SPACE SHUTTLE MAIN ENGINE THE FIRST TEN YEARS[1]

Robert E. Biggs[2]

Part 1 – The Engine

In July 1971, the Rocketdyne Division of Rockwell International[3] 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 began in April 1972. The engine, to be developed under contract with the Marshall Space Flight Center of the 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 when STS-1 was launched from pad 39 at the Kennedy Space Center.

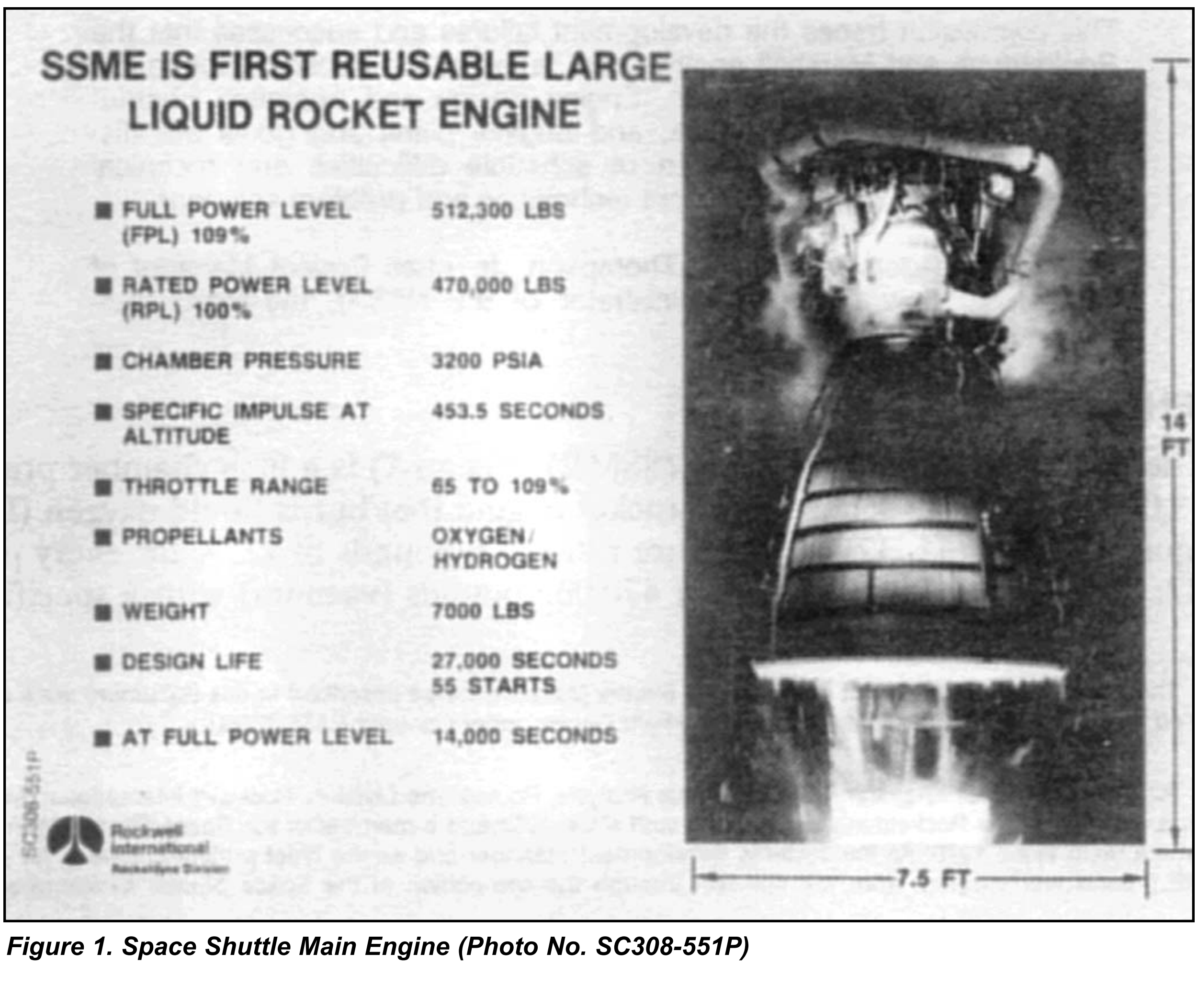

Figure 1. Space Shuttle Main Engine (Photo No. SC308-551P)

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 the NASA), the team "persevered and pressed on."

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 a half times the main combustion chamber pressure.

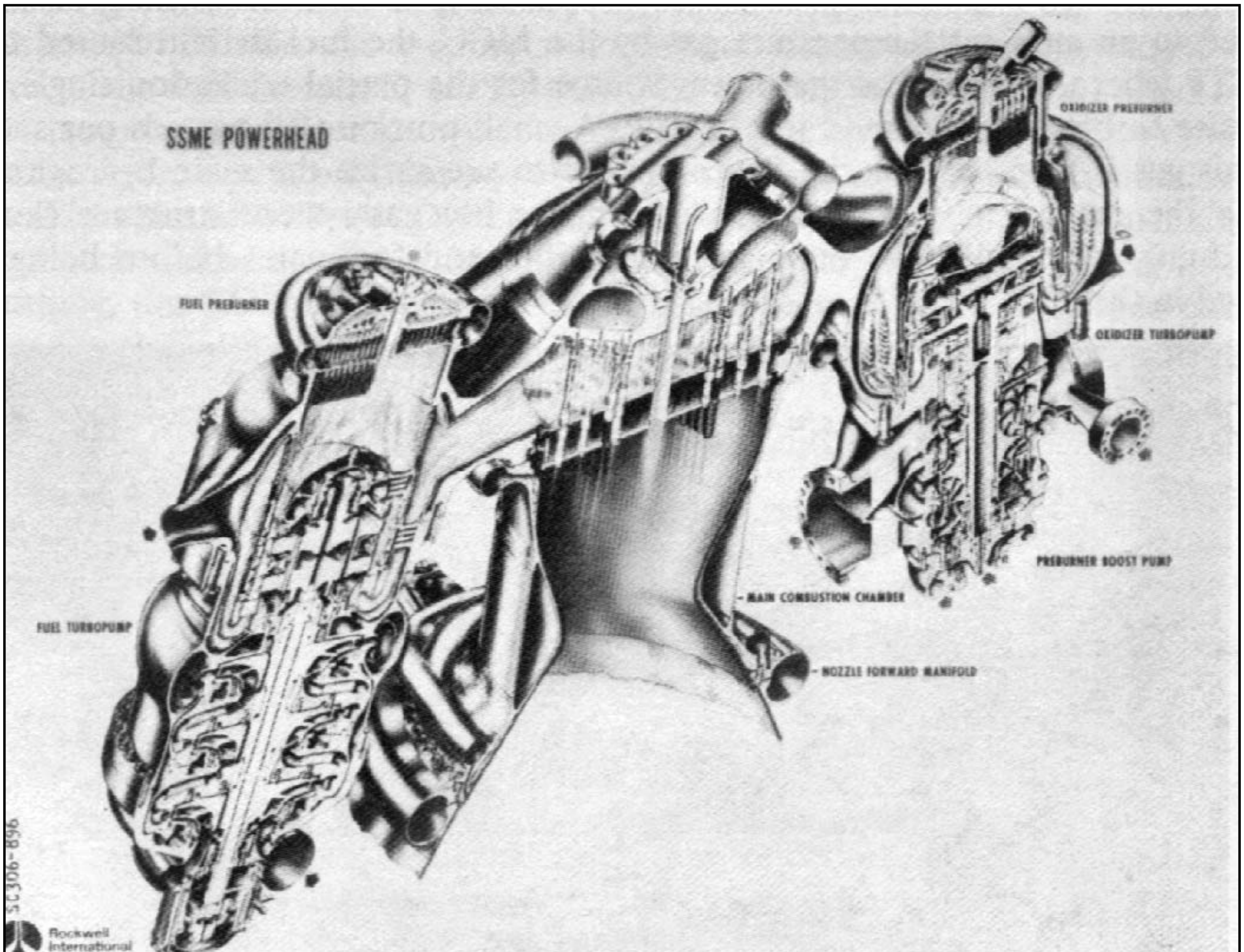

Figure 2. SSME Powerhead (Photo No. SC306-896)

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, 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

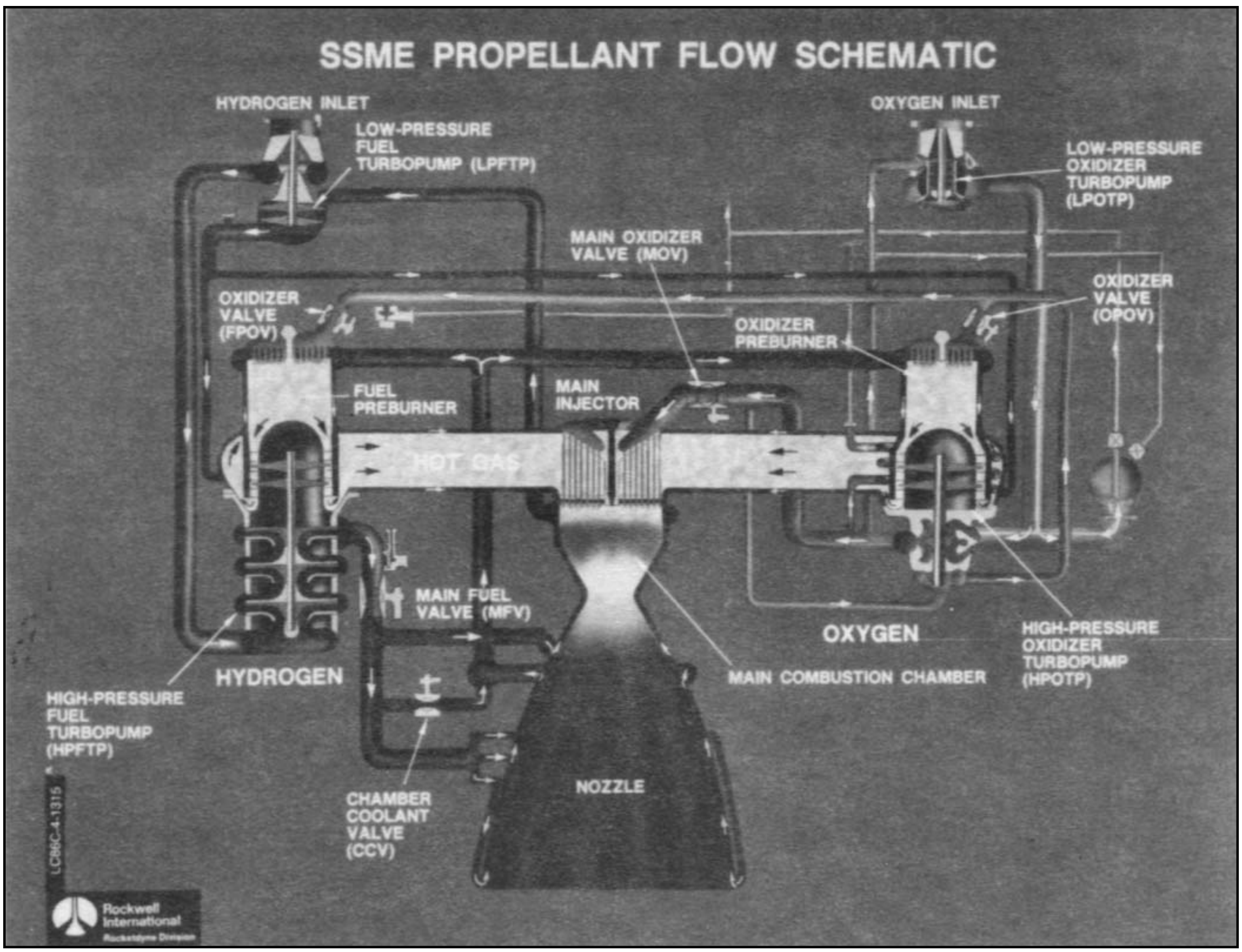

Figure 3. SSME Propellant Flow Schematic (Photo No. LC86C-4-1315)

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 on the oxidizer preburner (OP13) 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

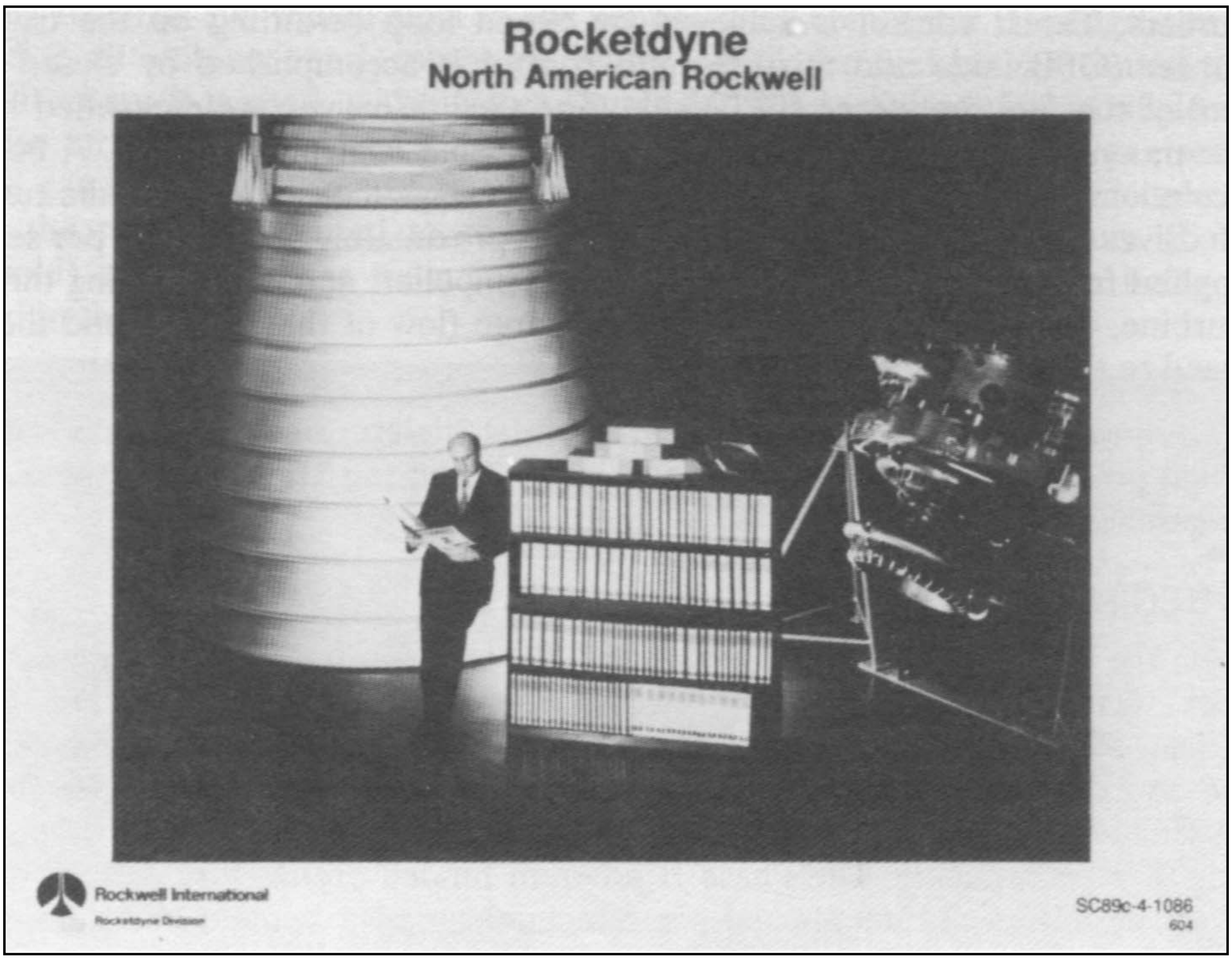

Figure 4. Rocketdyne Space Shuttle Main Engine Proposal (Photo No. SC89c-4-1086)

was required to be a bell-type nozzle to prevent the 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 sevenvolume 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 the 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 the engine redefinition until the matter of the protest was resolved. A cost-plus-fixed-fee levelof-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 the engine redefinition and certain sensitive issues relating to the

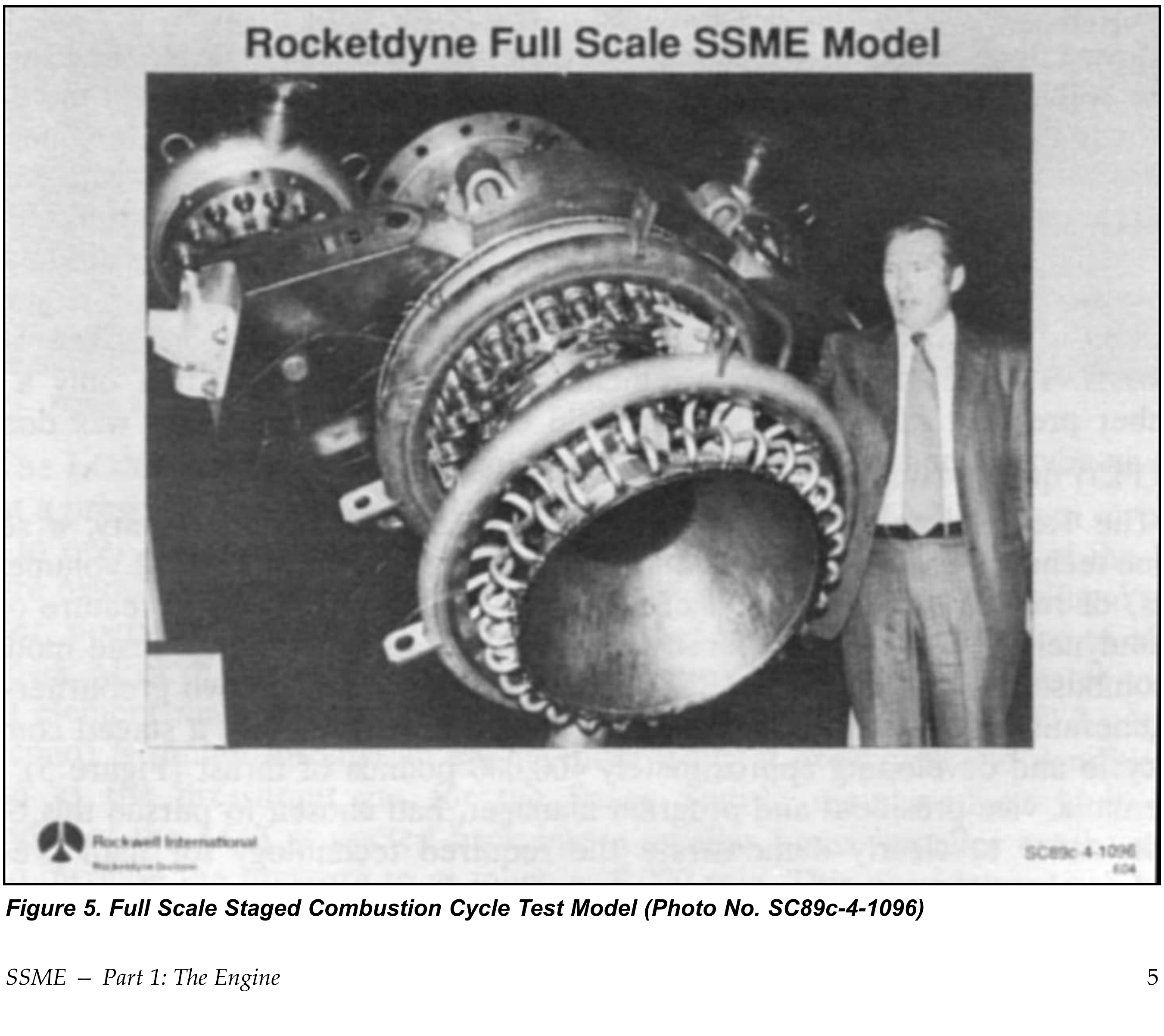

Figure 5. Full Scale Staged Combustion Cycle Test Model (Photo No. SC89c-4-1096)

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 the conversion of the 550,000 pound thrust engine to a 470,000 pound thrust engine [71 and other technical changes relative to a parallel burn Space Shuttle. A definitive cost-plusaward-fee contract was signed on August 14, 1972 [8].

THE REQUIREMENTS

The 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 the baseline release of the two major design requirements documents. The Interface Control Document (ICD) [91 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 DVSSSME-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 the completion of the DVS program (after the first flight) a total of 4,566 laboratory tests and 1,418 subsystem hot-fire tests were completed.[12]

[1] The design and development of the Space Shuttle Main Engine as described in this document were directed and funded by the NASA George C. Marshall Space Flight Center under Contract NASS-27980.

[2] R. E. Biggs, Project Engineer, SSME Systems Analysis, Rocketdyne Division, Rockwell International. 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.

[3] North American Rockwell became Rockwell International Corporation on February 16, 1973.