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Memorandum

Subject: **<u>INFORMATION</u>**: Policy Statement; Installation of Electronic Engine Control for Reciprocating Engine; PS-ACE100-2004-10024

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1. Summary

This policy statement identifies appropriate certification requirements for installation of an Electronic Engine Control (EEC) into a small airplane with a reciprocating engine. It includes guidance related to methods of compliance as well as potential Equivalent Level of Safety findings (ELOS) and special conditions.

The EEC must be approved for use as part of the certificated engine intended for installation per 14 CFR part 33 to be approved for installation in a 14 CFR part 23 airplane. The EEC part 33 approval can be through a Supplemental Type Certificate (STC) or Amended Type Certificate (ATC) process, and this certification may occur with the 14 CFR part 23 certification. This policy statement addresses the certification requirements to install an EEC that has been approved for use on a part 33 engine into a part 23 airplane.

Installation of an EEC into part 23 airplanes may include design features not envisioned when 14 CFR part 23 was created. This policy highlights areas where special conditions may be appropriate for these installations. However, appropriate special conditions for each installation must be determined on a case-by-case basis under 14 CFR part 21, §§ 21.16 and 21.17, and 14 CFR part 11.

Installing an EEC in a small certificated airplane design is not considered a design change so substantial that it would require a new airplane Type Certificate (TC) under 14 CFR part 21, § 21.19. Therefore, it is considered appropriate to install an approved EEC into a certificated airplane using the STC or ATC process.

Proposed EEC installations, whether supplemental, amended, or new TC projects, will be considered significant as defined in Order 8100.5, paragraph 103j.

Note: Federal Aviation Administration (FAA) Order 8100.5 has been cancelled and Order 8100.5A has been issued. However, the definition of “significant” has been omitted in Order 8100.5A. Therefore, the reference to Order 8100.5 is necessary to define “significant.” See Figure 1 for the definition of “significant” from Order 8100.5.

j. The term "significant," when used to describe a type or supplemental type certification project, for the purpose of initial project notification to an accountable directorate, means:

(1) Any new type certificate application.

(2) Any application for amended type certificate or new/amended supplemental type certificate in which:

(a) The design appears to require special conditions, exemptions, or equivalent safety findings or a certification basis derived from an unusual application of FAR 21.101 (a)(2) or 21.101(b).

(b) The design uses novel or unusual methods of construction.

(c) The design changes the kinematics, dynamics, or configuration of either the flight control or rotordrive system.

(d) The design change would substantially alter the aircraft's flight characteristics. Note: Flight items for which compliance can be qualitatively recorded and/or documented are considered to be nonsignificant for project description purposes. However, the accountable directorate may apt to participate in any aspect of flight testing.

(e) The design affects an area that has been the subject of a major service difficulty or accident.

(f) The design changes the engine configuration from reciprocating to turbopropeller or turbojet powered or changes from one engine or propeller model to a completely different engine or propeller model (does not include dash number changes).

(g) The integrity of the basic load bearing structure necessary for continued safe flight and landing or operation of the aircraft within approved limits is affected.

(h) The design consists of new state-of-the-art systems or components which have not been previously certificated or for which adequate certification criteria have not been published.

(i) The certification is likely to be controversial or highly visible.

Figure 1. FAA Order 8100.5, Paragraph 103j

Given the significance of the change, early program coordination between the Small Airplane Directorate Standards Office and the Aircraft Certification Office (ACO) is necessary. The ACO is expected to notify the Small Airplane Directorate Standards Office of such projects promptly and forward certification project notifications and associated certification plans as soon as practical after project application. The ACO will identify the technological areas of concern identified in this policy statement, as well as any additional concerns, and develop a G-1 issue paper to establish the certification basis. Signature authority for certificate issuance on these projects is retained by the Small Airplane Directorate Standards Office and will be re-delegated on a case-by-case basis as this new technology is understood and integrated into aerospace products.

2. Discussion of Significant Issues

a. General

New technology for aircraft piston engines is being developed to enable the use of a Full Authority Digital Engine Control (FADEC) or EEC. Such systems are not unique to aircraft piston engines and are increasingly common on turbine engines and in the automotive world. The FADEC or EEC controls any combination of engine control subsystems such as fuel delivery, ignition, turbocharger boost, or propeller speed. This policy statement covers installing piston engine EECs.

b. Nomenclature

EECs will be used to describe engine control systems that are software driven, regardless of whether they are full authority or supervisory over a mechanical system.

The following is a list of definitions applicable to EECs and their installation:

(1) Airplane Flight Manual (AFM). The document that contains the relevant airplane information about limitations, operating procedures, performance and systems information as specified in 14 CFR part 23, §§ 23.1581 through 23.1589.

(2) Aircraft-Supplied Data. Aircraft-supplied data is information that is generated in the aircraft systems and used by the engine control system, but whose source is not controlled under the design authority of the engine certification applicant. This does not include input from those sensors that are used by, and normally dedicated to, the engine control system but may be mounted in the airframe.

(3) Aircraft-Supplied Power. Aircraft-supplied power is any electrical power source that is an integral part of the aircraft electrical system whose primary function is to power aircraft systems (for example, an electrical bus).

(4) Alternate Control Mode(s). For this policy, an alternate control mode is characterized by engine control operating characteristics or capabilities that are sufficiently different from the "normal mode." Also, this mode may significantly affect the operating characteristics or capabilities of the aircraft, crew workload, or anything that constitutes appropriate crew procedures.

(5) Backup System. A backup system is a different type of system that is used as a standby or alternate control mode to the primary or normal control mode or system.

(6) Commercial and Industrial Grade Electronic Parts. Commercial and industrial grade electronic parts are not manufactured to military standards.

(7) Degraded System or Configuration. A degraded system or configuration is an EEC system that has faults or failures present that have no effect on engine operation but may cause a loss of redundancy in the system.

(8) Destructive Event. A destructive event is any malfunction or failure of the EEC system that could result in a condition in which the engine, or any of its components, causes physical damage to the airplane structure or its occupants that could potentially impact continued safe operation of the airplane. (Examples: engine separation from its mounts, uncontrolled fire, inability to shutdown the engine, generation of toxic products, or destructive engine explosion.)

(9) Electronic Engine Control (EEC) System. The EEC system consists of systems or subsystems, with full or limited-authority functional capability, which have the following characteristics:

(a) Modulation of one or more engine control parameters;

(b) Logic that discriminates the desired condition of an engine control parameter; and

(c) Electronic implementation of logic.

(10) Equivalent Level Of Safety (ELOS). ELOS findings will be granted when literal compliance with a certification regulation cannot be shown and compensating factors exist which can be shown to provide an equivalent level of safety (see 14 CFR part 21, § 21.21(b)(1)).

(11) Failure Condition. A failure condition is an abnormal or non-operating state of the engine that is caused by or contributed to by one or more faults or failures.

(12) Failure Event. See definition for "failure condition."

(13) Fault or Failure. A fault or failure is an occurrence that affects the operation of an engine component, part, or element so that it cannot function as intended (this includes both loss of function and malfunction). Software errors may cause failures, but are not considered faults.

(14) Fault or Failure Accommodation. This term refers to the capability of the control system or flight crew to mitigate, either entirely or in part, the fault or failure.

(15) Fault or Failure Detection. This term refers to the discovery of a fault or failure or the resulting condition.

(16) Fireproof. The capability of a material or component to withstand, as well as or better than steel, a 2000 °F flame (±150 °F) for 15 minutes minimum, while still fulfilling its design purpose. The term "fireproof," as used here, refers to materials and parts used to confine fires within designated fire zones. It means that the material or part will perform this function under conditions likely to occur in such zones and will withstand a 2000 °F flame (±150 °F) for 15 minutes minimum.

(17) Fire Resistant. The term "fire resistant," as used here, applies to powerplant installations such as fluid carrying lines, flammable fluid system components, wiring, air ducts, fittings and powerplant controls. In this case, "fire resistant" means the capability of a material or component to perform its intended functions under the heat and other conditions likely to occur at the particular location and to withstand a 2000 °F flame (±150 °F) for 5 minutes minimum. (Example: A fire resistant hose that will withstand a 2000 °F flame for 5 minutes.)

(18) Full Authority Digital Engine Control (FADEC). FADEC is an engine control system in which the primary functions are provided using digital electronics and the EEC unit has full-range authority over the engine power.

(19) Full-up System or Configuration. The full-up system or configuration is an EEC system that has no faults or failures present, detected or undetected, that affect the control of engine power, engine protection systems, indication of critical engine operating parameters or other safety features of the engine control system.

(20) Local Events. Local events are failures of aircraft systems and components, other than the EEC system, that may affect the installed environment of the EEC.

(21) Loss of Power Control (LOPC). LOPC is an EEC failure event resulting in the inability to attain at least 85 percent of rated power, or the inability to modulate power at all normal operating conditions.

(22) Minor Power Loss (MPL). MPL is an EEC failure event resulting in the ability to attain greater than 85 percent, but less than 95 percent, of rated power.

(23) Primary Mode. This is the mode of operation intended to control the engine. This is often referred to as the "normal mode."

(24) Programmed Logic Device. A programmed logic device is a custom micro-coded component, such as application specific integrated circuits (ASIC) or programmable logic devices (PLDs).

(25) Single-Fault Tolerant. Single-fault tolerant refers to the capability of the EEC system design architecture to accommodate the occurrence of any single EEC system fault to prevent an unsafe condition.

(26) Time-Limited Operations (TLO). TLO refers to the duration of flight operations permitted with the EEC system in a degraded condition.

(27) Unacceptable Change in Power. For this policy, an unacceptable change in power is defined as an LOPC.

(28) Uncovered Fault. An uncovered fault is a fault or failure for which either a detection mechanism does not exist or, if there is a detection mechanism, an accommodation for the fault does not exist.

(29) Unsafe Condition. For this policy, an unsafe condition refers to any malfunction or failure of the EEC system that could result in a destructive event or EEC system LOPC and MPL rates above those mentioned in AC 33.28-2, Tables 5-1 and 5-2. This definition is intended to apply to certification activities for reciprocating engine EEC systems and is not applicable to continued airworthiness activities related to these engines and systems.

c. Installation Manual

As part of the engine type certification effort, 14 CFR part 33, § 33.5, requires that an instruction manual for installation and operation of the engine be prepared and approved before issuance of the engine TC. The requirement for an instruction manual for installation and operation of the engine is required regardless of the method of engine and EEC certification (new, amended, or supplemental type certificate). 14 CFR part 23, § 23.901, requires that engine installations comply with these instructions. There may be aircraft specific requirements that are not addressed, or are not properly addressed, in the EEC installation instructions. Should such situations occur, the installer is required to coordinate with the EEC manufacturer and obtain approval for any deviations from the supplied installation instructions. The coordinated and approved deviations will then be part of the installation instructions for the aircraft. The applicable regulations identified in this policy statement (including potential special conditions), combined with the engine installation instructions, should provide a comprehensive list of installation requirements for most EEC installations. In addition, the methods of compliance and potential special conditions described in this policy statement should help applicants and ACO engineers when assessing methods to demonstrate compliance to applicable regulations.

Advisory Circular (AC) 33.28-2, “Guidance Material for 14 CFR 33.28, Reciprocating Engine, Electrical and Electronic Engine Control Systems,” provides additional guidance on the physical and functional integration of the EEC into an airplane electronic control system. Specifically, Chapter 8 should be reviewed as it contains guidance regarding EEC integration into an airplane.

The following is an excerpt of AC 33.28-2, “Guidance Material for 14 CFR 33.28, Reciprocating Engine, Electrical and Electronic Engine Control Systems,” Chapter 8. The following excerpt specifically identifies the need for an additional interface document supplied by the EEC manufacturer describing interface requirements and certification requirements for these integrated systems.

CHAPTER 8. INTEGRATION OF ENGINE, PROPELLER, AND AIRCRAFT SYSTEMS

8-1. EEC System Integration Certification Plan (SICP).

There must be a clear definition of the respective certification tasks of the various applicants: engine, propeller, and aircraft manufacturers, with the associated engine, propeller, and aircraft certification authorities. This should be documented in an EEC SICP, submitted by the applicant for engine certification. The plan should be included as an appendix to the instructions for installation and should include the following:

a. Distribution of Compliance Tasks. The tasks for the certification of the aircraft propulsion system equipped with electronic controls may be shared between the engine, propeller, and aircraft manufacturers. The distribution of these tasks between the manufacturers should be identified and agreed on by the appropriate engine, propeller, and aircraft authorities. The EEC SICP should summarize the engine applicant’s responsibilities for these certification tasks. The plan should list each task related to the EEC system certification and define those for which the engine applicant is responsible and those for which the aircraft or propeller applicant is responsible. The plan should address all analyses and tests required for EEC system certification.

b. Interface Definition and Other Data. The EEC SICP should include interface definitions and other data for the functional, hardware, and software aspects that have been integrated between the engine, propeller, and aircraft systems. The plan should describe integration aspects or provide cross-references to the instructions for installation for the following items:

(1) Functional requirements;
(2) Fault accommodation strategies;
(3) Maintenance strategies;
(4) Software quality level (per function if necessary);
(5) The reliability objectives for:
(a) LOPC and MPL events; and
(b) Transmission of faulty parameters.
(6) The environmental requirements, including the degree of protection against lightning or other electromagnetic effects (for example, the level of induced voltages that can be supported at the interfaces);
(7) Engine, propeller, and aircraft interface data and characteristics; and
(8) Aircraft electrical power supply requirements and characteristics (if relevant)

c. Design Change Control. The EEC SICP should describe the design change control system established to support post-certification activity. This system should ensure that changes to any control element that is integrated into the EEC system are evaluated by all design approval holders of that integrated system.

d. Electrical Power Requirements

Engines equipped with an EEC that relies on aircraft electrical power and has no mechanical backup system, will need a separate source of backup electrical power that is independent of the primary source. The separate source of electrical power can take one to the following forms:

(1) Backup Battery: This system is the simplest installation for an aircraft certified with a single battery and alternator electrical system. The backup battery would be connected to the EEC in a manner such that the failure of the aircraft primary electrical system would not affect the backup battery. Using a backup battery requires a method of determining the backup battery charge state in order to meet the minimum backup power requirements.

(2) Dual Electrical System: On aircraft with a dual battery, dual alternator/generator system with independent primary electrical busses, power from each of the electrical busses can be used when the failure of one electrical system is isolated from the other system.

(3) Backup Alternator/Generator System: This system differs from the Dual Electrical System because the backup alternator/generator is not used as a primary source of aircraft electrical power. The backup alternator/generator is used, as an alternate source of electrical power should the primary system fail. These systems are usually attached to an essential buss and offer reduced current capability when compared to the primary system.

The minimum requirements for the electrical power system are addressed in §§ 23.1309, 23.1351, 23.1353, 23.1357, 23.1359, 23.1361, 23.1365 and 23.1367. The backup power does not need to be wholly dedicated to the EEC since other critical systems may be supplied by the backup electrical power. However, following the loss of the primary power generation system, a minimum of 60 minutes of backup electrical power for the EEC is highly recommended.

3. Regulatory Review

The following is the criteria for a special condition as defined in § 21.16:

"If the Administrator finds that the airworthiness regulations of this subchapter do not contain adequate or appropriate safety standards for an aircraft, aircraft engine, or propeller because of a novel or unusual design feature of the aircraft, aircraft engine or propeller, he prescribes special conditions and amendments thereto for the product. The special conditions are issued in accordance with

Part 11 of this chapter and contain such safety standards for the aircraft, aircraft engine or propeller as the Administrator finds necessary to establish a level of safety equivalent to that established in the regulations."

The policy statement refers to special conditions because of the new and novel features of EECs. The existing regulations did not envision the use of electronic engine controls; therefore, the existing regulations do not directly address the certification requirements necessary to achieve the appropriate safety standards equivalent to those currently used by aircraft propulsion systems.

The table in Appendix 1 is taken from 14 CFR part 23 with an emphasis on the sections that require more consideration when incorporating an EEC for aircraft use. There are eleven potential special conditions identified in the text and in the regulations matrix that are included in Appendix 1. The sections noted with an asterisk (*) highlight requirements that may be applicable by special condition or may have more requirements applied by special condition. See the appropriate text paragraph describing the potential criteria for the special condition.

Appendix 2 contains an example of a special condition that has been applied to EEC installations.

4. Policy – Section Discussions

a. Part 21

(1) Primary Category

An aircraft engine EEC design not approved under part 33 may not be approved as part of an airframe TC under a primary category approval. The intent of primary category is a simple aircraft. The applicant may install an EEC equipped engine certified under part 33 in a primary category airplane.

(2) Airships

The EEC policy is applicable for airship projects.

(3) § 21.113, Requirement of supplemental type certificate

The EEC must be approved for use on a certificated engine per 14 CFR part 33. The approved EEC can then be installed in a small airplane. The applicant may install a certificated aircraft engine EEC into a certificated airplane using either the STC or ATC process. The applicant may not install a certificated aircraft engine EEC into a certificated airplane through a field approval.

All proposed EEC installations, whether supplemental, amended, or new TC projects, will be considered significant as defined in Order 8100.5, paragraph

103j. The design requirements appropriate to the aircraft engine EEC and its systems justify this rationale.

b. Part 23

(1) § 23.1, Applicability

The applicant will need to apply for a project number for an EEC installation by providing a certification plan.

(2) Subpart B – Flight

(a) § 23.33, Propeller speed and pitch limits

If the EEC controls propeller speed or pitch, incorporating an EEC would require complying with the applicable parts of this section. The failure modes of the EEC must also address any aspects of compliance with this section, such as propeller governing or overspeed protection, to ensure that no unsafe condition would be present.

Depending on the method of propeller control, an Issue Paper and an ELOS may be required to show compliance with § 23.33(d)(2). The most likely scenario would be a propeller governor that is integral to the EEC basic operation and was unable to be disabled without compromising the EEC basic operation. The applicant has to request an ELOS if the EEC control of the propeller uses a different means of controlling the engine overspeed rather than traditional methods such as reduced throttle settings.

(b) Subpart B – Performance, Flight Characteristics, Controllability and Maneuverability, Trim, Stability, Stalls, Spinning, Ground and Water Handling Characteristics and Miscellaneous Flight Requirements

The EEC may have operating modes in which the performance of the aircraft will be affected due to a change in available engine power. If the EEC has a negative effect on the aircraft performance, then the applicable Subpart B sections have to be evaluated. A five percent effect on the engine power available has been the allowable limit where additional aircraft performance testing was not required as long as the changes were not detrimental to the aircraft performance. An increase in available engine power will not be detrimental to a single engine aircraft. However, it may have a negative effect on the minimum controllable airspeed for a multi-engine aircraft. For example: If the EEC affects the idle thrust or the ability to achieve “zero thrust” due to minimum idle speeds, or mitigation for light polar moment of inertia propellers, the applicant will need to evaluate the effects on the stall speeds and mark the Airplane Flight Manual (AFM) accordingly.

(3) Subpart C -- Structure

(a) §§ 23.301, 23.303, 23.305, 23.307, 23.321

Basic compliance with the regulations is acceptable.

(4) Subpart D -- Design and Construction

(a) §§ 23.601, 23.603, 23.605, 23.607, 23.609, 23.611, § 23.613

Basic compliance with the regulations is acceptable.

(b) §§ 23.777, 23.779, 23.781

If either a single power lever or single power control is used, the applicant will need to request an ELOS using AC 23-17A guidance. The EEC may provide some or all of the combined control function. If this is the case, the EEC is to be identified and included within either an ELOS or special condition, as needed.

(c) § 23.867, Electrical bonding and protection against lightning and static electricity

Bonding, lightning protection, and static electricity are fundamental aspects of an EEC installation and must be addressed. If the EEC manufacturer provides installation instructions and approved substantiation data showing compliance with the requirements of § 23.867, then the applicant need not show additional compliance. Demonstration of compliance with this section is required regardless of the type of intended operation: Visual Flight Rule (VFR) or Instrument Flight Rules (IFR), day or night.

(5) Subpart E -- Powerplant

(a) § 23.901, Installation

The EEC is a component of the engine installation under § 23.901(a). The EEC functions may only be dependent on engine components or sensors that were certified under part 33, or the EEC may require interface with additional components or sensors that need to be certificated as part of the powerplant system under part 23. The EEC installation certification requirements will vary according to the complexity of the EEC installation.

As part of the engine type certification effort, 14 CFR part 33, § 33.5, requires that an instruction manual for installation and operation of the engine be prepared and approved before issuance of the engine TC. 14 CFR part 23, § 23.901, requires that engine installations comply with these instructions. The installation instructions must include the interface requirements, with the applicable aircraft systems, and any fault strategies necessary. One example would be EEC reliance on an aircraft Air Data Computer (ADC) for engine power setting. The interface between the EEC and the ADC needs to be defined to include required data, data transmission rates, and fault accommodations.

(b) § 23.903, Engines

In addition to basic compliance with this regulation, the EEC installation must comply with the engine isolation requirements of § 23.903(c). Operation for five minutes with either an engine or nacelle fire is addressed during the engine part 33 certification where it is substantiated that the EEC does not take unwanted action that could be hazardous to the aircraft. The results from the engine part 33 certification should be used for compliance with this regulation.

If the engine affected by fire is not shut down within five minutes, it will be required to have a benign shutdown that poses no hazard to the engine or the aircraft, while the remaining engine(s), if any, continue to operate normally. The benign shutdown refers to the failure modes of the EEC as the fire progresses. The requirement for a benign shutdown does not mean, in the event of a fire, that the EEC will automatically shut down the engine.

The restart envelope required by § 23.903(f) must be re-established, since the EEC may impact the restart capability of the engine.

The part 23 installation may require a software level that is higher than that certificated to date in part 33. AC 23.1309-1C shows the software requirements for the various classes of aircraft. The minimum software certification level for an EEC is DO178B level C. However, aircraft defined as Class III or higher will require DO178B level B software. If the EEC equipped engine is certified under 14 CFR part 33 with DO178B level C software, then a higher software certification level would be required to install this engine into a 14 CFR part 23 aircraft in the Class III category.

See the paragraph on § 23.1309 and AC 23.1309-1C for further clarification.

(c) § 23.904, Automatic power reserve [APR] system

The EEC will probably be a component of the APR system if such a system is available. The applicant will need to describe the operation of the APR system, as well as the triggering mechanisms.

(d) § 23.907, Propeller vibration

The EEC installation may affect the propeller vibration. An EEC may include control of fuel mixture and ignition timing. Both are known to affect the energy input to the crankshaft of a reciprocating engine. As a result, propeller vibration stress levels are also affected; therefore, evaluation of propeller vibration is required for a reciprocating engine EEC installation. The applicant is required to show that the EEC installation will not result in propeller vibration or strain levels that exceed the propeller manufacturer's limits.

You may use a previously tested and approved EEC equipped engine and propeller installation as substantiation for compliance to this regulation, if the same combination is being used for the new installation.

(e) § 23.909, Turbocharger systems

For EEC systems that include turbocharger control functions that were evaluated during the part 33 engine certification, only the interface issues need to be addressed during the powerplant installation certification. Issues to address include the following: fault accommodation, annunciation, and guidance to the pilot in the event of a failure to minimize exceeding the limits (either overboost or overspeed) in failure conditions.

For turbocharger systems that were certified as part of the airframe, per 14 CFR part 23, § 23.909, the EEC system must be evaluated for compatibility with the turbocharger system. The EEC turbocharger control, fault accommodations, fault annunciations and pilot operating procedures must be evaluated for normal and abnormal operating conditions. The evaluation of the EEC system is to ensure that the engine and aircraft are able to operate within the type certificated limits. The evaluation is also to ensure that any failure modes of the EEC do not result in a reduced margin of safety than previously demonstrated by the aircraft and engine.

See the paragraph on § 23.1309 and AC 23.1309-1C for additional guidance.

(f) § 23.933, Reversing systems

An EEC may include control of the reversing system. This would be an integration issue that should be addressed in the System Integration Certification Plan (SICP) described in paragraph 2.c. of this policy statement. The EEC control of the reversing system must preclude unwanted in-flight activation of the reversing system. The plan would need to include a description of the reversing system and identify all of the aircraft inputs and any EEC fault accommodations.

(g) § 23.939, Powerplant operating characteristics

The intent of 14 CFR part 23, § 23.939 is to evaluate the aircraft installation effects on the engine. Additionally, the requirement is used to validate that the installed behavior of the engine is compatible with the aircraft operating envelope.

The use of programmed schedules within an EEC may result in what was traditionally a non-critical operational aspect becoming the critical and limiting operation for certification. EEC systems use pressure and temperature inputs, and the fuel scheduling is directly affected by altitude and temperature. Consequently, flight test validation of acceptable engine operation throughout the altitude, temperature, power, and airspeed envelope by the applicant is fundamental for EEC installation certification. The regulation is currently required for turbocharged engines. However, the regulation will be applicable to normally aspirated engines via special condition.

(h) § 23.955, Fuel flow

Basic compliance with the regulations is acceptable.

(i) § 23.991, Fuel pumps

If the EEC system uses dual electric fuel pumps, instead of an engine driven pump, then the applicant must request an ELOS for compliance with § 23.991(a)(1).

Engines equipped with dual electric fuel pumps will need backup electrical power as discussed in section 2.d, Backup Electrical Power. The minimum requirement for the electrical power system is addressed in §§ 23.1309, 23.1351, 23.1353, 23.1357, 23.1359, 23.1361, 23.1365 and 23.1367.

(j) § 23.993, Fuel system lines and fittings

Basic compliance with the regulations is acceptable.

(k) § 23.997, Fuel strainer or filter

Basic compliance with the regulations is acceptable.

(l) § 23.1027, Propeller feathering system

Basic compliance with the regulation is acceptable.

(m) § 23.1041, General

The EEC and its related components that have a maximum temperature limit defined must be evaluated for sufficient cooling once installed in the aircraft. The EEC control of the mixture on reciprocating engines will have an effect on the cooling performance. The applicant must identify any EEC mitigation strategies for engine cooling. One example would be the EEC's ability to use fuel mixture auto-enrichment on reciprocating engines for temperature control that would then affect the engine power available.

The applicant must complete cooling tests to show that the temperature limits for the engine, the EEC, and its associated hardware components are not exceeded during the required flight maneuvers.

(n) § 23.1043, Cooling tests

Ignition timing and the mixture setting affect the cylinder and exhaust gas temperatures. The EEC may control one or both of these parameters as well as other parameters or functions that affect engine cooling.

The applicant must complete cooling tests to show that the limit temperature(s), including the EEC and its associated hardware, are acceptable. The applicant must consider each EEC parameter or function that affects cooling when establishing the critical temperature.

Ignition timing and spark may be variable and may have default modes of operation. The applicant must perform cooling tests with the EEC in any mode that is dispatchable. If there are Time Limited Operation (TLO) modes that allow certain reduced capability of the EEC, then those TLO modes must be tested. The additional testing will not be required if the applicant can demonstrate that the TLO modes do not affect either a parameter or a function that affects cooling. The applicant may use the results from 14 CFR part 33 testing if those results are applicable to the 14 CFR part 23 requirements.

EEC fuel enrichment may be creditable for compliance with cooling testing. The applicant must substantiate the EEC mitigation strategies for engine cooling before certification testing for any credit on engine cooling test results.

An example of evaluating the mitigation strategy is shown below:

$\Delta CHT / \Delta W_{Fuel}$	=	$1^{O} F / \frac{lbm}{hr}$
Measured Fuel Flow	=	172 lbm/hr
Target Fuel Flow	=	155 lbm/hr (normal operation)
Maximum Fuel Flow	=	195 lbm/hr
Measured CHT	=	430 ^{o}F
Limit CHT	=	450 ^{o}F
Measured OAT	=	70 ^{o}F
Pressure Altitude	=	5000 ft
CHT Corrected	=	CHT + 1.0(100-0.0036(Palt)-OAT)
	=	430 + 1.0(100-0.0036(5000)-70)
	=	442 ^{o}F
Fuel Cooling Correction	=	$\left(W_{Fuel(measured)} - W_{Fuel(Target)}\right) * \Delta CHT / \Delta W_{Fuel}$
	=	(172-155)* $1^{O} F / \frac{lbm}{hr}$
	=	17 ^{o}F
Fuel Corrected CHT	=	CHT Corrected + Fuel Cooling Correction
	=	442 ^{o}F+17 ^{o}F
	=	459 ^{o}F
Max Fuel Cooling	=	$\left(W_{Fuel(maximum)} - W_{Fuel(Target)}\right) * \Delta CHT / \Delta W_{Fuel}$
	=	(195-155) * $1^{O} F / \frac{lbm}{hr}$
	=	40 ^{o}F
Max Fuel Cooling CHT	=	Fuel Corrected CHT - Max Fuel Cooling
	=	459 ^{o}F-40 ^{o}F
	=	419 ^{o}F
CHT Margin	=	CHT Limit - Max Fuel Cooling CHT
	=	450 ^{o}F-419 ^{o}F
	=	31 ^{o}F

(o) § 23.1047, Cooling test procedures for reciprocating engine powered airplanes

The EEC control of the mixture and ignition timing will have an effect on the cooling performance. The applicant must review the EEC functions to determine if they affect cooling and address these functions during cooling tests.

(p) § 23.1093, Induction system icing protection

The applicant must verify each EEC controlled component and EEC control mode affecting induction system icing protection/air temperature control for proper operation and support it with substantiating data. Either flight testing, static testing, or analysis, or all, may be required. The applicant will have to substantiate any method of compliance with engineering data.

See § 23.1157 for related requirements.

Section 23.1309 is applicable.

(q) § 23.1141, Powerplant controls: General

An EEC installation may require an ELOS since the means for controlling the engine may be different from what is specifically written in this and subsequent engine control sections.

The requirements of § 23.1141(f) are applicable to EEC installation, as the effects of an engine/nacelle fire will have a direct impact on the EEC function. Any EEC components and associated wiring on the engine side of the firewall that are necessary to prevent a destructive engine event must be fire resistant. The EEC system may not burn such that it contributes to the fire hazard. The EEC system was evaluated for fire resistance during the part 33 engine certification program; therefore, evaluation of the effects of fire should be limited to components that are not part of the certificated engine.

See § 23.903 for related requirements concerning benign shutdown in the event of engine/nacelle fire.

See § 23.1183 for related requirements.

Also refer to § 23.1305 and § 23.1309.

(r) § 23.1143, Engine controls

The use of a single power lever is envisioned with incorporation of an EEC. The EEC control interface must have at least a single power lever that controls engine power or thrust. EEC control of either super or turbo chargers is envisioned. The EEC control interface with the engine may be accomplished by a mechanical or electrical connection.

The applicant must account for the EEC effect on the engine controls. The EEC installation may introduce either single or other failure modes not present in the traditional throttle control system configuration. The EEC control interface with the engine may not be mechanical but rather an electrical connection. The EEC is required to comply with this regulation as it requires separate controls for each engine power or thrust control. However, an ELOS may be required to satisfy the requirements of § 23.1143(g).

An ELOS or special conditions may be warranted for EEC installations that combine control of the superchargers or turbochargers requiring a control and the engine power control.

(s) § 23.1145, Ignition switches

The applicant must provide a means to stop the engine; this may mean shutting off the EEC or appropriate components. Depending on specific system details, a special condition, exemption, or ELOS for § 23.1145 could be used. For EECs that supplement the magneto ignition, basic compliance would be acceptable as the traditional interface with the magneto ignition would still be present.

See § 23.1165 for related requirements.

(t) § 23.1147, Mixture controls

The existing requirements of this section are applicable to manual mixture controls, should they be included in the EEC configuration.

For EEC's incorporating manual mixture control, the interface with the aircraft system, and any unique installation requirements, such as the need for mode switches, auto-enrichment activation switches, etc., need to be identified.

The applicant must provide a means to stop the engine; this may mean shutting off the EEC or appropriate components. Depending on specific system details, a special condition, exemption or ELOS for § 23.1147 could be used.

(u) § 23.1149, Propeller speed and pitch controls

The applicant must account for the EEC's effect on the propeller pitch and speed controls. Failure of the propeller speed and pitch control may lead to a destructive event of the engine or propeller. For EEC's that incorporate propeller control functions, verification of each EEC controlled component and EEC control mode affecting propeller speed and pitch control for proper operation should be addressed in the SICP.

Section 23.1309 is applicable in assessing the EEC control of the propeller speed and pitch control.

(v) § 23.1153, Propeller feathering controls

This requirement is the same as § 23.1149, except feather replaces speed and pitch.

(w) § 23.1157, Carburetor air temperature controls

For EECs that incorporate inlet air temperature controls, verification of each EEC controlled component and EEC control mode that affects inlet air temperature for proper operation should be addressed in the SICP.

See § 23.1093 for related requirements.

Section 23.1309 is applicable.

(x) § 23.1163, Powerplant accessories

Basic compliance with the regulation is acceptable. The installation instructions, as cited in § 23.901, should provide guidance to address the requirements.

(y) § 23.1165, Engine ignition systems [Backup electrical power]

Engines equipped with an EEC that relies on aircraft power and has no mechanical backup system will need backup electrical power, as discussed in section 2.d, Backup Electrical Power. The minimum requirement for the electrical power system is addressed in §§ 23.1309, 23.1351, 23.1353, 23.1357, 23.1359, 23.1361, 23.1365 and 23.1367.

(z) § 23.1183, Lines, fittings and components

Basic compliance with the regulation is acceptable.

(aa) § 23.1191, Firewalls

Basic compliance with the regulation is acceptable.

(6) Subpart F -- Equipment

(a) § 23.1301, Function and installation

EEC's have incorporated functions that are not required for certification such as maintenance data information, engine synchronization, etc. The applicant is required to show that the function(s) and associated failure(s) with the installed system are not hazardous and that the functions are performed as intended. Part 33 certification is acceptable only for those functions that can be adequately addressed in their entirety by part 33 certification. For example, engine synchronization would need to be shown to perform its intended function once installed: limits of synchronization authority, master/slave configurations, master engine failure effects on slave engine, etc. The failure analysis of the installed system would be conducted under the requirements stated in § 23.1309.

If the EEC manufacturer provides installation instructions and approved substantiation data showing compliance with the requirements of § 23.1301, then the applicant need not show additional compliance.

(b) § 23.1305, Powerplant instruments

The instrumentation requirements in § 23.1305 were written for conventional reciprocating engines and turbine engines. The requirement did not envision the use of EECs. Additional instrumentation may be necessary for a particular installation that is not specified in the current requirements or the engine installation manual.

The applicant is required to specify the necessary instrumentation for the EEC installation. If the aircraft engine limiting parameters are different from those specified in § 23.1521, special conditions would be developed to require those parameters to be used as limitations (see § 23.1521). The limiting parameters will be displayed in the instrument panel, as required by § 23.1305.

Special conditions will be developed if any additional items to those named in § 23.1305(b) are required.

Deviations from the requirements of § 23.1305(b) could be administered by an exemption, an equivalent level of safety, or in combination with any special conditions requiring additional parameters.

The EEC system will probably include some method of fault detection and engine parameter annunciation. The EEC may interface directly with the indicator or may require adding a separate interface unit. The interface unit would then be required to meet the requirements of § 23.1309 and its applicable special conditions.

See § 23.1322 for related requirements.

(c) § 23.1307, Miscellaneous equipment

Basic compliance with the regulation is acceptable.

(d) § 23.1309, Equipment, systems and installations

The applicant is required to provide a Functional Hazard Assessment (FHA) and System Safety Assessment (SSA) as part of the certification of an EEC installation. The FHA and SSA must address the system functionality, fault tolerance of the EEC, and any associated aircraft inputs to the EEC.

The EEC FHA and SSA need to address the following items:

1. Failure of aircraft supplied power: If the EEC relies on aircraft power, it will be necessary to evaluate the failure of aircraft supplied power. Since an EEC without mechanical backup requires backup electrical power, the criticality of the power supply system may be higher than on a conventional magneto ignition system. If the electrical power after loss of the generation system is limited to a specific time (for example, if supplied by battery), this remaining time should be demonstrated and the information included in the AFM. The time required in § 23.1353(h) is the minimum time for EEC backup electrical power in the event of a failure of available aircraft electrical power.

The intent of installing an EEC is to increase the reliability of the overall engine control system. The inherent redundancy of an independent power source for engine operation is a desirable feature when considering aircraft electrical power failure. Following the loss of the primary power generation system, a minimum of 60 minutes of backup electrical power for the EEC is highly recommended. The FHA, SSA combined with the EEC Safety Analysis (EECSA), as defined in AC 33.28-2, should result in an engine control system with greater reliability than the current magneto based systems. The probability of total failure of aircraft supplied power to the EEC must be shown to be lower than the probability of both magnetos failing.

If a total aircraft power failure occurs, the pilot must be informed in the AFM that the engine will stop if the electric power is lost and the engine is operated beyond the limits of the backup power.

2. Failure of aircraft data: The installation of an EEC in an airplane may have interfaces with other systems on the aircraft, such as an air data computer (ADC), weight on wheels (WOW) sensor, etc. Their failure rate must be considered when performing an aircraft system level assessment of the EEC installation.

3. EEC safety analysis (EECSA) assumptions: The results of the 14 CFR part 33 certification of the EEC safety analysis need to be evaluated in the aircraft SSA for the assumptions that were made about aircraft operation, maintenance, installed environment and latent failures.

4. EEC safety analysis (EECSA) results: The aircraft SSA should use the EECSA results of the estimated reliability of, or the failure rates for, safety-significant failure conditions and the other events associated with the control system. This includes events that the control system causes or is involved in preventing. These results are determined from the system safety analysis, such as the following: loss of power control, minor power loss, control mode transfers as a result of a failure, transmission of faulty parameters, loss of protection schemes, engine mechanical failures that affect EEC functionality, EEC fault accommodation logic, etc.

5. Environmental Testing: The aircraft SSA will address the levels of environmental exposure for which the EEC has been successfully qualified. The effects of lightning must be addressed. The applicant must specify the level of upset acceptable for the engine control aspects of the EEC as well as the additional functions and features of the EEC. For example, a lightning strike on an EEC equipped airplane may produce momentary interruptions of either the engine indications or engine operation, or both. The applicant must show that the interruptions of either the engine indications or engine operation, or both, do not cause an unsafe condition. Additionally, the applicant must show that the engine control functions recover the engine without hazard or increase to the crew workload.

6. Software validation and verification: The airplane system safety assessments provide the criticality and associated software development assurance level. Accomplishing this effort in a timely manner ensures that an adequate level of software development is used during engine selection/engine certification. Software developed and certified within part 33 to the correct assurance level does not establish part 23 certification of the software installation. Part 23 EEC functions without an equivalent part 33 requirement will be required to comply with 14 CFR part 23, § 23.1301. Compliance requirements for EEC functions that are applicable to part 23 requirements should be addressed in the SICP document. Application of current AC 23.1309-1C guidance information will result in classifications of failure conditions and identify the DO-178B criticality of engine software required.

The part 23 installation may require a software level that is higher than that certificated to date in part 33. AC 23.1309-1C shows the software requirements for the various classes of aircraft. The minimum software certification level for an EEC is DO178B level C. However, aircraft defined as Class III or higher will require DO178B level B software. If the EEC equipped engine is certified under 14 CFR part 33 with DO178B level C software, then a higher software certification level would be required to install this engine into a 14 CFR part 23 aircraft in the Class III category.

Should TLO be intended, early coordination by the applicant with the ACO is required to enable timely certification of both the engine and the airplane. The operational data reporting requirements for TLO are significant.

AC 33-28.2 discusses the requirements for EEC integration into a 14 CFR part 23 airplane by suggesting the EEC manufacturer provide a Certification Data Interface Document (CDID). The data contained in the CDID should provide the applicant the necessary data to show compliance to the requirements of 14 CFR part 23, § 23.1309. The description of the CDID from AC 33-28.2 is shown below:

<u>Certification Data Interface Document (CDID)</u>. The applicant should submit a document that includes data from safety analyses, environmental testing, and software validation and verification that is necessary for systems analysis of the powerplant installation and any other associated aircraft system. In lieu of a separate document, this data may be included in the instruction manual. If the applicant submits this data in a separate document, the instructions for installation should reference the document to ensure that the data is integrated into the airplane designer's powerplant system safety analysis. The CDID should include, but is not limited to, the following specific data:

(1) Failure of Aircraft Power Analysis. The CDID should include the assumptions and results from this analysis that should be considered in the aircraft design. Chapter 4 of this AC (AC 33-28.2) provides more information on this analysis. The CDID should also include, but is not limited to, the following specific data:

a. Data for the engine response to the transition from primary to backup power sources, or from primary to mechanical backup controls modes; and

b. Effects on engine power and the duration of those effects.

(2) Failure of Aircraft Data Analysis. The CDID should include the assumptions and results from this analysis that should be considered in the aircraft design. Chapter 4 of this AC provides more information on this analysis. The CDID should include, but is not limited to, the following specific data:

a. Fault accommodations taken for these aircraft-supplied signals;

b. Those air data signals whose failure can cause an engine MPL;

c. EEC system limitations for aircraft-supplied data, and signals for protection against common mode faults; and

d. Hardware and software requirements for aircraft components in the data path between the EEC system and the aircraft.

(3) EEC Safety Analysis (EECSA) Assumptions. The CDID should include assumptions relating to airplane installation and operation conditions, such as the following:

a. Operating procedures;

b. Maintenance actions and associated intervals;

c. Installed environment; and

d. Latent failure exposure period.

(4) EECSA Results. The CDID should specify the estimated reliability of, or the failure rates for, safety-significant failure conditions and the other events associated with the control system (that is, events that the control system causes or is involved in preventing), as determined from the system safety analysis, such as the following:

a. LOPC rate from the numerical fault tree analysis (FTA) and failure modes and effects analysis (FMEA);

b. MPL rate from the numerical FTA and FMEA;

c. *Failures that result in a system transfer to an alternate control mode;*
d. *Transmission of faulty parameters that affect cockpit-located engine displays or other safety critical functions identified in the EEC system output data content analysis;*
e. *Loss of any critical safeguards, such as CHT or overboost limiting control, identified as latent failures in the FMEA or FTA;*
f. *Failures of existing engine mechanical systems that could produce significant EEC system failure modes or effects as identified in the existing subsystems analysis; and*
g. *Fault accommodation logic data (for EEC system faults) as identified in the fault accommodation logic analysis:*
    i. *Tabulation of the fault accommodation logic for the critical parameters used by the control; and*
    ii. *Tabulation of the "default" or "fail-safe" states of all output effectors and the rationale for selection.*

(5) *Environmental Testing. The CDID should specify the types and levels of environmental exposure for which the EEC system has been successfully qualified (for example, vibration, temperature, electro-magnetic interference (EMI), HIRF, and lightning). For the HIRF, lightning, and EMI qualification tests, the CDID should describe the interfacing aircraft cables used for the tests.*

(6) *Software Validation and Verification. The CDID should identify the software level and documentation submitted in support of the software certification.*

The information provided in the CDID by the EEC manufacturer should be used in showing compliance with 14 CFR part 23, § 23.1309. If the EEC manufacturer does not provide a CDID, or other documentation containing the information cited above, then it becomes necessary for the applicant to perform the necessary testing and analysis to adequately satisfy the requirements of 14 CFR part 23, § 23.1309.

The current Small Airplane Directorate Standards Office policy on EEC installation in small airplanes, under § 23.1309, has been to issue two special conditions. The first special condition applies § 23.1309(a) through (e) to the propulsion system installation. The second special condition is protection of the EEC from exposure to HIRF. The evaluation should be limited to the interfaces of the engine/control system and verification that none of the assumptions made for part 33 certification of the engine are invalidated by the installation. The analysis should not extend into data submitted and approved as part of the engine certification program.

Should additional electronic equipment be installed into the airplane once the EEC has been installed, the applicant must ensure that the additional equipment will not interfere with the EECs proper operation. The newly installed electronic equipment must not have radiated HIRF levels that exceed the levels to which the EEC system has been certified.

Examples of the two special conditions that have been issued follow:

(aa) Special Condition [§ 23.1309]

The EEC system installation must comply with the requirements of § 23.1309(a) through (e). When showing compliance with this requirement, the reliability of the control system should either (1) be equivalent to or better than the reliability of the mechanical systems the control is replacing or (2) meet accepted § 23.1309 hardware reliability levels used for other airplane electronic systems. Software assurance levels used for the control should meet accepted § 23.1309 assurance levels used for other airplane electronic systems. When appropriate, engine certification data may be used when showing compliance with this requirement. However, the effects of the installation on this data must be addressed.

(bb) Special Condition [High Intensity Radiated Field (HIRF)]

In showing compliance with 14 CFR part 21, and the airworthiness requirements of 14 CFR part 23 for protection against hazards caused by the exposure to HIRF fields, electrical and electronic systems that perform critical functions must be considered. The hazards addressed include those that would result in a catastrophic failure condition to the airplane. To prevent this occurrence, airplane systems that perform critical functions must be designed and installed to ensure that the operation and operational capabilities of these critical systems are not adversely affected when the airplane is exposed to high-energy radio fields.

There is no specific regulation that addresses protection of electrical and electronic systems from HIRF; therefore, a special condition is necessary. The special condition will be written to ensure that each electrical and electronic system that performs critical functions is designed and installed properly. Proper design and installation means the operation and operational capabilities of the system to perform critical functions are not adversely affected when the airplane is exposed to HIRF external to the airplane. The term “critical” means those functions whose failure would contribute to, or cause, a failure condition that would prevent the continued safe flight and landing of the airplane. The critical functions may be determined using the FHA provided it is reviewed and approved by the FAA.

The FAA policy is contained in Notice 8110.71, dated April 2, 1998, which establishes the HIRF energy levels that airplanes will be exposed to in service. The guidelines set forth in this notice are the result of an Aircraft Certification Service review of existing policy on HIRF. The review occurred because of the continuing work of the Aviation Rulemaking Advisory Committee (ARAC) Electromagnetic Effects Harmonization Working Group (EEHWG).

The EEHWG adopted a set of HIRF environment levels in November 1997 that were agreed on by the FAA, Joint Aviation Authorities (JAA) and industry participants. As a result, the HIRF environments in this notice reflect the environment levels recommended by this working group. An issue paper must be written and must include how the airplane complies with HIRF and lightning requirements in accordance with Notice N8110.71, AC 23.1309-1C, and AC 23-17. Compliance with the HIRF requirements must address the airframe interface and installation of the HIRF approved engine. The installation issues of the EEC controlled engine into the airframe must be addressed. The applicant should specify what HIRF levels the engine was certified to and make a comparison of those requirement levels to the airplane in accordance with § 23.1309.

The FAA defines the following two acceptable methods for complying with the requirement for protection of systems that perform critical functions.

1. The applicant may demonstrate that the operation and operational capability of the installed electrical and electronic systems that perform critical functions are not adversely affected when the aircraft is exposed to the external HIRF threat environment defined in the following table:

	Field Strength (volts per meter)	
Frequency	**Peak**	**Average**
10 kHz - 100 kHz	50	50
100 kHz - 500 kHz	50	50
500 kHz – 2 MHz	50	50
2 MHz – 30 MHz	100	100
30 MHz – 70 MHz	50	50
70 MHz - 100 MHz	50	50
100 MHz - 200 MHz	100	100
200 MHz - 400 MHz	100	100
400 MHz - 700 MHz	700	50
700 MHz – 1 GHz	700	100
1 GHz – 2 GHz	2000	200
2 GHz – 4 GHz	3000	200
4 GHz – 6 GHz	3000	200
6 GHz – 8 GHz	1000	200
8 GHz – 12 GHz	3000	300
12 GHz – 18 GHz	2000	200
18 GHz – 40 GHz	600	200
The field strengths are expressed in peak root-mean-square (rms) values.		

or,

2. The applicant may demonstrate by a system test and analysis that the electrical and electronic systems that perform critical functions can withstand a minimum threat of 100 volts per meter peak electrical strength, without the benefit of airplane structural shielding, in the frequency range of 10 KHz to 18 GHz. When using this test to show compliance with the HIRF requirements, no credit is given for signal attenuation due to installation.

(e) § 23.1311, Electronic display instrument systems

The EEC interaction with electronic display components ranges from no system interaction to EEC generation and supply of display information. Consequently, the range of system configuration is broad. The applicability of multiple requirements must be considered. Some of these requirements include the following: §§ 23.901, 23.903, 23.1141, 23.1305, 23.1309, and 23.1353.

(f) § 23.1321, Arrangement and visibility

Basic compliance with the regulation is acceptable.

(g) § 23.1322, Warning, caution, and advisory lights

The EEC installation requirements for warning, caution, and advisory lights need to be identified by the applicant in the installation instructions required by § 23.901 and in compliance with § 23.1322. This includes operation, advisory, and fault annunciation schemes.

See § 23.1305 for related requirements.

(h) §§ 23.1331, 23.1337, 23.1351

Basic compliance with the regulations is acceptable.

(i) § 23.1353, Storage battery design and installation

Engine configurations with dedicated EEC electrical power generation have been certified. The applicant must address the reliance, if any, on airframe electrical power and applicable certification requirements, including § 23.1353(h).

Engine configurations without dedicated EEC electrical power have been certificated and, as a result, require reliance on airframe electrical power (and batteries) to enable compliance with § 23.1353(h). For aircraft with operations within close distance of suitable landing sites, engine operation that relies on backup battery duration may be sufficient. For aircraft with intended operations that are a significant flight time from a suitable landing site, the FAA recommends that the airplane be configured such that engine operation does not rely on airframe electrical sources.

The time required in § 23.1353(h) is the minimum time for EEC backup electrical power in the event of a failure of available aircraft electrical power. The intent of the installation of an EEC is to increase the reliability of the overall engine control system. The inherent redundancy of an independent power source for engine operation is a desirable feature when considering aircraft electrical power failure. Following the loss of the primary power generation system, a minimum of 60 minutes of backup electrical power for the EEC is highly recommended. The FHA, SSA combined with the EECSA, as defined in AC 33.28-2, should result in an engine control system with greater reliability than the current magneto based systems. The probability of total failure of aircraft supplied power to the EEC must be shown to be lower than the probability of both magnetos failing.

See § 23.1165 for related requirements.

(j) § 23.1357, Circuit protective devices

The EEC installation must include guidance on the ability to reset any circuit breakers, the applicability of the reset, and a limit on the number of times a circuit breaker can be reset before any maintenance action. The instructions for circuit breaker reset, and any limitations, must be in the AFM, and must comply with § 23.1521.

(k) §§ 23.1359, 23.1361, 23.1365, 23.1367, 23.1381, 23.1431, 23.1437

Basic compliance with the regulations is acceptable.

(7) Subpart G -- Operating Limitations and Information

(a) § 23.1501, General

Basic compliance with the regulation is acceptable.

(b) § 23.1521, Powerplant limitations

The applicant must define any EEC associated limitations and provide the pilot with a means of ensuring that the EEC is operated within those limitations. The applicant must define the dispatchable condition of the EEC installation, and those dispatchable configuration(s) for an EEC installation must be certified. Fault conditions considered dispatchable must be considered from the airplane perspective that also includes the engine limitations. TLO operations must also be considered if they are intended.

If the aircraft engine limiting parameters are different from those specified in § 23.1521, special conditions are developed to require those parameters to be used as limitations. The limiting parameters will be displayed in the instrument panel, as required by § 23.1305.

See § 23.1305 and § 23.1322 for related requirements.

(c) §§ 23.1525, 23.1527

Basic compliance with the regulations is acceptable.

(d) § 23.1529, Instructions for continued airworthiness

EEC functions will include basic engine operation. The EEC may also include fault detection, engine condition and performance, and other auxiliary functions. The applicant is required to provide Instructions for Continued Airworthiness (ICAs), and the instructions must address the EEC installation in its entirety. Specific guidance for fault resolution and disposition of any TLO conditions must be included.

(e) §§ 23.1541, 23.1543, 23.1549, 23.1555, 23.1559

Basic compliance with the regulations is acceptable.

(f) § 23.1581, General

The applicant must include EEC annunciations and associated limits in the AFM. Not all EEC annunciations belong in the cockpit. However, the AFM must include all flight crew related, cockpit-displayed annunciations. Additional EEC annunciations, such as maintenance diagnostic data, may be included in the AFM or in a supplemental document such as a maintenance manual.

(g) § 23.1583, Operating limitations

The applicant must include the EEC related limitations in the Operating Limitations section of the AFM. EEC unique operating limits, such as degraded mode of operation, if applicable, must be included in the operating limitations.

(h) § 23.1585, Operating procedures

The applicant must include the EEC related operating procedures in the Operating Procedures section of the Airplane Flight Manual. EEC unique operating procedures, such as a power-up, self-test, and channel switching, if applicable, must be included in the operating procedures.

(i) § 23.1587, Performance information

The applicant must include, in the Performance Information section, any EEC installation effects of EEC dispatchable modes that result in degraded engine performance. One example of this would be the dispatchable EEC degraded mode where the fuel mixture is at its full rich setting, resulting in a reduction in engine power. Additionally, if the EEC negatively affects the fuel flow when operating in a normal or degraded mode, then the effect will need to be included in the AFM.

5. Effect of Policy

The general policy stated in this document does not constitute a new regulation or create what the courts refer to as a "binding norm." The office that implements policy should follow this policy when applicable to the specific project. Whenever an applicant's proposed method of compliance is outside this established policy, it must be coordinated with the policy issuing office, for example, through the issue paper process or equivalent. Similarly, if the implementing office becomes aware of reasons that an applicant's proposal that meets this policy should not be approved, the office must coordinate its response with the policy issuing office.

Applicants should expect that the certificating officials would consider this information when making findings of compliance relevant to new certificate actions. Also, as with all advisory material, this policy statement identifies one means, but not the only means, of compliance.

s/ James E. Jackson
for
David A. Downey
Acting Manager, Small Airplane Directorate
Aircraft Certification Service

Appendix 1
Generic List of Certification Regulations Applicable to EEC Installations

14 CFR Part and Section	Subpart A – General	Guidance
23.1	Applicability.	See paragraph for guidance.
23.2	Special retroactive requirements.	
23.3	Airplane categories.	
	Subpart B -- Flight	
	GENERAL	
23.21	Proof of compliance.	
23.23	Load distribution limits.	
23.25	Weight limits.	
23.29	Empty weight and corresponding center of gravity.	
23.31	Removable ballast.	
23.33	Propeller speed and pitch limits.	See paragraph for guidance.
	PERFORMANCE	
23.45	General.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.49	Stalling period.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.51	Takeoff speeds.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.53	Takeoff performance.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.55	Accelerate-stop distance.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.57	Takeoff path.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.59	Takeoff distance and takeoff run.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.61	Takeoff flight path.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.63	Climb: General.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.65	Climb: All engines operating.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.66	Takeoff climb: One-engine inoperative.	See paragraph for guidance. See AC 23-8B for additional guidance.

23.67	Climb: One engine inoperative.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.69	Enroute climb/descent.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.71	Glide: Single-engine airplanes.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.73	Reference landing approach speed.	See paragraph for guidance.
23.75	Landing distance.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.77	Balked landing.	See paragraph for guidance. See AC 23-8B for additional guidance
	FLIGHT CHARACTERISTICS	
23.141	General.	See paragraph for guidance. See AC 23-8B for additional guidance.
	CONTROLLABILITY AND MANEUVERABILITY	
23.143	General.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.145	Longitudinal control.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.147	Directional and lateral control.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.149	Minimum control speed.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.151	Acrobatic maneuvers.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.153	Control during landings.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.155	Elevator control force in maneuvers.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.157	Rate of roll.	See paragraph for guidance. See AC 23-8B for additional guidance.
	TRIM	
23.161	Trim.	See paragraph for guidance. See AC 23-8B for additional guidance.
	STABILITY	
23.171	General.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.173	Static longitudinal stability.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.175	Demonstration of static longitudinal stability.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.177	Static directional and lateral stability.	See paragraph for guidance. See AC 23-8B for additional guidance.

23.181 Dynamic stability.	See paragraph for guidance. See AC 23-8B for additional guidance.
STALLS	
23.201 Wings level stall.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.203 Turning flight and accelerated turning stalls.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.207 Stall warning.	See paragraph for guidance. See AC 23-8B for additional guidance.
SPINNING	
23.221 Spinning.	See paragraph for guidance. See AC 23-8B for additional guidance.
GROUND AND WATER HANDLING CHARACTERISTICS	
23.231 Longitudinal stability and control.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.233 Directional stability and control.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.235 Operation on unpaved surfaces.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.237 Operation on water.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.239 Spray characteristics.	See paragraph for guidance. See AC 23-8B for additional guidance.
MISCELLANEOUS FLIGHT REQUIREMENTS	
23.251 Vibration and buffeting.	See paragraph for guidance. See AC 23-8B for additional guidance.
23.253 High speed characteristics.	See paragraph for guidance. See AC 23-8B for additional guidance.
Subpart C -- Structure	
GENERAL	
23.301 Loads.	Basic Compliance. See AC 23-19 for additional guidance.
23.302 Canard or tandem wing configurations.	
23.303 Factor of safety.	Basic Compliance. See AC 23-19 for additional guidance.
23.305 Strength and deformation.	Basic Compliance. See AC 23-19 for additional guidance.

23.307 Proof of structure.	Basic Compliance. See AC 23-19 for additional guidance.

FLIGHT LOADS

23.321 General.	Basic Compliance. See AC 23-19 for additional guidance.
23.331 Symmetrical flight conditions.	
23.333 Flight envelope.	
23.335 Design airspeeds.	
23.337 Limit maneuvering load factors.	
23.341 Gust loads factors.	
23.343 Design fuel loads.	
23.345 High lift devices.	
23.347 Unsymmetrical flight conditions.	
23.349 Rolling conditions.	
23.351 Yawing conditions.	
23.361 Engine torque.	
23.363 Side load on engine mount.	
23.365 Pressurized cabin loads.	
23.367 Unsymmetrical loads due to engine failure.	
23.369 Rear lift truss.	
23.371 Gyroscopic and aerodynamic loads.	
23.373 Speed control devices.	

CONTROL SURFACE AND SYSTEM LOADS

23.391 Control surface loads.	
23.393 Loads parallel to hinge line.	
23.395 Control system loads.	
23.397 Limit control forces and torques.	
23.399 Dual control system.	
23.405 Secondary control system.	
23.407 Trim tab effects.	
23.409 Tabs.	

23.415 Ground gust conditions.

HORIZONTAL STABILIZING AND BALANCING SURFACES

23.421 Balancing loads.
23.423 Maneuvering loads.
23.425 Gust loads.
23.427 Unsymmetrical loads.

VERTICAL SURFACES

23.441 Maneuvering loads.
23.443 Gust loads.
23.445 Outboard fins or winglets.

AILERONS AND SPECIAL DEVICES

23.455 Ailerons.
23.459 Special devices.

GROUND LOADS

23.471 General.
23.473 Ground load conditions and assumptions.
23.477 Landing gear arrangement.
23.479 Level landing conditions.
23.481 Tail down landing conditions.
23.483 One-wheel landing conditions.
23.485 Side load conditions.
23.493 Braked roll conditions.
23.497 Supplementary conditions for tail wheels.
23.499 Supplementary conditions for nose wheels.
23.505 Supplementary conditions for ski planes.
23.507 Jacking loads.

23.509 Towing loads.
23.511 Ground load; unsymmetrical loads on multiple-wheel units.

WATER LOADS

23.521 Water load conditions.
23.523 Design weights and center of gravity positions.
23.525 Application of loads.
23.527 Hull and main float load factors.
23.529 Hull and main float landing conditions.
23.531 Hull and main float takeoff condition.
23.533 Hull and main float bottom pressures.
23.535 Auxiliary float loads.
23.537 Seawing loads.

EMERGENCY LANDING CONDITIONS

23.561 General.
23.562 Emergency landing dynamic conditions.

FATIGUE EVALUATION

23.571 Metallic pressurized cabin structures.
23.572 Metallic wing, empennage, and associated structures.
23.573 Damage tolerance and fatigue evaluation of structure.
23.574 Metallic damage tolerance and fatigue evaluation of commuter category airplanes.
23.575 Inspections and other procedures.

Subpart D -- Design and Construction

23.601 General.	Basic Compliance. See AC 23-19 for additional guidance.
23.603 Materials and workmanship.	Basic Compliance. See AC 23-19 for additional guidance.
23.605 Fabrication methods.	Basic Compliance. See AC 23-19 for additional guidance.

23.607 Fasteners.	Basic Compliance. See AC 23-19 for additional guidance.
23.609 Protection of structure.	Basic Compliance. See AC 23-19 for additional guidance.
23.611 Accessibility provisions.	Basic Compliance. See AC 23-19 for additional guidance.
23.613 Material strength properties and design values.	Basic Compliance. See AC 23-19 for additional guidance.
23.619 Special factors.	
23.621 Casting factors.	
23.623 Bearing factors.	
23.625 Fitting factors.	
23.627 Fatigue strength.	
23.629 Flutter.	
WINGS	
23.641 Proof of strength.	
CONTROL SURFACES	
23.651 Proof of strength.	
23.655 Installation.	
23.657 Hinges.	
23.659 Mass balance.	
CONTROL SYSTEMS	
23.671 General.	
23.672 Stability augmentation and automatic and power-operated systems.	
23.673 Primary flight controls.	
23.675 Stops.	
23.677 Trim systems.	
23.679 Control system locks.	
23.681 Limit load static tests.	
23.683 Operation tests.	
23.685 Control system details.	

PERSONNEL AND CARGO ACCOMMODATIONS

23.771	Pilot compartment.	
23.773	Pilot compartment view.	
23.775	Windshields and windows.	
23.777	Cockpit controls. *	See paragraph. See AC 23-17A for additional guidance.
23.779	Motion and effect of cockpit controls. *	See paragraph. See AC 23-17A for additional guidance.
23.781	Cockpit control knob shape. *	See paragraph. See AC 23-17A for additional guidance.
23.783	Doors.	
23.785	Seats, berths, litters, safety belts, and shoulder harnesses.	
23.787	Baggage and cargo compartments.	
23.791	Passenger information signs.	
23.803	Emergency evacuation.	
23.805	Flightcrew emergency exits.	
23.807	Emergency exits.	
23.811	Emergency exit marking.	
23.812	Emergency lighting.	
23.813	Emergency exit access.	
23.815	Width of aisle.	
23.831	Ventilation.	

PRESSURIZATION

23.841	Pressurized cabins.
23.843	Pressurization tests.

FIRE PROTECTION

23.851	Fire extinguishers.
23.853	Passenger and crew compartment interiors.
23.855	Cargo and baggage compartment fire protection.
23.859	Combustion heater fire protection.
23.863	Flammable fluid fire protection.

23.865 Fire protection of flight controls, engine mounts, and other flight structure.	
ELECTRICAL BONDING AND LIGHTNING PROTECTION	
23.867 Electrical bonding and protection against lightning and static electricity.	See paragraph. See AC 23-17A for additional guidance.
MISCELLANEOUS	
23.871 Leveling means.	
Subpart E -- Powerplant	
GENERAL	
23.901 Installation.	See paragraph. See AC 23-16A for additional guidance.
23.903 Engines.	See paragraph. See AC 23-16A for additional guidance.
23.904 Automatic power reserve system.	See paragraph. See AC 23-16A for additional guidance.
23.905 Propellers.	
23.907 Propeller vibration.	See paragraph. See AC 23-16A for additional guidance.
23.909 Turbocharger systems.	See paragraph. See AC 23-16A for additional guidance.
23.925 Propeller clearance.	
23.929 Engine installation ice protection.	
23.933 Reversing systems.	See paragraph. See AC 23-16A for additional guidance.
23.934 Turbojet and turbofan engine thrust reverser systems tests.	
23.937 Turbopropeller-drag limiting systems.	
23.939 Powerplant operating characteristics. *	See paragraph. See AC 23-16A for additional guidance.
23.943 Negative acceleration.	
FUEL SYSTEM	
23.951 General.	
23.953 Fuel system independence.	
23.954 Fuel system lightning protection.	

23.955	Fuel flow.	Basic compliance. See AC 23-16A for additional guidance.
23.957	Flow between interconnected tanks.	
23.959	Unusable fuel supply.	
23.961	Fuel system hot weather operation.	
23.963	Fuel tanks: General.	
23.965	Fuel tank tests.	
23.967	Fuel tank installation.	
23.969	Fuel tank expansion space.	
23.971	Fuel tank sump.	
23.973	Fuel tank filler connection.	
23.975	Fuel tank vents and carburetor vapor vents.	
23.977	Fuel tank outlet.	
23.979	Pressure fueling systems.	
	FUEL SYSTEM COMPONENTS	
23.991	Fuel pumps.	See paragraph. See AC 23-16A for additional guidance.
23.993	Fuel system lines and fittings.	Basic compliance. See AC 23-16A for additional guidance.
23.994	Fuel system components.	
23.995	Fuel valves and controls.	
23.997	Fuel strainer or filter.	Basic compliance. See AC 23-16A for additional guidance.
23.999	Fuel system drains.	
23.1001	Fuel jettisoning system.	
	OIL SYSTEM	
23.1011	General.	
23.1013	Oil tanks.	
23.1015	Oil tank tests.	
23.1017	Oil lines and fittings.	
23.1019	Oil strainer or filter.	
23.1021	Oil system drains.	
23.1023	Oil radiators.	

23.1027	Propeller feathering system.	Basic compliance. See AC 23-16A for additional guidance.
	COOLING	
23.1041	General.	See paragraph. See AC 23-16A for additional guidance.
23.1043	Cooling tests.	See paragraph. See AC 23-16A for additional guidance.
23.1045	Cooling test procedures for turbine engine powered airplanes.	
23.1047	Cooling test procedures for reciprocating engine powered airplanes.	See paragraph. See AC 23-16A for additional guidance.
	LIQUID COOLING	
23.1061	Installation.	
23.1063	Coolant tank tests.	
	INDUCTION SYSTEM	
23.1091	Air induction system.	
23.1093	Induction system icing protection.	See paragraph. See AC 23-16A for additional guidance.
23.1095	Carburetor deicing fluid flow rate.	
23.1097	Carburetor deicing fluid system capacity.	
23.1099	Carburetor deicing fluid system detail design.	
23.1101	Induction air preheater design.	
23.1103	Induction system ducts.	
23.1105	Induction system screens.	
23.1107	Induction system filters.	
23.1109	Turbocharger bleed air system.	
23.1111	Turbine engine bleed air system.	
	EXHAUST SYSTEM	
23.1121	General.	
23.1123	Exhaust system.	
23.1125	Exhaust heat exchangers.	

POWERPLANT CONTROLS AND ACCESSORIES

23.1141	Powerplant controls: General.	See paragraph. See AC 23-16A for additional guidance.
23.1142	Auxiliary power unit controls.	
23.1143	Engine controls. *	See paragraph. See AC 23-16A for additional guidance.
23.1145	Ignition switches. *	See paragraph. See AC 23-16A for additional guidance.
23.1147	Mixture controls. *	See paragraph. See AC 23-16A for additional guidance.
23.1149	Propeller speed and pitch controls.	See paragraph. See AC 23-16A for additional guidance.
23.1153	Propeller feathering controls.	See paragraph. See AC 23-16A for additional guidance.
23.1155	Turbine engine reverse thrust and propeller pitch settings below the flight regime.	
23.1157	Carburetor air temperature controls.	See paragraph.
23.1163	Powerplant accessories.	See paragraph. See AC 23-16A for additional guidance.
23.1165	Engine ignition systems.	See paragraph. See AC 23-16A and AC 23-17A for additional guidance.

POWERPLANT FIRE PROTECTION

23.1181	Designated fire zones; regions included.	
23.1182	Nacelle areas behind firewalls.	
23.1183	Lines, fittings, and components.	Basic compliance. See AC 23-16A for additional guidance.
23.1189	Shutoff means.	
23.1191	Firewalls.	Basic compliance. See AC 23-16A for additional guidance.
23.1192	Engine accessory compartment diaphragm.	
23.1193	Cowling and nacelle.	
23.1195	Fire extinguishing systems.	
23.1197	Fire extinguishing agents.	
23.1199	Extinguishing agent containers.	
23.1201	Fire extinguishing systems materials.	
23.1203	Fire detector system.	

Subpart F -- Equipment

GENERAL

23.1301 Function and installation.	See paragraph. See AC 23-17A for additional guidance.
23.1303 Flight and navigation instruments.	
23.1305 Powerplant instruments. *	See paragraph. See AC 23-17A and AC 23.1311-1A for additional guidance.
23.1307 Miscellaneous equipment.	Basic compliance.
23.1309 Equipment, systems, and installations. *	See paragraph. See AC 23-17A and AC23.1309-1C for additional guidance.

INSTRUMENTS: INSTALLATION

23.1311 Electronic display instrument systems.	See paragraph. See AC 23-17A for additional guidance.
23.1321 Arrangement and visibility.	Basic compliance. See AC 23-17A for additional guidance.
23.1322 Warning, caution, and advisory lights.	See paragraph. See AC 23-17A for additional guidance.
23.1323 Airspeed indicating system.	
23.1325 Static pressure system.	
23.1326 Pitot heat indication systems.	
23.1327 Magnetic direction indicator.	
23.1329 Automatic pilot system.	
23.1331 Instruments using a power source.	Basic compliance. See AC 23-17A for additional guidance.
23.1335 Flight director systems.	
23.1337 Powerplant instruments installation.	Basic compliance. See AC 23-17A for additional guidance.

ELECTRICAL SYSTEMS AND EQUIPMENT

23.1351 General.	Basic compliance. See AC 23-17A for additional guidance.
23.1353 Storage battery design and installation.	See paragraph. See AC 23-17A for additional guidance.
23.1357 Circuit protective devices.	See paragraph. See AC 23-17A for additional guidance.
23.1359 Electrical system fire protection.	Basic compliance. See AC 23-17A for additional guidance.
23.1361 Master switch arrangement.	Basic compliance. See AC 23-17A for additional guidance.
23.1365 Electric cables and equipment.	Basic compliance. See AC 23-17A for additional guidance.
23.1367 Switches.	Basic compliance. See AC 23-17A for additional guidance.

LIGHTS

23.1381	Instrument lights.	Basic compliance. See AC 23-17A for additional guidance.
23.1383	Taxi and landing lights.	
23.1385	Position light system installation.	
23.1387	Position light system dihedral angles.	
23.1389	Position light distribution and intensities.	
23.1391	Minimum intensities in the horizontal plane of position lights.	
23.1393	Minimum intensities in any vertical plane of position lights.	
23.1395	Maximum intensities in overlapping beams of position lights.	
23.1397	Color specifications.	
23.1399	Riding light.	
23.1401	Anticollision light system.	

SAFETY EQUIPMENT

23.1411	General.	
23.1415	Ditching equipment.	
23.1416	Pneumatic de-icer boot system.	
23.1419	Ice protection.	

MISCELLANEOUS EQUIPMENT

23.1431	Electronic equipment.	Basic compliance. See AC 23-17A for additional guidance.
23.1435	Hydraulic systems.	
23.1437	Accessories for multiengine airplanes.	Basic compliance. See AC 23-17A for additional guidance.
23.1438	Pressurization and pneumatic systems.	
23.1441	Oxygen equipment and supply.	
23.1443	Minimum mass flow of supplemental oxygen.	
23.1445	Oxygen distribution system.	
23.1447	Equipment standards for oxygen dispensing units.	
23.1449	Means for determining use of oxygen.	

23.1450 Chemical oxygen generators.	
23.1451 Fire protection for oxygen equipment.	
23.1453 Protection of oxygen equipment from rupture.	
23.1457 Cockpit voice recorders.	
23.1459 Flight recorders.	
23.1461 Equipment containing high energy rotors.	

Subpart G -- Operating Limitations and Information

23.1501 General.	Basic compliance. See AC 23-8B for additional guidance.
23.1505 Airspeed limitations.	
23.1507 Operating maneuvering speed.	
23.1511 Flap extended speed.	
23.1513 Minimum control speed.	
23.1519 Weight and center of gravity.	
23.1521 Powerplant limitations. *	See paragraph. See AC 23-8B for additional guidance.
23.1522 Auxiliary power unit limitations.	
23.1523 Minimum flight crew.	
23.1524 Maximum passenger seating configuration.	
23.1525 Kinds of operation.	Basic compliance. See AC 23-8B for additional guidance.
23.1527 Maximum operating altitude.	Basic compliance. See AC 23-8B for additional guidance.
23.1529 Instructions for Continued Airworthiness.	See paragraph.

MARKINGS AND PLACARDS

23.1541 General.	Basic compliance. See AC 23-8B for additional guidance.
23.1543 Instrument markings: General.	Basic compliance. See AC 23-8B for additional guidance.
23.1545 Airspeed indicator.	
23.1547 Magnetic direction indicator.	
23.1549 Powerplant and auxiliary power unit instruments.	Basic compliance. See AC 23-8B for additional guidance.
23.1551 Oil quantity indicator.	
23.1553 Fuel quantity indicator.	
23.1555 Control markings.	Basic compliance. See AC 23-8B for additional guidance.

23.1557	Miscellaneous markings and placards.	
23.1559	Operating limitations placard.	Basic compliance. See AC 23-8B for additional guidance.
23.1561	Safety equipment.	
23.1563	Airspeed placards.	
23.1567	Flight maneuver placard.	
	AIRPLANE FLIGHT MANUAL AND APPROVED MANUAL MATERIAL	
23.1581	General.	See paragraph. See AC 23-8B for additional guidance.
23.1583	Operating limitations.	See paragraph. See AC 23-8B for additional guidance.
23.1585	Operating procedures.	See paragraph. See AC 23-8B for additional guidance.
23.1587	Performance information.	See paragraph. See AC 23-8B for additional guidance.
23.1589	Loading information.	

Appendix 2

Example of an Issue Paper for a Special Condition

ISSUE PAPER

PROJECT: Applicant Technology, Inc Models M-1, M-2 Project No. XXXXXX	**ITEM:** XX-1 **STAGE:** 2
REG. REF.: 14 CFR Parts 11, 21, and 23; 14 CFR Sections 21.16, 21.17, 21.21(b)(2), 23.1301, 23.1305, 23.1309, 23.1311, 23.1321, 23.1322, 23.1331, 23.1529, CAR 3.411, 3.652, 3.655	**DATE:** **PAGE:** 1
NATIONAL POLICY REF: FAA Policy Statement; Installation of Engine Electronic Engine Control (EEC) For Reciprocating Engine; PS-ACE100-2004-10024, FAA Notice N8110.71: Guidance for the Certification of Aircraft Operating in High Intensity Radiated Field (HIRF) Environments, dated April 2, 1998 and AC 23.1309-1C	**ISSUE STATUS:** Open
SUBJECT: High Intensity Radiated Fields (HIRF) Protection for the Applicant Technology EEC	**BRANCH ACTION:** ACE-110, 111, 112 **COMPLIANCE TARGET:** Pre-STC

EEC- HIRF – SPECIAL CONDITION

STATEMENT OF ISSUE: Applicant Technology has requested an amendment to Type Certificate XXXX to include Models M-1 and M-2. The proposed engine installation for these new models is the Engine Products E-1 engine, which includes a Full Authority Digital Electronic Control (FADEC) system, also known as an Electronic Engine Control (EEC). The EEC system performs critical functions, such as the control of the ignition and fuel injection functions throughout the operational envelope.

The applicable 14 CFR part 21 and 14 CFR part 23 airworthiness regulations for general aviation airplanes, including 14 CFR part 23, §§ 23.1301 and 23.1309, do not adequately consider failures due to the effects of High Intensity Radiated Fields (HIRF) on electrical and electronic systems that perform critical functions. Therefore, a special condition is proposed to provide HIRF protection for any electrical and electronic components that may be installed in the Applicant Technology M-1 and M-2.

BACKGROUND:

Changes in technology have resulted in advanced airplane electrical and electronic systems and more frequent use of high-energy radio frequency transmitters, such as radio and television broadcast stations, radar, and satellite uplink transmitters. The combined effect of these developments has been an increased susceptibility of electrical and electronic systems to the negative effects of electromagnetic fields.

Many advanced electronic systems, which perform critical functions, are prone to upsets and/or damage at energy levels lower than the analog systems previously used to perform the same functions. The use of composites has also increased the susceptibility of electronic systems to HIRF effects, since composites structures to not provide the same level of shielding as metal structure. For critical systems, provisions for protection from the effects of HIRF fields should be considered, and if necessary, incorporated into the original aircraft design data.

The Applicant Technology airplanes will incorporate an electronic engine control system that includes sensors, harnesses, and multi-port fuel injection systems required to control the fuel and ignition systems throughout the operational envelope. The functions of the EEC are considered critical. Additionally, the EEC system may be susceptible to disruption of both command/response/engine health-monitoring signals because of electrical and magnetic interference. This disruption of signals could result in the loss of critical engine functions, flight displays and annunciations, or present misleading information, including the health of the engine, to the pilot.

FAA POSITION:

In showing compliance with 14 CFR part 21 and the airworthiness requirements of 14 CFR part 23, the protection of critical electrical and electronic systems against hazards caused by the exposure to HIRF fields must be considered. The hazards addressed include those that would result in a catastrophic failure condition to the airplane. To prevent such failures, airplane systems that perform critical functions must be designed and installed to ensure the operation and operational capabilities of these critical systems are not adversely affected when the airplane is exposed to HIRF.

The term "critical" refers to those functions whose failure would contribute to, or cause, a failure condition that would prevent the continued safe flight and landing of the airplane. The critical functions may be determined using the Functional Hazard Assessment (FHA) provided it is reviewed and approved by the FAA.

Existing FAA policy on HIRF is contained in Notice 8110.71, dated April 2, 1998, which establishes the energy levels that airplanes will be exposed to in service. The guidelines set forth in this notice are the result of an Aircraft Certification Service review of existing policy on HIRF because of the ongoing work of the ARAC Electromagnetic Effects Harmonization Working Group (EEHWG). The EEHWG adopted a set of HIRF environment levels in November 1997, which were agreed upon by the FAA, JAA and industry participants. As a result, the HIRF environments in this notice reflect the environment levels recommended by this working group.

Compliance with the HIRF requirements must address the installation issues of the EEC and the engine itself into the airframe, such as proper grounding and strapping of the HIRF compliant engine. The applicant should specify what HIRF levels the engine was certified to and make a comparison of those requirement levels to the airplane IAW §§ 23.1309. Also, the continued integrity of the HIRF requirements for the approved engine and installation must be addressed. The applicant must prepare and deliver with each aircraft Instructions for Continued Airworthiness containing information that assures the continued HIRF compliance and integrity of the engine.

At this time the FAA and other airworthiness authorities are unable to precisely define or control the HIRF energy level to which the airplane will be exposed in service. Therefore, the FAA hereby defines two acceptable interim methods for complying with the requirement for protection of systems that perform critical functions.

(1) The applicant may demonstrate that the operation and operational capability of the installed electrical and electronic systems that perform critical functions are not adversely affected when the aircraft is exposed to the external HIRF threat environment defined in the following table:

	Field Strength (volts per meter)	
Frequency	**Peak**	**Average**
10 kHz - 100 kHz	50	50
100 kHz - 500 kHz	50	50
500 kHz - 2 MHz	50	50
2 MHz - 30 MHz	100	100
30 MHz - 70 MHz	50	50
70 MHz - 100 MHz	50	50
100 MHz - 200 MHz	100	100
200 MHz - 400 MHz	100	100
400 MHz - 700 MHz	700	50
700 MHz - 1 GHz	700	100
1 GHz - 2 GHz	2000	200
2 GHz - 4 GHz	3000	200
4 GHz - 6 GHz	3000	200
6 GHz - 8 GHz	1000	200
8 GHz - 12 GHz	3000	300
12 GHz - 18 GHz	2000	200
18 GHz - 40 GHz	600	200
The field strengths are expressed in terms of peak root-mean-square (rms) values.		

or,

(2) The applicant may demonstrate by a system test and analysis that the electrical and electronic systems that perform critical functions can withstand a minimum threat of 100 volts per meter peak electrical strength, without the benefit of airplane structural shielding, in the frequency range of 10 KHz to 18 GHz. When using this test to show compliance with the HIRF requirements, no credit is given for signal attenuation due to installation. For VFR rotorcraft, 200 volts per meter is required for the peak electrical strength.

If other critical systems are installed (now or at a later date), such as an Electronic Flight Instrument System (EFIS), the applicant should identify these systems and incorporate them into this issue paper regarding how compliance to the HIRF requirements will be determined.

APPLICANT POSITION:

CONCLUSION:

Special conditions for HIRF protection for the electrical and electronic systems which perform critical functions will be written for the Applicant Technology M-1 and M-2 in accordance with the guidelines set forth in Action Notice 8110.71, dated April 2, 1998.

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