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Aircraft Brake Temperature Monitor Systems (BTMS)

RATIONALE

AS1145B has been reaffirmed to comply with the SAE five-year review policy.

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1. SCOPE:

This specification covers minimum requirements for brake temperature monitoring equipment whenever used on any type and model of civil aircraft. It shall be the responsibility of the purchaser to determine the compatibility of these requirements with the application aircraft and to specify requirements in excess of these minimums as necessary.

1.1 Purpose:

To specify minimum requirements for brake temperature monitoring systems for civil aircraft applications.

2. APPLICABLE DOCUMENTS:

Materials, processes, and standard products shall be selected in accordance with specifications or standards from the sources indicated below in the order of precedence shown. The issue in effect on date of invitation for bids shall apply.

(a) Industry Specifications and Standards as listed in indexes published by recognized industrial associations including but not limited to the following, without order of precedence:

(1) Aerospace Material Specifications (AMS), Aerospace Standards (AS), Aerospace Recommended Practices (ARP), and Aerospace Information Reports (AIR) published by Society of Automotive Engineers.

(2) National Aircraft Standards (NAS) as published by the National Aircraft Standards Committee of the Aerospace Industries Association.

(b) Federal specifications and standards as listed in the index of Federal Specifications, Standards, and Handbooks published by the General Service Administration.

(c) Military Specifications and Standards as listed in the Department of Defense Index of Specifications and Standards.

The following documents are listed as a minimum and are considered the more important of the reference list given below. The documents that apply shall be of the issue in effect at the time of invitation for bids (including change notices) and shall apply to the extent specified herein.

RTCA/DO-160	Environmental Conditions and Test Procedures for Airborne Equipment
MIL-E-5400	Electronic Equipment, Airborne, General Specification for.
MIL-C-83723	Connector Electrical, Circular, Environmental Resisting, Receptacle and Plug, General Specification for
MIL-STD-810	Environmental Test Methods
MIL-STD-461	Electromagnetic Interference Characteristics, Requirements for Equipment
MIL-STD-130	Identification Marking of U.S. Military Property
MIL-STD-462	Electromagnetic Interference Characteristics, Measurements for
MIL-STD-883	Test Methods and Procedures for Micro-electronics
MIL-STD-701	List of Standard Semiconductor Devices

3. GENERAL REQUIREMENTS:

3.1 Function:

The temperature monitor system covered by this specification shall provide a means of measuring brake temperature and conveying information based on this measurement to the pilot and/or flight engineer. Three areas of information are important.

(a) Very high temperature, indicating overheated brake conditions exist which may cause the fuse plugs to blow, or necessitate a tire, wheel, and brake inspection at the ramp. This condition may be caused by severe usage or excessive drag.

(b) Moderate temperature, indicating that brakes contain energy levels which do not leave sufficient capacity to perform an R.T.O. Under this condition, take-off should not be attempted. This condition can exist due to normal braking or slightly dragging brake.

(c) Temperature Display. In the event that the pilot would desire to evaluate the brake thermal condition, the monitored temperature of each individual brake should be made available. This is most important if the indications for (a) or (b) are provided as warning indicator lamps. The display shall indicate over the range to which the sensor is exposed or as specified by the purchaser. The display shall indicate in degrees Celsius.

3.1.1 Design Objectives:

(a) Provide a high temperature indication when the temperature of any brake exceeds a predetermined safety threshold or reaches a temperature that may cause deterioration to the metallic chassis (as may be the case with a carbon heat sink brake). This temperature condition, if required by purchaser, can be displayed by a continuous lamp illumination or memory device indicating brake position and being resetable by ground/flight personnel. The temperature probe should be placed as close as possible to the heat source to provide minimum delay between the peak temperature of the heat sink and the peak temperature measured by the heat sensor element.

(b) Provide an indication when, due to high temperature, the energy capacity of any one brake is below RTO requirements. It is recommended that it be supplied as a lamp indication.

3.1.1 (Continued):

(c) The monitored temperature of the heat sink should be made available to the pilot and/or flight engineer at all times. The display can be of the needle indicator or of the digital readout type. If, at the purchasers option, it is desired to increase the brake maintenance utilization of the BTMS, consideration should be given to include the capability of recording/displaying the highest brake temperature obtained during a stop with a resetable device. There are three configurations for displays that can be utilized for the BTMS requirements.

(1) Single Display:

In this configuration only one readout is provided. As used normally, this readout continuously displays the temperature of the hottest brake. However, when utilizing the single display configuration, some provision should be made to select and display the temperature of each individual brake.

(2) Multiple Brake Display:

A display can be provided for more than one brake. For example, one display can be provided for each group on a strut. The readout should also display the temperature of the hottest brake in its group. Provisions should also be made to select and display the temperature of any brake of the group.

(3) One Display for Each Brake:

This configuration is the least desirable of the three configurations. It is not recommended that this system be used on aircrafts with more than 4 brakes. Normally the highest temperature is all that is required. If the lowest temperature is also desired, the technique described in Design Objective (d) is recommended.

(d) As an optional feature a delta temperature (ΔT) capability may be provided. The ΔT lamp indicates that one brake temperature differs either higher or lower from the average brake temperature by an excessive amount. Illumination of this lamp will indicate a brake problem. A dragging brake will be hotter than the other brakes. A brake that is defective and not providing its share of braking will be colder than the other brakes. Care must be exercised in selecting the brake groups for ΔT, so that the lamp does not become a problem by giving false indications. Use of the brakes for steering could cause one side of the aircraft to have brakes much hotter than the other side, giving a false ΔT indication.

(e) For aircraft that include monitoring equipment for maintenance purposes, an output from the overheat circuit shall be provided. This output shall be disabled during “BITE” test.

3.1.1 (Continued):

(f) Provide a total system accuracy of ±25 °C (±45 °F) at an ambient temperature of 25 °C (77 °F). The system accuracy shall be within ±50 °C (±90 °F) over the operating temperature -40 °C to +71 °C (-40 °F to 160 °F).

(g) The system shall be designed so that any shorts or opens in the sensor or sensor wiring will result in a zero brake temperature reading.

3.1.1.1 Self Test: System test provisions shall be included as an integral part of the BTMS design. Tests may be automatic or manual. Test operation or failure shall not reduce the system accuracy or response. The test shall, as a minimum, simulate a hot brake so that over temperature circuitry will also be evaluated as well as the analog or digital display. The test shall include evaluation of all brake channels and shall be capable of detecting sensor failures.

Capability shall be included which will isolate a failure to a wheel channel.

3.2 Material and Workmanship:

3.2.1 Materials: Materials shall be consistent with good commercial usage and suitable for the purpose intended.

3.2.1.1 Metals: All metals used in the construction of the BTMS shall be corrosion resistant unless suitably plated or treated to resist corrosion during stocking and normal service life.

3.2.1.2 Connectors: Connectors shall be selected in accordance with MIL-C-83723. The size of the connector shall be selected such that the number of pins will provide a minimum of 10% spares or 1 pin whichever is greater.

3.2.1.3 Electronic Components: All semiconductor microcircuits shall be screened to MIL-STD-883 Level B. All discrete semiconductors shall be selected where possible from MIL-STD-701. All semiconductors shall be hermetically sealed and no plastic encapsulated semiconductors shall be used.

3.2.2 Workmanship: Workmanship and finish shall be in accordance with high grade aircraft accessory manufacturing practice.

3.3 Identification:

All major components shall be plainly and permanently marked for identification in accordance with Standard MIL-STD-130 or per requirements of the purchaser.

3.3.1 Units too small to carry the complete identification on the part shall be suitably tagged or stored in suitable containers. The unit, such as the sensor, shall carry as a minimum the name of the manufacturer, part number, and serial number.

3.4 Design and Construction:

Shall conform to the requirements specified herein.

3.5 System Components:

The brake temperature monitor system consists of the following major components.

(a) Brake mounted sensor(s).

(b) Reference junction (if applicable) or compensator.

(c) Indicator unit, including meter(s) (or lights), required amplifiers, bridge, networks, level detectors, or other elements. Located for use by pilot and/or flight engineer.

3.6 Electrical Operation:

Electrically operated components of the brake temperature monitor system shall operate satisfactorily on a voltage and power specified by the purchaser. Where a choice is given, an effort should be made to standardize by utilizing 28V DC aircraft supply.

3.7 Sensor:

It shall be a design goal to replace, or service, the sensor assembly with minimum difficulty. If the sensor cable or tube is not permanently attached to the brake, the sensor assembly shall be replaceable without wheel or brake removal. Wiring to sensors shall be as continuous as possible.

The sensors shall be capable of 50 actual removal/installation cycles on the associated aircraft wheel/brake unit when installed on the aircraft without sustaining cracks or damage to the sensor assembly.

3.8 Operating Conditions:

3.8.1 Non-Operating Temperatures: All system components shall be designed and constructed to withstand, as a minimum, ambient temperatures from -54 °C to 71 °C (-65 °F to 160 °F), or the operating temperatures, whichever are more severe.

3.8.2 Operating Temperatures: The system components shall be designed and constructed for operation at the following temperatures:

(a) Indicator unit shall be designed to operate over temperature extremes of -40 °C to 71 °C (-40 °F to 160 °F).

(b) Reference junction or compensator (unless part of the indicator assembly) shall be designed to operate within the temperature zone dictated by its location. These temperatures shall be supplied by the purchaser but not less than -54 °C to 71 °C (-65 °F to 160 °F).

(c) Sensor assembly shall be designed and constructed to operate over a temperature range consistent with its location within the brake. Temperature exposure for each part of the assembly to be specified by the purchaser. As a minimum the sensor shall be capable of operating in the temperature range of -54 °C to 1093 °C (-65 °F to 2000 °F).

3.8.3 Other Environmental Conditions: Sensor reference junction or compensator, cables and connectors shall remain functional after exposure to hydraulic or aircraft cleaning fluids and lubricants, aircraft fuels as well as the environmental requirements listed in paragraph 4.3. All conduits which protect cable runs on landing gear shall be environmentally sealed or provided with drain holes at their lowest point.

3.9 Interchangeability:

All major components containing the same part number shall be directly interchangeable with no adjustment.

4. QUALIFICATION:

4.1 Sensor:

4.1.1 Visual Inspection: Inspect for good workmanship, absence of sharp edges obvious defects and for compliance with Sections 3.2 and 3.3.

4.1.2 Accuracy: The output of the sensor unit including the cold junction monitoring circuit (if a thermocouple is used) shall not vary from sensor to sensor more than ±5 °C (±9 °F) from 0 °C to 500 °C (32 °F to 932 °F).

The plot of the sensor output versus temperature between 0 and 500 °C (0 and 932 °F) shall not vary from a straight line curve more than ±5 °C (±9 °F) below the overheat threshold temperature and not more than ±20 °C (±36 °F) for temperatures above the overheat threshold temperature. The overheat threshold temperature shall be used as the mid point calibration temperature for the indicator unit electronics.

4.1.3 Environmental Tests:

4.1.3.1 Humidity Test: Test in accordance with Method 507, Procedure I, MIL-STD-810. Dry connector area with air blast and test per 4.1.2 and 4.1.6 within one hour.

4.1.3.2 Fungus Test: This test shall be per Method 508 of MIL-STD-810. Unit need not operate during test. Test per 4.1.6 after test is completed. Vendor supplied affidavits that all materials used in components contain no nutrients to fungus are acceptable in lieu of fungus test.

4.1.3.3 Salt Spray Test: Test per Method 509, Procedure I, MIL-STD-810, except that the duration of test shall not be less than 24 hours. Rinse connector end with tap water and dry with air blast. Test per 4.1.2 and 4.1.6 within one hour. Inspect and test per 4.1.2 and 4.1.6 again in 48 hours.

4.1.3.4 Shock Test: Test in accordance with Method 516, Procedure I, of MIL-STD-810. Use 60g peak acceleration with a time duration of 11 ± 1 milliseconds. Pulse shape per 516-1. Test per 4.1.2 and 4.1.6 within one hour.

4.1.3.5 Thermal Shock Test: Stabilize the sensor contact temperature at ambient temperature. Increase contact temperature to the value specified by the purchaser or 760 °C (1400 °F) in not more than 40 seconds. Hold for not less than 5 minutes. Cool to ambient temperature within 2 minutes with an air blast. Repeat 4.1.3.5 for a minimum of 1,000 cycles and then test per 4.1.2 and 4.1.6.

4.1.4 Temperature Cycling: Cycle those parts of the sensor assembly exposed to the brake heat sink from 93.3 °C (200 °F) to 537.8 °C (1000 °F) and back for 1000 cycles and then test per 4.1.2 and 4.1.6.

4.1.5 Vibration Test (See Note Figure 1): Mount the sensor in a suitable bracket to simulate brake service installation.

4.1.5.1 It is difficult to set a minimum vibration requirement for the sensor since the vibration it is exposed to depends on the aircraft and brake characteristics, the location of the sensor, and the method of mounting. Because these conditions cause the vibration requirements to vary substantially, it is recommended that each application be analyzed for the frequencies and G forces seen by the sensor for that task. However, as an absolute minimum the following specification is recommended:

(a) The Vibration Test shall be performed at two temperatures, the first at mid-range (approximately 350 °C or 662 °F) and at high range (approximately 500 °C or 932 °F).

(b) The unit shall be vibrated with a frequency varying from 5 to 2000 Hertz with an amplitude corresponding to Figure 1 50g curve with the exception that at lower frequencies, the test shall be performed with no less than 2g's.

4.1.5.2 Resonance: The unit shall be vibrated plus or minus 50g minimum for 15 minutes at the three most significant resonant points noted during 4.1.5.1. The unit shall also be vibrated at the brake resonant point (as determined from dynamometer tests). Test per 4.1.2 and 4.1.6.

4.1.6 Response Time: The sensor contact temperature shall be increased from ambient to 315.6 °C (600 °F) within 10 seconds. The sensor shall read 90% of its steady state value within 60 seconds.

4.1.7 Hydraulic Fluid Compatibility: Unit shall be submerged in the specified fluid for 7 days and then observed 24 hours later. No visible damage shall be noted. Test per 4.1.2 and 4.1.6.

4.2 Indicator Unit:

The indicator unit will consist of electronic components, meters, lights, etc., necessary to monitor the sensor temperature.

4.2.1 Visual Inspection: Inspect product for compliance with 3.2.1, 3.2.2, and 3.3.

4.2.2 Accuracy: The indicator unit shall indicate in response to a simulated sensor temperature of 0 °C (32 °F), 350 °C (662 °F) and 500 °C (932 °F) for temperatures of indicator unit of 20 °C (68 °F) and 43.3 °C (110 °F) within the tolerance of ±15 °C (±27 °F). This tolerance assumes ±10 °C (±18 °F) meter movement and ±5 °C (±9 °F) for the electronic circuitry.

4.2.3 Environmental Tests:

4.2.3.1 Humidity Test: Test per Method 507 Procedure I MIL-STD-810. Dry with air blast and test for accuracy per 4.2.2.

4.2.3.2 Shock Test: Test in accordance with Method 516, Procedure I, MIL-STD-810. Use 5g peak acceleration with a duration of 11 ± 1 millisecond pulse shape per 516-1.

4.2.3.3 Temperature Tests: Test at high temperature 71 °C (160 °F) per Method 501 Procedure I, MIL-STD-810. Test at low temperature per Method 502, Procedure I, MIL-STD-810.

4.2.3.4 Vibration Test: The indicator assembly shall be mounted in a suitable bracket to simulate service installation.

4.2.3.4.1 Cycling: The indicator assembly shall be vibrated with a frequency varying from 5 to 500 Hz and shall be accomplished in approximately 15 minutes with at least 2 minutes in the region below 25 Hz. The test shall continue for 3 hours in each of the three perpendicular axes. Note resonant frequencies of the unit but disregard fixture resonant frequencies.

The following schedule should be followed (See Figure 1):

5 to 10 Hertz: 0.20 inch (5.08 mm) DA
10 to 18 Hertz: ±1g
18 to 26 Hertz: 0.06 inch (1.52 mm) DA
26 to 500 Hertz: ±2g

4.2.3.4.2 Resonance: The unit shall be vibrated for 15 minutes at each resonant point noted in 4.2.3.4.1. The amplitude should not exceed the 2g line in Figure 1. Limit the test to the four most significant resonant points. At the conclusion of 4.2.3.4.1 and 4.2.3.4.2, test for 4.2.2.

4.2.3.5 Electromagnetic Interference (EMI) Tests: The EMI specifications for the BTMS equipment are so uniquely related to the various aircraft design and wire routing that the EMI requirements must be written to fit the particular installation. For this reason it is recommended that specifications MIL-STD-461 and MIL-STD-462 be used as a guide tailored to the purchaser's requirements.

If the aircraft EMI requirements are not defined, it is recommended that RTCA specification D-160, Environmental Conditions and Test Procedures for Airborne Electronic/Electrical Equipment and Instruments, be used as the guide for EMI design and testing.

4.2.3.6 Life Cycle Tests: The indicator assembly display and panel controls shall be cycled for 40,000 duty cycles without sustaining any malfunctions or failures.

4.3 Reference Junction or Compensator:

A reference junction will be required for thermocouple type heat sensors. It may be located on the bogie, strut, or other portion of the aircraft not environmentally controlled. This junction may also be included within the indicator unit.

4.3.1 Located in or Near Indicator Unit: When located within the indicator unit or designed to be located in the same environment as the indicator unit, complete all tests listed under 4.2.

4.3.2 Located Below Gear Shock Absorber: Complete tests listed below.

4.3.2.1 Visual Inspection: Inspect product for compliance with 3.2.1, 3.2.2, and 3.3.

4.3.2.2 Accuracy Test: Subject a precalibrated sensor contact to a constant temperature of 537.8 °C (1000 °F); with the indicator unit at ambient temperature, vary the temperature of the reference junction or compensator from -54 °C to 135 °C (-65 °F to 275 °F) or as specified. The indicator shall show the sensor temperature within ±22.2 °C (±40 °F).