Advanced Health Management System

In 2000, NASA's MSFC began development of the **Advanced Health Management System** (AHMS), a modification of the existing Block II main engine controller. The AHMS became active on mission STS-117 in June 2007. This final enhancement of the SSME included the addition of advanced digital signal processors, radiation-hardened memory, and new software. These changes to the main engine controller provided the capability of monitoring the vibrations of the high-pressure turbopumps in such a way that made it possible "to analyze and discriminate true rotor unbalance from erroneous sensor readings."[1177] They could detect and track a very subtle shift in the engine's vibration levels in a split second, allowing the engine to be safely shut down.

SSME Physical and Functional Descriptions

SSMEs by the Numbers

With the final mission of the SSP, forty-six engines were flown in 135 launches for a total of 405 engine missions. Of the total engine missions, 273 were completed with the FMOF, Phase II, or Block I configuration engines; forty-nine were with the Block IIA configuration containing the new large throat main combustion chamber; and eighty-three were Block II configuration featuring both the large throat main combustion chamber and the new high-pressure fuel turbopump.[1178] Typically, existing engines were modified to incorporate the newest design. All seven newly manufactured Block I configuration engines (Engines 2036 through 2042) were upgraded, and all fourteen Block IIA configuration engines were modified to Block II when the new high-pressure fuel turbopump became available. Many components from the earlier Phase II and Block I engines were used for the upgraded engines.[1179] Two original FMOF engines, 2007 and 2015, each underwent two successive rebuilds to the Phase II and Block I configurations. Engine 2007 began service with STS-1, and flew on *Columbia's* initial five missions. It ended service with STS-52, launched in October 1992, its thirteenth flight.

The SSP lost six engines as the result of the *Challenger* and *Columbia* accidents. Of the three SSMEs lost on Challenger, Engines 2020, 2021, and 2023, Engines 2020 and 2021 had flown together on four of their five previous flights. Engines 2049, 2053, and 2055 were lost with *Columbia*. This had been the maiden flight of Engine 2055.

[1177] NASA MSFC, *Space Shuttle Main Engine Advanced Health Management System*, NASA Facts, (Huntsville, AL: George C. Marshall Space Flight Center, August 2007), http://www.nasa.gov/centers/marshall/pdf/186582main_REV_B_AHMS_Fact_Sheet_STS-118.pdf; Jue and Kuck, "Options for the Future," 2.
[1178] Pratt & Whitney Rocketdyne, Inc., "Space Shuttle Main Engine KSC Processing Nominal Flow (Landing to Launch)," no date, 26, presentation materials provided to Joan Deming and Patricia Slovinac, KSC, June 2010.
[1179] Jenkins, *Space Shuttle*, 420.

The SSP ended with fourteen SSMEs in the active fleet. All were Block II engines with a two-duct powerhead. In the heyday of the program, twelve engines were kept flight-ready. At any time, an average of two to three engines were out of service, and there were always six engines ready, including three on the orbiter vehicle and three ready to be swapped out, if needed.

Engine 2019 was the fleet leader during the SSP. It flew nineteen missions, beginning with the launch of STS-9 (*Columbia*) on November 28, 1983, and completed its service with the landing of STS-93 (*Columbia*) on July 27, 1999. The newest addition to the SSME fleet, Engine 2061, arrived at KSC on December 19, 2008.[1180] It flew only two missions, STS-130, launched on February 8, 2010, and STS-134, launched on May 16, 2011. Engines 2045, 2047, and 2060 were the last to fly out the program on STS-135 (*Atlantis*). Of these, Engines 2045 and 2047 were both veterans of fourteen previous missions. Engine 2017 was the only unmodified engine to fly on all five orbiters, on flights dating from STS-6 (*Challenger*) in April 1983, through STS-75 (*Columbia*) in February 1996.

General Description

Each SSME measured approximately 14’ in length and 7.5’ in diameter at the exit of the nozzle, and weighed approximately 7,775 pounds. The engine powerhead, the portion located above the nozzle, included the two high-pressure turbopumps and the main combustion chamber, plus the main injector and the two preburner injectors.

SSME Major Components

The SSME contained approximately 50,000 parts, of which 7,000 were tracked periodically for replacement.[1181] The major components included the low-pressure fuel turbopump, the high-pressure fuel turbopump, the low-pressure oxidizer turbopump, the high-pressure oxidizer turbopump, the hot gas manifold, the oxidizer and fuel preburners, the main combustion chamber, the oxidizer heat exchanger, the nozzle, and five propellant valves. Physical and functional descriptions of each major Block II engine component follow.

<u>Low- Pressure and High-Pressure Turbopumps</u>

Each SSME had two high-pressure turbopumps that supplied LO2 and LH2 to the engine’s main combustion chamber. A turbopump is a single unit consisting of a pump, driven by a turbine, that boosts the pressure of the propellant. The low-pressure oxidizer and low-pressure fuel turbopumps were mounted 180 degrees apart on the engine. The ducts from the low-pressure

[1180] Helen Lewin, “SSME Planned Assignments Including Performance Impacts,” Pratt & Whitney Rocketdyne, September 17, 2009, 20, http://rkdn.ksc.nasa.gov. Engine 2062 was also finished, but never acceptance tested. VanHooser, personal communication.
[1181] NASA KSC, *Space Shuttle Main Engine Processing Facility*, NASA Facts (Florida: Kennedy Space Center, 2006), http://www.nasa.gov/centers/kennedy/pdf/ 167449main_SSMEPF-06.pdf.

turbopumps to the high-pressure turbopumps contained flexible bellows that enabled them to flex when loads were applied.[1182]

The **Low-Pressure Oxidizer Turbopump** (Figure No. C-19) contained an axial-flow inducer driven by a six-stage hydraulic turbine. It boosted the LO2 pressure from 100 psia to 422 psia. The flow was supplied to the high-pressure oxidizer turbopump to permit it to operate at higher speeds without cavitating.[1183] The low-pressure oxidizer turbopump operated at approximately 5,150 rpm. It measured approximately 18” x 18”, and was flange-mounted to the orbiter propellant ducting.[1184] A triple-redundant, magnetic-type, speed transducer was located on the turbine end.

The **Low-Pressure Fuel Turbopump** (Figure No. C-20) contained an axial-flow inducer driven by a two-stage, axial-flow turbine powered with gaseous hydrogen. It boosted LH2 pressure from 30 psia to 276 psia and supplied the high-pressure fuel turbopump. During engine operation, this pressure increase allowed the high-pressure fuel turbopump to operate at high speeds without cavitating. The low-pressure fuel turbopump operated at approximately 16,185 rpm. It measured approximately 18” x 24”, and was flange-mounted to the SSME at the inlet to the low-pressure fuel duct.[1185] Foam insulation encased in a Kevlar jacket covered the pump housing.

The **High-Pressure Oxidizer Turbopump** (Figure No. C-21), which debuted in July 1995, contained a mainstage pump for all of the oxidizer flow and another for a portion of the oxidizer flow used to supply the preburners. The mainstage pump was a double entry centrifugal impeller flanked by two inducers. The preburner pump was a single centrifugal impeller. The turbopump had a common shaft and was driven by a three-stage, hot gas turbine. The main pump boosted LO2 pressure from 422 psia to 4,300 psia while operating at approximately 28,120 rpm. The turbopump provided 970 pounds of LO2 per second.

The high-pressure oxidizer turbopump discharge flow split into several paths, one of which was routed to drive the low-pressure oxidizer turbopump turbine. Another path was routed through the main oxidizer valve and entered the main combustion chamber. Another small path was tapped off and sent to the oxidizer heat exchanger, where it was vaporized and then used to pressurize the external tank. The final path entered the preburner impeller to raise the LO2’s pressure from 4,300 psia to 7,420 psia for use in both preburners. The high-pressure oxidizer turbopump measured approximately 24” x 36”, and was flange-mounted to the hot gas manifold.[1186]

[1182] USA, *Crew Operations*, 2.16-6.
[1183] Cavitation occurs when cavities of gas develop and collapse in liquid fuels.
[1184] USA, *Crew Operations*, 2.16-5.
[1185] USA, *Crew Operations*, 2.16-4.
[1186] USA, *Crew Operations*, 2.16-5, 2.16-6.

The **High-Pressure Fuel Turbopump** (Figure No. C-22), which debuted in July 2001, was the most complex component of the SSME. The three-stage centrifugal pump was driven by a two-stage, hot gas turbine. It supplied 162 pounds of LH2 fuel per second, boosted LH2 pressure from 276 psia to 6,515 psia, and operated at a speed of approximately 36,000 rpm, or 600 times per second. Because of the centrifugal force at this speed, the turbine blades, which normally weigh 13 ounces each, weighed the equivalent of 14 tons.[1187] The high-pressure fuel turbopump generated 70 hp for each pound of its weight, compared with an automobile engine, which generates about 0.5 hp for each pound of its weight. It measured approximately 22" x 44", and was flange-mounted to the hot gas manifold.

The discharge flow from the high-pressure fuel turbopump was routed through the main fuel valve and then split into three flow paths. One path was through slots in the jacket of the main combustion chamber, where the hydrogen was used to cool the chamber walls, and then delivered to the low-pressure fuel turbopump to drive its turbine. The second flow path, through the chamber coolant valve, supplied LH2 to the preburner combustion chamber and also cooled the hot gas manifold. The third hydrogen flow path was used to cool the engine nozzle. It then joined the second flow path from the chamber coolant valve.[1188]

Hot Gas Manifold

The hot gas manifold, the central component of the powerhead, was considered the structural backbone of the engine. It tied together and structurally supported the major components and almost all of the engine weight. Hot gas generated by the preburners, after driving the high-pressure turbopumps, passed through the hot gas manifold on the way to the main combustion chamber.[1189]

The hot gas manifold was manufactured in two halves which were joined together by electron-beam welding. The structural outer walls consisted of an alloy 903 sheet metal liner, with a space between the liner and wall cooled by hydrogen gas to reduce the outer wall temperature.[1190] The main injector was located in the center of the hot gas manifold. It included 600 coaxial elements which injected LO2 through their center posts. Flow shields, bolted to the outer row of elements, helped to protect them from damage and erosion from the high-velocity gas.

The redesigned two-duct hot gas manifold, first flown in July 1995, replaced the three small fuel ducts with two enlarged ducts. This modification significantly improved fluid flows in the system, decreased pressure and turbulence, and lowered temperatures in the engine during operation. As a result, the overall performance of the engine was enhanced and maintenance was reduced.

[1187] Goetz, interview.
[1188] USA, *Crew Operations*, 2.16-4, 2.16-5.
[1189] USA, *Crew Operations*, 2.16-7.
[1190] Jewett and Halchak, "Alloy 718," 754.

Preburners

Both the fuel preburner and oxidizer preburner were welded to the hot gas manifold. The first stage of combustion took place in the two preburners, where LO2 and LH2 were partially burned. The preburners produced hot gas that passed through the turbines to generate the power to drive the high-pressure pumps.[1191] The hot gas then passed through the hot gas manifold on the way to the main combustion chamber. Here, the addition of LO2 resulted in further combustion.

The structural body and inlet manifold of each preburner were machined from Inconel alloy 718 forgings and preformed sheet metal. These were joined by electron-beam and gas tungsten welding. The fuel preburner had an internal diameter of 10.43" and a combuster length of 4.37". The injector was made up of 264 coaxial elements, arranged in a concentric row pattern. Twenty-four of the elements supported and cooled three baffles that helped to stabilize combustion. An augmented spark ignition chamber was located in the center of the injector. The oxidizer preburner had an internal diameter of approximately 7.5" and a combuster length of 4.25". The injector was comprised of 120 coaxial elements, arranged in a concentric row pattern. Fifteen of the elements supported and cooled the three baffles. Of similar configuration to the fuel preburner, it contained a spark ignition chamber in the center of the injector. [1192]

Main Combustion Chamber

The main combustion chamber (Figure No. C-23), bolted to the hot gas manifold, was where the LH2 and LO2 from the fuel and oxidizer preburners were mixed and burned to provide thrust.

The main combustion chamber had to tolerate hot gases at temperatures up to 6,000 degrees F. It also had to contain the internal pressure of 3,000 psi. To meet these demands, Rocketdyne developed NARloy-Z, a high conductivity copper-based alloy that contained silver and zirconium. The exterior of the liner was made from structural nickel which was applied by an electroforming process. The support jacket of the main combustion chamber was made from Inconel alloy 718. The main combustion chamber was cooled by super-cold hydrogen, which flowed through 430 channels machined into the liner inner wall.

A small augmented spark igniter chamber was located in the center of the main combustion chamber's injector. The main injector measured approximately 17.7" in diameter at the end, and featured a barrel-shaped collection of 600 identical, non-baffle injector elements, arranged in concentric rings.[1193] Each element was a hollow cylindrical post through which hot gases flowed. The dual-redundant igniter was used during the engine start sequence to initiate combustion. The

[1191] USA, *Crew Operations*, 2.16-7.
[1192] Baker, *Manual*, 104.
[1193] Steven J. Wofford, personal communication with James M. Ellis, MSFC, August 31, 2011.

igniter was turned off after approximately three seconds because the combustion process was self-sustaining.[1194]

Heat Exchanger

Mounted in the oxidizer side of the hot gas manifold, the single-coil heat exchanger was made from a continuous piece of coiled stainless steel alloy tubing measuring 41' in length, and with an outer diameter of 0.50". It drew on engine heat from the turbine discharge flow from the high-pressure oxidizer turbopump to produce a flow of GO2 that pressurized the ET oxygen tank. Until mid-1995, the heat exchanger featured seven welds. The redesigned exchanger eliminated all seven welds and tripled the wall thickness of the tube. The increased thickness, to 0.032 inches compared with as thin as 0.0125 inches previously, served to reduce wear, and thus make catastrophic failures less likely. Maintenance time and post-flight inspections also were minimized.

Nozzle

The engine nozzle (Figure No. C-24) extended below the main combustion chamber. The velocity of the combustion gas was governed by the nozzle area ratio.[1195] The SSME nozzle measured 10.3" in diameter at the throat, and 90.7" at the nozzle exit. Total length of the nozzle was 121". The throat area measured approximately 93 square inches and the nozzle area was 50.265 square feet. The nozzle configuration underwent a number of successive design changes to meet requirements specifying an area ratio of 77.5:1 and a length equal to 80 percent of a fifteen degree conical nozzle.[1196] At 100 percent power level, propellants flowed through the nozzle at a rate of 1,035 pounds per second. "The nozzle accelerates the combustion products to 17,000 feet per second at the nozzle exit, generating 470,000 pounds of thrust at vacuum."[1197]

Coolant feed lines were located at the aft end of the nozzle. The inside wall of the nozzle was lined with 1,080, 1/8" stainless steel cooling tubes that carried hydrogen. The tubes were brazed to the surrounding structural jacket. During flight, a portion of the fuel was first circulated through the tubes before it was directed to the combustion chamber. Nine hatbands were welded around the jacket for hoop strength, and a hydrogen feed line ("steerhorn") measuring 1.625" in diameter also was attached to the nozzle exterior. Coolant manifolds were welded to the top and bottom of the nozzle, along with three fuel transfer ducts and six drain lines.

A support ring welded to the throat of the nozzle was the attach point for the engine heat shield. For protection from the high temperatures during the launch, ascent, on-orbit, and entry phases,

[1194] USA, *Crew Operations*, 2.16-7.
[1195] The nozzle area ratio is derived by dividing the nozzle exit area by the throat area. R.A. O'Leary and J.E. Beck, "Nozzle Design,"1992, http//:www.engineeringatboeing.com.
[1196] O'Leary and Beck, "Nozzle Design."
[1197] O'Leary and Beck, "Nozzle Design."

portions of the nozzle were insulated with four layers of metallic batting covered with a metallic foil (Nichrome) acting as a thermal shield, and closed out by a layer of fine weave Nichrome screen.[1198]

Propellant Valves

Each engine had five major valves: the oxidizer preburner oxidizer valve, the fuel preburner oxidizer valve, the main oxidizer valve, the main fuel valve, and the chamber coolant valve. These valves were hydraulically actuated and controlled by electrical signals from the engine controller.

The **oxidizer preburner oxidizer valve** and the **fuel preburner oxidizer valve** were used to control the thrust level of the engine. The speeds of the high-pressure oxidizer turbopump and high-pressure fuel turbopump depended on the position of these two valves. The valves increased or decreased the LO2 flow into the preburners, thereby increasing or decreasing preburner chamber pressure and high-pressure oxidizer turbopump and high-pressure fuel turbopump speed. This directly affected LO2 and gaseous hydrogen flow into the main combustion chamber, which in turn increased or decreased engine thrust. The fuel preburner oxidizer valve was used to maintain a constant six-to-one propellant mixture ratio.[1199]

The **main oxidizer valve** controlled LO2 flow into the engine combustion chamber. The **main fuel valve** controlled the total LH2 flow into the engine cooling circuit, the preburner supply lines, and the low pressure fuel turbopump turbine. When the engine was operating, the main valves were fully open. A **chamber coolant valve** was located on each engine combustion chamber coolant bypass duct. It regulated the amount of gaseous hydrogen allowed to bypass the nozzle coolant loop to control engine temperature.[1200]

Other SSME Components and Systems

Main Engine Controller

Each SSME had its own on-board digital computer, which monitored and controlled all engine functions and diagnostics. It could shut an engine down if it detected a problem. Instructions to the engine control elements were updated 50 times per second, or every twenty milliseconds. The pressurized, thermally conditioned controller, manufactured by Honeywell, was attached to the thrust chamber and nozzle coolant outlet manifolds on the low-pressure fuel turbopump side of the engine. Each controller contained two redundant digital computer units, and each Block II computer used Motorola 68000 32-bit microprocessors. The double-redundant system contained a total of four processors per controller. All the sensors and actuators were connected directly to

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[1198] USA, *Crew Operations*, 2.16-7.
[1199] USA, *Crew Operations*, 2.16-8.
[1200] USA, *Crew Operations*, 2.16-8.

the controller. The microprocessors operated in "lock-step" within the dual central processing units (A and B). Prior to replacement by the Motorola processors, the controller used two redundant Honeywell HDC-601 computers.

The controller, operating in conjunction with the engine sensors, valves, actuators, and spark igniters, formed a self-contained system for engine control, checkout, and monitoring. It provided "engine flight readiness verification, engine start and shutdown sequencing, closed-loop thrust and propellant mixture ratio control, sensor excitation, valve actuator and spark igniter control signals, engine performance limit monitoring, and performance and maintenance data," as well as "onboard engine checkout, response to vehicle commands, and transmission of engine status." [1201]

The SSME controller processed four critical engine operating parameters and closely monitored them to see whether they remained within the specified limits (or "redlines"). A redline violation sensed by the controller caused it to automatically shut down the engine.[1202] In-flight parameters included:

- The high-pressure fuel turbopump's turbine discharge temperature not to exceed 1,860 degrees Rankine (R)[1203]
- The high-pressure oxidizer turbopump's turbine discharge temperature not to exceed 1,660 degrees R or fall below 720 degrees R.
- The high-pressure oxidizer turbopump's intermediate seal purge pressure not to fall below 159 psia.
- During steady state operation, the main combustion chamber's pressure not to fall more than 200 psia (400 psia, during throttling) below the reference chamber pressure.

Additional parameters were monitored on the ground prior to engine start, or following engine start but prior to SRB ignition. Exceedance of specified values for these parameters could also initiate a shutdown or inhibit engine start.

<u>Bleed Valves</u>

Two bleed valves were contained in each SSME, including one LH2 bleed valve and one LO2 bleed valve. The **liquid hydrogen bleed valves** were used to circulate LH2 through the engines

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[1201] USA, *Crew Operations*, 2.16-9, 2.16-10.
[1202] USA, *Crew Operations*, 2.16-26. Redlines were designed to avert catastrophic failure by initiating engine shutdown. Synchronous vibration redlines were later added, with the incorporation of AHMS, for the high-pressure oxidizer turbopump and high-pressure fuel turbopump, bringing the total of active, in-flight redlines to six. The Phase II and earlier SSMEs had two more redlines. These were a secondary seal redline on the high-pressure oxidizer turbopump seal package, and a coolant liner redline on the high-pressure fuel turbopump. Wofford, personal communication; Jon D. Reding, personal communication.
[1203] Rankine is a temperature measurement unit equal to one Fahrenheit degree, and zero on this scale is an absolute zero. Under the standard atmospheric pressure 0 Rankine equals -459.67 Fahrenheit. This scale does not have any temperature below zero; Aqua-Calc. "What is Rankine," http://www.aqua-calc.com.

during prelaunch thermal conditioning. They also served to dump the LH2 trapped in the engines after MECO. The **liquid oxygen bleed valves** connected the engine internal LO2 lines to an overboard port. They were used only during prelaunch thermal conditioning.

Helium System

Helium was used to pneumatically close the five main hydraulically-actuated valves in the propellant lines should a hydraulic failure occur. The helium system also was used to purge the high-pressure oxidizer turbopump intermediate seals. Helium was injected between the seals to keep the hydrogen used to cool the turbine-end bearings from mixing with the LO2 in the pump end.[1204]

Pneumatic Control Assembly

Each SSME had one pneumatic control assembly. The assembly contained solenoid valves which were energized by commands from the SSME controller to control and perform various functions. These functions included "the high-pressure oxidizer turbopump intermediate seal cavity and preburner oxidizer dome purge, pogo system postcharge, and pneumatic shutdown."[1205]

Thrust Vector Control Actuators

Two main engine TVC actuators were connected to the powerhead of each SSME. One was for yaw and the other for pitch. The pitch actuator could move the engine 10.5 degrees up or down and the yaw actuator a maximum of 8.5 degrees up or down.[1206] Each actuator had its own hydraulic switching valve and received hydraulic pressure from the orbiter hydraulic systems.[1207] The actuators provided attitude control and trajectory shaping by gimbaling both the SSMEs and SRBs during first-stage and the SSMEs alone during second-stage. They changed each main engine's thrust vector direction as needed during the flight sequence.

SSME Process Flow

Since the arrival of the first SSME at KSC in 1979, Pratt & Whitney Rocketdyne was responsible for SSME processing. Historically, the engines were built and assembled at Rocketdyne's facility in Canoga Park, California (Figure Nos. C-25 through C-30), with flight inspections performed at KSC. With the completion of the Space Shuttle Main Engine

[1204] Mark Kirkman, "Space Shuttle Systems 101 – More Than You Ever Needed To Know About the Space Shuttle Main Engines," *InterSpace News*, July 27, 2008, 4, http://www.interspacenews.com/ FeatureArticle/tabid/130/ Default.aspx?id=2130.
[1205] USA, *Crew Operations*, 2.16-24.
[1206] Baker, *Manual*, 105.
[1207] USA, *Crew Operations*, 2.16-25.