SPACE SHUTTLE MAIN ENGINE THE FIRST TEN YEARS

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PREFACE

"SPACE SHUTTLE MAIN ENGINE - THE FIRST TEN YEARS" was written at the request of Dominick J. (Dom) Sanchini so that this most important chapter of the United States space program would be documented for posterity. Mr. Sanchini was the leader of the team of experts that prepared the Rocketdyne proposal for the Space Shuttle Main Engine (SSME) in 1970 and 1971 and was the Rocketdyne vice president and program manager of SSME from 1975 until 1983. He died on November 17, 1990 at the age of 63, following a prolonged battle with cancer. He will be remembered as the dynamic leader of the Rocketdyne SSME team through its most trying times.

The Space Shuttle Main Engine program was the single most important contract in the history of Rocketdyne. The contract dollar value and the overall impact on the company's reputation and future success are all without parallel in the company's history. Although recognized as the world's leading producer of rocket engines, the company was in a state of severe retrenchment at the beginning of the 70's decade. From an employment of over 20,000 people during the Apollo program of the 60's, the total work force had dropped to less than 3,000. The continuation of Rocketdyne as a viable separate division of North American Rockwell Corporation was in jeopardy. The years 1970 and 1971 saw Rocketdyne in fierce competition with two other companies for the only significant rocket engine development program in the foreseeable future, the reusable Space Shuttle Main Engine (SSME). Rocketdyne was clearly not favored to win the SSME contract and did so only as a result of truly remarkable efforts by the company's management and proposal team. "SPACE SHUTTLE MAIN ENGINE - THE FIRST TEN YEARS" chronicles the trials, tribulations and triumphs associated with the design and development of the SSME through its maiden flight on April 12, 1981.

Throughout the decade to follow, the SSME performed with a 100 percent mission success rate while Rocketdyne grew to a billion dollar a year company. By the end of 1990, twenty-one engines had logged over sixteen hours of powered flight with three engines being used on each of thirty-eight Space Shuttle missions. Reusability of the engines had been demonstrated with all twenty-one engines, with an average of over five flights per engine and as many as eleven flights on one of them. Except for a malfunction of a sensor in the engine limit monitoring system on one of the earlier flights, not a single engine failure has been experienced. The SSME flight operation has been flawless.

FOREWORD

The design and development of the Space Shuttle Main Engine as described in this document were directed and funded by the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration (NASA) under Contract NAS8-27980.

ABSTRACT

In July 1971 the Rocketdyne Division of Rockwell International won the competitive bid to design, develop and produce the Space Shuttle Main Engine (SSME). After 10 months of delay due to a protest lodged by a competitor, the work was started in April 1972. The engine, to be developed under contract with the Marshall Space Flight Center of NASA, was a significant departure from the Apollo man-rated rocket engines of the 60's. A liquid oxygen/liquid hydrogen engine, it was rated at approximately a half million pounds thrust, with capability to throttle from 50% to 109% of rated power. It was to be computer controlled with a fully redundant, fail-operate, fail-safe control system and reusable for up to 100 flights. Nine years later, three of these engines successfully contributed to the new era of the Space Transportation System (STS) when STS-1 was launched off pad 39 at the Kennedy Space Center.

This discussion traces the development failures and successes that the Rocketdyne and Marshall engine team faced in the decade prior to the first flight of the Space Shuttle. Engine design and operating characteristics, program requirements, and original plans and goals are discussed. A history is presented of schedule difficulties and technical problems along with management techniques and problem solutions.

Under the leadership of J. R. Thompson, Jr. (then Project Manager of SSME and now Deputy Administrator of NASA), the team "persevered and pressed on".

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Mr. Biggs has been a member of the Rocketdyne Engineering staff since 1957 and a member of the Space Shuttle Main Engine management team since 1970. As the systems development manager and as the chief project engineer, He directed the SSME ground test program from the first test through the completion of the Space Shuttle development flight tests.

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THE ENGINE

The Space Shuttle Main Engine (SSME) (Figure 1) is a high chamber pressure (over 3,000 pounds per square inch) rocket engine that burns liquid oxygen (LOX) and liquid hydrogen (LH2) at a mixture ratio of 6 pounds of LOX for every pound of LH2. It produces a rated thrust of 470,000 pounds (vacuum) with a specific impulse greater than 453 pounds of thrust per pound of propellant per second. This very high efficiency is achieved by the utilization of a "staged combustion cycle" wherein a portion of the propellants, partially combusted at a fuel-rich mixture ratio, is used to drive the high pressure turbopump turbines prior to being completely burned in the main combustor. Figure 2 is a representation of the major components of the powerhead. As can be seen in this cutaway view, the turbine drive gases are produced in two "preburners" to provide the power for the two high pressure turbines, they then exit into the main fuel injector, and are burned with the remainder of the propellants in the main combustion chamber. This results in maximum propellant efficiency because all the propellant is used in the main combustor, and none is wasted by being dumped overboard from a low pressure turbine exhaust system as was the case with all prior large liquid rocket engines. This improved efficiency is achieved at a significant cost in system pressures. With the turbines in series with the main combustor, the turbine exhaust pressure has to be higher than the main combustion chamber pressure. Although the turbines are designed for low pressure ratio (approximately 1.5 to 1), the turbine inlet pressure has to be about 50 percent higher than the exhaust pressure in order to provide sufficient power. The preburners that provide the turbine drive gases have propellant injectors that require a minimum differential pressure in order to assure stable combustion. This further increases the required turbopump discharge pressures for the propellant pumps to as much as two and one-half times the main combustion chamber pressure.

It was the high combustion chamber pressure combined with the amplification effect of the staged combustion cycle that made this engine a quantum jump in rocket engine technology and created a significant challenge to the contractor and government team charged with its design and development. [1]

Figure 3 is a schematic representation of the engine system showing the interrelationship of the major components and the flow path of the propellants. To provide turbine power in the staged combustion cycle, 80 percent of the fuel (LH2) is burned in the two preburners with 12 percent of the oxidizer (LOX). The turbine exhaust gases are then burned in the main combustion chamber (MCC) with the remainder of the propellants.

The LH2 enters the engine at the low pressure fuel turbopump (LPFTP) inlet at a pressure of 30 psia and is increased in pressure by the 15,000 RPM turboinducer to over 250 psia. This pressure is required to prevent cavitation of the high pressure fuel turbopump (HPFTP). The three-stage centrifugal pump, operating at 35,000 RPM,

SSME IS FIRST REUSABLE LARGE LIQUID ROCKET ENGINE

■ FULL POWER LEVEL (FPL) 109%	512,300 LBS
■ RATED POWER LEVEL (RPL) 100%	470,000 LBS
■ CHAMBER PRESSURE	3200 PSIA
■ SPECIFIC IMPULSE AT ALTITUDE	453.5 SECONDS
■ THROTTLE RANGE	65 TO 109%
■ PROPELLANTS	OXYGEN/ HYDROGEN
■ WEIGHT	7000 LBS
■ DESIGN LIFE	27,000 SECONDS 55 STARTS
■ AT FULL POWER LEVEL	14,000 SECONDS

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Figure 1. Space Sh[illegible]e Main Engine

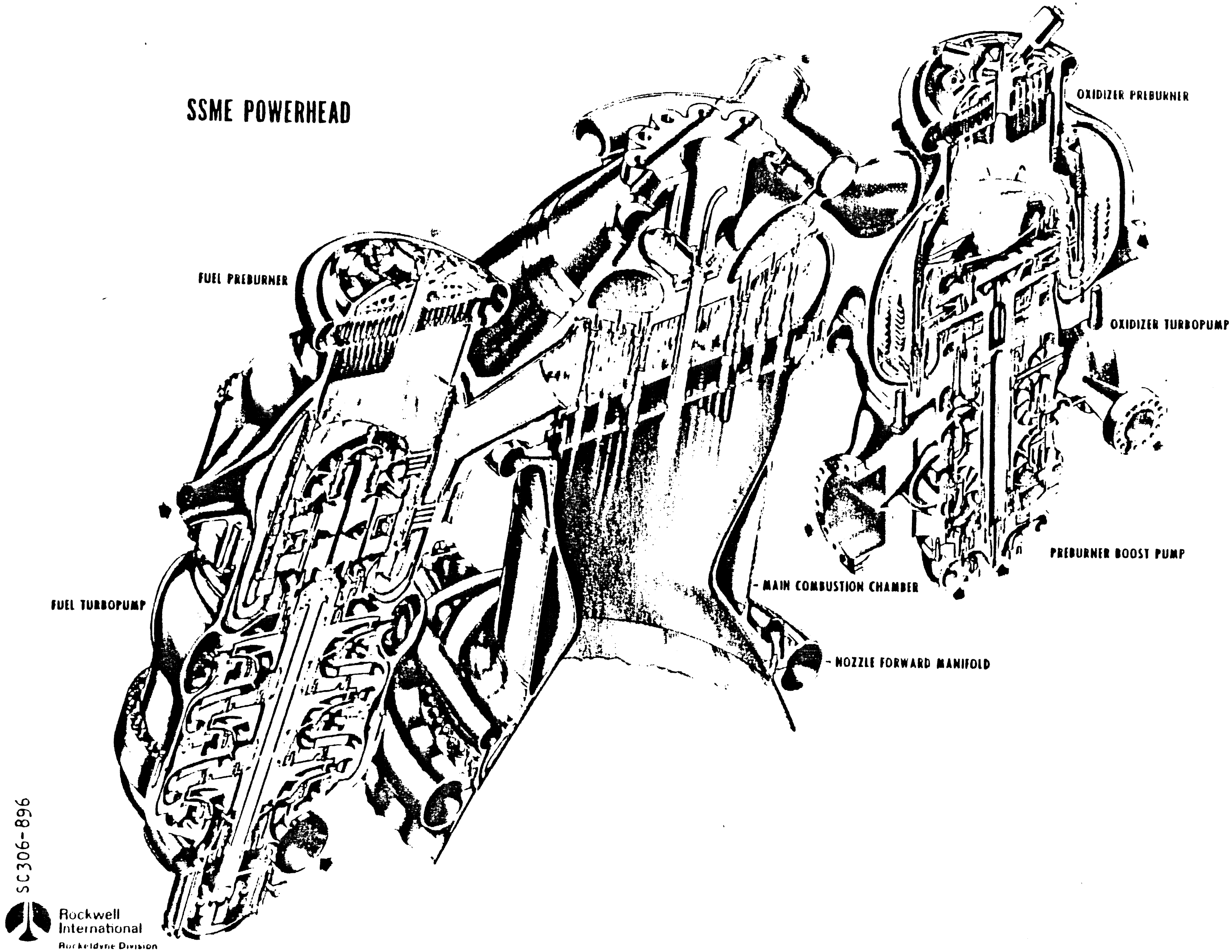

Figure 2. SSME Powerhead

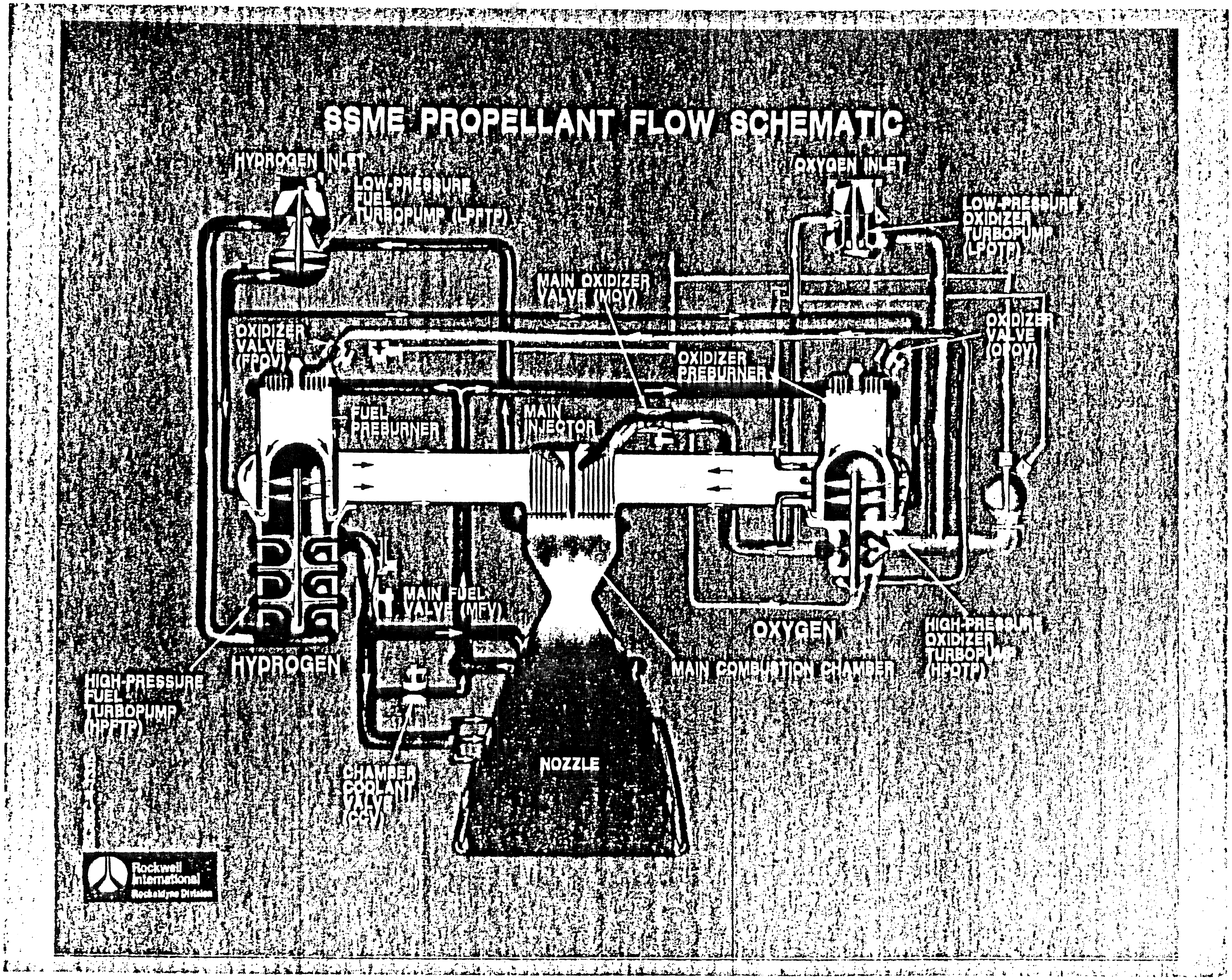

Figure 3. SSME Propellant Flow Schematic

further increases the pressure to over 6,000 psia. The LH2 is then divided into three separate flow paths. Approximately 80 percent of the fuel flows to the two preburners; half of this, however, is used to cool the thrust chamber nozzle and then mixed with the other half prior to entering the preburners. The remaining 20 percent of the fuel is used in the major component cooling circuit. The LH2 is first routed to the MCC where it provides coolant for the main combustion process by flowing through 390 milled slots in the copper alloy combustor. Having been converted to an ambient temperature gas by the MCC, the fuel is then routed to the LPFTP where it is used as the power source for the partial admission single stage impulse turbine which drives the LPFTP. A small portion (0.7 pounds per second) of this gas is then used by the Space Shuttle to pressurize the main hydrogen tank while the rest of it is used to cool the major hot gas system structure (hot gas manifold) and, finally, the main injector baffles and faceplates before being consumed in the MCC.

The LOX enters the engine at the low pressure oxidizer turbopump (LPOTP) inlet at a pressure of 100 psia and is increased in pressure by the 5,000 RPM turboinducer to over 400 psia. This pressure is required to prevent cavitation of the high pressure oxidizer turbopump (HPOTP). The dual inlet single stage centrifugal main impeller, operating at almost 30,000 RPM, further increases the pressure to about 4,500 psia. Most of the LOX is then routed through the main oxidizer valve to the coaxial element main injector of the MCC. A small amount of LOX (1.2 pounds per second) is routed through an engine-mounted heat exchanger and conditioned for use as the pressurant gas for the Space Shuttle main oxidizer tank. The remainder of the LOX is ducted back into a smaller boost impeller on the same shaft to increase the pressure to as much as 8,000 psia. This provides enough pressure to allow the use of throttle valves to control the LOX flow rate into the two preburners. Thrust control is achieved by closed loop throttling of the oxidizer preburner (OPB) side, and mixture ratio control is accomplished by closed loop control of the fuel preburner (FPB) side. The throttle valves are controlled by an engine-mounted computer known as the main engine controller (MEC). A built-in recirculation flow path provides power for the six stage axial flow hydraulic turbine which drives the LPOTP. A LOX flow rate of approximately 180 pounds per second is supplied from the discharge side of the main impeller; and, after passing through the turbine, this LOX is mixed with the discharge flow of the LPOTP and thereby returned to the HPOTP inlet.

The two preburners produce a hydrogen-rich steam that is used to power the two high pressure turbines that drive the HPFTP and the HPOTP. Combustion of these gases is completed in the MCC.

THE BEGINNING

On the 13th of July, 1971, the National Aeronautics and Space Administration (NASA) announced that it had selected the Rocketdyne Division of North American Rockwell Corporation, Canoga Park, California, for negotiations leading to the award of a contract to design, develop, and manufacture the Space Shuttle Main Engines.[2] The selection was made after a one-year "Phase B" competition among three contractors. The Phase B program funded preliminary design studies, program definition documents and some technology advancement and demonstration test programs. This, along with contractor discretionary resource funded programs and prior experience, formed the basis for the SSME proposals submitted by the three contractors on April 21, 1971. The request for proposal [3] was based on a Space Shuttle vehicle which employed two reusable stages, a manned fly-back booster vehicle with a piggy-back mounted orbiter. NASA had specified the design of a single powerhead that would be used as both a booster engine (12 engines with 550,000 pounds sea level thrust each) and an orbiter engine (3 engines with 632,000 pounds vacuum thrust each) by simply changing the thrust chamber nozzle for the different applications. The only engine design feature that was clearly defined was the thrust chamber nozzle. It was required to be a bell-type nozzle to prevent inclusion of the more technologically advanced aerospike nozzle in any of the proposals. The performance requirements, however, were such that only a high chamber pressure staged combustion cycle could satisfy them. This was done to force an advancement of rocket engine technology.

The Rocketdyne proposal [4] consisted of an executive summary, a seven-volume technical proposal, a five-volume management proposal and 81 volumes (87 books) of related data for a total of 100 books (Figure 4). The key feature of the Rocketdyne proposal was the construction and test of a nearly full-scale model of the combustion devices for the SSME powerhead. It contained two preburners and a regeneratively cooled main combustion chamber, operating in a staged combustion cycle and developing approximately 400,000 pounds of thrust (Figure 5). Paul Castenholz, vice president and program manager, had chosen to pursue this objective in order to clearly demonstrate the required technology for high pressure staged combustion.

During the engine competition phase, the Space Shuttle program underwent continuing reevaluation and redefinition. Fiscal funding was not to be provided at levels consistent with the original concept of the Space Shuttle; therefore, the reusable fly-back booster was discarded in favor of more cheaply developed recoverable solid rocket boosters.This meant that the SSME no longer was required to do double duty as both a booster and an orbiter engine and could be optimized for just the orbiter. The engine rated thrust level was reduced to 470,000 pounds (vacuum) with 109 percent emergency power level capability. The Space Shuttle vehicle was to have the orbiter engines burn in parallel with the booster rockets which would require starting and operating at sea level. This would limit the nozzle expansion area ratio to 77.5 to 1.

Redefinition of the engine could not proceed, however, because three weeks after the contract award announcement, a formal protest was lodged with the General Accounting Office (GAO) by one of the competitors. It was decided that Rocketdyne could not be allowed to expend any contract funds on engine redefinition until the matter of the protest was resolved. A cost-plus-fixed-fee level-of-effort contract [5] was issued by the George C. Marshall Space Flight Center (MSFC) to allow Rocketdyne to provide support to the still competing vehicle contractors and to help resolve technical and management issues between Rocketdyne and MSFC. A "fact finding" negotiation resolved all of the issues except those related to engine redefinition and certain sensitive issues relating to the protest. The protest was finally resolved by the GAO [6] on March 31, 1972; and the process of redefining the engine was allowed to continue.

On April 5,1972, a letter contract was effected principally for conversion of the 550,000 pound thrust engine to a 470,000 pound thrust engine [7] and other technical changes

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North American Rockwell

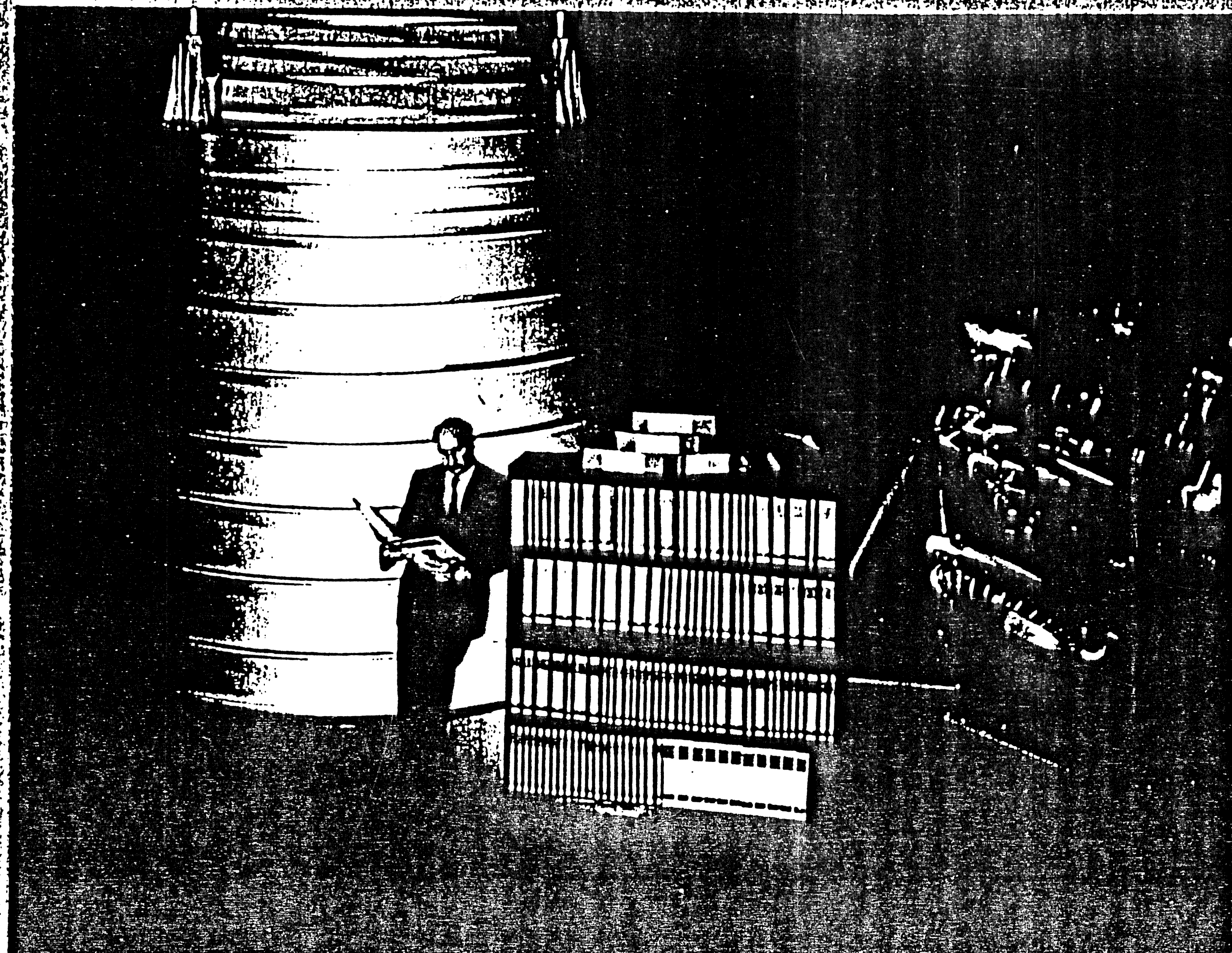

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Figure 4. Rocketdyne Space Shuttle Main Engine Proposal

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Rocketdyne Full Scale SSME Model

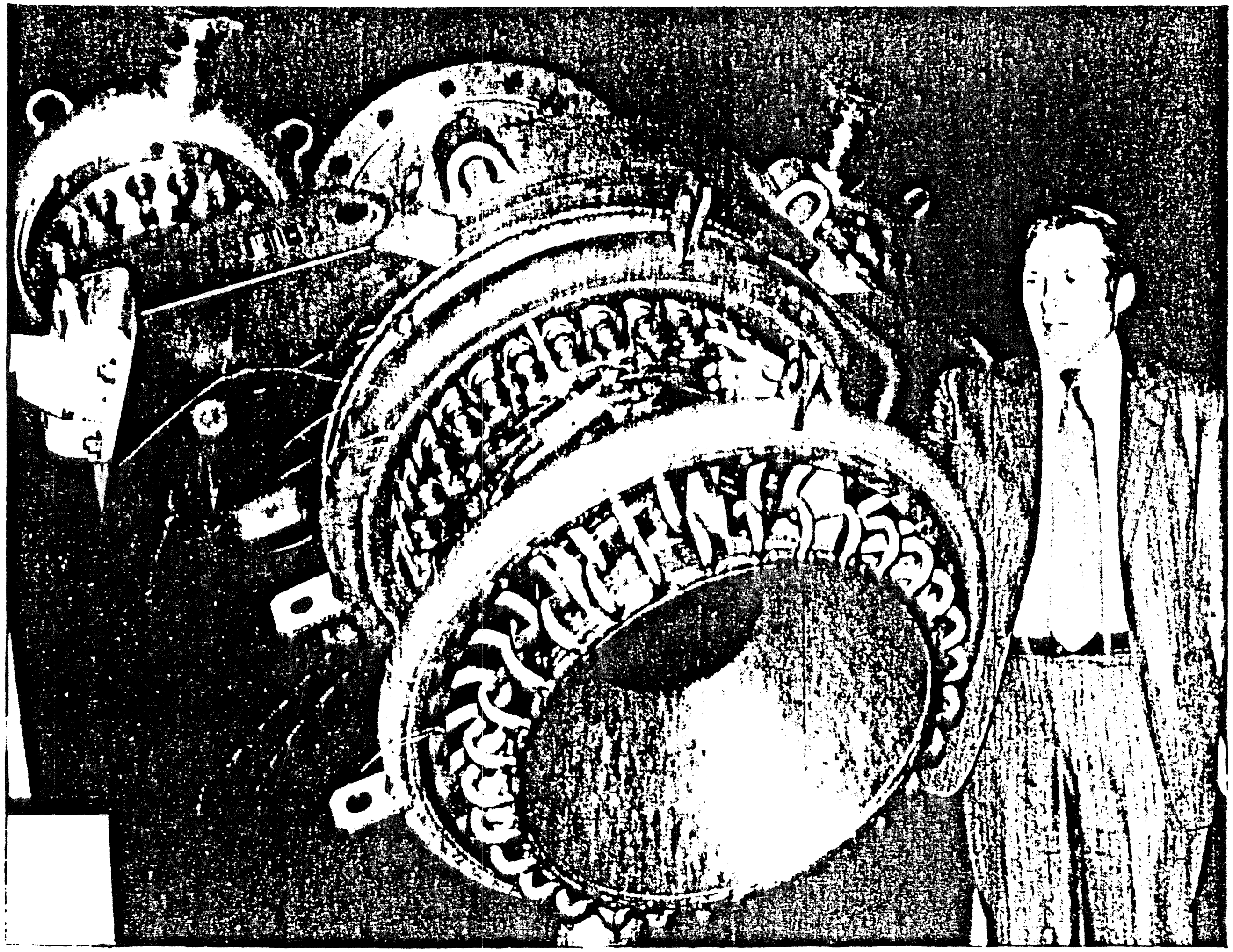

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Figure 5. Full Scale Staged Combustion Cycle Test Model

relative to a parallel burn Space Shuttle. A definitive cost-plus-award-fee contract was signed on August 14, 1972 [8].

THE REQUIREMENTS

Finalization of the engine design requirements began in May 1972 with the continuation of the fact-finding negotiation of the prior year. Over 250 separate issues were identified and resolved in a two month period. With the NASA selection of the orbiter contractor (Space Division of North American Rockwell Corporation) negotiations could begin to define the physical, functional and electronic interfaces between the engine and the orbiter. The first such meeting took place at Rocketdyne on August 10, 1972. In a series of technical meetings throughout the rest of 1972, fact-finding and interface issues were sufficiently resolved between the various contractor and NASA organizations to enable baseline release of the two major design requirements documents. The Interface Control Document (ICD) [9] was released on February 9, 1973, containing SSME design requirements relating to engine/vehicle interfaces. These included: engine envelope, weight and center of gravity; dimensions, tolerances and structural capabilities of all physical interfaces; electrical power, frequencies and phase requirements; computer command and data formats and failure responses; and lastly, engine environment and performance requirements. The Contract End Item (CEI) Specification [10] was released on May 10, 1973. The CEI specification contained: detailed requirements for engine checkout, prestart, start, operation and shutdown; engine service life and overhaul requirements; design criteria for thermal, vibration, shock, acoustic and aerodynamic loads; material properties, traceability and fabrication process control; control system redundancy requirements; and required safety factors. Few changes were made in these requirements after the baseline release; however, three changes that came about later as a result of further Space Shuttle system definition are worthy of mention before proceeding.

1. The original life requirement was for 100 missions and 27,000 seconds, including 6 exposures at the "Emergency Power Level" (EPL) of 109 percent. NASA requested a change that would maximize the allowed number of such exposures within the existing design. With the redefined Shuttle, 27,000 seconds was equivalent to 55 missions. A fatigue analysis concluded that if the total number of missions were reduced to 55 then no limit need be placed on the number of exposures at 109 percent. Because of this change, EPL was renamed "Full Power Level" (FPL). [11]

2. Engine mixture ratio was to have been controlled by vehicle command to any value from 5.5 to 6.5. As the Space Shuttle mission was refined, this requirement was first reduced in range to 5.8 to 6.2 and then eliminated altogether in favor of a fixed mixture ratio of 6.0. To take advantage of this, the engine design was modified by reducing various system resistances; and, as a result, system pressures and turbine operating temperatures were reduced.

3. Early in 1978, a definitive shuttle trajectory analysis revealed that throttling all the way to 50 percent power level during the period of maximum aerodynamic loading was not required. The Minimum Power Level (MPL) was raised from

50 percent to 65 percent which allowed further system resistance reductions in subsequent engines.

A series of design verification specifications (DVS) was developed which contained all of the engine design requirements derived from the ICD, CEI, contract statement of work [8], and other sources such as company design standards and good industry practice. The engine level requirements were contained in DVS-SSME-101. The engine component DVSs had similar identifications [12]. Each detailed requirement was listed, its source was identified and the methods of verification (proof that the design meets the requirement) and validation (proof that the requirement is valid) were specified. The methods to be employed for verification and validation were analysis, hardware inspection, laboratory or bench tests, subsystem hot-fire tests, and engine hot-fire tests. Emphasis was placed on obtaining the required proofs at the lowest possible level. These requirements formed the basis for the SSME development program until well into the flight program. Individual DVS task completions were used as benchmark control points or gates to allow continuation of the program for certain critical preplanned activities. The most significant of these was the first flight of the Space Shuttle for which 991 DVS tasks had to be closed. [13] At completion of the DVS program (after the first flight) a total of 4,566 laboratory tests and 1,418 subsystem hot-fire tests had been completed.[12]

THE OBSTACLES

Getting Started

Engine development testing was planned to be conducted at the NASA rocket test site in Mississippi and to begin late in 1974. The Mississippi Test Facility (MTF) had been used for static testing the Saturn (Apollo) launch vehicle stages, and the Saturn test facilities were modified, at NASA direction, to accommodate the SSME. {MTF was later renamed the National Space Technology Laboratory (NSTL) and more recently, the Stennis Space Center (SSC).} In the meantime, component and subsystem testing was also planned for the Coca area of the Rocketdyne Santa Susana Field Laboratory (SSFL) at Chatsworth, California. Existing test facilities were to undergo major modifications to accommodate the turbopumps and combustion devices and various combinations of components arranged in subsystems. During 1973 and early 1974, unforeseen difficulties were encountered with the Coca construction project that eventually led to a schedule slip of about six months. At the same time, procurement delays, weight reduction design changes and required structural improvements caused fabrication of major components to fall behind schedule. [14]

In the summer of 1974, the SSME program was realigned under the leadership of Bob (J. R.) Thompson as the MSFC SSME project manager and Norm Reuel as the Rocketdyne vice president and program manager. Program schedules were adjusted by about six months and increased management emphasis was provided to assure timely completion of the remaining development tasks. The most significant of these involved the Integrated Subsystem Test Bed (ISTB).

The ISTB was originally planned as a "bobtail" engine. It was to be made up of the four turbopumps and two preburners with associated plumbing and controls, but without a thrust chamber assembly (TCA). (The TCA consists of the main injector, the main combustion chamber and the nozzle). The control system included all the required valves and actuators; however, the controller was a remotely located, laboratory-type, rack-mounted computer operating in "single string" (no redundancy). This configuration was one of the subsystems originally planned for subsystem testing; however, during the fact-finding negotiations of July 1971, it was agreed that a shortened version of the TCA would be added to the ISTB. The shortened TCA was to have an area ratio (throat area divided by exit area) of 35 to 1 rather than the flight configuration of 77.5 to 1. The ISTB, then, became essentially an engine assembly which, because of the area ratio reduction, could throttle to 50 percent power level without requiring a vacuum chamber.

With the program realignment, activation of the ISTB test facility planned for the Coca test area (Coca-1C) was deferred in favor of testing the ISTB at NSTL, with the first test scheduled for May 1975. A special management team was formed to determine and implement system and operational changes that would ensure achievement of this very key objective. The team was headed by Dom Sanchini as associate program manager and included Ted Benham as the manufacturing project manager and Dr. Ed Larson as the engineering project manager.The team investigated in detail the production release and fabrication system being used. With the concurrence and help of MSFC project management, Quality and Engineering, changes were made to simplify paperwork and provide quick turnaround for hardware modifications without sacrificing quality or configuration control. This was achieved largely due to the assignment of 25 top design engineers to on-the-floor manufacturing support with authority for on-the-spot approval of material review dispositions and design change rework modifications [15]. The chief engine designer, Bob Crain, supervised the ISTB assembly.

The ISTB first full-up ignition test date had been selected by the Management and Budget Office of the White House as one of the major Space Shuttle program milestones by which that office would monitor the program progress and health. This very important milestone was achieved on schedule. The ISTB was installed in NSTL test stand A-1 (Figure 6) and a countdown demonstration test (Test 901-001) was conducted on May 19,1975. After five short exploratory ignition tests, the full thrust chamber ignition test was conducted on June 23, 1975. The engine development test program was underway.

Component Testing

This history of the SSME contains a review of the major problems encountered during engine testing from the first test of the ISTB to the first flight of the Space Shuttle and, as such, does not go into detail concerning the component and subsystem test program. The history, however, would be incomplete without at least a summary of this important part of the SSME development.

Within the program realignment of 1974, it was decided that the first article of each major component would be allocated to the ISTB. This action would accelerate engine testing and the discovery of any potential major system problems, but would delay the

ISTB Rollout

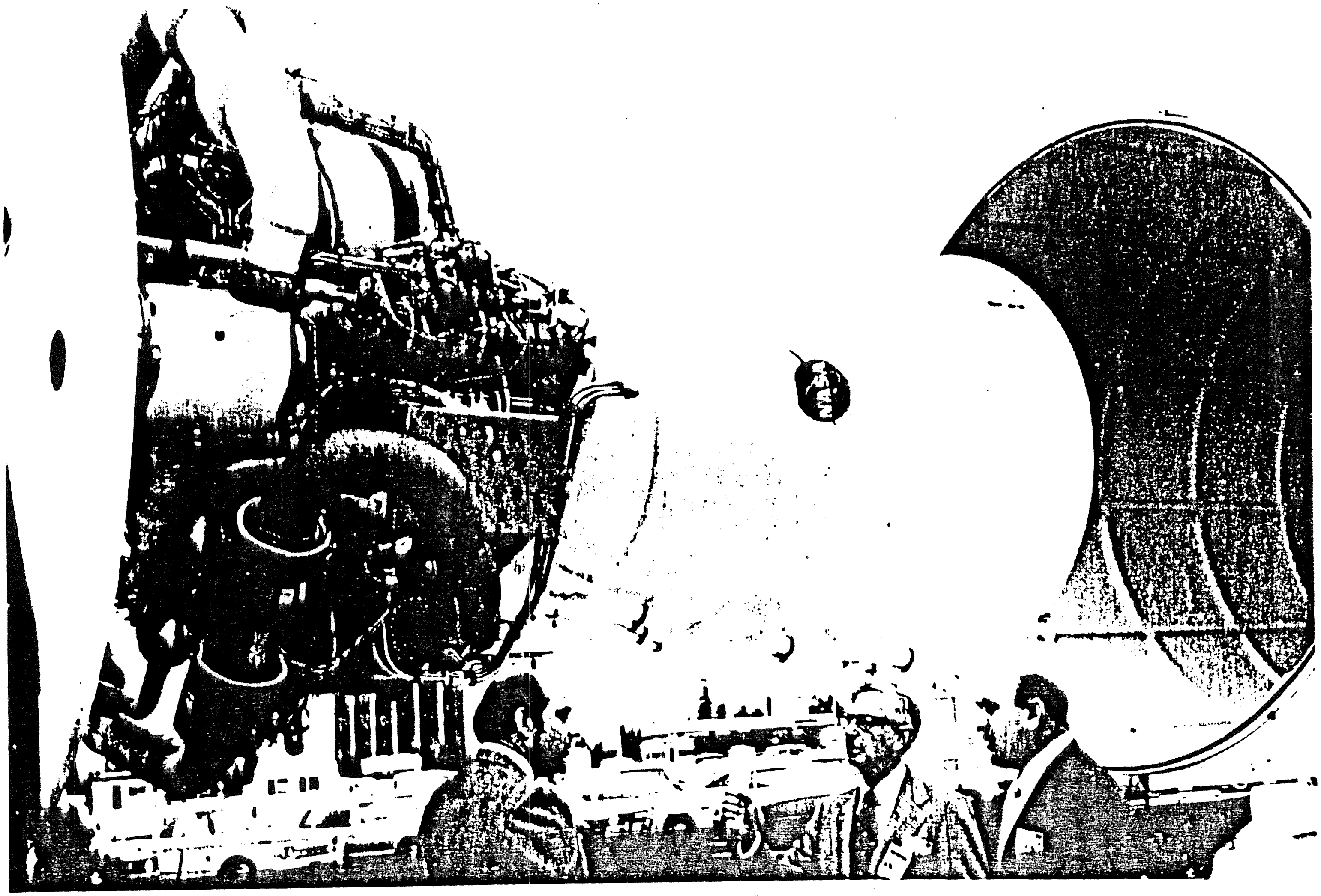

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Figure 6. The Integra[illegible] Subsystem Test Bed

beginning of the component test program until after the second article had been assembled. The component test program began in the same month as the engine test program (May 1975) with the low pressure turbopumps (LPOTP and LPFTP). The two high pressure turbopumps (HPOTP and HPFTP) began testing three months later, in August 1975. The combustion devices test program actually began during Phase B in 1971. The test program planned for the Coca area, however, began with ignition system tests and progressed to preburners and then preburners with the MCC, and, finally, culminated with the TCA (MCC with a 35 to 1 nozzle) in August 1975.

As mentioned before, component tests were used to great advantage in the design verification program delineated in the DVSs. Most of the problems that were encountered, however, were due to the complexity of the test facilities rather than the discovery of component failure modes. The test facilities were designed to accept various combinations of components arranged in subsystems and used facility devices (usually servo-controlled valves) to simulate the engine environment. The turbopump test stand had approximately 2,000 valves including 24 which were servo-operated. Preburner propellants were supplied from a 14,000 psi system with valves weighing as much as five tons. One of the more significant problems occurred on Coca-1A early in 1976. The oxidizer subsystem, which consisted of an LPOTP, HPOTP and OPB (actually a half powerhead which included the preburner) was being tested. At 19 seconds into Test 740-007, a facility rotary flowmeter failed, releasing flowmeter blades into the LOX flow stream. The blades initiated a fire at a downstream throttle valve which burned, causing a decrease in flow resistance. The decrease in flow resistance caused enough of a change in the operation of the HPOTP that it cavitated, lost axial thrust control and began to rub internally. This resulted in a major fire which caused significant damage to the components and the facility [16]. A similar failure occurred on a fuel subsystem test on Coca-1B the following year. Test 745-018 experienced a major fire beginning with a fire in a facility throttle valve caused by cavitation-induced erosion [17].

With the advent of engine and component testing and its attendant loss of hardware, it soon became evident that the planned hardware was inadequate to support the scheduled test program and keep up with the attrition realized from the development problems. This deficiency was to remain with the program for many years. The component test program, if pursued as originally planned, would have drained valuable resources from the engine test program to develop the complicated test facilities. The NASA administrator, Dr. Robert Frosch, stated in testimony to the Senate Subcommittee on Science, Technology and Space, that "...we have found that the best and truest test bed for all major components, and especially turbopumps, is the engine itself." [18] Largely due to the lack of sufficient resources to pursue an aggressive component test program in addition to the engine test program, the Coca area test facilities were gradually phased out from November 1976 to September 1977.

Engine Testing

The engine test program is summarized in Figure 7. It shows the number of engine tests as a function of calendar time from the first test in May 1975 to the first flight in April 1981. Also shown are the total test seconds and the test seconds at rated power level. Superimposed on the accumulated tests plot are indicators showing the initial

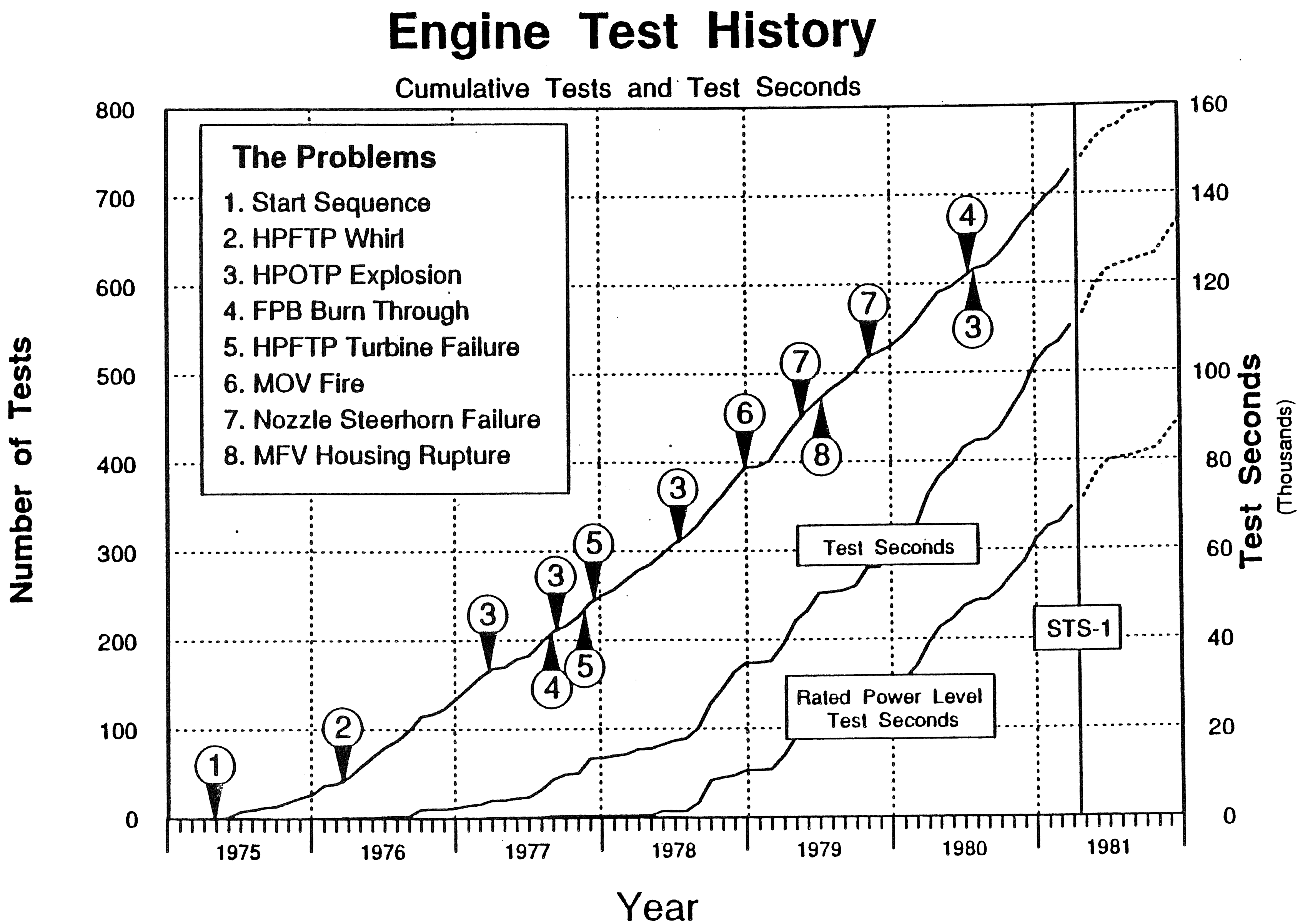

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Figure 7. Engine Test Problem Summary

dates of the major engine problems that are to be discussed in the following chapters. The eight problems listed were chosen as being the most significant as they relate to flight safety; and, even though some other problems also caused loss of resources during the program, the recovery effort for these eight was judged to be the most difficult in terms of what had to be done to allow program continuation and to assure a safe first flight.

Problem Management

Before continuing into the discussion of development problems encountered during the engine test program, it seems appropriate to summarize the management techniques employed by the SSME program to expedite the solution of these problems. The two consistent management devices were the "special team" and the "5 o'clock meeting". Although not unique to Rocketdyne or the SSME, both devices proved to be quite valuable in aiding the timely solution of these problems.

For every significant problem, a special team was formed under an autonomous team leader to whom appropriate organizational authority was delegated. Sometimes a separate MSFC team was formed and at other times the Rocketdyne and MSFC teams were combined. Full time dedicated team members were assigned to each team, representing all the technical disciplines required to solve the problem and return the program to normalcy. Team members were usually technical managers and included such specialties as structural analysis, dynamics, materials, thermodynamics, metallurgy, systems, quality control, data handling, component design and test planning. The teams were charged with the multiple tasks of identifying the problem cause, establishing problem control sufficiently to enable safe resumption of testing, determining and implementing the ultimate redesign or other action to prevent further recurrence of the problem, and providing proof to Rockwell and NASA management that the problem was eliminated or controlled. The team leader was chosen for each problem as it occurred and was usually the engineering director with the most appropriate background and expertise. Of the first 20 special teams formed, Ed Larson, director, Design Technology, was assigned as team leader for half of them. It is likely that he would have been assigned to others except for the fact that he had not yet concluded an investigation of a previous problem.

From the first test until long after the first flight, the five o'clock meeting was a daily ritual set aside to recap that day's activity and progress (or lack thereof) on the most significant current problem. Dom Sanchini, who was appointed vice president and program manager after the ISTB was delivered to NSTL for testing, conducted the meeting in his office with key members of his staff and selected individuals associated with the problem. In addition to problem team leaders, the regular attendees were: Willy Wilhelm or Paul Fuller (chief program engineer), Jerry Johnson (associate program manager, Engine Systems) and Bob Biggs, chief project engineer, who was responsible for directing the engine test program. For turbomachinery problems, the meeting included Jim Hale (associate program manager, Turbomachinery) and Joe Stangeland (director, Turbomachinery). If the problem involved combustion devices, then Don Mikuni (associate program manager, Combustion Devices) and Erv Eberle (director, Combustion Devices) would attend. Quite often, the meeting would be attended by Bob Thompson who, in spite of the fact that he lived in Huntsville, Alabama, spent a lot of time in the Rocketdyne Canoga Park, California, plant, and maintained a permanent office next door to Dom Sanchini.

Start and Shutdown

The first hurdle that had to be overcome in the engine test program was to learn how to safely start and shut down the engine. Five years of analysis had produced sophisticated computer models that attempted to predict the transient behavior of the propellants and engine hardware during start and shutdown. With these models, the basic control concepts were defined and initial sequences were developed [19]. The models had shown the engine to be sensitive to small changes in propellant conditions and that timing relative to opening the propellant valves was critical. Expecting difficulties, a cautious step-by- step plan was followed to explore the start sequence in small time increments. Using this approach, it required 19 tests, 23 weeks and 8 turbopump replacements to reach 2 seconds into an eventual 5 second start sequence. It took an additional 18 tests, 12 weeks and 5 turbopump replacements before momentarily touching MPL. A safe and repeatable start sequence was eventually developed by making maximum use of the engine mounted computer (MEC) to control the propellant valve positions. Without the precise timing and positioning allowed by the MEC, it is doubtful that a satisfactory start could have been developed.

Prior to starting the engine, there is a period of time referred to as the start preparation phase. At the beginning of this time period, the oxidizer side of the engine is purged with dry nitrogen to eliminate moisture and the fuel side is purged with dry helium to eliminate air as well as moisture. This is done because the temperature of liquid hydrogen (LH2) is cold enough (less than 40 R) to freeze air into a solid block of ice. After the engine is properly purged, the cryogenic propellants are allowed to flow into the engine to begin thermal conditioning.

Figure 8 is an SSME schematic showing the propellant flow paths and the location of the primary propellant valves relative to the other components. During the propellant conditioning period, LH2 fills the fuel side of the engine down to the main fuel valve (MFV), which is a single shutoff valve for all of the fuel. A small recirculation flow is maintained by flowing through a bleed valve located at the MFV to an overboard dump line or pumped back to the LH2 inlet. Liquid oxygen (LOX) fills the oxidizer side of the engine down to the three oxidizer valves. The main oxidizer valve (MOV), oxidizer preburner oxidizer valve (OPOV) and fuel preburner oxidizer valve (FPOV) act as three parallel shutoff valves for the LOX. Recirculation flow of LOX is maintained by flowing through a bleed valve located at the FPOV to an overboard dump system. The small recirculation flows are maintained for an hour or more to chill the four turbopumps to cryogenic temperatures and to eliminate gas pockets in the propellant feed system.

During the propellant system chill down, the MEC continually monitors the engine to assure that all valves are in the proper position and conducts an automatic checkout of the control system 50 times every second to verify proper operation and retention of full redundancy. About four minutes before the engine start command, the final engine purge is turned on. Dry helium is introduced downstream of the main fuel valve to displace any gas that would freeze at LH2 temperature. The MEC uses engine-mounted sensors to measure propellant temperatures and pressures. When the engine

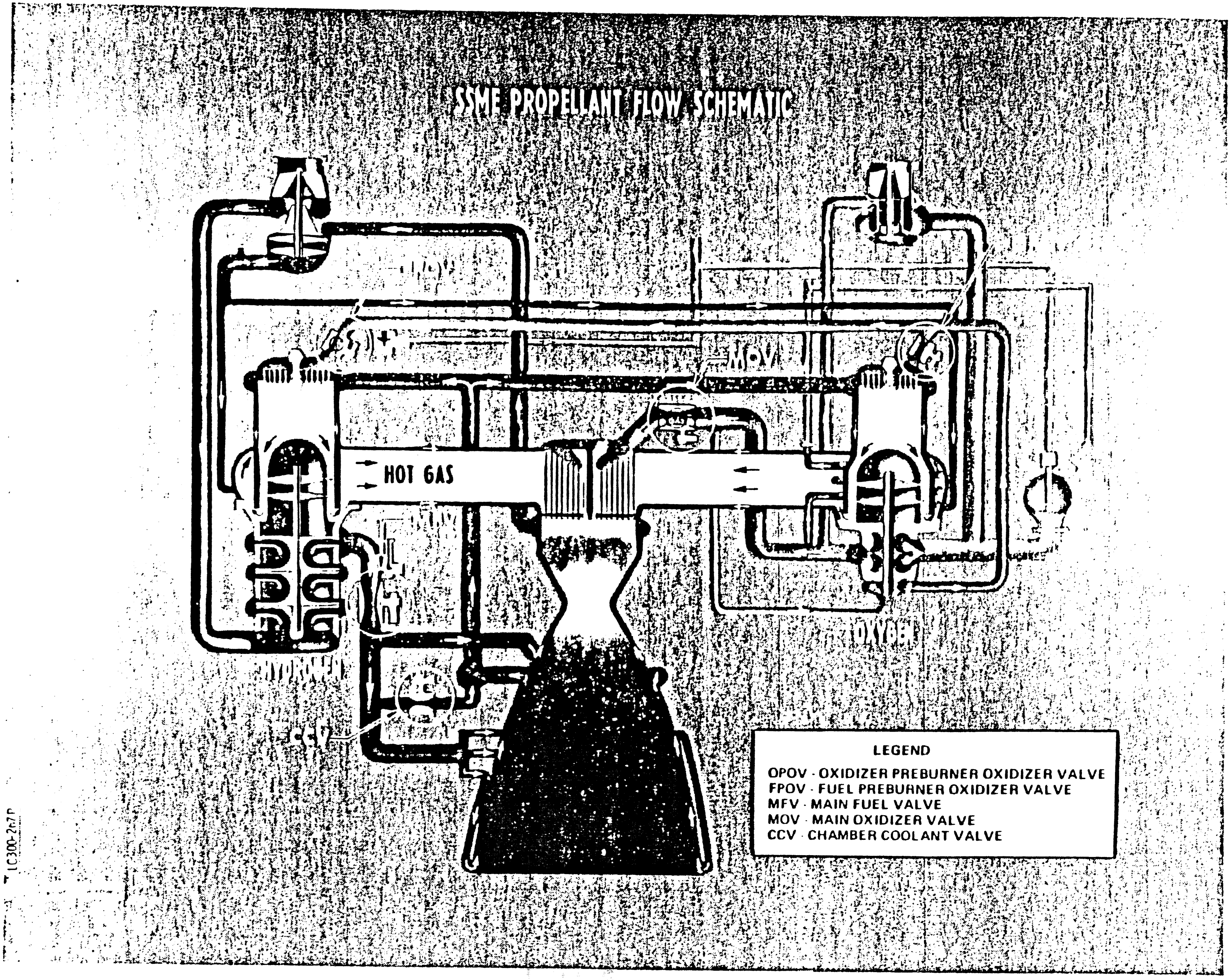

Figure 8. SSME Control Valves

has been purged and all parameters are in an acceptable range for starting, and if the control system checkout finds no failures, the MEC adopts an "engine ready" status. The status word in the data stream being relayed to the vehicle or test facility is changed to reflect that all conditions are acceptable for starting the engine. About three seconds before engine start, a "start enable" command is sent to the MEC. The MEC then closes the two bleed valves and waits for a start command.

When a start command is received, the MFV is immediately ramped to its full open position in two-thirds of a second (see Figure 9). This enables the LH2 to fill the downstream system and begin to power the high pressure turbines. The latent heat of the hardware imparts enough energy to the hydrogen to operate as an "expander-cycle" engine for the early part of the start sequence. This eliminates the need for any auxiliary power to initiate the start sequence, however it also creates a thermodynamic instability which is referred to as the fuel system oscillations. When the cold LH2 begins to flow into the thrust chamber nozzle, the hardware latent heat causes the hydrogen to expand rapidly, creating a flow blockage and momentary flow reversal. The result is a pulsating fuel flow rate with an unstable pressure oscillation at a frequency of approximately 2 Hz. The oscillations continue to increase in magnitude with dips (reductions in pressure) occurring at approximately 0.25 ,0.75 and 1.25 seconds, until the establishment of MCC chamber pressure causes it to stabilize after 1.5 seconds. Events prior to stabilization had to be made to conform to the idiosyncrasies of the fuel system oscillations.

Simultaneously with the opening of the MFV, electrical power is provided to the spark plugs in the augmented spark igniters (ASI) included in each of the three combustors. The ASI will then ignite the combustors when both fuel and oxidizer are present in the proper mixture ratio. The fuel is provided first by the MFV being opened and then the oxidizer is provided later for each combustor separately through the three oxidizer valves. Each valve has an ASI LOX supply line that allows LOX to flow to the ASI upon initial valve motion (about 5 percent). The proper mixture ratio for ignition is achieved by the second dip in pressure caused by the fuel system oscillations.

After the MFV starts to open, the three oxidizer valves are separately subjected to a series of position commands intended to precisely control the oxidizer system priming times for the three combustors. Priming is the process of filling the system with liquid, as with an old hand-cranked water pump. An oxidizer system is said to be "primed" when it is filled with liquid down to the combustor such that the flow rate entering the injector is equal to the flow rate leaving the injector to be burned in the combustor. This event generally results in a rapid rise in combustion chamber pressure. The target priming times for the three combustors are a tenth of a second apart; FPB prime at 1.4 seconds, MCC prime at 1.5 seconds and OPB prime at 1.6 seconds. Although part of the valve positioning is accomplished under a limited form of closed loop control, it is merely a convenient method of commanding the valves to a predetermined position and therefore will be treated as if it were all done as open loop commanded positions as a function of time.

The first oxidizer valve to be commanded is the FPOV. After a delay of 0.100 seconds, the FPOV is ramped to 56 percent open at its maximum slew rate. At 0.72 seconds, the FPOV is given a "notch" command to close about 10 percent and then reopen.

SSME Start Sequence

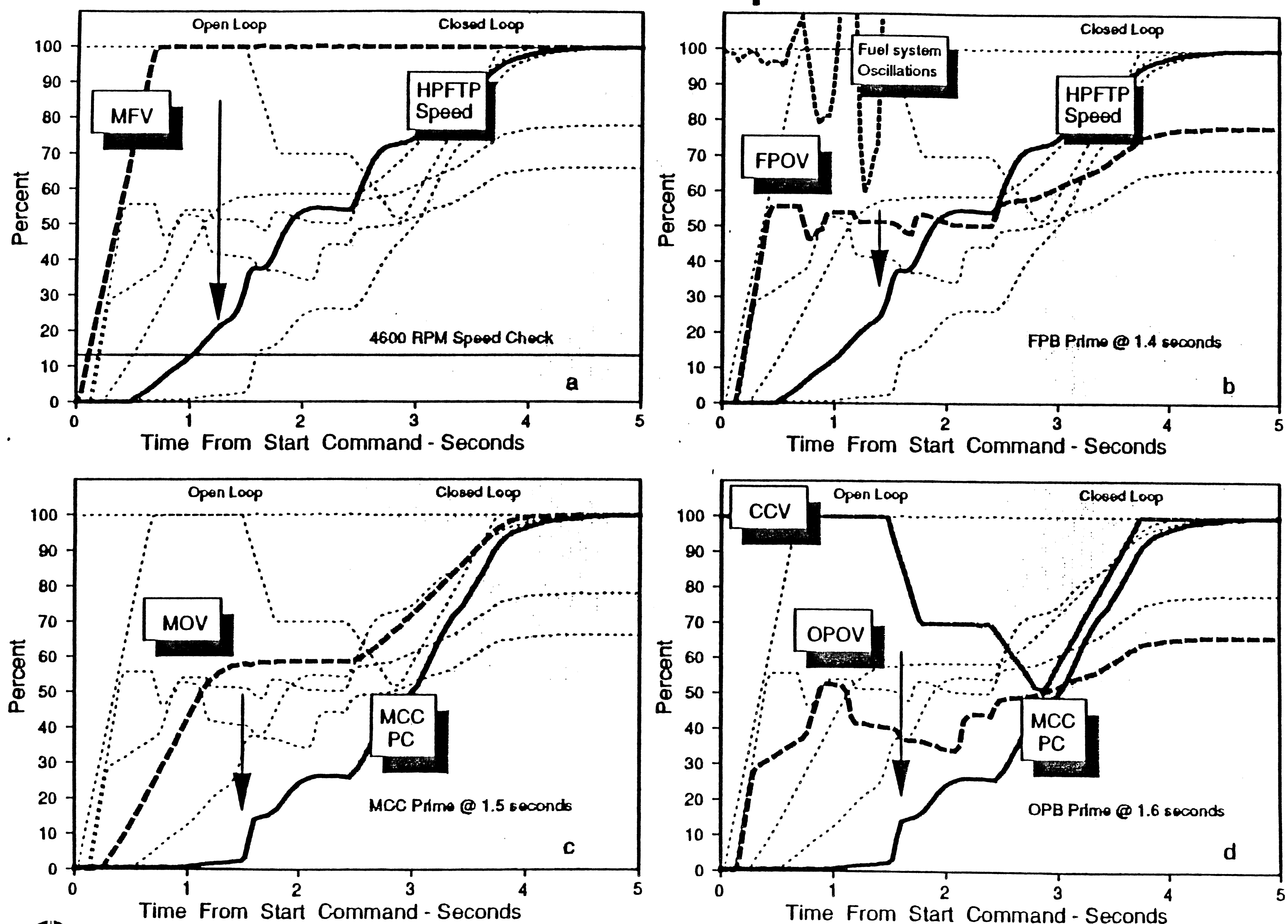

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Figure 9. Start Sequence

This is done to compensate for the second pressure dip caused by the fuel system oscillations and avoid damaging temperature spikes in the HPFTP turbine. During this dip, the FPB is ignited and the additional power causes a slight acceleration in the HPFTP speed. Just prior to the third fuel system oscillation pressure dip, the FPOV is given another notch command which is maintained throughout the priming sequence.

A safety check is made at 1.25 seconds to assure that the HPFTP speed is high enough to safely proceed through the priming sequence. The speed must be high enough at MCC prime to be able to pump hydrogen through the downstream system against the back pressure rise created by the MCC prime or an engine burnout will occur due to the resulting oxygen-rich combustion. It was determined from test experience that if the speed were to be less than 4,600 RPM at 1.25 seconds (Figure 9a), then it would likely be too low at MCC prime to maintain pumping capability. The engine must be shut down at 1.25 seconds because if the speed is discovered to be too low later in the start sequence, there is insufficient time to react and shut down safely.

When the FPB prime occurs at 1.4 seconds, there is a rapid rise of pressure at the inlet to the HPFTP turbine. Since the turbine back pressure is not provided until MCC prime, this pressure rise causes a high turbine pressure ratio and a significant acceleration in the HPFTP speed (Figure 9b). The higher HPFTP speed is desirable for a cool fuel-rich start, however, the turbine back pressure must be applied (MCC prime) soon to prevent a runaway condition.

MCC prime is primarily controlled by positioning of the MOV. After an initial delay of 0.200 seconds, the MOV is slowly ramped to just under 60 percent open. This combination of time delay, ramp rate and position provides a LOX flow rate that causes MCC prime to occur at 1.5 seconds and creates an engine system balance that will produce a safe low mixture ratio (between 3 and 4) for the stabilized operation just prior to activating the closed loop thrust control system at 2.4 seconds. When MCC prime occurs at 1.5 seconds, it causes a rapid rise in MCC chamber pressure (Figure 9c) which, because it increases the turbine back pressure, acts as a break to decelerate the HPFTP (Figure 9b).

The OPOV is used to control OPB prime. Its initial opening is after a delay of 0.120 seconds; however, the opening only retracts the valve inlet seal, which is designed to provide sufficient oxygen to ignite the ASI and to have a small leakage flow into the OPB injector. The valve is designed so that the major flow path does not start to open until an indicated position of 46 percent. The slow ramp shown in Figure 9d has no effect on the OPB LOX flow rate except to delay until 0.84 seconds when the main flow path through the valve starts to open. This flow path is partially open for about a third of a second before it recloses and the OPB is again run on valve leakage flow. The timing for this opening is scheduled to provide sufficient oxygen to allow the ASI to ignite the OPB before the second fuel oscillation pressure dip recovers and causes a significant decrease in mixture ratio. The next opportunity for ignition would be about a half a second later. With valve leakage flow, OPB prime occurs at 1.6 seconds and causes an increase in drive power to both high pressure turbines. The power increase stabilizes at about 2 seconds with the MCC chamber pressure at approximately 25 percent of RPL. During this time the chamber coolant valve (CCV), which was full open at start, is throttled down to 70 percent in order to force additional coolant flow through the MCC. The engine is allowed to run at this condition until 2.4 seconds to assure stable operation. The additional time period of 0.4 seconds is to allow for and absorb normal variations in propellant pressures and temperatures.

By using the engine-mounted sensors, the MEC verifies proper ignition and operation of the three combustors at 1.7 seconds and again at 2.3 seconds. If no malfunctions are discovered, the closed loop thrust control system is activated at 2.4 seconds. The MEC compares the measured MCC chamber pressure to a pre-programed chamber pressure ramp to RPL and modulates the OPOV in an attempt to zero out any differences. During this time, the FPOV is simply moved by the MEC with position changes that are proportional to the amount of OPOV movement, and the CCV is commanded open at a rate commensurate with the commanded chamber pressure ramp rate. Because of the engine dynamic response characteristics, the resulting chamber pressure lags behind the command by about 0.200 seconds. At 3.8 seconds the closed loop mixture ratio control system is activated using the FPOV to adjust fuel flow rate until the commanded mixture ratio is achieved. At 5 seconds the engine has achieved stabilized operation at RPL with a mixture ratio of 6.

Significant constraints were placed on the start sequence by the engine design characteristics. The priming sequence is the most critical. Very high (damaging) temperature spikes occur if any combustor prime coincides with the pressure dips caused by the fuel system oscillations. The timing of the sequence relative to each other is also critical. If the FPB prime were late or the MCC prime early, the insufficient fuel pump speed would cause very LOX-rich operation with major burning of the engine hardware. If the OPB prime were early or the MCC prime late, a rapid acceleration of the HPOTP could lead to its destruction. Because of the very compact design of the high pressure pumps (highest horsepower to weight ratio ever achieved), the very low inertia causes them to accelerate and decelerate extremely quickly under abnormal conditions. If only the normal operating torque were applied to the HPOTP without the fluid load applied (gas in the pump or in cavitation) it could accelerate from a dead stop to a destructive overspeed condition in less than a tenth of a second. The acceleration rate under this condition is almost 400,000 RPM per second [20].

The initial start sequence development tests on the ISTB were limited to starting to MPL (then 50 percent of RPL). The first test to achieve MPL was Test 901-037, a 3.36 second start transient test, at the end of January 1976. The first test to achieve stabilized operation with the closed loop mixture ratio control system activated was Test 901-042 on March 8, 1976. Operation at RPL was not achieved until January 1977 (Test 901-095). Although the ISTB start development tests resulted in a start sequence that would allow the continuation of the ground test program, the final start sequence was not arrived at until the end of 1978. The current operation of the preburner valves evolved over that time period to better compensate for variations in external conditions and in response to specific problems as they occurred.

The last significant start problem occurred on October 3, 1978. Test 902-132 on Engine 0006 began with an HPFTP breakaway torque that was slightly higher than normal. At the same time, the MOV actuator was misaligned such that the valve was 2 percent further open than indicated by the valve position measurement. The combination of these two unrelated events led to a slight reduction in HPFTP speed and an early MCC prime. The HPFTP was unable to pump against the downstream pressure, so the turbine horsepower was dissipated by heating up the LH2 and causing it to vaporize. This resulted in a major burnout of the turbines and hot gas system [21].

The evolution of the SSME start sequence has resulted in a repeatable and reliable start; however, it must be remembered that the SSME is a very high-powered, low-inertia system that is susceptible to extreme energy releases if subjected to abnormal conditions. An error in valve position of 2 percent (1 percent for the OPOV) or a timing error of a tenth of a second can lead to significant damage to the engine. Because of the inability to automatically compensate for unexpected variations in external conditions (such as the Engine 0006 incident), it is required that all new engines undergo a 1.5 second priming sequence verification test before a start is attempted.

As with the start, the SSME shutdown must contend with high power and low inertia; however, it does not have the further complications of fuel system oscillations and critical priming sequences. Although some problems were encountered during the evolution of the shutdown sequence, they were not as significant as the start problems. This discussion will be limited to a brief summary of the reasons for the various features of the final shutdown sequence. The shutdown sequence, which is totally open loop, is shown graphically in Figure 10.

The goal of the shutdown sequence is to shut the engine down as quickly and as safely as possible. The initial step is to remove power from the HPOTP turbine to reduce the LOX flow faster than the fuel flow reduction. This reduces the engine mixture ratio and, thereby, the combustion temperatures. The initial OPOV closing rate is limited to 45 percent per second because a faster rate would violate the ICD requirement for a maximum thrust decay rate of 700,000 pounds per second, the orbiter structural limit. The FPOV initial closing rate was chosen to assure that the oxidizer side will power down first. The OPOV and FPOV positioning for the rest of the shutdown maintains a balance with low mixture ratio and maximum oxidizer pressure decay short of allowing hot gas backflow into the oxidizer.

The MOV is scheduled to close as quickly as possible to terminate all LOX flow. The closing rate, however, is limited to 40 percent per second because the MOV must allow enough LOX flow to keep the MCC chamber pressure high enough relative to the turbine inlet pressures. Any further reduction in turbine back pressure would cause a turbine pressure ratio increase and a potential overspeed condition.

The CCV is partially closed to force more coolant flow into the MCC and nozzle to accommodate the increased heat load due to throttling. The MFV is held open for more than a second to assure a very fuel-rich shutdown, and then the MFV and CCV closing schedules are the fastest possible without causing damage to the HPFTP. While the HPFTP is coasting to a halt, it is necessary for the pump to continue pumping. If the flow rate through the pump were to be reduced to below the critical value, the excess power would be dissipated by vaporizing the LH2. The conversion from a liquid to a gas would cause the loss of axial thrust control and significant internal rubbing. After five seconds the HPFTP speed is low enough (below 7,000 RPM) that the "boilout" effect is no longer damaging.

SSME Shutdown Sequence

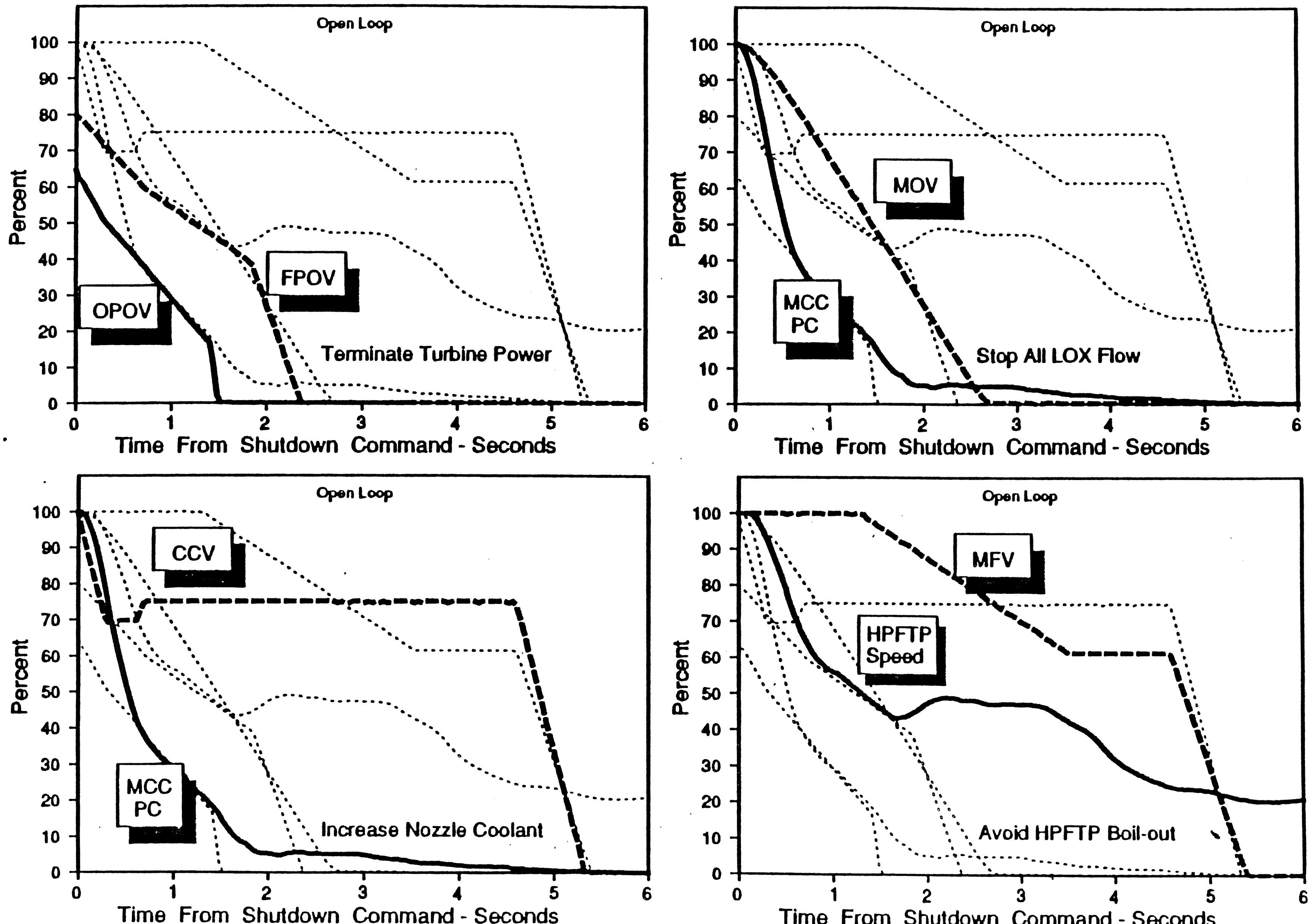

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Figure 10. Shutdown Sequence

High Pressure Fuel Turbopump Subsynchronous Whirl

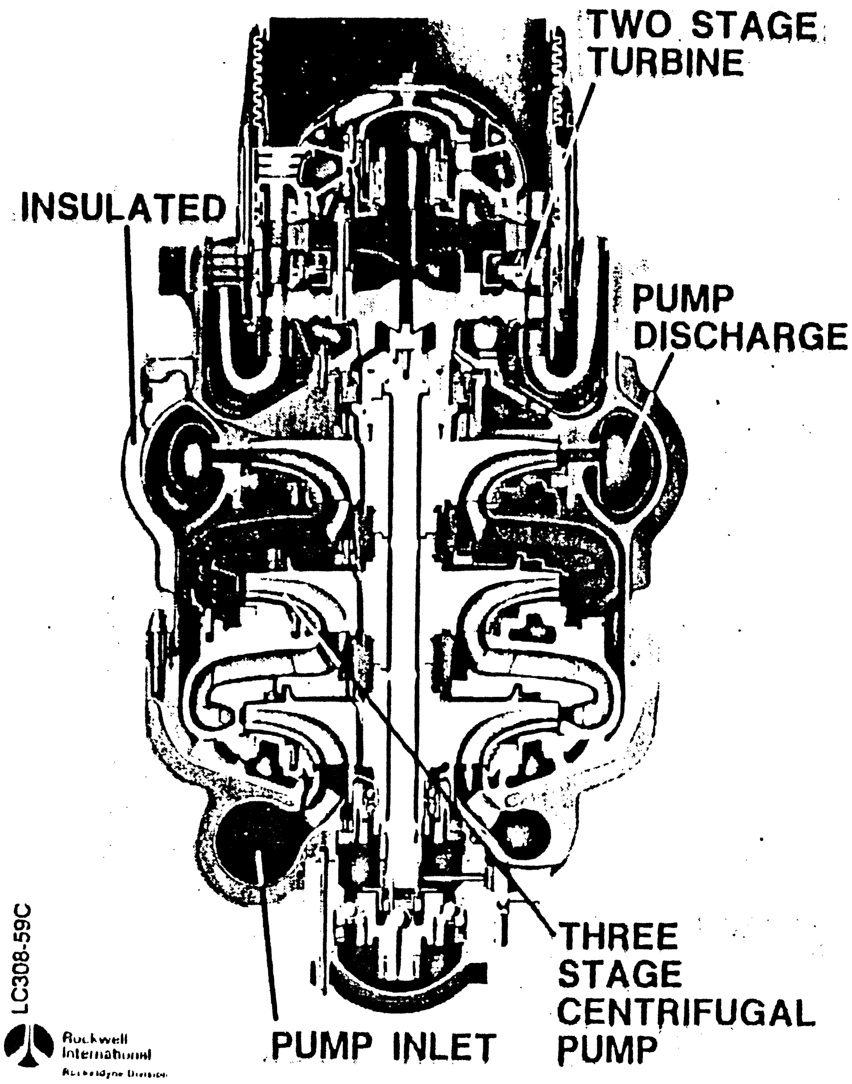
On March 12, 1976, four days after the first stabilized test on the ISTB, Test 901-044 was scheduled for a sixty-five second exploratory test at 50 percent power level with one second at 65 percent power level. Although 65 percent power level was successfully demonstrated (the highest achieved up to that time), the test was terminated at 45.2 seconds due to loss of axial thrust in the HPFTP. After the test, the HPFTP was bound up and could not be rotated with the turbopump torque test tool (normal post-test checkout). This condition was later found to be caused by the failure of the HPFTP turbine end bearings. A review of the test data revealed two major abnormalities. The HPFTP turbine gas temperature increased by almost 200 R during the test. This and other measurements indicated a significant loss of turbine efficiency during the test. In addition, high frequency vibration measurements on the HPFTP indicated a large amplitude vibration at a frequency of about one-half of the fuel pump speed. This vibration characteristic was immediately recognized as a rotordynamic instability known as subsynchronous whirl. Although the phenomenon had not been predicted to occur, the potential for this instability had been hypothesized as much as three years earlier [21]. A meticulous review of prior engine and component test data revealed that the phenomenon was present to some extent in most of the tests that exceeded 17,000 RPM; however, it was overshadowed by the transient nature of the tests and the persistence of an HPFTP axial balance piston problem [22].

To expedite solution of this problem, a combined Rocketdyne-MSFC team was formed under the leadership of Matt Ek, Rocketdyne vice president and chief engineer, and Otto Goetz, MSFC's leading turbomachinery expert. The team was ultimately expanded to include the foremost experts in the field of rotordynamics from industry, government, and the academic community in the United States and Great Britain [22].

The HPFTP is shown in cross section in Figure 11. It is a three-stage centrifugal pump driven by a two-stage reaction turbine which is part of the same rotating assembly. When operating at FPL, each of the three twelve-inch impellers develops over 60,000 feet of head rise at an LH2 flow rate of 17,000 GPM and a speed in excess of 36,000 RPM. The two-stage, eleven-inch diameter turbine delivers 75,000 horsepower with an efficiency greater than 80 percent at a pressure ratio of 1.5. With a total assembly weight of 775 pounds, and a length of just over three feet, the HPFTP is the most compact, highest power density rotating device known today. The power density of almost 100 horsepower per pound was an order of magnitude increase over prior turbopump designs.

The 130 pound rotating assembly is designed around a central drawbolt threaded into the second stage turbine disc. The turbine disc and the three impellers are rotationally piloted by splined sleeves which also perform the functions of interstage seals and journal bearings. This stack-up is drawn together by a nut on the first-stage impeller end of the drawbolt. The first-stage turbine disc is bolted to the second-stage turbine disc with a curvic coupling. Radial positioning is accomplished by two sets of angular contact, duplex spring-loaded 45 millimeter ball bearings spaced 23.3 inches apart. The bearings are not lubricated but are cooled with LH2 during operation. Axial positioning is maintained by an LH2 pressure balance with the back side of the third-stage

SSME HIGH PRESSURE FUEL TURBOPUMP



KEY PERFORMANCE PARAMETERS	RPL (100%)	FPL (109%)
PUMP INLET FLOWRATE (LB/SEC)	149.1	162.5
PUMP INLET PR (PSIA)	223	238
PUMP DISCH PR (PSIA)	6110	6872
PUMP EFFICIENCY	.763	.760
TURBINE FLOWRATE (LB/SEC)	159	178
TURBINE INLET TEMP (°R)	1795	1900
TURBINE PRESSURE RATIO	1.411	1.400
TURBINE EFFICIENCY	.839	.842
TURBINE SPEED (RPM)	34,390	36,600
TURBINE HORSEPOWER	61,400	74,900

DIAMETER: 21.7 INCHES
LENGTH: 37.8 INCHES
WEIGHT: 775 LBS

Figure 11. High Pressure Fuel Turbopump

impeller serving as a double-acting self-compensating balance piston. Balance piston pressure is supplied through an axially sensitive overlapping orifice at the impeller discharge and vented through an axially sensitive orifice at the hub of the impeller rear shroud. When at rest and at low speed, the rotor is supported by a thrust-bearing assembly at the bottom of the rotor (pump inlet end).

The HPFTP subsynchronous whirl was a violent instability which caused a gyration of the rotor in the direction of normal rotation at a frequency of about half of the pump speed (see Figure 12). This caused a forward precession of the rotor, which was actually an orbiting of the normal rotating axis. Being a true instability, the whirl was self-initiating and would usually start when the pump speed exceeded twice the first critical speed of the rotating assembly, with an inception frequency equal to the first critical speed (originally about 8,500 RPM). The amplitude would increase rapidly; and within half a dozen cycles, with bending of the rather flexible rotor, the normal clearances would be breached and internal rubbing would occur at many locations. With clearances closed and bearing supports bottomed out, the system stiffness increased significantly, preventing further increase in amplitude (limit cycle) and raising the first critical speed and, therefore, the whirl frequency. Bearing loads in the limit cycle condition were higher on the turbine end than the pump end by a factor of three, and a significant number of turbine bearing failures were experienced.

A multidisciplined approach was pursued by the team, which included historical research, literature surveys, mathematical models, and consultations with universities and other companies with related knowledge or experience. A vigorous test program included laboratory, component, subsystem and engine tests. Twenty-two potential drivers were identified and analyzed; however, it was eventually concluded that two factors were far more significant than all the others. The most significant destabilizing effects were hydrodynamic cross-coupling of the pump interstage seals combined with the low natural frequency of the rotating assembly [22]. These effects were attacked by a series of design changes to decrease cross-coupling drivers and provide damping at the seals and to increase the rotor critical speeds by stiffening the shaft and bearing supports. Over a ten-month time period, the whirl inception speed was gradually increased from 18,000 RPM (below MPL) to above 36,000 RPM, which allowed whirl-free operation to above RPL.

As the design changes allowed operation at higher speeds, it became evident that the turbine bearings were being overheated by some mechanism unrelated to whirl. Instrumentation added to the turbopump to gain data for the whirl problem indicated that inadequate hydrogen coolant flow was allowing hot gas to backflow into the bearing. Detailed analysis of the cooling system finally identified the existence of a free vortex at the base of the pump shaft through which the coolant flow was provided. The addition of a baffle at this location changed the free vortex to a forced vortex and reduced the pressure loss from 500 psi to 12 psi [23]. With this change, and the whirl problem eliminated, the HPFTP was capable of supporting the engine test program in mid-January 1977. For the first time, the SSME could be tested for extended durations.

HPFTP Whirl Vibration Characteristics

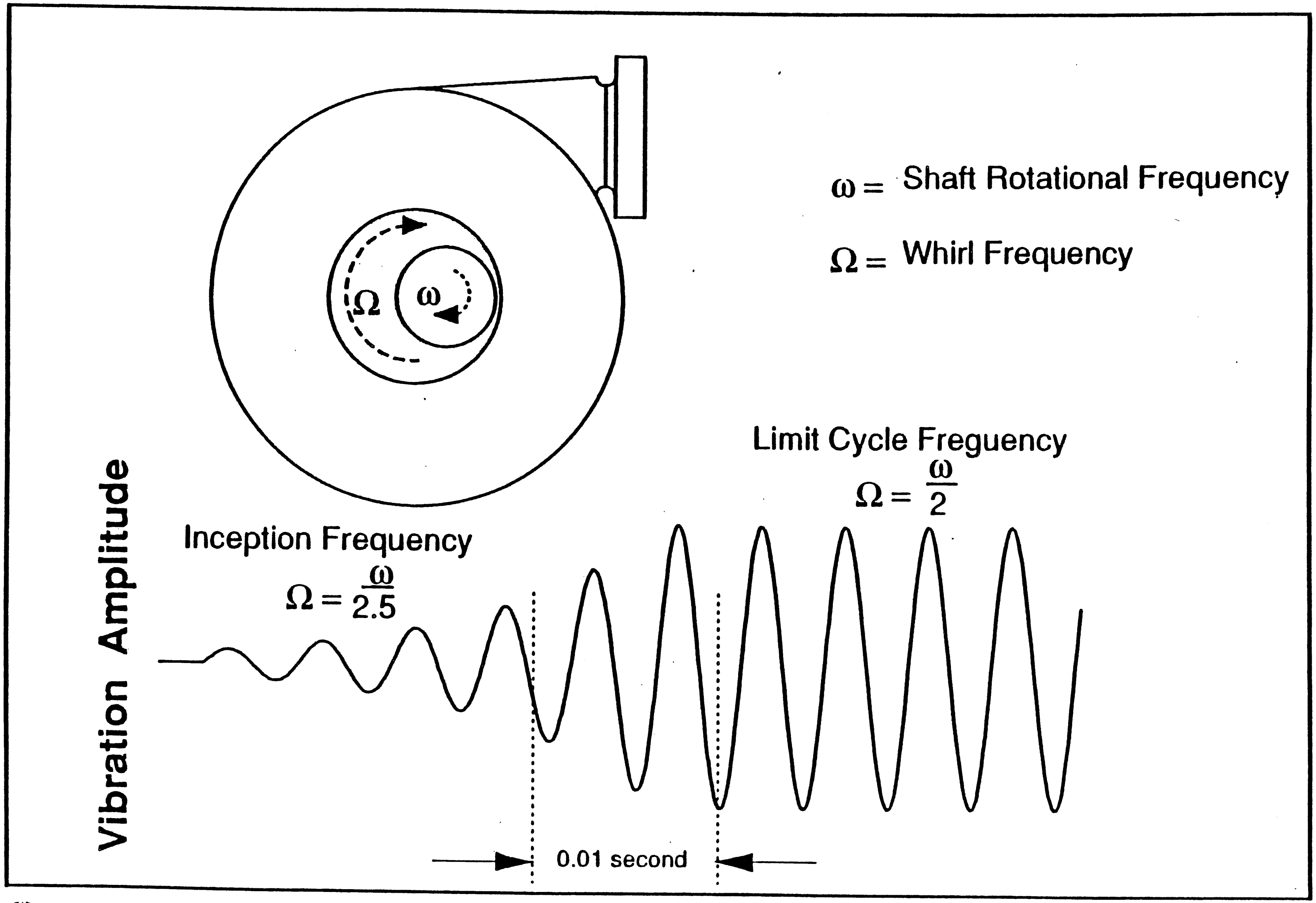


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Figure 12. HPFTP Whirl Vibration Characteristics

High Pressure Oxidizer Turbopump Explosions

LOX pump explosions are nightmarish events in rocket engine development programs. The cost in program resources is quite severe because the turbopump assembly and surrounding hardware are usually lost to the program for any future use. But even more significant is the fiendish nature of the failure. Once a fire has been ignited in the high pressure LOX environment, it readily consumes the metals and other materials that make up the hardware. In most cases, the part that originated the failure is totally destroyed, leaving no physical evidence as to the failure cause. Program management is often left in a quandary as to what to do to prevent further occurrence of the failure. This leads to a process of speculating on possible failure causes and fixing everything that it could be.

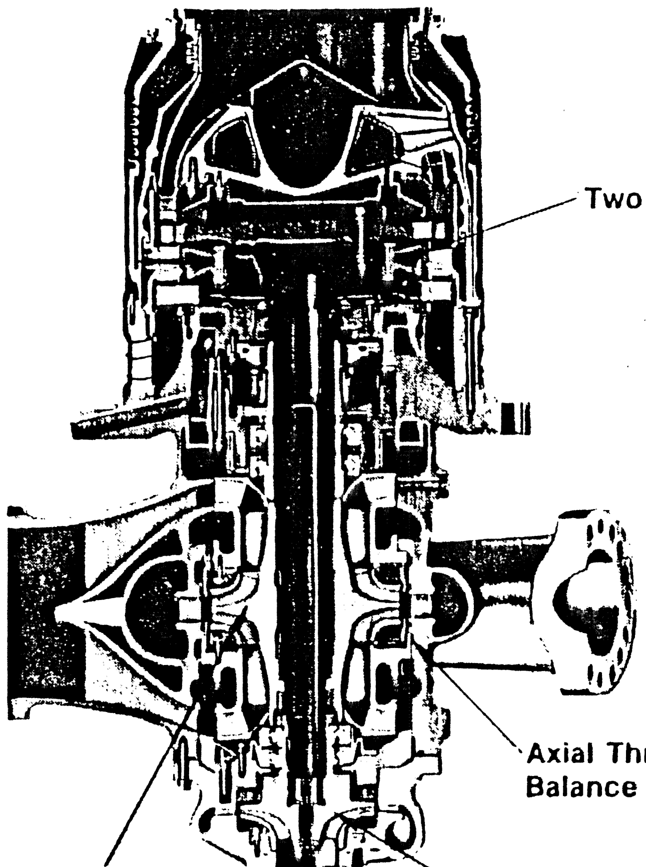
In the time period between the solution of the HPFTP whirl problem and the first shuttle flight, the SSME program experienced four HPOTP explosions. Two of them were caused by internal design flaws which had to be rectified. The other two, while not representing design problems, did significantly impact the program resources in terms of available hardware and required recovery time. All four are discussed following a brief description of the HPOTP.

The HPOTP assembly contains two pumps and a turbine on a common shaft which rotates at a speed of almost 30,000 RPM (see Figure 13). The main pump is a double-entry, single-discharge centrifugal pump with a built-in inducer on each side of the dual-inlet impeller. It has an overall diameter just under seven inches and pumps up to 7,500 GPM of LOX at a pressure in excess of 4,500 psia. The smaller (five inch diameter) boost pump is a single-stage impeller without an inducer and is separately mounted on the bottom end of the turbopump shaft. In supplying LOX to the preburners, it pumps about one tenth as much LOX as the main pump, while increasing the pressure another 3,000 psi. The eleven-inch diameter two-stage reaction turbine delivers over 28,000 horsepower with an efficiency of almost 80 percent while operating with a pressure ratio slightly over 1.5.

The turbopump shaft and the second-stage turbine wheel are made as a one piece construction, with the first-stage turbine wheel bolted to the second-stage wheel and piloted by a curvic coupling. The rotating assembly is radially positioned by two sets of angular contact duplex spring-loaded ball bearings. One set is located between the main pump and the boost pump and the other set is located between the main pump and the turbine. The pump end bearings are 45 millimeter ball bearings and are cooled by a small LOX flow rate which is taken from the boost pump impeller outlet and discharged into the main pump lower inducer inlet. The turbine end bearings are 57 millimeter bearings and are cooled by LOX which is supplied through the pump shaft from a hole in the boost pump inlet bolt. This LOX is discharged into the main pump upper inducer inlet. The bigger diameter bearings used on the turbine end are required for support of the rather large overhung mass of the turbine and the large shaft diameter needed for torque transmission.

The overriding design concern with the HPOTP was the absolute separation of the LOX being pumped from the hydrogen-rich steam that drives the turbine. This separation was accomplished by an elaborate set of shaft seals, drains and purges

SSME High Pressure Oxygen Turbopump



Key Performance Parameters

	RPL (100%)		FPL (109%)	
	MAIN	BOOST	MAIN	BOOST
PUMP INLET FLOWRATE (LB/SEC)	1069	107	1162	126
PUMP INLET PRESS (PSIA)	377	4173	392	4430
PUMP DISCHARGE PR (PSIA)	4291	7280	4580	7940
PUMP EFFICIENCY	0.681	0.803	0.681	0.805
TURBINE FLOWRATE (LB/SEC)	64		65	
TURBINE INLET PR (PSIA)	4965		5630	
TURBINE INLET TEMP (°R)	1495		1625	
TURBINE PRESS RATIO	1.511		1.547	
TURBINE EFFICIENCY	0.733		0.789	
TURBINE SPEED (RPM)	28,350		29,200	
TURBINE HORSEPOWER	23,970		28,200	

Diameter: 15.4 in.
Length: 31.7 in.
Weight: 574 lbs

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Figure 13. High Pressure Oxidizer Turbopump

located between the turbine end bearings and the second-stage turbine wheel (Figure 14). The turbine seal assembly is shown on the right-hand side of Figure 14, next to the second-stage turbine wheel. The turbine seal is comprised of two controlled-gap, self-centering carbon floating ring seals that are used with a shaft-mounted mating ring. The space between the two seals is vented though fourteen flow passages to the primary turbine seal overboard drain line. All of the primary turbine seal leakage is vented through this system to the thrust chamber nozzle exit plane. The primary LOX seal is a three-step, shaft-mounted labyrinth seal used with a stationary plastic (Kel-F) wear ring. A slinger is used at the inlet of the primary LOX seal to isolate it from the turbine end bearing coolant inlet pressure and to convert potential leakage flow from a liquid to a gas. The downstream side of the primary LOX seal is vented through eleven radial flow passages to the primary LOX seal overboard drain line. Additional isolation is provided between the turbine seal assembly and the primary LOX seal by the intermediate seal assembly. The intermediate seal assembly is a pair of controlled-gap, self-centering carbon floating ring seals with a shaft-mounted mating ring. The annular space between the two rings is purged with high pressure helium at a flow rate of 260 standard cubic feet per minute (scfm). The helium flow is split between the two intermediate seals with part of it flowing out the primary LOX seal drain line and part of it out of the intermediate seal drain line (a vent between the intermediate seal and the secondary turbine seal). This system is backed up by the control system by monitoring two pressure measurements for engine shutdown if the predetermined limits (redlines) are exceeded. The system is designed to operate safely with any one of the seals missing and with both redlines at the limit simultaneously.

On March 24, 1977, Test 901-110 on Engine 0003 experienced a major fire [24]. With the instrumentation being used for the Engine 0003 test series, it was possible to conclude that the fire originated in the general area of the HPOTP primary LOX seal drain cavity. Due to the severity of the fire, however, little physical evidence remained (Figure 15). The configuration of the primary LOX seal was significantly different from the eventual flight (and current) configuration. The seal was a bellows-loaded hydrodynamic liftoff seal. Under static conditions, a positive seal was maintained by the load applied to the seal face from the bellows. Under operation, the design of the seal face was such that fluid was forced into small depressions (Rayleigh steps), which provided a hydrodynamic lift and caused the seal to operate in an almost closed but not touching position. Several failures of this seal were hypothesized to have caused the HPOTP fire. They were: loss of hydrodynamic lift, resulting in rubbing and subsequent ignition; failure of the bellows weld, allowing gross LOX leakage; seal instability, causing interference with the shaft axial balance; and seal contamination. Other hypothesized failures were all related to communication between the LOX drain and the hot gas system because of sneak leak paths, manufacturing defects, or unknown high pressure differentials [24].

No clear, concise redesign was evident or forthcoming. It was, therefore, decided to take steps to guard against all of the hypothesized failure modes and resume testing with additional instrumentation directed toward better understanding of the primary LOX seal environment. Low resistance drain lines were installed and the intermediate seal purge flow rate was increased by a factor of ten, from 50 scfm to 500 scfm. (This was later dropped to 260 scfm by reducing the diameter of the intermediate seal [25]). The instrumentation included nine new redlines to provide detection of abnormalities.

HPOTP SEAL GROUP
(ENGINES 2004 AND SUBSEQUENT)

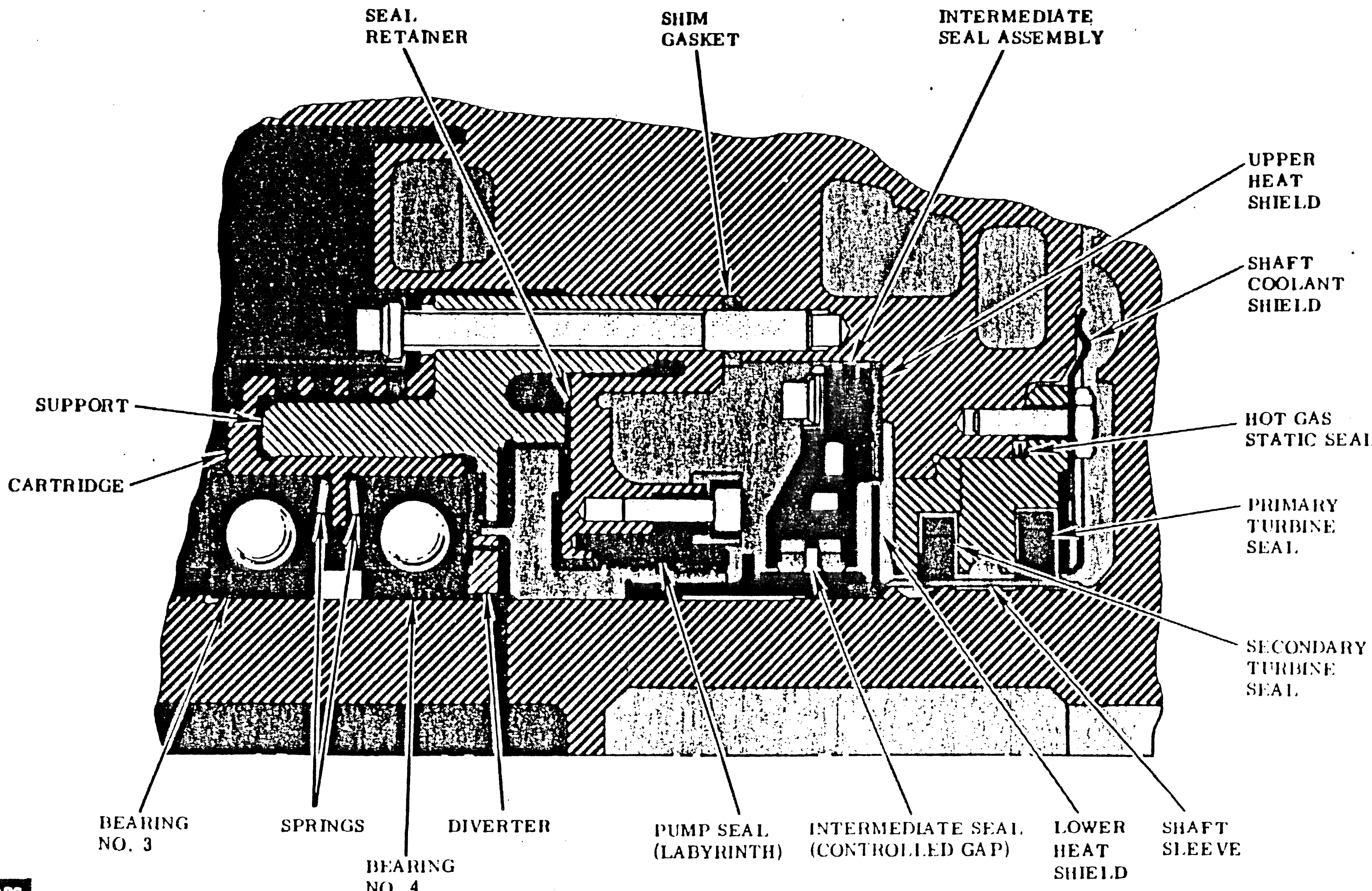

Figure 14. HPOTP Shaft Seals and Drains

High Pressure Oxidizer Pump

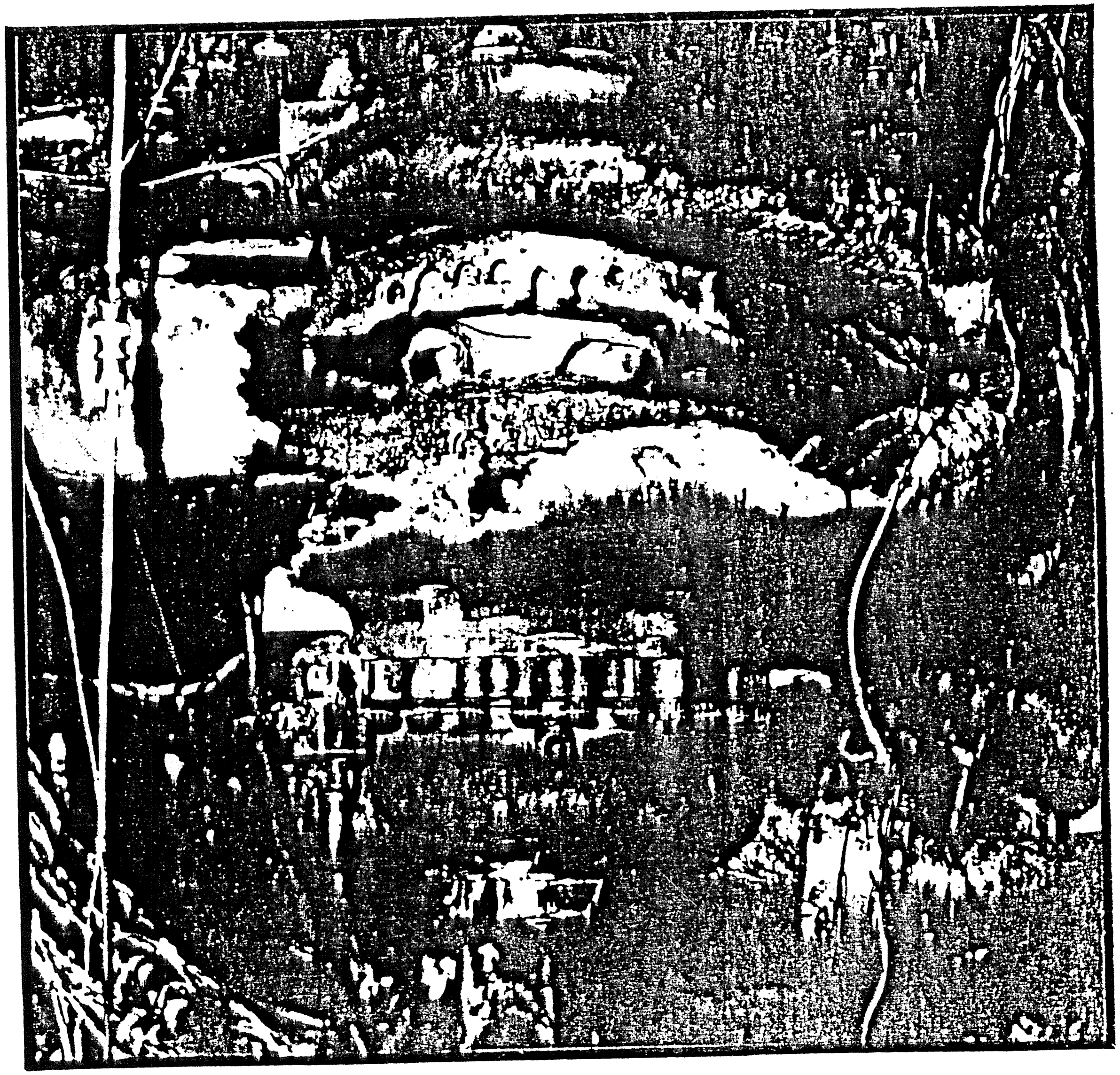

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Figure 15. HP[illegible]P After Test 901-110

On April 27, after one month of deliberation, the engine test program was resumed. Over the next three months, 25 tests were run on two engines with this configuration. A considerable amount of knowledge was gained by this approach, including the realization that the 50 scfm purge did not provide an adequate barrier to prevent co-mingling of the LOX and hot gas drain fluids. Twelve of the 25 tests were safely terminated by one of the new redlines (some real and some erroneous) and several seal failures were observed. The hypothesized failure of the bellows and face rubbing were both experienced; and on one of the tests (901-114), sparks were seen emanating from the primary LOX drain line.

On July 25, 1977, a new design primary LOX seal was introduced on Test 901-124. It was a three-step shaft-mounted labyrinth seal with a stationary plastic (Kel-F) wear ring. This configuration was introduced as an interim measure to be used while the seal was being redesigned, but it proved so successful that it remained as the permanent configuration.

Test 901-136 on Engine 0004 was scheduled for 320 seconds duration on September 8, 1977. The test was prematurely terminated at 300.22 seconds due to loss of engine electrical control, which was caused by a major fire originating in the HPOTP [26]. Through a detailed analysis of the test data, the investigating team was able to trace a series of events leading to the final conflagration that began almost three minutes earlier.

The engine was started to 90 percent power level and then throttled up to RPL. At 133 seconds, the power level was reduced back to 90 percent. Coincident with the power level change, a significant change was observed in the HPOTP vibration characteristics. All accelerometers indicated a slight increase in activity and the three accelerometers on the turbine end began to show a gradual rise in vibration amplitude. This was interpreted as a degradation of the turbine end bearings. At 185 seconds, the turbine end accelerometers had stopped increasing and began decreasing. At the same time, the pump end accelerometers began to increase in amplitude, indicating that radial loads were being transferred from the turbine end bearings to the pump end bearings. At 193 seconds, an increase in the LOX temperature at the main pump discharge indicated internal heat generation. At 200 seconds, the facility-measured LOX flow rate began to deviate from the engine LOX flow rate. This was conjectured to be caused by wearing of the boost pump impeller rear shroud labyrinth seal, resulting in increased recirculation flow from the boost pump back to the main pump inlet. Many measurements showed an increasing turbine power requirement for the rest of the test, indicating a continuing increase in internal friction. At 275 seconds, measurements in the HPOTP drain lines began to show evidence of increased clearances and heat generation in the seals. The condition of the HPOTP continued to degrade until 300 seconds when the rotor attempted to seize up, leading to a failure of the low pressure LOX duct. This caused the HPOTP to cavitate and overspeed, and an internal fire burned through to outside the pump and destroyed control system wiring, which led to a cutoff command.

The failure scenario was fairly conclusive but incomplete. Neither the data nor the remaining hardware contained clues as to what caused the gradual failure of the turbine end bearings. (See Figure 16.) An in-depth analysis was conducted covering

Main Impeller, Preburner Pump Impeller
(Preburner Pump End Bearing Inner Races After Fire)

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Figure 16. HPOTP Rotor After Test 901-136

31 potential failure modes [26]. Each failure mode was evaluated using a fault-tree diagram to describe how it could have caused or contributed to a fire in the HPOTP. Supporting and refuting evidence was obtained from test data, prior experience, hardware inspections, previous pump conditions, analytical models and other sources. The team concluded that the most probable initial failure cause was wearing of a turbine end bearing due to uneven load sharing and inadequate cooling. It was also concluded, however, that it was almost as likely that the first failure was in a pump end bearing. Because of the lack of conclusive evidence, all failure modes that were judged to have sufficient supporting evidence and, more importantly, those with insufficient refutative evidence, were placed in the final potential cause list. Seven basic failure modes were identified, with an average of three root causes each. For each of these, short-term and long-term action was taken to reduce or eliminate the potential for a similar event in the future.

On September 26, 1977, just 18 days after the incident, testing was resumed with Engine 0002 using an HPOTP with enhanced internal instrumentation. Changes were incorporated that would improve bearing coolant, equalize bearing load sharing and reduce bearing loads. Improved dynamic balancing of the rotor resulted in an immediate reduction in vibration levels and synchronous loads. Although additional improvements were to be made later in the program [25], these changes effectively resolved the failure mode of Test 901-136.

Prior to Test 901-136, the HPOTP turbine end vibration was monitored as a redline parameter. Because of the characteristics revealed on this test, the redline was changed to monitor the pump end vibration. The vibration amplitude on the pump end increased by a factor of six and exceeded 20 g rms by the end of the test. With a pump end redline of 12 g rms or less, the test would have been terminated at least 30 seconds earlier. This redline, while not used in flight, has been used throughout the rest of the ground test program with a maximum value varying from 8 to 12.

HPOTP unit number 0301 was assembled with a significant amount of special internal instrumentation, including pressure and temperature measurements to evaluate seal and bearing flows, strain gages and accelerometers to determine bearing loads, and a capacitance device to determine shaft and turbine end bearing movements as well as pump speed. The instrumented turbopump was installed in Engine 0101 in July 1978 for a series of tests designed to increase technical understanding of the internal HPOTP environment. On July 18, the fourth test in this series (Test 901-120) was prematurely terminated at 41.81 seconds by the HPOTP vibration monitor for a vibration level that exceeded the redline. Simultaneously with the cutoff signal, a major fire was apparent in the area of the HPOTP.

Eleven failure modes were postulated and analyzed with supporting and refuting data. The unusual amount of data provided by the instrumented turbopump enabled exclusion of all but two, and both involved failures of the special capacitance device that caused the rotor to repeatedly strike the device and eventually ignite it [27]. Although no design flaw was at fault, one change was made prior to test resumption. The outside diametric clearance for the primary LOX seal inlet slinger was increased from 0.020 inch to 0.040 inch.

Testing was resumed with Engine 0005, Test 901-185, on August 12, 1978. Not all of the 24 days down-time can be attributed to the incident, however, because Engine 0005 was undergoing major modifications to all three injectors in the test stand. The engine had been fitted with a new powerhead on the previous test and had experienced anomalous fluid resistances. All of the injector LOX element orifices in both preburners were enlarged, a one-square-foot rag was removed from the OPB fuel manifold, and the fuel flow was increased on 214 of the 600 main injector coaxial elements. (The rag was removed by using external heat with an internal oxygen purge which caused it to burn away. This was later referred to as the "rag roast"). Except for the increased slinger clearance, the only concession to the incident was a temporary redline placed on the HPOTP main pump discharge temperature. Since the temperature sensor presented a risk in itself, it was removed five tests later.

The last HPOTP explosion occurred on July 30, 1980, on the second test of Engine 0010. The first test (901-183) was a successful 1.5 second priming sequence verification test. Test 901-284 was prematurely terminated by the HPOTP vibration redline at 9.89 seconds when an internal HPOTP fire propagated and caused the failure of the high pressure oxidizer duct [28]. The investigating team concluded that extreme off-design operation of the HPOTP caused the fire and that no HPOTP design flaws were evident [28]. The off-design operation was the direct result of two unrelated failures, not associated with the turbopump, which caused an erroneous measurement of the MCC chamber pressure.

The MCC chamber pressure measurement is quad-redundant. Four independent measurements are taken by utilizing two strain gage bridges in each of two separately mounted sensors. Within the dual-redundant SSME control system, one sensor is assigned to channel A (two measurements) and the other is assigned to channel B. The first failure that occurred in Test 901-284 was loss of the channel B power supply at 3.28 seconds which automatically disqualified all channel B measurements. This loss of redundancy would have been sufficient cause to shut the engine down if it had been on the launch pad. The flight mission rules require full redundancy at liftoff; however, the ground test program at that time allowed test continuation with loss of redundancy.

Each of the chamber pressure sensors is ported to a small cavity which opens into the MCC combustion zone just below the injector. The sense port is purged with hydrogen to prevent accumulation and subsequent freezing of the water produced by the combustion process. A very small hydrogen flow rate is provided from the MCC fuel outlet manifold through a tiny orifice contained in a device (Lee Jet) that was pressed into the manifold and retained with a snap ring. The second failure occurred at 3.92 seconds when the channel A Lee Jet was dislodged, which exposed the chamber pressure sensor to the full fuel manifold pressure. Since the fuel manifold pressure is 65 percent higher than the MCC chamber pressure, the engine control system reacted and throttled the engine down to the desired pressure. This resulted in the engine operating at 60 percent power level with a mixture ratio of 3.5. At such a low mixture ratio, the HPOTP turbine gas temperature stabilized at an average temperature of -30 F, which caused the steam to freeze. A gradual ice buildup in the turbine ultimately caused all axial thrust balance capability to be exceeded, which led to internal rubbing and ignition within the LOX pump.

Because of the MCC design, it was not possible to use the Lee Jet in the preferred installation wherein the pressure differential acts as a holding force rather than a dislodging force. The redesign, however, eliminated the snap ring and incorporated a positive retention feature. As a result of this failure, a permanent minimum redline was established for the HPOTP turbine gas temperature to prevent turbine gas freezing.

High Pressure Fuel Turbopump Turbine Blade Failures

Late in 1977, two failures of HPFTP turbine blades occurred just two weeks apart [25]. On November 17, Test 902-095 on Engine 0002 was cut off prematurely, while operating at 70 percent power level, by the HPOTP vibration redline monitor. The average HPOTP vibration level had increased from 3 g rms to over 70 g rms. However, it was subsequently discovered that the vibration originated in the HPFTP, which violently shook the entire engine. The test was not shut down by the HPFTP vibration monitor because it had a built-in time delay of 0.240 seconds, and the HPOTP redline time delay was only 0.100 seconds. A post-test inspection revealed that a first-stage turbine blade had broken off and inflicted significant damage to both turbine stages. Figure 17 shows the damage to the first-stage wheel. The engine was shut down safely with no other engine damage. Two weeks later, on December 1, 1977, Test 901-147 on Engine 0103 experienced a similar failure at slightly above 80 percent power level. This time the damage was more severe. The turbine blade debris caused the rotor to seize up, resulting in the cessation of fuel flow and very LOX rich operation. Major burning throughout the hot gas system followed; but, although significant damage was sustained, it was contained within the engine with no external burnthrough. From this and other fuel side failures, it was concluded that engine failures on the fuel side would be self-contained and therefore fail-safe in the flight environment where there is engine out capability. It would be four years later (after the first flight) that a similar turbine failure at a higher power level would cause the rupture of the low pressure fuel duct and change the fail-safe conclusion.

The HPFTP turbine (Figure 18) is a two stage reaction turbine powered by hot hydrogen-rich steam produced by the FPB. The turbine drive gas is directed by 41 first-stage nozzle vanes into 63 first-stage turbine blades. The first-stage exhaust is gathered by 39 second-stage nozzle vanes and redirected through 59 second-stage turbine blades. (The number of elements was purposely made different to minimize the possibility of frequency reinforcement among the various parts.) The turbine exhaust gas is then guided by sheet metal structure through a 180 degree turn to be consumed in the MCC. The turbine blades are about one inch long by half an inch wide with a ribbed extension called a "fir tree", which it resembles (Figure 17). The blades are installed by inserting the fir trees into matching slots in the gold plated turbine disc, and they are held in place by pressure loading and centrifugal force. The blades are structurally independent of each other except for small spring-like devices that fit between the blade shanks in slots provided at the blade platforms. These are called dampers and are used to dampen blade vibration. Because the power generated by the HPFTP amounts to about 600 horsepower per blade, the blades are under tremendous stress. The power bending stress and the centrifugal stress are each approximately 50,000 psi for every blade. The blades are uncooled and operate at a temperature of about 2,000 R. The turbine blade material for this very severe environment was chosen based on tests conducted in 1971. The blades are

Blade Failure

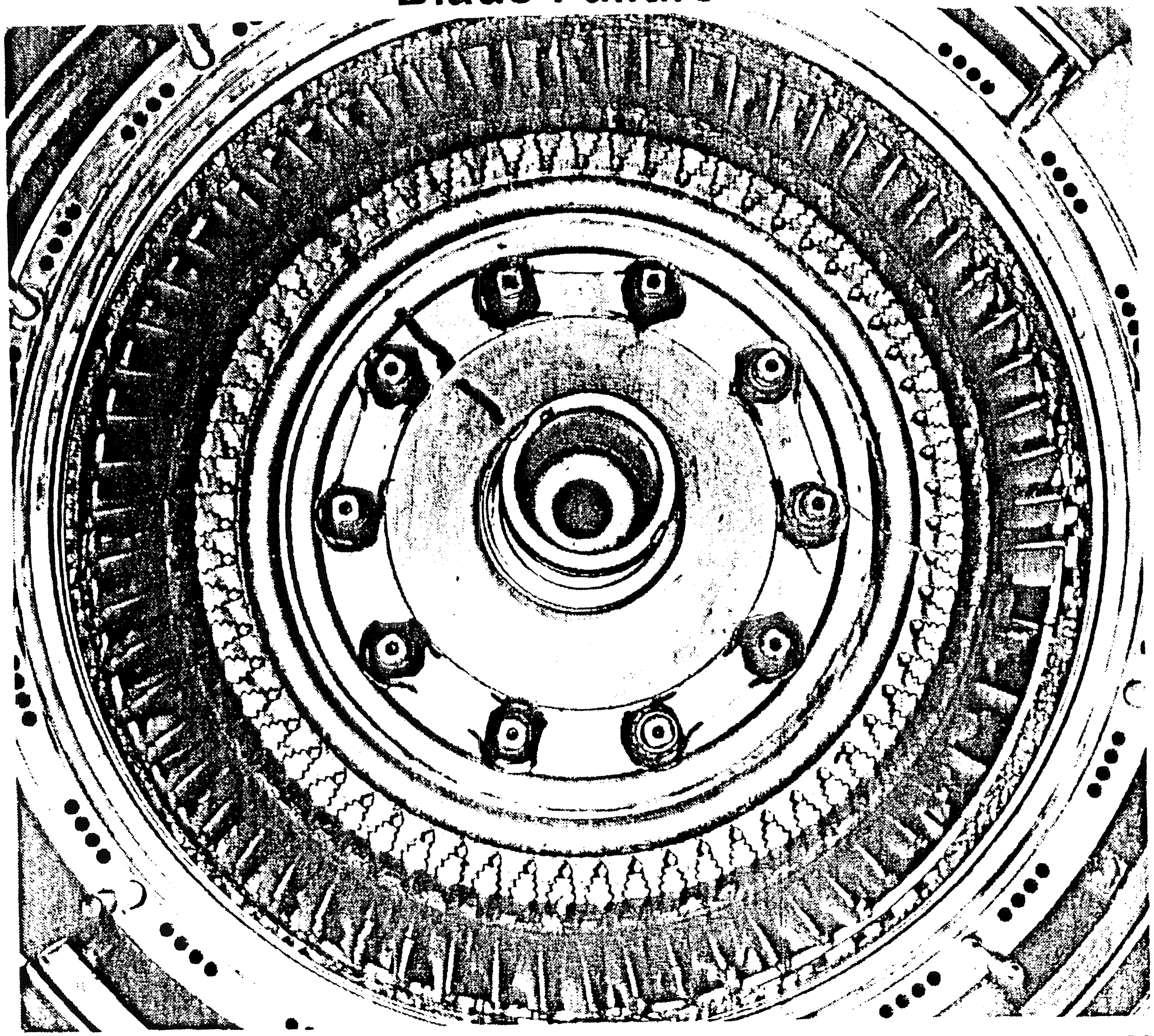

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Figure 17. HPFTP First Stage Turbine - Test 902-095

HPFTP TURBINE SECTION

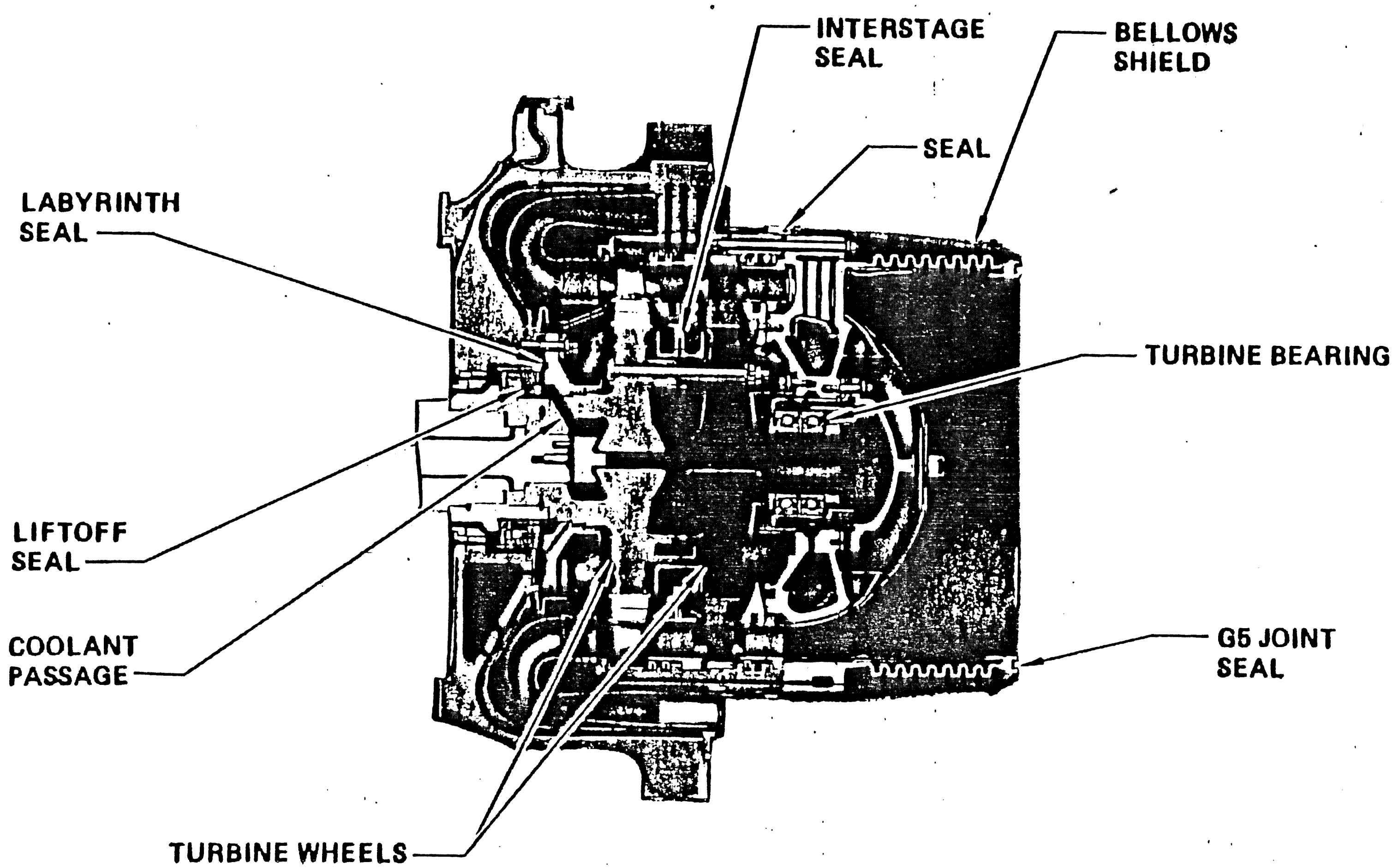

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Figure 18. HPFTP Turbine Section

directionally solidified castings of a nickel-based super alloy known as MAR-M-246 (developed by Martin Metals).

With the hardware evidence, data analysis and other blade samples, it was shown that the failures were initiated by a high cycle fatigue crack in the air foil of a first-stage turbine blade, close to the root. Consultation with government, industry and academic experts led to a comprehensive laboratory test program at Rocketdyne, General Electric, TRW and AiResearch [25]. Tests were conducted to evaluate blade vibration; static loads; high-temperature, high-cycle fatigue; blade platform loading and damper performance. At the same time, material property tests were run to determine thermal shock effects, heat effect on microstructure and characteristics of fatigue and stress rupture failures.

A centrifugal stress and dynamic response evaluation machine known as the Whirligig was activated at Rocketdyne to test the turbine wheels with blades up to 38,000 RPM while measuring the blade stress with strain gages. Damping rating tests were conducted with various damping designs while using high-pressure gas jets to pulse the blades over a range of frequencies. From these tests, it was discovered that the turbine blade fatigue was caused by locked-up blades and that all damper configurations tested were effective in reducing the effect of engine-induced vibration modes that could contribute to blade failure. An optimized, lightweight, precision-tolerance damper design was selected after a review of the Whirligig test data (Figure 19). The new damper design was incorporated along with changes to preclude blade lock-up either in the wheel or blade-to-blade. A looseness verification was added to the assembly procedures.

Although turbine blades would receive high priority attention for many more years, the specific failure modes associated with Tests 902-095 and 901-147 were eliminated by these modifications. [29]

Main Oxidizer Valve Fire

Engine 2001, Test 901-225, was scheduled for a 520 second flight mission simulation test on December 27, 1978. The test ended with a major fire when the HPOTP discharge duct ruptured at 255.63 seconds. Abnormal operation was apparent in the data for about 0.120 seconds before the failure, showing an rapid increase in LOX side power which culminated in exceeding the redline value for the HPFTP turbine temperature. The failure progressed too rapidly for the redline to provide protection and the high pressure LOX duct ruptured simultaneously with the engine shutdown command. The resulting fire caused sufficient damage to the engine control system, such that an engine-controlled shutdown was not possible; and propellant flow was ultimately terminated by closing the facility prevalves. The investigating team concluded that a fire had started in the MOV and created a flow blockage downstream of the HPOTP main pump discharge. Since the boost pump supply line is upstream of the MOV, the blockage caused a significant diversion of LOX to the boost pump and subsequently to the two preburners. This, in turn, created a drastic increase in power to both high pressure turbines, which led to the overpressurization of the main oxidizer duct in just over one-tenth of a second [30].

HPFTP First Stage Blade Precision Damper

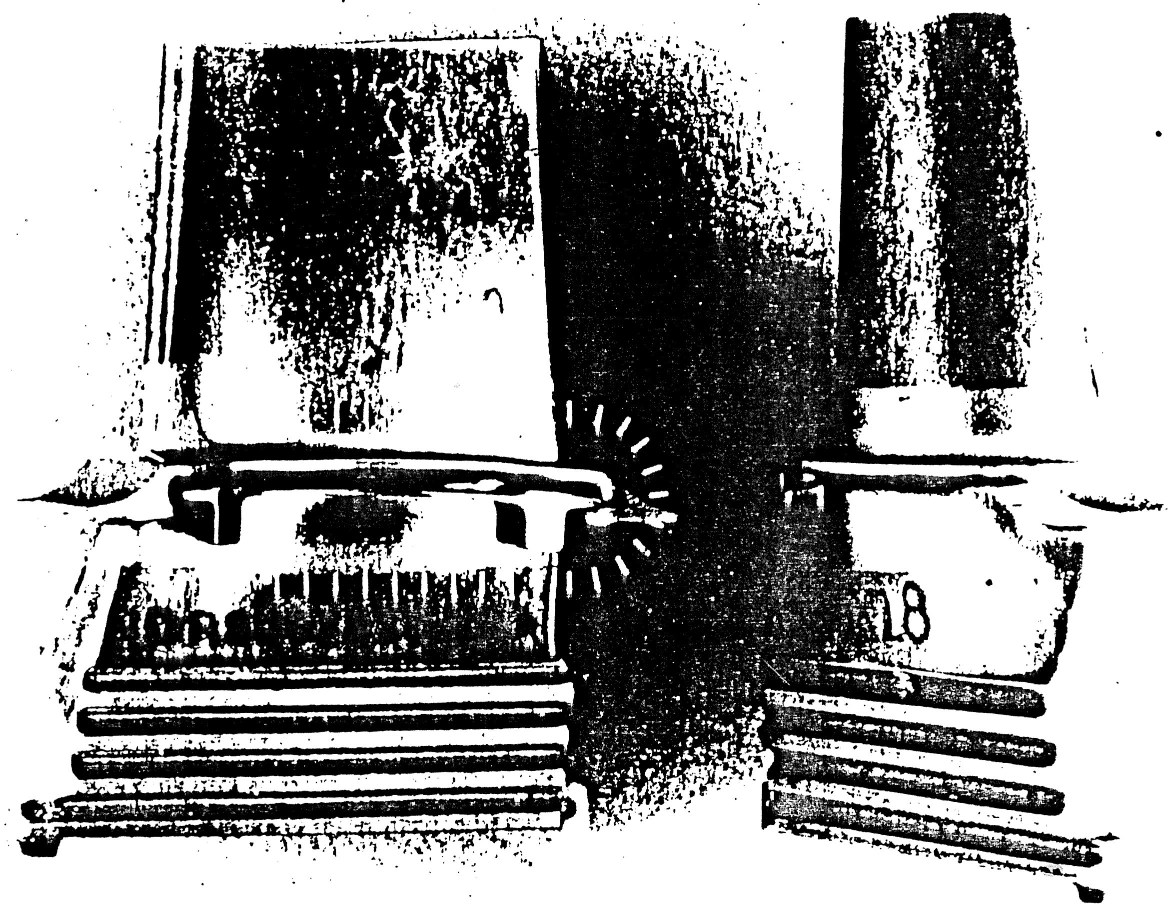

Figure 19. First Stage Blade Precision Damper

The MOV is shown in a cutaway representation in Figure 20. It is a ball valve fabricated with an integral ball and shaft. The valve is rotated by a hydraulic actuator (not shown) spline-coupled to the shaft at the top of the valve assembly. The hollow ball is approximately 5 inches in diameter with a 2.5 inch tubular flow passage. At the FPL flow rate of 850 pounds per second, the fluid velocity through the valve exceeds 350 feet per second. The ball seal is a machined plastic ring which is loaded against the inlet side of the ball by a bellows. The shaft has two integrally machined cams which push against a cam follower mechanism which lifts the seal from the ball during the first six degrees of valve rotation. The LOX flow for the MCC ASI is supplied from a port immediately downstream of the seal and begins to flow as soon as the seal is lifted. To minimize turbulence, a 2.5 inch diameter inlet sleeve is bolted to the valve inlet, isolating the bellows from the flow environment and aligning the flow stream with the ball tubular flow passage in the open position. The sleeve was positioned to a fixed gap between the ball and sleeve by using a stack of 0.002 inch thick stainless steel shims under the bolted inlet flange.

Disassembly of the MOV from the incident test revealed that the bellows had been burned away and the inlet sleeve was 50 percent consumed. (See Figure 21.) A dynamic analysis using two-dimensional and three-dimensional models showed that the inlet sleeve had a fundamental natural frequency of about 1,900 Hz, which corresponded to a high energy vibration generated by the HPOTP (four times pump speed). This was augmented by data obtained from a simulated LOX feed system test facility set up by NASA at MSFC to study flow under controlled laboratory test conditions [31]. It was shown that the energy generated by the HPOTP was enough to excite the sleeve frequency and loosen the preload on the bolts attaching the sleeve to the valve inlet flange. The team concluded that excessive vibration in this loosened condition caused the very thin steel shims to ignite, causing the fire. Disassembly of valves from prior engines confirmed this potential by showing evidence of fretting and broken shims.

Several design changes were incorporated, the most important being replacement of the shims with a single machined spacer having a minimum 0.040 inch thickness. To change the natural frequency of the sleeve assembly, the material was changed from 21-6-9 CRES to INCO 718 and the wall thickness was increased by about 75 percent. To minimize the potential for fretting, the sleeve diameter was increased to be an interference fit; the spacer material was changed from 302 CRES to INCO 718; and the attaching screws were countersunk with cup washers to provide positive locking.

These changes were made to the OPOV and the FPOV as well as the MOV. Testing was resumed utilizing the new LOX valves with Engine 0201 on January 30, 1979, after a total downtime of 33 days.

Main Fuel Valve Fracture

In April 1978, at the NASA NSTL test site, a test series was initiated on a simulated Space Shuttle orbiter aft section, including a cluster of three main engines. This combination of hardware was known as the Main Propulsion Test Article (MPTA) and was used by NASA to conduct full-up system tests of the entire orbiter propulsion system. On July 2, 1979, MPTA Test SF06-01 was scheduled for a 520 second flight

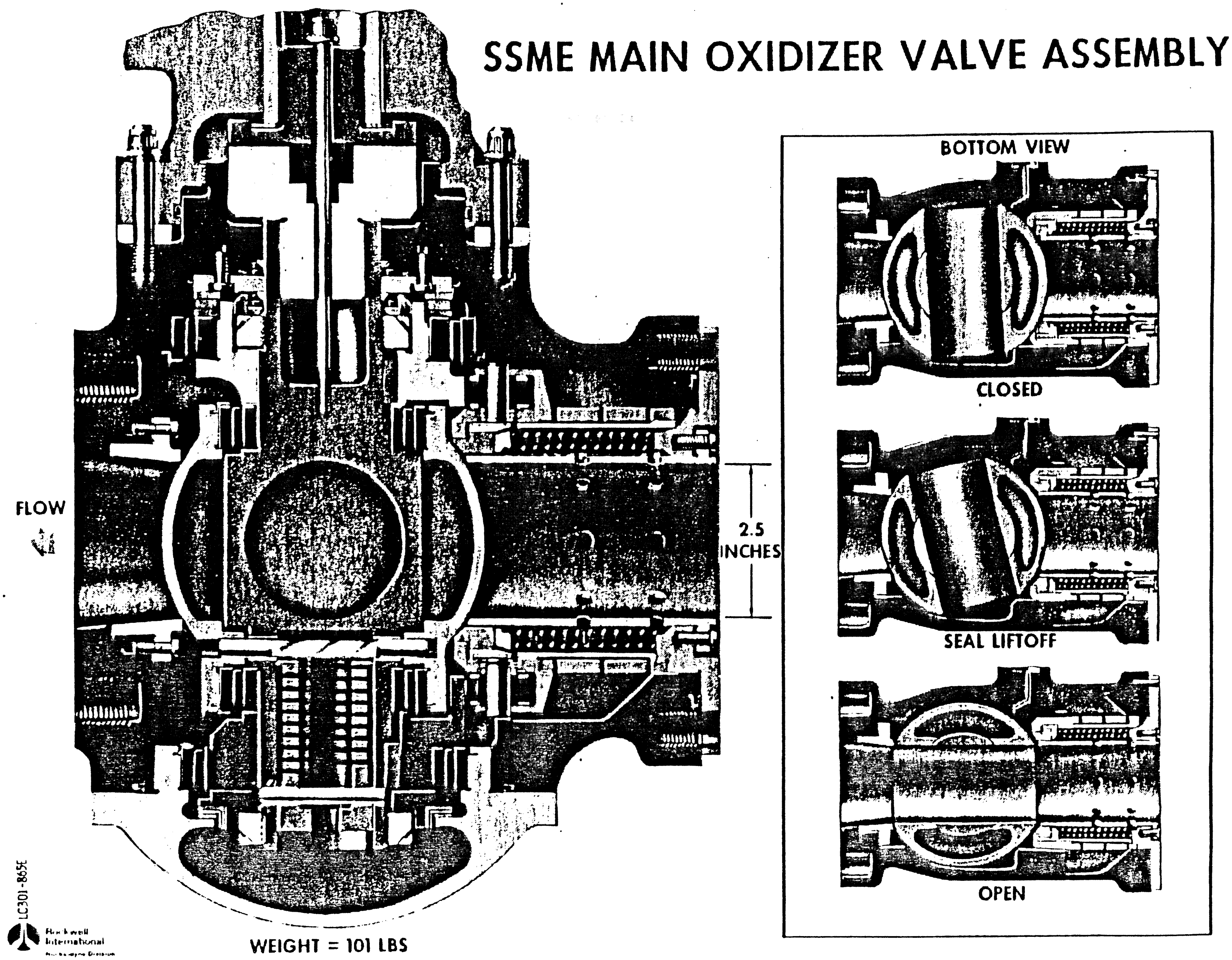

Figure 20. Main Oxidizer Valve

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Figure 21. MOV[illegible]t Sleeve After Test 901-225

mission simulation test. At about 18 seconds, the MFV housing on Engine 2002 developed a major fracture which allowed hydrogen to leak into the enclosed aft compartment. The loss of fuel in the engine caused both turbine temperatures to increase; and the HPFTP turbine temperature exceeded its redline value, which caused engine shutdown to be commanded for all three engines. During this time the pressure in the aft compartment increased as a result of vaporizing the hydrogen. Almost at the same time as the shutdown command, the aft compartment pressure reached 3.2 psi [32], which exceeded the structural capability of the aft compartment heat shield supports. Figure 22 shows how the heat shields around each engine were blown off. Major structural damage was sustained in the aft section of the MPTA. An external fire ensued which caused minor damage, mostly to instrumentation wiring; but there was no fire damage inside the aft compartment.

The MFV is similar in construction to the MOV (Figure 23). It is the same size as the MOV, but it operates at 50 percent higher pressure, 100 R colder temperature, and three times the fluid velocity. Three major differences between the valves are due to the higher pressure and colder temperature. The forged MFV housing is made from an alloy of titanium instead of INCO 718 in order to withstand the high pressure with a minimum weight valve. Because titanium has a significant gain in strength at very low temperature, the MFV was turned around with the bellows-loaded seal at the valve outlet rather than the valve inlet. This orientation allows the valve to be thoroughly chilled before engine start, thereby assuring maximum strength prior to being subjected to operating pressure. The third difference is insulation. The LH2 temperature is below 40 R, which is well below the boiling point of liquid nitrogen. To preclude formation of liquid nitrogen on the valve surface, the housing is covered with an insulator.

Because the incident involved a fracture in the titanium housing, the investigating team was expanded to include fracture specialists and experts in the characteristics of titanium. Nine consultants were employed from nine different research centers around the country [33]. The investigators conducted a thorough visual inspection of the fracture surface using magnifying glass, light microscope and scanning electron microscope (SEM). Most of the fracture surface exhibited the characteristics of a simple ductile-type failure due to overload. The SEM analysis clearly indicated the fracture origin to be at the location of a cutout in the housing designed for the ball seal retraction cam follower (see Figure 23). There was no evidence of material or forging defects, and the microstructure was normal for this titanium alloy. The material composition was verified to be correct by chemical analysis. Proper mechanical properties were verified by testing forging samples which were made as extensions on the housing in the original forging and then using sample bars machined from the fractured housing. Although SEM analysis identified areas in the originating fracture surface that indicated propagation by fatigue, other features identified by the SEM were indistinguishable from stress corrosion, hydrogen embrittlement, or low amplitude fatigue. Fractography was unable to identify the exact failure mechanism [33].

Fifteen hypothetical failure modes were investigated in detail. Extensive testing and analysis at Rocketdyne, MSFC and the various research centers managed to disprove 11 of them; however, the available evidence was insufficient to narrow the failure cause down further than the remaining four hypothetical failure modes. Therefore, positive action was taken to eliminate all of them for future valves. Aside from process

Overall View of Orbiter Heat Shield Damage on Engine Positions 1 & 2

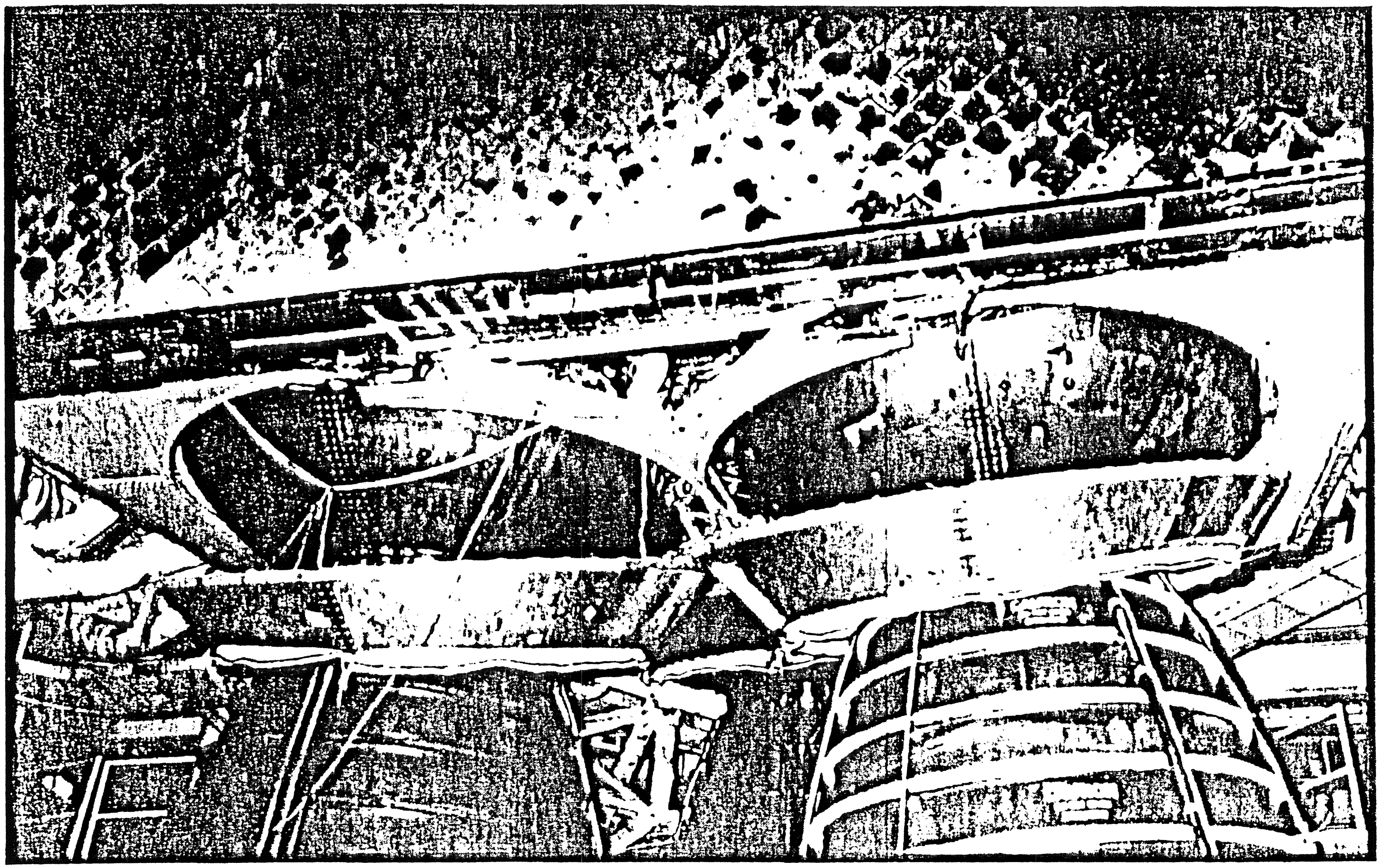

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Figure 22. MPTA Heat Shield Failure

SSME MAIN FUEL VALVE ASSEMBLY

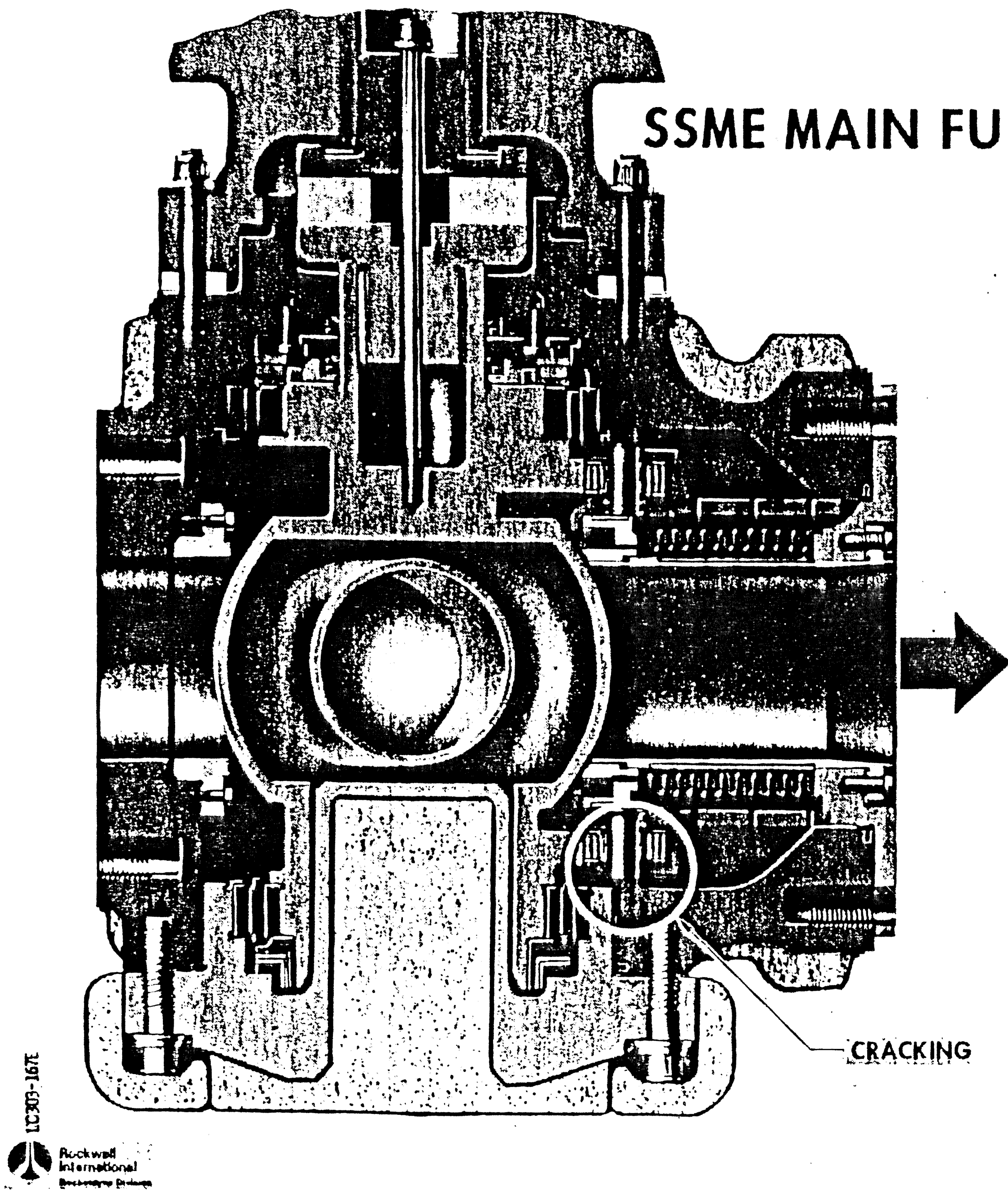

Figure 23. Main Fuel Valve

changes, the cam follower cutout area was reworked on all existing housings to provide generous radii for up to 30 percent reduction in stress concentration. In addition, for long-term recurrence control, the valve housing was redesigned to reduce the peak strains to 80 percent of yield.

Engine start transient testing was resumed on July 6, 1979 with Engine 0007; and mainstage testing began on July 12, 1979, ten days after the incident, with Engine 2007.

Nozzle Feed Line Failures

Engine 0201 Test 750-041 was scheduled for 100 seconds on May 14, 1979, at the Rocketdyne Santa Susana Field Laboratory (SSFL). The test was terminated at 4.27 seconds by the redline monitor for the HPFTP turbine gas temperature for a reason not associated with the major incident which followed. During the shutdown transient, when the MCC combustion chamber pressure dropped a little over 50 percent, a nozzle fuel coolant feed line ruptured close to the nozzle exit in a section of 1.625 inch diameter tubing known as the "steerhorn", which it resembles [34]. The massive fuel leak caused the engine to operate LOX rich, which caused significant hardware burning in both preburners, both high pressure turbines, the main injector, MCC and the nozzle. The external hydrogen fire caused damage to the engine control system and the facility instrumentation.

The nozzle is a regeneratively cooled extension bolted to the MCC which completes the combustion gas expansion from a 5 to 1 expansion ratio to a 77.5 to 1 expansion ratio. It has the contour of an optimized 80.6 percent bell nozzle (80.6 percent of equivalent 15 degree cone length) to minimize overall engine length. Figure 24 shows a cutaway representation of the nozzle with the configuration that was used through the first five flights. The nozzle is ten feet long, almost eight feet wide at the exit and weighs about 1,000 pounds. The tubular construction consists of 1,080 stainless steel (A286) tubes brazed together and to a surrounding structural jacket with coolant manifolds welded to each end of the tube assembly. The MFV is mounted on the nozzle inlet manifold which receives all of the MFV fuel flow. Immediately, 20 percent of the LH2 is rerouted to the MCC coolant circuit. One-half of the remaining LH2 is routed through the CCV (see Figure 3) to the two preburners. The remaining 40 percent of the fuel is distributed equally among three fuel transfer ducts (downcomers) for delivery to the nozzle aft manifold. Less than a foot from the end of the nozzle, the downcomers terminate in a tee fitting, which splits the flow into two tubes, each perpendicular to the downcomer. Each tube is then routed though ninety degree turns to enter the aft manifold at one of six equally spaced inlets around the circumference, creating three inverted "steerhorns". The LH2 flows from the aft manifold into each of the 1,080 tubes and provides nozzle wall cooling while flowing forward to the outlet manifold. After exiting the outlet manifold, the fuel is mixed with the outlet flow of the CCV prior to combustion in the preburners.

The failure occurred in the number one steerhorn, which is directly beneath the MFV. The entire right-hand section of tubing had broken off from the nozzle between the welds at the tee fitting and the aft manifold. Most of the tube was recovered as a single piece; however, extensive fragmentation had taken place at both ends, and

Nozzle Assembly

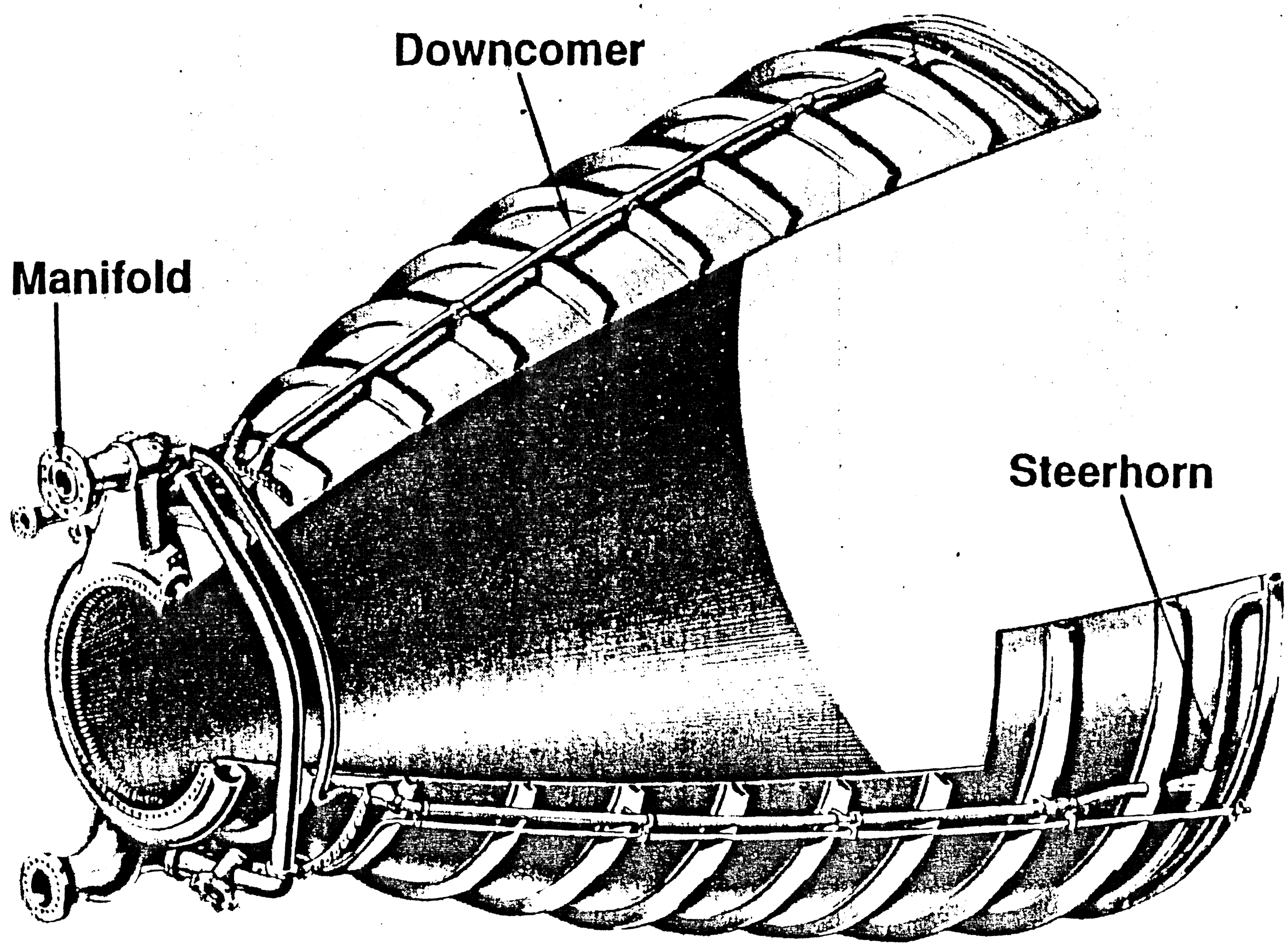

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Figure 24. SSME Flight Nozzle

some of the fragments were not found. Metallurgical analysis of the recovered pieces disclosed that most of the fracture surfaces were indicative of a simple ductile failure due to overload. It was concluded that the fracture at the tee end of the tube started at a low cycle fatigue crack 0.003 inch deep by 0.75 inch long located in the heat affected zone adjacent to the tee weld. Fracture initiation at the manifold end was uncertain because not all of the fragments were recovered, although no fatigue indications were evident in the fragments that were found. No material deficiencies or fabrication defects were found in any of the recovered pieces.

The nozzle had previously been subjected to 45 tests on other test facilities; however, this was the first test on the SSFL A-3 test stand with a full size flight nozzle. All the prior tests on A-3 had been conducted with a stub nozzle like the one used on the ISTB. Because of concern that some test facility interaction contributed to the failure, two investigating teams were formed [35]. One team concentrated on facility effects while the other investigated the nozzle structural capability and engine dynamics for potential causes of the fatigue failure. The facility was eventually cleared.

The nozzle structural analysis with the predicted thermal, pressure, and dynamic environments indicated a safety factor at ultimate of greater than two for the transient loads and predicted that no fatigue damage would be incurred from the start, operation and shutdown cycle (infinite life). The failure occurred during shutdown at a chamber pressure that corresponds to the maximum nozzle deflections due to internal asymmetric jet separation, the phenomenon known as side loads. A two-inch nozzle diametric ovalation was observed in the Test 750-041 motion picture coverage, but this was no more than that which was expected and which the design allowed. A laboratory test was performed to measure the actual strains with this amount of displacement, and it verified low stresses (45,000 psi or less versus the material capability of over 180,000 psi). With the material properties and the structural analysis clearly in conflict with the failed hardware, it had to be concluded that the predicted design loads were in error because the operating environment was not properly defined. To define this environment the engine test program was resumed eight days after the incident on Engine 2004, with special instrumentation on the nozzle to measure steerhorn strains and vibration.

Over the next six months, strain gage and accelerometer data were gathered from 41 tests on nine engines. From these data it was discovered that at the side-load conditions corresponding to the failure, high amplitude strains existed at the tee in the 200 to 400 Hz regime. This was determined to be a shock pulse with a few cycles of high amplitude strain, significantly higher than predicted levels, that could explain a low cycle fatigue failure [36]. Although there was considerable variation in the peak strain from test to test, the maximum recorded peak-to-peak strain was almost 20,000 microinches per inch. This was not high enough to cause failure in a single test, but it was high enough to sustain fatigue damage that would lead to a failure in a predictable number of tests. A fatigue damage model was developed from the strain gage data that predicted a fatigue life of 48 tests for the nozzle that had failed in 46 tests.

The failed steerhorn had a 0.049 inch wall thickness. A previous producibility design change had increased this dimension to 0.080 inch, and nozzles with the thicker wall had already started engine test and were committed for the flight configuration. The fatigue damage model predicted a life of 80 tests for the thick-wall steerhorn. It was judged acceptable to continue testing both the thin-wall and the thick-wall steerhorn configurations by establishing a life limit at which they would be removed from service. The life limits were set based on the fatigue damage model, using a factor of two for single engine tests and a factor of four for flight and MPTA tests. A redesign of the nozzle feed lines was undertaken which reduced the peak stresses in the steerhorn by at least 40 percent. The major contributor to the stress reduction was the inclusion of a "steam loop" in the downcomers to absorb longitudinal thermal contraction. Because of the required lead time for this type of change, the steam loop nozzle was not incorporated into the flight program until the sixth shuttle flight.

MPTA Test SF6-003 was scheduled for 510 seconds duration on November 4, 1979. The test was prematurely terminated at 9.7 seconds when an HPOTP seal redline was exceeded on the number three engine (0006), and all three engines were shut down. During the shutdown, the number one engine (2002) experienced a steerhorn failure which caused major internal engine damage and significant damage to the MPTA instrumentation systems. The previous investigating team was reconvened with a situation that seemed to contradict their previous findings. The nozzle on Engine 2002 had thin-wall steerhorns with a calculated life of 48 tests, yet it failed on the eighth test. In addition, the nozzle was instrumented with strain gages and the data showed that the maximum recorded strain magnitude was not enough to cause failure [37].

Metallurgical examination of the fracture surfaces showed a dimpled texture typical of a tensile overload, with no indications of fatigue striations. A microhardness survey of the fracture surface revealed that the welds at the tee were much softer than they should have been. Further examination, using an electron microprobe X-ray analyzer, indicated that the material was probably Inconel 62 rather than Inconel 718, as required. This finding was very significant because Inconel 62 has only one-half the strength of Inconel 718. An inspection method was developed using an electrolytic oxalic acid etch that would allow instant recognition of this material in other welds. Etching all the welds on the failed nozzle disclosed that eight of them were soft welds. Extending this inspection to all other nozzles showed that most of them had some soft welds. A survey of weld filler wire at Rocketdyne found two lots of wire with the wrong material, and they were both from the same vendor [38]. A review of 16,360 welds on 349 different parts revealed that 3,359 welds had been made using the filler wire from the suspect lots. Each of these was analyzed for possible corrective action. All weld wire (7,000 pounds) was removed from service and reverified prior to use. Stringent controls were put into place at Rocketdyne and then extended to 31 vendors.

Two actions were taken with the flight nozzles as a result of the soft weld wire incident. First, the remaining nozzles with thin-wall steerhorns were removed from service. Secondly, all nozzles with soft welds at the steerhorn tee were reinforced by electrodeposited nickel plating over the welds. This increased the strength of the weld by a factor of three.

Fuel Preburner Burn Through

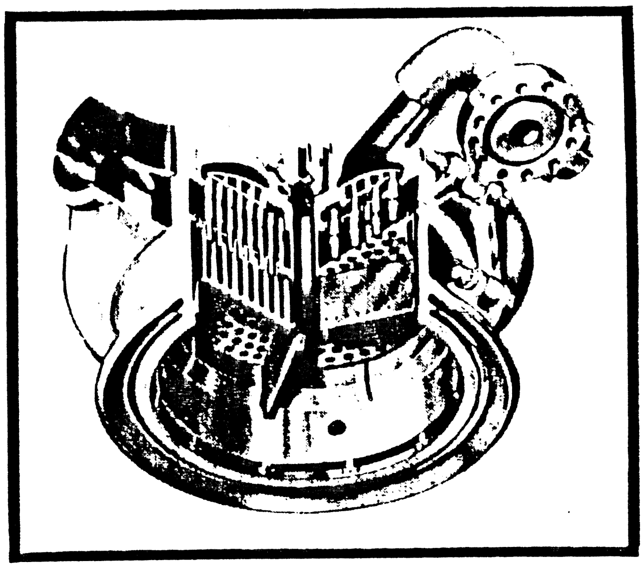
On August 27, 1977, Engine 0004 Test 901-133 was prematurely terminated by the primary observer because of an external fire. Hardware inspections after the test revealed that a hole had been burned completely through the FPB liner and outer body, allowing significant leakage of the hot gas to the outside. A simple design change was made that eliminated the specific failure mode; however, almost three years later, another failure occurred with the same result. MPTA Test SF10-01, on July 12, 1980, experienced a major fire in the aft compartment. It was determined that Engine 0006 FPB had a hole burned through the FPB liner and outer wall very similar to the incident on Engine 0004.

The FPB (Figure 25) provides the turbine drive gas for the HPFTP. It consists of two propellant manifolds, a centrally located ASI, an injector and a short combustor section welded together into the major hot gas manifold that also includes the oxidizer preburner, the high pressure turbine exhaust gas flow path and the injector for the main combustor. The FPB injector is a coaxial element injector with each element having low velocity LOX flowing in a center post, with high velocity gaseous hydrogen in a surrounding annulus to promote uniform mixing. Combustion takes place below the INCO 625 faceplate in three compartments which are separated by 2.25-inch-long copper alloy baffles to prevent tangential modes of instability. The baffles and the faceplate are cooled by hydrogen flowing through drilled holes into the combustion chamber. The outer structural body of the combustor is cooled by the use of a concentric cylindrical liner with hydrogen flowing between the liner and the combustor wall. This same hydrogen is used to cool the HPFTP turbine bellows by the utilization of a welded-in liner extension.

The failure that occurred on Engine 0004 in 1977 happened during a test series wherein the problem of generalized overheating of the FPB body was already being studied by the use of externally mounted thermocouples. The through hole was in line with a LOX post at the outer row corner of a baffle compartment. It had started by eroding through a small dead-end compartment called an acoustic cavity, which was attached to the inside of the liner. It then progressed though the liner and finally burned through the half-inch thick outer combustor wall. It was concluded that the burning was caused by a localized recirculation of LOX from the corner element, causing burning of the nearby acoustic cavity, which acted as fuel to propagate the burning. Two design changes were adopted immediately. The acoustic cavities were eliminated, and all six of the outer row baffle corner LOX posts were deactivated (plugged). The hole in the preburner was repaired by welding and testing was resumed five days after the incident.

The second failure, in 1980, was located six elements away from a baffle (Figure 26) and was determined to be caused by a different mechanism [39]. Inspection of the preburner elements showed no evidence of contamination which could have caused fuel blockage; however, it was discovered that the individual element LOX posts were not concentric with the fuel annuli, causing a fuel restriction on the outboard side of the outer row elements. Further inspection showed that the lack of concentricity was caused by a deformity of the faceplate in which it was bowed outward almost a tenth of an inch, halfway between the center and the outer row. The investigating team [39]

Fuel Preburner



Geometry

- Internal Diameter — 10.43 in.
- Combustor Length — 4.37 in.
- INCO 625 Faceplate
- Injector Configuration — concentric orifice
- Number of Elements — 264
- Baffle Length — 2.25 in.
  Material — NARloy-A

Operating Parameters (FPL, MR-6.0)

- Chamber Pressure — 5572 psia
- Combustion Temperature — 1918 R
- Hot Gas Mixture ratio (O/F) — 0.99
- Oxidizer Flowrate (including igniter) — 85.27 lb/sec
- Number of elements — 264
- Fuel Flowrate (including igniter) — 86.15 lb/sec

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Figure 25. Fuel Preburner

Engine 0006 Fuel Preburner

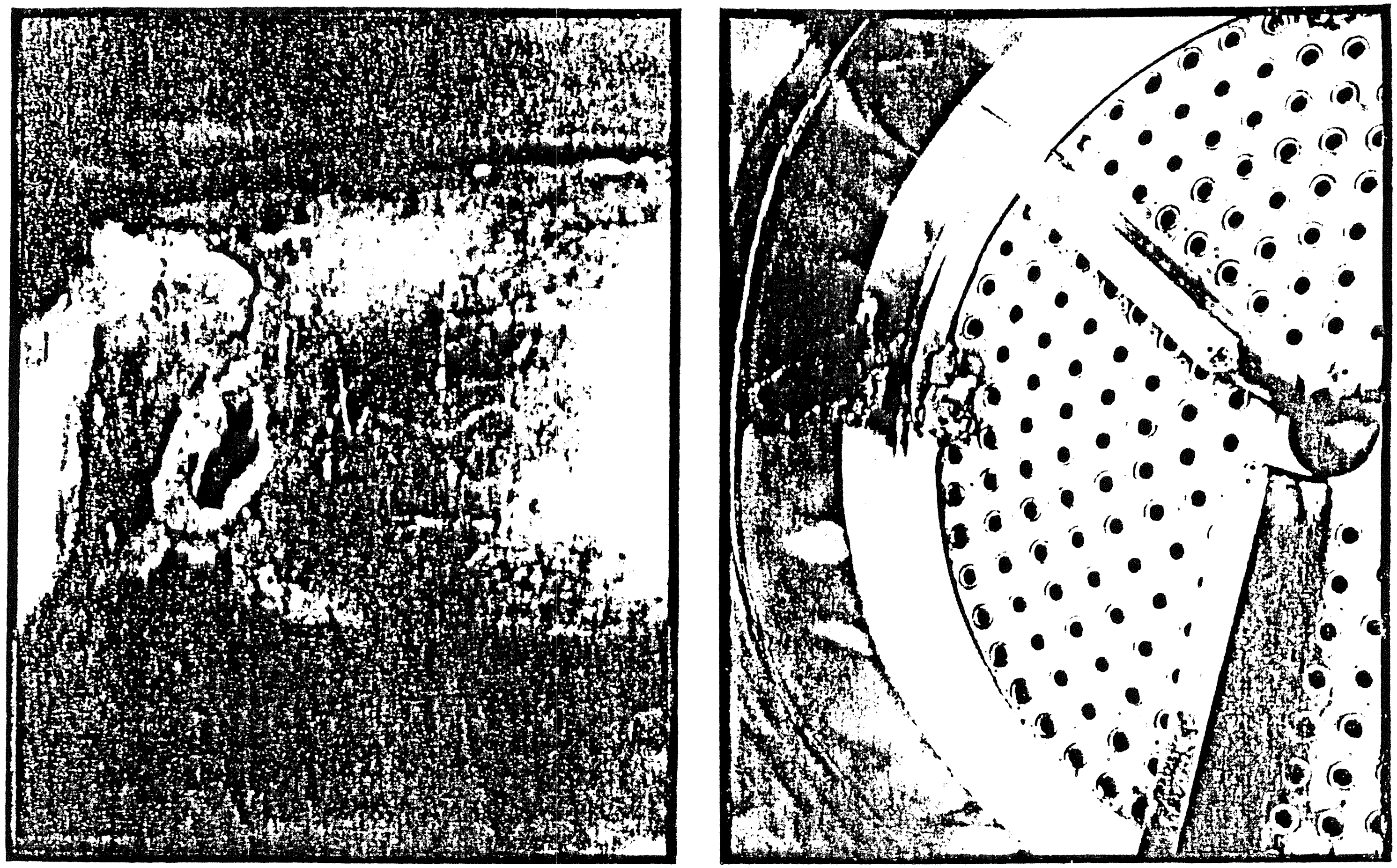

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Figure 26. Engi[illegible]006 Fuel Preburner

concluded from inspection of all other preburners that the bowed condition was unique to the failed unit. In addition, a review of historical problem reports disclosed that this FPB had experienced more reported cases of overheating or minor erosion than all other preburners combined. The cause of the deformity was never identified; however, periodic inspections were added for all preburners to verify outer row element concentricity in the future.

Even though the failure was caused by a unique hardware condition, a major effort was undertaken to preclude additional occurrences of this type of problem by making the preburner insensitive to maldistribution of the propellants. A two-dimensional, four times scale, water-flow model was constructed and tested to evaluate propellant streamlines in the outer row and along the liner wall. Two flow paths were discovered which could contribute to burning if the localized gases were at a higher mixture ratio. A recirculation field existed along the liner wall for about three inches (twelve inches on the water table) which disrupted the local boundary layer and increased the potential heat transfer. The flow path was found to be caused by the existence of an empty space along the faceplate between the element and the liner. The recirculation was eliminated by the incorporation of a new liner which added a divergent section on the inside of the liner to occupy the empty space. To further reduce the potential heat transfer to the liner, the new liner was coated with zirconium oxide. Another flow path was found which would cause hot gas to flow into the coolant circuit between the liner and the FPB body. With a small axial gap between the liner and the faceplate, the fluid at the faceplate would be forced by an adverse pressure gradient to flow through the gap into the coolant circuit. The new liner included coolant flow control orifices which assure that the coolant flow pressure is always higher than the hot gas pressure. This favorable pressure gradient prevents flow through any gap and also acts as an erosion inhibitor by automatically cooling any small hole in the liner with additional hydrogen.

Tests with purposely misdirected elements were conducted to verify the effectiveness of these design changes in preventing liner erosion. It has also been shown that even if a hole were placed in the liner, hot gas would not flow into the coolant circuit. Even so, a "belt and suspenders" approach was taken with this problem by adding a thermal barrier on the inside of the FPB body which would withstand the previously experienced failure conditions for a long enough time to complete a flight mission without failing. A ceramic thermal shield made of molybdenum and coated with disilicide was designed to be bonded to the FPB body.

THE GOALS

The DVS program (see THE REQUIREMENTS) was planned to verify all design requirements in a logical fashion, using certain key task completions as benchmark control points or gates constraining the continuation of the program for some of the more critical activities [12]. Superimposed on that program were other significant milestones which were established by various NASA and other government agencies as an aid in tracking the general health of the SSME and shuttle program. These generally fell into one of three categories: design reviews, test progress and formal demonstrations.

A preliminary design review (PDR) was conducted by NASA with each of the three competing contractors during the SSME contract competition phase in 1970. Immediately after the establishment of a definitive contract, the official PDR was conducted with the Rocketdyne design. The major emphasis in this review involved technical concerns and issues that were too sensitive to discuss during the competition. The purpose of the PDR was to establish confidence in the design concepts and agree to further actions to pursue in areas of insufficient confidence. The PDR was concluded in September 1972 with the agreed-to actions and schedules. The next major step, a critical design review (CDR) was scheduled to be conducted in the first quarter of 1976; however, during the program realignment in 1974, the CDR was rescheduled and was actually completed in September 1976.

The primary purpose of the CDR was to demonstrate that the design was sufficiently mature to allow fabrication of the first deliverable flight engines to commence. The review was organized by MSFC under four separate teams headed up by MSFC task managers Carlyle Smith, John McCarty, Walt Mitchell and Zack Thompson. Weaknesses, questions, required design changes and other requested actions were documented as review item dispositions (RIDs) and approved by the team leaders to be dispositioned by the CDR board. A pre-board review was held with Jerry Thomson, MSFC chief engineer for SSME, as pre-board chairman. Recommendations by the pre-board were dispositioned by the full CDR board which was chaired by Bob Thompson, SSME project manager. In all, 105 RIDs were dispositioned with 86 of them requiring additional action to be performed by Rocketdyne. At the conclusion of the CDR on September 27, 1976, the SSME design was baselined so that any future design changes would be subject to formal configuration control. Fabrication of the first set of flight engines was allowed to proceed.

Early in 1979, NASA conducted an Orbital Flight Test Design Certification Review (OFT DCR) of all the Space Shuttle elements. The SSME portion of the OFT DCR was organized much the same as the CDR, with Joe Lombardo as the pre-board chairman and Bob Thompson as the SSME board chairman. The purpose of the DCR was to review the verification status of all design requirements and to certify to NASA Headquarters that the engine design was sufficiently mature as to be considered flight worthy. The SSME portion of the DCR was completed in April 1979 and approved by Bob Lindstrom, manager, Shuttle Projects Office, MSFC. The Space Shuttle DCR results and certifications were then presented to John Yardley, associate administrator for space transportation systems.

Test progress milestones were established for six individual "first tests" [12]. One of them was the first ISTB ignition test discussed in a previous section. The last one was the first SSME "all-up" throttling test which was accomplished on March 16, 1977, on Engine 0002 Test 902-056. The most significant test milestone was established in terms of total accumulated test duration of the single engine ground test program (excluding MPTA). A goal of 65,000 seconds was set by John Yardley as representing a sufficient level of development maturity to consider the engine flight worthy. NASA Headquarters considered the achievement of this goal to be a flight constraint. The goal of 65,000 seconds was reached on March 24, 1980, during a test on Engine 2004. Figure 27 shows the growth of the accumulated test time over the years and also the annual improvement in average test duration that made it possible to reach

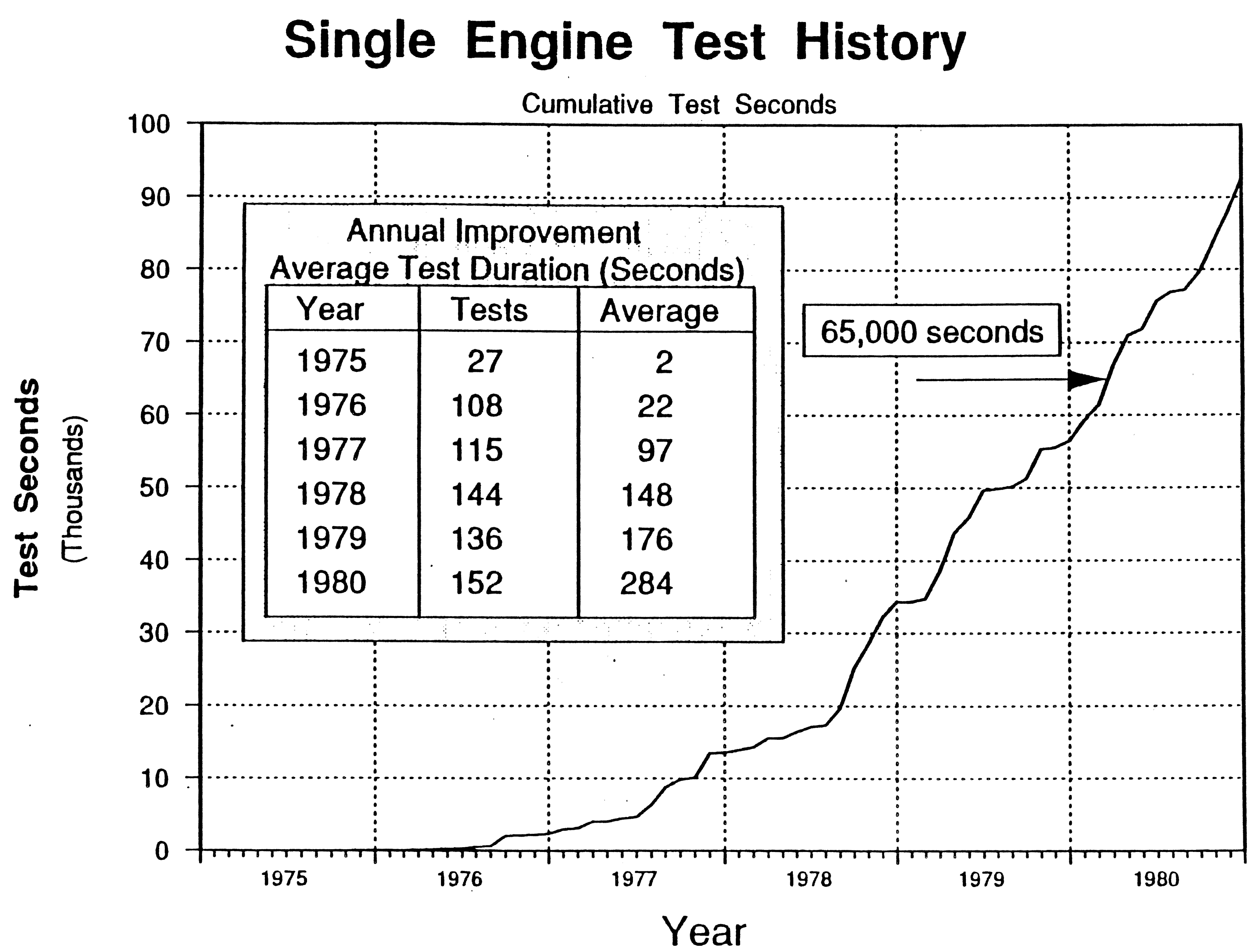

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Figure 27. Single Engine Test History

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the goal in that time period. The dramatic increase in average test duration was possible because the development problems were being solved and increasing confidence allowed more tests of longer duration to be scheduled.

The original SSME Program Plan included a Preliminary Flight Certification (PFC) demonstration test program to be conducted prior to the first flight. Specific requirements for the PFC evolved gradually during the program with the final requirements being established in early 1980 [40]. The PFC was defined in terms of a unit of tests that were called cycles. Each cycle consisted of 13 tests and 5,000 seconds of test exposure which included simulations of normal and abort mode flight profiles. It was required to conduct two PFC cycles on each of two engines of the flight configuration in order to certify that configuration for 10 shuttle missions. The PFC demonstration required 100 percent successful tests. If any test was shut down because of an engine problem, the PFC cycle did not count and had to be started over from zero.

Other PFC cycles were added to the program for the purpose of overstress testing and flight abort simulation. Eventually eight PFC cycles were completed prior to the first flight. A summary of the PFC cycles with their completion dates is given in Figure 28. At the time of the first flight, the SSME test program had accumulated 110,253 seconds during 726 tests.

THE FIRST FLIGHT

The first four flight configuration engines were assembled and acceptance tested in the first half of 1979. Engine acceptance testing included a 1.5-second start verification, a 100-second calibration firing and a 520 second flight mission demonstration test. Engine 2004 was allocated to the PFC demonstration program and Engines 2005, 2006 and 2007 were installed in the orbiter Columbia for the initial Space Shuttle flight. Several shuttle program problems (such as orbiter tile replacement) ensued which caused the first flight to be delayed. During this time significant changes were made to the three flight engines as a result of the test problems previously discussed. Because of the number and complexity of the changes, it was decided to repeat the final engine acceptance test. Engines 2005, 2006 and 2007 were removed from the orbiter and shipped to the engine test site at NSTL. In June 1980, all three engines successfully completed a 520 second flight mission demonstration test and were subsequently reinstalled in the orbiter Columbia.

A successful 20 second Flight Readiness Firing (FRF) was conducted on February 20, 1981. All three main engines were operated simultaneously at RPL with the entire Space Shuttle, including the solid rocket booster (SRB), on the launch pad in the launch attitude. The normal launch sequence was used including starting the main engines at T minus 6.6 seconds (staggered by 0.120 seconds). Liftoff was precluded by not igniting the SRBs (normally at T minus zero). The FRF had been planned as the final "all-up" verification that the engines and all interfacing systems were capable of satisfactory operation. Engine performance was within expected limits; and posttest hardware inspections, leak tests and other required checkouts were satisfactorily completed. The engines were ready for flight.

RATED POWER LEVEL (100%) CERTIFICATION SUMMARY FOR STS-1

CERTIFICATION OBJECTIVES	CERTIFICATION CYCLE REQUIREMENTS	ENGINE 2004				ENGINE 0009		ENG 2008	ENG 0008
		1ST CYCLE	2ND CYCLE	100/109% ABORT	102% CYCLE	1ST CYCLE	102% CYCLE	102% CYCLE	100/109% ABORT
TOTAL TESTS	13	16	17	7	4	21	13	16	5
STARTS TO RATED POWER LEVEL	13	14	15	7	4	15	13	14	5
TOTAL DURATION (SEC)	5000	5246	5098	2406	1298	5608	5041	5881	2540
TIME AT RATED POWER LEVEL	3000	3568	3643	1316	258	4078	31	479	1032
TIME ABOVE RATED POWER LEVEL	425	425 (102%)	425 (102%)	1090 (109%)	880 (102%)	427 (102%)	4070 423 (104%)	4363 433 (104%)	1440 (107% & 109%)
MISSION DURATIONS	6	8	8	3 (610 SEC)	2	9	8	9	4 (610 SEC)
NOMINAL MISSIONS (520 SEC)	4	5	6	–	2	7	6	7	–
ABORT TO ORBIT (665 SEC)	1	1	1	–	–	1	1	1	–
RETURN TO LAUNCH SITE ABORT (823 SEC)	1	2	1	–	–	1	1	1	–
DATE COMPLETED		6 79	2 80	4 80	7 80	9 80	12 80	1 81	3 81

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Figure 28. Certification Summary for STS-1

The countdown for the launch of STS-1 was initiated on April 5, 1981. This initial attempt was aborted at T minus nine minutes because of a problem with the orbiter computer systems. The computer problem was resolved by reloading the software, and the countdown was resumed on April 11. This time the countdown was successful with liftoff occurring at 7:00 a.m. on April 12, 1981 (Figure 29). Engine operation was flawless throughout the flight, maintaining a constant mixture ratio while responding to the power level commands issued by the orbiter guidance and control (G&C) computers. Figure 30 is a plot of actual power level for all three engines, and it shows how close to the same performance the three engines were. The start command was given at T minus 6.6 seconds, and all three engines were stabilized at RPL prior to liftoff at T minus zero. Less than a minute into the flight, all three engines were commanded to throttle down to 65 percent power level to reduce vehicle acceleration during the time of maximum external aerodynamic loading. After about fifteen seconds at 65 percent, the engines were returned to RPL, where they remained until the vehicle acceleration approached its design limit of three g's. The G&C computers then gradually throttled the engines so that the reduction in thrust would match the mass reduction due to propellant consumption and thereby maintain a constant safe acceleration. As the vehicle approached the required terminal velocity, the engines were throttled to 65 percent power level, allowed to stabilize for a little over five seconds and then commanded by the G&C computers to shut down.

The average engine performance was well within specification requirements; however, near the end of the flight, a small drift in mixture ratio was observed [41]. The shift of about one percent was found to be caused by radiant heating of a pressure sensor on each engine, located near the warm HPOTP turbine seal drain lines. The radiation had no effect during ground testing or even in flight until the engines were operated in the vacuum of space. The anomaly was eliminated on future flights by adding a small amount of insulation and a radiation shield.

The Space Shuttle orbiter Columbia achieved its predicted orbit and remained there for two days. Americans were back in space after an absence of almost six years. Return from orbit occurred on April 14, 1981, with a safe landing at Edwards Air Force Base, California. After the post-flight inspections at the Dryden Flight Research Center found the engines to be in perfect condition, Columbia was returned to KSC on April 28 to prepare for the next flight. The first reusable spacecraft had been sent to space, safely returned to earth, and was ready to go again. The era of the Space Shuttle had begun.

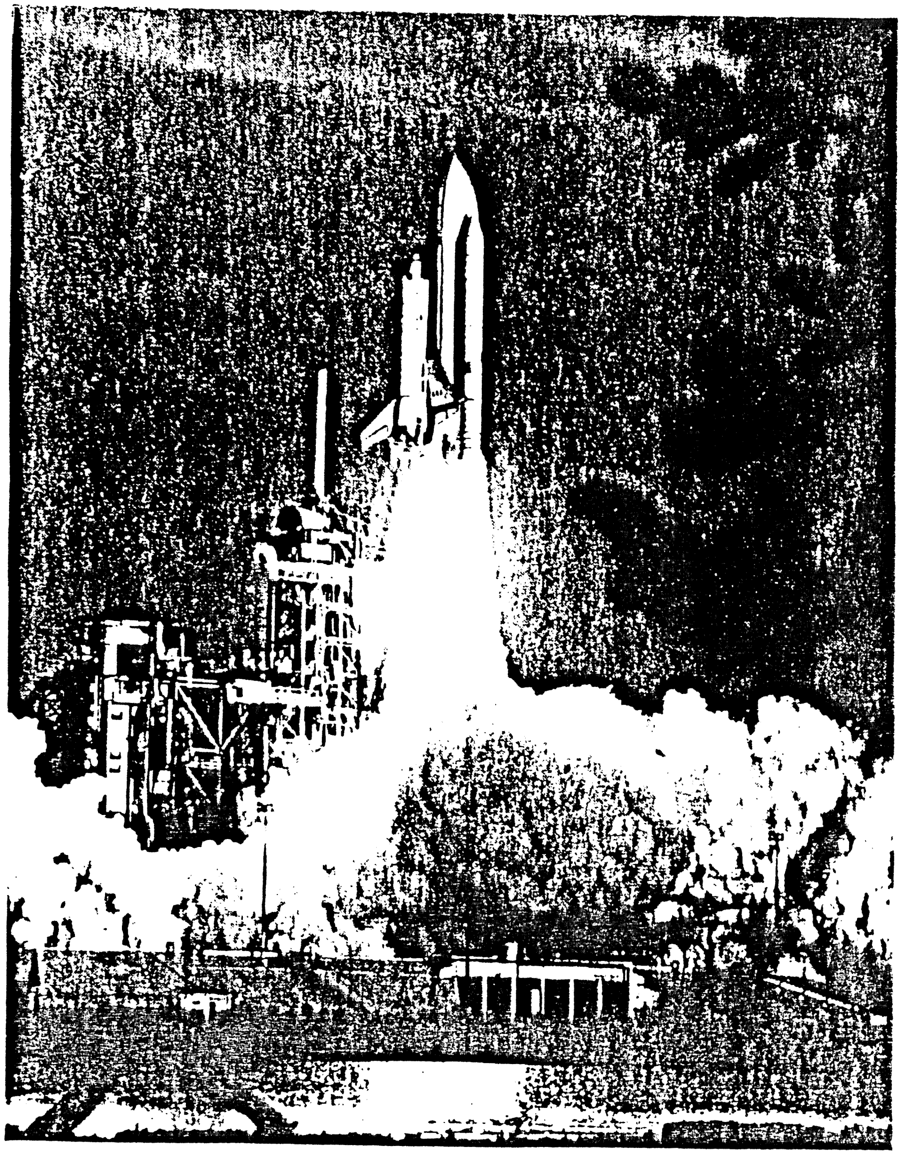

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Figure 29. The First Space Shuttle Flight - STS-1

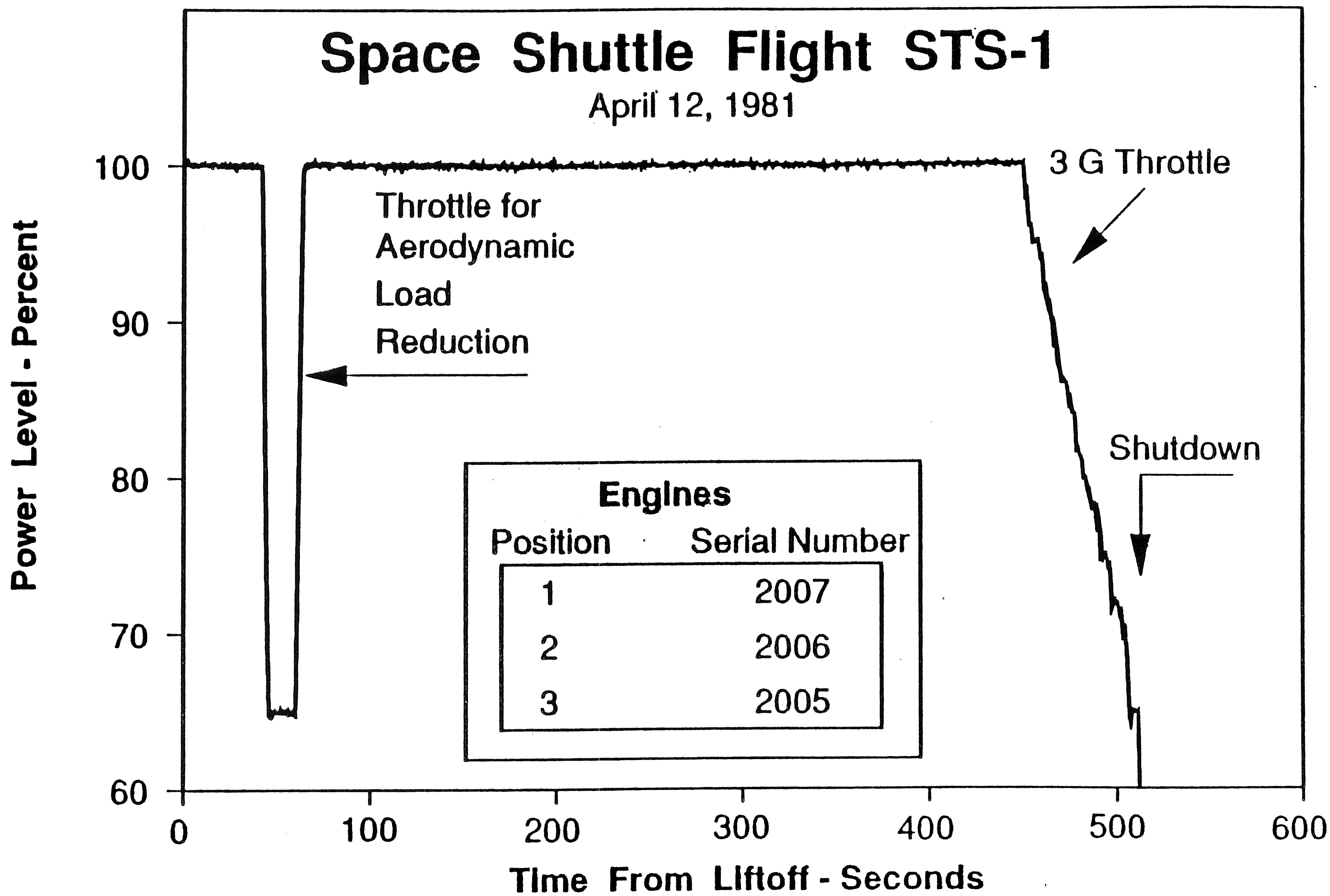

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Figure 30. STS-1 Engine Performance

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ACKNOWLEDGMENTS

The development of the Space Shuttle Main Engine was an arduous task beset with many technical, logistical and managerial difficulties. The team that solved these problems pushed and advanced the state of the art in many different fields. Thousands of specialists at Rocketdyne and the George C. Marshall Space Flight Center worked doggedly for a decade to produce a device which now ranks high on the list of mankind's greatest engineering achievements. For many, personal sacrifices took the form of long nights, weekends and holidays lost, and punishing cross-country air travel with days and weeks at a time away from their families and loved ones. The very fast-paced, high-stress environment caused others to pay a higher price. There were casualties along the way.

Many individuals deserve special recognition for significant contributions to the success of the program, as can be attested to by the hundreds of awards and certificates that have been presented by the government for this achievement. An attempt by the author to list only the most deserving of these was abandoned after surpassing 200 names and still finding that several very significant contributors were not listed. Wishing to avoid my embarrassment for having ignored someone truly deserving and not wanting to assign the task to a committee, such a list will not appear here.

ACRONYMS and ABBREVIATIONS

ASI	Augmented Spark Igniter
CCV	Chamber Coolant Valve
CDR	Critical Design Review
CEI	Contract End Item
CRES	Corrosion Resistant Steel
DCR	Design Certification Review
DVS	Design Verification Specification
EPL	Emergency Power Level
F	Fahrenheit (degrees of temperature)
FMOF	First Manned Orbital Flight
FPB	Fuel Preburner
FPL	Full Power Level
FPOV	Fuel Preburner Oxidizer Valve
FRF	Flight Readiness Firing
g	Gravitational Constant
GAO	General Accounting Office
GPM	Gallons per Minute
G&C	Guidance and Control
HPFTP	High Pressure Fuel Turbopump
HPOTP	High Pressure Oxidizer Turbopump
Hz	Hertz (Cycles per Second)
ICD	Interface Control Document
INCO	Inconel (iron, chromium and nickel)
ISTB	Integrated Subsystem Test Bed
KSC	Kennedy Space Center
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
LPFTP	Low Pressure Fuel Turbopump
LPOTP	Low Pressure Oxidizer Turbopump
MCC	Main Combustion Chamber
MEC	Main Engine Controller
MFV	Main Fuel Valve
MOV	Main Oxidizer Valve
MPL	Minimum Power Level
MPTA	Main Propulsion Test Article
MSFC	Marshall Space Flight Center
MTF	Mississippi Test Facility
NASA	National Aeronautics and Space Administration
NSTL	National Space Technology Laboratories
OFT	Orbital Flight Test
OPB	Oxidizer Preburner
OPOV	Oxidizer Preburner Oxidizer Valve
PDR	Preliminary Design Review
PFC	Preliminary Flight Certification
psi	Pounds per Square Inch
psia	Pounds per Square Inch (Absolute)
R	Rankine (degrees of absolute temperature)
RID	Review Item Discrepancy
rms	Root-Mean-Square
RPL	Rated Power Level
RPM	Revolutions per Minute
scfm	Standard Cubic Feet per Minute
SEM	Scanning Electron Microscope
SRB	Solid Rocket Booster
SSC	Stennis Space Center
SSFL	Santa Susana Field Laboratory
SSME	Space Shuttle Main Engine
STS	Space Transportation System
T	Time (liftoff)
TCA	Thrust Chamber Assembly

Authors note: To the reader who thinks that acronyms should be avoided at all costs, consider that these acronyms were an important part of the SSME program language and, therefore, have historical significance worthy of a place in the written history of the program.

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* *North American Rockwell became Rockwell International corporation on February 16, 1973.*