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Historical Evolution of the Space Shuttle Primary and Vernier Reaction Control Rocket Engine Designs

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In the early 1970's NASA embarked on the development of a reusable manned space shuttle vehicle. After early premature selections of reaction control systems using hydrogen and oxygen and then monopropellant hydrazine the hypergolic earth storable propellant combination of MON-3 (Nitrogen tetroxide with 3% NO addition) and monomethylhydrazine was baselined. Two thrust levels were required for this mission - 870 lbf (primary) and 25 lbf Vernier). The Marquardt Company (now part of Aerojet) located in Van Nuys, CA was selected to provide these engines by the prime contractor Rockwell (now Boeing) at Downey, CA. The basic engine designs proposed and tested, prior to the competitive procurement, by Marquardt have did not change significantly in the subsequent development phase. The Model R-40 engine was an evolution of the R-4D/R-23 100 and 300 lbf engines previously developed during the Apollo and Manned Orbiting Laboratory programs while the Model R-1E 25 lbf engine was a unique evolution of the R-1E engine flight qualified for the Manned Orbiting Laboratory program. This paper describes many of the unique design features of the engines which were required for 100 mission reuse and installation inside of the shuttle vehicle.

I. Introduction

THE space shuttle reaction control systems used on the forward and OMS systems pods uses earth storable MON-3 (nitrogen tetroxide with 3% NO) oxidizer and monomethylhydrazine fuel. Figure 1 shows the forward and aft module installation of these engines. The engines -thirty-eight 870 lbf primary thrusters and six 25 lbf Vernier thrusters – are installed inside of the vehicle structure/skin thus requiring the combustion camber and exit nozzle to be externally insulated to prevent overheating of the internal, shuttle structure and sensitive components. Figure 2 shows some typical (aft) primary and vernier thrusters.

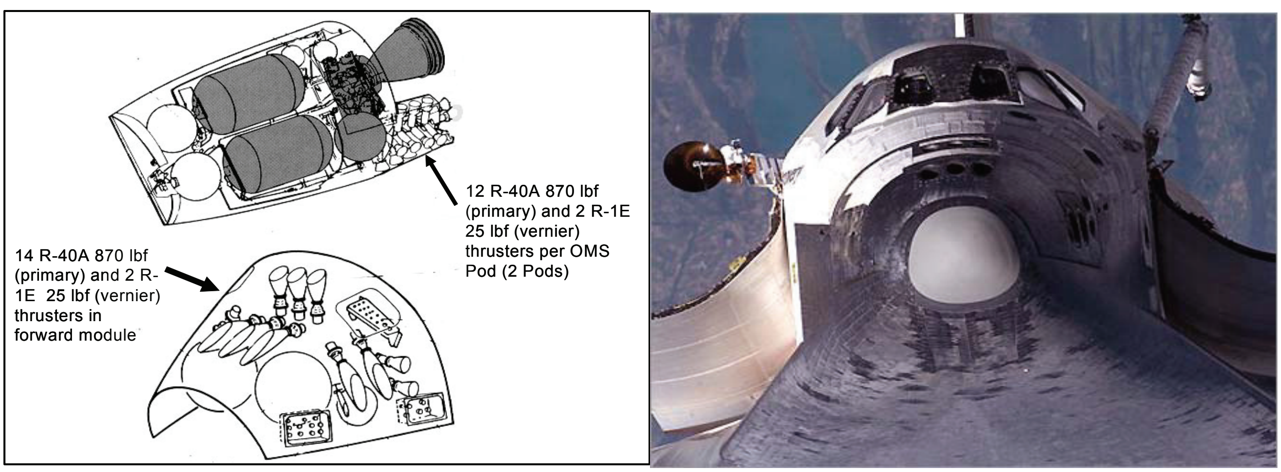

Figure 1 Primary and Vernier thruster locations

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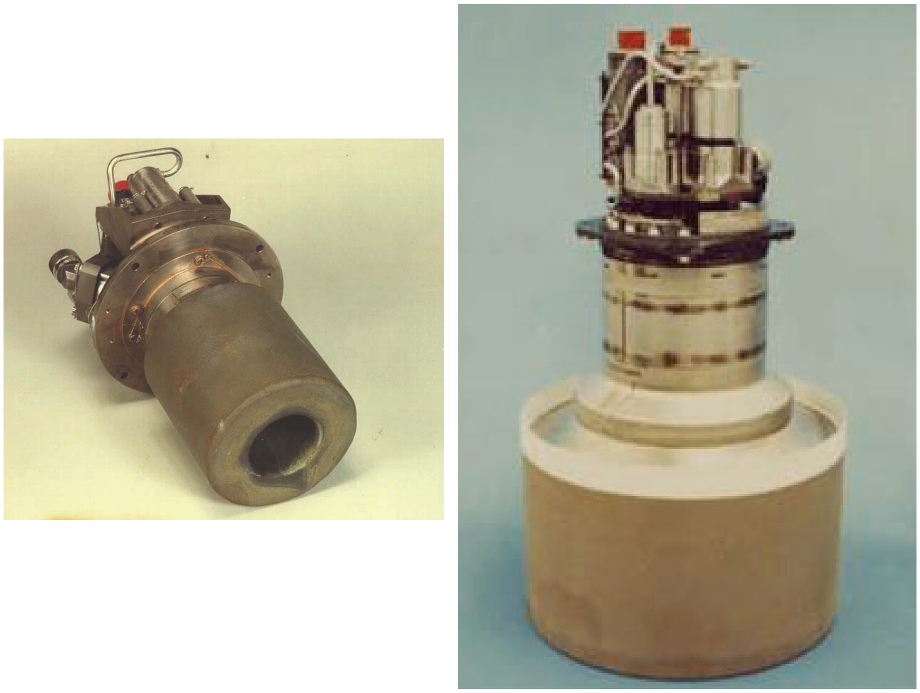

Figure 2 Aft SSRCS vernier and primary thrusters

The following sections summarize the evolution of the 870 lbf primary thruster and the 25 lbf vernier thruster that uses nitrogen tetroxide oxidizer with 3% NO (MON-3) and monomethylhydrazine fuel. This design and technical data is documented in the various AIAA technical papers that are listed in the bibliography at the end of this paper.

II. Early Development Efforts- Primary Thruster

The development of the primary and Vernier thrusters at The Marquardt Company (Marquardt) was initiated in the early 1970's after studies of the use of high and low pressure GO2/GH2 and hydrazine fueled engines were eliminated due to system and performance technical challenges. At the same time Marquardt was involved in with NASA in a small contract[1] to develop a low-power light-weight line pressure fluid actuated valve that would be capable of usage on the proposed 870 Lbf primary thruster.

Marquardt, concurrent with the valve development , initiated an internal research and development injector /chamber program designed to provide the basic design criteria necessary the space shuttle requirements which include operation at short (0.040) on times, large mixture ratio variations (1.2-2.5) and most importantly, installation inside of the vehicle (near adiabatic chamber). The original design criteria were based on historical design data from the Apollo RCS 100 lbf engine, and subsequent studies at the 300 lbf level[2].

- Basic design criteria
    - Unlike multiple doublet single row impinging injector with fuel film cooling
        - Minimum manifold volume downstream of valve seat.
        - C-103 niobium (columbium) alloy combustion chamber
    - No combustion stabilization devices
    - All welded
    - Valves thermally isolated from injector to maximize insensitivity to pulse mode operation
    - Incorporate line pressure actuated valves

The initial tests results were not encouraging. Figure 3 shows one of the early engine configurations.

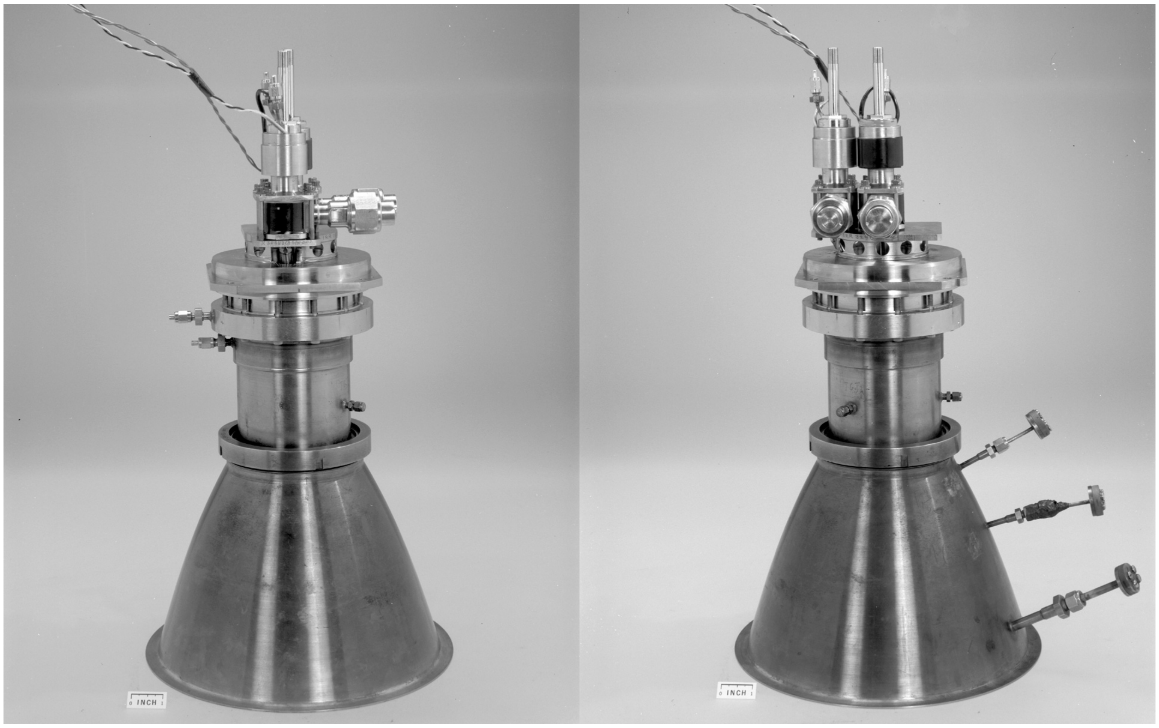

Figure 3 Early 870 lb engine configuration

The incorporation of 84 unlike doublets resulted in undesirable mixing prior to combustion and a low combustion efficiency was obtained. The injector was modified to eliminate the adjacent injector element mixing and the desired performance level consistent with the requirement to have an adiabatic (no radiation cooling) combustion chamber was achieved with a specific impulse approaching 290 seconds at an area ratio of 22:1. Subsequently the engine was performance de-rated to enable operation at the off mixture ratios resulting from a shuttle regulator failure.

During one of the test sequences after the injector low performance problem was solved unexpected 1st tangential instability was encountered which resulted in chamber failure. It had been hypothesized prior to that event that a 1st tangential instability would not occur because of the injection of the propellant at the mid-radius of the injector face which was similar to a single node of mass injection. This hypothesis was obviously false and a quick analysis program was accomplished to define and incorporate a combustion stabilization device using existing designs[3]. The ¼ wave acoustic resonator was selected due to its ease of incorporation into the existing injector. The figure 4 shows the first page and some test results from the original design based on reference 3 criteria. Note that this was before VisiCalc and Excel thus handwritten design criteria were acceptable. The most important data required for an accurate design which was not available was the combustion gas temperature that defines the speed of sound of the gas in the resonator. The initial estimates was later found to be in error based on actual measurement of the gas temperature in the resonators but did provide a basis for the initial design which proved to be adequate in damping the high frequency instability that occurred randomly during the ignition process. The engine was successfully subjected to a series of CPIA established criteria for stability stabilization (caseless detonators), however the engine was later found to be more sensitive from a 1st tangential instability mode using a “prime and purge” test where the engine was subjected to gas ingestion during the ignition process. This methodology was found to be a more effective method of verifying combustion stability.

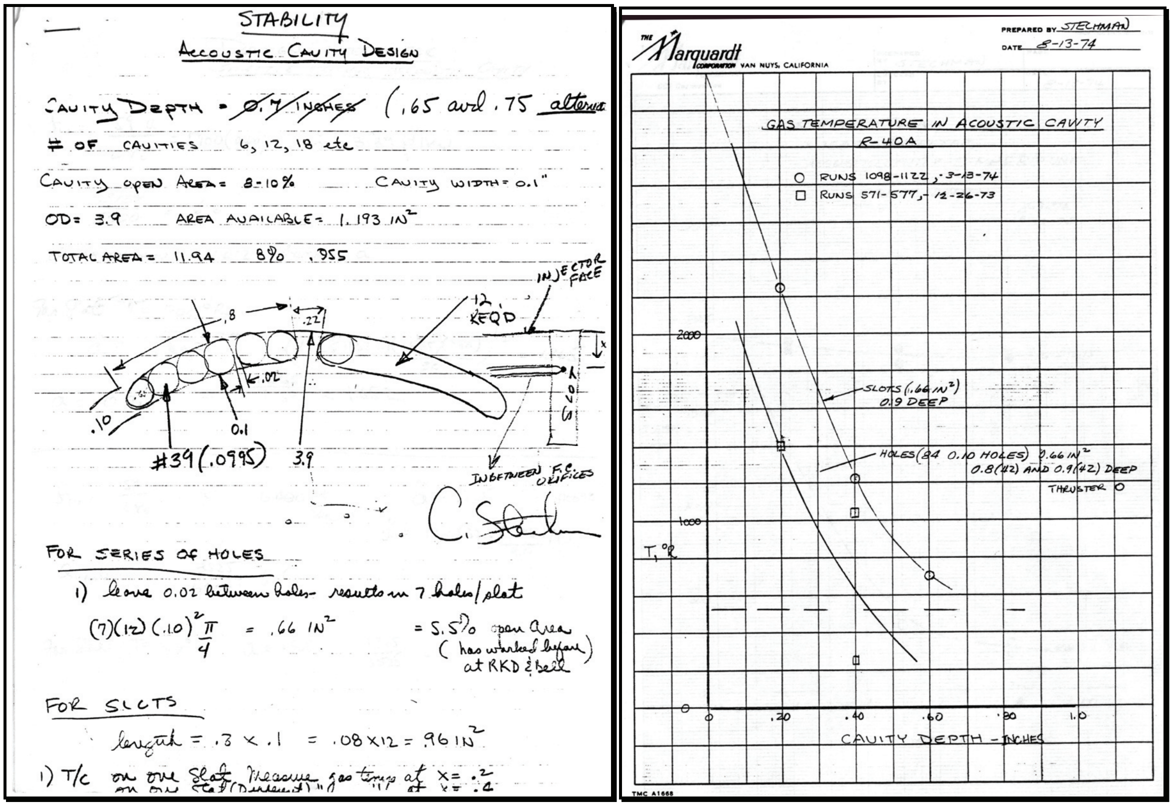

Figure 4 ¼ wave acoustic resonator design criteria for Shuttle Primary thruster (original)

The thermal and combustion efficiency characteristics of the engine were satisfactorily demonstrated during this early IR&D effort as was the development of the line pressure actuated valve developed under the NASA contract. Additional support of NASA JSC was accomplished by the production of an injector and chamber for testing at the NASA JSC test facility in Clear Lake, Texas. This injector was unique in that it contained two rows of unlike doublets rather then the single row under test at Marquardt. The performance of this injector was slightly more efficient than the single row injector design but was not considered for further test and evaluation Marquardt due to the hypothesized higher propensity to 1st tangential combustion instability.

III. The Proposal Phase- Primary Thruster[4,5,6]

The proposal phase which culminated in a technical and cost submittal to Rockwell on February 4, 1974 and a "Best and Final": on May 10, 1974 included a continuing design and test effort to insure that the best practical design was used. The forward module engines were originally designed to be identical and were retracted inside the vehicle prior to re-entry. That was the basis of the proposal; submittal. Figure 5 shows the proposal (prototype) test engine and the engine with the required chamber insulation in cartoon form. The proposed specific impulse of 291 seconds @ Ae/At=22:1 was considered to be the ultimate in performance for a rocket engine with an "adiabatic" combustion chamber. The engine had a demonstrated "safe" thermal operating margin over a mixture ratio range of les than 1.0 to more than 2.5 with 1.65 being the nominal operating point. During the development phase the engine was operated from less than 600 lbf to nearly 1300 lbf. The demonstrated operating range as obtained from the IR&D engine is shown in figure 6. In addition to the inlet pressure variation, a series of tests were successfully accomplished with gas bubble ingestion (up to 12 cubic inches helium). blocked injector holes, valve priming (and gas ingestion) and propellant temperatures up to 130°F.

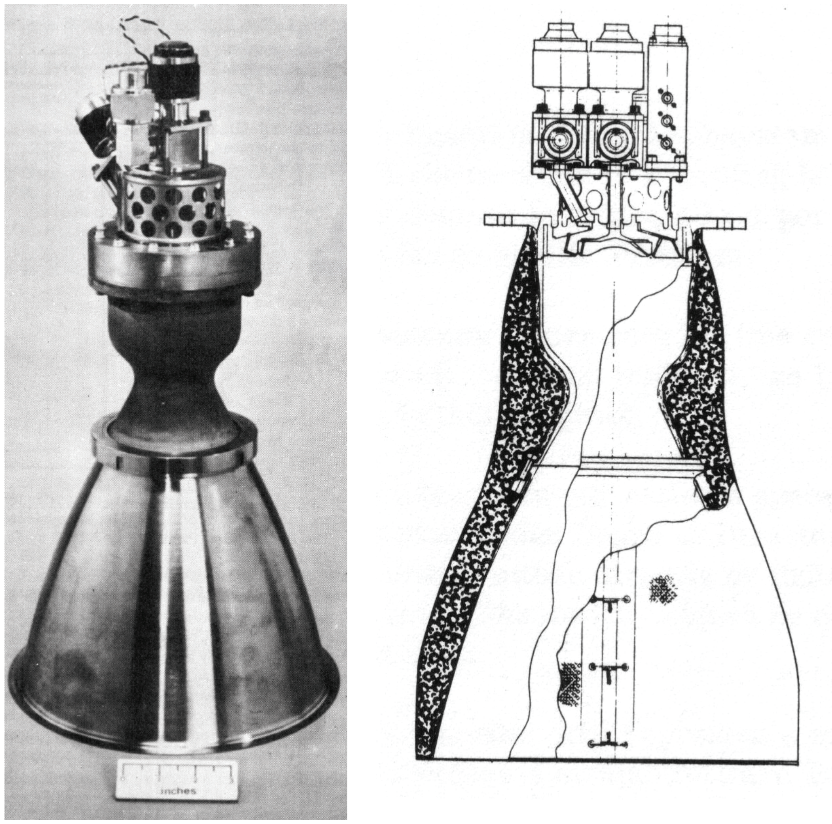

Figure 5 SSRCT prototype engine and proposal engine design with insulation

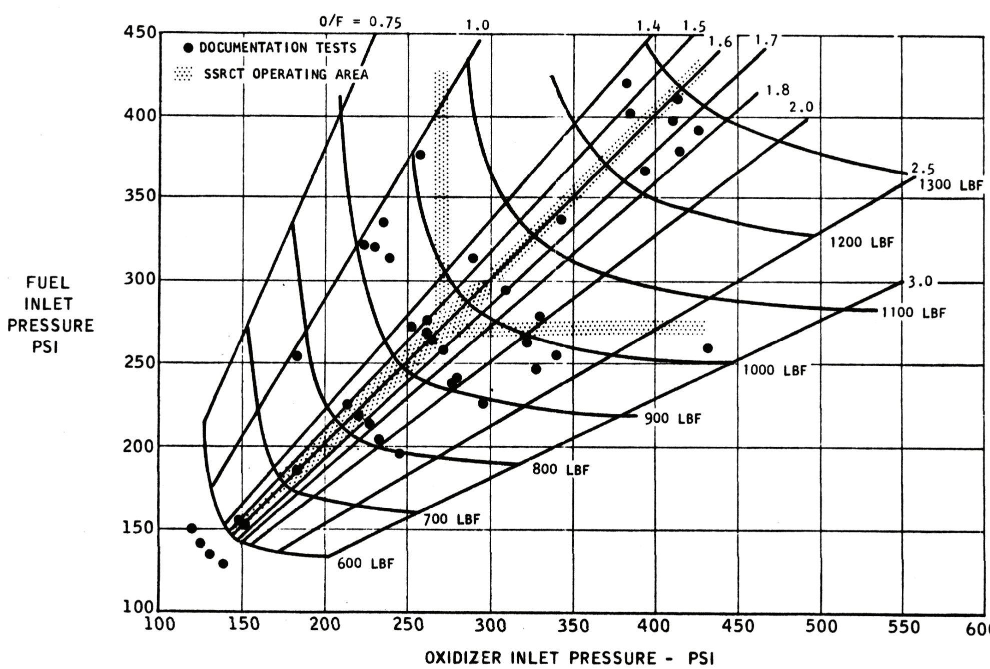

Figure 6 SSRCT demonstrated operating points

The Marquardt Company was awarded a contract by the Space Division of Rockwell International in July 1974. The contract included the technical effort to design, develop and qualify a 870 lbf engine (44 per shuttle) and the production of the first ship set of engines, The initial contract effort was in the range of $15 million[7] ($62 million in 2006 dollars). As the program progressed numerous changes were made to the engine design including the use of many different engine nozzle configurations (aft, short scarf and long scarf and variations there-in). The most significant change was the change in the design to incorporate engines whose exit nozzles were flush to the engine mold-line thus eliminating the retraction of the engines inside the vehicle during re-entry. This change probably resulted in the largest cost increase in the engine recurring price.

After contract award the primary effort was directed toward the integration of the design into the space shuttle and the construction of an engine design that was capable of multiple missions and repeated launch loads. Although the previous testing indicated that the specific impulse was in the range of 291 seconds this value seemed to erode as soon as the new test facility and thrust stand was fabricated. Eventually, after nearly a year of "tweaking" the injector design a value of 285 seconds was achieved. Further erosion of this specific impulse was encountered when more thermal margin was required for the "off-limits" mixture ratio operation. Eventually a specific impulse level of 281 seconds was baselined. The final engine configuration was subjected to formal test qualification including 100 missions of launch vibration and more than 15,000 seconds of operational firing time (150 seconds per mission).

The initial DDT&E (development and pre-qualification) engines were produced to define the manufacturing techniques unique to the insulated and scarfed nozzle as well as an attempt the manufacturer the injector in an economical way considering the fact that hundreds of holes, and slots were required to be drilled and Electric Discharged Machined. The figure 7 shows the face view of the injector with its 84 unlike impinging doublets and 42 film cooling holes and a cutaway of the injector showing the various characteristic that enable the engine to operate at most any duty cycle.

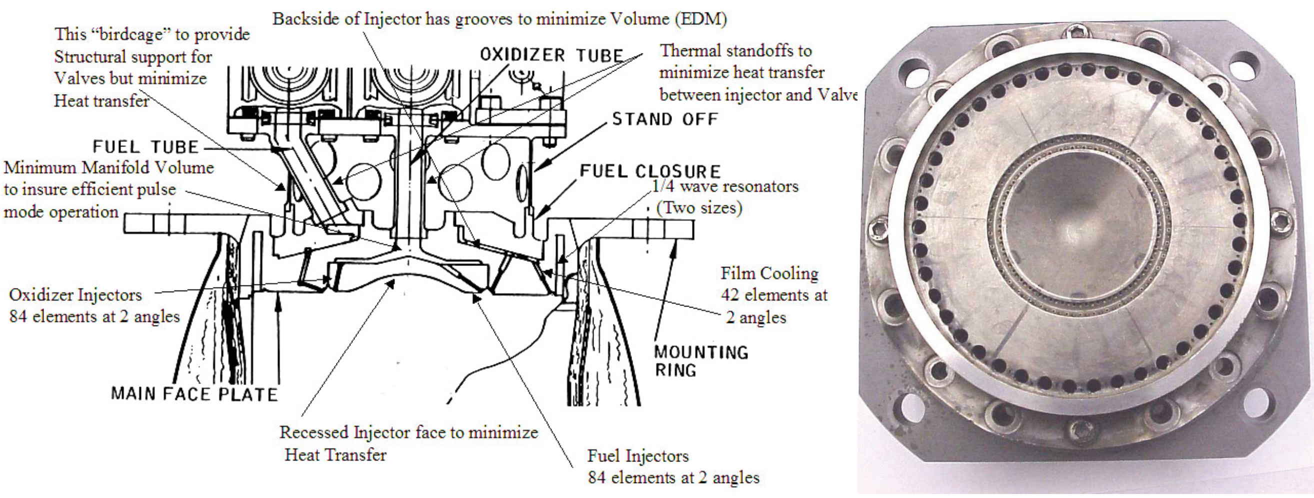

Figure 7 SSRCT injector and design characteristics[7]

Figure 8 shows one of the original engine configurations that were envisioned which included a bolt-on exit nozzle. The mass of the additional interface and the complexity of the design soon led to the realization that each engine would have to have an all an all welded chamber nozzle assembly. Figure 9 shows the final design configuration with a 60:1 exit nozzle[8]. Similar scarfed exit nozzles were used in the final shuttle design[9]. Approximately 5 years after the contract was awarded for the primary thrusters the engine design was successfully ground test qualified and successfully flight qualified on April 12, 1981 for STS-1[10]. Subsequent improvement in the engine include modification of the line pressure fluid actuated valve to improve tolerance to the ferric nitrate and other contamination that was generated internally by the shuttle due to the multiple re-use and the incorporation of a "wire wrap" internal to the insulation system to provide fail safe shuttle operation if an engine chamber failure (burn-through") should occur.

Figures 8 and 9 – R-40 Prototype with bolted exit nozzle (left) and R-40B with 60:1 all welded nozzle

IV. Vernier Thruster[11]

The 25 lbf vernier thruster has become one of the primary forces for the space shuttle. This engine model (the R-1) was the first bipropellant engine configuration developed by Marquardt in the early 1960's. The successful development and qualification of this thruster configuration fro the Manned Orbiting Laboratory program paved the way for the relatively easy adaptation to the Space Shuttle. The prior designs were radiation cooled and the incorporation of the insulated chamber did require a degree of additional development. The simple single unlike impinging doublet injector design was efficiently modified to provide the correct combustion gas temperature consistent with an insulated chamber and large mixture ratio variations that could result from a system malfunction.

The excellent operational history and performance of this engine has carried over to today. Subsequent engine flight configurations are shown in figure 10 where the external combustion chamber insulation has been removed and the interface simplified. The addition of the larger (100:1) exit nozzle results in a specific impulse of 280-285 seconds.

A complete description and development history of the Vernier thruster is contained in reference 11.

Figure 10 R-1E engine configurations (after Shuttle)

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