilot while in his normal panel-facing position and under all cockpit lighting conditions likely to occur. TSO approval does not assure instruments will be acceptable in a particular cockpit installation or for all lighting conditions. The instruments, switches, and placarding must be free from reflections. Malfunction annunciation devices should be conspicuous and clearly visible to the pilot. (See Advisory Circular 20-69 and §§ 29.1321, 29.771, 29.1381, and 29.1555(a).)

(iii) Load Analysis.

(A) Power Sources. It should be determined whether the electrical power source capacity is adequate for the system installation under all foreseeable operating conditions including engine failure on multiengine rotorcraft. System load reductions should be applied or power source capacity increased if necessary to assure

compatibility between load and source. Duplicate systems should be powered from separate buses and, in some cases, from independent sources if required by the airworthiness regulations. (Sections 29.1309, 29.1331, 29.1333, 29.1351, or 29.1355.)

(B) Navigation Course Deviation Circuit Loading. It should be determined that the deviation circuit source impedance is matched by its load and that the source capacity is not exceeded. When the system is capable of transfer, the transfer loads should also be considered (§ 29.1301).

(C) Malfunction Indicator Circuit Loading. It should be determined that the malfunction indicator source impedance is matched by its loads and that the source capacity is not exceeded. When the system is capable of transfer, the transfer loads should also be considered (§ 29.1301).

(D) Synchro Signal Loading. When parallel loads are added to Synchro's, the manufacturers' specifications should be reviewed to assure that the additional loads do not result in an overloaded synchro.

(iv) Interface. In many cases, the mating units of a system are designed by different manufacturers. For example, a brand-X gyro may be designed for operation with a brand-X flight director, but later a modifier decides to operate a brand-Y autopilot with the brand-X gyro. This applies just as well to NAV receivers, AREA NAV units, course indicators, omni bearing selectors, tachometer indicators, transmitters and many other equipment items. When this is the case, the applicant should provide data, in summarized form, describing those characteristics such as impedance, volts, etc., that are necessary to assure a compatible and reliable system. The data should also reference the source of the interface data (§ 29.1301).

(v) Flight Tests. An FAA/AUTHORITY engineering flight test is required during type certification or after modification that changes the established limitations, flight characteristics or performance of a rotorcraft or any of its required systems or operating procedures. New installations of equipment in the cockpit or modifications that affect existing equipment in the cockpit should be evaluated by appropriate flight test personnel, if it is necessary to evaluate operational aspects of the change. Where possible, cockpit arrangement, placards, markings, instrument visibility, and light reflections can be evaluated on the ground if the applicant opts to darken the windows. Electromagnetic compatibility functional checks, windshield glare, and pilot workload evaluations may be conducted in flight at the FAA/AUTHORITY flight test pilot's option.

(vi) Radio Master Switches. Some installations incorporate radio master switches to control special busses for the avionics systems. If this capability is provided it should be evaluated to assure failure modes are not introduced that will result in excessive or even total loss of all required avionics. One switch that controls all required avionics is not considered acceptable for IFR installations. The evaluation should include an assessment of the loss of the systems to be included on the radio master switch(es), and the subsequent effect on continued safe flight.

b. Test Procedures. Where the airworthiness or operating regulations require a system to perform its intended function, and/or not create a hazard to other required systems, sufficient testing should be accomplished to assure satisfactory performance. When ground testing is not sufficient to properly evaluate a system's performance, flight testing should be accomplished. Acceptable flight test criteria for specific navigation and communication equipment are contained herein. If the rotorcraft is to be approved for IFR operations, the additional criteria of AC 29 Appendix B should be satisfied.

(1) VHF Systems.

(i) General. Intelligible communications should be provided between the rotorcraft and ground facilities throughout the airspace within 100 NM of an FAA/AUTHORITY ground facility from radio line of sight altitude to the maximum altitude for which the rotorcraft is certificated. Communication should be provided with the rotorcraft at or above line of sight altitude in right and left bank up to 10 degrees and on all headings.

(ii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify by observation that no adverse effects are present in the flight systems.

(iii) Antenna Measurement. If satisfactory antenna measurement data are provided, the following flight test may be reduced to checks in right and left turns in the vicinity of the predicted bearings of worst performance. If antenna locations are symmetrical, tests may be conducted using only one direction of turn.

(A) Long Range Reception. Starting at a distance of at least 100 NM from the ground facility antenna, perform a right and/or left 360 degree turn at a bank angle of at least 10 degrees. Communicate with the ground facility every 10 degrees of turn to test the intelligibility of the signals received at the ground station and in the rotorcraft. For 100 NM, the minimum line of sight altitude is approximately 7,000 feet.

(B) Approach Configuration. With the landing gear down and with the rotorcraft in the approach configuration (at a distance of 10 NM from the ground station and in an idle power descent toward the station), demonstrate intelligible communications between the rotorcraft and the ground facility.

(2) HF Systems.

(i) Acceptable communications should be demonstrated by contacting a ground facility at a distance of at least 100 nautical miles. Single sideband equipment should also perform acceptably in the amplitude modulation mode of operation.

(ii) It should be demonstrated that precipitation static is not excessive when the aircraft is flying at cruise speed (in areas of high electrical activity, including

clouds and rain if possible). Use the minimum amount of installed dischargers for which approval is sought.

(3) VOR Systems.

(i) These flight tests may be reduced if adequate antenna radiation pattern studies have been made and these studies show the patterns to be without significant holes (with the rotorcraft configurations used in flight, i.e., landing gear retracted en route and extended for approach). Particular note should be made in recognition that certain rotor RPM settings may cause modulation of the course deviation indication (rotor modulation). VOR performance should be checked for rotor modulation in both approach and en route operation while varying rotor RPM throughout its normal range.

(ii) The airborne VOR system should operate normally with warning flags out of view at all headings of the rotorcraft (in level flight) throughout the airspace within 100 NM of the VOR facility while flying above the radio line of sight altitude to within 90 to 100 percent of the maximum altitude for which the rotorcraft is certified.

(iii) The accuracy determination should be made such that the indicated reciprocals agree within 2 degrees. Tests should be conducted over at least two known points, on the ground, such that data are obtained in each quadrant. Data should correlate with the ground calibration and in no case should the absolute error exceed ±6 degrees. Fluctuation of the course deviation indication should not be excessive.

(A) En route Reception. Fly from a VOR facility, rated for high altitude, along a radial to a range of 100 NM. The VOR warning flag should not come into view, nor should there be deterioration of the station identification signal. The course width should be 20 degrees (±5 degrees tolerance, 10 degrees either side at the selected radial). If practical, perform en route segment on a doppler VOR station to verify the compatibility of the airborne unit. Large errors have been found when incompatibility exists.

(B) Long Range Reception. Perform a 360-degree right and a 360-degree left turn at a bank angle of at least 10 degrees at an altitude just above radio line of sight (see b(1)(iii)(A) for line of sight altitude) and at a distance of at least 100 NM from the VOR facility. Signal dropout should not occur as evidenced by the malfunction indicator appearance. Dropouts that are relieved by a reduction of bank angle at the same relative heading to the station are satisfactory. The VOR identification should be satisfactory during the left and right turns.

(C) En route Station Passage. Verify that the To-From indicator correctly changes as the rotorcraft passes through the cone of confusion above a VOR facility.

(4) Localizer Systems.

(i) Flight test requirements may be modified to allow for adequate antenna radiation pattern measurements as discussed under VOR paragraph b(3)(i) flight test.

(ii) The signal input to the receiver presented by the antenna system should be of sufficient strength to keep the malfunction indicator out of view when the rotorcraft is in the approach configuration and at least 10 NM from the station. This signal should be received for 360 degrees of rotorcraft heading at all bank angles up to 10 degrees left or right at all normal pitch altitudes, and at an altitude of approximately 2,000 feet.

(iii) The deviation indicator should properly direct the aircraft back to course when the rotorcraft is right or left of course.

(iv) The station identification signal should be of adequate strength and sufficiently free from interference to positive station identification, and voice signals should be intelligible with all electric equipment operating and pulse equipment transmitting.

(v) Localizer performance should be checked for rotor modulation in approach while varying rotor RPM throughout its normal range.

(A) Localizer Intercept. In the approach configuration and a distance of at least 10 NM from the localizer facility, fly toward the localizer front course, inbound, at an angle of at least 50 degrees. Perform this maneuver from both left and right of the localizer beam. No flags should appear during the time the deviation indicator moves from full deflection to oncourse. If the total antenna pattern has not been shown by ground checks or by VOR flight evaluation to be adequate, additional intercepts should be made.

(B) Localizer Tracking. While flying the localizer inbound and not more than 5 miles before reaching the outer marker, change the heading of the rotorcraft to obtain full needle deflection. Then fly the rotorcraft to establish localizer on course operation. The localizer deviation indicators should direct the rotorcraft to the localizer on course. Perform this maneuver with both a left and a right needle deflection. Continue tracking the localizer until over the transmitter. At least three acceptable front course and back course flights should be conducted to 200 feet or less above threshold.

(5) Glide Slope Systems.

(i) Flight Test. The signal input to the receiver should be of sufficient strength to keep the warning flags out of view at all distances to 10 NM from the facility. This performance should be demonstrated at all aircraft headings from 30 degrees left to 30 degrees right of the localizer course. The deviation indicator should properly direct the aircraft back to path when the aircraft is above or below path. Interference with the navigation operation should not occur with all rotorcraft equipment operating

and all pulse equipment transmitting. There should be no interference with other equipment as a result of glide slope operation.

(ii) Glide Slope Intercept. While flying the localizer course inbound in level flight, intercept the glide slope below path at least 10 NM from the station. Observe the glide slope deviation indicator for proper crossover as the aircraft flies through the glide path. There should be no flags from the time the needle leaves the full scale fly-up position until it reaches the full scale fly-down position.

(iii) Glide Slope Tracking. While tracking the glide slope, maneuver the aircraft through normal pitch and roll attitudes. The glide slope deviation indicator should show proper operation with no flags. At least three acceptable approaches to 200 feet or less above threshold should be conducted.

(iv) Interference. With all rotorcraft electrical equipment operating and all pulse equipment transmitting, determine that there is no interference with the glide slope operation (some interference from the VHF may be acceptable), and that the glide slope system does not interfere with other equipment.

(v) Glide slope performance should be checked for rotor modulation during the approach while varying rotor RPM throughout its normal range.

(6) Marker Beacon System.

(i) The marker beacon annunciator light should be illuminated for a period of time representing 2,000 to 3,000 feet distance when flying at an altitude of 1,000 feet as it passes over a marker beacon (see table below).

Altitude = 1,000 feet (AGL)

Ground Speed Light Time (Seconds)

Knots	2,000 feet	3,000 feet
90	13	20
110	11	16
130	9	14
150	8	12

(ii) The audio signal should be of adequate strength and sufficiently free from interference to provide positive identification.

(iii) Technical: Approach the markers at a ground speed of 130 knots and at an altitude of 1,000 feet above ground level. While passing over the outer and middle markers with the localizer deviation indicator centered, the annunciators should be

illuminated for a period of 9 to 14 seconds. Check for acceptable intensity of the indicator lights in bright sunlight and at night. For slower rotorcraft, the interval should be proportionately longer.

NOTE: It is recognized that the normal altitude at the middle marker is on the order of 150 to 200 feet. Due to variations in both glide slope angle and position of the middle marker in relation to the runway, the on glide path marker width will vary considerably which in turn will give a widely varying light time. Therefore, the more clearly defined criteria at 1,000 feet altitude should be used for quantitative testing of the middle marker function.

(7) Automatic Direction Finding Equipment (ADF).

(i) Range and Accuracy. The ADF system installed in the rotorcraft should provide operation with errors not exceeding 5 degrees and the aural signal should be clearly readable up to the distance listed for any one of the following types of radio beacons:

(A) 50 NM from an H facility (transmitter power 50-2,000 watts).

(B) 25 NM from an MH facility (transmitter power less than 50 watts).

(C) 15 NM from a compass locator (transmitter power less than 25 watts).

(ii) Needle Reversal. The ADF indicator needle should make only one 180-degree reversal when the rotorcraft flies over a radio beacon. This test should be made both with and without the landing gear extended.

(iii) Indicator Response. When switching stations with relative bearings differing by approximately 175 degrees, the indicator should indicate the new bearing within ±5 degrees within 10 seconds.

(iv) Antenna Mutual Interaction. For dual installations, there should not be excessive coupling between the antennas.

(v) Technique.

(A) Range and Accuracy. Tune in a number of radio beacons spaced throughout the 200 - 415 kH range and located at distances near the maximum range for the beacon (see (a) Range and Accuracy). The identification signals should be clear and the ADF should indicate the approximate direction to the stations. Beginning at a distance of at least 15 NM from a compass locator in the approach configuration, fly inbound on the localizer front course and make a normal ILS approach. Evaluate the aural identification signal for strength and clarity and the ADF for proper performance with the receiver in the ADF mode. All electrical equipment on the aircraft should be operating and all pulse equipment should be transmitting. Fly over a ground check

point with relative bearings to the facility of 0, 45, 90, 135, 180, 225, 270, and 315 degrees. The indicated bearings to the station should correlate within 5 degrees.

(B) Needle Reversal. Fly the aircraft over an H, LOM, or LMM facility at an altitude of 1,000 to 2,000 feet above ground level. The indicator needle should make only one reversal.

(C) Indicator Response. With the ADF indicating station dead ahead, switch to a station having a relative bearing of approximately 175 degrees. The indicator should indicate within ±5 degrees of the bearing in not more than 10 seconds.

(D) Antenna Mutual Interaction. If the ADF installation being tested is dual, check for coupling between the antennas by using the following procedure.

(1) With #1 ADF receiver tuned to a station near the low end of the ADF band, tune the #2 receiver slowly throughout the frequency range of all bands and determine whether the #1 ADF indicator is adversely affected.

(2) Repeat (A) with #1 ADF receiver tuned to a station near the high end of the ADF band.

(8) Distance Measuring Equipment (DME).

(i) The DME system should:

(A) Continue to track without dropouts when the rotorcraft is maneuvered throughout the air space within 100 NM of the VORTAC station and at altitudes from the radio line of sight to the maximum altitude for which the rotorcraft is certificated. This tracking standard should be met with the rotorcraft in the cruise configuration, at bank angles up to 10 degrees, climbing and descending at normal maximum climb and descent attitude, and orbiting a DME facility.

(B) Provide clearly readable identification of the DME facility.

(C) DME operation should not interfere with other systems aboard the rotorcraft (some interference with the transponder may be acceptable) and DME operation should not be adversely affected by other equipment.

(D) DME Hold.
The DME should continue to operate and track when DME Hold is activated and the channel switch is varied.

(E) DME Override. When an override switch is provided, proper operation should be demonstrated.

(ii) Technique.

(A) Climb and Maximum Distance. Determine that there is no mutual interference between the DME system and other equipment aboard the rotorcraft. Beginning at a distance of at least 10 NM from a DME facility and at an altitude of 2,000 feet above the DME facility, fly the rotorcraft on a heading so that the aircraft will pass over the facility. At a distance of 5 to 10 NM beyond the DME facility, operate the rotorcraft at its normal maximum climb attitude up to an altitude of 7,000 feet maintaining the aircraft on a station radial (within 5 degrees). The DME should continue to track with no unlocks to a range of 100 NM. Record the maximum altitude flown.

(B) Long Range Reception. Perform two 360 degree turns, one to the right and one to the left, at a bank angle of 8 to 10 degrees at least 100 NM from the DME facility. A single turn will be sufficient if the antenna installation is symmetrical. There should be no more than one unlock not to exceed one search cycle (maximum 35 seconds) in any 5 miles of radial flight.

(C) Penetration. From an altitude of above 7,000 feet (AGL) perform a let-down directly toward a ground station (DME facility) at a normal maximum rate of descent so as to reach an altitude of 5,000 feet above the DME facility 5 to 10 NM before reaching the DME facility. The DME should continue to track during the maneuver with no unlocks.

(D) Approach. Make a normal approach to land at a field with a DME located on the airport. The DME should track without an unlock (station passage excepted).

(E) DME Hold. With the DME tracking, activate the DME hold function. Change the channel selector to a localizer frequency. The DME should continue to track on the original station.

(9) Transponder Equipment.

(i) Performance Criteria. The ATC transponder system should furnish a strong and stable return signal to the interrogating radar facility when the rotorcraft is flown in straight and level flight throughout the air space within 100 NM of the radar station from radio line of sight to within 90 to 100 percent of the maximum altitude for which the rotorcraft is certificated. The airborne system should be controllable so that objectionable ring-around, spoking and clutter will not persist. The transponder system should not interfere with other systems aboard the rotorcraft and other equipment should not interfere with the operation of the transponder system (some interference from DME operation may be acceptable). When the rotorcraft is flown in the following maneuvers within the air space described above, the dropout time should not exceed 20 seconds.

(A) In turns at bank angles up to 10 degrees.

(B) Climbing and descending at normal maximum climb and descent attitude.

(C) Orbiting a radar facility.

(ii) Technique.

(A) Climb and Distance Coverage: Beginning at a distance of at least 10 NM from and at an altitude of 2,000 to 3,000 feet above that of the radar facility and using a transponder code assigned by the ARTCC, fly on a heading that will pass the rotorcraft over the facility. At a distance of 5 to 10 NM beyond the facility, operate the rotorcraft to maintain an altitude above radio line of sight while maintaining the aircraft at a heading within 5 degrees from the radar facility to 100 NM from the radar facility.

(B) Communicate with the ground radar personnel for evidence of transponder dropout. During the flight, check the "ident" mode of the ATC transponder to assure that it is performing its intended function. Determine that the transponder system does not interfere with other systems (except possibly the DME) aboard the rotorcraft and that other equipment (except possibly the DME) do not interfere with the operation of the transponder system. There should be no dropouts, that is, when there is no return for two or more sweeps. The operation of the ATC transponder should be verified over the station, at 25 NM, and at 100 NM.

(C) Long Range Reception. Perform two 360-degree turns, one to the right and one to the left, at bank angles of 8 to 10 degrees with the flight pattern at least 100 NM from the radar facility. During these turns, the radar display should be monitored and there should be no signal dropouts (two or more sweeps).

(10) Weather Radar Equipment.

(i) Bearing Accuracy. The indicated bearing of objects shown on the display should be within 5 degrees of their actual magnetic bearing within the sectors 40 degrees right and left of the aircraft longitudinal axis. Beyond 40 degrees right and left, bearing accuracy should be ±10 degrees.

(ii) Distance of Operation. The radar should be capable of displaying prominent targets throughout the distance and angular range of the display.

(iii) Antenna Stabilization. When antenna stabilization is provided, it should eliminate blurring of the display for the ranges of pitch and roll for which it is designed.

(iv) Beam Tilting. The radar antenna should be installed so that its beam is adjustable to any position between 10 degrees above and 10 degrees below the plane of rotation of the antenna.

(v) Technique.

(A) Bearing Accuracy. Fly under conditions which allow visual identification of a target, such as an island, a river, or a lake, at a range within 10 percent of the maximum range of the radar. When flying toward the target, select a course that will pass over a reference point from which the bearing to the target is known. When flying a course from the reference point to the target determine the error in displayed bearing to the target on all range settings. Change heading in increments of 10 degrees and determine the error in the displayed bearing to the target.

(B) Contour Display (Iso Echo). If heavy cloud formations or rainstorms are reported within a reasonable distance from the test base, select the contour display mode. The radar should differentiate between heavy and light precipitation. In the absence of the above weather conditions, determine the effectiveness of the contour display function by switching from normal to contour display while observing large objects of varying brightness on the indicator. The brightest objects should become the darkest when switching from normal to contour mode.

(C) Stability. While observing a target return on the radar indicator, turn off the stabilizing function and put the aircraft through pitch and roll movements. Observe the blurring of the display. Turn the stabilizing mechanism on and repeat the roll and pitch movements. Evaluate the effectiveness of the stabilizing function in maintaining a sharp display.

(D) Ground Mapping. Fly over areas containing large, easily identifiable landmarks such as rivers, towns, islands, coastlines, etc. Compare the form of these objects on the indicator with their actual shape as visually observed from the cockpit.

(E) Mutual Interference. Determine that no objectionable interference is present on the radar indicator from any electrical or radio/navigational equipment when operating, and that the radar installation does not interfere with the operation of any of the rotorcraft's radio/navigational systems.

(11) Area Navigation. Advisory Circular 90-45A is the basic criteria for evaluating an area navigation system, including acceptable means of compliance to the FAR.

(12) Inertial Navigation. Advisory Circular 25-4, Inertial Navigation Systems, is the basic criteria for the engineering evaluation of an inertial navigation system (INS) and offers acceptable means of compliance with the applicable Federal Aviation Regulations which contain mandatory requirements in an objective form. The engineering evaluation of an INS should also include awareness of Advisory Circular 121-13, Self-Contained Navigation Systems (Long Range), which presents criteria to be met before an applicant can get operational approval. For flights up to 10 hours, the radial error should not exceed 2 nautical miles per hour of operation on a 95 percent statistical basis. For flights longer than 10 hours, the error should not

exceed ±20 NM crosstrack or ±25 NM along track error. A 2-nautical-mile radial error is represented by a circle, having a radius of 2 nautical miles, centered on the selected destination point.

(13) Doppler Navigation. Doppler navigation system installed performance should be evaluated in accordance with Advisory Circular 121-13. (See FAR 121, Appendix G).

(14) Radio Altimeters. Radio altimeter system installed performance should be evaluated in accordance with RTCA Document DO-123, Appendix A, Part II.

(15) Emergency Locator Transmitters (ELT).

(i) Emergency locator transmitter performance should be evaluated in accordance with TSO-C91. ELT installations should be examined for potential operational problems. There have been numerous instances of interaction between ELT and other VHF installations. ELT antenna installations in close proximity to other VHF antennas should be suspect. Antenna patterns of previously installed VHF antennas should be measured after an ELT installation. Some problems caused by ELT installations are:

(A) Loss of radiated power from VHF communications.

(B) Reradiation of VHF transmitter energy such that navigation crosspointers are affected.

(C) Reception of FM broadcast, at high level, in VHF communications.

(D) Inadvertent activation of the ELT by VHF transmitted energy. (See AD 72-22-3.)

(ii) ELT Installation. TSO-C91 specifies that the ELT be automatically activated when subjected to a force of 5.0 (+2,-0)g in the direction of the longitudinal axis of the aircraft. This recommendation for mounting is considered satisfactory for rotorcraft. In recognition of the significant vertical impact velocity that rotorcraft commonly have an optional placement of the ELT pitched down 30° from the horizontal axis of the rotorcraft is also satisfactory.

(16) Audio Interphone Systems. Acceptable communications should be demonstrated for all audio equipment including microphones, speakers, headsets, and interphone amplifiers. All modes of operation should be tested, including operation during emergency conditions (i.e., emergency descent, and oxygen masks) with all rotorcraft engines running, all rotorcraft pulse equipment transmitting, and all electrical equipment operating.

(17) Portable Battery Powered Megaphones (AC 121-6). Megaphone performance should be evaluated in accordance with AC 121-6.

(18) Omega and Omega/VLF Navigation Systems. Omega and Omega/VLF Navigation systems should be evaluated in accordance with the following AC's that apply to the type of approval requested:

(i) AC 120-37, Approval of Omega Systems, as a sole means of overwater long range navigation.

(ii) AC 120-31A, Approval of Airborne Omega Navigation Systems, as a means of updating self-contained navigation systems.

(iii) AC 20-101B, Approval of Omega and Omega/VLF Navigation.

(19) Rotorcraft Condition Monitoring System Installations.

(i) General. Avionics equipment and systems are being installed in rotorcraft to collect data to be used in assessing engine/rotorcraft performance and frequency of maintenance. Some of the items monitored are engine operating exceedances, hot starts, power assurance, and cycle counts. The monitoring systems being addressed by this paragraph are those used to collect data for maintenance purposes not those monitors which are utilized as part of the control system for autopilot/flight controls or engine controls. At present, optional approvals are being requested for most of these systems not performing any required functions. However, most of the applicants anticipate requesting approval for the systems to be used in the future to perform some required function or to allow required maintenance to be predicated on the operation of the system. This consideration becomes particularly important if the system is software based. A further discussion of system software is included in paragraph AC 29 MG 1 b(19)(iii)(B).

(ii) System Installation. The system installation should be shown to be free from hazards considering both normal operation and possible malfunctions. Malfunctions which might be caused by software errors are discussed under paragraph AC 29 MG 1 b(19)(iii)(B). The accuracy and response of the monitoring device/system should be sufficient to allow the operational and maintenance personnel to relate the data obtained to required maintenance actions. The exceedance (engine limit) information being acquired by these systems is or will be used in place of information previously acquired from field reports of operational personnel utilizing the basic aircraft instruments. In this case, the automated system will generally produce results which are more accurate than the basic aircraft instruments. However, in this circumstance, it is not appropriate to require the monitor system to be more accurate than the previously approved methods used to provide the required exceedance data. If the data collected by the system require filtering prior to use, it is equally acceptable to accomplish this filtering either as the data are being acquired (airborne function) or

when the data are analyzed (ground based function) and used in the maintenance of the rotorcraft.

(iii) System Components.

(A) Hardware. The hardware of the system when operating under the control of the imbedded software should be shown to comply with § 29.1301. Additionally, in showing compliance to § 29.1309(a), laboratory testing to the appropriate portions of the latest revision of RTCA Document DO-160 should be performed.

(B) Software. If the function of the monitor system depends on embedded airborne software to determine all or part of its functioning, Document DO-178 is the recommended standard to be used for the approval of the system software. A further discussion of the use of this document is included in paragraph AC 29.1309. The selection of the software level should be carefully considered because system approval is sometimes initially sought on the basis of the system being a non-required optional system. If it has further been shown that no dependence is made on the system software to preclude a hazardous failure mode, then a low software level would be acceptable. However, it is very difficult to qualify software to higher levels of "quality" once the software has been initially certified. Because of this, it is recommended that the software be chosen to the level consistent with the ultimate use to which approval of the system is planned. If the system is to be approved only as non-required optional equipment, then the choice of a low level of software qualification may be appropriate. However, when more experience is gained with the operation of the system, and it is ultimately planned to seek approval to perform required functions, then an appropriate higher level of software should be initially obtained.

NOTE: Extensive service experience should not be considered as a basis for level of criticality without accomplishing RTCA DO-178 procedures.

(20) Night Vision Goggles (NVG).

(i) Background. Night vision goggles (NVG) have been used by U.S. military pilots since the early 1970's. The first units (first generation or GEN I) were constructed from the rifle "Sniper-Scopes." These units did not provide much light amplification. The second generation (GEN II) were still primarily designed for ground use. Second generation high performance units (military designation AN/PVS-5C) had some consideration for flight use but were still lacking in several aspects. A light level of at least a quarter moon well above the horizon was required for operation of these NVG. At first the normally helmet-mounted units covered the pilot's entire upper face and the pilot could only see through the NVG. In order to protect the light amplification system these NVG had an automatic shutoff feature when brighter than relatively low levels of light were encountered. Normal incandescent and especially red incandescent lights would cause these NVG to shut down. Aircraft cockpit lights, especially the red warning lights, would cause "blooming" (an increased brightness of all or portions of the NVG

field of view with the disappearance of the “picture” in that area) or a total shutdown of the NVG. Military aircraft cockpits and lighting systems were significantly modified to avoid this problem. In the late 1980’s the military pushed technology for better and aircraft compatible NVG. Third generation (GEN III, military designation ANVIS or AN/AVS-6) NVG systems became available about 1988. These systems require only star light for satisfactory operation.

(ii) Procedures. As of January 1990, no approvals for civil rotorcraft operations with NVG have been issued. Since NVG are not installed in the rotorcraft, they are not required to be approved as part of the type design. However, since an operational approval would be required for use of NVG, they should meet some acceptable performance standard. The minimum standard recommended is the GEN III NVG. The performance of these NVG are rated as their spectral response to irradiated light sources, measured as density of incident photons per square meter. Third generation, AN/AVS-6, NVG have been evaluated for compatibility with a limited number of rotorcraft and were generally found to be usable during en route operations with no cockpit lighting systems modifications. It is anticipated, however, that some aircraft may require significant modifications to the existing cockpit lighting system. The FAA/AUTHORITY policy is that modification of the cockpit to a non-compliant configuration to accommodate NVG use is not acceptable. For instance, alteration of the required red warning annunciators to some other color is not acceptable. Since individual rotorcraft may have been modified with additional lights or systems, each rotorcraft being considered for use with NVG should be evaluated by an FAA/AUTHORITY representative during a night flight. If it is anticipated that cockpit lighting system modifications will be required to achieve an adequate level of NVG compatibility FAA/AUTHORITY involvement should be arranged as soon as possible. Preferably this evaluation flight would be made with two pilots or a pilot and safety observer, over a known area, where all the aircraft and cockpit lights are operated and their effect on the NVG determined. Reflections of landing or searchlights on windshields or other glass during approach or landing may affect NVG and may impose a minimum altitude restriction for use of NVG. Failure of the NVG should be evaluated during any critical flight phase.

Note that the above discussion is purposely limited in scope. Issues such as crew training and operating limitations would have to be addressed in detail to obtain an operational approval.

(21) Rotorcraft Health and Usage Monitoring Systems (HUMS).

(i) General. HUMS can be divided into two major categories: Health Monitoring Systems and Usage Monitoring Systems. The provisions of § 29.1301 are used to determine that the system performs its intended function. The provisions of § 29.1309(a) and (b) are used to look at the impact of environmental conditions and malfunctions. To date (mid-1990) HUMS have not been approved to replace service life or other specific physical limits but several systems are now in the process of seeking approval. Health monitoring systems are considered to be the serious applications of

this technology, and it will probably be some time before the necessary data base to allow full reliance on this technology is available. There have been numerous approvals of usage monitoring systems as optional equipment, and a good example of this technology is a condition monitoring system described in paragraph AC 29 MG 1b(19) above.

(ii) Health Monitoring Systems.

(A) It is anticipated these systems will begin as “optional” systems in order to build a data base to support expansion of the approval to achieve credit for extension of maintenance intervals, and so forth. Systems range from low to high integrity requirements depending on the determined criticality of application.

(B) Some systems that are being considered will utilize off aircraft processing of data. If this is to be pursued it should be assumed that the aircraft data will be lost or misplaced at the processing center, and the aircraft system design should consider this possibility. Some on-board data storage is one way to account for this lost data. The integrity of the processing center’s software should be equal to that of the aircraft software. In addition the intervals for processing the data from each flight should be specified as part of the approval.

(C) Due to the limited experience with these systems it is suggested the issue paper process be utilized to record the progress of the approval, and to provide information for later updating of this AC material.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 2. STANDARDIZED TEST PROCEDURE FOR ROTORCRAFT DC ELECTRICAL SYSTEM TESTS.

a. Test Requirements.

(1) General. The following functions and characteristics are to be evaluated:

(i) Normal System Operation.

(ii) Parallel Load Division.

(iii) Excitation.

(iv) Stabilization.

(v) Systems Malfunction.

(vi) Environmental Capability.

(vii) Electromagnetic Compatibility.

(viii) Cooling Capability.

(ix) Surge Characteristics, Ripple Voltage, and Voltage Spikes.

(2) Instrumentation. Calibration records should be available for all instrumentation. Current and voltage vs. time should be recorded in a permanent form. Enough specific currents and voltages should be recorded to allow reconstruction of any sequence of events that would happen as a result of any system testing described herein.

(3) Regulatory References. Sections 29.1301, 29.1307(c), (d), (e), 29.1309, 29.1351, 29.1353, 29.1355, 29.1357, 29.1363.

(4) Miscellaneous. The assigned FAA/AUTHORITY systems and equipment engineer normally witnesses these tests and should be notified as far in advance of the testing as possible to minimize scheduling problems. Conformity of the test setup must be established prior to conducting any testing. Most of the above test categories can be conducted on a bench test setup. A bench test setup is especially recommended in the case of the system malfunction tests. It is the applicant's option to demonstrate his

equipment either on the bench or installed for ground tests. When a bench setup is used, it should represent the actual aircraft installation to the extent that components and wiring (type, gage, and length) are duplicated. Some retesting may be necessary on the aircraft to verify the bench test results.

b. Ground and Bench Test Procedures.

CAUTION: Prior to disconnecting the battery and removing or adding large loads, either isolate the avionics systems or assure that transients induced are within limits of the avionics equipment.

(1) Normal System Operation.

NOTE: Equipment should be operated for at least 10 minutes prior to each test as a warmup.

(i) Minimum electrical load for paralleling and minimum engine RPM.

(ii) Vary RPM of all engines from low to high and back to low.

(iii) Repeat b(1)(ii) for maximum and 50 percent of maximum electrical loads.

(2) Parallel Load Division (if parallel system).

(i) Minimum electrical load for paralleling and minimum engine RPM.

(ii) Fifty percent of maximum electrical load and minimum engine RPM.

(iii) Maximum electrical load and minimum engine RPM.

(iv) Minimum electrical load for paralleling, vary No. 1 engine RPM from low to high and back to low while holding the RPM of the other engine at minimum (low).

(v) Repeat b(2)(d) for each other engine on the rotorcraft.

(vi) Repeat b(2)(d) and b(2)(e) procedures with 50 percent of maximum electrical load.

(vii) Repeat b(2)(d) and b(2)(e) procedures with a maximum electrical load.

(3) Excitation.

NOTE: All of these tests are to be conducted with the battery OFF since the purpose of the tests is to determine if the ship's battery is necessary for excitation of the alternator(s)/generator(s).

(i) Minimum anticipated electrical load, low engine RPM, and alternator(s)/generator(s) OFF. Demonstrate that when an alternator/generator is turned ON, it will come on the line. Repeat for any other alternators/generators in the system.

(ii) Maximum electrical load, low engine RPM, and alternator(s)/generator(s) OFF. Demonstrate that each alternator/generator will individually come on the line.

(iii) Minimum anticipated electrical load, high engine RPM, and alternator(s)/generator(s) OFF. Demonstrate that each alternator/generator will individually come on the line.

(4) Stabilization.

NOTE: All of these tests are to be conducted with the ship's battery OFF, since the purpose of the tests is to determine if the ship's battery is necessary for stabilization of the alternator/generator. In each case, if the ship's battery is not necessary for stabilization, the alternator/generator should be on the line and remain there at a satisfactory voltage level.

(i) Minimum anticipated electrical load, low engine RPM, alternator(s)/generator(s) ON. Switch on the heaviest electrical load that is anticipated to be installed on the aircraft.

(ii) Repeat b(4)(i) for a maximum electrical load and low engine RPM.

(iii) Repeat b(4)(i) for a minimum anticipated electrical load and high engine RPM.

(iv) Repeat b(4)(i) for a maximum electrical load and high engine RPM.

(5) System Malfunctions.

(i) Overcurrent faults (faults to airframe ground that are less than 5.0 Milliohms) should be applied to buses and feeders as necessary to demonstrate that the system's overcurrent circuit protective devices are properly coordinated and provide adequate protection/fault isolation.

(ii) Simulate an overvoltage condition on each alternator/generator to demonstrate satisfactory operation of the overvoltage sensing network. On a

multiengine configuration, the faulty alternator/generator should be removed without affecting operation of the remainder of the system.

(iii) The annunciation circuitry should be checked for indication of failures such as overvoltage, tripped generators, overcurrent, open feeders, open tie breakers, etc.

(6) Aircraft Ground Tests. If the above tests (reference b(1) through (4) inclusive) are conducted on a bench setup, enough tests should be repeated on the aircraft to validate the bench test results. The following tests should be conducted on the aircraft:

(i) Normal Battery Starts. Start all engines on the aircraft following the normal procedure prescribed in the flight manual. Record starter volts and amperes, time, and any other parameters deemed necessary.

(ii) Ground Power Cart Starts. If the aircraft is equipped with a plug for a ground power cart, use the procedure described in the flight manual and start all engines. Record starter volts and amperes, time, and any other parameters deemed necessary.

(iii) Emergency Battery Operation (if provided). The emergency battery mode of operation should be tested to assure at least proper switching, annunciation, and battery capacity. In some instances, an analysis of battery capacity may be adequate.

(iv) Other Tests. Conduct other tests as necessary to demonstrate proper operation of the specific design being evaluated.

(v) Distribution System Tests. With all systems operating individually, open and close feeder circuit breakers and system circuit breakers and assure separation of power sources for essential systems. For example, removing power from one bus by opening a feeder should not result in loss of both NAV 1 and NAV 2 or both COMM 1 and COMM 2 or both attitude gyros, or for example, opening NAV 1 circuit breaker should not affect NAV 2, etc. If the opening of the feeder protection has been satisfactorily demonstrated on a bench test facility, it should not be necessary to repeat that demonstration on the actual aircraft. The effect of loss of power sources should also be demonstrated on the aircraft. Reference §§ 29.1357(e) and 29.1309.

(7) Environmental Qualification. Each component of the system, such as relays, switches, alternator, generator, sensor, regulator, diode, etc., should be qualified to the critical environmental parameters. The temperature, altitude, humidity, and vibration expected in the approved aircraft operational envelope should fall within those limits the applicant substantiates for the electrical system components. (Refer to paragraph AC 29.1309.)

(8) Electromagnetic Compatibility. At no time during any of the qualification testing described herein should objectionable interference in the aircraft's radio, navigation, cockpit instrument, autopilot, or interphone system be considered acceptable.

NOTE: The quantitative type testing used for Items (7) and (8) above is outside the scope of this document. The latest revision of RTCA Document DO-160 is an acceptable standard.

(9) Transient Tests. The D.C. system should be tested and shown to exhibit surge, ripple, and spike voltages within the limits of the latest revision of RTCA Document DO-160.

(i) The surge and ripple voltage tolerance of avionic equipment is defined by the latest revision of RTCA Document DO-160. Category Z is considered applicable to rotorcraft D.C. systems.

(ii) The voltage spike tolerance of avionic equipment is defined by the latest revision of RTCA Document DO-160.

(10) Ground and Bench Test Report. At the conclusion of the ground and bench test program a report should be prepared and submitted that contains at least the following:

(i) System schematic (including instrumentation tie-in).

(ii) Instrumentation list (including calibration records).

(iii) Test result recordings.

(iv) Detailed procedures and results obtained.

(v) Conformity inspection records.

(vi) Other data, photographs, etc., to describe the test setup.

(vii) Summary of the test results. This summary should show the maximum load to which each bus, alternator/generator, etc., has been tested.

(viii) Analysis of test results. This should describe how compliance with the regulations has been shown. It should include consideration of the critical failure modes. Refer to paragraphs AC 29 MG 1 a(4)(ii) and AC 29.1309c for further information on failure analyses.

c. Flight Test Procedures.

(1) Alternator/generator cooling tests should be conducted in accordance with paragraph AC 29.1351.

(2) On multiengine rotorcraft, single-engine air starts should be conducted using the manufacturer's recommended procedures. This should be accomplished for each engine individually.

(3) A cockpit evaluation of the electrical system should be conducted to evaluate:

(i) Switch, circuit breaker, and annunciator identification.

(ii) Visibility of placarding, switches, etc., during bright sunlight and night operation.

(iii) Color of annunciators as related to the function/malfunction annunciated.

(iv) Load meter readability.

(v) Access to essential switches, circuit breakers, etc.

(vi) Electromagnetic interference.

(vii) Compatibility of the electrical system with the rotorcraft flight manual and the need for additional procedures in the RFM.

(viii) Clarity of functions such as opened feeder breakers, tie breakers, related annunciation, and necessary corrective action in the event of malfunction.

(ix) Absence of undesired functions in relation to switch combinations.

CHAPTER 3
AIRWORTHINESS STANDARDS
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MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 3. ROTORCRAFT AND SYSTEMS CERTIFICATION FOR CATEGORY II OPERATIONS.

a. Explanation.

(1) Category II instrument approach and landing minimums variations are based on ground facilities and environment, aircraft equipment, crew training, crew proficiency, and maintenance programs. For the pilot, the approach and landing minimums final consideration is the runway visibility which can be, and usually is, related to a cloud ceiling, although the concept is that if there is a runway visibility of 4,000 feet, as an example, there is a very high probability that the ceiling will be at least 300 feet. Therefore, Category I minimums are weather conditions of not less than a 200-foot ceiling and ½-mile visibility or runway visual range (RVR) of 2,400 feet. Category II minimums permit approaches at less than 200 feet decision height/RVR 2,400 to as low as 100 feet/RVR 1,200. Category III approach minimums are less than Category II but will not be discussed here.

(2) The ground facilities required for a Category II approach and landing include specific approach lighting extending more than 3,000 feet from the runway, thus eliminating any present heliports from being approved for Category II operations. Therefore, the following Category II approvals procedures for rotorcraft assume an approach to a runway at airspeeds at or above V_{MINI}.

(3) The regulations and advisory material covering the approval for IFR Category II operations are included in Part 91, Appendix A, and AC 91-16, Category II Operations - General Aviation Airplanes. Those references address airplanes; however, the concept is also suitable for the approval of rotorcraft for Category II operations. The equipment to be required and the procedures to be followed are basically the same for a rotorcraft as for an airplane. Additional reference material concerning Category II approval is contained in FAA Order 8440.5A, General Aviation Operations Inspection's Handbook, and AC 120-29, Criteria for Approving Category I and Category II Landing Minimum for FAR 121 Operations.

(4) Authority for rotorcraft to use Category A airplane minimums is contained in § 97.3(d)(1). FAA Order 8440.5, §§ 97.3, 91.6, and Appendix A of Part 91 provide authority to consider the rotorcraft as a small, Category A aircraft and relief from the requirement for two pilots and two sets of instruments and equipment. Any rotorcraft that is presented for Category II certification must first meet the requirements for rotorcraft instrument flight (Appendix B of Part 29 and paragraph AC 29.1543).

(5) In addition to the ground facilities and environment noted above, there are requirements in three other general areas to obtain Category II approval. These are certification of the aircraft and systems, certification and continuation training of flight crews, and a continuing maintenance program for the aircraft and Category II required systems. The entire Category II approval requires a Category II manual that covers all of these areas. FAA/AUTHORITY approval of this manual would normally be the responsibility of the operations and airworthiness inspectors that grant the approval to an operator for Category II operations.

(6) The additional equipment necessary for a Category II approval consists of the flight control guidance system. This system can be either a flight director system or an automatic approach coupler. A flight director system needs only to present computed steering data for the instrument landing system (ILS) localizer and should present at least raw glideslope data on the same instrument as the localizer steering commands. A single-axis steering autopilot could be used if it coupled to the ILS localizer. In a practical sense, however, contemporary rotorcraft flight director and automatic pilot systems use at least two-axes command guidance or coupling, and some provide coupling or guidance in three axes; localizer, glidepath, and airspeed. A marker beacon system or a radio altimeter is required for operations with decision heights of 150 feet or less. A rotorcraft flight manual (RFM) supplement is required to define the configuration limitations and procedures for Category II operation.

b. Procedures.

(1) Instrumentation. Test instrumentation is required to provide a time history of the following parameters throughout each approach:

- Localizer deviation.

- Glideslope deviation.

- Radar altitude (if available).

These parameters can be acquired from the cockpit display for each one. The localizer and glideslope deviations are normally recorded as a microampere deviation from the centerline on a continuous strip recording. The radar altitude is continuously recorded as feet above the ground on the same recording device. Any type of recorder that produces a time history of these parameters throughout the approach would be satisfactory. However, a recorder that can be read during, or immediately after, each approach is recommended. This will allow the acceptability of the tracking during the approach to be determined immediately after each approach.

In addition to the above data, cockpit data should be hand recorded on a format similar to that shown in AC 91-16, Attachment 3 (figures AC 29 MG 3-1 and AC 29 MG 3-2).

(2) Systems Evaluation.

(i) The major portion of a Category II approval is the evaluation of the flight guidance system. To certify the flight guidance system for a specific model rotorcraft, a demonstration of 50 ILS approaches with a 90 percent success rate (as defined in Part 91) must be accomplished. If the flight guidance system has not been previously certificated in the rotorcraft, a certification program should be completed for the system before the Category II evaluation is started. It should be determined that the flight guidance system does comply with all the certification requirements before 50 ILS approaches. This is particularly true of an autopilot system where hardover malfunctions must be considered.

(ii) The equipment to be installed for Category II operations must meet the performance criteria specified in AC 120-29, Appendix 1. This material details the criteria for approval of airborne equipment and its installations to meet Category II performance. This appendix covers the rotorcraft flight manual, the systems ground tests, and the installation requirements and tests. Transport category rotorcraft should meet the same systems performance requirements as transport category airplanes.

(iii) The flight demonstration required for Category II system approval is explained in Part 91, Appendix A, Paragraph (e). The accuracy requirements for the tracking equipment are included in Appendix 1 of AC 120-29. The usual method of determining the tracking accuracy is by measuring the localizer and glideslope deviations in microamperes and printing them on a continuous strip recorder. The observed cockpit date should also be recorded on a form similar to that in AC 91-16, Attachment 3 (figures AC 29 MG 3-1 and AC 29 MG 3-2). Each approach made during the evaluation should have a complete set of data.

(iv) Coupler systems that require manual trimming by the pilot to center the AFCS actuators should be carefully evaluated, especially in turbulent conditions or gusty crosswinds. These systems may not meet the trim requirements at the 100-foot decision height or may not provide sufficient tracking accuracy without excessive pilot attention and workload.

(v) The effects of coupler system hardover malfunctions should be evaluated in all axes to determine the minimum decision height. The altitude loss that would occur from a nose down hardover at the decision height should be determined. This altitude loss should be included in the rotorcraft flight manual with the appropriate limitation on the minimum height above the ground for operation with the coupler engaged.

(vi) It is recommended that the demonstration approaches be made to Category II ILS facilities, although this is not required by either Part 91, Appendix A, or AC 91-16. Many Category I ILS installations do not provide good enough signals at the lower altitudes for the precise tracking required for Category II operations. In many cases, this is due to the effects of terrain or buildings off the approach end of the runway. Nevertheless, if satisfactory accuracy can be attained, all the approaches

required for a Category II approval may be made at Category I facilities. During the flight test, especially if simulated IFR conditions are used in good weather, the approach control and control tower of the facility being used should be advised that Category II operations are being conducted. The Category II ILS clear areas must be kept unobstructed to allow satisfactory ILS signals. The air traffic control agencies should assure that taxiing aircraft, airfield maintenance trucks, and other airfield traffic are kept out of the critical areas during the data-gathering approaches. These agencies can also monitor the ILS facility for proper operation to Category II standards and can advise the test aircraft if abnormal operation occurs.

(3) Rotorcraft Flight Manual. Upon satisfactory completion of an engineering inspection and test program, the FAA/AUTHORITY Rotorcraft Flight Manual (RFM), or supplements thereto, should reflect the following:

(i) The limitations, if any.

(ii) Revision to the performance section, if appropriate.

(iii) A statement of Category II approval to the effect that “The airborne instruments and equipment meet the performance standards for Category II approaches” and the following note:

“NOTE: Compliance with the performance standards referenced above does not constitute approval to conduct Category II operations.

CATEGORY II APPROACH EVALUATION

Pilot in Command _____ Second in Command _____ Date _____

Registration No. _____ Airport _____ Runway _____ Weather _____ Wind _____

FAA Inspector _____

This form will be completed whenever an approach is attempted utilizing the airborne low approach system, regardless of whether the approach is abandoned or concluded succesfully.

APPROACH EVALUATION

SAMPLE

1. Was the approach successful? YES ☐ NO ☐
2. Flight control guidance system used
   a. Auto-coupler ☐
   b. Flight director ☐
   c. If equipped and used, did a and b agree? YES ☐ NO ☐
   Second in Command? YES ☐ NO ☐
   FAA Inspector? YES ☐ NO ☐
3. Airspeed at middle marker ± ____ at 100' ± ____ from programmed speed?
4. If unable to initiate ☐ or complete ☐ approach (indicate which), was reason due to:
   a. Airborne equipment ☐. Identify and describe nature of deficiency.
   b. Ground equipment ☐. Identify and describe nature of deficiency.
   c. Approach control or tower request ☐.
   d. Other ☐. State reason.
5. Was airplane in trim at 100' for continuation of flare and landings?
   YES ☐ NO ☐
6. If approach and landing abandoned, state altitude above runway: _____
   feet (State reasons) _____
   _____
7. Quality of overall performance: Good ☐ Acceptable ☐ Unacceptable ☐

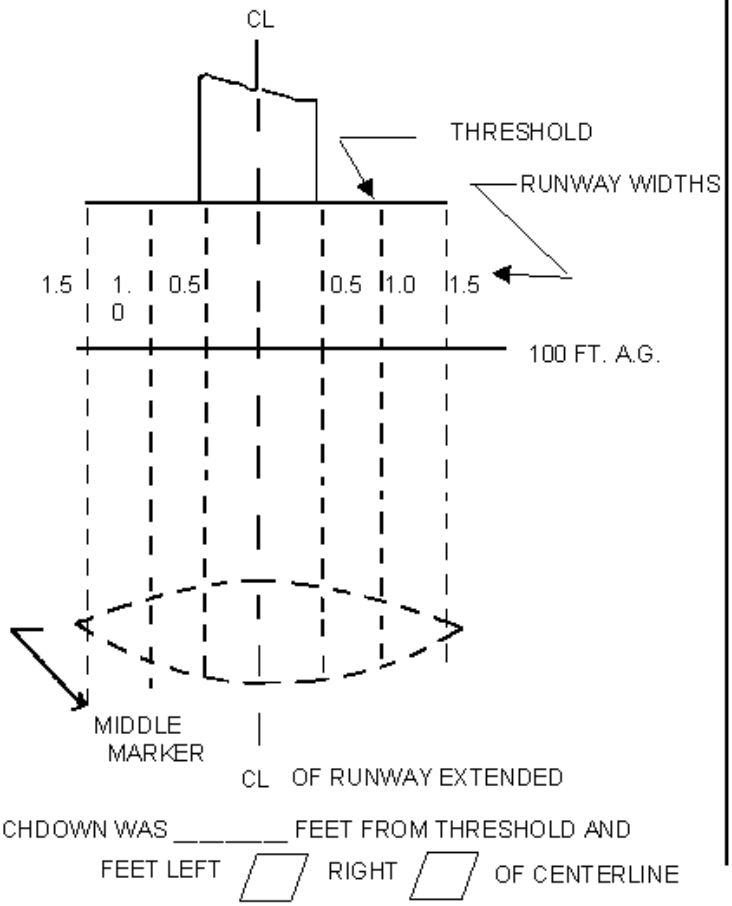
_____
Pilot in Command's Signature

FIGURE AC 29.MG 3-1

CATEGORY II APPROACH EVALUATION (cont.)

INDICATE GLIDESLOPE AND LOCALIZER DISPLACEMENT AT MIDDLE MARKER AND 100 FT. A.G. POINT

INDICATE AIRPLANE DISPLACEMENT & ORIENTATION WITH RESPECT TO RUNWAY CENTERLINE AT MIDDLE MARKER AND 100 FT. A.G. BY SYMBOL AT AIRPORTS AND RUNWAYS LISTED.



TOUCHDOWN WAS _____ FEET FROM THRESHOLD AND

FEET LEFT ☐ RIGHT ☐ OF CENTERLINE

REMARKS:

SAMPLE

FIGURE AC 29.MG 3-2

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MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 4 FULL AUTHORITY DIGITAL ELECTRONIC CONTROLS (FADEC)

a. FULL AUTHORITY DIGITAL ELECTRONIC CONTROLS (FADEC) FOR INSTALLATIONS WITH CATEGORY A ENGINE ISOLATION.

(1) Background. The advent of "microprocessor technology" has resulted in rotorcraft engine controls being implemented by digital process control rather than by conventional means. These digital, processor-based full authority engine controls offer many performance advantages (such as isochronous governing) which were not feasible with conventional technology, pneumatic or hydromechanical controls. Because of the incorporation of this advanced technology, some additional considerations must be made of the engine installation to ensure regulatory compliance.

(2) Requirements. The following is a discussion of some special attention areas for a Part 29 Category A FADEC engine installation.
Paragraph AC 29.1309(b)(3)(ii)(D) contains a general definition of what constitutes a "full authority" control.

(i) Software Qualifications.

(A) Paragraph AC 29.1309f contains a general discussion on the use of the RTCA/DO-178B document that is used for the approval of system software. FADECs are generally developed to Level A software under RTCA document DO 178B based on the hazard category of the FADEC failure condition(s). However, if an applicant proposes a FADEC with Level B software based on the Functional Hazard Assessment results, this will require the proposal to be reviewed and approved by both the Engine Directorate and the Rotorcraft Directorate.

(B) RTCA/DO-178A may still be applicable for those FADECs that were previously developed and approved under DO-178A and the applicant is proposing to make changes to the FADEC software. However, if the applicant proposes to make changes to a DO-178A approved FADEC, the determination on whether the changes should be made under DO-178B or DO-178A will need to be made by the Engine Directorate and Rotorcraft Directorate. When utilizing DO-178A, one might arrive at the conclusion that the engine control, as a required function, is essential; therefore, level 2 software under DO-178A would be appropriate for the control functions. However, for this level 2 category software, errors are presumed to exist, and a software error in a full authority control could result in simultaneous unacceptable malfunctions in all engines. The provisions of

§ 29.1309(b)(2)(i) for continued safe flight and landing and the engine isolation rule, § 29.903(b), would generally preclude the use of this classification.

(C) System designs which provide redundant distinctive software or an alternate technology control which is automatically selected and meets all of the minimum regulatory requirements would reduce the impact of software errors and may allow the level 2; i.e., essential software classification. At level 1, it is accepted that the software is sufficiently error free that the software does not require further verification in the installation evaluation.

(ii) Lightning Strike Protection. Paragraph AC 29.1309(b)(3) contains a complete discussion of an acceptable method of demonstrating that the FADEC, as installed, is adequately protected against the catastrophic effects of lightning.

(iii) Electrical Power System Considerations.

(A) Normal Operation. The system should be evaluated with all power sources operating normally. If additional power source capability is being provided that is above the minimum required for certification, a certain portion of the evaluation should be conducted while operating in the minimum configuration. The minimum power source configuration should consider the provisions of § 29.903(b).

(B) Malfunction Conditions. Beginning with the minimum configuration that is required for certification, electrical power system malfunctions should be introduced and the impact on continued FADEC operation determined.

(C) Circuit Protection Location. The circuit protective devices for the FADEC should be located in the cockpit such that they can be readily reset or replaced in flight. The operation of the FADEC system is considered to be essential to safety in flight. Reference § 29.1357(d). The definition for “essential to safety in flight” is given in AC 29.1357b(4).

(D) System Separation. On multiengine applications, each system should be separated from the other system to the maximum extent practical. Wiring should be routed separately. Power should be taken from independent busses and grounds, and system components should be independent of one another.

(E) Periodic Checks. Where periodic checks are appropriate, they should be made at reasonable intervals. This would normally range from preflight checks for certain items of greater concern to a tie-in with normal aircraft maintenance intervals for other items. If a crew check is specified, it should be evaluated to ensure it is a reasonable check. If items to be checked are located in an area that can be covered by interior upholstery, for example, a crew check would not be considered reasonable, and further design considerations may be in order.

(iv) Powerplant Installation Considerations.

(A) A demonstration of compliance with § 29.901(c) would generally include a failure mode and effects analysis (FMEA) of the powerplant systems as installed. When a FADEC is utilized, the analysis would consider the control's failure modes, the installed engine reaction, the effect on the aircraft, and the crew response to the situation. Combinations of undetected failures should be considered. Engine failures which may be escalated in severity by the FADEC's response to the initial failure should be analyzed. Potentially hazardous failures should be evaluated during flight testing. The requirements of §§ 29.903(b)(2) and §§ 29.1309(b)(2)(i) should be reviewed in determining acceptability of failures.

(B) Section 29.903(b)(2), Category A engine isolation, is intended to ensure that a failure will not prevent the continued safe operation of the remaining engine(s) or require immediate action of the crew to ensure continued safe operation. The FADEC's of the individual engines should be independent. Where communication between FADEC's is required (for example, for torque sharing), care should be exercised to ensure that failures which may occur will not result in a power loss to the extent that total power available is less than would be available under OEI conditions. The no-required immediate-crew-action provision would preclude credit for manually selected or operated backup systems in meeting the § 29.903(b) rule. These unrequired backup systems, which may offer the advantage of get-home multiengine capability rather than forced OEI operation, would be evaluated on a no hazard basis.

(C) Section 29.939, turbine engine operating characteristics, intends a flight investigation to ensure that no adverse characteristics are present to a hazardous degree during normal and emergency operation in the allowed flight envelope. The evaluation should include assessment of the minimum FADEC system certification configuration; i.e., the minimum proposed by the applicant to meet Part 29 requirements. Reduced capabilities (e.g., restrictions on normal collective movements, limited aircraft maneuvers, etc.) may be acceptable for degraded FADEC modes or backup systems not required to meet Part 29 requirements if those degraded capabilities are reasonable and not hazardous as determined by flight evaluation. The restrictions should be specified in the flight manual.

(D) The rotorcraft with FADEC engines must of course meet all of the Part 29 requirements, but the areas described herein are those which deserve special attention.

b. SINGLE CHANNEL FULL AUTHORITY DIGITAL ENGINE CONTROLS (FADEC) IN SINGLE ENGINE ROTORCRAFT APPLICATIONS.

(1) Background. The purpose of this appendix is to provide guidance for compliance to Part 27 and Part 29 Category B regulations when the powerplant installation is a single engine fitted with a single channel FADEC system. The

application of single channel FADECs in single engine helicopters requires special considerations because this combination can have a higher probability of FADEC-related malfunctions that could result in loss of ability to execute a controlled power-on landing or operate safely throughout the flight envelope, relative to dual channel FADEC systems or multiengine installations. The issues that should be addressed by the applicant are criticality level of failures as determined from the engine system safety analysis (SSA), the resulting integrity requirements, capability to detect and present failure/fault data to the crew, and the ability of the crew to manage any failures/faults. The term “must” in this policy is used in the sense of ensuring the applicability of these particular methods of compliance when the acceptable means of compliance described herein is used. This policy establishes an acceptable means, but not the only means of certifying a single channel FADEC for single engine application.

(2) Definitions.

(i) Fault or 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).

(ii) Integrity. The term “integrity” for the purpose of this policy includes the hardware reliability requirements as well as the software level requirements commensurate with the system criticality.

(iii) Single Channel FADEC. A single channel FADEC system is one which provides full authority control of the engine from below ground idle to 100 percent power and in some cases from engine start similar to more complex dual channel redundant FADEC systems, but without a fully capable second channel providing a dual redundant system. The backup for the single channel FADEC is provided by a less capable channel either by hydromechanical or electronic means, usually for “get-home” purposes rather than for dispatchability.

(3) References. FAR paragraphs 29.901, 29.903, 29.927, 29.939, 29.1143, 29.1309, 29.1581.

(4) Related Documents.

(i) Federal Aviation Regulations (FARs) paragraphs 21.21, 29.1301, 33.28, 33.75

(ii) FAA Advisory Circular AC 27-1B

(iii) Standards - Latest revision of RTCA/DO-178 and RTCA/DO-160; SAE documents

(iv) ARP4754 and ARP4761

(5) Design Requirements for Compliance with FAR 29.901. FAR paragraph 29.901(b)(2) requires that each component of the installation be constructed, arranged and installed to ensure its continued safe operation between normal inspections or overhauls. FAR paragraph 29.901(c) requires that no single failure or malfunction of the powerplant control system will jeopardize the safe operation of the rotorcraft. For an engine with a single channel FADEC some form of redundancy is needed to ensure the continued safe operation of the rotorcraft in the event of a random complete failure. This redundant system must be accessible and provide the pilot with the ability to perform a controlled power-on landing. In addition, FAR paragraph 29.939(a) requires that turbine engine operating characteristics be investigated in flight. Flight tests are required as noted below to demonstrate compliance with the FAR requirements. The following paragraphs provide guidance for meeting these general design requirements.

(i) Redundancy: Because of the random nature of electrical/electronic component failures, there is no assurance that the electronic systems will operate safely between established inspection periods. Therefore, some redundancy technique should be applied to the electrical/electronic part of the FADEC system to reduce the probability of losing the ability to land safely or continue safe flight. This redundancy is usually provided by some form of backup system or alternate method of control of the engine. The requirement for a backup system can be achieved with a number of approaches that include a simple mechanical/hydromechanical system, a simple electrical/electronic system that is not a completely redundant channel, or a completely redundant system.

(ii) Availability: A means must be provided either by system design or operational procedures to ensure that the primary and the backup or alternate system are available functionally to serve the intended purpose. The manufacturer's required interval for testing the backup or alternate system should be based on the expected failure rate established during the failure analysis of the system. However, the pilot should have the capability to test the backup system at the pilot's discretion. Additionally, failure of the primary system must not affect the safe operation of the backup or alternate system.

(iii) Capability of back-up system: Section 29.1143 requires that each power control provide a positive and immediately responsive means of controlling its engine. Additionally, § 29.903 requires that the powerplant systems associated with engine control systems are designed to give reasonable assurance that the engine operating limitations will not be exceeded in service. Although back-up control may be somewhat degraded, the system should allow for control of the engine and the aircraft within their operating limits. It should be demonstrated that upon failure of the primary control the aircraft can continue to be operated safely and execute a controlled power-on landing without creating an undue pilot workload. This includes demonstration of the ability to maintain rotor speed within

acceptable limits while transitioning to the backup mode and while using the backup control.

(iv) Ability of crew to switch to back-up: If crew action is required for switching to the back-up mode, this ability must be demonstrated during all phases of flight from any seat which may be occupied by the pilot in command or the copilot. The process to be used by the pilot to switch to the back-up mode should be clearly described in the Rotorcraft Flight Manual (RFM) as required by FAR paragraph 29.1581.

(v) Transfer to backup: The transfer to the back-up mode from the primary control mode or an intermediate mode (fixed position) must occur without excessive time delay or variation in power. Time delays and power variations experienced during the transfer should be evaluated during flight test for acceptability. A means should be provided to alert the pilot that transfer to the back-up mode has occurred.

(vi) Annunciation: Adequate annunciations should be provided to cue the crew of faults/failures and/or transfer of engine controls. These annunciations are of visual and aural types and must be distinct as to purpose and should not be misleading, especially under any fault/failure. Flight evaluation of these annunciations is required before final acceptance can be made.

(vii) Automatic Transfer: If the system is designed to accomplish automatic transfer between control modes, the transfer should occur without excessive variation in power and a means should be provided to alert the pilot that transfer to the back-up mode has occurred. Multiple automatic transfers between control modes may cause aircraft instability. A method to lockout the primary control after its initial failure and automatic transfer to the backup should be provided. If pilot reset is to be allowed, the procedure should be described in the RFM.

(viii) Calculated failure rate (with unannunciated faults present): Before a calculation of the failure rate can be attempted, the failure should be defined. The determination of failure rate, using the definition of failure, can be the product of a Failure Mode Effects Analysis (FMEA) combined with a reliability analysis, using individual part reliability figures. The figures should come from some recognized data base. The failure rate calculations should consider the worst case application limitations such as flight operation, environmental considerations, and time of operation. The flight operations to be considered for the worst case scenario include all flight segments (take off, cruise, hover, landing, etc.) together and separately for the various missions the aircraft is expected to be used in. Another way to determine failure rate is to use service history. However, service history is applicable only if a high degree of similarity exists for the FADEC and its installed application. The calculated failure rate is the direct result of the FMEA, and should

meet the integrity level requirement determined by the Functional Hazard Assessment.

(6) Certification Approach:

(i) Analysis Requirements: Functional Hazard Assessment: Compliance to the requirements of FAR paragraphs 29.1141 and 29.1309 for a single channel FADEC in a single engine application should be based on criticality of application for the system under consideration. This criticality of application may be determined by performing an aircraft level hazard assessment that starts with the type of possible failures and ends with the results of these failures. The results can be categorized into criticality levels and the required integrity levels can be obtained by matching the required integrity level to the criticality level. The main emphasis should be on determining the higher levels of criticality (Major and above) and their source. This process should include consideration of failures seen at the operational level and interaction of the failures with the airframe and crew as well as the system itself. The following subject areas are related to this assessment.

(A) Assumptions: Assumptions should be made about the airframe/crew interface in order to perform the aircraft level hazard assessment. These assumptions are prerequisites to perform an aircraft level hazard assessment and must be listed in this hazard assessment and validated by airframe testing when the airframe is available. If the assumptions cannot be validated, the actual airframe test data must be substituted for the invalidated assumptions (assumed prerequisites) and the hazard assessment re-evaluated with the new data supported prerequisites. The results of this new assessment would be the deciding factor for acceptance of the FADEC system for the installation as designed or provide the necessity for design changes.

(B) Criteria: Acceptance of an engine fitted with a single channel FADEC system in a single engine rotorcraft application requires that the integrity levels of the FADEC system be compliant with the criticality levels determined by the aircraft level Functional Hazard Assessment (FHA). In addition, final acceptance of the system at the aircraft level for the application is based on the integrity level(s) that match the criticality level(s) determined by the hazard assessment that uses data that has been validated during the aircraft flight test program. These assumptions/prerequisites would include operational aspects associated with the possible FADEC failures and would include as a minimum the following:

(1) Crew/aerodynamic response to failure.

(2) Worst case flight operation for failure to occur. (Landing, IFR, etc.)

(3) Duration of flight operation (exposure time).

(4) System interaction with shared Inputs/Outputs with other systems and/or with back-up systems.

(5) Adequate annunciation of failure.

(ii) Validation Criteria:

(A) General:

(1) Validation of the assumptions/prerequisites made by the engine manufacturer in developing the SSA, using aircraft level FHA requirements, must be validated by conducting flight testing during the certification of the installation. The possibility exists that if the assumptions cannot be validated during flight testing, then engine and/or FADEC redesign may be required.

(2) Failure management methods that are related to operational characteristics should be addressed. It should be determined that the FADEC/engine manufacturer's envisioned failure management is desirable and compatible with the operational requirements. Therefore, the following basic FADEC related information should be identified:

(i) The detected failures.

(ii) The failures that are not detected.

(iii) The action that the FADEC takes when failures are detected.

(iv) The failures that are annunciated to the crew and in what manner.

(v) The anticipated operational action required as a result of detected failures.

(vi) Possible operational results of the undetected failures.

(vii) Verification that the assumed worst case flight operation is the worst case.

(B) Manual Backup: Additional aircraft operational testing is required to specifically evaluate the manual backup system for compliance with the FAR requirements. The acceptability of the manual backup system depends substantially on its installation and interface with the airframe. The following items need to be demonstrated in accordance with § 29.927 and § 29.939 or accomplished on each application prior to the acceptance of the manual backup system:

(1) It should be demonstrated by flight test with the failure of the primary engine control, that the aircraft can be flown and a safe and controlled power-on landing executed without creating an undue pilot workload.

(2) It should be demonstrated by flight test that switching between control modes will not create an unsafe condition during any phase of operation within the aircraft operating envelope.

(3) The pilot action required as a result of a failure of the primary control and used as an assumption in the FHA and FMEA should be validated during flight tests and listed in the emergency procedures section of the flight manual.

c. FADEC RELIABILITY REVIEW DUE TO INCREASED ROTORCRAFT ENDURANCE

(1) Background. This advisory material is to provide guidance for reevaluation of the FADEC control system reliability due to extension of the aircraft mission endurance. During the initial type certification of an aircraft, an analysis is normally conducted on systems to determine their criticality category (e.g. catastrophic, hazardous, major, etc.) and reliability requirements. To establish a system's reliability, an exposure time is determined by making certain assumptions. In most cases, the exposure time is the average endurance based on the various flight scenarios in which the aircraft is to be used. When an aircraft's expected mission endurance is increased by adding fuel capacity, a new analysis for system reliability should be conducted taking into account the new increased mission endurance.

(2) Requirements.

(i) If the applicant has access to the initial analysis used for the type certification, one method to accomplish the new reliability analysis is by multiplying the exposure time used in the original reliability analysis by the ratio of the increased maximum endurance to the original maximum endurance. That is, if the aircraft endurance increases by 50 percent due to additional fuel capacity, the assumed exposure time should also increase by 50 percent. The applicant should then rework the analysis using this new exposure time.

(ii) If the applicant does not have access to the initial analysis it will be incumbent upon them to provide the rationale used for determining the new exposure time and to provide a complete analysis for the systems determined to be critical. The FAA engineer should compare this new analysis to the original.

d. CERTIFICATION GUIDELINES FOR COMPLIANCE TO THE REQUIREMENTS FOR ELECTROMAGNETIC COMPATIBILITY (EMC) TESTING FOR NON-QUALIFIED EQUIPMENT AND EQUIPMENT KNOWN TO HAVE A HIGH POTENTIAL FOR INTERFERENCE WHEN INSTALLED ON ROTORCRAFT WITH ELECTRONIC CONTROLS THAT PROVIDE CRITICAL FUNCTIONS.

(1) Background.

(i) Rotorcraft operations are varied and use a wide assortment of equipment. While some of this equipment is qualified to aircraft standards, particularly environmental standards, some of the equipment not qualified to such standards may be the source of harmful electromagnetic interference. Rotorcraft typically have not had electronic controls that perform critical functions, such as engine controls and flight controls and therefore there was no real concern about requiring equipment to be qualified to aircraft standards. Typically, this equipment was installed with only a cross matrix operational check for EMC. These tests consisted of operating the equipment in question and checking visually for an indication of interference. The equipment was, for the most part, non-required equipment and the primary concern was that interference may be emitted from the equipment.

(ii) Many more recent rotorcraft designs are using electronic engine controls, and fly-by-wire may be implemented in the near future, thus unqualified equipment and their effects on critical aircraft systems are a particular concern. Additionally, the physical proximity of unqualified equipment to the Full Authority Digital Engine Controls (FADEC) is inherent due to the size of most rotorcraft and represents greater potential for interference than for larger fixed wing aircraft.

(2) Requirements.

(i) The rules to assure that required functions are not subject to interference are provided in the certification basis for the rotorcraft. Although the certification basis may differ between aircraft, the requirements that address electromagnetic interference are quite similar and result in the same methods for compliance. A note has been added to type data sheets for rotorcraft that employ FADEC. This note was added to remind all modifiers that the requirement for addressing interference exists and that special test considerations must be addressed to show compliance. Most EMC considerations can be addressed by the operational interference checks addressed in the background discussion. However, when a critical function is electronically provided, additional special test considerations should be addressed, in addition to the previously described EMC tests. The determination of when these other, more rigorous tests are required is a simple concept, but complex in practice. More rigorous testing is required to satisfy the concern for the installation of equipment that would interfere with the FADEC's control or failure management. There are two types of equipment installations that would cause this concern. The first type is equipment known to have a potential to interfere and may or may not be qualified to an aircraft standard, such as HF radios, high powered radars, hoists,

transmitting antennas located near the controls systems, etc. The second type of equipment does not have a high potential to interfere and is not qualified to an aircraft standard. It is important to determine if an acceptable environmental qualification test for ElectroMagnetic Interference (EMI) has been conducted for this equipment. The concern associated with this equipment is the interaction with the electronic control, and it should be determined if the equipment has been tested to an acceptable standard. One acceptable standard is the RTCA document DO160 (latest revision), Section 21, Category Z for EMI, but there also may be other acceptable standards. If there is a question of the acceptability of a standard, the applicant should contact the FAA. Lower levels of testing may be accepted, if there is additionally some favorable installation test data. This acceptance of lower level testing plus some additional installation test data will be evaluated on a case-by-case basis.

(ii) Accomplishment. In addition to the following special testing considerations, all installed equipment should be tested for EMC by operating all equipment under consideration and observe that no hazard is created by interference to required equipment.

(A) Class of Equipment - Equipment Known to Have a High Potential to Interfere: This class of equipment should be tested in the installation as described in the “Installation Test” paragraph below. Since the concern of this class of equipment is its “high potential for interference, its qualification is not a factor to preclude the requirement for testing. The type of equipment will determine if ground testing alone is sufficient; however, due to the high power nature of this class of equipment, flight testing is usually required. Kinds of equipment in this class are HF radios, high powered radars, hoist, etc.

(B) Class of Equipment - Not Qualified to Acceptable Interference Test Standard: Once it has been established that unqualified equipment is proposed to be installed, then one of two methods to show compliance may be implemented. One method is to laboratory test the unqualified equipment to an acceptable standard for EMI, such as RTCA DO160 (latest revision), Section 21, Category Z. The other method is to test the unqualified equipment in the installation as described in the “Installation Testing” (paragraph (4) below) to determine if it is a source of interference to the critical control (FADEC, Fly-By-Wire).

(iii) Laboratory Testing. Laboratory testing to the RTCA standard DO160 (latest revision), Section 21, Category Z is an example of one of the options to satisfy the interference concern for the unqualified equipment, providing the equipment does not fall into the class of equipment that is known to have a high interference potential. Testing the unqualified equipment in the laboratory will require an FAA-approved test plan and some type of conformity to identify the test article. Conformity should at least consist of conformity by the vendors’ description/drawings and functional specifications. Most equipment of this type should be conformed to a serial number as well as part number since there is a high probability that the production changes are not documented under a controlled system. However, if reasonable assurance can be

provided that other equipment of the same part number is identical to that tested, then credit for the test can be issued.

(iv) Installation Testing.

(A) Installation testing is one method to show compliance for unqualified equipment that does not have a high potential to cause interference. For interference considerations, installation testing is the only method of testing to show compliance for the class of equipment known to have a high interference potential such as HF radios, high powered radars, hoists, antennas located 0.5 meters or closer to the FADEC or other control, transmitting systems, etc.

(B) To accomplish the installation tests, there should be an FAA approved test plan that requires the unqualified equipment or high interference potential equipment to be operated through all reasonable modes of operation, to determine if electromagnetic interference is entering the control system. Installation testing consists of interrogating the control, if it has such a feature, to determine if the control system is adversely affected (identify the recorded faults that occur during the test). Additionally, real-time monitoring of the control's input parameters should be accomplished. The pass/fail criteria is "no detected interference" for a pass state and conversely a fail state if any interference is detected entering the control. If interference is detected, the source of interference should be investigated to determine if the detected interference is the worst case. In some cases, the detection of interference may result in flight tests being required to determine if the interference is worse in flight. After the worst case is defined, the interference must be eliminated at the source, or the interference must be evaluated to assure that the control, its functions, and its related indications do not result in an unsafe condition. For FADECs, special test equipment developed by the engine manufacturer will be required to interrogate and monitor the parameters. Other type critical controls may also require special test equipment to perform this type of testing.

(C) Installation Test Conditions: When installation tests are required, ground tests are sufficient except for the type of equipment that is identified as a "known" potential for interference and other equipment that either requires large currents to operate or radiates strong electromagnetic fields, especially equipment that is prohibited to be operated on the ground. Examples of the type of equipment "known" to possess interference potential are HF radios, high powered radios, hoists, installation where radio transmission antennas are in close proximity to the FADEC, etc. Examples of the other equipment that require large currents to operate or radiate strong electromagnetic fields are some EMS equipment, night sun lights, some air conditioners, video and sound systems that require large currents (25 amps and up) to operate, FLIRS, some forward looking radars, some weather radars, some communication systems that transmit 30 watts or more, and some data link transmission systems. This type of equipment represents the main concern for radiated and inductive interference; therefore, ground and flight tests should be conducted. Tests for conductive interference should be conducted for all other

unqualified equipment. Tests for conductive interference may be performed by ground tests, using the same techniques as previously described. Another exception to installation testing is, if the equipment has been tested in relation to the critical control on another installation and the installation under consideration can be shown to be identical. The data showing identicality of equipment and installation with passing test data are acceptable in place of further testing on the same type rotorcraft.

(3) Summary. The concern for potential interference to the FADEC or any other critical control may be addressed by the methods contained within this document. To address the interference aspects of unqualified equipment, the equipment must either be laboratory tested or tested as a part of the installation. Ground testing for the most part is sufficient, except for certain equipment with a high potential to cause interference. Other equipment, that may or may not be qualified, should be ground and flight tested if there are operational limitations on the ground or it fits the “known” potential to cause interference.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 5. AGRICULTURAL DISPENSING EQUIPMENT INSTALLATION.

NOTE: This paragraph has been extensively revised and expanded to clarify the restricted category certification of agricultural dispensing equipment installations on rotorcraft.

a. Explanation. In the early development of the rotorcraft one of its primary usages was agricultural operation. The FAA recognized that the existing requirements, which were designed primarily to establish an appropriate level of safety for passenger-carrying aircraft, imposed an unnecessary economic burden and were unduly restrictive for the manufacture and operation of aircraft intended only for use in rural, sparsely settled areas. Therefore, a special document that established new standards for agricultural dispensing equipment and other special purposes was developed. Restricted Category CAM 8 became effective October 11, 1950.

(1) During the recodification of 1965, CAR 8 ceased to exist as a regulatory basis and selected portions addressing certification were incorporated into FAR 21. While the specific standards in CAR 8 were not changed substantially when adopted into FAR 21, the less restrictive philosophy of CAM 8 and the policy material that was stated in the preamble to CAM 8 were not clearly conveyed.

(2) Advisory material published in 1965 and revised in 1975, summarized the information contained in the advisory portions of CAM 8. This new advisory material indicated that the CAM advisory material would be applicable to the related FAR's. Unfortunately, this document specified that CAM 8 could be used in conjunction with certain FAR's for restricted category certification of small agricultural airplanes only. Rotorcraft were omitted.

(3) A survey of restricted category rotorcraft projects related to agricultural modifications indicates that the CAM 8 philosophy was interpreted to allow the use of AC 43.13-2A structural criteria for most STC's issued through the early 1980's. Since then more restrictive guidance based on CAR 6 and FAR 27 requirements has been applied by some ACO's to several STC applications. Since the more restrictive guidance imposed a significant economic burden on the industry, the HAI requested a meeting with the FAA during the 1990 annual convention in Dallas. As a result of the meeting, an Action Notice to clarify the interpretation of FAR 21.25(a)(1) for restricted category aircraft has been issued.

(4) The following advisory material is a result of a reassessment of past and present policy.

b. Procedures. The certification basis for agricultural dispensing equipment in the restricted category is FAR 21.25(a)(1) as interpreted by Action Notice 8110.22. The accountable Directorate guidance for the substantiation requirements for rotorcraft is as follows:

(1) Substantiation of the agricultural dispensing system hoppers or spray tanks to the load factors provided in figure AC 29 MG 5-1 provides for proof of structure. The load factors of figure AC 29 MG 5-1 address the critical structural load conditions of dispensing equipment mounted in or near the fuselage and provide adequate margins of safety.

FIGURE AC 29 MG 5-1
ACCEPTABLE ULTIMATE LOAD FACTOR FOR
AGRICULTURAL DISPENSING EQUIPMENT DESIGN

	UP	DOWN	SIDE	FORWARD	AFT
Tanks & Equipment Mounted In Or Near The Fuselage	1.5g	4.0g	2.0g	4.0g Note 1	- - - -
Spray Booms	1.5g	2.5g	- - - -	Note 1	2.5g Note 2

Note 1: An ultimate load factor of 2 G's is acceptable for externally side or under fuselage mounted tank and forward mounted spray booms where failure in a minor crash landing will not create a hazard to occupants or prevent exit from the rotorcraft.

Note 2: The aft loads for spray booms may be developed by the applicant based on the 111 percent of V_{NE} for which certification is requested or the load factors of figure AC 29 MG 5-1, whichever is greater.

(2) The applicant may elect to substantiate his/her product by either static or dynamic testing, by analysis, or any combination thereof.

(3) Lower load factors may be used only when justified by manufacturer's data, rational analysis, or actual rotorcraft flight and ground load demonstrations.

(4) Tank pressure test, while not mandated, is recommended for safety reasons. An acceptable procedure is included in paragraph AC 29 MG 5(c)(4).

(5) Dispensing equipment installation attach points. If attach points exist which are an integral part of the rotorcraft and these attach points have been certified to the standard category requirements no further substantiation of the attach point is required if an analysis indicates the dispensing system does not impose loads which exceed those for standard category certification.

(6) Ground clearance for dispensing equipment installation. A 5-inch ground clearance has typically been used for skid gear equipped rotorcraft which incorporate belly mounted supply tanks/hoppers or systems which have dual side mounted supply tanks/hoppers and the design incorporates cross tubes or other system components which are located beneath the bottom of the fuselage when these components are rigidly attached to the airframe structure. The 5-inch dimension is measured vertically from the ground to the lowest point of the installed system, with the rotorcraft in its operational configuration and gross weight (including disposable load) and while resting on a smooth, level asphalt surface. For rotorcraft equipped with wheels and/or landing gear struts, the maximum system deflections should be considered when determining the 5 inches of acceptable static ground clearance. The 5-inch ground clearance would only apply to original configuration of newly manufactured rotorcraft. However, a 3-inch ground clearance has been found acceptable and may be approved for skid gear equipped rotorcraft to account for the in-service permanent set allowed for skid gear members, (i.e., cross tube deflections allowed per the maintenance manual). Cable supported systems, (i.e., cargo hook installations) or dispensing systems utilizing flexible ducts (certain types of dry material dispensing equipment which may or may not be retractable) have been approved even though portions of the system may contact the surface during a normal landing.

(7) A number of rotorcraft are approved for external cargo operations that allow a gross weight higher than the approved internal gross weight limit. This difference is usually due to the allowable weight limit restriction of the landing gear. (The gear is not approved for the higher weight.) Those types of dispensing equipment, that can be loaded in flight to a weight that exceeds the allowable limit of the landing gear should incorporate a reliable means that rapidly reduces the total aircraft gross weight to within allowable landing gear limits. In most cases, this will involve jettison of the disposable load. The time interval for this operation should be demonstrated, and should not exceed a recommended 3 seconds from a level flight condition.

(8) A flight check or demonstration of the agricultural dispensing equipment installation is normally conducted. This flight check should also qualitatively determine that no hazardous deflection or resonance in the rotorcraft or dispensing system exists. For FAA flight operations approval, this flight check must be conducted in accordance with the requirements of FAR 133.41. In addition, recent service history has shown that external equipment and external fixture modifications can affect main rotor mast bending loads. In lieu of a mast bending survey, a pre and post modification flight test may be conducted at identical weights, center-of-gravity (CG), power, and density

altitude to compare a critical control position parameter (typically longitudinal cyclic stick position) at pre and post modification Vne airspeeds.

(i) If required, the post modification Vne should be reduced so that the post modification longitudinal cyclic stick position is slightly aft of (or less than) the pre modification stick position. This alternative procedure assumes that the static longitudinal stability of the helicopter has not been altered by the modification. For helicopters with neutral static stability, a more comprehensive investigation may be required.

(ii) In some cases, a control position parameter other than longitudinal stick position may be critical. For example, a heavy external device mounted to the side of the helicopter that gives a lateral CG close to the limit and an asymmetric yaw component would require pre and post modification lateral cyclic stick and pedal position measurements. Operating limitations other than Vne may need to be established, or reduced from pre modification limitations, to ensure pre modification mast bending is not exceeded.

(9) For rotorcraft certificated in dual categories, the inspection requirements of FAR 21.187(b) must be observed when converting from restricted to normal category.

c. Acceptable Means of Compliance.

(1) Analysis Method. Structural analysis (static) may be used if the structure is of a configuration for which experience has shown the method to be reliable. Structural substantiation of tanks that are designed to contain liquid materials may be accomplished by pressure testing. For tanks or hoppers designed to contain dry material, (e.g., dust or fertilizer) static load tests may be used to verify structural integrity. The tank/hopper, mounting hardware, and support structure should all be substantiated to the load conditions specified by this paragraph considering the effects of internal fluid pressures when applicable.

(2) Static Tests. Static tests of tank/hoppers, mounting hardware, and support structure for each critical load condition may be accomplished using conventional techniques; such as, dead weight loading, whiffletree systems, and hydraulic rams. If tests of the tank and its mounting hardware are conducted using a test fixture representing the rotorcraft, the rotorcraft support structure may be substantiated independently by means of test and/or analysis. Static test loads should be applied in combination with associated internal fluid pressure loadings. The ultimate loads specified in paragraph AC 29 MG 5 should be sustained for at least 3 seconds without failure.

(3) Dynamic Tests.

(i) If the applicant elects to test to the load factors noted herein, the maneuvering and gust loadings will be considered to be adequately substantiated. For each condition, the critical volume and density of fluid should be used.

(ii) The tank and mounting hardware should support ultimate loads without detrimental permanent set or failure, respectively. The rotorcraft support structure may be included in the dynamic tests, or it may be substantiated separately via static test and/or analysis for each condition specified by this paragraph.

(4) Pressure Testing. Internal pressure loads may be applied using the water standpipe technique. Standpipe water height should be accurately computed for each critical spray tank static test loading. Pressure testing of spray tanks is not absolutely essential but is recommended for safety reasons. This testing will also determine whether the joints and connections are tight and will not leak in addition to determining any weak spots in the construction. Where spraying is done with highly volatile and flammable liquids, or where the tank has a return line, such as in an engine oil tank where the fluid is pumped back into the tank, it is recommended that the tank be tested for a pressure of 5 pounds per square inch. For other liquids, and where no fluid return line is used, testing to 3 ½ pounds per square inch should be satisfactory. There are many ways of pressure testing a tank, however, it is believed that the simplest and easiest method is to fill the tank with water and use a standpipe filled with water. A 1 1/8-inch pipe can be connected to the venting tube or one adapted to the filler opening. In either case the height of the pipe would be the same. For a 3 ½ PSI test of the tank the height of the water in the pipe would only need to be 8 feet and for a 5 PSI test only an 11 ½ -foot height of water will be needed.

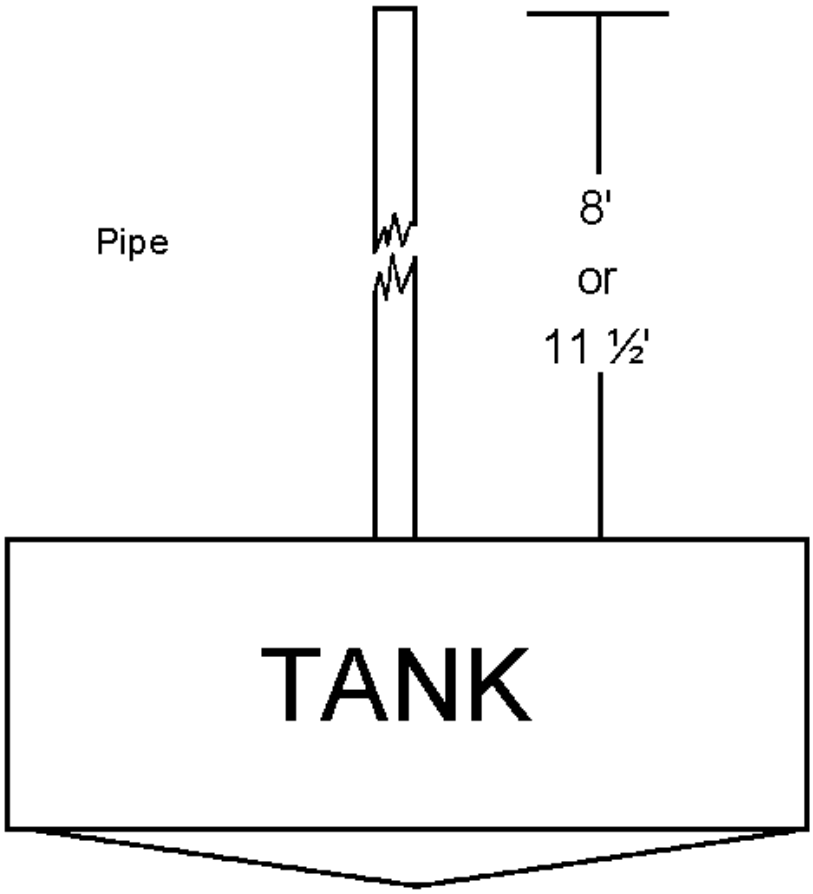

Figure AC 29.MG 5-2 Sketch of Tank Pressure Test

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 6. EMERGENCY MEDICAL SERVICE (EMS) SYSTEMS, INSTALLATIONS, INTERIOR ARRANGEMENTS, AND EQUIPMENT.

a. Explanation. This paragraph pertains to EMS configurations and associated rotorcraft airworthiness standards. EMS configurations are usually unique interior arrangements that are subject to the appropriate airworthiness standards, FAR Part 29 or its predecessor CAR Part 7, to which the rotorcraft was certificated. No relief from the standards is intended except by § 21.21(b)(1) and exemption. EMS configurations are seldom, if ever, done by the original manufacturer.

(1) The FAA/AUTHORITY has not specified in the airworthiness or operating rules the minimum equipment for an EMS configuration. Whatever equipment is presented for evaluation and approval is subject to compliance with the airworthiness standards. Any equipment that is not essential to safe operation of the aircraft is evaluated for a “no hazard approval,” i.e., it is optional equipment and may be approved provided the use, operation, and possible failure modes of the equipment are not hazardous to the aircraft. Safe flight, safe landing, and prompt evacuation of the rotorcraft, in the event of a minor crash landing, for any reason, are the objectives of the FAA/AUTHORITY evaluation of interiors and equipment unique to EMS.

(i) For example, a rotorcraft equipped only for transportation of a nonambulatory person (a police rotorcraft with one litter) as well as a rotorcraft equipped with multiple litters and complete life support systems and two or more trained attendants/medical personnel may be submitted for approval. These configurations will be evaluated to the airworthiness standards appropriate to the rotorcraft certification basis.

(ii) Transport rotorcraft should comply with many flightcrew and passenger safety standards which dictate features of the basic certified rotorcraft which are related to the interior arrangement, to the doors and emergency exits, and to occupant protection. Compliance with the airworthiness standards results in an emergency interior lighting system, placards or markings for doors and exits, exit size, exit quantity and location, exit access, safety belts, and possibly shoulder harnesses or other restraint or passenger protection means as a part of a rotorcraft type design. The features, placards, markings and “emergency” systems which are required as a part of the type design should be retained unless specific replacements or alternate designs are necessary for the EMS configuration to comply with the airworthiness standards.

(2) Many EMS configurations of transport rotorcraft are equipped with the following:

(i) Attendant/medical personnel seats which may swivel.

(ii) Multiple litters, some of which may tilt.

(iii) Medical equipment stowage compartments.

(iv) Life support and other complex medical equipment.

(v) Incubators for infants.

(vi) Curtains or other interior light shielding for the flightcrew compartment.

(vii) External loud speakers and search lights.

(viii) Special internal and external communication radio equipment.

b. Procedures.

(1) General.

(i) Original type design information and criteria may or may not be available from the manufacturer. This may be “public,” not proprietary, information that is pertinent for interior modifications. It may be appropriate to include “standard” features, placards, and markings for the rotorcraft type design by reference in the applicant’s modification design data.

(ii) The EMS modification presented for approval usually contains equipment of one manufacturer’s model or design. The type design of the modification will have features to power and restrain the equipment, maintain the rotorcraft systems integrity, and to otherwise protect the occupants. See paragraph b(15) which refers to equipment substitution.

(2) Evacuation and Interior Arrangements. Access to the emergency exits/doors from any location in the cabin/compartment, access to and use of the exit/door opening means or release device, and the unobstructed area of the “standard” or type design exit are potential problems that should be addressed in the early design stage. Multi-litter arrangements may be especially critical.

(i) The operation or use of devices for locking the position of swivel seats, etc., and for rapid installation and removal of litters (incubators, etc.) should be labeled unless they are simple and obvious, and do not require exceptional effort. The design features of the device(s) and the seat and/or litter will influence the extent of information in any label necessary to insure proper and safe installation for routine use and for prompt evacuation when appropriate or necessary for the interior arrangement. The requirement for labels or markings (instructions, etc.) that applies to operation of

seat or litter features, release devices, etc., is not relieved even if trained attendants are necessary for an evacuation as discussed in paragraph c(2)(v). Placards or instruction cards that contain evacuation procedures do not necessarily contain detail procedures for individual seats, litters, and so forth. Release devices that are simple and obvious and do not require exceptional effort are recommended. For example, a single central control for litter release would be preferred over multiple action release devices. Seats and litters which require multiple actions or steps to position or release for an emergency evacuation for the effect on achieving a rapid evacuation determines it is acceptable after minor crash landing may be acceptable if an evaluation determines it is acceptable.

(ii) The passenger compartment or cabin should not be partitioned to impede access to the exits. A person seated in the compartment should have access to each exit in the compartment. All persons must be able or have provisions to rapidly clear (evacuate) the rotorcraft as specified in § 29.803(a). A demonstration or a "walk-through" of appropriate procedures may be necessary to assure the means and procedures are feasible and adequate. In certain designs and arrangements, an evacuation demonstration may be necessary to prove questionable interior and emergency exit/door arrangements.

(iii) Although not a standard, 90 seconds to clear the rotorcraft exits should be used as the time interval whenever an evacuation demonstration is dictated. Attendants and the flightcrew, trained in the evacuation procedures, may be used to remove the litter patients. It is preferable for the patient to remain in the litter; however, the patient may be removed from the litter to facilitate rapid evacuation through the exit. The patients are not ambulatory during the demonstration. The demonstration may be conducted in daylight with an upright rotorcraft. Exits on one side (critical side) should be used. Exits on the other side are blocked (possibly by a fire).

(iv) Special evacuation procedures and trained attendants may not be required for simple and obvious means of evacuation for a single litter. Procedures may be prominently displayed in durable markings, placards, cards, and condensed or summarized in the emergency procedures section of the Rotorcraft Flight Manual (RFM) or an EMS configuration RFM supplement.

(v) If any medical attendants are required for evacuation, the attendants should be trained in these procedures and listed in the Limitations section of the RFM or a supplement as a required crewmember. If attendants are not essential for safe rotorcraft operation or rapid evacuation, then an attendant is not a required crewmember.

(3) Restraint of Occupants and Equipment. The minor crash conditions specified in § 29.561(c) usually dictate all but the vertical (down and possibly up) load conditions. The flight and landing loads, such as +3.5g limit (flight) vertical override the minor crash loads when they are larger. See paragraphs AC 29.561 and AC 29.785 for further information.

(i) Whether seated or recumbent, the occupants must be protected as prescribed in § 29.785. Swivel seats and tilt litters may be used provided they are substantiated for the appropriate loads for the position selected for approval. Placards or markings may be used to assure proper orientation for flight, takeoff, or landing. The seats and litters should be listed in the type design data for the configuration. See paragraph (b)(15) for substitutions.

(ii) For recumbent occupants, harnesses, straps, a padded headboard, a diaphragm, or safety belts may be used if adequate for the forward and lateral loads of § 29.561(c). Harnesses/straps are recommended, however. When harnesses/straps are used, they should prevent the occupant from significant forward motion (4 g condition) that would remove the support from the head as well as the shoulder for "head down" motion. Infants in incubators should be similarly protected by padding and containment within the incubator and the incubator restrained for the load cases noted. If the infant is strapped to a removable platform, the platform and infant should be properly restrained within the incubator for the load cases noted. The incubator should be listed in the type design data for the configuration.

(iii) Incubator materials are subject to the flammability standards of § 29.853. Evacuation procedures should include incubators if a part of the interior.

(iv) Galleys, medical supplies, and equipment compartments or modules should be restrained and the individual compartments must also contain the contents for the conditions noted in paragraph (b)(3). Durable placards, decals, or markings should be used where appropriate to limit the maximum weight of any compartment and the whole module. Compartment latches having sufficient strength and displacement/engagement should be used to contain the compartment contents for the conditions noted. If necessary, a static load test or analysis should be employed to ensure the container/compartment remains intact and the latch does not disengage for the most critical conditions. Unrestrained (loose) contents in an individual compartment, in combination with similar compartments, should require use of a magnification factor with the conditions noted. Prudent design and location of compartments having heavy, unrestrained (loose) equipment will mitigate the potential effects of minor crash impact loads.

(4) Flammability Standards for Materials. Interior materials shall meet the flammability standards appropriate for the rotorcraft type design; § 29.853.

(i) For rotorcraft certified prior to adoption of Amendment 29-17 (1978), the cabin materials shall be at least flash resistant and wall, ceiling linings, the covering of all upholstery, floors, and furnishings shall be at least flame resistant. Advisory Circular No. 23–2, Flammability Tests, dated August 20, 1984, contains test information about flash and flame-resistant material.

(A) Flash-resistant material may be characterized as that not exceeding a 20-inch-per-minute (horizontal) burn rate. See AC 23-2 for further information.

(B) Flame-resistant material may be characterized as that not exceeding a 4-inch-per-minute (horizontal) burn rate.

(C) For incubators, transparencies must be flash resistant and fabric (padding, covers) straps, etc., must be flame resistant according to the standard.

(ii) For rotorcraft certified to Amendment 29-17 adopted in 1978, the materials shall be self-extinguishing as specified in the standards. For example, transparencies shall be self-extinguishing as prescribed in § 29.853(a)(2).

(iii) Additionally, for rotorcraft certified to standards of Amendment 29-23 (1984), cushions of each passenger seat must have a "fire blocking layer" as prescribed in § 29.853(b).

(iv) The applicant is urged to use self-extinguishing materials regardless of the certification basis.

(v) For further information on materials, refer to paragraph AC 29.853. Advisory Circular 23-2, "Flammability Tests," dated August 20, 1984, also contains information about flash-resistant and flame-resistant material tests.

(5) Exit Signs/Markings and External Markings. The approved exits require certain signs, instructions, and identification. The rotorcraft type design contains the required data. The maintenance manual and the RFM should also contain this information. See paragraph AC 29.811 for more information. Alternates may be approved which then become part of the applicant's type design data. All U.S. transport rotorcraft presently in service should have an emergency interior lighting system to comply with § 29.811(f). (Refer to the certification basis of the rotorcraft.)

(6) Interior or "Medical" Lights. The view of the flightcrew must be free from glare and reflections that could cause interference. Curtains that meet flammability standards may be used. Complete partition or separation of the crew and passenger compartment is not prudent. Means for visual and oral communication are usually necessary. Paragraph AC 29.773 concerns pilot visibility.

(7) Patient Interference. When passengers or patients are located in close proximity to the pilot and the primary flight controls of the rotorcraft, a guard or shield should be installed or the patient restrained to prevent inadvertent or potential convulsive interference with safe operation of the rotorcraft. The guard may be a part of the rotorcraft interior features. In addition, rapid evacuation should be assured if a guard is used.

(8) External Devices.

(i) Search lights, loud speakers, baggage pods, etc., may be installed on the underside of or elsewhere on the rotorcraft. The strength of the attachments shall be substantiated for the flight and landing conditions, and for the minor crash conditions where applicable. Pilot visibility should not be affected adversely by lights or light reflection.

(ii) The device or pod located on the underside of the rotorcraft should not contact a level landing surface after "limit landing load" deflection of the landing gear. The gear should deflect without causing damage to the device. For example, if the limit landing load deflection is 8 inches, the device shall have at least 8-inch clearance to avoid contact with the landing surface or have an equivalent feature of design. The physical characteristics of the rotorcraft design dictate the necessary clearance for landing gear deflection. In addition, the device should be designed and located on the rotorcraft to preclude penetration of the device into a critical area of the fuselage. For example, the device should be located to minimize the potential of penetration into a fuel line, fuel cell, primary control tube, or occupant seat for any reason.

(iii) A flight evaluation is necessary to determine the effects of the device on the rotorcraft flight characteristics and on flight crew visibility. In addition, recent service history has shown that external equipment and external fixture modifications can affect main rotor mast bending loads. In lieu of a mast bending survey, a pre and post modification flight test may be conducted at identical weights, center-of-gravity (CG), power, and density altitude to compare a critical control position parameter (typically longitudinal cyclic stick position) at pre and post modification V_{ne} airspeeds.

(A) If required, the post modification V_{ne} should be reduced so that the post modification longitudinal cyclic stick position is slightly aft of (or less than) the pre modification stick position. This alternative procedure assumes that the static longitudinal stability of the helicopter has not been altered by the modification. For helicopters with neutral static stability, a more comprehensive investigation may be required.

(B) In some cases, a control position parameter other than longitudinal stick position may be critical. For example, a heavy external device mounted to the side of the helicopter that gives a lateral CG close to the limit and an asymmetric yaw component would require pre and post modification lateral cyclic stick and pedal position measurements. Operating limitations other than V_{ne} may need to be established, or reduced from pre modification limitations, to ensure pre modification mast bending is not exceeded.

(9) Miscellaneous. Various paragraphs in this AC contain guidance for the standards cited in the reference list (reference c(1)). These paragraphs should provide insight into designing an EMS configuration that would be acceptable under the standards.

(10) Oxygen. EMS oxygen installations are supplied by either liquid or gaseous oxygen. Both types of systems are discussed in this paragraph.

(i) Liquid Oxygen.

(A) System General Description. This section covers specific requirements for liquid oxygen systems. Most liquid oxygen systems in use are installed in military aircraft and, as a result, much of this material is based on experience with these systems. A rotorcraft liquid oxygen system should be comprised of a liquid oxygen converter, tubing, fittings, quantity gage, heat exchangers, and appropriate pressure and flow control components as shown in figures AC 29 MG 6-1 and AC 29 MG 6-2. The installation may provide for replenishing the liquid oxygen supply by use of a quick-removable converter or, in the case of a fixed installation converter, by providing external access for connection to a portable service trailer. More complicated systems such as those with multiple converter assemblies are not discussed here since installation of those systems are not envisioned in rotorcraft at this time.

(B) System Components. All components should be aircraft qualified and suitable for use in an EMS rotorcraft application.

(1) Liquid Oxygen Converter. A liquid oxygen converter assembly is a self-powered system for the storage of liquid oxygen and for its conversion to gaseous oxygen when required. A principal part of the converter assembly is a vacuum insulated container. Pressure relief valves should be provided to allow the escape of gas generated when oxygen is not being expended in the supply line. Oxygen losses from a converter assembly vary from 5 to 20 percent per 24 hours depending on the size of the container, its installation environment, and so forth. Aircraft qualified and approved converters suitable for EMS rotorcraft use are available in either 5- or 50-liter capacities. Size selection should be determined by flow rate and duration requirements. Performance characteristics of each converter size are available from the manufacturer.

(2) Shutoff Valve Assembly. This valve shall be accessible to a flightcrew member and be mounted in the supply line on or as close as possible to the outlet of the converter. This valve provides for the confinement of the remaining supply of liquid oxygen to the converter in the event of an emergency. Since the system pressure is low, the use of an electrically actuated shutoff valve is satisfactory to accomplish this function. In some installations, where the evaporating coil is immediately adjacent to the converter, a flow fuse has been used to accomplish this function. Use of a flow fuse must be supported by a system fault analysis and testing to show maximum normal flow will not result in nuisance trips, and reliable trips will be provided for malfunction conditions resulting in excess flow.

(3) Filler Valve. Some designs combine this function with the build-up and vent valve assembly as shown in figure AC 29 MG 6-2.

(4) Build-up and Vent Valve Assembly. This valve is positioned in the "vent" position when the system is being filled with oxygen and in the "build-up position at other times. Some designs combine this function with the filler valve as shown in figure AC 29 MG 6-2.

(5) Pressure Build-up Coil Assembly and Pressure Closing Valve. With the build-up and vent valve in the "build-up" position gas that is formed is allowed to apply pressure to the liquid to provide adequate flow through the check valve to the evaporating coil assembly. A connection to a pressure relief valve is also provided.

(6) Evaporating Coil Assembly. This is provided to convert the liquid oxygen into a gaseous form. The evaporating coil assembly should be of sufficient capacity to maintain the design flow quantity to the dispensing regulators at a temperature within +10 and -20° F of cabin ambient temperature. MIL-D-19326G contains a discussion of installation considerations for this unit.

(7) Vent Line. Gaseous oxygen escapes through this line. At the conclusion of the fill operation, liquid oxygen will flow overboard in a steady stream from this line to indicate the container is full of liquid oxygen. The vent line should be located to drain overboard at the bottom of the rotorcraft fuselage. Flow from the overboard vent should be directed so as not to create a hazard for personnel and not allow liquid oxygen to impinge on the rotorcraft. The vent lines should be insulated to prevent frosting and sweating if they pass over equipment which will be harmed by water dripping from the lines, or drip pans should be installed under the lines. There should be no hydrocarbon fills or drains, forward or above, in proximity to the vent outlet.

(8) Regulator. A regulator should be installed in the supply line downstream from the heat exchanger. The regulator should reduce the liquid oxygen converter operating pressure to a supply pressure of 50 pounds per square inch gauge (PSIG) to be compatible with the normal operating pressure of medical oxygen equipment.

(9) Flow Control Valve. This valve provides a calibrated flow of gaseous oxygen from an operating supply of 50 ± 5 PSIG. A valve whose proof pressure is specified at 80 PSIG and has a burst pressure rating of 350 PSIG would be considered satisfactory.

(10) Check Valve. This valve prevents gaseous oxygen in the supply system from backing up into the liquid oxygen in the container and increasing the vaporization rate of the liquid oxygen by exposure to the gas. This valve is normally an integral part of the liquid oxygen converter assembly.

(11) Quantity Indicators. A quantity indicator should be installed at the appropriate rotorcraft crew station to permit monitoring of the liquid oxygen supply. The indicator when installed in the rotorcraft should indicate the amount of liquid oxygen in the converter. Adequate clearance should be provided for the indicator connectors so

that they can be readily disconnected by servicing personnel. Provisions should be made for the storage of the rotorcraft connectors to the liquid oxygen converter when they are disconnected. Liquid oxygen quantity indicating equipment is available in three types: capacitance gauging, electro-mechanical transducer indication, and differential pressure type indication.

(12) Pressure Relief Valves. Pressure relief valves are provided to vent overboard through the overboard vent system any excess pressures developing within the system.

(13) Lines. Lines should be either solid tubing or flexible hoses. Examples of acceptable solid tubing are aluminum alloy conforming to AMS 4071 or corrosion resistant annealed steel (304) confirming to MIL-T-8506. Flexible hoses should be used for rotorcraft system connections to removable converters and to other applications where relative movement may occur. Flexible hoses should be wire-braid-covered bellows or wire-braid-covered tetrafluoroethylene. Flexible hose conforming to MS90457 or MS24548 would be considered satisfactory. MS90457 hose is flexible to -297° F (-183° C), and MS24548 hose is flexible to -65° F (-54° C). Synthetic lines such as plastic, nylon, or rubber should not be used for lines subjected to continuous pressure, or for application where the line will not be visible. Lines that are not visible are those that are located behind liners or in the walls of the fuselage.

(14) Fittings. If in contact, dissimilar metals should be suitably protected against electrolytic corrosion. Line assemblies should be terminated with “B” nuts or a similar manufactured terminating connection. Universal adapters (AN 807) or friction nipples used in conjunction with hose clamps should be avoided for use in pressurized systems.

(15) Drain Valve. Systems that have permanently installed containers should include a drain valve located to allow for complete draining of the liquid oxygen container. An acceptable drain valve would be one in accordance with MK-V-25962 that is suitably capped. A cap in accordance with AN 929-5 with a permanently attached chain is a suitable cap.

(16) Low Pressure, Low Level Warning System. It is recommended that provisions be included in the system to alert the appropriate aircraft crew member that the level of the oxygen supply has reached some low level. It is recommended that low level be actuated when less than 10 percent of the full container capacity is available. If low system pressure is also monitored, the low pressure valve selected should be such that any drop in supply line pressure upon inhalation should not activate the low pressure warning function.

(C) Component Installation. The following are typical installation considerations that should be addressed when designing the oxygen system.

(1) Location. The oxygen equipment, lines, and fittings should be located as remotely as practicable from sources of flammable fluids, high heat and electrical items, fuel, oil, hydraulic fluid, batteries, exhaust stacks, manifolds, and so forth. Oxygen lines should not be grouped with lines carrying flammable fluids. If possible, converters should not be in line with the plane of rotation of a turbine. System components should not be installed in an environment that will exceed the temperature limit of the component, and no part of the system should be installed in an area that will exceed 350° F (176° C). To minimize loss due to heat, the liquid oxygen converter should not be located near equipment that dissipates a high quantity of heat.

(2) Converter Mounting. The oxygen container should be readily accessible to servicing personnel. If the container is not removable for servicing, the filler should be external to the aircraft with adequate contamination protection. Mounting provisions for the converter and plumbing to the evaporating coil assembly should include a drain pan with an overboard drain.

(3) Flexible Hoses. Hoses should be of sufficient length to provide unstressed connections and be protected against chafing on surfaces or objects which may damage the wire covering. The bend radius imposed on the hoses by the installation and during remove and replace actions should not be less than the minimum established by the hose specifications.

(4) Lubricants. No lubricants should be used on liquid oxygen pipe fittings. MIL-T-27730 Teflon tape may be used on male pipe fittings when required. Teflon tape should not be used on flared tube fittings, straight threads, coupling sleeves, or on the outer side of tube flares. None of the tape should be allowed to enter the inside of a fitting. Krytox fluorinated grease by E.I. Dupont De Nemours and Company, or an equivalent, may be used sparingly on seals.

(5) Tubing Routing and Mounting. There should be at least 2 inches of clearance between the oxygen system and flexible moving parts of the rotorcraft. There should be at least a ½-inch clearance between the oxygen system and rigid parts of the rotorcraft. The oxygen system tubing, fittings, and equipment should be separated at least 6 inches from all electrical wiring, heat conduits, and heat emitting equipment in the rotorcraft. Insulation should be provided on adjacent hot ducts, conduits, or equipment to prevent heating of the oxygen system. In routing the tubing, the general policy should be to keep total length to a minimum. Allow for expansion, contraction, vibration, and component replacement. All tubing should be mounted to prevent vibration and chafing. This should be accomplished by the proper use of rubberized or cushion clips installed at 24-inch intervals (copper) or 36-inch intervals (aluminum) and as close to the bends as possible. The tubing, where passing through or supported by the rotorcraft structure, should have adequate protection against chafing by the use of flexible grommets or clips. The tubing should not strike against the rotorcraft structure during vibration and shock encountered during normal use of the rotorcraft.

(6) System Marking. The rotorcraft should be permanently and legibly marked, as applicable, in the locations specified below (a minimum letter height of ¼ inch is recommended):

(i) Adjacent to the overboard vent opening:

CAUTION
LIQUID OXYGEN VENT

(ii) On outside surface of filler box cover plate:

LIQUID OXYGEN (BREATHING) FILL ACCESS

(iii) On underside surface of filler box cover plate:

CAUTION - KEEP CLEAN, DRY, AND FREE FROM OILS

(iv) In prominent place when filler box is open, preferably near liquid oxygen drain valve:

DO NOT OPEN DRAIN VALVE UNTIL DRAIN HOSE
AND DRAIN TANK ARE CONNECTED

(v) Other placards, such as one at the converter cautioning about the presence of liquid oxygen, may also be appropriate.

(7) Other installation criteria are given in Chapter 6, AC 43.13-2A, Acceptable Methods, Techniques, and Practices-Aircraft Alterations, dated June 9, 1977, and should be given full consideration.

(D) Precautions. The referenced SAE report contains precautions peculiar to a liquid oxygen installation, and this material should be reviewed. It should also be emphasized that liquid oxygen equipment and the aircraft being serviced must be electrically grounded during servicing to prevent an accumulation of static electricity and discharge. The following considerations are included for special emphasis:

(1) System Cleanliness. The completed installation should be free of oil, grease, fuels, water, dust, dirt, objectionable odors, or any other foreign matter, both internally and externally prior to introducing oxygen in the system.

(2) Closures. Lines which are required to be disconnected, due to the location of the converter within the rotorcraft during rotorcraft maintenance checks or overhaul, should be capped to prevent materials which are incompatible with oxygen from entering the system when the system integrity is broken. Caps which introduce moisture and tapes that leave adhesive deposits shall not be used for these purposes.

All openings of lines and fittings shall be kept securely capped until closed within the installation.

(3) Degreasing. All components of the oxygen system should be procured for oxygen service use in an “oxygen clean” condition. Parts of the oxygen system, such as tubing, not specifically covered by cleaning procedures should be degreased using a vapor phase trichloroethane degreaser. Ultrasonics may be used in conjunction with vapor phase degreasing for the cleaning of components.

(4) Purging. The system should be purged with hot, dry 99.5 percent pure oxygen gas in accordance with the manufacturers recommendations after:

(i) Initial assembly of the oxygen system; and

(ii) After system closure whenever the oxygen system pressures have been depleted to zero, or the system has been left open to atmospheric conditions for a period of time or is opened for repairs.

(5) Maintenance and Replacement. All parts of the oxygen system should be installed to permit ready removal and replacement without the use of special tools. All tubing connections and fittings should be readily accessible for leak testing with a leak test compound formulated for leak testing oxygen systems and for tightening of fittings without removal of surrounding parts.

(ii) Gaseous Oxygen.

(A) General. This guidance is intended to supplement the existing guidance in AC 43.13-2A, Chapter 6. If there are any differences within the two AC’s, this guidance shall prevail since it pertains specifically to Part 29 requirements.

(B) System Components.

(1) High Pressure Cylinders. Many installations utilize hospital type cylinders rather than aviation type cylinders. A concern with the hospital type cylinders is the yoke and the hard plastic washer that is commonly used with these cylinders. It is very difficult to properly attach these yokes since the rotorcraft provides a high vibration environment and no positive lock is provided. Leaks are a continuous problem with this configuration. Yokes are available for these bottles that provide for a positive lock. Improved washers that provide for a good elastomeric seal and include a metal ring to limit crushing the washer are also available. If the hospital type bottles are to be used, only the modified yokes and improved seals should be considered for future installations. The preferred cylinder is the aviation type cylinder with the integral shut-off valve and regulator. All cylinders should be DOT approved.

(2) Lines.

(i) General. Any lines that pass through potential fire zones should be stainless steel.

(ii) High Pressure. Use of high pressure lines may be necessitated by the use of a pressure regulator that is remote from the cylinder. The intent is to locate the regulator as close as physically possible to the cylinder, and to minimize the use of fittings. Lines of 6-inch lengths are encouraged with 18-inch lengths being the maximum in unusual circumstances. Lines made of stainless steel are recommended.

(iii) Low Pressure. Although lines may only be subjected to low pressures, if they are located behind upholstery or for any reason are not 100 percent visible during normal operation, they should be solid metal lines or high pressure flexible lines such as Aeroquip 300 series hose, or Stratoflex 124, or 170 series hose assemblies. The so called “green lines” should only be used in locations that are 100 percent visible during normal operation. This would restrict their use to the run between the mask and the bulkhead disconnect in the aircraft cabin. Synthetic lines such as plastic, nylon, or rubber cannot be recommended for applications that will be exposed to continuous pressure (i.e., as opposed to pressurized when needed). These materials can cold flow.

(3) Fittings.

(i) High Pressure. Intercylinder connections are made with regular flared or flareless tube fittings with stainless steel. Usually fittings are of the same material as the lines. Mild steel or aluminum alloy fittings with stainless steel lines are discouraged. Titanium fittings should never be used because of a possible chemical reaction and resulting fire. An example of a series of fittings that has been accepted is the “SS” series Swagelok tube fittings (flareless).

(ii) Low Pressure. Fittings for metallic low pressure lines are flared or flareless, similar to high pressure lines. Line assemblies should be terminated with “B” nuts in a similar manner to a manufactured terminating connection. Universal adapters (AN 807) or friction nipples used in conjunction with hose clamps are not accepted for use in pressurized oxygen systems.

(4) Shut-off Valve. Each system should contain a shutoff valve that is located as close as practical to the high pressure cylinder(s), and it should be assessable to a flightcrew member. High pressure cylinders should use slow opening/closing system shut-off valves. Where the regulator is part of the cylinder, and low pressure oxygen is controlled, the emphasis in slow acting valves is not as significant, and use of a flow fuse may be possible. Use of a flow fuse must be supported by a system fault analysis and testing to show maximum normal flow will not result in nuisance trips, and reliable trips will be provided for malfunction conditions resulting in excess flow.

(5) Regulators. The regulator should be mounted as close as possible to the cylinders (reference b(9)(ii)(B)(2)(i)). If nonaviation qualified regulators are to be considered, their service history should be reviewed and careful consideration should be given to the manufacturer's environmental qualification. Radio Technical Commission for Aeronautics Document D0-160 is a recognized and accepted standard for environmental considerations. As a minimum, consideration should be given to operation during altitude, temperature, and vibration extremes.

(6) Placards. Appropriate placards should be provided with the installed system. Emphasis should be placed on any precautions that are appropriate during filling of the system and so forth.

(7) Filler Connections. When a filler connection is provided, it is recommended it be located outside the fuselage skin or isolated in a manner that would prevent leaking oxygen from entering the rotorcraft. Careful evaluation should also be made of any nearby sources of fuel, oil, or hydraulic fluid under normal or malfunction conditions. Each filler connection should be placarded. In addition, any valves (on aircraft or ground servicing equipment) associated with high pressure should be slow acting.

(C) "Provisions Only" Considerations. In some instances systems are approved that only include provisions for a supply system consisting of the high pressure cylinders, regulators, and their associated lines and fittings. In these instances, a placard should be provided that refers to a supply system that is considered satisfactory for the remainder of the installation. Other supply system configurations may very well be acceptable; however, they should be evaluated by the ACO that evaluated the original installation. An example of an acceptable placard for this situation is:

> Oxygen Supply System must be in accordance with the requirements given in STC SH _____. Deviations to the configuration specified must be evaluated and approved by the Manager (include reference to the appropriate FAA ACO).

(11) Medical Communication Equipment. This equipment is provided to allow for communication between the rotorcraft and ground medical personnel. It includes voice communication and may also include telemetry equipment for the transmission of graphic data. It should be demonstrated that this equipment functions, and the range at which this determination was made should be recorded in the project file. The functional demonstration should include a 360° turn (clockwise and counterclockwise) to assure no significant sections of signal blanking exists. The remainder of the emphasis on this equipment should be to assure that operation of this equipment does not interfere with normal operation of any avionic systems whose installation is required for safe operation of the rotorcraft.

(12) Cabin Lighting. EMS interiors normally include higher intensity cabin lighting than other interiors. This lighting capability should be carefully evaluated to ensure it does not interfere with operation of the rotorcraft. In some installations a special curtain is required to separate the cockpit from any interference by the lighting. The FAA/AUTHORITY project file should include a short discussion of how this evaluation was conducted. See paragraph b(6) for other curtain considerations.

(13) Other EMS Equipment. These items of equipment that are installed for the EMS mission are considered to be optional equipment and should be operated to assure they function properly. This evaluation would normally be done by someone that is knowledgeable about the particular type of equipment since correct operation of the equipment is essential to a valid determination that the required rotorcraft systems are not being interfered with. This includes all removable pieces of medical equipment that are used for patient care. The primary purpose of the evaluation of this equipment is to emphasize the possibility of any adverse interference between operation of the EMS equipment and the systems whose installation is required for safe operation of the aircraft, the adequacy of the installation provisions, and assurance that failure modes will not result in a hazardous condition for the rotorcraft.

(14) Miscellaneous. The following areas are not peculiar to EMS installations; however, their significance is enhanced by the complexity of an EMS installation.

(i) Compatibility. Many EMS installations are a collection of several STC's and may also include some "FAA/AUTHORITY field approvals." For this situation it should be shown that the overall installation provides for safe operation of the aircraft. Operation of a search light, if included, should be emphasized since this system can be difficult to keep out of the cockpit.

(ii) Electrical Load Analysis. An electrical load analysis should be conducted, and additional guidance is available in paragraph AC 29 MG 1. If the analysis indicates the generator(s) can be overloaded, appropriate measures should be taken to account for the problem. In some instances a placard that specifies certain operating limitations may be satisfactory while in other instances an electrical interlock may be in order. In general, if the amount of overload is relatively small, the placard solution will probably be satisfactory; whereas if the amount of possible overload is significant, it is more likely that an interlock scheme will be necessary.

(iii) Aircraft Grounding. It should be emphasized in an appropriate place in the STC data (RFM, maintenance information, etc.) that any time the EMS systems are being operated or serviced (oxygen for example) on the ground, the rotorcraft itself must be grounded.

(iv) Electrical Outlets. All electrical outlets provided in the cabin should be the three-prong grounded type. When not in use, these outlets should be suitably protected against the entry of fluids.

(v) Placards. All medical outlets should be placarded (air, oxygen, vacuum, etc.). Electrical power outlets should be placarded for type of voltage and amperage capacity. A placard stating no smoking when oxygen is in use should be included. Other placards would include information appropriate to the oxygen system, operation of special controls, etc.

(vi) Equipment in Cargo and Baggage Compartments. When components are added to the compartment, revisions should be made to protect the system components due to shifting cargo. In addition, when oxygen components are installed, the compartment should be placarded against the storage of oil or hydrocarbons. A smoke detector is recommended for a compartment if oxygen cylinders are installed in a closed, nonaccessible compartment. Also, the compartment weight limitations placard should be changed. Paragraph AC 29.787 pertains to cargo and baggage compartments.

(15) Equipment Substitution. The EMS modification that is presented for approval will contain specific items of equipment, and the approval will make reference to this equipment. If other equipment (new model, manufacturer, etc.) is to be substituted, then an evaluation should be made to assure the substitute equipment is also satisfactory. This evaluation would normally consist of comparing the attachment means, design features, failure modes, specifications, and operation of the two units. The purpose of the evaluation is to assure there are not differences that have an adverse effect on the airworthiness of the installation. Other differences would not be considered significant. A specific seat and litter design is approved as a part of the EMS configuration. Substitutions may be approved in accordance with the standards.

c. Related FAR Sections and References.

(1) FAR Sections. §§ 29.337, 29.471, 29.561, 29.773, 29.783, 29.785, 29.803, 29.805, 29.809, 29.811, 29.813, 29.815, 29.831, 29.853, 29.1301, 29.1309, 29.1353, 29.1357, 29.1411, 29.1431, 29.1557(d), 29.1583(d), 29.1585, and 29.1589.

(2) Other References.

(i) Helicopter Association International, Emergency Medical Services Recommended Guidelines, 1987, First Revision, 2 pages.

(ii) National Highway Traffic Safety Administration, Air Ambulance Guidelines, dated 1981.

(iii) FAR Part 91, General Operating and Flight Rules.

(iv) FAR Part 135, Air Taxi Operators and Commercial Operators.

(v) AC 67-1, Medical Information for Air Ambulance Operators, dated March 4, 1974.

(vi) AC 23-2, Flammability Tests, dated August 20, 1984.

(vii) Oxygen Equipment for Aircraft, Society of Automotive Engineers Aerospace Information Report No. 825B, Rev. 9/86.

(viii) Acceptable Methods, Techniques, and Practices--Aircraft Alterations, AC 43.13-2A, dated June 9, 1977.

(ix) Design and Installation of Liquid Oxygen Systems in Aircraft, General Specification for Military Specification MIL-D-19326G, dated October 1, 1985.

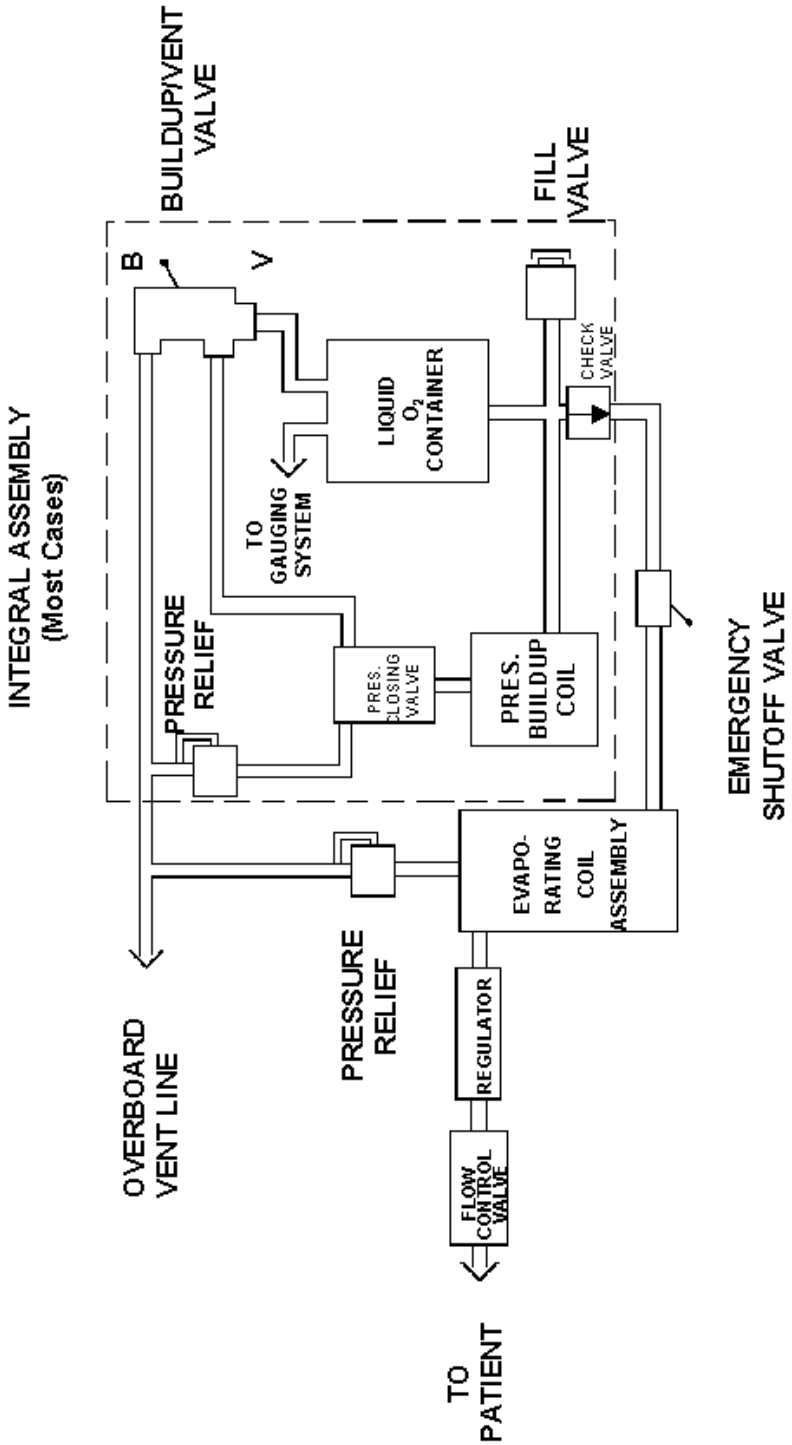

FIGURE AC 29.MG 6-1 TYPICAL LIQUID OXYGEN SYSTEM

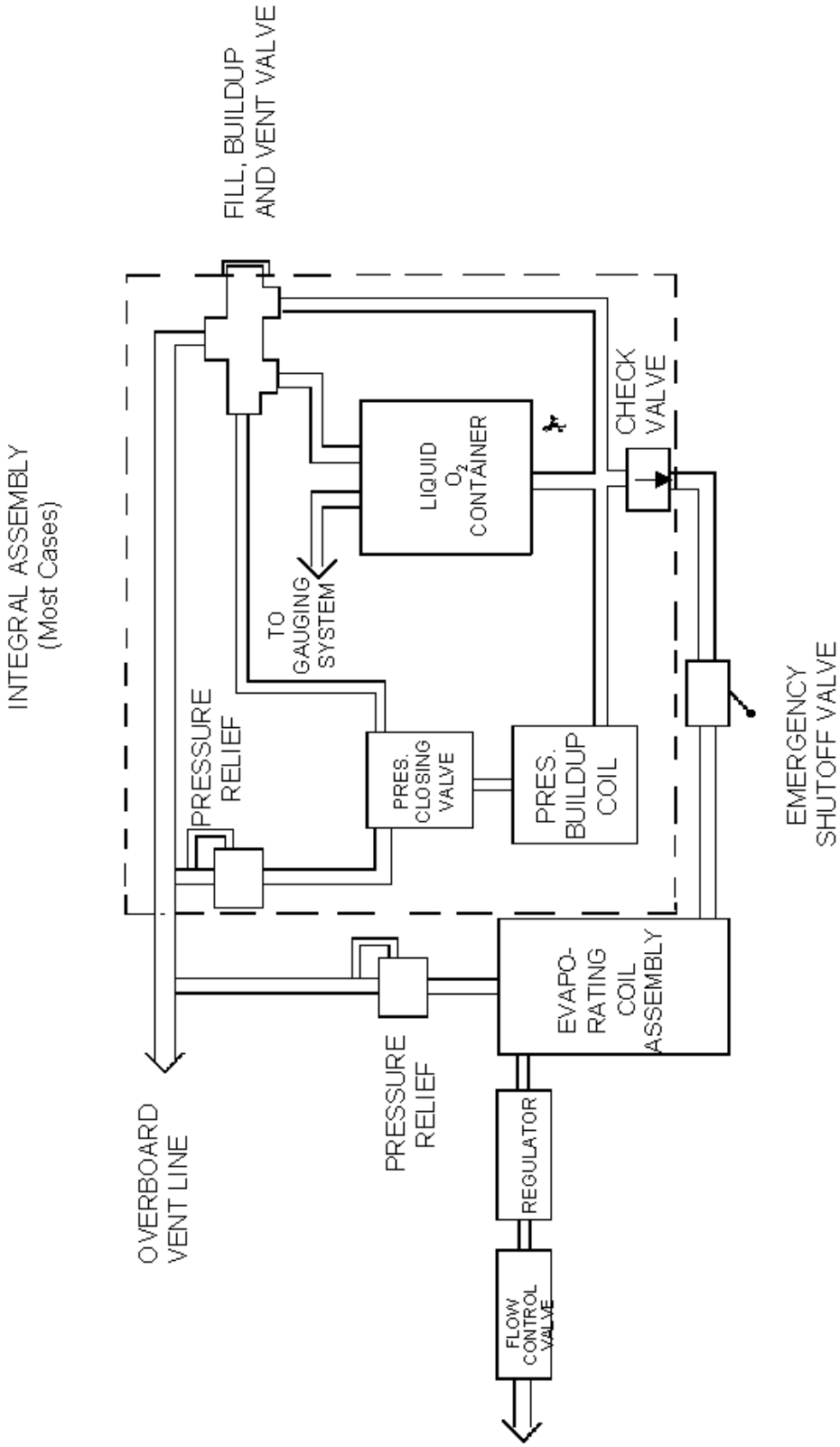

FIGURE AC 29.MG6-2 TYPICAL LIQUID OXYGEN SYSTEM - USING COMBINATION VALVE

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CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 7. STRUCTURAL CONDITION INDICATORS.

a. Related Sections. § 29.301 - Loads; § 29.305 - Strength and Deformation; § 29.571 - Fatigue Evaluation of Flight Structure; § 29.1301 - Function and Installation; § 29.1309 - Equipment, Systems and Installations; § 29.1321 - Arrangement and Visibility; § 29.1322 - Warning, Caution, and Advisory Lights; § 29.1355 - Distribution System; § 29.1503 - Airspeed Limitations: General; and § 29.1529 - Instructions for Continued Airworthiness.

b. Background.

(1) Structural condition indicators have been used on rotorcraft for several years in two main programs: as part of the basic type design and as part of airworthiness directive (AD) action. When approved as part of the basic type design, only limited “credit” has been given for the installation of structural condition indicators; i.e., components provided with a structural condition indication system were required to be designed to § 29.571 “safe-life” criteria considering the structural condition indicator system inoperative. So-called “nonhazard” approvals were granted. When used as part of the mandatory actions of ADs, structural condition indicators have had a degree of “credit” recognized, primarily in the recognition of “fail-safety” provided by the indicator system.

(2) Since structural condition indicators have been used during both original type design and AD issuance, and since there is movement toward increased damage tolerance in rotorcraft design, policy concerning condition indicator use is considered appropriate.

c. At present, the use of structural condition indicators alone on new type designs is not considered an acceptable substitute for providing the necessary safe life for each component. However, areas which may be considered when approving these indicators for fail-safety credit are delineated in the following paragraphs.

d. What, how, when, where, and who of structural condition indicators.

(1) Indication of what?

(i) Previous structural condition indicators have primarily been used for crack detection. Several types of through-the-thickness crack detection systems are currently in use. Two types which detect changes in pressure in an instrumented chamber due to gas movement through a cracked wall are known as the blade

inspection method (BIM) system and the integral spar inspection system (ISIS). These systems can only detect full-depth cracks which are large enough to allow loss (of gain) of pressure from the instrumented chamber. This presents a limitation since full-depth cracks may be fast growing before detection. Another through-the-thickness crack method is a pressurized, dyed fluid or oil system to detect through cracks in specially designed bolts (NASA patent), spindles, pins, or other closed chamber mechanical equipment.

(ii) Surface cracks can be found by systems such as surface-mounted crack detection wires. These systems would allow a greater safe crack growth period for assuring safe landing after detection than the through-crack-detection systems, but they have been used little in operations because of significant limitations; e.g., complexity of installation, durability problems, limited areas of coverage, and strain level limitations.

(iii) Some aircraft have had mast moment indicators or other load indicators to help prevent the pilot from inadvertently applying a high load to the instrumented system or to help the pilot reduce the load by control movements. These load indicators only indirectly give indications of structural condition; therefore, only limited “credit” is allowed for this use. “Credit” is limited in that the fatigue life substantiations of § 29.571 should consider a reasonable number of excursions into the higher ranges established for the load indicator, and special inspections, rework, or replacement instructions should be provided for any strength degradation associated with high range excursions.

(2) <u>How indicated</u>?

(i) Current BIM systems use two types of indicators. The visual blade inspection method (VBIM) uses a gauge mounted on the blade which must be read visually by maintenance personnel while the aircraft is parked. The cockpit blade inspection method (CBIM) uses lights mounted in the cockpit which may be monitored by the crew. Other pressurized chambers have used dyes or oils to improve visual inspection effectiveness. Mast moment indicators and other load indicators use instruments with marked ranges and needles.

(ii) No specific types of load indicators are required by the FAA/AUTHORITY but the type used should be evaluated for accuracy, readability, and overall effectiveness. Paragraphs AC 29 MG 7(e) and (f) cover, in more detail, the use of structural condition indicators.

(3) <u>When indicated</u>? Structural condition indicators are used before flight, during flight, and for normal maintenance inspections. Paragraphs AC 29 MG 7 (e) and (f) contain guidance for cockpit-mounted instruments which are monitored during flight. Indicators used for normal maintenance inspections are the preferred type since they can be scheduled to allow the most effective use of available maintenance personnel of well-equipped maintenance facilities and of parts available.

(4) Where indicated? Indications on the component are provided by VBIM systems and by systems utilizing dye or colored oil leakage. Cockpit-mounted lights and gauges may be used for certain critical structures which require frequent, but simple, checks. Maintenance panel locations (cabin, equipment bay, etc.) are the preferred locations for use in routine maintenance.

(5) Who reads indicators? The flightcrew, of necessity, monitors indicators mounted in the cockpit for use during flight. Gauges with ranges of values representing mast bending moments or other structural loads are monitored by the flightcrew, as necessary, to reduce or to prevent control operations from imposing excessive loads or to prevent too many high load applications. Maintenance personnel are generally responsible for reading component-mounted indicators and for monitoring indicators which are mounted on maintenance panels. The before-flight checks may be conducted by maintenance personnel or by flightcrew in certain cases (i.e., cockpit-mounted gauges or “push-to-test” checks).

e. Actions required by indicators.

(1) On-ground indications. Indications noted on the ground should be followed by a functional check of the indication system as provided for by its design. If indications persist after the system has been checked and found to be functional, further inspection of the affected component(s) should be conducted for damage assessment. Any damage found as a result of the detailed inspections should be repaired or replaced as appropriate.

(2) In-flight indications.

(i) Indications used for in-flight monitoring have in the past been used for two main reasons: to provide a structural load display (such as mast bending moment) and to help resolve a service problem (CBIM systems have been used to supplement conventional inspection methods in blind areas).

(ii) Structural load display systems should not be used instead of correcting deficient designs. Structural load display systems are appropriate for use in locating control positions, such as the cyclic stick, under transient conditions such as slope landings and hover in sidewinds, but structural load display systems are not considered appropriate for routine operations such as climbout or cruise with constant attention required by the flightcrew. If the load indicator provides a needed tool to the pilot in limited types of operations and does not significantly add to pilot workload otherwise, its use can be considered.

(iii) In the past, certain service problems have been solved by adding in-flight indicators such as CBIM systems. When retrofit of the affected structure is impossible or impractical, and when conventional inspection techniques are shown to be inadequate by themselves, CBIM or similar systems may be the only practical

solution, despite the increase in pilot workload and the potential for problems caused by overreaction by the pilot to a structural fault indication. When used for correction of service difficulties, the structural condition indicator system should be accompanied by clear, concise crew directions to prevent possible catastrophic overreaction. Load reduction measures such as rotor speed changes, airspeed reductions, altitude changes, etc., should be clearly provided, if needed. Crack propagation time from indication should be sufficient to allow continued safe flight to a safe landing area. For new designs, CBIM or similar systems which add to the pilot's workload are considered inappropriate. Proper redesign to provide the needed safe life, fail-safety, and inspectability is considered the appropriate action.

f. Complementary considerations of structural condition indicator use.

(1) Two basic programs are commonly used for approval of structural condition indicators. Basic type certification procedures are used for mast moment indicators and similar systems, and AD's (with appropriate type design changes) are used for CBIM systems which require pilot attention and corrective action when an indication of a structural fault is detected.

(2) The fatigue substantiation required by §§ 27.571 and 29.571 should consider a conservative number of excursions into the high load range monitored by a structural condition indicator such as a mast moment indicator. Static strength should not be adversely affected by a single excursion into the high load range monitored by the indicator.

(3) Complementary design provisions should accompany the use of a structural condition indicator system. Redundancy of load paths and inspection systems and indicator system failure analyses should be provided, as necessary, to meet the requirements of § 29.1309. The life remaining after the indicator system detects a structural failure should be calculated (with test verification), and compatible inspection and/or overhaul programs should be provided.

(4) The FAA/AUTHORITY approval of a structural condition indicator system requires evaluation by the airframe, systems and equipment, and flight test specialists. The airframe specialist has the responsibility to review effects of structural condition indicator system use on aircraft loads, strength and deformation, and structural fatigue evaluation as well as the instructions for continued airworthiness. The systems and equipment specialist needs to evaluate the system for function and installation as well as the reliability requirements of § 29.1309. Flight test evaluation of the instruments' arrangement and visibility, effect on crew workload, and possible changes for RFM is also needed. Care should be exercised to assure that responsibilities are not given to the flightcrew which would be more appropriately handled by a redesign or by the maintenance personnel. Early coordination between all specialists is necessary to prevent delays from last minute design changes.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 8. (Amendment 29-35) SUBSTANTIATION OF COMPOSITE ROTORCRAFT STRUCTURE.

a. Reference FAR Sections. §§ 29.305, .307, .571, .603, .605, .609, .610, .611, .613, .629, .923, .927, .931, .1529 and Appendix A.

b. Purpose. These substantiation procedures provide a more specialized supplement to the general procedures outlined by AC 20-107A, "Composite Aircraft Structure." These procedures address substantiation requirements for composite material system constituents, composite material systems, and composite structures common to rotorcraft. A uniform approach to composite structural substantiation is desirable, but it is recognized that in a continually developing technical area which has diverse industrial roots, both in aerospace and in other industries, some variations and deviations from the procedures described herein will be both necessary and acceptable. Significant deviations from this material should be coordinated in advance with the Rotorcraft Directorate.

c. Special Considerations. Since rotorcraft structure is configured uniquely and is inherently subjected to severe cyclic stresses, special consideration is required for the substantiation of all rotorcraft structure, including composites. This special consideration is necessary to ensure that the level of safety intended by the current regulations is attained during the type certification process for all structure with special emphasis on composite structure because of its unique structural characteristics, manufacturing quality and operational considerations, and failure mechanisms.

d. Background.

(1) Historically, rotorcraft have required unique, conservative structural substantiation because of unique configuration effects, unique loading considerations, severe fatigue spectrum effects, and the specialized comprehensive fatigue testing required by these effects. Rotorcraft structural static strength substantiation for both metal and composite structure is essentially identical to that for fixed wing structure once basic loads have been determined. However, rotorcraft structural fatigue substantiation for metals is significantly different from fixed wing fatigue substantiation. Since AC 20-107A, as developed, applies to both fixed wing aircraft and rotorcraft; it, of necessity, was finalized in a broad generic form. Accordingly, a need to supplement AC 20-107A for rotorcraft was recognized during type certification programs. One significant difference in traditional rotorcraft fatigue substantiation programs and fixed wing fatigue programs is the use of multiple full-scale specimen fatigue tests for rotorcraft programs rather than just one full-scale specimen test. Also, constant

amplitude, accelerated load tests are typically used rather than spectrum tests because of the high frequency loads common to rotorcraft operations. These rotorcraft fatigue tests have traditionally involved the generation of stress versus life or cycle (S-N) curves for each critical part (most of which are subjected to the cyclic loading of the main or tail rotor system) using a monotonic (sinusoidal) fatigue spectrum based on maximum and minimum service stress values. Unless configuration differences or flight usage data dictate otherwise, the monotonic fatigue spectrum's period is typically based on six ground-air-ground (GAG) cycles for each flight hour of operation. The S-N curves for the substantiation of each detailed part are typically generated by plotting a curved line through three data points (reference Appendix 1 of this AC, "Fatigue Evaluation of Transport Category Rotorcraft Structure (Including Flaw Tolerance)"). The three data points selected are a short specimen life (low cycle fatigue), an intermediate specimen life and a long specimen life (high cycle fatigue). Each raw data point is generated by monotonically fatigue testing at least two full-scale specimens (parts) to failure or run out for each data point on the S-N curve. The raw data point values are then reduced by an acceptable statistical method to a single value for plotting to ensure proper reliability of the associated S-N curve. Order 8110.9, "Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and Other Power Transmission Systems" and Appendix 1 contain comprehensive discussions of the S-N curve generation process. The rotorcraft S-N curve process contrasts sharply with the fixed wing process of using a single full-scale fatigue article (usually an entire wing or airframe, which constitutes a single full-scale assembly data point), generic material or full-scale assembly S-N data (e.g., MIL-HDBK-5 for metals, MIL-HDBK-17B for composites, or AFS-120-73-2 for full-scale assemblies), a non-monotonic spectrum and relatively large scatter factors to verify or determine the design fatigue life of the full-scale airplane.

(2) Also, rotorcraft have employed and mass produced composite designs in primary structure (typically main and tail rotor blades) since the early 1950's. This was 10 or more years before composites were type certificated for primary fixed-wing structure in either military or civil aircraft applications (with some notable limited production exceptions, such as the Windecker fixed wing aircraft). In any case, the early 1950 period was well before a clear, detailed understanding of composite structural behavior (especially in the areas of macroscopic and microscopic failure mechanisms and modes) was relatively common and readily available in a usable format for the average engineer working in this field. It also predated the initial issuance of AC 20-107. Currently, much composite design information is proprietary, either to government, industry or both, and many data gathering methods have not been completely standardized. Consequently, a significant variation from laboratory to laboratory in material property value determination methods and results can exist. The early rotor blade designs (as well as current designs) are by nature relatively low strain, tension structure designs. Also, by nature, these designs are not damage or flaw critical. Thus by circumstance as much as design, early composite rotor blade and other composite rotorcraft designs incorporated an acceptable fatigue tolerance level of safety. In the 1980's, more test data, analytical knowledge, and analytical methodology became available to more completely substantiate a composite design. Current

FAR's 27 and 29 contain many sections (reference paragraph a.) to be considered in substantiating composite rotorcraft structure, but this advisory material is needed to supplement the general guidance of AC 20-107A by providing specific rotorcraft guidance for obtaining consistent compliance with FAR sections applicable to rotorcraft.

e. Definitions. The following basic definitions are provided as a convenient reading reference. MIL-HDBK-17, and other sources, contain more complete glossaries of definitions.

(1) AUTOCLAVE. A closed apparatus usually equipped with variable conditions of vacuum, pressure and temperature. Used for bonding, compressing or curing materials.

(2) ALLOWABLES. Both A- basis and B- basis values statistically derived and used for a particular composite design.

(3) BALANCED LAMINATE. A composite laminate in which all laminae at angles other than 0° occur only in ± pairs (not necessarily adjacent).

(4) A-BASIS ALLOWABLE. The "A" mechanical property value is the value above which at least 99 percent of the population of values is expected to fall, with a confidence of 95 percent.

(5) B-BASIS ALLOWABLE. The "B" mechanical property value is the value above which at least 90 percent of the population of values is expected to fall, with a confidence of 95 percent.

(6) BOND. The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent.

(7) COCURE. The process of curing several different materials in a single step. Examples include the curing of various compatible resin system pre-pregs, using the same cure cycle, to produce hybrid composite structure or the curing of compatible composite materials and structural adhesives, using the same cure cycle, to produce sandwich structure or skins with integrally molded fittings.

(8) CURE. To change the properties of a thermosetting resin irreversibly by chemical reaction; i.e., condensation, ring closure, or addition. Cure may be accomplished by addition of curing (crosslinking) agents, with or without catalyst, and with or without heat.

(9) DELAMINATION. The separation of the layers of material in a laminate.

(10) DISBOND. A lack of proper adhesion in a bonded joint. This may be local or may cover a majority of the bond area. It may occur at any time in the cure or subsequent life of the bond area and may arise from a wide variety of causes.

(11) FIBER. A single homogeneous strand of material, essentially one--dimensional in the macro-behavior sense, used as a principal constituent in advanced composites because of its high axial strength and modulus.

(12) FIBER VOLUME. The volume of fiber present in the composite. This is usually expressed as a percentage volume fraction or weight fraction of the composite.

(13) FILL. The 90° yarns in a fabric, also called the woof or weft.

(14) GLASS TRANSITION. The reversible change in an amorphous polymer or in amorphous regions of a partially crystalline polymer from (or to) a viscous or rubbery condition to (or from) a hard and relatively brittle one.

(15) GLASS TRANSITION TEMPERATURE. The approximate midpoint of the temperature range over which the glass transition takes place.

(16) HYBRID. Any mixture of fiber types (i.e., graphite and glass).

(17) IMPREGNATE. An application of resin onto fibers or fabrics by several processes: hot melt, solution coat, or hand lay-up.

(18) LAMINA. A single ply or layer in a laminate in which all fibers have the same fiber orientation.

(19) LAMINATE. A product made by bonding together two or more layers or laminae of material or materials.

(20) LOW STRAIN LEVEL. As used herein, is defined as a principal, elastic axial gross strain level, that for a given composite structure provides for no flaw growth and thus provides damage tolerance of the maximum defects allowed during the certification process using the approved design fatigue spectrum.

(21) MATERIAL SYSTEM CONSTITUENT. A single constituent (ingredient) chosen for a material system (e.g., a fiber, a resin).

(22) MATERIAL SYSTEM. The combination of single constituents chosen (e.g., fiber and resin).

(23) MATRIX. The essentially homogeneous material in which the fibers or filaments of a composite are embedded. The resins used in most aircraft structure are thermoset polymers.

(24) MAXIMUM STRUCTURAL TEMPERATURE. The temperature of a part, panel or structural element due to service parameters such as incident heat fluxes, temperature, and air flow at the time of occurrence of any critical load case, (i.e., each

critical load case has an associated maximum structural temperature). This term is synonymous with the term “maximum panel temperature.”

(25) POROSITY. A condition of trapped pockets of air, gas, or void within a solid materials, usually expressed as a percentage of the total nonsolid volume to the total volume (solid + nonsolid) of a unit quantity of material.

(26) PRE-PREG, PREIMPREGNATED. A combination of mat, fabric, nonwoven material, tape, or roving already impregnated with resin, usually partially cured, and ready for manufacturing use in a final product which will involve complete curing. Prepreg is usually drapable, tacky and can be easily handled.

(27) RESIN. An organic material with indefinite and usually high molecular weight and no sharp melting point.

(28) RESIN CONTENT. The amount of matrix present in a composite either by percent weight or percent volume.

(29) SECONDARY BONDING. The joining together, by the process of adhesive bonding, of two or more already-cured composite parts, during which the only chemical or thermal reaction occurring is the curing of the adhesive itself. The joining together of one already-cured composite part to an uncured composite part, through the curing of the resin of the uncured part, is also considered for the purposes of this advisory circular to be a secondary bonding operation. (See COCURING).

(30) SHELF LIFE. The length of time a material, substance, product, or reagent can be stored under specified environmental conditions and continue to meet all applicable specification requirements and/or remain suitable for its intended function.

(31) STRAIN LEVEL. As used herein, is defined as the principal axial gross strain of a part or component due to the principal load or combinations of loads applied by a critical load case considered in the structural analysis (e.g., tension, bending, bending-tension, etc.). Strain level is generally measured in thousandths of an inch per unit inch of part or microinches/per inch (e.g., .003 in/in equals 3000 microinches/inch).

(32) SYMMETRICAL LAMINATE. A composite laminate in which the ply orientation is symmetrical about the laminate midplane.

(33) TAPE. Hot melt impregnated fibers forming unidirectional pre-preg.

(34) THERMOPLASTIC. A plastic that repeatedly can be softened by heating and hardened by cooling through a temperature range characteristic of the plastic, and when in the softened stage, can be shaped by flow into articles by molding or extrusion.

(35) THERMOSET (OR CHEMSET). A plastic that once set or molded cannot be re-set or remolded because it undergoes a chemical change; (i.e., it is substantially infusible and insoluble after having been cured by heat or other means).

(36) WARP. Yarns extended along the length of the fabric (in the 0° direction) and being crossed by the fill yarns (90° fibers).

(37) WORK LIFE. The period during which a compound, after mixing with a catalyst, solvent, or other compounding constituents, remains suitable for its intended use.

f. RELATED REGULATORY AND GUIDANCE MATERIAL.

Document	Title
(1) AC 27 MG-11	"Fatigue Evaluation of Rotorcraft Structure"
(2) AC 20-107	"Composite Aircraft Structure"
(3) AC 21-26	"Quality Control for the Manufacture of Composite Materials"
(4) MIL-HDBK-17	"Polymer Matrix Composites Volume 1: Guidelines"

g. PROCEDURES FOR SUBSTANTIATION OF ROTORCRAFT COMPOSITE STRUCTURE. The composite structures evaluation has been divided into eight basic regulatory areas to provide focus on relevant regulatory requirements. These eight areas are: (1) fabrication requirements; (2) basic constituent, pre-preg and laminate material acceptance requirements and material property determination requirements; (3) protection of structure; (4) lightning protection; (5) static strength evaluation; (6) damage tolerance and fatigue evaluation; (7) dynamic loading and response evaluation; and (8) special repair and continued airworthiness requirements. Original as well as alternate or substitute material system constituents (e.g., fibers, resins, etc.), material systems (combinations of constituents and adhesives), and composite designs (laminates, cocured assemblies, bonded assemblies, etc.) should be qualified in accordance with the methodology presented in the following paragraphs. Each regulatory area will be addressed in turn. It is important to remember that proper certification of a composite structure is an incremental, building block process which involves phased FAA/AUTHORITY involvement and incremental approval in each of the various areas outlined herein. It is strongly recommended that a FAA/AUTHORITY certification team approach be used for composite structural substantiation. The team should consist of FAA/AUTHORITY engineering, the MIDO inspector(s), the associated Designated Engineering Representatives (DER's), the associated Designated Manufacturing Inspection Representatives (DMIR's), and cognizant members of the applicant's organization. Personnel who are composites specialists (or are otherwise

knowledgeable in the subject) should be primary team member candidates. Once selected, it is recommended that team meetings be held periodically (possibly in conjunction with type boards) during certification to ensure the building block certification process is accomplished as intended.

(1) The first area is the fabrication requirements of § 29.605:

(i) The quality control system should be developed considering the critical engineering, manufacturing, and quality requirements and a guidance standard such as AC 21-26, "Quality Control For the Manufacture of Composite Materials." This ensures that all special engineering, or manufacturing quality instructions for composites are presented, evaluated, documented, and approved, using drawings, process and manufacturing specifications, standards, or other equivalent means. This should be one of the early phases of a composite structure certification program, since this represents a major building block for sequential substantiation work.

(ii) Specific allowable defect limits on, for example, fiber waviness, warp defects, fill defects, porosity, hole edge effects, edge defects, resin content, large area debonds, and delaminations, etc., for a particular material system component, laminate design, detailed part, or assembly should be jointly established by engineering, manufacturing, and quality and the associated inspection programs for defect detection created, validated, and approved. Each critical engineering design should consider the worse-case effects of the manufacturing process (maximum waviness, disbonds, delaminations, and other critical defects) allowed by the reliability limitations of the approved inspection program.

(iii) If bonds or bond lines such as those typical of rotorcraft rotor blade structure are used, special inspection methods, special fabrication methods or other approved verification methods (e.g., engineering proof tests, reference paragraph g(5)) should be provided to detect and limit disbonds or understrength bonds.

(iv) Structurally critical composite construction fabrication process and procurement specifications, for fabricating reproducible and reliable structure, must be provided and FAA/AUTHORITY approved early during the certification process and should, as a minimum, cover the following:

(A) Vendor and Qualified Parts List (QPL) Control. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both the manufacturing and inspection district office (MIDO) and FAA/AUTHORITY engineering) at any time, that their quality control systems ensure on a continuous basis, that only qualified suppliers provide the basic material constituents or material systems (e.g., pre-pregs) that meet approved material specifications. Recommended guidelines for qualification of alternate material systems and suppliers are contained in MIL-HDBK-17B, Volume I, Section 2.3.2. These methods can also be used, periodically for qualification status renewals of existing material systems and suppliers.

(B) Receiving Inspection and In Process Inspection. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their receiving and in-process quality control systems provides products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, such that the necessary statistical reliability and confidence levels for the items being inspected (that are specified by engineering) are continuously maintained. This will require periodic standard inspections and engineering characterization tests on basic constituent and material system samples which should be conducted, as a minimum, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY-witnessed.

(C) Material System Component Storage and Handling. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their composite material system (or constituent) storage and handling procedures and specifications provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, such that the necessary statistical reliability and confidence levels for the items being inspected (which are specified by engineering) are continuously maintained. This should require, as a minimum, periodic inspections to ensure that proper records are kept on critical parameters (e.g., room temperature “bench” exposure, shelf life, etc.) and that periodic basic constituent and material system characterization tests are conducted, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY-witnessed.

(D) Statistical Validation Level. It is necessary to maintain the minimum required statistical validation level of the quality control system (which should be specified for each critical item or constituent by the approved quality and engineering specifications). The statistical validation level should be defined and approved early in certification. Also, approval and proper usage should be continuously maintained during the entire procurement and manufacturing cycles.

(v) Alternate fabrication and process techniques should be approved and should comply with § 29.605. Any alternate techniques should provide at least the same level of quality and safety as the original technique. Any changes should be presented and FAA/AUTHORITY-approved well in advance of the change’s production effectivity.

(2) The second area is the basic raw constituent, pre-preg, and laminate material acceptance requirements and material property determination requirements of §§ 29.603 and 29.613. These criteria require application of the critical environmental limits such as temperature, humidity, and exposure to aircraft fluids (such as fuel, oils,

and hydraulic fluids), to determine their effect on the performance of each composite material system. Temperature and humidity effects are commonly considered by coupon and component tests utilizing preconditioned test specimens for each material system selected. Material “A” & “B” basis allowable strength values and other basic material properties (based on MIL-HDBK-17, or equivalent) are typically determined by small scale tests, such as coupon tests, for use in certification work. In the case of composites, determination of these basic constituent and material system properties will almost invariably involve the submittal, acceptance and use of company standards. This is currently necessary because MIL-HDBK-17 has not completed development of “B” basis allowables for inclusion in the handbook. Also, test methods vary somewhat from manufacturer to manufacturer; therefore, individual company results will exhibit some scatter in final material property values. Any company standard which is approved and used should meet or exceed related MIL-HDBK-17 requirements. Material structural acceptance criteria and property determination should, as a minimum, include the following:

(i) Property characterization requirements of all material systems (e.g., pre-pregs, adhesives, etc.) and constituents (e.g., fibers, resins, etc.) should be identified, documented, and approved. These requirements, once approved, should be placed in all appropriate procedures and specifications (such as those in paragraph (g)(1) above).

(ii) Moisture conditioning of test coupons, parts, subassemblies, or assemblies should be accomplished in accordance with MIL-HDBK-17, other similar approved methods or per FAA/AUTHORITY approved programs.

(iii) The maximum and minimum temperatures expected in service (as derived from test measurements, thermal analyses on panels and other parts, experience, or a combination) should be determined and accounted for in static and fatigue strength (including damage tolerance) substantiation programs considering associated humidity induced effects.

(iv) The glass transition temperature, Tg, is an important characteristic parameter of amorphous polymers, such as epoxies. It is the temperature below which the polymer behaves like a “glassy” solid and above which it behaves like a “rubbery” solid, i.e., it is the temperature at which there is a very rapid change in physical properties. In actuality, the change from a hard polymeric material to a rubbery material takes place over a narrow temperature range. A composite material will experience a drastic reduction in matrix controlled mechanical material properties when loaded in this temperature range. Since the resin (matrix) is the critical structural constituent in a composite and since Tg exceedance is critical to structural integrity; Tg determination is necessary. The Tg margin methodology of MIL-HDBK-17, Section 2.2.2.1, should be implemented, i.e., the wet glass transition temperature (Tg) should be 50° F higher than the maximum structural temperature (see definition). For any type of resin or adhesive, an acceptable temperature margin using MIL-HDBK-17 techniques (e.g., consideration of limited high temperature excursions) or equivalent methodologies based on tests

and/or experience should be established and approved early in the certification process. In no case should structural strength be degraded below limit load capability on a maximum world wide high temperature day.

(v) Local design values should be established by analysis and characterization tests and approved for specific structural configurations (point designs) which include the effects of stress risers (e.g., holes, notches, etc.) and structural discontinuities (e.g., joints, splices, etc.). Proper determination of these values for full-scale design and test should be considered one of the most critical building blocks in substantiating and evaluating a composite structure. These transitional load transfer areas typically produce the highest stresses (and strains) and serve as the nucleation sites for many of the failures (including those due to the relatively low interlaminar strength of composites) that occur in service in a full-scale part or assembly. Small scales tests (such as coupon, element, and subcomponent tests), or equivalent approved testing programs, and analytical techniques should be carefully designed, prepared, and approved to evaluate potential “hot spots” and provide accurate simulations and representations of full-scale article stresses and strains in the critical transition areas. Proper certification work in this area will ensure initial safety and continued airworthiness in full-scale production articles.

(vi) The design strain level for each major component and material system should be established and approved such that specified impact damage considerations are defined and properly limited. The effects of the approved strain levels should be established for each composite material using small scale characterization tests and the results should be used to establish or verify the maximum allowable design strain level for each full-scale article. The maximum allowable design strain values selected should also take into account the reliability and confidence levels established for the relevant portions of the quality control system. This methodology is necessary because the amount and size of flaws in the production article may restrict the allowable level of design strain. In a no-flaw-growth design, the maximum specified impact damage and manufacturing flaw size at the most critical location on the part will be a major factor in determining the maximum allowable elastic strain. This design approach is currently selected for nearly all civil and most military applications; since, under normal conditions, only visual inspections are required in the field (unless unusual external damage circumstances such as a hail storm occur) to maintain the initial level of airworthiness (safety). However, many military applications because of their demanding missions, employ scheduled field non-destructive inspection (NDI) maintenance, (such as comparative ultrasonics) to ensure that flaw growth either does not occur, is controlled by approved structural repair, or by replacement of affected parts. To date, civil applications have not been presented that desire a flaw growth, phased NDI approach. Therefore, selection of the full-scale article’s design strain limit based on small scale tests for a no flaw growth design is seen to be extremely important.

(vii) Composite and adhesive properties should be determined such that detrimental structural creep does not occur under the sustained loads and environments expected in service. Small scale characterization tests (such as coupon, element, and

subcomponent tests) and analysis, which verify and establish the full-scale design criteria and parameters necessary to ensure that detrimental structural creep in full-scale structure does not occur in service, should be conducted early in certification and should be FAA/AUTHORITY-approved.

(viii) Material allowable strength values for full-scale design and testing should be developed using the coupon procedures presented in MIL-HDBK-17 or equivalent. At least three batches of material samples should be used in material allowable strength testing. Company standards should be prepared, evaluated and FAA/AUTHORITY approved early in certification (as part of the building block process), that reflect the material property determination considerations recommended in MIL-HDBK-17 on a equal to or better than basis.

(3) The third area is the protection of structure as required by § 29.609. Protection against thermal and humidity effects and other environmental effects (e.g., weathering, abrasion, fretting, hail, ultraviolet radiation, chemical effects, accidental damage, etc.) should be provided, or the structural substantiation should consider the results of those effects for which total protection is impractical. Determination and approval of worst-case or most conservative operating limits, and damage scenarios should be accomplished. Appropriate flammability and fire resistance requirements should also be considered in selecting and protecting composite structure. Usually a hazard analysis is conducted early in certification which identifies the various threats and threat levels for which protection must be provided. This data is then used to construct and submit for approval the methods-of-compliance necessary to provide proper structural protection.

(4) The fourth area is the lightning protection requirements of § 29.610. Protection should be provided and substantiated in accordance with analysis and with tests such as those of AC 20-53A and FAA Report DOT/FAA/CT-86/8. For composite structure projects involving rotorcraft certified to earlier certification bases (which do not automatically include the lightning protection requirements of § 29.610), these requirements should be imposed as special conditions. The design should be reviewed early in certification to ensure proper protection is present. The substantiation test program should also be established, reviewed and approved early to ensure proper substantiation.

(5) The fifth area is the static strength evaluation requirements of §§ 29.305 and 29.307 for composite structure. Only conservative proven methods of static analysis and failure criteria should be employed. The material stress-strain curve should be clearly established, at least through the ultimate design load, for each composite design. Composite structure should be statistically demonstrated, incrementally, through a program of analysis, coupon tests, minor component ultimate load tests and major component ultimate load tests. The static strength substantiation program should consider all critical loading conditions for all critical structure including residual strength and stiffness requirements after a predetermined length of service, e.g., end of life (EOL) (which takes into account damage and other degradation due to

the service period). Analytical reports and tests should consider all possible failure modes and should include the critical, allowable effects of:

(i) Environment (reference paragraphs AC 29 MG 8 g(2) and g(3)).

(ii) Service Life (residual limit strength and stiffness demonstration).

(iii) Load path loss (fail-safe analysis and limit strength demonstration).

(iv) The standard fabrication process and its variability.

(v) Impact damage expected during service up to the established threshold of detectability of the field inspection methods to be employed.

(vi) Point design and structural discontinuity considerations (e.g., stress risers, joints, etc.).

(vii) Unless the ultimate strength of each critical bonded joint can be reliably substantiated in production by NDI techniques (or other equivalent, approved techniques), then limit load capability must be guaranteed by either of the following or a combination thereof:

(A) The maximum disbond of each critical bonded joint which will carry limit load is established by test, analysis, or both. Disbonds greater than these values are typically prevented by design features.

(B) Each critical bonded joint on each production article should be proof tested to the critical limit load.

(viii) For static strength analysis laminae and laminate “A” and “B” basis allowables (determined in accordance with paragraph g(2)) should be used subject to the following conditions unless lower material properties are required by point design considerations (e.g., stress risers, joints, etc.) stiffness requirements (e.g., flutter or vibration margins), fatigue strength (including damage tolerance), or other overriding considerations.

(A) When applied loads are distributed through a single load path or single member within an assembly, the failure of which would result in the loss of the structural integrity of the component involved or inability of the rotorcraft structure to carry limit load, the part should be designed, analyzed, and tested using “A” basis allowables.

(B) Redundant (fail-safe) structures in which the failure of individual elements would result in applied loads being safely redistributed to other load carrying members without exceeding the limit load capability of the rotorcraft structure may be designed, analyzed, and tested using “B” basis allowables.

(6) The sixth area is the fatigue evaluation requirements of § 29.571. The fatigue evaluation method for the rotorcraft being certified should consider damage tolerance in accordance with AC 20-107A.

(i) The safe-life method for composite structure as defined in AC 20-107A is a flaw tolerant safe-life method (e.g., the test specimens consider inherent production flaws and impact damage (reference paragraph (7)(ii)).

(ii) Large area disbonds, weak bonds, delaminations, or other defects should be considered in tests or be prevented or be limited by appropriate flaw tolerant special design features and by special manufacturing, maintenance, and inspection procedures. Special attention should be assigned to all pure bond lines (reference paragraph (5)).

(iii) Non-fail-safe or partially fail-safe dynamic component structure, which may employ bond lines as the only load path, should be designed to relatively small previously approved values of elastic, ultimate strain for the material system utilized, and should be subjected to full-scale S-N curve testing. Six or more specimens are recommended, as part of the substantiation process. Where practical, flight-by-flight spectrum testing should be used.

(iv) All critical safety of flight composite structure must be designed to be flaw (damage) tolerant. Environment degradation and in-service damage critical values are typically included in the flaw tolerance evaluation. All other key factors, such as material selection, manufacturing, and quality assurance controls, and in-service inspection and maintenance, as noted previously, are also to be accounted for.

(v) The fail-safe design features of the rotor heads and blade retention systems, other critical primary composite structure, and point design features (e.g., bonded metal-to-composite joints) should be assessed and appropriate inspection programs provided to prevent catastrophic failure from flaw/damage propagation.

(vi) The method of generating S-N curves using approved raw data should be demonstrated, evaluated, and approved.

(vii) Any limited life items must be identified and placed in the Airworthiness Limitations section of the maintenance manual in accordance with § 29.571.

(viii) Load spectra, load truncation methods and all other major aspects of the fatigue evaluation are documented in test proposals and approved.

(ix) Flaw growth rates (from initial detectability to the established value for residual strength) must be previously established and closely monitored during

substantiation. This data should be used to establish special phased inspections and maintenance intervals for critical structure, as required.

(7) The seventh major area is the dynamic loading and response requirements of § 29.629 for vibration and resonance frequency determination and separation for aeroelastic stability and stability margin determination for flutter critical flight structure. Critical parts, locations, excitation modes, and separations are to be identified and substantiated. This substantiation should consist of analysis supported by tests and tests which account for repeated loading effects and environment exposure effects on critical properties, such as stiffness, mass, and damping. Initial stiffness, residual stiffness, proper critical frequency design, and structural damping are provided as necessary to prevent vibration, resonance, and flutter problems.

(i) All vibration and resonance critical composite structure are identified and properly substantiated.

(ii) All flutter-critical composite structure are identified and properly substantiated. This structure must be shown by analysis to be flutter free to 1.1 V_{NE} (or any other critical operating limit, such as V_D, for a VSTOL aircraft) with the extent of damage for which residual strength and stiffness are demonstrated.

(iii) Where appropriate, crash impact dynamics considerations should be taken into account to ensure proper crash resistance and a proper level of occupant safety for an otherwise survivable impact.

(8) The eighth area is the special repair and continued airworthiness requirements of §§ 29.611, 29.1529, and FAR Part 29 Appendix A for composite structures. When repair and continued airworthiness procedures are provided in service documents (including approved sections of the maintenance manual or instructions for continued airworthiness) the resulting repairs and maintenance provisions must be shown to provide structure which continually meets the guidance of paragraphs (1) through (7) of this AC paragraph. All certification based repair and continued airworthiness standards, limits, and inspections must be clearly stated and their provisions and limitations defined and documented to ensure continued airworthiness. In general, no composite repair should be attempted which is out of scope to repairs stated in an approved Structural Repair Manual (SRM) without an engineering design approval by a qualified FAA/AUTHORITY representative (DER or staff engineer). The following minimum criteria should be met in any acceptable composite repair:

(i) The repair should be permanent.

(ii) The repair should restore the structure to the required strength and stiffness.

(iii) The repair should restore all functional requirements.

(iv) The repair should have a negligible weight penalty.

(v) The repair should be aerodynamically compatible.

(vi) The repair materials should be compatible in all essential aspects with the parent materials.

In summary, primary composite structure is an especially critical structure that requires a clearly defined, phased approval (building block) certification process. This process should involve the entire project certification team from a project's start to its finish so that proper certification is continuously and ultimately achieved. Also, in some special cases, involving new advanced state-of-the-art composite technology, an issue paper may be necessary. However, in the majority of cases (using current composite materials and design philosophy) the applicant's acknowledged use of this advisory material (as recorded in the type board minutes) should eliminate the need for a separate issue paper.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 9 ROTORCRAFT ONE-ENGINE-INOPERATIVE POWER ASSURANCE

a. Purpose. The purpose of this document is to establish an approach for an engine power assurance procedure which will assure that the required OEI power level can be achieved.

b. General. The data and methods described herein are intended to be utilized as a guide and not necessarily the only means of achieving the desired result.

c. Applicability. The applicability of the document is intended to be primarily in support of the new 30-second and 2-minute OEI rotorcraft engine rating scheme.

d. Partial Power Assurance (Engine "Run-Line").

(1) Fundamental to the concept of limited-use one-engine-inoperative (OEI) ratings is the requirement to be certain that the rated OEI power will indeed be available when needed. Conventional periodic power-assurance and topping checks are impractical with the limited-use rating concept because of the rapid expenditure of useful life during exposure at the engine speeds and temperatures consistent with limited-use ratings; therefore, we require a means of assuring the power available, other than by actual demonstration on each service engine. The advent of more sophisticated controls and engine developments catering to the 30-second/2-minute OEI rating concepts can provide the means to determine: (1) that the thermodynamic/mechanical capability of the engine as tested at the prevailing ambient conditions, will permit reaching a specified power level at any other ambient condition and (2) the fuel system and the various limiters will not prevent the engine achieving OEI power on demand. Pending availability of these new methods, the "parallel run line check" approach is recommended.

(2) The method commonly called the "parallel run-line check" that has been in use for two decades may require refinement for application to the new rating structure where the degree of extrapolation to the OEI power level is more extensive and the slope of the individual engine characteristic is important. As in any power assurance method, success is strongly dependent on the validity of the data base, the maintenance of the engines and sensor/indicating systems, and the care taken during the conduct of the power check. In addition, trending of individual engine performance by the operator and associated analyses can be used to avoid unnecessary flight delays and engine removals.

(3) Thermodynamic/mechanical capability can be addressed by test stand mapping of development engines over a range of ambient conditions to establish an adequate data base of engine characteristics. This will address characteristic slope variations between engines and establish correction factors necessary for extrapolation of data from a power assurance checkpoint to the 30-second OEI rating. Statistical verification and/or modification of the data base may be necessary during production by mapping of sample production engines. Performance data, at the 30-second OEI condition, taken during the supplementary block test and also during the “overhaul test” will demonstrate the capability of an engine and its control system near the end of an overhaul period to produce the required power. This will demonstrate capability with a deteriorated base-performance engine.

(4) The question of fuel system limitations and other various limiters, which could prevent the engine from achieving OEI power on demand, may be addressed by use of more sophisticated control systems, for example, electronic controls utilizing several engine parameter limiters each with automatic datum reset capability. Such control systems can sense an engine failure and automatically reset the operating limiters upward from “normal” to “OEI” limits. Conventional flow and electronic bench testing can be used to verify the function and limit setting of the units when new or after overhaul or repair. The reset features can be extended in function to include a fixed magnitude pulldown type reset for use in verifying new and field production engine/control combination function ability. Pulldown type resets are currently in use today for verification of limiter settings on some engines and can be utilized in this application to avoid unneeded exposure of the engine to the rapid life expenditure conditions.

(5) While the above is envisioned as the probable means in which assurance of capability will occur early in the application of such engines, there will be other means developed. One such means would be utilization of modern electronic engine condition or health monitors to display “go” or “no go” conditions relative to the ability of the engine and its control system to produce 30-second OEI power if required. In this application the device would be a “power assurance meter” and could be used with electronic, hydro-mechanical, and pneumo-mechanical control systems. It is entirely reasonable to expect that self-taught or self-programmed power assurance meters can be used that continually program the actual performance slope of the subject engine and extrapolate to the 30-second OEI with continuous engine monitoring. Self-programming occurs by sampling engine temperature, speed, torque, other characteristics (such as fuel pressure), and ambient conditions, resulting in the reflection of an actual characteristic for the installed engine. The availability of this information permits treating engines individually, whether it is a new or deteriorated engine or one with either minimum or maximum slope, without the necessary compromises to “best” engines that necessarily occurs using the earlier statistical approach. The question of instantaneous fuel system capacity could be addressed by fuel pump/control systems incorporating bypass systems equipped with flow meters. The health monitor or power assurance meter can continually integrate the fuel flow increment available in terms of power increment required in the event of OEI and would

include this intelligence in its pass-fail judgment criteria. Systems of this type would further be conducive to in-service ground checks by overt by-pass deactivation from low power settings to assure satisfactory mechanical function.

(6) Power assurance for the limited-use OEI ratings depends on a complete understanding of the engine model's operating characteristics. Two approaches have been discussed, one where, with the aid of a sophisticated fuel control system, the engine "learns" its own characteristics, and the other where the performance extrapolation is compared with a known minimum standard. The establishment of the standard is obviously a vital part of the procedure, which depends to a large extent on the existence of a reliable data base. In a mature program this is relatively easy to maintain, since it is possible to use the new production engine acceptance data to establish engine-to-engine variation and also to test engines prior to overhaul to determine the effects of deterioration. Thus, an up-to-date minimum or worst-engine characteristic can be maintained and service engines would be compared with this minimum engine.

(7) When the engine in question is a completely new design, or a remote derivative of an existing design, establishing the initial data base presents some problems which must be resolved. New production engines will eventually establish engine-to-engine variation, but initially an estimated worst variation must be assumed. The rate of deterioration and its impact on the base standard must be accounted for from the first engine delivered, yet it may be some time before an acceptable number of engines can be tested after service.

(8) A partial solution lies in the development and qualification cycle of the engine. A typical new-design program requires several development engines, of which more than half can be expected to be used for endurance or accelerated endurance testing. Furthermore, by the time certification is completed and production deliveries have commenced, these engines will normally have amassed several thousand hours of running usually to a schedule far more rigorous than normal service. The information gathered during these tests will provide the necessary data base for the assessment of in-service engines, and it can be progressively enlarged, and the derived data refined, as further production and service data are obtained.

e. Engine Considerations. This section describes the potential causes of an engine not delivering specification OEI power levels in spite of passing a parallel run-line power assurance check. Possible solutions are discussed in the context of one time use 30-second and 2-minute ratings.

(1) Fuel Flow.

(i) An engine may not achieve maximum power available or emergency rating because insufficient fuel is supplied. This condition has a number of possible causes:

(A) Low acceleration schedule

(B) Low maximum fuel stop

(C) Low fuel pump output

(D) Restrictions between the fuel control and the combustor

(ii) The proposed emergency ratings (OEI) may preclude the use of a topping check to uncover the above problems; therefore the following procedures are advanced which can be used either separately or in combination with other approved methods to assure that the required fuel flow is available.

(iii) During engine acceleration the fuel flow rate is considerably higher when compared with the normal steady state condition. This fact can be used to verify the availability of OEI fuel flow. The verification can be done by a direct measurement of fuel flow during an acceleration or derived indirectly from the engine acceleration rate. It is envisaged that the determination of fuel flow by these procedures should be done by some automatic means.

(iv) Figure AC 29 MG 9-1 is a bypass technique in which some of the fuel controls output is routed away from the engine and back to tank. This forces the fuel control onto the acceleration schedule in order to maintain gas generator speed. The design of the system should ensure that with the bypass flowing the fuel control outlet pressure and flow at the OEI ratings are simulated. The bypass system can be either permanently installed and operated in flight, (Failure Malfunction Effects Analysis must be provided), or as an item of ground test equipment. The quantity of fuel bypassed should be equivalent to the worst case difference between fuel flow at the 30-second rating and typical power assurance power levels. However, trend monitoring and service history may provide the basis of an alternative to periodic measurement.

(2) Limiters. A means must be provided to assure that a lower than required (for OEI power) limiter setting does not exist. Limiters that could prohibit reaching OEI power are as follows:

(i) Ng Limiter - (Maximum Compressor Speed Limiter or Governor)

(ii) Measured gas temperature limiter.

(iii) Output shaft torque limiter.

(iv) N_p limiter or power turbine governor - (Power turbine governors can be verified at lower than OEI power conditions.)

(v) Fuel flow limiter or maximum fuel flow stop - (Fuel flow limiting has been addressed in previous paragraphs.)

(3) Failure Modes and Effects Analysis. Failure modes and effects analysis, along with limited demonstration and suitable engine health monitoring procedures, may provide the basis of an acceptable solution to possible unexpected power limiting due to engine condition. It should be shown in the analysis that there is no probable event or combination of events which can cause a latent problem leading to inadequate fuel flow at high powers. The analysis should include all components of the fuel system such as: pump(s), control system (mechanical, hydromechanical, electronic, etc.) pipework, filters, fuel nozzle(s), and electrical interfaces. It should also address the probable effects of accumulated running time, dirty fuel, and hostile environment.

(4) High Corrected Gas Producer Speed.

(i) The proposed OEI ratings will cause the engine to run at high corrected gas producer speeds ($Ng/\sqrt{\theta}$). At high $Ng/\sqrt{\theta}$, performance characteristics of components, especially in the compressor, can change significantly and to an extent which would change the extrapolation of low speed run line data.

(ii) In operation, the effects of the accretion of dirt, FOD, component deterioration, and erosion of blading may also cause changes in the high-speed performance of an engine.

(iii) The above effects must be considered when developing power assurance procedures and data.

(5) Special Devices.

(i) The satisfactory operation of devices or systems whose functioning is required in order to achieve the OEI powers should be verified. Devices or systems, which in normal operations are not exercised through the range of travel needed to achieve the OEI powers, may require special checks to assure adequate capability.

(ii) Special devices that are required only in order to achieve the OEI powers (for example, solenoids to provide additional cooling flow to hot-section components or a water/anti-freeze mixture into the compressor), should be subjected to periodic checks and have a demonstrated high reliability.

f. Airframe Considerations.

(1) Instrumentation Accuracy.

(i) The accuracy of any power assurance check is strongly dependent on the air data and engine parameters. SAE ARP 1217 (May 1979) provides guidance on the desired measurement accuracy for parameters used for engine health and diagnostic monitoring. The parameters to be considered with their respective functions include:

Pressure Altitude Flight Speed Free Air Temperature (stagnation)	Air data basis for establishing power plant inlet pressure and temperature.
Torque Power Turbine Speed	Direct measurement of power output.
Gas Generator Speed(s) Measure Gas Temperature	Primary thermodynamic and limiting parameters
Fuel Flow	Secondary trend monitoring and potential limiting parameter

(ii) The overall power check accuracy can be assessed on a suitable statistical basis using equations that link the measured parameters and inserting system accuracy distributions for each value. This approach will provide an overall assessment of power check accuracy and will highlight major contributors to error. The accuracy assessment at each parameter should include the following elements:

Sensor error Indicator error Reading error	System error

(iii) This assessment might show that while conventional instrument displays of air data are acceptable, servo driven digital displays are desired for engine parameters. Further, displays that provide a “snapshot” of engine readings at a given moment may be useful in avoiding variation in power level during the finite period needed to manually read and log the set of parameters.

(2) Installation Loss Definition.

(i) Installation loss definition is an extremely important aspect of any form of rotorcraft engine performance. Engines are certificated and sold with uninstalled performance guarantees and estimates as to the power output capabilities. Installation of the engine in the rotorcraft imposes power output penalties that must be accounted for in any sort of power assurance check procedure. Normal practice dictates that the engine manufacturer provides a computer program that accurately predicts the engine power output capability throughout the approved flight envelope. This computer program has the capability to correct the power output for the losses incurred by the rotorcraft installation.

(ii) Losses that can reduce engine power available are as follows:

(A) Air intake total pressure loss

(B) Air intake total temperature rise

(C) Exhaust back pressure

(D) Accessory power extraction

(E) Compressor bleed air extraction

(F) Off-optimum power turbine output speed effects

(iii) The above items and methods of dealing with them are clearly defined in SAE Aerospace Recommended Practice (ARP) 1702. Typically, these losses will not be a fixed percentage but will vary with engine operating conditions and environment.

(iv) Any calculations involving power assurance data should use the approved engine performance program, and the rotorcraft losses should be input on a discrete basis so that the interaction between losses and their independent variability is properly considered. This approach is clearly defined in ARP 1702. Accurate consideration of the losses should produce a Power Assurance Check that will preclude premature removal of acceptable engines or continued operation of inadequate power plants.

g. Rotorcraft Flight Manual (RFM).

(1) The Power Assurance Check data for the installed engine (engine data adjusted for inlet losses, exhaust losses, bleed extraction, power extraction, and off-optimum output shaft speed operation) should be presented in the RFM in an easily useable format. The data format may consist of charts of engine torque (at constant power turbine shaft speed) versus allowable values of gas generator speed and gas path temperature covering the range of ambient conditions for takeoff operations. Associated limitations for the rotorcraft transmission and the engine should be noted.

(2) The RFM should also address the following:

(i) Include succinct statements of the reason for the Power Assurance Check and what must be done if the Power Assurance Check results are not acceptable.

(ii) Clearly state that Power Assurance Check either is a pre-takeoff or in-flight procedure, as required by operations, specifications and/or other approval authority documents.

(iii) Be kept simple, easy to use, and identify equipment operation limitations and requirements.

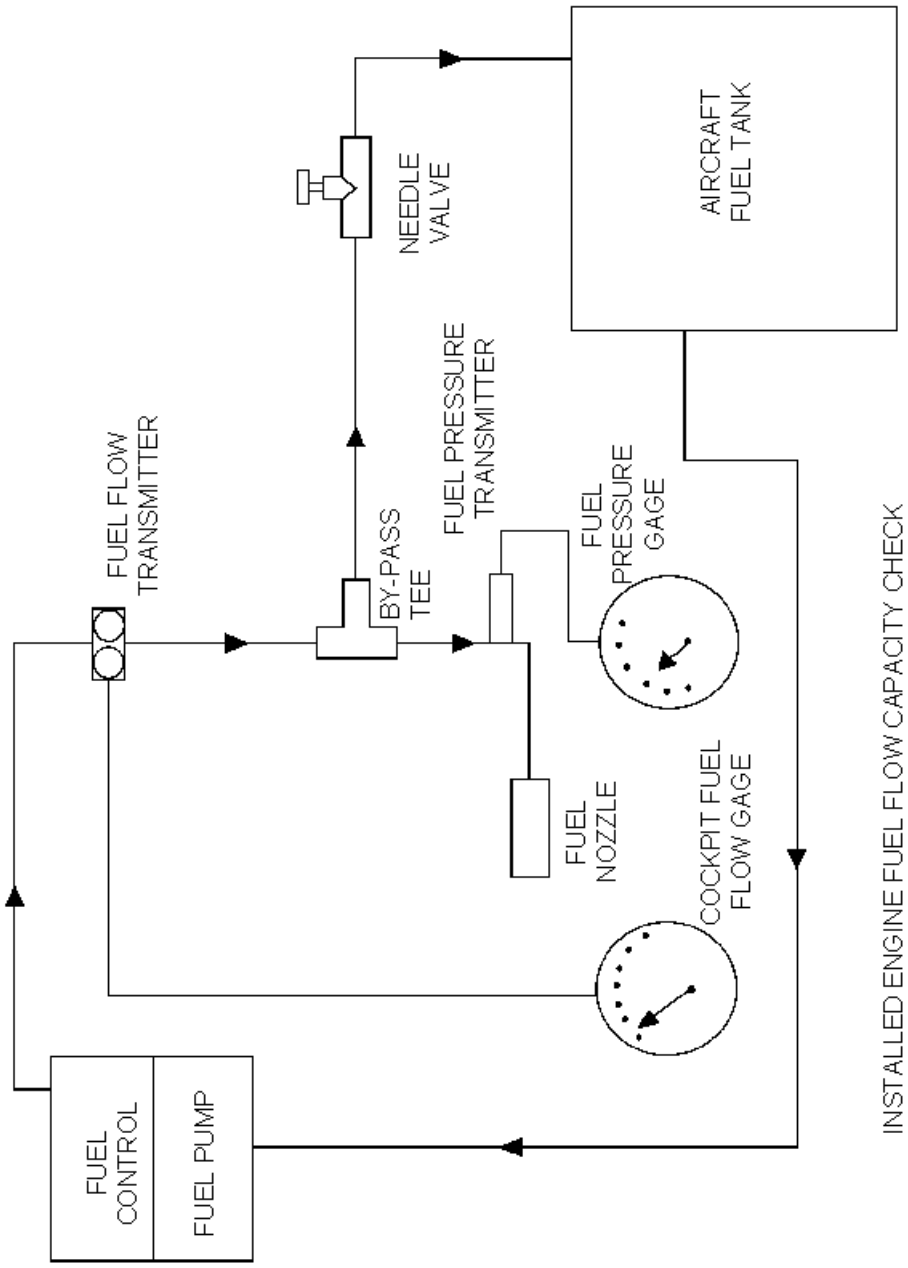

FIGURE AC 29.MG 9-1

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 10 ADVISORY MATERIAL FOR SUBSTANTIATION OF EMERGENCY FLOTATION SYSTEM

a. Reference. FAR sections 29.521, .563(b), .751, .753(a)(1), (a)(2), .801(b), (d), .807(d).

b. Explanation.

(1) This section pertains to emergency flotation systems used to provide buoyancy for rotorcraft not specifically certificated for ditching but performing over-water operations. According to paragraph AC 29.801, ditching may be defined as an emergency landing on the water deliberately executed with the intent of abandoning the rotorcraft as soon as practical. Currently, ditching certification is not required by FAR 29; however, certification requirements are prescribed for applicants requesting ditching certification approval. If a rotorcraft operates over water during a Part 135 operation, the rotorcraft must comply with FAR 135.183, which may require floats.

(2) There are no airworthiness rules specifying the minimum standards for emergency flotation systems on rotorcraft not certificated for ditching. Equipment presented for evaluation must perform its intended function and not create a hazard for the rotorcraft or occupants. The objective in evaluating emergency flotation systems is safe flight and evacuation of the rotorcraft in emergency situations. Adequate emergency flotation systems would aid in keeping rotorcraft sufficiently upright and in adequate trim to permit safe and orderly evacuation in an emergency water landing.

c. Procedures. The following guidance criteria is based on past certification policy and experience for emergency flotation systems. Demonstration of compliance to other criteria may produce acceptable results if adequately justified by rational analysis. Model tests of the appropriate emergency water landing configuration may be conducted to demonstrate satisfactory flotation and trim characteristics where satisfactory correlation between model testing and flight testing has been established. Model tests and other data from rotorcraft of similar configurations may be used to satisfy the water requirements where appropriate.

(1) Flotation Systems.

(i) Normally inflated. The flotation systems which are normally inflated and intended for **emergency use only**, should be evaluated for:

(A) Structural integrity when subjected to:

(1) Air loads throughout the approved flight envelope with floats installed,

(2) Water loads during water entry, and

(3) Water loads after water entry at speeds likely to be experienced after water impact.

(B) Rotorcraft handling qualities throughout the approved flight envelope with floats installed.

(ii) Normally deflated. Emergency flotation systems which are normally stowed in a deflated condition and inflated either in flight or after water contact during an emergency water landing should be evaluated for:

(A) Inflation.

(1) Proper Inflation. The inflation system design should minimize the probability of the floats not inflating properly or inflating asymmetrically. This may be accomplished by use of a single inflation agent container or multiple container system interconnected together. Redundant inflation activation systems will also normally be required. If the primary actuation system is electrical, a mechanical backup actuation system will usually provide the necessary reliability. A secondary electrical actuation system may also be acceptable if adequate electrical system independence and reliability can be documented.

(2) Inadvertent actuation. The inflation system should be safeguarded against spontaneous or inadvertent actuation for all flight conditions. It should be demonstrated that float inflation at any flight condition within the approved operating envelope will not result in a hazardous condition unless the safeguarding system can be shown to be reliable. Limitations to the approved envelope can be established so inadvertent actuation does not impose a hazard at the new envelope.

(3) Float actuation. The float activation means may be fully automatic or manual with a means to verify primary actuation system prior to each flight. If manually inflated, the float activation switch should be located on one of the primary flight controls. These activation means should be safeguarded against spontaneous or inadvertent actuation for all flight conditions.

(4) Flight Limitations. Maximum airspeeds for intentional in-flight actuation of the float system and for flight with the floats inflated should be established as limitations in the Rotorcraft Flight Manual (RFM) unless in-flight actuation is prohibited by the RFM.

(5) Inflation time. For floats inflated automatically by water contact, inflation time from actuation to neutral buoyancy should be short enough to prevent the rotorcraft from becoming submerged to the point where egress is impeded.

(6) Pressure checking. A means should be provided for checking the pressure of the gas storage cylinders prior to each flight. A table or device showing acceptable gas cylinder pressure variation with ambient temperature and altitude (if applicable) should be provided.

(7) Over inflation. A means should be provided to minimize the possibility of over inflation of float bags under any reasonably probable actuation conditions.

(8) No puncture inflation. The ability of the floats to inflate without puncture when subjected to actual water pressure should be substantiated. A full scale rotorcraft immersion demonstration in a calm body of water is one acceptable method of substantiation. Other methods of substantiation may be acceptable depending upon the particular design of the flotation system.

(9) Flotation bag containment. Float installations should be evaluated to ascertain that emergency exits are not blocked by the inflated floats when the float bags are inflated to their maximum inflation pressure or their most adverse inflation pressure for emergency exits and the rotorcraft at its most critical weight and center of gravity configuration.

(B) Structural Integrity. The flotation bags should be evaluated for loads resulting from:

(1) Airloads during inflation and fully inflated during the most critical flight conditions and water loads with fully inflated floats during water impact for the rotorcraft desiring float deployment before water entry; or

(2) Water loads during inflation after water entry.

(C) Handling qualities. Rotorcraft handling qualities should be verified by test or analysis to comply with the applicable regulations throughout the approved operating envelopes for:

(1) Deflated and stowed condition,

(2) In-flight inflation condition,

(3) Fully inflated condition, and

(4) Partially inflated condition, assuming the most critical float compartment fails to inflate.

(2) The float system attachment hardware should be shown to be structurally adequate to withstand critical air loads and water loads during water entry when both deflated and stowed and fully inflated (unless in-flight inflation is prohibited). The appropriate vertical loads and drag loads determined from water entry conditions (or as limited by flight manual procedures) should be addressed. The effects of the vertical loads and the drag loads may be considered separately for the analysis.

(3) Flotation and Trim should be investigated for a range of sea states from zero to the maximum selected by the applicant and should be satisfactory in waves having height/length ratios of 1:12.5 Category A rotorcraft, 1:10 Category B rotorcraft with Category A engine isolation, and 1:8 for Category B rotorcraft.

(i) Demonstrated to be satisfactory to at least sea state 4 water conditions.

(ii) Flotation tests should be investigated at the most critical rotorcraft loading condition.

(iii) Flotation time and trim requirements should be evaluated with a simulated, ruptured deflation of the most critical float compartment. Flotation characteristics should be satisfactory in this degraded mode to at least sea state 2 water conditions.

(iv) Probable rotorcraft door/window open or closed configurations and probable damage to the airframe/hull (i.e., failure of doors, windows, skin, etc.) should be considered when demonstrating compliance with the flotation and trim requirements.

(4) Float System Reliability. Reliability should be considered in the basic design to ensure approximately equal inflation of the floats to preclude excessive yaw, roll, or pitch in flight or in the water.

(i) Maintenance procedures should not degrade the flotation system (such as introducing contaminants which could affect normal operation, etc.).

(ii) The flotation system design should preclude inadvertent damage due to normal personnel traffic flow and excessive wear and tear. Protection covers should be evaluated for function and reliability.

(5) Buoyancy requirements for emergency flotation systems should be a minimum of 25 percent excess buoyancy at maximum internal gross weight. The weight of fresh water (density 62.42 lb/ft^3) displaced by fully submerged float or floats should be a minimum of 25 percent greater than the maximum certificated internal gross weight of the rotorcraft. Analysis may be used for buoyancy verification.

(6) Sufficient watertight compartments should provide an acceptable margin of positive stability with any single main float compartment flooded or deflated. The location of the floats, the most critical compartment, the rotorcraft weight, mass moment of inertia, and center of gravity location are also important considerations for stability. Analyses, tests, or a combination thereof may be used to substantiate a positive margin of stability with the most critical compartment flooded or deflated.

(7) The inflatable bag type floats should be designed for the maximum pressure differential developed at the maximum design altitude. That is, the resulting pressure difference between an operational altitude and a take-off site elevation should be established and substantiated. This resulting pressure differential may become an operating limitation.

(8) The float landing loads may be determined from the drop test of the float landing gear or the loads may be derived from landing gear drop test or loads may be determined from model or full scale water entry tests. The vertical loads are distributed over three fourths of the bag's projected area. Bag floats are not subject to the side loads. Rigid floats are to be designed for vertical, horizontal, and side loads distributed along the length of the float.

(9) Design and/or support of the forward part of bag type floats should be evaluated for maximum design speeds to prevent collapse or significant distortion of the bag while in flight.

(10) Resistance to puncture and abrasion at attach/wear points is an important design consideration. Girt or attachment design loads should be sufficient to withstand the maximum imposed design loads.

(11) Occupant Egress and Survival. Each practicable design measure should be taken to minimize the probability that the behavior of the rotorcraft would cause immediate injury to the occupants or prevent evacuation of the rotorcraft after an emergency landing on water. Emergency exits should be located such that they are above the waterline and will not be blocked by the inflated floats or partially inflated floats, impeding evacuation of the rotorcraft. The flotation time and trim of the rotorcraft should allow the occupants to evacuate the rotorcraft. i.e., the rotorcraft should remain sufficiently upright and in adequate trim to permit safe and orderly evacuation of all personnel. For configurations which are considered to have critical occupant egress capabilities due to float proximity, an actual demonstration of egress may be required. When a demonstration is required, it may be conducted on a full-scale rotorcraft actually immersed in a calm body of water or using any other rig/ground test facility shown to be representative. The demonstration should show that floats do not impede a satisfactory evacuation.

(12) Rotorcraft Flight Manual. The Rotorcraft Flight Manual should contain the information pertaining to the emergency flotation system. This material should include:

(i) The information pertinent to the limitations applicable to the emergency float system and operating limitations for the emergency float system,

(ii) Procedures and limitations for flotation device inflation,

(iii) Procedures for use of emergency flotation equipment, and

(iv) Procedures for emergency water landing occupant evacuation.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 11 FATIGUE TOLERANCE EVALUATION OF TRANSPORT CATEGORY ROTORCRAFT METALLIC STRUCTURE

a. Purpose. This advisory material provides an acceptable means of compliance with the provisions of § 29.571, Amendment 28, of the Federal Aviation Regulations (FAR) dealing with the fatigue tolerance evaluation of transport category rotorcraft metallic structure. For safe-life evaluations, AC 27-1B, MG 11 (Fatigue Evaluation of Rotorcraft Structure) provides in depth guidance on acceptable means of compliance. The fatigue evaluation procedures outlined in paragraph AC 29 MG 11 are for guidance purposes only and are neither mandatory nor regulatory in nature. Specific issues related to substantiation of composite rotorcraft structures are addressed in AC 29 MG 8. Although a uniform approach to fatigue evaluation is desirable, it is recognized that in such a complex problem, new design features and methods of fabrication, new approaches to fatigue evaluation, and new configurations may require variations and deviations from the procedures described herein. It is recommended that major deviations from the procedures be coordinated with the certifying regulatory authority to assure compliance with the regulatory requirements.

b. Special Considerations. The structure of rotorcraft is subject to cyclic stresses in practically every regime of flight. In addition, since rotorcraft are highly maneuverable and capable of forward, rearward, sideward, vertical, and rotational flight, operating limitations due to fatigue are possible in practically all flight situations. Corrosion and other environmental damage are also common in rotorcraft operations. For these reasons, special attention should be focused on the fatigue evaluation of rotorcraft structure.

c. Background.

(1) Fatigue substantiation of rotorcraft dynamic components was first implemented in the 1950's by means of Safe-Life Methodology. Many advances in design and analytical methods have been made in the state-of-the-art and in industry practices since that time. To date, Safe-Life Methodology, as described in AC 27-1B, MG 11, is considered successful in providing a high level of reliability, but could be improved by taking into account the strength-reducing effects of damage likely to occur in manufacturing, maintenance, and in service, including corrosion, accidental damage, or manufacturing/maintenance flaws. The introduction of composites led the manufacturers/certifying authorities to take into account the specific static and fatigue strength-reducing effects of aging, temperature, moisture absorption, impact damage, and recognition of a minimum quality standard. In parallel, crack growth methodology

in metal structure has been successfully used for solving short-term airworthiness problems in rotorcraft, and as the certification basis for civil and military transport aircraft applications.

(2) Recognizing that advances in state-of-the-art and industry practice warranted changes to the existing fatigue requirements in Part 29, the regulatory requirements of § 29.571 were substantially revised. The revision to § 29.571 requires new guidance material containing compliance provisions related to the changes. AC 29 MG 11 provides this material with respect to the flaw tolerance requirements for metallic structure and is supplemented by AC 27-1B MG 11. Guidance material specifically addressing *composite* rotorcraft structure is provided in Chapter 3, MG 8 of this AC and is supplemented by AC 20-107A.

d. Introduction.

(1) Definitions. These definitions are provided to define the terms used in applying the requirements specified in FAR 29.571, Amendment 28, and may differ from other definitions.

(i) Fatigue Tolerance. The capability of structure to continue functioning without catastrophic failure after being subjected to fatigue (repeated) loads expected during operation of the rotorcraft. Fatigue tolerance should be achieved by flaw tolerance design, or if impractical, safe-life design, or a combination.

(ii) Safe Life. The capability of as-manufactured structure as shown by tests, or analysis based on tests, not to initiate fatigue cracks during the service life of the rotorcraft or before an established replacement time.

(iii) Flaw Tolerance. The capability of rotorcraft structure to achieve fatigue tolerance accounting for the presence of flaws and damage which may occur in manufacturing and service use. Flaw tolerance can be achieved by either flaw tolerance safe-life or fail-safe designs. The term 'Damage Tolerance' is frequently used to describe the ability of a structure to tolerate the effects of flaws and damage; however, the terminology of FAR 29.571, Amendment 28, is used in this AC to maintain consistency.

(iv) Flaw tolerant Safe Life. The capability of as-manufactured structure with expected flaws as shown by tests, or analysis based on tests, not to initiate fatigue cracks during the service life of the rotorcraft or before an established replacement time.

(v) Fail-Safe. The capability of structure remaining after a partial failure to withstand design limit loads without catastrophic failure within an inspection period.

(vi) Multiple Load Path. Structure providing two or more separate and distinct paths of structure that will carry limit load after complete failure of one of the members.

(vii) Active Multiple Load Path. Structure providing two or more load paths that are all loaded during operation to a similar load spectrum.

(viii) Passive Multiple Load Path. Structure providing load paths with one or more of the members (or areas of a member) relatively unloaded until failure of the other member or members.

(ix) Crack Arrest Feature. Structure that does not provide completely separate and distinct load paths but does provide features of design such as bonded and/or riveted straps, changes in geometry, or special processing techniques such as rolling or coining to retard or arrest crack growth.

(x) Slow Crack Growth Feature. Structure (single element or multiple element) that provides for slow crack growth by material selection, material processing, limitation of stress levels, geometrical design features, or by other methods.

(xi) Flaw. Intrinsic imperfections such as inclusions, forging laps, or porosity, and discrete damage such as gouges, scratches, nicks, corrosion, fretting, wear, or impact, that could be expected during manufacture or operation.

(xii) Design Limit Loads. The maximum loads to be expected in service, as defined by § 29.301(a).

(xiii) As Manufactured. Product and/or component that has passed the applicable quality control process and has been found to conform to the approved design within the allowable tolerances.

(xiv) Residual Strength. The strength retained for some period of unrepaired use after a failure or partial failure due to fatigue, corrosion, and accidental or discrete source of damage.

(xv) Principal Structural Element (PSE). A structural element that contributes significantly to the carrying of flight or ground loads and whose failure due to fatigue can lead to catastrophic failure of the rotorcraft.

(2) Rotorcraft Fatigue Tolerance. Fatigue tolerant design as substantiated by fail-safe flaw growth or flaw tolerant safe-life means outlined in § 29.571 and paragraph AC 29 MG 11g is required for all PSE's, unless it entails such complications that an effective flaw tolerant structure cannot be achieved within the limitations of geometry, inspectability, or good design practice. Good design practice includes consideration of component complexity, component weight, methods of production, and component life cycle cost. Under these circumstances, a design that complies with

safe-life criteria should be used. Typical examples of structure that might not be conducive to flaw tolerance designs are swashplates, main rotor shafts, push rods, small rotor head components (i.e., devices, bolts, etc.), landing gear, and gearbox internal parts, including bearings. In addition, the need for the use of inspection techniques and equipment or highly trained personnel--resources not available (for economic or other reasons) to the small operator or in remote areas of operation--should be carefully considered.

(3) Test Background. Experience with the application of methods of fatigue evaluation indicates that a relevant test background should exist in order to achieve the design objective. It is the general practice within industry to conduct flaw tolerance tests for design information and guidance purposes. Flaw location and crack growth data based on test results and service history of similar parts, if available, should also be considered in establishing a recommended inspection program.

(4) Manufacturing and Maintenance Considerations. Assurance of structural adequacy also includes manufacturing, overhaul and repair, and service maintenance in accordance with design requirements, design specifications, maintenance procedures, overhaul and repair instructions, quality control to monitor compliance, and established manufacturing work processes including "frozen planning." The fatigue tolerance substantiation should include an evaluation of the details of the specific work processes used on each component to determine the potential sensitivities.

(5) Fatigue Tolerance Considerations. In the fatigue tolerance evaluation, the following items should be considered:

(i) Identification of the structure to be considered in each evaluation (a failure mode and effects analysis or similar method should be used).

(ii) The stresses and strains (steady and oscillatory) associated with all representative steady and maneuvering operating conditions expected in service.

(iii) The frequency of occurrences of various flight conditions and the corresponding spectrum of loadings and stresses.

(iv) The fatigue strength, fatigue crack propagation characteristics of the materials used and of the structure, and the residual strength of the damaged structure.

(v) Inspectability, inspection methods, and detectable flaw sizes.

(vi) Variability of the measured stresses, the actual flight condition occurrences, and the fatigue strength material properties.

e. Flight Loads Measurement Program. See paragraph c. of AC 27-1B MG 11.

f. Rotorcraft Usage Spectrum. See paragraph d of AC 27-1B MG 11.

g. Fatigue Tolerance Evaluation.

(1) General. A means should be established using the Safe-Life approach or a Flaw Tolerant approach to control the airworthiness of principle structural elements identified under § 29.571(a)(1)(i) While the safe-life approach is acceptable under certain circumstances as defined in § 29.571, the Flaw Tolerant safe Life and Fail-Safe (residual strength after crack/flaw growth) approaches are to be used unless shown to be impractical as stated in § 29.571. A Flaw Tolerance evaluation of structure is intended to ensure that even when expected flaws are present, the structure will withstand service loads without failure until the flawed parts are replaced or until the flaws (or resulting fatigue cracks) are detected and appropriate action taken. The Flaw Tolerant evaluation may be achieved by either Flaw Tolerant Safe Life, Fail Safe (residual strength after crack/flaw growth), or a combination thereof. Flaw Tolerant Safe Life includes the analyses and/or testing currently associated with safe-life substantiation, plus consideration of flaws to establish a replacement time, or a safe life greater than the service life of the rotorcraft. Crack growth methods include the analyses and/or testing currently associated with a Damage Tolerance Assessment (DTA) to establish an inspection program. Design features that should be used in attaining a fatigue tolerant structure are:

(i) Use of multi-path construction and the provision of crack stoppers to limit the growth of cracks and to provide adequate residual strength.

(ii) Selection of materials and stress levels that preclude crack growth or crack initiation from flaws or that provide a controlled slow rate of crack propagation combined with high residual strength after initiation of cracks. Tests are required to substantiate crack propagation rates.

(iii) Design to permit detection of cracks and other flaws, including the use of crack detection systems, in all critical structural elements before cracks can propagate and become dangerous or result in appreciable strength loss and to permit replacement or repair.

(iv) Use of multiple element structures may be provided so that damage or failure occurring in one element of the member will be confined to that element and the remaining structure will still possess adequate load-carrying ability until the failed element is discovered by inspection.

(v) Provisions to limit the probability of concurrent multiple damage, particularly after long service, should be provided. These provisions should ensure adequate independence of each failure mode of multi-path constructions. The use of full-scale fatigue test articles are recommended in this evaluation.

(vi) Identification of Principal Structural Elements. Typical examples of such elements are:

(A) Rotor blades, attachment fittings, and dynamic systems.

(B) Rotor heads, including hubs, hinges, some main rotor dampers, and support structures.

(C) Control system components subject to repeated loading, including control rods, servo structure, and swashplates.

(D) Rotor supporting structure (torque and lift path from airframe to rotor head).

(E) Fatigue critical fuselage structures, including stabilizers and auxiliary lifting surfaces.

(F) Main fixed or retractable landing gear and fuselage attachment structure.

(G) Gearboxes, driveshafts, and couplings.

(vii) Identification of Locations within Principal Structural Elements to be Evaluated. The locations to be considered for fatigue tolerance evaluation can be determined by analysis or by fatigue test on complete structures or subcomponents.

(A) The following should be considered:

(1) Strain gauge data on undamaged structure to establish points of high stress concentration as well as the magnitude of the concentration;

(2) Locations where analysis shows high stress or low margins of safety;

(3) Locations where permanent deformation occurred in static tests;

(4) Locations of potential fatigue damage identified by fatigue analysis;

(5) Locations where the stresses in adjacent elements will be at a maximum with an element in the location failed;

(6) Partial fracture locations in an element where high stress concentrations are present in the residual structure;

(7) Locations where detection would be difficult;

(8) Design details or similarly designed components that are prone to fatigue or other damage as shown by service experience; and,

(9) Components fabricated from materials of potentially low fracture toughness or high flaw growth rate.

(B) In addition, the areas of probable damage from sources such as a severe corrosive and/or fretting environment, a wear and/or galling environment, or a high maintenance environment should be determined from a review of the design and past service experience.

(viii) Extent of Flaws. Each particular design should be assessed to establish appropriate damage criteria in relation to inspectability and flaw extension characteristics. In any flaw determination, it is possible to establish the extent of flaws in terms of detectability with the inspection techniques to be used, the associated single element failure or initially detectable flaw size, the residual strength capabilities of the structure, and the likely flaw extension rate (after either an element failure or a partial failure) considering the expected stress redistribution under the repeated loads expected in service and with the expected inspection frequency.

(ix) Provisions for Inspection. The designer should strive to ensure adequate inspectability of all structural parts to qualify them under the fail-safe crack growth provisions. In those cases where blind areas or surfaces exist, suitable design features should be provided to allow inspection techniques (either visual or nondestructive testing, as necessary) to assure adequate residual strength is achieved unless shown to be impractical due to limitations of geometry and good design practice. In addition, the alternate safe-life approach to fatigue tolerance should be implemented if the inspection techniques are shown to be too complicated and impractical.

NOTE: Removed Figure AC 29 MG 11-1, "S-N or P-N Curve Usage," AC 29-2C, dated 9/30/99.

(x) Testing of Principal Structural Elements. Tests will 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. Test on complete structures are recommended as baseline for design validation and demonstration of compliance with the requirements. The nature and extent of tests on complete structures or on portions of the primary structure will depend upon:

(A) applicable previous design, construction and tests.

(B) service experience in connection with similar structures.

(C) extent and validation of the structural analysis (FEM models, design manuals, stress or strain measurements, …).

(D) conservatism and margin of safety of the validation, included assumptions on the loading conditions.

(2) Flaw Tolerant Safe Life Demonstration

(i) Flaw tolerant safe-life substantiations provide a safe period of operation of structure with expected flaws with only routine inspections necessary.

(ii) Designs, processes, materials and stress levels that preclude the occurrence of flaws, or that provide a tolerance to crack initiation from flaws, can be used to achieve a Flaw-Tolerant Safe Life structure.

(iii) Flaw-Tolerant Safe Life uses analyses and/or testing similar to that of Safe Life, except that structures with expected flaws are tested rather than as-manufactured structures. Tests will be necessary when the basis for analytical prediction may not be reliable, such as for complex components.

(iv) Evaluation of Flaw Types and Sizes (“Threat Assessment”).

(A) For each zone of the component, a systematic evaluation of the types and sizes of flaws should be conducted, based on the processes and practices used in design, manufacturing, maintenance, and overhaul, and service experience.

(B) Flaw types. The types of flaws considered should include intrinsic/discrete flaws, impacts, scratches, corrosion, fretting and wear. Consideration should also be given to factors that reduce scatter and deviations from nominal structures, such as "frozen processes", Flight Critical Parts programs, material selection to mitigate intrinsic flaws (inclusions and defects), and procedures to reduce manufacturing deviations. Implementation of a specific manufacturing inspection process can justify the elimination of some flaws from consideration if that process can be shown to be highly reliable, well-controlled and documented, and systematically

required. Where surface treatments, protective coatings, or shields are used, these should be considered when establishing the likely location and type of the flaw.

(C) Flaw size. The flaw sizes to be considered should be representative of those which are likely to be encountered during the structure's service life resulting from the manufacturing, maintenance, and service environment. An analysis may be used combining the distribution of likely flaw sizes, the criticality of location and orientation, and the likelihood of remaining in place for a significant period of time.

(v) Determination of Retirement Time.

(A) Flaw Characterization. S/N curve shapes and scatter factors should be defined for the specific materials selected for the Flaw Tolerant Safe Life structure. This may be accomplished by testing a number of small specimens ("coupons") or by reference to existing characterizations of the selected material. In addition, the effect of the flaws identified in paragraph g(2)(iv) above, on the S/N curve shape and basic fatigue strength of the selected material should be determined. This may be accomplished by testing a number of coupons incorporating the representative flaws, or by reference to existing characterizations of the same material and flaws. A coupon program incorporating representative manufacturing processes and relevant design features may also be appropriate to define "equivalent" flaws that produce the same strength-reducing effect as the representative flaws, but can be more easily applied and controlled.

(B) Flawed Full-Scale Specimens. In order to determine the mean fatigue strength considering flaws, a number of flawed full-scale specimens should be tested. Representative flaws should be imposed on these specimens in each critical location on the structure where flaws are likely to occur. Equivalent flaws may be used if they have the same or a more severe strength-reducing effect, as determined by a flawed coupon test program or by experience with similar applications.

(C) Mean Strength Determination. Conventional stress vs. number of cycles (S/N) or spectrum safe-life fatigue testing may be performed on flawed full-scale specimens to establish the mean fatigue strength. AC 27-1B MG 11 provides general guidance in the conduct of safe-life fatigue testing and the establishment of mean fatigue curves from the test data. The strength of flawed structure may also be derived from un-flawed structure test results by imposing the reductions in strength determined in paragraph g(2)(v)(A) above, or derived from experience with similar structure.

(D) Working S/N curve. Reduction factors should be applied to the mean curve in deriving a working S/N curve. These factors should include consideration of the number of specimens tested, variability (scatter), previous test data on the same materials or similar structures, as well as service experience. To preclude a dual penalty situation, reduction by two standard deviations rather than three (conventional safe life) may be used if justified by appropriate design features such as multiple elements or unmistakable flaw indications or by material properties that provide benign

types of failure modes. Where new materials or designs are being evaluated, it is recommended that a larger reduction factor be used until additional test data justifying a change are available.

(E) Replacement Time. Utilize the working S/N curve and loading spectrum of paragraph AC 29 MG 11f in substantiating a replacement time for the flawed structure by cumulative damage means. The replacement time established should be included in the airworthiness limitations section of the document established under § 29.1529.

(3) Fail-Safe Demonstration

(i) Safe Crack Growth. Safe crack growth (fracture mechanics) substantiation should show that the damage growth rate under the repeated loads expected in service (between the time at which the damage becomes initially detectable and the time at which the extent of damage reaches the value for residual strength evaluation) provides a practical basis for development of the inspection programs.

(ii) Inspection methods The designer should strive to ensure adequate inspectability of all structural parts to qualify them under the fail-safe flaw growth provisions. Inspection means that are appropriate for safe crack growth design follow:

(A) Routine Inspections. To support routine inspection programs, blind areas should be avoided, where practical.

(B) Special Inspections. These inspections will generally result from test results as well as the geometry of the design. Care should be given to special inspection techniques to be used in the field. Inspection techniques requiring facilities and resources beyond the capability of the small operator or not generally available in remote-area operations traditionally associated with rotorcraft operations should not be specified for field inspections. Conservative sizes for detectable cracks or other flaws should be used. Sufficient inspection intervals should be provided to detect cracks before they grow from a detectable size to a size that reduces the remaining strength below design limit strength.

(C) Pressurized Chambers. This design feature may be used to detect cracks that cause a chamber to lose its pressure (either positive or negative). Gauges can indicate the loss of pressure, or dye may be used if it is shown to be a dependable indicator.

(D) Vibration Generation. This characteristic should be considered both from the aspect of vibrations giving indications of a failure, and from the aspect of the increased fatigue loading resulting from the vibrations.

(E) Noise Generation. If initial failure will result in a clear and unmistakable noise that is sufficiently continuous and loud, this characteristic can be used in achieving flaw tolerance without additional special inspections.

(F) Crack Detection Wire, Foil, etc. Detection wire may be used in areas that are sufficiently well defined so that the wire can be properly located. This technique is appropriate in areas otherwise difficult to inspect.

(iii) Multiple Elements. Use of multiple element structures may be provided so that damage or failure occurring in one element of the member will be confined to that element and the remaining structure will still possess adequate load-carrying ability until the failed element is discovered by inspection. These provisions should be designed to provide adequate independence of each failure mode of multi-path constructions. The use of full-scale fatigue test articles is recommended in this evaluation. Examples of concurrent multiple damage to be avoided are:

(A) Simultaneous failure or partial failure of multiple path discrete elements working at similar stress levels.

(B) Failures or partial failures, in adjacent areas, due to redistribution of loading, following a failure of a single element.

(iv) Partial Failures. The following are typical examples of the type of partial failures that may be considered in the flaw growth fail-safe evaluation:

(A) Detectable skin cracks in the trailing edge sections of rotor blades.

(B) Detectable failures of individual straps in “strap packs.”

(C) Detectable skin cracks emanating from the edge of structural openings or cutouts.

(D) Detectable circumferential or longitudinal skin crack in the basic fuselage or tail boom structure.

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

(F) Presence of a detectable fatigue failure in at least the tension portion of the spar web or similar element.

(G) Detectable failure of a primary attachment, including blade attachment fittings and control surface hinge and fittings.

(H) Fretting, corrosion, and galling conditions expected in service.

(v) Initial inspection time. The initial crack size to be used for evaluation of the initial inspection time should be the flaw size controlled by manufacturing quality and damage that can go undetected for the life of the part.

(vi) Evaluation of the inspection intervals. For multiple load paths, a minimum of three inspection intervals is recommended between the initially detectable damage time and the time when residual strength is reduced to design limit load by crack growth. For single element structures, a minimum of four inspection intervals is recommended. The repeated loads should be 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 it is significant.

Tests of two or more specimens should be used to obtain crack propagation data using either a realistic load spectrum or an accelerated load (spectrum or single) associated with the use of propagation theory and data after cracks have been initiated. As far as applicable, crack growth tests should be carried out considering a fatigue crack naturally induced in testing. In the other cases, crack initiation points should be made by suitable methods, reference paragraph g(3)(viii). Unless a more rational method with an equivalent level-of-safety is applied for, the following methods of setting inspection intervals should be applied. In all cases, the inspection methods and intervals should adequately consider variables such as inspectability, type of inspection, crack growth behavior, and other scheduled maintenance considerations.

(A) For a single element (load path) structure, plot the data and set the inspection as shown in Figure AC 29.MG 11-1.

(1) Set the initial inspection at $L_1/3$.

(2) Set the repetitive inspection intervals at $L_2/4$.

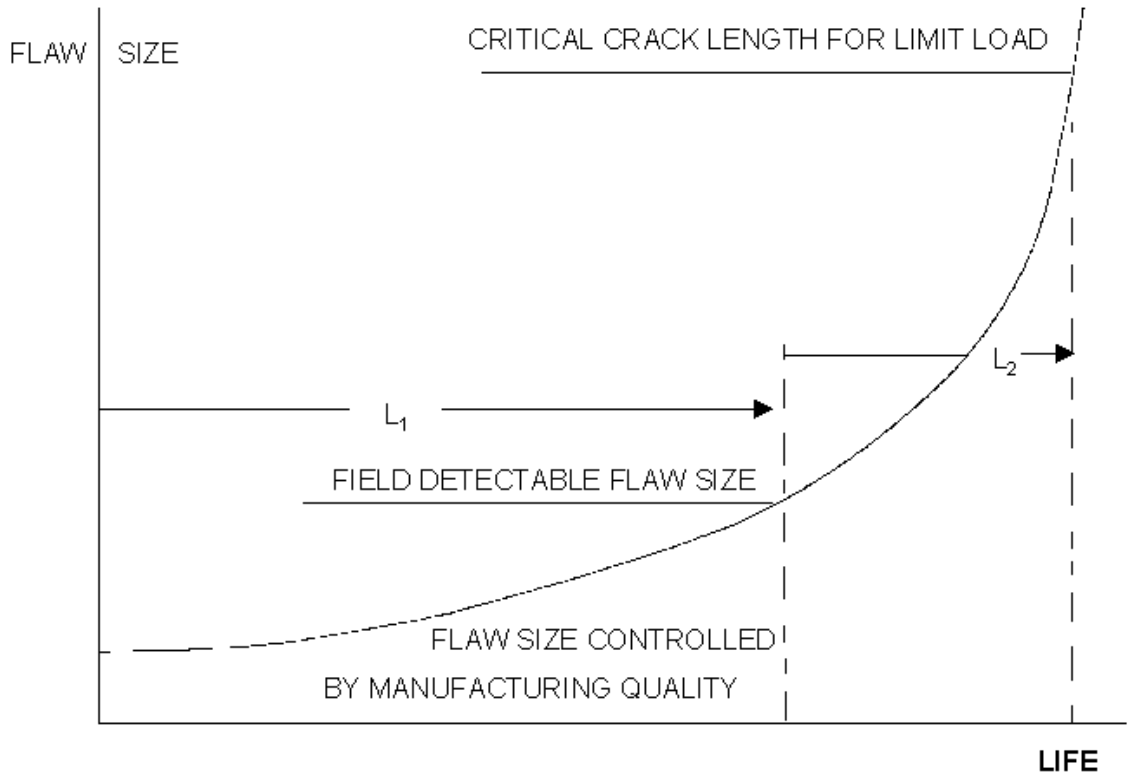

FIGURE AC 29.MG 11-1 CRACK GROWTH FOR SINGLE ELEMENT STRUCTURE

(B) For a multi-element load path structure substantiated by test:

(1) Obtain a complete failure of the critical load path, either by initiating and propagating a crack or by artificially disabling it.

(2) Damage (i.e. crack growth starting from nominal flaw or crack initiation) accumulated in the structure prior to load path failure should be considered in establishing L_r.

(3) Note when the residual strength of the remaining elements decreases to limit load due to crack growth, provided with a load margin if appropriate.

(4) From Figure AC 29.MG 11-2, set repetitive inspection intervals at $L_r/3$.

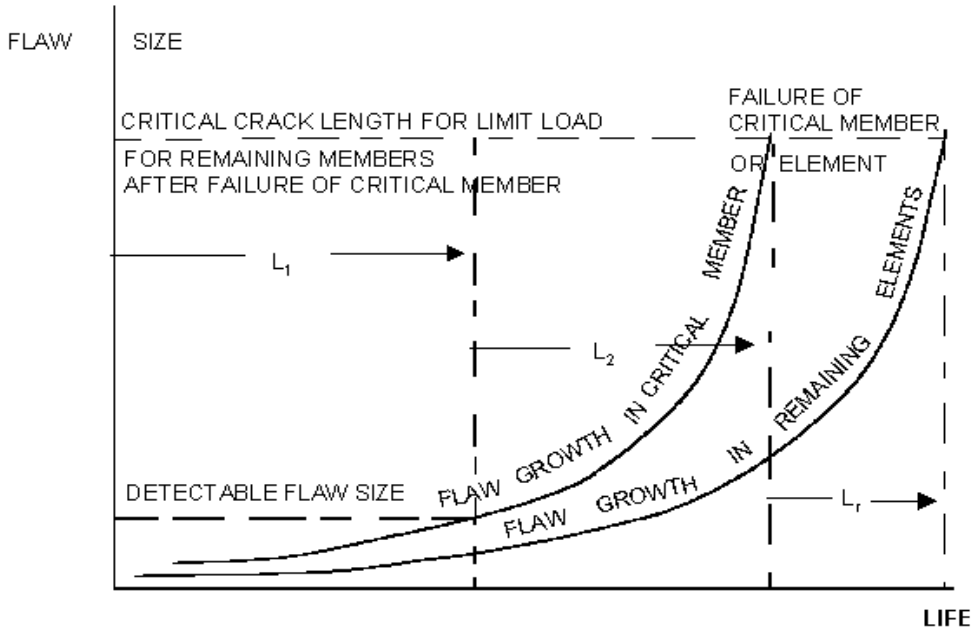

FIGURE AC 29.MG 11-2 CRACK GROWTH FOR REMAINING ELEMENTS OF MULTIPLE ELEMENT STRUCTURE

(vii) No/Benign Crack Growth

(A) This method depends on replacement rather than inspection to ensure the continued airworthiness of a PSE. The replacement time is established based on consideration of crack growth characteristics.

(B) To substantiate a structure in the no/benign crack growth category requires demonstration either by analysis, testing, or both, that the rogue crack (a_R), which is the most severe crack consistent with manufacturing, maintenance and service environment, will not grow or will not grow to critical size (a_{CRIT}) under the service loading and environment before the structure is replaced. The crack should be assumed at the critical location, as defined by the largest stress intensity factor range under the expected service loading range including the ground–air–ground cycle.

(C) To determine the replacement time, the rogue cracks should be assumed at the critical location and the crack growth characteristics should be determined for the expected load/environment spectrum. There are three different scenarios that could result from a crack growth assessment and be used for establishing a replacement time. These scenarios are illustrated in Figures AC 29.MG 11-3, -4 and –5.

(D) The no crack growth scenario is illustrated in Figure AC.29.MG 11-3. In this figure, the rogue flaw does not grow or growth is insignificant. In this case the replacement time should not exceed the design service life (L_{DES}).

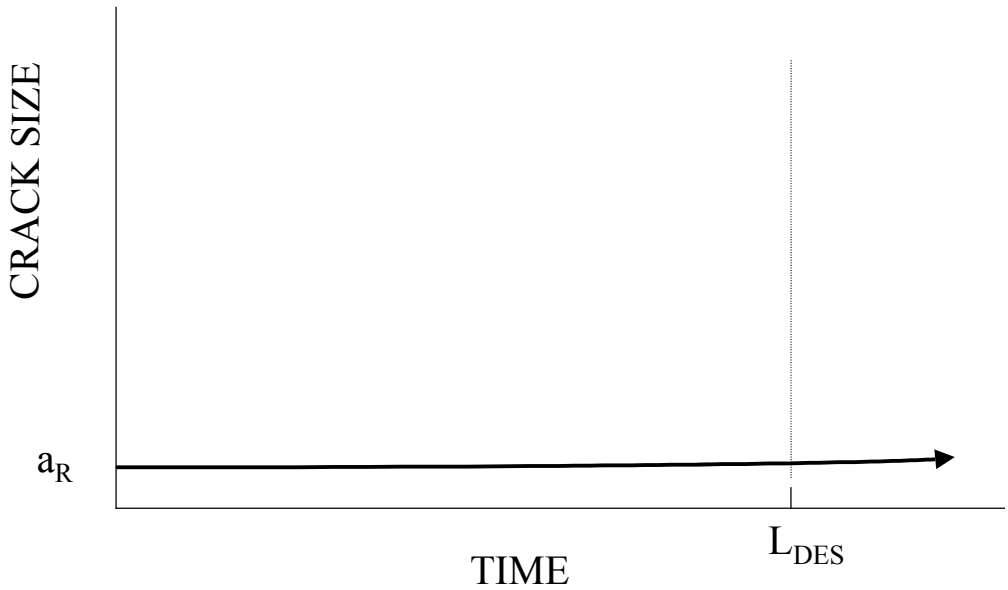

FIGURE AC 29.MG 11-3

(E) Figure AC 29.MG 11-4 illustrates the scenario where the rogue crack grows relatively slowly but becomes critical prior to becoming detectable (a_{DET}). In this case the replacement time should be set equal to the total crack growth life (L_T) divided by a factor N.

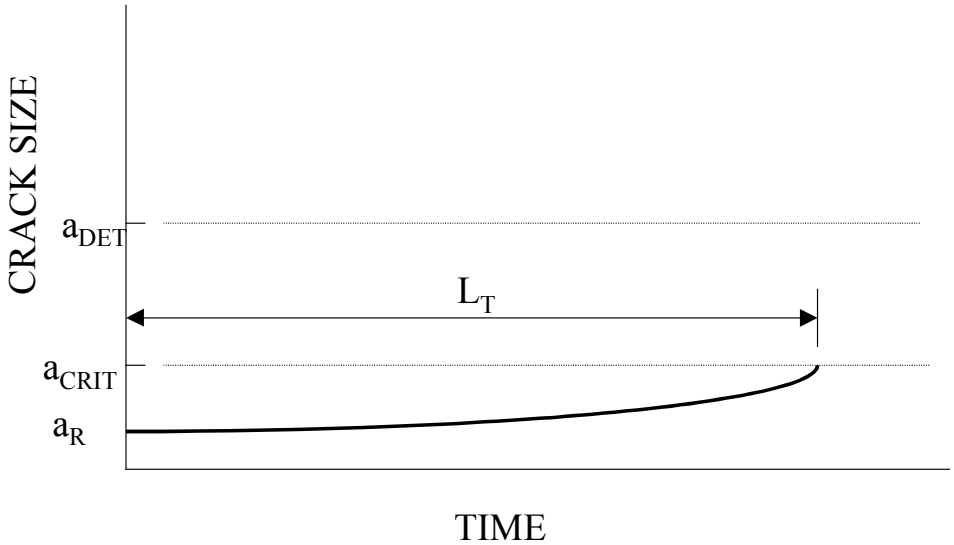

Figure AC 29.MG 11-4

(F) Figure AC 29.MG 11-5 illustrates the scenario where the rogue crack grows to a detectable size (at L_1) before becoming critical (at L_1+L_2). In this case the replacement time should be set equal to the total crack growth life (L_1+L_2) divided by a factor N.

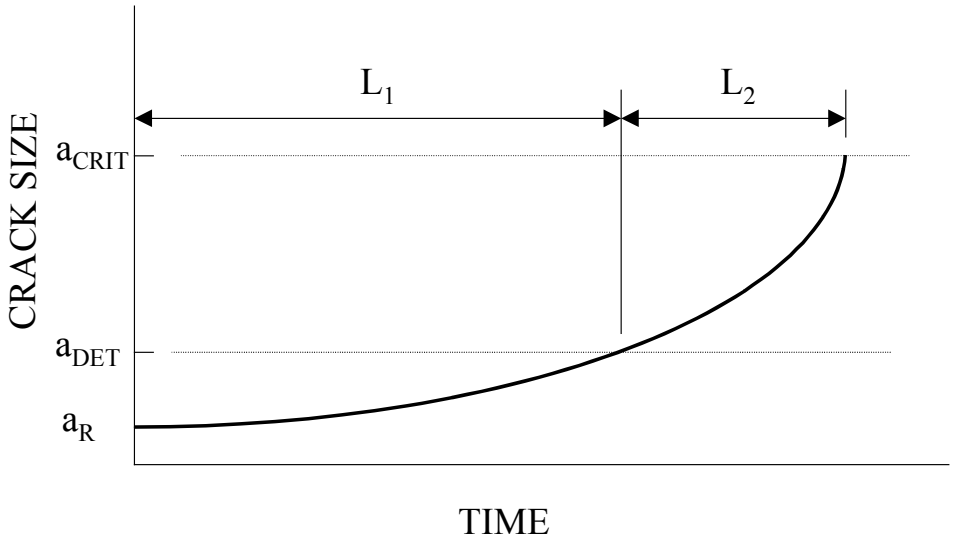

Figure AC 29.MG 11-5

(G) In determining the factor of N to be used for determining the replacement time, consideration should be given to the crack growth data used (e.g., top of scatter data versus average data, number of specimens used to generate data, etc.).

(H) The minimum suggested N value should be (1) N=2 in the case where the conservative top-of-scatter crack growth data are used in the crack growth analysis, or (2) N=4 when the average crack growth data are used in the crack growth analysis, or when the crack growth life is obtained from the crack growth test of one specimen (for two or more full scale specimens, N=3 of the shortest crack growth life can be used).

(I) It should also be noted that, in the no/benign crack growth category, the validity of the crack growth threshold, ΔK_{ITH}, is especially important since there is no element of inspection to ensure continued airworthiness. Consistent with this, additional attention may be required relative to validating the crack growth threshold value(s) used in the analyses. Consideration should be given to the influence of the test procedure used to develop values, microstructure, heat treatment, crack size, loading conditions, environment, grain size and orientation, etc. In general a coupon testing program may be necessary to develop a consistent ΔK_{ITH} data base and the use of bibliographic data might require additional conservatism.

(viii) Testing of Principal Structural Elements.

(A) The nature and extent of tests on complete structures or on portions of the primary structure will depend upon applicable previous design, construction, tests, and service experience in connection with similar structures. For fail-safe testing considering crack propagation, simulated cracks should be as representative as possible of actual fatigue damage. Where it is not practical to produce actual fatigue cracks, flaws can be simulated by cuts made with a fine saw, sharp blade, guillotine, Electrical Discharge Machine, or other suitable means. The validity of saw cuts, etc., should be verified by comparison to coupon tests of a cracked specimen of the same material. In those cases where bolt failure, or its equivalent, is to be simulated as part of a possible flaw configuration in joints or fittings, bolts can be removed to provide that part of the simulation.

(B) Other test and inspection programs may be used if shown to have comparable or better probability of assuring that a catastrophic fatigue failure will not occur.

(ix) Analytical Evaluation The fail safe characteristics can also be shown analytically, by reliable and conservative methods, by demonstrating that the repeated loads and limit load stresses do not exceed those of previously verified fail safe designs of similar configuration, materials, and inspectability. Analytical models should be properly validated to assure that:

(A) Stress analysis is sufficiently accurate. This can be achieved by similarity with full scale strain surveys, thermography, or other suitable testing approaches.

(B) Fracture mechanics is properly applied.

(C) Load interaction models are verified for the specific load histories.

(D) Consistent material data base is used.

(x) Inspection plan. Detection of flaws before they become dangerous is the ultimate control in ensuring the flaw tolerance characteristics of the structure. Therefore, the applicant should provide sufficient guidance information to assist operators in establishing the frequency, extent, and methods of inspection of the critical structure, and this kind of information should, under § 29.571(a)(2), be included in the maintenance manual required by § 29.1529. Due to the inherent, complex interactions of the many parameters affecting flaw tolerance, such as operating practices, environmental effects, load sequence on flaw growth, and variations in inspection methods, related operational experience should be taken into account in establishing inspection procedures. Comparative analysis can be used to guide the changes from successful past practice, when necessary. Therefore, maintenance and inspection requirements should recognize the dependence on experience and should be specified in a document that provides for revision as a result of operational experience, such as the one containing the operator's FAA/AUTHORITY-approved structural inspection program developed through the Maintenance Review Board (MRB) procedures for FAR Part 121 operators.

(4) Safe-Life Demonstration

(i) Information to guide fatigue evaluation based on the safe-life approach is described in details in AC 27-1B MG 11. The safe-life approach is used when both the fail-safe and flaw-tolerant safe-life methods are verified to be impractical due to considerations of inspectability, geometry, or good design practice as described above in paragraph d(2).

(ii) The safe-life approach may not account for flaws and imperfections due to manufacturing and in service conditions as compared to the flaw tolerant approaches. Therefore, conservative factors that adjust for variations in both load and strength are generally utilized when substantiating by the safe-life method.

(5) Combining Methods.

(i) Components may be managed by a retirement time based on the Flaw Tolerant Safe-Life method, paragraph g(2); by an inspection program based on the Fail-Safe method, paragraph g(3); or, if approved, a retirement time based on conventional Safe Life, paragraph g(4). In some cases it may be appropriate to

establish a retirement time based on the lowest from several methods, and in some cases it may be appropriate to establish both a retirement time and an inspection program for a given structure.

(ii) Retirement Times. The conventional Safe-Life retirement time determined from as-manufactured parts and conventional working curves may be lower than that determined from a Flaw Tolerant Safe-Life evaluation. In this event, the lower of the two retirement times should be used. Additionally, inspection intervals resulting from a Fail-Safe evaluation may be used as retirement times if practical, which removes the need to conduct that specific inspection. In this event, the conventional safe life retirement time should be used if lower.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 12. § 29.865 (Amendment 29-43) EXTERNAL LOADS.

a. Background. In the United States (U.S.), the external load attaching means standards for transport and normal category rotorcraft were originally contained in Subpart D, "Airworthiness Requirements of FAR Part 133, Rotorcraft External-Load Operations." Amendment 29-12, issued in 1977, added a new § 29.865, which moved these standards from Part 133 to Part 29. An identical transfer occurred in 1977 for Part 27. Amendment 29-26, issued in 1990, clarified the intent of Amendment 29-12 but did not change it substantively. Transport Categories A and B and Normal Category rotorcraft were initially used under Part 133 operations, and after Amendment 133-6, restricted category rotorcraft were also included under Part 133 operations. The carriage of persons external to the rotorcraft for hire first came about when a Part 29 operator, exempt from Part 133, transferred harbor pilots to and from ships by a hoist and sling. The exemption was granted to study the feasibility of passenger transfer outside of the cabin. Grant of the exemption was based, in part, on similar, prior operations that had been conducted in Europe and Africa, for hire, with helicopters certified by the appropriate authorities and, in part, on similar military and public helicopter operations, not for hire, in the U.S. Subsequently, Amendment 133-9, adopted in January 1987, established a new Class D rotorcraft load combination (RLC) for transporting loads other than Class A, B, or C that are specifically approved by the administrator external to the rotorcraft. Amendment 133-9 also provided for the limitations and conditions for transport of external loads other than Class A, B, or C and the necessary, associated safety requirements. Part 29 has recently been changed to reflect RLC Class D requirements. Also, the scope and thus the title of the standard have changed from "External load attaching means" to "External loads" to reflect the more comprehensive approach for external loads required to ensure the proper level-of-safety.

(1) In other Nations the operations standards have developed differently and more diversely and do not necessarily use the RLC Class A, B, C and D definitions of § 1.1 in the same way as FAA operations standards do. Thus the International commonality of this advisory material (like § 29.865) is based on whether or not an external load is jettisonable or non-jettisonable and whether or not the load is HEC or NHEC.

(2) Whenever possible, the more generic, internationally harmonized terminology (i.e., jettisonable or non-jettisonable and HEC or NHEC) is used in this material. However, references to U.S. operational terms are made in parentheses where deemed necessary and tabulated to ensure clarity of purpose and proper, consistent approvals to U.S. operations standards.

b. Explanation.

(1) This advisory material contains guidance for certification of helicopter external load attaching means and load carrying systems to be used in conjunction with operating rules such as Part 133, "Rotorcraft External Load Operations." Subpart D of Part 133 contains supplemental U.S. airworthiness requirements. FAR Part 1 defines four RLC classes that are approvable under the U.S. Part 133 operating rules and that are eligible for certification under § 29.865. The four U.S. RLC classes are summarized in figure AC 29 MG 12-1 and discussed in paragraph d. Under U.S. operating rules RLC Classes A, B, and C are eligible, under specific restrictions, for both human external cargo (HEC) and nonhuman external cargo (NHEC) operations. However, under U.S. operating rules, RLC Class D only is eligible for transporting HEC for compensation (see figure AC 29 MG 12-1). For further information, AC 133-1A, "Rotorcraft External-Load Operations in Accordance with FAR Part 133," October 16, 1979, may be reviewed. Also, paragraph AC 29.25 (reference § 29.25) concerns, in part, jettisonable external cargo.

(2) FAR 29.865 provides a minimum level of safety for rotorcraft designs to be used with operating rules such as Part 133. Certain aspects of operations such as microwave tower and high-line wire work may also be regulated separately by other Federal agencies such as DOE, EPA, and OSHA or by other international entities. For applications that could come under multiple agency regulation (or regulation by other entities), special certification emphasis will be required by both the applicant and the certifying authority to ensure all relevant safety requirements are identified and met. Potential additional requirements, where thought to exist, are noted herein.

(3) The methods of this AC are intended to apply only to either new designs or to major modifications that occur after the effective date of Amendment 29-43 (i.e., "ADD DATE"). Thus it is not intended that these requirements be imposed retroactively. However, after the effective date of Amendment 29-43, all applications to certify new rotorcraft systems for NHEC or HEC operations would be required to comply with the equipment standards, as well as, the operational requirements in effect at that time.

c. Definitions.

(1) Applicable cargo type. The cargo type (i.e., NHEC, HEC, or both) that each RLC Class is eligible to use by regulation (figure AC 29 MG 12-1 contains explicit definitions for U.S. Part 133 Operations).

(2) Backup Quick-Release Subsystem (BQRS). The secondary or "second choice" subsystem used to perform a normal or emergency jettison of external cargo.

(3) Cargo. The part of any Rotorcraft-Load Combination that is removable, changeable, and is attached to the rotorcraft by an approved means.

(4) Cargo hook. A hook that can be rated for both HEC and NHEC. It is typically used by being fixed directly to a designated hardpoint on the rotorcraft.

(5) Critical configuration. In cases where NHEC or HEC can have more than one shape, center-of-gravity, center-of-lift, and/or be carried at more than one distance in flight from the rotorcraft attachment, a critical configuration for certification purposes may or may not be determinable. If such a critical configuration can be shown to exist, then it may be examined for approval as a "worst case" (in lieu of examining the entire range of configurations that exist) to satisfy a particular certification criterion or several criteria, as appropriate.

(6) Dual actuation device (DAD). This is a sequential control that requires two distinct actions in series for actuation. One example is a covered switch that would require cover removal (or flip-up) followed by a switch activation for load release to occur. Another example is removal of a lock pin followed by a "then free" switch or lever activation for load release to occur. Under this definition, a load release switch protected by an uncovered switch guard is not acceptable.

(7) Emergency jettison (or complete load release). The intentional, instantaneous release of NHEC or HEC in a preset sequence by the QRS that is normally performed to achieve safer operation in an emergency (i.e., nonoptimum situation).

(8) External fixture. A structure external to and in addition to the basic airframe that does not have true jettison capability and has no significant payload capability in addition to its own weight. An example is an agricultural spray boom. These configurations are not "External Loads" certifiable under § 29.865.

(9) Fixed line flyaway. This is a helicopter extrication technique in which a person or persons in a PCDS are connected to a rope or cable attached to a helicopter. The aircraft lifts off with the HEC carried below it. The exact length of the line depends on the specific needs of the operation.

(10) Human external cargo (HEC). A person(s) that at some point in the operation is carried external to the rotorcraft. (Figure AC 29 MG 12-1 contains explicit definitions for U.S. Part 133 Operations).

(11) Nonhuman external cargo (NHEC). Any external cargo operation that does not at any time involve a person(s) carried external to the rotorcraft (figure AC 29 MG 12-1 contains explicit definitions for U.S. Part 133 Operations).

(12) Normal jettison (or selective load release). The intentional release, normally at optimum jettison conditions, of an NHEC.

(13) Personnel carrying device system (PCDS). The entire attached or suspended system used to carry HEC. This is any HEC carrying configuration such as a suspended (e.g., winch/hoist, cable, harness) HEC system or an attached (e.g., a rigid basket or cage attached to skids) HEC system.

(14) Primary Quick-Release Subsystem (PQRS). The primary or "first choice" subsystem used to perform a normal or emergency jettison of external cargo.

(15) Quick-release system (QRS). The entire release system for jettisonable external cargo, (i.e., the sum total of both the primary and backup quick-release subsystems). The QRS consists of all components including the controls, the release devices, and everything in between.

(16) Rescue hook (or hook). A hook that can be rated for both HEC and NHEC. It is typically used in conjunction with a winch/hoist or equivalent system.

(17) Spider: A spider is a system of attaching a lowering cable or rope or a harness to an HEC (or NHEC) RLC to eliminate unwanted flight dynamics during operations. A spider usually has four or more legs (or load paths) that connect to various points of a PCDS to equalize loading and prevent spinning, twisting, or other undesirable flight dynamics.

(18) True jettison capability. The ability to safely release an external load using an approved QRS in 30 seconds or less.

NOTE: In all cases, a PQRS should release the external load in less than 5 seconds. Many PQRS's will release the external load in milliseconds, once the activation device is triggered. However a manual BQRS such as a set of cable cutters could take as much as 30 seconds to release the external load. The 30 seconds would be measured starting from the time the release command is given and ending when the external load is cut loose.

(19) True payload capability. The ability of an external device or tank to carry a significant payload in addition to its own weight. If little or no payload can be carried, the external device or tank is an external fixture (see definition).

(20) Type inspection authorization (TIA). This is FAA Form 8110-1. It is used only for the purpose of authorizing official ground inspections and flight tests necessary to fulfill the requirements for type certification or supplemental type certification. Order 8110.4, Chapter 2, Section 1, Paragraph 16, states the criteria for TIA issuance.

(21) Winch/hoist. A winch is defined as a device that can employ a cable and drum or other means to exert a horizontal (i.e., x-rotorcraft axis) pull. A hoist is a similar device that exerts a vertical pull (i.e., a pull that does not typically exceed a 30-degree cone measured around the z-rotorcraft axis). The majority of "pull" devices used on rotorcraft are hoists. However, since a winch can be used to perform a hoist function by

use of a 90 degree cable direction change device (such as a pulley or pulley system), a winch system is approvable. Thus the terms "winch/hoist" and "winch/hoist system" are used throughout this AC.

(22) Winch/hoist demonstration cycle (or "one cycle"). This is the complete extension and retraction of at least 95 percent of the actual cable length, or 100 percent of the cable length capable of being used in service (i.e., that would activate any extension/retraction limiting devices), whichever is greater.

(23) Winch/hoist load-speed combinations. Some winch/hoist designs are such that the extension/retraction speed slows down as the load increases or near the end of a cable extension. Other winch/hoist designs maintain a constant speed as the load is varied. In the latter design, the load-speed combination simply means the variation in load at the constant design speed of the winch/hoist.

d. Procedures. Because of the technical detail contained in subparagraph (d); the following index is provided to assist in locating specific compliance procedures.

INDEX TO SECTION (d): "PROCEDURES"		
SUBSECTION	SUBSECTION TITLE	PAGE
d(1)	General Compliance Procedures for § 29.865	7
d(2)	General Static Structural Substantiation Procedures for § 29.865(a)	8
d(3)	Compliance Procedures for Functional Reliability and Durability Demonstration of Winch/Hoist Systems Under § 29.865(a) and § 29.865(b)(3)(i)	13
d(4)	Compliance Procedures for Cargo Hooks (or Equivalent Devices) and Their Related Systems under § 29.865(a), (b), and (c)	19
d(5)	Compliance Procedures for Maximum Limit Load Magnitude Determination for all Jettisonable RLC Applications under § 29.865(a)	21
d(6)	Compliance Procedures for Basic Load Distribution and Analysis under § 29.865(a)	21
d(7)	Compliance Procedures for General QRS Certification and Installation under § 29.865(b) and § 29.865(c)	23
d(8)	Compliance Procedures for Reliability Determination for Jettisonable NHEC and HEC QRS's and Devices under § 29.865(b)(3)	23
d(9)	Compliance Procedures for Electromagnetic Interference under § 29.865(b)(3)(i)	26
d(10)	General Compliance Procedures for HEC Applications under § 29.865(c)(1)	27
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d(12)	Compliance Procedures for QRS's under § 29.865(c)(1)	28
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INDEX TO SECTION (d): "PROCEDURES"		
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(1) General Compliance Procedures for § 29.865: For compliance with § 29.865, the applicant should clearly identify the Parts 1 and 133 RLC's (i.e., the type of operations) that are being applied for and all applicable cargo types (i.e., NHEC or HEC) that will be used (see figure AC 29 MG 12-1 following, for specific U.S. definitions). The structural loads and operating envelopes for each RLC class and applicable cargo type should be determined and used to formulate the flight manual supplement and basic loads report. The applicant should show by analysis, test, or both, that the rotorcraft structure, the external load attachment means, and (for HEC operations) the PCDS meet the specific requirements of §§ 29.865, 133.41, 133.43, 133.45, and the other relevant requirements of Part 29 for the proposed operating envelope.

In general, for compliance with § 29.865, the methods described by the following, procedural paragraphs are acceptable.

NOTE: It is possible, if approvable, to carry both HEC and NHEC externally, simultaneously as two separate external loads. However, in no case is it intended that

the approved Maximum Internal Gross Weight be exceeded for any approved HEC configuration (or combined NHEC/HEC configuration) in normal operations.

(2) General Static Structural Substantiation Procedures for § 29.865(a): The following static structural substantiation methods should be used (paragraph d(21) describes the fatigue substantiation methodology).

(i) Static structural substantiation: The following methods of static structural substantiation should be employed.

(A) NHEC applications. In most cases a standard static analysis alone is acceptable to show compliance.

(B) HEC applications. If a safety factor of 3.0 or more on the yield strength of the weakest component in the QRS, PCDS, and attachment(s) load path is used, only an analysis is required for certification. Otherwise, both an analysis and a full-scale ultimate load test of relevant parts of the QRS, PCDS, and its attachments that form the HEC load path(s) should be submitted.

(ii) NHEC applications. For NHEC applications, use of 2.5 g vertical limit load factor (N_{ZW}) at the maximum substantiatable cargo load (which is typical for heavy gross weight NHEC hauling configurations) is required by § 29.865(a). This 2.5 g limit load factor is based on an engineering evaluation and a rationalization of § 29.337 for high gross weight applications.

(iii) HEC applications. For HEC applications, which typically involve lower gross weight configurations, a higher limit load factor is required to ensure that limit load is never exceeded in service. The higher load factor for HEC applications should be the analytically derived maximum vertical limit load factor for the restricted operating envelope being applied for or, as a conservative option, a vertical limit load factor of 3.5 g's (reference § 29.337). Unless a more rational proposal is received, for HEC applications where maximum operating gross weight for the external load is between design maximum weight and design minimum weight, linear interpolation can be used between $N_{ZW\ MIN}$ and $N_{ZW\ MAX}$ versus gross weight to determine the design limit load factor. In no case may the vertical limit load factor be less than 2.5 g's for any RLC application for HEC. For example, an HEC external load-carrying attachment or PCDS that is certified to a limit vertical load factor of 2.5 g's and is installed in a minimum gross weight configuration rotorcraft capable of generating a vertical limit load factor of 3.2 g's could experience ((3.2/[2.5 x 1.5]) x 100) = 85 percent of ultimate load (i.e., 128 percent of limit load) under worst case conditions with new external hardware. However, if factors such as wear and corrosion have affected the structural integrity of the external load carrying hardware, the limit and ultimate load capability may decrease significantly and the current design standard could be exceeded. Certification policy is not to exceed limit load in service. Therefore, to meet the requirement of § 29.865(a), the external load carrying hardware would need to be designed to a higher design standard (i.e., to withstand a limit load factor of 3.2g's.).

FIGURE AC 29 MG 12-1
U.S. OPERATIONAL (PART 133) ROTORCRAFT-LOAD COMBINATION VERSUS APPLICABLE CARGO TYPE DATA AND DEFINITION SUMMARY

ROTORCRAFT-LOAD COMBINATION CLASS, CARGO TYPE	REQUIREMENT FOR CATEGORY "A" RATING AND OEI HOVER CAPABILITY
A, NHEC	NONE
A, HEC (SEE NOTE 2)	NONE
B, NHEC	NONE
B, HEC (SEE NOTE 2)	NONE
C, NHEC	NONE
C, HEC (SEE NOTE 2)	NONE
D, NHEC	NOT APPLICABLE (SEE NOTE 4)
D, HEC (SEE NOTE 1)	YES (SEE NOTE 3)

NOTES:

1. A person(s) (passenger OTHER than a crewmember and/or OTHER than a person who is essential to the external-load operation), when carried as an external load, can only be carried as a Class D RLC. These persons are being carried (transported).

2. A person WHO IS a crewmember or a person WHO IS essential and directly connected with the external-load operation is not being carried (transported) as a passenger. They are, instead, part of the operation. These persons are considered as RLC Class A, B, or C HEC as appropriate to the operation.

3. The rotorcraft are required to meet the Category A engine isolation requirements of Part 29 and have OEI/OGE hover performance capability, over the operating and weight envelopes applied for, to be eligible for certification to the Class D RLC.

(FIGURE AC 29 MG 12-1—continued)

4. NHEC Class D operations are not applicable. An alternate NHEC operational configuration, using the same rotorcraft, would become either a Class A, B, or C NHEC operation.

5. A Class D RLC operation may be conducted with an external cargo design having a physical configuration that meets the definitions of § 1.1 for RLC Class A, B, or C.

6. OEI power settings should not be used for certification credit for normal operations. However, they are available for the OEI emergency scenarios for which approval has been granted whether or not a NHEC or HEC is involved. For determination of the maximum rotorcraft gross weight approved for Class D operations (i.e., HEC operations performed with a multiengine rotorcraft capable of OEI HOGE, it is intended that use of the maximum OEI Power approved for the rotorcraft engine and drive system be allowed after failure of the critical engine (when applied in conjunction with an approved Class D operating procedure). Thus, it would be acceptable to base the required OEI/OGE hover performance capability for a Class D operation on a 30-second OEI power rating if the operator can demonstrate that the HEC can be safely transitioned to a flight condition where the HEC can be retrieved inside the rotorcraft for an execution of a normal OEI landing. If the specific operation for which the Class operation approval is requested does not provide for safe disposition of the HEC when using a time limited OEI rating, the Class D operation gross weight should be limited to a gross weight where OEI/OGE hover capability can be demonstrated for a continuous time period.

7. Figure AC 29 MG 12-1 is based on analogous information contained in Chapter 96 of FAA Order 8700.1. In case of conflicting information, Order 8700.1 takes precedence.

d(2) (continued)

(iv) Critical basic load determination. For all § 29.865(a) applications, obtain the gross weight range limits, obtain the corresponding limit load factors (N_{ZW}), and statically substantiate the system, in accordance with the applied for external cargo application(s) [Reference d(1)], for the critical load(s). This determines the critical basic loads and associated operating envelope for the RLC's and applicable cargo types applied for.

(v) Critical Structural Case. For § 29.865(a) applications involving more than one RLC class and/or cargo type, structural substantiation is required only for the most critical case (Reference d(1)) if accurately determinable from analysis.

(vi) Placards and markings. For all § 29.865(a) applications, appropriate placards, markings, and flight manual restrictions should be provided for items such as operating procedures, load capacities, and operational restrictions for all external load systems and devices (see also, d(13)(iii)(B)). Each placard, marking, and flight manual supplement should be checked during TIA flight testing (see also, d(20)).

(vii) Vertical Limit and Ultimate Load Factors. For all § 29.865(a) applications, the basic vertical limit load factor (N_{ZW}) from d(2) is converted to ultimate load by multiplying the maximum applied load [i.e., the sum of the carrying device load, its supporting external structure load, and the maximum cargo load] by 1.5. (For restricted category approvals, see guidance in paragraph AC 29 MG 5.) This ultimate load is used to substantiate all existing structure affected by and all added structure associated with the load carrying device, its attachments, and its cargo. Casting factors, fitting factors, and/or other dynamic load factors are to be applied where appropriate. For all HEC applications, the minimum weight of each occupant carried externally should be assumed, for analysis or test purposes, to be that of the 95 percentile 202-pound man (reference MIL-STD-1472, "Human Engineering Design Criteria for Military Systems, Equipment, and Facilities").

NOTE: If the HEC is engaged in special work tasks that would typically employ devices of significant added weight (such as heavy backpacks or fire extinguishers), the weight of these devices should be added to that of the 95 percentile 202-pound man and used in the structural analysis.

(viii) Winch/hoist system limit load. For all § 29.865(a) applications that employ winch/hoist systems to raise or lower either an HEC or NHEC from a hover, or other phase of flight, the system limit load is required to be properly determined based on the characteristics of the winch/hoist system and its installation such as mechanical advantage, static strength of the winch/hoist, static strength of its installation, allowable cable length, and the payload for any operating scenario being applied for. One acceptable method of determining the winch/hoist system limit load for any RLC and any applicable cargo type is by the following procedure:

d(2) (continued)

NOTE: In cases where either winch/hoist cables or long-line cables are utilized, a new structural system is established. Certain characteristics of this system should be examined during certification to ensure that either no hazardous failure modes exist or that they are acceptably minimized. For example, the cable or long line may (in conjunction with the rotorcraft) exhibit an unacceptable natural frequency that could be excited by sources internal to the overall structural system (i.e., the rotorcraft) or by sources external to the system. Another example is the loading effect of the cable or long line acting as a spring between the rotorcraft and the suspended external load or ground, respectively, either during flight or (when in ground contact) at the time of load release. These conditions should be reviewed and, if potentially hazardous, minimized by controlling relevant overall structural system parameters such as cable length.

(A) Determine the basic loads that fail and unspool the winch/hoist or its installation, respectively.

NOTE: This determination should be based primarily on static strength; however, any dynamic load magnification factors that are significant should be accounted for.

(B) Select the lower of the two values from (i) as the ultimate load of the winch/hoist system installation.

(C) Divide the selected ultimate load by 1.5 to determine the true structural limit load of the system.

(D) Determine the manufacturer's approved (or applicants applied for) "limit design safety factor." Divide this factor into the true structural limit load (from (c) above) to determine the winch/hoist system's working (or placarded) limit load. As a minimum, this factor should equal or exceed the value of all the factors defined under d(2)(vii) when multiplied together.

NOTE: Most winch/hoist manufacturers either use a "Limit design safety factor" of 4 to 5 on ultimate to determine their placarded limit load [i.e., allowable LL = UL/(4 to 5)]; or they use a safety factor of approximately 3 on yield to determine their placarded limit load [i.e., allowable LL = true LL/3.0]. In some cases, the load is swung through a cone of a 30 degree half apex angle. Typical structural design criteria is for the winch/hoist to remain in one piece and still function after experiencing true limit load, and to remain in one piece, but not necessarily function, after experiencing true ultimate load. These relatively large structural safety factors are used to conservatively account for phenomena such as casting factors in flight dynamic loading conditions, and wear and tear between phased inspections.

d(2) (continued)

(E) Compare the system's derived limit load to the applied for one "g" payload multiplied by the maximum downward vertical load factor (N_{ZWMAX}) from paragraph d(2) to determine the critical payload's limit value.

(F) If the critical limit payload is equal to or less than the system's derived limit load, the installation is structurally approvable as presented.

NOTE: For HEC applications, the critical limit payload should be equal to or more than the combined weight of the PCDS and its maximum number of passengers (See also d(2)(vii), for passenger weight values).

(G) If the critical limit payload exceeds the system's derived limit load, then one of the following options should be considered:

(1) Disapproval.

(2) Application for exemption.

(3) Reduction of the applied for critical limit payload to less than or equal to the system's derived limit load.

(4) Redesign of the winch/hoist system (and installation) to increase its derived limit load to equal to or greater than the critical payload.

(5) A combination of options *(3)* and *(4)*.

(6) Approvable operating restrictions to reduce N_{ZWMAX} and the corresponding critical limit payload to less than or equal to the system's derived limit load.

NOTE: Additional combinations of external load and operating restrictions may be subsequently approved under operational requirements as long as the FAR 29 structural limits of the basic certification are not exceeded, (i.e., equivalent safety is maintained).

(3) Functional Reliability and Durability Compliance Procedures for Winch/Hoist Systems under §§ 29.865(b)(3)(i) and (c)(2): It is recommended that winch/hoist systems and their installations in the rotorcraft should be designed, certified, and demonstrated as follows:

(i) General. Winch/hoist systems should be approved to acceptable aircraft industry standards. These standards and any related manufacturer's certificates of production/qualification, thereto, should be presented by the applicant as

d(3) (continued)

part of the approval package. Two typical winch/hoist approval scenarios exist. They are:

(A) For established, previously approved winch/hoist unit designs that are to be placed in a new rotorcraft installation, certification credit (to Amendment 29-43) for the unit itself can be given based on a successful unit design review (or a manufacturer's statement-of-certification accompanied by an FAA Form 8110-3 with appropriate DER approvals) that shows proper previous approval and that shows no new design changes have been made that adversely affect the reliability or function of the unit (i.e., an update of the Failure Modes and Effects Analysis (FMEA)). If so approved, then only the winch/hoist installation need be approved during certification.

(B) For new winch/hoist unit designs, the unit should be either certified to a standard aircraft industry specification that has been previously and successfully used to certify winch/hoist units, or an equivalent specification should be developed and met during the certification process.

NOTE: Background information. There are no generic industry, FAA, or military specifications currently available to apply to winch/hoist units. Thus, the detail specifications for winch/hoist unit certifications are typically generated as follows: 1) For military applications, the military dictates the basic winch/hoist unit specifications in the prime aircraft development specification. The airframe manufacturer then typically either writes or has a winch/hoist vendor write a detailed unit certification specification that includes all necessary, detailed certification criteria; 2) For commercial applications (that install the winch/hoist unit under either a new or amended type certificate or a supplemental type certificate), the airframe manufacturer typically either writes a detailed winch/hoist unit specification or has a vendor (usually the winch/hoist unit manufacturer) write the detailed certification specifications and procedures (based on the unit manufacturer's experience and the customer needs during the installation process). For either method, the FAA approves and adds the specification to the type data file during the installation approval process.

(ii) NHEC applications. The winch/hoist/rescue hook system should be reliable for the phases of flight in which it is operable, unstowed, partially unstowed and/or in which NHEC is carried. The primary electrical and/or mechanical failure modes that should be identified and minimized are unintended load release by any means, and loss of continued safe flight and landing capabilities due to a winch/hoist/rescue hook system failure. However, any other winch/hoist/rescue hook system failure that could lead to a catastrophic failure mode for the rotorcraft should also be minimized. Loss of winch/hoist operational control should also be considered. The reliability of the system should be demonstrated by completion and approval of the following:

d(3) (continued)

NOTE: It is assumed that only 1 winch/hoist cycle will typically occur per flight. This rationale has been used to determine the 10 demonstration cycles of d(3)(ii)(B) below. However, if a particular application should potentially involve more than one winch/hoist cycle per flight, then the number of demonstration cycles of d(3)(ii)(B) should be increased accordingly.

(A) A winch/hoist/rescue hook system level FMEA that identifies and minimizes any potential catastrophic failures should be conducted.

(B) Unless a more rational test method is presented and approved, a repetitive test of all functional devices in accordance with d(3)(vii) that exercises the entire system's functional parameters at least 10 times should be conducted. These repetitive tests may be conducted on the rotorcraft, or by using a bench simulation that accurately replicates the rotorcraft installation.

NOTE: If a more rational method of compliance is presented that clearly shows that an equivalent level of safety can be achieved in fewer than 10 system test cycles, the method of compliance may be acceptable.

NOTE: For properly certified winch/hoist units (Ref. d(3)(i)) that have established acceptable service histories, full certification credit for the unit itself may be given. However, each new installation is required to be approved individually, unless the installation is either identical or similar to an existing approved installation with an acceptable service history. If the new installation is only similar to an acceptable existing installation, then, for a similarity approval, all differences should be clearly stated, rationalized, analyzed, and/or tested to show they do not adversely affect the new installation (i.e., equivalent safety should be provided).

(C) A winch/hoist unit environmental qualification program that includes consideration of high and low temperatures (typically -40F to +150F), altitudes to 12,000 feet, humidity, salt spray, sand and dust, vibration, shock, rain, fungus, and acceleration should be conducted. Testing should be conducted in accordance with RTCA/DO-160 and/or MIL-STD-810 for high and low temperature tests and for vibrations. The winch/hoist manufacturers should submit a test plan and follow-on test reports to the applicant and FAA following completion of qualification. It is intended that the winch/hoist itself either be prequalified to the EMI and lightning threat levels specified for NHEC and/or HEC or that it be qualified as part of the entire onboard QRS to these threat levels.

(D) All instructions and documents necessary for continued airworthiness should be provided.

d(3) (continued)

(E) The methods of compliance in other relevant paragraphs of this AC or equivalent methods should be employed.

(iii) HEC applications. The winch/hoist/rescue hook system should be reliable for the phases of flight in which it is operable, unstowed, partially unstowed and/or in which HEC is carried. The primary electrical and/or mechanical failure modes that should be identified and minimized are unintended load release by any means and loss of continued safe flight and landing capability due to a winch/hoist/rescue hook system failure. However, any other winch/hoist/rescue hook system failure that could lead to a catastrophic failure mode for the rotorcraft should also be minimized. The winch/hoist should be disabled (or an overriding, fail-safe mechanical safety device such as either a flagged removable shear pin or a load-lowering brake should be utilized) to prevent inadvertent load unspooling or release during any extended flight phases which involve HEC and in which winch/hoist operation is not intended. Loss of winch/hoist operational control should also be considered. The reliability of the system should be demonstrated by completion and approval of the following:

NOTE: It is assumed that only one winch/hoist cycle will typically occur per flight. This rationale has been used to determine the 30 demonstration cycles of d(3)(iii)(B) below. However, if a particular application should potentially involve more than one winch/hoist cycle per flight, then the number of demonstration cycles of d(3)(iii)(B) should be increased accordingly.

(A) A winch/hoist/rescue hook system level FMEA that identifies and minimizes any potential catastrophic failures should be conducted.

(B) Unless a more rational test method is presented and approved, a repetitive test of all functional devices in accordance with d(3)(vii) that exercises the entire system's functional parameters at least 30 times should be accomplished. These repetitive tests may be conducted on the rotorcraft or by using a bench simulation test that accurately replicates the rotorcraft installation.

NOTE: If a more rational method of compliance is presented that clearly shows that an equivalent level of safety can be achieved in fewer than 30 system test cycles, the method of compliance may be acceptable.

NOTE: For properly certified winch/hoist units (Ref. d(3)(i)) that have established acceptable service histories, full certification credit for the unit itself may be given. However, each new installation is required to be approved individually, unless the installation is either identical or similar to an existing approved installation with an acceptable service history. If the new installation is only similar to an acceptable existing installation, then for a similarity approval, all differences should be clearly

d(3) (continued)

stated, rationalized, analyzed, and/or tested to show they do not adversely affect the new installation (i.e., equivalent safety should be provided).

(C) A winch/hoist system environmental qualification program that includes consideration of high and low temperatures (typically -40F to +150F), altitudes to 12,000 feet, humidity, salt spray, sand and dust, vibration, shock, rain, fungus, and acceleration should be conducted. Testing should be conducted in accordance with RTCA/DO-160 and/or MIL-STD-810 for high and low temperature tests and for vibrations. The winch/hoist manufacturers should submit a test plan and follow-on test reports to the applicant and FAA following completion of qualification. It is intended that the winch/hoist itself either be prequalified to the EMI and lightning threat levels specified for NHEC and/or HEC or that it be qualified as part of the entire onboard QRS to these threat levels.

(D) All instructions and documents necessary for continued airworthiness should be provided.

(E) The methods of compliance in other relevant paragraphs of this AC or equivalent methods should be employed.

(iv) Cable attachment. Either the cable should be positively attached to the winch/hoist drum and the attachment should have ultimate load capability, or equivalent means should be provided to minimize the possibility of inadvertent, complete, cable unspooling.

NOTE: Even though the placarded winch/hoist system load rating is much less, most winch/hoist cables are rated to a minimum of 3,300 lbs. limit load. Typically, cables have a neutral twist to minimize load oscillation.

(v) Cable length and marking. A length of cable nearest the cable's attachment to the winch/hoist drum should be visually marked to indicate to the operator that the cable is near full extension. The length of cable to be marked is a function of the maximum extension speed of the system and the operator's reaction time needed to prevent cable run out. It should be determined during certification demonstration tests. In no case should the length be less than 3 1/2 drum circumferences.

(vi) Cable stops. Means should be present to automatically stop cable movement quickly when the system's extension and retraction operational limits are reached.

(vii) Winch/hoist system load-speed combination ground tests. The load versus speed combinations of the winch/hoist should be demonstrated on the ground (either using an accurate engineering mock-up or a rotorcraft) by showing repeatability

d(3) (continued)

of the no load-speed combination, the 50 percent load-speed combination, the 75 percent load-speed combination and the 100 percent (i.e., system rated limit) load-speed combination. If more than one operational speed range exists, the preceding tests should be performed at either all speeds, or at the most critical speed if it can be determined. [Reference d(3)(ii)(B) and d(3)(iii)(B)].

(A) At least 1/10 of the demonstration cycles (see definition) should include the maximum aft angular displacement of the load from the drum, applied for under § 29.865(a).

(B) A minimum of six consecutive, complete operation cycles should be conducted at the system's 100 percent (i.e., system limit rated) load-speed combination.

(C) In addition, the demonstration should cover all normal and emergency modes of intended operation and should include operation of all control devices such as limit switches, braking devices, and overload sensors in the system.

(D) All quick disconnect devices and cable cutters should be demonstrated at 0 percent, 25 percent, 50 percent, 75 percent, and 100 percent of system limit load or at the most critical percent, if it can be determined.

NOTE: Some winch/hoist designs have built-in cable tensioning devices that function at the no load-speed combination, as well as at other load-speed combinations. This device should be demonstrated to work during the no load-speed and other load-speed cable-cutting demonstrations.

(E) All electrical and mechanical systems and load release devices for any jettisonable NHEC or HEC RLC should be shown to be reliable by both analysis and by testing done in accordance with the combined criteria of d(8) and this paragraph.

(F) Any devices or methods used to increase the mechanical advantage of the winch/hoist should also be demonstrated.

(G) During a portion of each demonstration cycle, the winch/hoist should be operated from each station from which it can be controlled.

NOTE: A reasonable amount of starting and stopping during demonstration cycles is acceptable.

(viii) Winch/hoist system continued airworthiness. The design life of the winch/hoist system and any limited life components should be clearly identified, and the

d(3) (continued)

Airworthiness Limitations Section of the maintenance manual should include these requirements. For STC's, a maintenance manual supplement should be provided that includes these requirements.

NOTE: Design lives of winch/cable systems are typically between 5,000 to 8,000 cycles. One major manufacturer uses a specification requirement of 7,500 cycles. Some winch/hoist systems have usage time meters installed. Others may have cycle counters installed. Cycle counters should be considered for HEC operations and high load or other operations that may cause low-cycle fatigue failures (see also d(24)).

(ix) Winch/hoist system manual proofing. Operating manuals, flight manuals, maintenance manuals, and associated placards should be used and proofed during the demonstration.

(x) Winch/hoist system flight tests. An in-flight demonstration test of the winch/hoist system should be conducted for helicopters designed to carry NHEC or HEC. The rotorcraft should be flown to the extremes of the applicable maneuver flight envelope and to all conditions that are critical to strength, maneuverability, stability, and control, or any other factor affecting airworthiness. Unless a lesser load is determined to be more critical for either dynamic stability or other reasons; the maximum winch/hoist system rated load or, if less, the maximum load requested for approval (and the associated limit load data placards) should be used for these tests. The minimum winch/hoist system load (or zero load) should also be demonstrated in these tests. (See also d(19)(x).)

(4) Compliance Procedures for Cargo Hooks (or Equivalent Devices) and their Related Systems under §§ 29.865(a), (b), and (c): Cargo hooks or equivalent devices and their related systems, used for any external cargo type, should be approved to acceptable aircraft industry standards. These standards and any related manufacturer's certificates of production/qualification, thereto, should be presented by the applicant as part of the approval package.

(i) General. Cargo hook systems should have the same reliability goals and should be functionally demonstrated under critical loads for NHEC, HEC, or both in a manner identical to winch/hoist/rescue hook systems (reference d(3)). All engagement and release modes should be demonstrated. If the hook is used as a quick-release device, then release of critical loads should be demonstrated under conditions that simulate maximum allowable bank angles and speeds and any other critical operating conditions. Demonstration of any relatch features and any safety or warning devices should also be conducted. Demonstration of actual in-flight emergency quick-release capability may not be necessary if the quick-release capability can be acceptably simulated by other means.

d(4) (continued)

NOTE: Cargo hook manufacturers specify particular shapes, sizes, and cross sections for lifting eyes to ensure compatibility with their hook design (e.g., Breeze Eastern Service Bulletin CAB-100-41). Experience has shown that, under certain conditions, a load may inadvertently hang up because of improper geometry at the hook/eye interface that will not allow the eye to slide off an open hook as intended. See also the discussion of hook dynamic roll out (i.e., the converse-an unintentional load release) under d(8).

NOTE: Some cargo hook systems may employ two or more cargo hooks for fail safety (i.e., after a failure of any single hook the remaining system is capable of carrying limit load). These systems are approvable. However, loss of load by any single hook should be shown to not result in loss of control of the rotorcraft. In a dual hook system, if the hook itself is the quick-release device (i.e., if a single release point does not exist in the load path between the rotorcraft and the dual hooks), the pilot should have a dual PQRS that includes selectable, collocated individual quick releases that are independent for each hook used. A BQRS should also be present for each hook. For cargo hook systems with more than two hooks, either a single release point should be present in the load path between the rotorcraft and the multiple hook system or multiple PQRS/BQRS's should be present. The former arrangement would only require a single PQRS and BQRS. A single release point can be a single or multiple cable cutter or release.

NOTE: If possible (within the rotorcraft configuration's restrictions), a simple set of approved cable cutters can satisfy the requirement for either a PQRS or BQRS in a cargo hook system installation. However, in many cargo hook system installations, unless a special access panel or an equivalent means is present, a crewman typically cannot reach and cut the cable with a standard set of cable cutters.

(ii) NHEC cargo hook systems. For jettisonable NHEC applications, each cargo hook-

(A) Should have a sufficient amount of slack in the control cable to permit cargo hook movement without tripping the hook release.

(B) Should be shown to be reliable in a manner identical to winch/hoist systems under d(3)(ii).

(iii) HEC cargo hook systems. For jettisonable HEC applications, each cargo hook-

(A) Should have a sufficient amount of slack provided in the control cable to permit cargo hook movement without tripping the hook release.

d(4) (continued)

(B) Each cargo hook should be shown to be reliable in a manner identical to winch/hoist systems under d(3)(iii).

(C) Unless the cargo hook is to be the primary quick-release device, each cargo hook should be designed such that the load cannot be inadvertently released by operationally induced loads. For example, a simple cargo hook should have a one-way, spring loaded gate (i.e., "snap hook") that allows load attachment going into the gate but does not allow the gate to open (and subsequently lose the HEC) when an operationally induced load is applied in the opposite direction. For HEC applications, cargo hooks that double as quick-release devices should be carefully reviewed to ensure they are reliable. Paragraph d(8)(iii) discusses means of increasing the reliability of devices such as cargo hooks for HEC applications.

(iv) Other cargo hook system safety requirements. DOE, EPA, OSHA, and other Government Agencies may have special safety requirements for cargo hook design over and above the FAR's, such as a dual cargo hook requirement for certain HEC operations under multi-agency regulation.

(5) Compliance Procedures for Maximum Limit Load Magnitude Determination for all Jettisonable RLC Applications under §29.865(a): For all jettisonable RLC applications for any applicable cargo type seeking § 29.865(a) approval, the maximum limit external load for which certification is requested (even though it may otherwise be much less than the maximum system capacity; e.g., cargo hook capacity, etc.) should not exceed the rated capacity of the QRS release devices used in the applicant's design or, for HEC, the rated capacity of either the QRS devices, the PCDS, or its attachments--whichever is less. Relevant parts of the QRS and the entire PCDS should be analyzed and strength tested, with FAA witness, or otherwise structurally substantiated to determine their allowable limit load capacity (reference (d)(2)) if not previously FAA approved or produced to a recognized, approvable industry and/or military standard.

(6) Compliance Procedures for Basic Loads Analysis under § 29.865(a): For all jettisonable RLC applications of any applicable cargo type seeking compliance with § 29.865(a), the maximum ultimate external load is required to be applied at sling-load-line to rotorcraft vertical axis (Z axis) angles up to 30 degrees, in any geometric direction, in substantiating analyses or tests. The 30-degree angle may be reduced in some or all directions if impossible to obtain due to physical constraints or operating limitations.

(i) Maximum cable angle. The maximum allowable cable angle (from either a winch/hoist/rescue hook, cargo hook system, or other acceptable system configuration) should be determined and approved (reference d(3)). The angle approval should be based on structural requirements, mechanical interference limits,

d(6) (continued)

and flight handling characteristics over the most critical conditions and combinations of conditions in the approved flight envelope.

NOTE: In an emergency, in some cases, the combined design of the rotorcraft and the suspended system may be such that the 30-degree angle can be exceeded, to a limited extent, without catastrophic failure. The flight manual should clearly state this maximum angle value (in the aft direction relative to the Rotorcraft Z axis; for both maximum and minimum cable lengths) that should never be exceeded in any emergency in order to minimize the hazard of a related, catastrophic failure.

(ii) 30-degree maximum angle value. In no case should the design angle for HEC exceed 30 degrees from the vertical rotorcraft axis (i.e., Z axis). If the angle is reduced, appropriate placards and flight manual changes are required (reference d(2)).

(iii) Special cases. In some special NHEC jettisonable RLC operations, such as wire stringing, the 30-degree angle can be exceeded. These cases should be approved on a case-by-case basis by an engineering certification office. An issue paper should be used to document the exact limit operational parameters determined during certification. This is necessary because of the large variability of external loads and flight maneuvers that should be considered to establish safe operating limits for these operations. As a minimum, the maximum allowable load, the maximum allowable cable angles, the maximum flight envelope, the necessary limitation placards, and the necessary RFM procedures/restrictions should be accurately determined and documented. The maximum allowable structural load envelope should be clearly identified and determined. The fatigue spectrum created by this load envelope and its frequency of use (considering in particular the possibility of low cycle fatigue failures and significantly reduced component life limits) should be clearly identified, documented, and approved.

NOTE: There are two typical configurations that have been previously approved for attaching jettisonable NHEC loads in operations such as wire stringing. They are:

Weighted-Line Sidepull Configuration. In this method, a heavy dead weight is suspended below the cargo hook. The sidepull line (jettisonable load) is then attached to the dead weight or just above the weight. The rotorcraft then proceeds in forward or sideward flight and the weight pulls the sidepull-line (jettisonable load). This method is very inefficient for payload utilization since much of the rotorcraft load capacity is used to move the dead weight rather than pulling the sidepull line. Load limiting devices such as approved fuseable/frangible links should be considered for these applications to ensure limit load is not exceeded in service.

Sidepull-Fixture Sidepull Configuration. In this method, a QRS device is attached at the side of the rotorcraft (or in another equivalently functional location) and is

d(6) (continued)

arranged so the sidepull-line's (jettisonable load's) load path is through (or nearly through) the rotorcraft center of mass at a typical working fuel condition. This configuration is more payload efficient and has much better controllability characteristics than the deadweighted sidepull-line configuration. At least one STC has been issued for a sidepull-fixture configuration for use in operations.

(7) Compliance Procedures for General QRS Certification and Installation under § 29.865(b) and § 29.865(c): For jettisonable RLC's for any applicable cargo type, a PQRS is mandated that features an approved primary quick-release device to be installed on one of the pilot's primary controls, or in an equivalently accessible location. The use of an "equivalent accessible location" is intended to be applied/reviewed on a case-by-case basis and to be used only where equivalent safety is clearly maintained. For example, if a helicopter is certified for the operating weight and can provide hover capability with one-engine-inoperative at that operating weight and altitude, then the need for an immediate jettison capability is reduced. A BQRS with a backup quick-release device is also required. The PQRS, the BQRS, and their load release devices and subsystems (such as electronically actuated guillotines) should be separate (i.e., physically, systematically, and functionally redundant). Also, for the BQRS, the backup release control and release need not be mechanical. It is intended that less sophisticated BQRS's and load release devices (such as manual cable cutters) would, if separate, be acceptable. However, if separate devices of this type are to be used, they should be listed in the flight manual as a required device and have a dedicated, placarded storage location. Each quick-release device should be designed and located to allow the pilot or a crewmember to accomplish external cargo release without hazardously limiting the ability to control the rotorcraft during emergency situations. The flight manual should reflect the requirement for a crewmember and the related functions. For jettisonable HEC operations, further QRS requirements are contained in § 29.865(c). (See paragraphs d(8), d(9) and d(12) of this AC.) In all cases, a PQRS should release the external load in less than 5 seconds. No BQRS should require more than 30 seconds from the time an emergency is declared and the BQRS quick-release device is located and activated until the NHEC or HEC load is released.

(8) Compliance Procedures for Reliability Determination for Jettisonable NHEC and HEC QRS's and Devices under § 29.865(b)(3): Jettisonable NHEC QRS's and devices and jettisonable HEC QRS's and devices are required to be reliable. One acceptable method of achieving the intended reliability goals is described as follows:

NOTE: For both NHEC and HEC designs, the phenomena of hook dynamic roll out should be considered, to the maximum practicable extent, to ensure that QRS reliability goals are not compromised. This is of utmost concern for HEC applications. Hook dynamic roll-out occurs during certain ground handling and flight conditions that may allow the lifting eye to work its way out of the hook (Reference figure AC 29 MG 12-2)

d(8) (continued)

Some commercial hook shape and keeper designs are quite prone to hook dynamic roll-out. Military Standard hook designs have not been as prone to hook dynamic roll-out as have some commercial designs. Hook dynamic roll-out typically occurs when either the RLC's sling or harness is not properly attached to the hook, is blown by down draft, is dragged along the ground, is dragged through the water; or is otherwise placed into the dangerous hook/eye configuration, shown by figure AC 29 MG 12-2. This can occur during ground handling or can be caused by relative motion of the hook and eye in flight. The potential for hook dynamic roll-out can be minimized in design by specifying particular hook-and-eye shape and hook-and-eye cross-section combinations. For non-jettisonable RLC's, a push-pull pin (or an equivalent device) can be used to lock the hook keeper in place during operations. The hook dynamic roll-out service history of any off-the-shelf components to be utilized should also be reviewed to minimize the use of potential "bad-actors."

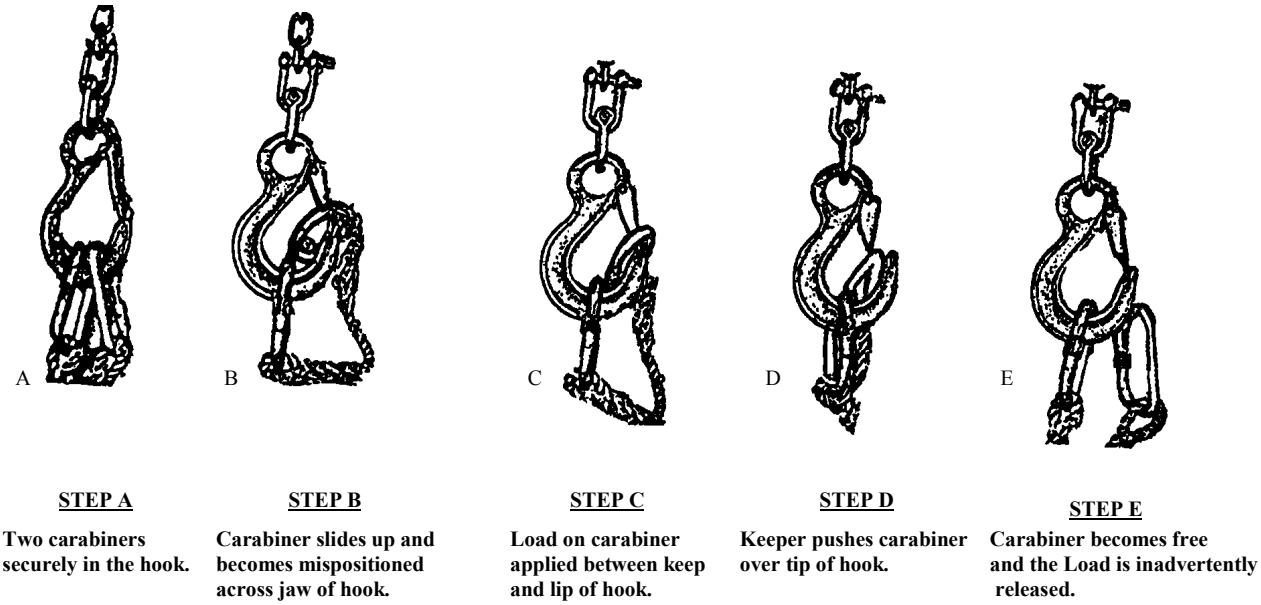


FIGURE AC 29 MG 12-2: SEQUENTIAL SIMULATION OF HOOK DYNAMIC ROLL-OUT

(i) Jettisonable NHEC designs. The QRS and the load suspension and retention designs should be reliable. The primary electrical and/or mechanical failure modes that should be identified and minimized are load release by any means and loss of continued safe flight and landing capability due to a QRS failure. However, any other failure that could lead to a catastrophic failure mode for the rotorcraft and its occupants should also be identified and minimized. The reliability of the system should be demonstrated by completion and approval of the following:

(A) A QRS level FMEA that identifies and minimizes any potential catastrophic failures.

d(8) (continued)

(B) A repetitive test of all functioning devices that affect or comprise the QRS and that tests all critical conditions or combinations of critical conditions at least 10 times each, using both the primary and backup QRS subsystems.

(C) An environmental qualification program such as that described in d(3)(ii)(C).

(D) Use of the methods of compliance in other relevant paragraphs of the AC or equivalent methods.

(ii) Jettisonable HEC designs. The QRS and the load suspension and retention designs should be reliable. The primary electrical and/or mechanical failure modes that should be identified and minimized are unintended load release by any means and loss of continued safe flight and landing capability due to a QRS failure. However, any other failure that could lead to a catastrophic failure for the rotorcraft and its occupants (either internal, external, or both) should also be identified and minimized. The reliability of the system should be demonstrated by completion and approval of the following:

(A) A QRS level FMEA that identifies and minimizes all failure modes, including any potential catastrophic failures.

(B) A repetitive test of all functioning devices that affects or comprises the QRS and that tests all critical conditions or combinations of critical conditions at least 30 times each, using both the primary and backup subsystems.

(C) An environmental qualification program such as that described in d(3)(ii)(C).

(D) Use of the methods of compliance in other relevant paragraphs of the AC or equivalent methods.

(iii) Special Cases. In some cases, an acceptable reliability for jettisonable HEC operations can be shown by temporarily deactivating a particular QRS, PQRS, and/or BQRS subsystem used for NHEC that is not otherwise reliable enough for use with jettisonable HEC. For example, this could be accomplished by adding an approved reliable QRS device for HEC such as alternate, ultimate load path across a relatively low reliability, jettisonable NHEC quick-release device or by adding another reliable fail-safe device (e.g., adding an approved, reliable safety strap as a parallel ultimate load path). The same reliability goal for HEC use could also be achieved by adding another, reliable fail-safe device such as a safing pin to an electronically actuated guillotine cutter to upgrade the system reliability to be acceptable for HEC carriage. For some designs, cargo hooks can be made more

d(8) (continued)

reliable by wiring them shut with an approved gage of safety wire. All other regulatory requirements for HEC carriage must still be met after an approved modification of the QRS to achieve the reliability necessary for HEC carriage. In the preceding examples, a replacement PQRS such as an additional set of cable cutters would need to be added to provide a complete QRS (i.e., both the PQRS and the BQRS must be present). In all cases, an HEC reliability demonstration in accordance with d(8)(ii) should be conducted and approved. Operational acceptability of these special case configurations also needs to be demonstrated.

(iv) Other load release types. In some current configurations, such as those used for high line operations, a load release may be present that is not on the rotorcraft but is on the PCDS itself. Examples are a tension release device that lets out line under an operationally induced load or a personal rope cutter. These devices are acceptable if:

(A) The off-rotorcraft release is considered a "third release"; i.e., an approved QRS (i.e., PQRS and BQRS) is present on the rotorcraft;

(B) The release meets other relevant requirements of § 29.865 and the methods of this AC or equivalent methods; and

(C) The release has no operational or failure modes that would affect continued safe flight and landing under any operations, critical failure modes, conditions, or combination of either.

(9) Compliance Procedures for Electromagnetic Interference under § 29.865(b)(3)(ii): Protection of any critical portions of the QRS against potential internal and external sources of electromagnetic interference (EMI) and lightning is required. This is necessary to prevent inadvertent load release from sources such as lightning strikes, stray electromagnetic signals, and static electricity.

NOTE 1: For "on-the-shelf" QRS system components (that may be used on different rotorcraft and in different installation configurations in the same rotorcraft) a one-time bench test, if FAA approved, can be used to test the EMI capability of the component itself. However, the EMI effect of each individual installation must be taken into account on a case-by-case basis when certifying the component's installation. This is especially critical for HEC applications.

(i) Jettisonable NHEC systems - should be able to absorb a minimum of 20 volts per meter (i.e., CAT U) RF field strength per RTCA/DO-160.

(ii) Jettisonable HEC systems[1] - should be able to absorb a minimum of 200 volts per meter (i.e., CAT Y) RF field strength per RTCA/DO-160.

d(9) (continued)

NOTE 1: These RF field threat levels may have to be increased for certain special applications such as microwave tower and high voltage high line repairs. Separate criteria for special applications under multi-agency regulation (such as IEEE/OSHA standards) should also be addressed, as applicable, during certification. When necessary, an issue paper can be used to establish a practicable level of safety for specific high voltage or other special application conditions. For any devices or means added to meet multi-agency regulations, their failure modes should not have an adverse effect on flight safety. Other certification authorities may require higher RF field threat levels than those required by § 29.865 (e.g., the European Joint Aviation Authorities Interim HIRF policy).

NOTE 2: An approved, standard rotorcraft test that includes the full HIRF frequency/amplitude external and internal environments on the QRS and PCDS (or the entire rotorcraft including the QRS and PCDS) could be substituted for the jettisonable NHEC and HEC systems tests defined by d(9)(i) and d(9)(ii), respectively, as long as the RF field strengths directly on the QRS and PCDS are shown to equal or exceed those of d(9)(i) and d(9)(ii).

NOTE 3: The EMI levels specified in d(9)(i) and d(9)(ii) are total EMI levels to be applied to the QRS (and/or effected QRS component) boundary. The total EMI level applied should include the effects of both external EMI sources and internal EMI sources. All aspects of internally generated EMI should be carefully considered including peaks that could occur from time-to-time due to any combination of on-board systems being operated. For example, special attention should be given to EMI from winch operations that involve the switching of very high currents. Those currents can generate significant voltages in closely spaced wiring that, if allowed to reach some squib designs, could activate the device. Shielding, bonding and grounding of wiring associated with operation of the winch and the quick-release mechanism should be clearly and adequately evaluated in design and certification. This evaluation may require testing. One acceptable test method to demonstrate adequacy of QRS shielding, bonding and grounding, would be to actuate the winch under maximum load together with likely critical combinations of other aircraft electrical loads and demonstrate that the test squibs (that are more EMI sensitive than the squibs specified for use in the QRS) do not inadvertently operate during the test.

(10) General Compliance Procedures for HEC Applications under § 29.865(c): For HEC applications, the safety requirements for HEC carriage for all applicable RLC's are addressed. This ensures that HEC certification requirements are clearly and properly identified.

(11) General Compliance Procedures for Jettisonable HEC Operations under § 29.865(c)(1): For jettisonable HEC operations, it may be required by Operations Requirements, that the rotorcraft meet the Category A engine isolation requirements of

d(11) (continued)

Part 29 and that the rotorcraft have OEI OGE hover performance capability in its approved, jettisonable HEC weight, altitude, and temperature envelope. OEI vertical climb capability may be needed in some operational circumstances for flight safety. Such instances should be identified and the necessary OEI vertical climb capability assessed and verified during the certification process.

(12) Compliance Procedures for QRS's under § 29.865(c)(1): For jettisonable HEC operations, both the PQRS and BQRS are required to have a DAD (i.e., see definitions, they are required to have a sequential control with two distinct actions) for external cargo release. Two distinct actions are required to provide a higher level of safety to minimize inadvertent jettison of HEC. The DAD is intended for emergency use only during the phases of flight that the HEC is carried (and/or retrieved) externally. The DAD can be used for both NHEC and HEC operations. However, because it can be used for HEC, its continued airworthiness should be carefully reviewed and documented in accordance with prescribed (or mandated) instructions. The DAD (i.e., either the primary or backup release) can be operated by the pilot from a primary control or, after a command is given by the pilot, by a crewmember from a remote location. If the backup DAD is a cable cutter, it should be properly secured but readily accessible to the crewmember intended to use it.

NOTE 1: OEI power settings should not be used for certification credit for normal operations. However, they are available for the OEI emergency scenarios for which approval has been granted whether or not a NHEC or HEC is involved. For determination of the maximum rotorcraft gross weight approved for Class D operations (i.e., HEC operations performed with a multiengine rotorcraft capable of OEI HOGE), it is intended that use of the maximum OEI Power approved for the rotorcraft engine and drive system be allowed after failure of the critical engine (when applied in conjunction with an approved Class D operating procedure). Thus, it would be acceptable to base the required OEI/OGE hover performance capability for a Class D operation on a 30-second OEI power rating if the operator can demonstrate that the HEC can be safely transitioned to a flight condition where the HEC can be retrieved inside the rotorcraft for an execution of a normal OEI landing. If the specific operation for which the Class operation approval is requested does not provide for safe disposition of the HEC when using a time limited OEI rating, the Class D operation gross weight should be limited to a gross weight where OEI/OGE hover capability can be demonstrated for a continuous time period.

(13) Compliance Procedures for PCDS's under § 29.865(c)(2): For all HEC applications, an approved PCDS is required. The PCDS is either required to be previously approved or is required to be approved during certification (reference d(14) for information on current designs). In either case, its installation should be approved. PCDS designs can vary from simple single occupant donut "lifesaver" devices to relatively complex multiple occupant cages or gondolas. However, the basic occupant

d(13) (continued)

hazard design philosophy is the same. It is to provide injured (conscious or possibly unconscious) occupants or uninjured occupants the level-of-safety necessary to minimize the possibility of any further or new injuries under any flight conditions that could occur while they are carried external to the rotorcraft.

(i) Static strength. The PCDS should be substantiated for the allowable ultimate load and loading conditions as determined under paragraphs d(2) through d(5) above.

(ii) Fatigue. The PCDS is required to be substantiated for fatigue in accordance with § 29.865(f) (Reference d(21)).

(iii) Personnel safety. For each PCDS design, a documented design evaluation should be submitted by the applicant (and presented to the Certification Authority) that ensures that the necessary level of personnel safety is provided (i.e., all potential, relevant occupant hazards are acceptably minimized). As a minimum, the following should be evaluated.

NOTE: It is intended that the evaluation should be comprehensive. However, it is not necessarily intended that the PCDS be required to have all the personnel safety design features of, for example, a transport aircraft interior. Only those personnel safety design features necessary to minimize new or further injury to PCDS occupants during the relatively short time interval the PCDS is utilized on a given mission are necessary.

(A) The PCDS should be easily and readily ingressed or egressed.

(B) It should be placarded for proper capacity, internal arrangement/location of occupants, and ingress and egress instructions (See also, d(2)(vi)).

(C) For door latch fail-safety, more than one fastener or closure device is recommended. The latch device design should provide direct visual inspectability to ensure it is fastened and secured.

(D) Any fabric used should be durable and should be at least flame-resistant.

(E) Safety harnesses and belts should meet TSO C-22 and TSO-C-114 requirements.

(F) Sharp corners and edges should be avoided and padding should be used, as necessary, to protect the occupants.

d(13) (continued)

NOTE: Acceptable sources of detailed design criteria and standards for PCDS webbing and harness can be found in sources such as U.S. AAVSCOM TR 89-D-22D, "Aircraft Crash Survival Design Guide, Volume IV - Aircraft Seats, Restraints, Litters, and Cockpit/Cabin Delethalization."

(G) Occupant retention devices and related design safety features should be used as necessary. In simple designs, only a lack of sharp corners and edges with adequate strapping (or other means of HEC retention relative to the PCDS) and head supports/pads may be all the safety features that are necessary. However, in more complex PCDS designs, safety features such as seat belts, hand holds, shoulder harnesses, placards, and/or other personnel safety standards may be required.

(H) The PCDS design should use methods of compliance in other relevant paragraphs of this AC or equivalent methods.

(iv) Reliability. The reliability level goal for the PCDS and its attachments to the rotorcraft is extremely improbable (i.e., 1×10^{-9} failures per flight) for all failure modes that could cause either catastrophic failure, serious injuries, and/or fatalities anywhere in the total airborne system. All significant failure modes of lesser consequence should be rendered improbable (i.e., 1×10^{-5} failures per flight). One acceptable method of achieving this goal is to submit and achieve approval of the following:

(A) A PCDS level FMEA that minimizes any potential catastrophic failures that are not extremely improbable and minimizes any other lesser, significant failures that are not improbable.

(B) A repetitive test of all functional devices that cycles these devices under critical structural conditions, operational conditions, or a combination at least 30 times.

(C) An environmental qualification review over the proposed operating environment.

NOTE: A complete environmental qualification test as described in d(3)(iii)(C) is necessary unless the design features would clearly not necessitate employment of all or part of the test program of d(3)(iii)(C).

(v) EMI and lightning protection. All essential, affected components of the PCDS, such as intercommunication equipment, should be protected against RF field strengths to a minimum of RTCA/DO-160 CAT Y. (Reference d(9)(ii).)

d(13) (continued)

(vi) Continued airworthiness. All instructions and documents necessary for continued airworthiness, normal operations, and emergency operations should be completed, reviewed, and approved during the certification process.

(vii) Flotation devices. PCDS's that are intended to have a dual role as floatation devices or life preservers should meet the requirements of TSO-C13f, "Life Preservers." Also, any PCDS design to be used in the water should have a floatation kit. The kit should support the weight of the maximum number of occupants and the PCDS in the water and minimize the possibility of the occupants floating face down.

(viii) Aerodynamic considerations. Litters and other types of PCDS designs may (because of effects from sources such as down drafts, maneuvers, or gusts) spin, twist or otherwise respond unacceptably in flight. These designs should be structurally restrained with devices such as a spider, a harness, or an equivalent device to minimize undesirable flight dynamics.

(ix) Medical design considerations. The PCDS should be designed to the maximum practicable extent and placarded to maximize the HEC's protection from medical considerations such as blocked air passages induced by improper body configuration and excessive loss of body heat during operations. HEC (especially injured and/or water soaked persons) may be exposed to high body heat loss from sources such as rotor wash and the airstream. PCDS occupant safety from transit induced medical considerations can be greatly increased by proper design.

(x) Special PCDS configurations. Certain PCDS configurations may be submitted for approval that have special design considerations. Known configurations and their special design considerations are described, as follows:

Net type PCDS's. A well-designed net type PCDS has the advantage of being able to quickly evacuate several combinations of able and/or disabled HEC. Net type PCDS's should be designed such that enough rigid or semi-rigid components are present so that the net does not close in and entrap, injure, further injure, and/or create panic from claustrophobia to the HEC occupants during rescue. Secondly, if intended for water use, the net type PCDS should have proper flotation so it does not drag the HEC underwater. Thirdly, the net type PCDS should be easily ingressed so that the HEC will readily climb into the net and not try to hang onto the outside of the net.

(14) Summary of Current PCDS Designs that relate to § 29.865(c)(3): In relation to § 29.865(c)(3), several commercial and military PCDS's exist and are used for emergency rescue work involving HEC. Known devices are summarized in figure AC 29 MG 12-3. Some devices are not approved; however, applications that involve them may be submitted for approval.

(15) Compliance Procedures for QRS Design, Installation, and Placarding under § 29.865(c)(3): For jettisonable HEC applications, the QRS design, installation,

and associated placarding should be given special consideration to ensure the proper level of occupant safety.

(16) Compliance Procedures for Intercom Systems for HEC Operations under § 29.865(c)(4): For all HEC operations, the rotorcraft is required to be equipped for or otherwise allow direct intercommunication under any operational conditions among crewmembers and the HEC. It is intended that for simple systems, voice or hand signals to PCDS occupants (if not in conflict with operations requirements) would be acceptable. In more complex systems, it is intended that more sophisticated devices such as intercoms be provided.

(17) Compliance Procedures for Flight Manual Procedures and Limitations for HEC Operations under § 29.865(c)(5) and (c)(6): All appropriate flight manual procedures and limitations for all HEC operations are required to be present and to be approved. These instructions and manuals should be proofed during flight tests (reference d(19)).

(18) Compliance Procedures for Special Conditions Encountered in Operations: If special conditions will be encountered in operations such as low visibility and night use, then provisions such as night lighting that provide the proper level of safety for both the rotorcraft and HEC when operating under these special conditions should be identified, considered, and approved during certification. This determination should be made on a case-by-case basis during either initial or supplemental certification using the proposed operating environment scenario.

(19) Compliance Procedures for Flight Test Verification Work under § 29.865(d): Flight test verification work (or an equivalent combination of analysis and ground testing, either in conjunction with or in addition to operations rules such as Part 133 for the U.S.) that thoroughly examines the operational envelope should be conducted with the external cargo carriage device for which approval is requested (especially those that involve HEC). The flight test program should show that all aspects of the operations applied for are safe, uncomplicated, and can be conducted by a qualified flight crew under the most critical service environment and, in the case of HEC, under emergency pressure. Flight tests should be conducted for the simulated representative NHEC and HEC loads to demonstrate their in-flight handling and separation characteristics.

(i) General. Flight testing (or an equivalent combination of analysis and testing) should be conducted under the critical combinations of configurations and operating conditions for which basic type certification approval is sought. Additional combinations of external load and operating conditions may be subsequently approved under relevant operational requirements as long as the structural limits and reliability

d(19) (continued)

considerations of the basic certification approval are not exceeded (i.e., equivalent safety is maintained). The qualification flight test work of this subparagraph is intended to be accomplished primarily by analysis and/or bench testing. However, at least one in-flight, limit load drop test should be conducted for the critical load case. If one critical load case cannot be clearly identified, then more than one drop test might be necessary. Also, in-flight tests for the minimum load case (i.e., typically the cable hook itself) with the load trailing both in the minimum and maximum cable length configurations should be conducted. Any safety-of-flight limitations should be documented and placed in the rotorcraft flight manual. Also, in certain low-gross weight, jettisonable HEC configurations, the PCDS may act as a trailing airfoil (i.e., exhibit lift characteristics above certain airspeeds) that could result in entangling the PCDS and the rotorcraft. These configurations should be assessed on a case-by-case basis by analysis and/or flight test to assure any safety-of-flight limitations are clearly identified and placed in the rotorcraft flight manual.

(ii) Determination of one engine inoperative (OEI) hover performance. FAR 29.865(c)(6) and 133.45(e)(1) require the rotorcraft to be type certificated under Transport Category A for the operating weight and provide hover capability with one engine inoperative at that operating weight, altitude, and temperature. It is intended that the rotorcraft be able to withstand an engine failure during hover and continue the hover operation.

(A) In determining OEI hover performance, dynamic engine failures should be considered. Each hover verification test should begin from a stabilized hover at the maximum OEI hover weight, at the requested in-ground effect (IGE) or out-of-ground-effect (OGE) skid/wheel height, and with all engines operating. At this point, the critical engine should be failed and the aircraft should demonstrate the capability to maintain a stabilized hover condition without exceeding any rotor limits or engine limits for the operating engine(s). As with all performance testing, engine power should be limited to minimum specification power. Engine failures may be simulated by rapidly moving the throttle to idle provided a needle split is obtained between the rotor and the engine RPM.

(B) Normal pilot reaction should be used following the engine failure to maintain the stabilized hover flight condition. When hovering OGE or IGE at maximum OEI hover weight, an engine failure should not result in an altitude loss of more than 10 percent or 4 feet, whichever is greater, of the altitude established at the time of engine failure. In either case, sufficient power margin should be available from the operating engine(s) to regain the altitude lost during the dynamic engine failure and to transition to forward flight.

(C) The time required to recover an external load (especially HEC loads) and to transition to forward flight should also be considered. This time increment may limit the use of short duration, OEI power ratings. For example, for a helicopter that

d(19) (continued)

sustains an engine failure at a height of 40 feet, the time required to restabilize in a hover, recover the external load (given hoist speed limitations), and then transition to forward flight (with minimal altitude loss) would likely preclude the use of a 30-second engine ratings and may encroach upon 2 1/2-minute ratings.

(D) In addition, for those helicopters that incorporate engine driven generators, the hoist should remain operational following an engine/generator failure. A hoist should not be powered from a bus that is automatically shed following the loss of an engine/generator. Maximum two engine generator loads should be established such that when one engine/generator fails, the remaining generator can assume the entire rotorcraft electrical load (including maximum hoist electrical load) without exceeding approved limitations.

(E) The Rotorcraft Flight Manual (RFM) should contain information that describes the expected altitude loss, any special recovery techniques, and the time increment needed for recovery of the external load when establishing maximum weights and skid heights. The OEI hover chart may be placed in the performance section of the RFM or RFM supplement. Allowable altitude extrapolation for the hover data should not exceed 2,000 feet.

(iii) Separation characteristics of jettisonable external loads. For any RLC for any applicable cargo type, satisfactory post-jettison separation characteristics of all loads should meet the minimum criteria that follow:

(A) Immediate "clean" operation of the QRS, including "clean" separate functioning of the PQRS and BQRS.

(B) No damage to the helicopter during or following actuation of the QRS and load jettisoning.

(C) A jettison trajectory clear of the helicopter.

(D) No inherent instability of the jettisonable (or just jettisoned) HEC and/or NHEC while in proximity to the helicopter.

(E) No adverse or uncontrollable helicopter reactions at the time of jettison.

(F) Stability and control characteristics after jettison should be within the originally certified limits.

FIGURE AC 29 MG 12-3

DEVICE	FAA APPROVED[2]	SOURCE
Stokes litter (one Person)	No	U.S. Coast Guard
Rescue Basket	No	U.S. Coast Guard
Rescue Sling (one Person)[1]	Yes	U.S. Coast Guard
Rescue Net (STC7586SW)[2]	Yes	Billy Pugh Co., Inc. P.O. Box 802 1415 N. Water Street Corpus Christi, TX 78403
LII (STC7731SW)[2]	Yes	Life Industries International, Inc. 4170 Rogers Avenue Suite D, Box 3284 Fort Smith, AR

NOTES:

1. The "rescue sling" or "rescue strop" is a "horse collar" device that requires a person to exert some effort to remain in the collar. Some versions of the rescue sling have retainer straps to help secure an occupant in the horse collar. These straps are typically located in pockets on each side of the collar and are usually marked "pull." The straps go around the occupant's back and clip together with a "V" ring and a quick ejector fitting. This device should only be used on a fully conscious individual, unless the individual is fully retained by devices such as retention straps. Even an alert, well-trained individual may have nerves impinged on by pressure from this device. Nerve impingement may result in loss of sensation in the arms, loss of grip, and inadvertent fall from the harness. The retainer strap version of the rescue sling should only be used in conjunction with properly written instructions and placards and with trained personnel.

2. FAA approval is for a specific installation only; each new installation is required to still be approved.

3. Other types of emergency rescue devices that are not listed but have been successfully used by the military are the Screamer Suit and the Jungle Penetrator. The screamer suit or harness (full body fishnet) is a PCDS constructed of mesh and webbing. It was originally designed to physically encompass the torso of HEC rescue subjects who are disabled or unconscious to prevent them from inadvertently falling out of the PCDS. It is a relatively simple device for a rescuer to use. The Jungle Penetrator is a heavy device (typically metal) with a tapered end. It will break light timber and brush when dropped in free-fall from the rotorcraft to an evacuee. It typically has arms that swing down on which HEC can ride and a webbing loop to hold the HEC onto the device.

d(19) (continued)

(G) No unacceptable degradation of the helicopter performance characteristics after jettison.

(iv) Jettison requirements for jettisonable external loads. For representative cargo types (low, medium, and high density loads on long and short lines), emergency and normal jettison procedures should be demonstrated (by a combination of analysis, ground tests, and flight tests) at sufficient combinations of flight conditions to establish a jettison envelope which should be placed in the flight manual.

(v) QRS demonstration. Repetitive jettison demonstrations should be conducted that use the PQRS. Except, the BQRS should be utilized at least once.

(vi) QRS reliability (i.e., failure modes) affecting flight performance. The FMEA of the QRS (reference d(7) and d(8)) should show that any single system failure will not result in unsatisfactory flight characteristics. For any QRS failures resulting in asymmetric loading conditions, the helicopter should be shown to be safely flyable. Performance characteristics should not be adversely affected by any QRS failure mode.

(vii) Flight test weight and CG locations. All flight tests should be conducted at the extreme or critical combinations of weight and longitudinal and lateral CG conditions within the applied for flight envelope. The rotorcraft should remain within approved weight and CG limits both with the external load applied and after jettison of the load.

(viii) Flight Speed Envelopes. Emergency and normal jettison demonstrations should be performed at sufficient airspeeds to establish any airspeed restrictions for satisfactory separation characteristics. The maximum and minimum airspeed limits for safe separation should be determined. The sideslip envelope as a function of airspeed should be determined.

(ix) Altitude. Emergency and normal jettison demonstrations should be performed at altitudes consistent with the approvable operational envelope and with the maneuvering requirements necessary to overcome any adverse effects of the jettison.

(x) Attitude. Emergency and normal jettison demonstrations should be performed from all attitudes appropriate to normal and emergency operational usage. Where the attitudes of HEC and/or NHEC with respect to the helicopter may be varied, the most critical attitude should be demonstrated. This demonstration would normally be accomplished by bench testing.

d(19) (continued)

(xi) Winch/hoist/rescue hook systems and/or cargo hook systems. These articles should be flight demonstrated per d(3)(x).

(20) Compliance Procedures for External Loads Placards and Markings under § 29.865(e): Placards and markings should be installed next to the external load attaching means, in a clearly noticeable location, that state the primary operational limitations - specifically including the maximum authorized external load. Not all operational limitations need be stated on the placard (or equivalent markings) only those clearly necessary for immediate reference in operations. Other more detailed and/or operational limitations of lesser immediate reference need should be stated either directly in the RFM or in a supplement thereto (See also, d(2)(vi)).

(21) Compliance Procedures for Fatigue Substantiation under § 29.865(f): The fatigue evaluation of § 29.571 should be applied as follows:

NOTE: The term "hazard to the rotorcraft" is defined to include all hazards to either the rotorcraft, to the occupants thereof, or both.

(i) Fatigue evaluation of NHEC applications. Any critical components of the suspended system and their attachments (such as the cargo hook or bolted or pinned truss attachments), the failure of which could result in a hazard to the rotorcraft, should include an acceptable fatigue analysis in accordance with AC 27 MG 11, paragraph e.

(ii) Fatigue evaluation of HEC applications. The entire PCDS and its attachments should be reviewed on a component-by-component basis to determine which, if any, components are fatigue critical or damage intolerant. These components should be analyzed and/or tested (per AC 27 MG 11, AC 29 MG 11, or other equivalent methods) to ensure their fatigue life limits are properly determined and placed in the limited life section of the maintenance manual.

(22) Compliance Procedures for Agricultural Installations (AI's): AI's can be certified for either jettisonable or non-jettisonable NHEC or HEC operations as long as they meet relevant certification and operations requirements and follow appropriate compliance methods. However, most current AI designs are external fixtures (see definition) - not external loads. External fixtures are not certifiable as jettisonable external cargo because they do not have a true payload (see definition), true jettison capability (see definition), or a complete QRS. Many AI designs can dump their solid or liquid chemical loads by use of a "purge port" release over a relatively long time period (i.e., greater than 30 seconds). This is not considered true jettison capability (see definition) since the external load is not released by a QRS and since the release time span is typically greater than 30 seconds (reference c(20) and d(7)). Thus, these types of AI's should be certified as a

d(22) (continued)

non-jettisonable external load. However, other designs that have the entire AI (or significant portions thereof) attached to the rotorcraft, that have short time frame jettison (or release) capability provided by a QRS that meets the definitions herein and that have no post-jettison characteristics that would endanger continued safe flight and landing may be certified as a jettisonable external load. For example, if all the relevant criteria are properly met, a jettisonable fluid load can be certified as a NHEC external cargo. AC 29 MG 5 discusses other AI certification methodology.

(23) Compliance Procedures for External Tank Configurations: External tank configurations that have true payload (see definition) and true jettison capability (see definition) should be certified as jettisonable NHEC. External tank configurations that have a true payload capability but do not have true jettison capability should be certified as non-jettisonable NHEC. An external tank that has neither a true payload capability nor true jettison capability is an external fixture; it should not be certified under § 29.865 (i.e., as an external load). If an external tank is to be jettisoned in flight, it should have a QRS that is approved for the maximum jettisonable external tank payload and is either inoperable or is otherwise rendered reliable to minimize inadvertent jettisons above the maximum jettisonable external tank payload.

(24) Compliance Procedures for Logging Operations: These operations are very susceptible to low-cycle fatigue because of the large loads and relatively high load cycles that are common to this industry. It is recommended that load measuring devices (such as load cells) be used to ensure that no unrecorded overloads occur and to ensure that cycles producing high fatigue damage are properly considered. Cycle counters are recommended to ensure acceptable cumulative fatigue damage levels are identifiable and are not exceeded. As either a supplementary method or alternate method, maintenance instructions should be considered to ensure proper cycle counting and load recording during operations.

(25) Compliance Procedures for Noise Certification: FAR 36 is the noise certification standard. Section 36.1(a)(4) specifically exempts helicopters that are designed exclusively for agricultural work, carrying firefighting materials, or external loads activity from the noise standards. FAR 21.93(b)(4) also contains specific information regarding external loads and what configurations constitute/do not constitute an acoustical change.

(26) Compliance Procedures for Inspection and Maintenance Considerations. Maintenance manuals (and supplements thereto) developed by applicants for external load applications should be presented for approval and should include all appropriate inspection and maintenance procedures. The applicant should provide sufficient data and other information to establish the frequency, extent, and methods of inspection of critical structure, systems and components thereof. This information must be included

d(26) (continued)

in the maintenance manual as required by § 29.1529. For example, maintenance requirements for sensitive QRS squibs should be carefully determined, documented, approved during certification, and included as specific mandatory scheduled maintenance requirements that may require either "daily" or "pre-flight" checks (especially for HEC applications).

FIGURE AC 29. MG 12-4: SUMMARY OF RELEVANT INFORMATION FOR U.S. PART 133 ROTORCRAFT LOAD COMBINATIONS CERTIFIABLE UNDER § 29.865

Basic Definition and Intended Use Class A	Typical Load Limits	Quick Release Requirements	Certification Requirements and Considerations
Fixed External Cargo Container - Is defined by § 1.1 as a load combination in which the external load cannot move freely, cannot be jettisoned, and does not extend below the landing gear. This category usually features multiple attachments (loadpaths) to the airframe. A typical example is a hard mounted cargo basket attached to the rotorcraft crosstubes which is used to carry external cargo from point A to point B. A non-typical example is a removable advertising sign that is in a folded configuration during take-off and landing, but is extended during flight. Maximum rotorcraft gross weight with external load may not exceed the maximum internal load gross weight approved under § 29.25(a).	Certification limit load is N_{ZW} X Maximum Substantiable External Load. N_{ZW} is 2.5 per § 29.865 (See Procedure, paragraph d(2)(ii)) for NHEC cargo. For HEC, $2.5 \leq N_{ZW} \leq 3.5$ depending on gross weight (see Procedure paragraph d(2)(iii)).	None. Cargo and its container are not jettisonable.	• For HEC and NHEC external cargo. (See figure AC 29 MG 12-1) • Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions and emergency procedures from § 29.865 should be a part of this supplement. • The rotorcraft does not need Category A and OEI hover capability to carry HEC. • Load limit placards are required by § 29.865(c). • Flight envelope restriction placards may also be required for gross weight limitations, elimination of dangerous maneuvers, HEC requirements, etc. • Cargo tiedowns to prevent load shifting relative to airframe and for inflight load retention may be required. • Effect of external cargo carrier and its maximum cargo weight on load paths, loads and fatigue of existing structure should be determined. • Type Inspection Authorization (TIA) testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate. • The applicant should test the aerodynamic effect of several representative load shapes and include applicable information in the flight manual supplement. If such information is not in the RFM, then the operator may be required to obtain an operations approval under Part 133. • PCDS (i.e., the entire attached HEC carrying device) should be reviewed for relevant occupant safety criteria and placarding. • If all relevant criteria are met, non-jettisonable external tank loads (i.e., fluid or other loads) can be certified as a Class A RLC [Reference d(22) and d(23)]. • To be certified under § 29.865 as a Class A RLC, the external load and its carrying device should have true payload capability (see definition) (i.e., it should be an external load, not an external fixture).

FIGURE AC 29 MG 12-4: SUMMARY OF RELEVANT INFORMATION FOR U.S. PART 133 ROTORCRAFT LOAD COMBINATIONS CERTIFIABLE UNDER § 29.865 (continued)

Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements and Considerations
Class B			
Single or Multiple Point Suspension External Load Airborne Is defined by § 1.1 as a load combination in which the external load is jettisonable and is lifted free of land or water during the rotorcraft operation. The payload is typically suspended from a hook or a similar device. The hook may be attached to the rotorcraft structure, or it may be attached to a movable hoist cable with the hoist itself attached to the rotorcraft. A typical use is to lift a cargo load until it is completely airborne and fly it from point A to point B. The external hoist load may be stowed in the fuselage (in some cases) while being transported. The rotorcraft maximum gross weight with external load attached may exceed the maximum internal gross weight approved under § 29.25(a) as long as all weight above the maximum internal weight is jettisonable.	Certification limit load is N_{ZW} X Maximum Substantiatable External load. N_{ZW} is 2.5 per § 29.865 (See Procedure, paragraph d(2)(ii) for NHEC). Load may be limited by winch/hoist allowables. For HEC, $2.5 \leq N_{ZW} \leq 3.5$ depending on gross weight (see Procedure paragraph d(2)(iii)).	Yes - § 29.865(b)(1) requires that a primary quick release subsystem control device be installed on a primary control or in an equivalently accessible location. Also, a backup quick release system actuation device should be available and readily accessible.	• For HEC or NHEC external cargo (See figure AC 29 MG 12-1). • Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions and emergency procedures from § 29.865 should be a part of this supplement. • The rotorcraft does not need Category A and OEI hover capability to carry HEC. • Load limit placards are required by § 29.865(c). • Flight envelope restriction placards may also be required for HEC. • Certifiable external cargo load capacity may be further limited by §§ 133.41 and 133.43. • Quick release subsystems and devices should be approved and be operable on a nonhazard basis by the pilot per § 29.865(b). • Quick release backup subsystems should be reliable but need not be overly sophisticated (cable cutters, axes, etc., used by crewmembers). • Effect of maximum suspended load and its attachment to rotorcraft structure on load paths, loads and fatigue of existing structure should be determined. • TIA testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate. • PCDS (i.e., the entire attached human external cargo carrying device) should be reviewed for relevant occupant safety criteria and placarding. • If all relevant criteria are met, jettisonable loads (i.e., fluid or other loads) can be certified as a Class B RLC [reference d(22) and d(23)].

FIGURE AC 29 MG 12-4: SUMMARY OF RELEVANT INFORMATION FOR U.S. PART 133 ROTORCRAFT LOAD COMBINATIONS CERTIFIABLE UNDER § 29.865 (continued)

Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements and Considerations
Class C			
Single or Multiple Point Suspension External Load Partially Airborne - Is defined by § 1.1 as an RLC in which the external load is jettisonable and remains in contact with land or water during the rotorcraft operation. The payload is typically partially suspended by a net or cables from a cargo hook or a similar device. The cargo hook may be attached to the rotorcraft structure or may be attached to a movable hoist cable and the hoist itself attached to the rotorcraft. A typical use is for stringing wire or laying cable where the payload is only partially suspended from the ground. (Note: Many applications combine both Category B and C operations because of the obvious utility involved.) The rotorcraft maximum gross weight with external load attached may exceed the maximum internal gross weight approved under § 29.25(a) as long as all weight above the maximum internal weight is jettisonable.	Certification limit load is N_{ZW} X Maximum Substantiable External load. N_{ZW} is 2.5 per § 29.865 (See Procedure, paragraph d(2)(ii) for NHEC). Load may be limited by hoist allowables. For HEC, $2.5 \leq N_{ZW} \leq 3.5$ depending on gross weight (see Procedure paragraph d(2)(iii)).	Yes - § 29.865(b)(1) requires that a primary quick release subsystem control device be installed on a primary control or in an equivalently accessible location. Also, a backup quick release subsystem control device should be available and readily accessible.	• For HEC or NHEC external cargo (See figure AC 29 MG 12-1). • Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions and emergency procedures from § 29.865 should be a part of this supplement. • The rotorcraft does not need Category A and OEI hover capability to carry HEC. • Load limit placards are required by § 29.865(c). • Flight envelope restriction placards may also be required for HEC. • Certifiable external cargo load capacity may be further limited by §§ 133.41 and 133.43. • Quick release subsystems and devices should be approved and be operable on a nonhazard basis by the pilot per § 29.865(b). • Quick release backup subsystems should be reliable, but need not be overly sophisticated (cable cutters, axes, etc., used by a crewmember). • Effect of the maximum suspended/attached load and its attachment to rotorcraft structure on load paths, loads and fatigue of existing structure should be determined. • TIA testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate. • PCDS (i.e., the entire attached HEC carrying device) should be reviewed for relevant occupant safety criteria and placarding.

FIGURE AC 29 MG 12-4: SUMMARY OF RELEVANT INFORMATION FOR U.S. PART 133 ROTORCRAFT LOAD COMBINATIONS CERTIFIABLE UNDER § 29.865 (continued)

Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements and Considerations
Class D Single or Multiple Point Suspension External Airborne Load. Is defined by § 1.1 as an RLC in which one or more persons who are passengers OTHER than crewmembers and/or persons who are essential to the external load operation are carried as an external load for compensation. Such passengers carried external to the rotorcraft in approved devices that meet the configuration definition of any other rotorcraft-load combination are defined as a Class D rotorcraft-load combination. This RLC is for HEC transport. The payload which typically consists of personnel and their PCDS can be configured in any safe manner. PCDS's may transport one or more persons. Typical PCDS's devices are vest and straps, baskets, life preservers with straps and attachment devices, cages, or a suspended container. (See Procedures d(13) and d(14). The maximum gross weight with external load attached should not exceed the OEI OGE Hover Performance capability for the operational ambient conditions (altitude and temperature).	For HEC, N_{ZW} varies from 2.5 at max gross weight to 3.5 at minimum gross weight. (See Procedures d(2)(iii)). Load is usually limited by hoist allowable, attachment allowable or by PCDS allowable.	A PQRS control DAD (requiring two distinct actions) should be installed on a primary control or be in an equivalently accessible location such as near a designated primary crewmember's station. Also, a BQRS DAD should be available and readily accessible.	• Used only for HEC other than Class A, B, or C. Only an HEC load that consists of a person other than a crewmember or a person who is essential and directly connected with the external load operation may be carried as an approved Class D RLC. These persons are being carried (i.e., transported) externally (See figure AC 29 MG 12-1). • This RLC combination cannot be used for NHEC (See figure AC 29 MG 12-1). • Rotorcraft should meet the Category A engine isolation requirements of Part 29 and should be certified for an OEI/OGE hover performance weight, altitude and temperature envelope that becomes the maximum envelope that can be used for Class D HEC operations. This is required for a Class D rating by § 133.45(e)(1). • PCDS's should be approved separately or as part of the certification project. • PCDS's should carry personnel internally or secure them safely in a harness or equivalent device. • Flight Manual Restrictions - § 133.47 requires a rotorcraft load combination flight manual supplement. Any flight envelope restrictions and emergency procedures from § 29.865 should be a part of this supplement. • Load limit placards are required by §29.865(c). • Flight envelope restriction placards may also be required. • Certifiable external load capacity is further limited by §§ 133.41, 133.43 and 133.45(e)(3), the load limit of the PCDS and its attachment to the rotorcraft. • QRS subsystem release devices should be approved and be operable on a nonhazard basis by the pilot or a designated primary crewmember per §§ 133.44(c)(6) and 29.865(b). • The PQRS should have an emergency release (DAD) requiring two distinct actions. • The BQRS subsystem should be

Basic Definition and Intended Use	Typical Load Limits	Quick Release Requirements	Certification Requirements and Considerations
			accessible and reliable. • Rotorcraft should be equipped to allow direct intercom among all crewmembers (per § 133.45(e)(2)). • Effect of maximum external load and its attachment to rotorcraft structure on load paths, loads and fatigue of existing structure should be determined. • TIA testing may be necessary to determine whether or not the system performs as intended and if placards and flight manual supplements are adequate.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 13. SYSTEMS CERTIFICATION CONSIDERATIONS.

a. Supporting Systems.

(1) Purpose. The purpose of this AC paragraph is to provide guidance on how to show compliance to Part 29 regulations as they apply to supporting systems for other systems that provide required functions. The applicability of this material to "systems" is defined by AC 29.1309.a(1). The systems that require support from supporting systems are defined as dependent systems in the following guidance. Application of recent technology is one of the predominant causes of more dependent/supporting systems relationships. More systems are employing technology that is dependent on supporting systems such as electrical, hydraulic, and/or other power sources or signal inputs. Certification of systems that are dependent on supporting systems to provide required functionality must consider the issues associated with this interdependent relationship.

(2) Definitions.

(i) Integrity. The term "integrity" for the purpose of this AC paragraph includes the hardware quality requirements, including reliability (availability); as well as the software level requirements, as defined in RTCA/DO-178B.

(ii) Criticality. The term "criticality" refers to the five levels of criticality addressed in this document in paragraph AC 29.1309f(2)(i).

(iii) Supporting System(s). The term "supporting system(s)" as used in this paragraph means any system(s) that provides an input to another "dependent" system, such that these dependent system(s) cannot function correctly without that input being present/correct.

(iv) Dependent System(s). The term "dependent system(s)" as used in this paragraph means any system/s that receives an input from another system or sensor.

(3) Related Documents.

(i) Federal Aviation Regulations (FARs) Paragraphs 29.1301, 29.1309, 29.1351, and 29.1435.

(ii) Standards - Latest revision of RTCA/DO178 and RTCA/DO 160; Parts of SAE documents ARP4754 and ARP4761.

(4) History: Applications of recent technology for systems have, in many cases, resulted in systems that are dependent on one or more supporting system(s) for inputs such as power or signal sources of any type. This relationship creates concerns for recognition that the criticality of the supporting system may be higher because of its role of supporting a dependent system of high criticality. The example of Liquid Crystal Displays (LCD) for engine instruments, in particular, has caused concern about the integrity of the supporting electrical power system. Past designs for engine instruments did not require electrical power for operation, but present designs with LCDs do require electrical power. Additionally, engine instruments of those older helicopter designs were driven from a mixture of sources such as independent wet line, pneumatic, and electrical drive (Tach Generator) inputs, and thus had independent failure modes from the sensor/power input aspect(s). This means that the integrity of the electrical power supplied to the dependent system must be commensurate withthe level of integrity as required for the highest criticality engine instrument application(s). This also stimulates the concern that the electrical power system can become a common point for failure of all engine instruments simultaneously, as well as anything else powered by the electrical system. For this example, these considerations represent an increase of integrity requirements for the electrical power system over previous designs of electrical systems for VFR helicopters. In the past, these electrical systems did not support required functions of higher criticality and were allowed to be simple in design with low design integrity and susceptible to single point faults. Application of recent technologies for systems resulting in dependent systems that require supporting systems must address the concerns for higher integrity and single point faults.

(5) Discussion: Integrity of supporting systems must be sufficient to support the required integrity of their associated dependent systems. The relationship of supporting systems to dependent systems is similar to an analogy of them being links in a chain, where the weakest link must be able to support the required integrity level that is consistent with the associated criticality category assessment. This principle is not new, but there may not be recognition that systems previously accepted at low integrity levels may not be acceptable because of their new role as supporting system(s). New emphasis must be applied to determine acceptability of previous designs that have become supporting systems through application of new technology or changes of system architecture. This is particularly true for derivative designs, or changes to existing design by either supplemental type certificate (STC) or field approval, where new technology system applications, or system architecture changes have been applied, that created a dependent/supporting system relationship. The main concerns are for systems such as electronic displays that are installed and supported by non-upgraded systems, such as single source and/or low reliability power generation and distribution systems. However the concerns related to supporting systems are not limited to displays for dependent systems, since control systems could also be affected. Integrity for fault considerations must be addressed for supporting systems in relation to dependent systems when the dependent system's provided functions are assessed at criticality levels of Major, Hazardous/Severe-Major, or Catastrophic. These integrity and fault considerations must address not only a particular dependent system, but also the

accumulative effect on other systems that the same supporting system's malfunction may affect. This may, in turn, affect the aircraft level functional hazard assessment (FHA) as the supporting system could act as a common point for simultaneous failures of more than one system. Additionally, the design of the supporting systems should preclude single point failures/faults for systems that support dependent systems functions assessed to have a criticality of Catastrophic.

(6) Certification Approach. There are three basic parts to the supporting systems concerns addressed herein and they are the inclusion of supporting systems in the integrity determination process, the design considerations for supporting systems relating to more than one system, and single point faults for the supporting systems themselves.

(i) A two-step procedure should be used to determine the adequate integrity level for supporting systems. The first step is to determine the level of criticality associated with loss/malfunction of all or any combination of the dependent system's functions and all combinations of other dependent system's functions that require support from the same supporting system. This can be achieved through the use of an FHA and associated FTA's. The criticality category level determined from this assessment must be a product of failure/malfunction possibilities for all of the involved dependent/supporting systems combinations and the worst case operational consideration for the function(s) provided by the dependent systems. The second step is to determine whether the supporting system's design integrity is sufficient to address the determined criticality category. The design integrity should address failures/malfunctions results of the dependent system(s), any combination of failures/malfunctions due to effects on more than one dependent system, and single point failures of the supporting system itself.

(ii) Analyses may be used to meet the criteria outlined in the second step above, for systems that support one or more dependent system functions whose loss or malfunction is assessed to be Hazardous/Severe-Major or Catastrophic. Analyses, such as a fault tree analysis, in combination with a common cause analysis to validate assumptions regarding the independence of faults, should be performed to show compliance. Testing may be required to validate the analysis, if the system is complex or dynamic in nature.

NOTE: Showing high reliability for a single thread system is not sufficient to meet the requirements for a Catastrophic failure condition category, thus reliability cannot be used to substitute for the preclusion of single point failures/faults.

(7) Summary: Supporting systems should be considered an integral part of the dependent system(s) that provides the required functions, for the purpose of addressing design integrity requirements.

b. **Complex System Integration.**

(1) Explanation. Complex integrated systems addressed by this paragraph are those systems that provide more than one function from a single electronic device or from more than one inter-related electronic devices/components. The inter-relationship is based on common aspect(s) of providing the functions. The definition of complexity as it applies to integrated systems, is a design condition that exhibits the characteristic of possible combinations of simultaneous failures/faults, as opposed to simple systems where there exists only failures/faults that can be considered individually. Integration that results in providing several functions from one design source inherently increases complexity, thus the two describing terms of "complex" and "integrated" are not independent from one another. Computers have become more powerful with recent increases in technology. Also, related sensors and servomechanisms have greatly improved. This has created an atmosphere from which complex integrated systems have spawned. Integrated systems can have the effects of reduced weight, economic advantages, and system enhancements. However, with these advantages there are some concerns that must be addressed, as this concept inherently creates problems with showing compliance to system independence requirements.

(2) Procedure.

(i) Integrated Systems typically compromise the concept of independence for failures/malfunctions and system/function separation. Using this as a given, the approach for showing compliance for systems that have a requirement for independence is to provide elevated system integrity to make up for the loss of independence. The requirements for independence are both direct and inferred. The direct requirements are defined by requirements for system separation and for specific systems. The inferred requirements for independence are those that inherently have independence by their method of implementation until integration of dissimilar functions by recent technology. They are inferred since past methods of implementation provided independence and therefore no direct requirement was defined.

(ii) The elevated integrity typically consists of high software levels and high system reliability that addresses the failure condition categories determined by a Functional Hazard Assessment (FHA). This approach basically states that the independence for failures and function separation is absent, but the probability is small for loss or malfunction, and the software level will match the threat level identified by the FHA. Provided the integrity is a reality, this approach works pretty well for the loss and malfunction aspects. However, system separation requirements may not be satisfied as easily, as they are mostly concerned with common mode failures from external sources. Some common mode concerns are temperature, fire, water, EMI, and physical mechanical threats.

(iii) The combinations of systems/functions that comprise a single integrated system are important. If any of the systems or combinations of systems that make up the composite integrated system are assessed to have a high criticality level, then the design integrity of the composite integrated system should match the highest of those assessments. In infrequent cases, partitioning and unique hardware/software

architecture may be exceptions to this determination. In cases where the FHA has determined low criticality for all combinations of systems/functions, many of the concerns associated with complex integration may be minimized.

(A) Concerns to be addressed for complex integrated systems that address failures and malfunctions:

(1) An FHA must be performed that considers each function individually and all combinations of functions for loss and malfunction. Additionally, all supporting systems, all combinations of supporting systems, and all combinations of supporting systems and dependent systems must be considered for loss and malfunction

(2) After the criticality has been determined for all functions and combinations of functions, the design integrity can be defined in terms of reliability and software level. The reliability must match or exceed the requirements derived from the FHA and associated FTA results. This includes any supporting system as well as the primary system.

(3) The software integrity level must match or exceed the requirements derived from the FHA and associated FTA results. This is true generally, for all of the software, if a single computer is utilized and no software or architecture scheme is implemented to provide partitioning/protection.

(4) If redundancy is required to meet the reliability requirements, adequate redundancy failure management must be provided. Redundancy failure/malfunction management is required to eliminate latent failures or undetected malfunctions. Redundancy management must address latent failures. Without the detection and management of latent and unannounced failures and malfunctions, duplication of subsystem components may not be creditable redundancy. If the first failure can result in unknown loss of one of the system's functionally duplicated parts, then the second failure in combination with the first failure must be treated as a single failure and no design credit can be given for redundancy.

(5) Redundancy design must consider similarity of software between redundant system components.

(6) Electronic Devices (EDs) such as the Central Processing Unit (CPU), Programmable Logic Device (PLD), Application Specific Integrated Circuits (ASIC), or other types of data storage or computing devices must be considered to have common mode failure potential, especially for control systems. This concern may be addressed in a variety of ways. One way would be to use dissimilar EDs between redundant implementations. Some other approaches may involve architectures with monitors that are dissimilar to the systems supplying the redundant functionality. Other hardware potential common mode failures must also be considered, such as power supplies, signal sources, and common Input/Output (I/Os) chips.

(B) The intent of system separation requirements is to minimize the possibility of total system failure/malfunction as a result of internal system failures or external influences. In some cases, system separation addresses systems that provide similar information by dissimilar means. An example of this type of system separation is the requirement for independence between the fuel quantity display system and the fuel low indication. This is a case where increased integrity can be accepted in lieu of total independence for small parts of an integrated system, depending on the extent of loss of independence and the associated failure condition category. Concerns to be addressed for system separation requirements are as follows:

(1) Internal concerns include common mode failures/malfunction that could result in unacceptable loss of satisfactory system functionality. Some of the sources of these failures/malfunctions include common electrical power supplies, common sensor sources, filtering referenced to common ground planes, common processing, and common threats from Electro–Magnetic Interference (EMI) sources.

(2) EMI from internal sources (Electro–Magnetic Compatibility (EMC)) and external sources (High Intensity Radiated Fields (HIRF)) must be addressed from systems separation aspects. Unless complete system immunity to EMI can be shown for designs of systems that provide functions to address catastrophic failures, the design should preclude influence from EMI events. This is of particular concern when redundancy is used to meet the criticality requirements. Designs should address the possibility of EMI affecting the required function because of close physical proximity between all or parts of the redundant sections of the system. Areas of design that have the most concerns are those that include redundant system sections in the same enclosure and the redundant sections have a common cavity for penetration of wiring connectors. Another significant area of concern is for redundant system sections that employ wiring cables with little physical separation between cables for the respective redundant sections. In these cases, different lengths of cables between these redundant sections would reduce the possibility that radiated EMI would affect the system sections simultaneously at the same frequency.

(3) Other separation concerns are associated with external physical installation aspects. These physical aspects include protection from fire, water, excessive thermal variations, excessive vibration damage, and any mechanical failure of another system/component that could possibly impair the integrated system's functionality and result in an unacceptable decrease in safety.

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 14 CERTIFICATION PROCEDURE FOR INSTALLATION OF VAPOR CYCLE AIR CONDITIONING SYSTEMS.

a. FAA/AUTHORITY Approval Philosophy. The vapor cycle (Freon) air conditioning system is generally considered "nonessential"; that is, its function is not necessary for safe flight. Therefore, the FAA/AUTHORITY looks at it from the standpoint of its potential of posing a hazard to the aircraft in the course of its normal function/malfunction or in case of a failure. 14 CFR Part 29.1309 thus becomes the dominant regulation concerning the system. However, if an air conditioning system is required for electrically powered equipment cooling, then a criticality assessment may show that the criticality level may be higher than nonessential.

b. Type of Refrigerant/Regulations/Environmental Impact.

(1) The refrigerant commonly used in automobiles and aircraft is known as Freon R-12 (home air conditioners use R-22). This Freon is one of the CHLOROFLUROCARBONS (CCl F) or CFCs. This compound is blamed for eroding the ozone layer in the Stratosphere (the chlorine in CFCs attacks and destroys the ozone molecules). The U.S. Clean Air Act restricts the production of CFCs. In 1992, production was restricted to 50%. The United States and most other industrial countries have agreed to phase out CFC production by 1995. CFC is prohibited beginning in the year 2000. Beginning in June 1992, CFCs required recovery.

(2) The new refrigerant HYDROFLUROCARBONS (CH FCF) HFC-134a or R-134a does not deplete ozone. Automobile industries as well as some small aircraft manufacturers are designing air conditioning systems with this non-ozone-depleting refrigerant. This HFC-134a is currently available and manufactured by the Dupont Company.

c. Suggested Compliance Checklist: R=Report, D=Drawing, T=Test.

REGULATION	SUBJECT	METHOD
29.301	Loads	R
29.303	Factor of safety	R
29.305	Strength & Deformation	R
29.307	Proof of structure	R
29.561	Emergency Landing conditions	R
29.603	Materials and Workmanship	D
29.605	Fabrication methods	D

Suggested Compliance Checklist: R=Report, D=Drawing, T=Test (continued)

REGULATION	SUBJECT	METHOD
29.607	Fasteners	D
29.609	Protection of Structure	R
29.613	Material Strength	R
29.685a,c	Control system details	D
29.831	Ventilation	R
29.853	Compartment interiors (a, b, c)	T
29.855	Cargo compartments (a, b)	D
29.863	Flammable fluid protection	D
29.1301	Function and Installation	T
29.1307	Miscellaneous Equipment, 29(b)	D
29.1309	Equipment systems and Installation	R
29.1435	Hydraulic system	T
29.1461	Equipment containing High energy rotors	D, R
29.1541	Markings and placards	D

d. Electrical System Considerations

(1) An electrical wiring diagram showing interconnections of all electrical components should be provided. The wiring diagram should show adequate circuit protection (circuit breakers). It should also indicate the use of wiring of adequate size and length to take maximum currents to which the system would be exposed. Power to air conditioning electrical system should be connected to an electrical power source that provides adequate power and does not interfere with essential electrical loads and provides solid electrical ground to airframe.

(2) An electrical load analysis should be provided to demonstrate the availability of adequate current to the air conditioning system from the helicopter electrical power source during all phases of flight and system operation. The system should also be powered from the helicopter electrical power source that provides adequate power and does not interfere with essential electrical loads.

(3) The air conditioning system should be capable of a successful functional test and electromagnetic compatibility test. It needs to be shown that air conditioning equipment will not be a source of interference with the essential equipment. Reference §§ 29.1351(a)(b), 29.1357, 29.1365, and 29.1367.

e. Structural Considerations

(1) Overall aircraft structure should be substantiated for the increased weight of the air conditioner modification. Each air conditioner component, its backup structure, and its attachment to the aircraft structure should be substantiated to the strength requirements of Subpart C (14 CFR Part 29) and the design requirements of Subpart D

(14 CFR 29). Load factors should be chosen considering the most critical of limit maneuvering load factor (§ 29.337), gust load factor (§ 29.341), or if applicable, emergency landing conditions (§ 29.561). Load paths must be substantiated for the distribution of static and/or dynamic load conditions. Fatigue substantiation may be required depending on the installation (§ 29.571).

(2) The modifications done on the structure, due to air conditioner equipment installation, should not create any adverse qualities to the overall structural integrity of the aircraft. Any access holes cut in the aircraft structure for routing refrigerant/electrical line skin/stringer cutouts for intake or exhaust holes, etc., should be substantiated for overall structural integrity.

(3) All attachment hardware used for the air conditioner modification should be substantiated to meet the increased structural requirements.

f. System and Equipment Considerations

(1) The vapor cycle (Freon) system is properly considered to be a gaseous system. Granted, during some portion of the cycle, the Freon is in a liquid or liquid/vapor state; however, it is a gas under standard atmospheric conditions. Therefore, the proper system test would be a pressure test for a gaseous (pneumatic) system.

(2) 14 CFR Part 29 does not call out specific testing criteria but instead relies on § 29.1309 to address potential hazards due to pressurized gas systems. A proof pressure test of 1.5 times the maximum normal operating pressure of the system is appropriate to satisfy the intent of § 29.1309.

(3) The Freon pressures vary throughout the system during operation, but the maximum normal operating pressure of the components upstream from and including the high-pressure side of the expansion valve can be regarded as the value limited by the overpressure switch. The condenser and receiver-dryer are of special concern as they are of relatively large volume and a failure could cause damage to the aircraft structure or essential mechanical components.

(4) The highest pressure normally experienced by the low pressure portion of the Freon system (downstream of the expansion valve to the suction side of the compressor) occurs when the system is shutdown; hence that pressure can be used as the "maximum normal operating pressure" for the proof and burst tests in this portion of the system.

(5) The burst pressure tests can be done on a component basis or on the entire system. The proof pressure test is best done on the entire system to allow observation of any movements of the flex hoses and other components under pressure, which may interfere with essential helicopter components.

(6) System capacity or efficiency does not affect the review of the system from the standpoint of safety; however, § 29.1301(a) requires "Each item of installed equipment be of a kind and design appropriate to its intended function". Hence the calculations of heat load of the cabin to be cooled in British thermal units (Btu) and the cooling capacity of the air conditioner system may be required. Care should be taken to assure that the § 29.831 required fresh air ventilation rate is maintained. The vapor cycle is a closed system (recirculating of existing air) with no fresh air make up capability.

g. Powerplant Considerations (for mechanically-driven air conditioning compressors):

(1) The drive systems and the drive system component supporting structure should be adequate both statically and in fatigue to handle any loads associated with the air conditioning drive mechanism. Both normal operating and failure conditions (compressor lockup) should be considered. For example, on systems which are belt driven off the tail rotor drive, all components should be substantiated at the highest torque, shear, and moment loads that can be imposed by the belt drive (compressor seizure as well as compressor start), in combination with the loads associated with max transient tail rotor drive torque.

(2) The compressor drive must not affect the normal function of the drive system. Additional load on the gearbox or the drive pulley should not cause gearbox temperature increase (§ 29.1041). Exceeding the gearbox temperature limit can cause stud loosening.

(3) The mounting of a bracket (for a compressor) or an idler pulley on the transmission top case should not affect the structural integrity or the corrosion protection of the transmission. If a compressor/blower is mounted to the gearbox, the overhang moment (which is created by the weight of the compressor and its center of gravity distance), should not exceed the mount limit. The addition of a bracket which picks up several existing top case-to-main case attachment studs may provide an additional path for water to enter and corrode the cases around the studs. In some cases, the studs may not be adequate to carry the additional load, because the thickness of the top case where each of these studs is typically individually controlled could result in the bracket warping upon installation. This warping results in unequal axial clamp-up at the studs, which further aggravates the stud loading and may also lead to significant case fretting. Also, the original studs may not be of adequate length to maintain proper thread extension through the nut after the bracket is installed.

(4) In the event of a compressor seizure, it should be substantiated that no damage to the primary drive system or aircraft structure can occur. On belt driven installations, no damage to the primary drive system should occur due to burning or flailing belts. On a shaft driven compressor/blower installation, the shaft shear section should fail before exceeding the gearbox torque limit. When the shear section fails, the

shaft should be contained or otherwise prevented from interfering with the drive system or flight controls. Similarly, a failure of the compressor or drive belts should not damage the drive system or interfere with the flight control mechanism.

(5) The mechanically-driven compressor should not adversely affect the (vibration) dynamic characteristics of the drive system in any operating condition (§ 29.907(b)). Maximum vibration levels should be given in engine/drive system installation data (§§ 29.901(b) and 29.927(a)).

h. Flight Analyst/Pilot Considerations.

Update Rotorcraft Flight Manual supplements (RFM) to show performance effects. If the installation is such that it interferes with engine inlet airflow, then determine any performance loss, evaluate inlet distortion, and validate turbine engine operating characteristics. Reference §§ 29.45 and 29.939. TIA should include operational tests such as intended function and abnormal/emergency operation. Conduct EMI tests and evaluate the RFM supplement. Reference §§ 29.1581, 29.1583, 29.1585, and 29.1587.

i. Safety Devices / Failure Mode Effect Analysis

(1) SAFETY DEVICES:

(i) Automatic Load shedding
(ii) Current Limiter
(iii) Compressor Temperature Limiter
(iv) Compressor Electric Motor temperature limiter
(v) Compressor Discharge pressure limiter
(vi) Oil separator / Injector
(vii) Containment shrouds
(viii) Belt guard
(ix) Pressure line gallery cover
(x) Ignition source protection (Freon is flame suppressant)

(2) FAILURE MODE EFFECT ANALYSIS: A Failure Mode Effect Analysis is crucial to the safety evaluation of the systems. Consider the areas given below:

(i) The overpressure safety system assures there is a means to shut the system down prior to a critical pressure developing (overpressure switches, blowout plugs, redundant circuit breakers, etc.).

(ii) An electrical load analysis should show that the failures in the air conditioning system do not jeopardize the safe operation of flight essential and flight critical airborne systems.

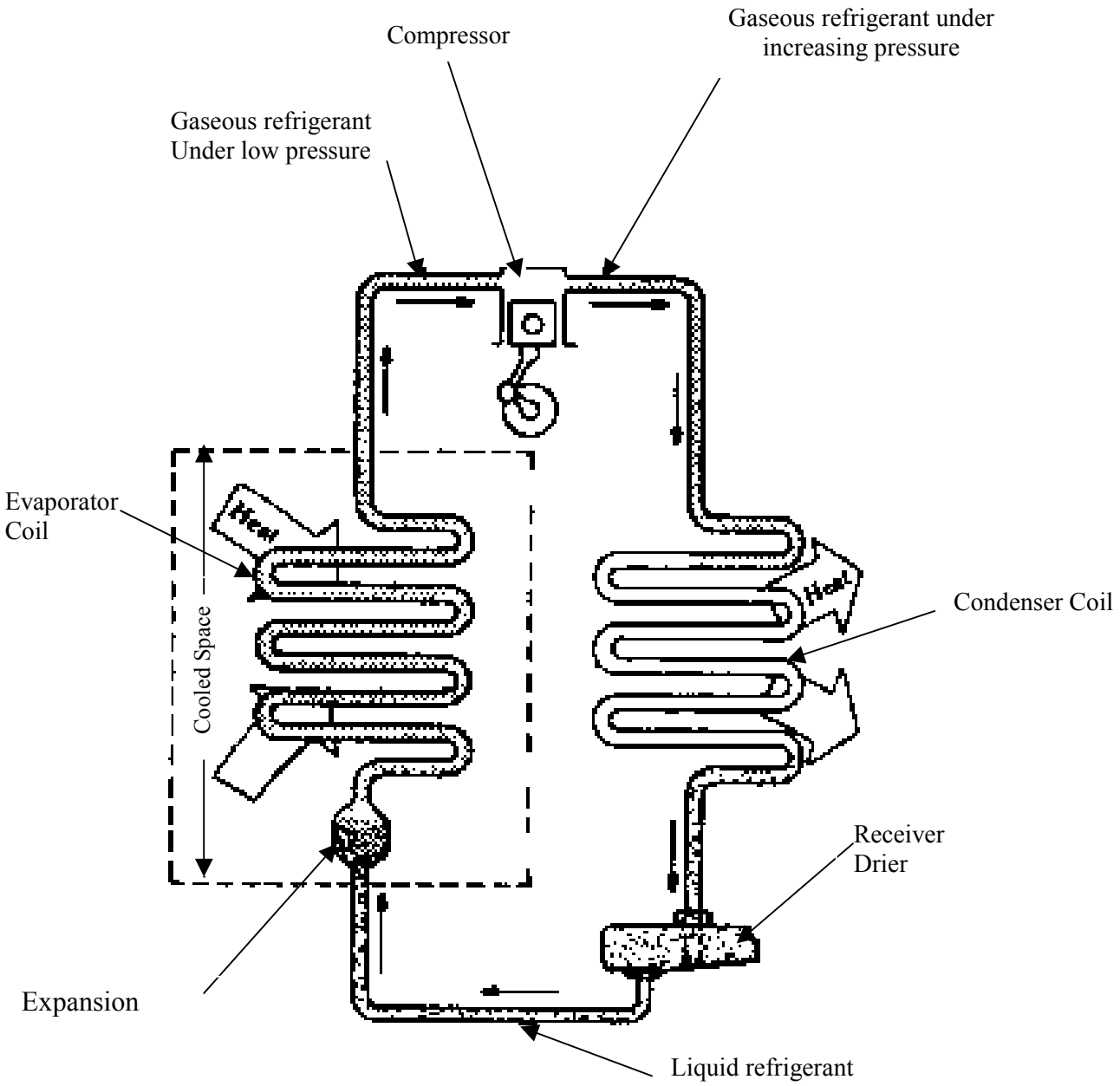
(iii) For systems driven by engine/transmission/drive shaft, a powerplant evaluation should be made to determine power available, vibration characteristics, etc.

(iv) Coupling/drive belt failure and its effect on adjacent components.

(v) The area around the condenser and receiver dryer (and any other high-pressure components) to determine if there are any critical components in the vicinity that could be damaged if a burst line occurs.

(vi) Assure that the Condenser blower/fan construction is such that if the fan or impeller fails, the pieces will not damage other components or helicopter structure (§ 29.1461).

(vii) The design should be such that Freon leakage cannot be ingested into the engines (§ 29.1309). Freon is an excellent fire suppressant. Ensure that no pressure relief valve or blow out plug (on the receiver dryer) is located inside the cabin. Quantities of Freon should also be prevented, as much as possible, from entering the cabin in the event of a leak. The rapid expansion of liquid Freon to its gaseous state in the close proximity of the flight crew could be disconcerting (could fog up the cabin). Liquid refrigerant, if allowed to strike the body, could cause frostbite, and if allowed to strike the eye, can cause blindness.



REFRIGERATION CYCLE

Figure AC 29.MG 14-1

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CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 15. AIRWORTHINESS APPROVAL OF ROTORCRAFT HEALTH USAGE MONITORING SYSTEMS (HUMS)

a. Purpose. The purpose of this section of the AC (AC 29 MG 15) is to provide guidance to achieve airworthiness approval for rotorcraft Health and Usage Monitoring System (HUMS) installation, credit validation, and Instructions for Continued Airworthiness (ICA) for the full range of HUMS applications. Mandatory terms used in this section of the AC, such as “must", are terms used only in the sense of ensuring the applicability of these particular methods of compliance when the acceptable means of compliance described herein are used. This section of the AC does not change regulatory requirements and does not authorize changes in, or deviations from, regulatory requirements. This section of the AC establishes an acceptable means, but not the only means of certifying a rotorcraft HUMS. AC 29 MG 15 addresses the most complex/extensive HUMS; systems of lesser complexity may be addressed by use of only the parts of this section of the AC that are pertinent. HUMS applications in the Catastrophic criticality category are not addressed herein.

b. References and Related Documents.

(1) Federal Aviation Regulations (FAR) Parts 21, 29, 33, 91, 125, 127, 129, 133, 135, 145 – Corresponding European Joint Aviation Requirements (JAR) 21, 29, JAR E, JAR-OPS 3.

(2) FAA Advisory Circular AC 29-2C and the European corresponding ACJs, AMJs where applicable.

(3) Standards - Latest revision of RTCA/DO-160/ED-14, RTCA/DO-178/ED-12, SAE documents ARP 4754, and ARP 4761.

c. Background.

(1) Various types of HUMS have been developed, and they are likely to be used more in the future. Initially, these systems were installed to show the feasibility of gathering meaningful data to modify required maintenance and/or operational actions. The degree of qualification required for this type of installation is relatively low. However, there is an increasing number of certification applications to install HUMS and use its data to intervene in maintenance and/or operations of the rotorcraft. This type

of installation requires a higher degree of qualification, commensurate to the criticality of the most severe effect of the intervention action(s) on the rotorcraft.

(2) HUMS typically consists of a variety of onboard sensors and data acquisition systems. The acquired data may be processed onboard the rotorcraft or on a ground station (or a combination of both) providing the means to measure against defined criteria and generate instructions for the maintenance staff and/or flight crew for intervention.

(3) The certification of HUMS must address the complete process, from the source of data to the intervention action. There are three basic aspects for certification of HUMS applications: installation, credit validation, and Instructions for Continued Airworthiness (ICA). These aspects are not totally independent and do have varying interactions with each other.

d. Definitions.

(1) END-TO-END: The term "end-to-end" as used in the text is intended to address the boundaries of the Health Usage Monitoring System (HUMS) application and the effect on the rotorcraft. As the term implies, the boundaries are the starting point that corresponds with the airborne data acquisition to the result that is meaningful in relation to the defined credit without further significant processing. In the case where credit is sought, the result must arise from the controlled HUMS process containing the three basic requirements for certification as follows:

(i) Equipment installation/qualification (both airborne and ground),

(ii) Credit validation activities, and

(iii) Instructions for Continued Airworthiness (ICA) activities.

(2) HUMS: Equipment, techniques, and/or procedures by which selected incipient failure or degradation and/or selected aspects of service history can be determined.

(i) Health Monitoring System: Equipment, techniques, and/or procedures by which selected incipient failure or degradation can be determined.

(ii) Usage Monitoring System: Equipment, techniques, and/or procedures by which selected aspects of service history can be determined.

(3) Credit: To give approval to a HUMS application that adds to, replaces, or intervenes in industry accepted maintenance practices or flight operations.

(4) Application(s): A HUMS process implemented for a distinct purpose(s).

(5) Criticality (1309): This term describes the severity of the end result of a HUMS application failure/malfunction. Criticality is determined by an assessment that considers the safety effect that the HUMS application can have on the aircraft. There are five criticality categories as follows:

(i) Catastrophic: Failure conditions, which would prevent continued safe flight and landing.

(ii) Hazardous/Severe Major: Failure conditions, which would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be:

(A) A large reduction in safety margins or functional capabilities,

(B) Physical distress or higher workload such that the flight crew could not be relied on to perform their tasks accurately or completely, or

(C) Adverse effects on occupants including serious or potentially fatal injuries to a small number of those occupants.

(iii) Major: Failure conditions which would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be, for example, a significant reduction in safety margins or functional capabilities, a significant increase in crew workload or in conditions impairing crew efficiency, or discomfort to occupants, possibly including injuries.

(iv) Minor: Failure conditions which would not significantly reduce aircraft safety, and which would involve crew actions that are well within their capabilities. Minor failure conditions may include, for example, a slight reduction in safety margins or functional capabilities, a slight increase in crew workload such as routine flight plan changes, or some inconvenience to occupants.

(v) No-Effect (Non-hazardous class): Failure conditions which do not affect the operational capability or safety of the aircraft, or the crew workload.

(6) Integrity: Attribute of a system or a component that can be relied upon to function as required by the criticality determined by the Functional Hazard Assessment (FHA).

(7) Mitigating Action: An autonomous and continuing compensating factor which may modify the level of qualification associated with certification of a HUMS

application. This action becomes a part of the certification requirements and, as such, is required to be performed as long as that certification requirement is not changed by a subsequent re-certification. An example of a mitigating action is a pilot's comparison of airborne HUMS data with aircraft instrument data.

(8) Commercial Off-the-Shelf (COTS): This term defines equipment hardware and software that is not qualified to aircraft standards. An example of COTS equipment hardware and software is a personal computer (PC) and its operational software.

(9) Independent Verification Means: An independent process to verify the correct functionality of a HUMS application on a ground station that utilizes COTS. The intent of independent verification is to gain some degree of confidence in the COTS operational reliability.

NOTE: This process may be discontinued when sufficient confidence in the application has been achieved.

(10) Synthesis: The process of evaluating service history and any other relevant data with the objective of validating and, if necessary, refining the performance of an approved credit.

e. Certification Approach.

(1) There are three basic aspects to Health Usage Monitoring Systems (HUMS) certification. Certification of HUMS must address all three. The three aspects are installation, credit validation, and Instructions for Continued Airworthiness (ICA). These aspects are not totally independent and do have varying interactions with each other. A method to address these aspects is provided by the approach herein. Installation includes all the equipment needed for the end-to-end application that is associated with acquiring, storing, processing, and displaying the HUMS application data, including airborne and ground based equipment. Credit validation includes evidence of effectiveness for the developed algorithms, acceptance limits, trend setting data, tests, etc., and the demonstration methods employed. A plan is needed to ensure continued airworthiness of those parts that could change with time or usage and includes the methods used to ensure continued airworthiness.

(2) The certification process should begin with the declared application intent, and determination of the resultant criticality. This declared intent should consider whether this application is for credit, that it adds to, replaces, or intervenes in maintenance practices or flight operations. When the declared intent is for credit, the end-to-end criticality for such an application should be determined and used as an input to establish the integrity criteria. If the declared intent is for non-credit, it may be certified as long as it can be shown that the installation of the equipment will not result in a hazard to the aircraft.

(3) The end-to-end criticality can be determined by performing a Functional Hazard Assessment (FHA). The integrity level is required to be equivalent to the determined end-to-end criticality. Compliance with the criticality level established by the FHA must be demonstrated. This may be achieved by a combination of application qualification plus appropriate mitigating actions.

(4) Applications are often qualified to a low level of integrity due to the assessment of criticality; however, it may be desirable to transition to a higher qualification level for future uses. Transition from one level of integrity to another will require re-evaluation.

NOTE: A certification plan may be provided to assist in the certification process. At a minimum, this plan should address the proposed means of compliance to each applicable paragraph of this advisory circular for a given application. Early submittal of this plan to the regulatory Authority is recommended.

f. Installation. Installation approval must cover systems and equipment that acquire, store, process, and display HUMS data and includes the airframe installation, or any one of these functions for a particular application. AC 29 MG 15 will address the most complex/extensive HUMS; systems of less complexity may be covered by use of only the parts of this AC that are pertinent. Different systems exhibit varying capabilities and configurations. Additionally, there may be different functional distributions between airborne and ground based equipment. HUMS equipment requirements consist of common requirements plus the unique requirements of airborne and ground based equipment.

(1) Common Requirements. A common requirement is one that applies to airborne, ground based, and installation equipment. These common requirements are discussed below.

(i) Criticality Determination.

(A) Criticality determination is a primary decision point relating to the depth of requirements for certification. The intended application can range from systems that acquire data for proof of concept only, to a system that acquires and processes data to determine if a life-limited part should be replaced. This range of applications will have a corresponding range of criticality for the systems from No Effect to Hazardous/Severe-Major. Systems in the Catastrophic criticality category are not addressed in AC 29 MG 15.

(B) If any credit is to be gained, the general guidelines for determination of criticality levels will be either Minor, Major, or Hazardous/Severe-Major. They will be in agreement with the resulting effect of the end-to-end criticality assessment.

(C) Typical examples of applications which may be classified as Catastrophic are as follows :

(1) Applications providing cockpit warning(s) which are the only means of detection with associated flight manual instructions to land immediately.

(2) System applications, for which constantly misleading information could be assessed as leading to a Catastrophic condition, must be designed to either detect these errors (e.g. Built-In-Test, system redundancy, etc.) and/or be tolerant to these errors (i.e., procedural, etc.).

(D) The Functional Hazard Assessment (FHA) may be a preliminary document to the Preliminary Safety Assessment (PSA) or a part of the PSA. The FHA is a top down analysis (which should involve pilots and flight analysts as well as engineers) that starts with the hazards to the rotorcraft and traces these hazards to the system, subsystem, and component level in the areas affected by HUMS. This type of analysis starts with the determination of what undesirable effects can occur as a direct or indirect result of using HUMS for maintenance or operational actions. The level of severity associated with this effect will result in assigning a criticality level that uses the definitions of criticality contained herein.

(E) The final level of equipment qualification may not only be the result of technical considerations, but also of other mitigating actions, of which there are many types. Many of these actions can result in a reduction of qualification levels for equipment.

(ii) Mitigating Actions.

(A) A mitigating action is an autonomous and continuing compensating factor which may modify the level of qualification associated with certification of a HUMS application. These actions are often performed as part of continued airworthiness considerations and are also an integral part of the certification. As such, the continuation of certification limitations, where appropriate, must be included in the Instructions for Continued Airworthiness (ICA). Mitigating actions are subjective in nature and are an intended method(s) of application where the pre-mitigated levels of integrity are defined.

(B) Applications that use COTS software and therefore may not be fully qualified applying RTCA/DO-178/ED-12 methodology may be accepted by alternative qualification methods as stated in paragraph f(3). Therefore, the subsequent use of mitigating actions that are of themselves of a subjective nature should be approached with caution. A mitigating action must be based upon the integrity level derived from the FHA.

(C) If the mitigating action is an operational consideration, the same concerns apply for continuing the mitigating action. The mitigating action should be recorded in the certification limitations and in the approved flight manual.

(iii) Performance. There must be minimum end-to-end performance criteria consistent with the application's intended use. Performance criteria, as a minimum, should consider accuracy, timing/sampling, resolution, event recognition, and consistency. The HUMS signal source must be compatible with the determined qualification level. Tests should be conducted to demonstrate that these criteria are met.

(2) Airborne Equipment Installation. Airborne equipment and the associated installation qualification procedures are the same as for any other airborne equipment. The installation qualification and the equipment qualification may be considered two separate activities although there is an obvious relationship between them. Signal independence, irrespective of method of implementation, should exist to the extent that acquisition of HUMS signals should not compromise the level of safety or reliability of functions provided by other equipment as a result of signal sharing.

(i) Equipment Installation. Equipment not approved by other methods must be approved as part of the installation and must consider overall system requirements.

(A) Equipment Qualified as Part of Installation. Equipment qualified all or in part as a part of the installation includes minor and major parts. Examples of minor parts are: connectors, common usage relays, diodes, etc. Examples of major parts are non-prequalified equipment (equipment not TSO'd or not qualified under the TSO to the required level for installation approval), consisting of significant system components and as transducers with their interfaces. Equipment qualification must consider environmental qualification (RTCA/DO-160/ED 14) including high intensity radiated fields (HIRF) and lightning.

(B) Software. RTCA/DO-178/ED-12 should be used for the software development standard. (See following figure for typical airborne application process for software not containing COTS.)

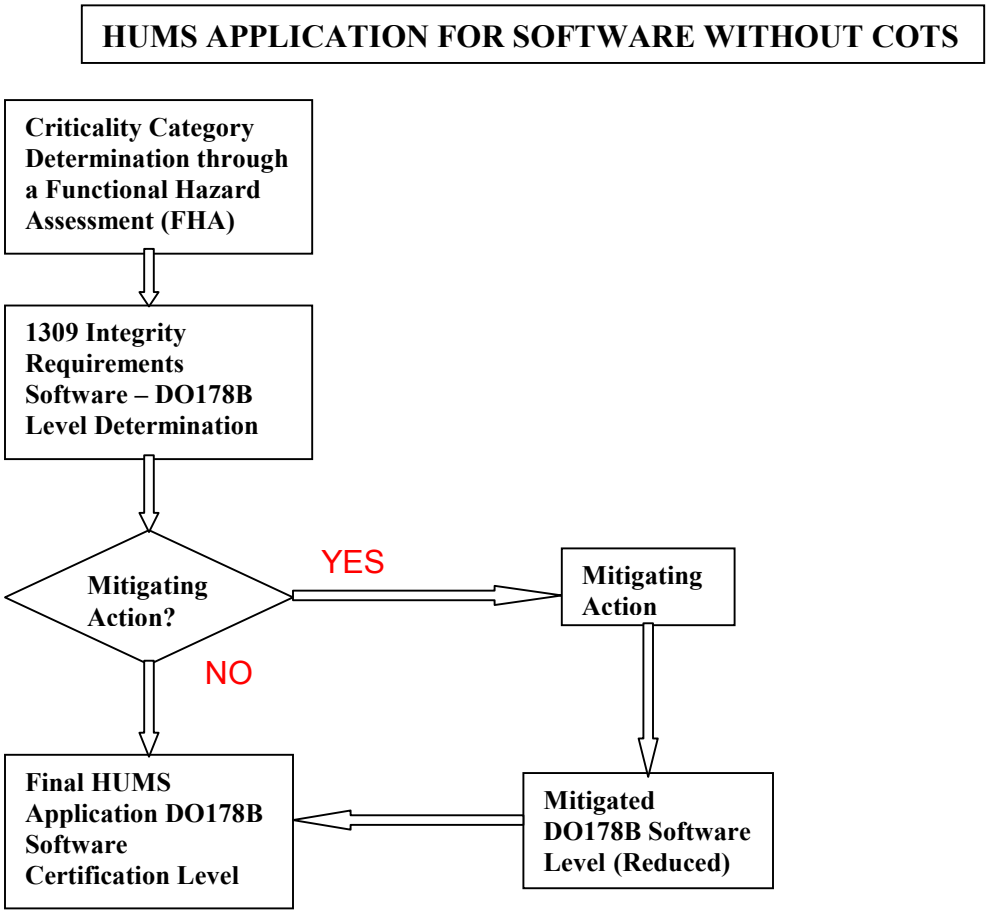

FIGURE AC 29 MG 15-1. Flow chart for use of Mitigating Actions applied for reduction of Software Level.

(ii) Installation Specific Considerations. The overall installation considerations should include, as a minimum, supply of electrical power, environmental conditions, system non-interference, and human factors if operations are affected.

(A) Supply of Electrical Power. An adequate source of electrical power for HUMS must be provided. The reliability of the power source must be commensurate with the required equipment qualification level. There should be no unacceptable reduction in the level of safety or reliability for other equipment as a result of acquiring power for HUMS.

(B) Electromagnetic Compatibility. Electromagnetic compatibility (EMC) must be addressed. Complex systems may require an EMC test plan, which includes a matrix of aggressors versus victims. The end result should be to assure that HUMS does not interfere with or is not affected by any other installed equipment.

(3) Ground Based Equipment Installation.

(i) Ground based equipment is typically used to process and display the data collected by airborne means. This processed data will ultimately be used to make decisions pertaining to some intervention action or provide data to other processing means to make the intervention action determination. Since the ground based equipment may be an important part of the process for determination of intervention actions, its integrity and accuracy requirements must be the same as any other part of the HUMS process.

(ii) The determination of compliance to the integrity requirements for ground based equipment is difficult when it is recognized that this equipment may, for the most part, be commercial and not necessarily designed specifically for the HUMS application. This section is intended to allow for the possibility of systems that contain COTS hardware and software, where the hardware is likely to be a personal computer and the operational software is COTS. The determination of compliance to the integrity requirements for COTS is based on equivalence, which is subjective. COTS service history alone will not be sufficient to comply with the requirements herein. Any ground based processing equipment that consists of commercial hardware and software must have satisfactory service history and an independent means of verifying the results of the processing. This independent verification means may be discontinued with the certifying Authority's agreement to modify the original HUMS approval and remove this requirement after significant quantities of the processed data consistently agree with the verifying means.

NOTE: The suggested processes contained in this document for acceptance of a ground based system that possibly includes COTS hardware and software is limited to ground based equipment for HUMS applications only. The integrity determination methods for systems that do not contain COTS is the same as described for the airborne systems.

(A) Independent Verification Means. The required independent verification means may consist of any one of many methods. Independent verification means may parallel only the ground based system processing or parallel all or any portion of the process that includes the COTS equipment processing. Some acceptable methods may include the following:

(1) Physical inspection(s).

(2) Redundant processing by a second dissimilar PC with different COTS from the primary processor.

(3) A combination of physical inspection(s) and independent dissimilar processing.

(4) Satisfactory comparison of processed directed action to actual maintenance performed as a result of inspection. This approach would require data collection on the system prior to actual credit application. The amount and duration of data collection should be agreed between the applicant and the certifying authority at the beginning of the project on a case-by-case basis.

(5) Any other independent means of verifying the accuracy/integrity of the equipment including software by a satisfactory comparison to the directed action of the HUMS processed data.

(B) Integrity Level Considerations. The methodology is the same for different integrity level requirements as they relate to COTS hardware/software, but the compliance requirement will vary. The processes described in the previous and subsequent paragraphs of f(3) should be applied to meet the initial integrity requirements for the criticality categories of Hazardous/Severe Major and Major. Minor criticality category level will also require qualification by this process, except that independent verification can be performed after certification, provided that an approved plan is submitted for this activity. Other applications that do not employ COTS will use standard engineering practices to satisfy the integrity level considerations. Modification of Approved Systems. Changes to the equipment including software should be qualified on a case-by-case basis that is dependent on the effect on the integrity and functionality of the system. If mitigation had been successfully demonstrated for the original configuration, the mitigation must be shown to provide the same level of integrity for the changed configuration.

(C) Ground Based Equipment Hardware. This hardware may consist of data processing, display, and possibly printing equipment or other accessories. The hardware must be compatible with the intended application and software. The independent means of verification activity is required due to the use of COTS hardware.

(D) Software. Most systems will employ two types of software. One type is the operational software and the other is the HUMS specific software. The operational software may be COTS. (See following figure for typical ground based application process for HUMS specific software using COTS as an operational software.)

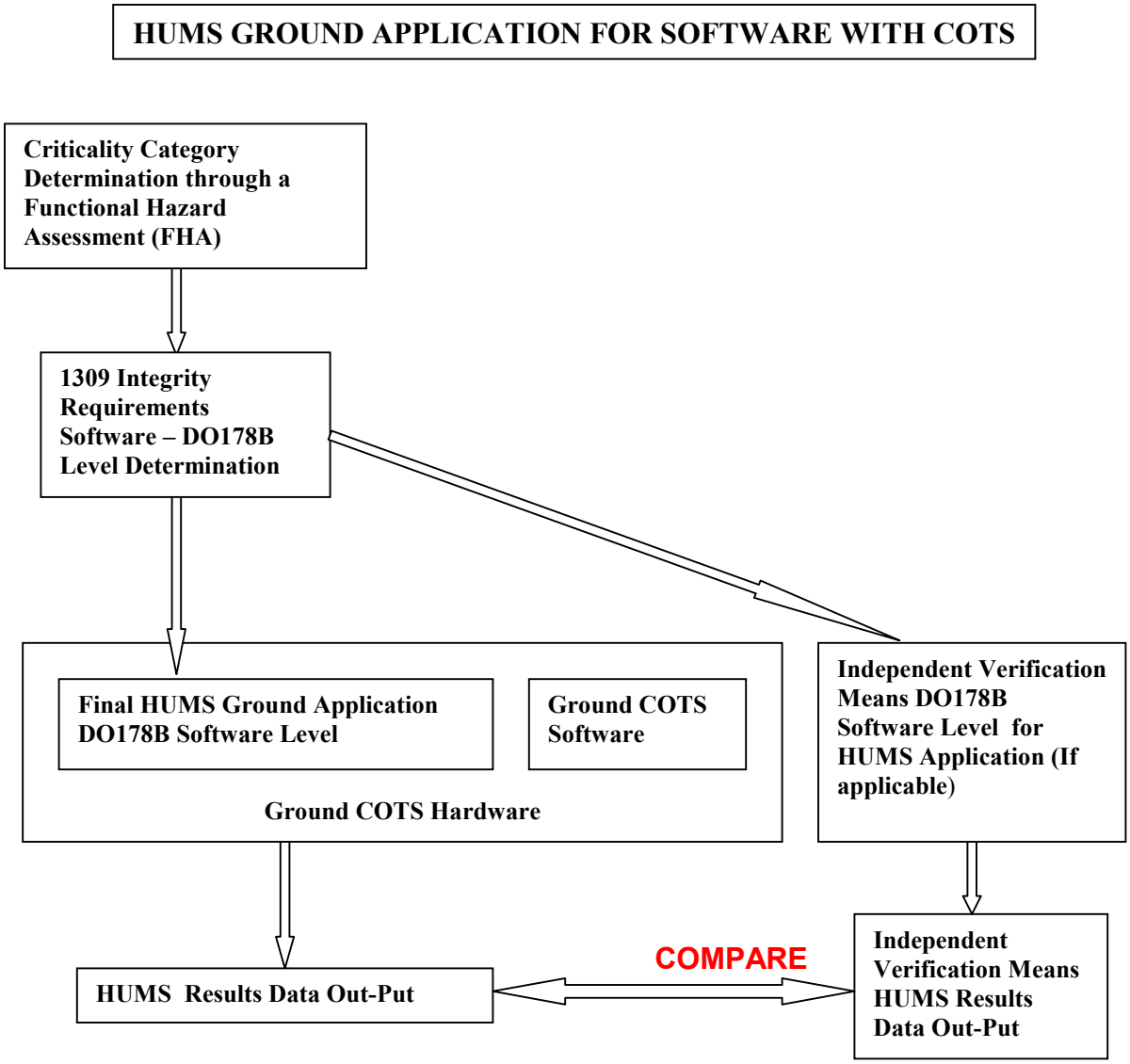

FIGURE AC 29 MG 15-2. Flow chart for application of HUMS specific software with a ground base that uses COTS software for operational software.

(1) COTS. This type of software can only be accepted by subjective considerations, such as service history, independent verification means, and design of the system to limit access to the operational COTS software to make changes. The independent means of verification activity is required due to the use of COTS software.

(2) HUMS Specific Software. This software should be developed to the integrity level required by the system criticality assessment using RTCA/DO-178 as the

standard. This system determined level should be a result of the end-to-end criticality assessment and, in general, the same as the airborne software. Use of mitigating actions is dependent on constraints stated in paragraph f(1)(ii).

(E) Data Processing. Data processing equipment and software should have the capacity to process the amount of data required. It should not introduce errors or provide out of specification accuracy for any parameter. The speed of processing should not be limited, by the hardware or software, to an unacceptable rate. The acceptability of speed will depend on the amount of data to be processed and the specified performance for HUMS data processing. The speed should be reasonable to accomplish data processing in a reasonable time for the particular HUMS application. Hazardous/Severe Major or Major criticality applications that contain COTS should be part of a dedicated system or demonstrate adequate protection for the higher level processing from anything else processed on the same equipment. Subject to a favorable comparison to the required independent verification means, Minor criticality applications need not be part of a dedicated system.

(F) Display and Peripheral Equipment. The display, for most cases, may be a part of the processing equipment or closely interface with it. It must be compatible with other parts of the system and provide a clear usable presentation.

(G) Data Communications. Network applications, modem interfaces, and other system sharing and transmission features may be utilized for integrity levels associated with Major and Minor criticality categories, provided that the independent verification means covers the use of these features. Integrity levels associated with Hazardous/Severe Major criticality categories may utilize these features only if sufficient protection can be shown to assure that this level of integrity is maintained throughout any foreseeable failure/malfunction or mistake in any associated application, in addition to required independent verification means.

g. Credit Validation. HUMS applications for which credits are sought must be validated. For each application, evidence shall be provided that the physics involved is understood and therefore that the monitoring technique/algorithm/parameter, rejection criteria, and associated intervention actions are well chosen. The designer of the component/equipment to be monitored is the most logical choice for this determination. However, in some cases the source can be from any organization as long as the validation criteria herein can be satisfied. If changes are proposed to an approved system, re-evaluation is required to ensure existing credit(s) are not invalidated. The degree of effort will vary and depend on the application type, the credit sought, and the consequences of failure or any other malfunction. The validation process would generally need to include the following:

- Description of application and associated credit.
- Understanding of the physics involved.

- Validation methodology.
- Introduction to service.
- Continued airworthiness (synthesis).

NOTE: Early notification to the regulatory Authority of the credit type and the proposed method for validation is recommended.

(1) Description of Application and Associated Credit.

(i) There are many types of HUMS credits with different levels of criticality. Some may be the introduction of new maintenance practices, in place of the established maintenance practices, and others may be the introduction of additional safeguards for safety where all standard practices are retained.

(ii) It is important to fully evaluate and describe the proposed credit and the worst effect on the rotorcraft should the application fail or malfunction. This evaluation is needed to determine the system criticality, the system installation integrity requirements, and the depth and scope of the credit validation effort.

(2) Understanding the Physics Involved.

(i) The mechanisms of failure and/or degradations associated with the requested credit should be understood. This includes how a failure occurs and/or at what rate the degradation progresses and a determination of the point where intervention action is necessary. For some complex applications, this may include supporting information from validated analytical tools such as finite element analysis and fracture mechanics.

(ii) These understandings should be used to determine the four important characteristics of a HUMS application.

(A) The technique to be used.

(B) The appropriate alert limits, including trending where appropriate.

(C) The appropriate intervention action.

(D) How often to monitor to give optimum opportunity for the intervention action to be effective.

(iii) This should also recognize the different characteristics of the failure/degradation and determine when trending or a step function is most appropriate.

(3) Validation Methodology. All HUMS applications should have their validation process based on suitably representative physical data. This process may use direct or indirect evidence, or a combination of the two, depending upon the credit type and the criticality on the aircraft of any HUMS failure or malfunction.

(i) Direct Evidence.

(A) When the HUMS application is classified as Hazardous/Severe Major, then direct evidence must be gathered. Examples of where this might be the most appropriate method include maintenance tasks such as vibration checks for imbalance/misalignment of high energy rotating equipment, fatigue life counting, or going "on-condition" for flight critical assemblies.

(B) Direct evidence is required for establishing that the HUMS application is sensitive to and obeys predicted response rules for the damage type, giving consistent alerts. This evidence may be gathered from several sources as follows:

(1) Actual service experience on HUMS equipped aircraft,

(2) "Seeded tests" (where the wear, defect, or deterioration is introduced, allowed to develop, and the technique response verified), and

(3) On- aircraft trials, investigating cause and effect (for example, introducing degrees of imbalance and calibrating the techniques response).

(C) Tests should be representative of the aircraft for which the credit is being sought and of test conditions representing the flight regime that would prevail when data is normally gathered (e.g., cruise). It should be established that the evidence gathered from on-aircraft ground trials or rig based seeded tests is valid in flight.

(ii) Indirect Evidence.

(A) When the HUMS application is classified as "Major" or lower, indirect evidence may be gathered. Criteria for this approach includes a criticality determination of Major or lower and either or both; application to "on-condition" maintenance actions, and/or lowering the probability of undetected failures. Monitoring of a high number of potential failure modes can collectively determine the probability of undetected failures. Here, it may not be practicable to generate direct evidence for each failure.

(B) Proven analytical methods may be combined with sound engineering judgment to provide calculated/derived criteria; tests can be performed to validate these criteria. Model based analytical methods for predicting damage progression (e.g., finite element analysis and fracture mechanics) may allow for a validation by claiming

analogy with 'direct' evidence generated for other aircraft types or equipment. However, to more fully validate this analogous data set, a degree of direct evidence for the actual equipment being monitored is still likely to be necessary to prove similarity of application. This might be achieved by performing an appropriate number of seeded defect tests and, in effect, "sampling" the range of failure types contained.

NOTE: For both direct and indirect evidence, the whole system must be validated end-to-end.

(4) Controlled Introduction to Service.

(i) For some credit applications, full validation and implementation may be possible during the development period. However, for many HUMS techniques, a plan for a controlled introduction to service may be necessary to fully validate the credit.

(ii) There must be provisions in the certification process to instruct the continued airworthiness effort to ensure compliance with the aforementioned plan.

(iii) During the implementation of this plan, data is accumulated by operational aircraft, and from this data, refinements and adjustments to the original criteria can be made. This period may also allow a proposed credit to be operated in parallel with alternative or standard procedures when it is necessary to gain additional in-service validation by way of back-to-back comparison

(iv) The plan should include procedures and provisions for this controlled period and should include clear goals by which progress and ultimately termination of this phase can be measured. The plan may include a multi-credit HUMS that will require a phased introduction of credits.

(5) Continued Airworthiness and Synthesis of Credit. Normal and established procedures will prevail for HUMS as for all other continued airworthiness matters. Arrangements should be made to validate the performance of an approved credit throughout its service use. Provisions should be made to allow for the synthesis of the service experience with relevant engineering evidence from rejected components, development testing, seeded testing, etc. Any necessary or desired modifications to the HUMS application or the component/equipment being monitored must be re-evaluated.

h. Instructions for Continued Airworthiness (ICA) and Other Requirements for Health and Usage Monitoring System (HUMS). This section addresses the ICA, operator's HUMS program, HUMS training, and Master Minimum Equipment List (MMEL) revision to incorporate HUMS.

(1) Instructions for Continued Airworthiness. The applicant for HUMS is required to provide ICA developed in accordance with FAR/JAR Part 29 and Appendix A. This section provides supplemental guidance with addressing aspects unique to HUMS. The applicant may be an airframe manufacturer, HUMS equipment manufacturer, or an operator. The ICA should address HUMS integration with the aircraft. This section addresses both airborne and ground based systems and equipment.

(i) HUMS ICA Items. The applicant must address the following subjects in addition to FAR/JAR Part 29. These subjects should address both airborne and ground based systems and equipment unless specifically indicated otherwise.

(A) Control and operating instructions must be provided for each element of HUMS, and where applicable, include data acquisition, transfer processing, display, configuration management, and resulting actions.

(B) Acceptance and rejection criteria and associated actions must be defined.

(C) A procedure is required when the system becomes inoperative because data is missing.

(D) When required, there must be a procedure for collecting and transferring HUMS data when the aircraft is away from the main HUMS data processing base.

(E) Provide a procedure for independent verification as defined in paragraph f(3), if applicable.

(F) Provide a procedure for implementing mitigating actions, when mitigating actions are applied.

(G) Provide a procedure for implementing controlled introduction to service instructions as defined in paragraph g(4), if applicable.

(H) Provide a training program on HUMS airborne and ground based systems and equipment.

(I) The airworthiness limitation section must be amended to address the following, if required:

(1) Requirements for independent verification and associated procedures.

(2) Requirements for mitigation actions and associated procedures.

(3) Requirements for controlled introduction to service and associated procedures.

(ii) Ground Based System and Equipment. A procedure must be defined to ensure the security of the ground-based system and equipment and the integrity of the HUMS data.

(2) Owner/Operator's HUMS Program.

(i) General. An owner/operator that installs and utilizes health and usage monitoring equipment on aircraft and intends to request maintenance credit will need a program. This program and revision to existing maintenance and/or inspection programs must be submitted to the aviation Authority for approval. This is due to the fact that maintenance credit may change existing maintenance inspection, overhaul requirements, and/or life limits.

(ii) HUMS Program Items. Regardless of the size and complexity of the health and usage monitoring equipment, the HUMS program must contain the following:

(A) A system must be provided for tracking the HUMS monitored component/system, including identification of component/system, recording requirement, tracking procedure, and other related activities.

(B) A system to assure that a maintenance credit must be maintained. The historical HUMS data must be traceable when such components/assemblies are transferred between aircraft.

(C) A procedure for new or overhauled HUMS monitored components.

(D) A procedure to address inoperative HUMS in accordance with paragraph j(1)(i)(C).

(E) A means for implementing procedures specified in paragraph j(1)(ii).

(F) A procedure for adjusting maintenance credits.

(G) An organization with clearly defined responsibilities to collect, analyze, and act upon the HUMS data.

(H) A procedure for implementing the training program specified in paragraph j(1)(i)(H).

(I) Where appropriate, a procedure for implementing the controlled introduction to service plan. See section g(4), Controlled Introduction to Service.

(iii) Ground Based System and Equipment.

(A) A procedure for troubleshooting and testing of the HUMS.

(B) A procedure for revising and using the operator's Minimum Equipment List (MEL) for HUMS.

(iv) Master Minimum Equipment List (MMEL)/Minimum Equipment List.

(A) The MMEL may need to be revised to include the HUMS equipment. Once the MMEL contains the HUMS equipment, the operator can revise their MEL to include HUMS and submit the MEL to the aviation Authority for approval.

(B) The aviation Authority should coordinate with engineering in evaluating the revised MEL.

NOTE: Any MMEL allowance should be determined considering the criticality of the 'credit' effect resulting from the HUMS application(s). MMEL allowances should be substantiated based on a Functional Hazard Assessment (FHA).

CHAPTER 3
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

MISCELLANEOUS GUIDANCE (MG)

AC 29 MG 16. CERTIFICATION PROCEDURE FOR ROTORCRAFT NIGHT VISION IMAGING SYSTEMS (NVIS) EQUIPMENT.

a. Background.

(1) The decreased ability of a pilot to see and avoid obstructions at night has been a subject of discussion since aviators first attempted to operate at night. Continuing technological improvements have advanced the capability and reliability of Night Vision Imaging Systems to the point that they are receiving increasing scrutiny, are generally accepted by the public, and are viewed by many as a tool for night flight.

(2) Simply stated, Night Vision Imaging Systems provide enhanced visual cues for night visual flight rules (VFR) flight. For the purpose of this advisory section, only image intensification systems are being considered. Currently, such systems consist of a set of night vision goggles and associated aircraft and lighting modifications. These sub-system elements are interdependent and, as technology advances, the characteristics associated with each element are expected to evolve. The complete description and performance standards of the night vision goggles and cockpit lighting modifications appropriate to civil aviation are contained in the Minimum Operational Performance Standards for Integrated Night Vision Imaging System Equipment, RTCA DO-275.

NOTE: Numerous references are made throughout this section to RTCA DO-275, Minimum Operational Performance Standards for Integrated Night Vision Imaging System Equipment, which provides a method of compliance for NVIS certification. This document may not be all-inclusive for NVIS operations in Joint Aviation Authorities (JAA) countries; therefore, other equivalent documents may apply.

(3) An increasing interest on the part of civil operators to conduct night operations has brought a corresponding increased level of interest in employing night vision imaging systems. However, night vision imaging systems do have performance limitations. Therefore, it is incumbent on the operator to employ proper training methods and operational procedures to minimize these limitations to ensure safe operations. In turn, operators employing night vision imaging systems must have the guidance and support of their regulatory agency/authority in order to safely train and operate with these systems. The purpose of this document is not to provide detailed operational guidance; however, the installer and certification authorities must understand the operational considerations to ensure a design that minimizes the effects of the limitations.

NOTE: Forward looking infrared (FLIR) devices, synthetic vision devices and aircraft head-up displays (HUDs) can also provide imagery and information useful during night operations. However, the scope of this document is limited to NVIS.

NOTE: Aircraft that are intended for NVG operations need to go through this NVIS certification process, whether modified or produced by the manufacturer. A night VFR approval does not constitute NVIS certification. In addition, some rotorcraft may not be suitable for NVIS certification (due to factors such as insufficient field of view, poor handling qualities, etc).

b. Explanation

(1) Operational Considerations. The fundamental purpose of using NVIS is to provide the operator the means of acquiring enhanced visual information when flying VFR at night. However, NVIS will not provide the user with visual cues equal to that observed during daylight. The following critical elements are the underlying assumptions in the operational considerations for NVIS (reference RTCA DO-268, Concept of Operations for Night Vision Imaging Systems for Civil Operations or equivalent EUROCAE document):

(i) Aircraft internal and external lighting is compatible in that it does not adversely affect the night vision goggles (NVG) image.

(ii) Incompatible light, especially inside the aircraft, can significantly degrade the NVG image with corresponding loss of external cues.

(iii) The NVIS has been properly maintained in accordance with the minimum operational performance standards or instructions for continued airworthiness (ICA).

(iv) A proper preflight has been performed on the NVIS confirming operation in accordance with the continued airworthiness standards and training guidelines.

(v) The pilot can maintain VFR flight unaided in the event that NVG imagery is lost or degraded.

(vi) Viewing imagery provided by a NVG will degrade one's depth perception and distance estimation.

(vii) The NVG does not provide adequate imagery under all lighting, scene contrast, and atmospheric conditions.

(viii) It is possible to "see through" areas of light moisture when using NVGs, thus increasing the risk of inadvertently entering instrument meteorological conditions (IMC). However, the NVGs are not designed to be used for IFR.

(2) NVIS Limitations/Characteristics.

(i) There are certain limitations inherent in the current NVG design. The following are some examples of NVG limitations that should be considered: Visual Acuity, Field Of View, Field of Regard, NVG Weight and Center of Gravity, Monochromatic Image, Amount and Type of Illuminance, Fatigue, Spatial Orientation, Depth Perception and Distance Estimation, and Ergonomics.

(ii) However, despite these characteristics, NVIS operations can be performed safely with proper equipment, operational procedures and training. System design and testing should consider these factors when certifying an NVIS. Due to the fundamental effect the NVGs have on visual perception and the inherent characteristics of NVIS technology, these type of modifications should always be considered major.

c. Definitions. (Reference DO-275, Section 1.0 for a more complete list of definitions.)

(1) Night Vision Goggle (NVG). A head-mounted, lightweight, self-contained binocular system consisting of two independent monocular assemblies.

(2) Night Vision Imaging System (NVIS). The integration of all elements (including the NVG, windshield, lighting system, etc) required to successfully and safely operate an aircraft with the aid of NVGs.

NOTE: The evaluation to determine acceptance of a NVIS should include both the integration of the elements (components) and the specific regulatory requirements for the component(s), i.e. cockpit, windshield, lighting, etc.

(3) Image Intensifier. An electro-optic device used to detect and intensify optical images in the visible and near infrared region of the electromagnetic spectrum for the purpose of providing visible images. It is composed of the photocathode, microchannel plate, and phosphor screen. This component performs the intensification process in an NVG. It does not include the objective and eyepiece lenses.

(4) NVIS Lighting Component. Any component that emits or transmits light within the flight deck or other crew compartments, or that is attached to the aircraft exterior, and does not degrade NVG performance.

(5) NVIS Lighting System. An aircraft lighting system that has been modified or designed to incorporate NVIS lighting components. It provides adequate illumination, under day and night conditions, of instruments/displays/controls for the unaided eye without degrading NVG performance.

(6) Windshield. A transparent screen in front of the pilot(s) of an aircraft or vehicle usually made of polymers, plastics, or glass.

(7) Transparency. Any transparent screen in front of occupants in an aircraft or vehicle (e.g., side windows, chin bubbles, etc.).

(8) NVG Compatible. Lighting components or systems that comply with applicable standards and do not adversely affect NVG performance.

(9) Aided. Flight with NVGs in an operational position.

(10) Unaided. Flight without NVGs or with the NVGs in a non-operational position.

(11) Crewmember. A person assigned to perform duties in an aircraft during flight using NVIS.

d. Procedures.

(1) General. This certification guideline has been prepared as an aid in the evaluation of a specific rotorcraft configuration for NVIS operations. The criteria presented are not to be considered all-inclusive, but are offered as one method of evaluating design and performance. The testing and qualification of an NVIS installation should be considered as consisting of three phases: design and development (plans, FHA's, component tests, etc.), ground test, and flight test. The amount of testing necessary during each phase will vary with the amount of testing performed on previous phases. Also, a thorough ground-testing program should result in a successful flight test, thus reducing the need for extra flights. NVIS certification should be substantiated by following the recommended standards of RTCA DO-275.

(2) Certification Plan. A Certification Plan describes how the applicant proposes to plan, manage, and document their product certification. The plan should clearly state whether the approval is intended for Category A or B design. The following subjects are not intended to provide a complete list of the items that should be included in the certification plan, but rather highlight some of the areas that should receive consideration:

(i) System Description.

(A) A comprehensive system description (to include cabin modifications) that includes a brief summary of the design as it relates to existing displays, switches, annunciator lights, control panels, electrical components, interior arrangement, other interfaces, etc.

(B) A layout and description of any changes to the flight instrument panels, lighting controls, and accessory panels.

(ii) Project Schedule. A detailed project schedule that identifies all major milestones and schedules for any required deliverables (i.e., test plans). Schedules for ground and flight evaluations for certification and operational approval should be coordinated with consideration for night illumination levels.

(iii) Certification Basis and Means of Compliance. A certification matrix (compliance checklist) that identifies the applicable regulations, AC's, current policies, certification basis, and the procedures or methods that will be used to comply with those requirements. Any testing or analyses applicable to the project that have been previously approved by the FAA/AUTHORITY under an STC, TC, or TSO should be provided. The approval date, letter reference number, and references as to how the specific approval was granted (i.e., STC, TSO, letter of approval, etc.) should be included in this section. See AC 29 MG 16 Paragraph d6(i) for a detailed explanation of certification basis.

(iv) Communication and Coordination. Identification of all delegated functions, which should include any stipulations, coordination, and limitations that are placed upon those delegations. Identify all qualified and approved personnel who are performing specific tests related to the NVIS certification.

(v) Human Factors Plan. This section should address human factors and ergonomic issues and provide human factors support for decisions regarding the crewmember interface and/or coordination issues resulting from the NVIS operation and installation. This should include any special/safety considerations or emergency procedures required by the NVIS installation.

(vi) Test Plan. This section should contain the requirements for the planning, preparation, and conduct of all certification testing, including any delegations. Proposed ground and flight test plans should be coordinated with the certifying authority.

(vii) Conformity Plan. This section should describe the activities associated with conformity of parts and aircraft installations.

(viii) Continued Airworthiness Plan. This section should provide the instructions for continued airworthiness for the NVIS equipment and installation. A clear procedure should be identified to ensure that any subsequent modification or maintenance to the aircraft (e.g., light emitting or reflecting device, transparencies, and avionics equipment) is shown to be in full compliance with the NVIS certification.

(ix) Compliance Documentation. This section should describe the procedures and documents for submittal and processing of compliance documentation.

(3) References.

(i) Regulatory references. Sections 29.1, 29.21, 29.141c, 29.561, 29.771, 29.773, 29.777, 29.779, 29.785, 29.803, 29.811, 29.812, 29.853, 29.1301, 29.1303, 29.1305, 29.1307, 29.1309, 29.1321, 29.1322, 29.1331a(3), 29.1333, 29.1351, 29.1355, 29.1357, 29.1359, 29.1381, 29.1383, 29.1385, 29.1387, 29.1389, 29.1391, 29.1393, 29.1395, 29.1397, 29.1399, 29.1401, 29.1413, 29.1501, 29.1523, 29.1525, 29.1529, 29.1541, 29.1543, 29.1545, 29.1549, 29.1553, 29.1555, 29.1557, 29.1561, 29.1581, 29.1583, and 29.1585.

(ii) Other references. RTCA DO-268, Concept of Operations, Night Vision Imaging Systems for Civil Operators; RTCA DO-275, Minimum Operational Performance Standards for the Integrated Night Vision Imaging System Equipment; SAE ARP 4754, Certification Considerations For Highly-Integrated or Complex Aircraft Systems; SAE ARP 4761, Guidelines and Methods For Conducting the System Safety Assessment Process on Civil Airborne Systems and Equipment; TSO-C4C, Bank and Pitch Indicators; TSO-C8D, Vertical Speed Indicators; TSO-C67, Radar Altimeters; AC 25-11, Electronic Displays; AC 20-74, External Lighting; and AC 20-88, Instrument Markings and Limits.

(4) System Design. Systems or equipment presented for installation approval should meet the appropriate description in section 1.0 in RTCA DO-275. The following are the typical NVIS equipment that may be submitted for approval.

(i) NVGs. The Night Vision Goggles consist of the following components: binocular assembly, monocular assembly, objective lens assembly, image intensifier, eyepiece lens, power source assembly, and helmet or head mount.

(ii) NVIS Lighting System. See definitions in c(4) and c(5).

(iii) Transparencies. See definitions in c(6) and c(7)

(iv) Additional NVIS equipment and components. Due to limitations of NVGs, some additional instruments or controls could be necessary to the crewstation design to accomplish a successful NVG operation. Some examples of additional information (to a night VFR certified only aircraft) are attitude indicator(s), vertical speed indicator(s), radar (radio) altimeter(s), generator or alternator of adequate capacity, and two-way radio communication equipment. For a better understanding for the need for these instruments, refer to RTCA DO-268.

(5) System Performance. Each system component is required to perform its intended function. System performance standards for the NVGs and NVIS lighting components can be found in sections 2.0 and 3.0 respectively in RTCA DO-275 In addition, any components installed should be qualified for aviation use by TSO or other approval means. Otherwise, performance data will need to be submitted to the FAA/AUTHORITY. In addition, the following data should be submitted.

(i) Environment. An appropriate means for environmental testing is set forth in RTCA document DO-160 or that recommended in DO-275. The applicant should submit test reports showing that the laboratory-tested categories such as temperature, vibration, altitude, etc., are compatible with the environmental envelope of the rotorcraft.

(ii) System Safety Assessment.

(A) The applicant should perform a System Safety Assessment (SSA), in accordance with the guidance provided for paragraph AC 29.1309, to establish the hazards associated with the proposed installation. Additionally, the Society of Automotive Engineers (SAE) Aerospace Recommended Practice (ARP) 4761 (Guidelines and Methods for Conducting the Safety Assessment Process on Civil Airborne Systems and Equipment) provides guidance for development of the SSA.

(B) The NVIS criticality can be determined by performing a Functional Hazard Assessment (FHA) and the integrity level of the system is required to be equivalent to the assessed criticality. Compliance with the criticality level established by the FHA should then be demonstrated during the certification process. The FHA is a top-down analysis (should involve pilots, engineers, etc.) that starts with the potential hazards to the rotorcraft and traces these hazards to the system, sub-systems and component level, and the areas affected by the NVIS. This type of analysis starts with the determination of what undesirable effects can occur as a direct or indirect result of NVIS failure, both annunciated and non-annunciated, during anticipated operational scenarios. The criticality of the failures can be mitigated through aircraft design, system redundancy, or crew composition.

(C) Since NVGs provide enhanced visual cues during night VFR operations, the complete loss of the NVG image or the display of hazardously misleading information from the NVG is classified as severe-major/hazardous. The complete loss of the NVIS lighting system has been classified as major. Uncommanded actuation of NVG incompatible light sources that adversely affect the NVG image may be classified as severe-major/hazardous, depending on the type of NVIS operation and location of light source. Due to the limitations of NVIS, additional equipment should be required and its loss should be analyzed through the SSA.

(iii) NVG Mechanical Assembly. The mechanical assemblies should be qualified for the environmental conditions intended for operation. The NVG design should consider automatic breakaway, image stability, crash-worthiness, weight, center of gravity, single-hand adjustments, removal, and operations in-flight. The NVG design must comply with § 29.561.

NOTE: Consideration of helmet/NVG mount integration should be included. The NVG mount and helmet should have the following desirable characteristics under normal and emergency conditions: stable, individual fitted, secure battery pack, and appropriate

chin and nape straps. A description of the helmet/NVG mount integration and how it relates to crash survivability is highly recommended.

(iv) NVG Power Source. The design of the power supply should be commensurate with the failure condition/classification stated in the Functional Hazard Assessment. If action is required to regain a backup or secondary source of power regarding complete loss of NVG image, then an indicator or annunciator should be provided to allow sufficient time for corrective action prior to complete loss and/or degradation of the NVG image. (Reference RTCA DO-275, Section 2.2.2.6)

(6) Installation Design. (In addition, refer to section 4.0 in RTCA DO-275)

(i) Certification Basis. The use of NVIS is a new type of operation, which require special considerations. The same aircraft design can be certified for other types of operation, such as Category A, Day VFR, Night VFR, and IFR. The certification basis (reference § 29.1) of the aircraft must be appropriate for all certified types of operation. The NVIS design must comply with the aircraft certification basis, and may require adoption of later amendment levels.

NOTE: NVIS operations in conjunction with Category A takeoff and landings are beyond the scope of this section. Other special flying operations, such as agriculturalor external loads will require special considerations.

(ii) Mechanical Installation. Aircraft modifications should be made to (1) ensure compliance with the airworthiness regulations and (2) comply with the equipment manufacturer's recommendations. The designer should observe good engineering practices, in specifying material type, wire routing, fastener type, light switches and controls, edge distance, and ergonomic considerations (§§ 29.561, 29.777, 29.1307, and 29.1359).

(iii) Arrangement and Visibility. The mounting position of all instruments, switches, position labels, and controls should be visible to the pilot while wearing NVGs and during all cockpit lighting conditions (day and night) likely to occur. TSO approval does not assure instruments will be acceptable in a particular cockpit installation or for all lighting conditions. The instruments, switches, emergency signs or exits, and placards should be free from reflections. Warning, caution and advisory annunciation devices should be conspicuous, NVIS compatible, and clearly visible to the pilot (day and night). (See RTCA DO-275, Advisory Circular 20-69, and §§ 29.771, 29.773, 29.777, 29.811, 29.812, 29.853, 29.1301, 29.1321, 29.1322, 29.1381, 29.1413, 29.1555, 29.1557 and 29.1559.)

(iv) Interior Lighting. Internal lighting systems should not adversely affect the performance of the NVIS. Compliance of the instrument lighting and illuminated markings with §§ 29.811, 29.812, 29.1381 and AC 20-88 should be demonstrated during aided and unaided operations.

(v) Exterior Lighting. External lighting systems should not adversely affect the performance of the NVIS. Compliance with §§ 29.1383 to 29.1401 and AC 20-74 should be demonstrated. External lighting should be evaluated under probable terrain, illumination, and environmental conditions (e.g., rain, snow, fog, dust, etc.).

(vi) Reflections and Glare. There should be no objectionable glare or reflections that interfere with NVG aided and unaided operations. Sources of reflection and glare could include aircraft interior surfaces (floor, panels, etc), equipment, seats, etc. Compliance with §§ 29.771, 29.773 and 29.1381 should be demonstrated in the ground and flight test.

(vii) Accessibility and Workload. The NVIS configuration should not compromise the ability to perform normal and non-normal duties. Scanning and control accessibility should be considered. The applicant should determine the representative flight profiles and environmental conditions under which workload should be evaluated. The purpose of this evaluation is to ensure that the workload associated with NVIS operations does not compromise safety of flight. Compliance with §§ 29.771, 29.773, 29.777, and 29.1523 should be demonstrated.

NVIS configuration should also consider inadvertent IMC and unusual attitude recovery. Inadvertent IMC is defined for these purposes as loss of visual references, such as brownout, whiteout, etc. It is important to assess the transition from aided flight to instrument flight. Therefore, the applicant should plan this evaluation during all requested phases of flight (such as cruise flight or takeoff/landing, if requested). The rotorcraft characteristics need to be such that workload is acceptable and the required instruments are within the pilot's primary scan.

(viii) Controls.

(A) The NVIS installation should include consideration of the detrimental effects of the inadvertent selection of incompatible light sources and the design should minimize the probability of such occurrence. In addition, the lighting should have a dimming range consistent with aided and unaided operations. Compliance with §§ 29.777, 29.1301, 29.1309, and 29.1381 should be demonstrated.

(B) Some installations incorporate master light switches to control special busses for the lighting systems. If this capability is provided, it should be evaluated to assure failure modes are not introduced that will result in the illumination of incompatible light sources or loss of all required lighting. It should be noted that one switch that controls all required (primary and backup) lighting may not be considered acceptable (§ 29.1307).

(ix) Power Sources. It should be determined whether the electrical power source capacity is adequate for the system installation under all foreseeable operating conditions including engine failure. Duplicate systems should be powered from

separate buses and, in some cases, from independent sources if required by the airworthiness regulations (§§ 29.1309, 29.1331, 29.1333, 29.1351, or 29.1355).

(x) Cooling Test. Systems and components (e.g., addition of filters to light sources) that are added or modified for NVIS operations should be verified with AC 29.1309 (cooling test procedures) for continued acceptability.

(xi) Interface and Integration. It is important to check the NVIS lighting for readability, uniformity, balance, light leakage, reflections, and degradation of NVG performance. Reference RTCA DO-275, Section 4.0, for an expanded explanation.

(7) Ground and Flight Tests. The purpose of FAA/AUTHORITY certification tests is to verify that the rotorcraft meets the certification and operational requirements for both aided and unaided operations. This will include an assessment of crewstation(s) to ensure that the crew can effectively use all systems for both aided and unaided operations. These assessments may result in changes to the established limitations, or operating procedures of a rotorcraft (reference §§ 29.1581, 29.1583, and 29.1585). These ground and flight tests should be conducted by appropriate flight test personnel.

Prior to requesting FAA/AUTHORITY certification flight tests, the applicant should submit the appropriate data, drawings, bench tests, component tests, environmental qualifications, installation configurations, and test equipment (including calibration) that were used to demonstrate compliance to the appropriate certifying office. This will also include any non-installed equipment that may be required for the NVIS operation (e.g., helmets, head mounts, NVG mounting brackets, etc.). Once submitted, the FAA/AUTHORITY will review this package along with the certification plan to ensure compliance has been met. After package acceptance, the applicant should coordinate final approval for the test plan and test schedule. The test plan and schedule should include all proposed tests to show compliance. One method of demonstrating compliance is described in RTCA DO-275, Section 4.0, and Appendices. Upon test plan acceptance, prior to commencing any certification tests (ground or flight), a conformity inspection should be performed. Once the conformity is performed, any changes from the conformed configuration (e.g., removing filters or changing bulbs in light sources) may, at the certification authority's discretion, require another conformity inspection.

The applicant is responsible for providing the appropriate pre-test requirements and informing the FAA/AUTHORITY certification office of the following:

- List of test equipment to perform test (tri-bar charts, illumination source, illumination measuring device, NVG test set, or NVG eye lane).
- Test facilities (type of darkened location).
- Flight area (cultural lighting, terrain, obstacles, landing areas).
- Weather (moon cycle, anticipated climatic conditions, etc.).
- Additional training (novel or new technology briefing).

- Flight equipment (helmets, mounts, NVGs, flashlights/torches, etc.).

(i) Ground Test Procedures. To optimize the test program and minimize flight test risks, ground testing should be accomplished prior to flight test. Flight-testing should only be commenced when ground testing is satisfactorily completed. It is highly recommended that the schedule allow sufficient time between ground and flight test to resolve deficiencies discovered during ground testing. A detailed test procedure and test sequence is provided in RTCA DO-275, Section 4.4.1, and Appendices. As a minimum, the following should be completed:

(A) Aircraft (Internal and External) Equipment Assessment.

(B) NVG Assessment/Verification.

(C) Daylight Readability for NVIS lighting modification.

(D) Crew Station Ergonomics (e.g., controls, displays, NVG, egress, etc.).

(E) Night Readability for NVIS lighting modification.

(F) NVG Visual Acuity for NVIS lighting modification (to include transparencies).

(G) Light Leakage/Reflections.

(H) Failure Modes Evaluation.

(I) Electro-Magnetic Interference Effects.

This evaluation should be conducted from all crewstations intended to be used (including cabin) for the NVIS operation.

NOTE: During ground tests, the applicant should ensure that the appropriate voltage is provided to represent flight conditions.

(ii) Flight Test Procedures. Prior to commencing the flight test evaluation, a daylight familiarization flight of the flight test area should be performed. Any specific flight regimes, maneuvers, terrain, or landing areas that will be used for NVIS evaluation should be familiarized for hazards to flight.

NOTE: If determined from the ground test that the normal configuration of the aircraft is required for the flight evaluation, it may not be possible to install dual controls. In this case, appropriate training/authorization may be required for the FAA/AUTHORITY engineering test pilot to conduct the in-flight evaluation. In addition, special test equipment may be installed during the evaluation to enhance flight test safety. If

additional personnel are required for the evaluation, the appropriate crewstations will be installed.

The flight test profile should consist of maneuvers representative of those performed during normal or special operations over terrain and cultural areas in various illumination and weather conditions. This test should include evaluation under low illumination conditions (e.g., no moon, and overcast sky, with little cultural lighting). A detailed test procedure and test sequence is provided in RTCA DO-275, Section 4.4.2. As a minimum, the following should be completed:

(A) Aircraft Equipment Assessment.

(B) Normal Operating Procedures Assessment.

(C) NVG Visual Performance (e.g., terrain, cultural lighting, etc.).

(D) NVG Functionality (adjustment, single-handed operation, removal).

(E) Daylight Readability for NVIS Lighting Modification.

` (F) Crew Station Ergonomics (e.g., controls, displays, NVG, maneuvers, etc.).

(G) Night Readability for NVIS Lighting Modification.

(H) NVIS Lighting Compatibility.

(I) Light Leakage/Reflections.

(J) Failure Modes and Emergency Procedures (Aircraft, NVG and NVIS equipment) Evaluation.

(K) Unusual Attitude Recovery and Inadvertent IMC Assessment

(L) Electro-Magnetic Interference Effects.

(M) External Lighting (position lights, landing lights, anti-collision lights, etc.).

NOTE: After flight test, identify any operational limitations that should be imposed on NVIS operations and coordinate with appropriate operational authorities.

(8) Rotorcraft Flight Manual (or Supplement).

(i) Limitations. The following instructions should be included in the Limitations Section of all RFM/RFMS:

(A) The following statement should be added: "NVIS operations are approved, and must be conducted under the appropriate night VFR regulations".

(B) Define minimum crew for NVIS operations. If minimum crew is more than single pilot, then the crew composition and appropriate crewstation should be defined. Coordinate with appropriate operational authority for proper crew qualification.

(C) If additional personnel or operational limitations are appropriate to a specific design, then they should be included (e.g., minimum NVG height en route using NVGs, additional personnel required for takeoff and landings, phases of flight restricted from NVG use, etc).

(D) State limitations related to operating environment (snow, cultural lighting, terrain, etc.).

(E) NVG remote area (does not qualify as an airport, heliport, or vertiport) requirements (lights, flares, etc.) should be defined if required. Reference RTCA DO-268, Section 4.5.1.3.2 for more information.

(F) State which crewstation(s) are approved for NVIS operations.

(G) State which Special Operations are approved (e.g., Category A, hoisting, etc.).

(H) Define equipment that is incompatible with NVIS operations and associated procedure (e.g., select off or disable).

(I) The following NOTE should be included in the Limitations Section of the RFM/RFMS:

NOTE: NVIS Installation approval does not constitute an approval for NVG operations; such approval requires coordination with the appropriate operations regulatory authority.

(ii) Required Equipment for NVG Operation (Per RTCA DO-268 and RTCA DO-275). As a minimum, the following equipment is required for NVIS operations. Additional equipment may be defined during certification.

(A) NVIS Lighting System.

(B) Compatible NVG (specify type, to include class and approval basis).

(C) Helmet/Head Harness with NVG Mount.

(D) Radar Altimeter.

(E) Communications and navigation equipment necessary for the successful completion of an inadvertent IMC procedure in the intended operational area.

(F) Attitude Indicator or equivalent.

(G) Gyroscopic Direction Indicator or equivalent.

(H) Vertical Speed Indicator or equivalent.

(I) A statement if required, "NVG Operator's Manual (specify), must be available to the pilot during NVG operations".

(J) External light source(s) required for NVIS operations (Define and Specify).

(K) If a flashlight/torch is an operational requirement, then it must be NVIS compatible.

(L) Additional equipment/instruments required for NVIS operations (Define and Specify).

(iii) Emergency and Abnormal Procedures. The following instructions should be included in the Emergency and Abnormal Procedures:

(A) NVG Malfunction/Failure.

(B) NVIS Lighting Malfunction/Failure.

(C) Basic Aircraft Emergencies (include any changes to aircraft procedures).

(D) Failure of Additional NVIS Equipment.

(E) Inadvertent IMC Procedures.

(iv) Normal Procedures. The following instructions should be included in the Normal Procedures Section:

(A) Preflight.

•Check Windshield/Transparencies for suitability (scratches, cleanliness, etc.).

•Check NVIS lighting for light leakage and compatibility.

•NVG adjustment and alignment.

•Check function of additional NVIS equipment.

•Interior Configuration check for NVIS equipment (e.g., deselect incompatible light sources).

•Exterior Configuration check for NVIS equipment (e.g., ensure exterior lights are in accordance with type design definition).

(B) In-flight.

•Transition to aided from unaided operations (and vice versa).

•Description of radar altimeter or additional equipment procedures.

(C) Post Flight. Report Discrepancies (NVG, NVIS lighting and equipment, and windshield) and record information required for continued airworthiness.

(D) Special Procedures. Describe any unique procedures for all phases of flight if required.

(v) Weight and Balance. The basic weight and balance should include the installation of NVIS equipment.

(vi) Additional Information in RFM(s). It is recommended that a sufficiently detailed system description be provided.

(9) Instructions for Continued Airworthiness.

(i) The Instructions for Continued Airworthiness (ICA) should contain the information necessary for carrying out ongoing maintenance and inspections on NVIS equipment installed in the rotorcraft. Refer to Paragraph AC 29.1529, AC 29 Appendix A, and RTCA DO-275, Section 5.0, for detailed instructions and procedures. As a minimum, the following should be included:

(A) Appliance, System or Accessory Maintenance Manual, or Section.

(B) Maintenance Instructions and Inspection Requirements.

(C) Airworthiness Limitations.

(D) Illustrated Parts Breakdown.

(ii) The ICA's should cover (as a minimum) aircraft transparencies, NVIS lighting, NVG, and any additional aircraft equipment that support NVIS operations.

NOTE: It is recommended to have a storage location/compartment on the aircraft to protect the continued airworthiness of the NVG. In addition, it is prohibited to store the NVG in a location that could cause damage to aircraft or component, hinder crashworthiness, or result in loss of intended function of a component.

(10) Post-TC/STC Approved Modifications. Any subsequent aircraft modifications (internal and external), including operational equipment (FLIR, EMS equipment, etc.) involving a light emitting or reflecting device should be re-assessed against the original requirements for the NVIS certification.

(11) Multiple TC/STC Approvals. If the approval is granted for multiple aircraft, installation/production procedures should be provided to ensure all aircraft comply with the type design. This information should be provided in the initial certification data package, so that the FAA/AUTHORITY is well informed for multiple approvals. The production procedures should be sufficiently detailed to detect minor differences between the different aircraft. It is expected that post-production tests will include some or all of the ground and flight tests described above for each aircraft.