ADS-50-PRF

AERONAUTICAL DESIGN STANDARD

ROTORCRAFT PROPULSION PERFORMANCE AND QUALIFICATION REQUIREMENTS AND GUIDELINES

15 APRIL 1996

UNITED STATES ARMY AVIATION AND TROOP COMMAND

ST.LOUIS, MISSOURI

AVIATION RESEARCH AND DEVELOPMENT CENTER

DIRECTORATE FOR ENGINEERING

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PREPARED BY:_________________________
DENNIS S. POWELSON
Chief, Propulsion
Integration Branch

REVIEWED BY:_________________________
VERNON R. EDWARDS
Chief, Propulsion
Technology Division

APPROVED BY:_________________________
BARRY J. BASKETT
Director of Engineering
Standardization Executive

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SECTION 1: SCOPE

1.1 This ADS establishes the performance and verification which constitute qualification requirements for rotorcraft propulsion systems. For the purposes of this ADS, propulsion systems includes engine and auxiliary power unit installations, start, fire detection/extinguishing, drive, fuel, environmental control, and hydraulic systems.

1.2 The **requirements** for each system are stated in paragraph 1 of each Section and are in **bold and larger letters** to distinguish them from the **guidelines** paragraphs.

1.3 The performance guidelines for each propulsion system are addressed in paragraphs 3.0 within each section herein. These performance criteria are intended to be performance guidelines that reflect accepted industry design practices and incorporate lessons learned from prior programs to achieve safety and durability and minimize operating and support costs of fielded systems. Should the contractor elect to utilize design/performance criteria other than those contained herein, the contractor is expected to explain his design selection. The agreed to design criteria/parameters will form the basis for the subsequent qualification test program. The qualification test guidelines in paragraphs 4.0 of each section represent the framework for the contractors test plans. The contractor is expected to address each of the qualification guidelines or explain why a particular guideline is not applicable or how it has been tailored to reflect the design configuration. The contractor may submit data to demonstrate that a particular component can be qualified by similarity. Successful completion of the qualification requirements will establish that the systems/components meet the agreed to performance criteria and will thus be considered qualified components, i.e. airworthy and specification compliant.

1.4 This ADS addresses the performance and verification which constitute qualification of component, module, or assembly level hardware. Criteria for design analysis, modeling, simulations, and test planning and reporting are covered in ADS-9C, Propulsion System Technical Data. Aircraft level propulsion system ground and flight testing requirements are addressed in ADS-1B, Rotorcraft Propulsion Systems Airworthiness Qualification Requirements.

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SECTION 2: ENGINE/APU INSTALLATION

2-1.0 REQUIREMENTS.

2-1.1 PERFORMANCE. The Engine/APU installation shall meet its allocated performance and function in such a manner that the aircraft shall be able to be operated safely throughout the operating envelope and meet the performance requirements as defined in the applicable weapon system specification.

2-1.2 QUALIFICATION. The qualification requirements for the Engine/APU installation are required to verify compliance with the performance requirements above as applicable to the aircraft design configuration. Qualification of the Engine/APU installation is accomplished at the aircraft system level in accordance with ADS-1B-PRF.

2-2.0 **REFERENCES**.

2-2.1 Military Specifications.

ADS-1B-PRF	Rotorcraft Propulsion Systems Airworthiness Qualification Requirements, Ground and Flight Test Surveys and Demonstrations
AV-E-8593	Engine, Aircraft, Turboshaft and Turboprop general Specification For
AV-P-85573	Power Unit Aircraft, Auxiliary, Gas Turbine, General Specification For
SD-24L	General Specification For Design And Construction of Aircraft Weapon Systems, Volume II - Rotary Wing Aircraft

2-3.0 **PERFORMANCE GUIDELINES.**

2-3.1 **Engine/APU Installation.** The engine or auxiliary power unit installation design should accommodate the installed engine or APU in accordance with the specific engine or APU system specification, or AV-E-8593 and AV-P-85573, respectively. The Reliability, Availability, and Maintainability (RAM) of the installed engine or APU should not be degraded by the installation nor should the engine or APU limits--e.g., vibration, temperature, etc.--specified in the engine or APU specification, be exceeded unless approved by the PCO. Installation design/integration requirements for engine/airframe interface for the engine or APU should be as defined in the Interface Control Document (ICD) and Air Vehicle Engine Integration Plan (AVEIP) mutually written and agreed to between the engine or APU and the air vehicle contractor.

2-3.1.1 Compartment(s). Due to the combustible fluids and heat release inherent in accommodating air-breathing power plants, dedicated compartments are commonly employed to achieve the isolation needed to minimize the hazards associated with operating the engines and APU. The engines and the APU should be separated from each other and from all of the surrounding internal areas of the airframe. Refer to Section 4 for specific fire containment performance guidelines.

2-3.1.2 Drains. All Contractor furnished parts used in propulsion compartment(s) should not be adversely affected by fuel, lubricants, oil, or cleaning materials. Drains in each compartment should be capable of draining fuel at a rate equal to or greater than the maximum fuel flow rate which could result from a severed fuel line. All drains should be free of traps and configured to prevent accumulation of vapor, fluid, or other contamination; and drains should discharge clear of the air vehicle structure. A drainage collection system to prevent fuel, oil, or hydraulic fluids from discharging on the ground or for use during shipboard operations should be provided. The compatibility guidelines for shipboard operations are specified in SD-24L. The drainage collection system should consist of a container, electrically actuated discharge valve, and appropriate plumbing and wiring. The drainage fluid collection system should be used to collect the fluids discharged from the engines, APU, and hydraulic pumps during normal or ship deck operations. The collected fluids should be manually emptied from the container via an electrically actuated discharge valve. Pressure differentials on all drains should not adversely affect engine or APU performance.

2-3.1.3 Cooling. The engine and APU cooling systems should maintain the temperatures of all the equipment in the compartments and all the components on the engines/APU within their maximum allowable temperature limits for all engine/APU power levels at all ground and flight conditions, including idle power and soak-back after shut-down. The maximum allowable temperature limits should be those established in the engine model specification. If the engine utilizes an air-oil heat exchanger, the cooling flow through the exchanger should be consistent with the exchanger losses and oil temperature limits of the model specification. The airflow requirements for cooling should include consideration of engine seal leakage, engine flange leakage, compressor vent flows, breather flows, engine-mounted oil tank and oil sump heat rejection, accessory drives, and all other sources of heat from power plant operation.

2-3.1.4 Washing. An engine and APU washing capability should be provided through appropriate plumbing lines and connections at easily accessible point(s). Easily accessible point are those which do not require the removal of aircraft panels. This installation should provide for the attachment of a hose from a ground based pressurized liquid container to the on-board connection. A means to motor each engine without ignition should be provided.

2-3.1.5 Mounts. Engine/APU mounts should be detachable, accessible, provide provision for engine/drive train alignment, and offer no interference to engine/APU or associated accessory installations. Engine mounts should react axial, vertical, lateral, and torsional loads as required. The mounts should accommodate maximum flight maneuver and landing loads without failure or any permanent deformation. The primary mount struts should provide mounting redundancy such that the engine will be held securely in place in the event of loss of any of the struts. The engine mounts should allow for engine thermal growth and should provide alignment of the engine with the transmission input module at all times. The mounts should provide engine retention when the engine is subjected to the maximum crash load factors (single axis and combined axis) or sudden engine seizure. The mounting system should prevent structural deflections of the airframe from imposing loads upon the engine.

2-3.1.5.1 Vibration. The engine/APU mounting system must limit the vibrations induced by the airframe rotor combination (including the tail rotor) at critical engine frequencies to levels acceptable to the engine. It must also limit the resonant structural vibration induced by the engine to levels which the airframe structure is designed to accommodate.

2-3.1.6 Access. The aircraft cowlings which close the engine/APU compartments should provide access by removal or by hinging outward. The resulting access should be adequate for engine installation, servicing, maintenance, and removal. The access doors, when open, may double as work platforms to provide easier servicing for the engines and APU. The cowling design should provide positive locking sections such that vibrations and/or deflections should not result in opening or loss overboard during ground or flight operation. In the closed position, the cowling sections should be sealed against the airframe and each other to minimize leakage of the compartment cooling airflow.

2-3.1.7 Engine Bleeds, Vents and Leakage. The high pressure, high temperature airflow exiting the engine constitute heat sources which must be accommodated and accounted for. It may be necessary to duct or diffuse the vent and leakage flows to minimize the potential adverse effects on compartment cooling and to prevent impingement on temperature-limited components such as wire bundles, electrical switches, fuel lines, etc. When the engine utilizes bleeds or vents for design, performance or stability purposes(such as seals, acceleration bleeds etc.), the back pressure restrictions established by the engine manufacturer must be observed and accommodated in the installation design.

2-3.1.8 Survivability/Vulnerability. Tanks and lines within the engine compartment which contain flammable fluids should be located such that any leakage will not flow down upon the engine nor into the engine inlet airstream. In multi-engine aircraft when the engine compartments are adjacent to one another, special efforts are essential to maximize isolation and prevent communication between the compartments. Combustible fluid shut-off valves must be located outside the firewalls to insure their functionality in the event of a fire.

2-3.2 **Air Induction System**

2-3.2.1 Inlets. The air induction subsystem should prevent any erratic or adverse airflow distribution at all operating conditions and attitudes. The air induction subsystem should have minimal aerodynamic losses. A 0.5-1.0% pressure loss should be attainable. The air induction system should not exceed the maximum acceptable engine inlet distortion limits as prescribed by the engine specification. The local total pressure should not differ from the average by more than 5.0%. In addition to acceptable inlet pressure distortion characteristics, the inlet flow at the engine face should not result in local swirl that exceeds the swirl limits of the engine specification. The inlet location should minimize the potential for ingestion of foreign objects, armament parts and/or fragments, or exhaust gases. Engines should be protected from sand/dust, debris, or other foreign objects by the engine IPS. All air induction inlets should be protected/sealed during aircraft nonuse. Inlets and ducts should be free of traps/pockets and configured to prevent accumulation of vapor, fluids, or other contamination.

2-3.2.2 Attachment and Loads. If the engine is directly connected to the inlet duct, the aircraft structural design should be such that excessive loads will not be imposed upon the engine face flange due to aircraft deflections under the maximum flight maneuvering or landing conditions. These same design concerns must be satisfied if a flexible seal is employed to join the inlet duct to the engine front face.

2-3.2.3 Pressure Recovery. The aircraft inlet design usually represents a compromise between maximum recovery at hovering and at maximum forward flight speed. Proper design of the diffuser permits the selection of a reduced inlet capture area and avoids excessive spill drag penalities at high flight speeds. Flow field effects due to aircraft attitude and rotor downwash can adversely impact inlet recovery and the full range of flight conditions, including maximum rate of climb flight, for example, must be considered. In addition to maximizing the pressure recovery, the inlet location on the aircraft is influenced by the flow fields and exhaust plumes during weapons firing, potential for ingesting recirculated engine exhaust gas, and susceptibility to FOD generated by the rotor downwash close to the ground.

2-3.2.4 Pressure Distortion/Swirl. To minimize pressure distortion and swirl, when the engine output shaft is at the front of the engine and the inlet duct must wrap around the shaft or a nose gearbox, the inlet duct must be of adequate length to permit a relatively gradual diffusion and lowering of the velocity of the air entering the engine. These type inlets may employ vanes or splitters in the duct for flow straightening ahead of the engine face. Immediately before the engine face, a very short accelerating section of the inlet duct is often used to reduce flow distortion. Special inlet design features may be required to accommodate the high rate of change of pressure and temperature that can be experienced during ingestion of armament gases caused by weapons firings. The limits for rate of change of inlet pressure and temperature are usually contained in the engine specification.

2-3.2.5 FOD Protection. For non bypass inlet systems, a FOD screen and actuated FOD/ice shield should prevent hard objects larger than 6 mm in diameter and soft objects such as grass, leaves, and rags from entering the engine inlets or the engine air/oil cooler inlets. Screens should either be designed to not accrue ice formation or should be heated to prevent ice accumulation. The FOD screen should be compatible with aircraft RCS design requirements, if applicable. For bypass inlet systems, sufficient direct airflow and/or scavenge/ejector air flow should be provided to assure that foreign objects, including ice, do not enter the engine.

2-3.2.6 Anti-icing. An anti-icing subsystem is required for the engine inlet(s), lips and ducts, at the conditions specified in environmental section of the applicable weapon or the engine system specification. The anti-icing system may be either electrical or bleed air type systems. Either system should maintain a surface temperature of at least 40° F. Provisions should be made to drain melted ice from the interior of the inlet to prevent ice water from refreezing and subsequently being ingested into the engine. The anti-icing system should be capable of operating during all flight and ground operations at all engine power conditions. If operational failure of the anti-icing system occurs, it should remain in, or revert to, the anti-icing mode. One control should activate all ice protection systems. Cockpit indication should verify operation of the anti-icing subsystem.

2-3.3 **Exhaust System**. Attachments should be designed to allow for thermal expansion and tailpipe deflections throughout the aircraft operating envelope. All exhaust outlets should be provided with watertight plugs and/or covers for protection during aircraft nonuse. If the exhaust system includes an infra-red suppresser (IRS), the IRS should:

a. Meet the engine exhaust system hot metal and plume IR signature requirements.

b. Cool and direct engine exhaust such that the system does not represent a fire hazard to ground vegetation or a safety hazard to personnel refueling or re-arming the aircraft with the engines running.

c. Prevent loss of tail rotor efficiency due to hot exhaust gas flowing through the tail rotor.

d. Prevent loss of power due to heating of inlet air and/or reingestion.

2-3.4 **Propulsion System Controls**.

2-3.4.1 Fuel System Controls. The pilot should have complete control of any combination of fuel tanks, including external/internal auxiliary tanks. The engine shut-off fuel feed system and the engine power control off position should be interlocked with the fire extinguisher and fire handle such that movement of the latter shuts off fuel flow both at the engine control and at the firewall shut-off valve. Control of external auxiliary tank fuel, such as transfer and shutoff should be provided in the cockpit display. Control should be provided for any combination of fuel tanks (internal and external). The display should provide both graphic and alphanumeric presentation of the active fuel system, depicting valve position, fuel remaining in each tank, total fuel remaining, fuel boost/prime (if applicable), and transfer in progress. Normal fuel shutoff should be initiated when the engine control lever (ECL) is retarded to the detent (STOP) position in any cockpit station. Emergency fuel shutoff should be initiated in any cockpit at the appropriate fire warning annunciator. Upon indication of engine fire, the annunciator is pressed to effect a shutoff of fuel at the firewall of the engine and at the fuel control valve, and to arm the fire extinguisher for routing to the selected engine. This methodology should permit the crew to maintain eyes up and out of the cockpit during the transition of reducing power to a single engine, and should simplify crew response to pressing a single illuminated push-button to achieve fuel shutoff.

2-3.4.2 APU Controls. APU start/stop control should be provided by a switch in the crewstation. The switch should provide a control signal to the APU engine control unit (ECU) to initiate the engine start sequence. The ECU should provide a signal for crewstation indication when the SPU has a successful start and is operating. The switch should provide a control signal to the ECU and to initiate an orderly shutdown of the APU. The APU should be capable of being started without electrical power (i.e., back-up starting).

2-3.4.3 Starter Controls. The aircraft should be secured via an unique "ignition key" such that, with the key in the "off" position, the ignition source can not be activated. Fuel flow should be controlled via the engine throttle "stop cock" position. This configuration should allow for engine water wash (without ignition spark) and ignition checks (without fuel flow) while maintaining the functionality of the ignition key. Start software/logic should prevent actuation of ignition out-of-sequence; i.e., if fuel flow is initiated by moving the engine throttle with the ignition key in the off position, software should prevent ignition spark should the ignition key be turned to "starter on," thereby preventing hard/hot starts. Cockpit indication should be provided to define a "starter on" condition. Airframe supplied electrical power should be provided for engine starting and as back-up power in the event of an engine electrical power system failure. Automatic restart capability, as part of the propulsion system design, should be provided. Upon achieving APU operation, all cockpit controls and displays should be enabled to initiate and monitor engine start and succeeding levels of operation. Once initiated, the engine start sequence should be automatic, including shutdown in the event a failure or "hot start" is detected.

2-3.4.4 Engine Controls. Each engine should have power control devices that will not allow inadvertent actuation to the "off" position.

2-4.0 **QUALIFICATION GUIDELINES**.

2-4.1 **System Test Guidelines**. Engine/APU installation design criteria should be verified at the aircraft system level IAW ADS-1B.

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SECTION 3: START SYSTEM

3-1.0 REQUIREMENTS.

3-1.1 PERFORMANCE. The Start system shall meet its allocated performance and function in such a manner that the aircraft shall be able to be operated safely throughout the operating envelope and meet the performance requirements as defined in the applicable weapon system specification.

3-1.2 QUALIFICATION. The following qualification requirements for the Start system are required to verify compliance with the performance requirements above as applicable to the aircraft design configuration.

3-1.2.1 Analysis. Start system design and performance analysis shall be documented, using ADS-9C as a guide. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

3-1.2.2 Component Tests. The following tests shall be conducted and a subsequent teardown inspection, to determine the post test condition, shall be performed. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

a. **Cycling, Overrunning, and Endurance**
b. **Valve Compatibility**
c. **Free Run**
d. **Vibration**
e. **Running Engagement**
f. **Environmental (Fungus, Salt Fog, Humidity, Temperature, Icing, Dust)**
g. **Dielectric Strength**
h. **Containment**
i. **Attitude**
j. **Structural Load**
k. **Emergency Disengagement Mechanism**
l. **Proof and Burst Pressure**
m. **Leakage**
n. **Explosion Proof**

3-1.2.3 Assembly/System Level Tests. Aircraft system level tests shall be conducted in accordance with ADS-1B-PRF.

3-2.0 **REFERENCES**.

3-2.1 Commercial Specifications.

AIR 713A	Determining, Presenting, And Substantiating Turbine Engine Starting And Motoring Characteristics, Guide For

3-2.2 Military Specifications.

ADS-1B-PRF	Rotorcraft Propulsion Systems Airworthiness Qualification Requirements, Ground and Flight Test Surveys and Demonstrations
ADS-2	Starters and Starter-Generators, 30 Volts, Direct Current, Aircraft Engine Driven
ADS-9C	Propulsion System Technical Data
MIL-H-5606	Hydraulic Fluid, Petroleum Base; Aircraft, Missile And Ordnance
MIL-STD-810	Environmental Engineering Considerations and Laboratory Test, Test Method Standard For

3-3.0 **PERFORMANCE GUIDELINES**.

3-3.1 **Environmental Conditions.** A self-contained starting capability should be provided for the engines and APU. The starting system should be capable of providing adequate starts with any combination of fuels with fuel no colder than the temperature corresponding to a fuel viscosity of 12 Centistokes, MIL-H-83282 hydraulic fluid (or MIL-H-5606 hydraulic fluid in winterization kit), and MIL-L-7808, DOD-L-85734, and MIL-L-23699 oils and under the worst case operating conditions specified for the aircraft in the applicable weapon system specification.

3-3.2 **Provisions.** The starting system should include provisions for starting from an external power source (Mobil Electric Power 360A, AGPU) or being started by other companion aircraft (buddy start). The starting system provisions for starting from an AGPU should include a pneumatic/pressurized air receptacle and an external electrical power receptacle. An AGPU power adapter cable (peculiar support equipment) is required to interface between AGPU and aircraft external power receptacle. Buddy start electrical receptacles should be provided.

3.3.3 **Characteristics.** The airframe should provide any additional interface signals, as necessary, to accomplish automatic engine starting and ensure compatibility with the engine control system. Starter(s) should be capable of providing adequate torque/speed characteristics for engine/APU starting and motoring requirements. AIR 713A should be used as guidance where applicable. Engine starting torque requirements should be IAW with the applicable engine specification. The start envelope (ambients, altitudes) for the engines and APU should be IAW the applicable weapon system specification. Start time lines for normal and emergency starts should be IAW with the applicable weapon system specification. Automatic restart capability should be provided by the engine IAW the applicable engine specification.

3-3.4 **Pneumatic Starters**. The starters covered by this document should be designed to operate on compressed air from ground support equipment, engine interbleed or on board air supply, for the purpose of starting aircraft jet engines. The starter should be constructed to withstand the strains, jars, vibrations, and other conditions incident to shipping, storage, installation, and service usage.

3-3.4.1 Overspeed Protection. The starter should be capable of containment of all fragments within its envelope and remain on its mount should a failure occur when the starter is supplied with the maximum combination of pneumatic inlet conditions of pressure and temperature possible in the system from sea level to the maximum aircraft operation altitude. Fragments may be emitted from the starter’s exhaust provided they do not constitute a fire hazard or contain sufficient energy to harm equipment, structure, or personnel. In addition, any rotation of the starter case which could shear ducts or other attachments should not be permitted.

3-3.4.1.1 For starters restricted solely to ground operation, induced turbine rotor hub burst containment demonstration should be required at a speed not less than the maximum operating speed of the starter under normal operating conditions. (Maximum starter cut-out speed.)

3-3.4.1.2 For starters to be used in flight, induced turbine rotor hub burst containment demonstration should be required at maximum free run speed, assuming starter control valve failure when being supplied with maximum pneumatic inlet conditions. The starter should be capable of containing all fragments and remain on its mount should a free-run failure occur, with cessation of rotation at the conditions that produce maximum free run speed, without possibility of the spread of fire because of structural failure, overheating, leaking flammables or other causes. Also in the event the unit is allowed to free run until a turbine bearing failure occurs, a means should be provided to prevent an axial shift of the turbine wheel that would negate the containment provisions.

3-3.4.2 Rotor Integrity

3-3.4.2.1 Low Cycle Fatigue. The starter rotor should be designed and constructed to minimize the probability of low cycle fatigue. As a minimum, the rotor hub should be designed to withstand cyclic stresses which occur when cycling up to the maximum normal operating speed with maximum inlet temperature, for two times the number of start cycles which can occur throughout the operational life of the starter. The rotor hub should also withstand 25 percent of the number of operational life start cycles under maximum free-run conditions. In those applications where the starter is also required to operate from different types of gas i.e., AGE compressed air, monopropellant and engine bleed air, the rotor hub should be designed to withstand these normal operating and free running requirements under the maximum conditions occurring with each type of gas. The number of life start cycles with each type of gas used for design and test purposes should be specified in the detail specification. The number of design operational life start cycle occurring with compressed air should not be less than 10,000.

3-3.4.2.2 High Cycle Fatigue. The starter rotor should be designed such that it is capable of operation to infinite life up to the maximum normal operating speed without fatigue failure. It should be demonstrated that high cycle fatigue stress is below the endurance limit of the material. For demonstration purposes, infinite life may be considered as 10^7 cycles for ferrous material and 10^9 for non-ferrous material. The starter rotor should also be designed to operate for two minutes of maximum free run speed per cycle for 25 percent of the operational life start cycles. In those applications where the starter is also required to operate from different types of gas, i.e. ACE compressed air, engine bleed air and monopropellants, the rotor should withstand free running under the maximum conditions of each type of gas for a duration that is equal to 2 minutes per cycle for 25 percent of the operational life start cycles occurring with each gas.

3-3.4.3 Engaging Mechanism Failure. Unless it can be shown that the starter engaging mechanism is fail safe, the unit should incorporate at least one of the provisions defined below to prevent a hazardous condition in the event the starter rotor is driven by the engine due to a failure of the engaging mechanism:

a. An automatic emergency starter drive disengagement mechanism should be incorporated external of the starter gear section to effect emergency disengagement of the starter from the engine in the event normal disengagement does-not occur. The disengaging mechanism should be so designed that removal of the starter from the engine pad should be required to effect re-engagement of the starter drive. An attempted start, after an emergency disengagement has been experienced, should not cause damage to the starter or the engine.

b. Containment of a maximum energy tri-hub burst (3-piece, 120° segments) should be provided up to the starter rotor burst speed or to maximum speed at which the engine can drive the starter rotor due to a failed engaging mechanism.

c. The starter rotor should incorporate rim and blade fusing such that at a predetermined speed the rim and blades should separate from the turbine hub. The minimum fuse burst speed should be greater than the maximum free run speed and the fused parts should be contained at the fuse burst speed. Containment should also be provided for the remaining rotating parts up to the maximum possible driven speed.

3-3.4.4 External Surface Temperature. The external surface temperature of the starter should not exceed 371°C (700°F) during or after any operating condition throughout the ambient temperature range specified herein. Provisions necessary to meet this guideline should be integral with the starter and should not require external covering or insulating blanket(s).

3-3.4.5 Flight and Ground Loading Conditions. The starter should withstand, without permanent deformation or failure, the largest forces resulting from all critical combinations of loads and rotational accelerations. For design purposes, minimum yield strength should provide for at least 1.5 times the largest forces resulting from the loading conditions and a minimum ultimate strength of at least 1.67 times the largest forces resulting from the loading conditions. The design limit torque load should be the highest value of torque specified in the model specification and should also include the torque loading conditions resulting from turbine wheel seizure during containment. For design purposes, the weight of the starter should include all components and parts that make up the complete starter assembly.

3-3.4.5.1 External Loads. The load factors and rotational accelerations specified below are to be considered acting separately and in combination with the design limit torque load and overhung moment:

a. Applicable loads and accelerations resulting from maximum flight maneuvers and ground loading conditions of 10g, unless otherwise specified in the detail specification, acting in any direction through the center gravity of the starter.

b. Duct attachment load. Unless otherwise specified in the component detail specification, the starter duct connecting flange(s) should be capable of withstanding, without permanent deformation or failure, a force of 222N (50 pounds) direct thrust acting either inward or outward or a moment of 16.9 NM (15O inch pounds) acting in any direction around the axial centerline of the flange.

3-3.4.6 Dielectric Strength. All parts of electrical equipment, except critical components such as transistors and capacitors, should be capable of withstanding, at commercial frequency, a voltage of 1,OOOV (rms) plus twice the working voltage. Transistors and capacitors should withstand twice the peak voltage to which they will be subjected during service or 100V, whichever is greater, for a period of 1 minute.

3-3.4.7 Starter Adapter Flange. Unless specified otherwise in the component detail specification, the starter should incorporate a quick-attach-detach (QAD) type of mounting flange. The adapter should provide for indexing and torsional restraint of the starter. The QAD should not require special tools for its use. Any adapter required to modify the engine accessory drive in order to mount the starter or to make the starter compatible with the engine drive, should be furnished with the starter. Mounting of the starter should be accomplished without requiring any measurements or adjustments of the engine accessory drive or starter prior to installation. The device for actuation of the QAD mounting should be automatically safetied when not in use.

3-3.4.8 Overhung Moment. Maximum overhung moment of the complete starter, filled with lubricant, should be specified in the model specification and should not exceed that specified in the detail component specification when measured from the face of the mounting flange. The location of the starter's center of gravity, weight, and overhung should be specified on the starter vendor's outline drawing.

3-3.4.9 Polar Moment of Inertia. The starter effective mass polar moment of starter drive should be specified in the model specification.

3-3.4.10 Safety. All threaded connections should be locked to prevent loosening in service by means of self-locking nuts or other methods. The use of lockwashers or staking is prohibited if used as a primary locking device.

3-3.4.11 Drainage. The starter should incorporate provision for drainage or be sealed to prevent water from accumulating within the unit. Drainage should be such that any condensed or accumulated water will not cause malfunction or cause delay in operation if frozen.

3-3.4.12 Fitting Identification. The starter should be permanently marked to indicate pneumatic connections, fill and drain ports, electrical connections, and safety devices such as burst diaphragms and safety valves.

3-3.4.13 Index Marking. All components of the starter, such as the pneumatic plenum and gear train housing, should be marked relative to each other to prevent mispositioning of the components with respect to the mounting pad during starter assembly.

3-3.4.14 Cover Plates. Cover plates or plugs, suitable for transient or storage conditions, should be provided for all openings on the starter.

3-3.4.15 Overhaul Life. The unit should be capable of performance throughout 2000 start cycles and 3000 hours of overrunning (normal aircraft flight operating time) or as specified in the detail specification.

3-3.4.16 Reliability. The starter should be designed and developed to achieve the highest operational reliability commensurate with the weapon system design specification.

3-3.4.17 Performance Characteristics. The starter, after being subjected to the tests specified in paragraph 4 below, should meet the minimum performance guidelines specified herein when operated either with or without a starter control valve. The actual performance characteristics should be specified in the component specification and should be predicated on the use of production hardware with adequate allowance for tolerance variations. Performance characteristics should be based upon the pneumatic inlet conditions listed below:

INLET CONDITIONS TO THE STARTER:

- Rated
- Maximum Temperature
- Minimum Pressure
- Maximum Pressure
- Maximum Engine Interbleed

3-3.4.17.1 Starter Output Torque. The starter output torque versus rpm should be as measured at the output drive of the starter for each inlet condition defined in the detail specification. The minimum torque output when defined in the model specification should not be less than that required in the detail specification.

3-3.4.17.2 Duty Cycle. The starter should be capable of making consecutive start cycles when expose to the environmental conditions specified herein with a maximum interval of 60 seconds between the completion of one cycle and the beginning of the next cycle. In addition, the starter should be capable of motoring the engine for a minimum of 5 minutes followed by a 5-minute rest period. A starting cycle should consist of unit initiation and acceleration of the output drive shaft from zero rpm to cutout speed. Duty cycles and engine motoring capability of the unit should be specified in the model specification.

3-3.4.17.3 Automatic Starting. The starter should be so designed that initiation of a single switch-or other device should provide automatic starter operation from initiation-to starter cutout. Initiation of the automatic starting cycle should be from the aircraft cockpit. The automatic starting provisions should be specified in the model specification.

3-3.4.17.4 Stopping. It should be possible to terminate the starting cycle at any time during the start cycle without damage to the starter.

3-3.4.17.5 Running Engagement. The starter should suffer no detrimental effects when running engagements at any speed up to cutout rpm.

3-3.4.17.6 Performance Curves. The starter performance should be defined by the following curves which should be part of the model specification. The curves should include minimum and maximum performance limits. Performance should be defined for each pneumatic inlet condition over the complete altitude and temperature range specified in the detail specification.

a. Starter output torque versus output shaft speed.

b. Typical torque transients at starter output drive versus time showing maximum impact torque.

c. Starter maximum no load speeds versus time.

d. Typical starter output torque at various pressure and temperature inlet conditions.

e. Starter airflow consumption at each inlet condition versus speed.

f. Starter output efficiency at each inlet condition.

3-3.4.17.7 Initial Calibration. Initial calibration should be conducted after the break-in run and before initiation of the scheduled test program. The starter should not be operated between completion of the initial calibration runs and start of the scheduled test program.

3-3.4.17.8 Performance Deterioration. Performance deterioration should not be more than 5 percent of the initial calibration values throughout 2000 cycles of operation. Performance during initial calibration of the unit should meet or exceed the minimum model specification requirements.

3-3.4.17.9 Noise Level. The starter should operate such that under all operating conditions the discrete frequency and broad band noise components should be minimized.

3-3.4.18 Environmental Conditions. The starter should suffer no detrimental effects and should operate satisfactorily during and after exposure to the environmental conditions specified herein or any natural combination of environments as might be anticipated in world-wide operating conditions.

3-3.4.18.1 Altitude. The starter should meet the performance guidelines of this specification from sea level to the maximum starter operating and maximum aircraft operational (starter overrunning condition) altitudes specified in the detail specification.

3-3.4.18.2 Temperature. The starter should meet the guidelines of this specification when subjected to the extreme thermal conditions of its installation that are associated with the following temperature conditions:

a. Ambient temperature operating range -54°C to 149°C (-65°F to +300°F).
b. Exposure temperature range -73°C to 201°C (-100°F to +400°F).

3-3.4.18.3 Humidity. The starter should meet the performance guidelines of this specification and should suffer no detrimental effects when exposed to any relative humidity conditions from 0 to 100 percent within the temperature range of -54°C (-65°F) to 71°C (160°F) including conditions wherein moisture freely condenses on the starter.

3-3.4.18.4 Salt fog. The starter should meet the guidelines of this specification and should suffer no detrimental effects during and after exposure to salt laden moisture.

3-3.4.18.5 Sand and Dust. The starter should meet the guidelines of this specification and suffer no detrimental effects during and after exposure to sand and dust laden air as may be encountered in desert operation.

3-3.4.18.6 Fungus. The starter should suffer no detrimental effects from being exposed to fungus growth as encountered in tropical climates.

3-3.4.18.7 Explosive Atmosphere. Operation of the starter in an explosive atmosphere should not create a hazard. The details necessary to meet this guideline should be defined in the model specification.

3-3.4.18.8 Vibration. The starter should withstand without damage the vibration environment at the engine mounting pad.

3-3.4.18.9 Attitude. The starter should be capable of meeting the starting guideline of this specification in attitudes from 90° down to 105° up and 20° of roll in either direction.

3-3.4.19 Details of Components.

3-3.4.19.1 Turbine Characteristics. The maximum operating and free running speeds of the turbine wheel, at sea level and maximum operating altitude should be specified in the detail component specification. The minimum yield speed of the turbine should be greater than the maximum free running speed. The turbine minimum burst speed should be specified in the detail component specification. The turbine proof speed should be less than the minimum yield speed and greater than the maximum free running speed.

3-3.4.19.2 Speed Reduction. The gear ratio between the energy conversion mechanism and the starter output shaft should be specified in the detail component specification. The amount of gear backlash should be the minimum consistent with the application and should be specified in the detail component specification.

3-3.4.19.3 Clutch and Output Assembly. An engage-disengage clutch should be provided which will automatically engage or disengage the starter from the engine under all engine starting conditions. The clutch operating limits should be specified in the detail component specification. In the event a slip clutch is used, the operating limits should be specified in the detail component specification.

3-3.4.19.4 Output Shaft. The starter output shaft should be replaceable without disassembly of the starter. The maximum starter output shaft torque and direction of rotation should be defined in the detail component specification, and the actual maximum torque should be specified. A shear section should be provided as part of the starter output shaft. The shearing torque for new shafts and for shafts that have accumulated life cycle fatigue degradation should not be less than 80 percent nor more than 90 percent of the shearing torque of the engine starting drive train shear section. The maximum applied torque value at which shear will occur should be specified in the detail specification. The maximum and minimum values at which the shaft will shear should be specified in the model specification.

3-3.4.19.5 Mounting Flange and Drive. The starter mounting flange and drive spline should conform to the detail specification requirement and should mount on the corresponding engine accessory drive or remote gearbox. The accessory drive or remote gearbox type number should be specified in the model specification.

3-3.4.19.6 Control System. The starter should be provided with a speed responsive cutout device which will terminate the power supplied to output shaft at that speed at which the engine no longer requires starter assist (-O percent, +10 percent). The cutout device should be defined in the model specification.

3-3.4.20 Starter Control Valve. The starter control valve should not be a part of the starter and should not be furnished with the starter unless specified by the procuring activity. However, if a valve is required in the starting system, the starter control valves should be one of the following two types:

Type I: Regulating (pressure).

Type II: Non-Regulating (on-off).

3-3.4.20.1 Design. The valve should be designed to automatically close in the event of loss of electrical power or pneumatic energy source. Manual override, remote position indication, and other such features should be specified.

3-3.4.20.2 Flight and Ground Loading Conditions. The valve should withstand, without permanent deformation or failure, the largest forces resulting from ground loading conditions and the largest forces resulting from all critical combinations of maximum flight maneuver conditions specified, acting in any direction through the center of gravity of the valve. In addition, unless otherwise specified, the valve's air inlet connecting flanges should be capable of withstanding, without permanent deformation or failure, a force of 222N (50 pounds) direct thrust acting either inward or outward or a moment of 169NM (150 inch pounds) acting in any direction around the axial centerline of the flange.

3-3.4.20.3 Safety. All threaded connections should be locked to prevent loosening in service by self-locking nuts or other methods. Lockwashers or staking, if used as a primary locking device, should not be used.

3-3.4.20.4 Drainage. The valve should incorporate provisions for drainage or be sealed to prevent water from accumulating within the unit. Drainage should be such that any condensed or accumulated water will not cause malfunction or cause delay in operation if frozen.

3-3.4.20.5 Direction of Flow. The direction of air flow through the valve should be prominently and permanently marked on the valve housing.

3-3.4.20.6 Snap Ring Retainer. The use of snap ring retainers is prohibited except where failure of the retainer will not affect operation of the valve.

3-3.4.20.7 Pneumatic Bleed. Bleed ports and passages should be as large as is practical to preclude blockage by contamination.

3-3.4.20.7.1 Continuous Bleed. If the valve incorporates any requirement for continuous bleed, it should be considered as a part of the total valve leakage as defined in the detail specification.

3-3.4.20.7.2 Filter. If filters are required, the maximum opening should be sufficient to protect the valve from contamination. The filter should be of sufficient size to minimize the need for cleaning and should be readily removable for replacement or cleaning without performing a major overhaul on the valve.

3-3.4.20.8 Electrical Guidelines. The valve should meet the guidelines of this ADS with electrical power provided by the Aircraft. Steady state operating or holding current of the unit should be specified.

3-3.4.20.8.1 Solenoid Preload. The solenoid armature and pilot valve parts should be designed to provide sufficient preload in the closed position to overcome the forces associated with the environmental conditions (vibrations, etc.) specified . If pressure assist is utilized in accomplishing this, the conditions must be met with the minimum operating pressure specified

3-3.4.20.8.2 Dielectric Strength. All parts of electrical equipment, except critical components such as transistors and capacitors, should be capable of withstanding, at commercial frequency, a voltage of 1,OOOV (rms) plus twice the working voltage. Transistors and capacitors should withstand twice the peak voltage to which they will be subjected during service or 1OOV, whichever is greater, for a period of 1 minute.

3-3.4.20.9 Lubrication. The valve should not require relubrication during the overhaul life.

3-3.4.20.10 Position Indicator. The valve should be equipped with an external position indicator which should include suitable markings to indicate the valve (open) and (closed) positions.

3-3.4.20.11 Proof and Minimum Burst Pressure. The valve should meet the operational guidelines of this ADS and should not have evidence of damage or permanent deformation after having been subjected to a proof pressure equal to two times the maximum operating pressure specified applied at the inlet port with the valve in both the open and closed positions and while at maximum operating temperatures. Minimum burst pressure should be one and one-half times the proof pressure.

3-3.4.20.12 Performance Characteristics. The valve, after being subjected to the tests specified in paragraph 4.0, should meet the operating guidelines specified herein. All performance characteristics should be based upon the use of production hardware.

3-3.4.20.12.1 Valve Operation. The valve should be normally closed with the inlet side of the valve pressurized to any inlet condition as specified. Application of electrical power to the unit should cause the valve to open. The valve should remain open as long as pressure in maintained and the electrical circuit is energized.

3-3.4.20.12.2 Opening-Closing Time. With the inlet side of the valve pressurized to any inlet condition specified, and when the electrical circuit to the valve is energized, the valve should move from full closed position to the required regulating pressure if the valve is the regulating type, or to the full open position for the nonregulating type. With the valve inlet pressurized to the maximum pressure inlet condition, the rate of pressure rise should not exceed that specified. Unless otherwise authorized by the procuring activity, the valve should not take longer than 2 seconds from time of electrical actuation to reach the required open position at any ambient or condition of pressure and temperature. When the electrical circuit is interrupted, the valve should close in not less than .4 seconds nor more than 2 seconds under any condition.

3-3.4.20.12.3 Breakaway Torque. Available breakaway torque of the flow controlling mechanism at normal operating pressure should be at least 20 percent higher than that required, after accounting for the most severe mechanism icing condition that could be encountered in service.

3-3.4.20.12.4 Duty Cycles. The valve should be capable of continuous cycles of 60 seconds open followed by 60 seconds closed and also 5 minutes open followed by 5 minutes closed.

3-3.4.20.12.5 Endurance. The valve should be capable of sustaining 10,000 cycles without overhaul or replacement of parts, and should operate within the design limits when subjected to the Cycling and Endurance Test.

3-3.4.20.12.6 Explosion Proof. The valve should be made explosion proof (it should not cause ignition of an ambient explosive gaseous mixture).

3-3.4.20.12.7 Temperature Extremes. The valve should meet the guidelines of this ADS when subjected to the extreme thermal conditions of its installation that are associated with the following temperature conditions:

a. Ambient temperature operating range -54°C to +149°C (65°F to +300°F).

b. Exposure temperature range -73°C to 204°C (-100°F to +400°F).

3-3.4.20.12.8 Performance Curves. The valve performance should be defined by the following curves:

a. Curves showing maximum pressure rise versus time at each specified inlet condition at sea level.

b. For pressure regulating valves, curves showing maximum peak pressures and regulated pressures (with allowable tolerance limits) for each specified condition of inlet pressure greater than the regulated pressure.

c. Curves showing variations in pressure rise time and regulated pressure with altitude and temperature.

3-3.4.20.13 Environmental Conditions. The valve should suffer no detrimental effects and should operate satisfactorily during and after exposure to the natural and induced environments including altitude, temperature, temperature shock, rain, salt, fog, dust, fungus, humidity, vibration, electromagnetic interference, shock, and other environmental conditions as anticipated in service.

3-3.4.20.14 Interchangeability. All parts having the same manufacturer's part number should be functionally and dimensionally interchangeable. Matched parts or selective fits will be avoided wherever possible; however, where required these parts should be serialized.

3-3.4.20.15 Lubrication System. The lubricant used, whether grease or oil, should be specified in the model specification. The lubricating system should be integral with the unit and should be adequate to lubricate the starter throughout its operating range. The starter design should be such that no change in lubricant should be required for operating throughout the ambient temperature range specified in the detail specification. The starter should be capable of completing the number of starting cycles and hours of overrunning required by the detail specification without changing or adding to the lubricant.

3-3.4.20.15.1 Oil Supply. If oil is used as a starter lubricant, the oil reservoir should be furnished as a component part of the starter. The consumption rate should be specified in the model specification. It should not be necessary to disassemble the starter or remove it from the engine in order to refill the reservoir. The lubricant used should be in accordance with MIL-L-7808 unless otherwise specified. The minimum and maximum amount of oil necessary for satisfactory starter operation should be specified in the model specification. Positive engine or gear-box oil lubrication should be provided to the spline. The spline should include an appropriate seal to prevent loss of lubricating oil.

3-3.4.20.15.2 Oil Loss. All sources of oil loss from the starter and their rate of oil loss should be specified in the model specification.

3-3.4.20.15.3 Oil Flow and Heat Rejection. In the event pressure lubrication is utilized, the oil flow and heat rejection rates should be specified in the model specification.

3-3.4.20.15.4 Oil Filter. A suitable oil filter element, if required, should be provided as a component of the lubrication system and should be specified in the model specification.

3-3.4.20.15.5 Oil Fill and Drain Provisions. Suitable oil fill and drain provisions should be provided. The unit should be adequately marked as to capacity and type of oil to be used. A magnetic type oil drain plug should be provided.

3-3.4.20.15.6 Oil Level Measurement. The oil reservoir of the starter should be provided with a means for manual measurement of the oil level. The method of oil measurement should be defined in the model specification.

3-3.5 **Hydraulic Starters**. The starters covered by this document are classified as Type I and Type II. Type I starters are designed to operate as a part of a dual pressure, nominal 3000-psi and 4000-psi hydraulic system. They should incorporate a flow-limiting device and should be capable of operating satisfactorily from power sources having unlimited flow and from power sources limited to flow as specified in the applicable specification sheet and herein. Type II starters should be designed to operate as a part of a nominal 4000-psi flow-limited hydraulic system. Type I and II hydraulic starter should be designed to conform to the hydraulic system components as defined in section 8 of this ADS and should be designed to operate in hydraulic systems conforming to the guidelines defined in section 8 of this ADS.

3-3.5.1 Environmental Conditions. The hydraulic starter should be capable of being operated, without impairment of function or change in adjustment, throughout the ambient temperature range specified in the detailed component specification and should not be damaged when exposed to temperatures specified thereon. Consideration should be given in the design of the hydraulic starter to satisfactory operate during and after exposure to any of the following conditions in worldwide operation:

a. Humidity
b. Salt fog
c. Sand and dust
d. Fungus
e. Mildew

3-3.5.1.1 Acceleration Forces. While mounted rigidly during non-operating periods, starters should be capable of withstanding non-vibratory linear acceleration forces of 10 g applied at the starter center of gravity in any direction, without structural damage. Starters should be capable of normal operation after exposure to this force. During starter operation, starters should not generate acceleration forces of sufficient magnitude to exceed any of the design requirements of the respective engine accessory drive pad and mounting pad.

3-3.5.1.2 Vibration. Hydraulic starters should be inherently balanced and designed to meet the vibration test guidelines specified herein.

3-3.5.2 Operating Fluid. The hydraulic starter should operate using hydraulic fluid conforming to MIL-H-5606 or MIL-H-83282. Temperature of the fluid should be as encountered as the result of ambient temperature specified in the detailed component specification, but not higher than 275°F maximum. The acceptable contamination level should be that provided by the continuous filtration of the starter inlet fluid through a 15-micron absolute filter.

3-3.5.3 Component Parts. The starter should consist of a hydraulic motor suitable for converting fluid energy into mechanical energy in the form of rotary motion. Variable displacement units should be equipped with suitable devices to provide a controlled change of displacement during the operating cycle. In addition, the starter should incorporate a suitable clutch for coupling the starter drive with the engine drive, a shear section or slip clutch in the output drive, and an automatically operated cutout device. Devices for limiting input power and speed should be supplied with Type I starters and for Type II starters.

3-3.5.4 Leakage.

3-3.5.4.1 Drive Shaft Seal. Where applicable, drive shaft seals should be designed so that positive leakage, past the seal, should not exceed the maximum leakage rates permitted in the detailed component specification.

3-3.5.4.2 Static Seal. Leakage in the form of drops from the starter housing or from static seals should not be permitted.

3-3.5.5 Functional Guidelines. The starters should be designed and constructed to satisfy the functional guidelines specified herein when operating under rated conditions and under conditions specified on the detailed component specification.

3-3.5.5.1 Rated Conditions and Cutout Speed. Rated conditions and cutout speed for the starters are as defined herewith and are specified on the detailed component specification.

3-3.5.5.1.1 Rated Inlet Pressure. The rated inlet pressure is the maximum pressure required at the starter inlet port when operating at rated torque, rated outlet pressure, and rated temperature. The inlet pressure at zero fluid flow should be as specified on the detailed component specification.

3-3.5.5.1.2 Rated Outlet Pressure. The rated outlet pressure is the pressure at the starter outlet port against which the starter is designed to produce rated torque at rated inlet pressure and rated temperature.

3-3.5.5.1.3 Rated Inlet Fluid Temperature. The rated inlet fluid temperature is the maximum fluid temperature at the starter inlet port when operating at rated torque and rated pressures.

3-3.5.5.1.4 Rated Fluid Flow. The rated fluid flow is the maximum fluid flow needed to meet the torque guidelines at any starter drive speed from zero rpm to cutout speed.

3-3.5.5.1.5 Rated Torque. The rated torque is the minimum torque at the starter drive when operating at rated pressures, rated temperature, and rated speed.

3-3.5.5.1.6 Rated Speed. The rated speed is the maximum starter drive speed at which rated torque is produced and is usually related to engine light-off speed.

3-3.5.5.1.7 Cutout Speed. The cutout speed is the starter drive speed at which the cutout device will signal the termination of starter operation.

3-3.5.5.2 Engagement. Engagement of the starter drive with the engine accessory drive should be accomplished with a suitable clutch engagement, with minimum shock, and independently of any external signal, provided sufficient hydraulic energy is available to cause rotation of the hydraulic starter.

3-3.5.5.3 Disengagement and Re-engagement. Disengagement should be rapid and complete at the end of a starting cycle. Re-engagement of the starter with the engine should be accomplished rapidly and without damage to starter or engine even though the starter or engine is rotating. Disengagement and re-engagement of the output shaft should be independent of any external signal.

3-3.5.5.4 Rotation. The rotation of the starter drive should be as specified on the detailed component specification.

3-3.5.5.5 Torque Limiting Device. Starters should incorporate a slip clutch or shear section to prevent damage to the engine in the event of excessive starter torque. The output drive should be designed so that if failure of the output drive occurs, the resulting parts will be retained within the starter and engine accessory drive.

3-3.5.5.5.1 Shear Sections. The shear section should be designed to permit replacement without disassembly of the starter and to limit axial movement of the output shaft. The maximum permissible static shear torque to be transmitted to the engine by the starter drive under all operating conditions, including engagement and re-engagement, should not exceed the value specified on the detailed component specification.

3-3.5.5.5.2 Slip Clutch. Slip clutch, when provided, should have a maximum breakaway torque that will prevent the starter from transmitting torque to the engine starter drive in excess of the values specified in 3-3.5.5.5.1 under all conditions specified herein.

3-3.5.5.6 Lubrication. The starter should be self-lubricating with hydraulic fluid. Overrunning clutches should be designed to incorporate adequate lubrication to permit continuous operation at the design conditions and at the attitude and environmental conditions. Only the specified hydraulic fluid should be used to lubricate seals during assembly of the starters.

3-3.5.5.7 Automatic Shutoff Control. The starter should be equipped with a cutout switch that will function in the manner and at a starter output shaft drive speed specified on the detailed component specification. The switch should be designed to withstand the electrical load specified on the detailed component specification. The switch cutout speed setting should be externally adjustable on the assembled starter and when the starter is installed on the mating accessory mounting pad. The switch should not be polarity sensitive.

3-3.5.5.8 Overrunning. The starter should not be damaged as result of overrunning under the conditions specified on the applicable specification sheet with the maximum misalignment allowed for the engine accessory drive.

3-3.5.6 Performance.

3-3.5.6.1 Output. The starter should meet the output torque characteristics at rated inlet hydraulic fluid temperature as specified on the detailed component specification.

3-3.5.6.2 Overspeed Limit. Type I starters should be designed to prevent overspeed when subjected to unlimited flow with the cutout speed switch inoperative. Type II starters should be designed to prevent overspeed when subjected to the flow rate as specified on the applicable specification sheet with the cutout speed switch inoperative. The no-load speed of the output shaft of the starters at the above conditions should be as specified on the detailed component specification.

3-3.5.6.3 Hydraulic Fluid Flow. When operating at rated inlet and outlet conditions, the fluid flow should be as specified on the detailed component specification at the speeds indicated thereon.

3-3.5.6.4 Pressure Application. The rate of pressure application to the starter should be as specified in Figure I herein.

3-3.5.6.4.1 Surge Pressures. During operation, starters should be capable of withstanding inlet surge pressures not to exceed 4500 psig and outlet surge pressures not to exceed 400 psig. Inlet and outlet surge pressures may occur simultaneously or separately and should not exceed one second in duration.

3-3.5.6.4.2 Proof Pressure. At rated speed, torque and inlet temperature, the starter should be designed to operate at the inlet pressure and outlet pressure specified on the detailed component specification.

3-3.5.6.5 Sustained Motoring. At rated inlet conditions, the starter should produce and maintain rated torque and speed for the time specified on the detailed component specification.

3-3.5.6.6 Succession of Starts. There should be no limitation of sequence or number of starts performed in succession at starter rated conditions. The starter should meet the endurance test specified below.

3-3.5.6.7 No-load Speed. The starter should be capable of running without damage at rated inlet conditions for the period specified on the detailed component specification with the starter output shaft unloaded and the cutout switch inoperative.

3-3.6 **ELECTRIC STARTERS**. Electric starter design/performance and qualification guidelines are contained in ADS-2.

3-4.0 **QUALIFICATION GUIDELINES**.

3-4.1 **Test Conditions**. Unless otherwise specified, the following conditions should apply for all tests.

3-4.1.1 Temperature. Ambient temperature tests should be conducted at the conditions existing within the test cell at the time of test. Low and high temperature tests should be conducted at the limits of the operating environment as specified in the applicable weapon system specification.

3-4.1.2 Pressure. Unless otherwise specified herein, all tests should be conducted at ambient atmospheric pressure.

3-4.1.3 Inlet Conditions. Unless otherwise specified, all pneumatic cycling tests should be conducted at rated inlet conditions. The minimum and maximum pressure and temperature parameters for the inlet conditions should be as specified in the detail component specification for the applicable starter installation. Pressure and temperature limits for all acceptance and calibration cycles should be $\pm$.07 Bar ($\pm$1 psia) and $\pm$8.4°C ($\pm$15°F).

3-4.1.4 Conditioning Time. Conditioning time for the starter should be such that all parts should have reached a temperature within $\pm$2.8°C ($\pm$5°F) of the specified temperature. A starter should be considered conditioned when it has been continuously exposed to the specified temperature for the conditioning time, making suitable allowance for the starting temperature. During the conditioning time, the temperature of the conditioning chamber should not vary more than $\pm$2.8°C ($\pm$5°F) from the specified temperature. The starter should be maintained at the low temperature for a period of 72 hours.

3-4.1.5 Transfer Time. The starter and component transfer time should not exceed 5 minutes.

3-4.1.6. Accuracy. For all starter and component tests, the test apparatus should be such as to insure that recorded parameters will have the accuracy specified in MIL-STD-810. Other parameters will have a steady-state accuracy within ±2 percent, except that the weight should be accurate within ±0.2 percent. All apparatus should be calibrated as required to insure that this degree of accuracy is maintained.

3-4.1.7 Recalibration. The point at which an unusual difference occurs in any parameter should be recorded and the starter should be recalibrated. If this point occurs prior to the last cycle of testing, the test should be stopped.

3-4.1.8 Test Flywheels. Two flywheels should be used in the preproduction test. They should have different moments of inertia as described below and should be referred to as flywheel A and flywheel B.

3-4.1.8.1 Flywheel A. Flywheel A should have a polar moment of inertia which will subject the starter to a loading condition similar to a normal engine starting cycle. This polar moment of inertia should be defined in the detail specification and should be specified in the model specification.

3-4.1.8.2 Flywheel B. Flywheel B should have a polar moment of inertia which will subject the starter to a loading condition similar to an engine exhibiting early light off and rapid acceleration after light off. This polar moment of inertia should be defined in the detail specification and should be specified in the model specification.

3-4.1.9 Starter Attitude During Testing. All starter tests should be conducted with the starter mounted in its installed operational attitude. The mounting attitude should be described in the model specification.

3-4.2 **Test Design.** Tests should be designed to demonstrate complete compliance with all the requirements of paragraph 3-1.0 above. The following test schedule is provided as an example outline for the tests to be conducted three test units. Specific tests required to demonstrate compliance with each requirement should be integrated into the individual unit test schedules.

3-4.2.1 Test sample number one:

a. Calibration.
b. Endurance cycling (at all operating and all inlet conditions - 2000 cycles minimum).
c. Valve compatibility (at maximum pressure inlet conditions - 100 cycles minimum).
d. Free-running.
e. Vibration.
f. Recalibration.
g. Teardown and inspection.
h. Containment demonstration.

3-4.2.2 Test sample number two:

a. Calibration.
b. Endurance cycling(at all operating and inlet conditions - 500 cycles minimum).
c. High temperature cycling overrunning test (1000 hr. minimum). Cycling should consist of 10-hour cycles.
d. Running engagements (60 cycles minimum). Engagements to be made at 100, 500, 1000, 1500, and at each successive 500 rpm increment up to starter cut off speed.
e. Environmental tests.
f. Dielectric test.
g. Recalibration.
h. Teardown inspection.
i. Containment demonstration.

3-4.2.3 Test sample number three:

a. Calibration.

b. Endurance cycling at all operating and all inlet conditions - 250 cycles minimum.
c. Endurance overrunning test (3000 hours minimum at the mounting pad temperature specified above).
d. Endurance cycling (at all operating and all inlet conditions - 250 cycles minimum).
e. Recalibration.
f. Teardown and inspection.
g. Engaging mechanism failure test.
h. Structural load test.

3-4.2.4 Maintenance, Adjustment, or Replacement of Components or Parts. Maintenance, adjustment, or replacement of components or parts on preproduction test samples should not be permitted after initial acceptance of the starters. Disassembly of any part of the starter to any extent prior to the final teardown inspection should not be permitted.

3-4.2.5 Lubrication. Lubrication, if required, should be accomplished prior to the cycling and endurance testing. After initiation of the testing, addition of oil should not be permitted.

3-4.2.6 Design Changes During Testing. If the design is changed after start of testing, testing associated with those performance characteristics that could be affected by design changes should be re-run.

3-4.3 **Test Procedures.**

3-4.3.1 Examination of Product. Each starter should be subjected to a careful inspection to insure proper assembly, configuration, workmanship, materials, finishes, processing, weight, identification, and compliance with applicable specifications.

3-4.3.2 Initial Cycling. Each starter should be subjected to a minimum of four cycles, two of which should be at rated inlet conditions, and two at maximum inlet pressure conditions. Performance should be within the limits specified in the model specification.

3-4.3.3 Over-running Test. The starter output shaft should be driven for 10 minutes at a speed equivalent to the maximum engine speed at the starter mounting pad as specified in the detail component specification. Failure of the starter to remain disengaged should be cause for rejection. All starters submitted for inspection should be subjected to a total of 30 minutes of overrunning at the specified speed before teardown.

3-4.3.4 Teardown Inspection. After completion of the initial cycling and over-running tests, all preproduction starters and the starters tested in accordance with sampling plan A should be disassembled sufficiently to allow a detailed inspection of all working parts. The extent of the disassembly should be at the option of procuring activity or its authorized representative. If any time the disassembly disclosed internal deficiencies after the initial cycling and overrunning tests which can only be detected by disassembly, the original schedule for teardown inspection should be re-initiated. Final cycling should be accomplished and should consist of two cycles at rated inlet conditions.

3-4.3.5 Highspeed Components. Prior to conducting any starter tests, the following should be accomplished on all high speed components:

a. Subject each part to X-ray inspection. Any cracks or occlusions that will adversely affect starter performance should be cause for rejection.

b. Subject each part to fluorescent penetrant inspection. Any cracks or occlusions that will adversely affect starter performance should be cause for rejection.

c. Proof spin each turbine wheel for 1 minute at room temperature at the proof speed.

d. Upon completion of items a, b, and c above, subject each turbine wheel to X-ray inspection to determine the extent of deformity, structural damage, and growth of cracks or occlusions. Any wheel growth, deformity, structural damage, growth of cracks or occlusions, or any new cracks should be cause for rejection.

3-4.3.6 Calibration. Two cycles of operation should be completed at each of the following inlet conditions: Rated, maximum temperature, minimum pressure, and maximum pressure inlet conditions with pressure limits of ±.07 Bar (±l psia) and temperature limits of ±8.4°C (±15°F). Individual test units should be rejected if initial preproduction test unit calibrations do not meet the performance specified in the detail component specification.

3-4.3.7 Starter Cycling and Endurance Test. Including the initial calibration, the starter should be subjected to 2000 pneumatic cycles, and 3000 hours of endurance running of the starter drive in the overrunning condition as specified in the detail component specification. The order of testing and the test conditions should be as specified herein and in the detail specification. Following each group of approximately 200 test cycles, the starter output should be driven (overrunning) at a speed equal to the maximum speed of the engine accessory drive pad, ±100 rpm, for a period of time equal to one-tenth of the total endurance overrunning time. Each cycle should be conducted as follows:

a. The starter should be operated such that the load will be accelerated to the cutoff speed of the starter.

b. Upon completing the operation specified above, the output shaft of the starter should be driven in excess of 105 percent of the starter cutoff speed for a period of 1 min.

3-4.3.8 Valve Compatibility. If the intended application for the starter requires the use of a starter control valve in the installation, the starter should be subjected to a minimum of 100 test cycles with pneumatic inlet conditions to the starter equivalent to the maximum rise rate of the valve at maximum regulated pressure condition, as applicable. These cycles may be conducted in conjunction with the pneumatic endurance cycles.

3-4.3.9 Free Run. With the cutout switch made inoperative, the starter should be operated at stabilized free run speed for a period of two minutes at maximum inlet conditions. Ten cycles should be interspersed and conducted during the endurance cycles.

3-4.3.10 Vibration Test. Unless otherwise specified in the detail specification, the starter should subjected to a vibration test in accordance with MIL-STD-810, method 514, equipment category (b), mounted without vibration isolators and mounted directly on the aircraft equipment that duplicates the vibration characteristics of the aircraft. The output shaft of the starter should be accelerated from rest to cutout speed at a normal start cycle acceleration rate, a minimum of 3 times during each resonant frequency endurance test to check the speed sensing devices for sensitivity to vibration. An oscillograph trace of each cutout switch actuation should be taken for each acceleration cycle. Operation of the speed sensing devices should be normal, and chatter during actuation or actuation outside of the specified cutout speed ranges should be cause for rejection. The starter may be overrun during vibration if desired. The starter should be subjected to two cycles of operation before and after the vibration test and should meet specified performance. Data obtained should be corrected to show sea level performance on a 15°C (59°F) day.

3-4.3.11 Running Engagement Test. The starter should be subjected to five series of starts into a moving load. The engagements should be made at 500 rpm increments up to minimum starter cutout speed. Each engagement should be made by accelerating Flywheel A to a speed above that designated for the engagement. As the flywheel coasts down to the engaging speed, a normal start cycle should be made so that the starter will pick up the rotating load at the proper test speed. The starter output shaft should be marked so that any permanent twist can be noted in the test report. Failure of any part of the starter including the shear shaft should be cause for rejection.

3-4.3.12 Environmental Tests.

3-4.3.12.1 Fungus Test. If the starter contains fungus nutrients, the starter should be subjected to a Fungus Test in accordance with Method 508. Procedure 1. of MIL-STD-810. If no fungus nutrients exist in the starter, a statement to this effect should be in the test report.

3-4.3.12.2 Salt Fog Test. The starter should be subjected to a 50-hour Salt Fog Test in accordance with Method 509, Procedure I, of MIL-STD-810, while mounted in a manner similar to its intended installation. All openings normally open on the aircraft installation (exhaust, etc.) should be left open during the test; however, the salt fog should not be caused to impinge directly into the opening(s). The starter should be operated before and after the test period as specified in MIL-STD-810 and once every 5 hours during the test. Rated inlet conditions should be used for all cycles.

3-4.3.12.3 Humidity Test. With the starter mounted in a manner similar to its intended installation, the starter should be subjected to a 25-hour Humidity Test. The humidity should be maintained at 95 percent and the temperature at 51.7°±14°C (125°±25°F). All openings normally open on the aircraft installation (exhaust, etc.) should be left open during the test. The starter should be operated before and after the test as specified in MIL-STD-810 and at the end of 5, 10, 20, and 25 hours. At the end of 15 hours, the test should be interrupted and the starter should be subjected to an ambient temperature of -53.9° ±2.8°C (-65° ±5°F) for 5 hours without drying, shaking, or otherwise removing condensed water from the starter. At the end of the 5-hour cold soak period, the starter should be operated for one cycle. Rated inlet conditions should be used for all cycles. Data should be recorded for each-cycle.

3-4.3.12.4 Dust Tests.

a. Non-operating. The starter should be subjected to a dust test in accordance with MIL-STD-810, Method 510. All openings normally open on the aircraft installation (exhaust, etc.) should be left open during the test; however, the dust should not be caused to impinge directly into the opening(s). The starter should be subjected to two cycles of operation before and after the test using rated inlet conditions.

b. Operating. The starter should be subjected to four cycles of dust ingestion at ambient temperature and pressure conditions. The concentration of sand and dust should be .0529 gram per cubic meter (.0015 grams per cubic foot) of inlet air. The sand and dust containment should consist of crushed quartz with the total particle size distribution as follows:

Particle Size, Microns	Quantity Percent by Weight Finer than Size Indicated
1,000	100
900	98-99
600	93-97
400	82-86
200	46-50
125	18-22
75	3-7

3-4.3.13 Dielectric Strength Test. The electrical equipment should be subjected to a test voltage of 1,OOOV (RMS) plus twice the working voltage at commercial frequency for a period of 1 minute. The test voltage should be applied between the terminals (shorted together) and ground at sea level atmospheric pressure. If the preceding method of testing is not feasible, the dielectric tests may be conducted on the components prior to final assembly with the critical components disconnected. The test voltage for critical components should be twice the peak voltage to which they will be subjected during service or 100V, whichever is greater, for a period of 1 minute.

3-4.3.14 Containment Demonstrations. The containment demonstration required on sample number 1 should consist of an induced tri-hub-burst turbine wheel failure at or above maximum cutout speed. Demonstrations which result in fire external of the starter, external surface temperatures in excess of 371°C (700°F), or failure of the starter to contain all fragments and remain on its mounting, should be cause for rejection. Parts may fall from the starter's exhaust provided they contain no destructive energy. This should be demonstrated by placing a sheet of soft aluminum [.08cm (.032 inches) or thinner] within three feet of the starter's exhaust such that the exhaust gases will impinge on the aluminum. The aluminum sheet should be supported such that it will not have a solid backing within one inch of the under side. Any pronounced dent or puncture of the aluminum should also be cause for rejection. After the containment demonstration, the starter should be disassembled and inspected for damage resulting from the test. Photographs should be taken of the starter before disassemble showing any exterior damage and during disassembly showing all internal damage. The containment demonstration required on sample number 2 should consist of free run to failure with unregulated maximum interbleed inlet conditions to the starter.

3-4.3.15 Attitude Test. During the high temperature overrunning test, the starter should be subjected to the following attitude cycling once during each 100-hour period of overrunning:

a. Horizontal position - 15 minutes

b. 45° nose up position - 30 minutes

c. 105° nose up position - 3 minutes

d. Horizontal position - 15 minutes

e. 45° nose down position - 30 minutes

f. 90° nose down position - 3 minutes

3-4.3.16 Structural Load Test. Unless, otherwise specified, the starter should be subjected to a static load equivalent to 15 g's acting through the center of gravity in 20° increments from 90° to 270°. The top of the starter vertical center line as installed on the aircraft should be defined as 0°. A complete examination for structural failure of the unit should be made at completion of the load test. This examination should include X-ray inspection of the unit and mounting flange. Any evidence of deformity or structural damage should be cause for rejection.

3-4.3.17 Recalibration. After completion of all non-destructive testing, a calibration test should be conducted. The starter should be within initial calibration after accounting for the allowance for deterioration.

3-4.3.18 Emergency Disengagement Mechanism Test. If the starter incorporates an emergency disengagement feature, the starter should be disassembled sufficiently to render the clutch inoperative. After reassembly, the output shaft should be accelerated until the emergency disengagement feature is actuated. The starter should then be subjected to a normal acceleration cycle with the disengagement feature actuated. Damage to the starter should be cause for rejection. If a jaw type decoupler is used, the starter should be installed on a test stand with a weakened shear shaft installed. A start cycle should be run that will cause the shaft to shear and the unit to shut down in its normal manner. A second start cycle should then be performed with the shaft sheared. Any damage to the starter or the mounting pad should be cause for rejection.

3-4.3.19 Teardown and inspection. The starter should be completely disassembled for inspection. A complete examination of all parts should be made to determine wear as well as distress or failure. The procuring activity should be notified at least five working days prior to the scheduled teardown in order that the procuring activity engineering personnel may witness the inspection. Complete photo coverage should be made of starter and all parts showing general condition of the parts and enlarged views of any wear, distress, or failure of the parts. After completion of the inspection, the replaceable shear shaft should be sheared to demonstrate that the shaft will fail within the allowable limits.

3-4.3.20 Shear section. Five shear sections shall be operated for 2,000 cycles from zero to maximum torque under normal starter operating conditions without failure. The rate of loading should simulate the maximum rate of loading to which the starter is subjected during normal operation. Upon completion of the 2,000 cycles, the section should be loaded to failure. The loads causing failure should, be as specified in the detail component specification.

3-4.3.21 Other Tests. Additional component tests may be required depending upon the starter's design.

3-4.3.22 Test Completion. The starter qualification test will be considered complete when the starter and the components have been subjected to the required tests and inspections.

3-4.4 **Qualification Guidelines for the Start Control Valve.**

3-4.4.1 Examination of Product. Each valve should be subjected to a careful inspection to insure proper assembly, configuration, workmanship, materials, finishes, processing, weight, identification, and compliance with applicable specifications.

3-4.4.2 Dielectric Strength Test. The electrical equipment should be subjected to a test voltage of 1,000V (RMS) plus twice the working voltage at commercial frequency for a period of 1 minute. The test voltage should be applied between the terminals (shorted together) and ground at sea-level atmospheric pressure. If the preceding method of testing is not feasible, the dielectric tests may be conducted on the components prior to final assembly or with the critical components disconnected. The test voltage for critical components should be as specified in 3-3.4.20.8.2.

3-4.4.3 Proof Pressure Test. With the valve in the closed position and the outlet port open to atmospheric pressure, apply the proof air pressure specified in 3-3.4.20.11 to the inlet port of the valve for a period of one (1) minute. Repeat this test, except the valve should be in the open position and the outlet port should be capped.

3-4.4.4 Cycling Test. The valve should be operated for 25 cycles using air at rated inlet conditions at room temperature and 21 volts dc electrical power.

3-4.4.5 Actuation Test. The valve should be connected to a 28 volt dc electrical power source and actuated twice at each of the inlet conditions specified in the detail specification. Flow rate may be reduced to a valve where air temperature can be maintained, provided the opening rate is not affected. Valve operation should be smooth and the opening and closing rates should be within the limits specified herein and in the detail specification. Regulated pressure, if applicable, should be within the limits specified in the detail specification. Current draw should not exceed that specified in the component detail specification.

3-4.4.6 Leakage Test. Internal and total leakage should be measured with the valve in the closed position and upstream pressure at both the maximum and minimum inlet pressures and temperatures specified in the detail specification.

3-4.4.7 Calibration. The valve should be subjected to four cycles at the following inlet conditions:

a. Rated
b. Maximum temperature
c. Minimum pressure
d. Maximum pressure
e. Maximum engine interbleed

Two cycles at each condition should be conducted using 21 volts dc and two cycles using 28 volts dc. Inlet pressure and temperature limits should be ±.07 Bar (±l psia) and ±8.3°C (±15°F). Airflow through the valve should be controlled by an orifice or nozzle downstream as defined in the detail specification. Individual test valves should be rejected if performance is not within the limits defined in the detail specification. Data should be recorded for each calibration cycle. In addition, it should be demonstrated that the valve will open (timing requirements need not be met) with rated inlet conditions and 17 volt dc power. The initial calibration should be conducted after the break-in run and before initiation of the schedule preproduction test program. The starter should not be operated between completion of the initial calibration runs and start of the scheduled test program.

3-4.4.8 Cycling and Endurance Test. Including the calibration cycles, the valve should be subjected to a total of 10,000 endurance cycles at the conditions specified in table 1. A cycle in this case is defined as actuation from close to open to close while the unit is exposed to the operational environmental conditions defined by this specification. Airflow through the valve should be controlled as specified in 3-4.6.7 and electrical power should be 28 volts dc. The valve should remain in the open and closed positions for a minimum number of seconds, as defined in the detail specification, during each cycle throughout this test, except that during period II the cycling rate should be __ seconds open, followed by __ seconds closed continuously, and during period III the cycling rate should be __ minutes open followed by __ minutes closed continuously, as defined by the detail specification. Data should be recorded for the first, last, and every 500th cycle during each period of testing to show compliance with the opening and closing time guidelines of this specification and internal and total leakage should be measured after every period. Failure of the valve to meet the operation and leakage requirements throughout the test should be cause for rejection.

TABLE I

PERIOD	NO. OF CYCLES	INLET CONDITION(I)	AMBIENT TEMPERATURE
I	2480	Maximum Pressure	Ambient
II	900	Maximum Temperature	High Operating
III	100	Maximum Temperature	High Operating
IV	5000	Rated	Ambient
V	1500	Maximum Temperature	High Operating

(1) The related inlet pressure or temperature conditions to those listed should be specified in the detail specification.

3-4.4.9 Proof Pressure Test. With the valve in the closed position and the outlet port open to atmospheric pressure, apply the proof air pressure specified in 3-3.4.20.11, to the inlet port of the valve for a period of 10 minutes. Repeat this test, except the valve should be in the open position and the outlet port should be capped. Following these tests, the valve should function properly, and leakage should not exceed the values specified in the detailed specification.

3-4.4.10 Vibration. Vibration tests should be conducted in accordance with the following method and procedure:

3-4.4.10.1 Method. The valve used in the Vibration Test should be the Endurance Test valve and should be tested as a complete assembly. Before and after this test, a calibration should indicate that no component subject to calibration has changed its calibration beyond the design tolerance range. All components not subject to calibration should be operated and should not have had any part of their function impaired by the test.

3-4.4.10.2 Procedure. The following procedure should apply.

3-4.4.10.2.1 Test Installation. The valve should be mounted on a test fixture to simulate airplane installed position. If the aircraft installation requires that the starter control valve be mounted directly on the starter pneumatic inlet flange, the vibration should be conducted with the applicable control valve mounted on a test fixture simulating the starter as in the airplane installation. To more closely simulate the airplane installation, the airplane pneumatic duct may be attached to the valve and supported as in the airplane. If the starter control valve is mounted remotely in the airplane installation, the valve should be vibrated as a separate unit. The test fixture should be rigid as possible and designed to eliminate or minimize fixture resonance's in the test frequency range. The test configuration should be such that rotational motion of the fixture is not induced due to any unsymmetrical weight or stiffness distribution of the equipment. Measurements of vibratory accelerations or displacement should be made at the mounting base of the valve. All frequency measurements should be accurate within ±5 percent. All displacement or acceleration measurements should be accurate within ±10 percent. The motion of the vibration table should be simple harmonic motion with not more than 20 percent distortion.

3-4.4.10.2.2 Frequency Scan. The equipment should be vibrated with essentially simple harmonic motion in each of three mutually perpendicular directions to explore for resonant frequencies of component parts over the ranges of frequencies required in the detail specification. Frequency scans should be conducted very slowly and carefully. The frequency range should be broken into small convenient intervals and each interval scanned at a constant applied force amplitude which produces approximately the fixture displacement or acceleration amplitude defined in the detail specification. The fixture motion should be observed closely during the frequency scans to detect the frequencies of minimum fixture motion which defines the frequencies at which major components of the unit are in resonance. The frequencies are quite sharply tuned and do not necessarily coincide with the frequencies at which maximum component amplitudes or noises occur when scanning at constant applied force amplitude. When the resonant components are small, the reduction of fixture motion at resonance may not be discernible, in which case the resonant frequencies may be determined from observations of maximum component amplitudes or noises, phase shift between fixture motion, and motion of a resonant part. In all cases, the resonant frequencies should be verified, if possible, by checking for minimum fixture motion. The test report must contain complete descriptions of all resonance's, including the resonant frequencies, parts in resonance and modes of vibration.

3-4.4.10.2.3 Resonance Endurance Tests. After completion of the frequency scan, resonance endurance tests should be conducted at all resonant frequencies of the components, with the applied force adjusted as necessary to obtain the applicable fixture displacements or accelerations specified in the detail specification. The valve should be continuously pressurized using air at a pressure equal to maximum engine interbleed pressure and should be cycles five times during vibration at each resonant frequency. (Flow rate optional.) Operation of the valve should be normal and any tendency for the valve to open without being actuated should be cause for rejection. The vibrator and power supply must produce the prescribed fixture motions at all frequencies. Endurance tests should not be conducted at any frequency at which the fixture amplitude abruptly increases when scanning at a constant applied force amplitude. Endurance tests should be interrupted periodically, to rescan for resonance. If a change in resonant frequency occurs during endurance testing at resonance, the frequency of vibration should be adjusted to follow the resonance. However, if large or abrupt resonant frequency shifts occur, the item should be examined for component failures. Resonance endurance tests should be conducted for one million cycles or eight hours, whichever occurs first, at each resonance. (See note I.) If the same resonance is excited by vibration applied in more than one direction, the endurance test for that resonance should be accomplished with vibration applied in the direction which produces the greatest response. If no resonance's are found, the unit should be vibrated at 100 cps and the corresponding amplitude as specified in the detail specification for one million cycles in each of the three mutually perpendicular directions. Note I: Where several resonance's are encountered, the procuring activity should be notified of the parts in resonance and, at the option of the procuring activity, the endurance tests may be limited to those resonance's considered most likely to produce failure. A detailed description of each resonance is required.

3-4.4.10.2.4 Cycling Endurance Test. The unit should be subjected to the following Cycling tests at room temperatures in addition to the frequency scan and resonance endurance tests. The unit should be vibrated at the displacements and accelerations defined in the detail specification with the frequency continuously varied between the specified limits in a cycle of approximately 15 minutes. The rate of change of frequency should be proportional to the frequency. (The rate of change of frequency may be a linear function with time). The total test time should be as specified in MIL-STD-810, method 514, procedure I. This test should be repeated with vibration along each of three mutually perpendicular axes. The cycling test may be broken into convenient frequency ranges, provided the cycling rates and test times for each range are not changed.

3-4.4.11 Recalibration. The valve should be subjected to a calibration. Failure of the unit to meet the performance requirements of the detail specification should be cause for rejection.

3-4.4.12 Teardown and Inspection. The valve should be completely disassembled for examination of all parts and for measurements as necessary to disclose excessively worn, distorted, or damaged parts. These measurements should be compared with those taken prior to the start of the Preproduction Test. The Preproduction Test should be considered complete when every component of the valve has been subject to, and has completed, the specified amount of preproduction testing without damage to an extent which would prohibit further operation of the unit. Photographs of the parts as disassembly should be made and furnished as a part of the Preproduction Test Report. Photographs of small parts and any wear points or failures should be enlarged to show detail.

3-4.4.13 Dust Test. The valve should be subjected to Dust Tests in both operating and non-operating conditions. The valve should be mounted in a manner similar to its intended installation. Data should be recorded both before and after each part of the test. Leakage should be measured both before and after the test. All cycles should be conducted using rated inlet conditions for non-regulating valves and maximum engine interbreed conditions for regulating valves. Failure of the valve to meet the performance requirements of the detail specification should be cause for rejection.

3-4.4.13.1 Part I, Non-operating. The valve should be subjected to a Dust Test in accordance with Method 510, Procedure 1, of MIL-STD-810 while the valve is inoperative.

3-4.4.13.2 Part II, Operating. The valve should be cycled 50 times while being supplied with contaminated air to the valve in accordance with Method 510, Procedure I, of MIL-STD-810, except the air should be supplied at either rated or maximum engine interbleed conditions as specified above, for the applicable type of valve.

3-4.4.14 Rain Test. The valve should be subjected to a Rain Test in accordance with Method 506, Procedure I, of MIL-STD-810. Data should be recorded for the test cycles both before and after the test. Rated inlet conditions should be used for the test cycles.

3-4.4.15 Low Temperature Test. The valve should be subjected to a Low Temperature Test in accordance with the following procedure:

a. Soak the valve at -53.9° ±2.8°C (-65° ±5°F) until stabilized at -53.9° ±2.8°C (-65° ±5°F).

b. While being maintained at the -53.9°C (-65°F) ambient, operate the valve for one cycle using rated inlet conditions.

c. Soak the valve at -53.9° ±2.8°C (-65° ±5°F) for 30 minutes and repeat the procedure in b above.

d. Repeat the procedure in c above.

e. The complete procedure above, with the exception that the initial soak may be reduced to 5 hours, should be repeated 2 times making a total of 6 cycles. Data should be recorded for each cycle.

3-4.4.16 Fungus Test. If the valve contains fungus nutrients, the valve should be subjected to a Fungus Test in accordance with Method 508, Procedure I, of MIL-STD-810. If no fungus nutrients exist in a valve, a statement to this effect should be in the test report.

3-4.4.17 Salt Fog Test. The valve should be subjected to a 50-hour Salt Fog Test in accordance with Method 509, Procedure 1, of MIL-STD-810, while mounted in a manner similar to its intended installation. The valve should be operated before and after the test period as specified in MIL-STD-810 and once every 5 hours during the test. Rated inlet conditions should be used for all cycles. Data should be recorded for each cycle.

3-4.4.18 Humidity Test. The valve, when mounted in a manner similar to its intended installation, should be subjected to a 25 hour Humidity Test. The humidity should be maintained at 95 percent and the temperature at 51.7° ±14°C (125° ±25°F). The valve should be operated before and after the test as specified in MIL-STD-810 and at the end of 5, 10, 20, and 25 hours. At the end of 15 hours, the test should be interrupted and the valve should be subjected to an ambient temperature of -53.9° ±2.8°C (-65° ±5°F) for 5 hours without drying, shaking, or otherwise removing condensed water from the valve. At the end of the 5-hour cold soak period, the valve should be operated for one cycle. Rated inlet conditions should be used for all cycles. Data should be recorded for each cycle.

3-4.4.19 High Temperature Test. The valve should be soaked at the high non-operating (exposure) temperature for 48 hours. The temperature should then be reduced to high operating temperature and, while being maintained at this temperature, the valve should be subjected to 10 continuous cycles of 60 seconds open followed by 60 seconds closed. The valve should then be subjected to 10 continuous cycles of 5 minutes open followed by 5 minutes closed. All cycles should be conducted using the applicable maximum temperature inlet conditions of table I.

3-4.4.20 Pressure Drop Test. Loss in static pressure from inlet to outlet port of the valve should be determined for a range of conditions so that a plot of

$$\{W * (T)**^{1/2}\}/ P \quad \text{versus} \quad \Delta P/P$$

on logarithmic (log-log) graph paper can be made where:
W = Actual weight flow
T = Temperature
P =Upstream pressure
ΔP = Pressure drop,

Pressure drop (ΔP) in excess of that specified in the detail specification should be cause for rejection.

3-4.4.21 External Load Test. The valve should be mounted by one of its connecting flanges and the external loads specified in the detail specification applied at the opposite flange. Bending moment loads should be applied in both directions in each of two mutually perpendicular planes whose intersection falls on the centerline of the connecting flanges. The test should be repeated with the valve mounted on the opposite flange. There should be no evidence of damage or permanent deformation to the valve as a result of the applied loads.

3-4.4.22 Burst Pressure Test. The valve should be subjected to a Burst Pressure Test as follows: With the valve in the closed position and the outlet port open to atmospheric pressure, apply an air pressure equal to the minimum burst pressure specified in the detail specification to the inlet port of the valve for a period of 10 minutes. Repeat the test, except the valve should be in the open position and the outlet port should be capped. The unit should show no evidence of structural failure upon completion of this test.

3-4.4.23 Icing Test. The valve flow controlling mechanism icing test should be specified in the detail specification. In addition, the following water contamination test should be conducted. The valve should be cycled five times at room temperature with saturated air. The temperature should then be reduced to -32°C (-25°F) and the valve cycled 50 times. The valve should operate within the limits specified in the detail specification.

3-4.4.24 Explosion Proof. The valve should be subjected to an explosion atmosphere test in accordance with MIL-STD-810, Method 511, procedure I with the exception that pressure or flow through the valve is not required. The unit should not create an explosion.

3-4.5 **Qualification Guidelines for Hydraulic Starters**.

3-4.5.1 Test Conditions. Unless otherwise specified in this document or the detailed component specification, all tests should be conducted with the starter at an ambient temperature range of 50° to 120°F, with rated conditions as specified on the detailed component specification. Tests may be conducted in the order deemed most desirable by the qualifying activity.

3-4.5.1.1 Pressure Rate Increase. Unless otherwise specified herein, the pressure rate increase during all tests should be within the limits of Figure I herein.

3-4.5.1.2 Filtration. For all tests except the low temperature test, an absolute 15-micron filter should be installed in the starter inlet and discharge hydraulic fluid lines. The filter should be inspected after every 100 starts and after every 100 hours of overrunning. Excessive accumulation of chips or other foreign material originating from the starter should be cause for rejection.

3-4.5.1.3 Hydraulic Fluid. Fluid conforming to MIL-H-6083 may be used during acceptance test at the discretion of the contractor.

3-4.5.2 Instrumentation. Instrumentation should be provided during qualification and inspection tests to observe or determine the following:

3-4.5.2.1 Outlet Pressure. The outlet pressure should be measured with a calibrated pressure gage located within two inches of the outlet port. Gage accuracy should be within ±1 percent psig.

3-4.5.2.2 Inlet Pressure. The inlet pressure should be measured with a calibrated pressure gage located within two inches of the inlet port. Gage accuracy should be within ±1 percent psig.

3-4.5.2.3 Hydraulic Flow. The flow from the outlet should be measured by any method which will give results within an accuracy of ±2 percent.

3-4.5.2.4 Speed. The speed should be measured within an accuracy of ±1 percent of output shaft speed.

3-4.5.2.5 Torque. The torque developed by the starter should be measured by a suitable method within an accuracy of $\underline{+}1$ percent.

3-4.5.2.6 Fluid Temperature. Hydraulic fluid temperature should be measured within an accuracy of ±5°F.

3-4.5.2.7 Ambient Temperature. Ambient temperature should be measured in the vicinity of the starter with an accuracy of 1 percent of maximum reading.

3-4.5.3 Initial Calibration. Initial calibration should consist of starter operation at starter drive speeds ranging from zero to free-running speed and observing the parameters listed in 3-4.2.2. A minimum of six stabilized speeds should be obtained. The starter oil flow versus speed curve should be within the limits specified on the detailed component specification. The starter performance should be within the limits specified on the detailed component specification. External cooling may be used during this calibration if excessive oil temperatures will result from prolonged operation. For Type I starters, this test should be conducted at the two pressure levels noted on the detailed component specification detailed component specification.

3-4.5.4 No-load Operation. The starter should be operated at no-load, with the cutout switch inoperative, without damage or failure for the time specified on the detailed component specification. The starter free-running speed obtained during this test, or the maximum speed attained before automatic cutback, should not be less than that specified on the detailed component specification.

3-4.5.5 Proof Pressure Test. With the loading device adjusted to limit starter output drive speed to rated speed rpm and at rated inlet temperature, the starter should be operated at the conditions noted on the detailed component specification.

Shaft seal leakage should not exceed the rate specified on the detailed component specification.

3-4.5.6 Endurance Test. The starter should be subjected to an endurance test as specified on the detailed component specification without damage to adjustment of, or replacement of any of the component parts. Each cycle should consist of the phases specified on the detailed component specification. The cutout switch should be electrically loaded to the value noted on the detailed component specification for each cycle and should function within the limits specified thereon.

3-4.5.7 Post-Endurance Calibration. After completion of the endurance test specified in herein, the rated torque speed test should be repeated. The starter overall efficiency should not have decreased more than 5 percent from the original values at rated load.

3-4.5.8 Leakage (operational). The shaft seal leakage should be determined during the endurance test herein and should not exceed a rate of 5 cc per 120 endurance cycles. Other external leakage should be of insufficient magnitude to form a drop.

3-4.5.9 Leakage (static). With hydraulic fluid at rated temperature and at rated outlet pressure and while the inlet port is blocked, external leakage and shaft seal leakage should be of insufficient magnitude to form a drop when measured over a 15-minute period.

3-4.5.10 Sustained Motoring. While coupled to a loading device, the starter should be operated at rated inlet conditions, rated speed, and rated torque load for the time period specified on the detailed component specification. No damage should result from this operation.

3-4.5.11 Stall Torque. The stall torque of the starter should be measured at rated inlet and outlet conditions at four equally-spaced positions within one revolution. The minimum torque measured should not be less than the torque specified for zero speed.

3-4.5.12 Overrunning. The starter output shaft should be driven for the time period specified and in accordance with the conditions noted on the detailed component specification. The starter motor case is to be completely filled with MIL-H-5606 hydraulic fluid and the test system containing at least two gallons of fluid with the case pressurized to rated outlet pressure. One pint of fluid may be drained from the starter after each 24 hours of overrunning; one pint of fluid may then be returned to the system. No damage to the starter should result from this test.

3-4.5.13 Extreme Temperature Operation.

3-4.5.13.1 Low Temperature Operation. The starter should be stabilized at -65°F. After fluid and starter have reached stabilized temperatures, the starter should be operated at rated inlet and outlet pressures and the fluid at rated inlet and outlet pressures. It should then be operated for three consecutive cycles without damage. Each cycle should be equivalent to one cycle as described for the endurance test herein except that the starter output should meet the requirements specified on the detailed component specification. Fluid temperature rise should be recorded during the three consecutive cycles.

3-4.5.13.2 High Temperature Test. The starter should be stabilized at an ambient temperature of 275 ±10°F. The starter should be operated for three consecutive cycles at rated pressures and rated fluid temperature without damage. Each cycle should be equivalent to one cycle as described for the endurance test herein.

3-4.5.14 Consecutive Operation. The starter should be operated for 10 consecutive endurance cycles without damage. Each cycle should be equivalent to one cycle as described for the endurance test herein.

3-4.5.15 Dynamic Torque Check. Cycling should be run, but using continuous torque recording instrumentation. Response rate of the instrumentation used should have a flat sinusoidal response to 1000 cps. The peak transient torque should not exceed the value listed on the detailed component specification.

3-4.5.16 Disengagement and Re-engagement Test. The starter should be disengaged and re-engaged at speeds in increments of 1000 rpm from zero rpm to cutout speed. No damage to the starter should result from this test. Recording instrumentation should be used to measure impact torque at re-engagement.

3-4.5.17 Shock Test. The starter should be subjected to the shock test in accordance with Method 516.2, Procedure I of MIL-STD-810 using Figure 516.2-1 at amplitude "a". As a result of the shock test the overall efficiency of the starter should not have deteriorated more than one (1) percent.

3-4.5.18 Vibration. A vibration test should be conducted in accordance MIL-STD-810, Method 514.2 and as follows:

a. Equipment category - b.1
b. Table - 514.2 - II
c. Procedure - 1
d. Figure - 514.2-2
e. Test curve - F

No damage to the starter should result from the test.

3-4.5.19 Shear Section Strength. After completion of all tests, the output shaft couplings of the qualification starters submitted should be sheared and the shearing section torque recorded. The shear value should not exceed the values specified in the detailed component specification.

3-4.5.20 Engine Cranking Tests. At the option of the qualifying activity, the starter must complete 75 actual starts of the engine for which it was designed. The starter may be one submitted for qualification or it may be an additional model furnished by the starter manufacturer for such tests. The tests may be performed in an engine test cell, at the engine manufacturer's plant, in a prototype aircraft, or in an aircraft specifically designed for such tests. Test conditions should be at the prevailing ambient conditions where the tests are performed and the inlet and outlet pressure should be as specified in the detail specification.

3-4.5.21 Teardown Inspection. A teardown inspection consisting of complete disassembly of the starter should be made at the completion of the qualification test. Dimensional inspection should be made of all wearing parts. Rotating parts should be subjected to Magnaflux or Zyglo inspection, except that assembled anti-friction bearings should not be subjected to Magnaflux inspection. Rotating parts subjected to centrifugal stresses should be measured on dimensions likely to show growth. Where practicable, seal clearances should be determined. Failure of or any abnormal wear or damage to any part, sufficient to impair continued use of the starter, should be sufficient cause for rejection of the starter.

3-4.5.22 Torque Calibration and Flow (quality conformance only). A calibration of torque versus speed and fluid flow should be conducted to determine that the starter meets the minimum starter drive torque and flow requirements specified on the detailed component specification.

3-4.5.23 Shaft Seal Leakage (quality conformance only). With the inlet port blocked off and the outlet port pressurized at rated outlet pressure, the starter should be overrun for a period of 30 minutes at the drive speed specified in the detailed component specification. This leakage check may be conducted during the overrunning test.

3-4.5.24 Over-pressurization. The starter should be operated at rated conditions except for the inlet pressure which should be as specified in the detailed component specification. The starter should complete 10 cycles in accordance with the endurance test specified herein without damage to or replacement of any of the component parts.

SECTION 4: FIRE DECTECTION/EXTINGUISHING

4-1.0 REQUIREMENTS.

4-1.1 PERFORMANCE. The Fire Detection/Extinguishing system shall meet its allocated performance and function in such a manner that the aircraft shall be able to be operated safely throughout the operating envelope and meet the performance requirements as defined in the applicable weapon system specification.

4-1.2 QUALIFICATION. The following qualification requirements for the Fire Detection/Extinguishing system are required to verify compliance with the performance requirements above as applicable to the aircraft design configuration.

4-1.2.1 Analysis. Design and performance analysis shall be documented, using ADS-9C as a guide. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

4-1.2.2 Component Tests. The following tests shall be conducted and a subsequent teardown inspection, to determine the post test condition, shall be performed. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

a. Fire containment
b. Detection alarm response
   1. Sunlight rejection
   2. Partial extinguishment
   3. Saturation
   4. Fuel/oil immersion
   5. Power variation
   6. Fault in conductor
   7. Field of view
c. Extinguishing
   1. Leakage
   2. Mount static load
   3. Discharge
   4. Proof pressure
   5. Overheat safety
   6. Hydrostatic bursting
   7. Environmental (temperature, humidity, salt spray/corrosion)
   8. Vibration

4-1.2.3 Assembly/System Level Tests. Aircraft system level tests shall be conducted in accordance with ADS-1B-PRF.

4-2.0 REFERENCES/DEFINITIONS.

4-2.1 References.

ADS-1B-PRF	Rotorcraft Propulsion System Airworthiness Qualification Requirements
ADS-9C	Propulsion System Technical Data
AIR 1262	Aircraft Fire Protection For Helicopter Gas Turbine Powerplant And Related Systems Installations

4-2.2 Definitions.

4-2.2.1 Fireproof: A material which will withstand heat as well as or better than steel which will withstand 2000°F (1093°C) for 15 minutes and still fulfill its design purpose. When applied to materials and parts used to confine fires within designated fire zones, "fireproof" means that the material or part will perform this function under the most severe conditions and duration of fire likely to occur in such zones.

4-2.2.2 Fire Resistant: When applied to sheet or structural members, "fire resistant" describes a material which will withstand heat as well as or better than aluminum alloy in dimensions appropriate for the purpose for which it is used. When applied to fluid-carrying lines, other flammable fluid system components, wiring, air ducts, fittings and power plant controls, this term refers to a line and fitting assembly, component, wiring, duct or controls which will perform the intended functions under the heat and other conditions likely to occur at the particular location. An example: Fire resistant hose which will resist 2000°F (1093°C) flame for 15 minutes.

4-2.2.3 Fire Zone: Any area within the aircraft that contains the ingredients for a fire, i.e. an ignition source (e.g. an electrical spark, a hot metal, or the potential for a flame (i.e engine combustion)), a combustible product or ingredient, and oxygen should be considered a fire zone.

4-2.2.4 Flame Resistant: Materials which will not support combustion to the point of propagating a flame beyond safe limits after removal of the ignition source.

4-2.2.5 Standard Test Flame: A standard test flame is produced by burning JP-4 fuel conforming to Specification MIL-J-5624, or 100 octane gasoline conforming to Specification MIL-G-5572, in a flat pan having an inside diameter of 5 inches and a depth of at least 3 inches. Unless otherwise specified, the air flow over the pan should not exceed 10 feet per second.

4-3.0 **PERFORMAANCE GUIDELINES**.

4-3.1 **FIRE CONTAINMENT**:

4-3.1.1 Fire Walls. All fire zones should be completely separated, one from the other, and from other parts of the structure, by firewalls or by tight fitting ventilation stops if the hazard is considered minor. If the ventilation stops or fireproof barriers are not airtight, the opening should be rigidly fixed to assure that the heat or over-pressures produced by the ignition do not increase the size of the openings and allow flame propagation into another compartment. Consideration should be given to the fact that the firewall may buckle severely due to heat; therefore any doors or joints should be closed by closely spaced fasteners of such a type that hazardous gaps will not open up during the fire. For transverse firewalls in turbo-jet installations, a failure in the tailpipe of the engine can release high temperature exhaust gas under high pressure into the rear engine compartments. The high pressure differential, for a sufficient time, should allow fire burn-out through some provided path (to relieve the pressure) or should allow use of normal fire control procedures. Such burn-out paths may require soft spots in cowling to provide fire egress in the most desirable spot. When seams or openings are located forward of the firewall, design provisions should prevent the propagation fire in the external airstream around the firewall and into another compartment. Each engine should be isolated by a firewall, shrouds, or equivalent means, from personnel compartments, structures, controls, rotor mechanisms, and other parts that are essential to controlled flight and landing. Each auxiliary power unit, combustion heater, and other combustion equipment to be used in flight, should be isolated from the rest of the rotorcraft by firewalls, shrouds, or equivalent means. Each firewall or shroud should be constructed so that no hazardous quantity of air, fluid, or flame can pass from any engine compartment to other parts of the rotorcraft. Each firewall and shroud should be fireproof, capable of withstanding impingement of 2000°F (1094°C) flame for 15 minutes without penetration and should also be protected against corrosion.

4-3.1.2 Material Compatibility/Flammability. None of the materials used in the engine and APU compartments should be adversely affected by fuel, oils, lubricants or cleaning materials. Flammable materials should not be used inside the compartments. Lines or components within a fire zone that carry or contain flammable fluids should withstand the same flame guidelines as for firewalls above. All grommets and fillers used at points where items pass through fire seals or walls should be made of material possessing the same fireproof characteristics as the fire seal or firewall material.

4-3.1.3 Compartmentation. It should be assumed that such items as fuel lines, booster pumps, etc., may fail at some time, and possibly start a fire. Separate compartments within the pylon, wing or fuselage will help to isolate such fires sufficiently to allow time for all possible countermeasures to be tried.

4-3.1.4 Air Scoops. Air scoops should be located and so constructed that fire from adjacent areas or other portions of the helicopter cannot enter them. Air supply scoops for inhabited compartments should be located so that combustible fluid or other vapor, fire or smoke cannot enter under any reasonable flight attitude.

4-3.1.5 Air Ducts. Air ducts should be fireproof where they pass through fire zones. When they discharge in fire zones, they should be provided with shutoff valves or sufficient extinguishing agent should be discharged so that the mixture will not support combustion. When ducts originate in fire zones they should be fireproof for a sufficient distance beyond the fire barrier to assure that any fire can be contained within the duct. A valve should be provided in the duct when it leads to another compartment in the helicopter.

4-3.1.6 Shutoff Valves. Combustible fluid supply lines which pass through the powerplant installation firewall should incorporate valves, except if the total amount of combustibles dumped into a fire is too small to be considered a hazard. Shutoff valves and their controls should be fireproof when located in the fire zones. The closing of any fuel shutoff valve for any engine should not make fuel unavailable to the remaining engines. The operation of any shutoff should not interfere with the later emergency operation of any other equipment, such as the means for declutching the engine from the rotor drive. There should be means to prevent inadvertent operation of each shutoff and to make it possible for the crew to reopen it in flight after it has been closed.

4-3.1.7 Lines and Fittings. Lines and fittings in fire zones which carry combustible fluids or vapors should be fireproof if rigid and at least fire resistant in the case of hose materials. This applies also to vent and drain lines, unless a failure of such lines and fittings will not result in or add to a fire hazard. Hose for emergency equipment in fire zones should be as fireproof as possible.

4-3.1.8 Cowling. Cowling enclosing fire zones should contain a fire to such a degree that no additional hazard will be caused by burning into other compartments, heating up surfaces of flammable tanks or walls of compartments with potential leakage of flammable fluids, or by heating up basic structure of basic control equipment beyond safe limits. Components or hinges, fasteners, and attachment supports for cowling and access panels which are located in fire zones and may be exposed to fire exiting around the joint should be made of fireproof materials.

4-3.1.9 Material Adjacent to Fire Walls. Materials used on the protected side of the firewall, and in other similar locations, should be of a type which will not burst into flames as a result of heat transfer or radiation from a fire in the fire zone. Structure and equipment on the protected side should not be affected in such a way that additional hazards evolve.

4-3.1.10 Combustible Tanks. Oil, hydraulic, and water-alcohol tanks and reservoirs located in fire zones should be constructed of fire resistance material. Their supports should be made of fireproof material, to preclude the dropping of the tank into a fire.

4-3.1.11 Cooling Air Outlets. Critical areas downstream and adjacent to cooling air outlets in fire zones should be protected against fire burning at and outside of the outlet.

4-3.2 **FIRE DECTECTION**.

4-3.2.1 Function. Whenever a fire zone is indicated by design analysis, a means of detection of a fire should be employed. The detector system should be designed for highest reliability to detect a fire and not to give a false alarm. It is desirable that it only responds to a fire and misinterpretation with a lesser hazard is impossible. Engine over-temperature, harmless exhaust leakage, and bleed air leakage should not be indicated by a fire detector system, unless indication of such conditions is deemed desirable. A fire detection system should be reserved for a condition requiring immediate measures such as engine shutdown, fire extinguishing, or bailout.

4-3.2.2 Flame Detection Characteristics. The system should indicate fire within 5 seconds after exposure to the standard test flame as defined. No alarm should occur as a result of exposure to steady or intermittent sources of extraneous radiation such as steady or chopped sunlight, hot engine parts or artificial lighting, or any other ambient light that may exist within an engine space.

4-3.2.3 Detection Capability. The system should signal an alarm when operating with the maximum design number of sensors, and only one sensor is subjected to the flame. The detection capability should not be adversely affected by accumulation of contaminants that may be encountered in normal aircraft operation.

4-3.2.4 Automatic Repeatability. Within 5 seconds after signal clearance, the system should be capable of signaling a re-ignition of the flame, without requiring manual resetting.

4-3.2.5 Abnormal Flight Conditions. The system should be capable of continuing operation during abnormal flight attitudes, rapidly changing altitudes from sea level to 20,000 feet, 95% relative humidity, temperatures varying between -65°F and 180°F, contaminating atmospheres, 15g accelerations and other conditions which may be encountered during take-off, flight, landing or servicing the aircraft.

4-3.2.6 Prevention of False Warnings. The design and installation of the system should be such as to prevent the occurrence of false fire warnings resulting from flight operations, environmental conditions, damage to components of the system, or loose connections (conductor of the system accidentally becoming broken, grounded or disconnected, or contacting another conductor of the system), or from transient voltage conditions, intermittent voltage between 0 and 124 volts for AC powered systems, or between 0 and 29 volts for DC powered systems.

4-3.2.7 Fire Warning Signal. The system, when actuating, should send a signal to the cockpit. Where more than one fire warning signal is provided, each signal, should indicate the specific engine or compartment in which the fire occurs.

4-3.2.8 Test Provisions. A test switch should be provided in the immediate vicinity of each fire warning signal to permit in flight and ground check of the continuity of the electrical circuits. Operational readiness should be shown by actuation of the warning signal when the test switch is operated.

4-3.2.9 Supply Power. The system should comply with the available power supplied by the aircraft.

4-3.2.10 Abnormal Voltage Protection. The system should be capable of accepting input voltages below 102 volts AC or 17 volts DC, as applicable, without being damaged, and should automatically resume operation when the input voltage returns within operating limits. If transistors are used in the system, protection should be incorporated to prevent damage by 200 volts rms. voltage transients of 50 milliseconds duration or by 56 volts DC voltage transients of 50 milliseconds duration, as applicable.

4-3.2.11 Electromagnetic Interference. The system should comply with the electromagnetic interference requirements of Weapon System Specification.

4-3.2.12 Explosion Proof. The system should be designed to operate in an explosive atmosphere without creating an ignition hazard.

4-3.2.13 Sensor Viewing Field. The viewing field of each sensor should include a conical volume whose envelope extends at least 45° from the center line of the cone axis.

4-3.2.14 Sensor Temperature. The sensor should function satisfactorily when subjected to temperatures between -65°F and 300°F over extended periods.

4-3.2.15 Design Operating Life and Reliability. The system should have an operating life of at least 1000 hours without requiring removal for bench servicing. As a reliability objective, less than two in flight failures per 100 service installations should occur per year.

4-3.2.16 Total Operating Life. The system should be designed for a minimum total operating life of 10,000 hours with reasonable servicing and replacement of parts.

4-3.2.17 Installation Guidelines.

4-3.2.17.1 Monitored Zones. The following potential fire zones, and such other ones that may be determined by the aircraft contractor, should be monitored by the sensors:

a. Power sections and accessory sections of reciprocating engine compartments.

b. Compressor, burner, tailpipe (if necessary) and afterburner compartments of the turbine engine installations.

c. Accessory sections of turbine engines, if flammable fluid system components and sources of ignition are both present.

d. Engine compartments of rocket engine installations.

e. Auxiliary power plant compartments.

f. Compartments containing electrical or electronic equipment in the vicinity of combustibles.

4-3.2.17.2 Location of Sensors. Sensors should be located and aimed to provide complete optical coverage of sources of combustibles such as fuel and hydraulic fluid components, and sources of ignition such as electrical equipment, where the proximity of these and other sources of combustion and ignition may be potential sources of fire. The selected locations should also comply with the following guidelines:

a. Sensors should be located to view the paths of most probable flame travel, including air exits from potential fire zones.

b. Sensors should be located out of the paths of normal exhaust gases and should not be located in positions where ambient temperatures may exceed the allowable operating temperature of the sensors.

c. Sensors should not be mounted on hot engine parts nor in any manner that will interfere with the ready repair or replacement of the engine, and should be positioned to avoid facing into direct sunlight.

d. Sensors should be located to minimize the probability of damage during engine removal and other aircraft maintenance.

4-3.2.18 Fire-resistance. Connectors, clamps and wiring located with potential fire zones should be fire resistant.

4-3.3 **FIRE EXTINGUISHING SYSTEM**.

4-3.3.1 Compartment Extinguishing Systems. A fixed fire extinguishing system should dilute all the atmosphere within and entering a compartment with sufficient inert agent that it will not support combustion, and continue the process for a duration sufficient that existing flame is extinguished and either the vapors are dissipated or ignition sources eliminated. The fire extinguisher system should also be designed to allow for any additional air leakage that could occur as a result of maintenance and service or as a result of a fire. HRD (High Rate of Discharge) systems utilize open end tubes to deliver a given quantity of agent within 1.35 seconds for CO2 and 1 second for all other agents. The HRD systems are recommended for use in compartments having high airflow where the required discharge rates can be more effectively provided by an HRD rather than a perforated tubing system. FAA tests indicate that unrestricted release through such an open end tube distribution system can be relied on for adequate distribution, provided the outlets are located properly. Although the discharge times given above are considered satisfactory, it is believed that any reduction in discharge time below that specified would improve system effectiveness. However, consideration should be given to the time requirements for draining accumulated combustibles, dissipating combustible vapors and cooling or eliminating ignition sources to assure that the minimum agent concentration is maintained for a duration sufficient to prevent re-ignition of the combustibles. An adequate fire extinguishing system should be installed in each fire zone region where other adequate means are not provided and which cannot be readily entered by a crew member with a portable fire extinguisher, or in which rapid extinguishment is essential.

4-3.3.2 Corrosion. Agents should not be used which will be injurious to the primary structure of the helicopter. It is desirable to use agents which will not be injurious to the contents of the compartment; however, effective fire extinguishment is the primary consideration.

4-3.3.3 Toxicity. Since all known fire extinguishing agents including CO2 are toxic to some extent, suitable safeguards should be taken to ensure that harmful quantities cannot enter habitable compartments by the ventilating or pressurization systems, intra-compartment leakage, or other means.

4-3.3.4 Agent Containers. Containers should provide a discharge valve(s), a pressure gage, a safety outlet and a filler port. The mounting system should have a static load capability of 14 times the loaded container weight.

4-3.3.5 Proof Pressure. The container and associated valves should be capable of withstanding an internal pressure of 1200 psi. without deforming.

4-3.3.6 Valve Passages. The valve passages should be of such size that 90% of the container contents at room temperature will be discharged under the tests conditions specified in paragraph 4 below.

4-3.3.7 Pressure Gage. A pressure gage should indicate the internal pressure of the container.

4-3.3.8 Safety Outlet. Each container should be furnished with a safety outlet incorporating a frangible disc type diaphragm or a fusible alloy type plug, in order to relieve excessive pressure that may occur in the container.

4-3.3.9 Filler Port. The container should have a filler port for introducing the agent and the pressurizing gas into the container.

4-3.3.10 Location of Agent Containers. The supply of agent or the valves controlling its flow should not be located in a fire zone.

4-3.3.11 Temperature Limitations of Agents and Containers. The agent containers should be either "winterized" for extreme temperature operation or so located in the helicopter that they should not be subjected to extreme temperatures. Safe limits for unwinterized carbon dioxide cylinders are approximately 0°F (18°C) to

140°F (60°C). Safe limits for "CB" and CH3BR containers are approximately -65°F (-54°C) to 200°F (93°C). The cartridge detonators have a variable age-with temperature limit. Contact should be made with the manufacturer for the latest information available for both installation and storage temperatures.

4-3.3.12 Control Systems. The fire extinguisher system may be either electrically controlled or cable controlled. The control system, either cables or electric wiring should not pass through any fire zones. Any portion of the controls which must be located in fire zones should be fireproof. If the system is electrically controlled, care should be taken to make certain the power source is not affected by fire control procedures.

4-3.3.13 Reserve Supply of Agent. A two shot system, capable of handling the largest hazard with release of one shot, must be installed for application to engine compartments on all multi-engine helicopters. Single shot systems are acceptable for baggage and cargo compartments, and for combustion heaters, etc.

4-3.3.14 Discharge Indicators. Indicators should be installed in the system to show when the agent containers have been discharged, either because of an overheat condition, or accidental or intentional discharge of the system. These discharge indicators should be so located that they may be easily inspected.

4-3.3.15 Eliminating or Inerting Fresh Air. When a compartment is to be flooded with agent and there is a source of fresh air entering the compartment, the incoming air should be either shut off prior to the release of the agent or rendered inert by directing extinguishing agent into the air blast or the quantity of agent should be increased to offset the incoming airflow.

4-3.3.16 Service Life. As a design objective, the components used in assembly of the fire extinguishing system should be so selected and installed that their expected service life is equal to that of the airframe. This guideline should not apply to explosive actuators.

4-3.3.17 Concentration of Agent. Actuation of the extinguishing system should produce a concentration of agent at least 6 percent by volume in all parts of the affected zone. This concentration should persist in each part of the zone for at least 0.5 second at normal cruising condition.

4-3.3.18 Duration of Discharge. The period of time required to discharge the calculated amount of agent should be 1 second or less, measured from the time the agent starts to leave the tubing ends until the required amount of agent has been discharged.

4-3.3.19 Agent Release. The release of agent from the container should commence within one-half second following actuation of the agent discharge switch.

4-4.0 **QUALIFICATION GUIDELINES**.

4-4.1 **Containment**. All components that are to contain a fire or components within a fire zone that carry combustible fluids should be subjected to the standard flame test as defined above. Any burn-through after the flame test such that the component fails to contain a flame or leaks flammable fluid should be considered as failing the test.

4-4.2 **Fire Detection Tests**.

4-4.2.1 Response Time. Each sensor should be exposed to the test flame and an alarm should occur in not more than __ seconds, as defined in the detail specification. The flame should then be blocked off and the alarm should clear in not more than __ seconds. For acceptance testing, a suitable colored light may be used in lieu of the standard flame test.

4-4.2.2 Sunlight Rejection. If the fire zone could be exposed to sunlight such that a detector could be inadvertently activated by solar radiation, a sunlight rejection test should be required. With the system operating at standby, one sensor should be exposed to the direct rays of sunlight for a period of __ seconds, as defined by the detail specification. The sunlight should not pass through a window or filter, and should be within __ degrees of the zenith, as defined in the detail specification. For test purposes, the illumination should be 5,000 foot-candles or greater, measured with a light meter facing into the sun. No alarm should occur.

4-4.2.3 Partial Extinguishment Test. A standard test flame should be applied for a period of 30 seconds. The test flame should then be masked so as to reduce its effective area by approximately 50 percent. The alarm signal should not clear. After an additional 30 seconds, the flame should be removed entirely, and the alarm signal should clear in __ seconds or less, as defined in the detail specification.

4-4.2.4 Saturation Test. The sensor should be mounted approximately 3 inches above the center of a flat pan, 2 feet in diameter, containing JP-4 fuel or 100 octane gasoline to a level 1/8 inch from the bottom. The fuel should be ignited by a source that cannot be detected by the sensor. An alarm should occur in 5 seconds or less, and should not clear while the sensor is exposed to this test for a period of one minute.

4-4.2.5 Fuel and Oil Immersion Test. The sensor should be thoroughly immersed in 100 octane gasoline (conforming to Specification MIL-G-5572) at room temperature, and then allowed to drain for one minute. The sensor should then be thoroughly immersed in oil (conforming to Specification MIL-L-7808), and removed and allowed to drain for one minute. No cleaning other than the drainage specified above should be permitted. The sensor should be exposed to the test flame and an alarm should occur in not more than __ seconds, as defined in the detail specification.

4-4.2.6 Repeat Response Test. The sensor should be exposed to the test flame for a period of one minute. The flame should then be blocked off. Within __ seconds, as defined in the detail specification, after the alarm has cleared, the sensor should again be exposed to the flame and an alarm should be signaled in __ seconds or less.

4-4.2.7 Power Variation Tests. Flame tests should be conducted under the input power conditions specified below. In each case, an alarm should occur in not more than __ seconds after exposure to flame, as defined in the detail specification, and should clear in not more than __ seconds after removal of the flame.

a. DC input systems: For system operating on DC power, the flame application and removal tests should be conducted with input voltages corresponding to the nominal aircraft voltage.

b. AC input systems: For systems operating on AC power, the flame application and removal tests should be conducted with combinations of aircraft representative input voltage and frequencies.

4-4.2.8 Fault in Conductor. With the system operating at standby, the integrity test switch should be actuated and the warning lights should illuminate. The test switch should be released. Each external connection should be grounded in turn to the control unit case and to the external ground lead, and a warning signal should not occur. Each external wire of the system should be individually disconnected and a warning signal should not occur.

4-4.2.9 Field of View Tests. The system should be operated with the maximum number of sensors for which it is designed. One sensor should be subjected to the test flame while all other sensors are kept completely shielded from light. The standard flame test should be performed, except that the viewing distance should be 36 inches. The test should be conducted with the exposed sensor aimed at appropriate viewing angles, measured from a reference line between the sensor and the flame center. The sensor should not be rotated about its viewing axis in the tests. In each test, an alarm signal should occur in __ seconds, as defined in the detail specification, or less after exposure to the flame.

4-4.3 **Fire Extinguishing Tests**.

4-4.3.1 Leakage. The container should be charged to its proper pressure and then heated to a temperature of 110 ± 5° F. While at this temperature, the container should be tested for leakage with a “sniffer” or any equivalent means a measuring leakage. There should not be any evidence of leakage or damage.

4-4.3.2 Mounting Provisions. A charged container should be mounted by its mounting provisions and loaded with weights which total __ times the total weight of the fully charged container. The container should remain loaded for not less than 30 minutes. At the end of the test, the mounting lugs should be examined. There should not be any failure, cracked weld, or permanent deformation.

4-4.3.3 Discharge. One charged container should be discharged through 10 feet of stainless steel tubing terminating in a tee having a total outlet area of 120 percent of the line area. Ninety percent of the contents of the container should be discharged in not more than __ second(s), as defined in the detail specification.

4-4.3.4 Proof Pressure. One container should be subjected to a proof pressure of ___ psi. as defined by the detail specification. The pressure should be retained for 10 minutes and then released. There should not be any permanent deformation or damage. Frangible disks, if used, should be replaced by steel disks furnished by the manufacturer.

4-4.3.5 Overheat Safety Outlet. Operation of the overheat safety device should be verified by one of the following tests, as applicable. The charged container should be subjected to internal pressure and the diaphragm should not rupture up to the proof pressure at 160°F. The charged container should be subjected to heat and the plug should not yield at a temperature up to 205° F.

4-4.3.6 Hydrostatic Bursting. One container should be hydrostatically tested to destruction and the bursting pressure recorded. The minimum bursting pressure should be not less than 1.5 times the proof pressure as defined in the detail spcification. Before pressure is applied, the frangible disk, if used, should be replaced by steel disks furnished by the manufacturer.

4-4.3.7 Environmental. The charged container should be subjected to the environmental tests below. Following each test, the container should be subjected to the Discharge test.

4-4.3.7.1 Temperature. The container should be subjected to the high and low temperature tests specified in MIL-STD-810.

4-4.3.7.2 Humidity. The container should be subjected to the Humidity test specified in MIL-STD-810.

4-4.3.7.3 Salt spray. The container should be subjected to the Salt spray test specified in MIL-STD-810 for 50 hours.

4-4.3.7.4 Vibration. The container should be subjected to the Vibration tests Specific in MIL-STD-810. One half of each resonant and cycling period should be conducted at room temperature and the other half of the periods should be conducted at 160°F.

4-4.3.7.5 Corrosion: Upon completion of the Environmental tests, the container should be examined for corrosion. There should not be any corrosion or pitting. Discoloration only should not be cause for rejection.

SECTION 5: DRIVE SYSTEM

5-1.0 REQUIREMENTS.

5-1.1 PERFORMANCE. The Drive system shall meet its allocated performance and function in such a manner that the aircraft shall be able to be operated safely throughout the operating envelope and meet the performance requirements as defined in the applicable weapon system specification.

5-1.2 QUALIFICATION. The following qualification requirements for the Drive system are required to verify compliance with the performance requirements above as applicable to the aircraft design configuration.

5-1.2.1 Analysis. Design and performance analysis shall be documented, using ADS-9C as a guide. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

5-1.2.2 Component Tests. The following tests shall be conducted and a subsequent teardown inspection, to determine the post test condition, shall be performed. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

a. **Lubrication Performance and Integrity**
    1. **Filters**
    2. **Debris Detection and Monitoring**
    3. **Heat Exchanger**
    4. **Heat Exchanger Blower**
    5. **Pumps**

b. **Drive Shaft System Integrity**
    1. **Shaft Critical Speed and Fatigue**
    2. **Coupling Fatigue**
    3. **Driveshaft Bearings**

c. **Bearings**

d. **Clutches**

e. **Rotor Brake/Gust Lock**

f. **Gears**

g. **Housings**

5-1.2.3 Assembly/System Level Tests.

a. **200 Hour Pre-Qualification Overstress Test**

b. **200 Hour Qualification Endurance Test**

c. **Oil Out Tests**

d. **Tie Down Tests**
    1. **50 Hour Pre-Flight Acceptance Test (PFAT)**
    2. **200 Hour Military Qualification Test (MQT)**
    3. **1250 Hour Reliability Endurance Test**

e. Aircraft Ground and Flight Tests in accordance with ADS-1B-PRF

5-2.0 **REFERENCES/DEFINITIONS.**

5-2.1 **Commercial Specifications**.

AS568	O-Rings, Aerospace Size, Standard For
AMS2438	Coating, Chromium, Thin Hard Dense Deposit
AMS2759	Heat Treatment, Steel Parts, General Requirements
AMS2759/6	Gas Nitriding and Heat Treatment of Low-Alloy Steel Parts
AMS2759/7	Gas and Vacuum Carburizing and Heat Treatment of Carburizing Grade Steel Parts
AMS2762A	Carburizing, Carbon and Low-Alloy Steel Parts

5-2.2 **Military Specifications**.

ADS-1B-PRF	Rotorcraft Propulsion System Airworthiness Qualification Requirements
ADS-9C	Propulsion System Technical Data
ADS-11B	Survivability Program, Rotary Wing
ADS-29	Structural Design Criteria For Rotary Wing Aircraft
MIL-HDBK-5	Metallic Materials And Elements For Aerospace Vehicle Structures
MIL-HDBK-17	Plastics For Aerospace Vehicles

5-2.3 **Definitions**.

5-2.3.1 Gearbox Power Rating is defined as the maximum input power level(torque and speed) at which the gearbox can operate for an unlimited duration.

5-2.3.2 Power For the purposes of life calculations, rotor shaft(s) power should equal the sum of all power input by the engine(s) or input shaft(s). Tail takeoff power should equal the rating of the associated tail drive gearbox.

5-3.0 **PERFORMANCE GUIDELINES.**

5-3.1 **Design**.

5-3.1.1 Drive System. The entire drive system, when operating as a unit, should transmit torque at the powers and speeds required by the aircraft specification from the engine(s) to all load absorbers (rotors, accessories, etc.)

5-3.1.2 System Criteria.

5-3.1.2.1 Life Limits. All drive system components should have a minimum 4500 hours life, based fatigue related failures, at any power level and duration allowed by the flight spectrum. There should be no fatigue or wear related failure to gearbox dynamic components for operation at any power level (and it's associated duration) which can be input (on a one-time basis) by the engine(s) or through the rotor system. Fatigue calculations should be based on 3σ working curves. The aircraft system design contractor should define the power and time limits associated with mission spectrum operation at powers above the Gearbox Power Rating up to the maximum input power.

5-3.1.2.2 Gears. At the Gearbox Rated Power, all gears should have infinite life (>10^7 cycles) in tooth bending, and 4500 hours life in tooth contact, using standard gear life calculation methodologies. For dual-engine drive gearboxes, the gears located before the gear which combines the load from each engine should have infinite life in tooth bending, and 4500 hours life in tooth contact, at the Gearbox OEI Rating. For operation above the rated power as specified in the flight spectrum, all gears should have a minimum 4500 hour life in tooth bending. The influence of gearbox deflections should be included in the life analysis. Gears should be insensitive to scoring for all possible combinations of load and temperature.

5-3.1.2.3 Bearings. Unless otherwise specified, all rolling element bearings should have a minimum B_{10} life of 4500 hours based on operation at 71% of the Gearbox Power Rating. Unless otherwise specified For dual-engine drive gearboxes, the bearings located before the gear which combines the load from each engine should have a minimum B_{10} life of 4500 hours based on operation at 71% of the Gearbox OEI Rating. Rolling element bearings in magnesium, aluminum, or composite housings should be installed in liners which are retained or locked by a positive method to prevent rotational and axial motion. Unless otherwise specified Drive shaft bearings should have a minimum B-10 life of 4500 hours based on operation at 71% of the associated gearboxes Gearbox Power Rating.

5-3.1.2.4 Pressurized Lubrication Systems. For pressurized lubrication systems, the lubricant should be provided at the required pressure and flow rate to all required components and accessories at all allowed gearbox attitudes. Suitable means should be provided for setting the gearbox internal pressure and flow to the required level during steady state ground operation, and for maintaining required gearbox internal pressures under all operating conditions and gearbox attitudes. Breathers should be equipped with filtration devices which remove air-borne particles of 10-microns or larger. Breathers should be arranged to prevent loss of oil from the gearbox under all operating conditions and gearbox attitudes. The operating oil temperature and pressure should be continuously monitored, and provisions should be made for display of the oil pressure and temperature, as well as the operating limits, on cockpit instrumentation.

5-3.1.2.5 Non-Pressurized Lubrication Systems. For non-pressurized lubrication systems, the lubricant should be provided to all required components and accessories at all allowed gearbox attitudes. Suitable means should be provided for maintaining required gearbox internal pressures and flows under all operating conditions and gearbox attitudes. Breathers should be equipped with filtration devices which remove air-borne particles of 10-microns or larger. Breathers should be arranged to prevent loss of oil from the gearbox under all operating conditions and gearbox attitudes. The gearbox design should maintain the gearbox oil temperature below the maximum allowed under all possible combinations of gearbox power level, ambient conditions, gearbox attitudes, and lubricant levels.

5-3.1.2.6 Loss of Lubricant. All gearboxes should continue operating for at least 30 minutes at drive system power required by ADS-11, paragraph 5.2.1.2.b, after loss of lubricant or lubrication system.

5-3.1.2.7 Vibration. The transmission drive system should be free of destructive vibration at all operating speeds and powers, including steady-state and transient operation. There should be no unfavorable dynamic coupling modes when the engine, engine accessories, the rotor system, and all transmission dynamic components are operated as a combined dynamic system. Vibration limits should be as defined by the contractor.

5-3.1.2.8 Dephasing. For multi-rotor rotorcraft, provisions should be designed into the drive system to ensure that phasing of the rotors is accomplished easily and in a fail-safe manner. Means should be included for remote indication that the rotors are locked in phase.

5-3.1.2.9 Torsional Stability. The aircraft should be free of any coupled engine/drive system/rotor system instabilities and resonance's at any operational rotor speed up to the limit rotor speed, power on or off (reference ADS-29).

5-3.1.2.10 Environmental Conditions. The drive system should not be degraded by continuous operation in any environmental condition allowed by the aircraft specification.

5-3.1.2.11 Torque Measurement. Operation at power levels specified in the flight spectrum which are above the Gearbox Power Rating should be monitored using a torque measurement system separate from the engine torque measurement system. The torque measurement system should allow accurate in-flight monitoring of input torque to each rotor gearbox. Excursions in power/time beyond that allowed by the flight spectrum should be recognized, recorded, and made available to the pilot, flight crew, and maintenance personnel as appropriate to track the life of the component/gearbox.

5-3.1.2.12 Crash Loads. All gearbox support structures will prevent the gearbox from being displaced into occupied space at crash load factors equal to or less than those specified for a survivable aircraft crash.

5-3.1.3 Transmission and Gearbox Ratings.

5-3.1.3.1 Single-Input Main Rotor Gearboxes. The Single-Engine (One Engine Inoperative - OEI) Gearbox Power Rating should be equal to the engine's uninstalled, sea-level Contingency Rated Power.

5-3.1.3.2 Dual-Input Main Rotor Gearboxes. The Dual-Engine Gearbox Power Rating should be equal to 110% of the engine's uninstalled, sea-level standard Maximum Continuous Power Rating times the number of engines.

5-3.1.3.3 Tandem-Rotor Gearboxes. For tandem-rotor helicopters, the Gearbox Power Rating of each rotor gearbox is equal to 60% of the engine's uninstalled, sea-level standard Maximum Continuous Power Rating times the number of engines.

5-3.1.3.4 Tail Rotor Drive Gearboxes. Unless otherwise specified The Gearbox Power Rating for tail rotor drive gearboxes should equal 15% of the helicopter's main rotor Gearbox Power Rating.

5-3.1.3.5 Engine Nose Gearboxes. The Gearbox Continuous Power Rating should be equal to 110% of the engine's uninstalled, sea-level standard Maximum Continuous Power Rating. The gearbox should be designed to be compatible with the engine's uninstalled, sea level Contingency Rated Power for same time duration and number of occurrences as the engine.

5-3.1.3.6 Tandem Rotor Combining Gearboxes. The Gearbox Power Rating should be equal to 110% of the engine's uninstalled, sea-level standard Maximum Continuous Power Rating times the number of engines; the Single-Engine (One Engine Inoperative - OEI) Gearbox Power Rating should be equal to the engine's uninstalled, sea-level Contingency Rated Power.

5-3.2 **Components.**

5-3.2.1 Standardization. Standardization principles should be applied in all phases of initial design and to all design changes. If standard parts are found to be unsuitable for use in specific applications, the variety of special parts used will be minimized to the most practical limit.

5-3.2.2 Standard Parts. AN or MS standard parts should be used whenever specified, and identified by their standard part numbers. Commercial Standard Parts developed specifically for use in aircraft propulsion systems are preferred and should be considered for use prior to consideration of other parts. Where general purpose standards, as defined by envelope dimensions or Qualified Products List (QPL's), are used in critical or high strength applications, parts should be identified by the manufacturer's part number. Parts derived from general purpose standards or Military Standards solely on an inspection or selection basis should be identified by contractor part numbers and all previous identification marks should be removed.

5-3.2.3 Materials and Processes. Commercial material and process specifications should be used in the manufacture of transmission drive system components whenever possible. The use of non-governmental specifications should not constitute waiver of Government inspections. When manufacturer's specifications are used for materials and processes, such specifications should be submitted to the using service for review prior to start of component testing and will be considered released upon successful conclusion of Qualification Testing.

5-3.2.3.1 Connection Identification. Transmission components should be permanently marked to indicate all instrumentation and lubrication connections.

5-3.2.3.2 Threads and Inserts. Standard screw thread forms should be used whenever possible. Taper threaded plugs should not be used in castings and nonferrous parts. Tapered pipe threads may be used only for permanent plugging of drilled passages or openings in steel.

5-3.2.3.3 Fasteners. Threaded fasteners and other connections should be self-locking or otherwise secured to prevent loosening under all operating conditions. Cotter pins and safety wiring is not permitted.

5-3.2.3.4 Processes. The use of processes which present a threat to the environment such as chrome and cadmium plating are prohibited.

5-3.2.3.5 Thin Dense Chrome Plating. The use of Thin Dense Chrome (TDC) for load bearings should be in accordance with AMS2438.

5-3.2.3.6 Shot Peening. No degradation of endurance stress allowables should be permitted.

5-3.2.3.7 Materials. Polyvinyl chloride, asbestos, mercury, Kapton wire, and silicone should not be used. Metals used for drive system components should be corrosion resistant, or should be suitably coated or treated to resist corrosion likely to be encountered during storage or normal service. Dissimilar metal combinations or assemblies should be avoided, unless when sealed or insulated to prevent galvanic corrosion. Physical and mechanical properties of metals should be in accordance with MIL-HDBK-5. Plain carbon and free machining steels should not be used. Vacuum melted steels should be used in applications requiring heat treatment to a minimum tensile strength of 1242 Mpa or above. Use of steels heat treated to 1380 Mpa or above should be discouraged. Heat treatment, carburizing, and nitriding of steel should be in accordance with AMS2759, AMS2759/6, AMS2759/7, and AMS2762A. Elastomeric materials should be resistant to ozone agents, fluids, and chemical agents. The use of age-sensitive elastomeric materials is prohibited. All nonmetallic O-ring seals and packing used in the transmission drive system should conform to AS568. Composite use should be in accordance with MIL-HDBK-17.

5-3.2.3.8 Castings. Castings should be clean, sound and free from blow holes, excess porosity, cracks, and other defects which may reduce the physical properties of the casting below the design requirements.

5-3.2.3.9 Lubricants. Lubricants should be functional at all transmission drive system operating temperatures.

5-3.2.3.10 Coatings and Finishes. Coatings and finishes should be environmentally safe and should be resistant to corrosion and surface damage when exposed to oil, fuel, and/or a salt water environment. Internal and external coatings for detail components should be Araldite Rockhard or equivalent. External coatings and finishes for assemblies should be in accordance with the external finish specification of the aircraft.

5-3.2.3.11 Sealant. Sealant should be used at parting planes of gearbox housings to prevent intrusion of moisture into faying surfaces, therefore helping to reduce corrosion. The sealant should be of low adhesion type with low peel strength to facilitate easy removal during disassembly.

5-3.2.4 Manufacturing and Fabrication Vendor or Fabrication Source Identification. All transmission drive system components should be marked directly with identification of the vendor or fabrication source of that particular part, unless the physical size of the part prevents doing so.

5-3.2.4.1 Part Serialization. All flight critical parts, life limited parts; and or parts requiring traceability should be serialized. This includes but is not limited to gears, bearings, shafts, flanges, and major housing assemblies. Records should be maintained so that each serialized component is traceable to the forging and heat treatment lot identification number, where applicable.

5-3.2.5 Accessory Drives. Accessories driven by the gearbox should be driven during an auto-rotation or whenever the rotor system is rotating. Accessory drive pads should allow for removal of accessories without removal of other components.

Accessory drive splines should be protected with spline inserts. Cover plates should be provided for use when accessories are not installed. Failure of accessories should not cause failure of the rotor drive system. The accessories should be designed so that there should be no damage to the MGB if an accessory should seize. Gears should not have any resonances which affect the strength of the gear at all possible operating conditions.

5-3.2.6 Housings. Deflections of main gearbox and transmission housings should not affect the durability or life of any internal components under any combination of input loading and aircraft operating condition.

5-3.2.7 Clutches. Clutches should be provided to permit engagement or disengagement of load absorbers, engines, or accessories as required for allowed modes of helicopter operation. During auto-rotation and single engine operation, the engine(s) not supplying torque should be immediately and automatically decoupled from the drive subsystem by an overrunning clutch(es) to permit continued operation of the main rotor, tail rotor and accessory drive. The bearings and lubrication subsystem for the clutches should be such that a two hour period of full overrunning can be accomplished without damage to the components. The location of the clutch(es) should allow for the continuation of necessary functions of the transmission system and safe operation of the aircraft for any combination of clutch engagements or disengagement's. Clutch engagements and disengagement's should not damage components of the clutch or transmission drive system. Overrunning for at least 30 minutes with complete loss of gearbox lubricant should be possible.

5-3.2.8 Rotor Brake/Gust Lock. A means of preventing rotation of the rotor in winds up to 45 knots should be provided. The system must be capable of restraining the rotor system at engine powers up to and including ground idle upon startup. The system must be able to be operated from the cockpit, and should be capable of 1000 engagements without failure of any of the parts. A mechanical interlock system should be provided to prevent actuation of the system if either engine control is forward of the ground idle position.

5-3.2.8.1 Gust Lock. The gust lock is a positively-locking mechanical device designed to prevent rotation of the rotor. If a gust lock is provided as the means to prevent rotation of the rotor, provisions must be made for incorporation of a rotor brake.

5-3.2.8.2 Rotor Brake. A rotor brake should be provided for any rotorcraft required to perform shipboard operations. The rotor brake is a hydraulically operated braking device consisting of a disc, friction surfaces, and an actuating mechanism designed to stop the rotor. With the engines off, the rotor brake should stop the main rotor from 50% speed in not more than 50 seconds and hold the rotor in the stopped position.

5-3.2.9 Lubrication Subsystem.

5-3.2.9.1 System. The lubrication system should provide lubricant to all required components and accessories under all attitude conditions which may be imposed by operation of the aircraft as allowed by the aircraft specification. Gearboxes and accessories which are lubricated by the gearbox lubrication system should be adequately lubricated during auto-rotation. Failure of a gearbox-driven accessory or an accessory lubricated by the gearbox lubrication system should not cause failure of the gearbox, and should not contaminate the gearbox lubrication system with debris from the accessory. All oil passages connecting points in the same gearbox should be located within the gearbox and should incorporate jet-protection screens which prevent the lubricating jets from becoming clogged with debris.

5-3.2.9.2 Cooling Fans. Failure of heat exchanger blowers should not cause failure of the drive system or any of its components. Suitable provisions should be made for monitoring of the blower condition.

5-3.2.9.3 Heat Exchanger. Suitable heat exchangers(s) should be provided to maintain the gearbox lubrication system oil-in temperature below the maximum allowed under all possible operating combinations (gearbox power level, ambient conditions, lubricant flow rates and pressures, gearbox attitudes, and air flow rates). An integral bypass should be provided to bypass the heat exchanger when the oil temperature is low.

5-3.2.9.4 Valves/Pressure Pump(s). No air traps should exist at the lubrication pump inlet(s). The pump(s) should provide the required oil flow rate and pressure without degradation of performance at all altitudes up to and including the helicopter's maximum operating altitude. The lubrication pump should be a line replaceable unit.

5-3.2.9.5 Sensors. Debris monitors should be utilized on all oil-lubricated gearboxes and transmissions. For pressurized oil systems, debris monitors should be located such that all gearbox lubricant passes through the debris monitor. For non-pressurized lubrication systems, the debris monitor should be located such that wear debris from any internal components is likely to migrate quickly to the debris monitor. The gearbox oil system should have no areas where generated wear debris can collect and not be passed to the debris monitor. Debris monitors should provide cockpit indications of abnormal debris generation rates or sizes. Debris monitors should be removable without draining gearbox lubricant. For gearbox lubrication systems with remote components and/or accessories, debris monitors should isolate the component or accessory generating the debris. Debris monitors should be placed before any lubrication pump(s) or filter(s). A removable, cleanable screen should be installed downstream of the debris monitor to protect the lubrication pump(s). A remote method of ensuring sensor circuit continuity should be provided.

5-3.2.9.6 Lubricant. Gearbox lubricants should use any oil conforming to and having any variation in characteristics permitted by DOD-L-85734 (preferred) or MIL-L-23699 for normal operating conditions, and MIL-L-7808 for cold weather conditions. Other oils may be used if substantiating data is provided which verifies its benefits and logistics impact to the field.

5-3.2.9.7 Filtration. Oil filters should be utilized on all pressurized lubrication systems. Oil filters should be installed such that all oil passing through the gearbox pressure pump(s) should immediately pass through the filter. The filtration ratio should be $\beta 3>200$ and $\beta 2>100$. Filter elements should be non-cleanable, throw-away type. Dirt holding capacity should be sufficient to hold 25 grams AC fine test dust. An integral bypass should be provided to bypass the filter element in the event the filter element becomes clogged. Suitable indications should be provided in the event of impending and actual bypass conditions. The impending bypass indication should be set such that a minimum of 10 hours of gearbox operation is possible before actual bypass occurs. The indicator should only be re-settable when the filter is replaced. The filter impending bypass indicator should have a thermal lockout feature that prevents actuation of the indicator when the oil temperature is low. The filter should be replaceable without removal of any other gearbox component, and without draining of all the lubricant. The filter housing should be oriented such that entrapped contaminants and unfiltered oil within the filter housing is removed when the filter element is removed. The contractor should specify the filter operating flow and pressure ratings.

5-3.2.9.8 Marking. Oil inlet and outlet connections in close proximity to each other should be oriented or sized so that reverse connection of lines is physically impossible.

5-3.2.9.9 Seals. Seal material and design should be selected to operate continuously at the maximum transmission and gearbox operating temperature and speed for 1500 hours. Transmission and gearbox lubricant should not be completely depleted in the event of a rotating shaft seal failure. No external leakage from the transmission or gearbox onto the aircraft structure should be permitted. The static seals should permit leakage. Rotating shaft seals should be selected for maximum reliability, and in those cases where a small amount of oil seepage is possible, provisions should be made for collection to overboard drains. Seals should be designed as a line replaceable units, where possible.

5-3.2.10 Drive Shaft Subsystem.

5-3.2.10.1 Drive Shafting. Shaft whirling critical speeds should be at least 30% from aircraft steady-state operating speeds, including idle, all flight conditions, and auto-rotation. Damping of supercritical shafts should be provided such that stress amplifications do not exceed the design limitations. The shafts should be dynamically balanced and should accommodate installation misalignment and aircraft frame deflections.

5-3.2.10.2 Couplings. The torque and misalignment capabilities of drive shaft couplings should be suitable for all possible combinations of torque and speed when installed in the aircraft at the maximum permissible misalignment. Couplings should be fail-safe. Replacement of couplings should not be cause for realignment of the associated shafting.

5-3.2.10.3 Bearings. Grease-lubricated bearings should incorporate methods for field checking, servicing, and replacement of the lubricant as required. A self-aligning feature should be provided for the bearing component of each hanger bearing assembly.

5-3.3 **Analysis**. The Following Analysis should be conducted in accordance with ADS 9:

Torsional Analysis
Lubrication System Analysis
Drive System Load /Life Analysis
Critical Speeds Analysis
Critical Parts Analysis

5-3.4 **Development**.

5-3.4.1 Tests. The following development testing should be accomplished in accordance with ADS 9 in support of the design of the system:

Lubrication System
Flow Rate / Oil Capacity
Jet Targeting
Windage Loss

5-3.4.1.1 Attitude Test. The contractor should conduct a no-load lubrication attitude test. The no-load attitude test setup should include the entire gearbox lubrication system. The no-load lubrication attitude test should be used to establish operating pressures, flow rates, jet targeting, scavenging, lubrication quantities, and cooling adequacy. Input speeds will range from 25% Nr to the maximum speed allowed by the aircraft specification. The transmission should be varied up to ± 30° from the neutral axis in roll and pitch. Testing should use combinations of roll and pitch to address all allowed aircraft flight attitudes. At each condition, lubrication system parameters and gearbox bearing and cooling system temperatures will be evaluated. Lubrication quantity and flow rates, jet sizes, and/or jet targeting and location will be modified as required until acceptable performance is attained. Performance of pressure bypass valves and breathers should be evaluated during this test.

5-3.4.1.2 Gear development. The contractor should also conduct the following tests:

a. Gear Pattern Development Test. The contractor should build-up a gearbox, using the latest configuration parts and assembly procedures, for the purpose of developing the gear bearing patterns of all internal gears. The power level at which the patterns are optimized should be defined and justified by the contractor. The patterns should be acceptable over the range of loading specified in the flight spectrum. There is no duration requirement for this test.

b. Gear Load Distribution Survey. Following successful completion of the gear pattern development test, a gear load distribution survey should be conducted. The contractor should build-up a gearbox, using the latest configuration parts and assembly procedures, for the purpose of performing a gear load distribution survey. The primary load gear teeth should be strain-gauged in the gear teeth roots, gear webs, and gear rims. The gearbox will be operated at input loads of up to at least 120% of Gearbox Power Rating and at speeds up to the maximum speed allowed by the aircraft specification. Gear loads should be under the design allowables.

c. Gear Tooth Bending Fatigue Test. The gearbox should be operated until 10^7 cycles are run on each primary drive gear in the gearbox at it's appropriate Gearbox Power Rating power level. There should be no gear tooth bending failures during this test.

5-3.4.2 Tradeoffs. Tradeoff studies should be accomplished as required as part of design effort to optimize the design in accordance with the overall aircraft criteria.

5-4.0 **QUALIFICATION GUIDELINES**.

5-4.1 **Qualification and Acceptance**. Testing can be classified as Acceptance tests and Qualification tests. Qualification tests of the system should be accomplished through testing of the components; the major sub assemblies such as transmission and gearboxes; as well as system testing conducted on both ground tie-down vehicles and flight test aircraft. Each production transmission and gearbox assembly should be subjected to an Acceptance test and inspection prior to delivery.

5-4.2 **Component Qualification Tests**. The contractor should perform component qualification testing on materials, components, and assemblies as specified in this Section. All test results, findings, data, and other pertinent information should be documented. No testing of a system may begin unless all applicable component tests have been completed.

5-4.2.1 General.

5-4.2.1.1 Connection Identification. The contractor should provide drawings showing the location and marking of all external gearbox lubrication and instrumentation connections.

5-4.2.1.2 Threads and Inserts. The contractor should provide a listing of all threaded connections used in the drive system, including threaded inserts and plugs.

5-4.2.1.3 Fasteners. The contractor should provide a listing of all threaded fasteners and other connection used on the drive system, and should specify the type of locking device used for each fastener or connection.

5-4.2.2 Lubrication System.

5-4.2.2.1 Filters. To substantiate compliance with the requirements and to verify the durability and integrity of the filter assembly, the following tests as a minimum should be completed on a filter assembly that is representative of the production filter assembly.

a. Filter element multi-pass filtration ratio test.
b. Maximum particle passed test.
c. Filter bypass performance test.
d. Impending bypass indicator performance test.
e. Cold start/temperature lockout test.
f. Filter assembly clean pressure drop test.
g. Filter assembly proof pressure test.
h. Filter housing burst test.
i. Filter element bubble point test.
j. Filter element collapse pressure test.

5-4.2.2.2 Debris Detection and Monitoring. To substantiate compliance with the requirements, a debris monitor(s) that is representative of the production unit(s) should be tested to substantiate that it can detect debris of the size, and shape, and material defined by the contractor to be characteristic of debris which may be considered abnormal.

5-4.2.2.3 Heat Exchangers. To substantiate compliance with the requirements, the following tests as a minimum should be completed on a heat exchanger that is representative of the production units.

a. Oil pressure drop test.
b. Decongeal test.

c. Anti-congealing test.
d. Static pressure test.
e. Pressure cycling test.
f. Thermostatic control valve performance tests.
g. Environmental Testing (including vibration)

5-4.2.2.3.1 Heat Exchanger Blowers. To substantiate compliance with the requirements, the following tests as a minimum should be completed on a heat exchanger blower that is representative of the production units.

a. Self-induced vibration test.
b. Performance tests.
c. 200-hour endurance test.
d. Containment test (damage tolerance)

5-4.2.2.4 Pressure Pump(s). To substantiate compliance with the requirements, the following tests as a minimum should be completed on a pressure pump that is representative of the production units.

a. Performance verification test.
b. 500-hour endurance test.
    (1) 200-hours with no contamination.
    (2) 150-hours with water contamination.
    (3) 150-hours with water/metal contamination.
c. 50-hour altitude endurance test.

5-4.2.3 Drive Shafting. The critical whirling speed of all shafting should be determined by demonstration. Demonstration of critical speeds on super-critical shafts should include measurement of stresses at that speed to insure they are within design limits.

5-4.2.3.1 Couplings. To substantiate compliance with the requirements, the following test as a minimum should be completed on a coupling that is representative of the production units.

a. Two couplings should undergo an endurance fatigue test run at the maximum permissible misalignment and at 110% of the maximum torque seen by the coupling in service.
b. One of the above couplings should be run until inspection reveals the coupling has become unserviceable. The test should continue until three times the normal inspection period for the coupling has elapsed. The coupling should not fail to transmit torque within that time interval. If the coupling is so designed as to not become unserviceable within an acceptable period of testing, then a determination as to the safe inspection interval will be made at that time.

5-4.2.3.2 Driveshaft Bearings. The contractor should demonstrate the method for checking and servicing drive shaft grease-lubricated bearings. The bearings should also demonstrate the ability to operate without failure at the maximum allowable shaft misalignment.

5-4.2.4 Bearings. The contractor should provide analysis in accordance with ADS-9 for all drive system bearings. Details of the bearing life calculation methodology used should be provided. In addition, the contractor should conduct a bearing temperature survey. The gearbox will be operated at input loads of up to 120% of Gearbox Power Rating and at speeds up to the maximum speed allowed by the aircraft specification. Bearing temperatures should be within design allowable.

5-4.2.5 Clutches. To substantiate compliance with the requirements, the following tests, as a minimum, should be completed on a clutch assembly that is representative of the production units.

a. 3000 Cycle Endurance Engage/Disengage Test
b. 2 Hour Overrunning Test
c. 30 Minute Loss of Lubrication Test

5-4.2.6 Rotor Brake/Gust Lock. A 1000 cycle engage/disengage endurance test should be performed to verify the ability of the gust lock to perform repeated startup cycles at the specified powers without failure. The performance of the mechanical interlock system to prevent actuation if either engine control is forward of the ground idle position should be demonstrated on the Tie Down System. The ability of the rotor brake to stop the rotor within the specified time from the specified power/rotor speed should be demonstrated on the aircraft or Tie Down Rig. The ability of the rotor brake to hold the rotor in the specified wind conditions should be demonstrated.

5-4.2.7 Gears. Gears should be tested to evaluate resonance characteristics. Test data should be used to determine the resonant frequency and mode shape of each gear. The contractor should provide analysis in accordance with ADS-9 for all drive system gears. Details of the gear life calculation methodology used should be provided.

5-4.2.8 Housings. A finite element analysis of the main gearbox and transmission housings should be provided. The deflections of the housings under worst-case conditions of input loading and aircraft flight condition should be determined. The main gearbox housings should be included as part of the transmission mounting system fatigue tests. Housings should be instrumented during bench tests to determine deflections/loads and thermal dissipation properties.

5-4.3 **Subsystem Bench Tests**.

5-4.3.1 Test Configuration. The gearbox configuration undergoing system development testing should reflect the latest changes as documented in the gearbox drawings. The contractor should document all gearbox components being tested for each specific test, including component part and serial numbers, vendor, and total test time. If a component is replaced during testing, the reason for replacement should be annotated.

5-4.3.2 Gearbox Test Stands. The gearbox test stand used for system testing should be capable of operating at variable speed (up to 125% Nr) and of imposing all power and load parameters (up to 130% of the maximum load) encountered under any rotor speeds and flight conditions allowed by the aircraft flight spectrum. When applicable, the test stand should be capable of imposing shaft loading consistent with operation of the aircraft as allowed by the flight spectrum. The gearbox lubrication system should not be augmented or bypassed. The test stand should provide for monitoring of all gearbox operating parameters. Accessory drives should be loaded to their maximum loading and overhung moments as specified in the aircraft specification.

5-4.3.3 Pre-Qualification (Overstress Tests). One 200-hour bench overstress test and one gear-tooth bending fatigue test (10 million cycles on all gear teeth) should be accomplished on the transmission and gearbox configuration to verify the tooth bending, surface stress, and fatigue life of all gears; to determine failure modes; and to identify potential limits of operation and growth capability. The overstress tests should be of the latest configuration gearbox, using the optimized patterns obtained during the pattern development test. The loading spectrum should be negotiated and should reflect all operational speeds. As a minimum, the test spectrum should include operation at:

Cond.	Input Power Level	Duration (Minimum)
1	120% of Gearbox Rated Power	80 hr.
2	120% of Gearbox Rated OEI Power (if applicable - each input)	20 hr.
3	100% of Gearbox Rated Power	60 hr.

Cond.	TTO Power Level	Duration (Minimum)
4	120% of Tail Takeoff rating (if applicable)	60 hr.

Rotor power levels should be equal to total input power minus TTO power (if applicable). The test should also include operation at 120% of any power level allowed by the flight spectrum that is above the Gearbox Rated Power for four times the duration specified in the flight spectrum, for both input and TTO loadings. Power levels for the remaining time should be at the contractors discretion. Accessory drives should be loaded at the nominal rating. Component replacement is allowed during the overstress test; however, the gearbox should not exhibit any failure or wear modes which might in any way restrict the operation of the rotorcraft in accordance with the aircraft specification. When component design changes are made during the test, the test duration should be increased until the new component is tested for the duration required for conditions 1, 2 (if applicable) and 4 (if applicable) above. The testing should verify operation with each lubricant at both normal and elevated temperature limits. A tear-down inspection, along with resolution of all discrepancies, should be required after each test and before initiation of the qualification bench test.

5-4.3.4 Qualification Tests. For each production source, a 200-hour bench qualification test should be accomplished on each transmission and gearbox configuration proposed for production. The configuration should be audited prior to the test to verify that the essential production configuration with all design configuration changes are incorporated. A loading spectrum should be established which reflects mission profiles and verifies latest design changes. A tear-down inspection along with resolution of all discrepancies should be required following the test.

5-4.3.5 Oil Out Test. Following completion of the test for each production source, two each, 30-minute duration, oil-out bench tests should be conducted on each transmission and gearbox configuration. Test article tolerances and clearances should be recorded prior to test and should be representative of the production configuration with respect to both "new" and "worn" units. The test load and ambient conditions should simulate the operation of the drive system as required by ADS-11, paragraph 5.2.1.2.b. The 30-minute time interval should start when the low-level warning system is activated. Successful completion should be defined as a single, 60-minute run at these same flight and landing loads and should be considered as satisfying the requirement for two, 30-minute tests.

5-4.4 Tie Down System Test. The specific Propulsion System Test Bed (PSTB) used (e.g., Iron Bird, Ground Test Vehicle, etc.) should be as specified in the contract, but should as a minimum consist of all components and systems that are associated with the propulsion system. This should include (but is not limited to) the engines, drive system, rotor system(s), rotor and engine control systems, hydraulic and pneumatic systems, environmental control system, starter system, fire detection and extinguishing systems, auxiliary/subsystem power units, electrical system, and fuel system. The test hours should accrue at a rate to lead the flight hours of the lead flight test vehicle by at least a 2 to 1 ratio until that high time vehicle attains 200 hours of flight time. The intent is to expose the propulsion components to actual powers/loads as part of the integrated aircraft system before they are encountered on the aircraft. This serves to reduce the flight test risk. During all phases of testing, degraded modes of operation should be investigated. A complete tear-down and inspection of all systems noted above should be accomplished after each segment of the test noted below. The test spectrums should be as defined below.

5-4.4.1 50 Hour Pre Flight Acceptance Test. A 50 hour Pre Flight Acceptance Test (PFAT) should be completed before the first aircraft flight. Tear-down inspection results should be utilized to establish inspections/limitations/restrictions for the flight release. The configuration of the propulsion components on the PSTB should be identical to the flight test aircraft configuration. The power spectrum described below is based on the uninstalled power ratings of the engine. The test spectrum should be 5 repetitions of the 10 hour tie down test cycle as follows:

a. Maximum Power (MP) Run (One Hour). One hour of alternate runs of 5 minutes at MP and speed, and 5 minutes at as low an engine idle speed as practicable with the engine declutched from the transmission system and the rotor brake applied during the first 1 minute of the idle run. When declutching the engine, it should be decelerated at the maximum rate possible in order to permit the operation of the overrunning clutch. During the remaining 4 minutes of the idle run, the engine should be clutched to the transmission system. Acceleration of the engine and the transmission system should be accomplished at the maximum rate.

b. Intermediate Power(IRP) Run(One Hour). One hour of alternate runs of 30 minutes at IRP and speed, and 30 minutes at 60 percent at normal rated speed.

c. Military Power (MCP) Run (Three Hours). Three hours of continuous operation at MCP and speed. During the normal rated run, the main rotor controls should be operated through a minimum of 15 cycles per hour consisting of the following main rotor pitch positions: (1) full vertical thrust, (2) maximum forward thrust component, (3) maximum aft thrust component, (4) maximum left thrust component, and (5) maximum right thrust component, except that this control movement need not produce loads or blade flapping motion exceeding the maximum loads or motion encountered in flight. The directional controls should be operated through a minimum of 15 cycles per hour consisting of (1) maximum right thrust, (2) neutral thrust as required by the power applied to the main rotor, and (3) maximum left thrust. Each control position should be held at maximum for at least 10 seconds.

d. 90% MCP Run (One Hour). One hour of continuous operation at 90 percent of MCP at normal rated speed.

e. 80 % MCP Run (One Hour). Two hours of continuous operation at 60 percent of MCP at the minimum desired cruising speed, as recommended by the contractor, or in lieu of this speed, at 90 percent of normal rated speed, whichever speed is lower.

f. 60% MCP Run (Two Hours). Two hours of continuous operation at 60 percent of MCP at the minimum desired cruising speed, as recommended by the contractor, or in lieu of this speed, at 90 percent of normal rated speed, whichever speed is lower.

g. Overspeed Run (One Hour). One hour of continuous operation at 110 percent of normal rated speed at MCP. If operation of the helicopter power plant(s) at 110 percent normal rated speed is prohibited, the speed of this run should be the highest speed permissible as recommended by the contractor.

5-4.4.2 200 Hour Military Qualification Test(MQT). The test spectrum should be 20 repetitions of the above 10 hour tie down test cycle.

5-4.4.3 Overspeed Test. After the completion of the 200-Hour MQT and without intervening major disassembly, the transmission system should be subjected to 50 overspeed runs, each 30 +3 seconds in duration at 120 percent of normal rated speed. If operation of the helicopter powerplant(s) at 120 percent of normal rated speed is prohibited, the speed employed should be the highest speed permissible, as recommended by the contractor. Overspeed runs should be alternated with stabilizing runs of 1 to 5 minutes duration each from 60 to 80 percent of normal rated speed. Acceleration and deceleration should each be accomplished in a period not longer than 10 seconds, and the time for changing speeds should not be deducted from the specified time for the overspeed runs. Overspeed runs should be made with the rotor(s) in the flattest pitch at which smooth operation can be obtained.

5-4.4.4 1250 Hour Reliability/ Maintainability Test. This test should commence following complete disassembly and inspection of the drive system components at the end of the MQT. The test spectrum for this endurance test should consist of a composite of the mission profile for the aircraft. The test spectrum should be proposed by the contractor.

5-4.5 **Ground/Flight Testing**. Ground and flight tests should be in accordance with ADS-1B-PRF.

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SECTION 6: FUEL SYSTEM

6-1.0 REQUIREMENTS.

6-1.1 PERFORMANCE. The Fuel system shall meet it’s allocated performance and function in such a manner that the aircraft shall be able to be operated safely throughout the operating envelope and meet the performance requirements as defined in the applicable weapon system specification.

6-1.2 QUALIFICATION. The following qualification requirements for the Fuel system are required to verify compliance with the performance requirements above as applicable to the aircraft design configuration.

6-1.2.1 Analysis. System design and performance analysis shall be documented, using ADS-9C as a guide. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

6-1.2.2 Component Tests. The following tests shall be conducted and a subsequent teardown inspection, to determine the post test condition, shall be performed. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

a. **Fuel tank crashworthiness - 65 foot drop test, full of water**
b. **Fuel tank slosh and vibration - 40 hour, 2/3 full**
c. **Fuel tank self-sealing - 12.7 and 14.5 mm gun fire**
d. **Pressurization/Explosion Suppression subsystem performance**
e. **Fuel tank pressure capability**
f. **Fuel tank static discharge**
g. **Fuel vent rollover and thermal relief**
h. **External surface fuel resistance**
i. **Fuel line self-sealing - .50 caliber gun fire**
j. **Fuel line breakaway fittings**
k. **Fuel level control valve performance/endurance**
l. **Boost and transfer pump performance/endurance**
m. **Shutoff valve performance/endurance**
n. **Environmental**
o. **Vibration**
p. **Lightning Hazard**

6-1.2.3 Assembly/System Level Tests. Aircraft system level tests shall be conducted in accordance with ADS-1B-PRF.

6-2.0 **REFERENCES**.

6-2.1 Federal Specifications.

VV-F-800	Fuel Oil, Diesel

6-2.2 Military Specifications.

ADS-1B-PRF	Rotorcraft Propulsion Systems Airworthiness Qualification Requirements, Ground and Flight Test Surveys and Demonstrations
ADS-9C	Propulsion System Technical Data
MIL-G-5572	Gasoline, Aviation: Grades 80/87, 100/130, 115/145
MIL-T-5624	Turbine Fuel, Aviation, Grades JP-4, JP-5, and JP-5/JP-8 ST
MIL-T-83133	Turbine Fuels, Aviation, Kerosene Types, NATO F-34 (JP-8) and NATO F-35

6-2.3 American Society of Testing and Materials.

ASTM-D-910	Standard Specification for Aviation Gasoline
ASTM-D-1655	Aviation Turbine Fuels

6-2.4 International Standards.

STANAG 3294	Aircraft Fuel Caps and Fuel Cap Access Covers
STANAG 3212	Diameters for Gravity Filling Orifices
STANAG 3105	Aircraft Pressure Fueling Connections
STANAG 3632	Electrical Safety Connections for Aircraft and Ground Support Equipment

6-3.0 **PERFORMANCE GUIDELINES**.

6-3.1 **Performance characteristics**.

6-3.1.1 Engine feed and transfer subsystem.

6-3.1.1.1 Fuel availability. Fuel should be available to the engines on an uninterrupted basis under all possible design ground and flight conditions for the aircraft. The main fuel feed tanks should have at least 99 percent available fuel at the normal landing attitude and power setting. For fuel transfer tanks, including external tanks, at least 99 percent fuel availability at cruise attitude should be used as a design goal.

6-3.1.1.2 Flow performance. Fuel flow performance of each engine feed system should provide 100% of the maximum fuel consumption of the engine(s) plus any fuel flow required for cooling purposes or motive flow for jet pumps.

6-3.1.1.3 Suction feed. The fuel feed system to the engines and APU (if installed), should operate as a suction system in the normal mode. The engine feed subsystem without boost pumps operating should provide fuel to the engines operating at maximum rated power setting for altitudes up to 20,000 feet with a fuel temperature of at least 135°F.

Rationale. A suction feed system reduces the risk a of post crash fire. Also, a suction feed system reduces the risk of fire in the event of a leak, due to reduced fuel leakage and spraying.

6-3.1.1.4 Priming. The fuel system should provide the capability for priming, as required by the engine(s) and APU with hot fuel up to 135°F. The fuel system should be designed to prevent loss of engine feed system prime in the event of an engine(s) shutdown (intended or unintended).

6-3.1.1.5 Hot restart. The fuel system should provide restart capability, following heat soak, with fuel up to 160°F in the engine compartment fuel line.

Rationale. Due to the temperature rise in the engine compartment immediately after engine shutdown, the fuel trapped in engine compartment fuel lines can significantly increase.

6-3.1.1.6 Pressure capability. The engine feed and transfer subsystems should be capable of withstanding a proof pressure of two times the maximum operating pressure, an ultimate pressure of one and one-half times the proof pressure, and a negative pressure of one atmosphere without air leakage into the system or collapse of any component.

6-3.1.1.7 Surge pressure. Surge pressure in the engine feed and transfer subsystem should not exceed the proof pressure of the system.

6-3.1.1.8 Contaminated fuel. Contaminant particles larger than 2,000 microns should be removed from the fuel by the airframe systems before delivery to the engine.

Rationale. The inlet from the fuel tank into the engine feed system should keep large size contaminants out of the engine feed line, which could damage or jam operating components. Typically, 8 mesh (approximately 2,000 microns) screens are used on boost pump/foot valve inlets.

6-3.1.1.9 Engine feed independence. A single fault in the engine feed subsystem will not result in the loss of more than one engine. It should be possible to cut off fuel flow to any engine or combination of engines without adversely affecting the fuel flow to the remaining engine(s). A separate feed tank and separate feed lines for each engine are desirable.

6-3.1.1.10 Engine cross feed. The engine feed system for each engine should be capable of supplying fuel to any other engine on the aircraft.

Rationale. Provides feed system redundancy allowing all engines to continue to operate with a failure in one of the engine's normal feed system.

6-3.1.1.11 Fuel transfer/management. On aircraft normally operated by a single pilot, the fuel feed and transfer subsystem should provide for automatic transfer of all fuel to the engine without requiring any action by the pilot to control sequencing, fuel center-of-gravity, or operation of pumps. For aircraft operated by a second pilot or flight engineer, a semiautomatic system may be used where the crew members provide control for tank sequencing, pump operation, and cross-feed. The system should not require the immediate attention or action of the pilot or crew member under normal flight conditions. The engine feed and transfer subsystem should provide two independent and isolated methods of moving fuel out of each feed tank on the aircraft, except for jettisonable external tanks where only one method is required. Transfer pump(s) should be shut off automatically when the tank is empty.

Rationale. Two independent and isolated methods to remove fuel from each feed tank minimizes unusable fuel in the event of a feed system failure. Automatic shutoff of transfer pumps eliminates the risk of a fuel pump initiated fire due to dry run operation.

6-3.1.1.12 Fuel center-of-gravity. Fuel management as a result of fuel depletion or loading should not cause the aircraft center-of-gravity to exceed safe operating limits.

6-3.1.1.13 Fuel center-of-gravity warning. A fuel imbalance warning device should be provided if the change of the fuel center-of-gravity can result in the aircraft C.G. exceeding a safe limit.

Rationale. An imbalance warning device will alert the pilot of a potentially unsafe center-of-gravity condition he may not be aware of until performing a maneuver which he may not be able to re-establish control or recover from.

6-3.1.1.14 Low fuel level warning. A low fuel warning device, for each engine feed tank, which will operate independent of the fuel gauging system, or operate after two failures in the gauging system, should be provided to indicate sufficient fuel remaining for 20 minutes of flight at maximum range cruise power and altitude, plus a normal descent and landing with one missed approach.

6-3.1.1.15 Fuel transfer to main tank. The rate of fuel transfer into the engine feed tanks from any fuel tank, internal or external, should be equal to or greater than the intermediate rated power fuel consumption for all engines a feed tank can feed.

6-3.1.1.16 Low fuel pressure indication. Low fuel pressure indication should be provided for each engine feed line and fuel transfer subsystem.

Rationale. Low fuel pressure indication provides the pilot information of engine feed system problems. If the pilot receives a low fuel pressure indication, he can use the cross-feed system or turn the boost pump on (if installed) to avoid a possible engine flameout.

6-3.1.1.17 Fire shutoff capability. Each engine feed line should have a fire shutoff valve. The fire shutoff valve should be mounted as near as possible to the engine compartment, but not in the compartment. The fire shutoff valve should be capable of being reopened from the cockpit. Electrical valves should be of a fail safe design to prevent uncommanded closure in the event of a short circuit. Some installations may require both a tank shutoff and a valve and a fire shutoff valve to minimize the quantity of fuel which can drain into the engine compartment after the valve is closed and to minimize the length of unprotected line which can drain the tank if the line is broken.

Rationale. The pilot must have the capability to stop fuel flow into an engine compartment in the event of an engine compartment fire or if the potential for an engine compartment fire exists.

6-3.1.1.18 Negative and zero "g". The engine feed system(s) should maintain uninterrupted fuel flow to the engine(s) for flight operations/maneuvers specified in the aircraft system specification which impose negative gravity or zero "g" conditions on the aircraft.

6-3.1.1.19 Crashworthiness. The fuel system should contain all fuel during and after all survivable crash impacts. The only fuel spillage allowed, is that associated with the actuation of self-sealing breakaway valves.

Rationale. Containment of fuel after a crash significantly reduces the risk of a post-crash fire.

3.2 **Fuel tank Subsystem**.

6-3.2.1 Location and separation. Fuel tanks should not be located over personnel compartments. Fuel tanks located adjacent to personnel, cargo or engine compartments should be separated from such compartments by a second liquid-tight and vapor-tight barrier in addition to the barrier provided by the tank. Fuel tanks should not be located in personnel or cargo compartments on a permanent basis.

6-3.2.1 R The absence of fuel tanks in the personnel compartments reduces the risk of injury due to fuel vapors and fuel leaks/fires. Also, the risk of personnel being crushed by a fuel tank breaking loose from its restraint system during a crash is eliminated.

6-3.2.2 Crashworthiness. All fuel tanks installed inside of the aircraft should be crashworthy. Each fuel tank configuration in the aircraft should be capable of withstanding, without leakage, a 65 foot free-fall drop, onto a nondeforming surface when filled with water to normal capacity. If desired, the test can be performed with a representative portion of aircraft structure surrounding the tank.

Rationale. The maximum survivable (sustaining serious injuries) impact velocity for humans is approximately 60 ft/sec. The impact velocity of fuel tanks from a 65 foot drop is slightly less than 65 ft/sec. Thus, fuel is contained during a survivable crash, reducing the risk of thermal injury or fatality.

6-3.2.3 Slosh and Vibration. The tank should withstand without damage or failure, a 25 hour simultaneous slosh and vibration and 15 hours additional slosh test, two-thirds full of Type III test fluid.

6-3.2.4 Self-sealing tanks. Metallic fuel lines should not be installed in contact with the walls of a self-sealing tank. Aircraft structure surrounding or supporting self-sealing tanks should be designed and constructed to withstand all flight conditions and forces produced by the passage of a projectile through the confined liquid. The structure surrounding self-sealing tanks should be liquid-tight except for the cavity drain(s). All fuel tanks, installed inside of the aircraft, should be self-sealing against 12.7 mm API projectiles on all surfaces. The tank should also be self-sealing against tumbled 14.5 mm projectiles for a volume equal to 30 minute reserve at VBR and MAGW.

6-3.2.5 Fuel resistance of external surface. Bladder type fuel tanks should be capable of withstanding a 60 day immersion test, in Type III test fluid.

Rationale. The old requirement for bladders to withstand 3 days of immersion in test fluid has proven to be inadequate because some leaks could not be detected within 3 days. All fuel tank manufacturers have developed tanks which can withstand 60 days of immersion in test fluid. Also, a DODIG investigation resulted in a recommendation that the U.S. Army procure fuel tanks with 60 day fuel immersion capability.

6-3.2.6 Tank capacity. The fuel tank(s) should contain enough fuel, after refueling at normal ground attitude, to meet the mission requirements of the aircraft.

6-3.2.7 Fuel expansion space. An expansion space to prevent tank overflow due to thermal expansion of fuel should be provided in each fuel tank. The expansion space
should be equal to or greater than 3 percent of the total fuel volume of the tank with the aircraft at normal ground attitude. For tank arrangements where a number of tanks or cells are connected to function as a single tank the expansion space for the total cluster can be provided in one of the cells.

Rationale. For aircraft which are fueled and left standing in the sun, a 56°F temperature increase can be encountered. For JP-4 fuel, this temperature increase requires approximately 3 percent of the tank volume as ullage.

6-3.2.8 Sump. Each fuel tank should be provided with a sump (which should be the lowest portion of the tank when the aircraft is at ground attitude) for the purpose of collecting sediment and water.

6-3.2.8.1 Sump volume. The sump of each fuel tank should have a capacity of 0.25 percent of tank volume at normal ground attitude. The capacity of the sump should not be less than 1 pint.

Rationale. The sump capacity is based upon 0.75 cubic centimeters (cc) of free water per gallon of fuel in excess of saturation at 80°F at the critical icing temperature.

6-3.2.8.2 Sump drains. Each sump should have a drain valve installed. Operation of the drain valve should remove all of the collected sediment and water in the sump. The fuel tank should be completely drainable through the drain valve, while the aircraft is at ground attitude.

6-3.2.8.3 Fuel sample. It should be possible to obtain a fuel sample from each sump without lying on the ground.

6-3.2.9 Tank pressure safety factors. The fuel tanks should have a proof pressure capability of 1.33 times the maximum operating pressure and a burst pressure capability of 1.5 times the proof pressure.

6-3.2.10 Bladder tank cavity sealing. The structure or cavity surrounding a bladder tank should contain any leakage from the bladder. A drain should be provided to remove any leakage which may occur.

Rationale. Containment of fuel prevents fuel from spreading to areas which may contain potential ignition sources. Also, clean-up of leaked fuel is easier since it is confined to a limited area.

6-3.2.11 Removable tank support. Rigid, internal, removable (i.e. internal auxiliary) fuel tanks should be supported to prevent rupture when subjected to the flight inertia load factors and emergency landing crash load factors specified for the vehicle. Brackets or lugs attached to the tank walls should not be used for supporting the tank.

Rationale. Mounting by lugs or screws in the tank's wall, even though this is may be reinforced, has proven to be the first area of failure in an overload condition. A failure of the lugs will generally pull a section from the tank wall resulting in severe leakage.

6-3.2.12 External tanks. External tanks should be structurally capable of satisfactorily reacting to flight induced aerodynamic and inertia loads. All external tanks should be jettisonable. The operational envelope for external tanks should be derived from the mission profile of the aircraft. All fuel, air, and vent lines should be of the quick disconnect, self-sealing, breakaway type. All electrical lines should be of the quick disconnect, breakaway type. All electrical connections should meet the environmental guidelines in of this ADS. Each external tank should have a tank empty sensor. The tank empty sensor should provide a signal to an advisory light in the cockpit, visible by the pilot and copilot. External tanks should not preclude the use of weapons on any store station not used by an external tank.

Rationale. Jettison capability allows the pilot to quickly remedy an out of center-of-gravity condition due to asymmetric external fuel transfer. Also, if flight conditions permit, the capability to jettison external tanks prior to a crash reduces the risk of a post crash fire, since crashworthy external tanks are not available.

6-3.2.12.1 Ground clearance. Ground clearance should be sufficient to prevent ground contact under any combination of the following static or dynamic ground conditions:

a. One or more flat tires.

b. One or more shock absorbers flat.

c. Pitching and/or rolling caused by variations in anticipated runway/taxiway surface.

6-3.2.12.2 Filler opening markings. The tank capacity and type of fuel should be stenciled adjacent to each filler opening.

6-3.2.13 Internal/main tank filler openings. The tank capacity and type of fuel should be stenciled adjacent to the filler opening.

3.3 Pressurization and Explosion Suppression Subsystem.

6-3.3.1 Pressurization subsystem. The pressurization subsystem should provide an independent primary and secondary means of pressure relief to prevent over-pressurization of the tank as a result of a pressure regulator failure in the full open position. The system should provide for automatic venting to atmosphere whenever the pressurization source is inoperative or shut off. Wherever possible, the fuel tank pressurization system should be independent of the cockpit air conditioning system. If a common line is installed, redundant means should be installed to prevent fumes from entering the cockpit.

6-3.3.2 Temperature of pressurization air. The temperature of pressurization air entering a fuel tank should not exceed 435°F.

Rationale. The auto-ignition temperature for JP-5 and JP-8 fuel is 435°F and JP-4 is 445°F.

6-3.3.3 Explosion suppression subsystem. All fuel tanks and vent systems, and dry bay areas in proximity to fuel lines and tanks of aircraft in which air-fuel vapors may be exposed to ignition sources (such as incendiary gunfire, missile fragments, hot engine fragments, high wall temperatures, etc.) should be so equipped that fires and explosions cannot occur in these areas. When baffle material (reticulated foam) or nitrogen inerting are used, the guidelines of 6-3.3.3.1 and 6-3.3.3.2, respectively should be met.

6-3.3.3.1 Baffle material. The baffle material should not degrade the performance of the fuel system beyond the limits specified in the fuel system detail specification. The fuel system should meet all performance requirements with or without the baffle material installed, excluding the explosion protection provided by the baffle material. It should be possible to remove the baffle material from the tanks and operate the aircraft without removing or adding any other hardware. Any components fastened in the tank because of the baffle material, should be sufficiently tested to verify that these components can be retained if the baffle material is removed. The baffle material should be included in all simulator and ground and flight tests.

6-3.3.3.2 Nitrogen inerting system. Nitrogen inerting should be provided for all main fuel tanks. The nitrogen inerting system should be completely automatic and should require no attention from the flight crew during flight except for the monitoring of associated caution/advisory lights. The system should prevent explosions and fire by diluting and maintaining the oxygen concentration below 9 percent, in the fuel tank(s) ullage space, without the use of ground equipment.

6-3.3.3.2.1 Pressurization. Nitrogen gas should pressurize the ullage and vent spaces during decreases in altitude to maintain a safe differential pressure between the tanks and ambient.

6-3.3.3.2.2 Damage. The nitrogen inerting system should maintain inert ullage and vent spaces with no electrical power applied for a minimum of 5 minutes with a 100 square inch hole in any one fuel tank.

6-3.3.3.2.3 Pressures. At no time should the positive or negative pressures in the fuel tanks and vents exceed the design pressure limits of the aircraft regardless of failure of any component.

6-3.4 **Ground Refueling and Defueling Subsystem**.

6-3.4.1 Refueling time. The refuel subsystem, with a steady state pressure of 55 psig. at the aircraft refueling adapter, should permit refueling the aircraft with 90% of a full load in a flow time not to exceed that in Table I. The flow rate to each tank should be balanced so that all tanks obtain their capacity shut-off point at approximately the same time.

TABLE I. PRESSURE REFUELING TIME

Total Aircraft Fuel Weight	No. of Servicing Adapters	Maximum time for 90% load (Minutes)*
25,000	1	15
10,000	1	10
4,900 or less	1	5

*Times are for flow times only and do not include time for positioning the aircraft or refueling equipment. Times may be interpolated between divisions.

6-3.4.2 Gravity refueling. The aircraft should be capable of gravity refueling. Filler orifice diameter should be 2.95 inches minimum. Filler openings should be located to permit filling the tanks from outside the aircraft without overfilling into the expansion space. The use of special adapters, such as neck extenders or funnels, is not recommended. If special adapters are used, they should be crashworthy and prevent fuel from being trapped outside of the fuel tank after refueling.

Rationale. Filler orifice diameter based on NATO Standardization Agreement (STANAG) 3212.

6-3.4.3 Refueling subsystem pressure capability. The aircraft should be capable of pressure refueling at steady state pressures between 20 and 55 psig., measured at the inlet to the aircraft refueling subsystem. The pressure refueling subsystem should be capable of withstanding a proof pressure of 180 psig. and an ultimate pressure of 270 psig.

6-3.4.4 Maximum surge pressure. During pressure refueling, surge pressures in the aircraft refueling system should not exceed the proof pressure of the system, with a steady state pressure of 55 psig. at maximum flow at the aircraft refueling adapter.

6-3.4.5 Maximum capacity refueling attitude. Maximum capacity refueling should be accomplished in the normal ground attitude of the aircraft with the required mission loads.

6-3.4.6 Pressure refueling adapter.

6-3.4.6.1 Location. The aircraft should be equipped with a readily accessible, single-point, ground-pressure refueling adapter. The ground refueling adapter should be located to permit simultaneous armament loading (as ammunition or cargo, when applicable) and ground refueling. The location of the refueling adapter should also permit safe hot refueling of the aircraft.

6-3.4.6.2 Installation. The refueling adapter should be compatible with pressure fuel servicing nozzles having a straight or 45° inlet. The adapter should be installed with the adapter face as nearly as possible in the vertical plane. The adapter should be located on the aircraft so that the use of ground support elevating devices will not be required for connecting the nozzle. The installation should provide ample clearance for connection and operation of the nozzle by personnel wearing heavy/arctic gloves.

6-3.4.6.3 Trapped fuel. No fuel should be trapped in the refueling system, outside of the fuel tank(s).

Rationale. Fuel trapped outside of the fuel tank(s) poses a fire hazard in the event of the line being hit by gunfire or shrapnel.

6-3.4.7 Filler caps and filler cap access. All gravity and pressure refueling adapter filler caps, as well as any filler cap access covers, should not require any tools for their removal or replacement.

Rationale. This guideline expedites the refueling process and avoids the need for refueling personnel to have/carry a tool to remove/replace the filler cap(s).

6-3.4.8 Refueling power guideline. The aircraft, including external tanks, should be capable of being refueled, by gravity and pressure, to any intermediate quantity or to the aircraft capacity and automatically shut off without external power applied to the aircraft.

Rationale. The use of external power during refueling increases the risk for fire in the event of fuel leakage/spillage. External power will increase refueling time due to connection and disconnection of power harness. Also, increased logistic support is undesirable.

6-3.4.9 Refueling controls location. The refueling controls and fuel quantity gages should be located adjacent to the pressure refueling adapter.

Rationale. This criteria allows one person to refuel the aircraft from the single point servicing adapter.

6-3.4.10 Required refueling personnel. The aircraft should be capable of being refueled or defueled by one person, this does not include the refueling truck operator or safety observer. Refueling operations should not require any personnel to be in the aircraft.

Rationale. This guideline minimizes personnel required to refuel an aircraft. For safety reasons, the number of personnel required to refuel an aircraft should be kept to a minimum. Personnel inside of an aircraft being refueled, are more susceptible to injury or death in the event of fuel spill/fire.

6-3.4.11 Hot refueling. The aircraft should be capable of being safely refueled with all engines operating and the auxiliary power unit operating.

Rationale. The capability to keep all engines and the APU running during hot refueling reduces refueling time and reduces wear and tear on turbine engines.

6-3.4.12 Tank selection. The refueling system controls should have the capability to fill any individual tank or to avoid filling any tank.

Rationale. Under emergency conditions it is beneficial to have the capability to select individual tanks for filling or to exclude tanks from filling to allow flying of an aircraft with a battle damaged or leaking fuel tank.

6-3.4.13 Fuel level control valve pre-check. A pre-check capability should be provided for each fuel level control valve. The pre-check system should permit isolation of a failed level control valve.

Rationale. A pre-check system provides greater assurance the fuel level control valve will work properly. If a level control valve fails, fuel will be spilled on the ground from the tank vent system. Spilled fuel poses a safety problem as well as a clean-up problem.

6-3.4.14 Defueling methods. Defueling should be accomplished by suction from both the pressure refueling adapter and the gravity refueling port(s). Electrical power should not be required for defueling, unless the aircraft pumps are required for defueling.

Rationale. Defueling from the pressure refueling adapter significantly reduces defueling time.

6-3.4.15 Defueling with failures. It should be possible to defuel each tank with any single component failure in the system.

6-3.4.16 Defueling crashed aircraft. In the event of a wheels-up or collapsed skid landing, it should be possible to defuel each tank through the normal fuel servicing adapters or by suction through accessible openings in each tank.

Rationale. In a crash landing the defueling adapter could be damaged preventing defueling. Defueling of crashed aircraft is desired in order to remove fuel to reduce the hazard and to lighten the weight of the aircraft for removal from the crash site.

6-3.4.17 Static discharge in fuel tanks. The refueling system should minimize static electricity discharge within the fuel tanks during refueling by electrical bonding of all fuel components to structure with a bond resistance of 10 megohms or less, or until proven safe based on the components capability not to produce the minimum ignition energy of 0.25 millijoule. Fuel velocities within lines should be limited to no more than 30 ft/sec, with 20 ft/sec or less preferred. Fuel tank entry velocity should be limited to no more than 10 ft/sec. Fuel flow should enter at the tank bottom with induced flow circulation along the tank bottom surface to prevent frothing and foaming.

Rationale. The refueling system must minimize the possibility of sparks within the fuel tank to reduce fire and explosion hazards. Air Force testing has proven the above bond resistance and fuel velocities reduce static build-up to acceptable/safe levels.

6-3.4.18 Refueling nozzle bonding. A bonding receptacle should be provided adjacent to each refueling receptacle/port for bonding the refueling nozzle to the aircraft. The bonding receptacle should be located no more than 41 inches from the gravity filler opening and no closer than 12 inches from fuel tank vent openings or filler cap openings.

Rationale. The dimensions stipulated correspond to the length of the bonding wire/cable on standard ground support equipment refueling nozzles and required nozzle/hand clearance for nozzle operation.

6-3.4.19 Refueling adapter isolation. The ground refueling adapter should be isolated from fuel pressure during all flight modes.

Rationale. Fuel pressure applied to the back side of a damaged ground refueling adapter will result in a fuel leak. The damage to the adapter can go undetected during ground refueling.

6-3.4.20 Engine and engine feed isolation. The engine and engine feed-lines should not be subjected to the pressures imposed on the refueling system during refueling.

Rationale. Surge pressures encountered during refueling, can damage the engine and engine feed system due to exceeding the proof pressure of the engine feed system.

6-3.5 **Aerial Refueling Subsystem**. TBD

6-3.6 **Fuel Vent Subsystem**.

6-3.6.1 Vent pressure. The vent subsystem should prevent each fuel tank pressure from exceeding the proof pressure of the tank during ground and flight conditions, including single fuel system component failure, especially a level control valve during refueling. The vent outlet terminal should be such that the pressure in the vented space of the fuel tanks is never below ambient pressure. For bladder-type tanks, the cell cavity pressure should never be greater than the pressure within the tank.

6-3.6.2 Overboard spillage through vents. The vent subsystem should prevent siphoning and spillage of fuel. Fuel spillage through the vent line during any in-flight maneuver should not exceed 0.05 percent of the fuel from any one tank or 1 gallon, whichever is smaller. No leakage is preferred. Overboard fuel spillage through the vents during all ground operations and conditions is prohibited, except for a high level shutoff valve failure during fuel transfer or refueling. Fuel spillage through the vent system should not occur during or after a rollover or survivable crash.

Rationale. Fuel spillage through the vent system at anytime is a safety hazard and should be eliminated.

6-3.6.3 Vent outlet location. Liquids or vapors emitted from the vent outlets should not blow back onto the aircraft or come into contact with the engine exhaust, aircraft brakes or air inlets, or air pressure sensor ports. The outlet should be located so that any moisture collecting in the line will drain to the outlet and prevent the entrance of water into the fuel system. The vent outlet location and configuration should prevent the formation of ice which will block the outlet.

Rationale. Liquid or vapors emitted from the vent outlets which come in contact with the aircraft increases the risk of fire.

6-3.6.4 Vent line size. The vent line size should be selected by the length of the vent line and the amount of air, vapor, or liquid that must be handled. The volume of air, vapor, or liquid that must be handled will depend on the fuel consumption of the engine, size of the fuel tank, pressure rating of the tank, rate of refueling, type of fuel being used, rate of climb, rate of dive, and service ceiling of the aircraft.

6-3.6.5 Interconnected vents. Fuel should not be transferred through one tank vent to another tank. Several fuel tank vents may be terminated at the same outlet location in the aircraft.

Rationale. Using the tank vent system as a transfer system can cause tank over-pressurization if the vent line is full of fuel and the tank is being refueled or receiving transfer fuel. Also, nitrogen inerting system pressure may not be vented, causing tank over-pressurization.

6-3.7 **Fuel Dump Subsystem**. TBD

6-3.8 **General Criteria**.

6-3.8.1 Materials. The use of magnesium, cadmium, and copper on parts that come in contact with fuel should be avoided.

Rationale. Cadmium and copper breakdown when exposed to fuel causing contamination of the fuel. Magnesium corrodes when exposed to fuel.

6-3.8.2 Metals. Metals should resist corrosion due to fuels, salt spray, atmospheric conditions, or wear likely to be encountered in transportation, storage, or during normal service life.

6-3.8.3 Dissimilar metals. Dissimilar metals should not be used in intimate contact with each other, unless protected against electrolytic corrosion.

6-3.8.4 Protective treatment. When materials are used that are subject to deterioration when exposed to climatic and environmental conditions likely to occur during storage or service usage, they should be protected against such deterioration in a manner that will in no way prevent compliance with the performance guidelines of this ADS.

6-3.8.5 Fuel. The fuel system should be suitable for use with the following fuels: JP-4 and JP-5 in accordance with MIL-T-5624, JP-8 in accordance with MIL-T-83133, Jet A, Jet A-1 and Jet B in accordance with ASTM-D-1655, aviation gasoline in accordance with MIL-G-5572 and ASTM-D-910, and VV-F-800 (DF-A, DF-1 and DF-2). Degradation of performance or any special maintenance activity caused by the use of other than primary fuel(s) should be identified by the contractor.

6-3.8.6 Primary fuel designation. Primary fuel for the aircraft should be JP-8.

Rationale. JP-8 has been designated as the standard NATO fuel.

6-3.8.7 Icing. The fuel system should be designed so that icing will not adversely affect system operation. Safety of flight of the aircraft should not be jeopardized when operating with 0.75 cubic centimeter of free water per gallon of fuel in excess of saturation at 80°F at the critical icing temperature. The fuel system should be capable of continuous operation with fuel initially saturated with water at 80°F and cooled to any temperature down to and including -65°F. The system should perform satisfactorily under these conditions with JET B fuel containing no anti-icing additive and without draining or cleaning the system.

Rationale. The allowable level of water contamination is 10 parts per million (ppm). The fuel system must be capable of normal operation with the specified level of water contamination.

6-3.8.8 Hose and tubing.

6-3.8.8.1 Clearance. Clearance between all fuel system hoses and tubing should be provided to prevent contact with surrounding components, hardware, and structure during all fuel transfer and refueling operations on the ground and in-flight. Fuel lines should be supported so that the lines will not deflect out of position as a result of internal pressure or aircraft maneuvers. Fuel lines which may be subjected to damage due to maintenance, cargo handling, personnel traffic, or normal aircraft use should be protected.

6-3.8.8.2 Hose assemblies. Flexible hose should not be stretched or twisted during installation.

6-3.8.8.3 Metal tubing. Forcing, bending, or stretching of metal tubing to accomplish installation should not be permitted.

6-3.8.8.4 Self-sealing fuel lines. All pressurized fuel lines, with an inside diameter of 5/8 inch or greater, which are exterior to the fuel tank(s) should be self-sealing against 0.50 caliber armor-piercing projectiles and 0.50 caliber ball ammunition. This guideline also applies to suction feed lines on aircraft utilizing a nitrogen inerting system.

6-3.8.8.4 R The smallest, qualified self-sealing fuel line has a 5/8 inch inside diameter. Self-sealing fuel lines minimize fuel spillage/leakage and risk of a fire in the event projectile or shrapnel punctures the line.

6-3.8.8.5 Fuel line fittings. Self-sealing breakaway valves should be used at all external fuel and vent line connections to the fuel tank and at locations where fuel line displacement adequate to cause line, or line to fitting failure, is possible during a crash.

Rationale. Self-sealing breakaway valves prevent fuel spillage/leakage in the event of a crash, thus reducing the risk of a post-crash fire.

6-3.8.9 Fuel drains. All drained/leaked fuel which does not drain to the onboard container (refer to paragraph 2-3.1.2), such as from bladder tank cavities, dry bays, and sheet metal pockets and traps, should be drained to the exterior of the aircraft. No drained/leaked fuel should be allowed to collect on the aircraft, other than in the onboard container.

Rationale. Overboard drainage of fuel, not collected in a container, is required because drained or leaked fuel poses a fire hazard.

6-3.8.10 Electrical bonding. The fuel system tubing and components should be electrically bonded to eliminate static charge accumulation, provide controlled current return paths, and provide lightning protection.

Rationale. Electrical bonding of tubing and components reduces the risk of ignition of fuel vapors from static discharge and lightning.

6-3.8.11 Electrical leads. Component electrical wiring leads should be of sufficient length to connect to the terminals without splicing. Electrical wire splices should be avoided inside of conduit.

Rationale. Fuel components with wire splices which must be pulled through conduit have caused difficulty in installing and removing components. Also, the integrity of a splice inside of conduit is difficult to verify.

6-3.8.12 Thermal relief. Pressure relief for fuel thermal expansion should be provided for all closed plumbing segments.

Rationale. Thermal expansion of fluid in a fixed volume can cause extreme pressures capable of bursting the system.

6-3.8.13 External fuel leakage. There should be no external fuel leakage from the fuel system during normal operating conditions including ground or aerial refueling.

6-3.9 **Components**.

6-3.9.1 Component features.

6-3.9.1.1 Pipe threads. Tapered pipe threads should not be used in the fuel system except for permanent closures.

6-3.9.1.2 Lubrication. Fuel system components should not require added lubrication for the normal life of the component. The use of hermetically sealed bearings containing lubricant is acceptable.

Rationale. Lubricants do not hold up well in fuel and periodic replacement of lubricant is an undesired maintenance action.

6-3.9.1.3 Threaded safety. All threaded parts should be positively locked. The use of lockwashers or staking is prohibited.

Rationale. Threaded parts will loosen or separate unless positive locking is provided. Lockwashers and staking cannot insure locking.

6-3.9.1.4 Electrical fault and explosive atmosphere. All fuel system components should be capable of operating in an explosive atmosphere without initiating an explosion. Component housings should be capable of containing any internal explosion without failure of the housing. No electrical short or internal explosion should propagate to the outside of the component housing or generate an unsafe condition.

6-3.9.1.5 Electrical equipment isolation. Where possible, electrical equipment should be isolated from the fuel to minimize the possibility of fuel leakage and fuel vapor coming into contact with electrical equipment.

6-3.9.1.6 Electromagnetic interference. Fuel system electrical components should meet the electromagnetic interference requirements of the aircraft system specification.

6-3.9.1.7 Electrical insulation. The electrical components should withstand application of test voltages without damage, breakdown or excessive leakage of current. The AC dielectric current should not exceed 2 milliamperes. Insulation resistance should be no less than 100 megohms for motors or 200 megohms for all other electrical components.

6-3.9.1.8 Operation with contaminated fuel. All components should continue to operate with fuel contaminated in accordance with Table II.

TABLE II. CONTAMINANT MIXTURE

Contaminant	Particle Size (Microns)*	Quantity(grms per 1000 liters)
Iron Oxide	0-5 5-10	19 1.0
Sharp Silica Sand	150-300 300-420	0.7 0.7
Prepared dirt conforming to AC Spark Plug Co. Part No. 1543637 (Coarse Arizona road dust)	Mixture as follows: 0-5 (12%) 5-10 (12%) 10-20 (14%) 20-40 (23%) 40-80 (30%) 80-200 (9%)	5.3
Cotton Linters	Staple Below 7 U.S. Dept. of Agri. Grading Stds.	0.07
Iron Chips	150-500	10
Aluminum Chips	150-500	10

*NOTE - The contamination used for testing is graded by the sieve method. Particles considered larger than the 500 microns size can pass through the sieve. Particles in the 700-800 micron range have been found in certified test contamination samples.

Rationale. TABLE II is based on inspection of contamination found in actual aircraft tanks over several years of operation.

6-3.9.2 Fuel level control valve.

6-3.9.2.1 Level control. The pilot control devices for the level control valves should be level sensing and should not be operated by sensing tank pressure or by a wetting action, such as thermistors. The valves should be located so that during all normal ground and in-flight refueling attitudes and fuel transfer attitudes, the fuel level will not exceed the three percent normal expansion space allowance of the fuel tanks.

Rationale. Level sensing control valves provide a more positive and reliable means of sensing fuel

6-3.9.2.2 Valve seat leakage. Leakage past valve seats or pilot valves, inside of fuel tanks, should not exceed 50cc per minute; except for dual pilot valves which should not exceed 100cc per minute.

6-3.9.2.3 Surge pressure/valve closure rate. Level control valve rate of closure should prevent refueling surge pressures from exceeding the proof pressure of the refueling subsystem.

6-3.9.3 Boost and transfer pumps.

6-3.9.3.1 Boost pumps. For high altitude and/or high temperature operations, a boost pump(s) can be installed in the engine feed system(s). The pump should be capable of providing rated performance with hot (135°F) or cold (-65°F) fuel at any operational altitude with a fuel level two inches above the pump inlet. The pump should provide the required flow and pressure in a time interval sufficient to meet the aircraft engine feed or transfer requirements. Pumps installed in fuel tanks of 1,000 gallons or greater capacity should be removable without draining the tank.

6-3.9.3.2 Dry operation. All transfer and boost pumps should be capable of accumulating 100 hundred hours of dry operation, in 5 hour cycles.

Rationale. Dry run capability of fuel pumps reduces the risk of pump failure and fires.

6-3.9.3.3 Reprime. Pumps should reprime within 5 seconds (unless a system performance analysis justifies a longer period), up to the maximum operating altitude of the aircraft, following a period when the inlet is uncovered and then resubmerged in fuel.

Rationale. Special procedures to reprime a pump cannot be tolerated.

6-3.9.4 Shutoff valves.

6-3.9.4.1 Flow control. The engine shutoff valve(s) should not be operated by a switch(es) mounted on the throttle quadrant. Any linkage for a mechanical valve should be subjected to the environmental and endurance tests with the valve unless the linkage is qualified by separate tests. The valve should incorporate a position indicator and manual override capability.

Rationale. A position indicator and manual override feature simplify maintenance and troubleshooting.

6-3.9.4.2 Electrical failure operation. Shutoff valves should not change flow control position as a result of an electrical failure or an electrical short across the leads.

Rationale. A change or stoppage of fuel flow resulting from electrical failure or short will cause an engine flameout.

6-3.10 **Environmental**.

6-3.10.1 Temperatures. The fuel system should operate with fuel temperatures of -65°F to 135°F. The fuel system should operate with ambient temperatures of -65°F to 160°F. The fuel system should not be damaged under non-operating conditions down to an ambient temperature of -80°F.

Rationale. These temperatures represent anticipated temperature ranges aircraft will be operated in during their life cycle.

6-3.10.2 Operating environment. The fuel system should withstand or operate with or after the following conditions:

Environment	Requirement
a. Acceleration	MIL-STD-810E, Method No. 513.4
b. Shock	MIL-STD-810E, Method No. 516.4
c. Humidity	MIL-STD-810E, Method No. 507.3
d. Rain	MIL-STD-810E, Method No. 506.3
e. Sand and dust	MIL-STD-810E, Method No. 510.3
f. Fungus	MIL-STD-810E, Method No. 508.4
g. Salt fog	MIL-STD-810E, Method No. 509.3
h. Vibration	MIL-STD-810E, Method No. 514.4
i. High Temperature	MIL-STD-810E, Method No. 501.3
j. Low Temperature	MIL-STD-810E, Method No. 500.3

6-3.11 **Interface**.

6-3.11.1 Power. All fuel system components should operate on standard electrical, hydraulic, and/or pneumatic power as defined in the aircraft system specification.

Rationale. The use of special converters or transformers to operate fuel system components is undesirable.

6-3.11.1.1 Pneumatic subsystems. Pneumatic subsystems which are used in/interface with the fuel system, for power, pressurization and/or transfer, should eliminate the collection and freezing of water.

Rationale. Collection of water in pneumatic subsystems can cause corrosion of components. Freezing of water can cause components with moving parts to bind or stick and restrict/block air flow.

6-3.11.2 Instrumentation.

6-3.11.2.1 Low fuel pressure. A low fuel pressure indicator for each engine should be provided for each pilot.

6-3.11.2.2 Engine shutoff valve(s). Cockpit indicating lights should be provided for electrically operated engine fuel shutoff valves. These lights should illuminate when the valve is in transition, a fault exists in the valve circuitry, or the valve limit switch fails to operate.

Rationale. Providing the pilot engine shutoff valve status allows the pilot to rapidly troubleshoot problems/failures and take immediate corrective action.

6-3.11.2.3 Fuel quantity gauging. Cockpit fuel quantity instrumentation should indicate the weight of fuel on the aircraft. Accuracy of the fuel quantity indicating system should be ±2 percent of indication and ±0.75 percent of full scale.

6-3.11.2.4 Fuel filter. A fuel filter impending bypass indicator should be provided on each engine filter . Provisions should be provided for cockpit indication of impending bypass for each engine filter.

Rationale. Fuel filter impending bypass provides indication of excessive filter contamination, before the filter goes into bypass. Early indication of excessive contamination allows maintenance crews to determine the cause/source of the contamination, before causing engine damage or flameout.

6-3.12 **Hazards and Failure Concept**.

6-3.12.1 Failure analysis. A complete fuel system failure analysis should be conducted and a report submitted to the Government IAW the applicable contract data items list. The study should not be limited to single failures but should account for multiple failures in critical flight modes and during emergency conditions. A single failure of a component in the fuel system or any other subsystem supplying power to the fuel system should not prevent the completion of the aircraft's primary mission. Two failures in the fuel system should not cause critical structural failure or prevent recovery of the aircraft. High reliability components should be excluded from the single failure criteria and should be identified by the contractor. Components should be identified as the design for a particular aircraft develops.

6-3.12.2 Fire hazard reduction. Fuel and vent lines should be routed so that a single failure in the line or single failure in another subsystem will not cause a fuel fire. Sources of fuel leakage, such as joint couplings, should not be located in the vicinity of materials or conditions that will permit ignition of fuel. Fuel lines should not be routed through any nacelle or powerplant compartment, except the one they feed. Also, fuel lines should not be routed through personnel or cargo compartments; however, in some instances an occupied portion of the aircraft is the only space available for routing of an aerial refueling line or ferry tank fuel line. In this case, the joints in the fuel line should be held to a minimum, and the joints should be shrouded and their compartments drained.

6-3.12.3 Lightning hazard. The fuel system should be safe from fire and explosion hazards caused by direct lightning strikes, cross-field streamering or static electricity within the fuel.

6-3.12.4 Component temperature. The temperature of components inside a fuel tank wall should not exceed 435°F.

Rationale. Refer to paragraph 6-3.3.2.

6-3.13 **Reliability.** The reliability program for the fuel system, fuel subsystems, and components should be integrated with the overall aircraft system reliability program.

6-3.14 **Maintainability**.

6-3.14.1 Maintainability program. The maintainability program for the fuel system, fuel subsystems and components should be integrated with the overall aircraft system maintainability program. All aircraft fuel system components should be accessible for inspection, cleaning, and adjustment or replacement while installed on the aircraft, with tools normally found in a mechanic's tool kit and without removal of the engine, fuel tanks, or important parts of the aircraft structure.

6-3.14.2 Component checkout. Components whose operability cannot be determined during normal operation and service should be provided with the means for periodic assurance of proper functioning. Components operating in unison through a common control switch should be provided with the means for ground testing to insure proper operation of each component.

6-3.14.3 Component Interchangeability. Subsystem operation should not depend on components which require precise adjustment prior to installation. Fuel system components should be interchangeable without adjustment or field calibration. All parts having the same manufacturer's part number should be functionally and dimensionally interchangeable.

Rationale. Field adjustment and/or calibration of components prior to installation increases aircraft downtime and support equipment requirements. Interchangeability of parts having the same manufacturer's part number reduces maintenance and logistic support.

6-3.14.4 Filter element removal. Fuel filters or strainer elements should be replaceable without draining the fuel tanks.

Rationale. Draining fuel tanks significantly increases aircraft downtime.

6-3.14.5 Tank access doors. Each fuel tank should have an access door(s) to permit inspection, cleaning, and repair of the entire interior surface of the tank and access to components.

Rationale. Good access to all interior parts of a fuel tank is desired in order to improve maintainability of fuel tank components.

6-3.14.6 Component construction. Components of modular or "plug-in" construction should be considered for each subsystem to minimize draining fuel, entering fuel tanks, and disrupting line and manifold connections during replacement.

6-3.14.7 Fixed housing. Fixed housings for plug-in components which are intended to last the life of the aircraft should not contain moving parts except those that function as secondary sealing or line shutoff mechanisms.

6-3.14.8 Murphy proof connections. Component connecting points should not permit installation of a component in an improper orientation. Where two hose connections, tube connections, or electrical connectors are in the same area, the design of the connecting points should not permit cross connecting of the components.

6-3.15 **International standards**. Due to drawings/sketches being included in the following NATO Standardization Agreements (STANAG's), they are referenced in this ADS, in lieu of including their content herein. The following international standards should be used in the design of the fuel system:

a. STANAG 3105
b. STANAG 3632

6-4. **QUALIFICATION GUIDELINES**.

6-4.1 **General**.

6-4.1.1 Verification Inspections. The inspections specified herein should verify the ability of the aircraft fuel system and its components to meet the performance as defined in the weapon system specification and/or the component detail specification.

6-4.1.2 Fuel Availability. Refer to ADS-1 for aircraft test requirements.

6-4.1.3 Flow Performance. Fuel flow performance of each engine feed subsystem, including the cross-feed system, should be verified by analyses. Refer to ADS-1 for simulator and aircraft test requirements.

6-4.1.4 Suction Feed. Refer to ADS-1 for test requirements.

6-4.1.5 Priming. The capability of the fuel system to prime the engine(s) and APU should be verified by analyses. Refer to ADS-1 for simulator and aircraft test requirements.

6-4.1.6 Hot Restart. The capability to restart each engine with a fuel temperature of 160°F in the applicable engine compartment fuel line, should be verified by tests on a simulator and the aircraft.

6-4.1.7 Pressure Capability. The proof pressure capability of the engine feed and transfer subsystems should be verified by component test. Ultimate pressure capability should be verified by component tests of every component in the subsystem. Negative pressure capability should be verified by component test. Refer to ADS-1 for installed system tests prior to first flight for proof and negative pressure.

6-4.1.8 Surge Pressure. Surge pressure levels in the engine feed and transfer subsystems should be verified by analyses. As a minimum surge pressures resulting from rapid engine power reduction, high-level shutoff resulting from fuel transfer and closure of shutoff and cross-feed valves should be analyzed. Refer to ADS-1 for aircraft test requirements.

6-4.1.9 Contaminated Fuel. The ability of the fuel system to limit particle size in fuel should be verified by document review and/or component inspection.

6-4.1.10 Engine Feed Independence. Engine feed independence should be verified by analysis. Refer to ADS-1 for aircraft test requirements.

6-4.1.11 Engine Cross Feed. Refer to ADS-1 for aircraft test requirements.

6-4.1.12 Fuel Transfer/Management. Verification of the automatic feed and transfer features should be accomplished by analysis. Refer to ADS-1 for aircraft test requirements.

6-4.1.13 Fuel Center-of-Gravity. Refer to ADS-1 for aircraft test requirements.

6-4.1.14 Fuel Center-of-Gravity Warning. The operation of the fuel center-of-gravity warning system (when required) should be verified by analysis. Refer to ADS-1 for aircraft test requirements.

6-4.1.15 Low Fuel Level Warning. Refer to ADS-1 for aircraft test requirements.

6-4.1.16 Fuel Transfer to Main Tank. The rate of fuel transfer from all tanks transferring fuel to the engine feed tank(s) should be verified by analysis. Refer to ADS-1 for aircraft test requirements.

6-4.1.17 Low Fuel Pressure Indication. Refer to ADS-1 for test requirements.

6-4.1.18 Fire Shutoff Capability. Refer to ADS-1 for test requirements.

6-4.1.19 Negative and Zero "g". Fuel flow during negative or zero "g" conditions should be verified by component bench tests. Refer to ADS-1 for aircraft test requirements.

6-4.1.20 Crashworthiness. The capability of the fuel system to contain fuel during and after survivable crash impacts should be verified by analysis and component bench tests. It is desirable to verify the crashworthiness of an aircraft by actual aircraft crash tests. However, if funds are not available, analysis and component bench testing is acceptable for verification.

6-4.2 **Fuel Tank Subsystem**.

6-4.2.1 Location and Separation. Refer to ADS-1 for verification requirements.

6-4.2.2 Crashworthiness. The capability of each fuel tank configuration to withstand the 65 foot free-fall drop test should be verified by component test. The production tank with all openings suitably closed should be filled to normal capacity with water and the air removed. The fuel tank should be placed upon a platform and raised to a height of 65 feet. A light cord may be used to support the tank in its proper attitude. Tanks installed in aircraft structure should be raised to a height of 65 feet; no platform should be used. The platform/structure should be released and allowed to drop freely onto a non-deforming surface so that the tank/structure should impact in a horizontal position ±10°. After the drop, there should be no leakage.

6-4.2.3 Slosh and Vibration. The slosh and vibration capability of the fuel tank should be verified by component test. The production tank, including all hardware (except baffle material, if applicable) to be installed inside the tank, should be installed in an actual section of the aircraft structure or structure simulating the aircraft structure. The entire fuel cell compartment should be lined with brown paper. The tank should be slosh and vibration tested for 25 hours with the tank two-thirds full of Type III test fluid containing a staining agent at a temperature of 110°F. During the entire test, the tank should be subjected to maximum stabilized vapor pressure encountered in any prescribed stabilized level flight. The tank should be mounted to simulate pitching in the actual aircraft. Special fixtures, such as baffles, should also be tested if applicable by mounting the aircraft structure on the rocker in another position for a portion of the test time. There should be no evidence of leakage or failure of the fuel tank or the attachment of its components during this test. After the 25 hour simultaneous slosh and vibration test, a 15 hour slosh test should be conducted. Following these tests, a pressure test should be conducted on the tank. For the pressure test, test pressure should be 1.5 times the normal head measured at the bottom of the tank.

6-4.2.4 Self-Sealing Tanks. All but the ballistic/gunfire capability of fuel tanks should be verified by document review and inspection of the aircraft. The ballistic/gunfire capability of the fuel tank should be verified by component test. For the gunfire test, the tank should be mounted in an actual section of aircraft structure containing any backing board that will be used in the aircraft. The test tank need not include accessories for this test. The tank should be filled two-thirds full of type I fluid. All shots should be below the test fluid level. The tank internal pressure throughout the test should be maintained at the maximum stabilized vapor pressure encountered during any prescribed stabilized level flight conditions. The tank should be subjected to the gunfire test in accordance with its capacity. The number of 12.7 mm AP rounds to be fired should be determined on the basis of one round for each 15 gallons of tank capacity up to a maximum of 10 rounds. Four of the rounds should be 14.5 mm, of which two rounds should be 3/4 to fully tumbled and two should be unyawed, all to be fired at 0 degrees obliquity to the cell surface. A damp seal should occur within two minutes.

6-4.2.5 Fuel Resistance of External surface. The fuel tank's external surface, fuel resistance capability should be verified by component test. A two foot test cube of identical construction to that of the actual aircraft tank(s) should be placed in a container sufficiently large enough to permit immersion of the bottom half of the cube. One fitting in the cube should be completely immersed during the entire test. The cube should be filled with and immersed in type III test fluid for a period of 60 days at ambient temperature. The cube should then be drained, removed from the immersion container and examined. Both the internal and external surfaces of the cube should show no swelling, separation, blistering, or dissolution. For self-sealing tanks, there should be no evidence of activation of the sealant material.

6-4.2.6 Tank Capacity. Refer to ADS-1 for test requirements.

6-4.2.7 Fuel Expansion Space. Refer to ADS-1 for test requirements.

6-4.2.8 Sump. The capability of each fuel tank to collect sediment and water should be verified by component document review.

6-4.2.8.1 Sump Volume. Refer to ADS-1 for test requirements.

6-4.2.8.2 Sump Drains. Refer to ADS-1 for test requirements.

6-4.2.8.3 Fuel Sample. Refer to ADS-1 for test requirement. The capability to obtain a fuel sample from each sump without lying on the ground should be verified by aircraft ground test.

6-4.2.9 Tank Pressure Safety Factors. Refer to ADS-1 for test requirements.

6-4.2.10 Bladder Tank Cavity Sealing. Refer to ADS-1 for test requirements.

6-4.2.11 Removable tank support. Removable tank support capability should be verified by component tests on the tank.

6-4.2.12 External tanks. The structural capability of the external tank should be verified by static tests. Verification of fuel, air, vent, and electrical disconnect and breakaway capabilities should be verified by component tests. Verification of the electrical connectors to meet environmental requirements should be verified by component tests. Refer to ADS-1 for aircraft test requirements.

6-4.2.12.1 Ground clearance. Refer to ADS-1 for test requirements.

6-4.2.12.2 Filler Opening Markings. Refer to ADS-1 for verification requirement.

6-4.2.13 Internal/Main Tank Filler Openings. Refer to ADS-1 for verification requirement.

6-4.3 **Pressurization and Explosion Suppression Subsystem**.

6-4.3.1 Pressurization. Refer to ADS-1 for test requirements.

6-4.3.2 Temperature of pressurization air. The maximum temperature of pressurization air entering a fuel tank should be verified by analysis and component test. Refer to ADS-1 for aircraft test requirements.

6-4.3.3 Explosion Suppression Subsystem.

6-4.3.3.1 Baffle Material. The baffle material should not be installed in the fuel tank during the slosh and vibration test. Refer to ADS-1 for aircraft test requirements.

6-4.3.3.2 Nitrogen Inerting System. Refer to ADS-1 for test requirements.

6-4.3.3.2.1 Pressurization. Refer to ADS-1 for test requirements.

6-4.3.3.2.2 Damage. The capability of the nitrogen inerting system to maintain inert ullage and vent spaces with no electrical power applied for a minimum of 5 minutes with a 100 square inch hole in any one fuel tank should be verified by analysis and component tests.

6-4.3.3.2.3 Pressures. Refer to ADS-1 for test requirements. Verification that the positive and negative pressures in the fuel tanks do not exceed the design pressure limits of the aircraft, regardless of failure of any component should be accomplished by simulator and flight tests.

6-4.4 **Ground Refueling and Defueling Subsystem**.

6-4.4.1 Refueling time. Refer to ADS-1 for test requirements.

6-4.4.2 Gravity refueling. Verification that filler orifices have a minimum diameter, compatible with standard current inventory refueling nozzles, should be by component drawing review. Verification that the location of filler openings permit refueling from outside the aircraft without overfilling into the tank(s) expansion space should be accomplished by analysis; refer to ADS-1 for aircraft test requirements. Verification that special adapters, if installed, are crashworthy and prevent trapping of fuel outside of the fuel tank should be by analysis and document review; refer to ADS-1 for aircraft test requirements. Refer to ADS-1 for additional aircraft test requirements.

6-4.4.3 Refueling subsystem pressure capability. The ultimate pressure capability of the refueling subsystem should be verified by component tests. Refer to ADS-1 for aircraft test requirements.

6-4.4.4 Maximum Surge Pressure. Refer to ADS-1 for test requirements.

6-4.4.5 Maximum Capacity Refueling Attitude. Refer to ADS-1 for test requirements.

6-4.4.6 Pressure Refueling Adapter.

6-4.4.6.1 Location. Refer to ADS-1 for test requirements.

6-4.4.6.2 Installation. Refer to ADS-1 for test requirements.

6-4.4.6.3 Trapped Fuel. Refer to ADS-1 for test requirements.

6-4.4.7 Filler Caps and Filler Cap Access. Refer to ADS-1 for test requirements.

6-4.4.8 Refueling Power Guideline. Refer to ADS-1 for test requirements.

6-4.4.9 Refueling Controls Location. Refer to ADS-1 for verification requirements.

6-4.4.10 Required Refueling Personnel. Refer to ADS-1 for verification requirement.

6-4.4.11 Hot Refueling. Refer to ADS-1 for test requirement.

6-4.4.12 Tank Selection. Refer to ADS-1 for test requirements.

6-4.4.13 Fuel Level Control Valve Precheck. Refer to ADS-1 for test requirements.

6-4.4.14 Defueling Methods. Refer to ADS-1 for test requirements.

6-4.4.15 Defueling with Failures. Refer to ADS-1 for verification requirement.

6-4.4.16 Defueling Crashed Aircraft. Refer to ADS-1 for verification requirement.

6-4.4.17 Static Discharge in Fuel Tanks. The freedom of sparks within fuel tanks during refueling should be verified by document review and component test. Refer to ADS-1 for aircraft test/verification requirements.

6-4.4.18 Refueling Nozzle Bonding. Refer to ADS-1 for verification requirement.

6-4.4.19 Refueling Adapter Isolation. Refer to ADS-1 for verification requirements.

6-4.4.20 Engine and Engine Feed Isolation. Refer to ADS-1 for verification requirements.

6-4.5 **Aerial Refueling Subsystem**. TBD

6-4.6 **Fuel Vent Subsystem**.

6-4.6.1 Vent Pressure. Refer to ADS-1 for test requirements.

6-4.6.2 Overboard Spillage through Vents. The capability of the vent subsystem to prevent fuel spillage during or after a rollover or survivable crash should be verified by component test. Refer to ADS-1 for aircraft test/verification requirements.

6-4.6.3 Vent Outlet Location. Refer to ADS-1 for verification/test requirements.

6-4.6.4 Vent Line Size. Refer to ADS-1 for verification requirement.

6-4.6.5 Interconnected Vents. Refer to ADS-1 for verification requirements.

6-4.7 **Fuel Dump Subsystem**. TBD

6-4.8 **General Guidelines**.

6-4.8.1 Materials. Verification that magnesium and copper parts do not come in contact with fuel should be by review of component drawings.

6-4.8.2 Metals. The capability of metals to resist corrosion due to fuels, salt spray, atmospheric conditions, or wear should be verified by component test.

6-4.8.3 Dissimilar Metals. Verification that dissimilar metals are not in intimate contact with each other should be by document review.

6-4.8.4 Protective Treatment. The capability of protective treatments to prevent deterioration of materials should be verified by component tests. Refer to ADS-1 for simulator and aircraft test requirements.

6-4.8.5 Fuel. Verification of the fuel system suitability with the specified fuels should be by component test.

6-4.8.6 Primary Fuel Designation. Refer to ADS-1 for test requirements.

6-4.8.7 Icing. Operation of the fuel system with the specified fuel conditions should be verified by component tests. During the tests the accumulation of ice should not obstruct moving parts, plug orifices or bleed holes, or block screens or filters. Refer to ADS-1 for simulator test requirement.

6-4.8.8 Hose and Tubing.

6-4.8.8.1 Clearance. Refer to ADS-1 for verification requirements.

6-4.8.8.2 Hose Assemblies. Refer to ADS-1 for verification requirements.

6-4.8.8.3 Metal Tubing. Refer to ADS-1 for verification requirements.

6-4.8.8.4 Self-Sealing Fuel Lines. Verification of all pressurized fuel lines outside of the fuel tank(s) being self-sealing against 0.50 caliber projectiles should be by document review and component test. The gunfire test should be conducted by filling four hose assemblies with JP-8 and two hose assemblies with JP-4 for seven days at 75°F±5°. After soaking, two of four JP-8 hose assemblies should be subjected to -20°F and two to -40°F one each at 40 psig. internal pressure and a negative pressure of 12 inches of water, the two JP-4 hose assemblies should be subjected to -50°F, one at 40 psig. internal pressure and one at a negative pressure of 12 inches of water. After fluid temperature stabilization, each unit should be subjected to two impacts of 0.50 caliber armor piercing ammunition and one impact of ball ammunition. All rounds should be fired from a distance of 50 to 60 yards. The projectiles should take effect at least 2 inches apart. Each hole should dry seal within two minutes.

6-4.8.8.5 Fuel Line Fittings. The capability of self-sealing breakaway valves to prevent fuel spillage/leakage from the fuel system in the event of a crash or hard landing should be verified by component test. Refer to ADS-1 for aircraft verification requirements.

6-4.8.9 Fuel Drains. Refer to ADS-1 for test requirements.

6-4.8.10 Electrical Bonding. Refer to ADS-1 for verification requirements.

6-4.8.11 Electrical Leads. Refer to ADS-1 for verification requirement.

6-4.8.12 Thermal Relief. The incorporation of adequate thermal relief provisions should be verified by component testing. Refer to ADS-1 for aircraft verification requirements.

6-4.8.13 External Fuel Leakage. Refer to ADS-1 for verification and test requirements.

6-4.9 **Components**.

6-4.9.1 Component Features.

6-4.9.1.1 Pipe Threads. The use of tapered pipe threads only for permanent closures should be verified by document review and component inspection.

6-4.9.1.2 Lubrication. Freedom from lubrication of components should be verified by document review and component qualification.

6-4.9.1.3 Threaded Safety. The presence of positive locking for threaded connections should be verified by inspection and by component tests.

6-4.9.1.4 Electrical Fault and Explosive Atmosphere. The capability of components to operate in an explosive atmosphere and contain an internal explosion should be verified by component tests.

6-4.9.1.5 Electrical Equipment Isolation. Refer to ADS-1 for verification requirement.

6-4.9.1.6 Electromagnetic Interference. Component electromagnetic interference compatibility should be verified by component tests. Refer to ADS-1 for aircraft test requirements.

6-4.9.1.7 Electrical Insulation. Component electrical insulation should be verified by component test.

6-4.9.1.8 Operation with Contaminated Fuel. The capability of fuel system components to operate with contaminated fuel in accordance with TABLE II should be verified by component test.

6-4.9.2 Fuel Level Control Valve.

6-4.9.2.1 Level Control. Verification that pilot control devices are level sensing should be by document review. The capability of the level control valve(s) to operate over the aircraft's range of flow, pressure, and attitude should be verified by component tests.

6-4.9.2.2 Valve Seat Leakage. Level control valve leakage rate should be verified by component tests.

6-4.9.2.3 Surge Pressure/Valve Closure Rate. The level control valve rate of closure and surge control should be verified by component test. Refer to ADS-1 for simulator aircraft test requirements.

6-4.9.3 Boost and Transfer Pumps.

6-4.9.3.1 Boost Pumps. Boost pump performance under the specified conditions should be verified by component tests.

6-4.9.3.2 Dry Operation. Pump assembly dry operation should be verified by component test.

6-4.9.3.3 Reprime. The reprime capability of each pump should be verified by component test.

6-4.9.4 Shutoff Valves.

6-4.9.4.1 Flow Control. The capability to meet environmental and endurance tests for any mechanical valve linkage should be by component tests. Verification of a position indicator and manual override should be component inspection. Refer to ADS-1 for aircraft verification requirement.

6-4.9.4.2 Electrical Failure Operation. Verification that a shutoff valve does not change flow control position as a result of an electrical failure or short across the leads should be by component tests.

6-4.10 **Environmental**.

6-4.10.1 Temperature. The capability of the fuel system to operate under the fuel and ambient temperature extremes should be verified by analysis and component tests. Refer to ADS-1 for aircraft simulator test requirements.

6-4.10.2 Operating Environment. The effects of the specified environmental conditions on the operation of the fuel system should be verified by components tests.

6-4.11 **Interface**.

6-4.11.1 Power. Operation of fuel system components at the specified power should be verified by component tests.

6-4.11.1.1 Pneumatic Subsystems. The capability of pneumatic subsystems, which are used in, or interface with the fuel system, to eliminate the collection and freezing of water should be verified by component tests. Refer to ADS-1 for simulator test requirements.

6-4.11.2 Instrumentation.

6-4.11.2.1 Low Fuel Pressure. Verification of proper operation of the low fuel pressure indicator should be by component test. Refer to ADS-1 for simulator and aircraft test requirements.

6-4.11.2.2 Engine Shutoff Valve. Refer to ADS-1 for test requirements.

6-4.11.2.3 Fuel Quantity Gauging. Verification of the accuracy of the indicator(s) should be by analysis and component test. Refer to ADS-1 for aircraft test requirements.

6-4.11.2.4 Fuel Filter. Refer to ADS-1 for verification requirement.

6-4.12 **Hazards and Failure Concept**.

6-4.12.1 Failure Analysis. Refer to ADS-1 for verification requirements.

6-4.12.2 Fire Hazard Reduction. Refer to ADS-1 for verification requirements.

6-4.12.3 Lightning Hazard. The capability of the fuel system to avoid fire and explosion hazards caused by direct lightning strikes, crossfield streamering, or static electricity within the fuel, should be verified by component test. Refer to ADS-1 for further verification requirements.

6-4.12.4 Component Temperature. Verification that the maximum temperature of components and tank walls does not exceed the flash point of the fuel, should be by component test. Refer to ADS-1 for aircraft verification requirements.

6-4.13 **Reliability**. Refer to ADS-1 for verification requirements.

6-4.14 **Maintainability**.

6-4.14.1 Maintainability Program. Refer to ADS-1 for verification requirements.

6-4.14.2 Component Checkout. Refer to ADS-1 for verification requirements.

6-4.14.3 Component Interchangeability. Interchangeability of components without precise adjustment or calibration should be verified by component tests and analyses. Refer to ADS-1 for aircraft test requirements.

6-4.14.4 Filter element removal. Refer to ADS-1 for verification requirement.

6-4.14.5 Tank access doors. Refer to ADS-1 for verification requirement.

6-4.14.6 Component construction. Verification that the use of modular or "plug-in" components were considered should be by document review.

6-4.14.7 Fixed housing. The absence of moving parts on fixed housings should be verified by component inspection.

6-4.14.8 Murphy proof connections. The freedom from possible improper installation of components should be verified by inspection of component drawings. Refer to ADS-1 for aircraft verification requirement.

6-4.15 **International standards**. Incorporation of the required international standards should be verified by component drawing inspection. Refer to ADS-1 for aircraft verification requirement.

SECTION 7: ENVIRONMENTAL CONTROL SYSTEM

7-1.0 REQUIREMENTS.

7-1.1 PERFORMANCE. The Environmental Control system shall meet its allocated performance and function in such a manner that the aircraft shall be able to be operated safely throughout the operating envelope and meet the performance requirements as defined in the applicable weapon system specification.

7-1.2 QUALIFICATION. The following qualification requirements for the Environmental Control system are required to verify compliance with the performance requirements above as applicable to the aircraft design configuration.

7-1.2.1 Analysis. System design and performance analysis shall be documented, using ADS-9C and the criteria below as a guide. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

7-1.2.2 Component Tests. The following tests shall be conducted and a subsequent teardown inspection, to determine the post test condition, shall be performed. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

a. **Pneumatic actuated component dust and moisture, hot and cold**
b. **Air Cycle Machine**
    1. **Performance**
    2. **Endurance**
    3. **Vibration**
    4. **Mechanical shock**
    5. **Environmental**
c. **Vapor Cycle Machine**
    1. **Performance**
    2. **Endurance**
    3. **Vibration**
    4. **Mechanical shock**
    5. **Environmental**
d. **Controls/Software functionality**
e. **Heat Exchanger**
    1. **Proof and burst pressure**
    2. **Vibration and mechanical shock**
    3. **Pressure drop**
    4. **Thermal efficiency**
    5. **Thermal bypass**
    6. **Leakage**
f. **Couplings/Ducting**
    1. **Temperature**
    2. **Proof and burst pressure**
    3. **Vibration**
    4. **Leakage**

5. **Flame resistance**

g. **Fan Containment**

7-1.2.3 Assembly/System Level Tests. Assembly level tests on the following subsystems shall be conducted to demonstrate component and system performance, integrity, and endurance. Aircraft system level tests shall be conducted in accordance with ADS-1B-PRF.

a. **Bleed air subsystem**

b. **Pressurization subsystem**

c. **Heating and Cooling subsystem**

d. **Nuclear/Biological/Chemical subsystem**

e. **Microclimatic subsystem**

7-2.0 REFERENCES/DEFINITIONS.

7-2.1 Commercial Specifications.

SAE ARP 669	Color Coding Of Terminals/Wiring For Flight Equipment

7-2.2 Military Specifications.

ADS-1B-PRF	Rotorcraft Propulsion System Airworthiness Qualification Requirements
ADS-9C	Propulsion System Technical Data
MIL-E-87145	Environmental Control, Airborne
MIL-STD-1568	Materials And Processes For Corrosion Prevention And Control In Aerospace Weapons Systems
MIL-STD-1587	Materials And Process Requirements For Air Force Weapons Systems

7-2.3 Engineering Tests. These tests are restricted to single components, parts of components, or coupon testing to determine suitability of materials, functionality, and preliminary performance of individual components.

7-2.4 Component Qualification Test. This is actually a series of tests conducted on production hardware such as a heat exchanger, valve, Pressure Swing Adsorber (PSA), air cycle machine, etc. These tests include but are not necessarily limited to performance, endurance, proof pressure, burst pressure, Electro-Magnetic Interference (EMI), Sand, Dust, Salt Spray, etc. Some tests may be waived in favor of qualification by analysis or similarity to other components.

7-2.5 System Qualification. All components comprising the ECS system are assembled in the laboratory and are arranged in the aircraft’s configuration. The system is heavily instrumented to capture the performance parameters of selected components and end product performance parameters. ECS end product performance parameters are: cooling, heating, ventilation, pressurization, and air filtration. The testing demonstrates performance, endurance, controllability, and safety for all flight and ground operation. The use of production hardware is maximized, but it is not necessarily required. However, production-like system functionality is required to be demonstrated and tested. Also, any production software controlling the system must be used in the test.

7-2.6 Subassembly Manufacturing Acceptance Testing. These tests are conducted during manufacturing and assembly. These may include applying simulated signals to valves and observing functionality, pressurizing a run of ducting and checking for leaks, checking the cockpit pressurization, checking for proper inputs and outputs from the ECS controller, checking the functionality of solenoids, etc. They are characterized by testing parts of the system - not the whole. They are conducted at strategic points in time during the assembly to assure quality of manufacturing assembly at the subassembly level.

7-2.7 System Acceptance Testing. This test is conducted at the completion of the system assembly on the aircraft. It must include either real or simulated interfaces from the other relevant systems such as the fuel system, propulsion system, electrical system, avionics system, cockpit interfaces such as warnings, cautions, advisories, and the control panel. This test demonstrates system integrity, performance, maintenance ability, and proper interface characteristics with all relevant subsystems. Power supply for the ECS for this test must come from the aircraft power supply (all relevant sources separately as designed) and from ground power separately to demonstrate full functionality. Therefore, the system acceptance testing will be conducted two times at a minimum (i.e. once with aircraft engines or APU, and once with a low pressure ground cooling cart or with a high pressure bleed air cart).

7-2.8 System Trade Studies. Trade studies determine the optimum ECS for a specific application. Component performance test data may not be available at the time trade studies are conducted, so either predicted or assumed data is used to conduct the trade. The dominating measure of merit for system selection is the relative difference of aircraft Take Off Gross Weight (TOGW) between compared systems, but other factors such as cost, installability, safety, reliability, maintainability, survivability, vulnerability, and program risks may be vitally important in the selection process.

7-2.9 System Analysis.

7-2.9.1 Performance. This analysis is a model of the ECS that provides performance predictions for all modes of operation including cooling, heating, ventilation, humidity control, pressurization, defog, defrost, and NBC protection. It uses detailed component performance characteristics, has a detailed breakdown and accountability of the avionic, electrical, solar, aerodynamic, equipment, personnel heating and cooling loads, and other requirements. This analysis is used as the basis for determining if the system is meeting its performance requirements for those cases where the system laboratory or aircraft flight tests can not achieve the design day boundary conditions. Therefore, it must be able to accurately predict system performance. It is validated by comparing the predicted results with test data and revising the model to account for these differences. System performance is predicted with this model prior to CDR and model validation. Revisions occur periodically or continually throughout the development cycle, and the final report is submitted after the flight test program concludes and the model has been validated by the contractor. Both reports list all assumptions and contains all data used in the analysis model.

7-2.9.2 Schematics. The schematic is an essential part of the analysis. It shows all line routings (i.e. air, vapor, and liquid cooling lines as appropriate), and all planned components for the ECS. In addition, all sensors which provide feedback to the ECS controller, the controller's subsequent output to the actuators, discrete signals going to and from the ECS, and the ECS cockpit controls are displayed. Sensors not providing feedback to the ECS controller, but nonetheless providing control information to various ECS actuators or other subsystems, is also shown.

7-2.9.3 Description. The system description is also an integral part of the analysis. Various ECS functions and control set points are fully discussed and related to the system schematic presentation.

7-2.10 Engineering Analyses These analyses are necessary to solve a narrow spectrum of design problems for a part or component and to substantiate the particular design or method, but do not of themselves play a role in showing that higher level system performance requirements are achieved. For this reason they are regarded as necessary to complete or validate a particular design but fall short of necessity for government review unless specifically requested.

7-3.0 **PERFORMANCE GUIDELINES.**

7-3.1 **Hot and Cold Day Design Criteria.** The ECS should be sized and designed to meet its requirements during the following extremes - except as modified by the system specification.

7-3.1.1 Performance Envelope.

7-3.1.1.1 Sea Level Using a sea level psychometric chart, the sea level performance envelope is defined to be:

a. A series of straight lines connecting the points, (1) 125°F dry bulb, 0% RH, (2) 125°F dry bulb, 5% RH, (3) 95°F dry bulb, 75% RH, (4) saturation temperature for a specific humidity of .02733 gr. H_2O/lb dry air. Next, continue the envelope along the curved line of saturation temperature down to (5) a saturation temperature of -50°F. Complete the envelope with a series of straight lines connecting points (5), (6) -50°F, 0% RH, and (1).

b. For the extreme hot dry bulb temperatures (i.e. points (1), (2), (3), and (4)), a solar load of 360 BTU/Hr-Ft^2 should be imposed. For the extreme cold dry bulb temperature (i.e. point (6)), no solar load should be imposed and no credit for avionics/electrical loads should be taken unless it is shown that these loads are continuously providing power output (i.e. providing continuous heat to the cockpit) for every conceived configuration of the aircraft.

7-3.1.1.2 Altitude. Using a psychometric chart equivalent to the service ceiling of the aircraft, a performance envelope should be constructed as in paragraph 7-2.2.2.1.1 but using a standard temperature lapse rate of 2°C/1000 ft. for altitude correction for the hot day (i.e. point (1) of paragraph 7-2.2.2.1.1). No lapse rate should be used for the cold temperature. For the extreme hot dry bulb temperatures (i.e. points (1), (2), (3), and (4)), a solar load of 435 BTU/Hr-Ft^2 should be imposed. For the extreme cold dry bulb temperature (i.e. point (6)), no solar load should be imposed and no credit for avionics/electrical loads should be taken unless it is shown that these loads are continuously providing power output on every conceived configuration of the aircraft. The ECS should meet its heating, cooling, ventilation, and Nuclear, Biological, and Chemical (NBC) requirements while the aircraft is operating at any point within the volumetric envelope defined by paragraphs 7-2.2.2.1.1 and 7-2.2.2.1.2. This volumetric envelope is intended to describe the conditions under which the ECS should meet all its performance requirements as determined from this ADS or from the Weapon System Specification as applicable and all Contractor established flow-down requirements such as reliability, maintainability, weight, etc. without add on kits. Add on kits are permitted for any flight outside this region or as indicated in the Weapon System Specification.

7-3.2 **Global System Criteria.** The ECS should be designed to meet its performance requirements as stated in the System Specification, as dictated by the Contractor in Contractor established "flow down requirements i.e. maintainability, reliability, weight, repairability, etc.", and as dictated by the cooling, heating, ventilating, and pressurizing needs of the aircraft, personnel, and equipment. These requirements should be met for the service life of the aircraft. Furthermore, the aircraft mission(s) should be analyzed by the designer in order to determine and design the ECS to its expected percent usage in various environments. This process will ensure that the components of the ECS and the system as a whole has been optimally designed from a reliability, weight, and performance point of view. The ECS should be designed to complete the mission with any single failure (i.e. no mission critical failures), and it should be designed such that any safety critical failure can be mitigated automatically or by human intervention. The system should be designed to a "single failure" concept (i.e. any failure within a given component should not result in failure of another component). The system should be designed for no scheduled maintenance. If non-regenerable filters are necessary for functions like NBC or to maintain cleanliness in pneumatic lines, etc., all filters should be readily accessible by maintenance personnel.

Rationale. The primary purpose is to provide a system which allows the weapon system to complete its mission, assure safety of flight, and to minimize costs associated with failures.

7-3.3 **Bleed Air System.** If used, the bleed air system provides high pressure, high temperature air to the ECS. This air is the working fluid of the ECS, and it will be cooled to an extent that it is used to heat, cool, and pressurize the cockpit and avionics as required, and to provide a source of pneumatic pressure for other aircraft functions. Design criteria as outlined in Aerospace Recommended Practice ARP 669 should be adhered to as required for the bleed air system and any ECS ducting within the aircraft which uses air equaling or exceeding the pressures and or temperatures of the bleed air ducting. In particular, but not exclusionary to other criteria of ARP 669, the following apply:

7-3.3.1 General Design Criteria.

7-3.3.1.1 Shutoff Capability. The bleed air system should employ a bleed air shutoff feature which is selectable by any crewmember. The intent is to provide the crewmember(s) a means of shutting off the bleed air. This feature may be implemented as a valve which is normally designed to fail in the closed position. However, the specific failure position (open or closed) should be dictated by a safety hazard analysis of the bleed air system design. In addition, if bleed air temperatures exceed the Autogenous Ignition Temperature (AIT) of fluids which by any means may come in contact with leaking bleed air, or if structural damage could occur, or if other subsystems could be adversely affected (e.g. burnt wiring), a bleed air leak detection system should be used on portions of the ducting where leakage could cause damage. A system safety hazard analysis should be conducted to determine the appropriate course of action upon detection of a bleed air leak. Appropriate action will either be to automatically shutoff the bleed air source and provide indication to the crew, or to provide indication to the crew and allow the crew to close the shutoff device manually from the ECS control panel or displays. A second means to shutting off the bleed air into the cockpit should also be provided. It is permissible to shutdown the normal mode of ECS cooling to meet this requirement.

Rationale. Total flow shutoff capability prevents the introduction of smoke, fumes, toxic gases, etc. into the occupied compartments, when the source of the contaminants is the environmental control system and also, allows the system to be shut off when a failure (i.e. full hot) has occurred. Dual means for shutting off hot air sources are necessary to prevent a single failure from resulting in the inability to shut off hot air to occupied compartments. On cargo, bomber, and other large aircraft, means for manual operation of certain valves can significantly reduce mission aborts as faulty valves can be manually overridden.

7-3.3.1.2 Overbleed Protection. Protection against overbleeding of bleed sources should be incorporated.

Rationale. Overbleeding can be detrimental to the bleed air source, and result in a hazardous condition. So, the intent of this criteria is to assure that any failure of the environmental control system will not result in overbleeding of the bleed source(s). Frequently the bleed air source may contain its own flow limiting device in which case such a device (such as a flow orifice) can be deleted in the bleed air system. The flow limiting device must satisfy maximum system demand which must not exceed the maximum allowable flow from the source(s).

7-3.3.1.3 Bleed Source Isolation. An isolation valve(s) capable of segregating the bleed air sources when multiple sources are used should be employed.

Rationale. Incorporating isolation and crossover shutoff valves permits alternate bleed source utilization for multiple subsystem, multiple engine aircraft and can eliminate the need for a mission abort in the event of a bleed supply duct failure.

7-3.3.1.4 Reverse Flow Protection. Provisions should be incorporated to automatically prevent reverse flow from one bleed source into another source (e.g. the use of check valves). These provisions mitigate the potential for engine performance degradation or damage and to assure proper air flow control in the event of a failure of a single bleed source.

Rationale. Many problems have resulted on past aircraft due to failure of check valves in the engine bleed air system. Careful attention to their design is required, especially the flapper design. There have been instances where the flappers have become disengaged and then lodged in downstream components causing them to malfunction. Serious incidents have occurred from shutoff valves being held in the open position by failed check valve parts. Such an incident resulted in an engine starter going into an uncontained turbine wheel failure with a part of the wheel striking and killing a B-58 crew member during takeoff. In some cases, it is possible for check valve failures to go undetected for several flights if they do not cause problems with other components. As a result, means should be incorporated to allow for periodic inspection of check valves. Borescope provisions have been used for this purpose. Dual check valves have been used to provide the capability that a single check valve failure will not cause engine malfunction in applications where both low and high stage bleed are used.

7-3.3.1.5 Pressure Regulation. The pressure of the engine bleed air should be regulated to that pressure level determined as the minimum necessary for all using subsystems to meet operational performance requirements.

Rationale. This assures that any damage experienced by the aircraft as a result of a bleed system failure will be minimized, and it allows for lower system cost and higher reliability by providing a less severe component design and operating environment. In addition, pressure regulation aids in eliminating problems due to pressure surge. An immediate advantage to pressure regulation is that it lessens the hazard potential and the potential for secondary damage in the event of a bleed system failure. Lower air pressure also allows for the utilization of lower cost components, less rigorous testing, and simplified system design. Pressure regulation is often accomplished in stages. The first stage of pressure regulation is the regulation of the bleed air system pressure. Further reduction in pressure can sometimes be accomplished for other subsystems. The regulator schedule for the bleed air system must be set to accommodate the highest pressure and flow rate demands for the aircraft plus distribution duct losses. To illustrate, the highest pressures required for subsystem operation may be for an ejector designed to utilize 65 psig. The ducting to the ejector may create a 5 psi pressure drop at the design flow rate. Therefore, the first stage regulator should be scheduled for 70 psig. Pressure regulation should occur as near to the bleed source as possible.

7-3.3.1.6 Duct Surface Temperature. Ducts having normal maximum surface temperatures in excess of 750°F should be isolated from exposure to flammable fluids or should be treated as an ignition source. Ducts located in the vicinity of combustible fluids and having normal maximum surface temperature in the range of 500°F to 750°F should either be insulated sufficiently to result in an outer insulation surface temperature less than 500°F, or the following measures should be taken to avoid a fire hazard:

a. Provide an airflow velocity of at least 2 feet/second in the areas immediately surrounding the external duct surface.

b. Prevent direct impingement of combustible fluids on the ducts.

Rationale. When bleed air system components are located near combustible fluids, maximum surface temperatures must be controlled as a fire protection measure.

7-3.3.1.7 Pressure Relief. Aircraft compartments containing bleed air ducting should employ provisions to protect against detrimental compartment over-pressurization in the event of duct failure. Provisions should be incorporated to protect ducting and other components from detrimental positive and negative pressure following single failures.

Rationale. Pressure relief provisions are required to protect the aircraft and subsystem components from damage in the event of bleed air system failure.

7-3.3.1.8 Distribution Shutoff. An independent means should be provided for shutting off air flow to each subsystem which uses bleed air. These shutoff provisions should be controllable from the aircraft crew station.

Rationale. Subsystem shutoff provisions are required to enable the flight crew to eliminate airflow to a failed subsystem without deactivating other subsystems. Distribution shutoff valves can also be used for leakage isolation.

7-3.3.2 Materials and Part Design Criteria. Materials should be consistent with the application. Titanium or stainless steel is used in most bleed air applications. Whatever material is used an analysis should be conducted to define the stress on the parts. This stress analysis should consider (1) pressure, temperature, and aircraft induced loads at the normal operating condition(s), (2) loads from unsupported components such as valves, heat exchangers or other equipment, (3)loads due to a worst case failure of a component of the ECS which might increase the stress on the part (e.g. a support bracket failure for a component mounted to ducting, or a failure resulting in increased temperature or pressure or both), and (4) loads due to the specified vibration environment. Additionally, parts should be designed for installation and maintenance loads as sometimes these loads may exceed the normal operational loads. For example a maintenance worker might repeatedly use a section of ducting for leverage (because of its convenience or location) or to hold oneself up to conduct a maintenance action. Materials should resist corrosion or aging and should remain serviceable for the aircraft's life when used in the intended environment.

7-3.3.3 Couplings. Couplings chosen should be consistent with the loads experienced at the respective junctions. These couplings should employ a quick disconnect type latch that has a safety feature which maintains joint integrity in the event of tee bolt (or other primary latching feature) failure. The quick disconnect feature should be used where frequent removal and replacement is expected due to ECS or other subsystem maintenance activity. Couplings that are frequently removed should be readily accessible and should have the capability to be reused without leakage. If joints require seals, the design should require them to be replaced after having been subjected to thermal and or mechanical loading, otherwise it should be demonstrated that they be reusable without causing increased leakage. The acceptable bleed air leakage from any one joint should be established by the Contractor, and the leakage should be demonstrated to be below or equal to the established limit.

7-3.3.4 Expansion/Slip/Ball/Rotary Joints. The bleed air system should be designed to mitigate repeated thermal expansion and contraction loads and aircraft torsional and bending loads by using suitable compensation devices. It should be designed to withstand pressure cyclic loads, flight induced loads, and vibratory loads. Fixed and sliding brackets and expansion joints should be used as required to allow freedom of movement. Support brackets for bleed air ducting should be used as required.

7-3.3.5 Duct Design Criteria.

7-3.3.5.1 Flow. Bleed air ducting size should limit the airflow to a mach number of .25 in order to reduce the effects of noise, internal resonance, duct fatigue, and pressure drop.

7-3.3.5.2 Temperature. The duct should be insulated as required, or sufficient ventilation should be provided to keep ducts and other bleed air component's surface temperature(s) below the AIT and below any other critical temperature identified by the Contractor which can cause damage to adjacent structure, components, or systems, or otherwise create a safety hazard to the aircraft crew or ground maintenance crew.

7-3.3.5.3 Strength. Ducts should be structurally adequate to resist internal pressures of at least 2.5 (burst pressure) and 1.5 (proof pressure) times the most critical combination of operating pressure and temperature without rupture and without permanent deformation respectively. Furthermore, if column action, torsional, or bending loads exist in the ducting, the duct's strength should be adequate to adsorb these and the internal pressure loads simultaneously. The most critical operating condition is that combination of operating pressure and temperature which results in the lowest yield strength of the duct material. The Contractor should defined it by considering the system under the worst case normal and worst case single ECS component failure operation and selecting the worst case condition from these two choices. Additionally, the contractor should identify ducts which are susceptible to excessive handling and maintenance loads and either (1) design them to withstand these loads or (2) clearly and appropriately mark them to warn maintenance workers not to use excessive force on these items. No permanent deformation is permitted when designing to identified installation or maintenance loads.

7-3.3.5.4 Leakage. Individual duct sections should permit no leakage. Leakage at duct unions is permitted, subject to a system safety hazard analysis, and should be itemized and quantified by the Contractor, but the design of the bleed air system should not permit the total leakage or any specified joint leakage as determined by the Contractor to be exceeded for the service life of the aircraft. This criteria includes any portion of the ECS (e.g. bleed air system, refrigeration package, or distribution system) where leakage presents either a safety hazard or a performance impact on the ECS.

7-3.4 **Pressurization.**

7-3.4.1 Cockpit Pressurization Level or Schedule. The Contractor should establish the level of pressurization of the cockpit and identify other portions of the aircraft structure requiring pressurization and identify the level of pressurization for those areas and provide a design which is able to achieve them. The contractor should also establish the permitted cockpit leakage criteria in order that the required pressurization may be achieved. The cockpit structure should be designed to not exceed this leakage criteria throughout the service life of the aircraft. If pressurization is intended to provide a means of NBC protection, the level of pressurization should prevent ambient air infiltration into the pressurized areas of the aircraft for any flight condition of any mission. Cockpit pressurization should be maintained at a constant level or at the prescribed schedule as a function of altitude (or other variable as deemed appropriate by the specific design) whenever the ECS is operating in the normal mode (i.e. not a failure condition or a backup cooling condition).

Rationale. Pressure schedules are specified to minimize the discomfort (due to pressure changes) to the crew and passengers, and to prevent hypoxia. The pressurization system must be capable of quickly reacting to changes in flight conditions, air conditioning flow rates, etc., in order to maintain the pressure schedule and to avoid annoying or painful pressure bumps. This function can only be accomplished by a fast acting automatic system. Aside from physiological requirements pressurization may be employed as a means to prevent infiltration of outside air. In this instance the pressure may be as low as .5 psid. Even so, problems associated with cockpit leakage which prevents the achievement of the required pressure, and the possibility of over-pressurizing still exists. Therefore, to assure that minimum safe pressure levels can always be maintained, it is necessary that permissible leakage rates be established and demonstrated, and it is necessary to ensure that over-pressurization due to the failure of a component or controller can be prevented.

7-3.4.2 Pressure Release. The pressurization system should have both normal and emergency provisions for release of pressure. The normal pressure release provisions should be capable of dumping pressure without shutting off the pressurizing air source. The emergency pressure release provisions should be capable of dumping pressure from maximum differential to CTBD within CTBD seconds with the pressurizing air source shut off automatically at initiation of dump. Provisions should be incorporated which assure that the pressure differential following landing or following normal pressure release in-flight does not exceed CTBD with the pressurizing air source operating.

Rationale. Provisions for pressure release are required in order to be able to quickly purge smoke, fumes, toxic gases, etc. from the occupied compartments, to allow in-flight air drop of cargo, to allow the use of ram air for emergency ventilation, and to permit emergency escape. To simplify the system design, this provision can be an additional function of the over-pressure release provisions. The normal pressure relief provisions are intended for use in clearing the occupied compartments of smoke, fumes, etc. when the ECS or other pressurization source is not the source of the problem. This permits the most rapid clearing of the compartments. The emergency provisions are intended for use when the pressurization air or ECS is the smoke or fume source, when total ECS failure occurs and ram air ventilation is required, or an emergency condition exists which requires emergency escape. Small residual differential pressures, across large surface areas, can result in large forces which can injure personnel and damage the aircraft when hatches, doors, etc. are released under those conditions. The contractor should evaluate his design and the aircraft mission to determine if and where provisions are necessary for relieving possible residual pressures before opening any hatches, canopies, doors, etc. This requirement must be met with the ECS operating because under normal operating procedures it is not desired to turn off the ECS before operating canopies, hatches, etc.

7-3.4.3 Aircraft Fuselage/Equipment Bay Pressurization. Equipment bay and or aircraft fuselage pressurization may be required to prevent infiltration of outside air. The contractor should establish the required level of pressurization and the amount of airflow required to achieve it. This ADS-50 does not encourage the post use of pressurization air for cooling purposes, but if the air used to pressurize these aircraft locations is collected, routed, and used for cooling subsequent to its use for pressurization, the contractor must demonstrate the structural integrity. The pressurization system should be designed with increased leakage rates in mind. Actual airflow cooling requirements should exceed the design airflow cooling requirements by a factor of 1.6.

7-3.4.4 Positive and Negative Pressure Relief. Provisions should be incorporated to prevent structural damage due to positive or negative pressure in any area of the aircraft.

Rationale. The need for this requirement is proportional to the level of pressurization, and it is necessary for crew safety. In cases where the actual pressurization is small (e.g. .5 psid or lower), it may be very difficult to generate higher pressure differentials even if there is a component or system failure. In this instance it would not be logical to include additional components to meet this requirement. But, if significant pressure forces can be generated, the need for the requirement is also significant. Each system needs to be examined for this possibility.

7-3.4.5 Pressure Source. The pressurization source should provide a minimum flow rate to the pressurized compartment(s) for all flight conditions of at least CTBD times greater than the maximum allowable production leakage rate. The pressurization source should meet the allowable contamination levels of CTBD. If stored gas is used for pressurization, the partial pressure of oxygen should be CTBD. No single failure of a supply or control component should result in CTBD. Means should be incorporated for sealing all pressurization supply inlet openings into the occupied compartments to prevent rapid loss of compartment pressure in the event of pressure source failure. For multiple pressure sources, the pressure schedule should be maintained with one source inoperative.

Rationale. The purpose of this requirement is:

a. To assure that the pressurization system will perform satisfactorily throughout the operational life of the aircraft, and to be sure that the extra capacity is built in that is required to overcome the additional leakage that occurs as aircraft age, undergo modifications, etc.

b. To assure that the air supply meets minimum requirements for health and safety reasons.

c. Stored gas systems are rarely encountered in manned aircraft. However, they may occur in special applications, and provisions must be made to assure that adequate oxygen levels are maintained.

d. To assure the safety of the crew and passengers under failure conditions, and

e. More than one pressurization source is desirable on multi-engine aircraft so that single engine failure will not result in loss of pressurization.

7-3.5 **Refrigeration Package.**

7-3.5.1 Instruments and Controls. The ECS should have a control panel(s) or equivalent which is(are) accessible by the pilot and co-pilot or other crew member. The panel(s) should allow all ECS functions to be controlled and all warnings, cautions, and advisories to be displayed. These include but are not limited to:

(1) bleed air control (on/off) for each engine or APU/SPU providing bleed air,
(2) Bleed air temperature out of the engine compartment is too high
(3) ECS off and on,

(4) ram air selection (or other appropriate backup cooling scheme),
(5) all warnings, cautions, and advisories (e.g. low or inadequate cooling to avionics, bleed air leaks, etc.)6
(7) cockpit temperature control,
(8) cockpit pressure dump as required,
(9) air flowrate as required,
(10) individual zone temperature control as required,
(11) NBC system on/off as required,
(12) defog selection as required,
(13) windshield anti-ice as required,
(14) safety critical functions as required,
(15) mission critical functions as required.

Rationale. The intent of these features is to ensure that all relevant functions of the ECS are controllable by the crew. This ensures that safety measures can be taken, such as turning off bleed air, turning off the ECS, selecting backup cooling, providing ventilation for smoke removal, etc., that normal functioning and control of the system can be accomplished, and that a means is provided to allow the mission to be completed in case of an ECS failure. The contractor should identify all warnings, cautions, and advisories relevant to the ECS, and the appropriate indication should be provided to the crew, via the instrument panel or appropriate displays. The need to notify the crew that the maximum bleed air temperature out of the engine compartment has been exceeded assures that any damage experienced by the aircraft as a result of a bleed system failure will be minimized. This criteria also allows for lower system cost and higher reliability by providing a less severe component design environment. The crew should be alerted to a failure of the bleed air temperature control provisions since a safety hazard could exist. Furthermore, the maximum temperature of bleed air being ducted from the engine compartment should be limited to the autogenous ignition temperature of combustible fluids which, through a fluid system failure could come in contact with the bleed air system. By this criteria, fuels, oils, and hydraulic fluids usually limit bleed air temperatures to 450°F outside of the engine compartment.

7-3.5.2 Test Provisions. The ECS should be testable during ground operation and maintenance checkout. Testability should include but not necessarily be limited to the following:

(1) All compartments which are pressurized to a prescribed schedule should have a means to manually close the automatic pressure regulating device in order that the structure may be pressurized and leak checked. Additionally, a means should be provided (e.g. warning, caution, advisory (WCA) indication on the control component, WCA via cockpit displays, inhibiting the pressure source from functioning, etc.) to prevent leaving the control device in the closed position (i.e. causing the structure to be pressurized without relief).

(2) If the aircraft's fuselage is used as a pressure vessel to trap and route air for avionics cooling, the structure's capability to contain the air, or an equivalent means to show that the equipment is receiving the appropriate air flow at the appropriate temperature should be testable during ground operation.

(3) With the ECS operating on the ground, sensed information from all ECS sensors which provide feedback to the ECS controller for control or diagnostic purposes, should be accessible by maintenance personnel on a continual basis (e.g. time history). If the information accessed is accomplished digitally, the minimum sampling rate should be consistent with the normal operating frequency response of the system and the particular variable involved. Discrete signals used by or sent by the ECS should also be accessible. (The particular means to download this information to maintenance personnel should be determined by the aircraft maintainability requirements. The project engineer should ensure that these testability criteria and the maintainability requirements of the system specification do not conflict.)

7-3.5.3 Cooling Occupied Compartments. Air-conditioning should be provided for occupied compartments in accordance with the following criteria:

7-3.5.3.1 In-flight and Ground Cooling (Flight Suit Environment). The air-conditioning system should be capable of cooling the pilot's and other crew member's envelope to the following levels under the environmental conditions specified in the Weapon System Specification:

Dry Bulb Temperature	Flight Conditions
70°F	All flight conditions of over 30 min. duration
80°F	All flight conditions of 10-30 min. duration
90°F	Flight conditions of less than 10 min. duration and ground operation

Rationale. The intent of cooling is to provide a climate for the crew conducive to mistake free work. Providing a thermally neutral environment helps to achieve this goal by not distracting the pilot or crew.

7-3.5.3.2 In-flight and Ground Cooling/Heating with NBC Gear (Microclimate Cooling/Heating). A means for individually cooling/heating each crew member should be provided. The design should permit individual temperature control, and the amount of cooling provided to each individual should be equivalent to a minimum of 300 watts. Furthermore, the cooling should be applied to a minimum of 15% of the body area including the head and torso, and it should prevent condensation of water on the inside of the transparent surfaces of the face shield. The provided maximum cooling capacity to the NBC ensemble should be available within 60 seconds after the ECS starts. In any case, the provided cooling/heating must prevent condensation from obscuring the crew member’s vision at all times the gear is being worn. Cockpit heating should be possible while cooling and or heating is provided via the Microclimate system. The level of heating of the cockpit should not be colder than 50°F.

Rationale. NBC (MOPP Gear) will trap the crew member’s body heat and induce thermal stress in a short amount of time. Therefore, it is necessary to provide individual cooling to each crew member wearing this gear. It is necessary to have the ability to provide cockpit heating while the NBC gear is being worn so the crew can maintain finger dexterity, have foot heating capability, and operate controls effectively.

7-3.5.3.3 Transient Cool Down and Heat Up (Cockpit). The cockpit effective temperature (i.e. Pilot Envelope Temperature, Wet Bulb Globe Temperature, or other description of the performance requirement as established in the system specification), should achieve 98% of the steady state performance requirement within 25% of the shortest mission profile time - not to exceed 30 minutes.

Rationale. The idea is to design a system which is able to achieve its performance requirements on the front end of the mission. Missions for fighter or attack aircraft will last for approximately two hours. The maximum heating and cooling capability of a system should be achievable on the front end of the mission rather than at the end.

7-3.5.3.4 Cockpit Air Distribution. The conditioned air entering the cockpit should have multiple outlets such that each crew member can adjust the flow-rate and direction of the air across his own envelope. There should be at least 4 outlets per person for cooling - two forward facing gaspers and two along side each crew member, and all should have the capability of directional control. Foot outlets for each crew member should be provided for heating.

Rationale. Proper air distribution is essential for adequate heating and cooling. Distribution requirements should be specified to prevent excessive temperature differences, to limit the velocity of air moving past crew members and passengers, and to prevent the airflow from being fixed directly onto crew members bodies in a distracting or irritating manner.

a. Pilot Envelope Temperature Approach: For proper cooling (which can be interpreted as adequate pilot comfort), proper flow distribution is essential. Air Force experience has shown that pilots are generally most satisfied when the cooling air is blown across their envelope, by controllable outlets that they can adjust for both direction and flow rate according to their personal preference.

b. Average Compartment Temperature Approach: In large occupied compartments, as in passenger or cargo aircraft, the distribution system should be designed to provide adequate mixing of the air to prevent excessive temperature differences. This is generally accomplished by an overboard distribution duct, along the center or both sides of the compartment. The air is exhausted through the sidewalls, near the floor. In addition to the overall approach to the distribution system, other factors must be considered:

1. Temperature Variation. The air should be uniformly distributed to prevent excessive temperature differences. Temperature variation between any two points in the pilots envelope (as defined in 7-3.2.2.1.1) should not deviate more than ± ( )°F from the "Pilot Envelope Temperature." Allowable variations of ±5°F to ±10°F are typical, reasonable requirements which minimize the temperature variations across the crew members. When using the average compartment temperature, the temperature variation between any two points in the envelope occupied by seated personnel should not deviate more than ± ( )°F from the average compartment temperature. Under certain circumstances, further details may be appropriate, such as excluding the area within 6 inches of the floor because of low floor temperatures and the resulting influence on temperatures near the floor. It may be also be appropriate to specify maximum temperature difference for areas outside the envelope of seated personnel, where movement is possible. Values of ±100°F to +-15°F are typical for the envelope immediately occupied by personnel.

2. Foot Outlets. Foot outlets or foot warmers are necessary for proper heating (and cooling) of the crewmember’s feet in many crew positions. This is particularly true of pilot and copilot positions in most aircraft, aerial refueling operator's stations, etc. The feet are very sensitive to the cold (see Appendix C, TABLE I-1) and proper attention to foot outlets is essential.

3. Gaspers. Cold air outlets or "gaspers" are typically provided at crew stations and passenger positions. These are flow and directional controllable outlets, which provide cool air (usually 35°F) and can be adjusted to suit the individual occupying the position. Gaspers can be required for crew positions, passenger positions, galleys, lavatories, etc., where individual temperature and flow control is desired.

4. Miscellaneous. The conditioned air, defog air, or anti-fog air should be distributed to prevent the crew's exhalation from coming in contact with transparencies (to prevent fogging or frosting). This guideline, of course, does not apply to non-mission essential windows. The air flow should not be fixed directly into the crewman's eyes or onto the crewman's arms or shoulders. Defog or anti-fog air should not impinge on the crew. The impact of hot avionics around the crew must also be considered in designing the distribution system.

7-3.5.3.5 Air Velocity (Crew and Passengers). The air velocity should be limited to prevent the blowing of maps and papers inside the cockpit, and to avoid annoying or fatiguing the crew and passengers.

7-3.5.3.6 Temperature Control. Each occupied compartment should have automatic temperature control with a means to override it. The air from any outlet which can impinge on the crew or other wise can cause damage should be limited to a temperature which cannot cause injury, cause physical damage to a component, or otherwise result in condition where the crew member cannot touch a required control. (This apples for normal and single failure conditions.) Other types of source heat such as windshield strip heaters, or radiant heaters should be automatically controlled, but also have a manual override capability.

Rationale. Controls must be incorporated into the system to be able to select and maintain the desired environment. In manned aircraft, readily accessible controls must be included to permit the occupants to control the degree of heating or cooling. To prevent personnel injury or a hazardous condition, provisions are necessary for preventing excessively hot air from entering occupied compartments, even with any single failure condition.

7-3.5.3.7 <u>Surface Temperatures.</u> Areas normally contacted by personnel and cargo should be maintained at levels consistent with human comfort and the protection of cargo as required for-all steady-state flight conditions. The temperature of all surfaces which enter into radiant heat exchange with personnel should be maintained at levels which do not adversely affect human comfort. No surface which can be touched by personnel, should impart pain due the heat exchange.

<u>Rationale.</u> Maximum surface temperatures are specified to prevent injury to personnel, and to control or minimize radiant heat exchange with occupants. Radiant heat exchange is a significant factor in personnel comfort. Personnel can feel uncomfortably warm or cold in spite of proper air conditioning when in radiant heat exchange with hot or cold surfaces. This is particularly a concern in cockpits and flight decks where the crew is surrounded by warm, and often hot, instruments, control panels, etc.

7-3.5.3.8 <u>Normal Ventilation.</u> Fresh air ventilation for contaminant and odor removal should be provided to occupied compartments during all flight and ground conditions. Galley and toilet areas should be vented with fresh air and provided with direct overboard exhaust outlets sufficient to eliminate odors.

<u>Rationale.</u> Fresh air ventilation is required to provide the necessary oxygen content to prevent high CO_2 concentrations to remove moisture and to remove objectionable odors.

7-3.5.3.9 <u>Emergency Ventilation.</u> Emergency fresh air ventilation should be provided to occupied compartments during periods when the normal air conditioning/ventilation system has failed or is shutoff due to contamination of the normal ventilation air supply or temperature control failure. Means should be incorporated to remove smoke, or gases originating in the occupied compartments as a result of fire or smoke.

<u>Rationale.</u> Emergency ventilation provisions are usually required for all aircraft to provide ventilation when the normal method from the environmental control system has failed or is contaminated, or other emergency conditions exist. Also, rapid means for smoke removal from the cockpit and cargo compartments are required so as to provide the crew visibility and prevent nausea or asphyxiation.

7-3.5.3.10 <u>Suit Ventilation/Pressurization.</u> If a ventilation or anti-g suit is used, the air provided to it should be thermally consistent with human comfort and provided at conditions (i.e. flow-rate, pressure, and temperature) consistent with its mission function. Normally, the air source should be the ECS since that air has been conditioned for temperature, humidity, and contaminate removal.

<u>Rationale.</u> Anti-G suits, pressure suits, ventilation suits, etc. may be required to meet the mission requirements of the vehicle. The environmental control system must supply air at the pressure, flow, temperature, moisture, and contamination levels compatible with the respective suits.

7-3.5.3.11 <u>Contamination.</u> No component of the ECS should emit toxins into the ECS airstream or otherwise be a contributor to contaminants entering the cockpit. Other sources of contaminants such as engine oil in the bleed air ducting, gunfire exhaust infiltrating the cockpit, engine exhaust, combustor heater exhaust, fuel fumes, etc., should be eliminated.

<u>Rationale.</u> Contamination levels must be limited in order to assure the efficiency and safety of the crew, and gaseous concentration must be prevented because of the explosion hazard.

7-3.5.3.12 <u>Cargo Overboard Venting.</u> This should be provided as required.

<u>Rationale.</u> Special overboard venting provisions may be necessary for cargo compartments, other compartments, or for the troop or passenger configuration aircraft, so that vapors and fumes from volatile and hazardous cargo may be vented directly overboard. If no passenger carrying requirement exists for the aircraft, then the ventilation requirement must be established based upon the cargo requirements. An airflow of 15-20 cubic feet per minute per 300 cubic feet of cargo volume would be a typical requirement. The air flow rate also depends on the rate of

expansion of the material being carried in containers. Vents should be the same size as those on previous aircraft. The ECS vents are part of aircraft structure. To prevent a possible explosion or fire, it is desirable to have more than one vent on board a cargo aircraft. These vents will ensure that oxygen and other possible reactive materials can be vented separately.

7-3.5.3.13 Nuclear/Biological/Chemical (NBC) System. This system should be provided as required, and it should include a means to determine whether or not NBC contaminants are present and if so at what levels. The design of this system should not be a cause for reduced reliability of the other ECS components either in the normal operating mode or in the backup mode. Since outside, contaminated air can traverse the engine or SPU, ECS bleed air components, aircycle machine, and distribution ducting rapidly, it is assumed that this equipment should be operational full time in order to provide the necessary protection. However, characteristics of unique designs may negate this perceived necessity. If the NBC device is used for additional purposes other than NBC protection (i.e. water vapor removal), all functions and required performance should be achieved simultaneously. Furthermore, this equipment should be designed such that its failure should not result in a failure of another component.

Rationale. This equipment is needed for filtering ECS air of deadly or incapacitating agents.

7-3.5.3.14 Moisture Control. Means should be incorporated to prevent moisture contamination of the avionics equipment by cooling air.

Rationale. In order to prevent avionics failures due to free water, it is necessary to include features prohibiting the introduction of moisture into the avionics equipment by the cooling air.

7-3.5.3.15 Dust Control. Means should be incorporated to prevent dust contamination of the avionics equipment by cooling air.

Rationale. In order to prevent avionics failures due to dust accumulation within the equipment, it is necessary to include this criteria.

7-3.5.3.16 Equipment Temperature Control. The temperature of all avionics equipment and equipment compartments should be maintained within the temperature range as required by the allocated reliability requirements established by the Contractor or the Government. Furthermore, with respect to transient operation, the Contractor should demonstrate that a cold soak start up (i.e. warming up avionics and other equipment as required) and a heat soak start up (i.e. cooling avionics and other equipment as required) can be accomplished as the system specification requires.

Rationale. Adequate temperatures must be provided for avionics equipment so that internal components will always operate within the temperature range used for predicting reliability of the avionics. In order to achieve desired reliability, it is necessary to require that proper temperature control be maintained for all ground and flight operating conditions and ambient extremes of temperature and humidity. Controls must be incorporated which automatically maintain the desired avionics environment. In order to prevent adverse effects due to low temperature, it is necessary to limit both minimum inlet cooling air temperature and maximum flow rate to avionics equipment. Some equipment may be temperature limited and will not function adequately until temperatures are within a certain range. The Contractor must identify this sort of equipment and under these circumstances it must demonstrated that the ECS provides a climate which allows the equipment to function within the prescribed time as set forth in the system specification.

7-3.5.3.17 Equipment Growth. The air conditioning system should have sufficient capacity to allow an increase of 25% heat load growth of the avionics.

Rationale. The purpose of growth capacity is to provide for those avionics equipment changes and additions which normally occur during the service life of an aircraft.

7-3.5.3.18 Avionics Ground Operation. Ground operation of the avionics should be automatically prevented unless coolant is provided.

Rationale. Avionics failure or degraded reliability can result from operating the avionics on the ground during maintenance without providing proper cooling. Since ground operation can account for 50 percent or more of the avionics total operating time, a significant reduction in reliability can result from operating the avionics on the ground without cooling. Therefore, provisions should be incorporated to automatically prevent ground operation of avionics unless proper cooling is provided.

7-3.5.3.19 Avionics Cooling Distribution. Coolant distribution should not be adversely affected by the removal of avionics from their mounting racks.

Rationale. During ground maintenance, individual avionics units may be removed from their mounting rack. In order to not degrade reliability of other operating avionics, it is necessary that proper coolant flow be maintained to all remaining avionics when one or more units are removed from their mounting rack.

7-3.5.3.20 Emergency Cooling. In the event of failure of the normal mode of avionics cooling, an alternate cooling mode should be provided. This backup cooling should allow for the mission to be completed.

Rationale. An auxiliary means for cooling mission essential or "safe return" to base avionics should be provided for use when the normal air conditioning system is inoperative. This will prevent mission aborts or flight safety hazards due to environmental control system failure.

7-3.5.3.21 Fog and Frost Protection Fog and frost protection should be provided for all critical viewing areas for all steady-state ground and flight operating conditions. Critical viewing areas should also be maintained free of fog and frost during maximum rate descent from the operational ceiling of the aircraft. Preheating of the critical viewing areas by increasing occupied compartment air temperature above normal limits prior to descent should not be required in order to maintain fog free surfaces. The normal fog and frost protection provisions should not be disabled by closure of an air conditioning package shutoff valve.

Rationale. Fog and frost protection is required to assure adequate crew visibility. Fog and frost protection may also be required for other transparent areas such as sensor windows. Preheating of compartments in order to accomplish the fog and frost protection function should be prohibited for two reasons:

a. It unnecessarily heats the cabin.

b. It adversely affects the transient capability. If a sudden dive is required due to loss of cabin pressurization or combat maneuvers, the crew will not have the time to preheat the cabin.

Fog and frost protection provisions should not be affected by closing an air conditioning shutoff valve. In-flight failure of an air cycle machine or other component may necessitate turning off the air conditioning system. However, fog and frost protection is still required in order to permit the safe completion of the mission or return of the aircraft.

7-3.5.3.2 Rain Removal. Rain removal should be provided for the critical viewing areas, and it should maintain the visibility required through critical viewing areas for mission completion under the precipitation rate of paragraph 3.2.5 of MIL-E-87145. The system should maintain the required visibility during operation in rain for both ground and flight conditions. The rain removal system should not be damaged by flight at the maximum aircraft speed, and a single failure should not result in failure of the rain removal system for two windshields.

Rationale. Viewing area rain removal is required to insure adequate visibility for crew, camera, sensors, etc.

7-3.5.3.23 Transparency Cleaning. A cleaning system should be provided for the aircraft transparencies.

Rationale. Aircraft with low altitude missions, vertical takeoff and landing capability, and similar missions may require a transparency cleaning capability to assure adequate crew visibility. Camera windows, sensor windows, etc. may require cleaning systems to assure proper performance of the device. Cleaning systems must be able to remove salt, dust, insects, gun gas residue, etc. depending upon the environment in which the aircraft is to operate.

7-3.5.3.24 Ice Protection for Critical Viewing Areas. The exterior surfaces of all critical viewing areas should be kept free of ice during all steady-state and transient flight conditions in the icing environment defined by paragraph 3.2.5 of MIL-E-87145. Transparency anti-ice systems should also include provisions to protect the critical viewing areas from over heating as a result of component failure or inappropriate action from the crew such as turning on a heat source on a hot day. A single failure should not result in failure of the ice protection provisions for two windshields.

Rationale. Critical viewing areas require ice protection to preserve adequate crew visibility. Sensor, camera, and other transparencies with critical transmissibility requirements may require ice protection.

7-3.5.3.25 Noise. The sound pressure levels in occupied compartments due to operation of the airborne environmental control system should not exceed ____________________.

Rationale. Acoustical noise levels must be controlled in order to permit effective and efficient communication between crew members and for radio communication. High noise levels are distracting and fatiguing and can cause permanent hearing damage.

7-3.5.3.26 Component Characteristics.

7-3.5.3.26.1 Pneumatic Actuated Components. Provisions should be incorporated to prevent pneumatic actuated component malfunctions due to dust or moisture in the pneumatic air supply.

Rationale. Many of the problems with pneumatic components can be traced to contaminants or moisture in the pneumatic supply. The problems are usually the result of corrosion in the mechanism or freezing of the condensate.

7-3.5.3.26.2 Insulation. Ducting and components should be insulated or shrouded as required to prevent overheating of wiring, structure or other components, eliminate personnel hazard, or eliminate potential fire hazard. Entry and retention of combustible fluid under or within insulation should be prevented. All insulation should be flame resistant.

Rationale. Hot surfaces must be insulated or otherwise protected to prevent damage, fire hazard, personnel hazard, etc. Fuel (or other flammable fluid) soaked insulation is a serious fire hazard.

7-3.5.3.26.3 Air Cycle Machines.

7-3.5.3.26.3.1 Housing. Each turbine, fan, and compressor housing or scroll should be replaceable.

Rationale. Scrolls or housings are a high cost part of the air cycle machine. Experience has shown that overhaul costs can be minimized when the housing is not an integral part of the air cycle machine and can be individually replaced.

7-3.5.3.26.3.2 Turbine Nozzle. Means should be incorporated to avoid air cycle machine turbine nozzle erosion. The turbine nozzles should be replaceable and should not be a portion of the torus or other major throw away part.

Rationale. Turbine nozzle erosion is a common air cycle machine problem, and the nozzles must be frequently replaced during overhaul. Since the turbine torus or scroll is a high cost item, significant savings can be realized by having the nozzle ring be a separately replaceable item.

7-3.5.3.26.3.3 Balancing. Air cycle machines should have provisions to permit balancing.

Rationale. Balance of high speed rotating machines is critical for long life. Air cycle machines must have provisions for quick and easy balancing during overhaul.

7-3.5.3.26.3.4 Bearings. The air cycle machine should use air bearings.

Rationale. Usage of air bearing designs since the late 1970s has proven that air bearings are superior to the former ball bearing units. For this reason, a proposed ball bearing unit should be vigorously challenged as an inferior design.

7-3.5.3.26.3.5 Instrumentation. Each air cycle machine should incorporate instrumentation for use during testing.

Rationale. Bearing temperature is a normal acceptance test procedure measurement for both production and overhaul. Built in thermocouples permit quick and easy measurement of bearing temperature. Providing thermocouples will help in trouble shooting possible problems.

7-3.5.3.26.4 Controls. The controls used throughout the airborne environmental control system should not become unstable when subjected to any transient condition of the air vehicle or due to any changes in control settings.

Rationale. The purpose of this criterion is to assure that the proper design attention is given to preventing stability and other dynamic problems which could be distracting to the crew.

7-3.5.3.26.5 Heat Exchangers.

7-3.5.3.26.5.1 Ram Air Heat Exchangers. Ram air heat exchangers and associated inlets should be located to avoid functional problems due to operation in icing conditions as defined by paragraph 3.2.5 MIL-E-87145 and avoid foreign object ingestion. A means of access for inspection and cleaning of the ram air heat exchangers, while installed in the aircraft, should be provided.

Rationale. Icing of the heat exchanger core, and blockage due to ingestion of grass, dirt, etc. reduces the ram air flow through the heat exchanger, and significantly degrades the environmental control system performance. Heat exchangers and associated inlets and ducting should be designed and located to avoid or minimize these problems. For those aircraft where the planned operational use may make it impossible to prevent object ingestion, provisions for inspecting and cleaning the heat exchangers should be included.

7-3.5.3.26.5.2 Air to Liquid Heat Exchangers. Heat exchangers which dissipate heat from air into fuel or coolant fluids other than water, should have provisions which prevent a single structural failure from resulting in leakage of fuel or toxic coolant fluid into hot air or occupied compartment air.

Rationale. Leakage of fuel and some coolants into the air can create serious safety of flight problems, such as possibility of explosion or fire and the possibility of toxic fumes or odors entering the occupied compartments. Therefore, these incidents must be prevented by proper design practices.

7-3.5.3.26.5.3 Water Boiler Heat Exchangers and Storage Tanks. A means of access for inspection of water boiler heat exchangers and storage tanks, while installed in the aircraft, should be provided. The water storage tank should not be an integral part of the aircraft structure and it should be readily replaceable. The water storage tank should have a readily accessible water fill port and overboard drain. Water boilers and storage tanks should be capable of repeated freeze and thaw cycles unless the water is protected against freezing.

Rationale. The potential for serious problems exist with the use of water. As a result, a number of design features are necessary to reduce the possibility for serious problems. The likelihood of corrosion necessitates means for easy inspection and replacement of water boiler and storage tank. Frequent servicing requires water fill and drain provisions. The high freezing point of water requires that the boiler and storage tank not be adversely affected by repeated freeze and thaw cycles.

7-3.5.3.26.5.4 Liquid to Liquid Heat. Heat exchangers which dissipate heat from a coolant fluid into fuel should be designed to prevent leakage of fuel into coolant or coolant fluid into the fuel as a result of a single failure. A conservative design approach should be used to minimize the probability of any leakage during life of the unit.

Rationale. Leakage of fuel into a coolant loop can possibly create serious safety-of-flight problems since the addition of air could result in auto-ignition. Therefore, this type of leak should be prevented by proper design practice. Leakage of coolant into fuel is not considered hazardous, but does normally result in degraded cooling system performance.

7-3.5.3.26.6 Couplings. Quick-detachable couplings, if used in the engine bleed air ducting system, should have a safety feature which maintains joint integrity in the event of tie bolt (or other primary latching feature) failure.

Rationale. Quick-detachable couplings are frequently used to permit quick engine changes, easy maintenance, etc. Without a safety feature, failure of the primary latching mechanism can result in the separation of the joint and a high rate of bleed air leakage, which is a serious safety-of-flight hazard.

7-3.5.3.26.7 Ducting Flame Resistance. All non-metallic air distribution ducting should be flame resistant.

7-3.5.3.26.8 Fans. Exposed fans should be protected to prevent personnel injury by fan blades and to protect the fan from foreign object damage.

Rationale. Fans are used in many applications including avionics racks and panels, ground blowers, etc. Because these fans can be operating while personnel are working around them, provisions must be incorporated to prevent personnel injury. Fan installation should also minimize the possibility of foreign object damage, especially for ground blowers.

7-3.5.3.26.9 Pressurization Controls. The location of pressurization control valves, (outflow, safety, and negative relief) should minimize the entrance of dirt or foreign objects into the valves.

Rationale. Dirt and foreign objects can cause malfunction of the pressurization controls. This is a significant safety of flight consideration because the malfunction of a safety valve, due to dirt, can result in personnel injury or aircraft damage.

7-3.5.3.26.10 Liquid Cooling Loops - Level Indication. Provisions should be incorporated to indicate coolant loop liquid level over the entire operating and non-operating temperature and pressure range.

Rationale. Liquid level indication is required to permit quick and easy inspection and maintenance, and should provide an accurate reading under all ground operating and non-operating conditions.

7-3.5.3.26.11 Liquid Cooling Loops - Disconnects. All line replaceable units of liquid cooling loops should have self-sealing disconnects. Liquid coolant connections to the cooled equipment should be a self-sealing and a quick-disconnect type. It should not be necessary to have to refill and bleed liquid loops as a result of routine maintenance actions.

Rationale. Self-sealing, quick-disconnect connections are required for line replaceable units (such as liquid cooled avionics equipment) to permit quick and easy maintenance. Quick disconnects are preferred over valves because disconnects provide for an instantaneous sealing of hoses or liquid cooled components without loss of any coolant.

7-3.5.3.27 Design and Construction.

7-3.5.3.27.1 Materials. Materials selection and corrosion control should be in accordance with accepted design practice.

Rationale. Proper selection and application of materials and corrosion control provisions are necessary to assure the long range reliability, maintainability, and performance of the system.

7-3.5.3.27.2 Structural.

7-3.5.3.27.2.1 Proof Pressure. All system components which are exposed to positive or negative pressure should withstand, without permanent deformation at associated component temperature and pressure, the greater of the following proof pressures:

a. _________ times the gage pressure which occurs during normal operation, no malfunction, that would have the most adverse effect upon the components structural integrity.

b. _________ times the gage pressure which occurs in the event of failure of an upstream pressure or temperature control device that would have the most adverse effect upon the component's structural integrity.

Rationale. The intent of this paragraph is to assure that all pressurized components are adequately designed and tested for structural integrity under both normal operating conditions and upstream pressure or temperature control device failures.

7-3.5.3.27.2.2 Burst Pressure. All system components which are exposed to positive or negative pressure should withstand, without rupture at associated component temperature and pressure, the greater of the following burst pressures:

a. __________ times the gage pressure which occurs during normal operation, no malfunction, that would have the most adverse effect upon the component's structural integrity.

b. __________ times the gage pressure which occurs in the event of failure of an upstream pressure or temperature control device that would have the most adverse effect upon the component's structural integrity.

Rationale. The intent of this paragraph is to assure that all pressurized components are adequately designed and tested for structural integrity under both normal operating conditions and upstream pressure or temperature control device failures.

7-3.5.3.27.2.3 Rotating Equipment Containment. The housing and scrolls of all rotating machinery of the airborne environmental control system should completely contain all fragments from:

a. Fused drive rotor failures (including tri-hub burst) at the maximum fuse speed and at the pressure and temperature associated with this speed.

b. Non-fused drive rotor failures (including tri-hub burst) at the maximum speed that can result from any failure inducing - condition or __________ percent of the maximum normal speed, whichever is greater, at the pressure and temperature associated with the speed.

c. Driven rotor failures (including tri-hub burst) at the maximum speed that can result from "a" or "b" above at the pressure and temperature associated with this speed. Driven rotors should not burst at speeds lower than the drive rotors burst.

Rationale. The intent of this paragraph is to assure that all rotating components are adequately designed and tested for structural integrity under both normal operating conditions and under upstream pressure or temperature control device failures. To prevent further damage to the air vehicle or injury to crew members, the housing and scrolls of all rotating machinery should completely contain all fragments from blade failures of wheel bursts.

7-3.5.3.27.2.4 Overspeed Spin. All rotating equipment should be designed so that operation at the speed resulting from the worst case single failure or up to 120%

percent of normal maximum operating speed for a period 5 minutes is possible without rubbing or other adverse effect on the equipment. Provisions should be incorporated to prevent or limit the freewheeling of all rotating equipment.

Rationale. Rotating equipment should be designed to operate satisfactorily and without adverse effect, under certain failures conditions, such as pressure regulator malfunctions, short term ram air circuit blockage, or starvation, etc. This is consistent with the "failure concept" of 7-3.3.3. Freewheeling of rotating equipment should be limited or prevented because of the hazards associated with failure of the equipment under these conditions.

7-3.5.3.28 Ground Provisions. Provisions for ground testing should be incorporated for functions such as leakage testing, ECS functional checkout, and cooling, heating, pressurization, and ventilation performance checkout.

Rationale. Provisions for ground pressure leakage testing of duct systems, pressurized compartments (see 7-4.2.1.1.5) etc., should be provided for troubleshooting and checking integrity following maintenance. Provisions should be incorporated for troubleshooting and checking the system. Provisions should be incorporated for cooling, heating, and ventilating occupied areas, avionics equipment, etc. for ground operation, maintenance, checkout, etc.

7-3.5.3.29 Ground Connections. Air inlet connections for the external ground air conditioning source should conform to appropriate Military Standards. Ground pneumatic cart connections should be compatible with existing ground equipment intended to be used

Rationale. Air Force standardization and International standardization agreements require specific interfaces with ground support equipment.

7-3.5.3.30 Ground. The minimum diameter of aircraft gravity filling orifices should be as follows:

De-icing Fluids	1 3/4 inches
Coolant Fluids	1 1/2 inches
Methanol Water	1 1/2 inches
De-mineralized Water	1 1/2 inches

Rationale. International Standardization Agreements require minimum diameter for aircraft gravity filling orifices.

7-4.0 **QUALIFICATION GUIDELINES**.

7-4.1 **General**. In general, the intent of this section is to ensure that the performance criteria proposed by the contractor in the weapon system specification or component detail specifications are met. The contractor should demonstrate that these criteria are met by conducting design reviews and showing that specific features have been included, by conducting analyses (e.g., trade study, component and system performance, safety hazard, Failure Modes and Effects, engineering analyses as required), and by demonstrations and tests.

7-4.2 **Analysis/Data.**

(1) Engineering analyses need not be submitted to the Government as formal data. However, the Contractor (and subcontractors) should conduct these analyses as required and document and retain them through completion of the flight test program. The government may request these to be provided at Technical Interface Meetings (TIM) or other meetings for design evaluation purposes.

(2) Stand alone ECS Component Performance data need not be submitted to the Government as formal data. However, the Contractor should provide this data to the Government upon request during TIM's, telephone conferences, or other meetings. Further, this data should be made a substantiating part of the performance analysis report and submitted formally. This data should include but is not necessarily limited to:

(a) heat exchanger thermal efficiency, pressure drop, weight, shape, materials, fin configuration, and core geometry and dimensions. Thermal efficiency and pressure drop data for the core should be presented as the product of the Stanton and Prandtl number and Friction factor as a function of Reynolds number.

(b) performance data for air cycle machines, vapor compressors, fans, and liquid pumps

(c) Pressure drop data for all distribution ducting, interconnecting ducting for the refrigeration pack, bleed air ducting, and other components contributing to the pressure drop of the system. Ducting layout and system installation drawings are included.

(d) Heat load summary of the equipment and avionics. The summary should itemize the heat loads, distinguish between the forced and ambient cooled equipment, identify the equipment's location in the aircraft, and list the equipment's cooling requirement to satisfy its allocated reliability.

(e) System description and schematic. The schematic should show all ECS components, indicate those locations where performance predictions for flowrate, pressure, temperature, and humidity/enthalpy are made, indicate control set points, and correlate the analysis pressure drops to the respective locations on the schematic

(f) Efficiency of components removing water and their dry and wet pressure drops.

7-4.3 **Bleed Air Subsystem.** The contractor should conduct and document engineering analyses to define the thermal stresses in the bleed air ducting and use the guidance of ARP 669 as required to mitigate thermal stresses in all bleed air ducting. Results of these analyses should be presented at the CDR and the design should be consistent with the analysis results (i.e. if analysis indicates that expansion joints are required, then the design should have expansion joints).

7-4.3.1 General Design Criteria.

7-4.3.1.1 Shutoff Capability. Inspection of drawings, laboratory testing, and aircraft demonstration should be used to verify that independent means are incorporated for shutting off air flow from each bleed source and that they are controllable from the crew station. The aircraft demonstration should include functional testing of this capability during aircraft acceptance testing and during flight testing.

Rationale. Bleed source shutoff provisions require verification through aircraft demonstration to assure proper installation and functional interface. Drawings and schematics must be examined to verify that shutoff control features are independent.

7-4.3.1.2 Overbleed Protection Analyses, inspection of drawings and laboratory tests should be used to demonstrate protection against overbleeding of bleed air sources. In some instances the flow limiting device is an integral part of the engine or APU. Test results and or engineering data supporting this fact should be presented at the CDR. Also, maximum airflow requirements of the ECS should be compared to the flow limits of the bleed air source during the CDR.

7-4.3.1.3 Bleed Source Isolation Inclusion of bleed source isolation should be verified by inspection of drawings and an analysis of the bleed air system design. Functionality of the bleed source isolation should be demonstrated with aircraft testing, and it should include all applicable combinations of normally operating valve positions and with valve failures.

7-4.3.1.4 Reverse Flow Protection Inspection of drawings should verify incorporation of provisions to automatically prevent reverse flow from one bleed source into another source. The Contractor should also conduct a safety analysis of the bleed air system assuming a bleed air check valve fails. This analysis should indicate whether or not additional design effort is needed for the bleed air system.

Rationale. This feature may be safety critical, and the assessment is necessary because of the difficulty of knowing when a bleed air check valve failure has occurred.

7-4.3.1.5 Pressure Regulation. Analyses, laboratory tests and aircraft ground and flight tests should be conducted to demonstrate that bleed pressure is regulated to the minimum level required by all using subsystems.

Rationale. The critical nature of this requirement necessitates interim verification by analyses during the initial system design phase and final verification by laboratory and aircraft tests.

7-4.3.1.6 Duct Surface Temperature Duct surface temperatures below levels which could cause auto-ignition of flammable fluids or below other critical levels should be demonstrated by heat transfer analyses. At locations where surface temperatures are allowed to be above critical levels, it should be demonstrated via inspection of drawings and a safety hazard analysis that such conditions pose insignificant probability of a safety hazard.

7-4.3.1.7 Pressure Relief Pressure relief device installation should be verified by drawing inspection, and laboratory tests should be conducted to verify pressure relief device performance. A failure mode and effects analyses or safety hazard analysis should verify the need for protection from excessive positive and negative pressure. This analysis should verify and document the size of pressure relief areas and pressure relief settings as required.

Rationale. Verification of this requirement is left to analysis and laboratory testing since aircraft demonstrations are not feasible.

7-4.3.1.8 Distribution Shutoff Inspection of drawings and aircraft demonstration should be used to verify that independent means are incorporated for shutting off air flow to each subsystem which uses bleed air and that these means are controllable from the crew station.

Rationale. Distribution shutoff provisions require verification through aircraft demonstration to assure proper installation and functional interface. Drawings and schematics must be examined to determine that shutoff control features are independent.

7-4.3.2 Materials and Part Design Criteria Proper choice of material and part design should be verified by an engineering analysis defining the stress on the part(s). This stress analysis should consider (1) pressure, temperature, and aircraft induced loads at the normal operating condition(s), (2) loads from unsupported components such as valves, heat exchangers or other equipment, (3) loads due to a worst case failure of a component of the ECS which might increase the stress on the part (e.g. a support bracket failure for a component mounted to ducting, or a failure resulting in increased temperature or pressure or both), (4) loads due to the specified vibration environment, and (5) handling loads as these loads may sometimes exceed the normal operational loads. Parts (i.e. support brackets, tubing, ECS components, etc.) expected to be subjected to handling loads greater than operational loads during maintenance should be addressed at CDR. Verification of and data demonstrating that parts should not be adversely affected by corrosion or aging should be presented at CDR.

7-4.4 **Pressurization.**

7-4.4.1 Cockpit Pressurization Level or Schedule. The cockpit pressurization level should be demonstrated with engineering analysis and tests. The analysis should consider the operating condition with the least air inflow to the cockpit or that condition for which the required pressurization is the most difficult to achieve. A manufacturing assessment of the expected maximum cockpit leakage area compared to the engineering established maximum limit should be a part of the analysis, and it should be documented and presented at CDR. The contractor should also present data at CDR to show that cockpit leakage will not increase with aircraft age. If data is not presented, the contractor should use the guidelines of paragraph 7-3.4.1 in establishing the maximum expected leakage rates of the aircraft for its service life and should show that the required pressurization is achievable using these guidelines.

Aircraft manufacturing and flight tests should also be conducted to verify that requisite pressurization is achieved. A manufacturing test should be conducted to determine the leakage rate and effective leakage area by pressurizing the aircraft to the maximum required pressure differential. Flight tests should be sufficiently instrumented to record cockpit pressure.

Rationale. Early analyses are required to establish leakage area limits. It is very difficult to construct cockpits, bulkheads, and skin panels which do not leak pressurized air. For this reason, it is necessary to alert manufacturing team leaders to the leakage requirements with appropriate analyses so there is sufficient time to plan for leakage containment. Providing a comparison between the manufacturing assessment of leakage and the engineering requirement during CDR ensures that necessary coordination is taking place between the teams and this maximizes the probability for success. However, tests are the only valid way of demonstrating that pressurization is being achieved.

7-4.4.2 Pressure Release. Pressure release should be demonstrated with manufacturing and flight tests. These tests should show that cockpit pressure has been brought to a safe level prior to unlatching the canopy. Furthermore, if doors or canopies are not fully latched, these tests should demonstrate that appropriate action, as determined by a safety hazard analysis, can be taken to preclude an unsafe condition from developing.

7-4.4.3 Aircraft Fuselage/Equipment Bay Pressurization. If the aircraft fuselage or any equipment bay is by design, pressurized, the contractor should demonstrate the ability to pressurize the volume to the required level by testing, and the testing should show that the required pressure level is achieved with the design air flow-rate. Furthermore, if the air used to pressurize the fuselage or equipment is collected internally to the aircraft structure and routed to a single or various destinations for added cooling of equipment, the Contractor should demonstrate by testing that the required amount of air (1.6 times the amount required for cooling the destination avionics or equipment) is provided. (The factor 1.6 is used to compensate for development of in-service leakage. This factor is required unless the contractor can prove to the contrary).

7-4.4.4 Positive and Negative Pressure Release The contractor should demonstrate by analysis that no damage to flight controls, flight surfaces, or other aircraft parts which could adversely affect safety of flight and landing can occur as a result of sudden decompression, over-pressurization, pressure differential, or negative pressure. For decompression analysis the Contractor should assume a hole size which results in the most severe condition. Hole sizes should be derived from:

(1) a door or canopy opening as a result of a faulty latch

(2) a window

(3) a failed engine turbine disk

(4) Enemy fire with subsequent loss of pressurization. For this analysis the Contractor should determine the maximum hole size sustainable (with respect to pressurization problems) for vulnerability purposes and verify that the vulnerability requirement is not compromised as a result of enemy fire and the loss of pressurization.

Analysis should be conducted to verify adequacy of positive and negative pressure relief settings. Laboratory and ground tests should verify positive pressure relief function. Laboratory and flight tests (rapid descent) should verify negative pressure relief function.

Rationale. Analyses are the only feasible means of verifying many of these requirements because of the inherent danger of personnel injury or aircraft damage while simulating a rapid decompression. This is especially the case since the requirement may permit some damage to occur during extremely rapid decompression emergencies. Ground and flight tests can easily verify most normal pressure relief functions.

7-4.4.5 Pressure Source. Analysis and flight test should verify that the minimum flow-rate to pressurized compartments is at least ___CTBD___ times greater than the maximum allowable production leakage rate. Air source contamination levels and partial pressure of stored gas sources should be verified during qualification testing of the pressure source. Failure mode and effects analyses should verify that no single failure of a supply or control component with a probability of failure greater than ___CTBD___ results in CTBD_________. Inspection of drawings and the aircraft should verify incorporation of means in air supply inlet openings to prevent rapid loss of compartment pressure in the event of pressure source failure. For multiple pressure source applications, analysis should verify ability to maintain pressure schedule with one source inoperative.

7-4.5 **Refrigeration Package Subsystem**

7-4.5.1 Instruments and Controls Inspection of drawings and the aircraft should verify incorporation of the functions described in paragraph 7-3.5.1. The function of the instruments and controls should be demonstrated during flight tests.

7-4.5.2 Test Provisions The Contractor should demonstrate pressurization testability by doing the test. This test should also show that indication is provided to the crew if a pressurization device is left inoperative, or that a means of preventing the pressurization device being left inoperative is provided. This test should show achievement of the required pressurization and required air flow-rate simultaneously. The contractor should demonstrate that all controlled parameters (i.e. currents to valve torque motors, discrete signals to solenoids or actuators, etc.), diagnostic and feed back information (i.e. outputs signals from ECS sensors and other subsystem sensors which interface with the ECS), and signals used between the ECS and other subsystems are accessible by maintenance personnel on a time history basis. This demonstration should include ground and flight operations.

Rationale. Typically, compartments utilize two control devices which assure proper pressurization. The compartment pressure level is maintained by a device which either controls to a constant absolute pressure or a constant differential to the ambient pressure. Additionally, a safety device must be utilized which will assure that, in the event of a system failure, the crew and aircraft structure will be protected from over-pressurization. Because structural damage can occur as a result of pressurization, it is necessary that these pressurization control devices have provisions for frequent testing.

7-4.5.3 Cooling Occupied Compartments.

7-4.5.3.1 In-flight and Ground Cooling (Flight Suit Environment). Cockpit temperature control to the level(s) specified in the weapon system specification should be demonstrated by analysis and testing. The analysis should assume characteristic pressure drops for all interconnecting ducting and components as determined by tests, individual component performance as determined by testing, and heat losses and gains of all bleed air ducting, interconnecting ducting between ECS components, and distribution ducting. The analysis model should use individual component performance characteristics and be updated as new data becomes available. The component performance data, pressure drop characteristics, and ducting heat loss/gain data used in the analysis should be made available to the government on an as requested basis (not a formal deliverable). The ECS analysis (formal data item report) should be documented and submitted before the CDR (i.e. per the contract requirements). If the contractor has subcontracted the performance analysis of the system, the contractor should include provisions in the subcontract to acquire component performance data so that it can be provided to government evaluators during TIM's conducted prior to and after the CDR. Part of the analysis should be a cockpit volume air distribution analysis which describes the velocity, direction, and momentum of the air in the cockpit. This analysis should be used to demonstrate that the distribution design in the cockpit is optimized for heating and cooling the crewmember's envelope, thus minimizing the weight impact on the ECS, and maximizing human comfort. This analysis should be able to predict spatial temperature throughout the cockpit. (See paragraph 7-4.5.3.4). The analysis model should undergo revision as required when individual component performances change, as laboratory system testing is accomplished, and as flight testing is accomplished. These stages of revision should serve to validate the model for predictive capability under observed laboratory and flight conditions and under conditions which may not be achievable as scheduled activities (extrapolated conditions). As new data which will be used to revise the model becomes available, it should be provided to the Government as requested, but not as a formal data item. The analysis report should undergo revision after the flight test is completed. If flight testing for the ECS is not conducted, the analysis report should be revised after laboratory system testing is completed. If flight testing or system laboratory testing can not achieve the specified design flight and ambient conditions required, then the validated analysis model should be used to verify that the required cockpit conditions are achieved. If the cockpit is not simulated (i.e. not geometrically similar and does not have the distribution ducting of the aircraft) during laboratory testing then the aircraft cockpit should be instrumented with thermocouples capable of recording the dry bulb temperature located at the crewmember’s head, torso, and foot levels. These thermocouples should be the basis for demonstrating that the required cockpit temperatures are achieved. Furthermore, the Globe Temperature should also be measured in the presence of maximum solar radiation, and the recorded value should be extrapolated to the design condition as required. The cockpit should also be instrumented to record the wet bulb temperature in the vicinity of the crew member(s). These temperatures should be used as required to demonstrate that the cockpit environment is maintained as required. Laboratory testing should include all aspects of the ECS. This includes but is not necessarily limited to:

(1) the bleed air ducting, valves, and heat exchangers,
(2) the refrigeration package which includes the heat exchangers, valves, air cycle machine or vapor compressor, sensors, interconnecting ducting, simulated ram air scoops, and controller,
(3) the NBC system components,
(4) components comprising the Microclimate cooling of crewmembers wearing NBC protective gear, and
(5) the distribution ducting down to the equipment interface. The cockpit distribution ducting need not be included, but the cockpit volume and the cockpit distribution ducting pressure drop should be simulated.

Rationale. Analysis is required to ensure that the initial design has a high probability of meeting the requirement. It is important the Government evaluators have access to the performance data and assumptions used in the Contractor’s analysis. This data can be provided to the government during TIM’s to expedite communication of design intention and system capability. If the contractor has subcontracted the performance analysis of the system, the contractor must include provisions in the subcontract to acquire component performance data so that it can be provided to government evaluators during TIM’s conducted prior to and after the CDR. A cockpit distribution model is critical since experience shows that aircraft modifications occur because of inadequate flow distribution in the cockpit and pilot complaints, and it is a necessary basis to support design changes when the existing design is inadequate. Ground and flight testing is the chief way of determining that the cockpit environment is maintained as required. However, design conditions are not usually encountered during the flight test, so cockpit thermal requirements are usually proven by analyses and backed up with limited flight data. The laboratory system test rig described above is necessary for a variety of reasons: it is the chief means to show that the ECS has the required heating and cooling capacity; it is used to demonstrate that the software control algorithms properly function and the ECS can be controlled safely; it is used to uncover and investigate control and design problems; and it is used to demonstrate NBC requirement compliance.

7-4.5.3.2 In-flight and Ground Cooling (NBC Gear Environment). Providing the required cooling to crew members while they wear NBC protective gear (i.e. Mission Oriented Protective Posture (MOPP)) should be demonstrated by testing. The test should include sufficient instrumentation to determine the quantity of cooling to each crew member and to record a time history of the temperature delivered to the crewmember’s NBC protective gear. The test should demonstrate:
(1) each crewmember’s capability to control his own microclimate temperature environment,
(2) that the crewmember’s face shield should be kept free of condensation on the internal side, and
(3) the maximum required cooling is achievable within 60 seconds after system start.

The test should also demonstrate that the cockpit and MOPP gear can be heated and cooled simultaneously. Analyses and tests should verify that the cockpit can be heated to a minimum of 50°F while heat is being provided to the MOPP gear. An analysis should show that cooling is applied to a minimum of 15% of the crewmember's body surface area and that there is a minimum of 300 watts of cooling capacity per crew member.

7-4.5.3.3 Transient Cool Down/Heat Up (Cockpit). Analysis supplemented with flight test data should be used to establish that the cockpit effective temperature (i.e. the performance requirement established by the system specification) should be achieved to within 98% of the final steady state value within 25% of the shortest mission profile time - should not to exceed 30 minutes. Similar to the steady state analysis described in paragraph 7-4.5.3.1, this analysis should undergo periodic revision as updated performance data becomes available, but an initial analysis predicting transient response should be completed prior to CDR. Furthermore, analysis data and assumptions should be made available to the government at TIM's or as requested prior to the CDR (i.e. not formal data submittal) so that an evaluation of the analysis and the system capability can be determined. The analysis should address both the heating and cooling capacity of the cockpit (i.e. the performance requirement) and the supply temperature to the MOPP equipment when that particular mode of heating/cooling is used.

Rationale. Since it is very difficult to conduct flight tests under design conditions, and frequently laboratory system demonstration tests are constructed and conducted with inadequate simulations of the cockpit, analysis is the only viable means to show expected performance.

7-4.5.3.4 Cockpit Air Distribution Inspection of drawings should show that each crew member has foot warming outlets and individual air gaspers for cooling and the torso and head levels. Analysis and flight or laboratory testing should demonstrate the adequacy of the design. The analysis should show that the distribution system is optimized to cool and heat the crewmember's envelope (i.e. torso and head) to the system requirement by controlling the air flow and direction (See paragraph 7-4.5.3.1). The analysis should also show that the temperature variation in the crewmember's envelope does not vary by more than ___°F. The laboratory or flight test should include sufficient dry bulb thermocouples to determine the conditions in the pilot envelope area. The tests should also show that the air can be directed to optimum locations and that flow-rates can be adjusted.

Rationale. A thorough analysis is needed because design conditions are difficult to achieve during flight tests.

7-4.5.3.5 Air Velocity. The air velocity as it impinges on the pilot envelope area should not exceed ___ ft/min as demonstrated by analysis (See Paragraph 7-4.3.5.1).

Rationale. The need for controlling the air velocity is discussed in the criterion section. Analysis is the most economical way of determining air velocity because system testing will measure the air flow-rate delivered to the cockpit. However, the cockpit distribution system is not usually included because of added complexity and cost to the system test. The results of the system tests, flow-rate having been measured, can be used to determine what the gasper outlet velocity would be. This data can be combined with the analysis of paragraph 7-4.3.5.1 to determine velocities in the vicinity of the pilot envelope.

7-4.5.3.6 Temperature Control. An analysis should be used to demonstrate that a means exists to prevent harmful or damaging high temperature air other cockpit heat sources from causing equipment damage or personnel injury during normal operation or for a single failure condition. The means may be automatic or manual, but assuming appropriate manual action is taken either method should prevent damage and/or injury, transient conditions notwithstanding. This analysis should be made a part of the System Safety Hazard Analysis and documented.

7-4.5.3.7 Surface Temperature. Flight tests should demonstrate that surfaces which can be contacted by the Crewmembers do not cause adverse discomfort because of heat exchange.

7-4.5.3.8 Normal Ventilation. An analysis should show that at least 20 cfm/crewmember of fresh air is provided to the cockpit on a continual basis during normal operation.

Rationale. An analysis is adequate to demonstrate that enough fresh air is introduced because the cockpit normal leakage will be this order of magnitude. For those cockpits which are pressurized enough fresh air has to be introduced to offset leakage and maintain pressure.

7-4.5.3.9 Emergency Ventilation. The provisions for emergency ventilation can be verified by inspecting drawings, but the performance should be verified by test. A small smoke generator can be used to release smoke in the cabin, and the time required to clear the smoke can be measured. This method has been used on many cargo aircraft. Whether or not a ground or flight test is conducted should be determined based upon the probability of smoke entering the area, possible causes, and the hazard associated with the actual test. The procedure for removal of smoke should be developed and tested early in the flight test program.

7-4.5.3.10 Suit Ventilation /Pressurization. Performance of the anti-g suit should be demonstrated by testing. The volume of the anti-g suit may be simulated by any suitable pressure vessel device. The test should demonstrate that the pressurization response time of the suit and the required pressure level is achieved as required. The cooling performance of the ventilation suit should be demonstrated by testing. The testing should record the temperature and the air flowrate, and these should meet the established design requirements of the vent suit. Control of the vent suit should be demonstrated by testing as required.

7-4.5.3.11 Contamination. Contamination of the cockpit supply air should be quantified by measurement under conditions that are intended to show that the ECS is not a source of contamination. This test should not be necessary if analysis shows that the ECS is not a source of contamination which can render the crew unable to function safely or complete the mission. Furthermore, analysis should show that the ECS is not a source of toxins that could result in death, in permanent, or in serious injury to the crew if they were to be exposed as a result of an ECS failure. If the ECS contains filter material for NBC purposes or general filtration such as odor removal, a laboratory system test should be conducted to demonstrate that the filter material is not contaminating the ECS air more than the established permissible limit. (Since the filtration medium is unknown, limits can not be established in this document. However, filtration mediums can be hazardous to the crew, and they also need to be evaluated for possible maintenance implications if they are released into the ECS. Therefore, the contractor should take the lead in studying the effects of the filtration medium on human physiology and the aircraft equipment assuming it is released in small quantities over a long period of time. These findings should be topics of discussion at TIM's and the government and contractor should jointly determine the appropriate permissible exposure limit of the medium. This limit should be adopted by the contractor as a requirement and should be covered at the CDR as required.) This test should include an air sample taken as close as possible to the bleed air source (i.e. upstream of the ECS filter) and a sample downstream of the ECS filter (i.e. cockpit supply or avionics supply air). Furthermore, a comparison should be made between the samples taken at the beginning of the system test to samples taken at the conclusion of the system tests or just prior to the rig dissemble. This will help in determining if the filter material is susceptible to breakdown over long periods of time. (This is assuming that the system rig is used for endurance testing. If not, samples from the aircraft should be taken at the beginning of the flight test program and at the conclusion of the flight test program.) The results of this comparison should be addressed in the System test report.

Rationale. Past ECS systems have used toxic materials. The intention is to eliminate their use, but this is not always possible. When they are used a means must be devised to test for the safety of the crew and to ensure that maintainability problems will not be created as a result of their use. Consequently, it may be necessary for the contractor to investigate the physiological effects endemic to the material and also possible maintainability impacts. An example of this is the use of 13x molecular sieve (Sodium Aluminosilicate - Na_2O: Al_2O_3: $2.8SiO_2$: xH_2O - also known as zeolite). This material has been observed to enter the ECS and cause severe erosion to aluminum. It is not expected that this material will cause permanent or serious injury to the crew under normal usage, but its erosive property could release other toxic substances if present. In addition, the long term effects of this substance on the equipment is unknown. Other unknown filtering agents may be used in the future. These must be evaluated as they are introduced, and conducting tests on the completed system is the only way of determining that permissible limits as established by investigation and study are not exceeded.

7-4.5.3.12 Cargo Overboard Venting. Venting requirements should be established by the Contractor on an as needed basis, and they should consider needs for passengers and cargo. If requirements are established these should be discussed at pre-CDR TIM's and the requirement as established jointly by the Government and Contractor should be presented at the CDR. Established requirement(s) should be demonstrated by an analysis that shows the ECS is capable of producing the required amount of vent air to purge equipment bays, cargo bays, or other area as required.

7-4.5.3.13 Nuclear/Biological/Chemical System. The performance of this system should be demonstrated by testing in the laboratory and during flight tests. If the system provides dual functions such as water or water vapor filtration in addition to NBC materials, each function should be tested individually and combined. For example, the NBC system should be tested for its water removal capability without NBC contaminants present. The system should also be tested with water and NBC contaminants present at the required challenge levels. Furthermore, ambient temperatures (i.e. bay temperatures where the equipment is located) should be simulated during the testing if it is known that the filter medium performance or system performance is dependent upon temperature. The ability of the NBC system equipment to detect chemical agents and record the levels as required should also be demonstrated by testing. For chemical challenges the system test rig should be used to demonstrate compliance to the performance requirement. While other ECS performance testing is being conducted (i.e. heating, cooling, pressurization, ventilation, etc.), the test rig should be used to test NBC characteristics with a suitable chemical simulant(s) that does not require extraordinary safety precautions to be taken. If simulant testing cannot be done in conjunction with the other ECS performance testing, chemical simulant testing should be scheduled towards the end of the system testing to ensure ample time for discovery of malfunctioning equipment, component infant mortality, etc., which may adversely affect the chemical protection performance outcome. The simulant should be introduced at the simulated bleed air source (i.e. APU or engine bleed port, etc.). The rig should be instrumented to take samples or record the simulant concentration at the source, before and after heat exchangers, before and after the air cycle machine, the cockpit inlet concentration, and sources of air that may be possible to bypass the filtration equipment, such as cooling air for air cycle machine bearings. Reductions in the concentration of the simulant as it traverses the ECS should be recorded as a function of time (i.e. concentration as a function of location and time). If a heat exchanger or other component failure can cause contamination of the cockpit, the rig should be instrumented to detect contamination due to a failure of these components. Filter medium integrity should also be demonstrated with laboratory testing. Air samples downstream of the filter should be taken at the onset of the system testing and be examined for filter medium material such as Sodium Aluminosilicate. Samples should be taken throughout the test program to determine if the rate of output of filter medium is changing. These samples should show that filter medium contamination does not increase with usage. Live agent testing should also be conducted. Preferably, this will be done in the system rig. If this is not feasible, a pseudo-ECS test rig should be constructed which encompasses as much of the ECS as practicable. The pseudo-ECS rig should be able to provide the interface parameters and boundary conditions of the absent ECS components (i.e. flowrate, pressure, temperature, liquid water content, humidity, and other persistent contaminants such as engine oil vapors) which represents the most severe operating condition for the filtration equipment. Engine or APU oil vapors at concentrations allowed by the respective specifications should be imposed simultaneously with water and chemicals for the duration of live agent testing. Challenge levels and time duration of the chemical agents should be as established by the system specification. If the specification does not distinguish between ambient challenge level(s) and the challenge level(s) which would exist at the interface of the NBC equipment, then the specification requirement (both chemical concentration and time) should be introduced at the bleed air source. If the NBC live agent testing cannot be conducted on the system test rig which would simulate the ECS in its entirety, then test samples taken from the system rig as described above should be used to establish the challenge level and time duration of the chemicals at the NBC interface on the pseudo-ECS rig. If this test data is not available at the time of the live agent testing, the full challenge level stated in the system specification should be applied at the inlet of the pseudo-ECS test rig. The flight test should be conducted with a suitable detectable simulant. The purpose of this test is not to show that the ECS/NBC system components work and perform as required, but rather it is to show that chemical infiltration to the occupied compartments does not occur under field operating conditions. However, if laboratory testing has not been conducted to demonstrate performance of the NBC equipment, then the flight test should be used as the platform to demonstrate it as well as stopping infiltration. As such, instrumentation described for the laboratory testing should be included for the flight test aircraft, and this instrumentation should be able to show that the ECS supply air is free from contamination and that infiltration does not occur.

Rationale. Live agent testing is the only way to prove that NBC equipment can work as required, but it is difficult to conduct because of the sheer size of an ECS (laboratory availability) and the need to take extraordinary safety precautions. Therefore, exhaustive laboratory testing with simulants, followed up with smaller scale live agent testing combines the proof of the concept (live agent testing) with the practicalities and limitations of real system operation. Both type tests must be conducted to conclude that the NBC system will perform as required.

7-4.5.3.14 Moisture Control. An analysis should show that moisture does not contaminate the avionics or does not create snow or fog in the cockpit.

7-4.5.3.15 Dust Control. An analysis should be conducted to determine the quantity and type of dust which can enter the ECS during periods when backup or emergency cooling is selected. This analysis should show that the contamination levels due to dust has no effect on the avionics, that it does not degrade the cooling capability of heat exchangers, and that it does not adversely affect any other component in the ECS. If this can not be demonstrated, then the analysis should describe the maintenance actions required to restore avionics operation, or the cooling capacity, or restore the other ECS components to their normal operating condition, the frequency with which said action must be done, and that this condition is allocated in the flowdown requirements for aircraft maintainability.

7-4.5.3.16 Equipment Temperature Control. Testing and analysis should be conducted to demonstrate that proper temperatures are maintained for the avionics and electrical equipment as required. The analysis should itemize the equipment and indicate the required environment to be maintained for each piece based on the reliability allocations established by the Contractor. Equipment which is performance sensitive to temperature should be itemized, and the analysis should show that transient temperature environments during cold soak start ups and hot soak cool downs for these selected pieces of equipment are consistent with their functional specification requirement (i.e. the equipment is allowed to function within the prescribed time and performance tolerance as set forth in the detailed equipment specification). The analysis should be conducted assuming a full complement of equipment (i.e. maximum heat load) and soak conditions at the extreme hot and cold day cases for the start up requirements. The validated ECS performance model described in paragraph 7-4.5.3.1 should be used to predict the equipment flowrate and temperature supply conditions. Then these supply conditions should be used to predict avionics and electrical equipment reliability, and lastly the predicted reliability numbers should be compared to the allocated requirements. The Contractor may consider the equipment duty cycle in the reliability predictions. The laboratory system test rig as described in paragraph 7-4.5.3.1 should be used to show that adequate cooling conditions exist at all necessary locations for the proper operation of the equipment.

7-4.5.3.17 Equipment Growth. Paragraph 7-3.5.3.17 of this document or the weapon system specification specifies the growth heat load. Laboratory testing should demonstrate that the baseline ECS can cool the baseline and growth heat loads within the allocated equipment reliability requirement. The system test rig should simulate the nominal avionics and electrical loads of the baseline aircraft, and it should be modifiable to increase the avionics and electrical loads by the specified amount. Any feature of the growth equipment that causes a performance perturbation of the baseline system or necessary modification should be simulated and tested in the rig (i.e., additional distribution pressure drop, added components, etc.). Also, the testing should show that the backup cooling provisions can provide the required cooling for the growth load under all applicable missions..

7-4.5.3.18 Avionics Ground Operation. The laboratory system test rig and ground testing should demonstrate that required cooling is provided to the avionics as a precondition for its use.

7-4.5.3.19 Avionics Cooling Distribution. Laboratory system and ground testing should show that removal of one or more unneeded avionics Line Replaceable Units (LRUs) during maintenance or aircraft reconfiguration does not cause an imbalance in coolant flow distribution.

7-4.5.3.20 Emergency Cooling. Laboratory system and ground testing should show provisions for an emergency cooling system and enough cooling to the avionics and electrical equipment for mission completion under the design day specified in paragraph 7-3.1 or the weapon system specification.

7-4.5.3.21 Fog and Frost Protection. Laboratory testing should demonstrate that the ECS can prevent the formation of fog and frost on all critical viewing areas under normal and backup cooling operations and that thermal protection for all critical viewing areas prevents overheating for normal and backup cooling modes.

7-4.5.3.22 Rain Removal. Flight tests should demonstrate that the rain removal system maintains the critical viewing areas clear of rain.

Rationale. Laboratory tests are essential in the design, development, and testing of rain removal systems because design conditions are difficult to encounter and measure during flight testing.

7-4.5.3.23 Transparency Cleaning. Laboratory/flight tests should demonstrate the transparency cleaning system.

Rationale. Laboratory tests are essential in the design, development, and testing of cleaning systems because design conditions are difficult to encounter and measure during flight testing.

7-4.5.3.24 Ice Protection for Critical Viewing Areas. Laboratory and flight tests should demonstrate that the anti-ice system maintains the critical viewing areas clear of ice.

Rationale. Analyses are necessary to establish the heat requirements, spray requirements, etc. and can be used to evaluate the adequacy of the system. Similarly, laboratory test results can be used to evaluate the system. The final evaluation of the full system installed on the aircraft is accomplished during flight tests behind a spray tanker.

7-4.5.3.25 Ice protection for Critical External Surfaces. Laboratory and flight tests should demonstrate that the anti-ice system maintains the critical flight surface areas clear of ice.

Rationale. Analyses are necessary to establish the heat requirements, spray requirements, etc. and can be used to evaluate the adequacy of the system. Similarly, laboratory test results can be used to evaluate the system. The final evaluation of the full system installed on the aircraft is accomplished during flight tests behind a spray tanker.

7-4.5.3.26 Noise Laboratory and flight tests should verify the ECS does not exceed the specified noise requirement for any of its control modes. These tests should determine the noise sources and volume levels, extrapolate this data to predict the aircraft's expected acoustical environment, and compare that with the requirement.

7-4.5.3.27 Component Characteristics. Separate laboratory performance tests should be conducted on all ECS components such as heat exchangers, fans, pumps, compressors, turbines, etc. to fully map their performance envelopes. Engineering reliability development tests should also be conducted as required on each component to synthesize its optimal design.

Rationale. The performance of an ECS cannot be tested until the system is assembled in the laboratory. The lead time can often exceed 24 months. Therefore, analysis must be used to evaluate system performance, and the analysis is predicated on individual component performance. To accurately assess system performance in a timely manner, component performance testing must be accomplished as soon as possible in the development cycle.

7-4.5.3.27.1 Pneumatic Actuated Components. Laboratory tests should demonstrate that dust and moisture does not cause a malfunction in pneumatically actuated components while they operate in the normal or backup cooling mode. Pneumatic components supplied exclusively with filtered dry air do not require testing. Laboratory tests should determine the quality of the filtered air delivered to the pneumatic components. The tests should record particulate concentration, size distribution, and moisture content. Tests should be conducted with the pneumatic components installed in the production configuration so that problems of moisture condensation in pneumatic lines, etc. can be identified. The following are recommended tests for verifying adequacy of pneumatic components:

a. Accelerated Internal Corrosion and Humidity. All pneumatically operated control components of the subsystem should be subjected to the following tests:

1. Components should be oriented in the same attitude as they will be installed in the air vehicle during all phases of testing.
2. All internal surfaces which are exposed to pneumatic air should be thoroughly wetted by supplying a solution of 5 percent (by weight) of sodium chloride in water to the component. Valves should be cycled five times from closed to open during the wetting operation.
3. The components should then be purged by use of factory air and all valves should be cycled as in item (2) above.
4. The components should then be placed in 130°F ± 5°F, 100 percent relative humidity environment, and baked for 1 hour. At the conclusion of each bake period, the internal surfaces should be flushed with clear water and valves should be simultaneously cycled as in item (2) above. A functional check should then be conducted to determine if a malfunction or degradation has occurred.
5. Items (2) through (4) constitute one cycle. All components should be cycled as follows:
(a) Ten cycles with a 1-hour bake period.
(b) Ten cycles with a 2-hour bake period.
(c) Ten cycles with a 5-hour bake period.

Each component should be disassembled and inspected at completion of each ten cycles. Any evidence of corrosion, damage, malfunction should be considered failure of the test.

b. Freezing Condensate. All pneumatically operated components of the subsystem should be subjected to the following test procedure:

1. Components should be oriented in the same attitude as they will be installed in the air vehicle.
2. The pneumatic components should be connected to an air source with a specific humidity of 154 grains of water/pound of d-y air, and all valves should be cycled five times from closed to open to closed.
3. Immediately after conclusion of step (2), the components should be de-pressurized and de-energized and placed in a cold chamber for 1 hour at 0°F or lower until the entire unit has stabilized at the low temperature.
4. At conclusion of step (3) and with components still in the cold chamber at 0°F, the components should be subjected to a functional tests to determine that all components perform satisfactorily.

7-4.5.3.27.2 Insulation. Laboratory tests should demonstrate that insulation protects personnel and critical components as required.

7-4.5.3.27.3 Air Cycle Machines (ACM). Laboratory tests should demonstrate the following design features and performance of the ACM:

Feature	Test Method
(1) Replaceable Housings	Demonstration
(2) Replaceable Turbine Nozzle	Demonstration
(3) Replaceable Bearings	Demonstration
(3) Erosion Resistant Nozzle*	Test
(4) Erosion Resistant Fan/Compressor**	Test

(5) RPM Measurement	Demonstration
(6) Bearing Temperature Measurement	Test
(7) Bearing Cooling Flow Accountability***	Test
(8) Performance	Test
(9) Shaft Balancing	Demonstration
(10) Endurance	Test
(11) Proof and Burst	Test
(12) Overspeed Spin	Test
(13) Turbine,Fan,Compressor Containment	Test
(14) Vibration	Test
(15) Mechanical Shock	Test

* These tests are not necessary if the ECS has a built in filtering system which is used to clean the air provided to the turbine and if the air provided to the turbine results in no condensation in the turbine nozzle section, but the filtration system should be tested per the guideline of paragraphs 7-4.3.5.11 and 7-4.3.5.13. (Note: Condensed water in the turbine nozzle section can cause erosion if untreated aluminum is used for construction.)

** The fan/compressor erosion test is not necessary if a ram air/bleed air filtration system is used, but the filtration system should be tested for its effectiveness.

*** The objective of the bearing cooling flow accountability test is to show that NBC contamination of the cockpit and avionics supply air does not occur as a result of air bypassing the NBC filter either in the normal heating and cooling or emergency backup cooling modes. A coupon test should also be conducted on the bearing materials to demonstrate that the specified chemical agents do not adversely affect the mechanical integrity and reliability of the bearings. This test should be conducted at temperatures equal to the highest observed for the bearings.

All indicated tests should demonstrate that the ACM can achieve its reliability allocation requirement as established by the contractor for the service life of the aircraft or until the designated overhaul period occurs. The contractor should establish the sand and dust concentration, particle distribution size, and frequency and duration of the challenge for the erosion testing based on the mission and expected location of operations.

7-4.5.3.27.4 Controls. A laboratory system test using the production configuration control software should demonstrate that the ECS control system functions without distracting the crew, displays stable characteristics, and controls all ECS functions safely and as required. The laboratory ECS rig should test all revisions to the ECS software affecting the control functions. A software development simulator for the ECS hardware may be used instead of the test rig if the Contractor proves that the simulator response characteristics match those of the ECS hardware, and the simulator incorporates all the required functions. The test rig should incorporate interface data busses to simulate control data transfer between systems (or within the ECS) or provide this data to the controller with the specified protocol. (e.g., A digitized temperature signal from the fuel system may be used by the ECS controller. This feedback information should be provided to the controller at the contractor specified frequency to simulate actual system performance). Aircraft flight tests should demonstrate that the installed ECS functions as required and that crew compartment control settings vary linearly with the controlled parameter.

7.4.5.3.27.5 Heat Exchangers. Laboratory tests should demonstrate the following guidelines are met for all heat exchangers:

(1) proof and burst pressure: The contractor should establish the heat exchanger’s worst case operating pressure and temperature points based on system analysis and use the factors discussed in 7-3.5.3.27.5 to establish the proof and burst limits. If subsequent laboratory system testing or a revised validated performance model reveals that the actual operating condition is more severe than initially predicted, the component should be re-tested to the actual operating or newly predicted condition or the Contractor should provide a safety assessment prior to lowering the requirement.

(2) vibration and mechanical shock
(3) pressure drop (all sides)
(4) thermal efficiency
(5) thermal bypass functionality as required
(6) leakage

A heat exchanger leakage test should demonstrate mechanical integrity at proof pressure levels after completion of vibration, shock, and endurance tests. This test is required for all heat exchangers. The heat exchanger should be submerged in water, oriented in a manner that would reveal the formation of bubbles, and pressurized to its proof level. The test technician should watch for bubble formation in the heat exchanger for a 5 minute period. No discernible bubbles should be permitted for those heat exchangers, that if failed, could contaminate the crew or equipment with NBC or other toxic agents. If the heat exchanger cannot be oriented to reveal bubbles, then the header or a portion of the header should be removed to render bubble observance possible. The contractor should establish leakage requirements and conduct the test described above for non-contamination critical heat exchangers. For those heat exchangers that are susceptible to icing because of entrained water in the bleed air, the contractor should conduct sufficient laboratory tests to show that the control system either averts or controls ice development on the heat exchanger core. For either case, testing should show that the control system operates in a manner that requires no human intervention during ice formation or prevention and that resultant cooling and/or heating supply temperatures do not change by more than __°F or flowrates change by more than __ lbm/min. If heat exchangers are susceptible to clogging due to foreign object ingestion, the Contractor should conduct a test to determine the severity of the problem. This includes heat exchangers using water spray for added cooling. With water spray occurring the Contractor should inject dirt at a rate that simulates "brown out" field conditions. Changes in system performance should be observed and extrapolated as required to determine the allowable number of hours of operation in such a condition before the system must be cleaned.

7-4.5.3.27.6 Couplings. A laboratory test should determine the suitability of couplings. This includes high temperature, high pressure bleed air and refrigeration pack applications, and it applies to metal couplings with a latching mechanism as well as to composite material type couplings that use "radiator hose type" clamps for fastening. Additionally, the contractor should submit coupling designs for review at CDR and indicate their previous usage. The test should simulate the coupling's application by connecting two duct sections. The Contractor should pressurize the system to proof levels at the required operating temperature and at the same time impose the required vibration environment. The test should compare the allowable leakage as determined by the Contractor to observed leakage. To demonstrate service life integrity the Contractor should devise an accelerated test, conduct it, and repeat the leakage test described above. The accelerated test may include elevating the temperature environment, vibration environment, pressure environment, stressing the ducting and couplings by deformation, or other suitable method determined by the Contractor. A leak test on aircraft ducting should demonstrate coupling integrity after installation, but before flight. This test should include the bleed air system, the refrigeration pack, and the distribution system. The Contractor should pressurize the ducting to proof levels and examine the joints for leakage by applying a soapy solution or by conducting an internally developed Standard Operating Procedure. This test should demonstrate that the total leakage allowed as determined by the Contractor is not exceeded.

Rationale. Couplings are critical for safety and optimum ECS performance. Significant leaks in the bleed air system can cause fires, explosions, burnt insulation and subsequent loss of serviced subsystems, and structural damage if the leak pressurizes the compartment. Leaks in the distribution system are more benign with respect to safety, but avionics and cockpit cooling will still suffer. Excessive leakage can render cooling performance inadequate to support flight, and significant flight schedule delays may result because of poor access to ducts and couplings and lengthy repair times. Resolving these problems during manufacturing is more efficient than discovering and resolving them during the flight test program.

7-4.5.3.27.7 Ducting Flame Resistance. If ducting passes through a fire zone or can otherwise be exposed to a fire, a laboratory test should demonstrate that distribution ducting will not allow a fire to propagate. Previous tests that demonstrate this requirement for the material used or Contractor assessments that demonstrate fires can not originate at the Equipment source should negate the need for this test. However, the Contractor should submit the results of previous testing and demonstrate its applicability to the current program and also submit evidence proving that fires can not originate at the equipment source for Government acceptance.

7-4.5.3.27.8 Fans. Laboratory containment tests should be conducted on fans. These tests should induce blade failures at the blade hub with 130% overspeed conditions, and the containment ring should prevent the penetration of blade fragments.

Rationale. Containment is a safety issue which must be demonstrated by tests or detailed analysis.

7-4.5.3.27.9 Pressurization Controls.

7-4.5.3.27.10 Liquid Cooling Loops - Level Indication.

7-4.5.3.27.11 Liquid Cooling Loops - Disconnects. Laboratory tests should demonstrate that quick disconnects do not leak more than the Contractor specified amount during connection, during disconnection, during use, or during the disconnection period. These tests should demonstrate acceptable leakage in the pressurized and unpressurized states and under the applications operating environment for pressure, temperature, and vibration simultaneously.

Rationale. These tests are necessary to show that maintenance problems will not occur as a result of leakage. These may include cleanup of spilled coolant on avionics, hazards to personnel, or air infiltration to the coolant system.

7-4.5.3.28 Design and Construction. Laboratory tests should demonstrate that materials used are appropriate for the given application.

7-4.5.3.29 Ground Provisions. Ground testing for duct and coupling leakage, ECS performance with engines and ground cart, cockpit pressurization, avionics coolant distribution, etc. should demonstrate that adequate ground test provisions exist.

SECTION 8: HYDRAULIC SYSTEM

8-1.0 REQUIREMENTS.

8-1.1 PERFORMANCE. The Hydraulic system shall meet its allocated performance and function in such a manner that the aircraft shall be able to be operated safely throughout the operating envelope and meet the performance requirements as defined in the applicable weapon system specification.

8-1.2 QUALIFICATION. The following qualification requirements for the Hydraulic system are required to verify compliance with the performance requirements above as applicable to the aircraft design configuration.

8-1.2.1 Analysis. System design and performance analysis shall be documented, using ADS-9C and the criteria below as a guide. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

8-1.2.2 Component Tests. The following tests shall be conducted and a subsequent teardown inspection, to determine the post test condition, shall be performed. Consult the applicable Contract Data Requirements List (CDRL) for submittal requirements.

a. Proof and Burst Pressure
b. Overspeed
c. Leakage
d. Pressure Drop
e. Temperature Extremes
f. Cycling/Endurance
g. Impulse
h. Vibration/Shock
i. Environmental
j. Dielectric Strength
k. Electromagnetic Interference

8-1.2.3 Assembly/System Level Tests. Aircraft system level tests shall be conducted in accordance with ADS-1B-PRF.

8-2.0 REFERENCES.

8-2.1 Commercial Specifications.

NAS 1638	Cleanliness Requirements Of Parts Used In Hydraulic Systems
SAE ARP 584	Coiled Tubing, Corrosion Resistant Steel, Hydraulic Applications
SAE ARP 603	Impulse Testing Of Hydraulic Hose, Tubing And Fitting Assemblies
SAE ARP 1281	Actuators: Aircraft Flight Controls, Power Operated, Hydraulic, General Specification For
SAE ARP 1383	Impulse Testing Of Aerospace Hydraulic Actuators, Valves, Pressure Containers, And Similar Fluid System Components

8-2.2 Federal Specifications.

FED-STD-791	Lubricants, Liquid Fuels, And Related Products; Methods of Testing

8-2.3 Military Specifications.

ADS-1B-PRF	Rotorcraft Propulsion System Airworthiness Qualification Requirements
ADS-9C	Propulsion System Technical Data
MIL-H-5606	Hydraulic Fluid, Petroleum Base; Aircraft, Missile And Ordnance
MIL-H-6083	Hydraulic Fluid, Petroleum Base, For Preservation And Operation
MIL-H-83282	Hydraulic Fluid, Fire Resistant, Synthetic Hydrocarbon Base, Aircraft, Metric, NATO Code Number H-537
MIL-STD-461	Control Of Electromagnetic Interference Emissions And Susceptibility, Requirement For The
MIL-STD-462	Electromagnetic Interference Characteristics, Measurement Of
MIL-STD-463	Definitions And Systems Of Units, Electromagnetic Interference And Electromagnetic Compatibility Technology
MIL-STD-810	Environmental Engineering Considerations and Laboratory Test, Test Method Standard For

8-2.3 International Standards.

STANAG 3212	Diameters for Gravity Filling Orifices

8-3.0 **PERFORMANCE GUIDELINES**.

8-3.1 **Fluid.** Hydraulic systems and associated ground equipment should be designed to use fluid conforming to MIL-H-5606 or MIL-H-83282 as these are the standard fluids in the DOD inventory.

8-3.2 **General System Design.** The hydraulic systems and components thereof should be designed to operate satisfactorily under all conditions that the aircraft may encounter within the structural limitations of the aircraft, including forces or conditions caused by acceleration, deceleration, zero gravity, negative g, or any flight attitudes obtainable with the aircraft, structural deflection, vibration, or other environmental conditions. The hydraulic system(s) should be configured such that any two fluid system failures due to combat or other damage which cause loss of fluid or pressure will not result in complete loss of flight control. For rotary-wing aircraft, the surviving system(s) should provide sufficient control for return to the intended landing area (including shipboard areas and land). Hydraulic systems should be as simple and foolproof as possible and in accordance with design, operation, inspection, and maintenance objectives specified in the aircraft design requirements.

8-3.2.1 Fluid Temperature Limitations. The hydraulic system(s) should be capable of operating under any condition encountered within the operating envelope, including climatic extremes, without exceeding a bulk fluid temperature limit in any portion of the system(s) of 135° C (275° F). Operation at these temperatures should not result in any degradation of system(s) or component performance.

8-3.2.1.1 Climatic Extremes. Ground operation, flight operation and storage climatic extremes should be in accordance with the weapon system procurement specification.

8-3.2.2 Fire Hazards. The hydraulic system should be integrated with other systems that will eliminate or isolate the system(s) from fire hazards caused by proximity of combustible gases, heat sources, bleed-air ducts or electrical equipment, etc. Hydraulic lines and equipment located in the vicinity of heat and ignition sources that will cause spontaneous ignition or sustained fire of hydraulic leakage from these lines or equipment should be protected by devices such as firewalls, shrouds, or equivalent means that will prevent fluid ignition.

8-3.2.3 Strength.

8-3.2.3.1 Additional Loads. All hydraulic systems and components which are subjected, during operation of the aircraft, to structural or other loads which are not of hydraulic origin should withstand such loads when applied simultaneously with appropriate proof pressure without exceeding the yield point at the maximum operating temperature.

Rationale. Components should be capable of withstanding not only the hydraulic pressure loads, but the loads the aircraft will put on the component in addition to the hydraulic pressure loads. For example, the rotor actuators will see not only hydraulic pressure loads, but loads introduced from the rotor forces.

8-3.2.3.2 Accelerated Loads. Actuating cylinders and other components and their attaching lines and fittings, subject to accelerated loads, should be designed and tested on the basis of a pressure equal to the maximum pressure that will be developed, without exceeding the yield point at the maximum, operating temperature.

8-3.2.4 Pressure Limitations.

8-3.2.4.1 System Pressures. System proof and burst pressures should be in accordance with Tables I and II. Peak pressure resulting from any phase of the system operation should not exceed 135 percent of the main system, subsystem, or return system pressure for normal operation. Lines, fittings, and equipment in return circuits should be designed for one-half the nominal system pressure.

8-3.2.4.2 Back Pressure. The system should be so designed that proper functioning of any unit will not be affected by the back pressure or changes in the back pressure in the system. The system or systems should also be so designed that malfunctioning of any unit in the system will not render any other subsystem, emergency system, or alternate system inoperative because of back pressure.

8-3.2.4.3 Brakes. Back pressure resulting from the operation of any unit while the aircraft is on the ground should create no greater back pressure at the brake valve return port than 90 percent of that pressure which will cause contact of braking surfaces. In addition, supply pressure to the brake system should not drop below the maximum brake-operating pressure during the operation of any other subsystem in the aircraft during taxiing, landing, or takeoff.

8-3.2.4.4 Pressure Regulation. Systems employing power-operated pumps should utilize a pressure-regulating device and an independent means of limiting excessive pressure. When the pump-driving mechanism is in continuous operation, such as engine or transmission drives, a variable-displacement pump should be used. When the pump is driven by an electric motor, a pressure switch may be used to deactivate the electric motor as the primary method of pressure regulation. In any case, an independent safety relief valve should be provided.

Typical Proof Pressure (minimum)

Characteristics	**Proof Pressure**	**Percent system Pressure**	**Remarks**
a. Lines, fittings and hoses	6,000	200	Proof pressure values for hose to be in accordance with the applicable detail specification
b. Components containing air and fluid under pressure	6,000	200	
c. Pump suction and case drain line components and reservoir			
(1) Non-pressurized reservoirs	50	---	
(2) Bootstrap reservoirs	---		150% of reservoir operating pressure
(3) Gas pressurized reservoirs	---		200 % of reservoir operating pressure
d. Components under system pressure only and pressure circuits (including lines, fittings and hoses)	4,500	150	
e. Components under return pressure only and return circuits (including lines, fittings and hoses)	2,250	75	Except hose, which should be 125% of nominal system pressure

Table I

Typical Burst Pressure (minimum)

Characteristics	**Burst Pressure**	**Percent system Pressure**	**Remarks**
a. Lines, fittings and hoses	12,000	400	Proof pressure values for hose to be in accordance with the applicable detail specification
b. Components containing air and fluid under pressure	12,000	400	
c. Pump suction and case drain line components and reservoir			
(1) Non-pressurized reservoirs	100	---	
(2) Bootstrap reservoirs	---		300% of reservoir operating pressure
(3) Gas pressurized reservoirs	---		400 % of reservoir operating pressure
d. Components under system pressure only and pressure circuits (including lines, fittings and hoses)	7,500	250	
e. Components under return pressure only and return circuits (including lines, fittings and hoses)	4,500	150	Except hose, which should be 250% of nominal system pressure

Table II

8-3.2.5 Reservoir Pressurization. Systems should be designed so that air does not contact the fluid during the normal function of the system(s). The reservoir pressure should be adequate to prevent cavitation at the inlet to the pump under all operating conditions.

Rationale. Because of the problems caused by gas entrapped in the circulating loop of aircraft hydraulic systems, separated reservoirs are preferred.

8-3.2.5.1 Reservoir Bootstrap Pressurization. Reservoir pressurization should be maintained in the event normal system pressure (reservoir bootstrap pressure) is lost.

8-3.2.6 Fluid Flow Effects. The systems should be so designed that malfunctioning of any unit or subsystem will not occur because of reduced flow, such as created by single-pump operation of a multi-pump system, or reduced engine speed. The systems should be designed so that increased flow will not adversely affect the proper functioning of any unit/subsystems, e.g. increased flow rate caused by accumulator operation or units affected by airloads.

8-3.2.7 Subsystem Isolation. Two or more subsystems pressurized by a common pressure source, one of which is essential to flight operation and the other not essential, should be so isolated that the system essential to flight operation will not be affected by any damage to the nonessential system.

8-3.2.8 Ground Test Provisions. Each hydraulic system should include a set of self-sealing couplings for attachment of ground test equipment. System ground test provisions should be so designed that pressurization of any hydraulic system in the aircraft is not necessary in order to test another hydraulic system. In particular, use of only one hydraulic test stand should be necessary to test the system, without use of Y connections between the test stand and the aircraft or use of a second ground test stand connected to another hydraulic system in the aircraft. (For filtration criteria, see 8-3.11.9.) A central ground servicing station should be provided for each system that includes connections for attachment of ground test equipment for system checkout and flushing, reservoir bleeding/filling, and accumulator air-nitrogen charging.

8-3.2.8.1 Ground Test Connections. A set of self-sealing couplings consisting of bulkhead halves and protective caps should be provided at a convenient location in the aircraft, easily accessible from the ground, for attachment of ground test equipment. The ground connections should be compatible with those connections supplied on ground test units in use by the procuring activity. Electric-motor-driven pumps used in emergency or auxiliary systems should not be used for ground test purposes unless the motor is designed for continuous operation.

8-3.2.8.1.1 Reservoir Supercharging Connection. When reservoirs are normally pressurized by either compressed air or nitrogen, a ground supercharging connection should be provided and should consist of a fitting end for attachment to a ground supercharging unit. A protective cap with a safety chain should be provided to protect the end connection when not in use.

8-3.2.8.1.2 Reservoir Filling Connection. Reservoirs should be filled by low-pressure replenishment methods.

8-3.2.8.2 Ground Test Data. The following data should be attached in a permanent manner on the aircraft near the ground test connections:

Set ground test reservoir pressurizing valve to_____ psi. 1/
Set ground test stand relief valve to_____psi. 1/
Set ground test stand volume output to_____gpm. 1/
Set ground test stand pressure compensator to_____psi. 1/
Use hydraulic fluid conforming to_____. 1/
Ground test stand output filter should be_____microns absolute. 1/
(Any other precautions or information considered necessary.)

1/ The contractor should fill in these values.

Caution note: The aircraft manufacturer should provide a caution note stipulating the maximum safe flow for landing-gear retraction checks during the time the aircraft is on jacks.

8-3.2.9 Removal of Entrapped Air. Suitable means, such as bleeder valves, should be provided for removal of entrapped air where it interferes with the proper functioning of the hydraulic system. Disconnection of lines or loosening of tubing nuts does not constitute suitable means. Equipment and system configuration should, insofar as practicable, be designed to automatically scavenge free air to a reservoir or other collection points where operation will not be affected and where release can be conveniently accomplished. (See 8-3.11.4.)

8-3.2.9.1 System Air Tolerance. The system should be designed and configured such that the presence of entrapped air should not cause sustained loss of system pressure or degradation of system performance during all conditions of intended aircraft operation.

8-3.2.10 Power Pumps. The hydraulic system pump(s) should be compatible with the installed aircraft system and should not cause abnormal or undesirable effects on the installed system in the aircraft. All pumps qualified for a given application should be physically and functionally interchangeable and should be compatible with the system and with each other allow mixed use in multiple pump systems.

8-3.2.10.1 Emergency Power Pumps. Hydraulic pumps required to provide emergency power for direct application to flight controls or other essential hydraulic flight requirements should not be used for any other function.

8-3.2.10.2 Multiple Pumps, Engine-Driven. Multiple engine aircraft hydraulic systems using engine-driven pumps should have pumps driven by at least two engines. A sufficient number of engine-driven pumps, augmented where necessary by pumps driven from other sources of power (e.g., electric motors, auxiliary power units, ram-air turbine, or pneumatic drives), should be provided to assure operation of control surface boost or power systems with any minimum combination of engines which will maintain flight and to assure operation of power brake systems and any other services needed during taxiing with any minimum combination of engines which may be used for taxiing.

8-3.2.10.3 Pump Pulsation. For all power-generating components (engine pumps, power packages, transfer units, etc.), pump pulsation’s should be controlled to a level which does not adversely affect the aircraft system tubing, components, and supports installation. The contractor should determine by test the effect of pump pulsation’s (pump ripple) on the hydraulic system.

8-3.2.10.4 Pump Rotation Reversal. For equipment not designed to withstand reverse rotation, the system and components should be designed so that no single failure will permit reverse rotation.

Rationale: On the CH-47, a check valve which isolates the flight control pump from PTU (Power Transfer Unit) pressure failed, causing the outlet port of the pump to be exposed to PTU pressure. Depending on the rate of leakage, this could run the pump like a motor, generating enough torque to cause it to rotate in the opposite direction.

8-3.2.11 Pump Supply Shutoff Valves. Pump supply (suction) shutoff valves should be provided if the fire protection requirements of the particular model aircraft specify the need for such equipment in other systems, such as fuel or lubricating oil systems, or both. These valves, when required, should not be located on the engine side of firewalls or flame-tight diaphragms but should be located as close as practicable to these members. However, the valves should be so removed from the engine that the loss of the engine from the attaching structure will not impair the operation of the valve. These valves should be operable from the cockpit, to both the closed and open positions.

8-3.2.12 Special Tools. Hydraulic systems should be so designed that special tools will not be required for installation or removal of components unless it can be shown that use of special tools is unavoidable.

Rationale: Weapon systems have much better maintainability in the field when special tools are not required.

8-3.2.13 System Pressure Indication. Pressure indicating equipment acceptable to the procuring activity should be provided to indicate the system pressure in hydraulic systems or subsystems. On engine-driven multi-pump systems, pressure indicating equipment should be provided for each pump to enable the flight crew to check for proper operation of each pump without shutdown of any engine. The pressure indicators should be so located as to be readily visible by the flight crew.

8-3.2.13.1 System Low-Pressure Warning Light. In addition, but not as a substitute for the requirement of 8-3.2.13, a suitable warning light should be installed in the cockpit in a conspicuous location to warn the pilot of low hydraulic system pressure. The light should be actuated by a pressure switch in the system. There should be a separate warning light for each hydraulic system. The warning light, or lights, should not be actuated by any combination of flight-control operations under normal operations. A momentary flicker of the warning light during ground checkout only is permissible, provided such condition is adequately described in the appropriate aircraft operation and maintenance manuals and provided such condition does not occur during flight unless a system malfunction exists.

Rationale: Flight crews need a quick visual indication of low hydraulic system pressure so precautionary measures can be taken in a timely manner.

8-3.2.13.2 Maintenance Check Gages and Indicators. Pressure gages/indicators that require a preflight, postflight, or daily check should not require work-stands or platforms in order to read the gages or indicators.

8-3.3 **Utility System Design.** All hydraulically operated services (excluding flight controls covered by 8-3.2) that are essential to the accomplishment of the basic aircraft mission (bomb-bay doors, in-flight refueling, etc.) or essential to land and stop the aircraft (landing gear, brakes) should have provisions for emergency actuation. No single failure of the utility system should result in loss of the aircraft. Wheel brake systems should perform to the requirements of the system specification.

8-3.4 **Flight-Control System Design.** In dual flight-control systems, both systems should be so designed that a ground test stand may be connected to either one of the flight-control systems and that system may be operated without adverse effect on the dead system, such as overflow of the system or failure of any part thereof. In order to accomplish this objective, automatic bypass of the fluid in the dead system from one side of the actuator to the other side should be provided.

8-3.4.1 System Isolation. Whenever hydraulic power is required for primary flight controls, a completely separate, integral, and adequate hydraulic system should be provided to supply only the primary flight controls. This hydraulic system should not be used to supply any other system or component in the aircraft. This hydraulic system should be as simple as practicable and should contain a minimum number of components. Dual actuator systems may employ the combined flight-control/utility system for one-half of the power, in which case the flight-control system should be given pressure priority. In addition, the combined flight-control/utility system should be so designed that the portions of the system required for operation only during the takeoff and landing phases of flight (e.g., landing gear or wing flaps) may be isolated from the rest of the system by means of a suitable shutoff valve in the pressure line, controllable from the cockpit, and check valves in the return lines so located that a rupture in any portion of the utility system will not cause loss of fluid from the reservoir when the system isolation valve is closed. When isolation valves are used in a combined flight-control/utility system to isolate nonessential flight functions, the system should be designed to preclude inadvertent isolation, during taxi or ground operation, that would result in loss of wheel braking, nose-gear steering, or other critical functions.

8-3.4.2 Hydraulic Power Failure. In aircraft where direct mechanical control sufficient to maintain emergency aircraft control cannot be accomplished following hydraulic power failure, an emergency power source should be provided, supplying controllability requirements.

8-3.4.2.1 Emergency System Application. The means of engaging the emergency power system should be either manual or automatic; however, they should be of the simplest and most reliable nature possible consistent with the requirements of the aircraft. Manual engagement is preferred, when suitable. Automatic engagement of the emergency power system should not be used. If the aircraft has a single engine, the emergency power source should be independent of the operation of this engine. On multiple-engine aircraft, the emergency source of power should be on a different engine than the primary source of power. In some cases, it is permissible to utilize the utility hydraulic system as the emergency source of power, if it is accomplished in such a way that there is no interconnection with the flight-control power system and no single failure can cause loss of both systems. Consideration should be given to the possibility of out-of-fuel landings wherein none of the engines are operating. Inasmuch as some turbojet engines will not windmill enough to provide adequate flight-control power supply during landing, it may be necessary to provide emergency power sources not dependent upon engine operation. In aircraft which are capable of being landed without engine power, this condition should not be considered an emergency, and provisions should be made for landings with one of the power systems failed while out of fuel. When designing for this condition, extreme care must be exercised not to reduce the reliability of the power systems. It should always be possible to reengage the flight controls or return them to normal following operation of the emergency hydraulic system, and where a ram-air turbine is used as the source of emergency power, it should be capable of extension, operation, and retraction under any flight conditions.

8-3.4.2.2 Disengagement and Bypassing. Where direct mechanical control is utilized following primary hydraulic system failure, or as made necessary by other system design conditions, provisions should be made for automatic, direct bypassing of the fluid from one side of the primary flight-control actuator to the other. Where the actuator can be disengaged from the system, bypassing will not be required. For dual actuator systems, where necessary, both systems should provide automatic bypass.

Rationale: Fluid must be able to be bypassed from one side of the actuator to the other without hydraulic pressure to prevent actuator “lockup”.

8-3.4.3 System Separation. Where duplicate hydraulic systems are provided, these systems should be separated as far as possible to obtain the maximum advantage of the dual system with regard to vulnerability from gunfire or engine fires. Where practicable, dual systems should be on opposite sides of the fuselage, the wing spar, or similarly separated. In any case, the systems necessary for safety of flight should be separated, preferably in a plane perpendicular to ground fire. Where it is deemed necessary, in a particular aircraft model, for these systems to come together, as in a dual tandem surface actuator, that actuator should be protected from the threat to a degree specified by the procuring activity. Adequate consideration should be given to the clearance between moving flight-control-system components and structure or other components to insure that no possible combination of temperature effects, airloads, or structural deflections can cause binding, rubbing, or jamming or any portion of the primary flight-control system.

8-3.4.4 System Pressure. Systems which use a pressure lower than the full system pressure should be designed to withstand and operate under the full pressure or should have an adequate relief valve installed downstream of the pressure reducing valve if the full pressure would be detrimental or dangerous. This relief valve may be incorporated into the same housing as the pressure reducer, provided that the relief valve mechanism is independent of the mechanism of the pressure reducer.

8-3.4.5 Power Sources. Aircraft primary flight-control hydraulic systems should have engine-driven pumps as their source of power, unless sufficient justification exists for using other power sources. Helicopter primary flight-control systems should have transmission-driven pumps as a source of power so that power will be available during the auto rotation.

8-3.4.6 System Temperature. The hydraulic flight control actuators should provide the required actuation rates under minimum and maximum in-flight fluid and ambient temperatures. The flight critical components should not bind or jam under any combination of in-flight fluid and ambient conditions including single failures, such as relief valves, worn pump, failed valves and other heat generating failures. The effects of differential fluid temperature in tandem units should also be demonstrated.

8-3.5 **Emergency System Design for Utility System**

8-3.5.1 Types. Where emergency devices are required in hydraulic systems, the emergency systems should be completely independent of the main system up to, but not necessarily including, the shuttle valve, the actuating cylinder, or motor. The system should be so designed that failure of an actuator in one subsystem should not prevent the operation of or cause the failure of both normal and emergency actuation of another subsystem. These emergency systems should utilize hydraulic fluid, compressed gas, gas-generating devices, direct mechanical connection, or gravity. Mechanical connections may include electromechanical units.

8-3.5.1.1 High-Lift Devices. Where safe operational landings cannot be accomplished without the use of hydraulically operated high-lift devices, they should be powered by dual actuator hydraulic systems or be provided with a suitable emergency system.

8-3.5.2 Emergency-Line Venting. The emergency line from the shuttle valves should be vented to the reservoir or to a low-pressure (15 psi gage maximum, above reservoir pressurization) nonsurging return line when the emergency system is not in use, except as authorized by the procuring activity. When shuttle valve leakage is not critical, the line may be vented to the atmosphere. After use of a compressed-gas emergency system, the system should be bled directly to the atmosphere rather than to the reservoir.

8-3.6 **Component Design.**

8-3.6.1 Standard Components. Standard components should be used in preference to nonstandard components wherever they will perform the function required by the system operating needs. Where no applicable standard component exists, a minimum size envelope compatible with the performance, installation, inspection, and maintenance requirements should be used.

8-3.6.2 Orifices. Orifices larger than 0.012 inch but smaller than 0.070 inch in diameter should be protected by adjacent strainer elements having screen openings 0.008 to 0.012 inch in diameter. Orifices smaller than the above range should be protected by adjacent elements having openings smaller than the orifices. Orifices and strainer elements should be strong enough to absorb system design flow and pressure without rupture or permanent deformation.

8-3.6.3 Actuators Essential to Safe Operation of the Aircraft. Where two or more independent hydraulic systems are utilized to power services essential for safe flight (e.g., primary flight controls), the actuation and control devices should be designed and constructed (either parallel or series configuration) so that no single structural or hydraulic failure will cause loss of more than one hydraulic system or allow transfer of fluid from one system to another. SAE ARP1281 should be used for design, performance, and test requirements for flight control system servoactuators.

8-3.6.4 Structural Strength. The components should have sufficient strength to withstand all loads or combinations of loads resulting from hydraulic pressure, vibration, temperature variations, actuation or operations, and torque loads for connection of tube fittings.

8-3.6.5 Reverse Installation. All components should be designed such that reverse or incorrect installation in the aircraft or sub-assembly cannot be made. Internal parts which are subject to malfunction or failure due to reversed or rotated assembly should be designed to render improper assembly impossible.

8-3.6.6 Metals. All metals should be compatible with the fluid and intended temperature, functional, service, and storage conditions to which the components will be exposed. The metals should possess adequate corrosion-resistant characteristics or be suitably protected to resist corrosion which may result from such conditions as dissimilar metal combinations, moisture, salt spray, and high temperature deterioration.

8-3.6.7 Threaded Parts. All threaded parts should be securely locked or safetied.

8-3.6.8 Pumps.

8-3.6.8.1 Power Pumps, Variable-Delivery and Fixed-Displacement. For variable-delivery pumps, the pressure differential between the pump-case cooling port and the reservoir should be such as to permit the pump to maintain adequate cooling flow in any pump flow condition.

8-3.6.8.2 Functional Guidelines.

8-3.6.8.2.1 Rated Case Drain Pressure. Rated case drain pressure is that maximum pressure at which the pump is required to operate in the system. Rated case drain pressure should be stated in the detail specification.

8-3.6.8.2.2 Case Drain Flow. At rated discharge pressure, rated inlet temperature, and at any speed from 50 to 100 percent of rated speed, the pump should be capable of producing a minimum case drain flow at a given maximum differential pressure between case pressure and inlet pressure as specified in the detail specification. Minimum and maximum case drain flow should be stated in the detail specification under conditions as specified in the detail specification.

8-3.6.8.2.3 Rated Temperature. The rated temperature of the pump should be the maximum continuous fluid temperature at the inlet port of the pump. It should be expressed in degrees Fahrenheit (°F).

8-3.6.8.2.4 Maximum System Temperature. The rated temperature of the pump should be 225°F or the value listed in the detail specification.

8-3.6.8.2.5 Minimum Fluid Temperature. The minimum continuous fluid temperature at the pump inlet port is not related to the rated temperature by this specification. A minimum continuous fluid temperature may be specified in the detail specification.

8-3.6.8.2.6 Maximum Displacement. The maximum displacement of the pump should be the maximum theoretical volume of hydraulic fluid delivered in one revolution of its drive shaft. It should be expressed as cubic inches per revolution (cu. in./rev). The maximum displacement of any pump should be determined by calculation from the geometry and dimensions of the pump. The effects of allowable manufacturing tolerances, of deflections of the pump structure, of compressibility of the hydraulic fluid, of internal leakage, and of temperature should be excluded from the calculation, because the maximum displacement is intended to be an index of the size of the pump rather than of its performance.

8-3.6.8.2.7 Rated Delivery. The rated delivery of the pump should be the measured output of the pump under conditions of rated temperature, rated speed, and maximum full-flow pressure, using the hydraulic fluid specified in the detail specification at rated inlet pressure. It should be expressed as compressed flow in U.S. gallons per minute (gpm), and its value specified in the detail specification.

8-3.6.8.2.8 Rated Speed. The rated speed of the pump should be the maximum speed at which the detail specification requires the pump to operate continuously at rated temperature and rated discharge pressure.

8-3.6.8.2.9 Rated Endurance. The value of the rated endurance should be specified in the detail specification and should be not less than 1000 hours.

8-3.6.8.3 Performance.

8-3.6.8.3.1 Torque and Heat Rejection. The performance requirements should be stated in the detail specification. The minimum performance requirements may be stated as maximum input torque at rated delivery and maximum heat rejection at rated discharge pressure.

8-3.6.8.3.2 Efficiency. Where the detail specification states a required minimum efficiency, it should be the ratio of output power to input power when the pump is operated at rated speed, and maximum full-flow pressure, using the hydraulic fluid specified in the detail specification. It should be stated as percentage. The above ratio is commonly referred to as "over-all efficiency" and includes volumetric efficiency. For the purposes of this specification, volumetric efficiency should not be segregated. In the determination of output power by calculation from flow rate and pressure change, only the net pressure difference between inlet and outlet ports of the pump should be used, and the flow rate may be as measured in the low pressure side of the discharge line at the option of the detail specification, provided adequate compensation is made for compressibility in calculating efficiency.

8-3.6.8.3.3 Pressure Pulsations. Pressure pulsations should be the oscillations of the discharge pressure, occurring during nominally steady operating conditions, at a frequency equal to or higher than the pump drive shaft speed. These pulsations should not exceed ±10 percent of rated discharge pressure or a pressure band specified in the detail specification under any condition, when the pump is tested in the circuit which simulates the actual system in which the pump is to be installed. The system volume may be simulated using tubing of the discharge line diameter being careful to avoid a line length whose natural frequency is resonant with pulsation frequency.

8-3.6.8.4 Variable Delivery Control. All pump models should incorporate delivery control means which should act to increase the delivery of the pump from zero to its maximum full-flow value for any given operating speed as the discharge pressure is reduced from rated discharge pressure to maximum full-flow pressure and vice versa.

8-3.6.8.4.1 Response Time. The response time of the pump should be the time interval between the instant when an increase (or decrease) in discharge pressure change initiates; and the subsequent instant when the discharge pressure reaches its first maximum (or minimum) value. The oscillographic trace of discharge pressure versus time should be employed as the criterion of movement of the delivery control mechanism. Pumps should have a response time specified in the detail specification.

8-3.6.8.4.2 Stability. The stability of the pump should be the freedom from persistent or quasi-persistent oscillation or "hunting" of the delivery control mechanism at any frequency that can be traced to the pump delivery control means. The oscillographic trace of discharge pressure versus time should be employed as the criterion of stability. The time required to recover steady-state operation after being disturbed should be specified in the detail specification.

8-3.6.8.4.3 Maximum Transient Pressure. The maximum transient pressure should be the peak value of the oscillographic trace of discharge pressure, made during operation of a pump. The value of the maximum transient pressure should not exceed 135 percent or rated discharge pressure or the maximum pressure defined in the detail specification.

8-3.6.8.4.4 Depressurization. When it is a requirement of the detail specification that the pump be depressurized either automatically or remotely as by an electrical signal the depressurization control should not when de-energized interfere with the normal operation of the variable delivery control.

8-3.6.8.5 Balance. The moving parts of the hydraulic pump should be inherently balanced and the pump should not vibrate in such a manner as to cause failure of any part in the pump or drive mechanism at speeds up to and including 125 percent of rated speed (maximum speed at which the detail specification requires the pump to operate continuously at rated temperature and discharge pressure).

8-3.6.8.6 Adjustment. Means should be provided to adjust the delivery control mechanism to cause zero flow to occur at rated discharge pressure. This adjustment should be preferably continuous, or acceptably in steps of less than 1 percent of rated discharge pressure, over a minimum range from 95 percent to 130 percent of rated discharge pressure. The adjustment means should be capable of being positively locked; and it should be possible to accomplish adjustment and locking by the application of standard hand tools. Where practicable, the arrangement of the adjustment means should permit adjustment to be made while operating under full system pressure with negligible loss of fluid.

8-3.6.8.7 Lubrication. The hydraulic pump should be self-lubricated with no provisions for lubrication other than the circulating fluid.

8-3.6.8.8 Drive Coupling. A replaceable part of the pump assembly, incorporating a shear section, should be interposed between the pump drive shaft and the engine accessory drive shaft by which the pump is to be driven. This shear coupling part should be held in place by a positive retainer. Both ends of the coupling should be designed to include a plastic spline bushing to minimize spline wear. A plastic spline bushing at only the gear box drive interface may be used if the plastic bushing is not adaptable to the pump interface.

8-3.6.8.9 Maintainability. All wear surfaces should be replaceable or repairable. Disconnects, mounting and wiring provisions should be designed to prevent erroneous connections.

8-3.6.8.10 Environmental. Pumps should not be affected by change of altitude, and should be capable of withstanding vibrations excited by the driving means. All pumps should be designed to withstand sustained accelerations applied in any direction and able to withstand continuous exposure, in the configuration as installed in aircraft, and either operating or non-operating, to salt spray as encountered in marine or coastal areas and to sand and dust as encountered in desert areas.

8-3.6.8.11 Structural Strength. The structural design of the ports and of the affected sections of the pump housing should be such as to withstand the application of a torque 2.5 times the maximum installation torque.

8-3.6.8.12 Markings. Inlet, outlet, and case drain ports should be identified on each pump in clear and permanent markings. In addition, the direction of rotation of the pump should be clearly and permanently marked on an exposed surface of the pump housing.

8-3.7 **Component Installation.** If a fluid such as MIL-H-6083, which is different from the specified fluid, is used for component testing and shipping, it should be drained prior to the installation of the component in the aircraft.

8-3.7.1 Design Practice and Installation. The hydraulic system component installation requirements specified in the following subparagraphs are considered to be representative of good design practice; however, it is recognized that variations from these practices will, in many cases, be necessary due to specific installation exigencies. All installation of standard parts or components should be designed to accommodate the worst dimensional and operational conditions permitted in the applicable part or component specification. All components should be installed and mounted to satisfactorily withstand all expected acceleration loads, wrench loads, vibration effects, etc.

8-3.7.2 Accumulators. Accumulators should be installed with the utmost consideration given to the protection of crewmembers in the case of rupture due to gunfire. Accumulators should be provided with a fluid port and a gas port. The accumulator should be provided with a suitable piston type separator to separate the fluid and gas within the accumulator. Accumulators should contain a safety provision to assure dissipation of the accumulator gas pressure and fluid pressure before any component parts can be disassembled. Each accumulator should be provided with a permanent, legible, attached warning in red letters stating:

MAXIMUM OPERATING PRESSURE ____PSI.

RELEASE GAS AND FLUID PRESSURE BEFORE DISASSEMBLING, STORING, OR SHIPPING ACCUMULATOR.

8-3.7.2.1 Measurement of Accumulator Gas Pressure. When accumulator gas charge is critical to the functioning of the hydraulic system or subsystem, a permanent pressure gage should be attached to the gas side of the accumulator. In non-critical installations, the use of the pressure gage should be at the discretion of the procuring activity. A gage indicating accumulator gas pressure should never be used to indicate equivalent hydraulic pressure to the crewmembers. Standard hydraulic gages should be used and should be attached with the shortest practicable length of line and minimum number of fittings.

8-3.7.2.2 Accumulator Accessibility. In all accumulator installations space should be provided around the gas charging valve for use of a high-pressure, gas-testing gage assembly and for standard fitting connections to charge accumulators.

8-3.7.2.3 Accumulator Instructions. Instructions for servicing the accumulator with gas pressure with the accumulator oil chamber discharged should be provided immediately adjacent to the accumulator. Adequate information should be included to indicate the proper gas preload pressure throughout the temperature range for which the accumulator will be serviced.

8-3.7.2.4 Gas Guidelines. Accumulators should be charged with inert gas only. All rotary-wing aircraft hydraulic system accumulators should be charged with either air or inert gas (nitrogen).

8-3.7.3 Actuating Cylinders. Hydraulic actuating cylinders should be so installed that they do not interfere with the adjacent structure and are readily accessible for maintenance and inspection. If possible, the cylinder should be installed in a protected area, or if exposed, be protected from flying debris during landing and takeoff. Bearings or bushings used in the actuators should be replaceable.

8-3.7.4 Bleeder Valves. Where required, bleeder valves should be so located that they can be operated without necessitating removal of other aircraft components. Such installations should permit attachment of a flexible hose so that fluid bled off may be directed into a container.

8-3.7.5 Directional Control Valves. The installation of directional control valves should be compatible with the control valve performance such that the system operation will not be affected by back pressure, interflow, or pressure surges which might tend to cause the valves to open or move from their setting or cause them to bypass fluid in other than the intended manner. No hydraulic control valves should be installed in the pilot’s cockpit or compartment.

8-3.7.5.1 Directional Control Valve Handle Installation. When the effective length of the directional control valve handle exceeds 0.8 inch, limiting stops, external to the valve, should be provided. These stops should be capable of withstanding 75r pound-inches limit load (where, r, equals the effective handle length) and should be so positioned that no load will be applied to the internal valve stops, unless the valve has been specifically designed to handle subject loads.

8-3.7.5.2 Multiple Control Valve Systems. In systems which incorporate two or more directional control valves, provision should be made to prevent fluid from being transferred inadvertently, at any possible valve setting, from the cylinder ports of one valve into the cylinder ports of another valve.

8-3.7.5.3 Control Valve Actuation. Control valve operation may be direct, such as push-pull rods or cable control, or indirect, such as electrically operated controls. Push-pull rods should require a minimum or no adjustment. Sheathed flexible controls should not be used. Cable control should be designed to provide minimum adjustment and positive control. All controls should be designed to prevent overtravel or undertravel of the valve control handle by use of external or internal stops. Electrically operated valves should be provided with mechanical override control mechanisms wherever practicable.

8-3.7.6 Filters. All vent openings or fluid exposed to breathing action through vents should be protected by filters. Line filters, when installed in the aircraft system in close proximity to an accumulator, should be installed upstream of the accumulator. Where pressure-drop indicators are provided on the filter assembly, the indicator should be easily visible to servicing personnel. In addition, replacement of the filter should be required before the indicator can be reset. When a screen or filter is provided either internally or in close proximity to a component, suitable provisions should be made for removal of the screen or filter for cleaning or replacement. Sintered-type elements should not be used.

8-3.7.6.1 Aircraft Filters. Filters should be provided in all hydraulic systems. Filters incorporating elements having absolute ratings lower (finer) than 15 microns may be used. These filters should be used to filter all circulating fluid in the system. The acceptable contamination level for aircraft delivery should be specified in the detail specification.

8-3.7.6.2 Filter Locations. Filters should be provided in the following locations as a minimum guideline.

8-3.7.6.2.1 Pressure Line Installation. A no-bypass-type line filter(s) should be installed in the system pressure line and should be so located that all fluid from the aircraft pump(s) and the ground test equipment pressure connection will be filtered prior to entering any major equipment or components of the system. In multi-pump systems, each pump should have a separate filter installation.

8-3.7.6.2.2 Return Line Installation. A bypass-type line filter should be installed in the return line. All fluid entering the return circuit should be circulated through the filter prior to entering the return line to the pump(s) and reservoir.

8-3.7.6.2.3 Reservoir Fill Line Installation. A no-bypass-type line filter should be installed to filter all fluid entering the system through the reservoir fill connection.

8-3.7.6.2.4 Pump-Case Drain Line Installation. Each pump-case drain (bypass) line should have a bypass-type filter installed.

8-3.7.6.2.5 Pump Suction Line. Filters should not be installed between the system reservoir and the pump suction port unless specifically authorized by the procuring activity.

8-3.7.6.3 Hydraulic Sequencing. Where hydraulic sequencing is critical, and where contamination can prevent proper sequencing, each sequence valve should be protected from contamination in each direction of flow by a suitable screen-type filter element. This element may be included as a part of the sequence valve assembly

8-3.7.7 Fittings. All tube fittings (other than connections at production break points, to removable components, and where needed to facilitate maintenance) should be permanently joined employing no screw threads or similar mechanical means. In addition, suitable repair and replacement methods involving failed tubing and fittings should be established for each aircraft model and should be included in the applicable aircraft publications. Removable components should accept standard fittings. No thread lubricant other than hydraulic fluid specified in 3.1 should be used on hydraulic fittings.

8-3.7.8 Flow Dividers. Flow dividers should not be used if the effect of a malfunction of the flow divider would result in an unsafe flight condition.

8-3.7.9 Flow Regulators. Flow regulators may be installed in the hydraulic system to limit the rate of fluid flow. The direction and rate of fluid flow should be clearly indicated on the flow regulator and the adjacent structure. Regulators used under continuous dynamic conditions should not adversely affect the operation of the hydraulic system.

8-3.7.10 Protective Devices. Hydraulic fuses, circuit breakers, reservoir level sensors, or other similar devices may be used to meet survivability requirements established by the procuring activity. Pre-mature or inadvertent shutoff or any other malfunction of such devices should not occur during any flow or pressure variations or any conditions of system operation. The function and reliability of such devices should be demonstrated in the functional mockup and simulator

8-3.7.11 Snubbers. Pressure snubbers should be used with all hydraulic pressure transmitters, hydraulic pressure switches, and hydraulic pressure gages. Pneumatic pressure gages are excluded from this requirement.

Rationale: Snubbers provide dampening for more accurate pressure readings.

8-3.7.12 Manually Operated Pumps. Where a manually operated pump is required, either a hand-actuated or foot-actuated pump should be selected, based on trade-off studies. In installations where the pump can be operated by personnel in a standing position, strong consideration should be given to a foot pump to minimize physical exertion.

8-3.7.12.1 Manually Operated Pump Suction Line. No screen or filter should be used in the suction line of the pump. The suction line should be of suitable diameter and length to insure priming a dry pump and obtaining full-rated flow at -54°C (-65°F) temperature with 12 complete cycles at a rate of 20 cycles per minute. The pump circuit should be capable of full priming and rated flow in flight at the highest altitude at which pump operation is essential and intended.

8-3.7.12.2 Manually Operated Pump Check Valve. A standard check valve should be provided in the pump pressure line.

8-3.7.12.3 Hand Pump Handle Length. The effective operating handle length of hand pumps should be such that the handle load should not exceed 67 pounds. The length of this handle travel at the handgrip should not exceed 18 inches.

8-3.7.13 Flexible Connections. Whenever relative motion exists between two points metal coiled tubing in accordance with ARP 584 is preferred.

8-3.7.14 Hose Assemblies. Hose assemblies should not be subjected to torsional deflection (twisting) when installed or during system actuation. No hose clamp type installation should be used in hydraulic systems.

8-3.7.14.1 Hose Support. The support of a flexible line should be such that it will never tend to cause deflection of the rigid lines under any possible relative motion that may occur. Flexible hose between two rigid connections may have excessive motion restrained where necessary but should never be rigidly supported as by a tight rigid clamp around the outside diameter of the flexible hose. Extreme care should be used in the selection and placement of the supports to assure the flexible line is not restricted and does not rub on structure or adjacent members during any portion of its excursion.

8-3.7.14.2 Hose Bend Radii. The minimum radius of bend of hose assembles should be a function of hose size and flexing range to which the hose installation will be subjected. The minimum bend radii for hoses should be as listed in the applicable hose specification.

8-3.7.14.3 Hose Protection. Hose should be suitably protected against chafing where necessary to preclude damage to the hose and to the adjoining structure, tubing, wiring, and other equipment. Hose assemblies including end fittings of amphibious aircraft which are subject to salt water immersion should be suitably protected.

8-3.7.14.4 Provisions For Hose Elongation and Contraction. Hose assemblies should be so selected and installed that elongation and contraction under pressure, within the hose specification limits, will not be detrimental to the installation either by causing strains on the end fittings or excessive binding or chafing of the hose.

8-3.7.15 Lock Valves. Where lock valves are used, provisions should he made for fluid expansion and contraction through-out the temperature range. Where several actuating cylinders are mechanically tied together, only one lock valve should be used to hydraulically lock all actuators so tied together.

8-3.7.16 Motors. All motors should be accessible for maintenance and inspection and should be located in an appropriately cool or warm space depending upon their service. Proper case overflow connections to the reservoir should be provided. Shaft seal drains should be vented overboard.

8-3.7.17.9 Electric-Motor-Driven Pumps. Electric-motor-driven hydraulic pumps may be used, as necessary, for either normal, emergency, or auxiliary operation of hydraulic systems.

8-3.7.18 Pressure Regulators (Unloading Valves). When an unloading valve is used, the return line to the reservoir should be as short as possible and should not be subjected to back pressure or pressure surges in excess of allowable values stated herein. The unloading valve should have the drain port piped directly to the reservoir or routed to some return line in which the maximum pressure does not exceed the reservoir pressurization and is not subjected to pressure surges. The tubing connecting the unloading valve should be so designed and installed that the pressure surges in the system will not affect the operation of the unloading valve at any flow rate of the system. In addition, provisions should be made in the system to eliminate any harmful shocks caused by pressure surges due to the operation of the unloading valve.

8-3.7.19 Relief Valves, System and Thermal Expansion. Relief valves may be incorporated as part of another unit. Relief valves are designed to be used as a safety device to prevent bursting of, or damage to, the system in the event the normal pressure regulation device in the system malfunctions; or, in blocked line condition, to relieve excessive pressure due to either thermal expansion of the fluid or overload forces on actuating units. Therefore, relief valves should not be used as the sole means of limiting pressure in a power circuit but should function only as a safety valve.

8-3.7.19.1 System Relief Valves. Provisions should be made to insure that pressure in any part of the power system will not exceed a safe limit above the cutout pressure of the hydraulic system. Pressure relief valves, as specified herein, should be located in the hydraulic system wherever necessary to accomplish this pressure relief. The system relief valve should have a capacity equal to or greater than the combined rate of flow of pumps where fixed-displacement pumps with common cutout regulators are used, or equal to or greater than the rated flow of the largest pump when variable-volume pumps with a common pressure line are used. The systems should be designed so that excessive temperature over 135°C (275°F) does not occur due to fluid flowing through a relief valve. As an alternative, a temperature warning and indication system may be used.

8-3.7.19.2 Thermal Expansion Relief Valves. Relief valves should be installed as necessary to prevent excessive pressure rise and system damage resulting from thermal expansion of hydraulic fluid. Internal valve leakage should not be considered an acceptable method of providing thermal relief.

8-3.7.20 Reservoirs. When a hydraulic emergency system is used in any military aircraft except trainer types, a separate emergency reservoir should be provided. The emergency reservoir should be located as remote as practicable from the main reservoir to minimize gunfire damage. It should be possible to fill the main and emergency reservoirs through a common filler port, unless the two reservoirs are so far distant as to make this guideline impracticable. The feed and vent lines in the two reservoirs should be so located that rupture of either of the reservoirs or the feedlines will not cause loss of sufficient fluid from the other reservoir to impair the system operation. Reservoirs should be suitably protected (i.e., return line relief valve) to prevent failure or damage when rapid discharge of the main or emergency system into the reservoirs is encountered. Installations of pressurized reservoirs should include a suitable depressurization valve for maintenance purposes.

8-3.7.20.1 Reservoir Location. It is desired that the reservoir should be so located that the following conditions will be obtained.

a. A static head of fluid will be supplied to the hand pump and the power-driven pump or pumps in all normal flight attitudes of the aircraft

b. The length of suction line to the pump is a minimum

c. The best available temperature and pressure is utilized, but must not be installed in engine compartments

d. Protection from combat damage

e. If practicable, suction lines should be so routed as to prevent breaking of the fluid column caused by gravity after engine shutdown and during the parking period. Where such routing is not possible, or where the reservoir cannot be located above the pump, suitable provisions should be installed to maintain the fluid column to the pump after engine shutdown. A swing type check valve in the suction port of the reservoir should normally maintain theca fluid column to the pump.

f. If routing of the pump bypass cannot be accomplished so that breaking of the fluid column by gravity after engine shutdown is prevented, check valves should be incorporated in the lines.

8-3.7.20.2 Reservoir Venting. If a vent is provided in the reservoir, it should be so arranged that loss of fluid will not occur through the vent during flight maneuvers or ground operations of the aircraft. A filter should be incorporated into the vent line if the temperature requirement is suitable. If a filler cap is used, the act of removing the filler cap should automatically vent the reservoir in such manner that the energy contained in the pressurizing air is not dissipated by imparting kinetic energy to either theca filler cap or the fluid contained in the reservoir or elsewhere in the system.

8-3.7.20.3 Gas-Pressurized Reservoirs. The air (or inert gas) pressure should be controlled by an externally nonadjustable pressure-regulating device to control the gas pressure in the reservoir. A relief valve should also be connected to the airspace to protect the reservoir and pump from excessive pressure. If the air pressure regulator and relief valve are combined into one housing, a single failure in that unit should not permit over-pressurization of the reservoir. Devices, such as aspirators, that introduce air into the hydraulic fluid should not be used. When the air is separated from the fluid by a piston or other device, operation of the system should not introduce air into the hydraulic fluid. Provision should be made to remove entrained air which may have entered the system during servicing or operation.

8-3.7.20.4 Reservoir Air Pressurization Moisture Removal Equipment. When engine bleed air is used for reservoir pressurization, a suitable moisture removal unit should be so located as to protect the pressure regulation lines and equipment. An adequate filter should be provided.

8-3.7.20.5 Reservoir Fluid Level Indication. Indicators should be incorporated in the reservoir design to automatically provide a continuous visual indication of reservoir fluid level. Temperature compensation for accurate fluid indication should be included in the design. The use of a dip stick, or the necessity for the manipulation or movement of any of the component parts in order to obtain the reading should not be acceptable. Reservoir fluid level indication should be provided in the cockpit. Indicator fluid level markings should correspond with the direct-reading fluid level indicator markings provided on the reservoir and should be lighted in accordance with applicable cockpit lighting requirements. A suitable warning light should also be provided to advise the pilot of a low fluid level condition. The cockpit fluid level indicator should not eliminate the requirement for the direct-reading fluid level indicator on the reservoir itself, as this is required for reservoir servicing with power OFF.

8-3.7.20.6 Reservoir Capacity. Hydraulic reservoirs should be designed to provide a minimum total available capacity sufficient to provide fluid for all combinations of system use, including but not limited to accumulator charging, thermal considerations, fluid compression, and maximum system usage.

8-3.7.20.6.1 Emergency Reserve Capacity. In those cases where an emergency reserve capacity of fluid is required, the minimum volume should be equal to 125 percent of the fluid required for the emergency purposes. The volumetric capacity of those subsystems which can be operated other than hydraulic fluid should not be considered, provided that no fluid is drawn from the reservoir during such operation. No allowances for return of fluid to the reservoir should be made in determining emergency reserve capacity. The emergency reserve capacity should be provided in a separate reservoir.

8-3.7.20.7 Filling and Refill. The reservoir design should be such that initial filling of the reservoir installed in the vehicle, and any subsequent refilling or replenishment of the fluid, should be accomplished only through the use of an external force device such as a hand pump or power pump. No provision should be incorporated which permits the transfer of hydraulic fluid into the reservoir by gravity flow from an external container. The reservoir filling ports should be so designed and installed at locations where it should prevent the above procedure. If a “fill module” is used for transfer of fluid into the reservoir, the gravity fill orifice should have a minimum internal diameter of 1.5 inches.

Rationale: NATO standardized agreement (STANAG) 3212 established a minimum internal diameter for hydraulic fluid gravity fill orifices.

8-3.7.20.8 Draining. The reservoir should contain provision for draining hydraulic fluid completely from the reservoir when the vehicle is at rest, without the necessity of disconnecting any tubing connections.

8-3.7.20.9 Bleeding. Reservoir design and installation should be such that if entrained has bubbles in the hydraulic fluid allowed to accumulate in any space in the reservoir, means should be provided for removing entrapped air in a bleeding operation without any tubing disconnection. Also, provisions should be provided in any re-circulating system reservoir for either automatic or periodic manual removal of any hydraulic fluid from any cavities in the reservoir in which fluid is not present by design, such as the gas side of a gas-pressurized piston type reservoir.

8-3.7.21 Restrictor Valves. Adjustable orifice restrictor valves may be used in experimental aircraft, but only fixed orifice restrictor valves should be used in service test and production aircraft. For one-way restrictors, the direction of restricted and unrestricted flow should be indicated on the restrictor valves and adjacent structure. (For orifice filtration requirements, see 8-3.6.2.)

8-3.7.22 Self-Sealing Couplings. Hydraulic systems should be provided with self-sealing couplings for each engine-driven pump and so located that the powerplant section can be readily removed for servicing. A suitable coupling should be used in each line going to each pump. Self-sealing couplings should also be provided on all hydraulically operated brake installations where it is necessary to disconnect the brake line in order to remove the wheel. The self-sealing coupling should be attached to the brake, and it should be possible to remove the wheel without damaging the coupling. Self-sealing couplings should also be provided at all other points in the hydraulic system which require frequent disassembly or, where convenient, to isolate parts of the system as in jacking and servicing one landing gear only. Sufficient clearance should be provided around the coupling to permit connection and disconnection. Self-sealing couplings installed adjacent to each other should be of different size or be otherwise designed that inadvertent cross connection of the lines cannot occur.

8-3.7.22.1 Airframe Break Points. When self-sealing couplings are provided at airframe break points, especially in flight-control systems, and where disconnection of such a coupling or couplings will adversely affect the operation of any of the systems, adequate means should be provided to prevent the inadvertent disconnection of the couplings. Such means should also provide adequate indication when a coupling connection is incomplete. If the means of preventing inadvertent disconnection are not absolutely positive, the system should be so designed that a hydraulic lock resulting from an inadvertent coupling disconnection will not be theca cause of an aircraft accident

8-3.7.23 Shuttle Valves. Shuttle valves should not be used in installations in which a force balance can be obtained on both inlet ports simultaneously which may cause the shuttle valve to restrict flow from the outlet port. Where shuttle valves are necessary to connect an actuating cylinder with the normal and emergency systems, the shuttle valve unit should be built into the appropriate cylinder head. Where the above installation cannot be made, a standard shuttle valve may be located at the actuator port. In the event neither of the above installations is possible, a length of rigid line is permissible between the cylinder port and the shuttle valve, provided that the rigid line and shuttle valve are firmly attached to the actuating cylinder. Flexible hose should not be used between the actuating cylinder port and the shuttle valve.

8-3.7.24 Pressure Switches. Pressure switches may be installed in hydraulic systems where the regulation of hydraulic pressure is required by controlling an electric-motor-driven pump or other applications. Adequate precautions should be taken to prevent chatter or cutoff.

8-3.7.25 Swivel Joints. Where lines or fittings are used to drive swivel joints, they should be adequately supported and should be of sufficient strength to insure a satisfactory operating installation.

8-3.7.26 Tubing.

8-3.7.26.1 Tubing Bends. Forcing, bending, or stretching of tubing to accomplish installation should not be permitted.

8-3.7.26.2 Installation of Small Size Tubing. If tubing in sizes smaller than 1/4-inch outside diameter (-4 size) is used in hydraulic systems, particular care should be taken to properly install, support, and protect it, and it must be shown that proper operation of the service in which such tubing is used will be possible at -54°C (-65°F) temperature.

8-3.7.26.3 Designed Motion in Tubing. Looped or straight aluminum-alloy tubing should not be used between two connections where there is designed relative motion. Coiled tube and torsion tube installations should be designed and installed in accordance with the data given in ARP 584. Rigid tubing end connections should not be used.

8-3.7.26.5 Tubing in Fire Hazard Areas. Aluminum tubing should not be used within powerplant compartments and at other locations where fires are likely to occur. Where separable tube fittings are required, they should be corrosion-resistant or carbon steel.

8-3.7.26.6 Tubing and Fitting Identification. All hydraulic fluid lines should be permanently marked. A sufficient number of hydraulic lines should be marked in conspicuous locations throughout the aircraft in order that each run of line may be traced. This marking should indicate the unit operated and the direction of flow, such as "LANDING GEAR UP —>" or "FLAPS DOWN —>". These markings should be repeated as often as necessary, particularly on lines entering and emerging from closed compartments, to facilitate maintenance work. Where fittings are located in members, such as bulkheads and webs, each fitting location should be identified (placarded) as to system function, using the same terminology as on its connecting line.

8-3.7.26.7 Tubing Supports. All hydraulic tubing should be supported from rigid structure by cushioned steel clamps or by suitable multiple-tube block-type clamps. Supports should be placed as near as practicable to bends to minimize overhang of the tube. Where tubes of different diameter are connected together, an average spacing distance may be used. In any event, the installation should be in compliance with 8-3.7.28.6 and 8-3.7.28.7. Provisions should be made in support locations to accommodate change in tubing length caused by expansion and contraction.

8-3.7.26.8 Location of Hydraulic Tubing. Hydraulic lines should not be installed in the cockpit or cabin and should be remote from personnel stations. In addition, hydraulic lines should be located remotely from exhaust stacks and manifolds; electrical, radio, oxygen, and equipment lines; and insulating materials. In all cases, the hydraulic lines should be below the aforementioned to prevent fire from line leakage. Hydraulic lines should not be grouped with links carrying other flammable fluids in order to prevent inadvertent cross connection of different systems. Hydraulic drain and vent lines should exhaust in areas where the fluid will not be blown into the aircraft, collect in pools in the structure, or be blown onto or near exhaust stacks, manifolds, or other sources of heat. Tubing should be located so that damage will not occur due to being stepped on, used as handholds, or by manipulation of tools during maintenance. Components and lines should be so located that easy accessibility for inspection, adjustment, and repair is possible. Hydraulic tubing should not be used to provide support of other aircraft installations, such as wiring, other aircraft tubing, or similar installations. Attachment of so-called marriage clamps for spacing of such installations is likewise prohibited.

8-3.7.26.9 Tubing Flares and Assembly. When installing tube connections, care should be exercised to keep the wrench torque used to assemble each joint within the specified limits. Male threaded aluminum-alloy flared fittings should not be used with stainless-steel lines below size -8.

8-3.7.26.10 Tubing Pre-Stress. Each titanium tube assembly should be pre-stressed by the application of pressure to approximately 5 percent of the minimum yield strength. This test may be performed on the bench or in the aircraft.

8-3.7.27 Design of System Installations.

8-3.7.27.1 Component Lines. Two or more lines attached to a hydraulic component should be sufficiently different to prevent incorrect connection to the component.

8-3.7.27.2 Drain Lines. Drain or vent lines coming from the pump, reservoir, or other hydraulic components should not be connected to any other line or any other fluid system in the aircraft in such manner as to permit mixture of the fluids at any of the components being drained or vented.

8-3.7.27.3 Mounting Lightweight Components. Lightweight components that do not have mounting provisions may be supported by the tubing installation, provided that the component is rigidly installed and does not result in destructive vibration or cause other adverse conditions and the tubing installation. Clamps or similar devices may be used to support such units to structure, provided that nameplates, flow-direction arrows or markings, or other data is not obscured and that the supporting member(s) do not affect the operation of the unit.

8-3.7.27.4 Bonding. The aircraft hydraulic system components and lines should be bonded to the aircraft.

8-3.7.27.5 Vibration. The complete hydraulic system, including lines and components, should be designed to withstand the effect of vibration, pump pulsation, and shock loads encountered during service operation of the aircraft.

8-3.7.27.6 Tubing Clearance. Where tubing is supported to structure or other rigid members, a minimum clearance of 1/16 inch should be maintained with such member. A minimum clearance of 1/4 inch should be maintained with adjacent structure, tubing, or other installations. In areas where relative motion of adjoining components exists, a minimum clearance of 1/4 inch should be maintained under the most adverse conditions that will be encountered.

8-3.7.27.7 Corrosion Protection. All tubing in exposed areas, such as wheel wells, weapons bays, and cove areas, should be adequately protected against corrosion, particularly under the sleeve at the fittings.

8-3.7.27.8 Suction Line to Power-Driven Pumps. The supply line to the power-driven pump(s) should be designed to provide adequate flow and pressure at the pump inlet port. This guideline should include operating the pump at the maximum output flow required of the pump and should include all ground and flight conditions the aircraft will encounter. Zero g and negative g conditions and low-temperature start and operation should also be included in the above guideline.

8-4.0 **QUALIFICATION GUIDELINES**.

8-4.1 **Qualification Test Conditions.** The following test conditions apply to qualification testing, unless otherwise modified by, or added to, in the detail specification.

8-4.1.1 Adverse Tolerance Conditions. The component should be capable of functioning when assembled with adverse tolerance parts without any degradation in component performance or life. The manufacturer should verify compliance with this guideline by mathematical analysis.

8-4.1.2 Test Fluid. Fluid conforming to MIL-H-5606 or MIL-H-83282 should be used as a test fluid. MIL-H-6083 may be used for quality conformance testing only.

8-4.1.3 Temperature Conditions. Unless otherwise specified, the ambient and outlet fluid temperatures should be within the range indicated in each individual test. For the components with appreciable heat generating characteristics such as relief valve, solenoid-operated units, etc., the outlet fluid temperature should be as specified, and the inlet fluid and ambient temperature may be decreased to compensate for this heat generation. However, in no case should the inlet fluid temperature or ambient temperature be decreased by more than 25°F. For zero flow condition tests, the ambient temperature should be as specified. The ambient inlet and outlet fluid temperature should be stated in the test report. Fluid temperature should be measured as near as practicable to the component ports. During all soaking periods, the system should be bled of air and should be maintained full of fluid. Unless otherwise specified, the following tolerances should be applied respectively to the following basic temperatures referred to throughout the tests specified:

275° ± 5°F	100° ± 5°F
225 ± 5°F	-65° +0°,-5°F
160° +5°, -0°F	

8-4.1.4 Filtration. For qualification testing, the test fluid should be continuously filtered through a filter element with a micron rating equivalent to the micron rating of the filter element used in the aircraft.

8-4.1.5 Qualification of Similar Units. In the case of a series of devices which are intended to serve the same general function in a hydraulic system, qualification of one device of the series may, at the discretion of the procuring activity, be applied to any other devices of the series if all the internal working parts are identical in every detail with the corresponding internal working parts of the qualified device, and provided it meets the proof, burst pressure, and such operational requirements as may be designated by the activity.

8-4.2 **Component Qualification Tests.**

8-4.2.1 Quality Conformance Tests. Unless otherwise specified in the detail specification, the qualification tests for a component should consist of all the tests listed in paragraph 8-1.2.2 above and should be conducted using the guidelines below.

8-4.2.2 Quality Conformance Test Conditions Unless otherwise specified in the detail specification, fluid and ambient temperatures should be between 70° and 120°F.

8-4.2.1 Examination of Product. Each component should be carefully examined to determine conformance to the requirements of applicable design specifications for design, weight, workmanship, marking, conformance to applicable government and manufacturer’s drawings, specifications and standards for any visible defects.

8-4.2.2 Immersion.

8-4.2.2.1 Nonmetallic Parts Components containing nonmetallic parts other than plastic parts or standard seals in glands known to be compatible with hydraulic fluid should be immersed in hydraulic fluid for a period of 72 hours at a temperature of 275° ± 2°F prior to conducting the qualification tests specified herein or in the detail specification. All internal parts should be in contact with the fluid during this period.

8-4.2.3 Pressure Tests.

8-4.2.3.1 Proof Pressure A proof pressure, as specified in TABLE I or the detail specification, should be applied at the temperature specified in the detail specification for at least two successive times and held 2 minutes for each pressure application. The rate of pressure rise should not exceed 25,000 psi per minute. The equipment should be operated in its normal function between applications of the test pressure. There should be no evidence of external leakage, other than a slight wetting at seals insufficient to form a drop, excessive distortion, or permanent set. Components which require varying test pressures in different elements may have these pressures applied either separately or simultaneously as specified in the detail specification. Components that are subject to pressure in the reverse direction such as check valves, shut off valves or accumulators should be pressurized in both directions, either separately or simultaneously as specified in the detail specification.

8-4.2.3.2 Burst Pressure. A burst pressure, as specified in table II or the detail specification, should be applied at the temperature specified in the detail specification to the component at a maximum pressure rise rate of 25,000 psi per minute. The component should not rupture under this pressure nor should leakage exceed that permitted in external leakage test specified herein. The pressure may be increased above that specified in order to secure data on actual rupture pressure. This should be the last test performed because of its destructive nature. Components that require different test pressures in different elements should have these pressures applied either separately or simultaneously, whichever is the most critical. Components that are subject to pressure in the reverse direction such as check valves, shut off valves or accumulators should be pressurized in both directions, either separately or simultaneously as specified in the detail specification.

8-4.2.4 Leakage Tests.

8-4.2.4.1 External Leakage. During the course of all the tests listed in this specification, external leakage, other than a slight wetting insufficient to form a drop through static seals, should be cause for rejection. Where external, dynamic seals are utilized, permissible leakage past such seals should be no greater than that specified in the detail specification.

8-4.2.4.2 Internal Leakage. These tests should be performed with the component held in the position most conducive to leakage. Pressure of 5 psi and working pressure should be held for a period of 5 minutes each, unless otherwise specified in the detail specification. In each case, the leakage measurement should consist of the last 3 minutes of the 5 minute period. The rate of leakage should not exceed that specified in the detail specification for the qualification test.

8-4.2.5 Pressure Drop. Pressure drop characteristics for a low range of 0 to 150 percent of rated flow or as specified in the detail specification should be determined for the component. The piezometer or manometer across the component may be used for accurate measurement where the pressure drop range is low enough to permit its use. The pressure drop observed at rated flow should not exceed the value permitted by the applicable detail specification.

8-4.2.6 Extreme Temperature Functioning Tests.

8-4.2.6.1 Low Temperature. The component should be connected to a static head of 1 to 3 feet of the test fluid or rated working pressure, whichever is the more critical condition. This arrangement should be maintained at a temperature not warmer than -65°F for 3 hours after the temperature has stabilized at -65°F. After this period the component should be actuated at least two times. Variation of actuating forces or regulation, as applicable should not exceed that permitted by the detail specification. The quality conformance tests for leakage should be performed after each actuation and the requirements of the detail specification satisfied.

8-4.2.6.2 Intermediate Temperature. Immediately following the low temperature test (8-4.3.6.1), the test arrangement should be warmed rapidly to a temperature of 160°F. While the temperature is being raised, the component should be actuated at maximum increments of 36°F to determine satisfactory operation throughout the temperature range. These check tests should be made without waiting for temperature of the entire component to stabilize. For complex components, the 36°F increment may be increased in the detail specification to allow for time to perform functional tests.

8-4.2.6.3 High Temperature. In the case of standard components, the temperature should be maintained at 275°F, or in the case of nonstandard components, the temperature should be maintained at the highest value which the component is expected to encounter for a length of time sufficient to allow all parts of the component to attain the temperature. In no case should the temperature at which this test is conducted be less than 275°F. The component should then be actuated at least two times. In the case of pressure actuation or regulation, the variation from room temperature actuation or regulation should not exceed that permitted by the detail specification. The qualification test for leakage should be performed after each actuation and the requirements of the detail specification satisfied.

8-4.2.6.4 Differential Temperature. For components utilizing fluid from the two systems, the component should be operated with the fluid temperature maintained at a differential temperature of the maximum differential possible for the system. The component should be actuated at least two times. Variation of actuating forces or regulation, as applicable should not exceed that permitted by the detail specification for the differential temperature condition.

8-4.2.6.5 Temperature Limits. The solenoids should be subjected to high temperature and low temperature test procedure of MIL-STD-810.

8-4.2.7 Temperature Rise. Components should be tested at the dc or ac voltage and at the frequency as specified in the detail specification. The dc test source should be used to measure coil resistance prior to and immediately after operation. The dc resistance measurements should be used to determine temperature rise.

8-4.2.8 Endurance.

8-4.2.8.1 General. The component should be subjected to cyclic operation and to other fatigue tests, such as hydraulic impulse, in accordance with the requirements of the detail specification which should indicate number of cycles, schedule of cycling, cycle rate, stroke, rate of flow, loads, temperature, impulse peaks, etc. When applicable, leakage should be checked at 25, 50, 75 and 100 percent of the number of cycles required. At the conclusion of the endurance test, the component should operate satisfactorily and should be disassembled and carefully inspected. There should be no evidence of excessive wear in any part of the component.

8-4.2.8.2 Aircraft Applications. The number of cycles selected should be based on the duty cycle over the anticipated life of the aircraft or the component, whichever is greater, multiplied by an appropriate safety factor. In either case, the cycles should be not less than the values specified in Table III.

TABLE III. ENDURANCE TEST

Type and Usage of Component	Cycles
Standard	See detail specification
Nonstandard -Emergency	5,000
-Infrequent (less than 10 cycles per flight)	20,000
-Frequent (more than 10 cycles per flight)	50,000
-Flight control, steering, anti-skid, etc.	See detail specification

8-4.2.8.3 Endurance. The endurance tests should be governed by the following general test cycle, as well as the test methods specified in 8-4.3.8.2. The following tests should be performed in the sequence indicated:

General Test Cycle

(a) Fill the component with hydraulic fluid to 90 percent of the total fluid volume of the unit. Cap the ports and place the component in heating chamber in which the ambient temperature is maintained at 275°F. Hold the component at the ambient temperature of 275°F for a period of 72 hours.

(b) Conduct the test specified in 8-4.3.3.1 at 275°F.

(c) Conduct the test specified in 8-4.3.6.1 at -65°F for a minimum of 10 cycles. Test specimen to remain at -65°F for at least 4 hours, prior to conducting test. Increase in temperature during the test due to operation is permitted.

(d) Immediately following the test specified in 8-4.3.6.2 at 160°F, warm test arrangement rapidly to 275°F and actuate component at increments of approximately 36°F to determine satisfactory operation.

(e) Conduct the test specified in 8-4.3.6.3 at 275°F for a minimum of 10 cycles.

(f) Conduct 25 percent of the cycles of the test specified in 8-4.3.8 at 275°F, with the consequent length of exposure at 275°F, unless either condition is modified as a result of 8-4.3.1.1.

(g) Soak component at 275°F for 2 hours. Pressure is to be maintained during the first hour and reduced to approximately zero psi for the second hour.

(h) Repeat the low temperature test specified in 8-4.3.6.1 at -65°F and the high temperature test specified in 8-4.3.6.3 at 275°F.

(i) Conduct 75 percent of the cycles of the test specified in 8-4.3.8 at 225°F unless modified as the result of 8-4.3.1.1.

8-4.2.8.3.1 The contractor should determine the percentage of the cycles a component is to operate at elevated temperatures and the cumulative elevated temperature exposure of the component during its life or the life of the aircraft, whichever is greater. The extent of elevated temperature endurance cycling of a component should be based on the above determination.

8-4.2.8.4 Impulse.

8-4.2.8.4.1 Actuators, Valves, Pressure Containers and Similar Components. These components should be subjected to an impulse test in accordance with SAE ARP 1383 and as specified in the detail specification. Where the SAE ARP and the detail specification conflict, the detail specifications should take precedence.

8-4.2.8.4.2 Hose Assemblies, Tubing, Fittings, Quick Disconnect Couplings, Filters and Other Transmission Line Components. These components should be subjected to an impulse test in accordance with SAE ARP 603 and as specified in the detail specification. Where the SAE ARP and the detail specification conflict, the detail specification should take precedence.

8-4.2.9 Vibration, Shock and Acceleration. Components should be subjected to vibration, shock and acceleration test procedure of MIL-STD-810 (methods 514, 516 & 513 respectively) when specified in the detail specification.

8-4.2.10 Humidity. Moisture resistance should be established by the humidity test procedure of MIL-STD-810. At the conclusion of this test, the component should operate normally through 25 cycles at rated voltage. The solenoids should be subjected to dielectric strength test.

8-4.2.11 Fungus. Components which include materials that are not classified as fungus inert by MIL-STD-454, Requirement 4, should be subjected to the fungus test of MIL-STD-810, Method 508.

8-4.2.12 Sand and Dust. The components should be subjected to the dust test procedure of MIL-STD-810. This test may be omitted if all moving parts of the component are exposed only to internal fluid.

8-4.2.13 Salt Fog. The components should be subjected to salt fog test procedures of MIL-STD-810 unless it is established by the procuring activity that this test is not required.

8-4.2.14 Icing. The component should be subjected to an icing test if its design is such that accumulation of ice on external surfaces or inside of vent holes may cause malfunction. When required, this test should be performed as specified in the detail specification.

8-4.2.15 Explosion Proof. Components with a potential source of ignition should be subjected to an explosion proof test in accordance with the explosive atmosphere test procedure of MIL-STD-810.

8-4.2.16 Electromagnetic Interference. Components that cause electromagnetic interference should be subjected to a electromagnetic interference test in accordance with MIL-STD-461, MIL-STD-462 and MIL-STD-463.

8-4.2.17 Actuation Above System Pressure. Components should be tested for actuation, under a pressure equal to thermal relief valve maximum setting of the circuit in which they are installed. This test should be conducted as specified in the detail specification.

8-4.2.18 Reliability. Tests should be conducted to demonstrate compliance with reliability requirements, including MTBF, or equivalent, as specified in the detail specification.

8-4.2.19 Dielectric Strength. If the dielectric test follows the humidity test or the salt fog test, the solenoids should be baked for 6 hours at a maximum ambient temperature as specified in the detail specification prior to being subjected to the dielectric test. All solenoids should be subjected to a __ Hertz alternating test voltage between terminals and case for one minute at the following root mean square amplitudes:

(a) ____ volts at room temperature and pressure
(b) ____ volts at maximum operating temperature and altitude.

Leakage current should not exceed one milliampere during these tests.

8-4.2.19.1 Subsequent dielectric tests on assembled hydraulic component or dielectric test after environmental test on the solenoid should be performed at 75 percent of the above voltages for 1 minute. Flashover or leakage current greater than one milliampere should constitute a failure. There should be no distinction between test voltage on prototype and production units.

8-4.2.20 Drop Out Voltage Test. Solenoid operated components should be tested for drop out voltage by applying nominal activation voltage and slowly reducing the applied voltage to 10% of the nominal activation voltage. The solenoid should drop out between 10% of the nominal activation voltage and the minimum activation voltage specified in the detail specification.

8-4.2.21 Component Cleanliness. The component should be tested for internal cleanliness by subjecting a representative sample of the fluid contained in the component to a particle count using FED-STD-791, method 3009. The component should be actuated (if possible) during the sampling. The cleanliness level should be equal to or better than Class 8, Table I of NAS 1638. If this procedure is not practical, the detail specification should include a component cleanliness test.

8-4.3 **Qualification of Hydraulic Pumps.** The qualification tests outlined in this section are in addition to the guidelines of the component qualifications presented in section 8-4.2. In cases of the same test parameters, the tests of this section should replace the guidelines of section 8-4.2 for testing of hydraulic pumps. The hydraulic fluid used in all design integrity tests should be that specified in the detail specification. Accuracy of the instrumentation for operating conditions should be consistent with the state-of-the-art, and the degree of accuracy considered achieved should be attested to. The pump or pumps chosen for the Design Integrity Tests must be representative of those pumps to be produced in subsequent production. The qualification tests should consist of the following, performed in the suggested order listed on one sample pump, or the same test performed on two sample pumps (A and B) on a selected basis, as defined by the following:

a. Quality conformance - samples A and B
b. Fluid immersion - sample A
c. Proof pressure - samples A and B
d. Calibration - samples A and B
e. Maximum pressure - sample B
f. Response time - sample B
g. Pressure pulsation - sample B
h. Heat rejection - sample B
i. Vibration - sample B
j. Low temperature - sample B
k. Endurance - sample A
l. Cavitation - sample A
m. Drive coupling shear - sample A or B
n. Additional tests, if any, required by the model specification - sample A or B

8-4.3.1 Quality Conformance Inspection.

8-4.3.1.1 Examination Of Product. The pump should be examined to determine conformance with the applicable specifications

8-4.3.1.2 Break-In Run. The break-in run should be made with any desired pressure in the inlet and outlet lines and should consist of 1/2 hour minimum at 30 to 75 percent of rated rpm and 1/2 hour minimum at 80 to 100 percent of rated rpm.

8-4.3.1.3 Proof Pressure and Overspeed Tests.

8-4.3.1.3.1 Test Conditions.

a. Case drain inlet pressure should be maintained at 450 to 500 psig, at 125 percent rated speed, and with fluid temperature optional.

b. Pressure control should be adjusted for 125 percent rated discharge pressure at no flow.

C. System impedance should be IAW 8-4.3.5.1.2.

8-4.3.1.3.2 Load Cycles.

a. A step-function load should be imposed, causing the pump to cycle from 125 percent rated discharge pressure at no flow to 115 percent rated discharge pressure at 10 cpm for 5 minutes with equal dwells at each load condition.

b. A step-function load should be imposed, causing the pump to cycle from 125 percent rated discharge pressure at no flow to maximum flow at minimum practical system back pressure at 10 cpm for 1 minute.

c. There should be no external leakage sufficient to form a drop, except that the shaft seal may leak at a rate not to exceed 5 milliliters (ml.) per hour. Case drain flow should be monitored. There should be no evidence of malfunction.

Note: The overspeeds specified in 8-4.3.1.3.2 are subject to the same limitations and modifications defined in Note 7 of Tables IV and V.

8-4.3.1.4 Functional Test. During all of the functional tests, the hydraulic fluid used should be that specified in the detail specification. Rated inlet pressure should be maintained within the tolerance specified herein. Inlet temperature should be at the rated condition. The functional test should be comprised of the following:

a. One-half (1/2) hour at rated speed and maximum full flow pressure.

b. One-half (1/2) hour at rated speed with discharge. pressure varied between maximum full flow pressure and rated discharge pressure at a frequency of 6 cpm.

c. Four hours at rated speed and rated discharge pressure; except that at 10 minute intervals the discharge pressure should be reduced to maximum full flow pressure for 1 minute.

8-4.3.1.4.1 Run-In. The run-in after teardown inspection should be performed at 50 to 100 percent of rated speed for a period of one-quarter hour with discharge pressure varied between maximum full flow pressure and rated discharge pressure at a frequency of __ CPM.

8-4.3.1.4.2 External Leakage Test.

a. No external leakage, other than at shaft seal, of sufficient magnitude to form a drop should be permitted.

b. Shaft seal leakage During design integrity tests and during service life. Static leakage - should not exceed one drop in 2 minutes at rated pressure conditions. Dynamic leakage - should not exceed 5 ml. per hour.

8-4.3.1.4.3 Pressure Control Tests.

a. Rated discharge pressure should remain within detail specification limits as pump speed is varied from 50 percent to 100 percent of rated speed or to the speed specified in the detail specification and as fluid temperature is varied as defined in the detail specification.

b. There should be no indication of pressure control instability as pump speed is varied from 50 percent to 100 percent of rated speed throughout flow range. System conditions should be defined in the detail specification.

c. The hysteresis characteristics of the pressure control from zero to rated flow at rated speed should not exceed the value as specified in the detail specification.

8-4.3.1.4.4 Calibration. At the completion of the other quality conformance tests the torque required to drive the pump and the case drain flow should be measured and recorded for the following conditions: Rated speed, rated discharge pressure, rated inlet pressure, case pressure 20 psi or as specified in the detail specification above inlet pressure, and rated inlet temperature or as designated in the detail specification. The delivery of the pump at rated speed, rated maximum full-flow pressure, and at rated inlet pressure should be measured and recorded. The measured values for torque, case drain flow, and delivery should be within the limits specified in the detail specification. Flow may be measured in the low pressure side of the discharge line at the option of the detail specification, provided adequate compensation is made for compressibility in stating the delivery.

8-4.3.1.4.5 Filter Patch Test. The inlet fluid used during the test should be continuously filtered to 5 microns absolute. A filter should be installed in the inlet, outlet, and case drain or cooling port lines of the test setup. The fluid in the filter bowls downstream of the pump, should be checked by the procedure specified in 8-4.3.1.4.5.1 for contamination accumulated during the first 2 hours of the 4 hour functional test performed as per 8-4.4.2.4. A filter patch test should again be made during the last 2 hours of the 4 hour functional test.

8-4.3.1.4.5.1 Patch Preparation. The fluid in each filter bowl should be collected in clean containers. Rinse both the filter bowl and element with a minimum of 15 cubic centimeters (cc.) of a suitable fluid solvent and add to the applicable container. The total resulting fluid should be passed through a 47 millimeter (mm.) disc, #40 Watman paper. Wash the disc free of fluid with a minimum of 15 cc of fluid solvent. After drying, the resultant filter patch should be coated with clear lacquer and permanently attached to the log sheet of the test. All fluid solvent should be filtered through a 0.45-micron pore size membrane prior to use during the foregoing procedure.

8-4.3.1.4.5.2 Patch Comparison. Each filter patch resulting from the preceding test should be compared with the standard patch then in effect and any discrepancy noted in the test log. The second patch should show equal or less contaminant than the first patch, and should also show equal or less contaminant than the standard patch. If it does not, additional 2 hour patches may be run until a trend is established.

8-4.3.2 Fluid Immersion Test. Where electrical components are a part of the pump assembly, they, as part of their qualification test, will be separately subjected to a fluid immersion test prior to the start of the qualification test. This will consist of continuous immersion for 72 hours in the hydraulic fluid at rated temperature. After the 72 hour soak period, the component should remain in the fluid at normal temperature until ready for further tests.

8-4.3.3 Proof Pressure and Overspeed Test. The proof pressure and overspeed tests as outlined in 8-4.3.1.3 for quality conformance inspection should be performed, except the tests should be repeated 10 times. At the conclusion of the proof pressure tests, the pump delivery mechanism should be restored to its normal adjustment of configuration.

8-4.3.4 Calibration Test. Values of flow rate and driving torque should be determined at minimum operating speed, and at 25, 50, 75, 100 and 110 percent of rated speed. At each of these speeds, four sets of flow and torque recordings should be made at approximately the following pressures: 25, 50, 75, and 100 percent of maximum full-flow pressure and at 5 equally spaced increments of flow between maximum full-flow pressure and rated discharge pressure. Unless otherwise specified in the detail specifications, calibrations will be made at the inlet condition specified in 8-3.3.4 and flow measurements may be made in the line downstream of the load valve but must be corrected for fluid compressibility at the option of the detail specification.

8-4.3.4.1 Pump Inlet Pressurized. Regulate the pressure at the pump inlet port to the rated inlet pressure at full flow and rated speed conditions.

8-4.3.4.2 Minimum Operating Speed. The speed should be reduced below 25 percent of the rated speed to determine the speed at which the discharge flow or pressure becomes erratic. This point should be recorded and designated the minimum operating speed for that condition.

8-4.3.5 Maximum Pressure, Response Time, and Pressure Pulsations. Pressure pick-up and recording equipment should be used to provide an oscillographic record, or its equivalent, of the pressure-time function of the pump and its hydraulic circuit through the transient and steady state periods described in the following three tests. The pressure pick-up and recording equipment should be capable of static calibration with repetitive accuracy of 5 percent of rated pressure and readability of 3 percent of rated pressure. It should be considered essential that the dynamic calibration of the pick-up and recording equipment is valid for the dynamic conditions. The pressure pick-up should be located in the pump discharge line as close to the pump outlet fitting as physically possible.

8-4.3.5.1 System Impedance.

8-4.3.5.1.1 Response Time. The system impedance of the test circuit when determining pump response should be such that when the pump is operating at rated speed, maximum full flow, and rated inlet temperature, the rate of pump discharge pressure rise when the flow in the system is suddenly stopped (the minimum rise of the maximum slope of pressure rise) should be 50,000 psi/sec. as calculated from system volume, pump rated delivery, and fluid bulk modulus at rated temperature and rated discharge pressure.

8-4.3.5.1.2 All Other Test. The system impedance of the test circuit when determining maximum pressure, pressure pulsations, stability, and the remaining design integrity tests should be specified in the detail specification. Both inlet circuit and high pressure circuit should be simulated.

8-4.3.5.2 Maximum Pressure Test. The test circuit specified in 8-4.3.5.1.2 should be utilized. Flow changes should be initiated by a solenoid operated valve with a response time 0.020 second or less, or as specified in the detail specification. As the test pump is caused to operate between steady state maximum, full-flow pressure to steady state rated discharge pressure in both directions, an oscillographic record of the pressure-time function through the transient period should be made. The test should be run at 50 percent and 100 percent of rated pump speed. Air entrainment in the hydraulic fluid should be at a minimum. Unless otherwise specified in the detail specification, the peak pressure transient as measured on the above record should not exceed 135 percent of rated discharge pressure.

8-4.3.5.3 Response Time. With the test circuit specified in 8-4.3.5.1.1, and load valves set at a flow condition equivalent to maximum full flow pressure at each of the test speeds, the solenoid valve, which changes discharge line from full open to full closed, or vice versa, should then be used to execute the test. Runs should be made at 50, 75, and 100 percent of rated speed or as specified in the detail specification. With the solenoid valve open and the test pump operating at steady state maximum full-flow pressure, an oscillographic record should be made of the pressure-time function through the transient period associated with the closing of the solenoid valve and establishment of steady state rated discharge pressure. At 50% and 75%, and 100% of rated speed, the response time should not exceed the value specified in the detail specification. The response time for the change from rated discharge pressure to maximum full-flow pressure should be recorded and at 50%, 75%, and 100% of rated speed, should not exceed the value specified in the detail specification. Check the response time for small incremental changes of flow as follows: introduce a parallel flow path which includes an orifice and a downstream solenoid valve with .2 second response or as specified in the detail specification. This orifice should be adjusted to pass 5 percent of maximum full-flow and the main load throttling valve should be adjusted to pass 90 percent of maximum full-flow for each of the three pump speed settings. Check response time at each speed setting when the small flow path solenoid is opened and closed with the main flow path solenoid valve both opened and closed. The response time at rated speed should not exceed that specified in the detail specification. Check response time at rated speed, rated pressure, minimum inlet pressure and oil temperature as specified in the detail specification.

8-4.3.6 Pressure Pulsations. The test circuit specified in 8-4.3.5.1.2 should be equipped with a dynamic pressure transducer of zero volume and sensitive to 20 to 100 kilo-hertz. With pump at rated discharge pressure, vary the speed from 50 to 100% of rated speed at a rate of change not exceeding 100 RPM per second. During this period an oscillographic record should be made of the pulsation pattern. In addition, runs should be made at 50, 75, and 100 percent of rated speed at maximum full flow pressure and at 25, 50, and 75 percent of full flow at each speed. Values of pressure pulsation should not exceed ± 10 percent of rated discharge pressure, or a pressure band specified by the detail specification under any condition, when the pump is tested in the circuit which simulates the actual system in which the pump is to be installed, as defined in the detail specification.

8-4.3.7 Heat Rejection. The object of these tests is to measure the rate of heat rejection of the pump over the expected normal range of operating conditions. The rate of heat rejection at specified conditions should be considered equal to the difference between the input and output horsepower of the pump at those conditions. Output H.P. may be calculated based on flow measurements in the low pressure side of the discharge line at the option of the detail specification. provided adequate compensation is made for compressibility in calculating output power.

8-4.3.7.1 Heat Rejection Determination. To determine the rate of heat rejection, the pump should be run at rated speed and rated inlet temperature, and the input and output horsepower measured at rated discharge pressure, maximum full-flow pressure, and at least two additional flow points between those values. Should it be desired to determine the rate of heat rejection at operating conditions other than these, the additional requirements should be defined in the detail specification. The maximum acceptable value in British thermal units per minute (BTU/MIN) of heat rejection rate at specified operating conditions should be as specified in the detail specification.

8-4.3.8 Vibration tests.

8-4.3.8.1 Test Pump Mounting Orientation. The test pump should be mounted on a vibration generating mechanism successively in each of at least three positions. All of the testing specified should be performed in each of the mounting positions. One of these mounting positions should be such that the direction of vibratory motion should be parallel to the shaft axis of the pump. Another mounting position, if and when practicable, should be such that the direction of vibratory motion should be parallel to the axis of the compensating mechanism. When the pump is equipped with an electrical depressurization device, an additional mounting position should be such that the direction of vibratory motion should be parallel to the E.D.V mechanism.

8-4.3.8.2 Resonant Frequency Vibration. Resonant frequencies should be searched according to the double amplitude and frequency charts of MIL-STD-810. Applicable procedures and test values should be specified in the detail specification.

8-4.3.8.3 Cyclic Frequency Vibration. Upon completion of the resonant frequency vibration, a cycling vibration should be imposed in accordance with MIL-STD-810. Applicable procedures and test values should be specified in the detail specification.

8-4.3.8.4 Other Vibration Tests. The detail specification will require other vibration tests to be performed when a particular installation imposes severe environmental conditions peculiar to its system requirements.

8-4.3.8.5 Pump Operation. Throughout the above vibration tests the pump should be operated in the test circuit of 8-4.3.5.1.2. Oil inlet temperature should be maintained at 140°F regardless of the rated temperature of the pump . being tested, and ambient temperatures should be maintained at room ambient conditions. The pump discharge pressure should be continuously cycled from rated discharge pressure (zero flow) to a discharge pressure corresponding to approximately 50 percent of rated delivery. These pressure cycles should be abruptly accomplished by electrically controlled hydraulic valves at a rate of five cycles per minute (cpm). Transition from one condition of flow to the other condition of flow must be accomplished in a valve time of less than 1/2 second. Where the pump is equipped with a depressurizing means, cycle between pressurized and depressurized modes are required by the detail specification.

8-4.3.9 Low Temperature Test. All temperature requirements apply equally to the pump body, hydraulic fluid, and ambient environment. After at least 18 hours at the minimum inlet temperature specified in the detail specification (or at -65° ± 5°F, in the absence of such stipulation in the detail specification) the pump should be started and uniformly accelerated to 50% of rated speed in not more than 10 seconds, unless otherwise specified in the detail specification. Twenty runs should be made with the outlet pressure as low as practicable and the inlet pressure as specified in the detail specification. Rated speed should be reached within 20 seconds after startup. When rated speed has been reached, it should be maintained for at least 10 seconds; observations should indicate whether the pump displaces fluid through the hydraulic system. Then five starts and runs should be made, during which the pump discharge line terminates in a relief valve set to pass fluid at maximum full-flow pressure. In addition, five starts should be made with the pump discharge line completely closed so that the pump will operate at rated discharge pressure. Throughout these tests, after each run the pump and fluid should be allowed to stand idle long enough for them to be restored to the above soaking temperature before starting the next run. When the pump includes a depressurization device that is used as an engine starting aid, the device should be activated during starts and deactivated when the pump reaches 50% of rated speed. During al1 tests the test circuit should contain a volume simulating operating conditions in the aircraft system or as specified in the detail specification.

8-4.3.10 Endurance Test. As a minimum guideline, the sample pump should complete the following endurance test schedule consisting of:

a. Normal Test. 600 hours, consisting of the first eight phases specified in Table IV in the order listed, plus two calibrations, the stop-start cycles, filter checks, and air ingestion cycles as specified in the following subparagraphs.

b. Overload Test. 450 hours, consisting of the first eight phases specified in Table V in the order listed, plus one calibration, the stop-start cycles, thermal cycles, and thermal shock test. The test circuit for the low temperature and endurance test should be as specified in 8-4.3.5.1.2. Modification of any of the test conditions of Table IV or V or additional endurance testing, in the form of additional cycles in any of the phases, may be required by specifying such modification or additions in the detail specification.

Table IV. Endurance test conditions (normal test)

		Cycles								
Phase 1/	Percent of rated endurance 2/	Flow in percent of rated delivery	Percent of rated discharge pressure	Duration time in seconds	Flow in percent of rated delivery 7/	Percent of rated discharge pressure	Duration time in seconds	Inlet pressure (psia) 3/	Percent of rated speed 7/	Case drain port pressure (psig) 4/
1	6.67	0	100	540	66.7	5/	60	-	66.7	75
2	20.0	0	100	540	106.7	5/	60	-	106.7	100
3	6.67	0	100	6	120	5/	6	-	120	30
4	6.67	0	100	6	85	6/	6	-	120	100
5	13.33	100	5/	60	50	6/	60	-	100	30
6	13.33	70	6/	5	5	6/	5	-	100	100
7	13.33	100	5/	5	0	100	5	-	100	30
8	20.0	90	6/	5	106.7	5/	5	-	106.7	100

1/ One or more phases, totaling not less than 25 percent of rated endurance should be run at rated temperature. The remaining 75 percent of the rated endurance should be run at 80 percent of the rated temperature unless otherwise specified in the detail specification. The apportionment of endurance test time between rated inlet temperature and the lower temperature(s) should, where feasible, be based on a realistic appraisal of the mission profile of the aircraft in which the pump is to be installed.
2/ A time tolerance of ± 1 percent is permissible for ease of test implementation. The total test duration of all phases covered by this table should be as specified in 4.3.2.1.10 as a minimum
3/ The inlet pressure should be as defined in the detail specification.
4/ Case drain port pressure should be set by means of a fixed restriction at maximum drain flow condition. Pressures to be as stated except not less than 30 psig above inlet.
5/ Ninety-five percent of maximum full-flow pressure.
6/ Pressure indicated should be adjusted to provide stipulated flow.
7/ Phases 3 and 4 should apply as is to all pumps whose rated speed meets the requirements of 3.3.9. Where the rated speeds exceed those shown, subtract 20% of the overspeed percent for every 10% the pump is rated above 3.3.9 except the overspeed cannot be less than 110% of rated speed in Phases 3 and 4.

Table V. Endurance test conditions (overload test)

		Cycles								
Phase 1/	Percent of rated endurance 2/	Flow in percent of rated delivery	Percent of rated discharge pressure	Duration time in seconds	Flow in percent of rated delivery 7/	Percent of rated discharge pressure	Duration time in seconds	Inlet pressure (psia) 3/	Percent of rated speed 7/	Case drain port pressure (psig) 4/, 5/
1	6.67	0	125	540	66.7	110	60	-	66.7	75
2	20.0	0	125	540	106.7	110	60	-	106.7	100
3	6.67	0	125	6	125	110	6	300	125	330
4	6.67	0	125	6	110	6/	6	-	125	100
5	13.33	125	110	60	50	6/	60	-	125	100
6	13.33	70	6/	5	5	6/	5	-	125	100
7	13.33	125	110	5	0	125	5	300	125	330
8	20.0	90	6/	5	106.7	110	5	-	106.7	100

1/ One or more phases, totaling not less than 25 percent of rated endurance should be run at rated temperature. The remaining 75 percent of the rated endurance should be run at 80 percent of the rated temperature unless otherwise specified in the detail specification. The apportionment of endurance test time between rated inlet temperature and the lower temperature(s) should, where feasible, be based on a realistic appraisal of the mission profile of the aircraft in which the pump is to be installed.

2/ A time tolerance of ± 1 percent is permissible for ease of test implementation. The total test duration of all phases covered by this table should be as specified in 8-4.3.2.1.10 as a minimum

3/ The inlet pressure should be as defined in the detail specification.

4/ Case drain port pressure should be set by means of a fixed restriction at maximum drain flow condition. Pressures to be as stated except not less than 30 psig above inlet.

5/ The pump compensator may be reset to provide the specified discharge pressure for each phase at the case drain port pressure shown.

6/ Pressure indicated should be adjusted to provide stipulated flow.

7/ The overspeed and excess flow of Phases 3 thru 7 will apply to all pumps whose rated speed meets the requirements of 3.3.9. Where the rated speeds exceed those shown, subtract 20% of the overspeed percent for every 10% the pump is rated above 3.3.9 except the overspeed cannot be less than 110% of rated speed in Phases 3 thru 7.

8-4.3.10.1 Filtration for Endurance Test. The hydraulic fluid to be used should be passed through a 5 micron absolute filter before entering the test system. Filters, either 5 micron or 15 micron absolute as specified in the detail specification should be installed in the pump inlet, outlet, and case drain or cooling port lines throughout the endurance test.

8-4.3.10.2 Filter Check. At intervals of 50 ± 8 hours, during the endurance test, clean filter elements should be installed in all 3 filters, and the endurance test schedule resumed for 2 hours, at the end of which these filter elements should be removed and replaced with clean filter elements. The filter elements removed after 2 hours running should be checked in accordance with 8-4.3.1.4.5.

8-4.3.10.3 Calibration. Before starting phase 1 of both portions of the endurance test, and again upon completion of the complete test, the pump should be calibrated, using the procedure specified in 8-4.3.4, the results of these three calibrations should be plotted on one chart to show the effect of the endurance test on the performance of the pump.

8-4.3.10.4 Start-Stop Cycles. Start-stop cycles should be performed before the normal endurance test and after the overload endurance test. The system should be as specified in 8-4.3.5.1.2. Fluid temperatures may range from ambient to rated, but actual values should be recorded. For those cycles run following the overload test, the compensator should be left adjusted at its maximum pressure.

8-4.3.10.4.1 Full Load Cycles The pump should be accelerated to rated speed within 2 seconds with the load orifice adjusted to 95 percent of maximum full-flow pressure for the first half of the cycles, and 110 percent of rated pressure for the last half of the cycles. The pump should be allowed to coast to a stop immediately after reaching rated speed. Deceleration time should be recorded. One full load start-stop cycle should be performed for each 3 hours of endurance testing performed. Half the cycles should be performed prior to the start of the normal endurance test and the remaining half should be performed following the overload endurance test.

8-4.3.10.4.2 No-flow Cycles. Follow the same procedure as specified in 8-4.3.10.4.1, except the discharge line should be blocked during starts and stops, and in between cycles the discharge pressure should be reduced to approximately 10 psi. Two no-flow start-stop cycles should be performed for each phase. Half of the cycles should be performed prior to the start of normal test and the remaining half should be performed following the overload endurance test. When the pump includes a depressurization device that is used as an engine starting aid, the device should be deactivated when the pump reaches 50 % of rated speed.

8-4.3.10.5 Pump Case Pressure Cycles. Throughout all the 16 phases of the endurance test, the pump case should be subjected, at approximately equal intervals of running time, to internal pressure impulses originating in the case drain (or by-pass) line, and reaching at least 500 psi or 150 percent of actual case drain pressure, whichever is greater. Unless otherwise specified in the model specification, the time from the start of the pressure impulse to the initial return to normal case pressure should be not less than 0.1 second and not more than 0.5 seconds.

8-4.3.10.5.1 Total Cycles. At least 20,000 pressure cycles should be applied to the pump case, distributed with reasonable uniformity over the 16 phases of the endurance test. An oscillographic, or equivalent, record of the pressure impulse should be taken within the first hour of the endurance test and with each even hundredth hour thereafter to provide verification that the pressure-time history of the impulse is in accordance with 8-4.3.10.5.

8-4.3.10.6 Air Ingestion. The ingestion of air by the hydraulic system, for example, as a result of the replacement of components during line maintenance, should be simulated during the first eight phases of the normal endurance test by the following procedure: The line supplying oil to the pump should be arranged so that a 4-foot length of it immediately adjacent to the pump can be disconnected, vented, drained, and reconnected without draining the rest of the test system.. Each time the test is shut down to install the filter elements for the 2-hour patch run (or alternatively, to remove these elements at the end of the 2-hour patch run) it should be shut down while the pump is operating at full flow; the reservoir should be depressurized, the 4-foot length of inlet line should be disconnected, vented, drained and reconnected. During these operations, the filter elements should be replaced. The pump should then be started with the test system set to develop maximum full flow. Thirty seconds after starting, the discharge pressure should be recorded, then the reservoir should be pressurized and the endurance test schedule resumed. The aircraft reservoir or an agreed upon facsimile, the aircraft suction line sizes, lengths and configuration together with rated inlet pressure should be used.

8-4.3.10.7 Thermal Cycles. After completion of the overload test, eight thermal cycles of the pump from -20°F to rated temperature must be accomplished. With the pump not operating, the pump and test system should be stabilized at -20°F for 1 hour. The pump should then be started and operated at maximum full-flow pressure and rated inlet pressure until the fluid reaches rated temperature. This should comprise one cycle. Heating rate should be specified in the detail specification.

8-4.3.10.8 Thermal Shock. After completion of the overload test, a thermal shock cycle must be completed. The hydraulic pump and fluid should be cooled by some means to a temperature of -65°F or a temperature specified in the model specification. The hydraulic reservoir temperature should be maintained at rated temperature and contain a volume of fluid equal to that in the aircraft system or as specified in the model specification. The pump should be started and brought up to rated speed in a time interval specified in the detail specification. Discharge pressure should be set to cause the pump to deliver rated flow or as specified in the detail specification.

8-4.3.10.9 Fluid. The hydraulic fluid used in the endurance test should be that specified in the detailed specification. The endurance test system should be changed at the start of the endurance test and no fluid should be added before the endurance test is completed except:

a. The amount of fluid unavoidably removed from the system during the specified filter checks and air ingestion tests may be replaced.

b. In the event of failure of the test system external to the pump, resulting in loss of fluid or contamination not pertinent to pump endurance, the entire fluid supply may be replaced.

c. To maintain fluid within the physical and chemical property limits established by the procuring agency.

A record should be made of the time and the quantity of fluid added in each case, and entered in the log of the test.

8-4.3.10.10 Failure of Parts. If, during the design integrity test program, the test is terminated because of a part failure, the pump should be replaced or repaired using a redesigned part(s) or, in the case of faulty material or workmanship, the procuring activity may authorize the installation of a part of the original design and the defect over-come. The program should be considered complete when all parts within the pump have completed without failure the requirements of the program as specified in the applicable detail specification. Should pump tests be continued from point of failure with repaired or replaced part(s), subsequent failure of parts that have successfully completed total endurance requirements will not be considered cause of rejection.

8-4.3.10.11 Cavitation Test. The pump should be operated at rated speed, rated inlet temperature, and maximum full-flow pressure. The fluid pressure at the pump inlet port should be adjusted to 120% of rated inlet pressure. The rate of flow and delivery pressure should be measured as inlet pressure is reduced in steps small enough to clearly establish the onset of cavitation and its effect on output.

8-4.3.10.12 Drive Coupling Shear Test. The drive coupling should be removed from the pump and set up for the torsion test. It should be loaded torsionally until failure takes place, and the load producing the failure should be recorded. The failure should take place at the shear section of the coupling.

8-4.3.10.13 Tear Down Inspection. After completion of the endurance tests the pump should be disassembled and all parts visually inspected, or as specified in the detail specification. The general condition of the parts should be reported.

8-4.4 **Accumulators.** The qualification tests outlined in this section are in addition to the guidelines of the component qualifications presented in section 8-4.2. In cases of the same test parameters, the tests of this section should replace the guidelines of section 8-4.2 for testing of accumulators.

8-4.4.1 Separator Under Pressure. With the accumulator mounted in a vertical position and with the gas port down, the accumulator separator should withstand proof pressure applied to the fluid port with the gas port open for 2 minutes without leakage or damage. Also, the accumulator mounted in a vertical position and with the fluid port down, the accumulator separator should withstand proof pressure applied to the gas port with the fluid port open for 2 minutes without leakage or damage.

8-4.4.2 Volumetric Efficiency. The accumulator should be mounted in a vertical position, with the gas port down, and with the piston bottomed on the gas end. The accumulator should be filled with fluid and the volume of the fluid should be measured and recorded. Pressure should then be applied to the gas port and the piston will be allowed to travel until it is bottomed on the fluid end. The volume of fluid exhausted should be measured and recorded. The volumetric efficiency should be determined by the following formula:

$$\text{Volumetric efficiency percent} = 100 \times \frac{\text{Volume of exhausted fluid}}{\text{Volume of admitted fluid}}$$

The volumetric efficiency should not be less than that specified in the detail specification.

8-4.4.3 Fluid Immersion. The accumulator should be filled with hydraulic fluid conforming to MIL-H-5606 or MIL-H-83282, in such a manner that all internal parts of the unit are in contact with the fluid. The accumulators should be immersed continuously in hydraulic fluid for a period of 72 hours at a fluid temperature of not less than 225°F in a closed container prior to conducting the remainder of the qualification tests specified herein. After the 72 hour soak period, the accumulators should remain in the fluid at normal room temperature until ready for test.

8-4.4.4 Proof Pressure. With the piston in approximately mid-position, completely fill both ends of each accumulator with fluid and plug the gas end. Fluid pressure should be applied at the fluid port until a pressure of 200 percent of the accumulator operating pressure is obtained and maintained for 5 minutes. There should be no leakage of fluid or sign of failure in any part of the accumulator.

8-4.4.5 Cycling and Endurance. With all fluid exhausted, the accumulator should be precharged with pure dry nitrogen gas as specified in the steps of Table VI. Fluid should then be cycled to the accumulator in accordance with the corresponding pressures and temperatures specified. The fluid and the gas in the accumulator should attain the specified temperature before tests are started. The tests should be conducted in the order specified. The stabilized gas temperature, the pressure, and leakage of fluid and gas should be recorded and should not exceed that specified in Table VI. There should be no external leakage or malfunctioning of the accumulator during this test. Only one set of packing gland seals should be used throughout tests 1 through 6.

Table VI. CYCLING AND ENDURANCE TESTS (See note 1)

Step No.	Total Cycles (minimum)	Cycling Rate (cycles per minute)	Gas Charge (PSI ±25 psi)	Fluid Pressure Cycling Limits (psi) Lower Limit Max	Fluid Pressure Cycling Limits (psi) Upper Limit Min	Temperature (°F ±10°F) Fluid	Temperature (°F ±10°F) Ambient Air	Leakage of Fluid to Gas Side (% of accumulator's volume) (see note 5)	Leakage of Gas (% drop in gage pressure)
1	2,000	.2 to 2	1,000	200	3,000	275	(note 2)	2	1
2	Gas Charge Leakage Test (4.x.x)								
3	50	(note 3)	500	200	3,000	-65	-65	0.5	5
4	500	.2 to 2	500	200	3,000	-40	-65	1	3
5a	2,500	3 to 10	1,000	200	3,000	275	(note 2)	3	3
5b	7,500	3 to 10	1,000	200	3,000	225	(note 2)	3	3
6a	12,500	3 to 10	500	2,600	3,000	275	(note 2)	2	3
6b	37,500	3 to 10	500	2,600	3,000	225	(note 2)	2	3
7a	12,500	OPTIONAL	(note 4) N/A	10	3,500	100	(note 2)	N/A	N/A
7b	500,000	OPTIONAL	N/A	200	3,000	275	(note 2)	N/A	N/A
7c	1,000,000	OPTIONAL	N/A	2,00	3,000	100	(note 2)	N/A	N/A

Notes:

1. Pressures shown are for 3,000 psi accumulator. For higher pressure accumulators, test pressures should be per the detail specification. During all cycling tests, the gas side of the accumulators should be lubricated with an amount of fluid equal to approximately .50 of 1 percent of accumulator rated volume. Fluid leakage should be determined by draining through the gas port without disassembling the accumulator. Gas pressure should be measured at stabilized and identical temperature before and after each step of Table X.
2. Ambient temperature should be maintained such that the gas temperature equals or exceeds the fluid temperature at the end of each compression stroke.
3. The accumulators should be charged to a minimum of 3000 psi fluid pressure. Accumulators should be maintained in this condition for 24 hours at the temperature specified in step 3. The 50 cycles should be fast discharge followed immediately by recharge with oil at the specified temperature. A 2 hour minimum interval should elapse between each of the cycles.
4. The accumulator piston should be removed for all tests of step 7. The gas port should be capped during this test. This test is intended to prove design of the accumulator shell construction and end cap construction. The rate of pressure buildup and the peak pressures should be recorded at the start and finish and at least 10 equally spaced intervals during each test. The pressure buildup rate should be between 100,000 to 200,000 psi per second
5. Be sure not to count the lubrication fluid on the gas side as leakage.

8-4.4.6 Gas Charge Leakage. Internal gas leakage for pressures up to the proof pressure should be as specified in the detail specification for temperatures of -65° to 275°F. There should be no external leakage.

8-4.4.7 Fluid Leakage. Static and dynamic leakage should be as specified in the detail specification for pressures zero to proof pressure and temperatures -65° to 275°F respectively.

8-4.4.8 Seizing of Parts. All accumulators should be tested for seizing of parts throughout the temperature range of 275° to -65°F. The assembly should be maintained at a temperature no less than 275°F for at least 6 hours, after which 2 cycles of operation should be made. The sliding parts should be operated throughout the cooling period and at least 2 complete cycles after the 34-hour period. The assembly should then be rapidly warmed up to room temperature and operated during this period. There should be no evidence of malfunctioning or seizing of parts caused by thermal contracting or expansion of the parts. At least 10 full cycles of operation should be made during cooling and warm-up periods.

8-4.4.9 Burst Pressure. The piston should be approximately in mid-position and the gas chamber should be filled with fluid during this test. With the fluid and unit temperature stabilized at 275°F (minimum), pressure should be applied at the fluid port until a pressure of 400 percent of operating pressure is reached. The accumulator should not rupture.

8-4.4.10 Fragmentation. The accumulator should be pre-charged to the detail specification and should be hit with a .50 caliber projectile fired at a range of 25 to 50 yards. The projectile should have a muzzle velocity of 2700 to 2900 feet per second. The projectile should be tumbled and should not be considered tumbled unless the projectile produces an "in" hole at least .50 inch wide by 1.50 inch long. The accumulator should be supported in a manner similar to a typical aircraft mounting. Attached to the fluid port should be a length of tubing with a shutoff valve located 3 feet from the port. The projectile should be so directed that it will hit the fluid side of the accumulator approximately midway between the piston and the mounting strap. The accumulator when struck by gun fire as specified should remain in one piece and greatest dimension of the opening (cut and tear) created by the projectile should not exceed the dimensions of the hole (cut) created by the projectile by more than 3 inches in any one direction. Cutting should be considered as the actual section of the accumulator cut by contact with the projectile and a "tear" should be considered as an extension beyond the cut.

8-4.5 **Reservoirs.** The following test should be performed on reservoirs in addition to the qualification tests referenced in paragraph 8-4.2.

8-4.5.1 Negative Pressure. Reservoirs should be subjected to a negative pressure test. Starting with the separator held at the extreme over-full position, evacuate the system to 11 psia. There should be no influx of ambient air into the reservoir fluid storage chamber. The above check should be repeated at increments of travel up to full stroke. The reservoir should show no permanent deformation or influx of ambient air.

8-4.6 **Actuators, Flight Control.** Hydraulically powered servoactuators for use in aircraft flight control systems should be tested in accordance with SAE ARP 1281.