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REUSABLE ROCKET ENGINE MAINTENANCE STUDY

January 1982

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prepared for
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16. Abstract

Approximately 85,000 liquid rocket engine failure reports, obtained from 30 years of developing and delivering major pump feed engines, were reviewed and screened and reduced to 1771. These were categorized into 16 different failure modes. Failure propagation diagrams were established. The state of the art of engine condition monitoring for in-flight sensors and between-flight inspection technology was determined. For the 16 failure modes, the potential measurands and diagnostic requirements were identified, assessed and ranked. Eight areas have been identified requiring advanced technology development.

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FOREWORD

This work herein was conducted by the Engineering Department of Rocketdyne, a division of Rockwell International, under Contract NAS3-22652 from September 1980 through October 1981. Mr. J. P. Wanheinen and R. M. Masters, Lewis Research Center, were Project Manager and Assistant Project Manager, respectively. At Rocketdyne, Mr. F. M. Kirby as Program Manager, and Mr. C. A. MacGregor as Project Engineer, were responsible for technical direction of the program. Mr. M. Ionnitiu performed Task I; Mr. S. Barkhoudarian, Mr. J. R. McManus, Mr. J. Maram, and Mr. R. L. Phillips performed Task II; and Mr. B. D. Hines performed Task III.

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SUMMARY

Rocketdyne has reviewed the 85,000 failure reports which have resulted from

1. The development of eight different pump-fed liquid rocket engines
2. The delivery of about 2500 engines
3. The launch of over 1000 flight vehicles

over the last 30 years. These engine failure reports were reviewed, screened, categorized and were reduced to 16 failure modes and failure propagation diagrams, which were common to all engines.

A survey of the state of the art of sensors for in-flight and inspection techniques for between-flight engine condition moinitoring was performed. The in-flight sensors and the between-flight inspection techniques were assessed, matched, then ranked relative to their suitability for prognosis and diagnosis of the identified 16 failure modes. The highest ranked technology selections for both in-flight and between-flight were considered upgradable and the effort required to develop these technologies has been identified.

The eight technologies that are potentially applicable to rocket engines are:

1. Optical pyrometer for turbine blade temperature
2. Fiberoptic deflectometer for bearing condition
3. Isotope wear detector for wear particles
4. Tunable diode laser spectrometer for wear particles
5. Ultrasonic flowmeter for propellant flows
6. Ultrasonic thermometer for high temperatures
7. Holographic leak detector for fluid leaks
8. Scanning pyrometer for blocked fluid passages

INTRODUCTION

Future space transportation systems for low earth orbit must rely on reusable subsystems and routine ground operations to be cost effective. This can be achieved by avoiding high costs associated with maintenance on a basis other than for cause, and avoiding disassembly for routine inspection and premature component replacement. The approach to achieving substantial operations cost reductions by increasing rocket engine service life and reducing maintenance and turn-around time between flights is to incorporate engine condition monitoring. Engine condition monitoring includes both in-flight condition monitoring and between-flight inspection. This study was conducted for the purpose of identifying technology advancements in engine condition monitoring needed to minimize liquid rocket engine maintenance.

There has been a long history of development activity directed toward aircraft air breathing engine monitoring systems. Several Air Force aircraft/engine systems as well as engine-alone systems have been implemented recently through prototype and operational applications. These systems have been directed toward reducing propulsion support costs and improving aircraft operational availability. Similar activity has existed with commercial airlines. However, prior to the advent of the Space Shuttle and the Space Shuttle Main Engines, no large requirement for reusable liquid rocket engines existed. The Space Shuttle is bringing about new requirements.

This study was undertaken to identify needed technology advancements in engine conditioning monitoring. The efforts (1) reviewed past rocket engine failures modes, (2) identified state-of-the-art technology for in-flight engine condition monitoring sensors and between-flight inspection techniques to detect incipient component failures, and (3) identified areas where advancement in monitoring and inspection technology is required.

The study was performed in four tasks:

- Task I - Review and Characterization of Past and Present Rocket Engine Failures
- Task II - Identification and Evaluation of In-Flight Condition Monitoring Sensors
- Task III - Identification and Evaluation of Between-Flight Inspection Techniques
- Task IV - Eight Technologies Recommended for Additional Development Effort

The study was performed during part of 1980 and 1981.

DISCUSSION

CHARACTERIZATION OF ROCKET ENGINE FAILURES

The objectives of Task I of this study were to identify engine failures of main propulsion booster and space engines, regardless of propellant combination, to categorize these failures, and to investigate and evaluate the failure modes in order to conduct an assessment of state-of-the-art technology of the in-flight engine condition, monitoring equipment and inspection techniques.

To perform this task it was necessary to draw upon the Rocketdyne Reliability Data bank for applicable failures, to categorize the data in some meaningful way, to reduce it to a manageable size, and to unravel the propagation of the applicable failure modes to assist the investigation of the monitoring techniques.

These data were to be submitted in an agreed-upon format that would simplify the performance of the subsequent tasks of the study, and would record the results.

FAILURE ANALYSIS PROCEDURE

1. Definitions

Since Rocketdyne has had a relatively long and rich experience in rocket engine development, testing and production (Fig. 1), it was decided to review and evaluate the reports associated with engine failures that could provide a basis for the study.

The failure data accumulated at Rocketdyne over the years was estimated to be in the neighborhood of 100,000 pieces of information. It became apparent that some ground rules and screening were required to handle the mass of data in some consistent manner to obtain meaningful results to support the study.

The first decision was to select a definition for failure which would be consistent with the approach. The definition is:

> "Failure is the inability of equipment to satisfy performance or design specifications once the equipment has experienced successful operation or acceptance or has the expectation of successful performance without adjustment or rework."

This definition permits the reporting of failures, which have been noted during operation, as well as the reporting of conditions which would result in a failure if operation were permitted.

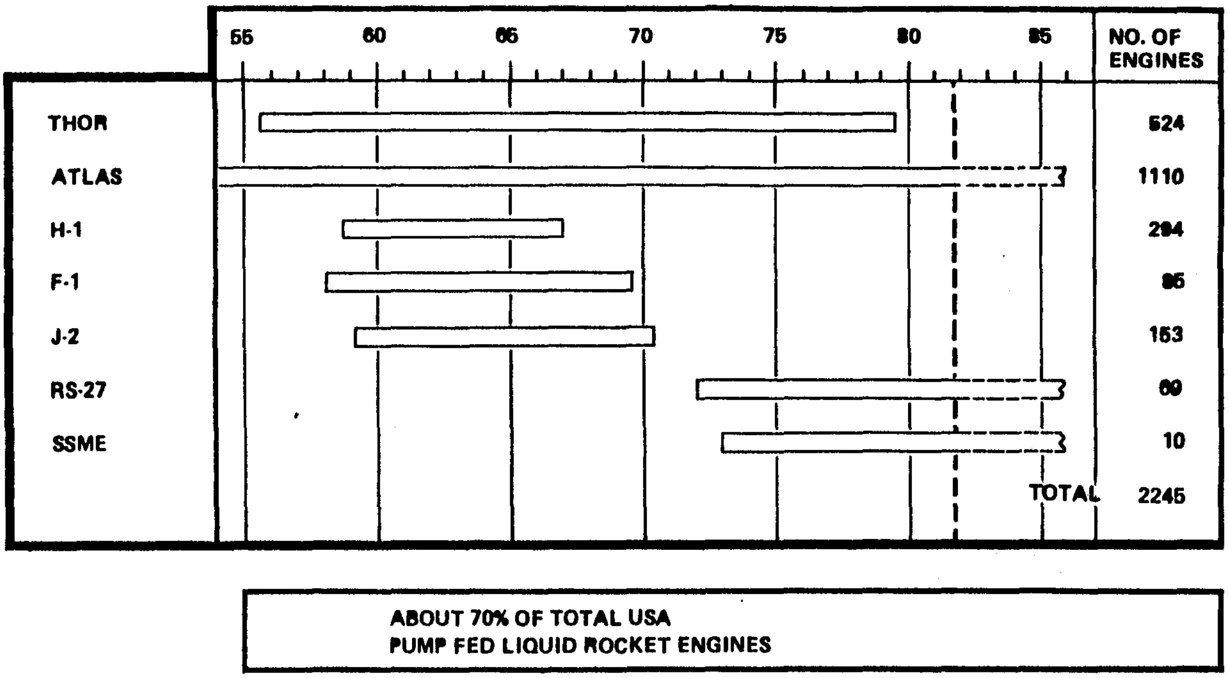

Figure 1. Rocketdyne--30 Years of Delivering Engines

The definition would become a first screen of the data by eliminating the trouble reports in the system which are generated as a result of rejection of hardware due to improper paperwork, cosmetic discrepancies (scratches on paint, lack of torque stripe, etc.). While the boundaries of the failure were determined, the criticality of the failure had also to be defined.

Since the format selected for presentation of the data required assigning criticality factors, these were defined as:

Category 1 = loss of life or vehicle

Category 2 = loss of mission (includes both post-launch abort and launch delay sufficient to cause mission scrub)

Category 3 = all others

2. Raw Data Base

Before examining the criteria for subsequent screens, it is helpful to describe the life cycle of an engine at Rocketdyne and in the field, up to the point where the engine is expended, and to establish at which point failure reports, or Unsatisfactory Condition Reports (UCRs) originate.

Figure 2 shows the typical activity to which the hardware, which constitutes an engine, is exposed. Separate pieces of hardware are received and tested prior to assembly in subsystems or assemblies in Receiving Inspection operations by the Quality Assurance organization. Once a component or parts thereof are deemed acceptable, they are ready for assembly into larger components, subsystems, systems and finally into a complex system. The engine UCRs are written only when the component, having once demonstrated its ability to function according to specification requirements, fails to meet these requirements.

Components are functionally checked during assembly and subsystems are further tested. Turbopumps are calibrated and assembled, and subjected to a so-called "green run" on the component test stand. The green run is the initial hot fire test of the assembled turbopump, verifying its ability to deliver the desired performance. The turbine receives its working fluid from a slave gas generator: while the fuel pump delivers fuel, the LOX side pumps water. After successfully passing the green run tests, the assembled turbopump is returned to the shop to be mounted on the engine. The thrust chamber and injector are also calibrated separately, in water tests, to determine the Delta-P and are then ready for assembly. For many years the gas generator, the component that delivers working gas to the turbine, was tested separately to determine its performance.

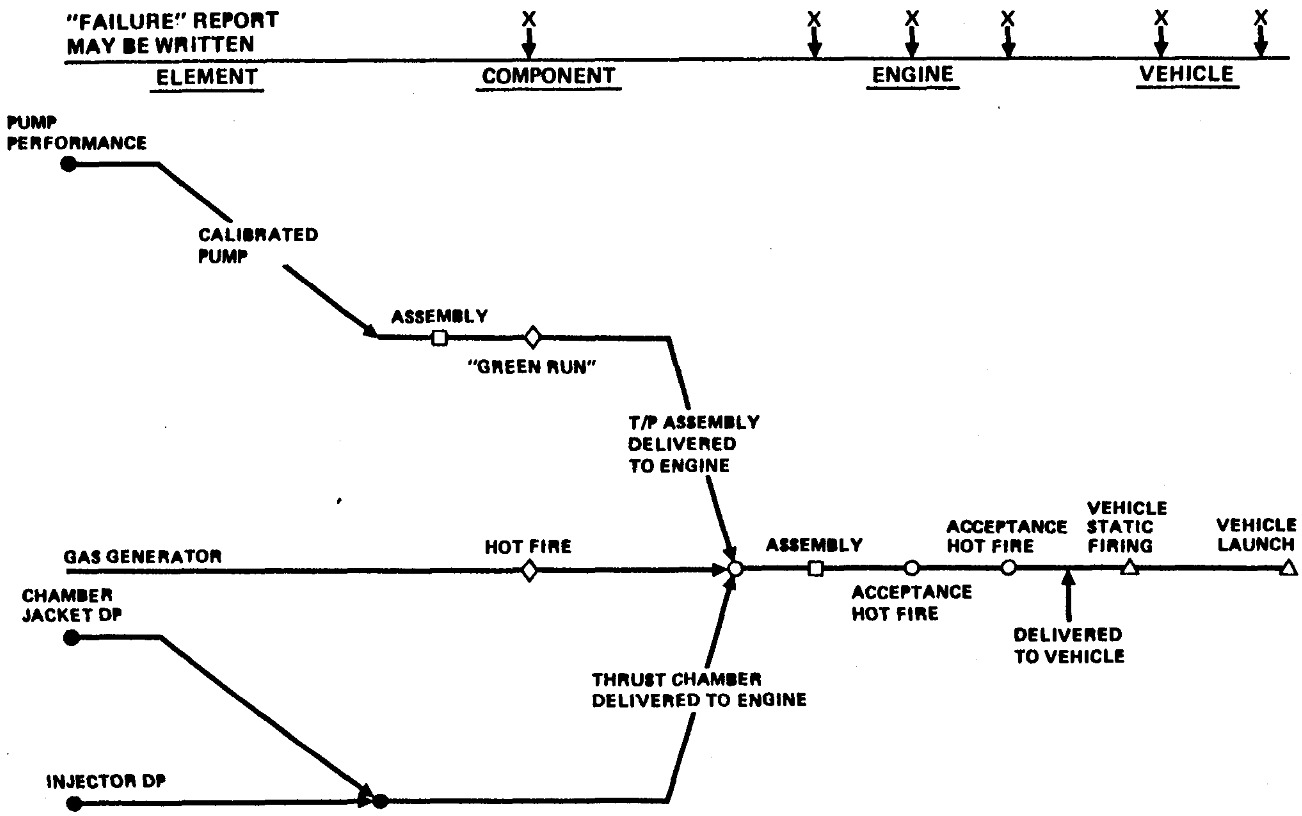

Figure 2. Gas Generator Cycle Rocket Engine Test Activity

All these components, together with the thrust mounts, valves, controls, lines and ducts, electrical components and harnesses are assembled into an engine. UCRs are always written when a component does not meet any requirements. Finally, the completed engine is subjected to a series of electromechanical and leak checks, usually called the first E&M check. After passing this series of tests, the engine is mounted on a test stand and subjected to a minimum of two hot fire tests to verify engine performance and operation. Any nonconformance is written up as an UCR. After completing the hot fire, or acceptance tests, the production engine is returned to the shop for another series of leak and electromechanical tests, the second E&M check. After successfully passing these tests, the engine is delivered to the customer. At this point, the engine is transported to another contractor, where it passes receiving inspection tests, is installed in the vehicle and subjected to a new series of electrical, mechanical and leak checks. During this time, any discrepancy is written up as an UCR.

Subsequently, the integrated vehicle is transported to the launch site, the payload is installed and the engine goes through the final series of checks prior to countdown sometimes including static firing. Further UCRs may be generated during this time, until, in conventional rocket engines, the engine is expended in launch.

As it can be seen, the Rocketdyne failure reporting system is designed to record on UCRs nonconforming conditions at various stages of the engine life. In addition, the UCR provides disposition for the discrepant hardware; it outlines corrective action against future similar occurrences and supplies trend data. Figure 3 shows two UCRs from the SSME data file, indicating all the information that is recorded regarding the discrepancy.

At this point, the selection of the engine systems, from which the UCR data were going to be evaluated, was also made. Based on the study requirements, it was decided to use failure data from large liquid rocket engines; that is, systems that utilize pumps for propellant feed, rather than being fed from pressurized tanks. The engine systems selected are based on similarity of engine operation and of component configuration with the hypothetical reusable rocket engine.

The information retrived from the computerized Reliability Data Repository was limited to current engine systems still in production as well as those that had been designed for manned application in the Apollo program. Data from discontinued pump-fed engines, such as the Navajo, Jupiter and Redstone, were not used since they are too far removed from the current concept of rocket engines. The selected engine systems are:

1. SSME used in the Orbiter Vehicle
2. J-2 used in the Saturn Ib and V Vehicles
3. H-1 used in the Saturn Ib Vehicle
4. F-1 used in the Saturn V Vehicle
5. RS-27 used in the Delta Vehicle
6. Thor used in the Thor Vehicle
7. Atlas used in the Atlas, Atlas-Centaur Vehicles

These systems are described briefly in Appendix A.

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UCR NO FAIL DATE ENGINE VEHICLE	PART NO SERIAL NO OPERATION TEST NO	PART NAME COMP CODE CC/RA/ST LOCATION ASSIGNMENT / DEPT	FAILURE MODE REMEDIAL ACTION CAUSE RECURRENCE CONTROL
A01C057 03-12-79 2003	RSC08255-161 4889291 HOT FIRE 901230	MAIN OXIDIZER VALVE D120 2 / B / CL NSTL #1 A OLSEN /585193	MOV VIBRATION CUTOFF TBC VIBRATION INDUCED MECH STRESSES REDESIGN: ECP 271, 278 & ECP C9686

PROBLEM DESCRIPTION

DURING ENGINE HOT FIRE TEST 901-230, A PREMATURE ENGINE CUT-OFF OCCURRED AT 18.32 SECONDS OF A SCHEDULED 60 SECOND TEST. ENGINE TEST WAS CUT-OFF BY MOV VIBRATION, SEQUENCES 691 & 692.

FAILURE ANALYSIS

HIGH VIBRATION LEVELS IN THE MOV ARE TRIGGERED BY ACOUSTICS CAUSED BY HIGH FLOW VELOCITY, VALVE GEOMETRY AND VALVE-LINE INTERFACE GEOMETRY.
RECURRANCE CONTROL: DESIGN CHANGES TO REDUCE THE G LEVELS INCLUDE A VALVE OPEN STOP, BALL SEAL MODIFICATIONS, LARGER BALL INLET RADIUS (MCR-SSME-2711) AND A THICK WALL INLET SLEEVE WITH A LARGE INLET CHAMFER (MCR-SSME-2781). ECR C9686 INCORPORATES A SHIM AT THE MOV INLET TO FILL THE INTERFACE CAVITY WHICH ACTS AS A NOISE SOURCE AND TRIGGERS MOV VIBRATION

IDCR# 128	901230	03-12-79	D12C	A01C057

PRINT DATE 09-11-80

UCR NO FAIL DATE ENGINE VEHICLE	PART NO SERIAL NO OPERATION TEST NO	PART NAME COMP CODE CC/RA/ST LOCATION ASSIGNMENT / DEPT	FAILURE MODE REMEDIAL ACTION CAUSE RECURRENCE CONTROL
A010816 12-27-78 2001	RSC08255-091 4054963 POST TEST 901225	MAIN OXIDIZER VALVE D120 1 / A / CL NST A OLSEN /585193	FIRE ORIGINATING IN MOV TBC FLOW INDUCED VIBRATIONS REDESIGN ECP SSME 248, 258 & 271

PROBLEM DESCRIPTION

DURING POST TEST INSPECTION 901-225, FIRE AND EVIDENCE OF AN EXPLOSION WAS DISCOVERED IN THE AREA OF THE MOV.

FAILURE ANALYSIS

FAILURE ANALYSIS HAS DISCLOSED THAT ONE SCREW SECURING THE INLET SLEEVE TO THE BELLOWS HAD BACKED OUT DUE TO HIGH FLOW VIBRATION CAUSING THE SHIMS BETWEEN THE SLEEVE AND BELLOWS TO FLUTTER RESULTING IN A FRETTING CONDITION. THE HEAT GENERATED BY THE FRETTING PRODUCED IGNITION OF THE SLEEVE OUTER DIAMETER RESULTING IN FAILURE AND EXTENSIVE DAMAGE TO THE ENGINE.
RECURRENCE CONTROL: DESIGN MODIFICATIONS TO REDUCE FLOW VIBRATIONS AND TO PREVENT FRETTING WERE INCORPORATED BY ECP-SSME-248, 258 & 271.

NONE	901225	12-27-78	D120	A010816

Figure 3. Sample of UCR (SSME Oxidizer Main Valve)

After excluding other data from programs that did not fit the initial criteria, that is UCRs originating in programs related to the engine systems listed above, the data base consisted of some 84,000 pieces of information.

3. Data Screening

Concentrating on the objective of the study, it was desirable to use only UCRs which could provide the basis for investigation of rocket engine sensors. Since the UCR search covered an extensive time period during the development stages and the production of these engine systems, it was desirable to concentrate on the failures originating during the operational phase of the engine.

Screen No. 1. The criteria to eliminate failure reports of components and engines of experimental configuration were established. The most cost-effective way was to retain the data originating from production and flight configuration engines. This allowed an automatic sort of the data stored on computer tapes. The criteria for sorting the failure data were by engine serial number denoting production/flight engine systems as follows:

SSME - All 2xxx series plus flight configuration engines 0006, 0008, 0009

Atlas - All 11xxxx and 22xxxx series engines

RS-27 - All 00xx series

F-1, J-2, H-1 - All 4 digit series

Thor - All 6 digit series

A further screening of the Atlas and H-1 engine data removed information related to earlier models as unsuitable for the analysis.

Figure 4, indicates the phase from which the failure data was drawn, relating it to reliability growth. Because data from mature engine was desirable, UCRs from the early life of the program were excluded.

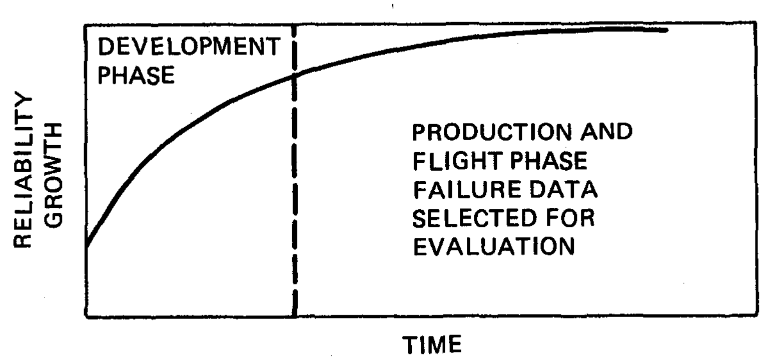

Figure 4. Failure Data Selection

Screen No. 2. The next screening step was to select failure data from hot fire tests, flight/launch operations and post hot fire tests. Each production engine, before being delivered, is subjected to one or more hot fire tests to determine its operational characteristics and to verify the integrity of the system.

These tests, performed at the Santa Susana Field Laboratory (SSFL) test stands (F-1 testing was performed at Edwards Air Force Base facility) and called acceptance tests, were found most suitable for the study. The acceptance tests duplicate, as much as possible, the operation of the engine systems during launch and flight with the exception of duration and acceleration.

Screen No. 3. Another screening step resulted from elimination of the failure reports due to causes that would obscure the goal of the investigation. The following categories were excluded as unsuitable:

1. Procedural problems
2. Human error
3. Facility and vehicle discrepancies
4. Low frequency failures (one-time occurrences)
5. Experimental hardware or procedures
6. Secondary failures
7. Obsolete hardware
8. Information type instrumentation failures

Screen No. 4. In addition, because the bulk of the failure data thus obtained originated from the expendable rocket engine experience, the UCRs were screened with respect to their impact on reusability. Also, where design information exists, the life of the component was compared to its design life. In addition, conditions that could have been monitored to detect the incipient failure were listed.

Figure 5 shows the original data base that was available for the study and the reduction in numbers after the successive screening operations.

4. Data Base After Screening

Figure 6 shows the distribution among the several engine systems of the 1771 UCRs that were left after screening. It is not surprising to note that over three-fourths of the retained UCRs come from engine systems that represent 85% of the delivered engines.

ENGINE SYSTEM	INITIAL NO. UCRs	AFTER SCREEN 1 & 2 **	AFTER SCREEN 3	4
SSME	5,600	951	288	101
J-2	16,321	2,288	148	127
F-1	13,140	1,279	103	102
H-1	9,751	1,521	326	326
THOR*	12,029	1,849	476	474
RS-27	1,264	260	108	108
ATLAS*	26,274	819	541	533
TOTAL	84,379	8,977	1990	1771

*ONLY LATEST MODELS CONSIDERED APPLICABLE
THOR MB 3-1, THOR MB 3-3
ATLAS MA-3, ATLAS MA-5
H-1 205K
** SCREENS 1 AND 2 WERE PERFORMED SIMULTANEOUSLY BY APPROPRIATE CRITERIA DURING COMPUTER RETRIEVAL OF DATA.

Figure 5. UCRs Applicable After Screening

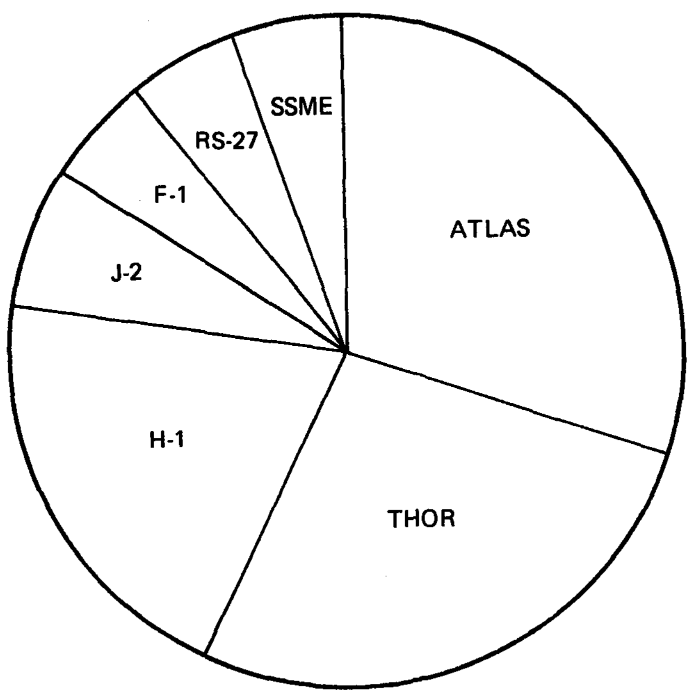

Figure 6. Distribution of UCRs by Engine Systems After Screening

FAILURE CHARACTERIZATION

Figure 7 presents the format for evaluation of each failure mode at the Engine System and Component Level, and for listing the viable in-flight condition monitoring systems and between-flight inspection techniques that would be capable of detecting an incipient failure. Each failure mode represents an event during which the respective engine system failed to perform according to specifications.

Each component failure mode was evaluated and the following was determined:

1. If the failure was predictable or unpredictable
2. If the failure would be detectable in flight, on the ground, or not at all
3. If the failure was functional or operational
4. If the failure was primary or secondary
5. If the failure caused performance degradation or was catastrophic.

In addition, where design information exists, the life of the component was compared to its design life. Conditions that could have been monitored to detect the incipient failures are also listed. The successive screenings of the UCRs written against the failures of the matured applicable engine systems have reduced the number of pieces of information from over 84,000 to 1771.

The UCRs that passed all successive screening steps were grouped by engine systems prior to making an individual assessment. Within each engine system, the UCRs were analyzed and the failure was evaluated with the aid of drawings, schematics, exploded views and test data, when available. In addition, to assist in understanding the failure mechanism, an analysis of the sequence of events leading to the incident was made using a graphic illustration. This is later described in Failure Propagation Block Diagrams, and shown in Fig. 8.

Based on the assessment, the UCRs were grouped in failure modes. The operation was completed separately for each of the engine systems selected for the study. The resulting failure modes were then compared and integrated. This effort resulted in 16 modes for 1771 UCRs.

All information collected from the failure reports was thus included in the forms, called summary sheets, that are submitted in Appendix B. A brief description of each failure mode is presented below.

1A. Bolt Torque Relaxation

A main oxidizer valve which controls the flow of propellant to the main combustion chamber, caught fire during engine operation, requiring premature engine shutdown. Investigation of the failure disclosed that a screw that secures one of the internal seals of the valve had become loose as a result of cavitation and vibration, and allowed fretting of aluminum assemblies in a liquid oxygen environment.

ENGINE SYSTEM/COMPONENT SSME/Nozzle/Combustor

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
2. <u>Coolant Passage Leakage</u> Nozzle Tube Splits and Ruptures Caused by material embrittlement from prior repairs, start and shutdown transient surges, contamination clogged tubes, intermittent braze with regions of non-braze.	4f 2.23%		Loss of fuel at nozzle tubes caused premature engine cutoff due to HPOT discharge temperature exceeding redline.	Overtemp and leakage.	Primary	3	Imm.	HPOT turbine Temp cutoff	No incorporated system could predict reliably and expeditiously	Metal Embrittlement Pressure Transient *Tube Splits Flow, Reduction Mixture Ratio Shift Temperature, Rise In Combustion
Caused by thermal strain and/or braze porosity because of prior repairs, leakage through braze joint due to insufficient bonding during braze cycle.	30f 16.75%		External fuel leakage would result in fire hazard and would cause performance degradation, and coolant loss.	Thermal strain	Primary	4 or potential 1	N/A	Post-test inspection		*Are Detectable Between Flight Only

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Ultrasonic Thermometer (Flame) Ultrasonic Flowmeter (Nozzle) Polarometer Tunable Diode Laser Spectrometer (Mixture Ratio)	Ultrasonic Leak Acoustic Holography X-ray Radiography Gamma Radiography Pentoxide Polarography Hydrogen Polarography Hygrometer Optical Pyrometry Holographic Leak Millimeter-wave Interferometry	

Figure 7. Sample Failure Summary

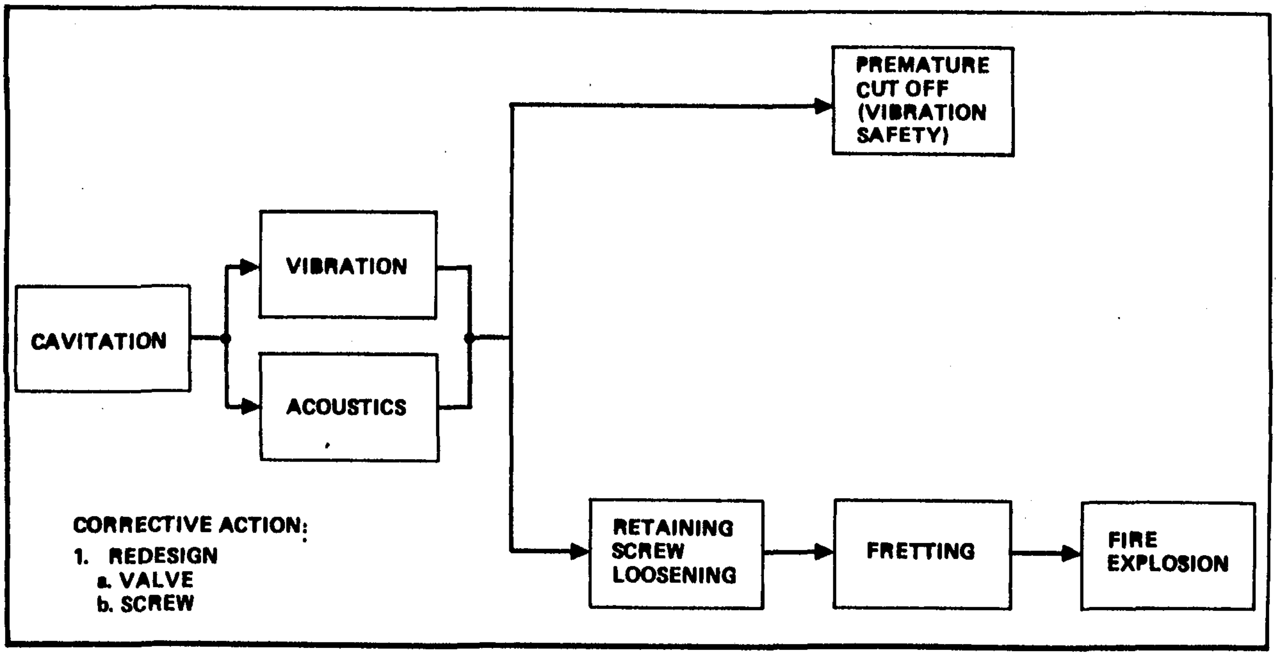

Figure 8. Sample - Failure Propagation Block Diagram

The rubbing of the two metal parts caused heat and subsequent fire, and extensive damage. The valve was redesigned to eliminate this failure mode.

1B. Bolt Torque Relaxation

In another engine system, the same failure mode, bolt torque relaxation, caused the loosening of a seal retainer in the sequence valve of the main oxidizer valve, as a result of excessive vibration due to high flow velocity and/or engine vibration. This, in turn, allowed the pressurant gas (helium) to escape prematurely to the gas generator control valve open port, actuating this valve out of sequence. Premature operation of the gas generator control valve caused a detonation in the combustor with damage to the assembly.

2. Coolant Passage Splits

The thrust chamber assembly, common to all engine systems, is the component that transforms the energy stored in the fuel into kinetic energy. This fluid is contained and directed by the sides of the thrust chamber. During engine operation, the thrust chamber is exposed to high pressure, to vibration, to high temperature all in a brief period of time. The strength and the cooling capability of the thrust chamber is achieved through an ingenious design.

Containment is accomplished by a series of circular bands that ring the thrust chamber, the lightness by using thin walls, and the cooling by recirculating fuel. The requirement of lightweight and high heat transfer capability is achieved by constructing the thrust chamber walls of tubes through which the fuel circulates. High thermal strains, stresses induced by vibration, surges during the ignition and transition stages, containment within the tubes causing obstructions and material deficiencies are some of the causes of failure of the cooling passages. Loss of coolant through the thrust chamber walls may cause loss in engine performance and loss of cooling capability which will lead to engine failure.

3. Joint Leakage

All liquid propellant engine systems suffer from leakages from the interfaces of the propellant and pressurant fluid ducting. Defects in material, improper installation causing damage to seals or sealing surfaces, warping or distortion of sealing surfaces due to thermal strains during engine start or operation, fastener torque relaxation during engine operation are causes of this failure mode. Effects on this failure mode vary depending upon the location and the type of leakage, and some of them have had catastrophic consequences.

4. Hot Gas Manifold Transfer Tube Cracks

This failure mode is peculiar to one engine, which utilizes double-walled ducts to convey hydrogen-rich hot gases from the preburners to the high pressure fuel turbopumps. Excessive high temperature transients have caused hot spots or cracks on the inner wall (liner) which may evolve into a complete failure of the component with catastrophic effect.

5. High Torque

High torque, as a result of rubbing the labyrinth seal in propellant pumps has been experienced in several engine systems. The seal consists of a series of land and grooves designed to minimize leakage from the high pressure side of the pump to the low pressure inlet side. Excessive temperature and vibration can lead to friction between the static and rotating parts thus increasing the torque of the turbopump with eventual subsequent failure.

6. Cracked Turbine Blades

Generally, turbine blades are subjected during start and main stage operation to high energy transients which could be due to pressure, temperature, or accoustical spikes leading to failure as a result of localized heating of turbine parts. Impact on turbine blades of debris and contaminants in the hot gases has also caused damage, with resultant loss of efficiency and imbalance of the turbine.

7. Failure of Bellows

Flexible ducting is used to convey propellants between some components to avoid problems that afflict rigid ducting. The bellows are damaged by high-cycle fatigue, which is caused by high energy transients and high flow velocity. The effects of this mode of failure vary depending on the location of the duct, as well as on the type of fluid conveyed through the bellows.

8. Loose Electrical Connectors

Some engines are more dependent on electrical controls than the previous generation of rocket engines, and have encountered instances where failures were reported caused by incomplete electrical circuits. The failures were due to connectors that loosened as a result of vibration from engine operation. The consequences of this mode of failure vary according to the affected electrical circuit.

8. Bearing Damage

Excessive loading of bearings in the highly stressed turbomachinery may lead to wear-out and eventual failure. Contributing factors are excessive axial and radial loads, vibration, and friction, and the effects are generally catastrophic if not detected in time.

10. Tube Fracture

The failure mode noted on one engine occurred on a particularly sensitive component that caused premature engine operation cutoff, and was the result of vibration-induced fatigue.

11. Turbopump Face Seal Leakage

In the engine systems that utilize turbopumps to convey the propellant under pressure to the combustion chamber, it is imperative to prevent leakage along the rotating shaft. The seal leakage is especially critical in the engine systems that use a common shaft to power the fuel and the oxidizer pumps, because the mixing of propellant at that location has catastrophic consequences. The cause of the seal leakage is generally due to excessive temperature gradients, vibration, friction, or interface material damage since the seal has to prevent leakage in both a static and in a dynamic condition.

12. Lube Pressure Anomalies

Several anomalies in the lube (oil) system have been noted during this investigation, and all have been grouped in the same category as they affected primarily a subsystem peculiar to certain types of engine systems. The lubrication subsystem delivers oil under pressure to the turbopump gearcase for lubrication and cooling gears and bearings through jets and nozzles. The failures included in this category consist mainly of obstruction of the flow due to contamination.

13. Valve Fails to Perform

In this category, failures of different valves were included. These failures were mainly caused by contamination or excessive friction. The effects upon the engine performance vary depending upon the function of the part.

14. Internal Valve Leakage

This category comprises all those incidents in which an engine failure was due to internal leakage within valves. The several failure mechanisms which can lead to this condition are so noted in the summary sheet.

15. Regulator Discrepancies

All regulator failures were grouped in this category since they pertain to a subsystem that is used in a few of the engine systems included in this study. The function of the regulator is to reduce pneumatic supply pressure to a required level and to maintain it at that level throughout engine operation. Malfunction was caused mainly by contamination.

16. Contaminated Hydraulic Control Assembly

The failures that were included in this category are peculiar to one engine system. The hydraulic control assembly receives and directs hydraulic control pressure in the proper sequence for the operation of the engine main valves during start and shutdown and controls also their position during mainstage operation.

Viewing the number of UCRs that fell into the different categories gives an interesting picture of relative magnitude. The pie chart (Fig. 9) depicts the distribution of UCRs by failure mode and verifies what was known from previous experience; that the major problem that plagues liquid propellant rocket engines is leakage. Over three-fourths of UCRs are related to leakage, either internal to components, or external from joints. Thus the dependence on leak testing the engine systems at various stages of their life, and the reluctance to break into a subsystem for minor reasons, disturbing proven joints, is justified.

All summary sheets for the 16 failure modes are presented in Appendix B.

Revision to Failure Modes Listing

Further evaluation of the 16 failure modes indicated that some should be dropped from the analysis. Consequently, failure modes 1 and 4 (bolt torque relaxation, and hot gas manifold transfer tube cracks) were excluded because the corrective action taken in both cases was redesign. Thus, under previously established ground rules they would have been eliminated as not meeting the mature engine definition. They had escaped the screening because the redesign occurred on flight engines, which were by definition considered to be mature.

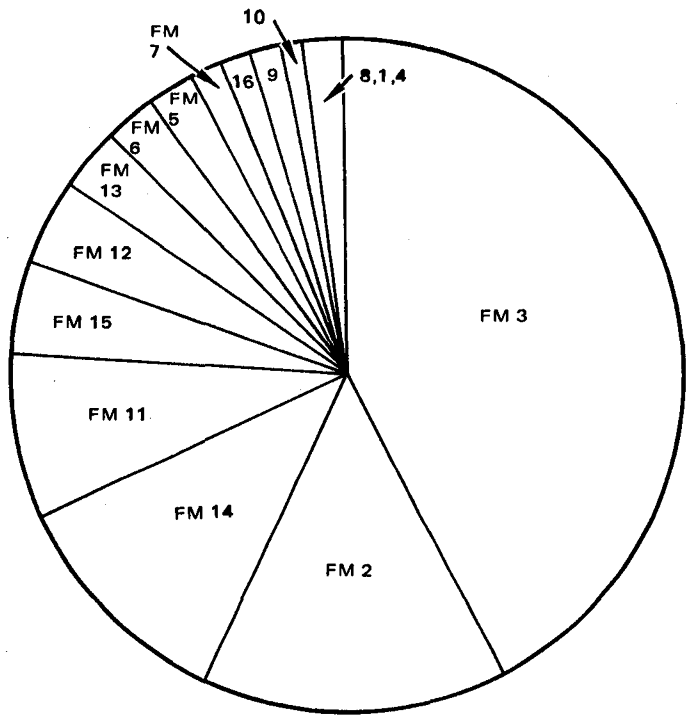

Figure 9. Distribution of UCRs Among Failure Modes

Failure Mode 12, Lube system anomalies, was also deleted because it is not applicable to a reusable engine.

It should also be noted that the Failure Mode and Effect Analysis of every engine system under consideration indicates other possible failure modes in addition to the 16 that were determined for this study. These additional failure modes have not been experienced in testing the mature engines, therefore, it has been established that further investigation was not warranted for an occurrence that has an extremely low frequency.

In conclusion, the screening process has been effective in reducing a large amount of failure data to a manageable number by following a series of logical steps saving only that information that could contribute significantly to the study. It also showed that most failure modes are common within the various engine systems. This gives more confidence in selecting these occurrences for determination of suitable in-flight condition monitoring devices and between flight inspection methods and equipment.

FAILURE PROPAGATION BLOCK DIAGRAMS

To support the study for the applicable in-flight condition monitoring devices, a method for depicting failure modes was provided to indicate the sequence of contributing events.

These events, usually an anomalous system performance, show the relationship between a symptom which could be monitored and the eventual failure of the engine system to perform. The analysis method was to attempt to slice the period of time in which the failure develops into small increments and survey the changes that occur. Isolating the contributing factors in time assisted in the selection of suitable sensors. As depicted in Fig. 10, the events are shown as rectangles, and the sequence, left to right, indicates passage of time.

The failure propagation block diagrams included in this report (Appendix C) are typical for each failure mode listed and were not repeated for each different engine system, which may have a similar mode for slightly different components.

Flight Failures

To complete the failure investigation of the concerned engine systems, it was deemed necessary to examine also the flight failures caused by these systems over the years, since these occurrences were not covered by UCRs but by special reports. This examination resulted in preparation of failure propagation block diagrams similar to those discussed in the previous pages. These charts do not indicate sensing devices, since most of the flights were boosted by engine systems developed for military use, which carried limited instrumentation. The theory under which these engine systems were designed and tested was to get the vehicle off the launch pad whether it was functioning properly or not. The engines were generally devoid of monitoring devices and the shutdown controls were inactive before a predetermined operating duration, the alternative being the destruct button.

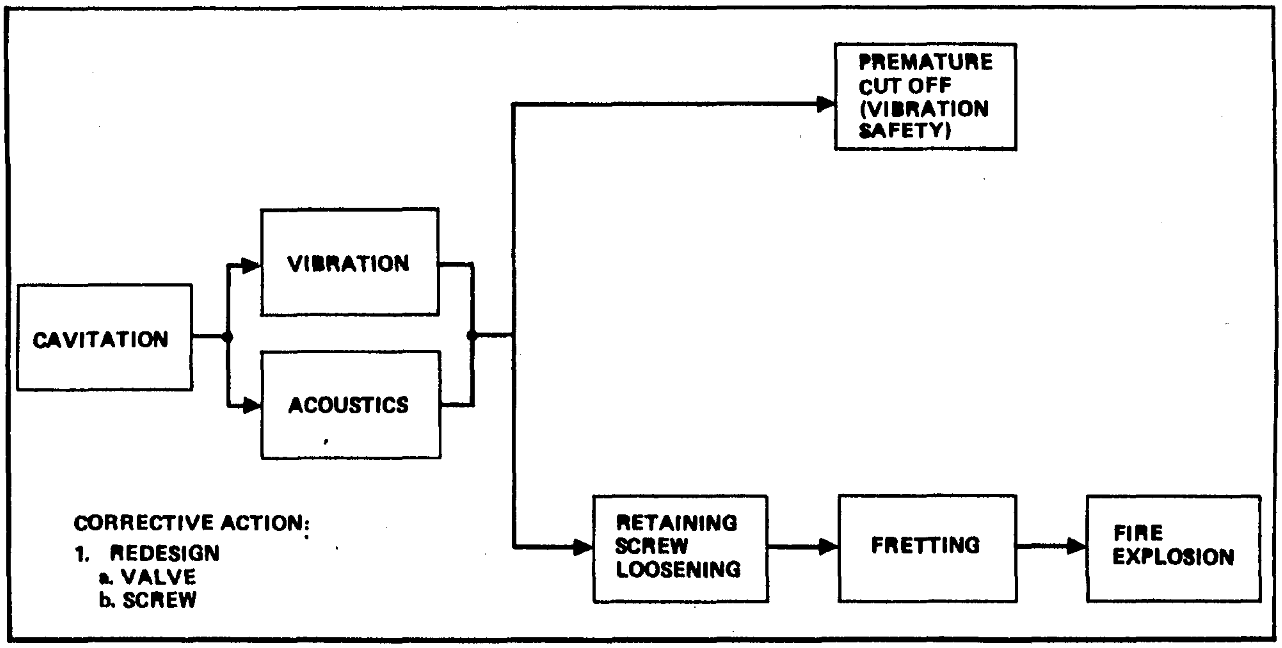

Figure 10. Sample Failure Propagation Block Diagram

It can be clearly seen that the charts illustrating the failures indicate possible points in the propagation of the failure where detection of the incipient failure could have limited damage to the engine and to the vehicle.

FAILURE DATA CONCLUDING REMARKS

The analysis and evaluation of the 1771 failure reports screened from the tens of thousands of UCRs in the Reliability Data Repository resulted in a summary presented in Fig. 11 and accounts for only 16 failure modes.

Examination of the data surveyed during the performance of Task I did not reveal any surprises, except that the expected number of failure modes encountered in 30 years of testing and flying rocket engines was rather small.

As expected, the Atlas engine system, which consists of 3 separate engines, exhibited most failure modes - eleven.

Another feature that appeared during this investigation is the commonality of failure modes among the various engine systems. That is not surprising, since the similarity in configuration between most of the engines. The failure modes that appeared only on one engine system are those that occurred on a component peculiar to one system (like hot gas manifold transfer tube cracks, or Hydraulic Control Assembly) or because the engine system is more dependent than others for successful operation on proper functioning of the component (Failure Mode 8, Loose Electrical Connectors).

FAILURE MODE CATEGORY	FAILURE MODE DESCRIPTION	ENGINE SYSTEM							TOTAL MODE
		SSME	J-2	H-1	F-1	RS-27	THOR	ATLAS	
1	BOLT TORQUE RELAXATION:								
	A MAIN OXIDIZER VALVE	3							3
	B SEQUENCE VALVE		3						3
2	COOLANT PASSAGE LEAKAGE	34		38	6		76	105	259
3	JOINT LEAKAGE:								
	A. HOT GAS	5	9	61	22	28	27	79	231
	B. PROP. & LUBE HYDR.	12		68	43	40	219	148	530
4	HOT GAS MANIFOLD TRANSFER TUBE CRACKS	3							3
5	HIGH TORQUE, T/P	20					11	10	41
6	CRACKED TURBINE BLADES	9	7	27					43
7	CRACK-CONVOLUTIONS BELLOWS	5					8	12	25
8	LOOSE ELECTRICAL CONNECTORS	6							6
9	BEARING DAMAGE	4	1	12			6	2	25
10	TUBE FRACTURE		17						17
11	TURBOPUMP SEAL LEAKAGE		13	28	2	12	19	65	139
12	LUBE PRESSURE ANOMALIES			37	4	2	14	21	78
13	VALVE FAILS TO PERFORM:								
	A. MOISTURE, ICE		13					2	15
	B. CONTAM/FRICTION		6	26	10				42
14	INTERNAL VALVE LEAKAGE:								
	A. CONTAMINATION		58	29		8	50	16	161
	B. COMPRESSION OF SPRING				9				9
	C. VIBRATION SEAT				6	2	7	3	18
	D. TRAPPED PRESSURE					11	4		15
15	REGULATOR DISCREPANCIES					5	33	44	82
16	CONTAMINATED HYDR. CONTR. ASSY							26	26
	TOTAL ENGINE	101	127	326	102	108	474	533	1771

Figure 11. Summary and Distribution Per Engine System

IN-FLIGHT CONDITION MONITORING

The Task II objective was to identify those in-flight condition monitoring devices, with an assessment of their maturity, which could detect rocket-engine generic failure modes resulting from Task I. Several sequential studies were conducted to this goal: A comprehensive literature search and review generated a list of novel, state-of-the-art (SOTA) and conventional sensors (Appendix E). Correlation with potential measurands applicable to the previously experienced failures, plus some practical considerations, pared this list to a manageable, relevant level for deeper analysis (Appendix F).

The comparison of competing technologies was evaluated with a screening system (Appendix G). The selected technologies were compared for technical, economical and temporal factors which yielded a final list of ranked failure-detection technologies.

SURVEY

The literature survey established a baseline for existing state-of-the-art and novel technology used for condition monitoring on in-flight systems. The survey revealed those sensors and monitoring systems used most frequently for diagnostic and prognostic purposes. The survey was well rounded in that it covered industrial processes, ground transportation, and the electronics field, as well as aircraft and aerospace. From the surveyed 89 articles, 20 novel and 14 state-of-the-art sensors were found that may be applicable to on-board rocket-engine condition monitoring. These sensors are listed in Table 1, along with the sensors already used in in-flight rocket-engine applications.

INTRODUCTION

A portion of the Task II effort was devoted to conducting a literature search for in-flight condition-monitoring technologies that would be applicable to a reusable rocket engine. The survey for condition-monitoring systems covered the fields of aircraft and aerospace, transportation, industrial processes, the medical industry and electronics. The results of the literature search, including uncovered novel and state-of-the-art condition monitoring devices, are presented in Appendix E.

FAILURE MONITORING SENSOR ASSESSMENT

The potential condition-monitoring detection technologies were to be examined to determine their suitability for detection of the 16 failure modes obtained from Task I. This was achieved by first transforming the failure modes into the measurands that could potentially be detected by sensors. To obtain all the measurands of each failure mode, the failure mode was analyzed to determine the stages leading to its incidence and identify corresponding measurands capable of detecting each stage of the failure.

TABLE 1. IDENTIFIED, IN-FLIGHT DIAGNOSTIC SENSORS

COMMONLY USED IN ROCKET ENGINES	STATE OF THE ART BUT NOT USED IN ROCKET ENGINES	NOVEL ADVANCED TECHNOLOGY DEVICES
● RESISTIVE TEMPERATURE DETECTOR	● SOLID STATE THERMOMETER	● BETA-RAY DENSIMETER, THERMOMETER
● STRAIN-GAGE PRESSURE	● DIGITAL QUARTZ PRESSURE	● ULTRASONIC THERMOMETER
● MAGNETIC PICKUP	● CORIOLIS MASS FLOWMETER	● FLUIDIC THERMOMETER
● POSITION (POTENTIOMETERS, RVDT, LVDT)	● ULTRASONIC FLOWMETER	● FIBEROPTIC PRESSURE
● THERMOCOUPLE	● TARGET FLOWMETER	● LASER DIGITAL PRESSURE
● PIEZOELECTRIC ACCELEROMETER	● HALL TACHOMETER	● SURFACE-ACOUSTIC-WAVE PRESSURE
● PIEZOELECTRIC PRESSURE	● WIEGAND TACHOMETER	● FERROMAGNETIC TORQUEMETER
● TURBINE FLOWMETER	● FIBEROPTIC TACHOMETER	● ISOTOPE WEAR DETECTOR
● THERMOPILE CALORIMETER	● MAGNETOSTRICTIVE TORQUEMETER	● FIBEROPTIC DEFLECTOMETER
● FOIL RADIOMETER	● HYDROPHONE	● EXO-ELECTRON EMISSION DETECTOR
	● ULTRASONIC EXTENSOMETER	● POLAROGRAPH
	● PYROMETER	● OPTICAL ACOUSTIC-EMISSION DETECTOR
	● EDDY-CURRENT DETECTOR	● ELECTRO-OPTICAL EXTENSOMETER
	● ULTRA-VIOLET FLAME DETECTOR	● TUNABLE DIODE LASER SPECTROMETER
		● RAMAN-LASER SPECTROMETER
		● LASER-SCATTERING DENSIMETER, VELOCIMETER
		● TUNGSTEN-CAP CALORIMETER
		● FIBEROPTIC HYGROMETER
		● EMAT (ELECTROMAGNETIC ACOUSTIC TRANSDUCER)
		● NEUTRON-RAY CORROSION DETECTOR

Figure 12 shows an example of the stages of a failure-mode propagation diagram depicted in rectangular blocks, and the in-flight and between-flight detection measurands presented in ovals and diamonds, respectively. The remaining failure modes, including their failure detection measurands, are presented in Appendix F. A total of 23 distinct in-flight measurands are derived and presented in Table 2 according to their failure modes.

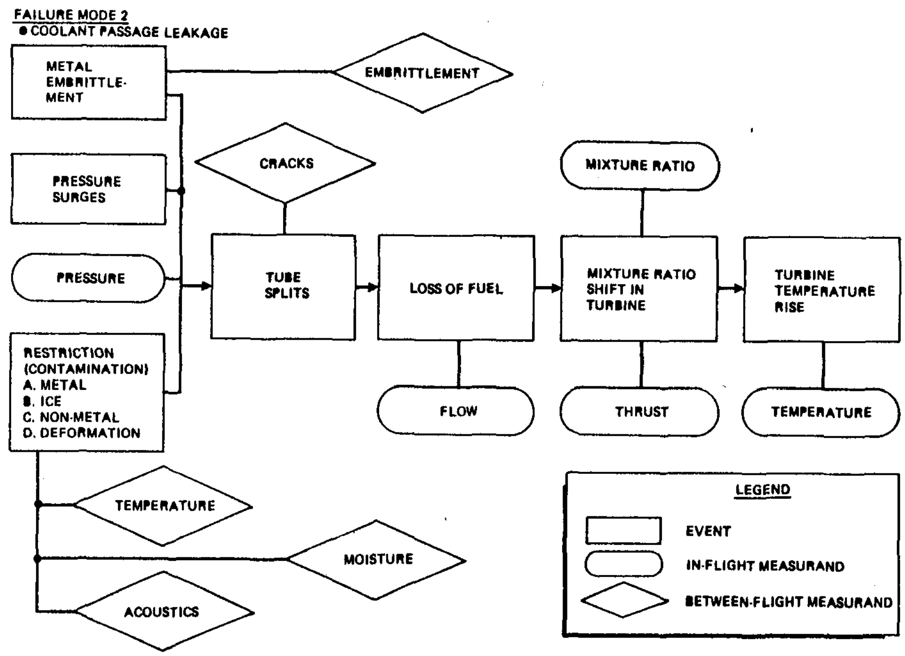

Figure 12. In-Flight and Between Flight Measurands for Detection of Nozzle Failure

Next, the novel, state of the art, and current advanced rocket-engine sensors, shown in Table 1, combined with conventional industrial sensors were matched with these measurands. The result was a matrix, shown in Table 3, that relates the in-flight potential failure-detecting devices to the 16 failure modes. In this matrix, N, S, R, and C denote novel, state of the art, rocket-engine and conventional sensors, respectively.

SENSOR SELECTION AND RANKING

The matched sensors of Table 3 were next graded and ranked. The in-flight condition-monitoring technologies required grouping them into direct and indirect condition-monitoring categories for application of clear-cut screens. A direct condition-monitoring technology detects how a component

TABLE 2. FAILURE DETECTING MEASURANDS

FAILURE MODES	MEASURANDS	
1 - BOLT TORQUE RELAXATION	VIBRATION ACOUSTICS LEAK	FRETTING EXTENSION
2 - COOLANT-PASSAGE LEAKAGE/ RESTRICTION	METAL EMBRITTLEMENT PRESSURE TRANSIENT FLOW, REDUCTION	MIXTURE RATIO SHIFT TEMPERATURE RISE IN COMBUSTION
3 - JOINT LEAKAGE	LEAK	FIRE
4 - TRANSFER TUBE CRACK	TEMPERATURE TRANSIENT MIXTURE RATIO SHIFT	FATIGUE
5 - HIGH TURBOPUMP TORQUE	TORQUE, RIPPLES TEMPERATURE, SEALS VIBRATION ACOUSTICS	WORN PARTICLES RPM TAILOFF CONTAMINANTS
6 - CRACKED TURBINE BLADE	FATIGUE TEMPERATURE TRANSIENT PRESSURE, TRANSIENT	VIBRATION ACOUSTICS BALANCE
7 - CRACKED CONVOLUTION, BELLOWS, SHIELDS	TEMPERATURE, TRANSIENT PRESSURE, TRANSIENT	ACOUSTICS VIBRATION
8 - LOOSE ELECTRICAL CONNECTORS	TORQUE, RELAXATION CONTINUITY, INTERMITTENT	SEPARATION
9 - BALL BEARING DAMAGE	TEMPERATURE, EXCESSIVE RACE VIBRATION ACOUSTICS TORQUE, RIPPLES WORN PARTICLES	RPM TAILOFF FATIGUE CONTAMINANT BALANCE
10 - SMALL TUBE FRACTURE	VIBRATION EXTENSIVE	DEFORMATION
11 - TURBOPUMP SEAL LEAKAGE	TEMPERATURE, EXCESSIVE VIBRATION WORN PARTICLES	RPM FAILOFF CONTAMINANT
12 - LUBE PRESSURE ANOMALIES	PRESSURE, DIFFERENTIAL FLOW, REDUCTION	CONTAMINANT
13 - VALVE FAILURE	MOISTURE, DEWING CONTAMINANT	PRESSURE, ACTUATION
14 - INTERNAL LEAKAGE	WORN PARTICLES ACOUSTICS	VIBRATION TEMPERATURE, TRANSIENT
15 - REGULATOR DISCREPANCIES	CONTAMINANTS LEAK	WORN PARTICLES
16 - CONTAMINATED HYDRAULICS	CONTAMINANT LEAK	WORN PARTICLES

TABLE 3. IN-FLIGHT CONDITION MONITORING DETECTION TECHNOLOGIES ACCORDING TO FAILURE MODES AND THEIR MEASURANDS

LEGEND

N = NOVEL TECHNOLOGY

S = STATE OF THE ART TECHNOLOGY

R = ROCKETDYNE TECHNOLOGY

C = CONVENTIONAL TECHNOLOGY

IN-FLIGHT SENSORS \ FAILURE MODES/MEASURANDS	1 - BOLT TORQUE RELAXATION	VIBRATION	ACOUSTICS	LEAK	FRETTING	EXTENSION	2 - COOLANT-PASSAGE LEAKAGE/RESTRICTION	METAL EMBRITTLEMENT	PRESSURE TRANSIENT	FLOW, REDUCTION	MIXTURE RATIO SHIFT	TEMPERATURE, RISE IN COMBUSTION	3 - JOINT LEAKAGE	LEAK	FIRE	4 - TRANSFER TUBE CRACK	TEMPERATURE TRANSIENT	MIXTURE RATIO SHIFT	FATIQUE	5 - HIGH TURBOPUMP TORQUE	TORQUE, RIPPLES	TEMPERATURE, SEALS	VIBRATION	ACOUSTICS	WORN PARTICLES	RPM TAILOFF	CONTAMINANTS	6 - CRACKED TURBINE BLADE	FATIGUE	TEMPERATURE, TRANSIENT	PRESSURE, TRANSIENT	VIBRATION	ACOUSTICS	BALANCE	7 - CRACKLED CONVOLUTION, BELLOWS, SHIELDS	TEMPERATURE, TRANSIENT	PRESSURE, TRANSIENT	ACOUSTICS	VIBRATION	8 - LOOSE ELECTRICAL CONNECTORS	TORQUE, RELAXATION	CONTINUITY, INTERMITTENT	SEPARATION	9 - BALL BEARING DAMAGE	TEMPERATURE, EXCESSIVE RACE	VIBRATION	ACOUSTICS	TORQUE, RIPPLES	WORN PARTICLES	RPM TAILOFF	FATIQUE	CONTAMINANT	BALANCE	10 - SMALL TUBE FRACTURE	VIBRATION	EXTENSION	DEFORMATION	11 - TURBOPUMP SEAL LEAKAGE	TEMPERATURE, EXCESSIVE	VIBRATION	WORN PARTICLES	RPM TAILOFF	CONTAMINANT	12 - LUBE PRESSURE ANOMALIES	PRESSURE, DIFFERENTIAL	FLOW, REDUCTION	PRESSURE, REDUCTION	CONTAMINANT	13 - VALVE FAILURE	MOISTURE, DEWING	CONTAMINANT	PRESSURE, ACTUATION	14 - INTERNAL LEAKAGE	WORN PARTICLES	ACOUSTICS	VIBRATION	TEMPERATURE, TRANSIENT	15 - REGULATOR DESCREPANCIES	CONTAMINANTS	LEAK	WORN PARTICLES	16 - CONTAMINATED HYDRAULICS	CONTAMINANT	LEAK	WORN PARTICLES
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PRESSURIZED LEAK DETECTOR														R																																																																				F				R	

is, whereas the indirect technology detects what the component does in regard to the engine operation (temperature, pressure, flow, speed and thrust). It was with the help of this distinction that it was possible to apply the speed (of a few milliseconds) screen to the indirect technologies, to detect the process transients. In contrast, the speed screen was not applicable to direct technologies. For example, relatively slow pressure build up in a contained joint, indicating a slow leak, is best sensed directly by a pressure sensor. Fast response time for this trending-type observation is not significant.

Upon thorough review of various screens only four distinct screens were determined to be unequivocal. Two of these screens were valid only for indirect condition-monitoring technologies; namely speed and failsafeness. The other two were applicable only to the direct condition-monitoring technologies, i.e., bulkiness and numerosity.

For indirect technologies it is necessary to capture the frequency and amplitude of transients of process (flow, pressure, temperature, rpm) measurements. In rocket engines these transients are typically a few milliseconds in duration. By totalizing the transients it may be possible to determine the maximum stress exposures, which provides information regarding the remaining life of the component. Conventionally, however, process sensors are designed with damping to generate an average signal. This eases controlling of the process measurand.

The second indirect screen is failsafeness, which implies no catastrophic hazard to the engine if the measuring device malfunctions. With the aid of these four go-no-go screens, the 33 direct condition-monitoring technologies were reduced to 12 and the 33 indirect technologies were reduced to seven as shown in Tables 4 and 5.

Upon examination of the acceptably screened direct-diagnostic sensors, five devices were recognized as possessing well established limitations regarding their rocket engine applicability. To preclude carrying these well-known conventional instruments any further in ranking, their utility was termined. The five sensors, comprised of strain gage and piezoresistive accelerometers and nickel, semiconductor and the thermocouple thermometers, are denoted by a deletion sign in Table 6 resulting in only 12 direct condition-monitoring technologies remained for grading and ranking.

In all, only 19 technologies, consisting of 11 novel, 6 state-of-the-art and 2 rocketry were acceptable for further grading and ranking.

GRADING AND RANKING

Upon successful application of the four screens, 19 technologies remained to be graded and ranked.

A consistent and methodic rationale was needed to grade all these technologies. To conceive such a rationale, each detection technology is depicted by the liabilities (or penalties of that technology) versus its

TABLE 4. IN-FLIGHT DIRECT-DIAGNOSTIC SENSOR SCREENING

SENSOR TYPES	SENSOR[1] STATUS	BULKY	NUMEROUS[2]	ACCEPTABLE SENSORS
ACCELEROMETERS:		-	-	ACCELEROMETERS:
STRAIN-GAGE	C	NO	NO	
PIEZOELECTRIC	R	NO	NO	PIEZOELECTRIC
PIEZORESISTIVE	S	NO	NO	
HYDROPHONES:		-	-	HYDROPHONES
PIEZOELECTRIC	S	NO	NO	PIEZOELECTRIC
FLAME DETECTORS:		-	-	
RADIOMETER	S	NO	YES	-
TORQUEMETERS:		-	-	TORQUEMETERS:
MAGNETOSTRICTIVE	S	YES	NO	-
RELUCTIVE	S	YES	NO	-
STRAIN-GAGES, AC OR DC	C	YES	NO	-
OPTICAL	C	YES	NO	-
DIGITAL, FERROMAGNETIC	N	NO	NO	DIGITAL, FERROMAGNETIC
DISPLACEMENT METERS:		-	-	
STRAIN-GAGES	C	-	YES	-
LVDT/RVDT3	R	-	YES	-
POTENTIOMETRIC	C	-	YES	-
DIGITAL ENCODER	R	-	YES	-
CAPACITIVE	C	-	YES	-
ULTRASONIC EXTENSOMETER	S	-	YES	-
EDDY CURRENT	S	-	YES	-
OPTICAL EXTENSOMETER	N	-	YES	-
ISOTOPE WEAR DETECTOR	N	NO	NO	ISOTOPE WEAR DETECTOR
FIBEROPTIC BEARING DETECTOR	N	NO	NO	FIBEROPTIC BEARING DETECTOR
EXO-ELECTRON MISSION DETECTOR	N	NO	NO	EXO-ELECTRON EMISSION DETECTOR
POLAROMETER	N	NO	NO	POLAROMETER
OPTICAL ACOUSTIC-EMISSION DETECTOR	N	YES	NO	-
TUNABLE-LASER SPECTROMETER	N	NO	NO	TUNABLE-LASER SPECTROMETER
RAMAN-LASER SPECTROMETER	N	YES	NO	-
PRESSURIZED LEAK DETECTOR	R	NO	YES	
EDDY-CURRENT DETECTOR	S	NO	NO	EDDY CURRENT DETECTOR
EMAT[5]	N	NO	NO	EMAT
THERMOMETERS:		-	-	THERMOMETERS
NICKEL RTD[4]	C	NO	NO	NICKEL RTD
PLATINUM RTD[4]	R	NO	NO	PLATINUM RTD
SENICONDUCTOR	C	NO	NO	SEMICONDUCTOR
THERMOCOUPLE	C	NO	NO	THERMOCOUPLE
PYROMETER	S	NO	NO	PYROMETER
TOTAL: 33				12

[1]C = CONVENTIONAL, R = ROCKET, S = STATE OF THE ART, N = NOVEL

[2]MORE THAN 10 SENSORS PER ENGINE IS CONSIDERED AS TOO MANY CLUTTERED

[3]LVDT = LINEAR VARIABLE DIFFERENTIAL TRANSFORMER

[4]RDT = RESISTIVE TEMPERATURE DETECTOR

[5]EMAT = ELECTROMAGNETIC ACOUSTIC TRANSDUCER

TABLE 5. IN-FLIGHT INDIRECT-DIAGNOSTIC SENSOR SCREENING

SENSOR TYPES	SENSOR[1] STATUS	FAST[2]	FAILSAFE	ACCEPTABLE SENSORS
PRESSURE TRANSDUCERS:				PRESSURE TRANSDUCERS:
PIEZORESISTIVE DIAPHRAGM	S	YES	YES	-
PIEZORESISTIVE BRIDGE/CIRCUIT	S	NO[3]	YES	-
DEPOSITED METAL BRIDGE	R	NO[3]	YES	-
BONDED STRAIN-GAGE BRIDGE	C	NO[3]	YES	-
DEPOSITED THIN FILM	R	NO[3]	YES	-
DIGITAL QUARTZ RESONATOR	S	YES	YES	DIGITAL QUARTZ RESONATOR
POTENTIOMETRIC	C	NO	YES	-
DIGITAL CYLINDRICAL RESONATOR	S	NO	YES	-
LVDT[4]	C	NO	YES	-
CAPACITIVE	C	NO	YES	-
PIEZOTRANSISTIVE	S	YES	NO	-
SILICON ON SAPPHIRE	S	YES	NO	-
FIBEROPTIC	N	YES	YES	FIBEROPTIC
LASER DIGITAL	N	YES	YES	LASER DIGITAL
SURFACE ACOUSTIC WAVE	N	YES	YES	SURFACE ACOUSTIC WAVE
THERMOMETERS:				THERMOMETERS:
NICKEL RTD[5]	C	NO	NO	-
PLATINUM RTD	R	NO	NO	-
SEMICONDUCTOR (THERMISTOR)	C	NO	NO	-
TERMOCOUPLE	C	NO	NO	-
BETA-RAY	N	YES	NO	-
UNTRASONIC	N	YES	YES	ULTRASONIC
FLUIDIC	N	NO	NO	-
TACHOMETERS:				TACHOMETERS:
OPTICAL	S	YES	YES	OPTICAL
MAGNETIC PICKUP	R	YES	NO	-
HALL EFFECT	S	YES	NO	-
WIEGAND EFFECT	S	YES	NO	-
FLOWMETERS:				FLOWMETERS:
OPTICAL	S	NO	YES	-
THERMAL	S	NO	NO	-
CORIOLIS	S	NO	YES	-
ULTRASONIC	S	YES	YES	ULTRASONIC
VORTEX SHEDDER	S	YES	NO	-
TURBINE	R	YES	NO	-
TARGET	C	YES	NO	-
TOTAL: 33				7

[1] C = CONVENTIONAL, R = ROCKET, S = STATE OF THE ART, N = NOVEL

[2] FAST = FAST RESPONSE (A FEW MILLISECONDS IS REQUIRED FOR TRANSIENTS)

[3] EXCESSIVE THERMAL LAG

[4] LVDT = LINEAR VARIABLE DIFFERENTIAL TRANSFORMER

[5] RTD = RESISTIVE TEMPERATURE DETECTOR

TABLE 6. VIABLE IN-FLIGHT CONDITION-MONITORING SENSORS

IN-FLIGHT SENSORS	FAILURE MODES								
	2-COOLANT-PASSAGE LEAKING/RESTRICTION	5-HIGH TURBOPUMP TORQUE	6-CRACKED TURBINE BLADE	7-CRACKED CONVOLUTION, BELLOWS, SHIELDS	9-BALL BEARING DAMAGE	11-TURBOPUMP SEAL LEAKAGE	13-VALVE FAILURE	14-INTERNAL LEAKAGE	15-REGULATOR DESCREPANCIES
DIGITAL QUARTZ PRESSURE SENSOR	S		S	S			S		
FIBEROPTIC PRESSURE SENSOR	N		N	N			N		
DIGITAL LASER PRESSURE SENSOR	N		N	N			N		
SURFACE ACUOSTIC WAVE PRESSURE SENSOR	N		N	N			N		
ULTRASONIC THERMOMETER	N	N	N	N	N	N	N		
OPTICAL TACHOMETER		S			S	S		N	
ULTRASONIC FLOWMETER	S								
PIEZOELECTRIC ACCELEROMETER		R	R	R	R	R		R	
PIEZOELECTRIC HYDROPHONE		S	S	S	S			S	
FERROMAGNETIC TORQUEMETER		N			N				
ISOTOPE WEAR DETECTOR		N			N	N		N	N
FIBEROPTIC BEARING DETECTOR			N		N				
EXO-ELECTRON DETECTOR			N		N				
POLAROGRAPH	N								
TUNABLE DIODE-LASER SPECTROMETER	N		N		N	N	N	N	N
EDDY CURRENT DETECTOR			S						
PLATINUM RTD**	R	R	R	R	R	R		R	
PYROMETER			S	S					
EMAT***			N		N				
PERFECT SCORE									

*FAILURES NO. 1, 4, AND 12 ARE OBVIATED BY IMPROVED DESIGN, FAILURES NO. 3, 8, 10, AND 16 ARE NOT APPROPRIATE FOR IN-FLIGHT DIAGNOSTICS.
**RTD = RESISTIVE TEMPERATURE DETECTOR
***EMAT = ELECTRO-MAGNETIC ACOUSTIC TRANSDUCER

LEGEND

N = NOVEL TECHNOLOGY
S = STATE OF THE ART TECHNOLOGY
R = ROCKET TECHNOLOGY

virtues (or rewards); the higher the net virtues the better the technology. Both liabilities and virtues were divided into two categories which facilitate their comparison, economic and technical. Thus, the economic liabilities consist of the expenditure to develop and integrate the technology and the economic virtues represent the return on the investment in terms of inspection-labor saving and hazard detection and prevention. The technical virtues on the other hand, consist of elements called lumped descriptors which describe the ability of the technology to detect accurately, correctly, constantly and safely.

The salient descriptors of each of these lumped descriptors were selected. The result is shown in Table 7. It should be noted that the signal-conditioning and data-processing requirements are included under the electronic group of technical liabilities and are not considered separately elsewhere.

Subsequently, quantitative values (scores) were assigned to each lumped descriptor based on a consistant grading scale.

One additional dimension was added to the comparative ratings. This factor, development time, is related to a program schedule which can influence the viability of the technology according to the need and application determined by specific programs.

TABLE 7. TECHNICAL AND FINANCIAL DESCRIPTOR CATEGORIES AND DESCRIPTORS

TECHNICAL REQUIREMENTS	TECHNICAL PERFORMANCE	EXPENDITURES
• PHYSICAL WEIGHT SPACE STRENGTH MATERIAL CHEMICALS RESONANCE FATIGUE • ELECTRONIC POWER, CONSUMPTION VOLTAGE CURRENT WIRING FILTERING AMPLIFICATION ANALOG/DIGITAL MEMORY REQUIREMENTS SIGNAL CONDITIONING LINEARIZATION SHIELDING • FUNCTIONAL INTRUSIVE POWER	• DETECTIBILITY SPEED ACCURACY REPEATABILITY SENSITIVITY RESOLUTION DRIFT ARTIFACTS SUSCEPTIBILITY • DURABILITY RECALIBRATION INSPECTION LIFE • SAFETY FAIL SAFETY FAILURE EFFECTS	• R&D • INTEGRATION

Using a 0 through 10 relative scale for each lumped descriptor, the detection technologies for each failure mode were numerically graded. To be as objective as possible, each technology was graded according to its own features, independent of its utility or need criticality. The functional, detectability, and safety categories were considered more important, hence were assigned a twofold weighting factor relative to the physical, electronic and durability categories; the rationale being that if you cannot measure the failure without burdening and hazarding the rocket engine, it does not matter how small, durable or electronically demanding the detection technology is. An example of each such rating is shown in Table 8.

This table is divided into direct and indirect condition monitoring groups which are henceforth ranked independently. Indirect condition monitoring, to detect a failure, requires extensive data processing to correlate engine operational parameters under varying loads, rpm, temperatures, pressures.

Direct condition monitoring, in contrast, requires very little data processing because it monitors the condition of the component independent of the propellant temperature, pressure, or flow.

In all, nine similar tables were completed and are included as Appendix G. It is noted that, of the original 16 failure modes, not all were developed into technology rankings. Three modes were eliminated and four were not detectable through any viable in-flight condition-monitoring means.

Next, the technical, financial, and development-time ranks of each technology were added together yielding the overall grade for the technology. The technical rank was weighted significantly higher than the other ranks: the rationale was based on the fact that any technology, regardless of how well developed it is for non-rocket industry application, still requires a significant amount of testing, adaptation, and modification efforts and expenditures before it can be flown in a rocket engine.

The outcome of the detection technology grading and the corresponding rankings is summarized in Table 9. Table 10 identifies the ultrasonic thermometer and flowmeter as to the two top-ranking, most promising indirect condition-monitoring technologies, followed by digital quartz pressure sensor and optical tachometer. These four technologies combined could indirectly detect eight generic failure modes, but they require extensive in-flight engine-parameter correlation, trending, thresholding, totalizing, data processing, etc. The same table identifies pyrometer, fiber-optic deflectometer, isotope wear detector and tunable diode-laser spectrometer as the most promising direct condition-monitoring technologies capable of in-flight detection of all nine failure modes.

CONCLUDING REMARKS

In summary, the computerized literature search, from an on-line six-million-citation data bank yielded a review of 289 abstracts and 78 articles. From this review, 20 novel and 14 state of the art in-flight condition-monitoring technologies were identified.

TABLE 8. TECHNICAL AND ECONOMICAL GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS FOR DETECTION OF COOLANT PASSAGE LEAKAGE/RESTRICTION (#2)

DESCRIPTORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				
SENSORS	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	\$1	\$1	\$1	10	0	10	110
PRESSURE SENSORS														
QUARTZ, DIGITAL	7	3	18	6	18	8	60	50	250	300	7	1	9	76
FIBEROPTIC	7	2	18	6	18	8	59	200	250	450	5	3	7	71
LASER DIGITAL	7	3	18	6	18	7	59	300	250	550	4	4	6	69
SAW, DIGITAL	7	3	18	6	18	7	59	200	250	450	5	2	8	72
ULTRASONIC THERMOMETER, FLAME	6	5	20	12	20	6	69	100	200	300	7	3	7	83
ULTRASONIC FLOWMETER, NOZZLE	10	5	20	6	20	9	70	50	150	200	8	2	8	86
POLAROGRAPH	2	4	10	14	10	4	44	250	450	700	3	6	4	51
TUNABLE DIODE LASER SPECTROMETER MIXTURE RATIO	8	5	19	12	19	7	60	300	300	600	4	6	4	68

1 - IN THOUSANDS

NOTE: THE REMAINING TABLES ARE PRESENTED IN APPENDIX G.

TABLE 9. IN-FLIGHT CONDITION-MONITOR TECHNOLOGY RANKING

IN-FLIGHT SENSORS	RANK	GRADE*				FAILURE MODES								
		TOTAL	TECHNICAL	ECONOMIC	DEVELOPMENT	2-COOLANT PASSAGE LEAKAGE/RESTRICTION	5-HIGH TURBOPUMP TORQUE	6-CRACKED TURBINE BLADE	7-CRACKED CONVOLUTION BELLOWS, SHIELDS	9-BALL BEARING DAMAGE	11-TURBOPUMP SEAL LEAKAGE	13-VALVE FAILURE	14-INTERNAL LEAKAGE	15-REGULATOR DISCREPANCIES
INDIRECT														
ULTRASONIC THERMOMETER	1	83	69	7	7	N	N	N	N	N	N		N	
ULTRASONIC FLOWMETER	2	81	65	8	8	S								
DIGITAL QUARTZ PRESSURE SENSOR	3	78	62	7	9	S		S	S			S		
OPTICAL TACHOMETER	4	76	62	6	8		S			S	S			
SURFACE ACOUSTIC WAVE PRESSURE SENSOR		74	61	5	8	N		N	N			N		
FIBEROPTIC PRESSURE SENSOR		73	61	5	7	N		N	N			N		
DIGITAL LASER PRESSURE SENSOR		71	61	4	6	N		N	N			N		
DIRECT														
PYROMETER	1	85	71	6	8			S	S					
ISOTOPE WEAR DETECTOR	2	82	71	4	7		N			N	N		N	N
FIBEROPTIC DEFLECTOMETER	2	82	71	4	7			N		N				
TUNABLE DIODE-LASER SPECTROMETER	4	79	71	4	4	N		N		N	N	N	N	N
PIEZOELECTRIC ACCELEROMETER		78	64	7	7		R	R	R	R	R		R	
PIEZOELECTRIC HYDROPHONE		78	62	8	8		S	S	S	S			S	
FERROMAGNETIC TORQUEMETER		76	72	3	3		N			N				
PLATINUM RTD (RESISTANCE TEMPERATURE DETECTOR)		73	58	7	8.	R	R	R	R		R			
EMAT (ELECTROMAGNETIC ACOUSTIC TRANSDUCER)		62	51	5	6			N		N				
EDDY CURRENT DETECTOR		62	49	6	7			S		S				
EXO-ELECTRON DETECTOR		58	51	4	3			N		N				
POLAROGRAPH		51	44	3	4	N								
PERFECT SCORE		110	90	10	10									

LEGEND
N = NOVEL TECHNOLOGY
S = STATE-OF-THE-ART TECHNOLOGY
R = ROCKET TECHNOLOGY
* = THE HIGHEST SCORE AMONGST VARIOUS FAILURE MODES

TABLE 10. IN-FLIGHT CONDITION-MONITOR TECHNOLOGY RANKING

IN-FLIGHT SENSORS	FAILURE MODES								
	2-COOLANT PASSAGE LEAKAGE/RESTRICTION	5-HIGH TURBOPUMP TORQUE	6-CRACKED TURBINE BLADE	7-CRACKED CONVOLUTION, BELLOWS, SHIELDS	9-BALL BEARING DAMAGE	11-TURBOPUMP SEAL LEAKAGE	13-VALVE FAILURE	14-INTERNAL LEAKAGE	15-REGULATOR DISCREPANCIES
<u>DIRECT</u>									
FIBEROPTIC DEFLECTOMETER			80		88				
PYROMETER			86	86		79		79	78
TURNABLE DIODE-LASER SPECTROMETER	78				82	82	82	82	82
ISOTOPE WEAR DETECTOR		79			79	79	75	79	78
<u>INDIRECT</u>									
ULTRASONIC THERMOMETER	83								
OPTICAL TACHOMETER		83			83	83		83	
ULTRASONIC FLOWMETER	81								
DIGITAL QUARTZ PRESSURE SENSOR	76		78	78			78		

Next, the 16 failure modes and failure propagation diagrams were analyzed, resulting in 23 distinct in-flight failure-detecting measurands. These measurands were then correlated with novel, state of the art, rocket-engine and conventional technologies resulting in 33 direct and 33 indirect potential condition-monitoring technologies.

A selection approach was applied successfully using four nonequivocal screens and several lumped descriptors. The screening process rejected inapplicable technologies. The lumped descriptors were employed for grading and ranking of the remaining 19 applicable in-flight condition-monitoring technologies.

The ranking was achieved by assigning relative numerical grades to each device feature. Since these technologies vary in their state of maturity and utility, they were graded and ranked assuming they are completely developed and are used for only one failure mode at a time.

Such an approach resulted in identifying four top-ranking direct condition-monitoring devices capable of detection of all failure modes and four top-ranking indirect condition-monitoring devices capable of detection of eight out of nine failure modes:

The direct condition-monitoring devices are:

1. Pyrometer detects rotating-blade temperature
2. Isotope wear detector detects bearing, rotary-seal and valve-seat wear.
3. Fiberoptic deflectometer detects bearing loading and deflection
4. Tunable diode-laser Spectrometer detects nonmetal wear

The indirect condition-monitoring devices are:

1. Ultrasonic thermometer
2. Ultrasonic flowmeter
3. Digital quartz pressure sensor
4. Fiberoptic tachometer

It should be noted here that other technologies such as Raman spectroscopy, ferromagnetic torquemetering and exo-electron detection, although eliminated by this screening and grading process, have unique condition-monitoring capabilities and should be carefully followed for any major breakthrough which could render them applicable to in-flight condition-monitoring.

BETWEEN-FLIGHT INSPECTION

This task determined the between-flight inspection requirements that would provide engine component reverification and remaining life assessment for those failure-prone components identified in Task I. The applicability of between-flight inspection technologies and their implications on engine design and operation were evaluated with respect to those requirements. The upgrading or development required of each technology was also identified.

An approach similar to the method used in Task II was taken. A survey was performed to identify existing inspection technologies which might be applicable to rocket engines. This included inspection procedures which have been in routine use for many years as well as experimental techniques used soley in a laboratory environment. The results of Task I were then examined to identify the between-flight-detectible measurands associated with each failure mode. This led to the development of general inspection requirements and their correlation with the surveyed technologies. The techniques corresponding to each failure mode were evaluated on an equal basis of development, resulting in scores which ranked the technologies and became inputs to Task IV. Accessibility requirements, engine configuration modifications and estimates of the effects on engine reliability and safety were determined for each inspection and included in the scoring.

INSPECTION TECHNOLOGY SURVEY

A survey was undertaken to find inspection technology which could be applicable to reusable rocket engines between flights. This survey included computer literature searches, periodical reviews, and personal visits. Representative literature was enumerated and the inspection techniques uncovered were then summarized (Fig. 13).

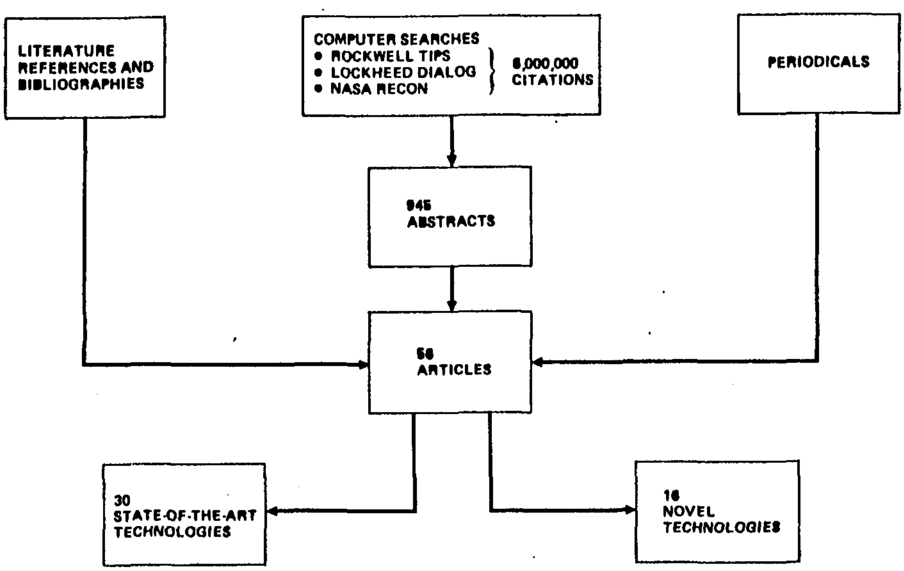


Figure 13. Literature Survey Utilized Multiple Resources to Uncover Inspection Technologies

It should be noted that the purpose of these searches was to provide a broad survey of between-flight engine condition monitoring technology, but not a complete bibliography of this subject. Documents were selected for enumeration if they contained different technology, applications, or approaches than had been previously encountered. Also, literature covering extensively used technology, or technology clearly not applicable to rocket engines was not chosen for examination.

Resources

Four computerized searches were made, producing a total of 945 listings. The search parameters were purposely left rather general so that technologies which have only seen limited or specialized use might be identified. Three of the searches looked for literature dealing with inspection technology for aerospace engines as well as various basic inspection concepts. The fourth search dealt with leak detection only. Each listing was examined to determine if the defined document might contain new and useful inputs to Task III. Literature which might pertain to the Task II effort was also identified. A brief description of each search follows.

The Rockwell TIPS search (Fig. 14) was an on-line examination of the combined database of five Rockwell International Divisions (Rocketdyne, Space Systems Group, Science Center, and North American Aviation, both Columbus and Los Angeles Divisions).

Two searches were conducted on Lockheed's DIALOG, a combined database of the NTIS, Engineering Index, Inc., and Data Courier, Inc. (Fig. 15): a search of leak detection technology and a more general inspection technology search.

The NASA RECON computer search yielded a total of 267 citations (Fig. 16), but because of duplications of citations in the Rockwell TIPS and Lockheed DIALOG searches, only 15 of these documents were selected for review.

Beyond the computer searches, discussions were held with Air Force personnel about their diagnostic and nondestructive (NDI) programs.

Additional information was obtained by reviewing the references of literature located by the computer searches and by examining recent periodicals for articles concerning nondestructive inspection methods.

Survey Results

Literature that was found to contain useful information was tabulated as shown in Table 11 and Appendix H. Information pertaining to in-flight diagnostics was forwarded to the Task II effort.

The number of different inspection techniques described in each document, as indicated in Table 11, have been divided into three categories. Rocket Engine refers to technology which is or has been successfully applied to liquid propellant rocket engine inspection. State of the art refers to

```
QN          _RR1-0248

TITLE       _INSPECTION TECHNIQUES

REQUESTR_B.D.  HINES

SEARCHER_JULIA KEIM

ADDRESS _ROCKETDYNE TIC, 3A29

SV1         _INSPECTION | CHECKOUT | EXAMINATION | NDI

SV2         _AIRCRAFT ENGINES | DIESEL ENGINES | GASOLINE ENGINES |
            _GAS TURBINE ENGINES | HELICOPTER ENGINES | JET ENGINES |
            _LIQUID PROPELLANT ROCKET ENGINES | SOLID PROPELLANT ROCKET ENGINES |
            _ROCKET ENGINES | MARINE ENGINES

SV3         _ACOUSTICEMISSION | X RAY ANALYSIS | FIBER OPTICS | TORQUEMETERS |
            _HALL EFFECT | FLOWMETERS

CB          _SV1 + SV2

CB1         _SV2 + SV3

CB2         _CB | CB1

DR          _CB2, RJE=D


N7500781
PUB DATE 74 B BKCL 629.4 P VOL34
ALSO R-043298
INSTRUMENTATION FOR AIRBREATHING PROPULSION: TECHNICAL PAPERS SELECTED FROM THE SYMPOSIUM, SEPTEMBER 1972, U.S. NAVAL
POSTGRADUATE SCHOOL, MONTEREY, CALIF.
BY FUHS, A. E., ED.; KINGERY, M., ED.
PROGRESS IN ASTRONAUTICS AND AERONAUTICS. VOL. 34.
NOTE THE MIT PRESS
MISC 547P.
DESC AERODYNAMICS; *AIR BREATHING ENGINES; *AIRCRAFT INSTRUMENTS; AXIAL FLOW TURBINES; BIBLIOGRAPHIES; COMBUSTION
CHAMBERS; COMPRESSORS; CONTROL SYSTEMS; FIBER OPTICS; FLOW FIELDS; FLOW MEASUREMENT; FLOWMETERS; FLUCTUATION; GAS
TURBINE ENGINES; HEAT TRANSFER; HOLOGRAPHY; INSTRUMENTATION; JET ENGINES; JET PROPULSION; LASERS; MEASURING INSTRUMENTS;
NOZZLES; PROPULSION SYSTEMS; RAMAN SPECTRA; RAMJET ENGINES; SPRAYERS; SUPERSONIC FLOW; SUPERSONIC PLANES; *SYMPOSIA;
SYSTEMS ENGINEERING; TEMPERATURE MEASURING INSTRUMENTS; TRANSDUCERS; TURBINE BLADES; TURBOFAN ENGINES; TURBOJET ENGINES;
TURBULENT FLOW; WIND TUNNEL TESTING
```

Figure 14. Example of Rockwell TIPS Search

```
User  676   Date: 4nov80   Time:15:21:55   File:  8

Set Items Description
  1   7214 TANK?
  2  17844 VIBRATION?
  3    177  1 AND 2
  4   4863 INSPECTION
  5  11118 EXAMINATION
  6  15348  4 OR 5
  7   2655 ROCKET
  8  15984 AIRCRAFT
  9   1836 HELICOPTER?
 10   4384 DIESEL
 11  23947  7 OR 8 OR 9 OR 10
 12   9641 ENGINE
 13  12960 ENGINES
 14  16521 MOTOR?
 15  30719  12 OR 13 OR 14
 16  45682  11 OR 15
 17   8984  11 AND 15
 18    177  6 AND 17
 19      0 FIBER?OPTIC?
 20     12 TORQUEMETER?
 21    741 X?RAY
 22   1292 FLOWMETER?
 23   2042  19 OR 20 OR 21 OR 22
 24     48 FIBEROPTIC?
 25  13084 FIBER
 26  40069 OPTIC?
 27   3506  25 AND 26
 28   5560  23 OR 24 OR 27
 29     22  17 AND 28
 30    196  18 OR 29

Print 30/5/1-196

Search Time:  0.120   Prints: 196   Descs.:  18


  984270   ID NO.- EI791184270
WEAR DEBRIS ANALYSIS.
Parr, N. L.; Ritchie, J.
AGARD Lect Ser n 103, Presented at London, Engl, Apr 23-24
1979; Milan, Italy, Apr 26-27 1979.  Publ by AGARD,
Neuilly-sur-Seine, Fr, 1979 p 4. 1-4. 20   CODEN: NAGLB5
  The factors controlling the cost of ownership of expensive
military equipment are outlined with specific reference to the
role of wear on scheduled and unscheduled maintenance.  The
value and limitations of established condition monitoring
techniques and procedures, based on study of the particulate
debris carried by the lubricating fluid, are explored for
engine, gearbox, and hydraulic systems.  An account is given
of current effort to improve these techniques and of research
to evolve meaningful monitoring measures for a more scientific
approach to the development and operation of new machinery
incorporating advanced engineering designs and materials.  An
idealized research and development program, centered on gear
profile failure demonstrator facilities, including a number of
supporting scientific, technological, and design exercises, is
presented.  33 refs.
  DESCRIPTORS: (*AIRCRAFT ENGINES, *Nondestructive Examination
   ), (AIRCRAFT MATERIALS, Wear),
  CARD ALERT: 653, 421, 415, 652
```

Figure 15. Example of Search from DIALOG

```
SEARCH NO.      004
SEARCH TITLE INSPECTION TECHNI
DATE/FILE     11-17-80/0
SEARCH BY      B.D. HINESIS
REQUESTER      JULIA KEIN
STREET        ROCKETDYNE., ACCO. 12214 LAKEWOOD BLVD.
CITY/STATE     DOWNEY, CA  90241
USER ID       ROCK
```

TERMINAL 52   11-18-80

TOTAL TIME PER COMMAND FOR THIS USER

RECON COMMAND	TIME MIN	NO	RECON COMMAND	TIME MIN	NO	RECON COMMAND	TIME MIN	NO
BEGIN SEARCH	000.33	0	COMBINE	000.41	6	PRINT	000.02	3
EXPAND	000.00	0	LIMIT	000.00	0	LIMIT ALL	000.00	0
DISPLAY	000.00	0	KEEP	000.00	0	END SEARCH	000.02	2
SELECT	001.02	20	TYPE	000.00	0	ERROR	000.00	0
DISPLAY SET	000.00	0	MESSAGE	000.00	0	ITEMS PRINTED		264

TOTAL ELAPSED TIME IS 006.23 MIN.

SET	NO.	DESCRIPTION
1	1465	1465 ST/INSPECTION
2	121	121 ST/INFRARED INSPECTION
3	512	512 ST/X RAY INSPECTION
4	44	44 ST/EXAMINATION
5	6863	6863 NONDESTRUCTIV/NONDESTR
6	1129	1129 ST/IN FLIGHT MONITORING
7	946	946 ST/GROUND SUPPORT SYSTE
8	10383	10383 1+2+3+4+5+6+7
9	446	446 ST/ENGINES
10	4187	4187 ST/GAS TURBINE ENGINES
11	1698	1698 ST/JET ENGINES
12	1309	1309 ST/RAMJET ENGINES
13	2015	2015 ST/TURBOJET ENGINES
14	947	947 ST/INTERNAL COMBUSTION
15	710	710 ST/DIESEL ENGINES
16	557	557 ST/HELICOPTER ENGINES
17	461	461 ST/PISTON ENGINES
18	2824	2824 ST/ROCKET ENGINES
19	2379	2379 ST/LIQUID PROPELLANT RO
20	4433	4433 ST/SOLID PROPELLANT ROC
21	1191	1191 ST/TURBINE ENGINES
22	10757	10757 9+10+11+12+13+14+15
23	11247	11247 16+17+18+19+10+21
24	17389	17389 22+23
25	272	272 8*24
26	264	264 25-7

```
      74A27444*   ISSUE 12   PAGE 1711   CATEGORY 28   RPT#:
      ASME PAPER 74-GT-51   74/03/00  9 PAGES   UNCLASSIFIED
      DOCUMENT
UTTL: Residual stresses in gas turbine engine components
      from Barkhausen noise analysis
AUTH: A/BARTON, J. R.;  B/KUSENBERGER, F. N.   PAA:
      B/(Southwest Research Institute, San Antonio, Tex.)
      SAP: MEMBERS, $1.00; NONMEMBERS, $3.00
      American Society of Mechanical Engineers, Gas Turbine
      Conference and Products Show, Zurich, Switzerland,
      Mar. 30-Apr. 4, 1974. 9 p.
MAJS: /*ENGINE NOISE/*ENGINE PARTS/*GAS TURBINE ENGINES/*
      NONDESTRUCTIVE TESTS/*RESIDUAL STRESS/*STRESS
      MEASUREMENT
MINS: / ACOUSTIC MEASUREMENTS/ CALIBRATING/ COMPRESSOR
      BLADES/ DOMAIN WALL/ ENGINE DESIGN/ JET AIRCRAFT NOISE
      / JET ENGINES/ MAGNETIC DOMAINS
```

Figure 16. Example of Search From NASA RECON

TABLE 11. LITERATURE SURVEY EXAMPLE

NUMBER	TITLE	AUTHOR	SOURCE	IN-FLIGHT			BETWEEN-FLIGHT			REMARKS
				SOTA* ROCKET	SOTA NONROCKET	NOVEL**	SOTA ROCKET	SOTA NONROCKET	NOVEL	
1	MAINTAINABILITY OF THE SPACE SHUTTLE ORBITER MAIN ENGINE	GOE, R.T.	ROCKETDYNE				3		1	EARLY SSME MAINTENANCE CONCEPTS
2	DIVERSIFICATION OF ACOUSTICAL HOLOGRAPHY AS A NONDESTRUCT INSPECTION TECHNIQUE TO DETERMINE AGING DAMAGE IN SOLID ROCKET MOTORS	COLLINS, DR. H.	HOLOSONICS, INC.						1	ACOUSTICAL IMAGING TECHNIQUES FOR CRACK DETECTION
3	WELDED ROTOR INSPECTION DEVELOPMENT PROJECT T55-J-027	SUSHIEL, J. VICTOR, S. PAUL, J.	AVCO LYCOMING					2		ULTRASONIC AND ACOUSTIC-EMISSION INSPECTION OF GAS TURBINE POWER SHAFTS
4.	USE OF LASER-POWERED OPTICAL PROXIMITY PROBE IN ADVANCED TURBOFAN ENGINE DEVELOPMENT	HARDY, H. D.	PRATT & WHITNEY AIRCRAFT			1			1	ROTATING COMPONENT CLEARANCE MEASUREMENT
5	ENGINE CONDITION MONITOR SYSTEM TO DETECT FOREIGN OBJECT DAMAGE AND CRACK DEVELOPMENT	HEGNER, H. R.	ITT RESEARCH INSTITUTE			2			2	DETECTION OF BLADE DAMAGE AND CRACK DEVELOPMENT IN AIRCRAFT ENGINES
6	A SYSTEMS ENGINEERING APPROACH TO EFFECTIVE ENGINE CONDITION MONITORING	LEIBY, D. W.	GENERAL ELECTRIC		1			1		INTEGRATED CONDITION MONITORING SYSTEM FOR AIRCRAFT ENGINES
7	FROM CRACKING CRACKS TO BREAKING BEAMS, A REVIEW OF ACOUSTIC EMISSION FOR AIRCRAFT STRUCTURE	BAILEY, C. D. LEWIS, W. H.	LOCKHEED - GEORGIA CO.		1				1	DETECTION OF CRACK INITIATION AND GROWTH IN AIRCRAFT STRUCTURES
8	STATE OF THE ART OF NONDESTRUCTIVE INSPECTION OF AIRCRAFT ENGINES	COMASSAR, D.M.	GENERAL ELECTRIC					3		RECENT DEVELOPMENTS IN ULTRASONIC, EDDY CURRENT, AND PENETRANT INSPECTIONS
9	HIGH RESOLUTION RADIOGRAPHY IN THE AERO-ENGINE INDUSTRY	PARISH, R. W.	AERE					1		X-RAY, GAMMA RAY, AND PARTICLE RADIOGRAPHY
10	WEAR DEBRIS ANALYSIS	PARR, N. L. RITCHIE, J.	ROYAL AIRCRAFT ESTABLISHMENT					1		LUBRICANT PARTICLE DETECTION AND ANALYSIS TECHNIQUES
11	HIGH RESOLUTION ULTRASONIC NONDESTRUCTIVE TESTING OF COMPLEX GEOMETRY COMPONENTS	MORAN, T. J.	AIR FORCE MATERIALS LABORATORY					1		DETECTION AND CHARACTERIZATION OF FLAWS

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY

**NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

techniques which are regarded as proven in concept and successful in regular application in some other industry. Novel refers to any other technology, ranging from the conceptual stage of development to having seen only limited success as a maintenance facility technique. The techniques uncovered through the survey are listed in Table 12 for each category. Table 13 gives a brief summary of each technique along with typical uses, advantages and limitations. Most of the techniques might, with development, be usable in situ; meaning with the engine installed in the vehicle. These in situ techniques are of significant interest because of the savings in turnaround time afforded with no requirements for engine removal.

TABLE 12. INSPECTION TECHNOLOGIES LOCATED BY LITERATURE SURVEY

ROCKET ENGINE	STATE-OF-THE-ART	NOVEL
• ULTRASONIC EXTENSIOMETRY	• ULTRASONIC LEAK DETECTION	• ACOUSTIC HOLOGRAPHY
• ULTRASONIC FLAW DETECTION	• ACOUSTIC EMISSION	• SCANNING ACOUSTIC FLOW DETECTION
• X-RAY RADIOGRAPHY	• PARTICLE RADIOGRAPHY	• ISOTOPE THERMOMETRY
• GAMMA-RAY RADIOGRAPHY	• FLUOROSCOPY	• ISOTOPE TRACER DETECTION
• MAGNETIC PARTICLE	• MAGNETIC PERTURBATION	• REMNANT MAGNETIZATION
• PENETRANT DETECTION	• BARKHAUSEN NOISE ANALYSIS	• PENTOXIDE POLAROGRAPHY
• CONNECTOR CONTINUITY CHECKING	• PARTICLE ANALYSIS	• HYDROGEN POLAROGRAPHY
• HYGROMETRY	• OPTICAL LEAK DETECTION	• LEAK TAPE/COATING
• FLOW LEAK DETECTION	• DIFFERENTIAL RADIOMETRY	• LASER SURFACE SCATTERING
• MASS SPECTROMETRY	• ELLIPSOMETRY	• LASER INTERFEROMETRY
• THERMAL CONDUCTIVITY LEAK CHECKING	• HOLOGRAPHIC MAPPING	• SCANNING OPTICAL PYROMETRY
• TORQUING	• OPTICAL PROXIMITY DETECTION	• HOLOGRAPHIC LEAK DETECTION
• LEAK SOLUTION	• RESISTIVITY MONITORING	• EXO-ELECTRON EMISSION
• BORESCOPING	• EDDY CURRENT	• POSITRON ANNIHILATION
	• HALOGEN LEAK DETECTION	• ELECTRIC CURRENT INJECTION
	• PRESSURE DECAY	• MILLIMETER-WAVE INTERFEROMETRY

TABLE 13. SUMMARY AND COMPARISON OF SURVEYED INSPECTION TECHNIQUES

TECHNIQUE	MEASURANDS	FAILURE TYPE	TYPICAL COMPONENTS	ADVANTAGES	LIMITATIONS
Ultrasonic Extensiometry	Acoustic Wave Propagation	Torque Relaxation Plastic Deformation	Bolts	Direct, Accurate, One-sided Measure of Deformation or Preload.	Individual Records Must Be Kept For Each Component.
Ultrasonic Flaw Detection	Anomalies in Acoustic Wave Properties	Fatigue Foreign Object Damage Crystallographic Changes	Blades Ducts Chambers Shafts	Good Sensitivity and Resolution of Internal Defects. Can be Applied With Access to Only One Side. Can be Readily Interfaced With Computer Processing.	Small Sensor Required for Detection of Small Flaws. Data Interpretation can be difficult.
Ultrasonic Leak Detection	Decrease in Acoustic Impedance at Leak Path	Leakage	Joints Valves	Fast Location of Leaks	Requires Transducer to be Placed Internally. Quantification of Leakage Difficult.
Acoustic Emission	Acoustic Noise Generated by Anomalies in Component Under Load	Fatigue	Ducts Chambers Blades	Excellent Sensitivity and Resolution of Internal Defects. Can be Interfaced With Computer Processing.	Component Must be Loaded Past Previous Maximum Stress Level.
Acoustic Holography	Anomalies in Acoustic Wave Properties	Fatigue Delamination	Chambers Valves	Visual Imaging of Internal Defects. Can Utilize Rapid-Scanning Laser Transducer.	Computer Processing Required. Resolution Limited by Ultrasonic Wavelength. Expensive.
Scanning Acoustic Flow Detection	Flow-generated Acoustic Noise	Restriction	Chambers	Non-intrusive, Rapid Location of Internal Flow Blockage.	Must be High Velocity Flow. Mechanically-coupled Sensor Usually Required.
X-ray Radiography	Anomalies in X-ray Attenuation	Cracks Thickness	Chambers Ducts	Detects Internal Flaws in Wide Variety of Materials. Permanent Record.	Detection of Fatigue and Delaminations Difficult. Expensive. Health Precautions Required.
Gamma-ray Radiography	Anomalies in Gamma-ray Attenuation	Cracks Thickness Restriction	Chambers Ducts Shafts	Isotope Placed Internally in Part Permits More Selective Inspection.	Less Sensitive Than X-rays. Long Exposure Times Needed. Health Precautions Required.

TABLE 13. (Continued)

TECHNIQUE	MEASURANDS	FAILURE TYPE	TYPICAL COMPONENTS	ADVANTAGES	LIMITATIONS
Particle Radiography	Anomalies in Particle Beam Attenuation	Corrosion Cracks Thickness	Composites	Good for Low-Density Materials.	Expensive, Bulky Equipment. Poor Flaw Definition. Health Precautions Required.
Fluoroscopy	Anomalies in X-ray Attenuation	Cracks Clearances	Turbopumps	Detects Flaws and Clearances of Operating Components	Expensive, Bulky Equipment. Health Precautions Required.
Isotope Thermometry	Rate of Beta-ray Emission	Peak Temperature	Blades Chambers	Post-facto Detection of Peak Operating Temperature. Minimal Health Hazard.	No Indication of Duration at Peak Temperature. Must be Impregnated Before Flight.
Isotope Tracer Detection	Radioactive Particles	Wear Galling	Blades Valves Bearings	Sensitive and Selective Wear Detection. Linear Wear/Count Relationship Provides Good Remaining Life Prediction. Minimal Health Hazard	Filter or Some Other Collection System Required In-Flight to Retrieve Particles For Analysis
Magnetic Particles	Preferential Orientation of Magnetic Particles at Surface Flows	Fatigue	Bearings	Simple, Low Cost, Sensitive Detection of Cracks	Component must be Ferromagnetic. Requires Post-Inspection Cleaning.
Remnant Magnetization	Impact-Induced Magnetization Anomalies	Foreign Object Damage	Blades Turbopumps	Simple, Low Cost Detection of Impact Damage	Component must be Ferromagnetic. Not Suitable for Internal Defects.
Magnetic Perturbation	Anomalies of Magnetic-Induction Field In Vicinity of Defect	Fatigue	Bearings	Good Sensitivity to Surface or Near-Surface Flaws	Component must be Ferromagnetic. Not Suitable for Internal Defects.
Barkhausen Noise Analysis	High Frequency Changes in Magnetic Flux Due to Residual Stresses	Fatigue	Bearings	Early Detection of Internal Defects. Applicable to Computer Processing.	Component Must be Ferromagnetic.
Pentoxide Polarography	Current Produced by Electrolysis	Moisture	Chambers Ducts	Good Indications of Water Vapor in a Wide Variety of Environments.	Intrusive Sensor

TABLE 13. (Continued)

TECHNIQUE	MEASURANDS	FAILURE TYPE	TYPICAL COMPONENTS	ADVANTAGES	LIMITATIONS
Hydrogen Polarography	Current Produced by Oxidation of Entrapped Hydrogen	Hydrogen Embrittlement	Chambers Ducts	Good Indication of Hydrogen Content of Material	Slow for Large Area Coverage
Leak Tape/Coating	Visual Color Change Caused by Reaction to Leaking Fluid	Leakage	Joints	Low Cost, Fast Indication of Leakage Produced During Engine Operating Conditions.	No Quantitative Data Produced Tape/Coating Must Cover Entire Leak Path in Extreme Environments.
Particle Analysis	Particles	Wear Galling Contaminants	Bearings Valves	Spectrographic Analysis Gives Good Indication and Life Prediction of Wear.	Filter or Some Other Collection System Required In-Flight. No Distinction Between Wear of Components of Same Materials.
Optical Leak Detection	Absorption of Light At Selected Wavelengths	Leakage	Joints Valves	Non-Contacting, Quantitative Leak Data Provided	Leaking Gas Must Be Distinguishable From Environment
Laser Surface Scattering	Dispersion of Incident Laser Beam	Wear Galling	Valves	Single, Non-Contacting Fiber-Optic Probe Gives Good Indication of Surface Condition	Factors Other Than Wear Can Affect Dispersion. Internal Access Required.
Holographic Deflection Prediction	Deformation Fringes	Wear Distortion	Joints Ducts	Prediction of Excessive Flight Deformations. Applicable to Computer Processing	Expensive Equipment
Borescoping	Visual Surface Anomalies	Cracks Deformation	Blades Valves Injectors	Versatile Detection of Flaws and Fractures. Can Be Film Recorded.	Operator Dependent. Internal Access Required. Not Highly Sensitive.
Differential Radiometry	Differential Absorption of Light At Two Wavelengths.	Leakage	Joints Valves	Quantitative Leak Detection. Better Distinction Between Environment & Leaking Gas	Longer Sample Path Required.
Ellipsometry	Changes in State of Reflected Polarized Light	Wear Surface Films	Bearings Valves	Extremely High Sensitivity. Non-contacting.	Requires Precise Alignment of Equipment & Skilled Operators

TABLE 13. (Continued)

TECHNIQUE	MEASURANDS	FAILURE TYPE	TYPICAL COMPONENTS	ADVANTAGES	LIMITATIONS
Penetrant Detection	Absorption and Emission of Penetrant Fluid In Defects	Cracks Porosity	Chambers Ducts	Low Cost, Highly Sensitive Indications of Surface Defects	Requires Post-Inspection Cleaning. Crack Must Be Open to the Surface
Holographic Surface Mapping	Fringe Patterns Produced By Dual-Wavelength Hologram	Fatigue Wear	Blades Valves	Simpler Than Interferometry. No Pre-Flight Reference Required	Changes in Surface Conditions More Difficult to Detect
Optical Proximity Detection	Reflection of Incident Laser Beam	Interference	Blades Valves	Minimal Access to Gap Required. Non-Contacting	Reflective Characteristics of Reflective Surface Must Be Known
Scanning Optical Pyrometry	Temperature Anomalies	Restriction	Chambers	Fast Location of Blocked Coolant Passages. Can Be Automated and Remote	Partial Restriction Difficult to Detect. Purging Gas Must Be Hot
Holographic Leak Detection	Leak-Induced Fringes of a Multiple-Pulse Laser	Leakage	Joints Chambers	In-Toto Detection of Multiple Leaks. Quantitative Data. Very Fast.	Expensive Equipment
Exo-Election Emission	Stimulated Emission of Electrons	Fatigue	Blades Bearings Ducts	Excellent Fatigue Characterization and Life Prediction. Non-Contacting	Surface and Near-Surface Fatigue Only
Positron Annihilation	Beta-Ray Emission	Fatigue	Blades Ducts	Good Fatigue Characterization and Life Prediction	Surface and Near-Surface Only. Must be Exposed to Vacuum. Requires Positron Source For Injection Into Part.
Electric Current Injection	Anomalites In Surface Temperature Induced By Defects	Fatigue	Blades Ducts	Thermal Mapping Of Electrically Heated Surface Provides Fast Indication of Flaws	Surface and Near-Surface Flaws Only. Poor Resolution.
Resistivity Monitoring	Resistance Changes Due To Cryogenic Leak	Leakage	Joints	Leakage Can Be Detected At Operating Temperature	Not Highly Sensitive. Little Quantitative Data Produced.
Eddy Current	Anomalies in Electric Conductivity	Fatigue	Blades Chambers Ducts	Good Sensitivity for Moderate Cost. Applicable to Computer Processing	Surface and Near-Surface Defects Only. Affected By Many Material Variables.

TABLE 13. (Concluded)

TECHNIQUE	MEASURANDS	FAILURE TYPE	TYPICAL COMPONENTS	ADVANTAGES	LIMITATIONS
Millimeter-Wave Interferometry	Differential Millimeter-Wave Reflection	Cracks	Chambers	Differential Approach Eliminates Most Material Variables Non-Contacting	Not Highly Sensitive
Connector Continuity Checking	Continuity	Connector Loose	Electrical Connectors	Direct, Low Cost, Verification of Connector Operation. Can be Automated	No Indication if Continuity Loss Is Imminent
Halogen Leak Detection	Rate of Ion Formation	Leakage	Joints Valves	Sensitive to Low Leak Rates	Requires Tracer Gas. Sensitive to Background Gases. Insensitive For High Leak Rates
Hygrometer	Impedance	Moisture	Chambers Ducts	Low Cost. Fast Response	Intrusive Sensor
Flow Leak Detection	Leakage Flow	Leakage	Joints Valves	Direct Measurement of Leakage Flowrate.	Time-Consuming Procedure. Many Possible Errors. Cannot Detect Low Leak Rates Location Difficult
Mass Spectrometry	Ion Concentration	Leakage	Joints Valves	Highly Sensitive	Becomes Saturated At Higher Leak Rates. Slow
Thermal Conductivity Leak Checking	Thermal Conductivity	Leakage	Joints Valves	Relatively Sensitive To Leak But Insensitive To Background Gas. Fast. Low Cost	Tracer Gas Required.
Torquing	Torque	Bolt Relaxation Excessive Friction	Bolts Turbopumps	Direct Indication of Insufficient or Excessive Torque	Operator Error Can Cause Damage. Slow
Leak Detection Solution	Leakage	Leakage	Joints	Direct, Visual Location of Leak	Requires Post-Inspection Cleaning. Operator-Dependent. Slow. No Quantitative Data
Pressure Decay	Pressure Loss	Leakage	Joints Valves	Simple, Low Cost. Indication of Leak.	Volume of Test Component Must Be Known. Slow. Location Of Leak Difficult.

DEFINITION OF CANDIDATE TECHNOLOGY

Analysis of Task I Failure Modes

Each of the sixteen failure mode categories identified by Task I were examined to determine what between-flight diagnostic requirements would be necessary to predict that incipient failure. The measurands associated with each failure mode were determined first as listed in Table 14. The measurands are shown in Fig. 17 and Appendix C, along with the propogation of the failure to give a clear indication of where they become detectible.

TABLE 14. INSPECTABLE FAILURE MODES AND MEASURANDS IDENTIFIED

2 COOLANT PASSAGE LEAKAGE
- METAL EMBRITTLEMENT
- RESTRICTION
- TUBE SPLITS

3 JOINT LEAKAGE
- WARPING DISTORTION
- TORQUE RELAXATION
- LEAK

5 HIGH TURBOPUMP TORQUE
- PHYSICAL INTERFERENCE
- EXCESSIVE TEMPERATURE
- EXCESSIVE FRICTION

6 CRACKED TURBINE BLADES
- HIGH TEMPERATURE TRANSIENT
- FOREIGN OBJECT DAMAGE
- FATIGUE

7 CRACKED CONVOLUTION, BELLOWS SHIELD
- HIGH TEMPERATURE TRANSIENT
- FOREIGN OBJECT DAMAGE
- FATIGUE

8 LOOSE ELECTRICAL CONNECTOR
- HIGH TEMPERATURE TRANSIENT
- TORQUE RELAXATION
- CONTINUITY

9 BALL BEARING DAMAGE
- EXCESSIVE TEMPERATURE
- EXCESSIVE FRICTION
- WEAR

10 TUBE FRACTURE
- LINE DEFLECTION
- FATIGUE

11 TURBOPUMP SEAL LEAKAGE
- PHYSICAL INTERFERENCE
- EXCESSIVE TEMPERATURE
- EXCESSIVE FRICTION

13 VALVE FAILURE
- MOISTURE
- INTERNAL FRACTURE
- GALLING
- CONTAMINATION

14 INTERNAL VALVE LEAKAGE
- FRETTING
- CONTAMINATION
- TORQUE RELAXATION
- DISTORTION
- STUCK COMPONENTS
- LEAKAGE

15 REGULATOR DISCREPANCIES
- CONTAMINATION
- LEAKAGE
- EXCESSIVE FRICTION

16 CONTAMINATED HYDRAULIC CONTROL
- CONTAMINATION
- LEAKAGE
- EXCESSIVE FRICTION

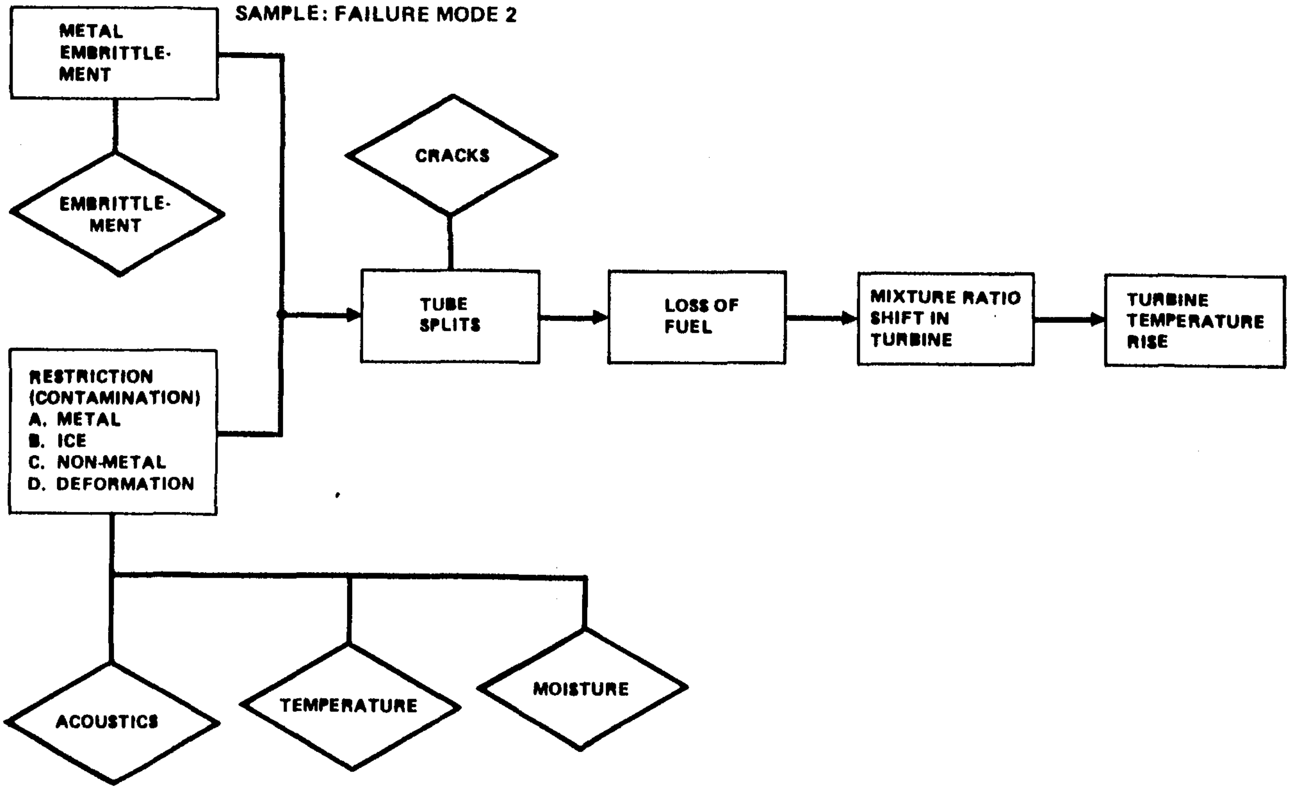


Figure 17. Task I Propagation Diagrams Reviewed to Determine Diagnostic Measurands

Correlation of Surveyed Techniques With Diagnostic Requirements

After locating between-flight inspection technology with the literature survey and defining diagnostic requirements based on the characterization of past failures, the inspection techniques were matched to applicable failure mode measurands. This correlation is shown in Table 15. The measurands are listed across the top, grouped by failure mode and the inspection techniques are listed at the left, grouped by detection type. Each possible inspection is indicated by an N, S, or R, depending on whether the use of that technique could be considered novel, state of the art, or rocket engine state of the art for that particular failure type. This table gives a singular summary of the multiplicity of use of each inspection technique, the number of techniques available toward each failure mode and the present level of detectibility of each failure type.

DESIGN AND INSPECTION COMPATIBILITY ASSESSMENT

Accessibility and engine configuration modification requirements to utilize each inspection technique were identified and included in the technology grading. In addition, engine configuration modifications which could provide enhanced inspection capability were identified and assumed in the grading.

TABLE 15. BETWEEN-FLIGHT FAILURE MODE DETECTION TECHNOLOGY

LEGEND

N = NOVEL TECHNOLOGY

S = STATE OF THE ART TECHNOLOGY

R = ROCKET TECHNOLOGY

FAILURE MODES/MEASURANDS

BETWEEN-FLIGHT INSPECTION TECHNIQUES	1) BOLT TORQUE RELAXATION ✦	VIBRATION	SCREW LOOSENING	LEAK	FRETTING	2) COOLANT PASSAGE LEAKAGE	METAL EMBRITTLEMENT	RESTRICTION	TUBE SPLITS	3) JOINT LEAKAGE	WARPING DISTORTION	TORQUE RELAXATION	LEAK	4) TRANSFER TUBE CRACKS ✦	HIGH TEMPERATURE TRANSIENT	FATIGUE
ACOUSTIC																
ULTRASONIC EXTENSIOMETRY												R				
ULTRASONIC FLAW DETECTION																
ULTRASONIC LEAK DETECTION													S			
ACOUSTIC EMISSION																
ACOUSTIC HOLOGRAPHY									N							
SCANNING ACOUSTIC FLOW DETECTION								N								
RADIOGRAPHIC																
X-RAY RADIOGRAPHY									R							
GAMMA-RAY RADIOGRAPHY								R								
PARTICLE RADIOGRAPHY ✦ ✦																
FLUOROSCOPY ✦ ✦																
ISOTOPE THERMOMETRY																
ISOTOPE TRACER DETECTION																
MAGNETIC																
MAGNETIC PARTICLE ✦ ✦ ✦																
REMNANT MAGNETIZATION																
MAGNETIC PERTURBATION ✦ ✦ ✦																
BARKHAUSEN NOISE ANALYSIS ✦ ✦ ✦																
CHEMICAL																
PENTOXIDE POLAROGRAPHY								N								
HYDROGEN POLAROGRAPHY							N									
LEAK TAPE/COATING													N			
PARTICLE ANALYSIS																
OPTIC																
OPTICAL LEAK DETECTION													S			
LASER SURFACE SCATTERING																
HOLOGRAPHIC DEFLECTION PREDICTION											N					
BORESCOPING																
DIFFERENTIAL RADIOMETRY													S			
ELLIPSOMETRY ✦ ✦																
PENETRANT DETECTION																
HOLOGRAPHIC SURFACE MAPPING																
OPTICAL PROXIMITY DETECTION																
SCANNING OPTICAL PYROMETRY								N								
HOLOGRAPHIC LEAK DETECTION									N				N			
ELECTRICAL																
EXO-ELECTRON EMISSION																
POSITRON ANNIHILATION																
ELECTRIC CURRENT INJECTION																
RESISTIVITY MONITORING													S			
EDDY CURRENT																
MILLIMETER-WAVE INTERFEROMETRY									N							
CONNECTOR CONTINUITY CHECKING																
HALOGEN LEAK DETECTION													S			
HYGROMETRY								R								
OTHER																
FLOW LEAK DETECTION													R			
MASS SPECTROMETRY													R			
THERMAL CONDUCTIVITY LEAK CHECKING													R			
TORQUING												R				
LEAK SOLUTION													R			
PRESSURE DECAY													S			

BETWEEN-FLIGHT INSPECTION TECHNIQUES	5) HIGH TURBOPUMP TORQUE	PHYSICAL INTERFERENCE	EXCESSIVE TEMPERATURE	EXCESSIVE FRICTION	6) CRACKED TURBINE BLADES	HIGH TEMPERATURE TRANSIENT	FOREIGN OBJECT DAMAGE	FATIGUE	7) CRACKED CONVOLUTION, BELLOWS SHIELD	HIGH TEMPERATURE TRANSIENT	FOREIGN OBJECT DAMAGE	FATIGUE	8) LOOSE ELECTRICAL CONNECTOR	HIGH TEMPERATURE TRANSIENT	TORQUE RELAXATION	CONTINUITY
ACOUSTIC																
ULTRASONIC EXTENSIOMETRY																
ULTRASONIC FLAW DETECTION							N	S			N	R				
ULTRASONIC LEAK DETECTION																
ACOUSTIC EMISSION																
ACOUSTIC HOLOGRAPHY																
SCANNING ACOUSTIC FLOW DETECTION																
RADIOGRAPHIC																
X-RAY RADIOGRAPHY																
GAMMA-RAY RADIOGRAPHY																
PARTICLE RADIOGRAPHY ✦ ✦																
FLUOROSCOPY ✦ ✦																
ISOTOPE THERMOMETRY			N			N				N				N		
ISOTOPE TRACER DETECTION				N			N									
MAGNETIC																
MAGNETIC PARTICLE ✦ ✦ ✦																
REMNANT MAGNETIZATION							N				N					
MAGNETIC PERTURBATION ✦ ✦ ✦																
BARKHAUSEN NOISE ANALYSIS ✦ ✦ ✦																
CHEMICAL																
PENTOXIDE POLAROGRAPHY																
HYDROGEN POLAROGRAPHY																
LEAK TAPE/COATING																
PARTICLE ANALYSIS				S												
OPTIC																
OPTICAL LEAK DETECTION																
LASER SURFACE SCATTERING																
HOLOGRAPHIC DEFLECTION PREDICTION																
BORESCOPING			R				R	R			R	R				
DIFFERENTIAL RADIOMETRY																
ELLIPSOMETRY ✦ ✦																
PENETRANT DETECTION												R				
HOLOGRAPHIC SURFACE MAPPING								S				S				
OPTICAL PROXIMITY DETECTION		S														
SCANNING OPTICAL PYROMETRY																
HOLOGRAPHIC LEAK DETECTION																
ELECTRICAL																
EXO-ELECTRON EMISSION								N				N				
POSITRON ANNIHILATION								N				N				
ELECTRIC CURRENT INJECTION								N				N				
RESISTIVITY MONITORING																
EDDY CURRENT								S				S				
MILLIMETER-WAVE INTERFEROMETRY																
CONNECTOR CONTINUITY CHECKING																R
HALOGEN LEAK DETECTION																
HYGROMETRY																
OTHER																
FLOW LEAK DETECTION																
MASS SPECTROMETRY																
THERMAL CONDUCTIVITY LEAK CHECKING																
TORQUING				R											R	
LEAK SOLUTION																
PRESSURE DECAY																

BETWEEN-FLIGHT INSPECTION TECHNIQUES	9) BALL BEARING DAMAGE	EXCESSIVE TEMPERATURE	EXCESSIVE FRICTION	WEAR	10) TUBE FRACTURE	LINE DEFLECTION	FATIGUE	11) TURBOPUMP SEAL LEAKAGE	PHYSICAL INTERFERENCE	EXCESSIVE TEMPERATURE	EXCESSIVE FRICTION	12) LUBE PARTICLE ANOMALIES ✦	CONTAMINATION	13) VALVE FAILURE	MOISTURE	INTERNAL FRACTURE	GALLING	CONTAMINATION
ACOUSTIC																		
ULTRASONIC EXTENSIOMETRY																		
ULTRASONIC FLAW DETECTION				S		N	R											
ULTRASONIC LEAK DETECTION																		S
ACOUSTIC EMISSION							S											
ACOUSTIC HOLOGRAPHY																N		
SCANNING ACOUSTIC FLOW DETECTION																		
RADIOGRAPHIC																		
X-RAY RADIOGRAPHY							R											
GAMMA-RAY RADIOGRAPHY																		
PARTICLE RADIOGRAPHY ✦ ✦																		
FLUOROSCOPY ✦ ✦																		
ISOTOPE THERMOMETRY		N								N								
ISOTOPE TRACER DETECTION				N							N						N	
MAGNETIC																		
MAGNETIC PARTICLE ✦ ✦ ✦				R														
REMNANT MAGNETIZATION																		
MAGNETIC PERTURBATION ✦ ✦ ✦				S														
BARKHAUSEN NOISE ANALYSIS ✦ ✦ ✦				S														
CHEMICAL																		
PENTOXIDE POLAROGRAPHY															N			
HYDROGEN POLAROGRAPHY																		
LEAK TAPE/COATING																		
PARTICLE ANALYSIS				S							S							S
OPTIC																		
OPTICAL LEAK DETECTION																		S
LASER SURFACE SCATTERING																	N	
HOLOGRAPHIC DEFLECTION PREDICTION						N												
BORESCOPING		R															R	
DIFFERENTIAL RADIOMETRY																		S
ELLIPSOMETRY ✦ ✦																		
PENETRANT DETECTION							R											
HOLOGRAPHIC SURFACE MAPPING																	S	
OPTICAL PROXIMITY DETECTION									S									
SCANNING OPTICAL PYROMETRY																		
HOLOGRAPHIC LEAK DETECTION																		
ELECTRICAL																		
EXO-ELECTRON EMISSION				N			N											
POSITRON ANNIHILATION				N			N											
ELECTRIC CURRENT INJECTION							N											
RESISTIVITY MONITORING																		
EDDY CURRENT				S			S											
MILLIMETER-WAVE INTERFEROMETRY																		
CONNECTOR CONTINUITY CHECKING																		
HALOGEN LEAK DETECTION																		
HYGROMETRY															R			
OTHER																		
FLOW LEAK DETECTION											R							
MASS SPECTROMETRY																		
THERMAL CONDUCTIVITY LEAK CHECKING																		
TORQUING			R								R							
LEAK SOLUTION																		
PRESSURE DECAY																		

BETWEEN-FLIGHT INSPECTION TECHNIQUES	14) INTERNAL VALVE LEAKAGE	FRETTING	CONTAMINATION	TORQUE RELAXATION	DISTORTION	STUCK COMPONENTS	LEAKAGE	15) REGULATOR DISCREPANCIES	CONTAMINATION	LEAKAGE	EXCESSIVE FRICTION	16) CONTAMINATED HYDRAULIC CONTROL	CONTAMINATION	LEAKAGE	EXCESSIVE FRICTION
ACOUSTIC															
ULTRASONIC EXTENSIOMETRY															
ULTRASONIC FLAW DETECTION															
ULTRASONIC LEAK DETECTION							S			S				S	
ACOUSTIC EMISSION															
ACOUSTIC HOLOGRAPHY															
SCANNING ACOUSTIC FLOW DETECTION															
RADIOGRAPHIC															
X-RAY RADIOGRAPHY															
GAMMA-RAY RADIOGRAPHY															
PARTICLE RADIOGRAPHY ✦ ✦															
FLUOROSCOPY ✦ ✦															
ISOTOPE THERMOMETRY															
ISOTOPE TRACER DETECTION		N													
MAGNETIC															
MAGNETIC PARTICLE ✦ ✦ ✦															
REMNANT MAGNETIZATION															
MAGNETIC PERTURBATION ✦ ✦ ✦															
BARKHAUSEN NOISE ANALYSIS ✦ ✦ ✦															
CHEMICAL															
PENTOXIDE POLAROGRAPHY															
HYDROGEN POLAROGRAPHY															
LEAK TAPE/COATING															
PARTICLE ANALYSIS			S						S		S		S		S
OPTIC															
OPTICAL LEAK DETECTION							S			S				S	
LASER SURFACE SCATTERING		N													
HOLOGRAPHIC DEFLECTION PREDICTION															
BORESCOPING		R													
DIFFERENTIAL RADIOMETRY							S			S				S	
ELLIPSOMETRY ✦ ✦															
PENETRANT DETECTION															
HOLOGRAPHIC SURFACE MAPPING		S													
OPTICAL PROXIMITY DETECTION						S									
SCANNING OPTICAL PYROMETRY															
HOLOGRAPHIC LEAK DETECTION															
ELECTRICAL															
EXO-ELECTRON EMISSION															
POSITRON ANNIHILATION															
ELECTRIC CURRENT INJECTION															
RESISTIVITY MONITORING															
EDDY CURRENT															
MILLIMETER-WAVE INTERFEROMETRY															
CONNECTOR CONTINUITY CHECKING															
HALOGEN LEAK DETECTION							S			S					
HYGROMETRY															
OTHER															
FLOW LEAK DETECTION							R			R					
MASS SPECTROMETRY							R			R					
THERMAL CONDUCTIVITY LEAK CHECKING							R			R					
TORQUING				R											
LEAK SOLUTION															
PRESSURE DECAY							S			S				S	

✦ FAILURE MODE OBVIATED BY IMPROVED DESIGN

✦ ✦ TECHNIQUE NOT APPLICABLE TO ANY FAILURE MODES

✦ ✦ ✦ TECHNIQUE NOT DIAGNOSTIC BECAUSE COMPONENT DISASSEMBLY IS REQUIRED

Accessibility Requirements

The accessibility requirements were determined for each inspection technique as applied to each failure mode. It was found that six types of accesibility could be defined, in approximately decreasing desirability, as follows:

A - Direct External Access: where no interference problems with other components would normally be encountered.

B - External Access with Interference: where considerations regarding the engine configuration, such as duct routing, would usually be required of the design in order to provide adequate inspection accessibility.

C - Internal Port Access: where an inspection port (or flight instrumentation sensor) would have to be removed to provide access to the interior or a component.

D - Component Removal Access: where the removal of a component would be necessary to provide adequate internal access for the inspection.

E - Component Disassembly Access: where, after removal, a component would require major disassembly to accomplish the inspection.

F - Component Addition Access: where the addition of an on-board component would be required in order to carry out the ground inspection.

Table 16 indicates the accessibility requirements, using the above accessibility codes, in the same format as was used in the technology/failure mode correlation matrix (Table 15). This information became an input to the technology grading, affecting the engine application, hazard, integration, and development descriptors as appropriate.

Configuration Modifications

Although most of the inspection techniques would impact on engine design configuration, few would affect it in a manner inconsistent with typical design requirements. Access ports and component removal needs are considerations normally encountered and, with prudent design practices, will have minimal impact on engine weight or performance. Access to components for inspection has not usually been a problem in regard to preventing failures. The lack of adequately-developed (for rocket engine use) inspection or prediction technology has been the major hinderance. There are two engine sub-components, however, that have been typically difficult to apply state-lf-the-art rocket engine inspection technology to. Both are subcomponents of a turbopump, a high speed precision machine which operates under extreme environmental conditions.

TABLE 16. BETWEEN-FLIGHT INSPECTION ACCESSIBILITY

LEGEND

EXTERNAL ACCESS CODES

A = DIRECT ACCESS
B = INTERFERENCE WITH OTHER COMPONENTS

INTERNAL ACCESS CODES

C = PORT REQUIRED
D = COMPONENT REMOVAL REQUIRED
E = COMPONENT DIS-ASSEMBLY REQUIRED
F = COMPONENT ADDITION REQUIRED

BETWEEN-FLIGHT INSPECTION TECHNIQUES \ FAILURE MODES/MEASURANDS	1) BOLT TORQUE RELAXATION ◆	VIBRATION	SCREW LOOSENING	LEAK	FRETTING	2) COOLANT PASSAGE LEAKAGE	METAL EMBRITTLEMENT	RESTRICTION	TUBE SPLITS	3) JOINT LEAKAGE	WARPING DISTORTION	TORQUE RELAXATION	LEAK	4) TRANSFER TUBE CRACKS ◆	HIGH TEMPERATURE TRANSIENT	FATIGUE
ACOUSTIC																
ULTRASONIC EXTENSIOMETRY												B				
ULTRASONIC FLAW DETECTION																
ULTRASONIC LEAK DETECTION													C			
ACOUSTIC EMISSION																
ACOUSTIC HOLOGRAPHY									A							
SCANNING ACOUSTIC FLOW DETECTION								A								
RADIOGRAPHIC																
X-RAY RADIOGRAPHY									B							
GAMMA-RAY RADIOGRAPHY								C								
PARTICLE RADIOGRAPHY ◆◆◆																
FLUOROSCOPY ◆◆																
ISOTOPE THERMOMETRY																
ISOTOPE TRACER DETECTION																
MAGNETIC																
MAGNETIC PARTICLE ◆◆◆																
REMNANT MAGNETIZATION																
MAGNETIC PERTURBATION ◆◆◆																
BARKHAUSEN NOISE ANALYSIS ◆◆◆																
CHEMICAL																
PENTOXIDE POLAROGRAPHY								C								
HYDROGEN POLAROGRAPHY							A									
LEAK TAPE/COATING													B			
PARTICLE ANALYSIS																
OPTIC																
OPTICAL LEAK DETECTION													B			
LASER SURFACE SCATTERING																
HOLOGRAPHIC DEFLECTION PREDICTION											B					
BORESCOPING																
DIFFERENTIAL RADIOMETRY													B			
ELLIPSOMETRY ◆◆																
PENETRANT DETECTION																
HOLOGRAPHIC SURFACE MAPPING																
OPTICAL PROXIMITY DETECTION																
SCANNING OPTICAL PYROMETRY								A								
HOLOGRAPHIC LEAK DETECTION									B				B			
ELECTRICAL																
EXO-ELECTRON EMISSION																
POSITRON ANNIHILATION																
ELECTRIC CURRENT INJECTION																
RESISTIVITY MONITORING													B			
EDDY CURRENT																
MILLIMETER-WAVE INTERFEROMETRY									A							
CONNECTOR CONTINUITY CHECKING																
HALOGEN LEAK DETECTION													C			
HYGROMETRY								C								
OTHER																
FLOW LEAK DETECTION													C			
MASS SPECTROMETRY													C			
THERMAL CONDUCTIVITY LEAK CHECKING													B			
TORQUING												B				
LEAK SOLUTION													B			
PRESSURE DECAY													C			

BETWEEN-FLIGHT INSPECTION TECHNIQUES \ FAILURE MODES/MEASURANDS	5) HIGH TURBOPUMP TORQUE	PHYSICAL INTERFERENCE	EXCESSIVE TEMPERATURE	EXCESSIVE FRICTION	6) CRACKED TURBINE BLADES	HIGH TEMPERATURE TRANSIENT	FOREIGN OBJECT DAMAGE	FATIGUE	7) CRACKED CONVOLUTION, BELLOWS SHIELD	HIGH TEMPERATURE TRANSIENT	FOREIGN OBJECT DAMAGE	FATIGUE	8) LOOSE ELECTRICAL CONNECTOR	HIGH TEMPERATURE TRANSIENT	TORQUE RELAXATION	CONTINUITY
ACOUSTIC																
ULTRASONIC EXTENSIOMETRY																
ULTRASONIC FLAW DETECTION							D	D			D	D				
ULTRASONIC LEAK DETECTION																
ACOUSTIC EMISSION																
ACOUSTIC HOLOGRAPHY																
SCANNING ACOUSTIC FLOW DETECTION																
RADIOGRAPHIC																
X-RAY RADIOGRAPHY																
GAMMA-RAY RADIOGRAPHY																
PARTICLE RADIOGRAPHY ◆◆◆																
FLUOROSCOPY ◆◆																
ISOTOPE THERMOMETRY			E			C				C				A		
ISOTOPE TRACER DETECTION				F			F									
MAGNETIC																
MAGNETIC PARTICLE ◆◆◆																
REMNANT MAGNETIZATION							C				D					
MAGNETIC PERTURBATION ◆◆◆																
BARKHAUSEN NOISE ANALYSIS ◆◆◆																
CHEMICAL																
PENTOXIDE POLAROGRAPHY																
HYDROGEN POLAROGRAPHY																
LEAK TAPE/COATING																
PARTICLE ANALYSIS				F												
OPTIC																
OPTICAL LEAK DETECTION																
LASER SURFACE SCATTERING																
HOLOGRAPHIC DEFLECTION PREDICTION																
BORESCOPING			C				C	C			C	C				
DIFFERENTIAL RADIOMETRY																
ELLIPSOMETRY ◆◆																
PENETRANT DETECTION												E				
HOLOGRAPHIC SURFACE MAPPING								D				D				
OPTICAL PROXIMITY DETECTION		C														
SCANNING OPTICAL PYROMETRY																
HOLOGRAPHIC LEAK DETECTION																
ELECTRICAL																
EXO-ELECTRON EMISSION								D				D				
POSITRON ANNIHILATION								E				D				
ELECTRIC CURRENT INJECTION								D				D				
RESISTIVITY MONITORING																
EDDY CURRENT								E				D				
MILLIMETER-WAVE INTERFEROMETRY																
CONNECTOR CONTINUITY CHECKING																A
HALOGEN LEAK DETECTION																
HYGROMETRY																
OTHER																
FLOW LEAK DETECTION																
MASS SPECTROMETRY																
THERMAL CONDUCTIVITY LEAK CHECKING																
TORQUING				C											A	
LEAK SOLUTION																
PRESSURE DECAY																

BETWEEN-FLIGHT INSPECTION TECHNIQUES \ FAILURE MODES/MEASURANDS	9) BALL BEARING DAMAGE	EXCESSIVE TEMPERATURE	EXCESSIVE FRICTION	WEAR	10) TUBE FRACTURE	LINE DEFLECTION	FATIGUE	11) TURBOPUMP SEAL LEAKAGE	PHYSICAL INTERFERENCE	EXCESSIVE TEMPERATURE	EXCESSIVE FRICTION	12) LUBE PARTICLE ANOMALIES ◆	CONTAMINATION	13) VALVE FAILURE	MOISTURE	INTERNAL FRACTURE	GALLING	CONTAMINATION
ACOUSTIC																		
ULTRASONIC EXTENSIOMETRY																		
ULTRASONIC FLAW DETECTION				E		B	B											
ULTRASONIC LEAK DETECTION																		C
ACOUSTIC EMISSION							B											
ACOUSTIC HOLOGRAPHY																C		
SCANNING ACOUSTIC FLOW DETECTION																		
RADIOGRAPHIC																		
X-RAY RADIOGRAPHY							B											
GAMMA-RAY RADIOGRAPHY																		
PARTICLE RADIOGRAPHY ◆◆◆																		
FLUOROSCOPY ◆◆																		
ISOTOPE THERMOMETRY		E								E								
ISOTOPE TRACER DETECTION				F							F						F	
MAGNETIC																		
MAGNETIC PARTICLE ◆◆◆				E														
REMNANT MAGNETIZATION																		
MAGNETIC PERTURBATION ◆◆◆				E														
BARKHAUSEN NOISE ANALYSIS ◆◆◆				E														
CHEMICAL																		
PENTOXIDE POLAROGRAPHY															C			
HYDROGEN POLAROGRAPHY																		
LEAK TAPE/COATING																		
PARTICLE ANALYSIS				F							F							F
OPTIC																		
OPTICAL LEAK DETECTION																		C
LASER SURFACE SCATTERING																	C	
HOLOGRAPHIC DEFLECTION PREDICTION						A												
BORESCOPING		C															C	
DIFFERENTIAL RADIOMETRY																		C
ELLIPSOMETRY ◆◆																		
PENETRANT DETECTION							B											
HOLOGRAPHIC SURFACE MAPPING																	C	
OPTICAL PROXIMITY DETECTION									C									
SCANNING OPTICAL PYROMETRY																		
HOLOGRAPHIC LEAK DETECTION																		
ELECTRICAL																		
EXO-ELECTRON EMISSION				E			B											
POSITRON ANNIHILATION				E			D											
ELECTRIC CURRENT INJECTION							B											
RESISTIVITY MONITORING																		
EDDY CURRENT				E			B											
MILLIMETER-WAVE INTERFEROMETRY																		
CONNECTOR CONTINUITY CHECKING																		
HALOGEN LEAK DETECTION																		
HYGROMETRY															C			
OTHER																		
FLOW LEAK DETECTION											C							
MASS SPECTROMETRY																		
THERMAL CONDUCTIVITY LEAK CHECKING																		
TORQUING			C								C							
LEAK SOLUTION																		
PRESSURE DECAY																		

BETWEEN-FLIGHT INSPECTION TECHNIQUES \ FAILURE MODES/MEASURANDS	14) INTERNAL VALVE LEAKAGE	FRETTING	CONTAMINATION	TORQUE RELAXATION	DISTORTION	STUCK COMPONENTS	LEAKAGE	15) REGULATOR DISCREPANCIES	CONTAMINATION	LEAKAGE	EXCESSIVE FRICTION	16) CONTAMINATED HYDRAULIC CONTROL	CONTAMINATION	LEAKAGE	EXCESSIVE FRICTION
ACOUSTIC															
ULTRASONIC EXTENSIOMETRY															
ULTRASONIC FLAW DETECTION															
ULTRASONIC LEAK DETECTION							C			C				C	
ACOUSTIC EMISSION															
ACOUSTIC HOLOGRAPHY															
SCANNING ACOUSTIC FLOW DETECTION															
RADIOGRAPHIC															
X-RAY RADIOGRAPHY															
GAMMA-RAY RADIOGRAPHY															
PARTICLE RADIOGRAPHY ◆◆◆															
FLUOROSCOPY ◆◆															
ISOTOPE THERMOMETRY															
ISOTOPE TRACER DETECTION		F													
MAGNETIC															
MAGNETIC PARTICLE ◆◆◆															
REMNANT MAGNETIZATION															
MAGNETIC PERTURBATION ◆◆◆															
BARKHAUSEN NOISE ANALYSIS ◆◆◆															
CHEMICAL															
PENTOXIDE POLAROGRAPHY															
HYDROGEN POLAROGRAPHY															
LEAK TAPE/COATING															
PARTICLE ANALYSIS			F						F		F		F		F
OPTIC															
OPTICAL LEAK DETECTION							C			C				C	
LASER SURFACE SCATTERING		C													
HOLOGRAPHIC DEFLECTION PREDICTION															
BORESCOPING		C													
DIFFERENTIAL RADIOMETRY							C			C				C	
ELLIPSOMETRY ◆◆															
PENETRANT DETECTION															
HOLOGRAPHIC SURFACE MAPPING		C													
OPTICAL PROXIMITY DETECTION						C									
SCANNING OPTICAL PYROMETRY															
HOLOGRAPHIC LEAK DETECTION															
ELECTRICAL															
EXO-ELECTRON EMISSION															
POSITRON ANNIHILATION															
ELECTRIC CURRENT INJECTION															
RESISTIVITY MONITORING															
EDDY CURRENT															
MILLIMETER-WAVE INTERFEROMETRY															
CONNECTOR CONTINUITY CHECKING															
HALOGEN LEAK DETECTION							C			C					
HYGROMETRY															
OTHER															
FLOW LEAK DETECTION							C			C					
MASS SPECTROMETRY							C			C					
THERMAL CONDUCTIVITY LEAK CHECKING							C			C					
TORQUING				C											
LEAK SOLUTION															
PRESSURE DECAY							C			C				C	

◆ FAILURE MODE OBVIATED BY IMPROVED DESIGN

◆◆ TECHNIQUE NOT APPLICABLE TO ANY FAILURE MODES

◆◆◆ TECHNIQUE NOT DIAGNOSTIC BECAUSE COMPONENT DISASSEMBLY IS REQUIRED

The bearings, because of the manner in which they must be structurally supported, have been difficult to inspect unless the turbopump is disassembled. In-flight vibration monitoring has not been a reliable indication of bearing condition either. The diagnostic technology identified by this study should alleviate this problem as well as provide much better failure prediction with only modest configuration considerations. The isotope tracer techniques require an inline device, sensor for in-flight monitoring or collector for between-flight analysis, which could be external to the turbopump. The fiberoptic bearing detector requires only fiberoptic access to the outside of the bearing outer race, a consideration which might be difficult to retrofit on an existing turbopump but which could be incorporated in a new design.

Access to turbopump turbine blades is typically difficult, especially for multiple-stage turbines, often requiring removal of the turbopump from the engine. These subcomponents operate at very high rotational speeds in a hot gas and thus are vulnerable to over-temperature conditions and fatigue lives which must be carefully monitored and predicted. Improved accessibility to the turbines through the use of removable turbine housings (not requiring the removal of the entire turbopump) or some other means would be highly desirable. This would enhance the use of a wide number of inspection technologies which are otherwise difficult to employ. The development of this design feature would be significant but the benefits of great improvements in inspectability, life prediction and repair would be a major advancement in the maintainability of reusable rocket engines.

TECHNOLOGY SELECTION AND UPGRADING

The method for evaluation and ranking of the technologies was developed in cooperation with the Task II effort. The evaluation method selected was, as in Task II, a two-step approach. First, two clear-out screens were identified and applied to the techniques, resulting in the elimination of six technologies. Lumped descriptors, each made up of many specific descriptors, were then defined. The technologies applicable to each figure mode were graded using these lumped descriptors, thus providing a ranking of the techniques. All techniques were assumed to be equally developed for use on rocket engines.

Identification of Unacceptable Techniques

Before ranking the techniques, it was desirable to eliminate from further consideration those technologies which were clearly not amenable to the goals of this study. Although many criteria with which to screen out these unacceptable techniques were considered, only two appeared to be unequivocal. They are:

1. Need - Is the technique applicable to any of the Task I failure modes? Although valuable diagnostic techniques which were identified by the literature survey, three technologies, particle radiography, fluoroscopy, and ellipsometry were eliminated by these screen.

2. Component Disassembly - Does the use of this technique necessitate major disassembly of a component? Such disassembly would defeat the goals of on-condition maintenance diagnostics. Three additional techniques, magnetic particle, magnetic perturbation, and Barkhausen noise analysis were eliminated from consideration by this screen. All three applied only to turbopump bearing inspection.

Technique Grading

Inasmuch as many specific descriptors were not available to evaluate the technologies, lumped descriptors were defined, each incorporating several specific parameters. With this approach, errors in the judgments of unknown information tend to balance with both other missing and known descriptors for a fair grade of each lumped category. This, in turn, reduces the possibility of biasing the overall ranking with a distorted grade, appearing as a seemingly well-substantiated score, for any one descriptor.

The lumped descriptors were grouped as being either technical, economic, or developmental in nature. The technical and economic groups were subdivided so as to differentiate between the required inputs and resulting effects of using the technology. The lumped descriptors are:

Technical Requirements

- Application - Those requirements concerning the use of the inspection equipment, including accessibility, human interfacing, and engine configurational needs. Weighted score to reflect the importance of these factors.
- Auxiliary - Auxiliary requirements involved in performing the inspection, including electrical, mechanical, and computational needs.
- Physical - Physical characteristics of the inspection equipment, such as size, weight, complexity, material and chemical.

Technical Features

- Detectability - How well the technique can identify the failure. This includes many factors such as accuracy, repeatability, sensitivity, resolution, drift, susceptability and level of failure progression. This is a key descriptor and is weighted accordingly.
- Durability - How rugged or reusable the inspection equipment is, as well as how much maintenance such as recalibration is needed to maintain the required level of detection.
- Speed - The time needed to perform the inspection, including equipment setup, use, removal, and data processing. Since a reduction in turnaround time is a major goal with associated savings in operational costs, this descriptor was given a weighted score.

Hazard	-	Danger of initiating a failure with use of technique, including the sensitivity to an improperly performed inspection.

Economic Expenditures

R&D Costs	-	The estimated cost to upgrade or develop the technique for use on rocket engines.
Integration Costs	-	Approximate cost of incorporating the use of the technique into an engine design after upgrading or development. Does not include equipment costs.

Economic Savings

Operational Savings	-	Approximate savings through reduced turnaround and labor or improved predictability with the use of the technique. Savings were assumed at $100.00 per hour for 1,000 inspections, i.e., a 1-hour improvement would result in $100,000.00 in savings for 1,000 inspections.

Development

Development	-	The number of years necessary to advance the technology to routine rocket engine use.

The technologies were graded as indicated in Table 17 and Appendix I for each failure mode. In order to achieve technical scores which could be fairly compared, the technologies were assumed to be equally developed for use on rocket engines. The scores given for technical lumped descriptors were summed to give an overall technical score. Economic costs were subtracted from savings, resulting in an overall savings figure. An economic grade was then assigned based on one point for each nearest $100,000 in savings. The development grade was determined by subtracting one point, from a maximum of ten points, for each year required for development. A total overall grade was obtained by summing the technical, economic, and development grades.

TECHNIQUE SELECTION

The results of the grading can be interpreted in many ways. In all cases, however, it should be remembered that the grades are inherently subjective, with disagreements over scores inevitable. For this reason, small differences between grades should be considered neither a decisive distinction nor an indication of nearly equal status. For the purpose of selecting technologies for further development, however, some ranking basis was needed, so the total overall grades, as calculated, were employed.

TABLE 17. BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING (SAMPLE)

FAILURE MODE 2	TECHNICAL								ECONOMICAL					DEVELOPMENT		
COOLANT PASSAGE LEAKAGE	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$∞K	$∞K	10	0	10	120
SCANNING ACOUSTIC FLOW	14	5	6	12	7	16	9	69	200	10	600	390	4	2	8	81
ACOUSTIC HOLOGRAPHY	15	3	5	15	5	18	9	70	200	10	500	290	3	4	6	79
X-RAY RADIOGRAPHY	7	1	2	8	5	8	7	38	100	10	200	90	1	2	8	55
GAMMA RADIOGRAPHY	6	2	4	15	7	8	3	45	100	20	400	290	3	3	7	55
PENTOXIDE POLAROGRAPHY	7	7	6	12	4	8	1	45	200	50	200	(50)	0	4	6	51
HYDROGEN POLAROGRAPHY	15	5	5	12	5	12	8	62	150	10	400	240	2	4	6	70
HYGROMETRY	7	7	6	10	5	7	1	43	50	20	200	130	1	1	9	53
SCANNING OPTICAL PYROMETRY	18	5	8	10	8	12	9	70	100	10	600	490	5	2	8	83
HOLOGRAPHIC LEAK	17	3	4	8	7	17	9	65	200	10	700	490	5	3	7	77
MILLIMETER-WAVE INTERFEROMETRY	15	3	4	8	6	12	8	56	200	10	400	190	2	4	6	64

Table 18 is a summary of the grading. It is important to note that the grades given are maximum and, in general, apply to only one failure mode. This table does give a good overall technology ranking because, as can be seen by studying grading tables in Appendix I, a technique which may rank highest in only one failure mode tends to rank high in others for which it applies; similarly, a low-ranking technique tends to rank low for all applicable failure modes.

In order to provide adequate detection or prediction for each failure type, the top ranking inspection technique, based on its total overall grade, was selected for each failure mode. As Table 19 shows, seven technologies were chosen, three of which were highest for more than one failure mode. Two of the techniques, thermal conductivity leak detection and connector continuity checking, are considered rocket engine techniques; one, particle analysis, is a stage-of-the-art technology; and four, holographic leak detection, exo-electron emission, scanning optical pyrometry, and isotope tracer detection, are considered novel.

When developed, these seven technologies could be used to detect or predict all of the failures identified by Task I. However, in some cases in-flight detection may provide better or more timely detection and existing rocket engine inspections, although not highest ranking, may be adequate. These factors were evaluated, with resulting recommendations, in Task IV.

CONCLUDING REMARKS

The Task III effort paralleled the Task II approach. A literature survey, over a database of over six million citations, was performed and yielded 56 tabulated documents. A review of these documents identified 14 rocket engine techniques, 16 state-of-the-art techniques in other industries and 16 novel techniques. The results of the Task I failure characterization were examined to identify between-flight measurands. The techniques identified by the survey were then correlated to the applicable failure mode measurands to provide a matrix of inspection possibilities.

A selection procedure was used to identify the top ranking inspection technique for each failure mode. First, six technologies were eliminated as being clearly inadequate. The remaining techniques were then graded using lumped descriptors which included factors such as accessibility, engine configuration requirements, detectibility, speed, costs, savings and safety. The techniques were graded assuming an equal state of maturity for each failure mode application. Based on the grades for each lumped descriptor, the technologies were assigned total overall grades which permitted a ranking of the applicable inspection technology for each failure mode. Seven technologies were identified as being top-ranking in one or more of the 16 failure modes. They are:

1. Holographic Leak Detection - Leak detection of multiple joints

2. Thermal Conductivity Leak Detection - Leak detection in localized areas.

3, Scanning Optical Pyrometry - Detection of coolant passage restrictions or leaks.

TABLE 18. GRADING OF BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY

LEGEND

N = NOVEL TECHNOLOGY

S = STATE OF THE ART TECHNOLOGY

R = ROCKET TECHNOLOGY

BETWEEN-FLIGHT INSPECTION TECHNIQUES	MAXIMUM TOTAL OVERALL GRADE	MAXIMUM TECHNICAL GRADE	MAXIMUM ECONOMIC GRADE	MAXIMUM DEVELOPMENT GRADE	FAILURE MODES	2) COOLANT PASSAGE LEAKAGE	3) JOINT LEAKAGE	5) HIGH TURBOPUMP TORQUE	6) CRACKED TURBINE BLADES	7) CRACKED CONVOLUTION,BELLOWS,SHIELD	8) LOOSE ELECTRICAL CONNECTOR	9) BALL BEARING DAMAGE	10) TUBE FRACTURE	11) TURBOPUMP SEAL LEAKAGE	13) VALVE FAILURE	14) INTERNAL VALVE LEAKAGE	15) REGULATOR DISCREPANCIES	16) CONTAMINATED HYDRAULIC CONTROL
HOLOGRAPHIC LEAK DETECTION	86	69	10	7		N	N											
THERMAL CONDUCTIVITY LEAK CHECKING	85	67	8	10			R									R	R	
SCANNING OPTICAL PYROMETRY	83	70	5	8		N												
SCANNING ACOUSTIC FLOW DETECTION	81	69	4	8		N												
EXO-ELECTRON EMISSION	79	68	4	7					N	N		N	N					
ACOUSTIC HOLOGRAPHY	79	70	3	6		N									N			
CONNECTOR CONTINUITY CHECKING	77	66	1	10							R							
EDDY CURRENT	77	64	4	9					S	S		S	S					
ULTRASONIC FLAW DETECTION	76	65	3	8					S	S		S	R					
ISOTOPE TRACER DETECTION	75	64	4	7				N	N			N		N	N	N		
HOLOGRAPHIC DEFLECTION PREDICTION	75	66	3	6			N						N					
PARTICLE ANALYSIS	74	61	4	9				S				S		S	S	S	S	S
HOLOGRAPHIC SURFACE MAPPING	74	57	10	7					S	S					S	S		
ULTRASONIC LEAK DETECTION	73	55	8	10			S								S	S	S	S
OPTICAL LEAK DETECTION	72	57	7	8			S								S	S	S	S
ELECTRIC CURRENT INJECTION	72	61	4	7					N	N			N					
HYDROGEN POLAROGRAPHY	70	62	2	6		N												
REMNANT MAGNETIZATION	69	50	10	9					N	N								
TORQUING	68	58	0	10			R	R			R	R		R		R		
OPTICAL PROXIMITY DETECTION	67	57	2	8				S						S		S		
HALOGEN LEAK DETECTION	66	50	7	9			S									S	S	
MILLIMETER-WAVE INTERFEROMETRY	64	56	2	6		N												
FLOW LEAK DETECTION	64	54	0	10			R							R		R	R	
MASS SPECTROMETRY	62	51	1	10			R									R	R	
BORESCOPING	61	51	0	10				R	R	R		R			R	R		
DIFFERENTIAL RADIOMETRY	60	52	0	8			S								S	S	S	S
PRESSURE DECAY	60	49	1	10			S									S	S	S
ACOUSTIC EMISSION	59	48	4	7									S					
PENETRANT DETECTION	59	49	0	10						R			R					
X-RAY RADIOGRAPHY	58	48	1	9		R							R					
POSITRON ANNIHILATION	57	51	0	6					N	N		N	N					
HYGROMETRY	56	47	0	9		R									R			
ISOTOPE THERMOMETRY	55	46	1	8				N	N	N	N	N		N				
GAMMA-RAY RADIOGRAPHY	55	45	3	7		R												
PENTOXIDE POLAROGRAPHY	55	49	0	6		N									N			
LASER SURFACE SCATTERING	54	47	0	7											N	N		
LEAK TAPE/COATING	51	38	5	8			N											
ULTRASONIC EXTENSIOMETRY	48	38	0	10			R											
LEAK SOLUTION	42	32	0	10			R											
RESISTIVITY MONITORING	38	29	1	8			S											
PERFECT SCORE	120	100	10	10														

TABLE 19. TECHNOLOGY RANKING

TECHNOLOGY	FAILURE MODE: 2-COOLANT PASSAGE LEAKAGE	3-JOINT LEAKAGE	5-HIGH TURBOPUMP TORQUE	6-CRACKED TURBINE BLADES	7-CRACKED CONVOLUTION, BELLOWS, SHIELD	8-LOOSE ELECTRICAL CONNECTOR	9-BALL BEARING DAMAGE	10-TUBE FRACTURE	11-TURBOPUMP SEAL LEAKAGE	13-VALVE FAILURE	14-INTERNAL VALVE LEAKAGE	15-REGULATOR DISCREPANCIES	16-CONTAMINATED HYDRAULIC CONTROL
HOLOGRAPHIC LEAK DETECTION	77	86											
THERMAL CONDUCTIVITY LEAK DETECTION		85									71	70	
SCANNING OPTICAL PYROMETRY	83												
EXO-ELECTRON EMISSION				75	70		45	79					
CONNECTOR CONTINUITY CHECKING						77							
ISOTOPE TRACER DETECTION			66	48			75		68	64	62		
PARTICLE ANALYSIS			64				74		64	62	61	67	68

XX = APPLICABLE TECHNOLOGY GRADES

◯ = TOP-RANKING TECHNOLOGY FOR EACH FAILURE MODE

4. Exo-Electron Emission - Fatigue monitoring and characterization

5. Connector Continuity Checking - Verification of electrical connector function.

6. Isotope Tracer Detection - Monitoring wear.

7. Particle Analysis - Monitoring wear and contamination.

FUTURE TECHNOLOGY DEVELOPMENT

The objectives of this task were to review the results from In-Flight Condition Monitoring (Task II) and Between-Flight Inspection (Task III) and to establish an objective prioritized list of technology development requirements.

TECHNOLOGIES SELECTED FROM TASK II AND TASK III

The procedures followed in Task II and Task III were similar; a survey of technologies was made and matched with the failure modes identified in Task I, which were then ranked and these technologies became the inputs to this task and are:

Task II: In-Flight Condition Monitoring

Direct In-Flight Sensors

1. Optical pyrometer
2. Fiberoptic deflectometer
3. Isotope wear detector
4. Tunable diode-laser spectrometer

Indirect In-Flight Sensors

1. Ultrasonic flowmeter
2. Ultrasonic thermometer
3. Digital quartz pressure sensor
4. Optical tachometer

Task III: Between-Flight Inspection

1. Holographic leak detection
2. Thermal conductivity leak checking
3. Scanning optical pyrometry
4. Exo-electron emission
5. Connector continuity checking
6. Isotope tracer detection
7. Particle analysis

APPLICATION OF SELECTED TECHNOLOGIES TO ROCKET ENGINES

In-Flight Condition Monitoring

Considerable reductions in cost and time in maintaining a long service life for reusable rocket engines can be realized by the use of in-flight condition monitoring systems. The search and screening of such systems yielded eight state of the art and novel technologies which can be used to detect symptons related to failure modes previously experienced in rocket engines. They include optical pyrometers, fiberoptic bearing deflection detectors, isotope wear detectors, tunable diode-laser spectrometers, ultrasonic flowmeters, ultrasonic thermometers, digital quartz pressure sensors, and optical tachometers.

A description of each of these technologies is presented in a discussion of the steps in development required for their integration into rocket engines.

Direct In-Flight Sensors

1. Optical Pyrometer for Remote Temperature Monitoring of Turbine Blades

Technical Description. All objects emit a spectrum of thermal radiation characteristic of their emissivity and their temperature. An optical pyrometer measures the frequencies and amplitudes of the spectrum of this thermal radiation, and thus provides a noncontacting, nonintrusive, fast means of temperature sensing. A system for temperature sensing by means of a pyrometer is shown in Fig. 18. The pyrometer consists of a semiconductor device in which electromagnetic quanta are converted into electrical current. The current magnitude is measured by an electronic system. In measuring the temperature of an object, the thermal radiation is optically filtered by a window into one or two narrow frequency bands to give a unique thermal fingerprint of the object. For observing the temperature of parts internal to a system, the filtered waves are transmitted to the pyrometer by means of optical fibers, allowing isolation of the detector and its electronics from a hot, hostile environment.

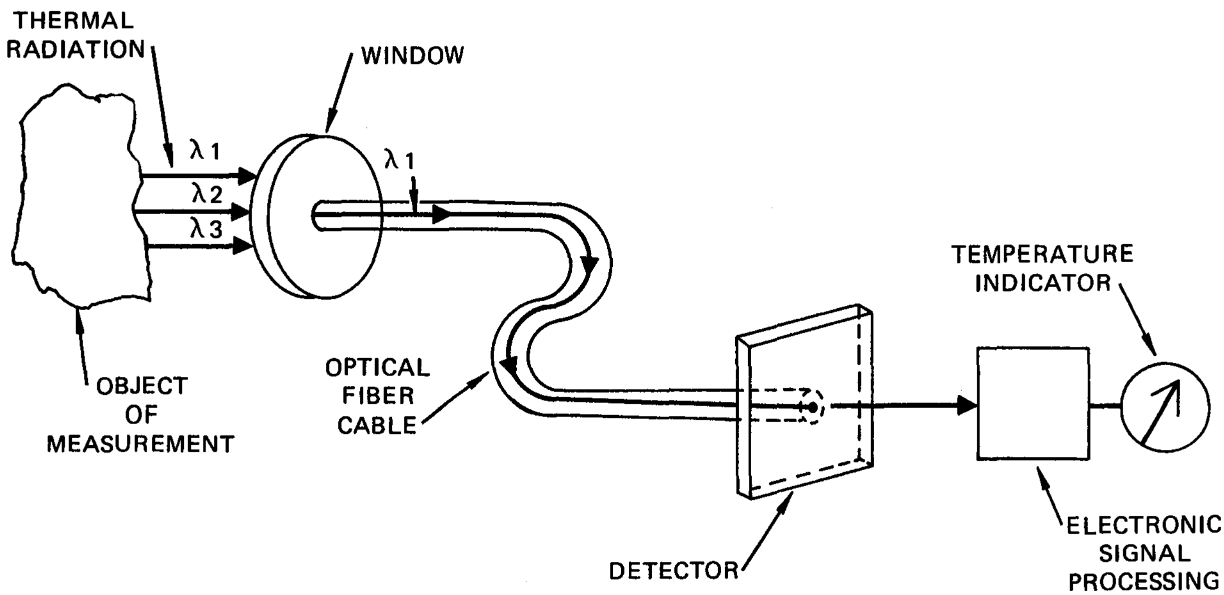


Figure 18. Block Diagram of Pyrometer for Remote Temperature Measurement

Further advantages of this pyrometer system for turbine blade measurement include the nonintrusiveness of the detector, preventing any hazard or aeration as well as its immunity to creep and fatigue, by virtue of its crystalline structure. Furthermore, a response time of 0.6 microseconds or less is possible, enabling transients to be accurately monitored. The versatility of today's electronics makes possible complete thermal profiling or imaging of a device, as well as strobed observation.

At the temperature range typical of rocket engine turbine blades (538 to 807 C), accuracies as good as 0.5% can be achieved by these devices. Pyrometers have, in fact, been used for remote in-flight monitoring of turbine blade temperature in airplanes.

Development Requirements. To be incorporated into a rocket engine, the following steps in development are required. The measurement requirements for the pyrometer must be defined in terms of its resolution, accuracy and range of frequency sensitivity desired, and a choice of pyrometer made. The proper fiberoptic and window materials for the high-temperature environment of the turbine must be chosen. Finally, the electronics must be laid out, the system integrated and a prototype fabricated and tested in the field.

2. Fiberoptic Bearing Deflection Detector

Technical Description. A fiberoptic deflection detector monitors the condition of a bearing by measuring the localized, cyclic deformations on the outer bearing race caused by the passage of the balls, by a detector mounted near the race. The key element of this detector, as shown in Fig. 19, is a bundle of optical fibers with its end in close proximity to the outside radial surface of the outer race. Light is both shined onto the race and reflected from the race to a detector through this fiber bundle. The magnitude of the reflected photoelectric signal is a function of the variation of distance between the race and the sensing element. The output of the probe for a normal bearing is very similar to a clean half sine wave, corresponding to the passage of each ball by the probe. When the surface of one of the balls becomes pitted or flawed, the photoelectric output of the detector is correspondingly distorted each time that ball passes over the fiberoptic probe, thus generating a distorted half sine wave once per revolution. On the other hand, a crack or pit on the inner surfaces of the bearing races is noted as a distortion in each of the output peaks, corresponding to the passage of each ball over the flaw in the race.

Fiberoptic bearing deflection detectors are well suited for bearing performance monitoring. The detector provides a noncontacting means of measuring bearing deflections and loads directly. High levels of sensitivity, such as 50 mV/micron, are commonly achieved. By virtue of the fiberoptic coupling, the device has a profile which facilitates installation and is immune to electromagnetic noise.

Development Requirements. Methods of mounting the detector should be evaluated which minimize the effect of structural vibrations on the output. The detector electronic circuitry should be laid out, fabricated and tested. A fiberoptic material should be chosen which best functions in a cryogenic environment.

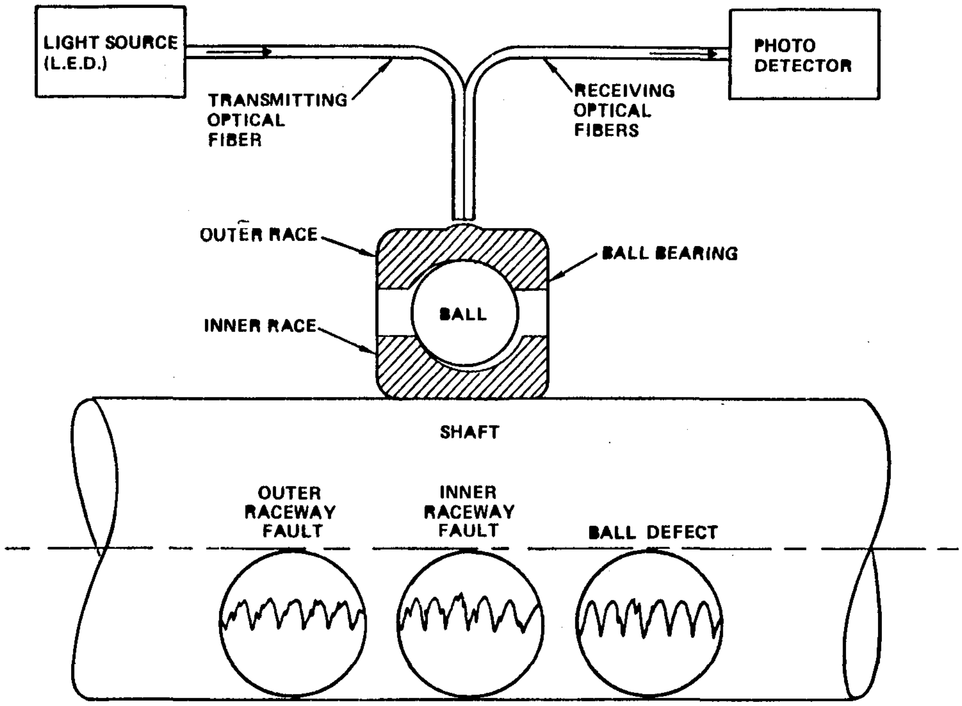


Figure 19. Fiberoptic Deflectometer

3. Isotope Wear Detectors

Technical Description. An isotope-wear detector measures the low-energy OSHA-approved gamma rays of wear particles of ball bearing or rotary seals. It consists of a gamma-ray detector and a particle catcher downstream of the isotope-tagged bearings or seals. The γ-photons emitted by the captured particles strike the detector and are converted into an electric current. Electronic circuitry then measures the magnitude of this current, determining the rate of γ-photon emission and thus the quantity of wear particles.

Tagging is effectively accomplished by immersing the part in a flux of neutrons (e.g., by placing it in a storage hold of a nuclear reactor), converting a small, uniformly distributed fraction of its atoms into radioisotopes. The part is then incubated for several months, leaving it tagged with only long-lived radioisotopes suitable for long-term monitoring of wear.

Materials chosen for bearings because of desirable mechanical properties are often ferromagnetic (such as the steel bearings in the SSME turbopump), and debris from parts made of such materials could be captured by a magnetic trap located on a duct at a bend in flow downstream of the part (Fig. 20). Centrifugal forces would move the debris radially outward at the bend into a bed of magnetic pins which capture these particles for measurement by a γ-ray detector. An alternative method, suitable for turbopump rotary seals, routes the purging or cooling fluid passing through the seal through a duct to a lower-pressure region, where a filter with a low pressure drop captures a representative amount of the wear particles.

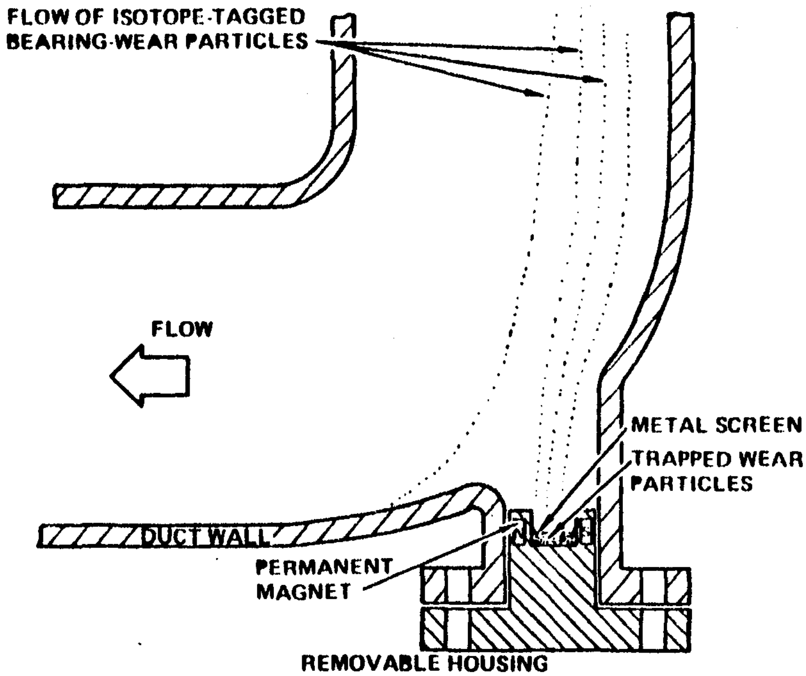


Figure 20. Isotope Detector for Steel Bearing Wear Particles

Isotope wear monitors are thus a quantitative, noncontacting means of monitoring the wear of rocket engine turbopump bearings and rotary seals. Semiconductor detectors are small (less than 1 cm^3) and lightweight (on the order of a few grams). Several semiconductor detectors are available which are 100% efficient in converting low energy γ-photons into electrical signals, so that parts can be tagged with small, safe concentrations of low γ-energy radioisotopes which are certified suitable for public handling by OSHA.

Development Requirements. In order to incorporate such a detector into a rocket engine turbopump, methods of mounting the semiconductor device and capturing the wear particles should be evaluated and tested which minimize the separation of wear particles from the detector, and thus attenuation of the γ-photons. The methods of capture should be further evaluated for their efficiency in trapping wear particles. The signal to noise ratio should be optimized by a proper choice of commercially available semiconductor detectors and proper design of the electronic circuitry.

4. Tunable Diode-Laser Spectrometer

Technical Description. A tunable diode-laser spectrometer is a highly compact and rugged means of measuring the infrared spectra of the constituents of rocket engine combustion gases. A p-n junction diode laser generates an infrared beam whose wavelength is varied by altering the diode-laser bias current. When transmitted through combustion gases, this beam is selectively absorbed at certain wavelengths which are characteristic of the constituent gas species (Fig. 21).

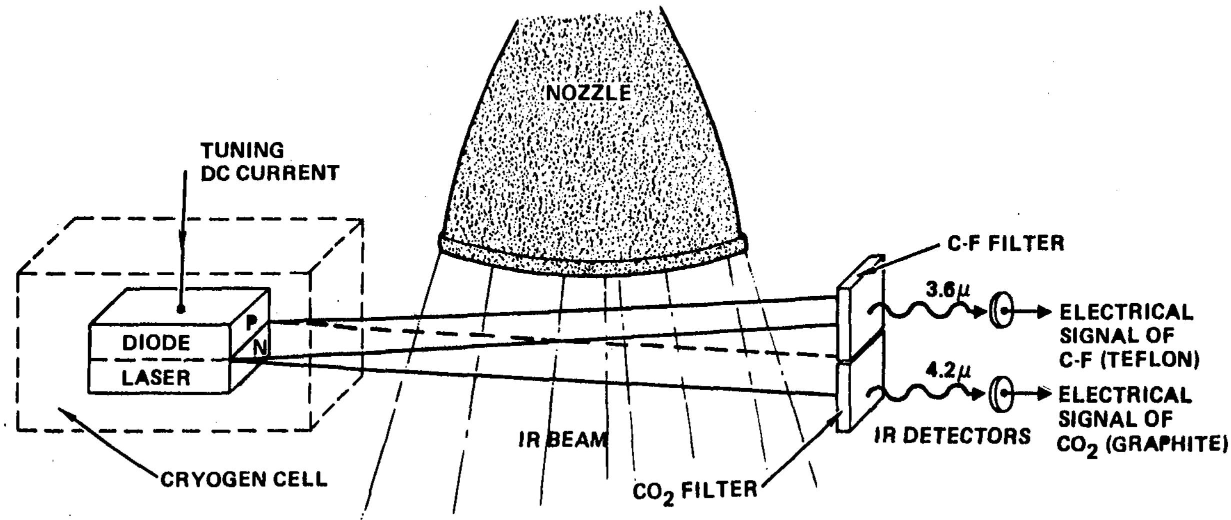


Figure 21. Tunable Diode Laser Spectrometer

To measure the concentration of a gas species, an optical filter (instead of a bulky monochromator grating), consistent with an IR absorption line of the gas, is placed in front of a wide-band photoelectric detector. The electrical current of the detector is related to the intensity of the absorption line, and thus to the concentration of the gas species. Thus, for example, by selecting the absorption lines of H_2 and O_2, the combustion mixture ratio of H_2-fueled engines can be monitored and controlled.

Alternatively, the spectrometer can be used for condition monitoring. For example, by selecting an absorption line of the C-F bond found in combusted shavings of Teflon valve seats and measuring the photocurrent in that band, the concentration of the Teflon shavings or wear particles can be measured. Similarly, an absorption band of CO_2 can be selectively monitored, allowing measurement of the concentration of the debris particles from worn graphite rotary seals.

One of the principle advantages of the tunable diode spectrometer is its size: the diode-laser itself can fit on top of a dime. Furthermore, while power of only about a milliwatt is required, the intensity of the laser output signal is quite high since it is monochromatic, allowing a high inherent signal-to-noise ratio. The diode-laser also has all the advantages characteristic of semiconductor devices, inlcuding ruggedness and immunity to creep and fatigue.

Development Requirements. To incorporate this detector into a rocket engine, a suitable housing must be designed and fabricated which isolates the detector both mechanically and thermally. Appropriate diode-laser materials and filters must be chosen which correspond to the combustion constituents to be monitored. The optical arrangement and electronics of the detector must be laid out, fabricated and field tested. Furthermore, a lightweight shielding for the detector should be designed which minimizes noise due to background cosmic rays. The effect, if any, of radioisotope labeling on the physical properties of the parts should also be determined.

Indirect In-Flight Sensors

1. Ultrasonic Flowmeter for Propellant Flow Measurement

Technical Description. The velocity of a sound wave is altered when it passes through a moving fluid, much like a boat crossing a flowing river. Figure 22 shows a transit-time ultrasonic flowmeter which takes advantage of this principle. A pair of transducers is mounted on the pipe, directed at one another and oriented diagonally to the flow. Ultrasonic pulses are alternately transmitted and detected by each of the transducers. Because the upstream signal velocity is decreased and the downstream velocity increased, there is a difference in their velocities and thus of their transit times. This difference in transit times is analyzed by electronic circuitry to measure the flow velocity. This measurement is rendered independent of properties influencing sound in the fluid such as temperature, pressure, density and viscosity, by the use of a judicious combination of the measured upstream and downstream transit times.

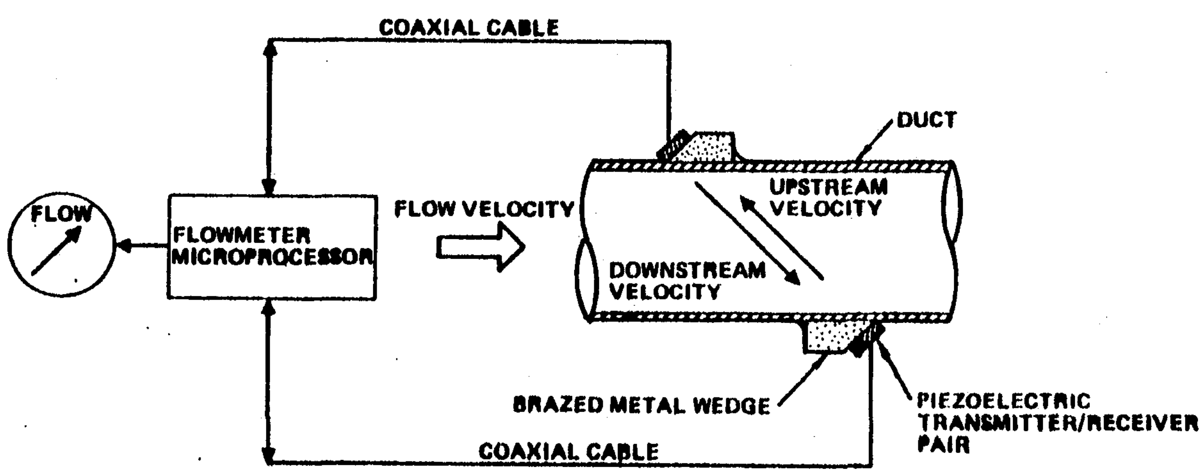


Figure 22. Block Diagram of Transit Time Ultrasonic Flowmeter

The flowmeter has the advantage of being completely nonintrusive and thus could never become an impedance to flow or a hazard in case of structural failure. Accuracies of 0.4% and repeatabilities of 0.2% have been achieved commercially.

Development Requirements. Ultrasonic flowmeters should be developed for suitability in a cryogenic environment. Methods of attachment of transducers and piezoelectric elements must be optimized. Furthermore, since structural vibrations and flow noise can both produce piezoelectric signals, the frequency and acoustic polarization must be optimized to minimize these effects.

2. Ultrasonic Thermometer

Technical Description. An ultrasonic thermometer measures the transit time of an ultrasonic pulse transmitted across a hot gas chamber. From this transit time, the acoustic speed and, in turn, the gas temperature are determined. Figure 23 diagrams the basic concept of this sensor. A pair of piezoelectric transducers alternately transmit (by means of an electronic driving signal) and receive ultrasonic pulses sent across the gas chamber. The transit time of these pulses is then determined by electronic circuitry, yielding the mean acoustic velocity across the chamber. From the acoustic velocity, the temperature of the gas is determined. By a judicious combination of the measured transit times of pulses in both directions across the chamber, the acoustic velocity determination and thus the temperature measurement will be independent of the gas flow speed.

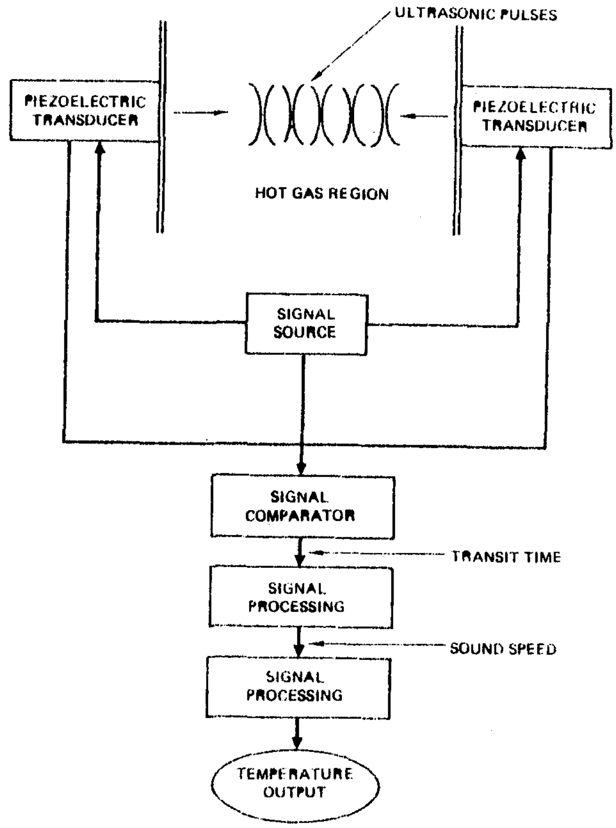


Figure 23. Block Diagram of Ultrasonic Thermometer

The advantages of this sensor include high temperature capability (greater than 1650 C with cooled transducers) and fast response. Unlike other temperature sensing probes, the ultrasonic thermometer measures the path-averaged temperature across the gas chamber. It is, furthermore, noncontacting and nonintrusive.

Development Outline. Steps for development of such a system for use in a rocket engine include design of electronics to optimize small-signal recognition and signal-to-noise ratio, and design of a system of mounting and cooling the piezoelectric transducer pair. The system would then be mounted on an engine and tested in the field.

3. Digital Quartz Pressure Sensor

Technical Description. With a digital quartz pressure sensor, one takes advantage of the pressure dependence of the resonant frequency of a quartz crystal. In the sensor, an electronic oscillator circuit connected to a quartz beam forces the beam into resonant ultrasonic oscillations. Compression of the beam alters its resonant frequency, which is then detected by the electronic circuitry, generating a time-domain digital signal. Figure 24 shows a design of a digital quartz pressure sensor for measuring absolute pressure. The input pressure is transformed through a bellows and a cantilever into a force imposed on the quartz crystal.

The advantages of digital quartz pressure sensors are manifold. Quartz crystal has long-term stability (no creep) ease of vibrational excitation (resulting in lower power consumption) and low-temperature sensitivity (sensitivities of 0.005%/C are experienced in quartz pressure sensors). It is for these very reasons that quartz oscillators are the most commonly used frequency standards for clocks, etc. Additional advantages include high repeatability (within 0.005%) and a dynamic range of up to 200,000:1. Miniaturized digital quartz sensors are available (see Fig. 25) which can measure as high as 6.9 megapascal and weigh only 57 grams.

Development Outline. The candidate miniaturized digital quartz pressure sensor must be tested for compatability to engine environments. Further tests should be made to determine its performance trends and durability. The device then can be adapted for incorporation in an engine.

4. Optical Tachometer

Technical Description. An optical tachometer measures the rpm of a rotating shaft by measuring the time between reflections generated by an encoder which is attached to and rotates with the shaft. The configuration of an optical tachometer is shown in Fig. 26.

An LED acts as a light source, transmitting through optical fibers onto the encoder. The encoder, in turn, reflects the light back through a set of optical fibers coaxial to the input fibers, onto a semiconductor element which converts these light pulses into an electrical signal. To this end, the encoder is appropriately bevelled and contoured to provide maximum contrast between the reflective and nonreflective areas. From the known geometry of the encoder and the measured time between the pulses, electronic circuitry connected to the sensor determines the rpm of the shaft. The optical fibers are terminated in a fused quartz rod whose end is fashioned into a lens which focuses both the incident and reflected light, thus allows the sensor to be placed a considerable distance from the encoder; i.e., outside the liquid flow and flush with the housing wall.

Contrast ratios well above 10 and input signal to noise ratios of about 38 have been calculated for optical tachometers of the above description. Being flush with the shaft housing, the detector is completely nonintrusive to cryogen in flow.

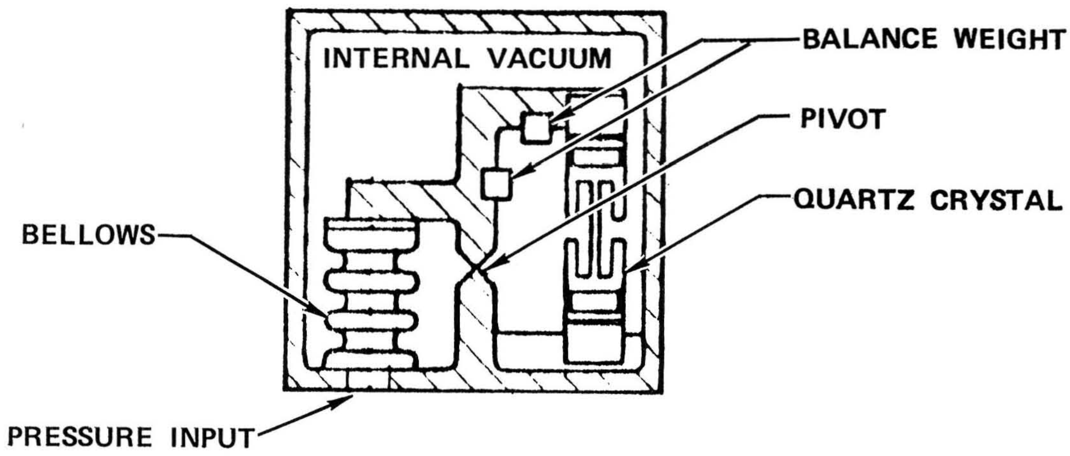


Figure 24. Digital Quartz Pressure Sensor

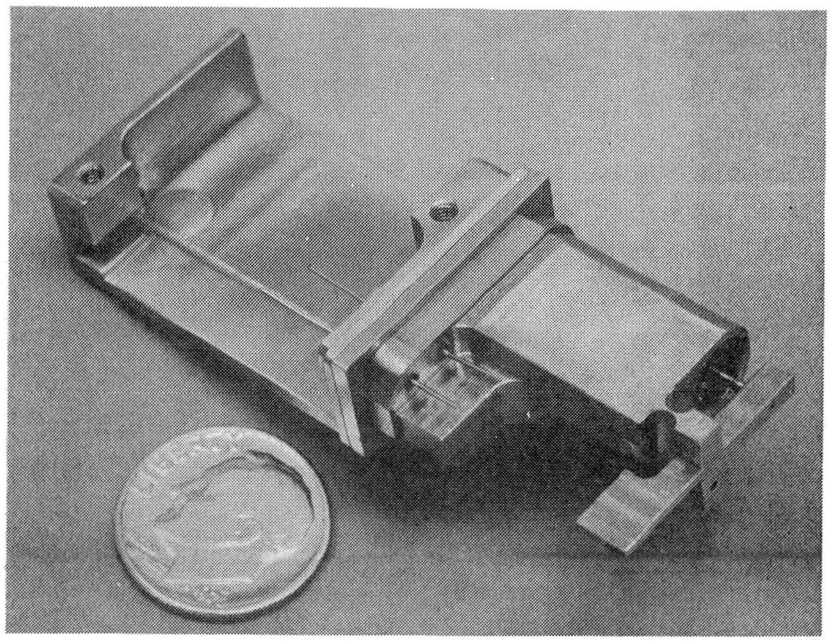


Figure 25. Housing for Miniaturized Digital Quartz Pressure Sensor

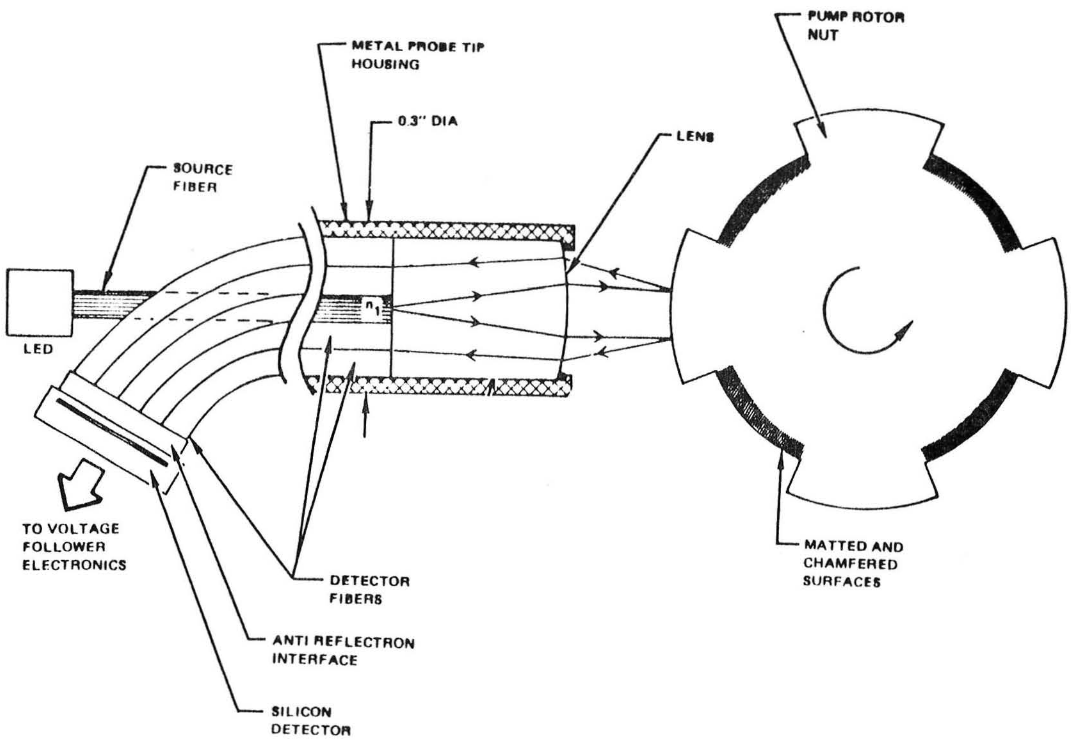


Figure 26. Block Diagram of Optical Tachometer

Development Outline. Before incorporation in a rocket engine turbopump, the lens, reflecting surfaces and fiber optics should be tested to determine their properties in the cryogenic and vibrational environment of the pump. Designs for sealing the optical fibers to the fused quartz lens and for sealing the lens to the metal sensor port need to be evaluated. Finally, tests must be made to determine the effect of bubbles and fluid turbulence which determine the ultimate performance of this optical sensor.

Between Flight Inspection

Seven technologies were chosen from the inspection techniques for the detection of prediction of the failures identified by Task I; two are considered rocket engine state-of-the-art, one state-of-the-art, and four are novel. The technologies are holographic leak detection, thermal conductivity leak detection, scanning optical pyrometry, exo-electron emission, connector continuity checking, isotope tracer detection, and particle analysis.

A brief descritpion of each of these technologies is made here.

1. Holographic Leak Detection (Figure 27)

Holographic leak detection is a technique capable of simultaneously locating and quantifying multiple leaks. This is achieved by illuminating the test area with a triple-pulse coherent light source. The first pulse reflects off the test object and is combined with a reference beam, obtained with optical beam splitters, thus forming a hologram of the object. The second optical pulse, shortly following the first one, forms another hologram in the same manner. These holograms, recorded on the same photographic film, interact to cancel each other if the object has not been altered at all. However, if the object has undergone even minute changes between the pulses, an interference pattern is formed. If the object is purged with a gas that is optically different from the ambient gas, any leak will produce such an interference pattern. The volume of the interference pattern is proportional to the volume of the leaking gas. The third pulse of the coherent source enhances the interference fringes, the spacing of which are proportional to the speed of the leakage. Since the product of volume and velocity is proportional to flowrate, the leakage rate can be determined simultaneously for all leaks in the line of sight of the detector.

Holographic leak detection can provide significant reductions in the time required for leak detection with the advantage of being a noncontacting technique capable of locating multiple leaks simultaneously. Both qualitative and quantitative data reproduced in a manner which can be readily interfaced with computer processing. As an optical technique, it has the virtue of providing simplified scanning ability and permanent image recording.

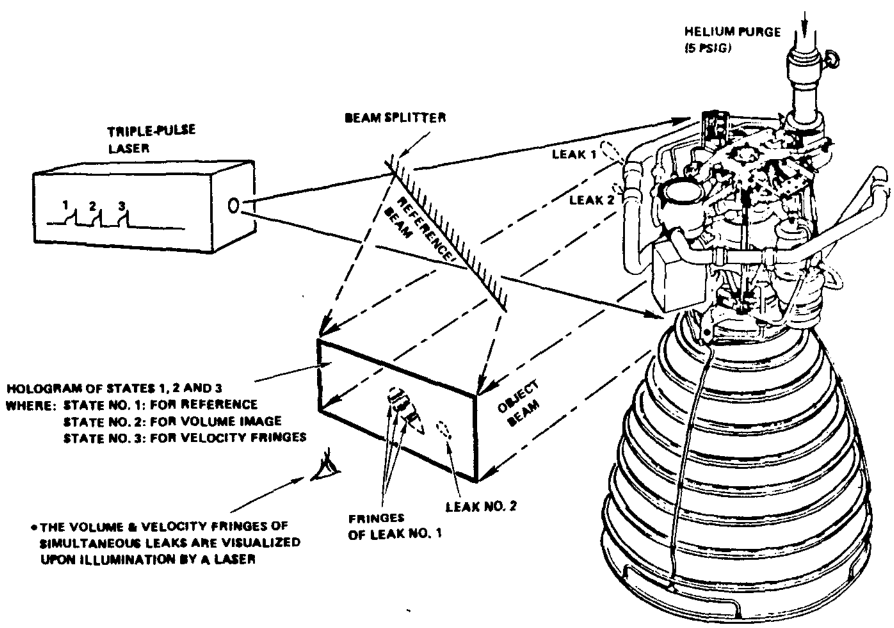
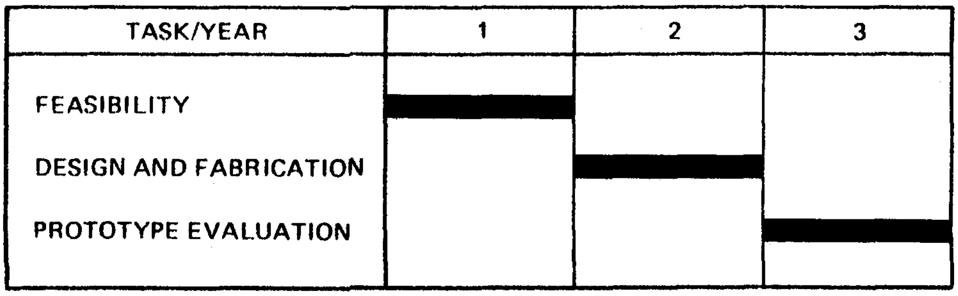


Figure 27. Holographic Leak Detection Schematic

Development Outline. The development required of holographic leak detection before its routine use on rocket engines includes establishing laser requirements and recording needs so that equipment may be selected. Real time imaging of the leakage hologram should also be investigated. Procedures to inspect the maximum number of ducts with a minimum of different viewing angles should be defined.



2. Thermal Conductivity Leak Detection (Figure 28)

Thermal conductivity leak detection uses a hot wire resistance bridge with one element of the bridge exposed to air (reference side) and the other element exposed to the leak (tracer gas). Conductivity of the air changes with concentration of the tracer gas which causes a change in resistance or temperature of the element. Leakages are detected with a probe. Helium gas may be used for the tracer. The instrument is sensitive to approximately 1×10^{-4} scc/sec, portable, relatively insensitive to background interference and has a good response (1 to 5 seconds). After saturation, the instrument returns to zero immediately when the tracer gas source is removed. This type of device, the Uson 500, has been successfully used on the J-2 engine.

Thermal conductivity leak detection provides a fast and reliable location of leakage at low cost. The device is easily used and interpreted by maintenance personnel. The technique is versatile and may be used to pinpoint leaks indicated by other means.

Development Outline. Thermal leak detection has previously seen limited application on the J-2 engine. Although these devices are commercially available, an investigation to compare tracer gases and explore means of reducing sensitivity to background effects would improve the usability of this technique on rocket engines.

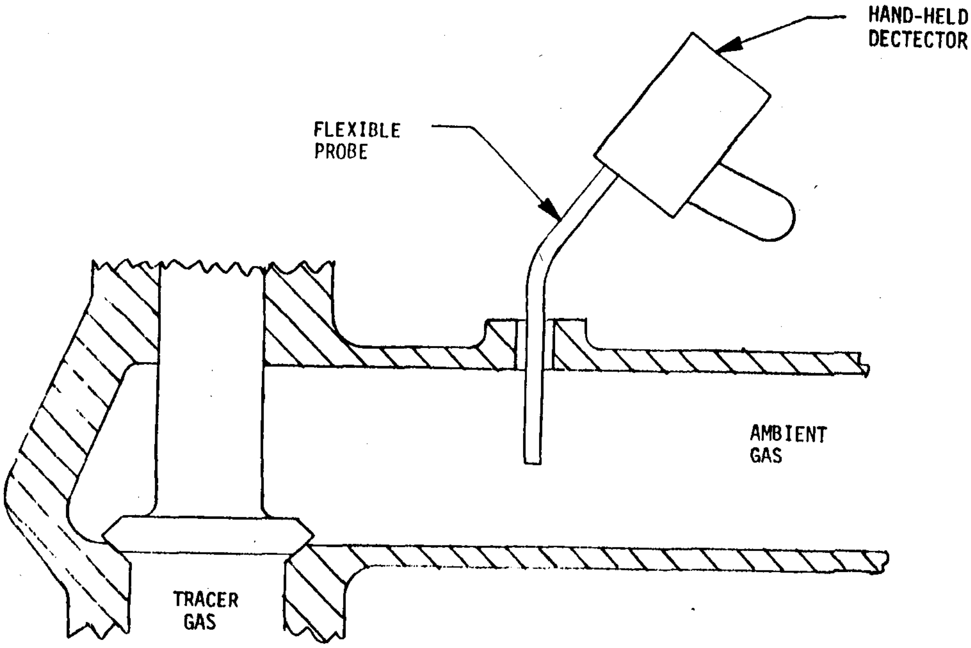


Figure 28. Thermal Conductivity Detection of Internal Leakage

TASK/YEAR	1	2	3
FEASIBILITY	■		
DESIGN AND FABRICATION	■■■		
PROTOTYPE EVALUATION	■■■■		

3. Scanning Optical Pyrometry (Figure 29)

Scanning optical pyrometry locates restrictions in chamber coolant passages by automatically sweeping the inner chamber walls and noting temperature profile anomalies. The optical pyrometer measures the frequencies and amplitudes of the spectrum of the thermal radiation, characteristic of the emissivity and temperature, emitted by the chamber wall. With blockage of a hot purge gas, localized cooling occurs downstream which can be detected by the pyrometer. The pyrometer incorporates optical filters to isolate the thermal radiation into selected narrow frequency bands. This radiation is then converted into electrical current by a semiconductor device. The current is measured by an electronic system and converted to a temperature indication. The pyrometer automatically scans the chamber wall by a combination of mechanical and optical means in both the azimuth and chamber axis directions. Data can be presented with imaging of the temperature profile and/or by having the coordinates of all anomalies automatically calculated and listed.

Scanning optical pyrometry provides fast, positive location of coolant passage restrictions. Automatic and remote operation can also provide enhanced safety to maintenance personnel and, because of the noncontacting nature of the scanning, less engine hazard. The technique can be used during purging operations to provide little impact on turnaround; its use might even permit reduced purging requirements.

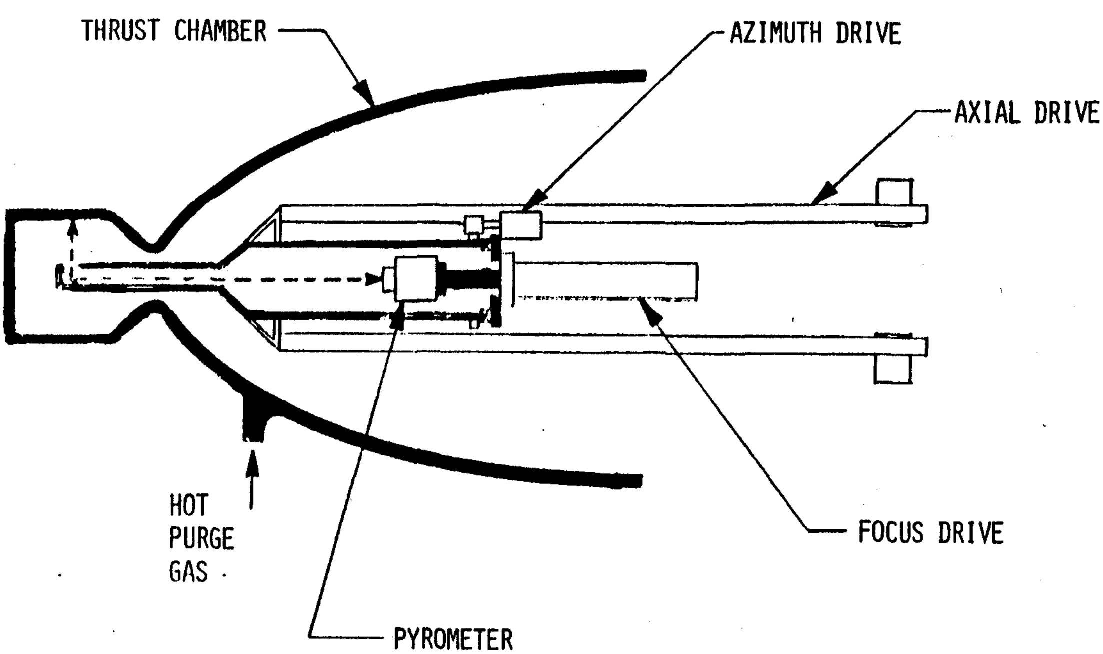
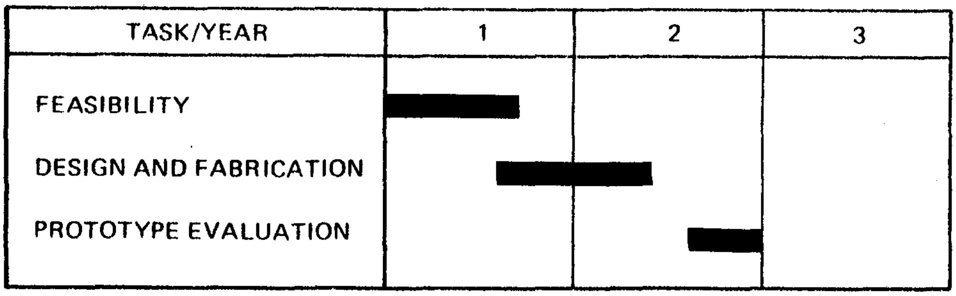


Figure 29. Scanning Pyrometer Concept for Detection of Coolant Passage Restrictions

Development Outline. In order to develop this technology for use on rocket engines, the measurement requirements for the pyrometer must be identified in terms of resolution, accuracy and range of frequency sensitivity so that a pyrometer may be selected. A suitable scanning mechanism must be worked out and its control circuitry defined. The manner of data display must also be selected and the appropriate electronics and software laid out.



4. Exo-Electron Emission (Figure 30)

Exo-electron emission is a means of locating and characterizing fatigue damage in order to provide remaining-life prediction. This technique refers to the photoelectron emission from a metal part which emanates through micro-cracks in the brittle surface oxide. By scanning the test surface with a small beam of ultraviolet radiation, this exo-electron emission may be collected and amplified by an electron multiplier and measured by a current meter. The intensity of the emission increases as fatigue-induced cracks grow with continued cycling. By this means, a clear, accurate indication of the location and progression of fatigue can be obtained.

Fatigue damage produces highly localized exo-electron emission very early, i.e., on the order of 1% of the fatigue life. Cracks, on the other hand, are not visible until about 10 to 15% of the fatigue life has expired. This very early detection, as well as the characteristic increasing emission as fatigue damage progresses, permits correspondingly earlier and more accurate prediction of remaining life with the use of this technique.

Development Outline. Before exo-electron emission may be used routinely for rocket engine inspections, equipment suitable for in-field use may be developed. Scanning methods and sensors need to be optimized for each application. The emission behavior of the materials to be inspected also must be carefully characterized. Finally, the best method of data presentation, such as plots or imaging, needs to be identified.

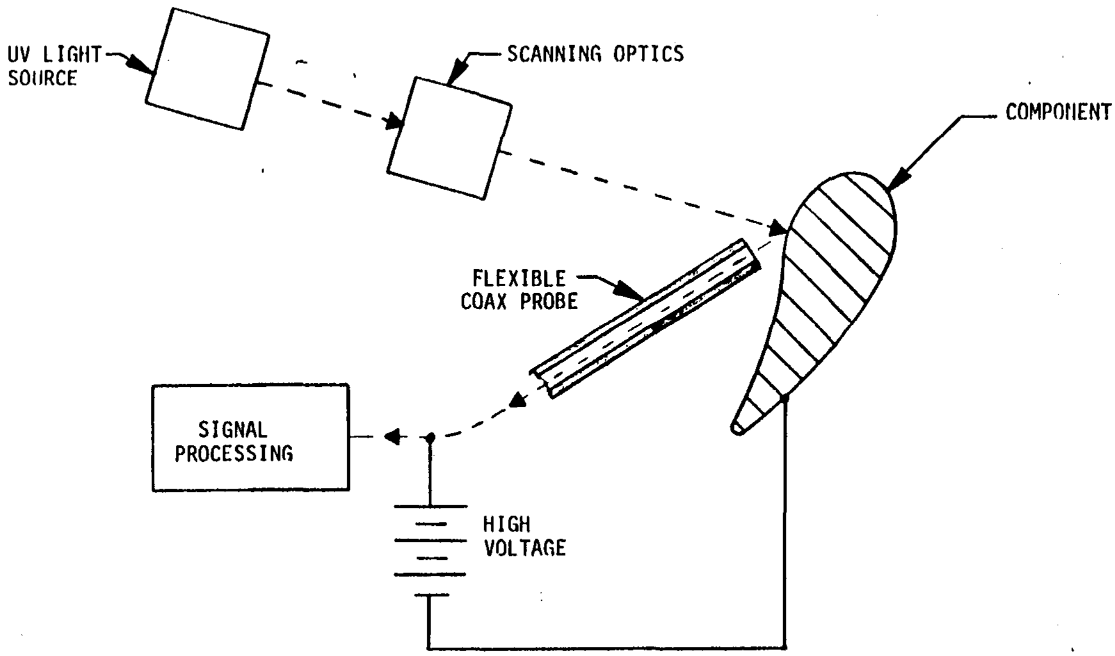


Figure 30. Schematic of Typical Exo-Electron Inspection

TASK/YEAR	1	2	3
FEASIBILITY	■■■		
DESIGN AND FABRICATION		■■■	
PROTOTYPE EVALUATION			■■■

5. Connector Continuity Checking

Connector continuity checking verifies proper functioning of connectors by simply determining the electrical continuity through them. This is done as part of the automatic checks that also confirm the functioning of valves, igniters, instrumentation, etc. The on-board controller executes a checkout program that sends signals to components (through connectors) and compares the output signal to stored limit valves. Any discrepancies are noted so that maintenance personnel may isolate and correct the faulty wiring, connector, or component. Connector continuity checking thus provides a fast, automatic determination of connector function.

Development Outline. Connector continuity checking is presently used on the Space Shuttle Main Engine. Because it performs satisfactorily, no special development needs are considered necessary.

6. Isotope Tracer Detection (Figure 31)

Isotope tracer detection measures the low-energy (OSHA approved) gamma rays of wear particles. These particles are captured by a catcher downstream of the isotope-tagged part during flight. The gamma-photons emitted by the captured particles are detected between-flight by removing the catcher from the engine and placing it in a gamma-ray detector. The magnitude of the electric current from the detector indicates the rate of gamma-photon emission and thus the quantity of wear particles.

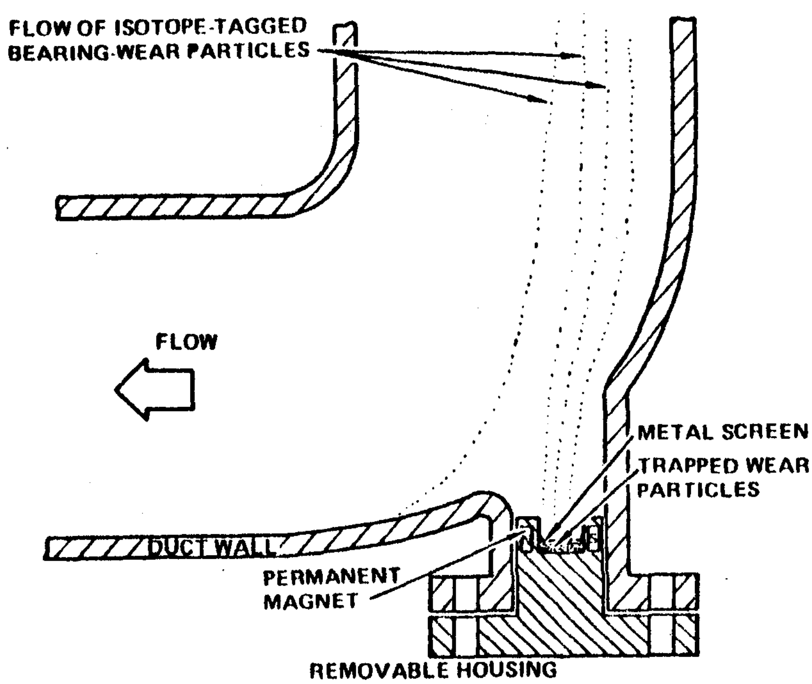


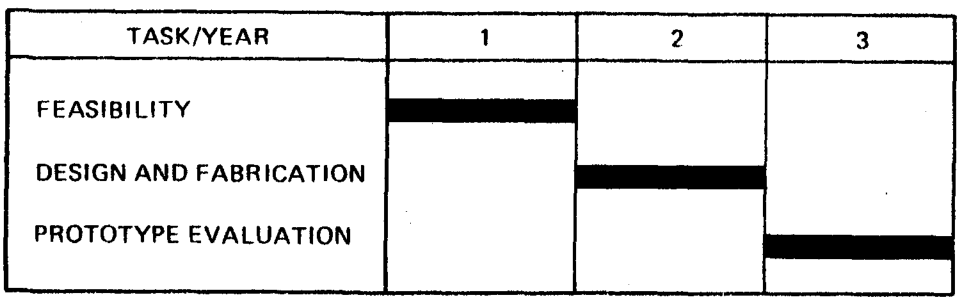
Figure 31. Magnetic Wear Particle Trap

Tagging is accomplished by immersing the part in a flux of neutrons (e.g., in a storage hold of a nuclear reactor), converting a small, uniformly distributed fraction of its atoms into radioisotopes. The part is then incubated for several months, leaving it tagged with only long-lived radioisotopes suitable for long term monitoring of wear.

Materials chosen for bearings are often ferromagnetic because of desirable mechanical properties, so debris from parts made of such material could be captured by a magnetic trap. An alternate method would route the purging or cooling fluid through a duct to a lower-pressure region where a low-pressure-drop filter catches a representative amount of the wear particles.

Isotope tracer detection thus offers a quantitative, noncontacting means of monitoring the wear of components. Isotope tagging provides excellent identification of the wear particles of a selected component in an environment where wear particles generated from several different parts might be present.

Development Outline. In order to apply isotope tracer detection to rocket engine components, methods of capturing the wear particles must be evaluated and tested for efficiency in entrapment and retention. The effect, if any, of radioisotope tagging on the physical properties of the components should be determined. Engine performance effects must be evaluated. Selection of a rapid and reliable ground-based gamma-ray measurement system should also be investigated.



7. Particle Analysis (Figure 32)

Particle analysis is a means of determining wear or contamination by examining the debris in a working fluid. These particles are captured in-flight and analyzed between-flight. The analysis identifies the concentration of each type of particle material so that, with the materials of engine components known, the upstream wear or contaminant source can be monitored. Selection of different materials for wear surfaces of different parts provides good wear isolation. By trending the wear history of each component, sudden excessive wear or incipient failure can be identified and appropriate actions taken.

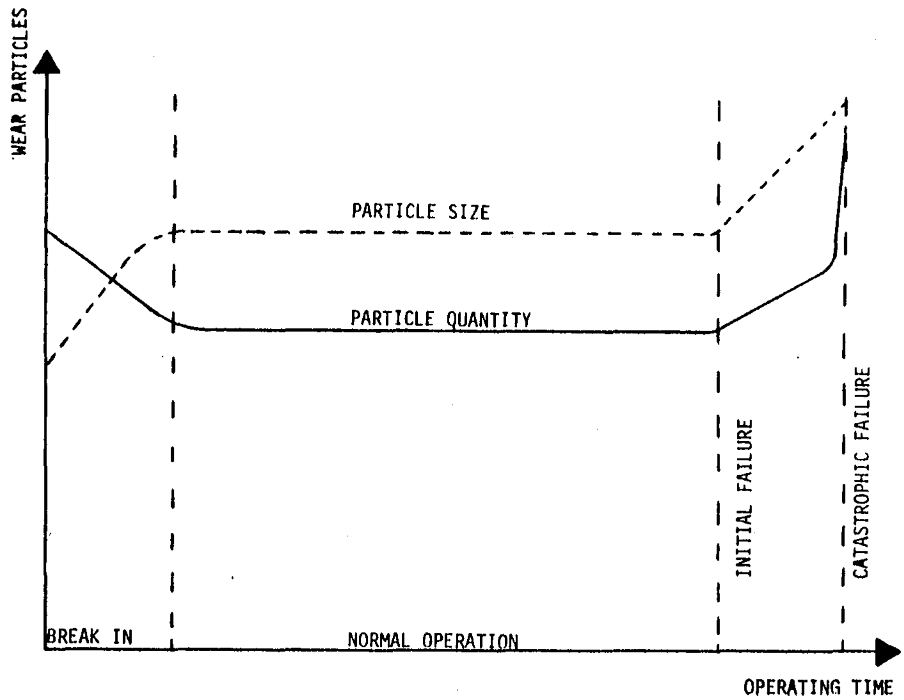


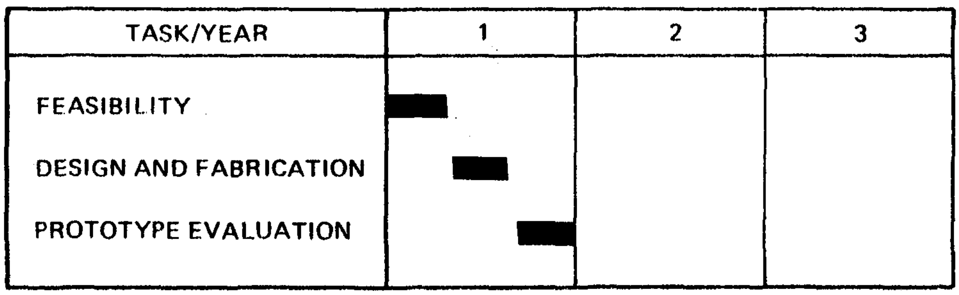
Figure 32. Typical Wear Particle Trending

Materials chosen for bearings are often ferromagnetic because of desirable mechanical properties, so debris from parts made of such material could be captured by a magnetic trap. An alternate method would route the purging or cooling fluid through a duct to a lower-pressure region where a low-pressure-drop filter catches a representative amount of the wear particles.

The analysis of the particles can be accomplished in several ways, the most common being spectrometry. The particles are identified as to their material makeup and the corresponding concentrations of each material. Trending of wear or identification of contaminants can then be used to make timely maintenance actions.

Particle analysis thus offers a quantitative noncontacting means of monitoring wear or contamination. Trending of wear data can permit use of components for their full life as well as enhanced maintenance planning capability.

Development Outline. Before particle analysis can be applied to rocket engines, methods of capturing wear particles and contaimnants must be evaluated and tested for efficiency in entrapment and retention. This would include an investigation of performance effects on the engine. Selection of a rapid and reliable ground-based particle analyzer should also be evaluated.



SELECTION OF PREFERRED IN-FLIGHT OR BETWEEN-FLIGHT TECHNOLOGY

The highest ranked technology selections that were considered upgradable for both in-flight and between-flight, and the effort required to develop these technologies have been identified. A comparison is now made between these technologies for a given failure mode and the preferred technology selected.

In two cases, however, technologies have been combined to form:

1. Wear Detection
2. In-Flight Indirect Condition Monitoring

WEAR DETECTION

The two technologies used to detect wear are:

1. Isotope wear detectors
2. Tunable diode-laser spectrometer

Wear can be monitored by tagging certain engine parts, i.e., turbopump bearings. The debris from these parts can be captured in-flight and recorded or the debris may be entrapped and retained and measured between flights. Details of the capturing system remain to be defined.

The tunable diode-laser spectrometer can be used for detecting wear for condition monitoring by selecting absorption lines for appropriate materials which would be indicative of wear, i.e., the C-F bond corresponding to Teflon valve seats.

The combination of these two technologies will provide data on surface wear and to identify the wear surface within the engine. The detection can either be in-flight and/or between-flight. In any event, detectable wear would be cause for further investigation between-flight.

The need exists to identify liquid rocket engine internal wear surfaces and wear rates. One of the differences between air breathing engines and liquid rocket engines is that the latter currently have no procedure to indicate surface wear, as contrasted to SOAP (Spectrographic Oil Analysis Procedure) for air breathing engine lubricating systems. The need exists to identify liquid rocket engine internal wear surfaces and wear rates. The combination of these two technologies provides such a possibility. The system would satisfy an existing need, be fail-safe, non-intrusive, with medium-to-long-development time and effort. It is strongly recommended.

IN-FLIGHT INDIRECT CONDITION MONITORING

An in-flight indirect condition monitoring system acquires performance data on the engine fluid path and rotating mechanical parameters. This in-flight data can be used to predict, detect, and diagnose failures after it has been validated, corrected, displayed and interpreted. The objective is to predict, detect and diagnose prior to loss in mission capability. The important result from an engine monitoring system is the effect of the information on the maintenance system. Air breathing engine monitoring systems have been designed with various degrees of sophisticated electronic systems, automatic recording, amount of data, etc., and with various degrees of success.

The basic required data of the fluid path is the flowrates, pressures, and temperatures at each desired station. The data for the rotating mechanical equipment is the rpm.

From Task II, four technologies were recommended for in-flight indirect condition monitoring:

1. Ultrasonic flow meter
2. Ultrasonic thermometer
3. Digital quartz pressure sensor
4. Optical tachometer

An assessment has been made that the current pressure sensors, even though they may have some less than desirable characteristics, are adequate to meet the engine condition monitoring requirements; and similarly, adequate tachometers also exist. Therefore, technologies No. 3 and 4, digital quartz pressure sensor and optical tachometer, will not be considered further.

With these two combined technologies, each failure mode will be discussed and the selected technology noted, as well as the reasons for the selection.

Failure Mode 2: Coolant Passage Leakage/Restriction

The in-flight sensor depends upon the measurement of several flowrates and finding a small difference of two large numbers, while the between-flight technology consists of locating coolant passage restrictions prior to the generation of a hot spot and the probable burn-through of the cooling passage.

The primary problem of locating a coolant passage restriction is the fact that the restriction must be located by viewing from the outside of the passage. This problem has always existed and, in general, is similar to the problem of restrictions occurring in air breathing engine turbine air cooled blades. The proposed solution approach is similar - locating an internal passage restriction by the measurement of the external surface temperature. Under some conditions, this solution approach has been successful for air breathing engine turbine air cooled blades.

The selection is the Scanning Optical Pyrometer of between-flight technology.

Failure Mode 3: Joint Leakage

Since this failure mode does not lend itself to an in-flight sensor, the holographic leak detection was selected.

Failure Mode 5: High Torque Turbopump

Both in-flight and between-flight suggest a wear detector. A combination of this technology, and with existing rpm tail-off and break-away torque would improve detection of a high torque.

Failure Mode 6: Cracked Turbine Blades

The selection was the optical pyrometer of the in-flight sensors and was the choice of a state-of-the-art technology over a novel technology. Currently the optical pyrometer is being used in several air breathing turbo-fan engines with a large degree of success. The pyrometer can detect, follow, and control turbine blade material temperature transients, as well as detect certain failure modes for cooled turbine blades. It is proving a very valuable sensor.

Rocket engines historically experience very large temperature transients (often thousands of degrees per second). This is associated with the rapid start and cut-off transient brought about by the application requirements and the engine control system. These large temperature transients result in large thermal stresses. They have existed in the past, they exist today, and probably will exist in the near future. The result is failed turbine blades. It is anticipated in the near future that cooled turbine blades will be incorporated into liquid rocket engines.

Therefore, the optical pyrometer is needed currently and within the foreseeable future. The optical pyrometers can be incorporated in liquid rocket engines with a large pay-off, non-intrusive, fail-safe, short development time, minimum effort, and is state of the art. It is very strongly recommended.

Failure Mode 7: Cracked Convolution Bellows

The in-flight sensor, a digital quartz pressure sensor, depends upon measuring the pressure difference between a cracked and an uncracked convolution. The between-flight technique would be exo-electron emission as a means of locating and characterizing fatigue damage. The current between-flight technique would be a boroscope inspection. The between-flight inspection technique is preferred and the existing procedure appears adequate.

Failure Mode 8: Loose Electrical Connector

No reasonable in-flight sensor or technique was determined. The procedure would then be performed between-flight and the recommended technique would be connector continuity checking.

Failure Mode 9: Bearing Damage

Damage to the bearing probably occurs under dynamic conditions and could be detected by the in-flight fiberoptic deflectometer permitting corrective action either in-flight or between-flight. The wear detector (isotope tracer detection) inspected between-flight would also indicate bearing damage. However, between-flight inspection does not permit the option of corrective action in-flight. Therefore, the in-flight sensor is preferred.

Failure Mode 10: Tube Fracture

No reasonable in-flight sensor or technique was determined. The between-flight technique was exo-electron emission. The current between-flight technique would be an optical inspection for fluid leaks and is the preferred between-flight inspection technique.

Failure Mode 11: Turbopump Seal Leakage

Both the in-flight and between-flight technologies are wear detection; however, the existing technology consists of pressurizing the seal cavity with an inert gas between flights and measuring the pressure decay, which can be related directly to a leakage rate. Historically, acceptable leakage rates have been established and if the leakage rate exceeds acceptable limits, corrective action is indicated. Corrective action in-flight based upon wear detection and wear rates is very remote. The existing technology will be retained and wear detection will be added for between-flight inspection.

Failure Mode 13: Valve Fails to Perform

The in-flight selection is for the "Indirect Condition Monitoring". The performance data along the fluid path would indicate the failure of the valve to perform and permit early corrective action. The between-flight inspection technology would not permit this possibility of early corrective action. Therefore, the in-flight technique is preferred.

Failure Mode 14: Internal Valve Leakage

The in-flight selection is for indirect condition monitoring. The performance data on the fluid path would indicate the internal valve leakage failure and permit early corrective action. The between-flight inspection technology would not permit this possibility of early corrective action. Therefore, the in-flight technique is preferred.

Failure Mode 15: Regulator Discrepancies

The in-flight selection is for indirect condition monitoring. The performance data along the fluid path would indicate a regulator discrepancy and permit early corrective action. The between-flight inspection technology would not permit this possibility of early corrective action. Therefore, the in-flight technique is preferred.

Failure Mode 16: Contaminated Hydraulic Control Assembly

Since this failure mode does not lend itself to an in-flight sensor, the particle analysis of between-flight inspection technology was selected. These selections are shown in Table 20 .

TABLE 20. UPGRADABLE TECHNOLOGY SELECTION

FAILURE MODE CATEGORY	FAILURE MODE DESCRIPTION	UPGRADABLE TECHNOLOGY		SELECTED
		IN-FLIGHT (I)	BETWEEN-FLIGHT (B)	
2	COOLANT PASSAGE LEAKAGE RESTRICTION	ULTRASONIC FLOWMETER	SCANNING OPTICAL PYROMETER	B
3	JOINT LEAKAGE	-	HOLOGRAPHIC LEAK DETECTOR	B
5	HIGH TORQUE TURBOPUMP	RPM TAIL-OFF (EXISTING) WEAR DETECTOR (TUNABLE DIODE SPECTOMETER)	BREAK-AWAY (EXISTING) WEAR DETECTOR (ISOTOPE TRACER DETECTION)	BOTH
6	CRACKED TURBINE BLADES	PYROMETER	EXO-ELECTRON EMISSION	I
7	CRACKED CONVOLUTIONS BELLOWS	DIGITAL QUARTZ PRESSURE SENSOR	BOROSCOPE (EXISTING) EXO-ELECTRON EMISSION	B (EXISTING)
8	LOOSE ELECTRICAL CONNECTOR	-	CONNECTOR CONTINUITY CHECKING (EXISTING)	B (EXISTING)
9	BEARING DAMAGE	FIBEROPTIC DEFLECTOMETER	WEAR DETECTOR (ISOTOPE TRACER DETECTION)	B
10	TUBE FRACTURE	-	OPTICAL INSPECTION (EXISTING) EXO-ELECTRON EMISSION	B (EXISTING)
11	TURBOPUMP SEAL LEAKAGE	WEAR DETECTOR (TUNABLE DIODE SPECTOMETER)	PRESSURE DELAY (EXISTING) WEAR DETECTOR (ISOTOPE TRACER DETECTION)	BOTH
13	VALVE FAILS TO PERFORM	INDIRECT CONDITION MONITORING	WEAR DETECTOR (ISOTOPE TRACER DETECTION)	I
14	INTERNAL VALVE LEAKAGE		THERMAL CONDUCTIVITY LEAK DETECTION	I
15	REGULATOR DISCREPANCIES		THERMAL CONDUCTIVITY LEAK DETECTION	I
16	CONTAMINATED HYDRAULIC CONTROL ASSEMBLY	-	PARTICLE ANALYSIS (EXISTING)	B (EXISTING)

PRIORITY OF SELECTION

Two priority selection bases have been identified for the upgradable technology: (Table 21)

1. The first selection basis is for a priority based-up frequency of occurance of the failure mode. Failure Mode No. 3, Joint Leakage, accounts for almost half the failures (45%). The other three large frequencies are Failure Mode No. 2, Coolant Passage Leakage/Restriction (15%), Failure Mode No. 14, Internal Valve Leakage (12%), and Failure Mode No. 11, Turbopump Seal Leakage (8.5%). The first two Failure Modes (3 and 2) result in 60% of the total failures and the first four Failure Modes in over 80% of the total failures. Three of these four are between-flight technologies. Clearly, these upgradable technologies are very important.

2. The second selection is based upon the magnitude of development effort required. Assuming limited developmental effort, which technology can be established for the least effort? The first is the pyrometer for detecting turbine blade temperatures in-flight. The second is the fiberoptic deflectometer for measuring turbopump bearing loads and damage in flight. The third is the indirect condition monitoring system. The first three are all in-flight technologies. The last three are all between-flight technologies and are wear detector, scanning optical pyrometer, and holographic leak detector.

TABLE 21. PRIORITY SELECTION LIST FOR UPGRADABLE TECHNOLOGY

FREQUENCY RATING	FAILURE MODE DESCRIPTION	FAILURE MODE CATEGORY	PERCENT	UPGRADABLE TECHNOLOGY		STATUS	DEVELOPMENT EFFORT RANKING
				WHERE DETECTED	TECHNOLOGY		
1	JOINT LEAKAGE	3	45	BETWEEN-FLIGHT	HOLOGRAPHIC LEAK DETECTOR	N	6
2	COOLANT PASSAGE LEAKAGE/ RESTRICTION	2	15	BETWEEN-FLIGHT	SCANNING OPTICAL PYROMETER	N	5
3	INTERNAL VALVE LEAKAGE	14	12	IN-FLIGHT	INDIRECT CONDITION MONITORING	N/S	3
4	TURBOPUMP SEAL LEAKAGE	11	8.5	BETWEEN-FLIGHT	PRESSURE DECAY WEAR DETECTOR	EXISTING N	- 4
5	REGULATOR DISCREPANCIES	15	5	IN-FLIGHT	INDIRECT CONDITION MONITORING	N/S	3
6	VALVE FAILS TO PERFORM	13	3.5	IN-FLIGHT	INDIRECT CONDITION MONITORING	N/S	3
7	CRACKED TURBINE BLADES	6	2.5	IN-FLIGHT	PYROMETER	S	1
8	HIGH TORQUE PUMP	5	2.5	IN-FLIGHT BETWEEN-FLIGHT	RPM TAIL-OFF BREAK-AWAY WEAR DETECTOR	EXISTING EXISTING N	- 1 4
9	CONTAMINATED HYDRAULIC CONTROL ASSEMBLY	16	1.5	BETWEEN-FLIGHT	PARTICLE ANALYSIS	EXISTING	-
10	BEARING DAMAGE	9	1.5	IN-FLIGHT	FIBEROPTIC DEFLECTOMETER WEAR DETECTOR	N N	2 4
11	CRACKED CONVOLUTION BELLOWS	7	1.5	BETWEEN-FLIGHT	BOROSCOPE	EXISTING	-
12	TUBE FRACTURE	10	1	BETWEEN-FLIGHT	OPTICAL INSPECTION	EXISTING	-
13	LOOSE ELECTRICAL CONNECTOR	8	0.5	BETWEEN-FLIGHT	CONNECTOR CONTINUITY CHECKING	EXISTING	-

CONCLUDING REMARKS

The following concluding remarks are made with regard to this report.

1. Approximately 85,000 liquid rocket engine failure reports, from 30 years of developing and delivering major pump fed engines, were reviewed and screened, and reduced to 1771. These were categorized into 16 different failure modes. These failure modes were common to all engines and historically consistent.

2. The state of the art of engine condition monitoring for in-flight sensors and between-flight inspection technology was determined.

3. Failure propagation diagrams for the 16 failure modes were established; the potential measurands and diagnostic requirements were identified and compiled. The sensors and inspection technology were matched with the measurands and requirements.

4. The sensor and inspection technology was assessed and ranked.

5. Areas requiring advanced technology development have been identified and are as follows:

   a. <u>Direct In-Flight Condition Monitoring</u>

      (1) Optical Pyrometer - turbine blade temperatures
          Very strongly recommended

      (2) Fiberoptic Deflectometer - bearing condition
          Very strongly recommended

      (3) Isotope Wear Detector - wear particles
          Strongly recommended

      (4) Tunable Diode Laser Spectrometer - wear particles
          Strongly recommended

   b. <u>Indirect In-Flight Condition Monitoring</u>

      (5) Ultrasonic Flowmeter - propellant flow
          Very strongly recommended

      (6) Ultrasonic Thermometer - high temperatures
          Very strongly recommended

   c. <u>Between-Flight Inspection Techniques</u>

      (7) Holographic Leak Detector - fluid leaks
          Strongly recommended

      (8) Scanning Pyrometer - blocked fluid passages
          Recommended

APPENDIX A. GENERAL DESCRIPTION OF CONVENTIONAL ROCKET ENGINES

APPENDIX A

GENERAL DESCRIPTION OF CONVENTIONAL ROCKET ENGINES

Conventional rocket engines are defined as bipropellant engines that use liquid oxygen and a hydrocarbon based fuel (RP-1) for combustion (Fig. 33). The propellants are fed into the thrust chamber (the major engine component that transforms, through combustion, the potential energy stored in the fuel into kinetic energy by means of centrifugal pumps powered by gas generator-driven turbines. The flow of propellants is regulated by valves which, in turn, are controlled by hydraulic, electrical and/or pneumatic systems to establish required valve actuation relationship for engine operation. An ignition system, either by means of hypergolic pyrotechnics or electrical spark plugs, initiates combustion in the thrust chamber and gas generator. Most engine systems are provided with a purge system that protects some components from contamination by the external environment or protect critical areas against mixture of potential hazardous leakage of propellants. Other subsystems might be found on conventional engines depending on their cycles of operation or state-of-the-art development.

The conventional rocket engines (Fig. 34) referred to in this study are: Atlas MA-3 and MA-5, Thor, RS-27, F-1 and H-1. The J-2 engine is a more advanced type of propulsion system since it uses hydrogen as fuel, yielding a higher specific impulse (426 sec/vacuum). The SSME propulsion system, again using hydrogen as fuel, is another step forward in state of the art, and uses a two-stage combustion process requiring higher pressures and making a more efficient use of propellants. A brief description and application of these propulsion systems follows.

SPACE SHUTTLE MAIN ENGINE (SSME)

The high performance requirements of the SSME engine demand the use of a staged combustion power cycle (Fig. 35) coupled with high combustion chamber pressures. In the SSME staged combustion power cycle, the propellants are partially burned at low mixture ratio, very high pressure, and relatively low temperatures in the preburners to produce hydrogen-rich gas to power the turbopumps. The hydrogen-rich steam is then routed to the main injector where it is injected, along with additional oxidizer and fuel, into the main combustion chamber at a high mixture ratio and high pressure. Hydrogen fuel is used to cool all combustion devices directly exposed to contact with high-temperature combustion products. An electronic engine controller automatically performs checkout, start, mainstage and engine shutdown functions.

The SSME was developed especially for the Space Shuttle Orbiter vehicle, which uses three systems for launch. The SSME is a reusable, high performance, liquid propellant rocket engine with variable thrust. The engine is ignited on the ground at launch and operates in parallel with the solid rocket boosters during the initial ascent phase and continues to operate for approximately 480 seconds total firing duration. Each of the rocket engines operates on a mixture ratio of 6:1, and a chamber pressure of 3000 psia to produce a sea-level thrust of 375K pounds of thrust. The engine is throttleable over a thrust range of 65 to 109% of design thrust level. This provides a higher thrust level during liftoff and the initial ascent phase, and allows orbiter acceleration to be limited to 3 g during the final ascent phase. The engines are gimbaled to provide pitch, yaw, and roll control during orbiter boost phase.

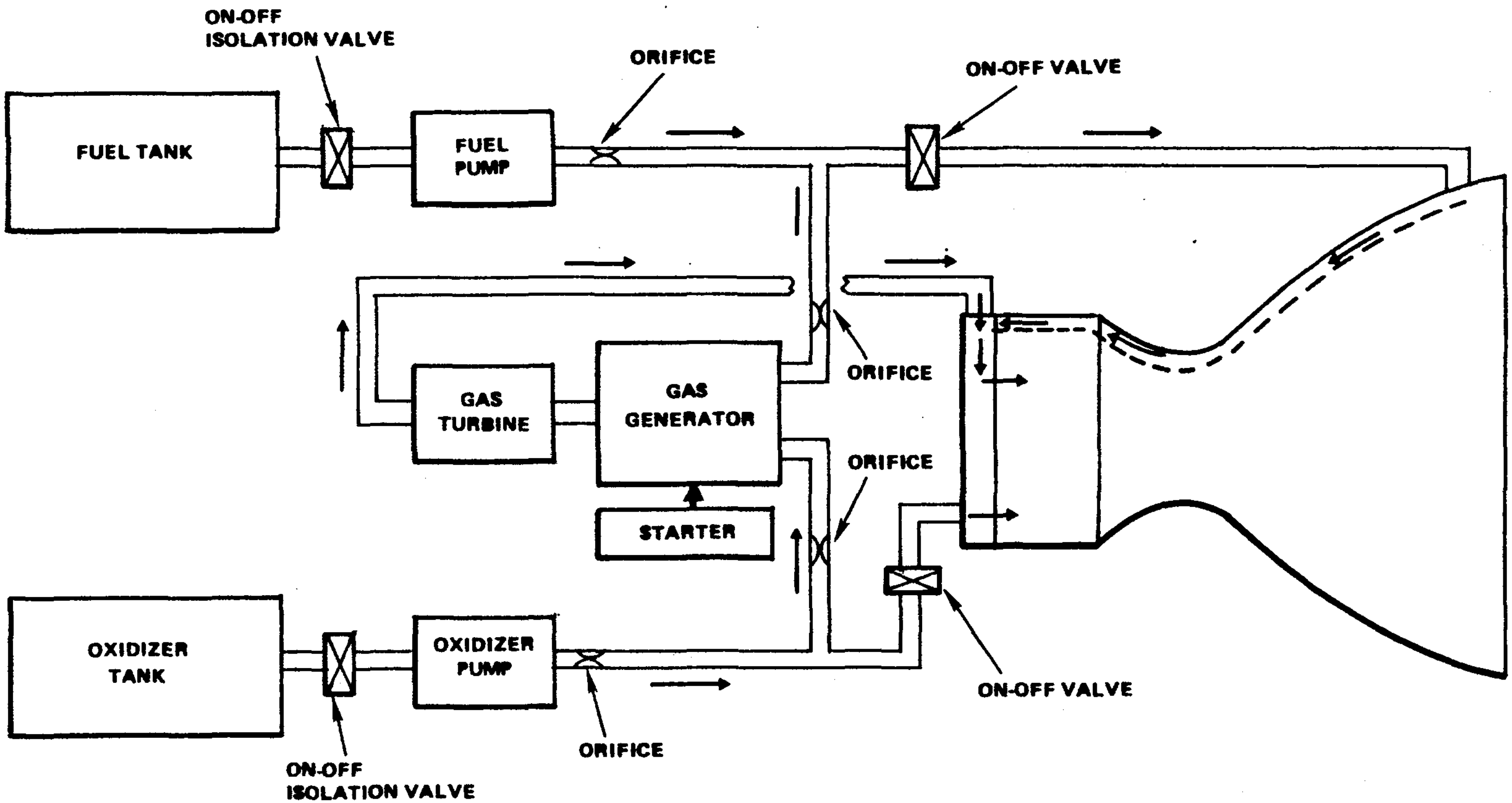


Figure 33. Simplified Existing Control System, Gas Generator Engine

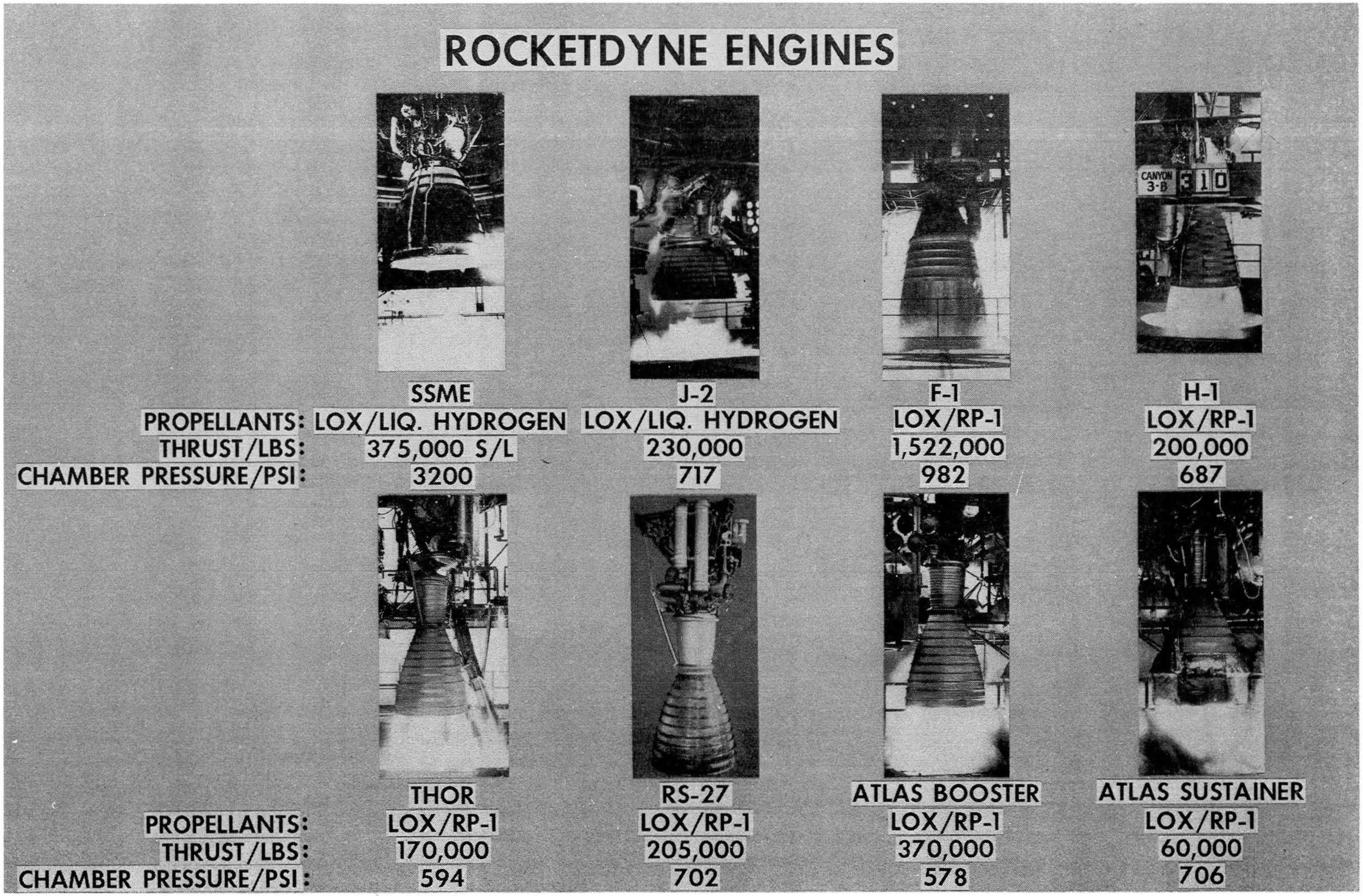
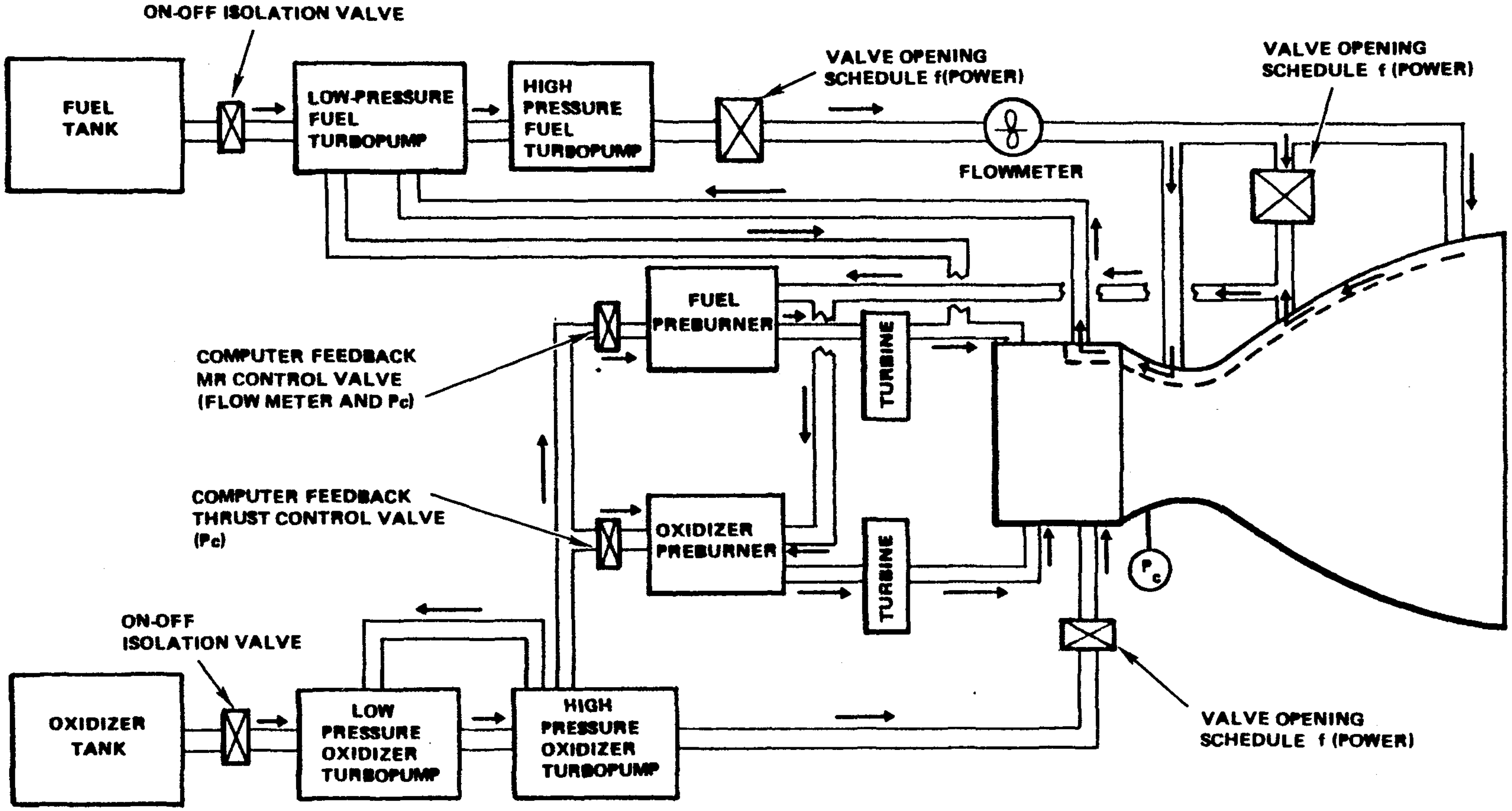


Figure 34. Conventional Rocket Engines



- **3 SCHEDULED AND 2 CONTROLLED VALVES FOR MR AND THRUST LEVEL**

Figure 35. Simplified Engine Control System, Staged Combustion Engine

J-2 ENGINE

The J-2 rocket engine is a high-performance, multiple-restart engine that uses liquid oxygen for oxidizer and liquid hydrogen for fuel. Each propellant is pumped into the thrust chamber by separate gas-turbine-driven, direct-drive turbopumps. The two turbopumps are powered in series by a single gas generator that uses the same propellants as the thrust chamber. The thrust chamber is tubular-walled and is regeneratively cooled by circulating fuel through the tubes before the fuel is injected into the combustion area. The engine has a refillable start tank from which pressurized gaseous hydrogen is routed to the turbopump turbines for starting the engine. This feature, combined with the augmented spark ignition system, makes the J-2 a multistart engine.

The J-2 engines were ignited at altitude as they powered the second stage of the Saturn V Vehicle and developed 225K pounds of thrust. The burn duration was 480 seconds with a mixture ratio of 5.50 (O/F). It yielded a specific impulse (I_{sp}) of 423.8.

The J-2 rocket engine was developed to provide the power for the SIVB stage of the Saturn IB vehicle and for the SII and SIVB stages of the Saturn V vehicle.

The SII stage is propelled by a cluster of five J-2 engines, four outboard engines and one inboard engine. Because only a single engine start is required for SII stage application, the engines are modified to delete the engine restart capability by blocking off the start tank refill lines. The inboard and outboard engines are basically identical.

The SIVB stage is propelled by one J-2 engine. The engine used in the Saturn IB, SIVB stage, and the engine used in the Saturn V, SIVB stage are basically identical.

H-1 ENGINE

The H-1 rocket engine is a single-start, fixed thrust, pump-fed, regeneratively cooled liquid bipropellant engine that uses liquid oxygen and RP-1 fuel. The propellants are supplied to the gimbal-mounted thrust chamber by a gas generator-driven turbopump that has two centrifugal pumps on a single shaft. The engine is calibrated to develop a sea-level rated thrust of 205,000 pounds of thrust with an I_{sp} of 263.4 seconds at a mixture ratio of 2.23:1 oxidizer/fuel. The H-1 engine was ignited on the ground and the scheduled burn duration was 150 seconds.

The H-1 engine was developed to boost the Saturn IB vehicles. In the first stage booster, eight H-1 engines were used in a two-concentric arrangement, four outboard, and four inboard. The outboard engines were capable of being gimbaled for pitch, yaw, and roll control. At launch the vehicle was held down until satisfactory mainstage combustion was established in all eight engines.

F-1 ENGINE

The F-1 engine is a single-start, fixed thrust, liquid bipropellant engine, calibrated to develop a sea-level-rated thrust of 1,522,000 pounds with an I_{sp} of 265 seconds during 150 seconds of operation. Engine propellants are liquid oxygen and propellant RP-1 (kerosene) fuel at a mixture ratio of 2.27:1. The propellant fuel is used as the working fluid for the gimbal actuators and for the engine control system, and is also used as the turbopump bearing lubricant.

The F-1 engine features a two-piece thrust chamber that is tubular-walled and regeneratively cooled to the 10:1 expansion ration plane, and double-walled and turbine gas cooled to the 16:1 expansion ration plane; a thrust chamber mounted turbopump that has two centrifugal pumps on a single shaft driven by a two-stage, direct-driven turbine; one-piece rigid propellant ducts that are used in pairs to direct the fuel and oxidizer to the thrust chamber; and a hypergolic fluid cartridge that is used for thrust chamber ignition.

Thrust vector changes are achieved by gimbaling the entire engine. The gimbal block is located on the thrust chamber dome, and actuator attach points are provided by two outriggers on the thrust chamber body.

The F-1 propulsion system was developed to provide the power for the booster flight phase of the Saturn V vehicle. Five engines are clustered in the S-IC stage of the Saturn V to obtain the necessary 7,610,000 pounds thrust. The ignition of the engine takes place on the ground and the vehicle was not released until satisfactory mainstage operation was established in all five engines.

RS-27 ENGINE

The RS-27 rocket engine is, like the H-1 engine, a single-start, fixed thrust, pump-fed, regeneratively cooled liquid bipropellant engine that used liquid oxygen and RP-1 fuel. The propellants are supplied to the gimbal-mounted thrust chamber by a gas generator driven turbopump that has two centrifugal pumps on a single shaft. The engine is calibrated to develop a sea-level thrust of 205K pounds of thrust with an I_{sp} of 262.7 seconds and a mixture ratio of 2.24:1 oxidizer/fuel.

The RS-27 engine system was developed from H-1 and Thor engines hardware, to serve as a booster engine for the Delta Launch vehicle.

The engine is ignited on the ground and for over 240 seconds operates in parallel with a number of solid propellant booster rockets clustered around the vehicle.

Its mission is to power the first stage of the Delta launch vehicle used to place in orbit a variety of commercial satellites.

THOR ENGINE

The Thor engine is like the RS-27, a single-start fixed thrust, pump-fed, regeneratively cooled liquid engine that used liquid oxygen and RP-1 fuel. The propellants are supplied to the gimbal mounted thrust chamber by a gas generator-driven turbopump that has two centrifugal pumps, on a single shaft. The latest version of the engine is calibrated to develop a sea-level thrust of 170K pounds of thrust with an I_{sp} of 247 seconds and a mixture ratio of 2.30:1, oxidizer/fuel.

The Thor engine, developed in the late 1950's, was designed to serve as the main propulsion system for the Thor missile.

Lately, the Thor, with a burn duration of 175 seconds, has been used to orbit payloads.

ATLAS ENGINE

The Atlas propulsion system is composed of two separate types of engines, booster and sustainer. The booster consists of two low altitude thrust chambers with their components, similar to the H-1 or RS-27 engines. The sustainer engine is a high-performance propulsion system designed for high-altitude operation. The design of the system is such that the booster engines deliver 330K of thrust and burn of 145 seconds, the sustainer engines have a thrust of 57K and operate for 320 seconds.

The Atlas engine, MA-3 version, was developed to power the ICBM missiles in the late 1950s. An uprated version, the MA-5 engine, was developed for use as an intermediate launch vehicle.

The Atlas engine systems yield a specific impulse of 258 seconds for the booster and 219 for the sustainer. The mixture ratio for both engines is 2.25.

Currently the Atlas MA-3 version E/F series is used to boost classified payloads. The MA-5 engine systems are incorporated into NASA's SLV-3D launch vehicle that is used to place satellites in orbit.

APPENDIX B. FAILURE SUMMARY SHEETS

APPENDIX B

FAILURE SUMMARY SHEETS

Each failure mode that passed the UCR screening process has been documented as one of the following sheets. Each sheet, one per each failure mode, records the failure mode and cause, the frequency of failure, the design life, if available, and the effect of the failure upon the subsystem and/or engine. Further identification is shown with respect to the failure type, criticality, reaction time, and the detection method used. An evaluation is made with respect to failure predictability, and potential measurands of the parameters that are affected by the failure. Based on these elements, a selection of suitable in-flight monitoring systems and between-flight inspection techniques is presented for each failure mode. All these data are recorded in a format that was approved and found appropriate for conducting this study.

FACTOR IDENTIFICATION	FACTORS
(a)	COMPONENT NAME - EVALUATION CONDUCTED AT THE ENGINE SYSTEM AND COMPONENT OR LINE REPLACEABLE UNIT (LRU) LEVEL. THE MAJOR CATEGORIES TO BE EVALUATED ARE: 1. ENGINE SYSTEM 2. THRUST CHAMBER 3. NOZZLE 4. INTERCONNECTS (a) DUCTS AND LINES (b) HEAT EXCHANGER (c) SEALS 5. COMBUSTION DEVICES (a) GAS GENERATOR (b) PREBURNER (c) IGNITERS • HYPERGOL • ELECTRICAL 6. CONTROLS (a) PROPELLANT CONTROL VALVES • FUEL VALVE • OXIDIZER VALVE • GAS GENERATOR AND PREBURNER VALVES • IGNITER VALVES (b) ELECTRONIC CONTROLLERS (c) PNEUMATIC SYSTEM (d) HYDRAULIC SYSTEM 7. TURBOMACHINERY (a) FUEL PUMP (b) OXIDIZER PUMP
(b)	FAILURE MODE AND CAUSE - A BRIEF DESCRIPTION OF THE FAILURE MODE AND CAUSE WHICH CONTRIBUTED TO THE FAILURE. APPLICABLE HYPOTHESIZED FAILURE MODES AND CAUSES WILL ALSO BE INCLUDED.
(c)	FREQUENCY OF FAILURE - THE NUMBER OF OCCURRENCES OF SUBJECT FAILURE MODE EXPERIENCED BY ENGINE SYSTEM AND/OR COMPONENT WITH RESPECT TO TOTAL ENGINE OPERATIONS
(d)	DESIGN/ACTUAL LIFE - WHERE DESIGN AND LIFE INFORMATION EXIST, THE LIFE OF THE COMPONENT UNTIL FAILURE OCCURRENCE AND DESIGN LIFE WILL BE DOCUMENTED
(e)	EFFECT OF FAILURE - EFFECT OF FAILURE INCLUDING DESCRIPTION OF THE FAILURE MODE SHOWING FREQUENCY OF EVENTS RESULTING IN THE ULTIMATE FUNCTIONAL EFFECT ON ENGINE OPERATION (i.e., PERFORMANCE DEGRADATION, PREMATURE SHUTDOWN, EXPLOSION - UNCONTAINED, ETC.).
(f)	FAILURE TYPE - LEAKAGE, STRUCTURAL, OVERPRESSURE, OVERTEMPERATURE, OPERATIONAL, EXCESSIVE WEAR, ETC.
(g)	PRIMARY OR SECONDARY FAILURE - FAILURE MECHANISM OR COMPONENT FAILURE BEING THE PRIMARY CAUSE OF THE FAILURE MODE OR FAILURE MODE MANIFESTED BY ANOTHER FAILURE OCCURRENCE
(h)	CRITICALITY - THE FAILURE MODE ASSESSED AS TO ITS FUNCTIONAL EFFECT ON ENGINE OPERATION: CRITICALITY CATEGORY — ENGINE EFFECT 1 — EXPLOSION, BURNTHROUGHS, UNCONTAINED FRAGMENTATION 2 — PREMATURE ENGINE SHUTDOWN - ENGINE DAMAGE 3 — PREMATURE ENGINE SHUTDOWN - NO DAMAGE 4 — PERFORMANCE DEGRADATION 5 — NO EFFECT
(i)	REACT TIME - ESTIMATE OF THE TIME FROM DETECTION OF EFFECT TO FAILURE OCCURRENCE Imm = IMMEDIATE (MILLISECONDS RANGE) Inst = INSTANTANEOUS (LESS THAN A FEW SECONDS) TIME GREATER THAN A FEW SECONDS TO BE AS NOTED
(j)	DETECTION METHOD (USED) - DESCRIBE METHOD OR INSTRUMENT BY WHICH FAILURE WAS DETECTED
(k)	FAILURE PREDICTABILITY - DESCRIBE WHETHER FAILURE MECHANISM COULD BE PREDICTED THROUGH REVIEW OF OPERATIONAL DATA OF PREVIOUS OPERATIONS AND/OR GROUND INSPECTION TECHNIQUES. (WEAR INDICATORS, TRENDS ANALYSIS OF PERFORMANCE PARAMETERS, ETC.)
(l)	RESOLUTION - BRIEFLY DESCRIBE RESOLUTION METHOD FOR PREVENTING RECURRENCE OF FAILURE
(m)	VIABLE IN-FLIGHT MONITORING - LISTING VIABLE ENGINE CONDITION MONITORING DEVICES CAPABLE OF DETECTING FAILURE (TASK II)
(n)	BETWEEN-FLIGHT INSPECTION - LISTING OF BETWEEN-FLIGHT INSPECTIONS AND TECHNIQUES THAT WOULD PROVIDE VERIFICATION OF ENGINE INTEGRITY PRIOR TO NEXT FLIGHT
(o)	REMARKS - ADDITIONAL INFORMATION NECESSARY FOR DESCRIPTION AND EXPLANATION

FAILURE SUMMARY SHEETS

SSME DATA

ENGINE SYSTEM/COMPONENT SSME Oxidizer Valve (Main)

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
1. Bolt Torque Relaxation A. Main Oxidizer Valve Caused by high flow velocity, valve geometry and valve-line interface geometry which trigger abnormal acoustic levels and subsequent vibration. Redesign of hardware obviates new detection devices.	2f 1.1% 1f .55%		Premature engine shutdown by MOV vibration safety cutoff with possible damage to engine. Vibration induced mechanical/structural failures resulting in fire and explosion in MOV area. Test was cut off at 255.6 Sec by HPFT turbine discharge Temp redline. There was severe damage to engine detected post test.	Cavitation Vibration Structural Acoustic	Primary	2 or 3	Imm N/A	Vibration safety cutoff Post-test inspection		Vibration Acoustics *Screw Loosening Leak Fretting Extension *Are Detectable Between Flight Only

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT SSME/Nozzle/Combuster

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
2. <u>Coolant Passage Leakage</u> Nozzle Tube Splits and Ruptures Caused by material embrittlement from prior repairs, start and shutdown transient surges, contamination clogged tubes, intermittent braze with regions of non-braze.	4f 2.23%		Loss of fuel at nozzle tubes caused premature engine cutoff due to HPOT discharge temperature exceeding redline.	Overtemp and leakage.	Primary	3	Imm.	HPOT turbine Temp cutoff		Metal Embrittlement Pressure Transient *Tube Splits Flow, Reduction Mixture Ratio Shift Temperature, Rise In Combustion
Caused by thermal strain and/or braze porosity because of prior repairs, leakage through braze joint due to insufficient bonding during braze cycle.	30f 16.75%		External fuel leakage would result in fire hazard and would cause performance degradation, and coolant loss.	Thermal strain	Primary	4 or potential 1	N/A	Post-test inspection		*Are Detectable Between Flight Only

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Ultrasonic Thermometer (Flame) Ultrasonic Flowmeter (Nozzle) Polarometer Tunable Diode Laser Spectrometer (Mixture Ratio)	Ultrasonic Leak *Acoustic Holography* X-ray Radiography Gamma Radiography Pentoxide Polarometry Hydrogen Polarometry Hygrometer Optical Pyrometry Holographic Leak Millimeter-wave Interferometry	

ENGINE SYSTEM/COMPONENT SSME/HP Fuel Turbopump-Static Seals

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage A. Hot Gas Caused by scratched or damaged seals and/or flanges-flange and/or seal distortion from manufacturing and installation-loosened plug in tap.	5f 2.8%		One instance resulted in premature cutoff due to fire in LPFTP area. This is primary concern of hot gas leakage which can result in fire/explosion or at best, damage to adjacent hardware.	Defects in material Improper installation Warping Torque relaxation	Primary	2 or potential 1	Imm	Operator observer cutoff-visual		*Distortion *Torque Relaxation Leak Fire *Are Detectable Between Flight Only

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Extensiometer Ultrasonic Leak Leak Tape/Coating Optical Leak Laser Interferometry Differential Radiometry Holographic Leak Resistivity Monitoring Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Leak Fluid Pressure Decay	

ENGINE SYSTEM/COMPONENT SSME Static Seals

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage Continued b. Propellant Leakage	12 6.6%		One instance resulted in premature cutoff when fuel turbine discharge Temp exceeded redline due to MFV leak.	Defects in material Improper installation Torque relaxation	Secondary	2	Imm	HPFT turbine discharge Temp cutoff.		
			Other propellant leaks can lead to fire and/or explosion in the presence of an ignition source or to damage to adjacent hardware. The other recorded five instances were pre/post test where corrective action was taken to prevent recurrence.		Primary	Potential 1 or 5	N/A	Pre/post test inspection		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT SSME Hot Gas Manifold

5

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
4. Hot Gas Manifold Transfer Tube Liner Cracks Caused by material fatigue property degradation due to excessive operating Temp of engine 2004 preburner configuration, resulting in high cycle fatigue. Redesign of hardware obviates new detection devices.	3f 3.3%		These occurrences were discovered post test and corrective action taken to prevent recurrence. Propagation of this failure if not detected and corrected, could possibly result in burn-through of tube with resultant hot gas leak and catastrophic consequences.	Overtemp	Primary	5 or potential 1	N/A	Post-test inspection		Transfer Tube Crack Temperature Transient Mixture Ratio Shift Fatigue

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT SSME Propellant Turbopump Labyrinth Seal

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
5. High Torque a. Excessive Vibration Caused by rubbing of the interstage seal with consequent abnormal G-levels.	2f 1.1%		Premature test cutoff by HPFT vibration monitor. No significant damage to engine.	Interference Excessive temperature Vibration	Primary	3	Imm	HPFT vibration monitor.		Torque Ripples Temperature, Seals Vibration Acoustics Worn Particles RPM Tailoff Contaminants
b. Excessive Torque Caused by rubbing of the interstage seal and/or malfunction of turbine tip seals.	13f 7.3%		These occurrences were detected during post-test torque checks with corrective action taken to prevent recurrence. Propagation of this failure, if not detected and corrected would result in effects as noted in "a" and possible engine damage.	Interference Excessive temperature Vibration	Primary	5 or potential 2 or 3	N/A	Post-test inspection.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
RTD Thermometer Optical Tachometer Accelerometer Isotope Wear Detector Hydrophone Ferromagnetic Torquemeter Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

ENGINE SYSTEM/COMPONENT SSHE Propellant Turbopump Labyrinth Seal

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
c. Excessive Temperature Conditions Caused by rubbing of the interstage seal and binding of third stage impeller with failed HRD ring resulting in incompatible start transient conditions. Possibly initiated by metal chip. Redesign of hardware obviates new detection devices	5f 2.8%		These five failures represent a progressive failure culminating in premature test cutoff with major damage to pump.	Excessive temperature	Primary	2	Imm	Turbine discharge over-temp redline cutoff.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT SSME/High Pressure Fuel Turbopump

Page

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
6. <u>Cracked Turbine Blades</u> Caused by thermal spikes during engine start/cutoff transients with resultant spalling of protective coatings and localized melting of turbine parts. Also due to debris and/or contamination impacting on nozzle or other turbine parts. Redesign of hardware obviates new detection devices.	9f 5%		These occurrences were all detected during post-test or removal inspection and corrective action taken to prevent recurrence. Propagation of these failures without detection could result in severe damage to pump and possibly to engine.	High-energy transients	Primary	5 or potential 1	N/A	Post test and/or removal inspection		Fatigue Temperature, Transient Pressure, Transient *Foreign Object Damage Vibration Acoustics *Fatigue Balance *Are Detectable Between Flight Only

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Sensors Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Pyrometer Vibration Hydrophone Fiberoptic Bearing Detector Exo-electron Detector	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Remnant Magnetization Optical Holography Borescoping Exo-electron Emission Positron Annihilation Electric Current Injection Eddy Current	

ENGINE SYSTEM/COMPONENT SSME/High Pressure Fuel Turbopump-Bellows & Shield

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
7. Crack - Convolutions Bellows Caused by high cycle fatigue	5f 2.8%		Detected during post test inspection and corrective action taken. Propagation of this failure could result in hot gas leakage with possible engine damage.	High-energy transients	Primary	5 or potential 1	N/A	Post-test inspection-visual.		Temperature, Transient Pressure, Transient *Foreign Object Damage Acoustics Vibration *Fatigue *Are Detectable Between Flight Only

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Sensor Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital RTD Thermometer Accelerometer Hydrophone	Ultrasonic Flaw Isotope Thermometry Remnant Magnetization Borescoping Penetrants Optical Holography Exo-electron Emission Positron Annihilation Electric Current Injection Eddy Current	

ENGINE SYSTEM/COMPONENT SSME/Electrical Harnesses

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
8. Loose Electrical Connectors Assume caused by improper torque combined with effects of vibration during test. Redesign of hardware obviates new detection devices.	6f 3.35%		All of these instances were detected during post-test inspection and corrective action taken. In two instances, FID displays were actuated by connector failure. *Dependent on the location* of the connector which could become loose or disengaged, any number of failures could occur, including engine cutoff. This would be catastrophic if occurrence was during flight.	High energy transients: Vibration heat acoustics	Primary	5 or potential 1, 2 or 3	N/A	Post-test inspection.		*High Temp. Transient Torque, Relaxation Continuity, Intermit. Separation *Are Detectable Between Flight Only

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Isotope Thermometry Continuity Checking Torquing	

ENGINE SYSTEM/COMPONENT SSME High Pressure Oxidizer Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
9. Bearing Damage Caused by incipient spalling and/or superficial wear associated with the load track-insufficient bonding of bearing cage wrap.	4f 2.23%		These occurrences were all detected during post-test inspection and corrective action taken. If not detected, propagation of these conditions could result in bearing failure with possible severe damage to pump and/or engine.	Material	Primary	5 or potential 1	N/A	Post-test inspection		Temp., Excessive Race Vibration Acoustics Torque, Ripples Worn Particles RPM Tailoff Fatigue Contaminant Balance

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Optical Tachometer Isotope Detector Fiberoptic Detector RTD Thermometer Accelerometer Hydrophone Ferromagnetic Torquemeter Exo-electron Detector Tunable Diodelaser Spectrometer	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Exo-electron Emission Positron Annihilation Eddy Current Torquing	

FAILURE SUMMARY SHEETS

J-2 ENGINE DATA

ENGINE SYSTEM/COMPONENT J-2/Main Oxidizer Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
1. Bolt Torque Relaxation b. Sequence Valve Internal Leakage Caused by low torque on lipseal retainer screws allowing leakage past the flange portion of the lipseal.	3f .186%		Resulted in partial opening of GG control valve at engine start. Premature introduction of propellants into GG will cause detonation at start with possible damage to GG.	Torque relaxation	Primary	5 or potential 2	N/A	Post-test checkout		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT J-2 Seals

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage a. Hot Gas Caused primarily by damaged/defective seals. Teflon extrusion of seal and off-center installation are also contributing factors. Corrective measures have been initiated to control these problems.	9f .560%		Two of these failures resulted in premature test cutoff due to fire in engine area-only minor damage occurred. While human error and poor manufacturing and handling procedures contribute greatly to the incidence of this type of failure, the potential consequences of hot gas leakages mandate all possible effort for their control.	Plastic deformation	Primary	3 - (2f) and 5 - (7f) or potential 1	Inst.	Observer cutoff when fire detection Temp exceeded redline		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Extensiometer Ultrasonic Leak Leak Tape/Coating Optical Leak Laser Interferometry Differential Radiometry Holographic Leak Resistivity Monitoring Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Leak Fluid Pressure Decay	

ENGINE SYSTEM/COMPONENT J-2/Oxidizer Turbopump - First Stage Wheel

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
6. Cracked Turbine Blades Caused by wheel rub due to axial vibration. Design change initiated to correct this problem.	7f .435%		These failures were detected during pre/post test inspection and corrective action taken. Propagation of this failure could result in additional damage to turbine area with performance degradation almost certain.	Structural and vibration	Primary	5 or potential 4	N/A	Pre/post test inspection		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Sensors Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Pyrometer Vibration Hydrophone Fiberoptic Bearing Detector Exo-electron Detector	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Remnant Magnetization Optical Holography Borescoping Exo-electron Emission Positron Annihilation Electric Current Injection Eddy Current	

ENGINE SYSTEM/COMPONENT J-2/Fuel Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
9. *Bearing Damage* No failure analysis is available for this failure but condition could be caused by contamination in bearing or by damage to bearing parts due to improper lubrication or handling during installation or servicing.	IF .062%		Detected post test during disassembly. Bearing retainer and bearing support bore also damaged as a result of this failure. Undetected *propagation of this* failure could result in pump seizure and possible damage to pump and engine.	Structural	Primary	5 or potential 2	N/A	Post-test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Optical Tachometer Isotope Detector Fiberoptic Detector RTD Thermometer Accelerometer Hydrophone Ferromagnetic Torquemeter Exo-electron Detector Tunable Diodelaser Spectrometer	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Exo-electron Emission Positron Annihilation Eddy Current Torquing	

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ENGINE SYSTEM/COMPONENT J-2/ASI Fuel Injection Hose/Fuel Line

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
10. Tube Fracture External fuel leakage Due to ASI fuel line rupture caused by inadequate line strength in vacuum environment. Also a problem of very little clearance during installation and removal procedures contributing to damage to hose. Design changes have been initiated for control of this problem.	4f .249%		One of these failures resulted in premature cutoff of one engine at 261 seconds into SA-502 mission and another in failure of engine to achieve mainstage operation at second burn during same mission, with consequent cutoff by vehicle logic when mainstage pressure switch did not actuate. Considerable thrust chamber and possible engine damage could follow as a result of this failure.	Leakage and fatigue	Primary	3 or potential 2	Imm	Vehicle logic cutoff device.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Flaw Acoustic Emission X-ray Radiography Penetrants Laser Interferometry Exo-electron Emission Positron Annihilation Electric Current Injection	

ENGINE SYSTEM/COMPONENT J-2 ASI Oxidizer Line

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Tube Fracture Continued Caused primarily by accumulation of tolerances and to installation procedures resulting in preloading of line. Vibration induced fatigue during hot fire testing then led to fracture of line. Four instances of attaching clamp for ASI oxidizer line-to-MOV flange breaking due to same causes are included in these ten failures. New installation procedures have been initiated to minimize problems of preloading.	13f .809%		One instance resulted in premature cutoff of test due to loss of M/S O.K. signal, also fire detection system and observer noted fire in area. Eleven of the 13 occurrences were detected pre/post test but potential failure as noted above would result from undetected propagation of the problem.	Structural and vibration fatigue	Primary	2 (2f) and 5 (11f)	Imm.	Mainstage O.K. cutoff monitor.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT J-2/ASI Oxidizer Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
13. Valve Fails to Perform a. Moisture, Ice Valve failed to open. Caused by icing condition in area of poppet and poppet guide resulting from moisture entering valve during component test.	1F .062% 6F .37%		Test abort since failure results in lack of ignition due to lack of oxidizer supply to ASI assembly. Pre-test checks detected valve failure.	Contamination (moisture)	Primary	3	Imm.	Visual - no start		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressor Sensors Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Isotope Wear Spectrometer Tunable Diode Laser Spectrometer	Ultrasonic Leak Acoustic Holography Isotope Tracers Pentoxide Polarometry Hygrometer Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography	

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ENGINE SYSTEM/COMPONENT J-2/Start Tank Vent and Relief Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Valve Fails to Perform Cont. Caused by freezing of moisture in control portion of valve actuator with resultant binding or seizure of valve *Initiation of improved* drying procedure has been made to attempt to alleviate this condition.	6f .373%		Results in decay and most likely loss of start tank pressure. Two of these failures caused cancellation of planned tests, others detected pre/post and corrective action taken. Loss of start tank pressure would result in inability to start engine	Contamination (moisture) and freezing	Primary	5 or potential 3	Inst.	Test cancelled by observer on decay of start tank pressure-visual.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

5-9

ENGINE SYSTEM/COMPONENT J-2 Oxidizer Bleed Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
13. <u>Valve Fails to Perform</u> b. Contamination/Friction Valve failed to close during tank purges following test. Caused by broken poppet retaining bolt due to excessive pressure buildup from trapped oxidizer in inner bellows. Design change initiated for control of this problem.	5f 31%		No significant effect in this case except for some continued flow of oxidizer. Subsequent engine operation, if not detected, could result in reductions in thrust, M/R, and GG temperature, with possibility of premature engine shut-down by dropout of main-stage O.K. pressure switch.	Structural and overpressure	Primary	5 or potential 3 restart only	N/A	Post-test inspection		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT J-2/P.U. Valve (MRC Valve-2 Positions)

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Valve Fails to Perform Continued Performance degradation Cause of performance shift could not be determined by test. Could be caused by corrosion resulting in binding of internal parts.	1f .062%		Resulted in shift in engine performance of approximately 2400 lbs thrust and 0.05 M/R units.	Corrosion	Primary	4	Inst.	Test instrumentation.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT J-2/Oxidizer Turbine Bypass Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage a. Contamination Slow-closing valve does not close completely. Caused by galling of gate rings and gate housing which increased valve friction force during last portion of valve travel. Some galling also present in area of drive shaft and retainer. Metal contamination from turbine exhaust gas may also have been a contributing factor. Redesign has been initiated to reduce friction and prevent galling to improve actuation characteristics.	24f 1.5%		All of these occurrences were detected during pre/post test inspection and checkout procedures and corrective action taken. Failure of valve to close during start sequence if not detected and corrected would result in premature engine cutoff at expiration of sparks deenergized timer since low oxidizer pressure would prevent injection pressure switch pickup.	Material and interference	Primary	5 or potential 3	N/A	Pre/post test inspection and checkout procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Ultrasonic Thermometer Accelerometers Isotope Detector Hydrophone Tunable Diode Laser Spectrometer	Ultrasonic Leak Isotope Tracers Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography Optical Proximity Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Pressure Decay	

ENGINE SYSTEM/COMPONENT J-2 Pressure-Acutated Purge Control Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued Internal leakage past valve seat Caused by extensively damaged inlet seal due to contaminants, most likely self-generated during test, which resulted in leakage past seal to valve vent port. Corrective design measures have been initiated for control of this problem.	11f 168%		These failures were all detected during pre/post checkout procedures and corrective action taken. Prolonged or extensive leakage of this nature (3 of these failures indicated gross leakage) could result in sufficient loss of helium to preclude restart of engine.	Contamination and galling	Primary	5 or potential 3	N/A	Pre/post test checkout procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT J-2/Various Assemblies - Check Valves

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued Reverse flow internal leakage Caused by fretting of poppet and galling between poppet and seat assembly augmented by presence of contamination in poppet/seat area. This results in partially open position during engine operation. Seal damage was also a contributing factor to some of these failures. Design modifications to valve have been initiated for control of this problem.	19f 1.182%		These occurrences were all detected during pre/post test procedures and remedial action taken. Failure analyses note that any reverse leakage through valve would be vented through purge control vent line, which provides a certain amount of redundancy for this failure mode. Prolonged or gross leakages (some of these were noted as beyond the range of the flow rater), however, could result in sufficient loss of helium and/or oxidizer/fuel to result in	Interference	Primary	5 or potential 3	N/A	Pre/post test inspection.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT J-2/Various Assemblies - Check Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued			degradation of GG/TC and consequently engine performance with the possibility of premature engine shutdown. Extensive loss of helium pressurant could also result in failure to restart engine when required.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT J-2/G.G. Control Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued Internal leakage past poppet/seat Caused by damage to poppet seal from reverse flow past the fuel poppet at engine start - damage to plastic seat by tie-wrap wedged between poppet and seat.	4f .249%		These incidents were detected pre/post test and remedial action taken. Undetected leakage would result in unplanned delivery of one or both propellants to GG with possible detonation at start and damage to gas generator and/or turbines	Material	Primary	5 or potential 2	N/A	Pre/post test inspection.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

FAILURE SUMMARY SHEETS

H-1 ENGINE DATA

C・1

ENGINE SYSTEM/COMPONENT H-1 Thrust Chamber

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
2. <u>Coolant Passage Leakage</u> Due to cracks or ruptures (mostly pin-hole type) in tubes, caused by overheating as the result of damage to tubes or disturbance of the exhaust stream from irregularities in T/C wall. Overheating of tubes has also been caused by restriction of free coolant flow by flush-mounted photocon bosses on the T/C. Also caused by cracks in exit manifold braze joint or in tubes just upstream of braze joint due to overheating - one cause of the overheating in this area is the	38F		All of these failures were detected during post-test checkout and inspection procedures and corrective action taken. External fuel leakage during engine operation, however, would always present the problem of the possibility of fire in the presence of an ignition source, with consequent possibility of severe damage to engine	Overtemp, Structural Material	Primary	5 potential 2, 3	N/A	Post-test checkout and inspection		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Ultrasonic Thermometer (Flame) Ultrasonic Flowmeter (Nozzle) Polarometer Tunable Diode Laser Spectrometer (Mixture Ratio)	Ultrasonic Leak Acoustic Holography X-ray Radiography Gamma Radiography Pentoxide Polarometry Hydrogen Polarometry Hygrometer Optical Pyrometry Holographic Leak Millimeter-wave Interferometry	

ENGINE SYSTEM/COMPONENT H-1/Thrust Chamber

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Coolant Passage Leakage Continued extension of the manifold into the hot gas stream. H-1 thrust chamber design change has been modified so that exit manifold will be moved further outboard from the main flame stream to correct this latter problem.										

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT H-1/ Turbopump-Thrust Chamber-Gas Generator Assy.

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage a. Hot Gas Caused by thermocouple blown out of boss; cracks in GG combustor body and turbine manifold; defective damaged or broken seals/gaskets; undertorqued bolts or relaxation of torque on bolts during engine operation. The majority of these failures (40 of the 59) were leakage past the seal between the G.G. and turbine assembly flanges. Suggestion made to incorporate Naflex seals in joints using spiral-would (flexitallic) gaskets.	59F 2F		Two of these failures resulted in premature termination of test by observer due to fire in turbine area and hot gas leakage at G.G. Nearly all these failures were detected and corrective action taken during pre/post test procedures. Hot gas leakage during engine operation or flight can result in fire in engine area with consequent possibility of explosion and/or major damage to engine and/or vehicle.	Structural Material Torque relaxation	Primary	3, 5 potential 1, 2	Inst.	Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Extensiometer Ultrasonic Leak Leak Tape/Coating Optical Leak* Laser Interferometry Differential Radiometry Holographic Leak Resistivity Monitoring Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Leak Fluid Pressure Decay	

ENGINE SYSTEM/COMPONENT H-1/Gas Generator & Propellant Feed System Seals, Fittings, Lines & Ducts-T/C Dome

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. <u>Joint Leakage</u> b. Propellant and Lube External fuel leakage caused by damaged or defective seals/flanges, undertorqued bolts or relaxation of torque on bolts during engine operation, loose or damaged bolts/B-nuts, cracked or damaged lines/ducts, braze and/or weld porosity	18f 4f		*Three of these failures* resulted in premature termination of tests by observers, two for fuel leakage and one for fire *noted. In two other* instances post-test inspection revealed a fire had occurred during main-stage firing tests. The *other 13 failures were* detected during pre/post test inspection & checkout and remedial action taken. The end result of fuel leakage *can always be fire* in the presence of an ignition source, with possible explosion and/or substantial engine damage.	Structural Material Torque relaxation	Primary	3, 5 potential 1, 2	Inst.	Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT H-1/Gas Generator & Propellant Feed System Seals, Fittings, Lines & Ducts-T/C Demo

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage (Continued) External Lox leakage caused by damaged, defective or contaminated seals and/or sealing surfaces, loose or undertorqued bolts/fittings, or relaxation of torque during engine operation, cracked lines/ducts, faulty braze joints.	25f .017%		All of these failures were detected during pre/post test checkout and inspection procedures and corrective action taken to return components to acceptable condition. Lox leakage into the engine compartment contributes to the possibility of substantial engine damage if fire is present, or could result in freezing of adjacent components with resultant failure of these components to operate as required.	Structural Material Torque relaxation	Primary	5 potential 2, 3	N/A	Pre/post test checkout and inspection.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued External Lox leakage caused by damaged, defective or contaminated seals and/or sealing surfaces, loose or undertorqued bolts/fittings, or relaxation of torque during engine operation, cracked lines/ducts, faulty braze joints.	25f		All of these failures were detected during pre/post test checkout and inspection procedures and corrective action taken to return components to acceptable condition. Lox leakage into the engine compartment contributes to the possibility of substantial engine damage if fire is present, or could result in freezing of adjacent components with resultant failure of these components to operate as required.	Structural Material Torque relaxation	Primary	5 potential 2, 3	N/A	Pre/post test checkout and inspection.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

C-7

ENGINE SYSTEM/COMPONENT H-1 Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
6. Cracked Turbine Blades Due to excessive Lox lead from GG caused by delayed opening of fuel poppet or by premature opening or leakage of Lox poppet, with consequent excessively high temperature and resultant erosion. Also can be caused by impact of foreign objects on the blades, or by rubbing of blades.	24f 3f		Three of these failures resulted in premature test cutoff by RCC device and by failure to achieve bootstrap. Other 21 failures were detected during pre/post test inspection and checkout, and corrective action taken. Engine operation with the level of erosion noted in most of these failures would prevent engine from attaining mainstage operation or, if propagated during engine operation, could result in severe damage to pump and probably to engine.	Overtemp Erosion Contamination	Primary	3, 5 potential 2	Imm. Inst.	RCC cutoff device. Observer cutoff.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Sensors Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Pyrometer Vibration Hydrophone Fiberoptic Bearing Detector Exo-electron Detector	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Remnant Magnetization Optical Holography Borescoping Exo-electron Emission Positron Annihilation Electric Current Injection Eddy Current	

C S

ENGINE SYSTEM/COMPONENT H-1/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
9. Bearing Damage Due to rubbing of rollers and inner/outer race with resultant scoring and eventual binding of bearings; contamination in bearings or lack, or restriction of lube to bearings could result in same failure of bearings to function.	8f 4f		Four of these failures resulted in premature test cutoff by observer and by failure to achieve mainstage. One of the four was a launch engine cutoff 116.8 Sec after liftoff. The other 4 failures were detected during post-test analysis and checkout. Continued operation with damaged or failed bearings could result in severe damage and possibly explosion of pump, with resultant major damage to engine.	Interference (Contamination)	Primary	3, 5 potential 1, 2	Imm.	M/S O.K. monitor Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Optical Tachometer Isotope Detector Fiberoptic Detector RTD Thermometer Accelerometer Hydrophone Ferromagnetic Torquemeter Exo-electron Detector Tunable Diodelaser Spectrometer	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Exo-electron Emission Positron Annihilation Eddy Current Torquing	

ENGINE SYSTEM/COMPONENT H 1/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
11. Turbopump Seal Leakage Internal leakage past primary fuel seal	28f		These failures were detected pre-test and corrective action taken. This leakage can be discharged from the lube overboard drain line and in the presence of an ignition source could result in fire and/or explosion, or substantial engine damage.	Material Contamination	Primary	5 potential 1, 2	N/A	Pre-test checkout		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
F-11 RTD Thermometer Optical Tachometer Accelerometers Isotope Seal Detector Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

ENGINE SYSTEM/COMPONENT H-1/Lube Oil Filter

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
12. Lube Pressure Anomalies Clogged filter caused by contamination of the filter element by foreign material in lube oil system to the extent that flow through the filter is substantially impaired.	37f		One of these failures resulted in premature termination of test due to lube pressure dropping below redline value. The others were detected during post test inspection and corrective action taken.	Contamination	Primary	3, 5	Imm.	Lube oil redline monitor cutoff device		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

C-11

ENGINE SYSTEM/COMPONENT H-1/Main Oxidizer Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
13. Valve Fails to Perform b. Contamination/Friction Fast, slow or erratic opening/closing time caused by galling of a actuator housing bore and piston, variations in spring constant and/or seal friction, cracked or damaged lip sealinterfering with valve movement, bearing malfunction, heater failure.	13f		No premature cutoffs resulted from these failures; however, one test resulted in destruction of thrust chamber and injector in a post-cutoff Lox fire. Other failures were detected during checkout and post-test investigation of performance data and corrective action taken. The occurrence of this failure during engine start could result in the initiation of performance anomalies with probable test cutoff and possible engine damage. Leakage	Structural Contamination Temperature Interference	Primary	4, 5 potential 2, 3	N/A	Post-test observation and checkout.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressor Sensors Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Isotope Wear Spectrometer Tunable Diode Laser Spectrometer	Ultrasonic Leak Acoustic Holography Isotope Tracers Pentoxide Polarometry Hygrometer Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography	

ENGINE SYSTEM/COMPONENT H-1/Main Oxidizer Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Valve Fails to Perform Continued			at test cutoff due to slow closing could allow external Lox leakage from T/C with resultant possibility of damage to engine.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT H-1 Check Valves

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Valve Fails to Perform Continued Fails to close-reverse leakage caused primarily by contamination lodged between poppet and seat preventing valve from closing properly; damaged or scratched poppet/seat; binding or sticking of valve poppet due to buildup of contamination on stem and/or bore, or to dimensional anomalies.	13f		Eight of these failures were on the gearcase pressure check valve, the others were on the lube drain check valve. All of these failures were detected during checkout and inspection procedures and corrective action taken. Occurrences of these failures during engine operation could result in reduction or depletion of gearcase pressure with consequent probability of test cutoff and possibility of damage to gearcase/pump.	Contamination interference	Primary	5 potential 2, 3	N/A	Checkout and inspection		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

C-14

ENGINE SYSTEM/COMPONENT H-1/Main Fuel Valve/Check Valves

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage Fuel leakage caused by severely damaged shaft seals; scratched and contaminated sealing surfaces of gate and lip seal.	29f		All of these failures were detected during pre/post test inspection and checkout procedures, and corrective action taken. The occurrence of this failure during engine start could result in fuel leak to T/C with possible detonation when oxidizer enters T/C. Leakage at test cutoff could result in fuel leakage from T/C with possibility of fire and consequent engine damage.	Material Contamination	Primary	5 potential 2, 3	N/A	Pre/post test inspection and checkout.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Ultrasonic Thermometer Accelerometers Isotope Detector Hydrophone Tunable Diode Laser Spectrometer	Ultrasonic Leak Isotope Tracers Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography Optical Proximity Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Pressure Decay	

FAILURE SUMMARY SHEETS

F-1 ENGINE DATA

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ENGINE SYSTEM/COMPONENT F-1/Thrust Chamber

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
2. Coolant Passage Leakage Internal fuel leakage caused by braze bond defects augmented by thermal and vibration stresses.	6f		These failures were all detected during pre/post test procedures and corrective action taken. Engine operation with significant internal fuel leakage, however, could result in reduction of return fuel flow to T/C injector and impingement of fuel into combustion flow through the T/C nozzle. If these conditions were severe enough, thrust output could be affected and premature engine cutoff triggered.	Structural Vibration Temperature	Primary	5 potential 3	N/A	Pre/post test procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Ultrasonic Thermometer (Flame) Ultrasonic Flowmeter (Nozzle) Polarometer Tunable Diode Laser Spectrometer (Mixture Ratio)	Ultrasonic Leak Acoustic Holography X-ray Radiography Gamma Radiography Pentoxide Polarometry Hydrogen Polarometry Hygrometer Optical Pyrometry Holographic Leak Millimeter-wave Interferometry	

ENGINE SYSTEM/COMPONENT F-1/Thrust Chamber & Extension Nozzle

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage a. Hot Gas External hot gas leakage caused by warped/damaged T/C-to-nozzle extension flanges due primarily to welding heat during installation of turbine exhaust manifold. Also due to damaged seals. New seal design is incorporated at the T/C-to-nozzle extension flanges precluding leakage if minor flange warpage exists.	22f		All of these failures were detected during pre/post test procedures and corrective action taken. Leakage of hot gas into engine/vehicle compartment during engine operation presents the possibility of severe fire and/or explosion or at best, damage to adjacent hardware.	Structural Material Torque relaxation	Primary	3, 5 potential 1		Post test checkout and inspection.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Extensiometer Ultrasonic Leak Leak Tape/Coating Optical Leak Laser Interferometry Differential Radiometry Holographic Leak Resistivity Monitoring Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Leak Fluid Pressure Decay	

ENGINE SYSTEM/COMPONENT F-1/Thrust Chamber, Seals, Adapters, Disconnects & Fuel Ducts, Lines and Fittings

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage b. Propellant and Lube External Fuel Leakage Cause of the leakage varies dependent on component and location in engine. For thrust chamber external fuel leaks, the primary causes were cut/damaged packing/seals/fittings, and erosion/holes in T/C tubes resulting from contaminant on tube surface during furnace brazing. For seals, the main causes were low torque on fasteners, imperfections in mating surfaces of flanges, contaminant between	43f		All of these failures were detected pre/post test, prelaunch, or during leak tests (*mainly thrust chamber leaks*). Corrective action was taken in all cases prior to subsequent operation. Engine operation with these conditions existent, however, would have varied effects dependent on component and location in engine. For thrust chambers, significant external leakage of fuel could affect propellant mixture ratio with consequent Lox-rich	Material Erosion Under torque *Relaxation* Structural Contamination	Primary	5 potential 2, 3	N/A	Prelaunch Pre/post test checkout and inspection procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

D-4

ENGINE SYSTEM/COMPONENT F-1/Thrust Chamber, Seals, Adapters, Disconnects & Fuel Ducts, Lines and Fittings

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued seal and flange, and/or damage to rubber sealing surfaces. For adapters, leakage was due to weld failure, and the most prevalent cause was the use of soft copper gaskets which allowed torque relaxation on fasteners following hot fire and, if fasteners were not retorqued prior to subsequent operation, leakage resulted. Welded in place adapters were adoped on later engines to eliminate the problem of soft copper gaskets.			burning in thrust chamber and possible damage from excessive temperatures. Also, external fuel leakage from thrust chamber would have a common effect with leakage from any of the other components. In that leakage of fuel presents a fire hazard in the presence of an ignition source, with resultant fire which could cause damage to engine and/or other engine components, the severity dependent on the magnitude of the leak and the location in the							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT E-1/Thrust Chamber, Seals, Adapters, Disconnects & Fuel Ducts, Lines and Fittings

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued For disconnects (engine half only), main causes of leakage were deformation of seat and/or poppet, and contamination between poppet and seat. For fuel ducts, lines, and fittings, the primary causes of leakage are undertorqued B-nuts, material defects resulting from casting and processing deficiencies, cracks resulting from fatigue failure.			engine or in the vehicle compartment. One other effect, where leakage is from lines directing hydraulic (fuel) pressure for operation of components, would be the possibility of sufficient loss of pressure to result in failure of these components to operate, with consequent premature cutoff of engine operation or damage to engine and/or components.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT F-1/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
11. T/P Seal Leakage Caused by discrepant primary fuel seal internal O-ring resulting in lack of proper O-ring squeeze and consequently low pressure seating capability.	2f		Failures detected during leak test and corrective action taken. Engine operation with this condition, however, could result in reduction of fuel flow from pump with consequent imbalance of propellant ratios and effect on engine operation with possible damage to engine. This fuel leakage is directed to fuel drain manifold and then to overboard fuel drain. This presents a possible fire hazard with consequent possibility of damage to engine and/or other components.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
RTD Thermometer Optical Tachometer Accelerometers Isotope Seal Detector Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

V-7

ENGINE SYSTEM/COMPONENT F-1 Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
12. Lube Pressure Anomalies Lox Pump Bearing Jet Pressure Excessive Caused by restriction of one or more bearing jet holes with consequent reduction in bearing lube flowrate and increase in pressure as noted. Restriction was due to contamination clogging jet holes. Special cleaning procedures have been instigated for engines F2060 and subs. to control the incidence of contamination in this area	4f		One instance resulted in premature test cutoff when Lox pump bearing jet pressure exceeded the maximum redline value. Others detected during review of test data and corrective action taken. Propagation of this condition could result in lack of lube to bearings with resultant damage to bearings and most likely to pump. Potential engine shutdown.	Restricted flow Contamination	Primary	3, 5 potential 2	Imm.	Lox pump bearing jet pressure redline cutoff device.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT F-1/Turbopump Bearing Coolant Valve/Main Oxidizer Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
13. Valve Fails to Perform b. Contamination/Friction Failure to open at required applied cracking pressure caused by multiple striations on the poppet assembly and the presence of fine-particle contamination on poppet and guide.	4f		These failures were detected and corrective action taken prior to launch. Engine operation with closed valve, however, would result in loss of lube to bearings with almost certain failure of bearings and seizure or excessive binding of pump. Damage to pump and most likely to other engine areas would ensue.	Binding Contamination	Primary	5 potential 2,3	N/A	Pre-launch checkout procedures.		
Hydraulic Pressurant (Fuel) Leakage from Open Sequence Valve Area. Leakage could be caused by damaged parts/seals,	6f		All of these failures were detected during prelaunch checkout and/or leak tests and corrective action taken. Sufficient leakage							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressor Sensors Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Isotope Wear Spectrometer Tunable Diode Laser Spectrometer	Ultrasonic Leak Acoustic Holography Isotope Tracers Pentoxide Polarometry Hygrometer Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography	

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ENGINE SYSTEM/COMPONENT F-1/Turbopump Bearing Coolant Valve/Main Oxidizer Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Hydraulic Pressurant (Fuel) Leakage from Open Sequence Valve Area Continued dimensional discrepancies, contamination, or improper seal and retainer installation. Wrong size O-ring seals is also a contributing factor to this leakage.			in this area, however, could result in failure to transmit adequate level of hydraulic pressurant to GG control valve for operation when sequence valve is actuated. Failure of GG to actuate due to this lack of hydraulic pressure would preclude further engine premature cutoff. External fuel leakage also presents a fire hazard in the presence of an ignition source with possibility of damage to engine.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

D-10

ENGINE SYSTEM/COMPONENT F-1/Main Oxidizer Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage b. Compression of Spring Internal oxidizer leakage past the poppet caused by a loose poppet skirt seal assembly, and by a permanently distorted compressor ring on the poppet skirt seal. On later engines, a poppet rotation test was added to the valve drawing and a vented seal retainer was incorporated to control this problem.	9f		These incidences were detected pre/post test and corrective action taken. Leakage of oxidizer past the poppet/seat, however, could result in an accumulation of oxidizer in the thrust chamber with consequent possibility of detonation or severe fire at time of ignition.	Torque relaxation Material	Primary	5 potential 2, 3	N/A	Pre/post test procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Ultrasonic Thermometer Accelerometers Isotope Detector Hydrophone Tunable Diode Laser Spectrometer	Ultrasonic Leak Isotope Tracers Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography Optical Proximity Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Pressure Decay	

ENGINE SYSTEM/COMPONENT F-1/Main Fuel Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage c. Vibration-Seat Internal fuel leakage due to seat misalignment caused by particle in area between the seat and the seat retainer. Engine vibration and fuel flow aggravated the original condition to allow seal and seat contact to become marginal.	6f		This failure mode was detected during leak check and corrective action taken. Leakage of fuel past the seat, however, could result in the accumulation of fuel in the thrust chamber area with possibility of severe fire at ignition and damage to thrust chamber and/or engine.	Plastic deformation Vibration	Primary	5 potential 2, 3	N/A	Pre-test leak check procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

FAILURE SUMMARY SHEETS

RS-27 ENGINE DATA

E-1

ENGINE SYSTEM/COMPONENT RS-27/Gas Generator, Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage a. Hot Gas Caused by damaged/defective seals, gaskets and/or sealing surfaces, under-torqued bolts or torque relaxation on bolts.	28f		All of these failures were detected during launch preparation and/or other engine checkout procedures and corrective action taken. Hot-gas leakage during engine operation always presents the possibility of severe fire and/or explosion, or at best, damage to adjacent hardware.	Torque relaxation Under torque Material	Primary	5 potential 1, 2, 3	N/A	Engine checkout procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Extensiometer Ultrasonic Leak Leak Tape/Coating Optical Leak Laser Interferometry Differential Radiometry Holographic Leak Resistivity Monitoring Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Leak Fluid Pressure Decay	

ENGINE SYSTEM/COMPONENT RS-27 Thrust Chamber, Turbopump, Main Fuel Valve, Fuel Start Tank, Fuel B/S Check Valve, Propellant Feed System Lines, Fittings, Flanges & Connections

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage b. Propellant & Lube External Fuel Leakage These failures were due to various causes, dependent on the component and location in the engine. The primary causes can be grouped as follows: Thrust Chamber - Top O-ring at T/C-to-injector interface found cut in two places. Turbopump - Scratched flange on inlet elbow, porosity through parent metal at plug in fuel inlet elbow.	20f		All of these failures were detected during pre-launch and other engine checkout procedures and corrective action taken. Engine operation with external fuel leakage, however, presents a fire hazard in the presence of an ignition source, with resultant fire which could cause damage to the engine and/or other engine components, the severity dependent on the magnitude of the leak and the location in the engine or in the vehicle compartment.	Material damage Material Contamination Torque relaxation Under torque	Primary	5 potential 2, 3	N/A	Engine checkout procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT RS-27/Main Oxidizer Valve, Start System & Propellant Feed System - Fittings & Connections.

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage Continued External Oxidizer Leakage Failures were due to scratched/damaged/defective seals, packing and/or sealing surface, or to under torque or torque relaxation on bolts and/or fittings.	12f		All of these failures were detected during engine checkout procedures and corrective action taken. External oxidizer leakage during engine operation could result in possible fire hazard and damage to engine and/or other engine components. Possible damage to adjacent hardware also exists due to the extremely low temperature of the leaking oxidizer.	Torque relaxation Under torque Material damage	Primary	5 potential 2, 3	N/A	Engine checkout procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

E-4

ENGINE SYSTEM/COMPONENT Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
11. Turbopump Seal Leakage Caused by low or relaxed seal bolt torques, damaged seals and/or mating ring surfaces, seal ring not seated properly, foreign material preventing carbon nose seating flush, could also result from initial pressure surge and/or start transients displacing carbon nose.	12f		Failures detected pre/post and corrective action was taken. In addition to the serious engine/vehicle damage, which could occur due to excessive leakage, adjacent hardware could also be harmed.	Torque relaxation Contamination Material damage	Primary	5 potential 2	N/A	Engine checkout procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
RTD Thermometer Optical Tachometer Accelerometers Isotope Seal Detector Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

ENGINE SYSTEM/COMPONENT RS-27 Turbopump, FABU Quick-Disconnect

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
12. Lube Discrepancies External lube leakage caused by under torque or torque relaxation on B-nut, and by damaged seal/sealing surface.	2f		Both of these failures were detected during engine checkout procedures and corrective action taken. Since the lube for the RS-27 is a mixture of fuel and a lubricant additive (contained in the FABU) the effects of external lube leakage would be the same as for fuel, with the possibility of fire in the presence of an ignition source and consequent possibility of damage to engine and/or other engine components.	Torque relaxation Material damage	Primary	5 potential 2, 3	N/A	Engine checkout procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT RS-27 Main Oxidizer Valve, Start System and Propellant Feed System - Fittings & Connections

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage a. Contamination Internal Oxidizer Leakage Six of the failures were MOV or fuel lip seal leakage due to localized gate seal lip wear resulting from rough finish condition on gate seal and/or undersize or eccentric lip seal I.D. The other failures occurred on drain quick disconnect valves as a result of contamination trapped between poppet and seat.	8f		All of these failures were detected during engine checkout procedures and corrective action taken. Internal oxidizer or fuel leakage during engine operation could result in possible fire hazard and damage to the engine and/or other engine components. Possible damage to adjacent hardware also exists due to the extremely low temperature of the leaking oxidizer.	Torque relaxation Under torque Material damage	Primary	5 potential 2, 3	N/A	Engine checkout procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Ultrasonic Thermometer Accelerometers Isotope Detector Hydrophone Tunable Diode Laser Spectrometer	Ultrasonic Leak Isotope Tracers Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography Optical Proximity Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Pressure Decay	

E-7

ENGINE SYSTEM/COMPONENT

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage c. Vibration Seat Leakage Past Oxidizer Poppet and/or Bellows Leakage past bellows (at upper Lox bellows end piece and bellows seal weld joint) caused by localized poor quality weld weakened by engine vibration and/or thermal shock. Leakage past poppet due to poppet hung open from prior test due to galling of Lox poppet and mating body bore, probably caused by metallic contamination lodging between poppet and bore.	2f		Both of these failures were detected during pre-test and/or other engine checkout procedures, and corrective action taken. Engine operation at these conditions could result in excessive oxidizer buildup in system prior to ignition and entry of fuel, and/or possible combination of Lox and fuel in GG control valve housing, either of which could cause severe damage to GG and/or engine.	Material Vibration Galling Contamination	Primary	5 potential 2	N/A	Engine checkout procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT RS-27 Isolation Check Valves

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage d. Trapped Pressure Reverse Leakage Through Redundant Check Valves Due to the effect of surface damage caused by, and the unseating tendency characteristics of, contaminants embedded in the Teflon O-ring seals of both valves. Damage most likely from self-generated fretting wear due to chattering or unstable operation during low-flow periods. In some instances it is believed that the failure was caused by low	11f		All of these failures were detected during pre-launch countdown/checkout and/or other checkout procedures, and corrective action taken. The effects of this failure can be quite variable, dependent on the magnitude of the leakage and the time of occurrence.	Contamination Dynamic	Primary	5 potential 1, 2, 3	N/A	Engine checkout procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

E-9

ENGINE SYSTEM/COMPONENT RS-27 Isolation Check Valves

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued pressure sealed between the poppet and the spring retainer during the previous operation, keeping the spring compressed and allowing reverse leakage. Transpiration of fuel vapor accross the check valve poppet from trapped fuel is believed to have been also noted in some instances due to exposure to fuel for extended periods of time.										

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

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ENGINE SYSTEM/COMPONENT RS-27 Pneumatic Regulator

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
15. Regulator Discrepancies Regulator-out Pressure High/Erratic Caused by contaminant (particles) trapped between the ball and retainer seat of the loader assembly (probably due to inadequate cleaning and contamination control) and/or by slivers temporarily wedged between loader seat and bore of housing retarding piston movement during dynamic operation. One instance was the result of a combination of a discrepant (oversize) piston and under-torqued screw and probe of the regulator valve creating a leak path around the packing and thru the threads.	5f		One of these failures resulted in premature test cutoff when regulator pressure spiked in excess of redline cutoff. The other four instances were detected during pre-test pneumatic control system checkouts and corrective action taken. It is unlikely that any consequence would occur as the result of this type of failure of a more critical nature than engine cutoff.	Contamination Dimensional	Primary	3, 5 potential 3	Inst.	Chart observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Tunable Diode Laser Spectrometer Isotope Wear Detector	Ultrasonic Leak Particle Analysis Optical Leak Differential Radiometry Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Pressure Decay	

FAILURE SUMMARY SHEETS

THOR ENGINE DATA

F-1

ENGINE SYSTEM/COMPONENT Thor/Thrust Chamber Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
2. <u>Coolant Passage Leakage</u> Leakage caused by tube ruptures as result of localized overheating, detonation and/or insufficient braze penetration at tube-to-end ring joint, intergranular corrosion and embrittlement due to the presence of high sulphur compounds in combination with high operating temperatures with resultant tube cracks, splits and pinholes.	76f 7.9%		Localized fire in one case, on outside of thrust chamber causing premature cutoff and leakages detected during pre/post test checkout procedures. Engine operation with external fuel leakage presents a fire hazard and possible decrease in performance. The magnitude of the leak will determine the severity of the performance loss and the damage to engine hardware.	Material Structural Stress corrosion High temperature	Primary	3 potential 2	Inst.	Observer cutoff, Pre/post test checkout procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Ultrasonic Thermometer (Flame) Ultrasonic Flowmeter (Nozzle) Polarometer Tunable Diode Laser Spectrometer (Mixture Ratio)	Ultrasonic Leak Acoustic Holography X-ray Radiography Gamma Radiography Pentoxide Polarometry Hydrogen Polarometry Hygrometer Optical Pyrometry Holographic Leak Millimeter-wave Interferometry	

ENGINE SYSTEM/COMPONENT Thor/Thrust Chamber Assembly/Gas Generator Assy

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage a. Hot Gas Thrust Chamber leakage due to damaged or discrepant gasket and/or port, insufficient braze alloy penetration at exit ring-to-tube joint resulting in subsequent cracks or tube separations. Gas generator leakage is caused by damaged seals, gaskets or flanges, torque relaxation of bolts.	27f 2.81%		One of these failures (on the thrust chamber) resulted in premature engine termination, while the balance was detected during pre/post test checkout procedures. Hot gas leakage always presents the possibility of severe fire and/or explosion hazard and damage to adjacent hardware.	Material Structural High temperature	Primary	3 potential 1, 2	Inst.	Observer cutoff. Pre/post test checkout and inspection.		

VIABLE IN FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Extensiometer Ultrasonic Leak Leak Tape/Coating Optical Leak Laser Interferometry Differential Radiometry Holographic Leak Resistivity Monitoring Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Leak Fluid Pressure Decay	

F-3

ENGINE SYSTEM/COMPONENT Thor/Various Engine Subsystems, Lines, Fittings & Seals

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage b. Propellant & Lube External Oxidizer Leakage These failures were due to various causes, dependent on the component and location in the engine. The primary causes can be grouped as follows: Thrust Chamber - Inner dome and inlet elbow-to-dome bolts undertorqued, discrepant washers preventing proper sealing at Lox dome inner bolts, inadequate finish on sealing surface of inner dome bolts.	43f 4.47%		Five of these failures resulted in premature termination of test by front bunker observer when significant Lox leakage occurred. Two instances were in the Lox high pressure duct area, two in the area of the main Lox valve, and the other in the T/C Lox dome area. In addition, one test was delayed due to Lox leakage at the Lox start tank vent valve. A seventh instance did not result in a premature cutoff, but 14 seconds after planned engine cutoff a Lox-rich	Low torque Material Contamination Fatigue Structural	Primary	3 potential 2	Inst.	Observer cutoff. Check out and inspection.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

F - 4

ENGINE SYSTEM/COMPONENT Thor/Various Engine Subsystems, Lines, Fittings & Seals

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued Turbopump - Excess removal and uneven build-up of dry film lube between inlet adapter bolt heads and wasners resulting in inadequate sealing of O-ring due to marred surface and non-uniform condition at sealing surface, inlet adapter-to-elbow gasket damaged/defective, under-torqued bolts. G.G. Blade Valve - Damaged seal and/or sealing surface of Lox-side valve cover plate.			fire was noted in the turbine exhuast duct area. The remaining failures were detected pre/during/post test and during other checkout and test procedures and corrective action taken. External oxidizer leakage during engine operation could result in possible fire hazard and damge to engine and/or other engine components.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

F. 5

ENGINE SYSTEM/COMPONENT Ther/Various Engine Subsystems, Lines, Fittings & Seals

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued Start Tank - Undertorqued fittings and/or bolts, O-ring damaged or improperly installed, damaged packing/improper lube application at fill head-to-adapter connection, defective sealing surface and/or packing at tank head-to-orifice fitting, defective gasket/flange damage between body and tank head, scratched seal/sealing surface at bottom cap-to-body mating surfaces, broken lip seal at vent port seal.										

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

F- 6

ENGINE SYSTEM/COMPONENT Joint Leakage Continued

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued Main Oxidizer Valve - Cracked/split seal at valve inlet and/or outlet contamination at valve shaft seals allowing leakage between housings. Also caused by cracked/split/damaged seals and/or gaskets, undertorqued bolts/B-nuts, scratched/damaged sealing surfaces.										

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

F-7

ENGINE SYSTEM/COMPONENT Thor/Lox Start Tank Bootstrap Area, Lines, Fittings & Check Valves

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
External Oxidizer Leakage Continued Caused by casting flaw in surface of fitting, under-torque relaxation during operation on fittings/bolts/B-nuts, excessive application of Lox lube on O-rings/packing, faulty O-ring or improper installation at flex hose-to-manifold connection, scratched/damaged seals and/or sealing surfaces, use of rubber O-ring and a standard boss at upstream end of check valve, misalignment of boss centerline.	121f 12.58%		These failures are peculiar to the design of the Lox bootstrap system and may not be applicable to the engine under evaluation, but are included due to the unprecedented number of failures and the potential consequences involved. All of these failures were detected during pre/post test and other checkout and inspection procedures and corrective action taken. External leakage of oxidizer during engine operation, however, always presents the possibility of fire hazard and damage	Material Torque relaxation Dimensional	Primary	5 potential 2, 3	N/A	Checkout and inspection procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Thor/Lox Start Tank Bootstrap Area, Lines, Fittings & Check Valves/Turbopump Assy.

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
External Oxidizer Leakage Continued External Lube Leakage Caused by scratches/slight indentation across sealing face of crush washer, snap-ring dimensional error resulting in seal leakage, scratched sealing surface, damaged gasket.	 8f 0.832%		to engine and/or other components. All of these failures were detected during pre/post test procedures and corrective action taken. Since the lube for the Thor pump gears and bearings is a mixture of fuel and a lubricant additive (contained in the FABU), the effects of external lube leakage would be the same as for fuel with the possibility of fire in the presence of an ignition source and consequent possibility of damage to engine and/or other engine components.	Material Dimensional	Primary	5 potential 2, 3	N/A	Pre/post test procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Thor/Various Engine Subsystems, Lines, Fittings & Seals

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage b. Propellant & Lube External Fuel Leakage These failures were due to various causes, dependent on the component and location in the engine. The primary causes can be grouped as follows: Thrust Chamber - O-ring at thrust chamber-to-injector joint out of groove for a distance of 12" due to improper installation procedure, damaged/defective O-ring. Turbopump & Main Fuel Valve- Defective/damaged seals.	47f 4.886%		Two of these failures resulted in premature test cutoff by observer, one due to fire on outside of thrust chamber and the other to substantial fuel leakage at the injector-to thrust chamber interface. The other 45 failures (the majority of which were in the thrust chamber area) were detected during pre/post test inspection and checkout procedures and corrective action taken. Engine operation with external fuel leakage, however, presents a fire hazard in the presence of	Material Structural Torque relaxation	Primary	3 potential 2	Inst.	Observer cutoff, checkout and inspection		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

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ENGINE SYSTEM/COMPONENT Thor/Various Engine Subsystems, Lines, Fittings & Seals.

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued Lines, Fittings & Seals - Several puntures by sharp ends of damaged braid in inner surface of flex hose, improper mating of flex hose in nipple. Also caused by low torque or relaxation of torque during operation on bolts/B-nuts, damaged/defective seals and/or gaskets.			an ignition source, with resultant fire which could cause damage to engine and/or other engine components, the severity dependent on the magnitude of the leak and the location in the engine or in the vehicle compartment.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Thor/Turbopump Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
5. High Torque, T/P Caused by binding of seal carbon due to combustion products in area, rubbing of labyrinth seal, slight shifting of 2nd stage nozzle during operation due to undertorque or torque relaxation on nozzle retaining screws.	11f 1.143%		All of these failures were detected during pre/post test inspection and checkout procedures and corrective action taken. Engine operation at these conditions could result in propagation of the problem to a stage where reduced pump output could affect thrust output or other monitored parameters with resultant premature termination of engine operation.	Binding Torque relaxation Dynamic	Primary	5 potential 3	N/A	Pre/post test inspection and checkout procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
RTD Thermometer Optical Tachometer Accelerometer Isotope Wear Detector Hydrophone Ferromagnetic Torquemeter Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

ENGINE SYSTEM/COMPONENT Thor/Oxidizer High Pressure Duct

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
7. Crack, Convolutions Bellows Cracks in convolutions of oxidizer high pressure duct due to fatigue, cold working, braid abrasion, and mismatch of elbow-to-hose joint.	8f .83%		Significant Lox leakage caused in some instances in premature termination of engine operation. External oxidizer leakage could result in possible fire hazard and damage to engine and/or components. It could also cause freezing of control, lube or fuel lines leading to premature shutdown of the engine.	Material Fatigue Contamination	Primary	3 potential 1	Inst.	Observer cutoff, pre/post test inspection and checkout procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Sensor Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital RTD Thermometer Accelerometer Hydrophone	Ultrasonic Flaw Isotope Thermometry Remnant Magnetization Borescoping Penetrants Optical Holography Exo-electron Emission Positron Annihilation Electric Current Injection Eddy Current	

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ENGINE SYSTEM/COMPONENT Thor/Engine Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
9. Bearing Damage Due to bearing malfunction resulting in excessive torque and consequent reduction in pump output.	6F 0.624%		Two of these failures resulted in premature cutoff of test by the thrust observer due to decay in engine thrust level, with damage to the gas generator and turbine. In the other instances, thrust decay was noted during test and post-test corrective action taken to remedy the discrepant conditions. Continued operation with decaying thrust, however, could result in catastrophic damage to engine and/or vehicle.	Interference Contamination Dimensional	Primary	2, 3, 5 potential 1	Inst.	Thrust observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Optical Tachometer Isotope Detector Fiberoptic Detector RTD Thermometer Accelerometer Hydrophone Ferromagnetic Torquemeter Exo-electron Detector Tunable Diodelaser Spectrometer	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Exo-electron Emission Positron Annihilation Eddy Current Torquing	

ENGINE SYSTEM/COMPONENT Thor/Turbopump Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
11. Turbopump Seal Failure Primary oxidizer seal failure caused by low or relaxed seal bolt torques, seal shim crimped or otherwise damaged, damaged Lox seal and/or mating ring, mating surfaces of carbon seal and seal ring not seated properly, foreign material lodged between carbon flange and mating ring preventing carbon nose from seating flush with mating ring. Could also result from initial pressure surge and/or start transients in Lox pump displacing carbon nose of	19f 1.97%		Four of these failures resulted in premature test cutoff by chart observer when primary Lox seal temperature dropped below redline as evidenced by Lox seal cavity drain line temperature below acceptable limits. One other instance resulted in cancellation of test at the control center for the same reasons. None of these five failures resulted in any engine damage, however, one other instance of below-normal Lox seal drain temperature did not	Torque relaxation Contamination Dynamic	Primary	1, 3 potential 2	Inst.	Chart observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
RTD Thermometer Optical Tachometer Accelerometers Isotope Seal Detector Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

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ENGINE SYSTEM/COMPONENT Thor/Turbopump Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Primary Oxidizer Seal Failure Continued seal from mating ring.			violate redline but, one second after planned engine cutoff, the turbopump exploded causing extensive damage to the rocket engine assembly. The other 5 instances were detected pre/post test and corrective action taken. In addition to the serious engine/vehicle damage which could occur due to this condition, adjacent lines/hardware could be rendered inoperative by freezing due to the extremely cold temperature of the leaking oxidizer.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

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ENGINE SYSTEM/COMPONENT Thor/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
12. Lube Pressure Anomalies Excessive temperature and increased pressure recorded caused by restriction of lube flow to one or more lube jets resulting in increase in recorded pressure and/or bearing temperature	8f .83%		Some discrepancies resulted in premature termination of engine operation as a result of either monitored temperature or pressure exceeding redline conditions. Continued operation at these levels will affect turbopump hardware and performance and will lead to gear and/or bearing failure.	Contamination	Primary	3 potential 1	Imm.	Bearing temperature and lube pressure monitor redline cutoff.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

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ENGINE SYSTEM/COMPONENT Thor/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Lube Pressure Anomalies Continued Decay or loss of lube pressure caused by obstruction in the protective screen area or by shearing of the lube pump drive shaft	6f .62%		Decay or loss of lube flow, if of sufficient duration, will lead to failure of gears, bearings and/or uncoupling of pump due to turbine shaft failure, with subsequent catastrophic failure of the engine. Redline conditions cause premature termination of engine operation on test stand with minimal damage.	Contamination Structural	Primary	3 potential 1	Inst.	Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT ____________________

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. <u>Internal Valve Leakage</u> a. Contamination Fuel leakage past G.G. blade valve seal due to carbon blow-back from combustion chamber at termination of previous test resulting in contamination of seal/sealing surface, scratched/damaged fuel blade and/or seal, low spots in seal resulting from surge pressure and dynamic loading of blade at engine cutoff, discrepant spring resulting in insufficient loading of blade against seal.	50f 5.198%		This failure is a special case, peculiar to the design of the G.G. blade valve, but is included due to the large number of failures and the potential consequences involved. All of these *failures were detected* during pre/post test or other checkout/inspection procedures and corrective action taken. If significant leakage occurred in this area during engine start sequence, accumulation of fuel in GG combustor could result in explosion and severe damage at the	Dynamic Contamination Material Plastic Deformation	Primary	5 potential 1, 2, 3	N/A	Checkout and inspection procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Ultrasonic Thermometer Accelerometers Isotope Detector Hydrophone Tunable Diode Laser Spectrometer	Ultrasonic Leak Isotope Tracers Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography Optical Proximity Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Pressure Decay	

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ENGINE SYSTEM/COMPONENT Thor/Gas Generator Blade Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued			introduction of Lox and initiation of GG ignition Also, since the primary manifestation of this failure is fuel leakage from the quick disconnect at bottom of GG, external fuel leakage could result in fire and possible damage to engine and/or other components.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Thor/Pressure Regulator

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued c. Vibration Also caused by hang-up of Lox check valve due to rubber O-ring catching between poppet shoulder and seat and temporarily holding poppet open, initiated by vibration from engine operation. A series-redundant check valve was added to the Lox check valve to alleviate this problem effective Eng. 4822 & Subs.	7f 0.73%		Five of these failures resulted in premature cutoff of test by chart observer. It is unlikely that any consequence would occur as the result of this type failure of a more critical nature than engine cutoff.	Interference Contamination Vibration	Primary	3	Inst.	Chart observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Thor/Engine Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued d. Trapped Pressure Fuel contamination of pneumatic system and Lox start tank caused by fuel flow past redundant fuel start tank pressure check valves. This reverse flow was due to low pressure being trapped in check valve spring cavity, holding the check valve poppet in the open position. Fuel then had a reverse flow path to contaminate the pneumatic system and the Lox start tank through the regulator.	4f 0.416%		This contamination is a systems problem with various possible effects. Two of these failures resulted in premature test termination, one by the engine regulator pressure chart observer when regulator out pressure exceeded redline; the other when an explosion occurred in the area of the Lox start tank. The other two instances resulted in significant increase in the pneumatic system pressure during test, following which corrective	Contamination Overpressure	Primary	2, 3, 5 potential 1	Inst.	Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

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ENGINE SYSTEM/COMPONENT Thor/Engine Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued A subsequent fix to prevent this fuel contamination was made by drilling a 1/4" hole in the downstream end of the spring case, which resulted in equalization of pressure between the spring cavity and the lines and allowed the spring to return the poppet to the closed position as required.			action was taken to return the involved systems to required operational standards. The consequences of this condition of a fuel-contaminated system and/or components could be extremely hazardous.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

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ENGINE SYSTEM/COMPONENT Thor/Pneumatic Control Assy.

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
15. Regulator Discrepancies Failure of oxidizer start tank pressurizing valve to close caused by foreign particles in control port area between valve and orifice, plugging or restricting control port orifice and preventing valve from fully reclosing following Lox start tank venting. Could also be caused by O-ring deformation in valve.	33f 3.43%		All of these failures were detected during pre-launch checkout and/or other inspection and checkout procedures and corrective action taken. Effect of this failure is to allow bleed down of vehicle/missile bottle pressure past the valve which has not reclosed following venting of the oxidizer start tank. If this occurred to any substantial degree following engine bootstrap operation insufficient bottle pressure could result in failure to properly	Contamination Plastic deformation	Primary	5 potential 3	N/A	Checkout and inspection procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Tunable Diode Laser Spectrometer Isotope Wear Detector	Ultrasonic Leak Particle Analysis Optical Leak Differential Radiometry Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Pressure Decay	

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ENGINE SYSTEM/COMPONENT Thor/Pneumatic Control Assy.

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Regulator Discrepancies Continued			control main propellant valves and GG control valve. Premature termination of engine operation could result.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

FAILURE SUMMARY SHEETS

ATLAS ENGINE DATA

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ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA 5/Thrust Chamber

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
2. <u>Coolant Passage Leakage</u> Fuel leakage caused by tube ruptures (probably the result of localized explosions within the tubes), insufficient penetration of braze alloy at tube-to-ring joints and between tubes, intergranular corrosion and embrittlement due to presence of high sulphur compounds (presumably from the fuel) in combination with high operating temperatures with resultant tube cracks, splits and pinholes.	105f		All of these failures were detected during pre/post test checkout and inspection procedures and corrective action taken. Engine operation with fuel leakage presents a fire hazard in the presence of an ignition source, with resultant fire which could cause damage to engine and/or other engine components. Substantial reduction in fuel flow back to T/C injector could cause M/R imbalance with possibility of premature cutoff.	Material Intergranular corrosion High Temp.	Primary	5 potential 2, 3	N/A	Pre/post test procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Ultrasonic Thermometer (Flame) Ultrasonic Flowmeter (Nozzle) Polarometer Tunable Diode Laser Spectrometer (Mixture Ratio)	Ultrasonic Leak Acoustic Holography X-ray Radiography Gamma Radiography Pentoxide Polarometry Hydrogen Polarometry Hygrometer Optical Pyrometry Holographic Leak Millimeter-wave Interferometry	

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Thrust Chamber, Gas Generator, Exhaust Systems

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. <u>Joint Leakage</u> a. Hot Gas Due to under-torqued bolts or torque relaxation on bolts, damaged/defective seals, gaskets, and/or sealing surfaces.	79f		Failures were detected during pre/post test procedures and checkouts and corrective action taken. Hot gas leakage during engine operation always presents the possibility of severe fire and/or explosion or at best damage to adjacent hardware.	Torque relaxation Material damage Vibration Material	Primary	3, 5 potential 1, 2	Inst.	Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Extensiometer Ultrasonic Leak Leak Tape/Coating Optical Leak Laser Interferometry Differential Radiometry Holographic Leak Resistivity Monitoring Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Leak Fluid Pressure Decay	

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/ Several Engine Subsystems

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage b. Propellant & Lube Hydraulics Oxidizer leakage due to scratched/damaged/defective seals, packing and/or sealing surfaces, or to inadequate torque or torque relaxation on bolts and/or fittings.	51f		These failures resulted in premature engine cutoff by observer due to oxidizer leakage. The other failures were detected during pre/post test checkout and procedures and corrective action taken. External oxidizer leakage during engine operation could result in possible fire hazard and damage to engine and/or other engine components. Possible damage to adjacent hardware also exists due to extremely low temperature of the leaking oxidizer.	Torque relaxation Undertorque Stress corrosion Fatigue Contamination Material damage	Primary	3, 5 potential 2	Inst.	Observer cutoff.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

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ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Several Engine Subsystems

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued b. Propellant & Lube, Hydraulics Fuel leakage causing cutoff failures were the result of loose line fittings, most likely due to undertorque or torque relaxation. Other failures due to scratched, damaged and/or defective seals, gaskets, packings and/or sealing surfaces, porosity leaks through parent metal in flanges, torque relaxation or undertorque on nuts, bolts and/or fittings.	31f		20 of these failures resulted in premature engine cutoff by observer due to fuel leaks at propellant line fittings. The other failures were detected during pre/post test checkout and procedures, and corrective action taken. Engine operation with external fuel leakage presents a fire hazard in the presence of an ignition source, with resultant fire which could cause damage to engine and/or other engine components, the severity dependent on the magnitude of the leak	Undertorque Torque relaxation Material damage Material	Primary	3 potential 2	Inst.	Observer cutoff.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Several Engine Subsystems

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Joint Leakage Continued			and the location in the engine or in the vehicle compartment							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/ Turbopump, Lube Oil Pump, Lines & Fittings

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage Continued b. Propellant & Lube Hydraulics Lube leakages caused by low torque or torque relaxation on lube line fittings and/or screws and bolts, damaged/ defective seals and/or sealing surfaces in lube oil pump or turbopump, damaged/ defective gasket at thermo-couple installations.	41f		All of these failures were detected during pre/ post test checkout and/or inspection procedures and corrective action taken. The greatest danger in engine operation with leakage of lube oil is the possibility of sufficient loss of lube oil to affect proper lubrication of gears and bearings in the turbopump. Should this happen, effects could be catastrophic.	Torque relaxation Undertorque Material damage	Primary	5 potential 1, 2, 3	N/A	Pre/post test procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

G-7

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/ Several Engine Subsystems

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
3. Joint Leakage Continued b. Propellant & Lube, Hydraulics Hydraulic leakages caused by damaged/defective O-rings, seals and/or sealing surfaces, undertorque or torque relaxation on screws, bolts and/or fittings, contamination between poppet and seat of hydraulic package relief valve. In one case, the mount holes were too shallow to permit mount screws to pull servovalve down on face of hydraulic package to attain proper O-ring sealing action.	25f		All of these failures were detected during pre/post test procedures and inspections, and corrective action taken. Significant hydraulic leakage could affect the operation of one or more control valves (e.g. servo valves) with various effects dependent on the magnitude of the leak and the location in the engine system.	Material damage Torque relaxation Contamination	Primary	5 potential 3	N/A	Pre/post test procedures		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

G. 8

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
5. High Torque Caused by seal rubbing shaft of the balance assembly due to shift of seal and nozzle with respect to shaft and manifold, turbine nozzle loose (due to torque relaxation on retaining screws) and binding on turbine wheel second stage nozzle labyrinth seal rubbing the seal land of the second stage wheel. Could also be caused by binding of seal carbon due to combustion products in area.	10f		All of these failures were detected pre/post test and corrective action taken. Engine operation at these conditions could result in propagation of the problem to a point where reduced pump output could affect engine thrust level and/or other monitored parameters with resultant premature termination of test.	Interference Torque relaxation Contamination	Primary	5 potential 3	N/A	Pre/post test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
RTD Thermometer Optical Tachometer Accelerometer Isotope Wear Detector Hydrophone Ferromagnetic Torquemeter Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

E-9

ENGINE SYSTEM/COMPONENT Atlas MA-3/MA-5/Oxidizer High Pressure Duct

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
7. Crack-Convolutions Bellows External oxidizer leakage was due to crack in the oxidizer high pressure bellows. The failures were due to fatigue and work hardening induced by flow vibration.	12f		External oxidizer leakage during engine operation could result in possible fire hazard with damage to the engine and possibility of freezing control and sensing lines with resulting premature termination of engine operation. Depending on the magnitude of the leak, engine mixture ratio shift might result leading to engine performance degradation.	Fatigue Vibration Material degradation	Primary	3 potential 1	Inst.	Observer cutoff.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressure Sensor Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital RTD Thermometer Accelerometer Hydrophone	Ultrasonic Flaw Isotope Thermometry Remnant Magnetization Borescoping Penetrants Optical Holography Exo-electron Emission Positron Annihilation Electric Current Injection Eddy Current	

C-10

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
9. Bearing Damage Caused by incipient spalling and/or superficial wear associated with the lead track, insufficient bonding of bearing cage wrap.	2f		Both of these failures were detected during test and corrective action taken post test. Engine operation with high bearing temperature could result in possibility of bearing/gear damage and consequent damage to pump and/or other engine components.	Contamination Material	Primary	5 potential 1, 2, 3	N/A	Chart observation Visual		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Optical Tachometer Isotope Detector Fiberoptic Detector RTD Thermometer Accelerometer Hydrophone Ferromagnetic Torquemeter Exo-electron Detector Tunable Diodelaser Spectrometer	Ultrasonic Flaw Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Exo-electron Emission Positron Annihilation Eddy Current Torquing	

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Several Engine Subsystems

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
11. Turbopump Seal Leakage Oxidizer Leakage Seal drain leakage is from the T/P primary Lox seal, and is the result of low or relaxed seal bolt torques foreign material lodged between carbon flange and mating ring, mating surfaces of carbon seal and seal ring not seated properly. Other failures due to scratched/damaged/defective seals, packing and/or sealing surfaces.	65f		30 of these failures resulted in premature engine cutoff by observer due to oxidizer leakage. The other failures (37 of which were T/P Lox seal drain cavity leakage) were detected during pre/post test checkout and procedures, and corrective action taken. Oxidizer leakage within the turbopump could result in possible mixing with lube and could form an explosive gel. Damage could occur to engine and/or engine components as a result of detonation. Possible T/P performance degradation.	Torque relaxation Undertorque Stress corrosion Fatigue Contamination Material damage	Primary	3, 5 potential 2	Inst.	Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
RTD Thermometer Optical Tachometer Accelerometers Isotope Seal Detector Tunable Diode Laser Spectrometer	Isotope Thermometry Isotope Tracers Particle Analysis Borescoping Optical Proximity Torquing	

G-12

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Turbopump

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
12. <u>Lube Pressure Anomalies</u> Lube pressure variation caused by contamination in lube jet orifices and/or pressure reducer fitting, erratic output of lube oil pump due to ruptured rubber seal.	21f		3 of these failures resulted in premature test cutoff by observer when lube manifold and/or bearing jet pressure dropped below redline. The other failures were detected during/post test and corrective action taken prior to subsequent testing. Operation at low lube pressure could propagate into turbopump failure due to inadequate lubrication of gears/bearings with consequent failure and extensive damage to engine and/or components.	Contamination Material	Primary	3 potential 1, 2	Inst.	Observer cutoff.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Oxidizer Bootstrap Check Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
13. Valve Fails to Perform a. Moisture, Ice Fails to fully open due to interference between gate and seat resulting from adverse conditions (moisture, ice), and causing gate Assy to hang-up in a partially open position. Failure of valve to open fully also attributed to stiff spring.	2f		One of these failures resulted in premature test termination by the main-stage limiter during start transient. Other failure resulted in abnormally slow buildup of thrust but not sufficient to activate cutoff. It is not likely that the effect of this failure would be of a more critical nature than premature engine cutoff as noted.	Interference Material	Primary	3	Imm.	Mainstage limiter cutoff device.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Pressor Sensors Quartz, Digital Fiberoptic Laser, Digital S.A.W., Digital Isotope Wear Spectrometer Tunable Diode Laser Spectrometer	Ultrasonic Leak Acoustic Holography Isotope Tracers Pentoxide Polarometry Hygrometer Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography	

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Head Suppression (H.S.) Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage a. Contamination Results in sticking or binding of shaft and/or gate, or hang-up of other moving parts in actuator assembly.	4f		All of these failures were detected during pre/post test inspection procedures and corrective action taken. Fast opening and/or closing or failure to open or close at required time in the engine sequence could result in erroneous oxidizer/fuel mixture ratio which could result in premature engine cut-off, or possibility of engine and/or component damage dependent on timing and severity of failure.	Binding Contamination	Primary	5 potential 2, 3	N/A	Pre/post test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Ultrasonic Thermometer Accelerometers Isotope Detector Hydrophone Tunable Diode Laser Spectrometer	Ultrasonic Leak Isotope Tracers Particle Analysis Laser Scattering Optical Leak Borescoping Differential Radiometry Optical Holography Optical Proximity Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Torquing Pressure Decay	

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ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Gas Generator

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
14. Internal Valve Leakage Continued a. Contamination Oxidizer gate valve leakage cutoff was caused by pieces of Lox pad (shattered by detonation in GG) lodged in injector and restricted Lox flow with resultant failure to bootstrap. Detonation is primarily caused by leakage of Lox past the gate with resultant combustion and explosion when SPGG's are initiated. Leakage is due to failure of blade to make positive seal due to scratched or damaged seal and/or seat, or to misfit	5f		One of these failures resulted in premature termination of test by the mainstage limiter when the engine failed to attain normal bootstrap. The other 4 failures were detected during post-test inspection of oscillograph and other test records, and corrective action taken. 3 of these 4 failures indicated pressure spikes and detonation in G.G. during transition. Leakage past the oxidizer gate seal presents a high probability of explosion in GG during transition,	Interference Material damage	Primary	3, 5 potential 2	Imm	Mainstage limiter cutoff device		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

G 16

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Gas Generator

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued caused by loose splines.			however, resulting damage would most likely be confined to G.G. interior.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Gas Generator

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued Lox poppet leakage past seat, binding of poppet in bore of housing due to galling of poppet and/or contamination of poppet stem or to misalignment also caused by scratched/damaged poppet and/or seat.	7f		All of these failures were detected during post-test inspection and investigation of oscillograph and other test records, and corrective action taken prior to any subsequent testing. One of the failures indicated a fire and explosion in the G.G. control valve/injector area but with minimal damage to G.G. oxidizer leakage past the Lox poppet presents the possibility of explosion and damage to G.G. during transition.	Interference Galling Material damage	Primary	5 potential 2, 3	N/A	Post-test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

G 18

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Heat Suppression (H.S.) Valve

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Internal Valve Leakage Continued c. Vibration, Seat Caused by low torque or torque relaxation on retainer bolts, resulting in pieces of Teflon flaking off seat and becoming lodged between seal and sealing surface.	3f		All of these failures were detected during pre/post test procedures, and corrective action taken. Oxidizer leakage past lip could result in accumulation of Lox in T/C area with possibility of detonation and/or damage to T/C, engine and/or other components at time of ignition.	Low torque Material Contamination Vibration	Primary	5 potential 2, 3	N/A	Pre/post test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

6 19

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Lox Regulator

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
15. Regulator Discrepancies Fails to Provide Proper Regulation of Oxidizer flow. Caused by misalignment of sleeve, moisture/contamination in regulator, lack of or inadequate lubrication, nicked/damaged spool, piston and/or bore. Could also be caused by excessive diametral clearance resulting in side loads and high friction causing oscillations. For MA-5 only - could also be caused by external leakage of helium control pressure.	44f		7 of these failures resulted in premature test cutoff, 4 by fuel manifold pressure switch and/or mainstage limiter when bootstrap was not attained, 2 by overspeed trip device when Lox regulation went out of control and 1 by observer due to pressure oscillations from malfunctioning regulator. The other 37 failures were detected during pre/post test procedures and inspections and corrective action taken. 2 of these 37 failures gave post-test	Contamination Material damage Dimensional	Primary	3, 5 potential 2	Imm. Inst.	Fuel manifold pressure switch, Mainstage limiter cutoff device Overspeed trip device Observer cutoff		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
Tunable Diode Laser Spectrometer Isotope Wear Detector	Ultrasonic Leak Particle Analysis Optical Leak Differential Radiometry Halogen Leak Flow Leak Mass Spectrometry Thermal Leak Pressure Decay	

G 22

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA 5/Lox Regulator

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Regulator Discrepancies Continued			indication (from run records) of chugging, which presents potential possibility of combustion instability. Lack of proper regulation of oxidizer flow can have varied effects, mostly related to gas generator operation and output, most of which would result at worst in premature engine cutoff. The possibility of engine and/or component damage is, however, potentially existent.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

C 21

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Mixture Ratio Control Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
16. Contamination, Hydraulic Control Assembly Excessive deadband is caused by contamination in MRC body producing a high friction between the piston and bore and resulting in excessive hysteresis, misalignment and/or improper torque on body bolts, side loading of piston due to misalignment of piston and diaphragm assemblies.	14f		All of these failures were detected during pre/post test procedures and/or inspection, and corrective action taken. Engine operation at this condition could result in delayed response of the servo-piston to changes in Delta pressure, which could cause corresponding delay in H.S. valve movement to correct the existent discrepancy. Dependent on the time of occurrence during engine operation and the magnitude of the failure, initiation of premature engine cutoff could result.	Contamination Torque Interference	Primary	5 potential 3	N/A	Pre/post test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS
	Ultrasonic Leak Particle Analysis Optical Leak Differential Radiometry Flow Leak Pressure Decay	

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Mixture Ratio Control Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Contamination, Hydraulic Control Assembly Continued Null shift is caused by change in setting as the result of vibration and/or shock during engine operation, incorrect null setting at assembly. Could also be due to unequal spring compression ratios, damaged/defective diaphragms.	7F		All of these failures were detected during pre/post test procedures and/or inspections, and corrective action taken. Engine operation at this condition could result in unplanned movement of the H.S. valve gate or failure of the H.S. valve to respond accurately to differential input pressures. This bias could result in improper propellant utilization or, dependent on time of occurrence and severity of the malfunction, could result in oxidizer-rich	Contamination Torque Interference	Primary	5 potential 3	N/A	Pre/post test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

G 7 5

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Mixture Ratio Control Assembly

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL, %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Contamination, Hydraulic Control Assembly Continued			burning conditions which could trigger premature engine cutoff and possibly result in damage to engine and/or components.							

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

ENGINE SYSTEM/COMPONENT Atlas MA-3 & MA-5/Hydraulic Control Valve (MOV Control Package)

FAILURE MODE AND CAUSE	FREQUENCY OF FAILURE, %	LIFE DESIGN/ACTUAL %	EFFECT OF FAILURE	FAILURE TYPE	PRIMARY OR SECONDARY FAILURE	CRITICALITY	REACT TIME	DETECTION METHOD USED	FAILURE PREDICTABILITY AND METHOD	POTENTIAL MEASURANDS
Output Unstable/Erratic Caused by internal leakage in the control package, limited travel of control piston due to overlength and canted spring, binding or sticking of spool, contamination in package affecting movement of spool or other internal parts.	5f		All of these failures were detected by chart observation during test and/or post test inspection procedures, with corrective action taken prior to any subsequent testing. Malfunctions of the control valve affect operation of the main oxidizer valve and, dependent on time of occurrence during engine operation and magnitude of the failure, could result in premature engine cutoff with possibility of damage to engine and/or components.	Binding Contamination Interference	Primary	5 potential 2, 3	N/A	During and post-test procedures.		

VIABLE IN-FLIGHT MONITORING SYSTEMS	BETWEEN FLIGHT INSPECTION TECHNIQUES	REMARKS/COMMENTS

APPENDIX C. FAILURE PROPAGATION BLOCK DIAGRAMS

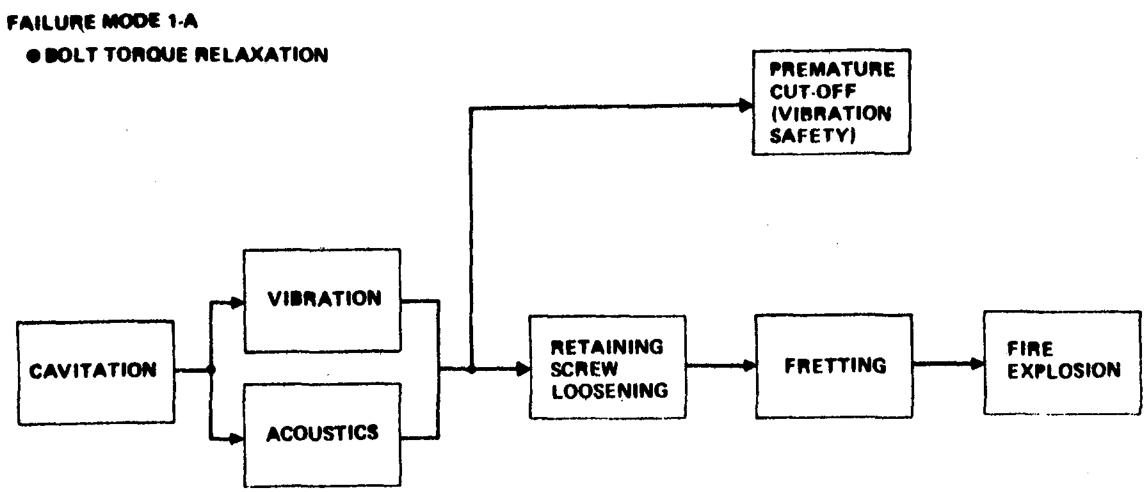
APPENDIX C

FAILURE PROPAGATION BLOCK DIAGRAMS

The Failure Propagation Block Diagrams were devised to obtain a better understanding of the failure mechanism of each of the sixteen failure modes encountered in the study.

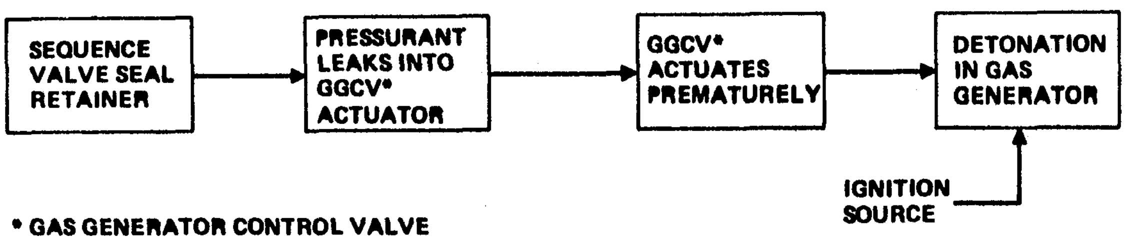
Each diagram attempts to illustrate the events which lead to the failure as described in the Failure Summary sheets (see Appendix B). It has been found that by indicating symptoms and events preceding the outright failure, the determination of appropriate monitoring devices is made easier. There is a Failure Propagation Block diagram for each of the sixteen failure modes, regardless of engine system.

The events are shown as rectangles and the time sequence from left to right.



Main Oxidizer Valve

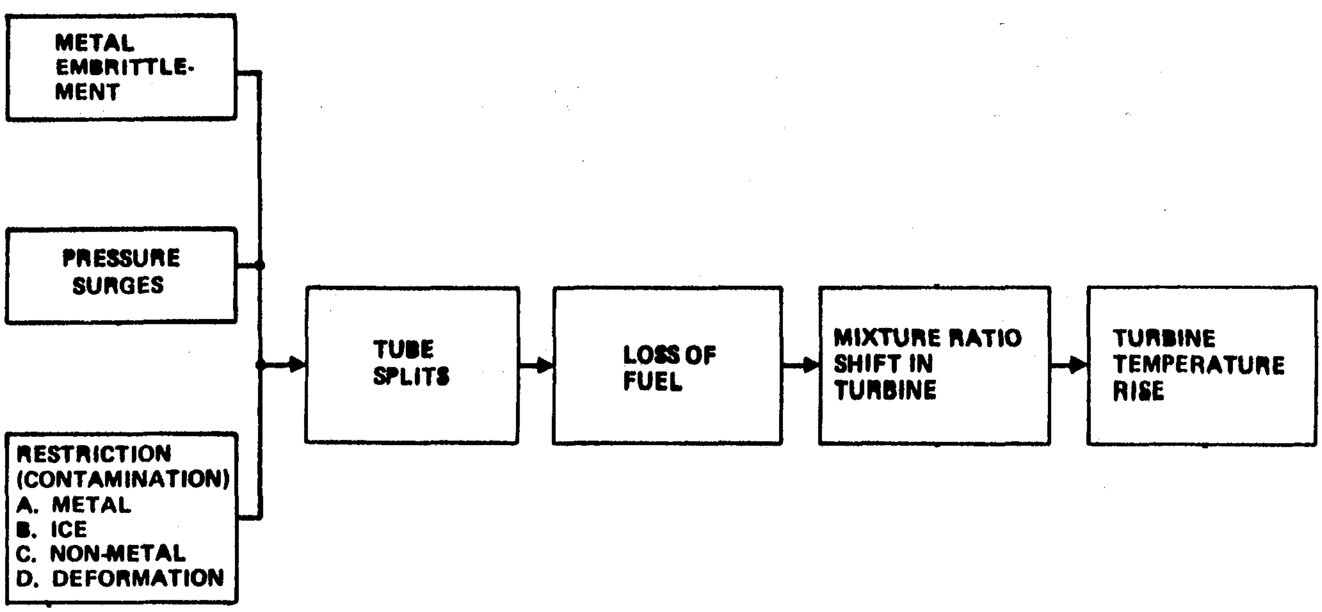
FAILURE MODE 1-B
- BOLT TORQUE RELAXATION



* GAS GENERATOR CONTROL VALVE

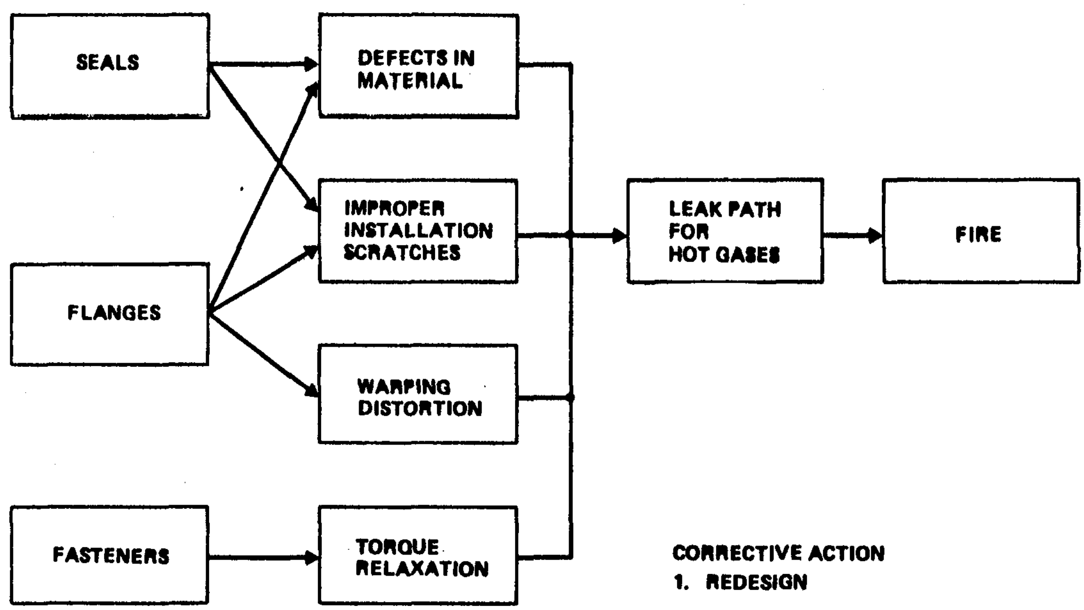
Main Oxidizer Valve

FAILURE MODE 2
- COOLANT PASSAGE LEAKAGE



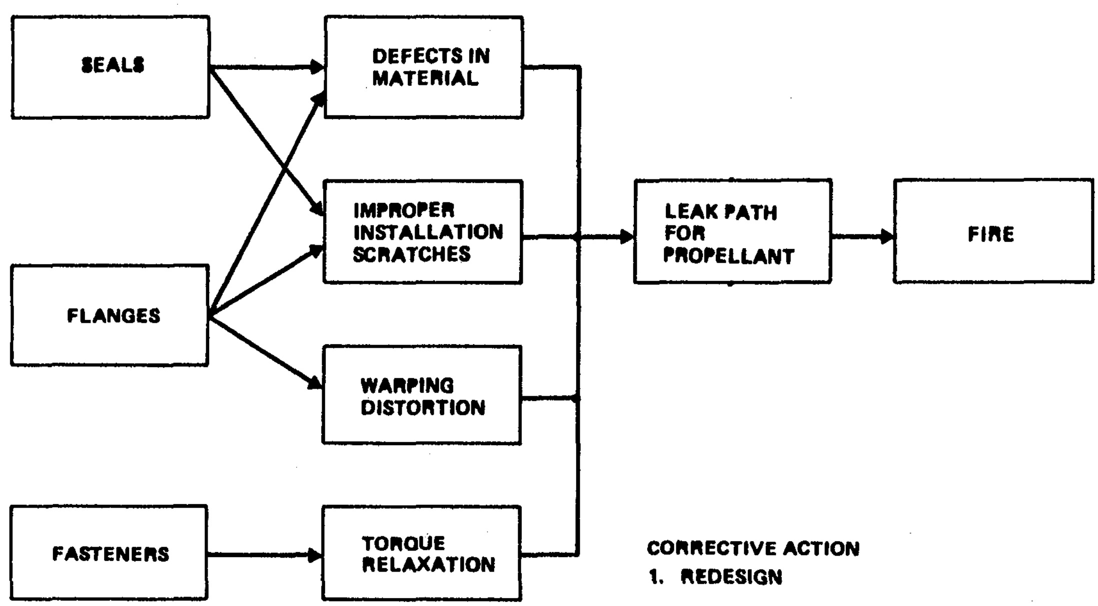
Nozzle-Combustor

FAILURE MODE 3-A
• JOINT LEAKAGE



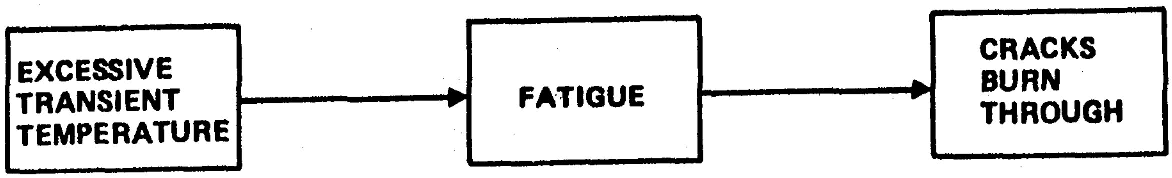
Hot Gas Leakage

FAILURE MODE 3-B
• JOINT LEAKAGE



Seals-Propellant Leakage

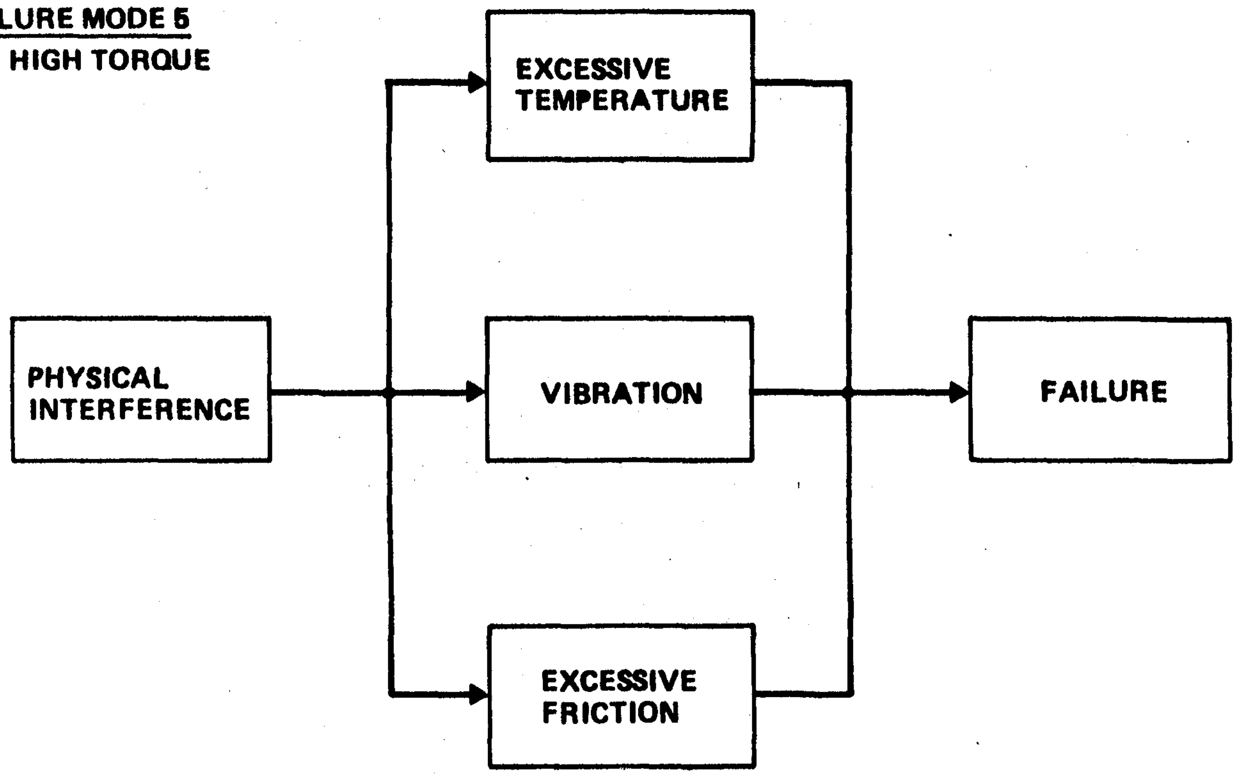
FAILURE MODE 4
- **TRANSFER TUBE CRACKS**



CORRECTIVE ACTION

1. **REDESIGN**

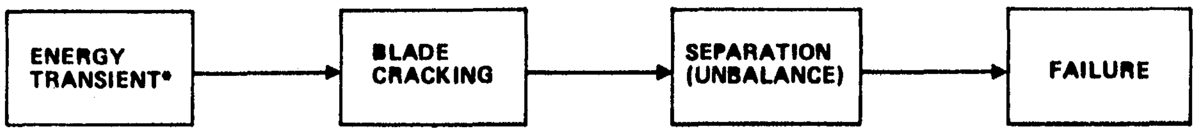
FAILURE MODE 5
- **HIGH TORQUE**



Propellant Turbopump Labyrinth Seal

FAILURE MODE 6

- **CRACKED TURBINE BLADES**



*TEMPERATURE
PRESSURE
ACOUSTICS
MECHANICAL

Turbopump

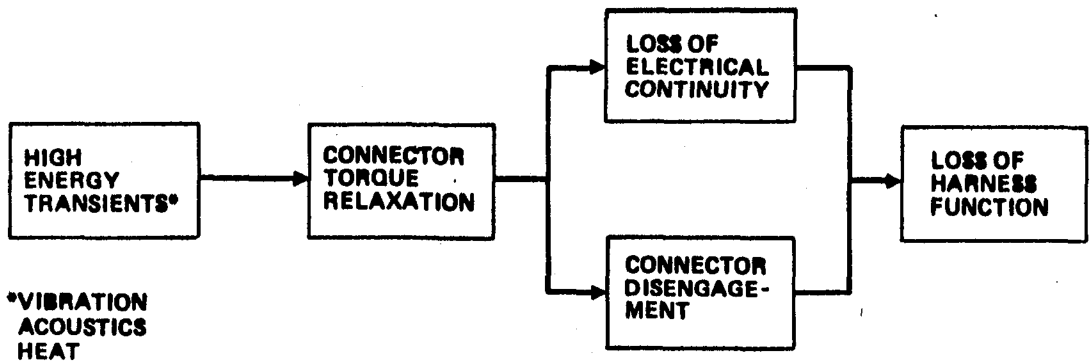
FAILURE MODE 7

- **CRACKED CONVOLUTION, BELLOWS & SHIELD**

*TEMPERATURE
PRESSURE
ACOUSTICS
MECHANICAL

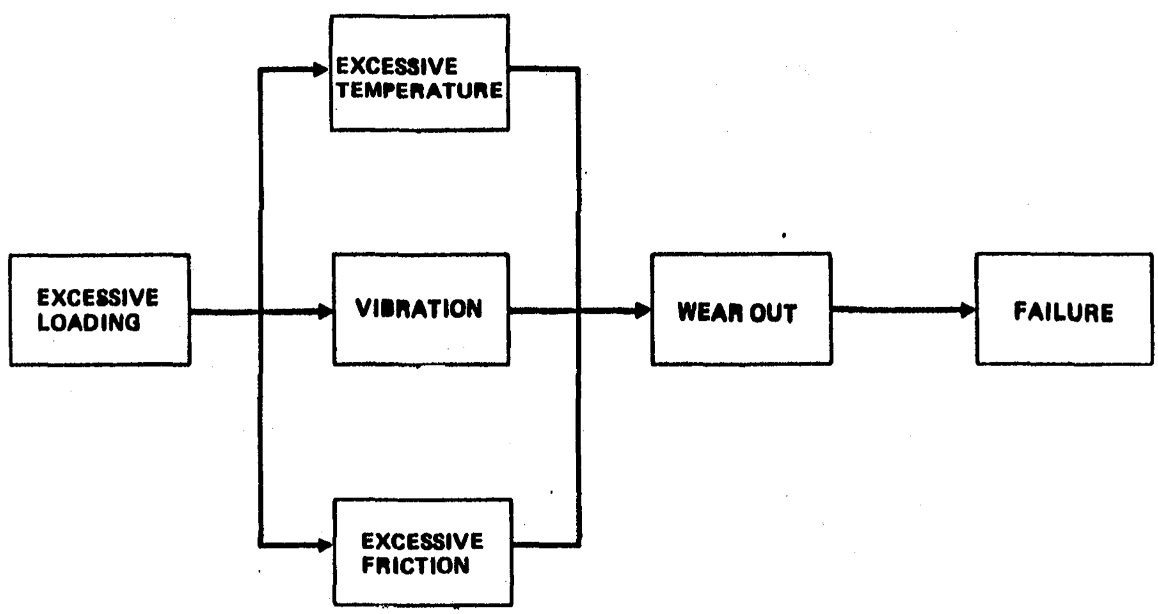
High-Pressure Fuel Turbopump

FAILURE MODE 8
- LOOSE ELECTRICAL CONNECTORS



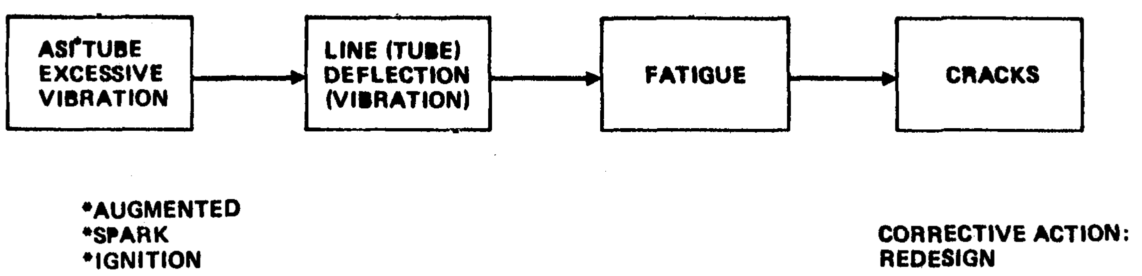
Electrical Harnesses

FAILURE MODE 9
- BALL BEARING DAMAGE



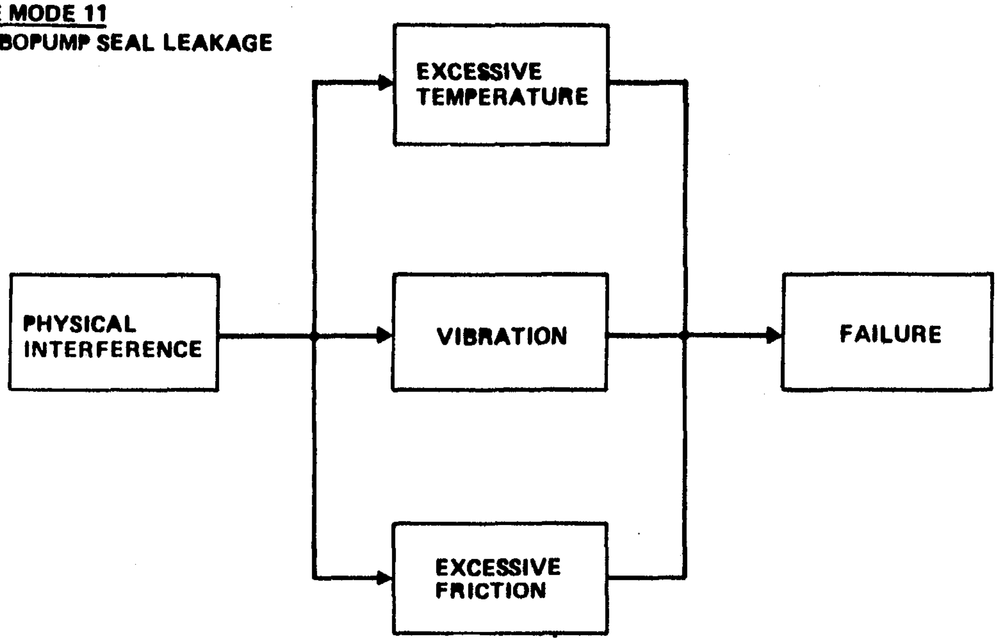
Turbopump

FAILURE MODE 10
• TUBE FRACTURE

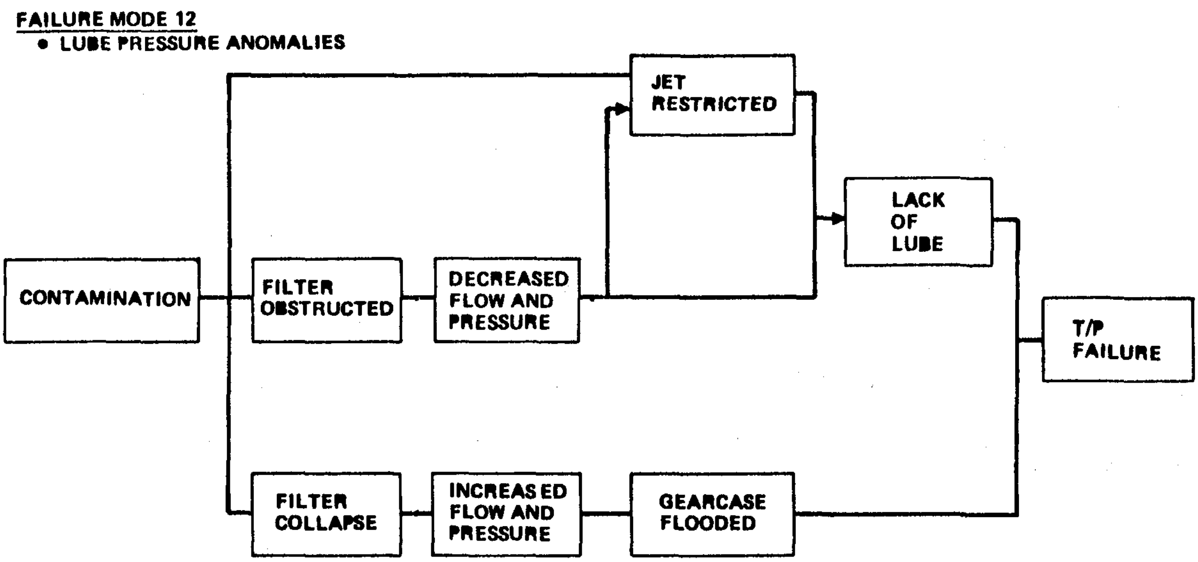


ASI Propellant Line (Tube)

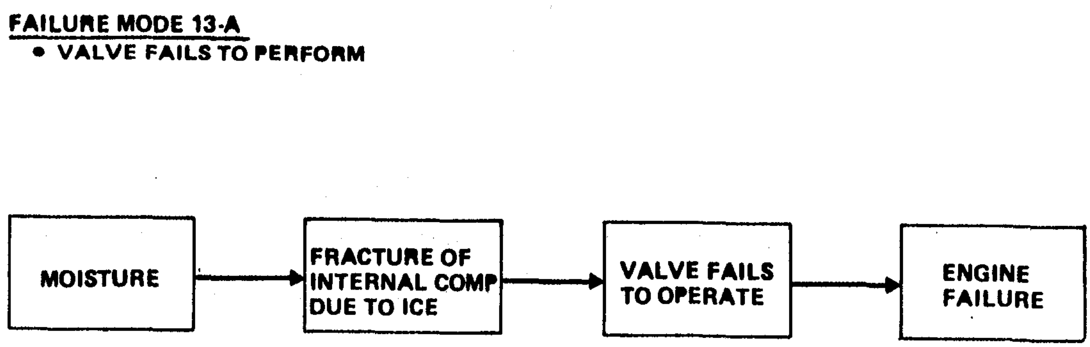
FAILURE MODE 11
• TURBOPUMP SEAL LEAKAGE



Primary Turbopump Seal



Lube Pressure Anomalies



Oxidizer Poppet Valve

FAILURE MODE 13 B

- VALVE FAILS TO PERFORM

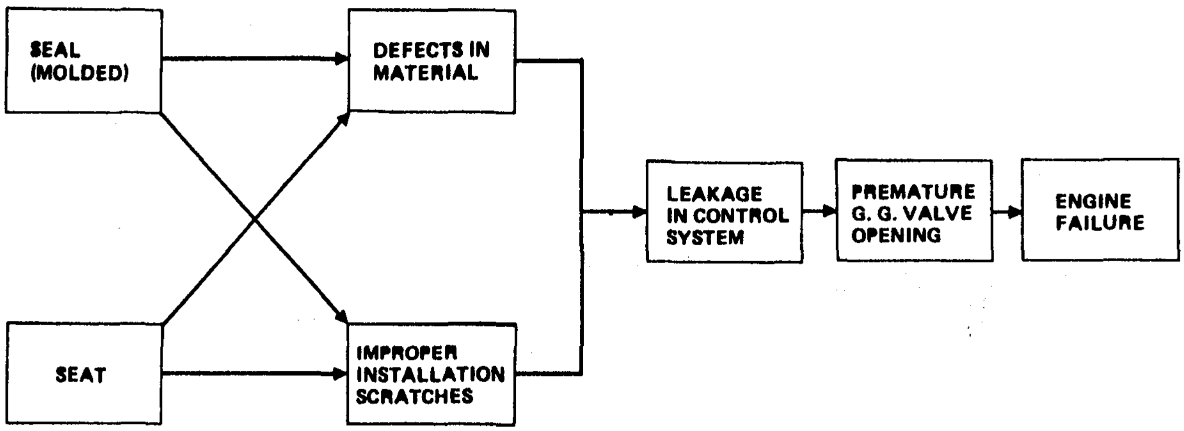
Main Propellant Valve

FAILURE MODE 14-A

- INTERNAL LEAKAGE

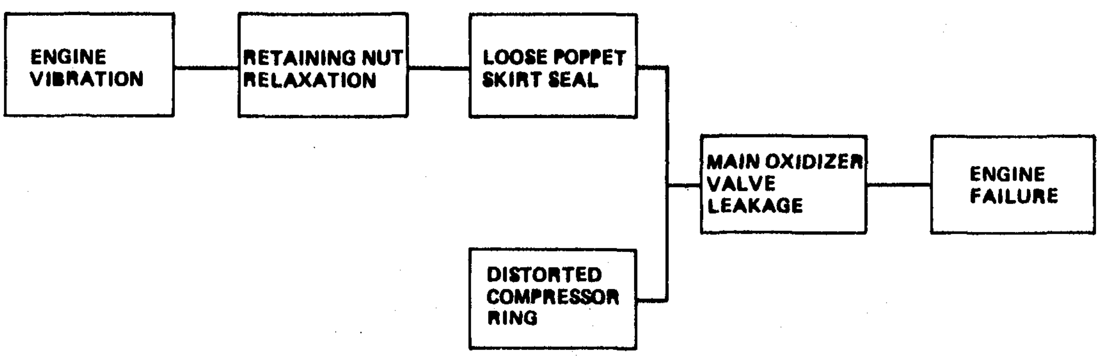
Poppet Valve

FAILURE MODE 14-B
- INTERNAL VALVE LEAKAGE



MOV Sequence Valve

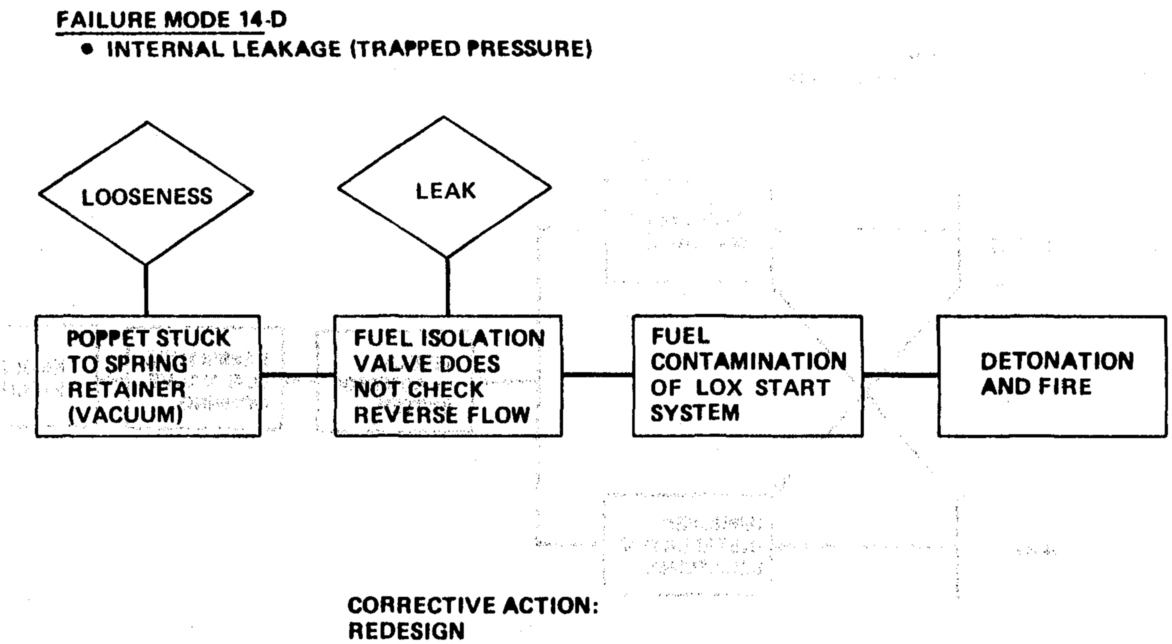
FAILURE MODE 14-C
- INTERNAL LEAKAGE



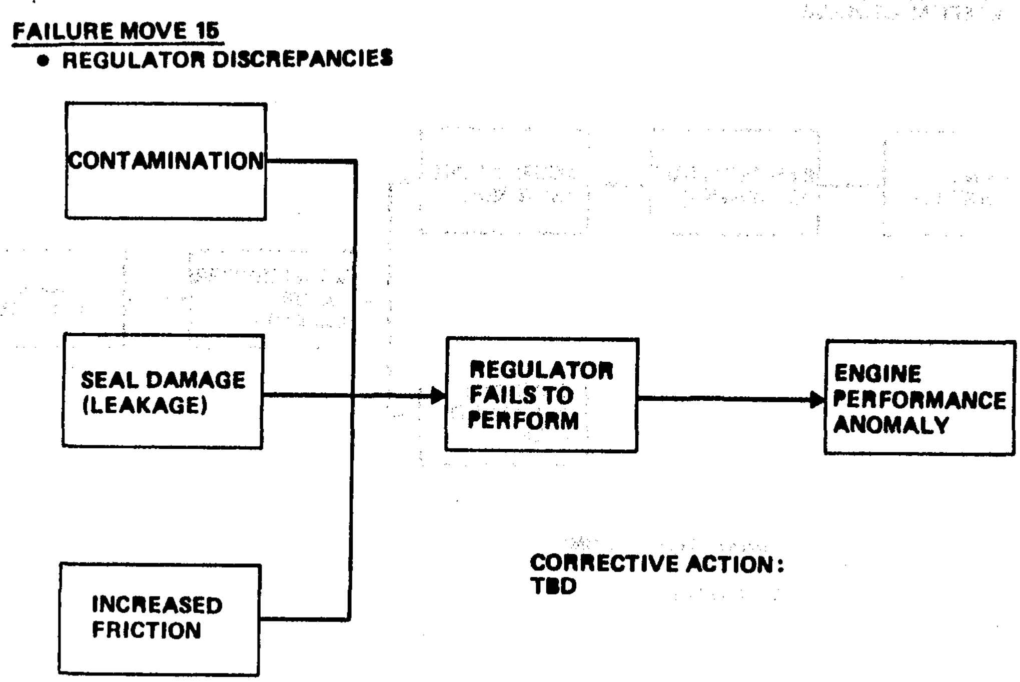
CORRECTIVE ACTION:

1. REDESIGN

Main Oxidizer Valve



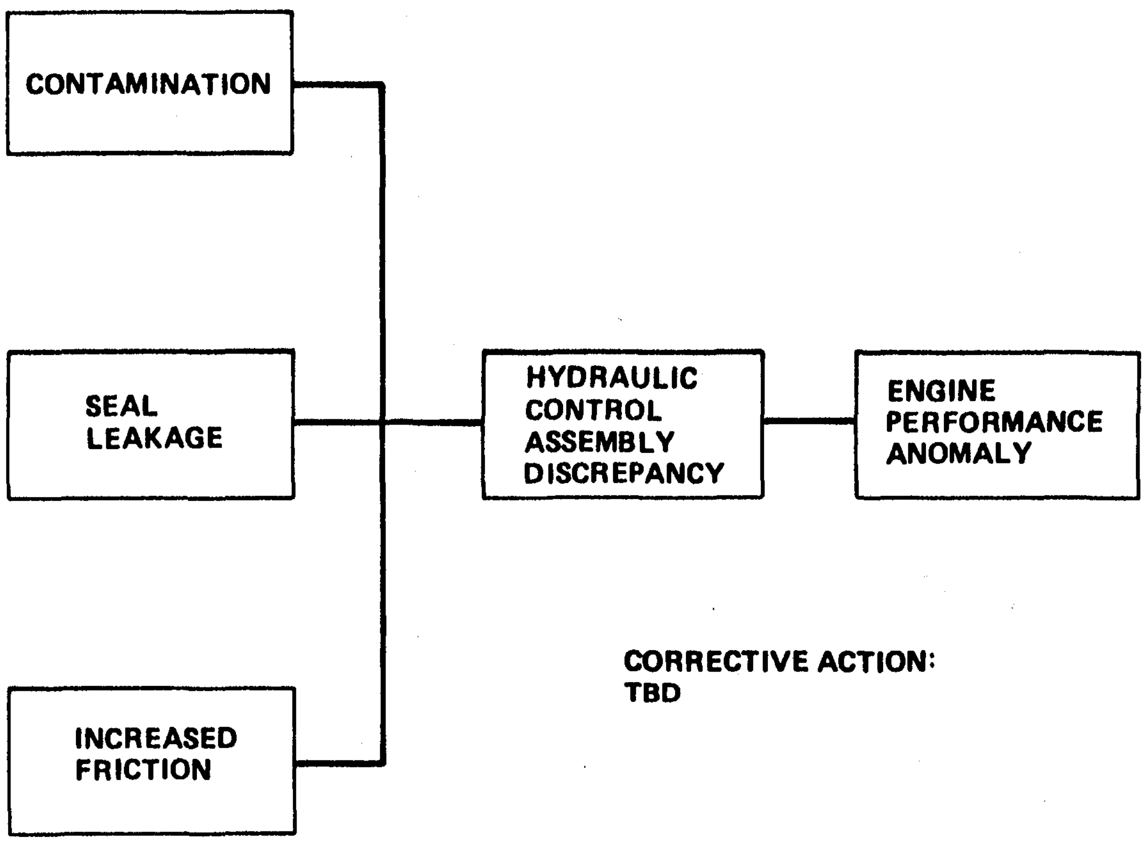
Redundant Isolation Valve



Regulator Failure

FAILURE MODE 16

• CONTAMINATED HYDRAULIC CONTROL ASSEMBLY



CORRECTIVE ACTION:
TBD

Hydraulic Control Assembly

APPENDIX D

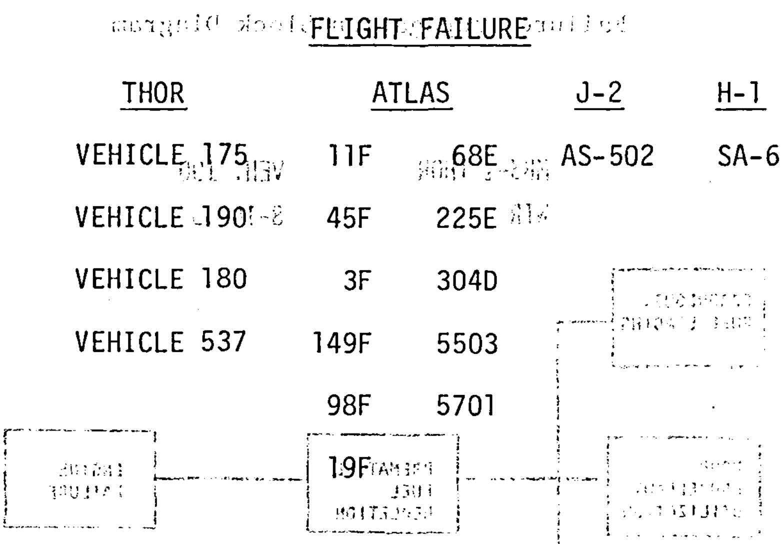
FLIGHT FAILURES

To support the study of failure modes, an analysis was made of flight failures of the engine systems selected for the study.

Again, the same technique used to slice into the failure modes detected by the analysis of UCR's was used in this assessment.

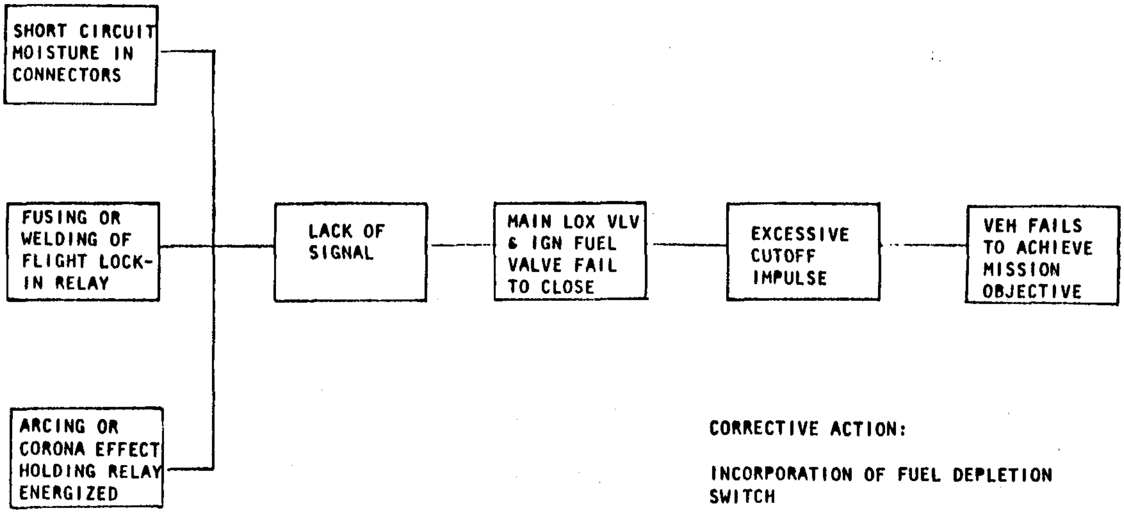
Not all flight failures had exhaustive reports detailing the incident. The events leading to the engine failure are shown as rectangle and the passage of time is shown from left to right. For purposes of illustration, the failure mechanism has been greatly simplified but, in each case, has retained sufficient characteristics to indicate how the incident developed.

The charts in this appendix do not show existing or possible monitoring devices. Most of the depicted flights carried limited instrumentation with no means to shut down the malfunctioning engine because once the vehicle left the pad, there was no way to recover the mission.



Failure Propagation Block Diagram

MB3-1 THOR VEH. 175
WTR 8-3-59

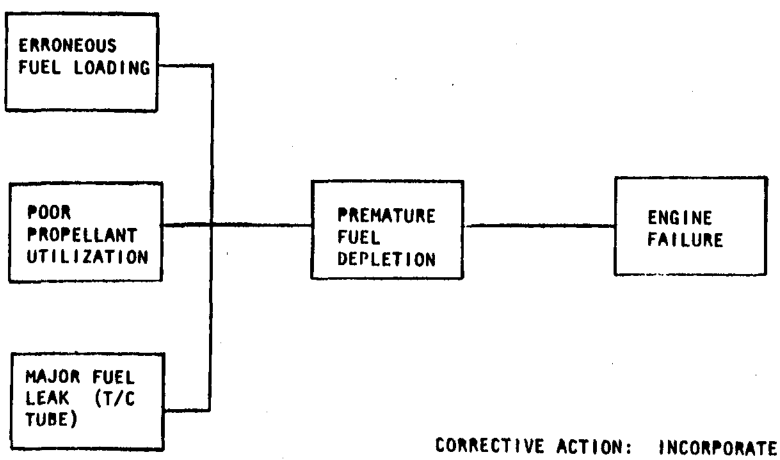


CORRECTIVE ACTION:

INCORPORATION OF FUEL DEPLETION SWITCH

Failure Propagation Block Diagram

MB3-1 THOR VEH. 190
WTR 8-14-59

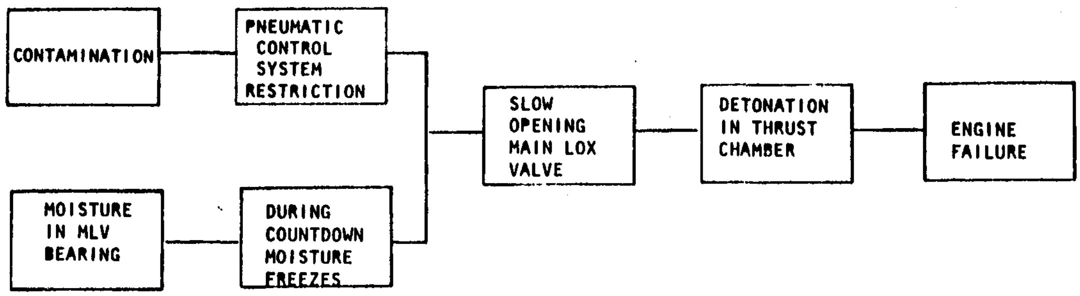


CORRECTIVE ACTION: INCORPORATE A FUEL DEPLETION SWITCH

Failure Propagation Block Diagram

MB3-1 THOR VEH. 180

WTR 7-26-62



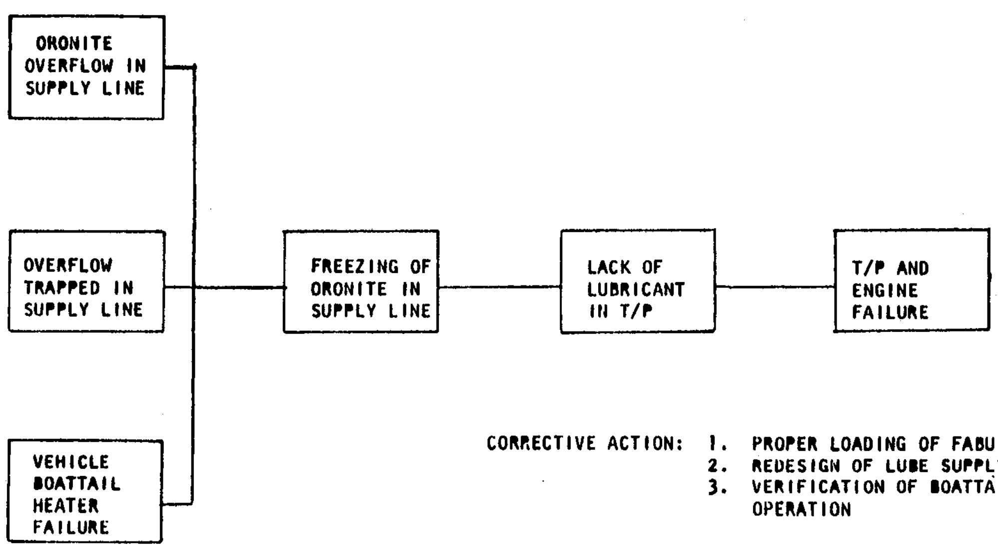
CORRECTIVE ACTION:

CHANGES IN PROCEDURE PRIOR TO LAUNCH TO INSPECT FOR CONTAMINATION AND MOISTURE.

Failure Propagation Block Diagram

MB3-3 THOR VEH. 537

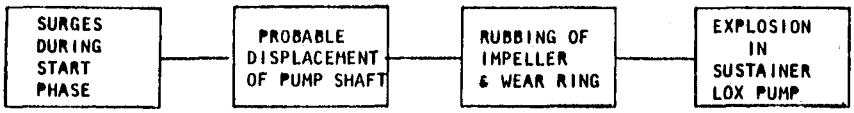
WTR 2-17-71



CORRECTIVE ACTION:
1. PROPER LOADING OF FABU
2. REDESIGN OF LUBE SUPPLY LINE
3. VERIFICATION OF BOATTAIL HEATER OPERATION

Failure Propagation Block Diagram

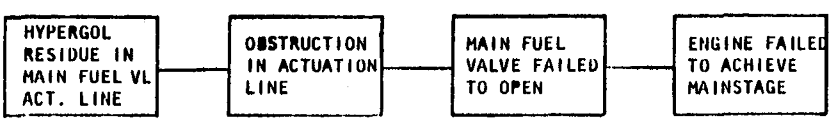
MA-3 ATLAS VEH. 11F
ETR 4-9-62



CORRECTIVE ACTION: INCORPORATION OF KEL-F LINER

Failure Propagation Block Diagram

MA-3 ATLAS VEH. 45F
WTR 10-3-63

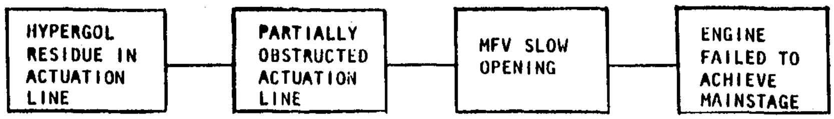


CORRECTIVE ACTION:

1. REPLACEMENT OF ACTUATION LINE PRIOR TO FLIGHT
2. INCORPORATE PURGE OF ACTUATION LINES AFTER HOT FIRE.

Failure Propagation Block Diagram

MA-3 ATLAS VEH. 3F
WTR 4-3-64

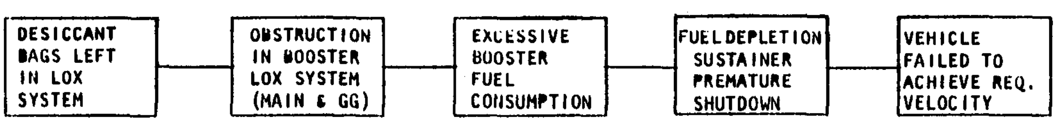


CORRECTIVE ACTION:

1. REPLACEMENT OF ACTUATION LINES PRIOR TO FLIGHT
2. INCORPORATE PURGE OF ACTUATION LINES AFTER HOT FIRE

Failure Propagation Block Diagram

MA-3 ATLAS VEH. 149F
8-8-66

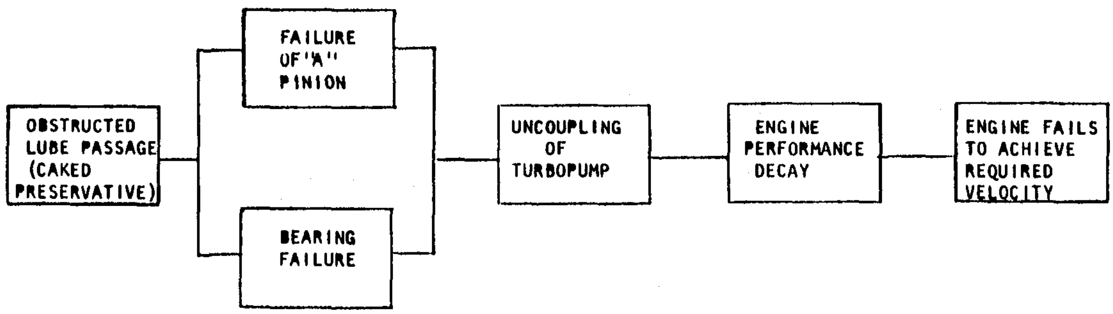


CORRECTIVE ACTION:

PROCEDURES MODIFIED TO VERIFY REMOVAL OF ALL DESICCANT BAGS FROM ENGINE PRIOR TO LAUNCH.

Failure Propagation Block Diagram

MA-3 ATLAS VEH. 98F
WTR 10-10-69

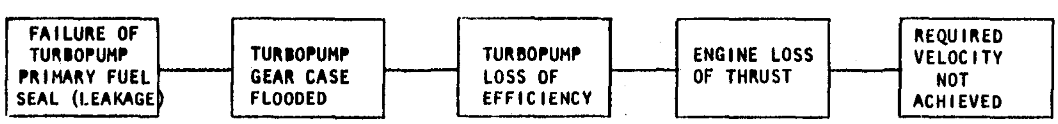


CORRECTIVE ACTION:

1. REPRESERVED ALL ENGINES WITH PROPER TYPE PRESERVATIVE OIL
2. INSPECTION AND FLOW CHECKS OF LUBE DISTRIBUTION SYSTEM

Failure Propagation Block Diagram

MA-3 ATLAS VEH. 19F
WTR 5-29-80



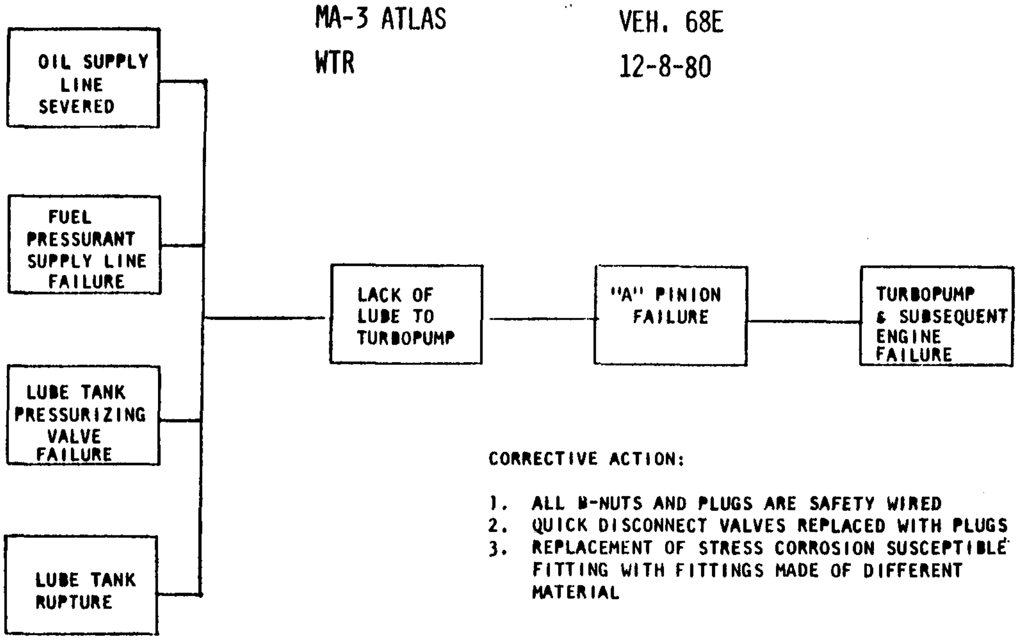
CORRECTIVE ACTION:

1. INCORPORATION OF T/P GEAR BOX PURGE
2. INSPECTION OF T/P DRAIN LINE FOR RESTRICTIONS

Failure Propagation Block Diagram

MA-3 ATLAS WTR

VEH. 68E 12-8-80



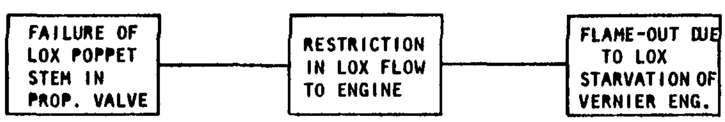
CORRECTIVE ACTION:

1. ALL B-NUTS AND PLUGS ARE SAFETY WIRED
2. QUICK DISCONNECT VALVES REPLACED WITH PLUGS
3. REPLACEMENT OF STRESS CORROSION SUSCEPTIBLE FITTING WITH FITTINGS MADE OF DIFFERENT MATERIAL

Failure Propagation Block Diagram

MA-5 ATLAS ETR

VEH. 225E 7-20-65



CORRECTIVE ACTION:

INCREASE POPPET STRENGTH AND CORROSION RESISTANCE CHANGE OF MATERIAL (2024-T6 TO 2024-T4)

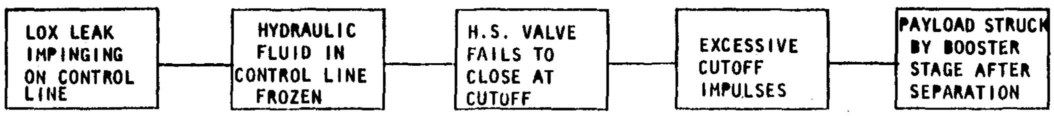
PROHIBIT REPETITIVE MOLDING OF KEL-F SEAT

VERIFY EXISTENCE OF GAP BETWEEN PNEUMATIC PISTON & POPPET

Failure Propagation Block Diagram

MA-5 ATLAS VEH. 304D

WTR 3-19-66



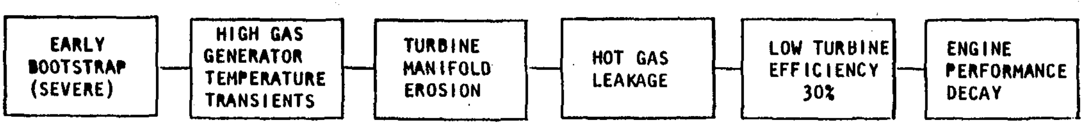
CORRECTIVE ACTION:

PROPELLANT UTILIZATION AND HEAD SUPPRESSION VALVE CONTROL LINES WERE INSULATED.

Failure Propagation Block Diagram

MA-5 ATLAS VEH. 5503

ETR 12-4-71



CORRECTIVE ACTION:

REDESIGN OF SUSTAINER GAS GENERATOR OXIDIZER SUPPLY SYSTEM:

1. ADAPTER REDESIGNED
2. INCORPORATION OF NEW CHECK VALVE
3. LARGER BOOTSTRAP LINE

Failure Propagation Block Diagram

MA-5 ATLAS　　VEH. 5701　AC-43

ETR　　9-29-77

CORRECTIVE ACTION:

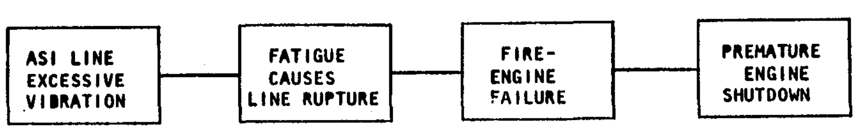
REVIEW OF PROCESS CONTROLS OF ITEMS MANUFACTURED FROM 300 SERIES STAINLESS STEEL.

MODIFICATION OF PROCEDURES TO PREVENT CARBON CONTAMINATION OF DUCT DURING BRAZING OPERATION.

Failure Propagation Block Diagram

J-2　　AS-502

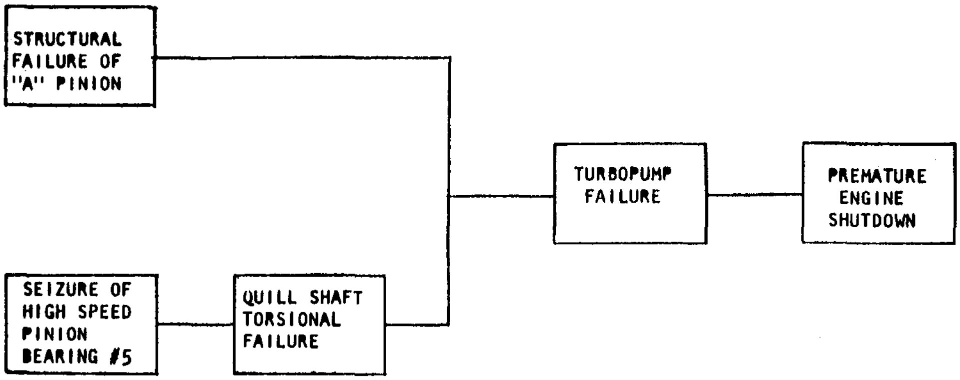
ETR　　4-4-68



Failure Propagation Block Diagram

H-1 VEH. SA-6

ETR 5-28-64



Failure Propagation Block Diagram

APPENDIX E. IN-FLIGHT CONDITION MONITORING LITERATURE SEARCH

APPENDIX E

IN-FLIGHT CONDITION MONITORING LITERATURE SEARCH

A computerized search and review of periodicals was employed to survey the literature for identification of novel and state-of-the-art in-flight condition-monitoring technologies. The in-house on-line capability included Orbit IV and Dialog systems, and Compendex, NTIS, and ISMEC data bases, totaling some six million citations.

The search addressed all those citations which were related to sensors, instruments and detectors, both diagnostic and prognostic. It was limited to industrial, aerospace and automotive fields and the result was 289 relevant citations. Upon reviewing these citations, 89 complete articles were requested. They are summarized in Table 22 according to article title, author, source, in-flight/between-flight novel, SOTA and rocket-engine categories. The table also shows the number of SOTA and novel, in-flight and between-flight condition-monitoring systems discussed in each article.

This search combined with a few other minor on-line searches and review of periodicals resulted in the 20 novel and 14 SOTA technologies.

TABLE 22. SUMMARY LITERATURE SEARCH FOR IN-FLIGHT CONDITION MONITORING TECHNOLOGIES

TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
ON-LINE DIAGNOSTICS CUT ENGINE MAINTENANCE	REASON, JOHN	POWER MAGAZINE		6					GAS TURBINE ENGINE DIAGNOSTIC SYSTEM
CHROMATOGRAPHY AUTOMATION: SYSTEM CONTROL AND CREDIT-BILITY IMPROVEMENT THROUGH MICROPROCESSORS	BAUMANN, FRED BROWN, A. C. CRAIN, S. P. HARTMANN, C. H. HENDRICKSON, JOEL	VARIAN INSTRUMENT DIVISION	-	-	-	-	-	-	MINICOMPUTER BASED AUTOMATED GAS CHROMATOGRAPH
MICROPROCESSOR-BASED AUTO-MATIC HETERODYNE INTERFEROMETER	MOTTIER, F. M.	UNITED TECHNOL-OGIES RESEARCH CENTER	-	-	-	-	-	-	INSTRUMENT INTENDED AS A DIAGNOSTIC TOOL IN ADAPTIVE OPTICS
TECHNICAL DIAGNOSIS - A SYS-TEMS APPROACH/AGARD CONFER-ENCE PROCEEDINGS NO. 165	BRACHMAN, R. J.	FRANKFORD ARSENAL, DEPART-MENT OF THE ARMY		10					TECHNICAL DIAGNOSIS OF ENGINES AT THE DEPOT AND VEHICLE USER LEVEL OF TACTICAL UNITS
IN-FLIGHT THRUST MEASUREMENT A FUNDAMENTAL ELEMENT IN ENGINE CONDITIONING MONITOR-ING AGARD CONFERENCE PRO-CEEDINGS NO. 165	CHAPPELL, M.S. GRAVELLE, J. A.	NATIONAL RESEARCH COUNCIL COMPUTING SERVICES CO.	1						IN-FLIGHT GROSS THRUST MEASUR-ING SYSTEM
AIRCRAFT ENGINE DESIGN AND DEVELOPMENT THROUGH LESSONS LEARNED AGARD CONFERENCE PROCEEDINGS NO. 215	KOFF, B. L.	GENERAL ELECTRIC AIRCRAFT ENGINE GROUP		2	1				INFRARED OPTICAL PYROMETER USED FOR MEASURING TEMPERATURE OF ROTATING TURBINE BLADES
METROLOGY AUTOMATED SYSTEM FOR UNIFORM RECALL AND REPORTING (MEASURE USERS MANUAL)	---	OFFICE OF CHIEF OF NAVAL OPERA-TIONS, DEPARTMENT OF THE NAVY	-	-	-	-	-	-	USERS MANUAL TO PROVIDE INFOR-MATION TO EFFECTIVELY USE THE NAVY'S METROLOGY AUTOMATED SYSTEM
AIRCRAFT GAS TURBINE CONDI-TION ANALYSIS INSTRUMENTA-TION: ITS USE FOR THE STATUS DIAGNOSIS OF NAVEL TURBINE ENGINES	ZIEBARTH, H.K. CHANGE, J. D.	AIRESEARCH MANUFACTURING CO.	-	-	-	-	-	-	TURBINE ENGINE DIAGNOSTIC TECHNIQUES FOR STATUS DETERM-INATION OF CRITICAL COMPONENT OF GAS TURBINE ENGINES
MONITOR MACHINERY CONDITION FOR SAFE OPERATION	BENTLY, D. E.	BENTLY NEVADA CORP.	-	-	-	-	-	-	PHILOSOPHY OF USING DIAGNOSTIC INSTRUMENTATION FOR PREVENTING ACCIDENTS INVOLVING ROTATING MACHINERY

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY **NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT			BETWEEN-FLIGHT			REMARKS
			SOTA* ROCKET	SOTA NONROCKET	NOVEL**	SOTA ROCKET	SOTA NONROCKET	NOVEL	
INSTRUMENTATION FOR RAMAN/RAYLEIGH LIGHT SCATTERING MEASUREMENTS OF GAS DENSITIES AND TEMPERATURES IN AEROSPACE TEST FACILITIES	POWELL, H. M. JONES, J. H. WILLIAMS, W. D. MCQUIRE, R. L.	ARO, INC.			1				INSTRUMENTATION SYSTEM DEVELOPED FOR MEASUREMENT OF GAS SPECIES DENSITIES AND TEMPERATURES IN AEROSPACE TEST FACILITIES. (RAMAN/RAYLEIGH LIGHT SCATTERING TECHNIQUES)
MODERN DIAGNOSTIC TECHNIQUES IMPROVE STEAM-TURBINE RELIABILITY	BANNISTER, R. L. OSBORNE, R. L. JENNINGS, S. J.	WESTINGHOUSE ELECTRIC CORP.	-	-	-	-	-	-	SOTA TURBINE SUPERVISORY INSTRUMENTATION AND A NOVEL LASER LIGHT PROBE TO MEASURE MOISTURE IN LOW-PRESSURE TURBINE
A NEW METHOD FOR ON-LINE SURVEILLANCE OF NUCLEAR POWER REACTORS BASED ON DECISION THEORY	SAEDTLER, E.	FEDERAL REPUBLIC OF GERMANY	-	-	-	-	-	-	METHOD FOR THE AUTOMATIC MONITORING OF REACTOR OPERATIONAL STATES BASED UPON DECISION THEORY
1975 IEEE INTERCON CONFERENCE RECORD		IEEE	-	-	-	-	-	-	VARIETY OF PAPERS PRESENTED AT THE 1975 INTERNATIONAL CONVENTION AND EXPOSITION OF THE IEEE, APRIL 1975
GAS TEMPERATURE-DENSITY (GTD) SENSOR FOR TURBINE INLET GAS TEMPERATURE MEASUREMENT	VANROBERTS, J. ROHY, D. A.	AIR FORCE FLIGHT DYNAMICS LABORATORY			1				B - RADIATION DENSITY - TEMPERATURE MEASUREMENT IN AIRCRAFT TURBINES
ADVANCES IN MEASURING TECHNIQUES FOR TURBINE COOLING TEST RIGS: STATUS REPORT	POLLACK, F. G.	NASA LEWIS RESEARCH CENTER			3				OPTICAL TEMPERATURE SENSORS AND ROTATING MEASUREMENT SYSTEMS
TURBINE BLADE PYROMETER SYSTEM IN THE CONTROL OF THE CONCORDE ENGINE	CURWEN, K. R.	KOLLSMAN INSTRUMENT LIMITED		1					PYROMETRIC TEMPERATURE SENSING SYSTEM FOR AIRCRAFT TURBINE BLADES
AN ULTRASONIC TURBINE INLET GAS TEMPERATURE SENSOR	SMALL, L. L. LONGSTREE, C. S.	BENDIX CORP.			1				ULTRASONIC TEMPERATURE SENSOR FOR AIRCRAFT GAS TURBINE
ENGINE CONDITION MONITORING AS A PART OF THE PROPULSION MANAGEMENT CONCEPT	SIBLEY, R. K.	PRATT & WHITNEY AIRCRAFT		1					AIRCRAFT ENGINE CONDITION MONITORING SYSTEM
INFLIGHT ENGINE CONDITION MONITORING SYSTEM	VANCLEVE, G. C.	DETROIT DIESEL ALLISON		1					AIRCRAFT ENGINE CONDITION MONITORING SYSTEM
FLOWMETER FOR SMALL ATTITUDE CONTROL PROPULSION SYSTEMS	THOMPSON, R. J. JR.	ROCKETDYNE	1						CANTILEVER STRAIN GAGE-TYPE FLOWMETER

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY **NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT: SOTA* ROCKET	IN-FLIGHT: SOTA NONROCKET	IN-FLIGHT: NOVEL**	BETWEEN-FLIGHT: SOTA ROCKET	BETWEEN-FLIGHT: SOTA NONROCKET	BETWEEN-FLIGHT: NOVEL	REMARKS
THE IN-LINE OIL MONITOR AND ITS ROLE IN ENGINE CONDITION MONITORING	SKALA, G. F.	ENVIRONMENT/ONE CORP.		1					CONTINUOUS OIL CONDITION AND PARTICULATE MONITORING FOR AIRCRAFT ENGINES
ADVANCED TORQUE MEASUREMENT SYSTEMS TECHNIQUE FOR AIRCRAFT TURBOSHAFT ENGINES	SCOPPE, F. E.	AVCO LYCOMING		1					AIRCRAFT TURBOSHAFT ENGINE TORQUE MEASURING SYSTEM
PERFECT MEACHINES REPLACE FALLIBLE MEN? CAVEAT EMPTOR!	NATKIN, H.	ELECTRONIC COMPONENT NEWS	-	-	-	-	-	-	OVERVIEW OF THE USE OF AUTOMATIC TEST EQUIPMENT
TRENDS - AN AUTOMATIC GAS TURBINE DIAGNOSTIC SYSTEM	PASSALACQUA, J. R.	HAMILTON STANDARD DIVISION OF UNITED AIRCRAFT	-	-	-	-	-	-	DEVELOPMENT, OPERATION AND PERFORMANCE OF AN AUTOMATIC ENGINE CONDITION MONITORING SYSTEM CALLED TRENDS
ON VEHICLE MOBILITY MEASUREMENT AND RECORDING SYSTEM	CHIN, F. K. WATTS, R.	GENERAL AMERICAN TRANSPORTATION CORPORATION AND MVCT-MAINTENANCE U.S. ARMY TANK AUTOMOTIVE COMMISSION		6					ON-BOARD ENGINE CONDITION MONITORING SYSTEM FOR U.S. ARMY M35A2, 2-1/2 TON CARGO TRUCK
TURBINE ENGINE SENSORS FOR HIGH TEMPERATURE APPLICATIONS	SMALL, L. L.	USAF AERO PROPULSION LABORATORY			4				NOVEL TURBINE ENGINE TEMPERATURE SENSORS INCLUDED: 1. FLUIDIC TEMPERATURE SENSOR USING EDGETONE RESONATOR 2. INFRARED PYROMETER 3. ULTRASONIC GAS GAP SENSOR 4. ELECTRON BEAM SENSOR
A HIGH SPEED AIRBORNE DATA ACQUISITION AND CONTROL SYSTEM WITH AN INTEGRATED DIGITAL COMPUTER	TROVER, W. F.	TELEDYNE CONTROLS COMPANY	-	-	-	-	-	-	AIFIDS-4000 SYSTEM FOR USE IN AIRCRAFT AND SYSTEM FLIGHT TEST
CALORIMETER PROBES FOR MEASURING HITHER THERMAL FLUX-IEEE 1979 INSTRUMENTED AEROSPACE SIMULATION	RUSSEL, L. D.	AMES RESEARCH CENTER		1					EXPENDABLE, TUNGSTEN-CAP CALORIMETER PROBE FOR MEASURING EXTREMELY HIGH HEAT FLUXES (10-30 KW/CM^2) IN ARC JET FACILITIES USED FOR SIMULATING PLANETARY ENTRY HEATING CONDITIONS

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TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT			BETWEEN-FLIGHT			REMARKS
			SOTA* ROCKET	SOTA NONROCKET	NOVEL**	SOTA ROCKET	SOTA NONROCKET	NOVEL	
OVERVIEW OF TRANSDUCERS AND SENSORS FOR DIAGNOSTICS	TOBIN, H. G.	IIT RESEARCH INSTITUTE	-	-	-	-	-	-	AN OVERVIEW OF SENSORS USED IN DIAGNOSTIC TECHNIQUES FOR AUTOMOTIVE PRUPOSES. TEMPERATURE, PRESSURE, VIBRATION AND ELECTRICAL IGNITION SYSTEM SENSORS ARE DISCUSSED
PROBE MEASUREMENTS IN FLAMES (EXPERIMENTAL DIAGNOSTICS IN GAS PHASE COMBUSTION SYSTEMS)	BOWMAN, C. T.	STANFORD UNIVERSITY	-	-	-	-	-	-	TEMPERATURE, SPECIES CONCENTRATION AND VELOCITY PROBES USED FOR MEASUREMENT IN LABORATORY AND INDUSTRIAL FLAMES
HYDRAULIC DIAGNOSTIC MONITORING SYSTEM	DUZICH, J. J.	GRUMMAN AEROSPACE CORPORATION		10	1				DIAGNOSTIC MONITORING SYSTEM FOR A HYDRAULIC FLIGHT SIMULATOR. SYSTEM WARNS OF IMPENDING FAILURE OF HYDRAULIC SYSTEM COMPONENTS BY ON-BOARD SENSORS. ONE NOVEL SENSOR WAS A FIBER-OPTIC APPROACH USED FOR DETECTING THE PRESENCE OF LIQUID IN A HIGH-PRESSURE PNEUMATIC BOTTLE.
STUDY OF ADVANCED AUTOMATIC DIAGNOSTIC/PROGNOSTIC TEST EQUIPMENT FOR MAINTENANCE OF MILITARY AUTOMOTIVE VEHICLES (REPORT NO. A-4712, TASK 53)	CRESWICK, F. A. WYLER, E. N.	BATTELLE, COLUMBUS LABORATORIES	-	-	-	-	-	-	REVIEW OF CURRENT TECHNOLOGY FOR AUTOMATIC DIAGNOSTIC/PROGNOSTIC TEST EQUIPMENT FOR USE IN MILITARY VEHICLE MAINTENANCE
SOME PROBLEMS OF EXPLOITATION OF JET TURBINE AIRCRAFT ENGINES OF LOT POLISH AIR LINES	SLODOWNIK, A.	TECHNIKA LOTNICZA	-	-	-	-	-	-	MENTIONS THE USE OF A RADIOACTIVE ISOTOPE FOR DETERMINING THE WEAR OF ENGINE ROTOR BEARINGS AND TURBINE TIPS ON COMMERICAL JET AIRCRAFT
SPACE SENSOR LOCATION AND ATTACHMENT	MAYER, T. C. SUTPHIN, H. W. HARRINGTON, J. T.	PARKS COLLEGE OF ST. LOUIS UNIVERSITY	-	-	-	-	-	-	THIS REPORT DISCUSSES THE SHOCK PULSE VIBRATION TECHNIQUE FOR DETECTING BEARING WEAR IN HELICOPTER GEAR BOXES. PLACEMENT AND MOUNTING METHODS FOR THE ACCELEROMETERS ARE DESCRIBED
A STATUS REPORT ON SENSORS AND THEIR APPLICATION TO BEARING CONDITION MONITORING (MECHANICAL FAILURES PREVENTION GROUP MEETING NO. 18, NOVEMBER 8 TO 10, 1972)	WHITTIER, R. M.	ENDEVCO	-	-	-	-	-	-	DISCUSSION OF ACOUSTIC EMISSION, VIBRATION SENSORS AND PIEZOELECTRIC TRANSDUCERS FOR BEARING CONDITION MONITORING

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TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
RESONANT STRUCTURE TECHNIQUES FOR BEARING FAULT ANALYSIS	BURCHILL, R. F.	MECHANICAL TECHNOLOGY, INC.	1			1			SENSING SYSTEM TO DETECT BALL BEARING FAILURE FOR A SPACE GYRO APPLICATION USING A BROAD BAND, MINIATURE ACCELEROMETER (INCLUDED A FAULT DETECTION CIRCUIT WITH 28K Hz FILTER AND ENVELOPE DETECTION DEVICE)
INSTRUMENTATION III MEDICAL EQUIPMENT	SHACKIL, A. F.	IEEE SPECTRUM JANUARY 1981	-	-	-	-	-	-	TWO NEW DEVICES DEVELOPED BY THE MEDICAL FIELD CALLED COMPUTERIZED AXIAL TOMOGRAPHY (CAT) AND POSITRON EMISSION TOMOGRAPHY PROVIDE BIOCHEMICAL AND STRUCTURAL INFORMATION. THESE SCANNERS ARE NONINVASIVE IMAGING SYSTEMS. A THIRD DIAGNOSTIC SYSTEM DESCRIBED IS A NUCLEAR MAGNETIC RESONANCE IMAGING SYSTEM.
DIAGNOSTICS OF WEAR IN AERONAUTICAL SYSTEMS	WEDEVEN, L. D.	NASA LEWIS RESEARCH CENTER		5					SOTA DETECTION TECHNIQUES FOR OIL ANALYSIS: 1. SOAP (SPECTROMETRIC OIL ANALYSIS PROGRAM) 2. CHIP DETECTORS 3. FERROGRAPHY 4. IN-LINE OIL MONITOR 5. RADIOACTIVE ISOTOPE TAGGING
AN EXPERIMENTAL INVESTIGATION OF CYLINDRICAL ROLLER BEARINGS HAVING ANNULAR ROLLERS	SUZUKI, A. SEIREG, A.	UNIVERSITY OF WISCONSIN TRANS OF ASME OCTOBER 1976	-	-	-	-	-	-	RADIOACTIVE TRACING OF BALL BEARINGS WITH GAMMA RADIATION AND USE OF A SCINTILLATION DETECTOR AND COUNTER TO MEASURE CHANGE IN RADIOACTIVITY OF BEARINGS AND THUS PRODUCE A MEASURE OF BEARING WEAR
F15/F100 ENGINE DIAGNOSTIC SYSTEM	SPETH, R. H. SCOTT, B. C. ROMOSER, B. K.	McDONNEL AIRCRAFT CO., PRATT & WHITNEY AIRCRAFT		20					ENGINE MONITORING AND DIAGNOSTIC SYSTEM TO DETECT AND DIAGNOSE ENGINE MALFUNCTIONS AND IDENTIFY FAULTY COMPONENTS
AIDS - AIRCRAFT INTEGRATED DATA SYSTEM	HUGHES, I.	HAMILTON STANDARD DIVISION OF UNITED TECHNOLOGIES CORP.	-	-	-	-	-	-	AIDS FUNCTION IS TO PROVIDE ON-BOARD MONITORING OF ENGINES, AIRCRAFT SYSTEMS AND AIRCRAFT PERFORMANCE

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TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT: SOTA* ROCKET	IN-FLIGHT: SOTA NONROCKET	IN-FLIGHT: NOVEL**	BETWEEN-FLIGHT: SOTA ROCKET	BETWEEN-FLIGHT: SOTA NONROCKET	BETWEEN-FLIGHT: NOVEL	REMARKS
A TRANSMITTER FOR DIAGNOSTIC IMAGING (VOL. 90, PROCEEDINGS OF THE PHOTO-OPTICAL INSTRUMENTATION ENGINEERS, AUGUST, 1976)	WANG, K. CHANGE, H. SHEN, H. WADE, G. SU, K. LO, K. ELLIOT, S.	UNIVERSITY OF HOUSTON UNIVERSITY OF CALIFORNIA SANTA BARBARA	-	-	-	-	-	-	ACOUSTIC IMAGING SYSTEM USING AN OPTO-ACOUSTIC TRANSDUCER (OAT) TO PRODUCE REAL-TIME ORTHOGRAPHIC DIAGNOSTIC IMAGING IN THE MEDICAL FIELD
EXPERIMENTAL DETERMINATION OF TRANSIENT STRAIN IN A THERMALLY-CYCLED SIMULATED TURBINE BLADE UTILIZING A NONCONTACT TECHNIQUE	CALFO, F. D. BIZON, P. T.	NASA LEWIS RESEARCH CENTER			1				A NONCONTACTING ELECTRO-OPTICAL EXTENSOMETER USED TO MEASURE DISPLACEMENT BETWEEN PARALLEL TARGETS MOUNTED ON LEADING EDGE OF SIMULATED TURBINE BLADE. THIS METHOD COULD BE EXTREMELY USEFUL IN DEVELOPMENT AND EVALUATION OF A THEORY FOR PREDICTING THERMAL FATIGUE LIFE OF STRUCTURAL COMPONENTS
CONCEPT FORMULATION STUDY FOR AUTOMATIC INSPECTION, DIAGNOSTIC AND PROGNOSTIC SYSTESM (AIDAPS) FINAL REPORT, VOL. 1	NORTHROP CORP. ELECTRONICS DIVISION	NORTHROP CORP.		1 SYSTEM					THIS PAPER PRESENTS THE RESULTS OF A CONCEPT FORMULATION STUDY FOR AN AUTOMATIC INSPECTION, DIAGNOSTIC AND PROGNOSTIC SYSTEM (AIDAPS) FOR ARMY AIRCRAFT
INTEGRATED ENGINE INSTRUMENT SYSTEM	SKOVHOLT, R. L.	GENERAL ELECTRIC COMPANY		1 SYSTEM					INTEGRATED ENGINE INSTRUMENT SYSTEM (IEIS) IS A COMPUTER DRIVEN DISPLAY AND PROCESSING SYSTEM FOR MONITORING AIRCRAFT ENGINE CONDITION. THIS REPORT COVERS THE ESTABLISHMENT OF REQUIREMENTS AND SYSTEM DESIGN OF IEIS
IMPROVED CAPABILITIES TO DETECT INCIPIENT BEARING FAILURE	ALCORTA, J. A. PACKER, L. L.	PRATT & WHITNEY AIRCRAFT GROUP			1				LOW-LEVEL RADIATION TECHNIQUE, USING IRON-55 AS THE RADIOACTIVE TAG, FOR DETECTION OF WEAR IN GAS TURBINE ENGINE MAINSHAFT BEARINGS. A GAS FLOW PROPORTIONAL COUNTER WITH COSMIC GUARD DETECTOR AND BACKGROUND SHIELDING CONSTITUTES THE LOW-LEVEL RADIOACTIVE MEASURING DEVICE FOR THE IRON-55 COUNTING

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TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
F101 (PV) OPERATION AND SERVICE MANUAL	--	GENERAL ELECTRIC		10	2				TWO NOVEL IN-FLIGHT SENSORS: 1. T4B PYROMETER, AN INFRARED RADIATION SENSING DEVICE (CONSISTING OF A SILICONE CHIP PROTODIODE SENSOR AND ELECTRONICS PACKAGE) PRODUCING AN OUTPUT THAT IS AN EXPONENTIAL OF TURBINE BLADE TEMPERATURE 2. FLAME SENSOR WHICH IS AN UNTRAVIOLET RADIATION SENSING DEVICE USED FOR DETECTING THE PRESENCE OF FLAME AT THE FLAMEHOLDER IN THE AUGMENTER
INTEGRATED ENGINE INSTRUMENT SYSTEM FINAL TECHNICAL REPORT PHASE IV	SKOVHOLT, R.	GENERAL ELECTRIC		1 SYSTEM					AREAS OF SYSTEM DESIGN, HUMAN FACTORS AND DISPLAY EQUIPMENT FOR THE IEIS PROGRAM ARE DESCRIBED IN THIS FINAL REPORT. DIGITAL PROCESSING TECHNIQUES FOR CONVERTING ANALOG PYROMETER AND ACCELEROMETER OUTPUTS TO DIGITAL WORDS FOR FURTHER PROCESSING.
SETE WORKSHOP PROCEEDINGS, ADVANCED TECHNIQUES FOR AUTOMATIC TESTING AND BUILT-IN TEST EQUIPMENT (BITE) FOR TEST, MEASUREMENT AND DIAGNOSTIC EQUIPMENT (TMDE)	GOODMAN, D. M.	FLEET MISSILE SYSTEMS	-	-	-	-	-	-	OVERVIEW OF R&D ACTIVITIES IN AUTOMATION AND BUILT-IN TEST EQUIPMENT SPONSORED BY DOD, NASA, DEPARTMENT OF COMMERCE AND INDUSTRY.
FEASIBILITY STUDY FOR A HYDROGEN GAS LEAK DETECTION SYSTEM AS REQUIRED FOR USE ON IN-FLIGHT EXPERIMENTS - FINAL REPORT	VARADI, P. ADAIR, R. SHABECK, J.	RAYTHEON COMPANY, SPACE AND INFORMATION SYSTEMS DIVISION			1				A VARADI MASS SPECTROMETER TUBE DESIGNED FOR USE IN A FLIGHT HYDROGEN LEAK DETECTOR SYSTEM. THE SYSTEM DETECTED LESS THAN 1% HYDROGEN IN A HELIUM ATMOSPHERE AT 10^{-5} TORR.
CONCEPT FORMULATION STUDY FOR AUTOMATIC INSPECTION, DIAGNOSTIC AND PROGNOSTIC SYSTEMS (AIDAPS) APPENDIX F-AIDAPS PARAMETER LISTS	--	NORTHRUP CORPORATION, ELECTRONICS DIVISION							INSTRUMENT PARAMETER LISTS FOR ARMY AIRCRAFT WHICH DEFINE THE INTERFACING REQUIREMENTS BETWEEN MAINTENANCE REQUIREMENTS AND AIDAPS DATA COLLECTION FUNCTIONS.

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TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
INSTRUMENTATION AND PROCESS CONTROL DEVELOPMENT FOR IN-SITU COAL GASIFICATION TWENTIETH QUARTERLY REPORT: SEPTEMBER THROUGH NOVEMBER 1979	GLASS, R. E.	THERMAL PROCESSES DIVISION, SANDIA NATIONAL LABORATORIES	-	-	-	-	-	-	SANDIA NATIONAL LABORATORIES TESTING OF AN INVERTED THERMOCOUPLE AND A SURFACE ELECTRICAL RESISTIVITY NETWORK FOR IN-SITU COAL GASIFICATION EXPERIMENTS
UH-1H AIDAPS TEST BED PROGRAM VOLUME I AND II	PROVENZANO, J. GAMES, J WYROSTEK, A. OSTHEIMER, A. YOUNG, J.	HAMILTON STANDARD		1 SYSTEM					SOTA HARDWARE TO PROVIDE AUTOMATIC INSPECTION, DIAGNOSTIC AND PROGNOSTIC MAINTENANCE FUNCTIONS ON SLECTED UH-1H HELICOPTER SYSTEMS.
NONDESTRUCTIVE INSPECTION PRACTICES, VOLUME I	BOLIS, E. EDITOR	NATO AGARDOGRAPH NO. 201	-	-	-	-	-	-	THE FOLLOWING SOTA TECHNIQUES FOR NONDESTRUCTIVE EVALUATION OF MATERIALS ARE DISCUSSED: 1. RADIOGRAPH 2. MAGNETIC PARTICLE 3. LIQUID PENETRANT 4. EDDY CURRENT 5. ULTRASONIC 6. ACOUSTIC EMISSION 7. HOLOGRAPHIC METHODS
USAAMRDL TECHNICAL REPORT 72-59 ADVANCED ENGINE CONTROL PROGRAM	WHITE, A. H. WILLS, D. F.	COLT INDUSTRIES CHANDLER-EVANS INC., CONTROL SYSTEMS DIVISION		8					AN ADVANCED ELECTRONIC ENGINE CONTROL SYSTEM FOR SMALL TURBO-SHAFT ENGINES. INSTRUMENTATION FEATURES A RADIATION PYROMETER MEASURING TURBINE BLADE TEMPERATURE - UTILIZES A FLEXIBLE FIBER OPTIC CABLE TO LINK HOT ZONE APERATURE ASSEMBLY TO DETECTOR ASSEMBLY
INSTRUMENTATION FOR NONCONTRACT IC ENGINE TEST AND MONITORING	HADDEN, S. C. HULLS, L. R. SUTPHIN, E. M.	RCA GOVERNMENT AND COMMERCIAL SYSTEM/AUTOMATED SYSTEMS DIVISION	-	-	-	-	-	-	A SINGLE NONCONTRACTING TRANSDUCER AND SPECIAL PURPOSE CIRCUITRY WHICH EXTRACTS ENGINE SPEED INFORMATION AND PERFORMS SPECTRAL ANALYSIS FOR DIAGNOSTIC PURPOSES FOR INTERNAL COMBUSTION ENGINES.
A RADIATION PYROMETER DESIGNED FOR IN-FLIGHT MEASUREMENT OF TURBINE BLADE TEMPERATURES (SOCIETY OF AUTOMOTIVE ENGINEERS)	BARBER, R.	LAND PYROMETERS, INC.		1					DESCRIBES THE PRINCIPLE OF OPERATION AND DESIGN OF A RADIATION PYROMETER DEVELOPED TO MEASURE SURFACE TEMPERATURES OF TURBINE BLADES DURING FLIGHT THE PYROMETER CAN MEASURE TEMPERATURES ABOVE 1300 F TO AN ACCURACY OF ±10 F.

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TABLE 22. (CONTINUED)

TITLE	AUTHOR	SOURCE	IN-FLIGHT: SOTA* ROCKET	IN-FLIGHT: SOTA NONROCKET	IN-FLIGHT: NOVEL**	BETWEEN-FLIGHT: SOTA ROCKET	BETWEEN-FLIGHT: SOTA NONROCKET	BETWEEN-FLIGHT: NOVEL	REMARKS
EFFECTIVENESS OF THE REAL TIME FERROGRAPH AND OTHER OIL MONITORS AS RELATED TO OIL FILTRATION	POPGOSHEV, D. VALORI, R.	NAVAL AIR PROPULSION CENTER			1				AN OIL MONITOR KNOWN AS A REAL TIME FERROGRAPH USED FOR DETECTING ROLLING CONTACT FATIGUE OR SCORING-TYPE FAILURES. FERROGRAPH IS EFFECTIVE IN DETECTING FAILURES WHEN OIL FILTRATION LEVEL IS ABOVE 40 MICROMETERS.
FIBER OPTIC AND LASER DIGITAL PRESSURE TRANSDUCER	MARGERUM, G. W. LEONARD, J. W. FURS, A. E.	NAVAL POST-GRADUATE SCHOOL			2				TWO FIBER OPTIC PRESSURE TRANSDUCERS: 1. FIBER OPTICAL DEVICE MEASURING OUTPUT LIGHT FLUX FROM A DIAPHRAGM WHICH IS A MEASURE OF PRESSURE. 2. DIGITAL PRESSURE TRANSDUCER EMPLOYING MODULATION OF LASER POWER BY USE OF A MIRROR ATTACHED TO THE SENSING DIAPHRAGM.
STATUS OF THE EVALUATION OF A CORIOLIS EFFECT MASS FLOWMETER FOR DENSE PHASE COAL FLOWS	BAUCUM, W. E.	UNIVERSITY OF TENNESSEE, SPACE INSTITUTE			1				MASS FLOWMETER UTILIZING CORIOLIS FORCES GENERATED BY FLOW OF A SUBSTANCE TO MEASURE THE MASS WHICH GENERATES THE FORCE
SURFACE ACOUSTIC WAVE UNDERWATER SOUND SENSORS	STAPLES, E. J. WISE, J SCHOENWALD, J. S. LIM, T. C.	ROCKWELL INTERNATIONAL, ELECTRONICS RESEARCH CENTER			1				ACOUSTICAL TYPE OF UNDERWATER SOUND DETECTOR USING SURFACE ACOUSTIC WAVE RESONATOR CONTROLLED OSCILLATORS
DIGITAL QUARTZ PRESSURE TRANSDUCERS FOR FLIGHT APPLICATIONS	PAROS, J. M.	PAROSCIENTIFIC, INC.			1				DIGITAL QUARTZ PRESSURE TRANSDUCERS USED ON THE F-111 AIRCRAFT IN THE INTEGRATED PROPULSION CONTROL SYSTEM
WIEGAND EFFECT: A NEW PULSE GENERATING OPTION	—	SOCIETY OF AUTOMOTIVE ENGINEERS, INC.			1				A NOVEL TRANSDUCER YIELDING DIGITAL PULSES IN RESPONSE TO MOTION, THE WIEGAND MODULE HAS SUCH POTENTIAL APPLICATIONS AS IGNITION TRIGGERS AND TACHOMETERS AND SPEEDOMETERS IN THE AUTOMOTIVE INDUSTRY.

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TABLE 22. (CONTINUED)

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TASK COMPLETION REPORT SSME FLIGHT INSTRUMENTATION 12 JUNE 1980 RI/RD80-170	SSME ENGINEERING	ROCKWELL INTERNATIONAL, ROCKETDYNE DIVISION			2				SURVEY AND ANALYSIS OF DIGITAL PRESSURE TRANSDUCERS AND FIBER OPTICAL SPEED SENSORS FOR SSME APPLICATIONS.
NERVA NUCLEAR SUBSYSTEM INSTRUMENTATION	SWOPE, R. R.	WESTINGHOUSE ELECTRIC CORPORATION			1				A NOVEL HIGH TEMPERATURE THERMOCOUPLE UTILIZING TUNGSTEN/TUNGSTEN-26% RHENIUM THERMOCOUPLE WIRE IN A MOLYBDENUM SHEATH WITH B&O VITRIFIED BEADS FOR INSULATION (TEMPERATURE RANGE 492 TO 4785 R)
TURBINE ENGINE INSPECTION WITHOUT DISASSEMBLY	McCORD, R. M.	PRATT & WHITNEY AIRCRAFT GROUP	-	-	-	-	-	-	FIBROSCOPE USED FOR TURBINE ENGINE INSPECTION.
APPLICATIONS OF ELECTRO-OPTICAL INSTRUMENTATION	ALWANG, W. G.	PRATT & WHITNEY AIRCRAFT GROUP			13				NOVEL GAS TURBINE ELECTRO-OPTICAL INSTRUMENTATION INCLUDES: 1. OPTICAL PYROMETERS 2. RAMAN SCATTERING VIBRATION AND STRAIN 1. HOLOGRAPHY 2. SPECKLE PHOTOGRAPHY 3. DIFFRACTION GRATINGS 4. ROTOR BLADE TIP ORIENTATION USING OPTICAL SENSORS 5. REFLECTED LASER BEAM FOR DETERMINING ROTOR BLADE VIBRATORY MODE SHAPES 6. OPTICAL HETERODYNING CLEARANCE AND DISPLACEMENT 1. OPTICAL PROXIMITY PROBES-INTENSITY TYPE 2. OPTICAL PROBES-TRIANGULATION TYPE 3. IMAGING TYPES OF DISPLACEMENT SENSORS FLOW 1. HOLOGRAPHIC FLOW VISUALIZATION 2. LASER VELOCIMETRY.

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TABLE 22. (CONTINUED)

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A NOVEL FIBER-OPTIC TEMPERATURE PROBE	DARKIN, J. P. KAHN, D. A.	PLESSEY RADAR RESEARCH CENTER			1				FIBER OPTIC TEMPERATURE PROBE CONSISTING OF A SILICA FIBRE WAVEGUIDE TERMINATED BY AN OPAQUE SHIELD (FOR USE IN 400 TO 1100 C RANGE)
FF41-A-2/A7E INFLIGHT ENGINE CONDITION MONITORING SYSTEM (IECMS)	DeMOTT, L. R.	DETROIT DIESEL ALLISON GMC		14					INFLIGHT ENGINE CONDITION MONITORING SYSTEM FOR THE TF41-A-2 AIRCRAFT GAS TURBINE ENGINE.
AIDS - EXPECTATIONS PAST, PRESENT AND FUTURE	ALLISON, J. W. DIECKMAN, T. W.	PRATT & WHITNEY AIRCRAFT GROUP	-	-	-	-	-	-	THIS PAPER DISCUSSES THE AIDS HARDWARE INSTALLED, SIGNIFICANT PARAMETERS MONITORED, AND AIDS PROGRAM HIGHLIGHTS FOR COMMERCIAL AIRCRAFT.
JT9D-7A(SP) JET ENGINE PERFORMANCE DETERIORATION TRENDS	RICHTER, G. P.	LEWIS RESEARCH CENTER	-	-	-	-	-	-	THIS PAPER PRESENTS A DISCUSSION OF THE TEST PROGRAM AND THE RESULTS OF THE DATA ANALYSIS CONDUCTED ON THE P&W JT9D JET ENGINE
NEUTRON RADIOGRAPHIC NONDESTRUCTIVE EVALUATION OF AEROSPACE STRUCTURES	DANCE, W. E.	ADVANCED TECHNOLOGY CENTER, INC.			1				
IMPROVED CAPABILITIES TO DETECT INCIPIENT BEARING FAILURE	ALCORTA, J. A. PACKER, L. L.	PRATT & WHITNEY			1				
APPLICATIONS OF ELECTROMAGNETIC ACOUSTIC TRANSDUCERS	ALERS, G. A.	UNIVERSITY OF NEW MEXICO		1					
METHOD FOR MEASURING THE SIZE AND VELOCITY OF SPHERES BY DUAL-BEAM LIGHT-SCATTER INTERFEROMETRY	BACHALO, W. D.	SPECTRON DEVELOPMENT LABORATORIES, INC.		1					
AN INSTRUMENT FOR SPRAY DROPLET SIZE AND VELOCITY MEASUREMENT	BACHALO, W. D. HESS, C. F. HARTWELL, C. A.	SPECTRON DEVELOPMENT LABORATORIES, INC.		1					
VISIBILITY OF LARGE SPHERES OBSERVED WITH A LASER VELOCIMETER: A SIMPLE MODEL	FARMER, W. M.	UNIVERSITY OF TENNESSEE, SPACE INSTITUTE		1					

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TABLE 22. (CONCLUDED)

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LASER - RAMAN DIAGNOSTICS OF TEMPERATUER AND NUMBER DENSITY IN THE MIXING REGION OF A ROCKET ENGINE EXHAUST AND A COFLOWING AIR STREAM	WILLIAMS, W. D., ET AL.	ARO, INC., ARNOLD AIR FORCE STATION			1				
TURNABLE DIODE LASER SULFURIC ACID STACK MONITORING SYSTEM	PEARSON, E. F., MANTZ, A. W.	LASER ANALYTICS		1					
COMBUSTION GAS MEASUREMENTS USING TURNABLE LASER ABSORBTION SPECTROSCOPY	HANSON, R. K.	STANFORD UNIVERSITY							
HIGH RESOLUTION SPECTROSCOPY OF COMBUSTION GASSES USING A TURNABLE IR DIODE LASER	HANSON, R. K., KNUTZ, P. A., KNIGER, G. H.	STANFORD UNIVERSITY		1					
ROTATING MACHINERY ROLLING ELEMENT BEARING PERFORMANCE USING THE FIBER OPTIC METHOD	PHILLIPS, G. J.	NAVAL SHIP RESEARCH AND DEVELOPMENT CENTER			1				
A STUDY OF PLASTIC DEFORMATIONS BY EXO-ELECTRON EMISSION	BAXTER, W. J.	GENERAL MOTORS RESEARCH LABORATORY		1					
ACOUSTIC EMISSION TECHNOLOGY 1979	GREEN, A. T.	ACOUSTIC EMISSION TECHNOLOGY CORPORATION		1					
A NEW TECHNOLOGY FOR BEARING PERFORMANCE MONITERING	PHILLIPS, G. J.	NAVAL SHIP RESEARCH AND DEVELOPMENT CENTER			1				FIBER-OPTIC MONITORING OF RACE DEFORMATIONS
ULTRASONIC MASS FLOWMETER FOR ARMY AIRCRAFT ENGINE DIAGNOSTICS	LYNWORTH, L. C. PEDERSEN, N. E. CARNEVALE, E. N.	PANAMETRICS, INC.			1				NONINTRUSIVE FLOWMETER FOR GAS TURBINE ENGINES
AN OPTICAL GAGE FOR STRAIN/DISPLACEMENT MEASUREMENT AT HIGH TEMPERATURE NEAR FATIGUE CRACK TIPS	SHARPE, W. N. JR. MARTIN, D. R.	MICHIGAN STATE UNIVERSITY, DIVISION OF ENGIEERING RESEARCH			1				OPTICAL HIGH TEMPERATURE STRAIN/DISPLACEMENT MEASUREMENT
ADVANCED TORQUE MEASUREMENT SYSTEM, FINAL REPORT	CHANGE, DR. J. D. KUKEL, DR. J.	GARRETT AIR RESEARCH		1					AIRCRAFT TURBOSHAFT ENGINE, MEASURING SYSTEM
HIGH RESPONSE LASER FLOWMETER FINAL REPORT	BALZEY, R. N. SCHNEIDER, J. R.	SPERRY RAND GYROSCOPE DIVISION			1				PULSED ROCKET LASER FLOWMETER

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY **NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

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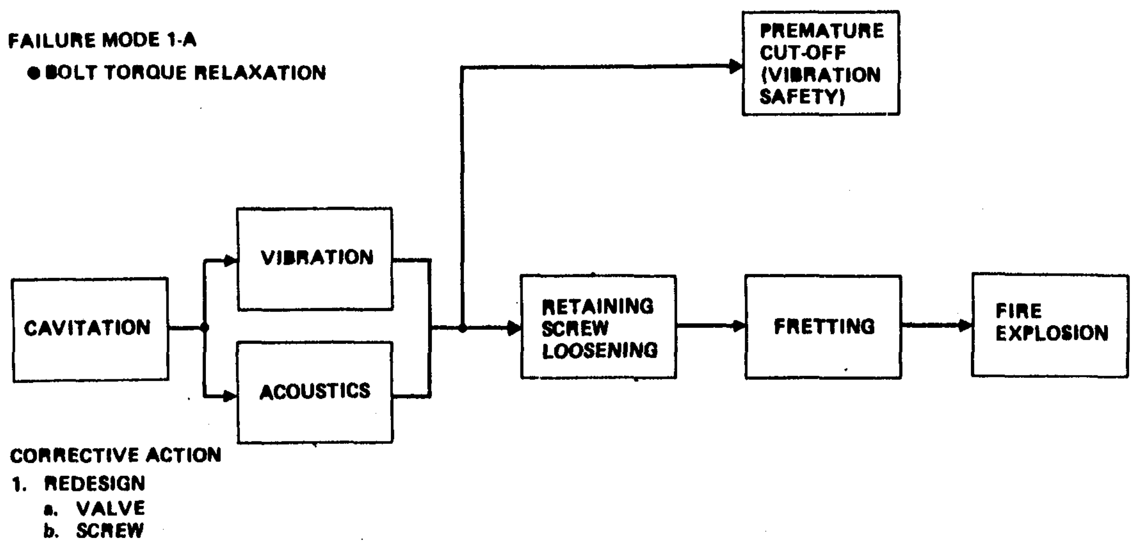
APPENDIX F. IN-FLIGHT AND BETWEEN-FLIGHT MEASURANDS FOR DETECTION OF FAILURE

APPENDIX F

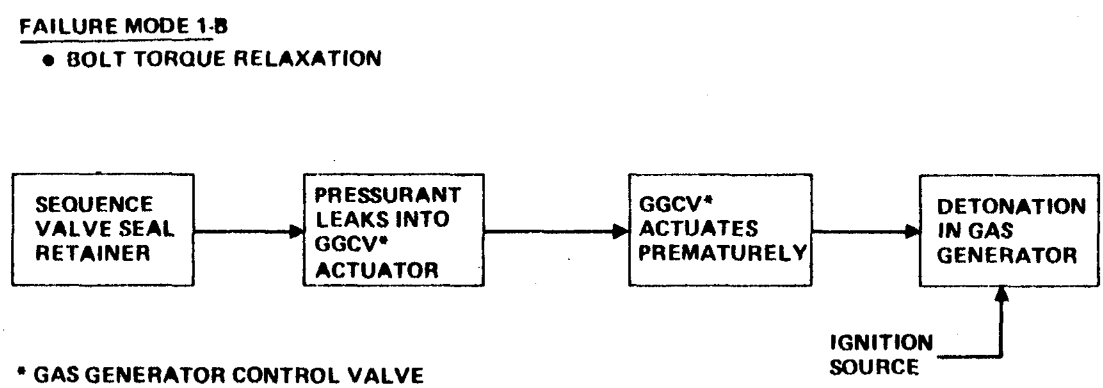
IN-FLIGHT AND BETWEEN-FLIGHT MEASURANDS FOR DETECTION OF FAILURE

The diagrams in this section show the in-flight and between-flight measurands which can be used for detection of each failure mode from Task I and shown in Appendix C.

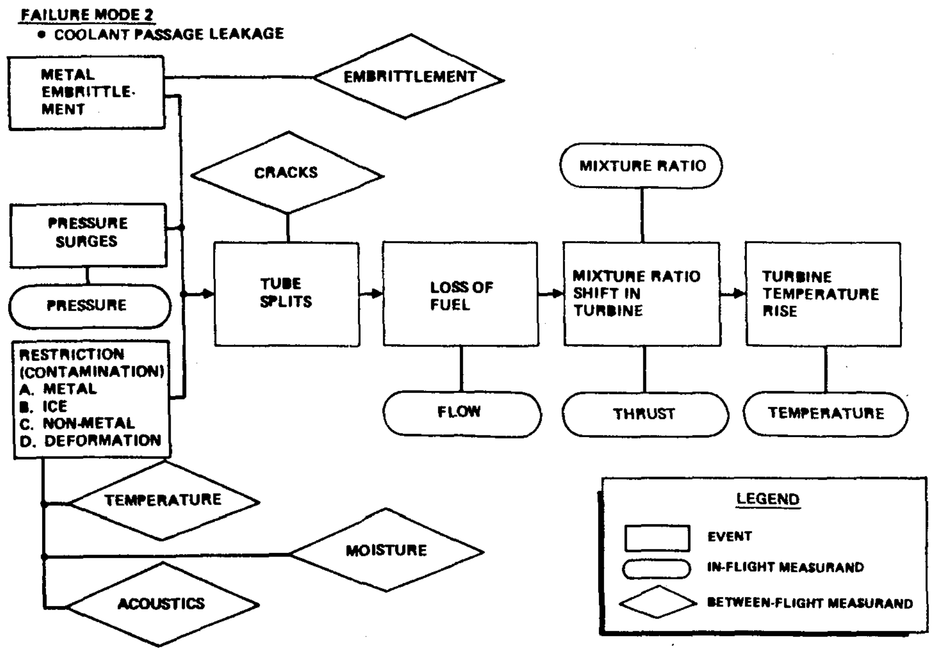
The 16 failure modes covered herein were determined by an analysis of 86,000 actually experienced failures.



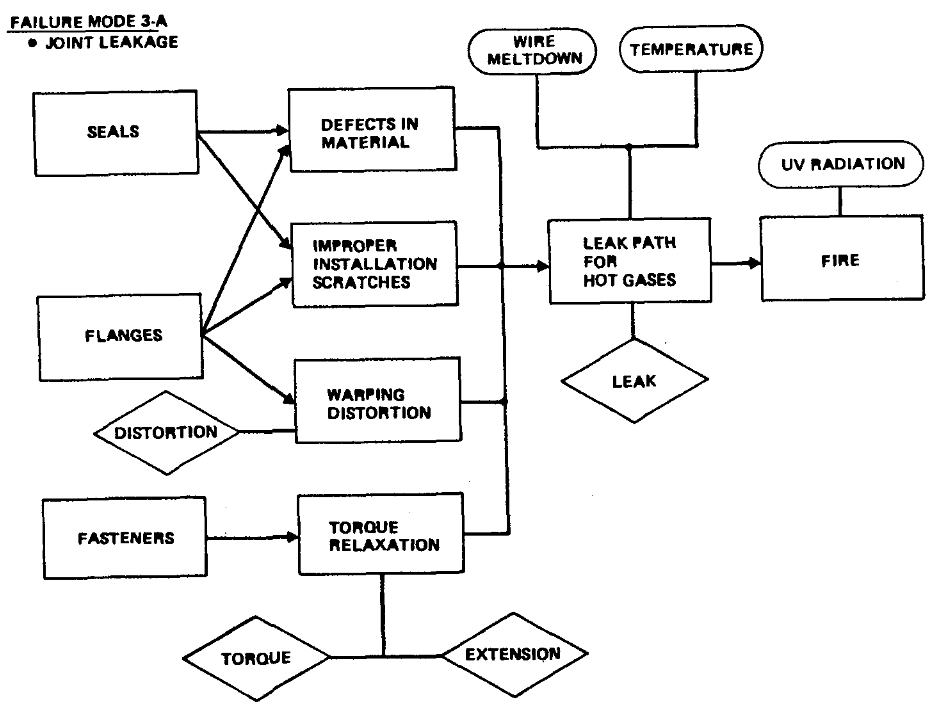
Main Oxidizer Valve Failure



Main Oxidizer Valve Failure

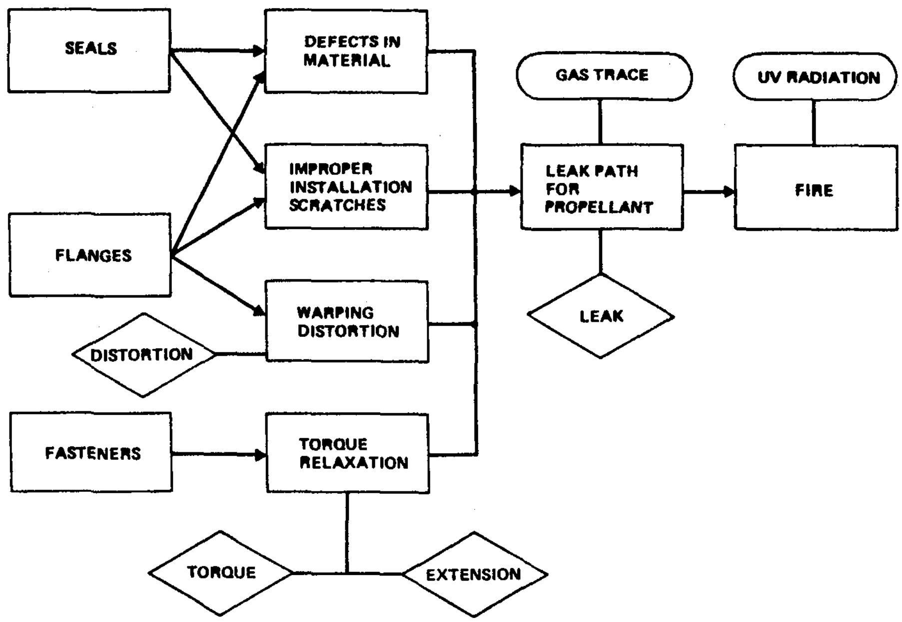


Nozzle-Combustor Failure



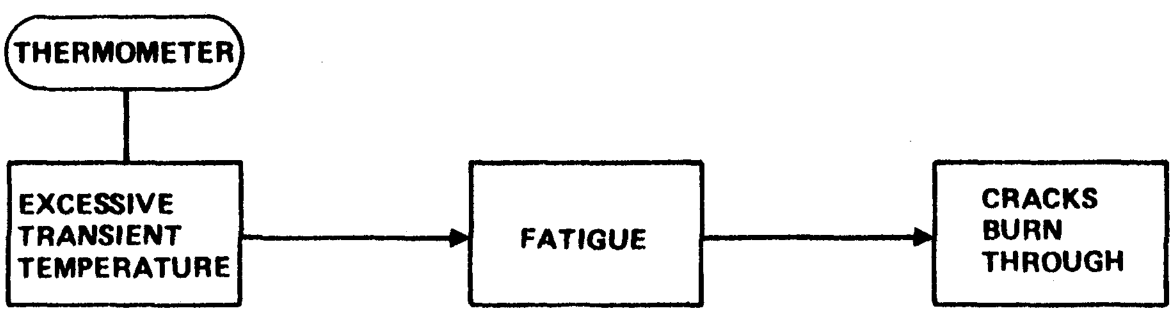
Hot-Gas Leakage Failure

FAILURE MODE 3-B
- JOINT LEAKAGE



Propellant Leakage Failure

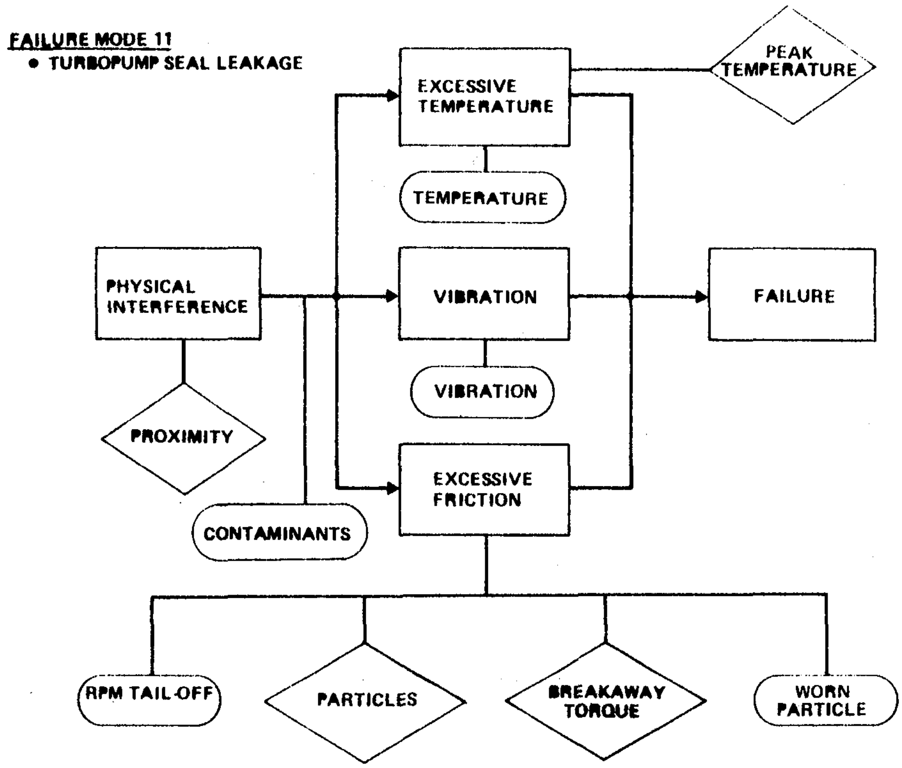
FAILURE MODE 4
- **TRANSFER TUBE CRACKS**



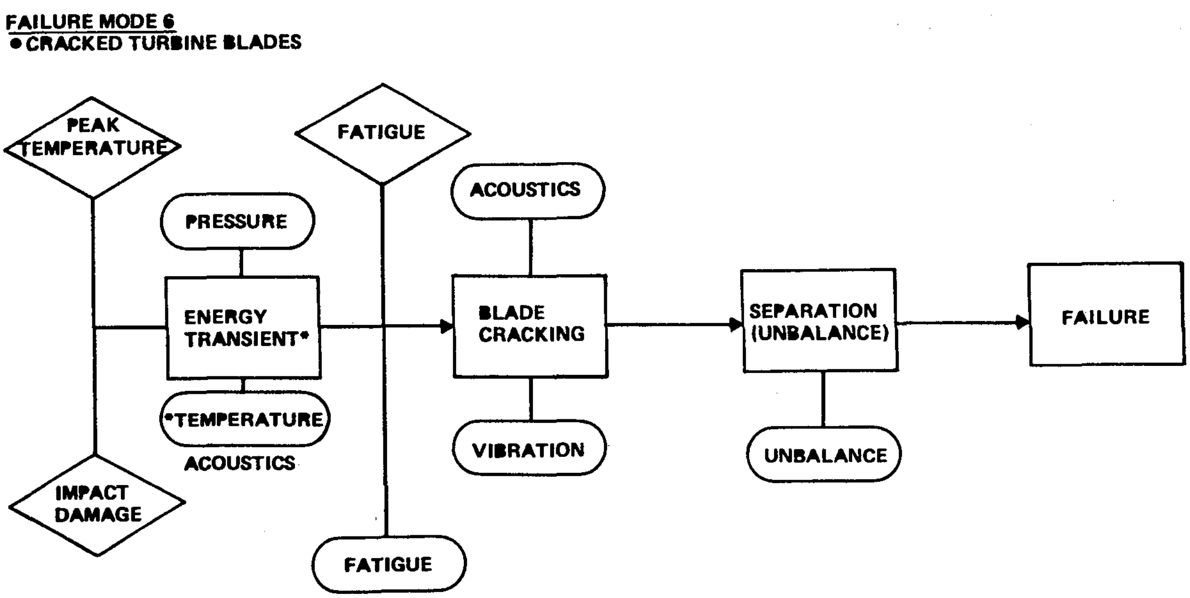
CORRECTIVE ACTION

1. REDESIGN

Hot Gas Manifold Failure



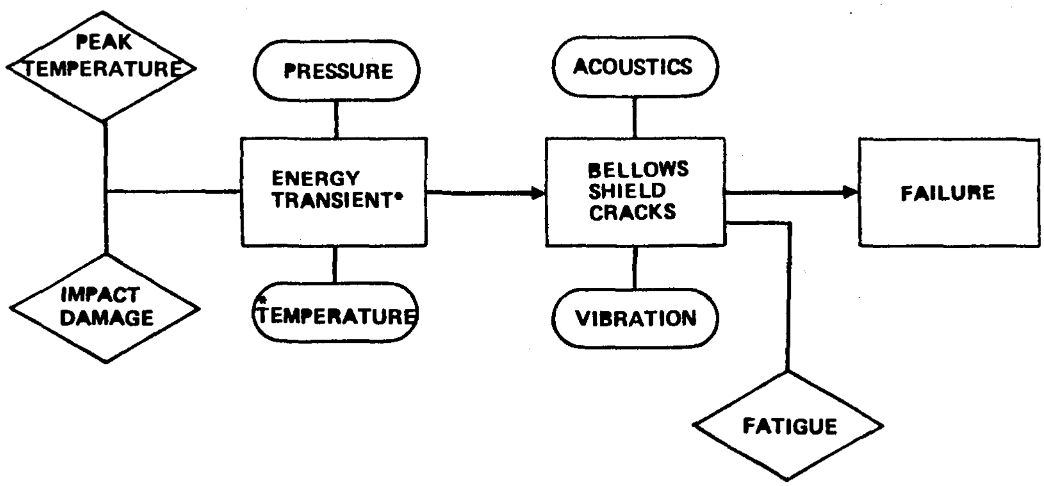
Propellant Turbopump Labyrinth Seal Failure



Turbine Blade Failure

FAILURE MODE 7

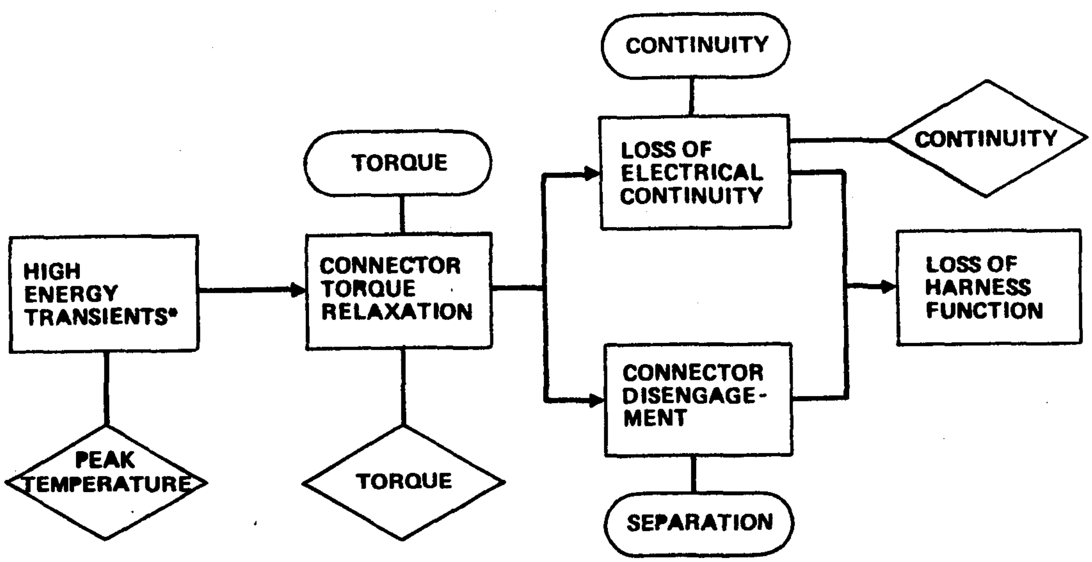
- CRACKED CONVOLUTION, BELLOWS & SHIELD



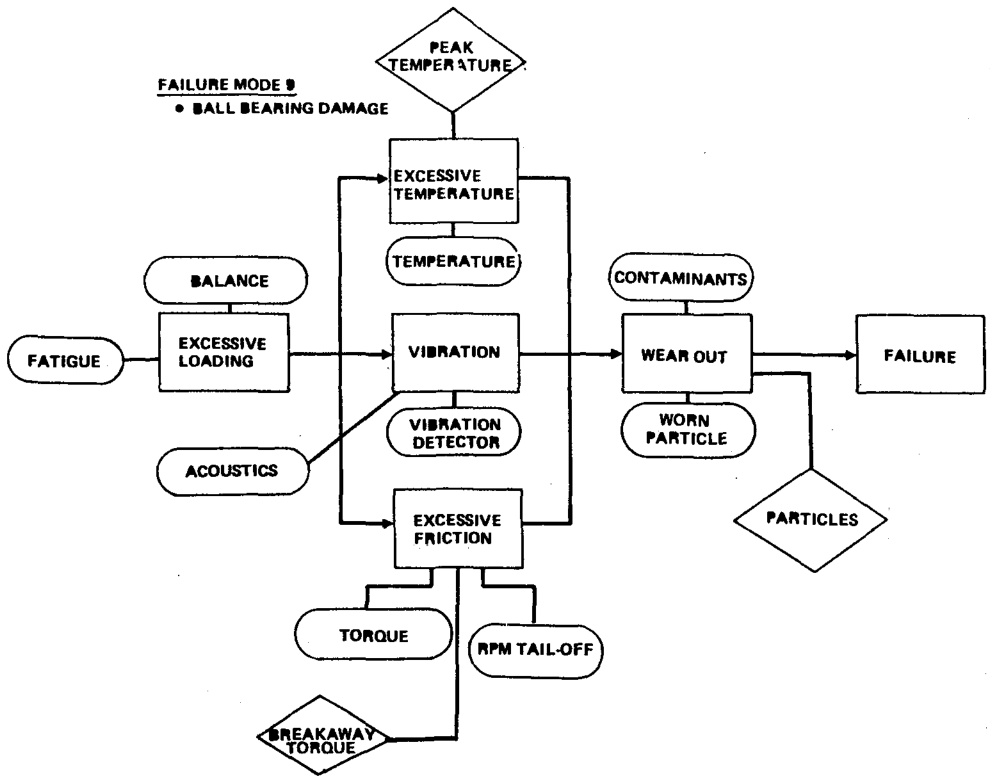
Cracked Convolutions, Bellows and Shields

FAILURE MODE 8

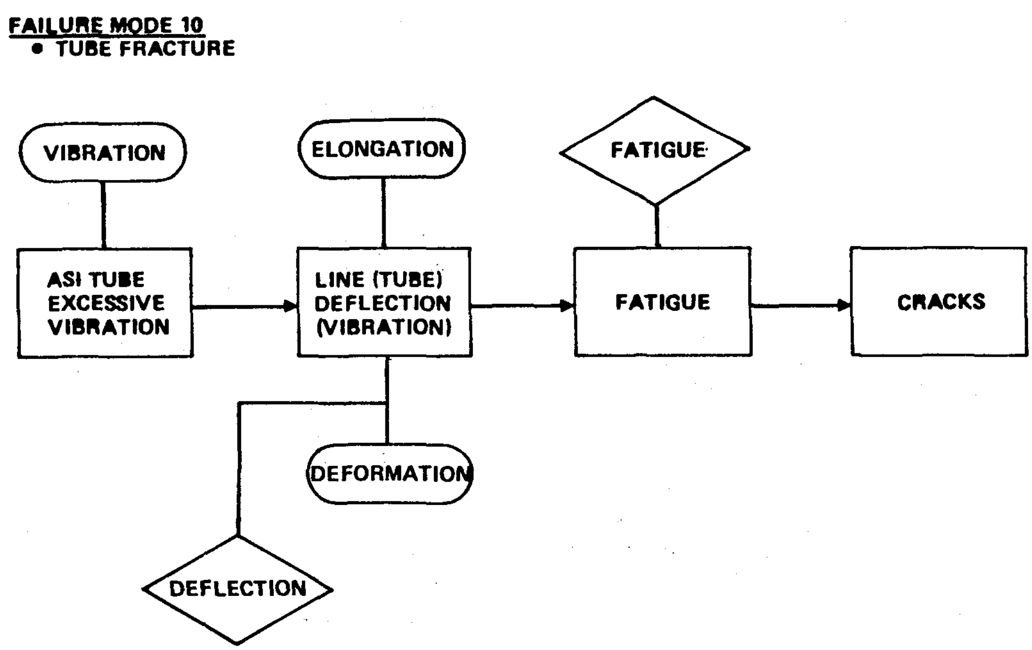
- LOOSE ELECTRICAL CONNECTORS



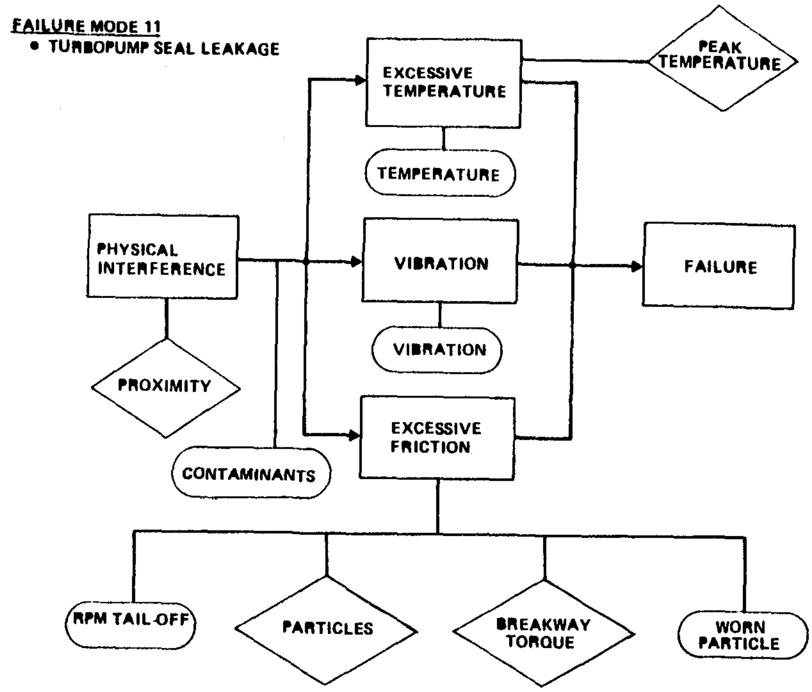
Loose Electrical Connectors



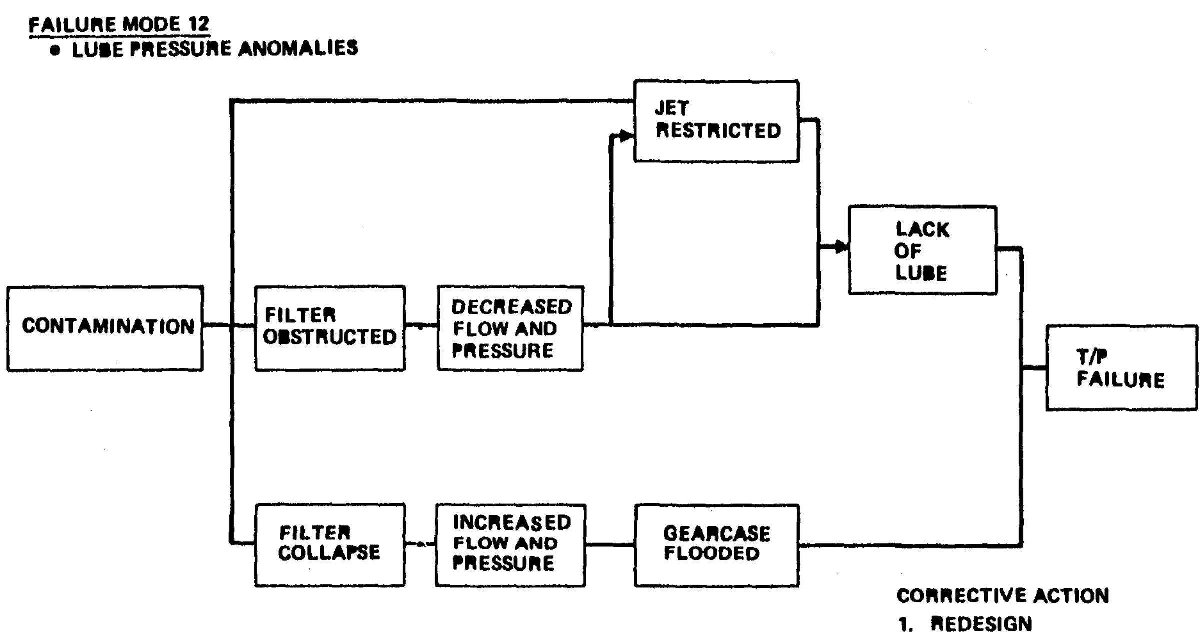
Ball-Bearing Damage



ASI Propellant Line (Tube) Failure

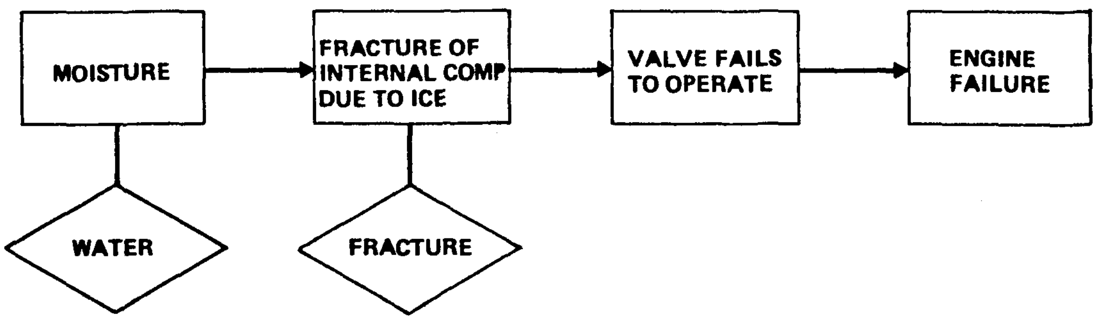


Primary Turbopump Seal Failure



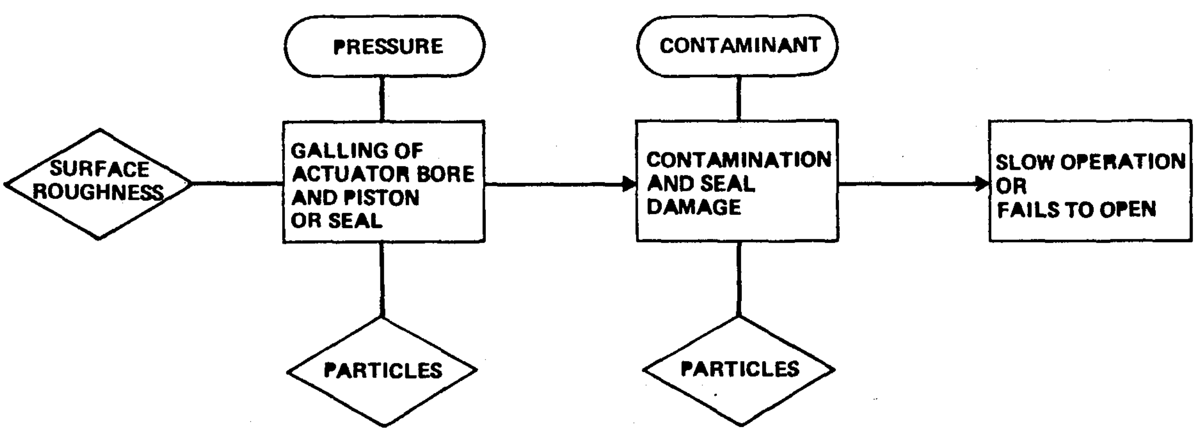
Failure due to Lube Pressure Anomalies

FAILURE MODE 13-A
• VALVE FAILS TO PERFORM



Oxidizer Poppet Valve Failure

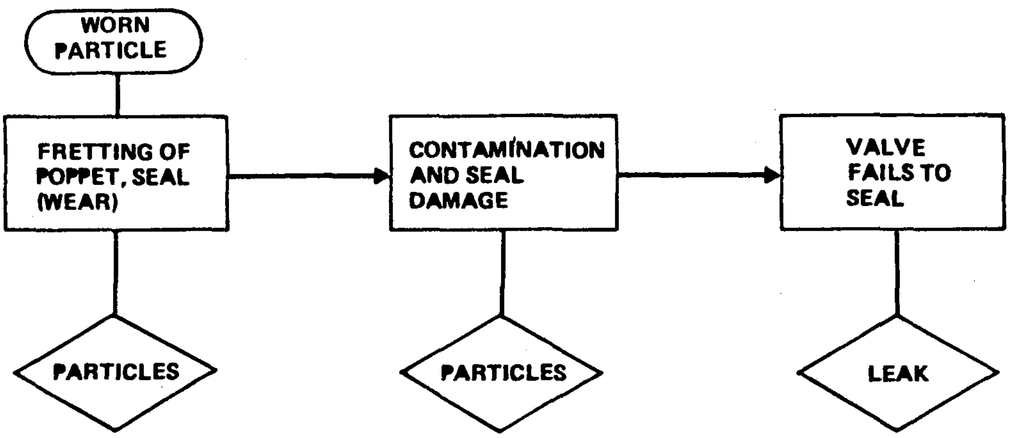
FAILURE MODE 13-B
• VALVE FAILS TO PERFORM



Main Propellant Valve Failure

FAILURE MODE 14-A

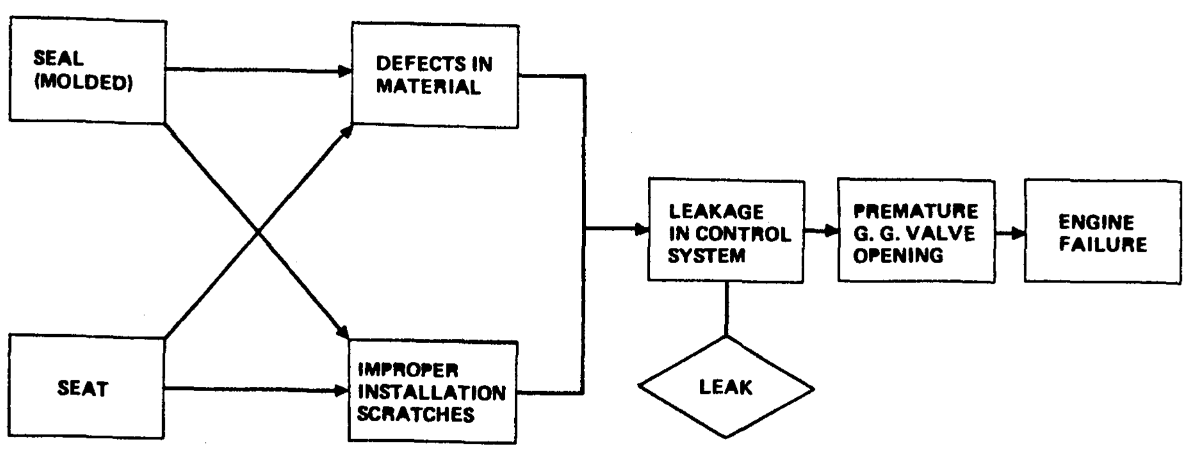
• **INTERNAL LEAKAGE**



Poppet Valve Failure

FAILURE MODE 14-B

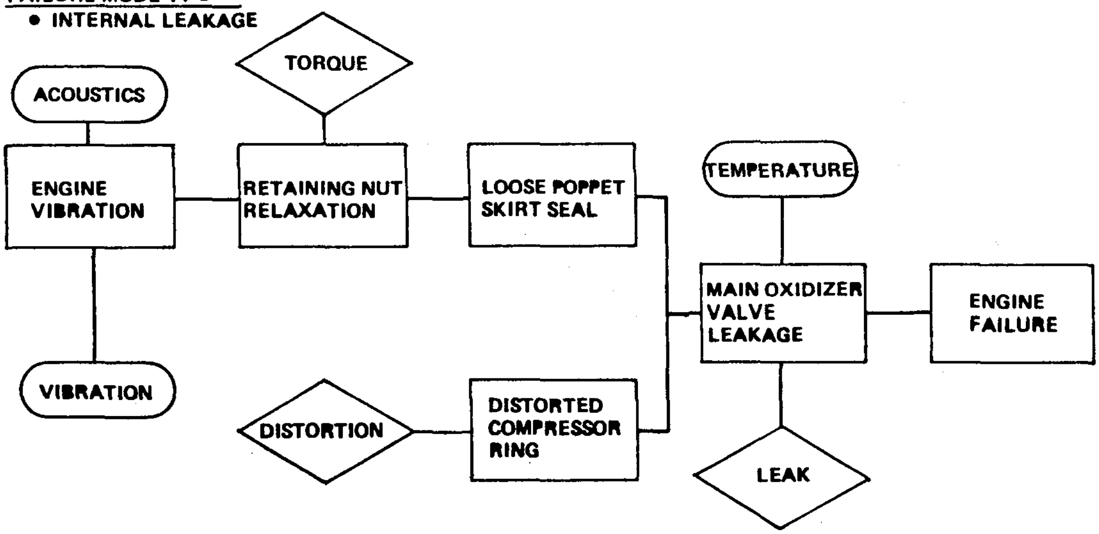
• **INTERNAL VALVE LEAKAGE**



MOV Sequence Valve Failure

FAILURE MODE 14-C

- INTERNAL LEAKAGE



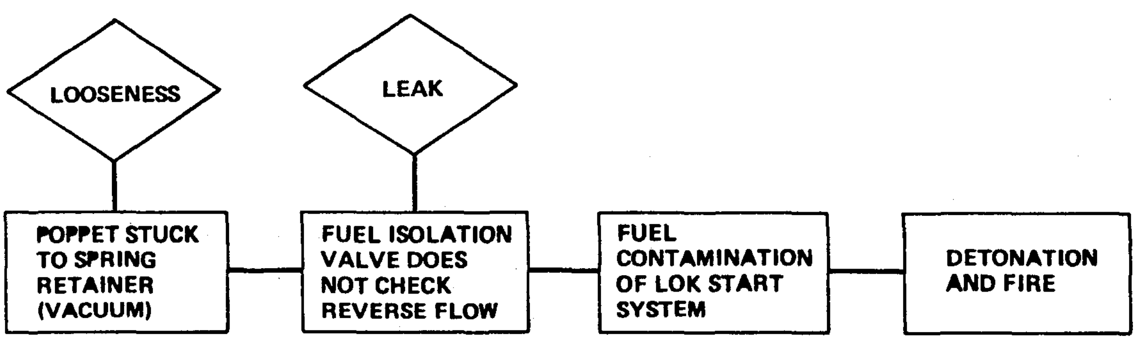
CORRECTIVE ACTION:

1. REDESIGN

Main Oxidizer Valve Failure

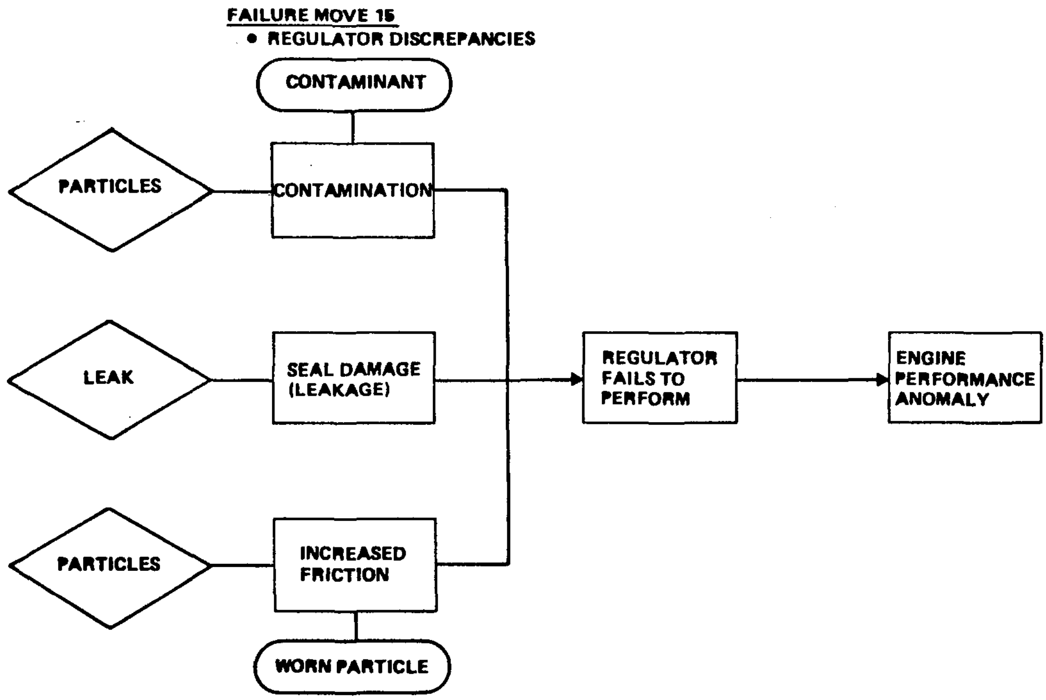
FAILURE MODE 14-D

- INTERNAL LEAKAGE (TRAPPED PRESSURE)

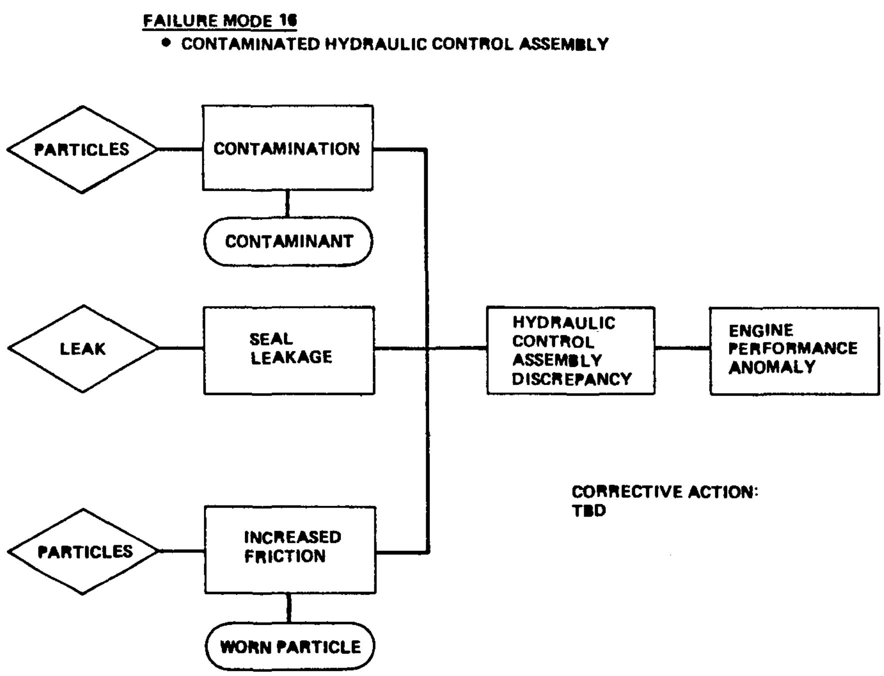


CORRECTIVE ACTION:
REDESIGN

Redundant Isolation Valve Failure



Regulator Failure



Hydraulic Control Assembly Failure

APPENDIX G. IN-FLIGHT CONDITION MONITORING SENSOR GRADING

APPENDIX G

IN-FLIGHT CONDITION MONITORING SENSOR GRADING

In this section, candidate sensors are graded with respect to ideal sensors for their usefulness in detecting each class of rocket engine failure. The grading is based on the lumped technical and economic descriptors described in Table 23, including physical, electronic and functional requirements, detectability, durability, safety and cost. The development time required for each sensor to be used for the given application is also listed, and an overall grade is given, again with respect to an ideal sensor.

TABLE 23. LUMPED DESCRIPTORS

TECHNICAL REQUIREMENTS:

- PHYSICAL
  - WEIGHT
  - SPACE
  - STRENGTH
  - MATERIALS
  - CHEMICALS
  - RESONANCE
  - FATIGUE
- ELECTRONIC
  - POWER, COMSUMPTION
  - VOLTAGE
  - CURRENT
  - WIRING
  - FILTERING
  - AMPLIFICATION
  - ANALOG/DIGITAL
  - MEMORY REQUIREMENTS
  - LINEARIZATION
  - SHIELDING
- FUNCTIONAL
  - INTRUSIVE
  - POWER LOSS (PARASITIC)

TECHNICAL FEATURES:

- DETECTABILITY
  - SPEED
  - ACCURACY
  - REPEATABILITY
  - SENSITIVITIY
  - RESOLUTION
  - DRIFT
  - ARTIFACTS
  - SUSCEPTIBILITY
- SAFETY
  - FAILSAFE
  - FAILURE EFFECTS
- DURABILITY
  - RECALIBRATION
  - INSPECTION
  - LIFE

ECONOMICAL:

- EXPENDITURES
  - R&D
  - INTEGRATION

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF COOLANT PASSAGE LEAKAGE/RESTRICTION (NO. 2)

DESCRIPTORS / SENSORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				TOTAL
	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	$[1]	$[1]	$[1]	10	0	10	110
PRESSURE SENSORS														
QUARTZ, DIGITAL	7	3	18	6	18	8	60	50	250	300	7	1	9	76
FIBEROPTIC	7	2	18	6	18	8	59	200	250	450	5	3	7	71
LASER DIGITAL	7	3	18	6	18	7	59	300	250	550	4	4	6	69
SAW, DIGITAL	7	3	18	6	18	7	59	200	250	450	5	2	8	72
ULTRASONIC THERMOMETER, FLAME	6	5	20	12	20	6	69	100	200	300	7	3	7	83
ULTRASONIC FLOWMETER, NOZZLE	10	5	20	6	20	9	70	50	150	200	8	2	8	86
POLAROGRAPH	2	4	10	14	10	4	44	250	450	700	3	6	4	51
TUNABLE DIODE LASER SPECTROMETER MIXTURE RATIO	8	5	19	12	19	7	60	300	300	600	4	6	4	68

1 - IN THOUSANDS

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF HIGH-TURBOPUMP TORQUE (NO. 5)

DESCRIPTORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				
SENSORS	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	\$[1]	\$[1]	\$[1]	10	0	10	110
RTD THERMOMETER	6	8	18	4	18	4	58	50	200	250	7	2	8	73
OPTICAL TACHOMETER	6	8	20	12	15	8	69	200	200	400	6	2	8	83
ACCELEROMETER	9	5	20	1	20	9	64	70	150	220	7	3	7	78
ISOTOPE WEAR DETECTOR	7	5	20	18	18	7	75	500	500	1000	0	6	4	79
HYDROPHONE	7	5	20	6	18	6	62	50	150	200	8	2	8	78
FERROMAGNETIC TORQUE-METER	5	10	20	15	15	10	76	500	400	900	1	7	3	80
TUNABLE DIODE LASER SPECTROMETER	8	5	19	16	18	8	74	300	300	600	4	6	4	82

1 - IN THOUSANDS

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF CRACKED TURBINE BLADE (NO. 6)

DESCRIPTORS / SENSORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				TOTAL
	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	$[1]	$[1]	$[1]	10	0	10	110
PRESSURE SENSORS[2]														
QUARTZ DIGITAL	9	3	18	6	18	8	62	50	250	300	7	1	9	78
FIBEROPTIC	9	2	18	6	18	8	61	200	250	450	5	3	7	73
LASER DIGITAL	9	3	18	6	18	7	61	300	250	550	4	4	6	71
SAW DIGITAL	9	3	18	6	18	7	61	200	250	450	5	2	8	74
PYROMETER	8	5	20	14	16	9	72	100	300	450	6	2	8	86
VIBRATION[3]	9	5	20	1	20	9	63	20	250	270	7	1	9	79
HYDROPHONE[3]	8	5	20	6	18	6	62	50	250	300	7	2	8	77
FIBEROPTIC BEARING DETECTOR	6	7	20	10	18	8	69	200	400	600	4	3	7	80
EXO-ELECTRON DETECTOR	3	4	10	15	12	7	51	300	300	600	4	7	3	58
EDDY CURRENT DETECTOR	3	4	10	12	12	8	49	50	300	300	6	3	7	62
EMAT DETECTOR[4]	3	4	10	14	12	8	51	150	300	450	5	4	6	62

1 - IN THOUSANDS
2 - TRANSIENTS
3 - CHIPPED BLADE
4 - EMAT - ELECTROMAGNETIC ACOUSTIC TRANSDUCER

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF CRACKED CONVOLUTIONS, BELLOWS, SHIELDS (NO. 7)

DESCRIPTORS / SENSORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				TOTAL
	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	$[1]	$[1]	$[1]	10	0	10	110
PRESSURE SENSORS														
QUARTZ, DIGITAL	9	3	18	6	18	8	62	50	250	300	7	1	9	78
FIBEROPTIC	9	2	18	6	18	8	61	200	250	450	5	3	7	73
LASER, DIGITAL	9	3	18	6	18	7	61	300	250	550	4	4	6	71
SAW DIGITAL	9	3	18	6	18	7	61	200	250	450	5	2	8	74
RTD THERMOMETER	6	8	18	4	18	4	58	50	200	250	7	2	8	73
ACCELEROMETER	9	5	18	1	18	9	62	70	150	220	7	3	7	76
GYDROPHONE	7	5	18	10	16	6	62	50	150	200	8	2	8	78

1 - IN THOUSANDS

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF BALL BEARING FEATURES (NO. 9)

DESCRIPTORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				TOTAL
SENSORS	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	$[1]	$[1]	$[1]	10	0	10	110
OPTICAL TACHOMETER	6	8	20	12	15	8	69	200	200	400	6	2	8	83
ISOTOPE DETECTOR	7	5	20	18	18	7	75	500	500	1000	0	6	4	79
FIBEROPTIC DETECTOR	6	7	20	19	18	8	77	200	400	600	4	3	7	88
RTD THERMOMETER	6	8	20	4	18	4	60	50	200	250	7	2	8	75
ACCELEROMETER	9	5	20	1	20	9	64	70	150	220	7	3	7	78
HYDROPHONE	7	8	20	10	18	6	69	50	150	200	8	2	8	85
FERROMAGNETIC TORQUEMETER	5	10	20	15	15	10	75	500	400	900	1	7	3	79
EXO-ELECTRON DETECTOR	3	4	10	15	12	7	51	300	500	800	2	7	3	56
TURNABLE DIODE LASER SPECTROMETER	8	5	19	16	18	8	74	300	300	600	4	6	4	82
EDDY CURRENT DETECTOR	3	4	10	10	12	8	47	50	300	350	6	3	7	60
EMAT[2]	3	4	10	12	12	8	49	150	300	450	5	4	6	60

1 - IN THOUSANDS
2 - EMAT - ELECTROMAGNETIC ACOUTIC TRANSDUCER

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS FOR DETECTION OF TURBINE SEAL LEAKAGE (NO. 11)

DESCRIPTORS / SENSORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				
	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	$[1]	$[1]	$[1]	10	0	10	110
RTD THERMOMETER	6	8	20	4	18	4	60	50	200	250	7	2	8	75
OPTICAL TACHOMETER	6	8	20	12	15	8	69	200	200	400	6	2	8	83
ACCELEROMETERS	9	5	20	1	20	8	64	70	150	220	7	3	7	78
ISOTOPE SEAL DETECTOR	7	5	20	18	18	7	75	500	500	1000	0	6	4	79
TUNABLE DIODE LASER SPECTROMETER	8	5	19	16	18	8	74	300	300	600	4	6	4	82

1 - IN THOUSANDS

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF VALVE FAILURES (NO. 13)

DESCRIPTORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				TOTAL
SENSORS	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	$[1]	$[1]	$[1]	10	0	10	110
PRESSURE SENSORS														
QUARTS, DIGITAL	9	3	18	6	18	8	62	50	250	300	7	1	9	78
FIBEROPTICS	9	2	18	6	18	8	61	200	250	450	5	3	7	73
LASER, DIGITAL	9	3	18	6	18	7	61	300	250	550	4	4	6	71
S.A.W., DIGITAL	9	3	18	6	18	7	71	200	250	450	5	2	8	74
ISOTOPE WEAR DETECTOR	7	5	16	18	18	7	71	300	300	600	4	6	4	79
TUNABLE DIODE LASER	8	5	19	16	18	8	74	500	500	1000	0	6	4	78

1 - IN THOUSANDS

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF INTERNAL LEAKS (NO. 14)

DESCRIPTORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				
SENSORS	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	\$[1]	\$[1]	\$[1]	10	0	10	110
ULTRASONIC THERMOMETER	6	5	20	12	20	6	69	100	200	300	7	3	7	83
ACCELEROMETERS	9	5	20	1	20	9	64	70	150	220	7	3	7	78
ISOTOPE DETECTOR	7	5	20	18	18	7	75	500	500	1000	0	6	4	79
HYDROPHONE	7	5	20	6	18	6	62	50	150	200	8	2	8	78
TUNABLE DIODE LASER SPECTROMETER	8	5	19	16	18	8	74	300	300	600	4	6	4	82

1 - IN THOUSANDS

TECHNICAL AND ECONOMIC GRADING OF IN-FLIGHT DIAGNOSTIC SENSORS
FOR DETECTION OF REGULATOR FAILURES (NO. 15)

DESCRIPTORS / SENSORS	TECHNICAL							ECONOMICAL				DEVELOPMENT TIME		TOTAL
	REQUIREMENTS			FEATURES			TOTAL	EXPENDITURE		TOTAL				
	PHYSICAL	ELECTRONICS	FUNCTIONAL	DETECTABILITY	SAFETY	DURABILITY	TECHNICAL	R&D	INTEGRATION	ECONOMICAL	GRADE	YEARS	GRADE	OVERALL GRADE
PERFECT SCORE	10	10	20	20	20	10	90	$[1]	$[1]	$[1]	10	0	10	110
TURNABLE DIODE LASER SPECTROMETER	8	5	19	16	18	8	74	300	300	600	4	6	4	82
ISOTOPE WEAR DETECTOR	7	5	20	18	18	7	75	500	500	1000	0	6	4	79

1 - IN THOUSANDS

APPENDIX H. LITERATURE SURVEYED FOR BETWEEN-FLIGHT INSPECTION

APPENDIX H

LITERATURE SURVEYED FOR BETWEEN-FLIGHT INSPECTION

A survey was undertaken to find inspection technology which could be applicable to reusable rocket engines between flights. This survey included computer literature searches, periodical reviews, and personal visits. Representative literature was enumerated and the inspection techniques uncovered were then summarized. This appendix indicates the 56 documents that were tabulated as a result of this survey. The sequence shown represents only the order in which the literature was reviewed.

Table 24 is a tabulation indicating the title, author(s), source organization which actually performed the study or tests, level of development (Novel, Rocket Engine, or state-of-the-art), and remarks for each document.

Table 25 gives the reference information needed to obtain the literature, listed to correspond to the tabulation numbering of Table 24.

TABLE 24. SUMMARY OF LITERATURE SEARCH FOR BETWEEN-FLIGHT CONDITION MONITORING TECHNOLOGIES

NUMBER	TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
1	MAINTAINABILITY OF THE SPACE SHUTTLE ORBITER MAIN ENGINE	GOE, R.T.	ROCKETDYNE				3		1	EARLY SSME MAINTENANCE CONCEPTS
2	DIVERSIFICATION OF ACOUSTICAL HOLOGRAPHY AS A NONDESTRUCT INSPECTION TECHNIQUE TO DETERMINE AGING DAMAGE IN SOLID ROCKET MOTORS	COLLINS, DR. H.	HOLOSONICS, INC.						1	ACOUSTICAL IMAGING TECHNIQUES FOR CRACK DETECTION
3	WELDED ROTOR INSPECTION DEVELOPMENT PROJECT T55-J-027	SUSHIEL, J. VICTOR, S. PAUL, J.	AVCO LYCOMING					2		ULTRASONIC AND ACOUSTIC-EMISSION INSPECTION OF GAS TURBINE POWER SHAFTS
4.	USE OF LASER-POWERED OPTICAL PROXIMITY PROBE IN ADVANCED TURBOFAN ENGINE DEVELOPMENT	HARDY, H. D.	PRATT & WHITNEY AIRCRAFT						1	ROTATING COMPONENT CLEARANCE MEASUREMENT
5	ENGINE CONDITION MONITOR SYSTEM TO DETECT FOREIGN OBJECT DAMAGE AND CRACK DEVELOPMENT	HEGNER, H. R.	ITT RESEARCH INSTITUTE						2	DETECTION OF BLADE DAMAGE AND CRACK DEVELOPMENT IN AIRCRAFT ENGINES
6	A SYSTEMS ENGINEERING APPROACH TO EFFECTIVE ENGINE CONDITION MONITORING	LEIBY, D. W.	GENERAL ELECTRIC					1		INTEGRATED CONDITION MONITORING SYSTEM FOR AIRCRAFT ENGINES
7	FROM CRACKING CRACKS TO BREAKING BEAMS, A REVIEW OF ACOUSTIC EMISSION FOR AIRCRAFT STRUCTURE	BAILEY, C. D. LEWIS, W. H.	LOCKHEED - GEORGIA CO.						1	DETECTION OF CRACK INITIATION AND GROWTH IN AIRCRAFT STRUCTURES
8	STATE OF THE ART OF NONDESTRUCTIVE INSPECTION OF AIRCRAFT ENGINES	COMASSAR, D.M.	GENERAL ELECTRIC					3		RECENT DEVELOPMENTS IN ULTRASONIC, EDDY CURRENT, AND PENETRANT INSPECTIONS
9	HIGH RESOLUTION RADIOGRAPHY IN THE AERO-ENGINE INDUSTRY	PARISH, R. W.	AERE					1		X-RAY, GAMMA RAY, AND PARTICLE RADIOGRAPHY
10	WEAR DEBRIS ANALYSIS	PARR, N. L. RITCHIE, J.	ROYAL AIRCRAFT ESTABLISHMENT					1		LUBRICANT PARTICLE DETECTION AND ANALYSIS TECHNIQUES
11	HIGH RESOLUTION ULTRASONIC NONDESTRUCTIVE TESTING OF COMPLEX GEOMETRY COMPONENTS	MORAN, T. J.	AIR FORCE MATERIALS LABORATORY					1		DETECTION AND CHARACTERIZATION OF FLAWS

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY　　**NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

TABLE 24. (CONTINUED)

NUMBER	TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
12	NONDESTRUCTIVE METHODS FOR THE EARLY DETECTION OF FATIGUE DAMAGE IN AIRCRAFT COMPONENTS	GREEN, R.E., JR.	THE JOHN HOPKINS UNIVERSITY					4	2	SURVEY OF FATIGUE DAMAGE DETECTION METHODS
13	FEASIBILITY DEMONSTRATION OF USING PULSE LASER HOLOGRAPHIC TECHNIQUES TO INSPECT NAVAL AIRCRAFT ENGINE COMPONENTS	JACOBY, J. L. WRIGHT, J. E.	TRW SYSTEMS GROUP						1	INSPECTION OF TURBINE BLADES ON FULLY ASSEMBLED TURBINE WHEELS
14	ACOUSTIC EMISSION TECHNOLOGY 1979	GREEN, A. T.	ACOUSTIC EMISSION TECHNOLOGY CORP.						1	STATE OF ACOUSTIC EMISSION METHODS IN 1979
15	AN OPERATIONAL 150KV MICRO-FOCUS ROD ANODE X-RAY SYSTEM FOR NONDESTRUCTIVE TESTING	FONTIJN, L. A. PEUGEOT, R. S.	INSTITUTE OF APPLIED PHYSICS NETH. & RIDGE INSTRUMENT CO.					1		HIGH-SENSITIVITY X-RAY TECHNIQUE
16	HOLOGRAPHY AS A ROUTINE METHOD OF VIBRATION ANALYSIS	HOCKLEY, B. S. BUTTERS, J. N.	ROLLS-ROYCE, LTD.						1	HOLOGRAPHIC INSPECTION OF AIRCRAFT ENGINE COMPRESSOR BLADES AND TURBINE WHEELS
17	CORRELATIONS BETWEEN ADVANCE NONDESTRUCTIVE EVALUATION METHODS AND FRACTURE MECHANICS PARAMETERS	TELLER, C.M. ET. AL.	SOUTHWEST RESEARCH INSTITUTE					2		PULSE-ECHO SURFACE WAVE ULTRASONIC AND MAGNETIC PERTURBATION DETECTION OF FATIGUE CRACKS
18	FATIGUE DAMAGE DETECTION IN 2024 ALUMINUM ALLOY BY OPTICAL CORRELATION	HAWORTH, W. L. HEIBER, A. F. MUELLER, R. K.	WAYNE STATE UNIVERSITY					1		FATIGUE MONITORING USING OPTICAL HOLOGRAPHY
19	ACOUSTIC HARMONIC GENERATION DUE TO FATIGUE DAMAGE IN HIGH-STRENGTH ALUMINUM	MORRIS, W. L. BUCK, O. INMAN, R. V.	ROCKWELL INTERNATIONAL SCIENCE CENTER					1		FATIGUE DAMAGE DETECTION WITH ACOUSTIC SECOND HARMONIC GENERATION
20	STUDY OF A FLIGHT MONITOR JET ENGINE DISK CRACKS THE CRITICAL LENGTH CRITERION OF FRACTURE MECHANICS	BARRANGER, J. P.	NASA LEWIS RESEARCH CENTER				1		1	EDDY CURRENT DETECTION OF CRACKS AND PREDICTION OF FAILURE
21	APPLICATIONS OF ELECTRO-OPTICAL INSTRUMENTATION IN TURBINE ENGINE DEVELOPMENT	ALWAG, W. G.	PRATT & WHITNEY AIRCRAFT					5		REVIEW OF SOTA OPTICAL INSTRUMENTATION

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY

**NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

TABLE 24. (CONTINUED)

				IN-FLIGHT			BETWEEN-FLIGHT			
NUMBER	TITLE	AUTHOR	SOURCE	SOTA* ROCKET	SOTA NONROCKET	NOVEL**	SOTA ROCKET	SOTA NONROCKET	NOVEL	REMARKS
22	NONDESTRUCTIVE TESTING, A SURVEY	VARIOUS	SOUTHWEST RESEARCH INSTITUTE					10	3	A SURVEY OF NDE TECHNIQUES
23	FIBER-OPTIC CAMERA	RADDING, A.	UNKNOWN						1	TIP-MOUNTED FIBER-OPTIC CAMERA CONCEPT
24	A STUDY OF PLASTIC DEFORMATION BY EXOELECTRON EMISSION	BAXTER, W. J.	GENERAL MOTORS RESEARCH LABORATORIES						1	EXOELECTRON DETECTION OF FATIGUE
25	MICROPROCESSOR BASED AUTOMATIC HETERODYNE INTERFEROMETER	MOTTIER	UNITED TECHNOLOGIES RESEARCH CENTER						1	VERSATILE PROGRAMMED SCAN INTERFEROMETER
26	CRITICAL INSPECTION OF BEARINGS FOR LIFE EXTENSION	BARTON, J. R. KUSENBERGER, F.N. SMITH, R. T.	SOUTH RESEARCH INSTITUTE						1	AUTOMATIC QUANTITATIVE NDI OF BEARING COMPONENTS
27	DEVELOPMENT OF LMFBR STEAM GENERATOR LEAK PROTECTION SYSTEMS	MAGEE, P.M. GERRELS, E.E. GREENE, D. A. McKEE, J.	GENERAL ELECTRIC						1	ACOUSTIC LEAK DETECTION SYSTEM
28	A CRYOGENIC LINE LEAK DETECTOR	ALLAN, D. S. SCHIFF, D. S.	ARTHUR D. LITTLE, INC.					1		CONTINUOUS LINE LEAK MONITORING
29	INDUSTRIAL APPLICATIONS OF ULTRASOUND - A REVIEW	LYNNWORTH, L. C.	PARAMETRICS, INC.					11		BROAD REVIEW OF APPLICATIONS OF ULTRASOUND
30	MONITORING OF LNG VAPOR	HINCKLEY, E. D.	JET PROPULSION LABORATORY					2		TWO-BAND DIFFERENTIAL RADIOMETER AND LASER LEAK DETECTORS
31	NONDESTRUCTIVE EVALUATION OF METAL FATIGUE	KUSENBERGER, F.N. ET. AL.	SOUTHWEST RESEARCH INSTITUTE					5		MAGNETIC PERTURBATION ULTRASONIC SURFACE WAVE, AND BARKHAUSEN NOISE ANALYSIS FLAW DETECTION
32	NONDESTRUCTIVE INSPECTION METHOD FOR JET ENGINE TURBINE BLADES	KRASKA, I. R. BERNDT, W. L.	GENERAL AMERICAN TRANSPORTATION					1		EDDY CURRENT INSPECTION OF TURBINE BLADES
33	DESIGN OF AN ENDOSCOPIC CARRIER WITH COMPLETE DIRECTIONAL CONTROL	LACHIVER, G. SUEFERT, W. D.	UNIVERSITY OF SHERBROOKE, QUEBEC					1		CONTROLLED ARTICULATION FIBER OPTIC CARRIER
34	FATIGUE DAMAGE DETECTION	BARTON, J. R. KUSENBERGER, F.N.	SOUTHWEST RESEARCH INSTITUTE					5		METHODS, USES, LIMITS AND FUTURE OF DETECTING FATIGUE

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY **NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

TABLE 24. (CONTINUED)

NUMBER	TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
35	THE USE OF OPTICAL PROCESSING OF ENGINE VIBRATION DATA AS A MEANS OF PREDICTING FAILURES	MARKEVITCH, B.V. RODAL, D. R. BROWN, H.	AMPEX CORP.						1	HIGH RESOLUTION ENGINE SPECTRAL ANALYSIS USING OPTICAL DEFRACTION
36	IMPROVED COMBUSTION CHAMBER OPTICAL PROBE	WALKER, J.	LTV AEROSPACE CORP.						1	SPHERICAL VIEWING PERISCOPE-TYPE PROBE FOR THRUST CHAMBERS
37	RESIDUAL STRESSES IN GAS TURBINE ENGINE COMPONENTS FROM BARKHAUSEN NOISE ANALYSIS	BARTON, J. R. KUSENBERGER, F.N.	SOUTHWEST RESEARCH INSTITUTE						1	RESIDUAL STRESS MEASUREMENTS USING BARKHAUSEN NOISE ANALYSIS
38	AUTOMATED JET ENGINE BLADE INSPECTION SYSTEM	ROTHFUSZ, R.W.	BENDIX RESEARCH LABORATORIES						1	AUTOMATED DYE-PENETRANT FLAW DETECTION
39	TURBINE ENGINE LUBRICATION AND MOVING PARTS CHECKOUT	ZIEBARTH, H. K. CHANG, J. D. KUKEL, J.	GARRETT AIR RESEARCH					10		TURBINE ENGINE CONDITION MONITORING TECHNIQUES STUDY
40	THE DETERMINATION OF HYDROGEN IN HIGH STRENGTH STEEL STRUCTURES BY AN ELECTROCHEMICAL TECHNIQUE	BERMAN, D. A. BECK, W. DeLUCCIA, J. J.	NAVAL AIR DEVELOPMENT CENTER						1	IN-SITU DETECTION OF HYDROGEN CONTENT IN METALLIC STRUCTURES
41	FLAT-BASED WATER VAPOR SENSOR OF THE PHOSPHORUS PENTOXIDE TYPE	WIEDIJK, P.	NV PHILIPS GLOEILAMPENFABRIEKEN						1	WATEP VAPOR DETECTOR FOR BOTH ATMOSPHERIC AND VACUUM SYSTEM
42	AN ANGULAR DISPLACEMENT TRANSDUCER	WELSH, B. L.	ROYAL AIRCRAFT ESTABLISHMENT					1		FIBER OPTIC ROTATION/DISPLACEMENT TRANSDUCER
43	RADIOACTIVE GAS PENETRANT SYSTEM: A REPORT ON INITIAL PRODUCT APPLICATION	EDDY, W. C., JR.	INDUSTRIAL NUCLEONICS CORP.						1	DETECTION OF FLAWS USING KRYPTON-85 PENETRANT
44	DIAGNOSTIC SONICS FOR GAS TURBINE ENGINES	ZABRISKIE, C.J.	CURTISS-WRIGHT CORP.					1		MONITORING OF TURBINE ENGINE COMPONENTS WITH SONIC ANALYSIS
45	A TRANSMITTER FOR DIAGNOSTIC IMAGING	WANG, K. ET. AL.	CULLEN COLLEGE OF ENGINEERING AND UNIVERSITY OF CALIFORNIA AT SANTA BARBARA						1	SCANNING-FOCUSED ACOUSTIC BEAM USING AN OPTO-ACOUSTIC TRANSDUCER

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY **NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

TABLE 24. (CONCLUDED)

NUMBER	TITLE	AUTHOR	SOURCE	IN-FLIGHT SOTA* ROCKET	IN-FLIGHT SOTA NONROCKET	IN-FLIGHT NOVEL**	BETWEEN-FLIGHT SOTA ROCKET	BETWEEN-FLIGHT SOTA NONROCKET	BETWEEN-FLIGHT NOVEL	REMARKS
46	MAXIMUM SURFACE TEMPERATURE BY MEANS OF KRYPTONATES	GOODMAN, P.	PANAMETRICS, INC.						1	POST FACTO DETERMINATION OF MAXIMUM SURFACE TEMPERATURE
47	MICROWAVE TECHNIQUES FOR NONDESTRUCTIVE EVALUATION OF CERAMICS	BAHR, A. J.	SRI INTERNATIONAL						1	DETECTION OF FLAWS IN CERAMICS USING MICROWAVES
48	TOWARDS PRACTICAL NONDESTRUCTIVE FATIGUE DAMAGE INDICATORS	WEISS, V. OSHIDA, Y. WU, A.	SYRACUSE UNIVERSITY					1	2	X-RAY DIFFRACTION, ULTRASONIC ABSORPTION AND PHASE-CHANGE FATIGUE DAMAGE MONITORING
49	ELECTROTHERMAL NONDESTRUCTIVE TESTING OF METAL STRUCTURES	McCULLOUGH, L.D. GREEN, D. R.	BATTELLE NORTHWEST LABORATORIES						1	THERMAL IMAGING OF FLAWS HEATED BY AN ELECTRICAL CURRENT PULSE
50	POSITRON ANNIHILATION	COLEMAN, C. F. HUGHES, A. E.	AERE						1	FATIGUE MONITORING USING POSITRON DECAY
51	IMAGING TECHNOLOGY: A EUROPEAN SURVEY	MEYER-EBRECHT, D.	UNKNOWN					1		INFRARED, ULTRASONIC AND X-RAY IMAGING SYSTEMS IN MEDICINE
52	PHOTODIODE ARRAYS: A CONVENIENT TOOL FOR LASER DIAGNOSTICS	SAKA, W. ZIMMERMAN, J.	INSTITUTE OF APPLIED PHYSICS					1		ON-LINE DIAGNOSTICS OF PULSED LASERS INCLUDING BEAM CROSS-SECTION AND PSEC-PULSE DURATION MONITORING
53	DIRECT CONTACT, HAND-HELD DIAGNOSTIC B-SCANNER	HOLASEK, E. SOKLLU, A.	CASE WESTERN RESERVE UNIVERSITY					1		HAND OPERATED, ULTRASOUND SCAN SYSTEM FOR OPHTHALMIC EVALUATION
54	DEPOT ATE ARCHITECTURE AND INSTRUMENTATION CONSIDERATIONS	KOLE, ROY. S.	AiRESEARCH CORP.							IDENTIFIES LIMITATIONS IN THE SYSTEM ARCHITECTURE TEST EQUIPMENT AND DESCRIBES PROPOSED ALTERNATIVES
55	APPLICATION OF ACOUSTICAL HOLOGRAPHY TO INDUSTRIAL TESTING	BRENDEN, B.	UNKNOWN					1		ACOUSTICAL HOLOGRAPHY SYSTEM USED FOR IMAGING, LOCATING AND SIZING FLAWS IN LARGE REACTOR PRESSURE VESSELS.
56	CIT IS STUDIED FOR INDUSTRY APPLICATIONS	UNKNOWN	INDUSTRIAL RESEARCH AND DEVELOPMENT JANUARY 1981						1	COMPUTERIZED INDUSTRIAL TOMOGRAPHY (CIT), FIRST USED IN MEDICAL SCIENCES, IS STUDIED FOR USE IN INSPECTIONS.

*SOTA = UP TO DATE, IN USE, PROVEN TECHNOLOGY **NOVEL = NOT PROVEN, PROTOTYPE TECHNOLOGY

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APPENDIX I. BETWEEN-FLIGHT TECHNOLOGY

APPENDIX I

BETWEEN-FLIGHT TECHNOLOGY GRADING

The method for evaluation and ranking of the technologies was developed in cooperation with the Task II effort. The evaluation method selected was, as in Task II, a hybrid approach: first, two clear-out screens were identified and applied to the techniques. Lumped descriptors, each made up of many specific descriptors, were then defined. The technologies applicable to each failure mode were graded using these lumped descriptors, thus providing a ranking of the techniques. All techniques were assumed to be equally developed for use on rocket engines. This appendix shows the technology grading performed for each failure mode. The scores given for technical lumped descriptors were summed to give an overall technical score. Economic costs were subtracted from savings, resulting in an overall savings figure. An economic grade was then assigned based on one point for each nearest $100,000 in savings. The development grade was determined by subtracting one point, from a maximum of 10 points, for each year required for development. Thus, three grades were obtained for each technology in each failure mode.

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 2 — COOLANT PASSAGE LEAKAGE / INSPECTION TECHNOLOGY (DESCRIPTORS)	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$∞K	$∞K	10	0	10	120
SCANNING ACOUSTIC FLOW	14	5	6	12	7	16	9	69	200	10	600	390	4	2	8	81
ACOUSTIC HOLOGRAPHY	15	3	5	15	5	18	9	70	200	10	500	290	3	4	6	79
X-RAY RADIOGRAPHY	7	1	2	8	5	8	7	38	100	10	200	90	1	2	8	47
GAMMA RADIOGRAPHY	6	2	4	15	7	8	3	45	100	20	400	290	3	3	7	55
PENTOXIDE POLAROGRAPHY	7	7	6	12	4	8	1	45	200	50	200	(50)	0	4	6	51
HYDROGEN POLAROGRAPHY	15	5	5	12	5	12	8	62	150	10	400	240	2	4	6	70
HYGROMETRY	7	7	6	10	5	7	1	43	50	20	200	130	1	1	9	53
SCANNING OPTICAL PYROMETRY	18	5	8	10	8	12	9	70	100	10	600	490	5	2	8	83
HOLOGRAPHIC LEAK	17	3	4	8	7	17	9	65	200	10	700	490	5	3	7	77
MILLIMETER-WAVE INTERFEROMETRY	15	3	4	8	6	12	8	56	200	10	400	190	2	4	6	64

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 3 — JOINT LEAKAGE	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ULTRASONIC EXTENSIOMETER	4	6	6	12	4	2	4	38	0	100	0	(100)	0	0	10	48
ULTRASONIC LEAK	12	8	7	8	6	10	4	55	20	0	800	780	8	0	10	73
LEAK TAPE/COATING	2	4	5	10	2	10	5	38	100	100	700	500	5	2	8	51
OPTICAL LEAK	12	5	5	10	5	12	8	57	100	0	800	700	7	2	8	72
HOLOGRAPHIC DEFLECTION	6	2	2	5	2	8	8	33	300	0	400	100	1	4	6	40
DIFFERENTIAL RADIOMETRY	6	3	4	9	5	8	7	42	200	0	600	400	4	2	8	54
HOLOGRAPHIC LEAK	18	3	3	12	6	18	9	69	200	0	1200	1000	10	3	7	86
RESISTIVITY MONITORING	2	6	5	8	2	3	3	29	100	100	300	100	1	2	8	38
HALOGEN LEAK	10	5	5	12	7	8	3	50	20	20	700	660	7	1	9	66
FLOW LEAK	1	3	3	16	6	1	1	32	0	20	0	(20)	0	0	10	42
MASS SPECTROMETRY	2	2	2	16	6	2	2	32	0	20	0	(20)	0	0	10	42
THERMAL CONDUCTIVITY LEAK	14	8	8	10	7	12	8	67	20	0	800	780	8	0	10	85
TORQUING	3	5	7	10	5	1	2	33	0	0	0	0	0	0	10	43
LEAK SOLUTION	3	8	2	10	2	5	2	32	0	0	0	0	0	0	10	42
PRESSURE DECAY	8	5	7	5	5	7	3	40	10	20	200	170	2	0	10	52

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 5: HIGH TURBOPUMP TORQUE	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ISOTOPE THERMOMETRY	5	6	3	8	3	8	2	35	200	50	0	(250)	0	3	7	42
ISOTOPE TRACERS	9	5	4	14	7	14	2	55	200	30	600	370	4	3	7	66
PARTICLE ANALYSIS	10	5	5	10	4	15	2	51	100	20	500	380	4	1	9	64
BORESCOPING	5	7	7	5	7	4	3	38	0	20	0	(20)	0	0	10	48
OPTICAL PROXIMITY	8	7	5	10	7	14	4	55	100	100	400	200	2	2	8	65
TORQUING	8	8	6	16	7	7	3	55	0	10	0	(10)	0	0	10	65

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 6: CRACKED TURBINE BLADES	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ULTRASONIC FLAW	7	5	6	14	5	11	2	50	50	30	2000	1920	10+	2	8	68
ISOTOPE THERMOMETRY	7	4	4	8	4	7	2	36	200	10	1000	790	8	3	7	51
ISOTOPE TRACERS	5	4	2	7	4	10	1	33	200	30	1000	770	8	3	7	48
REMNANT MAGNETIZATION	7	7	7	7	7	10	5	50	50	20	2000	1930	10+	1	9	69
HOLOGRAPHIC SURFACE MAPPING	10	4	4	15	7	14	3	57	200	20	3000	2780	10+	3	7	74
BORESCOPING	6	7	7	6	8	6	2	42	50	20	500	430	4	1	9	55
EXO-ELECTRON EMISSION	6	5	6	18	7	12	2	58	200	20	2000	1780	10+	3	7	75
POSITRON ANNIHILATION	6	1	3	10	6	6	2	34	300	40	1000	660	7	4	6	47
ELECTRIC CURRENT INJECTION	5	4	4	9	5	8	2	37	200	30	1500	1280	10+	3	7	54
EDDY CURRENT	6	5	6	14	7	6	2	46	100	30	2000	1870	10+	2	8	64

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 7: CRACKED CONVOLUTIONS, BELLOWS, AND SHIELD	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	\$0K	\$0K	\$ ∞ K	\$ ∞ K	10	0	10	120
ULTRASONIC FLAW	9	5	6	16	6	12	3	57	50	10	200	140	1	2	8	66
ISOTOPE THERMOMETRY	10	4	4	8	4	9	2	41	200	10	400	190	2	3	7	50
REMNANT MAGNETIZATION	10	7	6	6	7	9	3	48	200	10	300	90	1	3	7	56
BORESCOPING	10	7	8	6	8	10	2	51	0	10	0	(10)	0	0	10	61
PENETRANTS	5	7	3	14	4	4	1	38	20	20	0	(40)	0	1	9	47
HOLOGRAPHIC SURFACE MAPPING	12	4	4	14	7	15	4	60	200	20	400	180	2	3	7	69
EXO-ELECTRON EMISSION	11	4	4	18	7	14	3	61	200	20	400	180	2	3	7	70
POSITRON ANNIHILATION	8	1	3	10	6	10	3	41	300	20	300	(20)	0	4	6	47
ELECTRIC CURRENT INJECTION	8	4	3	10	5	9	2	41	200	30	300	70	1	3	7	49
EDDY CURRENT	12	5	6	12	7	10	2	54	100	10	300	190	2	2	8	64

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 8 — LOOSE ELECTRICAL CONNECTOR	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ISOTOPE THERMOMETRY	9	5	4	8	5	10	5	46	100	10	200	90	1	2	8	55
CONTINUITY CHECKING	12	4	4	16	8	15	7	66	0	30	100	70	1	0	10	77
TORQUING	10	8	8	14	8	7	3	58	0	10	0	(10)	0	0	10	68

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 9 — BALL BEARING DAMAGE	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$∞K	$∞K	10	0	10	120
ULTRASONIC FLAW	5	4	5	7	5	7	3	36	100	30	200	70	1	2	8	45
ISOTOPE THERMOMETRY	6	5	4	7	4	8	3	37	200	20	100	(120)	0	3	7	44
ISOTOPE TRACERS	12	7	5	17	5	16	2	64	200	30	600	370	4	3	7	75
PARTICLE ANALYSIS	13	7	7	12	6	14	2	61	100	20	500	380	4	1	9	74
BORESCOPING	5	6	8	4	8	7	3	41	50	20	0	(70)	0	0	10	51
EXO-ELECTRON EMISSION	7	4	4	8	5	7	3	38	200	30	200	(30)	0	3	7	45
POSITRON ANNIHILATION	6	1	4	7	5	6	3	32	300	30	200	(130)	0	4	6	38
EDDY CURRENT	6	5	5	6	7	6	3	38	100	30	200	70	1	1	9	48
TORQUING	12	7	7	10	8	10	2	56	0	10	0	(10)	0	0	10	66

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 10 — TUBE FRACTURE	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ULTRASONIC FLAW	14	5	6	14	6	12	8	65	50	10	400	340	3	2	8	76
ACOUSTIC EMISSION	10	5	5	6	7	13	1	48	200	20	600	380	4	3	7	59
X-RAY RADIOGRAPHY	8	5	4	10	5	10	6	48	100	10	200	90	1	1	9	58
PENETRANTS	10	8	3	14	4	7	3	49	20	10	0	(30)	0	0	10	59
HOLOGRAPHIC DEFLECTION	16	3	4	14	5	16	8	66	300	20	600	280	3	4	6	75
EXO-ELECTRON EMISSION	14	4	5	18	5	14	8	68	200	10	600	390	4	3	7	79
POSITRON ANNIHILATION	10	1	4	14	4	10	8	51	300	20	200	(120)	0	4	6	57
ELECTRIC CURRENT INJECTION	12	5	6	12	5	14	7	61	200	10	600	390	4	3	7	72
EDDY CURRENT	12	5	7	12	6	14	8	64	100	10	500	390	4	1	9	77

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 11 / TURBOPUMP SEAL LEAKAGE	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ISOTOPE THERMOMETRY	5	6	3	8	3	8	2	35	200	50	0	(250)	0	3	7	42
ISOTOPE TRACERS	9	5	4	14	7	16	2	57	200	30	600	370	4	3	7	68
PARTICLE ANALYSIS	10	5	5	10	4	15	2	51	100	20	500	380	4	1	9	64
FLOW LEAK DETECTION	10	7	6	14	7	7	3	54	0	10	0	(10)	0	0	10	64
OPTICAL PROXIMITY	8	7	5	12	7	14	4	57	100	100	400	200	2	2	8	67
TORQUING	8	8	6	16	7	7	3	55	0	10	0	(10)	0	0	10	65

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 13 VALVE FAILURE	TECHNICAL								ECONOMICAL					DEVELOPMENT		
DESCRIPTORS	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ULTRASONIC LEAK	12	6	7	10	5	12	3	55	100	20	200	80	1	2	8	64
ACOUSTIC HOLOGRAPHY	8	4	6	8	4	10	5	45	200	10	200	(10)	0	4	6	51
ISOTOPE TRACERS	12	6	4	15	4	14	1	56	200	30	300	70	1	3	7	64
PENTOXIDE POLAROGRAPHY	10	7	5	10	5	10	2	49	200	20	100	(120)	0	4	6	55
HYGROMETRY	10	7	6	8	5	9	2	47	50	20	100	30	0	1	9	56
PARTICLE ANALYSIS	12	6	4	12	4	12	1	51	100	30	300	170	2	1	9	62
LASER SURFACE SCATTERING	8	5	5	10	5	12	2	47	200	30	200	(30)	0	3	7	54
OPTICAL LEAK	8	5	4	14	5	12	3	51	200	20	200	(20)	0	3	7	58
BORESCOPING	7	7	7	8	8	8	2	47	0	10	0	(10)	0	0	10	57
DIFFERENTIAL RADIOMETRY	8	5	5	14	5	12	3	52	200	20	200	(20)	0	3	7	59
HOLOGRAPHIC SURFACE MAPPING	10	4	4	12	5	12	2	49	200	30	200	(30)	0	3	7	56

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 14 INTERNAL VALVE LEAKAGE	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K	$ ∞ K	10	0	10	120
ULTRASONIC LEAK	12	6	7	12	5	12	3	57	100	20	200	80	1	2	8	66
ISOTOPE TRACERS	12	6	4	15	4	12	1	54	200	30	300	70	1	3	7	62
PARTICLE ANALYSIS	12	6	4	12	4	12	1	51	100	30	300	170	2	1	9	61
LASER SURFACE SCATTERING	8	5	5	10	5	12	2	47	200	30	200	(30)	0	3	7	54
OPTICAL LEAK	8	5	4	14	5	12	3	51	200	20	200	(20)	0	2	8	59
BORESCOPING	7	7	7	8	8	8	2	47	0	10	0	(10)	0	0	10	57
DIFFERENTIAL RADIOMETRY	8	5	5	14	5	12	3	52	200	20	200	(20)	0	2	8	60
HOLOGRAPHIC SURFACE MAPPING	8	4	4	12	5	12	2	47	200	30	200	(30)	0	3	7	54
OPTICAL PROXIMITY	7	7	6	8	5	7	2	43	100	20	100	(20)	0	2	8	51
HALOGEN LEAK	10	6	7	12	5	10	3	53	50	10	200	140	1	1	9	63
FLOW LEAK	10	5	6	13	5	10	3	52	0	10	100	90	1	0	10	63
MASS SPECTROMETRY	10	5	6	13	4	10	3	51	0	10	100	90	1	0	10	62
THERMAL CONDUCTIVITY LEAK	14	7	7	12	5	12	3	61	50	10	200	140	1	1	9	71
TORQUING	6	7	7	6	7	9	1	43	100	20	100	(20)	0	1	9	52
PRESSURE DECAY	12	5	6	8	5	10	3	49	0	10	100	90	1	0	10	60

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 15 REGULATOR DISCREPANCIES INSPECTION TECHNOLOGY / DESCRIPTORS	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$↔K	$↔K	10	0	10	120
ULTRASONIC LEAK	12	6	6	11	5	12	3	55	100	20	200	80	1	2	8	64
PARTICLE ANALYSIS	14	6	5	14	4	12	1	56	100	30	300	170	2	1	9	67
OPTICAL LEAK	9	5	4	12	5	10	3	48	200	20	200	(20)	0	2	8	56
DIFFERENTIAL RADIOMETRY	9	5	5	12	5	10	3	49	200	20	200	(20)	0	2	8	57
HALOGEN LEAK	10	5	6	12	5	12	2	52	50	10	200	140	1	1	9	62
FLOW LEAK	9	5	6	12	5	10	3	50	0	10	100	90	1	0	10	61
MASS SPECTROMETRY	9	5	6	12	5	9	3	49	0	10	100	90	1	0	10	60
THERMAL LEAK	12	7	7	12	6	13	3	60	50	10	200	140	1	1	9	70
PRESSURE DECAY	10	5	5	8	5	10	3	46	0	10	100	90	1	0	10	57

BETWEEN-FLIGHT DIAGNOSTIC TECHNOLOGY GRADING

FAILURE MODE 16 CONTAMINATED HYDRAULIC CONTROL	TECHNICAL								ECONOMICAL					DEVELOPMENT		
	REQUIREMENTS			FEATURES												
INSPECTION TECHNOLOGY / DESCRIPTORS	APPLICATION	ANCILLARY	PHYSICAL	DETECTIBILITY	DURABILITY	SPEED	HAZARD	TECHNICAL SCORE	R&D COSTS	INTEGRATION COSTS	OPERATIONAL SAVINGS	TOTAL SAVINGS (COST)	ECONOMIC GRADE	TIME (YEARS)	GRADE	TOTAL OVERALL GRADE
PERFECT SCORE	20	10	10	20	10	20	10	100	$0K	$0K	$ ∞ K.	$ ∞ K	10	0	10	120
ULTRASONIC LEAK	12	6	6	9	5	12	4	54	100	20	200	80	1	2	8	63
PARTICLE ANALYSIS	14	6	5	14	4	12	2	57	100	30	300	170	2	1	9	68
OPTICAL LEAK	9	5	4	10	5	10	4	47	200	20	200	(20)	0	2	8	55
DIFFERENTIAL RADIOMETRY	9	5	5	10	5	10	4	48	200	20	200	(20)	0	2	8	56
PRESSURE DECAY	10	5	6	10	5	9	4	49	0	10	100	90	1	0	10	60

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APPENDIX J

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