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ORBITAL MANEUVERING SYSTEM DESIGN EVOLUTION

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ABSTRACT

NASA and industry Shuttle configuration studies conducted in 1969 and 1970 baselined a liquid oxygen/liquid hydrogen (LO_2/LH_2) orbital maneuvering system (OMS), for a series of 15 000-pound to 25 000-pound payload class vehicles. Although the initial OMS guidelines were limited and general in nature, they established the basic architecture for future trade studies. Multiflight reuse, fail-operational/fail-safe redundancy, and a 10-year/100-mission life were required. The propellant tanks were sized for a 2000-ft/sec delta velocity, which included a 1500-ft/sec on-orbit requirement. In 1970, conceptual design studies were conducted for a 50 000-pound, 15- by 60-foot payload bay Orbiter. To minimize overall vehicle length and reduce subsystem development costs, the OMS baseline was changed to Earth-storable nitrogen tetroxide/Aerozine-50 propellants using a single Apollo lunar module (LM) descent engine. In 1971, several Shuttle configurations using external main propulsion system propellant tanks were studied. In a continuing effort to reduce Orbiter length and weight, trade-off studies were conducted using the Apollo LM ascent engine and new 5000- and 6000-pound hypergolic-propellant engines. Because of increased interest in an Earth-storable-propellant OMS system, engine technology contracts were initiated in 1972 to investigate chamber cooling, injector design, and combustion stability concepts. The results of these and system storable-propellant studies provided critical design data that later proved to significantly reduce the mainstream development effort.

In 1972, the Rockwell International Company was awarded the Orbiter Shuttle contract and they selected the McDonnell Douglas Astronautics Company (MDAC) to design and develop the OMS pods. The initial configuration contained a wedge in the OMS pod envelope for a separate reaction control system (RCS) pod. The forward end of the pod interfaced with a nose fairing that was part of the Orbiter payload bay doors. The OMS propellant tanks employed an acquisition system but did not have an RCS feed requirement; the pod skin was of conventional aluminum aircraft construction. Configuration trade studies and design evolution continued after the MDAC contract was awarded. The most significant change was integration of the RCS module into the OMS pod structure. This concept reduced cost and weight and allowed easier interconnection of the OMS and RCS systems. The integrated pod was also redesigned to allow elimination of the payload bay nose fairing, and the structural material was changed to graphite epoxy. In 1974, the Aerojet Liquid Rocket Company was selected as the OMS engine contractor. The baseline engine incorporated a platelet injector, acoustic cavities for stability, a fuel regeneratively cooled combustion chamber, a pneumatic-operated quad redundant ball valve, and an all-columbium nozzle.

The current OMS consists of two identical pods that use nitrogen tetroxide (NTO) and monomethylhydrazine (MMH) propellants to provide 1000 ft/sec of delta velocity for a payload of 65 000 pounds. Major systems are pressurant-gas storage and control, propellant storage supply and quantity measurement, and the rocket engine, which includes a bipropellant valve, an injector/thrust chamber, and a nozzle. The subsystem provides orbit insertion, circularization, and on-orbit and deorbit capability for the Shuttle Orbiter.

PRELIMINARY DESIGN CONSIDERATIONS

In-house Space Shuttle configuration studies conducted at the NASA Lyndon B. Johnson Space Center in early 1970 addressed the concerns of the lack of convergence of requirements, the high development and total program cost, the high technical risk, and the long development time associated with previous studies. A revised set of guidelines and constraints was developed and continually modified during the study. The payload weight was established as between 10 000 and 15 000 pounds; a resizing of the vehicle resulted in a 15-foot-diameter by 30-foot-long payload bay. Extensive examination was made of all current spacecraft and aircraft hardware to minimize cost of new developments and to reduce technical risks. Where no existing system could satisfy vehicle requirements, a new system development was considered assuming 1970-71 state of the art. The Orbiter was designed for a life of 100 missions, or 10 years combined storage and operations. To obtain reuse, limited refurbishment was permitted, where practical. This refurbishment was accepted at the expense of slightly higher flight operational costs. A general redundancy philosophy of fail operational/fail safe was established for the orbital maneuvering system (OMS). The liquid oxygen/liquid hydrogen (LO_2/LH_2) orbital maneuvering system had propellant tankage sized for 2000-ft/sec delta velocity, with payload quoted at a propellant loading giving 1500 ft/sec. Additional propellant for missions requiring in excess

of 1500-ft/sec delta velocity was provided by offloading payload. The on-orbit engines and the main engines were not required to operate simultaneously.

Late in 1970, studies were continued to refine the OMS for Orbiter vehicles of alternate configurations. The airframe, avionics, crew station, and propulsion subsystems were evaluated, and a preliminary design was developed for a series of 009-1 configurations. The OMS is shown in figures 1 and 2. The tanks were sized for 22 000 pounds of usable propellant. With a 5-percent allowance for residuals and ullage, the resulting volumes were 1206 cubic feet for the LH_2 tanks and 261 cubic feet for the LO_2 tank. The LO_2 tanks were located on each side of the LH_2 tank. The propellant tanks included interior baffling and structure, propellant retention devices, and gaging systems. Two gimbaled RL10A3-3 engines mounted to a thrust bulkhead integral with the fuselage structure were installed at the forward end of the Orbiter and were forward firing. The feedline to the reaction control system (RCS) used LO_2/LH_2 drawn from the OMS tanks; the RCS gas generator was connected to the OMS engine feedline downstream of the shutoff valve. The OMS tanks were pressurized by the OMS engines, by the main propellant tanks, or by the gas-generating device for the RCS system. A forward umbilical plate contained the OMS LH_2 and LO_2 fill valves; the umbilical panel was located on the left side of the Orbiter. Each OMS tank was vented through 1-inch-outside-diameter lines overboard through a redundant vent valve located on the aft fuselage structure.

Configurations ranging from long slender fuselages to short stubby fuselages, and various fuselage taper ratios, tank arrangements, and payload access schemes were investigated in the 012 series of vehicles. The payload weighed 50 000 pounds and the payload bay was 15 feet diameter by 60 feet long. All of the configurations were generated by varying the type and location of the main propellant tanks as well as the OMS. Internal volume constraints and concerns regarding the complexity of the O_2/H_2 OMS and RCS led to consideration of storable hypergolic propellants (nitrogen tetroxide (NTO)/Aerozine-50) as used in the Apollo lunar module (LM) and command and service modules. The tankage was sized for a delta velocity of 2000 ft/sec with a specific impulse (I_{sp}) of 310 seconds with a 1:6 mixture ratio and contained 31 300 pounds of usable propellant. Single LM engines were also investigated. The 013 series of Orbiter configurations (fig. 3) was generated to investigate the possibilities of forcing the vehicle center of pressure aft by means of an arrowhead type of fuse-
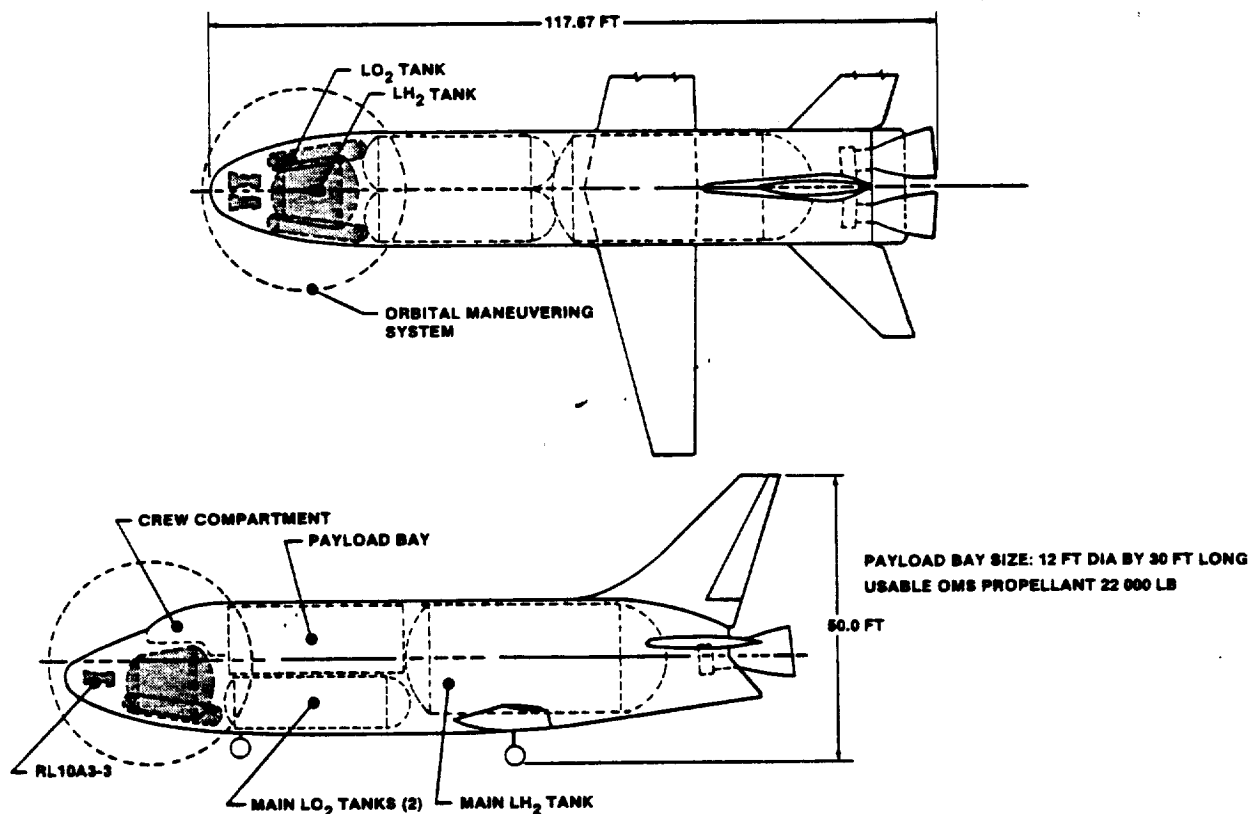

FIGURE 1.- ORBITER CONFIGURATION 009.

LEGEND

- 🔲 TURBINE
- ⬜ PREBURNER
- 🗘 PUMP
- ➤◁ LIQUID FLOW VALVE
- ➤◀ PRESS VALVE
- 🔲 PRESS REGULATOR
- ─┼─ ORIFICE
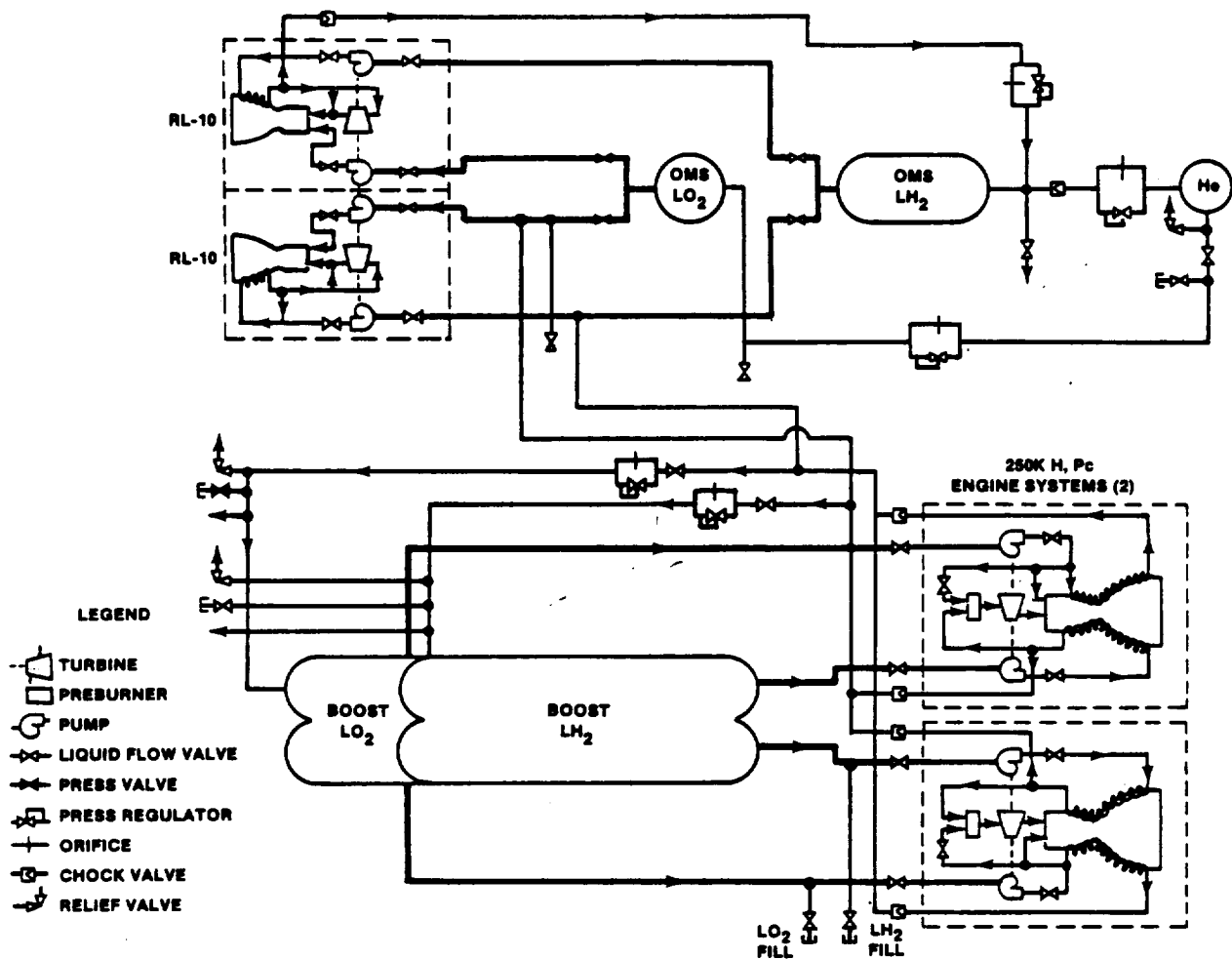- ◁⬤▷ CHOCK VALVE
- ➤⬦ RELIEF VALVE

FIGURE 2.- ORBITER FEED AND PRESSURIZATION SCHEMATIC.

lage. Four configurations were generated by varying length, diameters, and fineness ratios of the injection tanks, as well as, in the case of Orbiter configuration 013-1, the location of the crew compartment. The OMS propellants were storable, hypergolic (NTO/Aerozine-50), and sized for a delta velocity of 2000 ft/sec at an I_{sp} of 310 seconds. The OMS tankage was loaded to provide a delta velocity of 1395 ft/sec.

In 1971, a series of Orbiter configurations using external main propulsion system propellant tanks was evaluated. Results of these vehicle studies showed that smaller Shuttle Orbiters with external, expendable main engine tankage would provide a more cost-effective approach compared to the large vehicles used to generate previous baseline requirements. The Orbiter weight reduction resulting from this change enabled a significant reduction of OMS impulse requirements. This effect, together with a decision to allow scheduled OMS refurbishment, stimulated further consideration of Earth-storable propellant systems. For the smaller, lighter Orbiter with external main tanks, sufficient internal volume for an oxygen/hydrogen OMS was a significant penalty; higher density storable propellants were also attractive from this standpoint. To be consistent with Orbiter Shuttle philosophy at that time, only existing engines were considered. Earlier trade studies indicated an OMS using the LM ascent engine would provide the lightest system weight of the Earth-storable-propellant configurations. However, because of engine burn-time considerations, the Agena and the LM descent engine were also considered. The maximum engine mission-duty-cycle firing duration of the LM ascent engine (500 seconds) combined with its relatively low thrust (3500 pounds) dictated the need for the three engines to meet the 1550-second burn-time requirement associated with the 1500-ft/sec delta-velocity maneuvering capability. The LM ascent engine had demonstrated mission-duty-cycle firing durations as high as 900 seconds, but a detailed thrust chamber thermal analysis substantiated with hot
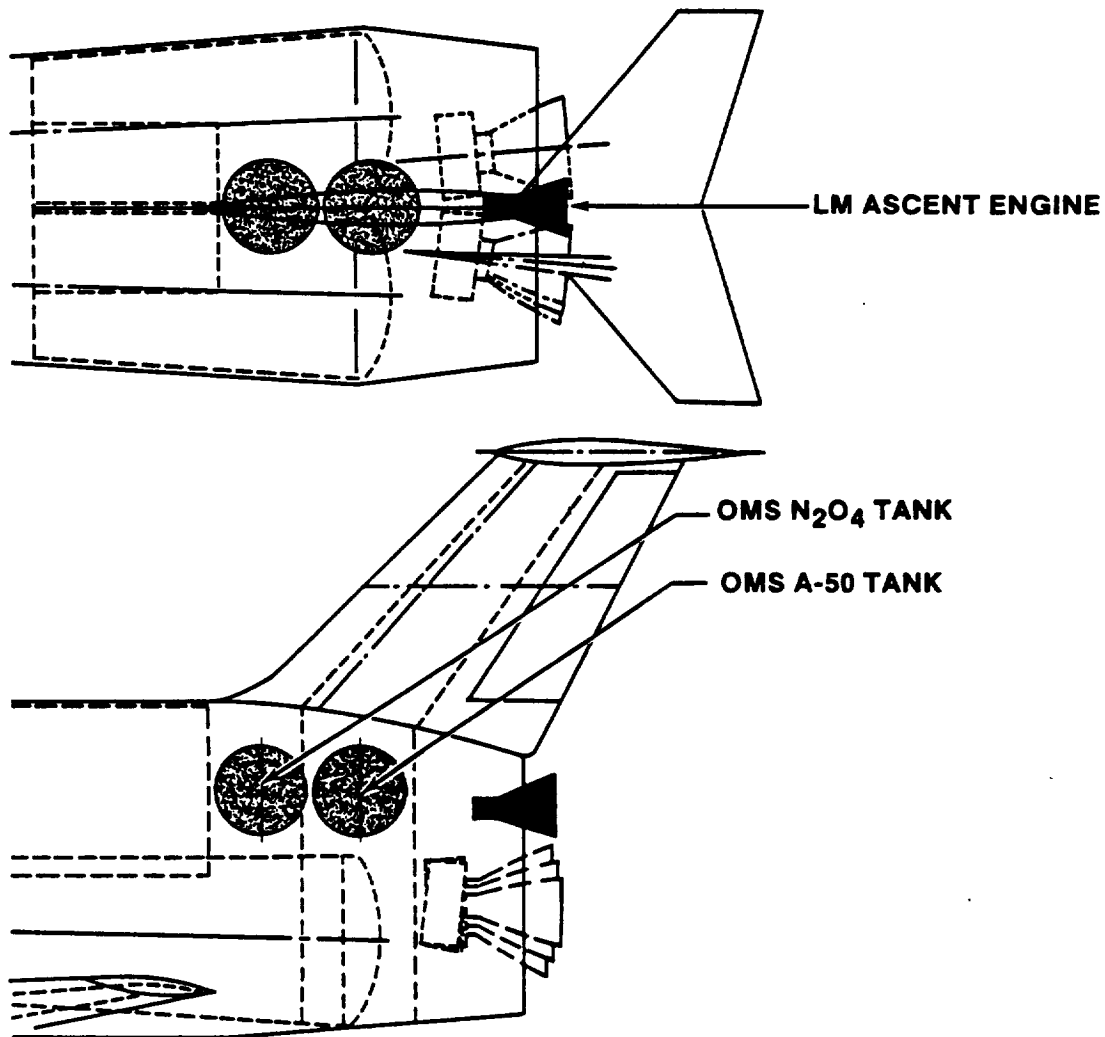
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FIGURE 3.- ORBITER CONFIGURATION 013.

firings would be required to justify using only three engines for a 2000-ft/sec delta-velocity capability.

Major perceived disadvantages of the storable-propellant system were the higher maintenance requirements resulting from the corrosive nature of the propellants and personnel exposure concerns resulting from propellant toxicity. These considerations were addressed by incorporating the OMS (and the RCS) into modular pods that could be readily removed from the Orbiter. Thus, OMS maintenance or refurbishment could be decoupled from Orbiter turnaround activities.

The 040C orbital maneuvering system (fig. 4) had two propulsion pods, each self-contained and capable of producing 500 ft/sec delta velocity for an on-orbit vehicle weight of 250 000 pounds, which included 65 000 pounds payload. The engine for each pod had the capability of using propellant from a storage system located in the payload bay and/or propellants from the other pod. Using auxiliary propellant tankage in the payload bay, the OMS would produce a total of 2500 ft/sec delta velocity. The baseline engine was a new 5000-pound-thrust, reusable, Earth-storable-propellant engine. This change was based on the following reasons.

1. The thrust-to-weight ratio (T/W) was marginal using the LM ascent engine (LMAE) and left no tolerance for vehicle weight growth.
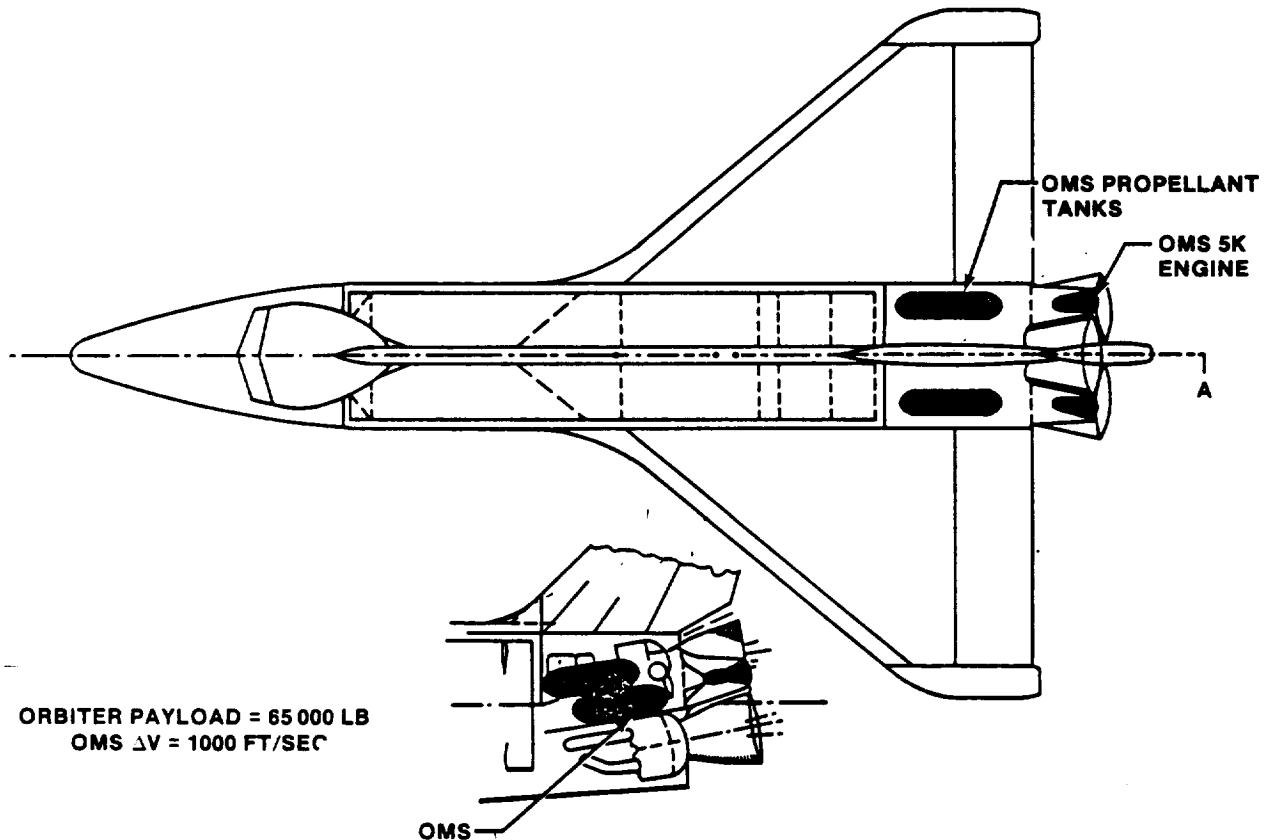
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ORBITER PAYLOAD = 65 000 LB
OMS ΔV = 1000 FT/SEC

FIGURE 4.- ORBITER 040C CONFIGURATION.

2. The burn time for the LMAE to perform the total delta-velocity requirement was 2800 seconds for each engine; this duration dictated a new ablative chamber.

3. The burn time for the LM descent engine (LMDE) to perform the total delta-velocity requirement was 995 seconds for each engine, an indication of marginal engine capability to perform an engine-out deorbit burn without chamber modifications.

4. The refurbishment cost of the LMAE and LMDE made building a new reusable engine cost effective.

5. The new OMS engine allowed for vehicle and requirement growths.

Figure 5 is a schematic of one of the two OMS pods. Each pod consisted of a pressurization system, a propellant storage and feed system, a liquid-propellant rocket engine, and the structure. The system used hypergolic propellants of NTO/Aerozine-50 and was pressure fed. The propellants were pressurized by gaseous ambient-temperature helium supplied by one tank per pod. The module was assembled, tested, checked out, installed, and removed independent of other vehicle systems. The design goal for the OMS was 15 hours life (100 missions) with maintenance-free operation for 1 year.

ENGINE CRITICAL ISSUE INVESTIGATIONS

During the post-Apollo period, several NASA technology contracts were conducted to resolve key technical issues associated with development of the orbital maneuvering engine (OME). The first of these contracts had a program objective of improving the Apollo service propulsion system (SPS) bipropellant valve. The SPS valve had two major problems: marginal life cycle characteristics (i.e., excessive leakage after cycling) and complicated assembly and repair procedures. The original scope of work consisted of design, fabrication, and test of one preprototype valve assembly. Primary
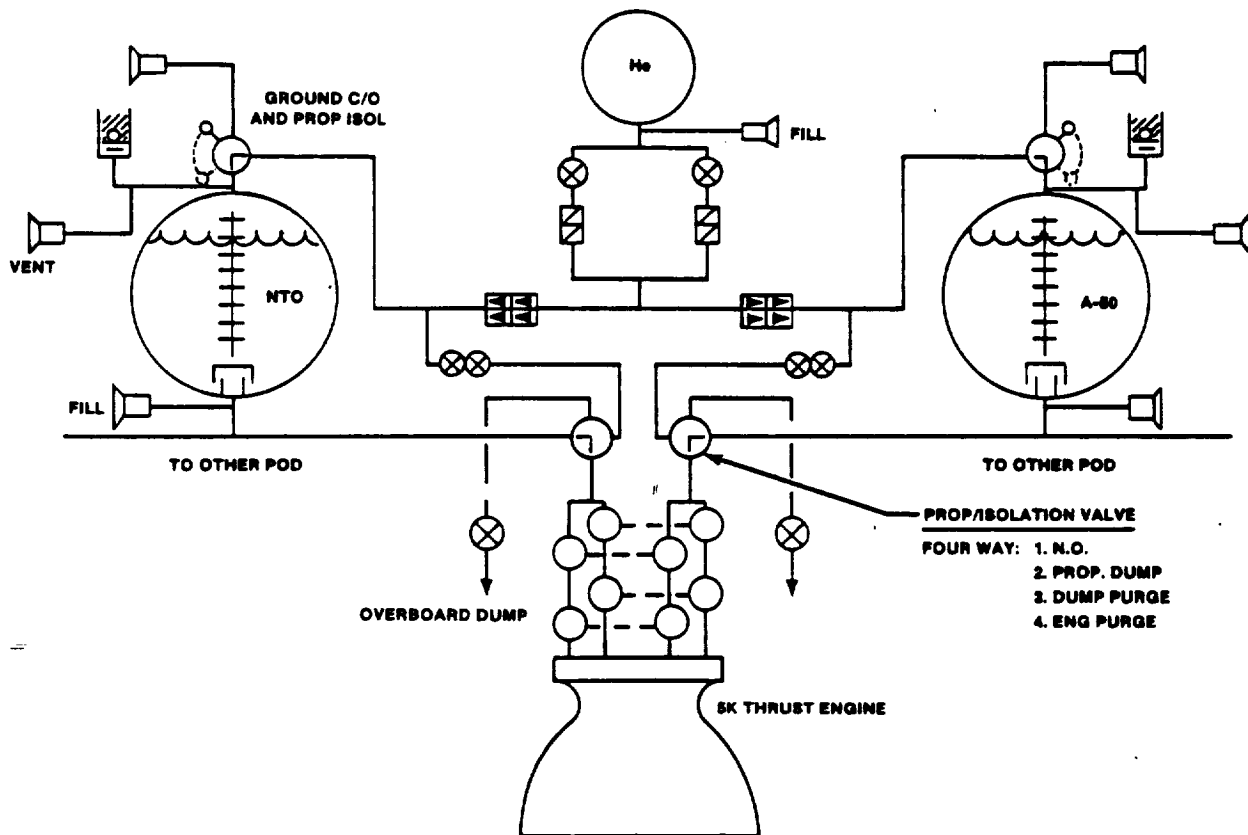
FIGURE 5.- ORBITER O4OC OMS SCHEMATIC.

emphasis was placed on improved leak rates and assembly procedures. The results were ultimately used to design the OMS valve.

Two contracts were awarded to investigate reusable thrust chamber concepts for determining the optimum configuration for OMS application. The primary objective of these programs was to evaluate and determine the feasibility of reusable thrust chambers with storable propellants. The programs provided basic engine data to potential vehicle contractors to assist them in evaluating and selecting various OME configurations. One contract was directed toward investigation of a fuel regeneratively cooled thrust chamber. The analyses and tests conducted during the program led to the general conclusion that a regeneratively cooled NTO/monomethylhydrazine (MMH) engine could provide a lightweight, stable, reusable propulsion system with high performance. The propellant and cooling method analyses led to the conclusion that a regeneratively cooled NTO/MMH engine was the preferred concept of various candidates studied based on considerations of performance, weight, development risk, cost, safety, maintainability, life, and reliability. It was concluded from results of subscale injector tests that the like-doublet injector would produce higher and more stable performance than injectors using either unlike-doublet or triplet elements. Full-scale injector test programs demonstrated that a like-doublet injector provided safe, stable operation with moderately high performance. The injector could be stabilized with an acoustic cavity having a gradual contoured entrance. Results of the thrust chamber cooling tests using electrically heated tubes and channels led to the conclusion that regenerative cooling at design and off-design conditions could be accomplished with reasonable design parameters and factors of safety. Data obtained with simple round-tube tests were used to define steady-state safety factors for chambers having channel wall construction. The engine could be started safely when the chamber was hot from a previous firing or from exhaust plume impingement from other engines. The chamber could also tolerate large continuous helium bubbles in the fuel, but the safety factor was degraded by the presence of frothlike propellant. A flight-weight thrust chamber was fabricated and safe operation was demonstrated at nominal and anticipated off-design conditions. It was shown that the thrust chamber and the injector could survive a fuel deple-

tion condition and that the engine was stable in the blowdown mode to approximately 70 psia chamber pressure. Propellant saturation did not significantly affect either the performance or the heat-transfer characteristics of the engine. Performance tests demonstrated an I_{sp} of 317 seconds with an OME-sized nozzle.

The second thrust chamber design study emphasized an insulated columbium chamber. The insulated columbium thrust chamber was tested to prove the capability both of fabrication and of the fuel film cooling to limit the chamber wall temperature to 2400° F. Several lengths of chamber were tested and a final performance of slightly more than 310 seconds I_{sp} was accomplished. Subsequent interest in the columbium chamber concept was reduced as further studies of the regeneratively cooled chamber indicated that cyclic requirements and high performance using the regeneratively cooled chamber could be achieved.

Injector designs combined with acoustic cavity concepts were investigated by several companies. Acoustic cavities, used either independently or in conjunction with baffles, had been demonstrated as an effective method of suppressing acoustic modes of combustion instability in rocket engines. In propulsion applications with requirements for both long-duration firings and reusability, cavities had an advantage over baffles because they were easier to cool and, therefore, less subject to failure from either burnout or thermal cycling. Acoustic cavities, therefore, were particularly attractive for use in the orbital maneuvering engine. Extensive tests had been made with LM ascent-engine-type hardware, an unbaffled injector, and the NTO/Aerozine-50 propellant combination. Dynamic stability was demonstrated with a relatively wide range of cavity configurations. Moreover, analytical design techniques had been developed for the design of cavities. Nevertheless, the stability of an engine with or without acoustic cavities could not be predicted analytically with confidence. As the regeneratively cooled engine concept advanced, data from extensive testing with LMAE-type hardware were available to aid in the design of cavities for the OME case, but all of these tests had been made with ambient-temperature propellants and with operating conditions near those of the LMAE. A principal concern was the effect of the high fuel temperature associated with regenerative cooling. Therefore, technology programs were begun to evaluate the effectiveness of acoustic cavities under conditions closer to those of the OME. The results from these programs clearly indicated that dynamic stability could be ensured in regeneratively cooled OMS engines through the use of acoustic cavities. Stable operation was demonstrated with a range of cavity configurations, an indication that a moderate stability margin could be obtained. Further, adequate suppression was demonstrated with doubly tuned cavity configurations that prevented occurrence of the first and third tangential modes and the first radial mode. All three of these modes were encountered when insufficient suppression was provided.

The OME platelet injector program was undertaken to evaluate a platelet face injector as a means of obtaining additional design margin and lower cost. The program was conducted in three phases. The first phase consisted of evaluating single injector elements; it involved visual flow studies, mixing experiments using propellant simulants, and hot firings to assess combustion efficiency, chamber wall compatibility, and injector face temperatures. In the second phase, subscale units producing 600 pounds thrust were used to further evaluate orifice patterns chosen on the basis of unielement testing. Full-scale testing of selected patterns at 600 pounds thrust was performed in the third — phase. Performance, heat transfer, and combustion stability were evaluated over the anticipated range of OMS operating conditions. Among these conditions were the effects of combustion stability on acoustic cavity configuration, including cavity depth, open area, and inlet contour.

INITIAL SUBSYSTEM REQUIREMENTS DEFINITION

As the Shuttle configuration became more defined in 1972, so did the OMS requirements and challenges that had to be met. An accurate evaluation and comparison of candidate OMS configurations required a consistent and well-defined set of requirements. The requirements that evolved were that the Shuttle system should, as a design objective, have a service life of 10 years and be capable of low-cost refurbishment and maintenance for 100 missions. The OMS turnaround time from landing and re-turn to launch readiness was less than 160 working hours, covering a span of 14 calendar days for any mission. The OMS had to be capable of launch readiness from a standby condition within 2 hours and hold in a standby status for 24 hours.

The Space Shuttle was planned to accomplish a wide variety of missions. The reference mission (satellite delivery/retrieval to a 100-nautical-mile circular orbit) for the Shuttle was used in conjunction with other requirements to size the OMS. The spacecraft was launched from the NASA John F. Kennedy Space Center due east and required a payload capability of 65 000 pounds with the Orbiter vehicle. The Orbiter was inserted into a 50- by 100-nautical-mile orbit; circularization at apogee required an OMS delta velocity of 90 ft/sec. The Orbiter remained on station for approximately 6 days and, in this timespan, 12 orbit maintenance burns were required to retain the 100-nautical-mile parking orbit; each burn required a 4.5-ft/sec velocity increment. Before satellite retrieval, a 32-ft/sec OMS burn was required for terminal phase initiation. The crossrange capability of the

delta wing Orbiter eliminated the need for predeorbit phasing, and, at the appropriate time, the Orbiter deorbited and returned to the launch site. The OMS also provided the 250-ft/sec deorbit burn. The anticipated on-orbit and descent OMS requirement was 372 ft/sec (neglecting the orbit maintenance burns), but a total delta-velocity capability of 1000 ft/sec was provided.

The required thrust of the OMS was principally determined by vehicle orbital maneuver and abort requirements. The orbital maneuvers performed by the OMS ranged from small velocity corrections to relatively large maneuvers such as plane changes and retrograde firings. From the standpoint of defining thrust requirements, the smaller maneuvers were more efficiently performed at a low thrust level, but low thrust increased the burn time and, consequently, the impulse required for large velocity corrections. Since the latter consideration was more significant for the Shuttle vehicle, a thrust level of approximately 6000 pounds was determined to be a reasonable nominal value with 4000 pounds as the lower limit.

The propulsion functions that the OMS had to perform set the design requirements, e.g., total impulse, thrust level, burn time, etc. However, the system designed to meet these requirements was strongly influenced by interpretation of the Shuttle reliability criterion. Two factors were of predominant importance: the number of engines and the propellant available for maneuvering and deorbit. Based on Apollo experience, the use of two OMS engines in conjunction with other component redundancy represented an acceptable level of safety and was a ground rule for system design. This rule meant that the system was designed for full mission capability after an engine failure, and, thus, definition of the thrust level and the total impulse was based on operation of a single engine for all mission functions. Systems using a common propellant supply inherently have the capability to use all the system impulse through either of the two OMS engines. However, modular-type (pod) systems, using separate propellant and pressurant supplies for each engine, would effectively be reduced to half the system impulse capability in the event of an engine failure unless each module was designed for full system capacity. Design of the separate modules with full capacity resulted in an excessive weight penalty. Since engine failures were entirely possible at times when the propellant remaining in one module was insufficient for retrograde firing, the capability to transfer propellant to the operational engine was necessary for crew survival. Hence, for the modular system, a requirement was that the OMS be designed for the capability to expend all propellant through either engine. The effect of this requirement was to dictate dry, isolated propellant interconnects between modules to achieve minimum weight. The interconnects allowed propellant supply components to be only doubly redundant and still provide the Shuttle with fail-operational/fail-safe OMS capability.

TRADE STUDIES AND DESIGN APPROACHES

After selection of the Rockwell International Company as the Space Shuttle prime contractor, the OMS was changed from an internal installation to a separate module mounted on the aft sides of the Orbiter and projecting into the fuselage, as shown in figure 6. After selection of the McDonnell Douglas Astronautics Company (MDAC) to build the pod, the configuration became shoulder mounted because of aerodynamic considerations. This configuration had the pod extending to the payload bay, and required a fairing on the payload bay doors.

Figures 7 and 8 identify major component parts for the MDAC configuration. Helium pressurant for propellant tank pressurization and for flight purge of the rocket engine assembly was contained in a single, composite aluminum bottle. From the bottle, the pressurant flow divided into two branches; each branch supplied helium to a pressurization panel containing series-parallel regulators, regulator isolation solenoid valves, quad redundant check valves, and a solenoid valve. The components were arranged so that redundant components were isolated from each other and were not subjected to identical dynamic environments. Primary and secondary relief mechanisms at the outlet of each pressurization panel protected the propellant tanks against an overpressure condition. Manual selector valves allowed independent checkout of the regulators without pressure cycling the propellant tanks. The propellant tanks were made of annealed titanium. Each contained point sensors for measuring propellant quantity (when the propellant was settled) and a refillable trap propellant acquisition assembly to assure gas-free propellant delivery to the rocket engine. A propellant line and quad redundant valve assembly on each tank provided the capability for rapid propellant dump during a launch abort. The engine was regeneratively cooled. Pneumatic two-position valves in the engine feedlines provided engine isolation and purge. Complete servicing or safing of the engine, propellant, and pressurant assemblies was accomplished from a ground servicing panel in the pod base heat shield. The pods were constructed of conventional aluminum, with emphasis on low cost and ease of field inspection and maintenance. The pod was attached to the Orbiter at four points with shear pins and threaded fasteners to provide quick mate and demate capability.

The cargo bay kit employed the same components and pressurization panels as the pods. It contained as many as six propellant tanks mounted on an all-aluminum structure for a maximum additional delta-velocity capability of 1500 ft/sec. Propellant was transferred to the pod by vehicle-mounted transfer lines, which joined the engine feedlines upstream of the engine isolation purge
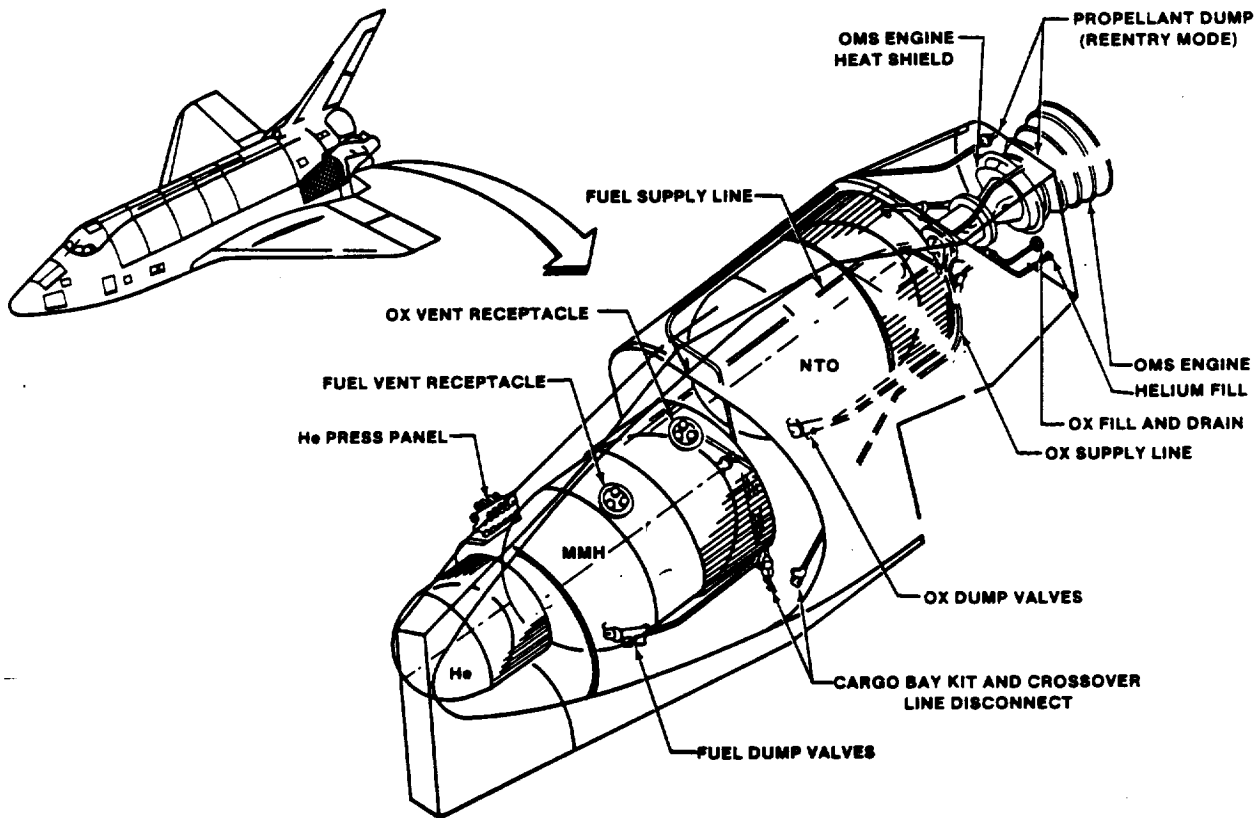
FIGURE 6.- OMS CONTRACTOR PROPOSAL BASELINE.

valves. Normally closed valves in the pod kept the vehicle-mounted transfer lines dry, unless the lines were needed for feed from cargo bay tankage or for crossfeed between pods.

Several trade studies performed and different design approaches investigated early in the program resulted in significant changes. Through design studies, it was found that both cost and weight would be saved with a common integrated structure for the OMS and the reaction control system. This change, combined with selection of RCS NTO/MMH propellants, resulted in initiation of trade studies — to investigate several degrees of integration between the two propulsion systems. The design study included common propellant tanks, clustered RCS engines for the OMS, and an interconnect system by which the RCS used OMS propellants. The interconnect system was baselined because of cost, weight, and low development risk considerations.

With the decision to interconnect the RCS and OMS propellant tanks, design requirements for the OMS acquisition and gaging system changed. A refillable trap was no longer adequate to supply propellant to the RCS. The basic requirement imposed on the OMS acquisition system was a capability to supply 1000 pounds of propellant to the RCS while maintaining a capability to restart the OMS 10 times. The design that evolved was a compartmentalized refillable trap as shown in figure 9. As a result of this change in the acquisition system, it was desirable to monitor propellant quantity in the lower compartment. Therefore, the gaging system was redesigned from point sensors to a capacitance probe. The OMS pod and later OMS/RCS pods were fabricated using aluminum and conventional aircraft construction. When it was recognized that a large weight savings could be accomplished by using a graphite epoxy skin similar to that being used for the payload bay doors, a design change was made to reduce each pod weight by 250 pounds.

Initially, the OMS and the RCS were considered as separate systems with redundancy requirements for fail operational/fail safe. After the OMS/RCS interconnect lines were incorporated, the criteria for the OMS were reduced to a fail-safe condition. This change resulted in removal of the third regulation leg of the pressurization system. It was later found to be cost and weight effective to have a common regulation source for both oxidizer and fuel. This configuration further reduced the regulated flow path from separate to coupled propellant tanks and allowed closer control of mixture
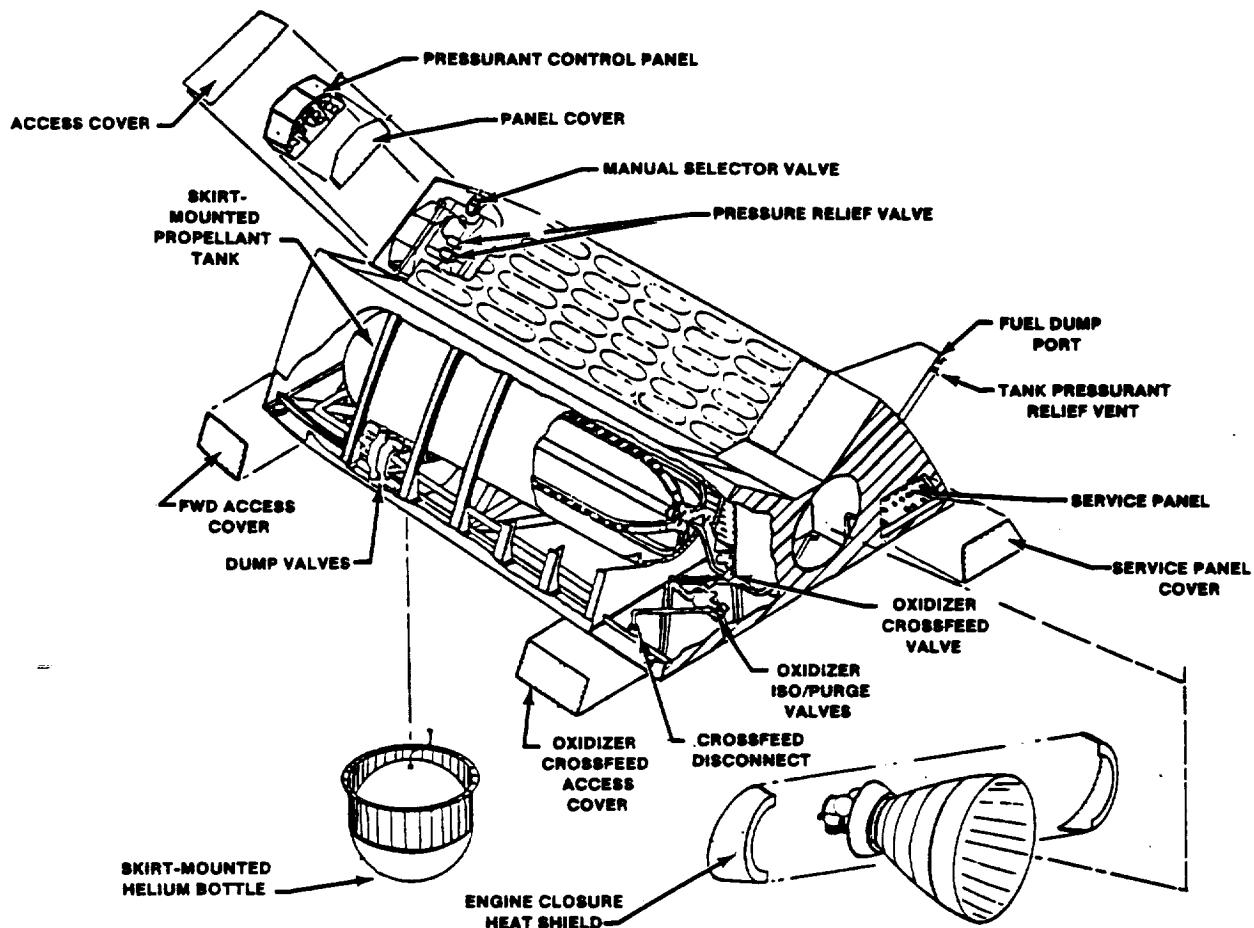
647

FIGURE 7.- OMS POD DESIGN.

ratio with a common pressure source. To assist in preventing the migration of vapor from the oxidizer tank to the fuel tank, parallel solenoid valves were installed upstream of the check valves on the oxidizer side. The initial design had a valve to purge the engines before reentry following the deorbit burn. The valve was later determined to not be required and was removed. During return to launch site (RTLS) abort, it is required to dump propellant in the OMS tanks to an acceptable level for landing. The configuration at the beginning of the program used quad redundant valves for propellant dump. This configuration was later changed to series valves with the ground rule modification that RTLS operation would not consider additional failures. Subsequent analysis indicated that the dump could be accomplished through the OMS and RCS engines by use of the interconnect, and the separate dump system was deleted. The final OMS design is shown pictorially in figure 10 and schematically in figure 11.

The OMS engine is illustrated schematically in figure 12. The design drivers were life, envelope, applied environment, specific impulse, combustion stability, reusability, and propellant inlet feed pressure. A single overriding consideration was that no single-point failure would result in a safety hazard to ground or flight personnel. The basic design concepts were direct applications of predevelopment technology activities. The key requirement was long life and this influenced design of the acoustically stabilized, flat-face, photo-etched injector; the regeneratively cooled, slotted combustion chamber; and the redundant ball valve. The engine nozzle area ratio was 55:1, which efficiently used the allocated envelope length but only 44 inches of the 50-inch envelope diameter. The resulting configuration reduced weight because the nozzle skirt was smaller. The engine used a fuel inlet torus-mounted gimbal ring which was selected on the basis of cost, weight, stiffness, and maintainability considerations. The side-mounted, series redundant propellant valve reduced feedline length, and engine length, and allowed for shock mounting to modulate engine input. Filters were contained at the inlets of the feedlines for ease of access. Static leakage was controlled by
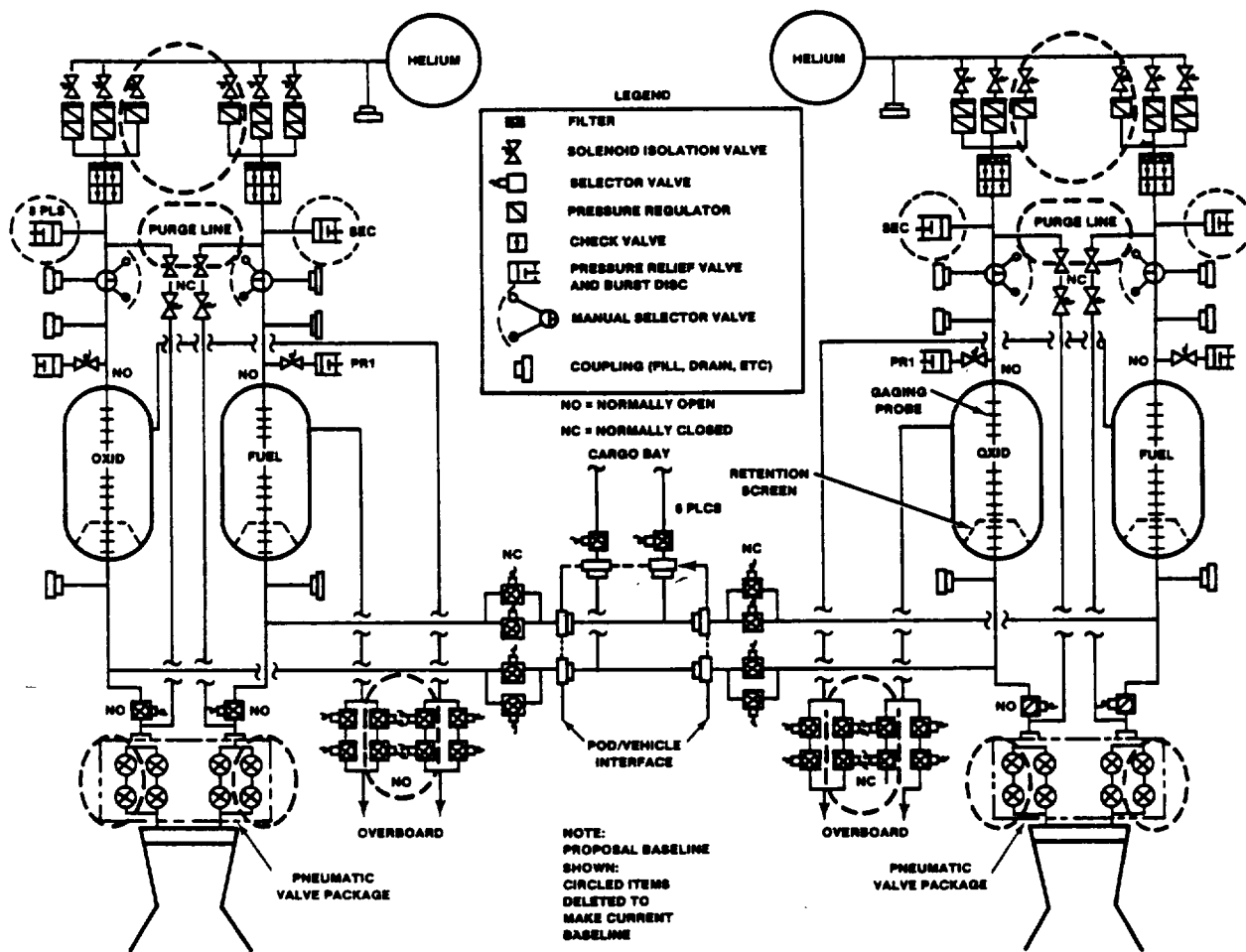
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FIGURE 8.- OMS CONFIGURATION CHANGES.

redundant seals at all flanges and interconnecting fluid joints. Inlet line routing to the engine was through propellant lines located in the same plane as the gimbal ring and chamber throat. Maintenance, installation, and servicing were simplified by designing the engine as a line replaceable unit (LRU) and mounting it on the inlet manifold-mounted gimbal ring. Large tolerance stackups were also eliminated and thrust alinement was simplified.

The development injectors were designed to be mechanically joined to a fuel manifold acoustic cavity assembly, which simulated the forward end of the regeneratively cooled thrust chamber. This design allowed stability assessment of the flight configuration to be accomplished with uncooled workhorse thrust chambers. Removal of the injector flange enabled welding of the same injectors directly to the regeneratively cooled chamber. The X-doublet injector developed under a technology contract was later replaced by a like-on-like pattern composed of eight photo-etched platelets but in all other respects identical. The basic reason for the change was the inability to reproduce the X-doublet injector. This discrepancy became obvious when the first like-on-like injector was tested and found to have impaired stability characteristics. An exhaustive inspection of both injectors disclosed a slight variation in platelet flatness in the technology injector that resulted in minor stream variations. Because of the subtle nature of differences between the two injectors, it was almost impossible to define all of the effects. As a consequence, the like-on-like pattern (fig. 13) was selected for the baseline engine.

The combustion chamber, shown in figure 14, was regeneratively cooled by fuel flowing in a single pass through nontubular coolant channels. Its design was based on regeneratively cooled thrust chambers fabricated on technology contracts. The design configuration was similar to that of the Space Shuttle main engine chamber. The chamber was constructed from a stainless steel liner with
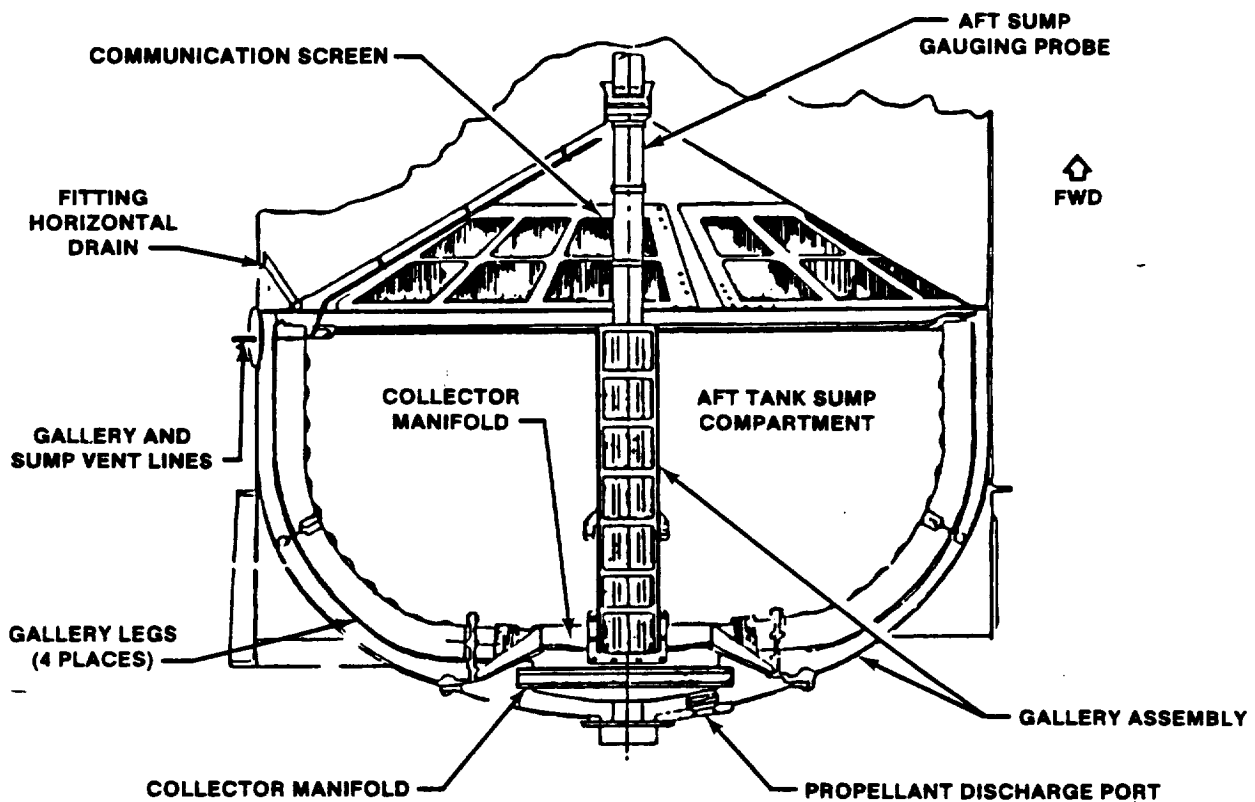
FIGURE 9.- PROPELLANT ACQUISITION SYSTEM.

rectangular coolant channels enclosed with an electroformed nickel shell. Because the electroformed shell and the liner were independently capable of withstanding the structural loads, the criticality of the interface bond was removed. The upper chamber was cylindrical with a contraction ratio of 1:9. The distance from the injection plane to the throat is 15.9 inches. The thrust chamber assembly extends to a 6:1 area ratio, where the nozzle extension was attached by means of a bolted joint. The chamber contained 120 longitudinal, milled, rectangular-shaped passages. The cooling channels had constant widths but varied in depth to provide an optimum configuration for cooling effectiveness, chamber life, and engine performance. The combustion chamber was composed of three main parts: a stainless steel liner, an electroformed nickel shell, and an aft flange and fuel inlet manifold assembly. The liner was fabricated from 304L stainless steel, selected because of its adequate strength properties at operating temperature, chemical compatibility with the combustion environment, and superior machining and electron beam (EB) welding characteristics. The chamber structural design was based on life cycle requirements; mechanical loads such as pressure, thrust, and aerodynamic loading on the nozzle; fabricability; and weight. The main structural consideration for the stainless steel liner was thermal fatigue due to temperature gradient and channel pressure. The nickel outer shell thickness was governed principally by the moment along the length of the chamber resulting from aerodynamic loading on the nozzle.

The nozzle extension (fig. 15) was radiation cooled and was constructed entirely of columbium commensurate with experience gained in the Apollo Program. The nozzle extended from the regeneratively cooled interface to an area ratio of 55:1. The aft section was reinforced by three external stiffeners to provide for the large external pressure loads encountered by the nozzle during ascent. The entire surface of the nozzle was constructed of three parts: a flange, a forward section, and an aft section. The mounting flange consisted of a bolt ring made from a forging and a tapered section which could either be made from a forging or spun. This tapered region provided a transition from the 0.100-inch-thick flange to the 0.050-inch-thick forward nozzle section. The forward and aft sections were made from two panels each; the aft section was 0.030 inch thick. The panels were butt-welded to form two cones; the cones were welded circumferentially to each other at the flange region. This assembly was bulge-formed to the final configuration, and the stiffening

OMS ENGINE CHARACTERISTICS	
THRUST	6000 LB (VACUUM)
SPECIFIC IMPULSE	313.2 SEC
CHAMBER PRESSURE	125 PSIA
MIXTURE RATIO	1.65
GIMBAL CAPABILITY	+6° PITCH +7° YAW

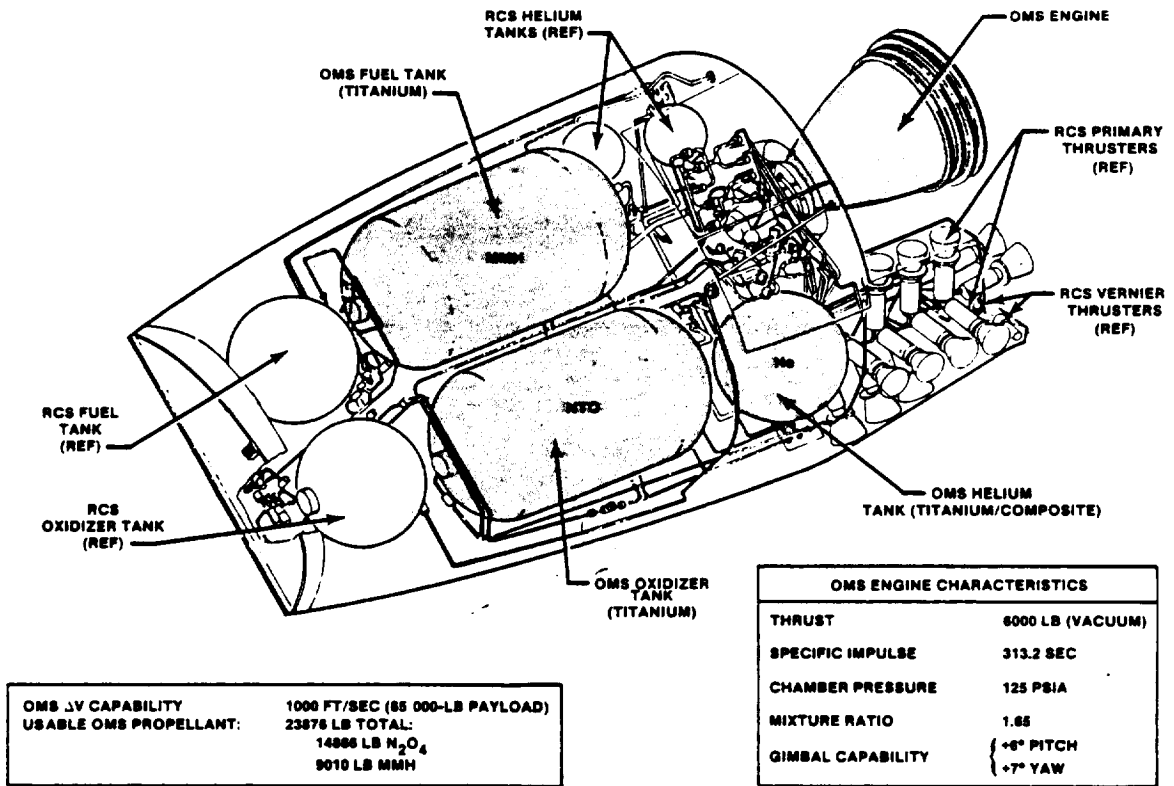| OMS ΔV CAPABILITY	1000 FT/SEC (65 000-LB PAYLOAD)
USABLE OMS PROPELLANT:	23876 LB TOTAL:
	14866 LB N_2O_4
	9010 LB MMH

FIGURE 10.- BASELINE ORBITAL MANEUVERING SYSTEM.
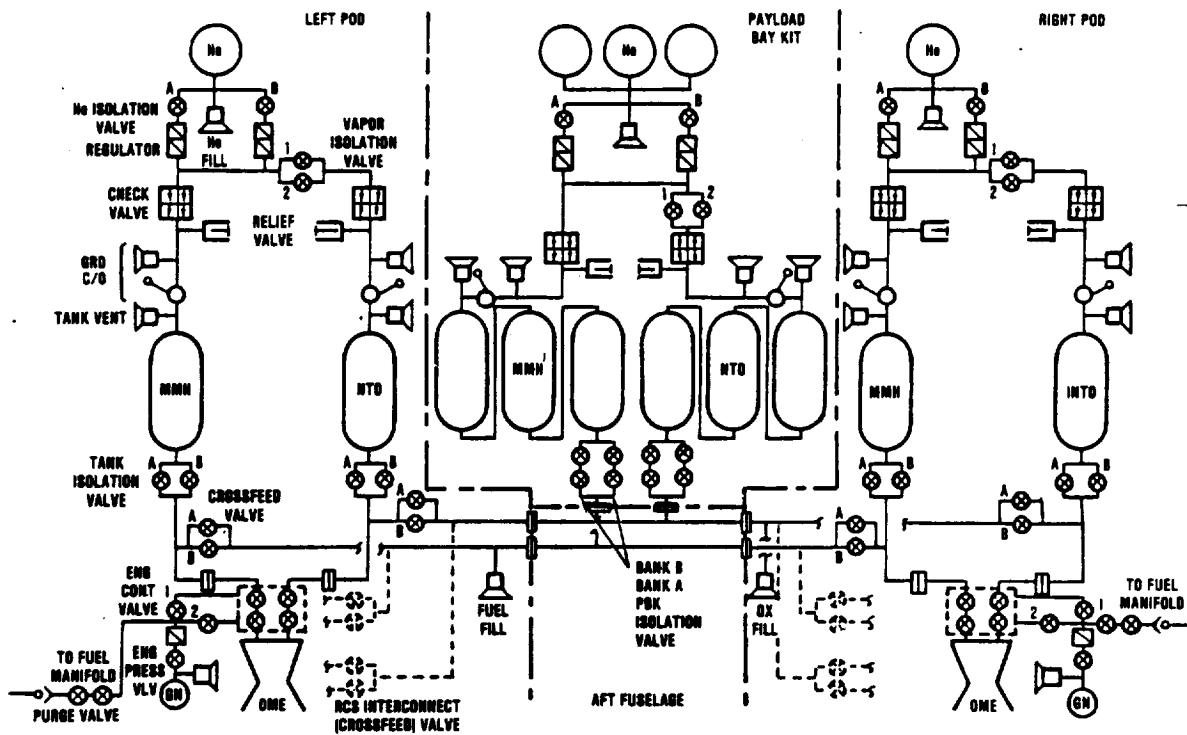
FIGURE 11.- BASELINE OMS SCHEMATIC.
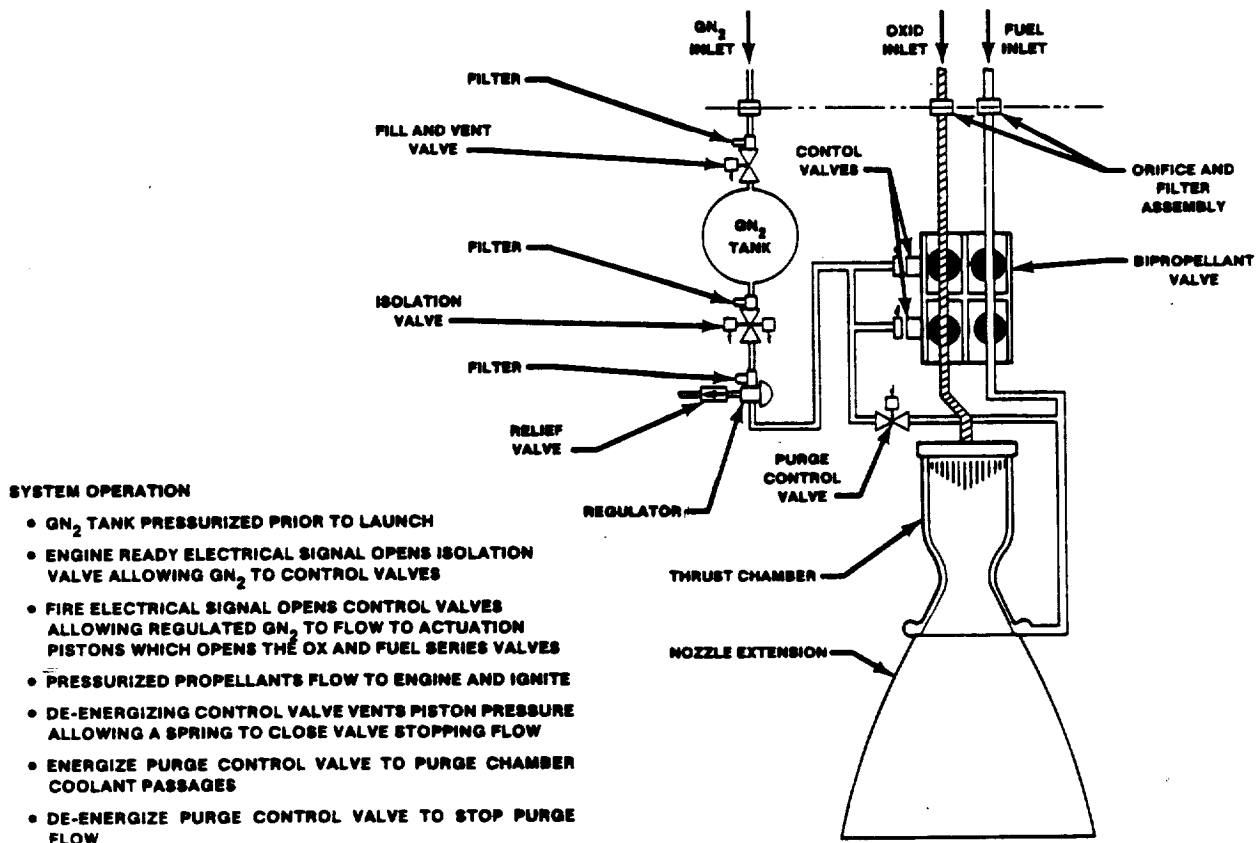
FIGURE 12.- OMS ENGINE SCHEMATIC.

SYSTEM OPERATION

- GN₂ TANK PRESSURIZED PRIOR TO LAUNCH

- ENGINE READY ELECTRICAL SIGNAL OPENS ISOLATION
  VALVE ALLOWING GN₂ TO CONTROL VALVES

- FIRE ELECTRICAL SIGNAL OPENS CONTROL VALVES
  ALLOWING REGULATED GN₂ TO FLOW TO ACTUATION
  PISTONS WHICH OPENS THE OX AND FUEL SERIES VALVES

- PRESSURIZED PROPELLANTS FLOW TO ENGINE AND IGNITE

- DE-ENERGIZING CONTROL VALVE VENTS PISTON PRESSURE
  ALLOWING A SPRING TO CLOSE VALVE STOPPING FLOW

- ENERGIZE PURGE CONTROL VALVE TO PURGE CHAMBER
  COOLANT PASSAGES

- DE-ENERGIZE PURGE CONTROL VALVE TO STOP PURGE
 FLOW

rings were attached by welding. The oxidation barrier diffusion operation was done after all machining was completed. The nozzle extension also underwent some minor design changes as a result of the development program. As previously mentioned, the original design included three stiffening rings located approximately at the midpoint of the nozzle. Changes in both the magnitude and the location of aerodynamic loading combined with changes to the expected aerodynamic noise level dictated the current design with a single flange at the nozzle exit. With the exception of minor changes in shell thickness and in number of circumferential welds, the nozzle and the manufacturing process are identical to those proposed.

A basic design consideration for the bipropellant valve (figs. 16 and 17) was the elimination or the reduction of problems that occurred with the Apollo quad redundant valves. The selection was accomplished first by subjective trade-offs, to narrow the number of candidates, and then by quantitative comparisons to evaluate major candidates. A basic design premise was requirement of a modular-type valve as a cost-effective approach, not only for maintenance and servicing but also for fabrication and test. Primary reasons for this selection were the potential for low-risk attainment of operating requirements and credible, substantiated costs. One fuel and one oxidizer valve were mechanically linked, with each of four linked pairs driven through a rack and pinion assembly by a piston actuator. Opening force was produced by pneumatic pressure, which was controlled by a close-coupled, three-way solenoid valve located at each actuator. Closing force was provided by nested, counterwound, helical compression springs. The three-way solenoid valves were included in the actuator module of the valve assembly to reduce fluid volumes and actuation delay times. Actuation gas control was achieved by a pneumatic pack. The pneumatic pack included a gas storage tank, a two-way solenoid valve, a pressure regulator, a pressure relief valve, related filters, and access ports for servicing and instrumentation. In addition to selection of the basic subcomponent parts of the valve assembly, there were options regarding the physical arrangement of these subcomponents relative to the engine. The primary aspect of the assembly design was modularization. The capability to preassemble subassemblies and install and remove them without disturbing other parts of the valve was the key to a cost-effective program. This capability would reduce fabrication problems, reduce development time, and allow servicing and maintenance goals to be met with lower inventory. The quad
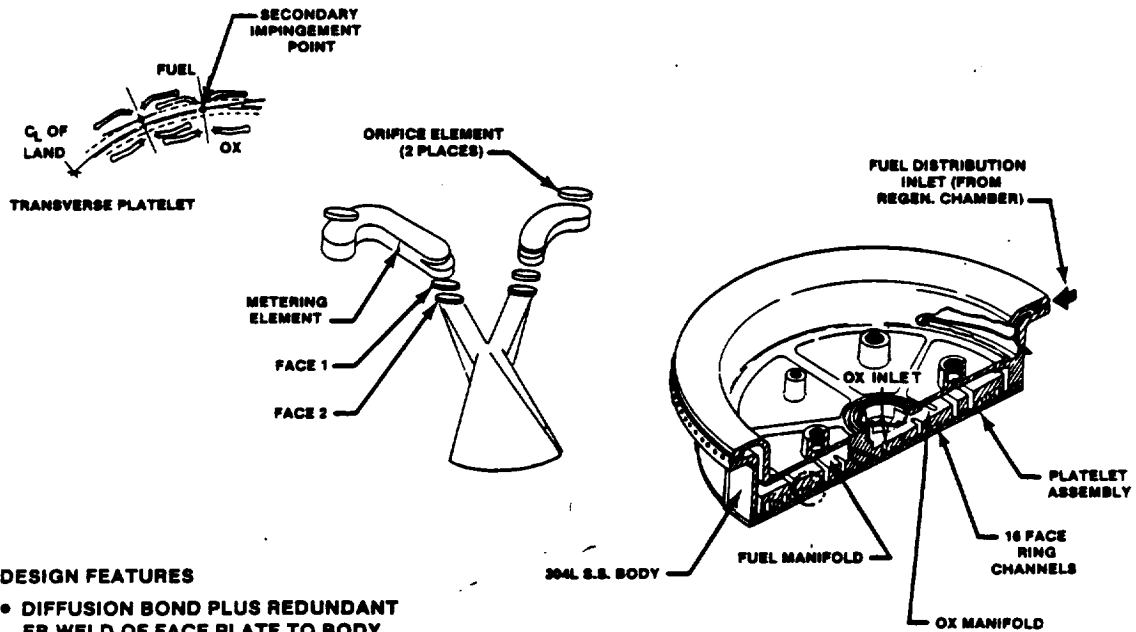
DESIGN FEATURES

- DIFFUSION BOND PLUS REDUNDANT EB WELD OF FACE PLATE TO BODY
- PHOTO-ETCHED INJECTOR PATTERN
- STABILITY ACHIEVED WITH ACOUSTIC RESONATORS

FIGURE 13.- OMS ENGINE INJECTOR.
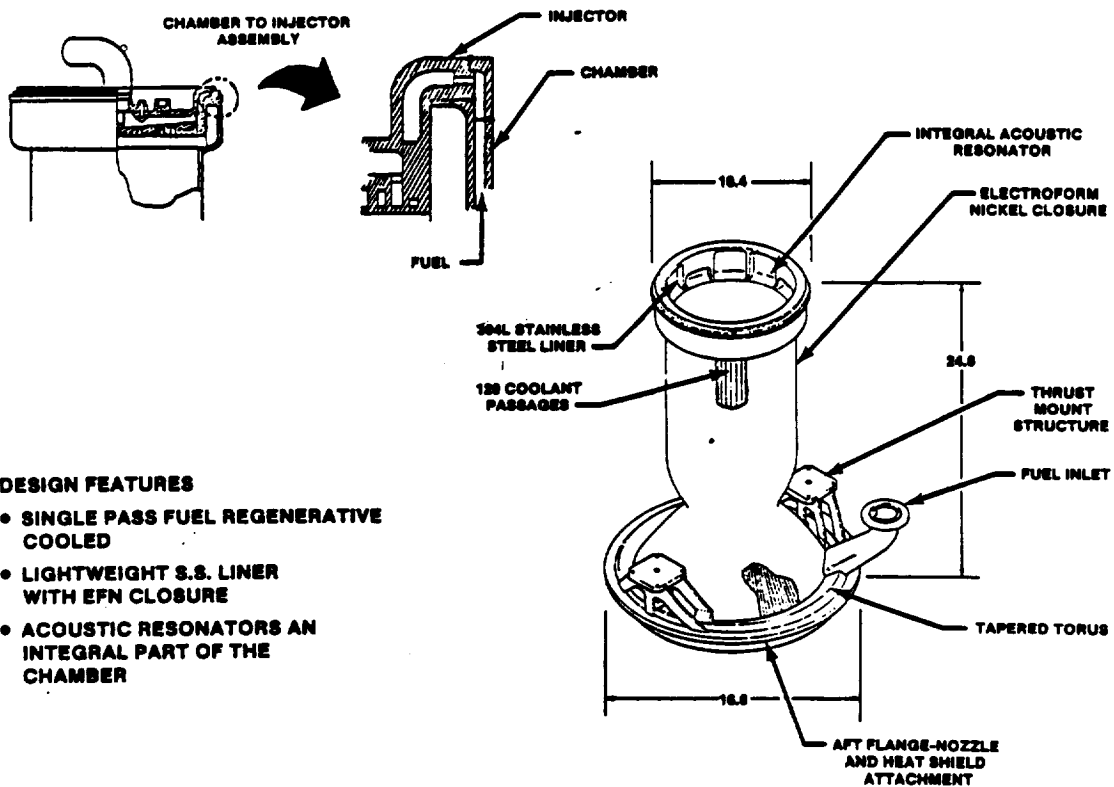
DESIGN FEATURES

- SINGLE PASS FUEL REGENERATIVE COOLED
- LIGHTWEIGHT S.S. LINER WITH EFN CLOSURE
- ACOUSTIC RESONATORS AN INTEGRAL PART OF THE CHAMBER
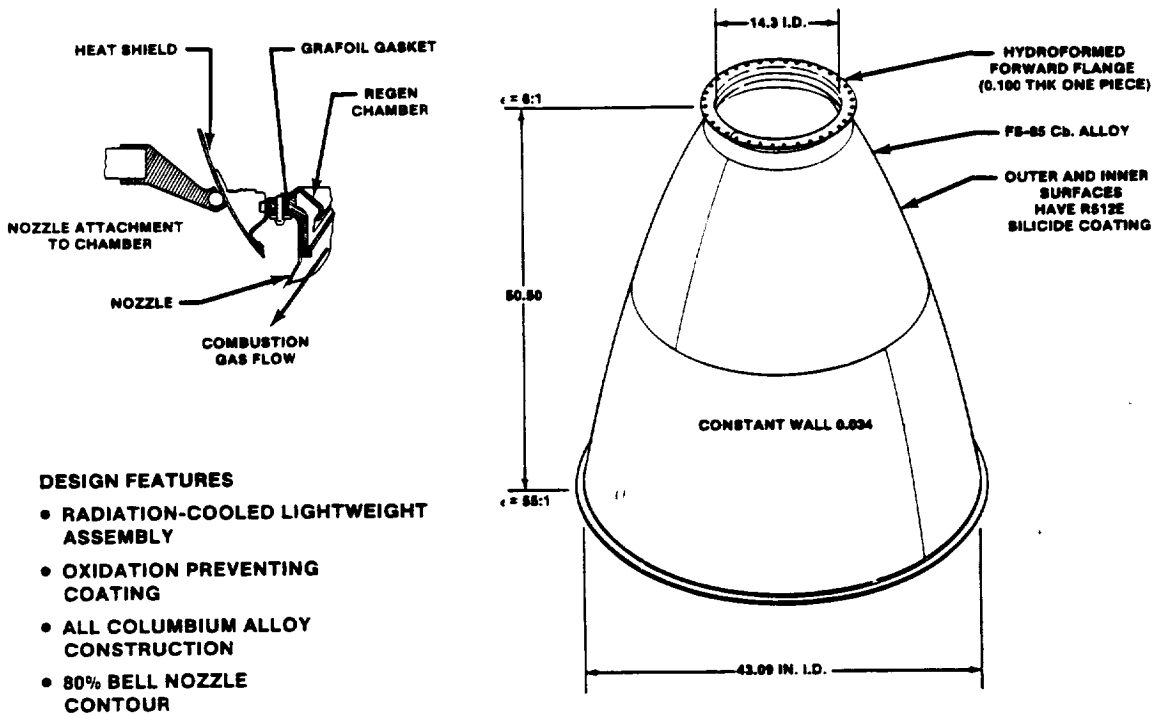
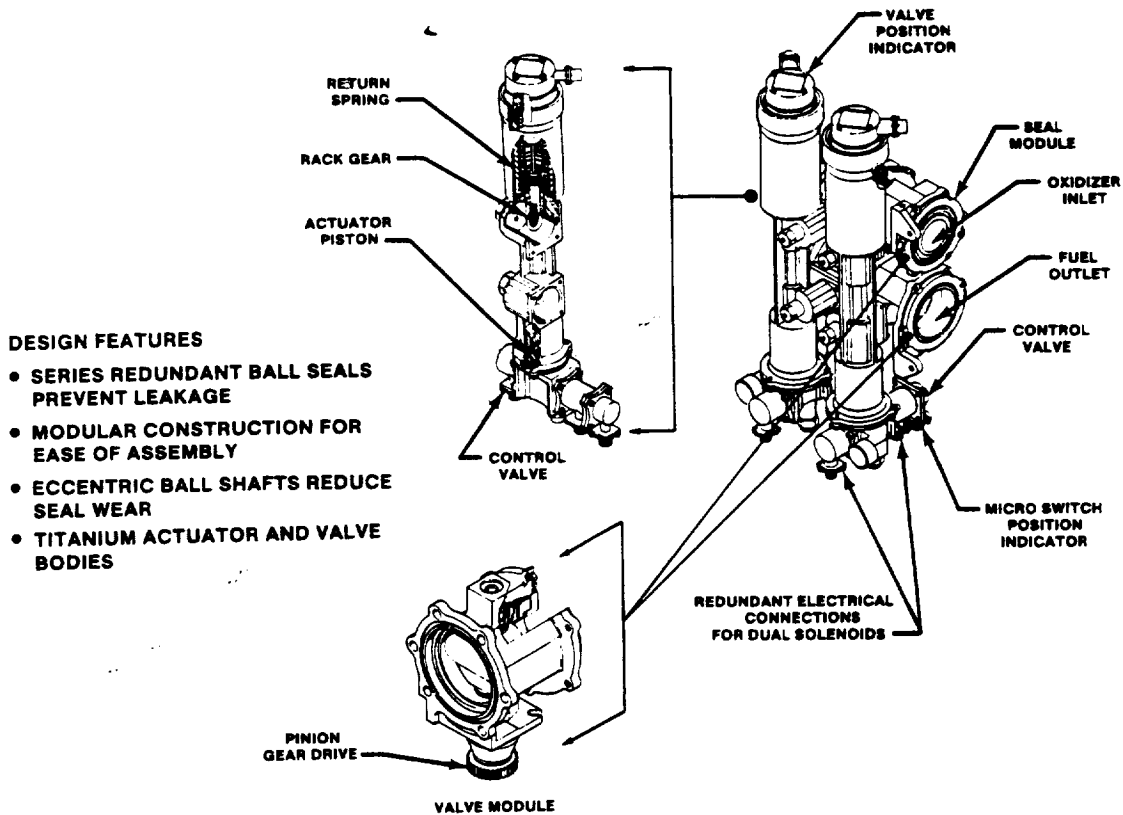FIGURE 14.- OMS ENGINE COMBUSTION CHAMBER.

DESIGN FEATURES

- RADIATION-COOLED LIGHTWEIGHT ASSEMBLY
- OXIDATION PREVENTING COATING
- ALL COLUMBIUM ALLOY CONSTRUCTION
- 80% BELL NOZZLE CONTOUR

FIGURE 15.- OMS ENGINE NOZZLE EXTENSION.

DESIGN FEATURES

- SERIES REDUNDANT BALL SEALS PREVENT LEAKAGE
- MODULAR CONSTRUCTION FOR EASE OF ASSEMBLY
- ECCENTRIC BALL SHAFTS REDUCE SEAL WEAR
- TITANIUM ACTUATOR AND VALVE BODIES

FIGURE 16.- OMS ENGINE BIPROPELLANT VALVE.

654

DESIGN FEATURES

- **INTEGRAL TITANIUM TANK AND MANIFOLD ASSEMBLY**
- **"PLUG-IN" COMPONENTS MOUNTED ON MANIFOLD**
- **REDUNDANT COIL SOLENOID ON ISOLATION VALVE**
- **PRESSURE RELIEF VALVE INTEGRAL WITH REGULATOR**
- **PROVIDES REGULATED PNEUMATIC PRESSURE FOR 10 ENGINE PURGES AND 40 VALVE ACTUATIONS**

FILL AND VENT VALVE

INLET PORT

GN_2 TANK (2500 PSIA)

PRESS. TRANSDUCER

REGULATOR (325 PSIA)

TWO-WAY ISOLATION VALVE

OUTLET PORT

TEST PORT

GN_2 STORAGE FOR ONE VALVE ACTUATION

FLOW SCHEMATIC

FILTER

REGULATOR

TANK GN_2

TO ENGINE

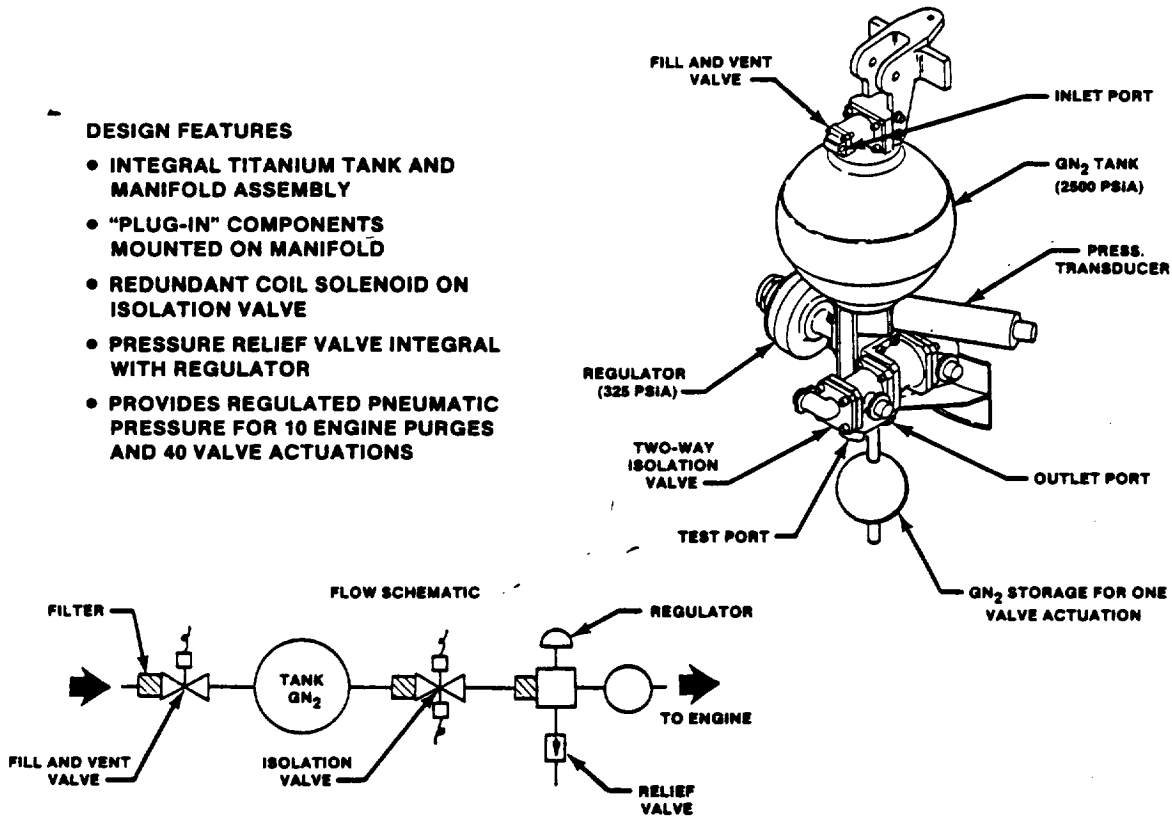FILL AND VENT VALVE

ISOLATION VALVE

RELIEF VALVE

FIGURE 17.- OMS ENGINE VALVE ACTUATION SYSTEM.

redundant bipropellant valve was subsequently changed to a series redundant valve in the interest of weight and complexity. The quad redundant concept is advantageous from the standpoint of a failure to open because parallel flow paths are provided; however, the added complexity is not desirable from the standpoint of the added leak paths. Upon reevaluation of the quad redundant concept, it was decided that series redundancy was appropriate to the OMS system because, unlike the Apollo SPS engine, the OMS engines were redundant to each other and the tankage and propellant supply lines could be configured to provide complete functional redundancy in the event of a failed-closed condition. In addition, it was determined that the most probable cause of failure of the bipropellant valve to open was a failure of one of the solenoid control valves which control the flow of gaseous nitrogen (GN_2) (the actuation gas) to the valve actuators. Consequently, all solenoid valves include dual coils to provide redundancy. Finally, a second GN_2 storage tank was added immediately downstream of the pressure isolation valve. This tank provides sufficient GN_2 to open the bipropellant valve once even if the pressure isolation valve fails to open.

CONCLUSIONS

The most significant lesson learned from the OMS program was the advantage of developing critical technology before initiating full-scale hardware designs. The successful completion of predevelopment studies undoubtedly reduced total contracted costs and minimized schedule delays that had been experienced in previous propulsion programs.

ACKNOWLEDGMENT