SPACE SHUTTLE MAIN ENGINE THE FIRST TEN YEARS

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Part 5 – High Pressure Oxidizer Turbopump Explosions

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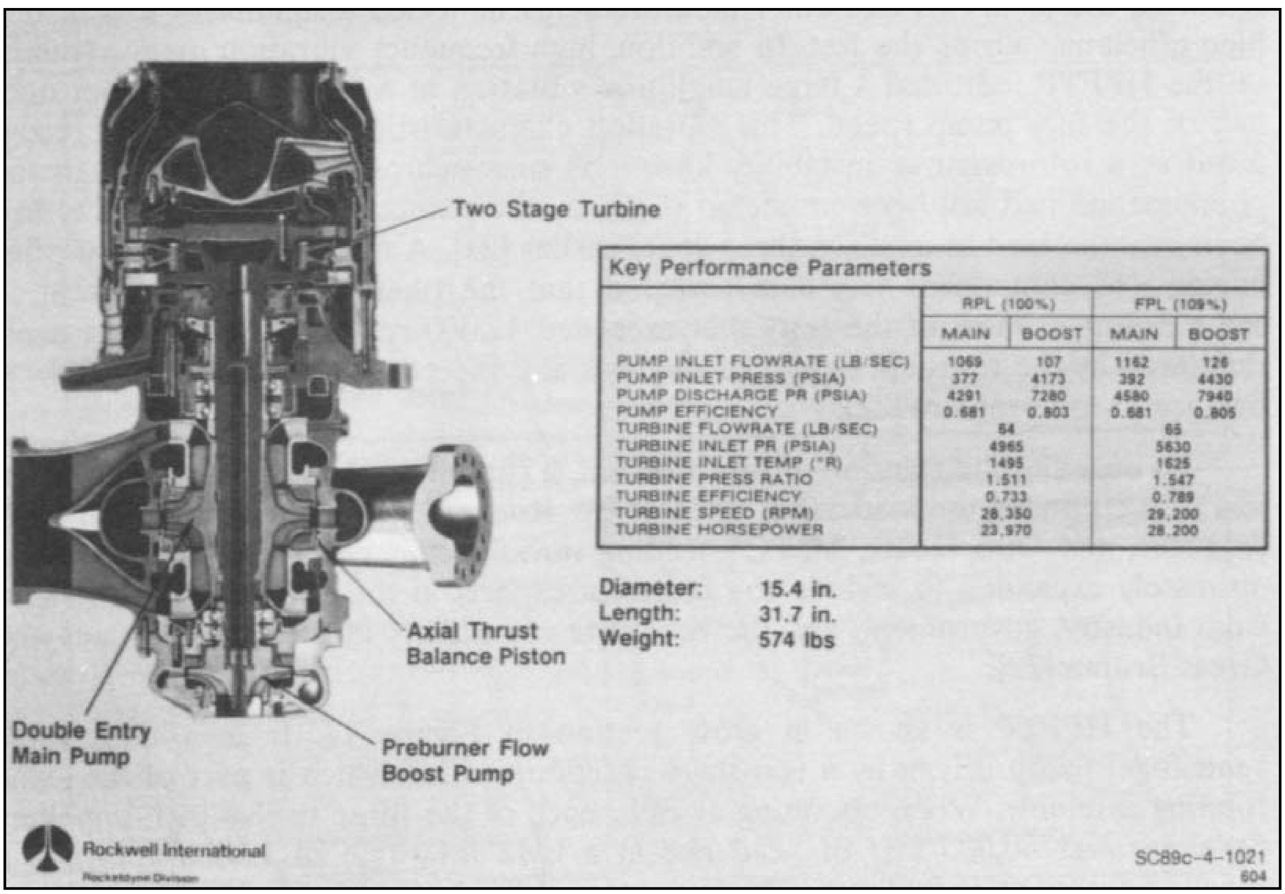
High Pressure Oxidizer Turbopump Explosions

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,



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	

Figure 13. SSME High Pressure Oxidizer Turbopump (Photo No. SC89c-4-1021)

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 located between the turbine end bearings and the second-stage turbine wheel (Figure 14). The turbine seal assembly is shown on the right-hind 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

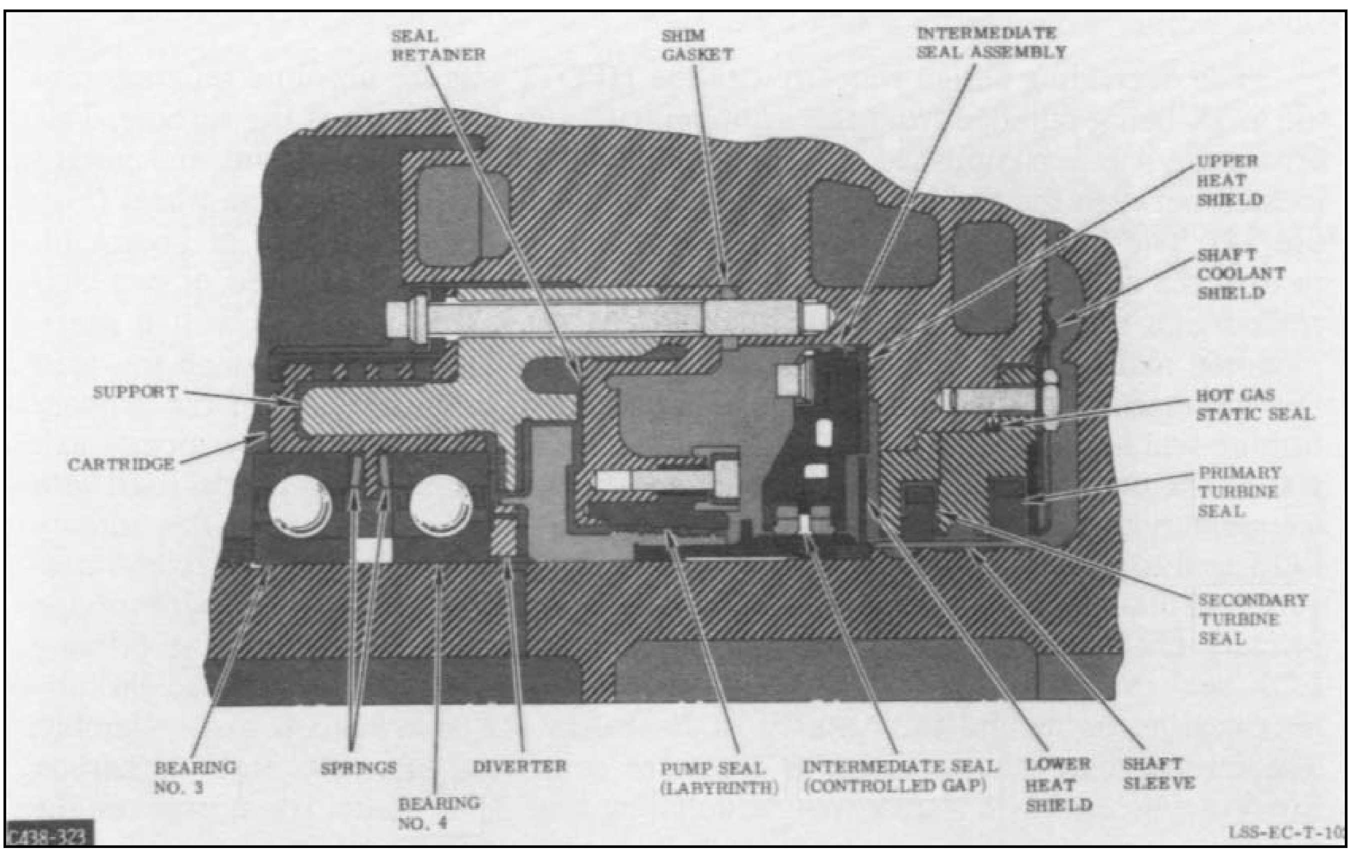


Figure 14. HPOTP Shaft Seal and Drains (Engines 2004 and Subsequent) Photo No. LC438-323 or LSS-EC-T-102)

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 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 the part of it out 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.

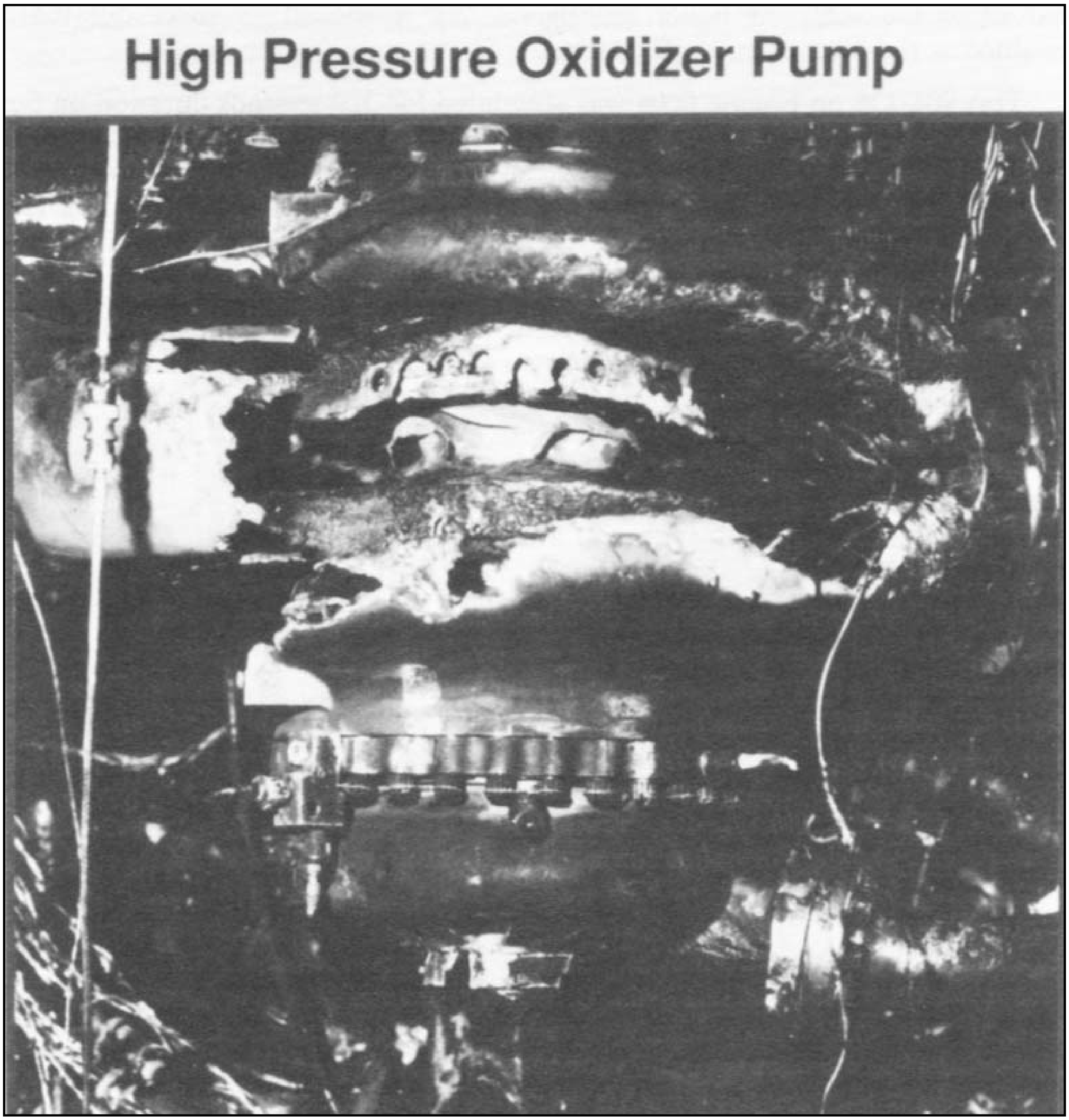


Figure 15. HPOTP After Test 901-110 (Photo No. SC89C-4-1009)

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 early detection of abnormalities.

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 commingling 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 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 lancing 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

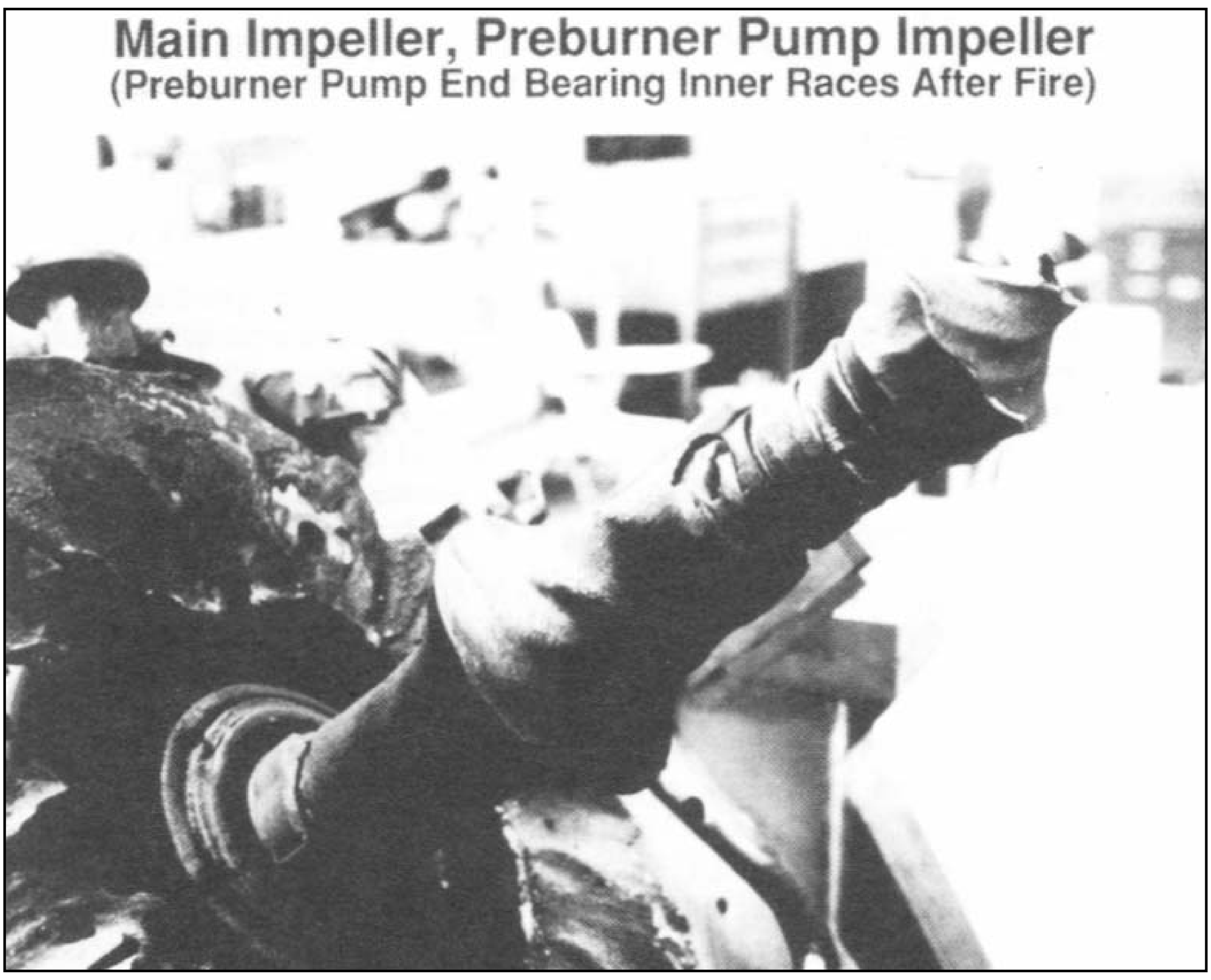


Figure 16. HPOTP Rotor After Test 901-136 (Photo No. 89c-4-1029)

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 cut-off 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 the exclusion of all but two of them, and they 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 bum 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 the 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 that opens into the MCC combustion zone just below the injector. The sense port is purged with hydrogen to prevent the 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.