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HYDROGEN-OXYGEN SPACE SHUTTLE ACPS THRUSTER TECHNOLOGY REVIEW

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Abstract

A comprehensive program, sponsored by NASA-LeRC during the past few years, has provided the technology groundwork for the use of hydrogen-oxygen propellants in the Space Shuttle Attitude Control Propulsion System (ACPS) thrusters. This work has concentrated on generation of technology for injectors, cooled thrust chambers, valves and ignition systems. The thrusters are designed to meet a unique and stringent set of requirements, including: long life for 100 mission reuses, high performance, light weight, ability to provide long duration firings as well as small impulse bits, ability to operate over wide ranges of propellant inlet conditions and to withstand reentry heating. The program has included evaluation of thrusters designed for ambient temperature and cold gaseous propellants at the vehicle interface. The culmination of the component technology work is the incorporation of the best component designs into a complete integrated thruster assembly for evaluation of interaction effects, life, and performance.

This paper summarizes the results of the above programs.

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Introduction

For the past few years, NASA has been sponsoring, through the Office of Aeronautics and Space Technology, an extensive program to advance the technology of hydrogen-oxygen thrusters applicable to the Space Shuttle Attitude Control Propulsion System (ACPS). This thruster technology program has been managed by the Lewis Research Center. Related feed system technology work has been handled jointly by Manned Spacecraft Center and Marshall Space Flight Center. At the time this effort was initiated, hydrogen-oxygen propellants were baselined for the Space Shuttle ACPS for both the Orbiter and Booster stages. These propellants were selected because of their high performance, relatively low cost, clean exhaust, and nontoxic, noncorrosive nature, which is vitally important for a fully reusable system. Additional benefit was expected from commonality of propellants with the main engines and Orbital Maneuvering System (OMS). Subsequently, the Shuttle concept changed from the reusable flyback Booster and internally tanked Orbiter to the present external-tank Orbiter configuration with twin solid rocket motor boosters. These changes eliminated hydrogen-oxygen commonality with the Booster and greatly reduced the total system impulse required of the ACPS for the Orbiter (from over 2×10^6 lb-sec to approximately 1.0-1.5×10^6 lb-sec). Consequently, earth storable propellants (nitrogen tetroxide-amine fuel) or monopropellant hydrazine were selected as baseline for the Orbiter in preference to hydrogen-oxygen. Since then, the hydrogen-oxygen technology work has been continued on a reduced scale to provide an alternate or backup approach for Shuttle as well as to fully evaluate the technology areas of importance so that the concepts would be available for other possible applications, such as Space Tug. It is the purpose of this paper to present a comprehensive review of the state-of-the-art of hydrogen-oxygen ACPS thruster technology, with principal emphasis on the gas/gas thruster area. Discussion of the relative merits of the various competing propellant combinations, which requires an extensive and detailed trade-off study of system weight, volume, operational problems, cost, and reliability will not be presented herein.

Attitude Control Propulsion System (ACPS) Concepts and Considerations

The Shuttle Orbiter ACPS must provide both angular and translational control from the time of main engine shutdown after launch until reentry into the earth's atmosphere where aerodynamic control surfaces become capable of vehicle attitude control. The ACPS also provides roll control during main engine firing and translation maneuvers in three axes in space for rendezvous and docking.

Of the various types of systems studied involving the use of hydrogen-oxygen propellants for the ACPS, the most promising are the gas/gas, gas/liquid, and liquid/liquid systems, shown schematically in Figures 1, 2 and 3, respectively.

The gas/gas feed system (fig. 1) has received the greatest amount of attention and technology effort. In this system, the propellants are stored at low pressure as cryogenic liquids, pumped to high pressure by centrifugal pumps and passed through heat exchangers where they are "conditioned." The gases are stored in high pressure accumulators, from which they are fed on demand through a distribution network to the many thrusters. Small H_2-O_2 gas generators, which also draw gases from the accumulators, are utilized to provide the hot gases to drive the turbopumps and to heat the incoming liquids. Various configurations of this basic system concept have been studied, including some where a single gas generator is employed to power the turbopumps and heat the propellants.

The gas/gas feed system offers the advantages of versatility, flexibility, reasonably light weight, and the ability to be developed into a reliable, high performance, fully reusable system with excellent thruster pulsing performance.(1) Its disadvantages are complexity, large propellant storage volume, and high development cost. The system performance is degraded from the thruster performance of 435 lb_f-sec/lb_m to about 380 lb_f-sec/lb_m specific impulse because of the losses associated with the gas generator exhaust gases. The

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system also imposed severe requirements on the turbopumps and the gas generator/heat exchanger thermal conditioning unit. Technology contracts were undertaken by NASA-MSC and NASA-MSFC to solve the problems associated with the high performance and long life required of these components.[2,3]

The gas/liquid feed system (fig. 2) was devised as an approach that eliminates some of the undesirable aspects of the gas/gas system by substituting a liquid propellant feed system on the oxygen side. In the simplest approach of this kind, the oxygen is stored supercritically and fed directly to the thrusters at moderate pressure and near liquid oxygen saturation temperatures. This eliminates the need for the turbopump, accumulator, and gas generator/heat exchanger on the oxygen side making the total system simpler and less costly to develop. It does, however, introduce the requirement for distributing a cryogenic fluid throughout the vehicle to all the thrusters.

A further step in the direction of simplification is made by going to the liquid/liquid feed system (fig. 3). In this concept, both the hydrogen and oxygen are fed to the thrusters as cryogenic liquids. The propellants would be stored as liquids, pumped by centrifugal or positive displacement pumps to moderate pressure, stored in liquid accumulators (bellows tanks), and distributed throughout the network of manifolds and lines to the thrusters.[4] Thus, the gaseous accumulators and gas generator/heat exchangers used for gasifying the liquids are eliminated, making a simpler, lighter weight, lower volume system that would be cheaper to develop. The problem areas associated with this system are ignition of the cryogenic liquids, pulsing operation performance with liquid propellants as opposed to gases, possible combustion instability, and distribution of the cryogens without excessive heat leaks causing off design operation. This latter question has been studied in detail[5] to determine the temperature rise of liquid hydrogen and liquid oxygen in typical distribution lines and manifolds and to ascertain whether vacuum jacketed lines are necessary. The results of this analysis indicated that relatively small, controllable temperature increases would occur even without vacuum jacketed lines, making the concept appear quite feasible from this standpoint. Also, the study considered various types of LH_2 pumps and various means of driving the pumps, including the gas generator/hot gas turbine drive (shown in fig. 3), hydraulic and electrical drive. The questions concerning ignition of liquid hydrogen-liquid oxygen and thruster steady-state and pulsing performance are being investigated presently in a technology contract started in 1972.[6] The thruster technology for the liquid/liquid and gas/liquid systems is not as far advanced as the gas/gas system, and they will not be discussed further in this paper. Evaluation of the above problems, as well as experimental system demonstrations are needed before the viability of these approaches will be fully established.

Thruster Requirements

The Shuttle operational characteristics and the characteristics of the feed systems described above imposed a unique and rather stringent set of requirements on the ACPS thrusters, including high performance, long life, reusability, ability to provide both long duration firings and short impulse bits, exposure to reentry heating and atmospheric conditions, and ability to operate over a wide range of propellant inlet conditions. The thruster operating conditions and design requirements that were evolved from these general require- and used for the various technology programs are given in Table I for the gas/gas feed system. The ranges of chamber pressure, mixture ratio and propellant inlet temperature over which the thrusters were tested are also given.

The various nominal values listed in Table I were selected as a design point at which to examine thruster technology problem areas and were considered reasonably close to correct values for the original Shuttle concept. System studies[7,8] of various hydrogen-oxygen ACPS were conducted to select the most promising system concept and to provide trade-off studies of various thruster operating parameters to select the optimum values. The nominal values of thrust, chamber pressure, mixture ratio, nozzle expansion ratio, and propellant inlet conditions listed were selected based on these studies and supporting NASA documentation of ACPS requirements.[9] The propellant temperatures were initially specified to be 540° R (ambient) and were later reduced to the colder temperatures listed.

The specification of thruster life was considered to be of great importance and is one of the truly distinguishing features of this type of engine. The design life requirement specified for all the technology programs was that the thrusters should be capable of 100 mission reuses with a minimum of maintenance or refurbishment. Assumed firing rates of 1000 pulses/mission plus 50 full thermal cycles/mission were used. A safety factor of five was employed to arrive at the design life because of the uncertainties of thermal fatigue analysis techniques and the sparsity of reliable basic materials data upon which to base the analysis. The precise thruster firing schedule for the various types of Shuttle missions is not well defined at present. Obviously, not all thrusters will be utilized equally; this will depend on the number of thrusters (approximately 30-40), their location on the Orbiter, level of redundancy, and control techniques. The selected life requirements were, therefore, chosen to provide a measure of conservatism but were considered to be reasonable design values with which to demonstrate the effects of long life requirements on thruster performance and design details. These effects will be further described in the section on Thrust Chamber Life.

The specified values of minimum impulse bit and response are necessary to provide precise Orbiter attitude control and minimize propellant usage when controlling attitude to a tight deadband. These values are achieveable with the hydrogen-oxygen thruster provided valve response is rapid (see section on Valves) and manifold volumes are kept small. The gas/gas thruster is particularly adept at providing high specific impulse at small impulse bits.

Certain thrusters will be exposed to severe reentry heating, particularly the down firing and forward firing engines, if protective doors are not provided. The specified exposure time and temperatures are based on preliminary estimates; firm values have yet to be determined. These requirements were not imposed upon the early technology programs but have been applied more recently as more information on reentry heating effects on the thrusters became available. The effect of this requirement is to legislate against the use of low melting point materials, such as copper, in the expansion portion of the nozzle. Another factor that dictates against the use of copper is repeated exposure to atmospheric conditions when hot, which causes surface corrosion and severely degrades thermal fatigue life.

The specified thruster assembly weight was not imposed upon the initial thruster programs but was applied to the Integrated Thruster Assembly (see Thruster Assembly Section). After consideration of all the requirements, the critical component technology areas were identified and specific programs formulated to evaluate them. Included in the overall technology programs were: investigations of the problems of injector design and performance; thrust chamber cooling, life, and fabrication techniques; ignition of ambient temperature and cold gaseous propellants; valve design, cycle life, and leakage; and thruster assembly steady-state and pulsing performance, heat soak back and component interaction effects. These subjects are discussed in detail in the following sections of the paper.

Component Technology

Injectors

The thruster design and operating requirements listed in Table I dictated that injector designs be developed that were capable of providing high combustion performance (over 97% energy release efficiency) and combustion stability at both steady-state and pulsing conditions, that would provide a benign combustion chamber environment with a minimum of temperature striations, that would have low face temperatures to promote long life and minimum heat soak back, and that would be relatively low in cost to fabricate. Another somewhat unique requirement was the use of gaseous fuel and oxidizer in the injector, whereas most rocket injectors are designed for either gas/liquid or liquid/liquid propellants.

A number of parallel ACPS thruster contracts (10, 11, 12, 13) were initiated in which different approaches to injector design were pursued in order to provide alternate solutions to the injector design problem and avoid direct duplication of effort. A number of injector types were evaluated analytically and by cold flow tests using simulant fluids to determine the basic element designs best able to meet the above requirements. This preliminary screening resulted in selection of the element types shown in Figure 4 which were used for full scale injectors for hot firing tests. These included: (a) concentric tube (coaxial) element, (b) premix, (c) triplet (with raised center post), (d) trislot, and (e) reverse flow/vortex cup.

The various injectors were tested initially in instrumented heat sink thrust chambers to obtain energy release efficiency and heat transfer data. Tests were made both with no film cooling and with up to 37% fuel film cooling, since many of the thrust chamber cooling schemes involved the use of varying amounts of film cooling. A comparison of the energy release efficiency (ERE) performance of the various injectors is shown in Figure 5 in which ERE is presented as a function of mixture ratio (O/F) for the case of no film cooling. The raised post triplet injector provided very high performance for these operating conditions, which is typical for this type of element. The coaxial and premix injectors provided high performance also and proved to be rugged, durable designs well suited to the long life requirements of the ACPS thrusters. The tri-slot injector was slightly lower in performance and also caused hot streaks in the chamber due to its rather coarse pattern. Consequently, it was not used for the cooled thruster testing. The reverse flow/vortex cup injector data shown is somewhat misleading since with this concept there is always some film cooling of the chamber occurring, which tends to reduce performance. Because of this, delivered specific impulse of the reverse flow thruster was better than would be predicted based upon the injector performance curve and met the performance requirements.

Performance data was obtained with the various injectors with varying film coolant flow, propellant inlet temperatures, and chamber pressure to determine the effects of these variables. The effect of percent film cooling on performance is presented in Figure 6 for the coaxial, triplet, and premix injectors. The performance differences are small and are partly due to the differences in thruster mixture ratio at which these data curves were taken. The trend of decreasing performance with increasing film coolant flow is consistent with predictions and was typically exhibited by all the injectors tested. The injectors also showed consistent trends of decreasing performance with decreasing chamber pressure and decreasing performance with decreasing propellant temperatures.

Chamber heat transfer data were obtained with the various injectors using copper chambers instrumented with thermocouples. Typical throat heat flux data obtained with a coaxial injector[10], a premix injector[11], and a triplet injector[12] are presented in Figure 7. Throat heat flux, referenced to a 500° F design point wall temperature, is shown as a function of percent fuel film cooling. As would be expected, the throat heat flux was reduced as film cooling flow was increased. Data curves are shown for the premix injector for two different film cooling rings, one of essentially zero length (equivalent to film coolant injection from the injector face) and one which extended 2.5 inches downstream from the injector face. It can be seen that the 2.5 inch ring was more effective in reducing throat heat flux than the zero length ring. This result would be expected since the film coolant has less length in which to mix with the hot gas core flow and, therefore, is more effective in cooling the throat. The data shown are not intended to provide generalized results in terms of the heat flux produced by the different injectors, since there were differences in film cooling injector design, chamber length, and chamber contour (all of which have subtle influences on throat heat flux and film cooling effectiveness). It is shown only to illustrate the type of data obtained and present some typical results. Similar results were obtained with the other injectors.

Face patterns of the various injectors are shown in Figures 8 through 12. The coaxial injector developed by Rocketdyne Division of North American Rockwell[10] is shown in Figure 8. The design shown resulted from extensive cold flow testing of different element configurations and hot fire testing of two different injectors. The highest performing coaxial element design (shown in fig. 4(a)) incorporates an oxidizer tube flush with the injector face which contains a 7° tapered outlet to enchance mixing between the oxygen and hydrogen. The injector contains 54 coaxial elements and has the igniter port in the center of the injector face. The injector has a copper faceplate and Nickel-200 oxidizer posts. The body and manifolds are of 304L stainless steel.

The premix injector developed by Aerojet Liquid Rocket Company[11] is shown in Figure 9 and the element detail is shown in Figure 4(b). In this novel approach, the propellants are premixed in a "cup" behind the injector face. Oxygen enters the cup region of each element through a central tube and is impinged upon by two hydrogen streams from the two sides of the cup. A number of different injector designs were tested in which the cross-sectional shape of the hydrogen orifices was varied. Rectangular, T-shaped, and I-shaped orifices were investigated and an I-shape was finally proven to be optimum. These variations in the hydrogen orifice shape are achieved by the unique fabrication method employed. A stack of thin (0.010 in.) nickel plates, in which the appropriate holes are photoengraved, are brazed together and brazed to the stainless steel body of the injector. This method provides very accurate control of tolerances for the small flow passages involved and provides great flexibility in designing the shape and size of the passages. The faceplate is cooled by a multiplicity of small hydrogen bleed holes. This injector has 72 premix elements and has the igniter port in the center of the injector face.

The raised post triplet injector developed by TRW Systems, Inc.[10] is shown in Figure 10 and the element detail is shown in Figure 4(c). The injector has three circular rings containing a total of 144 oxidizer-fuel-oxidizer triplet elements. The orifice diameters were varied for each ring to provide a relatively uniform mass distribution across the injection plane. The injector faceplate (rings) are OFHC copper and the injector body and manifolds are 304L stainless steel.

The trislot injector design developed by Rocketdyne[10] is shown in Figure 11 and the element detail is shown in Figure 4(d). The trislot injector consisted of 18 elements in each of which hydrogen flow from two rectangular slots impinged upon oxygen flow from a central rectangular slot. The objective of this approach was to provide an injector concept that would be low in cost to fabricate, provide high performance, and be well suited for gas/gas propellants. The slots were fabricated by electrical discharge machining (EDM) into a copper faceplate core, which was then electron beam welded onto a 304L stainless steel body.

The reverse flow injector concept developed by Bell Aerospace Company[13] is shown in Figure 13 and Figure 5(e). In this unique approach, the hydrogen is injected in a reverse direction in the convergent portion of the nozzle and used to film cool the spherical chamber. The oxygen is injected from the head end of the engine through a vortex cup which causes the oxygen to spray radially outward and impinge with the fuel coming off the chamber walls. The objective of this design approach was to provide low temperature chamber walls, thereby allowing use of common materials and reducing manufacturing cost, and to separate the fuel and oxidizer which normally are in close proximity in the injector. This injector design will be covered in more detail in the section on thrust chambers to follow since in this concept, the injector and chamber are combined.

All of the above injector concepts, except the trislot, were utilized for testing of cooled thrust chambers. As stated earlier, the trislot was eliminated because of its tendency toward streaking.

Thrust Chambers

To arrive at the most attractive thrust chamber concepts for the ACPS thrusters, a number of cooling schemes were analyzed, such as regenerative, film, dump, ablative, radiation, and combinations of these. The analysis included tradeoff studies of performance, weight, inlet pressure required, cost, external wall temperature, ability to withstand reentry heating and repeated exposure to atmosphere, nozzle scarfing, and most particularly, chamber life.

Ablative chamber cooling was quickly discarded because of life limitations and excessive weight. Radiation cooling cannot be used effectively in buried installations and it also is somewhat life limited by the refractory metals normally used in the construction of radiation cooled chambers. It was, however, selected for the nozzle skirt in one of the chamber designs. Detailed study of the remaining cooling methods by the various contractors led to selection of the concepts shown in Figure 13 for fabrication of cooled thrust chamber hardware and testing.

The selected chamber configurations all feature combined use of three basic cooling methods, all of which employ active cooling with the fuel as coolant: (a) regenerative, (b) dump, and (c) film. To avoid confusion, definitions were established for these cooling methods, as follows:
(a) Regenerative cooling refers to forced convective cooling of the hot thrust chamber wall by passing the coolant first through coolant channels or tubes built into the chamber walls and then routing the coolant to the injector after it leaves the chamber. Thus, the fuel flows through the chamber coolant circuit and the injector in series.

(b) Dump cooling is similar to regenerative in that it employs forced convective cooling and uses a double wall or channel-wall design to provide coolant passages. It differs from regenerative cooling in that the coolant does not pass through the injector and chamber in series, but instead these flow circuits are in parallel, i.e., after convectively cooling the chamber, the coolant is dumped directly into the combustion chamber or nozzle, where it may then be employed as film coolant. Depending upon the chamber design, the dump coolant may be partially or fully "regenerative" in the sense of a thermodynamic cycle in which heat is removed from the combustion process and absorbed by the incoming working fluid (fuel).

(c) Film cooling refers to the cooling of a single chamber wall (as opposed to double or channeled wall) by injecting a cooler fuel flow along the wall which acts as a barrier to heat flow from the hot gas core to the metal wall. In several of the designs, fuel is used first for regenerative or dump cooling and then for film cooling.

These three cooling methods are well known in the rocket industry, but occasionally the terms are used differently. Hence, the above more or less arbitrary definitions are provided. Regenerative cooling has the advantages of no performance loss, long chamber life, and cool outer chamber walls, but the disadvantages of high pressure drop, high weight, and high fabrication cost. Dump cooling has similar advantages and disadvantages to regenerative cooling, but has lower pressure drop and lower delivered performance. Film cooling has the advantages of low pressure drop, simple, low cost chamber design, and relatively light weight, but the disadvantages of performance loss, higher wall temperatures which, therefore, limit chamber life. The advantages of film cooling were a good choice for the skirt of the ACPS thruster from a nozzle area ratio of about 18-20:1 to the exit plane. As may be seen in Figure 13, all of the thrust chamber cooling concepts used film cooling for the skirt. This provided a lightweight, single wall nozzle design that facilitates scarfing of the nozzle and makes excellent use of materials such as stainless steel or Haynes 188 which can withstand the high temperatures due to combustion heating with very little film cooling needed, as well as reentry heating without active cooling.

Th six chamber concepts shown in Figure 13, all of which were fabricated and tested, differed significantly in the cooling approaches used for the chamber and nozzle up to the nozzle area ratio of 18-20:1. These different approaches represented a deliberate attempt to provide a broad technology background. In the chamber design considerations, variation among the desirable attributes of performance, life, and cost, resulted in quite different design approaches to achieving the performance and life goals of the technology efforts. These chambers are generally characterized or referred to first by the method used to cool the throat, since this is the most critical region, and then by the method used to cool the combustion chamber or nozzle expansion section.

In the next few paragraphs, each of the six thrust chamber concepts will be described in detail as to the materials and construction techniques employed. Following this, a brief description will be given of the considerably different methods used to analyze the life of two typical chambers, a regenerative design and a film cooled design. Delivered performance obtained with four of the thrust chambers is described in the section on Thruster Assemblies, to follow.

(1) Regenerative/Dump Cooled Chamber. - In Figure 14 is shown a one-half-pass regeneratively cooled thrust chamber designed, fabricated, and experimentally tested by Rocketdyne[10], which was shown conceptually in Figure 13(a). This design employed regenerative cooling for the combustion chamber, throat, and divergent section of the nozzle to an expansion area ratio of 3. Dump cooling was employed from an area ratio of 3 to 18, and film cooling from an area ratio of 18-40. The chamber is of channel wall construction from the injector face to a plane 1.7 inches upstream of the throat. From this plane to a nozzle area ratio of 3, the chamber is of double wall construction (closure not attached to liner). From area ratio 3 to 18, channel wall construction is used, followed by the film cooled nozzle extension. The chamber, from the injector face to an area ratio of 18, used a NARloy-Z (copper/silver/zirconium alloy) liner with an annealed electroformed nickel closure, while the film cooled conical nozzle extension was made of 310 stainless steel. The double wall chamber region was designed to decrease the stresses imposed on the inner wall by the restraint of the outer wall when the inner wall is heated, thereby increasing the cyclic life (thermal fatigue) of the chamber. In this chamber, the hydrogen flow enters the manifold located near the throat and is split. Approximately 15% of the fuel flows downward to dump cool the nozzle from an area ratio of 3 to 18 and film cool the skirt, while the remainder of the flow is routed upward to regeneratively cool the throat and combustion chamber. The nozzle film coolant was injected through small two-dimensional nozzles designed to provide supersonic injection of the coolant at a pressure matching the free stream static pressure in the nozzle. Figure 16 shows the NARloy-Z chamber liner after the coolant channels were machined but prior to electroforming the nickel outer wall. The film coolant injection nozzles are clearly evident. This chamber also had provision for the use of film coolant injection from the injector end to supplement the regenerative cooling and reduce the chamber wall temperatures.

(2) Regenerative/Film Cooled Chamber. - A second thrust chamber design classified as regenerative/film, shown in Figure 16, was developed by Aerojet.[9] This chamber design was shown conceptually in Figure 13(b). The design consists of a single pass counterflow regeneratively cooled copper chamber which extends from the injector face to a nozzle area ratio of 3, a short dump cooled downpass flow section from area ratio 3 to 8, and a spun stainless steel film cooled nozzle expansion section from area ratio 8 to 40. A novel fabrication approach was used which featured the use of 60 photoetched precontoured stainless steel truss members to close out the coolant channel slots and provide coolant manifolds in the throat region. Final channel and manifold closeout was accomplished by use of a shrunk-fit and brazed stainless steel cylindrical jacket for the chamber region and several match machined conical stainless steel sections for closing out the channels in the divergent section. The film coolant flow was injected through 100 small two-dimensional nozzles formed by photoengraving a 0.015-in. thick copper strip which was brazed in place between the copper inner wall and a match machined stainless steel closeout section. The spun stainless steel skirt was electron beam welded to the aft end of the jacket. The hydrogen

flow enters this chamber at the manifolds near the throat with 10% of the fuel flowing downward to dump cool the short section from an area ratio of 3 to 8 and film cool the skirt and the remainder flowing upward to regeneratively cool the chamber.

(3) Dump/Film Cooled Chamber. - In Figure 17 is shown a dump/film cooled thrust chamber developed by Rocketdyne[10], which is illustrated schematically in Figure 13(c). This chamber consists of an uppass dump cooled section from an area ratio of 3 to the injector end, a downpass dump cooled section from an area ratio of 3 to 18, and a film cooled skirt. In this design, 35% of the fuel enters the manifolds located near the throat while the reminder is routed directly to the injector. Fifteen percent of the fuel is used to downpass dump cool the nozzle expansion section and film cool the skirt and 20% is utilized to uppass dump cool the throat and chamber and is then dumped into the chamber as film coolant. Because of the splitting off of the coolant flow, the injector is designed to operate at a mixture ratio of 6.1 whereas the overall thruster mixture ratio is 4.0. The construction details for the dump cooled chamber are very similar to concept (1) above, except that no double wall section was employed. The chamber is of channel wall construction throughout having a NARloy-Z liner into which the coolant channels were machined and which were closed out with electroformed nickel. The manifolds and bolted on nozzle skirt are of 304L stainless steel.

(4) Film/Regenerative Cooled Chamber. - A chamber concept featuring the use of film cooling for the throat and nozzle expansion and regenerative cooling of the cylindrical chamber, developed by Aerojet[11], is shown in Figure 18. This chamber was shown conceptually as Figure 13(d). In this design, the fuel enters the chamber at the manifolds located near the start of nozzle convergence. Twenty percent of the fuel flows downward through a short dump cooled section and is injected as film coolant 1.3 in. upstream of the throat. The remainder of the fuel flows upward through a short cylindrical regeneratively cooled copper chamber section and discharges into the injector fuel manifold. The cylindrical chamber liner was fabricated of OFHC or zirconium copper (one of each were built) into which 80 coolant slots were machined. The coolant slots were closed out by a 304L stainless steel jacket, brazed in place. The throat section was fabricated by spinning Haynes 188 on a mandrel. The nozzle skirt, of 304L stainless steel material, was also spun to shape and welded to the Haynes 188 throat section. The conical shaped film coolant injection ring was made of copper.

(5) Film/Dump Cooled Chamber. - Shown in Figure 19 is a unique film/dump cooled chamber that was designed, fabricated, and tested by TRW Systems, Inc.[12] This chamber is shown conceptually in Figure 13(e). In this concept, 32% of the fuel is injected between a copper inner sleeve (or duct) and the stainless steel chamber. This fuel cools the duct in a downpass dump mode and is then injected as film coolant in the convergent portion of the nozzle. The throat and entire nozzle expansion section are film cooled. The duct is fabricated of Berylco copper with coolant channels machined into the outer wall. The combustion chamber/nozzle assembly is of spun A-286 steel. The duct is mounted within the chamber so that it is free to grow slightly axially and radially when heated. This concept has the unique advantages of: (a) a simple, lightweight, inexpensive chamber, (b) the structural loading imposed on the inner duct wall is low and, therefore, cyclic life is high, and (c) response and coolant pressure drops are minimal.

(6) Reverse Flow Chamber. - The reverse flow chamber concept shown in Figure 21 was developed by Bell Aerospace Company.[13] This concept, as shown in Figure 14(f), actually consists of a regeneratively cooled throat section, a spherical shaped, film cooled combustion chamber, and a film cooled nozzle expansion section. In this chamber, the fuel enters the manifold near the throat and flows upward from a nozzle area ratio of 10 through coolant channels machined into the copper liner. The fuel, thus, regeneratively cools the throat region and is injected into the nozzle convergent section in the reverse direction to film cool the spherical combustion chamber section. This throat cooling mode is regenerative (according to our earlier definitions) rather than uppass dump since the main fuel injector located in the nozzle convergence is in series with the coolant circuit. In this chamber design, the vortex cup oxygen injector parts and spherical combustion chamber are 304L stainless steel. The throat liner is OFHC copper with coolant channels machined into the outer wall. Closeout of the coolant passages is achieved by two aluminum shroud half shell members that are held in place by the 304L stainless steel fuel manifold assembly. A film cooling mainfold is incorporated into the aft end of the fuel manifold and throat section assembly from which 7.5% of the fuel is injected downward to film cool the nozzle extension. The bolted on nozzle extension section is of Columbium Alloy C103 with a silicide coating on the inner wall and a Dynaflex insulation blanket on the outer wall. The Columbium nozzle is essentially radiation cooled with supplemental film cooling. A chamber insulating blanket is provided to reduce the radiation heat transfer to the vehicle.

Thrust Chamber Life

The advent of reusable vehicles, such as the Space Shuttle, wherein the components should ideally be capable of 100 mission reuses, places a premium on thrust chamber life. The ACPS thruster life requirements, as listed in Table I, included 500,000 pulses, 25,000 ful thermal cycles, and 50 hours of operation. These are stringent requirements for a small, high performance rocket engine which require careful design considerations to produce a truly long-life engine. Compromises in engine performance and materials selections had to be made in some cases to meet the life requirements.

The entire technology related to rocket thrust chamber life is a new and difficult field that is essentially in its infancy. Reliable predictions of chamber life require extensive basic materials data on thermal fatigue, such as that shown in Figure 21, detailed analysis of the cyclic stress and strain behavior of each portion of the chamber, and intimate knowledge of the temperature profile throughout the chamber during the transient and steady-state phases of each firing cycle. To experimentally verify chamber life requires extensive testing at known operating conditions with avoidance of extraneous failures due to facility malfunctions, chamber fabrication discrepancies, or the like. Correlation of experimental results with theoretical predictions is also complicated by the inaccuracies in such areas as heat transfer analysis and material property data.

The fundamental theory used in life analysis is that failure prediction depends on the accumulation of fatigue damage and creep damage. The analysis involves the evaluation of the material's capability to resist damage from exposure to steady stresses, cyclic stresses, and elevated temperatures for the specified service life. Mansion's universal slopes method, adjusted for elevated material properties or actual experimental thermal fatigue property data were used to calculate fatigue damage. Stress rupture data were used to estimate creep damage.

A complete discussion of the methods and theory used for chamber life predictions is beyond the scope of this paper. However, a brief discussion is included of the approach used for two different chamber types, a film cooled (throat) chamber and a regenerative chamber. These chambers differ not only in the materials used and the allowable temperatures, but also in the wall temperature behavior under transient conditions. As shown in Figure 22, the hot side and back side wall temperatures for a film cooled chamber wall initially diverge when an engine firing begins and then converge as the heat pulse is absorbed and the wall comes to thermal equilibrium. The maximum temperature difference across the wall, and consequent maximum thermal strain, occurs very early in the firing, perhaps within 100 milliseconds. For a regeneratively cooled chamber, where a channeled wall construction is used and the wall is actively cooled, the hot side and back side wall temperatures continually diverge until thermal equilibrium is established. This occurs later in the firing than for the film cooled case and, consequently, the worst case condition, i.e., the maximum temperature difference, occurs only in a longer duration firing. Thus, the life limiting condition for film cooled chambers is any pulse of greater than say 100 milliseconds, whereas the life limiting case for the regenerative chamber is a full thermal cycle, i.e., a firing longer than about 2-3 seconds. This, of course, neglects other stress imposing factors, such as pressure loads, and creep, which tend to be secondary effects for the ACPS thruster operating conditions.

For the film cooled chamber, shown in Figure 18, the cyclic and creep rupture life of the single wall throat and skirt was computed based on experimental temperatures and transient heating rates measured during a test firing.[11] The analysis begins at the film coolant injection station and extends through the Haynes 188 throat and 304L stainless steel skirt to an area ratio of 40. Figure 23 shows the effective stress in the Haynes 188 throat region at 0.10 seconds after the thruster is started from a cold condition. The major component of this stress is due to the radial gradient through the wall. For the 0.045 in. thick wall Haynes 188 throat, the peak stress of 72,000 psi results in a strain of 0.27% which converts to a cycle life of 4.6×10^5 (fig. 21). The life at the chamber throat could be increased by using more film cooling or a thinner chamber wall. The region 0.4 in. upstream and downstream of the throat operates in the elastic region and has much greater life.[11] For longer firing times, the radial temperature gradient through the wall is greatly reduced and the remaining structural loads are due to thrust, internal pressure, and axial temperature gradients. These steady-state stresses are considerably below the 50-hour creep rupture limits of the chamber materials throughout the film cooled portion of the chamber.

Life predictions for the regenerative chamber designs are generally based upon steady-state measured temperature profiles throughout the chamber wall at or just upstream of the throat cross section obtained with heat-sink chambers. These data are then correlated with analytical predictions using two-dimensional heat transfer computer programs. For this analysis, the chamber wall cross section is divided into a grid network and the temperature at each nodal point is calculated. From these calculations, an average circumferential chamber wall temperature is established for which the material stress and strain in the chamber walls are calculated at various axial stations. Life predictions are then made taking into account thermal fatigue damage and creep damage.

Experimental data were obtained with a regenerative chamber of the type shown in Figure 16 at chamber pressures from 100 to 500 psia and with varying propellant inlet temperatures. The effect of these variables on the chamber inner wall temperature difference and the chamber life are shown in Fig. 24. The temperature measurements were made at a point 0.6 in. upstream of the throat, which is the most critical point for this chamber. The life predictions shown are for thermal fatigue damage alone, neglecting long term creep effects, which are of secondary importance. These predictions are based upon minimum properties for zirconium copper and are, therefore, conservative. The curves show that chamber pressure has a dramatic effect on chamber life while changes in fuel inlet temperature have little effect.

Ignition Systems

No clear-cut choice of the ignition system to accommodate the stringent ignition requirements of a small, pulsing hydrogen/oxygen thruster was evident. Spark ignition systems, for fixed-point operating condition hydrogen/oxygen engines have been successfully developed in the past for both the RL-10 and J-2 engines. However, the requirements for the Shuttle ACPS thrusters are more severe in that the igniter must have fast response, high cycle life, be capable of operating over a wide range of conditions, and the total number of igniter systems required is very high (≅38); therefore, power requirement for each igniter system must be very low. Table II lists the requirements that the ignition system must meet in addition to the general thruster requirements given in Table I.

In order to provide an adequate ignition system concept to fulfill the requirements of the

ACPS application, a number of different ignition schemes were considered. Some of the more attractive candidates that were selected for experimental testing as part of the LeRC technology program were: (1) electric spark plug/torch** igniter, (2) electric plasma/torch** igniter, (3) direct electric spark plug ignition (combustion chamber wall-mounted), (4) heterogeneous catalytic torch** igniter, and (5) auto ignition torch.**

(1) Electric Spark Plug/Torch Igniter. - The primary objective of the electrical spark plug/torch igniter investigation was to obtain a highly reliable ignition device capable of operating over the wide ranges of operating conditions given in Table II, while also achieving significant reductions in spark plug and exciter package input energy requirements, reduction in exciter weight and size, and minimum radio frequency interference (RFI). Elimination of the high-tension pressurized cable assembly between the exciter package and the spark plug, such as used on the J-2 and RL-10 engines, was an important design goal to reduce RFI and eliminate voltage loss and leakage problems associated with such cables.

A technology program was conducted by Rocketdyne(14) on igniter systems, which included extensive analytical and design effort, to fully delineate the effects of major variables on the ability to obtain fast reliable ignition of the spark igniter/torch itself and of the complete thruster. This effort began with analysis of the basic chemical and physical variables involved in the ignition process for various mixtures of gaseous hydrogen and gaseous oxygen over a range of temperatures.

Cold flow tests were conducted with scale model hardware from which evolved design critera for the igniter injector, igniter torch chamber, and spark plug location. The design which evolved for the spark plug/torch igniter is shown in Figure 25. Extensive tests were performed with the spark/torch igniter over ranges of propellant temperatures and pressures, fuel lead and lag, environmental temperature and pressure, and mixture ratio. Variations were also made in spark rate and energy, spark plug gap, electrode type and material, and plug location. Tests were made to verify that the RFI output was within the MIL-STD-461 specification.

A flight type exciter package was designed that features an integral spark plug, as shown in Figure 26, and weighs only 0.6 lbs. The exciter provides an inductive electrical circuit to provide energy to the spark discharge, as opposed to a capacitive discharge device. The exciter package requires only 8 watts electrical power input, and complies with the RFI requirement.

As a result of the testing done on the igniter and exciter unit, a complete list of nominal igniter design conditions was derived, given in Table III, which are applicable to the Rocketdyne unit.(14) The test results indicated that reliable ignition was obtained over all of the operating condition ranges except at low propellant temperatures (below 380° R) where some nonignitions occurred. Typical response data for this igniter design indicated that ignition usually occurred on the fifth spark 16 milliseconds (ignition delay) after inception of oxidizer flow, 25 milliseconds after start of fuel flow, and 33 milliseconds after electrical start signal. A fuel lead of 10 milliseconds was intentionally used to avoid autoignition. Thus the local spark plug mixture ratio had to go through a fuel-rich transient. Without a fuel lead, the response could be less than 25 milliseconds from electrical start to ignition.

A second electrical spark plug/torch igniter design concept was generated under contract by Aerojet.(15) Although the objectives and design requirements (Table II) of this effort were the same as discussed above, several noticeable differences are evident in the approaches taken. As may be seen in Figure 27, this igniter uses an air-gap spark plug and a cooled igniter body. Oxygen is injected around the plug center electrode so that the spark discharge occurs in an oxidizing atmosphere. Fuel is injected radially inward just downstream of the plug tip and provides a controlled mixing environment for reliable ignition. The remainder of the fuel passes through coolant channels in the double-walled body and is injected at the igniter tip giving an overall O/F of 6.5, whereas the core O/F is 45:1. Other characteristics of the Aerojet design include: an exciter spark rate of 500 sparks per second, capacitive discharge exciter unit, nickel electrodes, and electrical energy delivered to the spark plug of 5 millijoules. The igniter uses 20 KV breakdown voltage and a spark gap of 0.050 inches.

Experimental results with this electrical spark/torch igniter, employing the higher O/F in both the igniter core and overall igniter flows in conjunction with a higher spark rate, provided reliable ignition over the entire range of operating conditions tested and was also within the standard RFI requirements. Another significant experimental result with this igniter design was that optimum ignition with no pressure spikes was obtained with simultaneous injection of the main propellants. Both fuel leads and lags caused ignition overpressures, but fuel lags caused the more severe spikes. From the data from all tests conducted with this igniter assembly under vacuum conditions, it was determined that the average response time for the torch ignition to occur was 10 milliseconds from electrical start. Durability tests of both steady-state (10 seconds of operation) and pulse mode (1000 pulses of 1 sec duration) operation on the igniter assembly revealed no operational or physical degradation.

(2) Electric Plasma/Torch Ignition. - Another electric ignition system very similar to the spark/torch igniter was also evaluated by Aerojet(15) for potential ACPS thruster applications. This concept, shown in Figure 28, has the potential advantages over the spark/torch approach of longer electrode life and lower input power required. The basic dif-

**Torch means an ignition device that brings together ignitable mixtures of gaseous hydrogen and oxygen, in the presence of an ignition energy source, and expells the products of this combustion (effluent) into the main combustion chamber of the thruster to ignite the main propellant flow.

ference between the two concepts is in the primary ignition zone (ignition kernel area).

In the plasma pulse igniter, a portion of the hydrogen is passed through the plasma gap and is ionized by a pulsing electrical energy source of 0.2 millijoules at 5000 volts to provide the ignition source. In the spark plug igniter system, oxygen is passed through the spark gap or in some designs the spark discharge occurs directly in a gas mixture. The energy level of the spark plug system is much higher. This approach requires 5 millijoules of energy at 20,000 volts. The oxygen is injected in an annulus around the plasma plug tip. As in the spark plug/torch igniter, most of the hydrogen (85%) is used to cool the torch chamber wall. This coolant hydrogen then mixes and further reacts with the igniter core effluent at the igniter exit plane to provide a very hot torch for thruster main stage ignition. The core O/F is 45:1 while the overall O/F is 6.5 for plasma/torch igniter. Test results demonstrated that the plasma/torch igniter operates reliably over the ranges of operating conditions given in Table II. Best results again were obtained with simultaneous propellant valve sequencing. A portion of the testing was concerned with evaluation of candidate electrode materials. Based on electrode durability tests, copper was selected for the anode and 2% thoriated tungsten for the cathode.

(3) Direct Electric Spark Plug Ignition. - The possible use of a conventional aircraft engine airgap spark plug, simply mounted through the combustion chamber wall, was evaluated as a part of the hydrogen/oxygen thruster technology work performed under NASA-LeRC contract with Bell Aerospace Company.(13) The ignition effort performed was only that necessary to develop a workable ignition system to be used in thruster tests. A conventional G.L.A. spark plug exciter package and Champion A14-1-395-1 aircraft spark plug were used. The spark plug was mounted through the combustion chamber wall, as shown in Figure 28. Provision was made in the spark plug mounting arrangement to provide oxygen augmentation on the upstream side of the spark plug to enhance the ignition process. It was found that some chamber wall erosion occurred if the quantity of oxygen used was not very accurately controlled. This ignition scheme is somewhat simpler than the spark/torch or plasma/torch systems and apparently worked sufficiently well to complete the overall thruster evaluations, but does not provide the well controlled environment at the spark plug tip so necessary for reliable ignition, particularly over the range of propellant inlet temperatures and mixture ratios set forth in Table I.

(4) Heterogeneous Catalytic Torch Igniter. - For many years, rocket engineers have sought a reliable passive ignition system for the hydrogen/oxygen propellant systems (i.e., one requiring no external power input). Since the interest arose in hydrogen/oxygen propellant for the Shuttle ACPS thrusters, NASA-LeRC intensified this search for a passive ignition system. Two passive ignition systems have been investigated during the 1969-1971 time period under Lewis sponsorship. These two, to be discussed in the next two sections, are: (1) heterogeneous catalytic ignition, and (2) autoignition torch/resonance tube ignition.

By 1969, the use of noble metal catalysts had been demonstrated to create spontaneous ignition of both ambient temperature and chilled oxygen/hydrogen mixtures in a single thermal bed reactor.(16,17) However, the response time required for the spontaneous reaction to occur (from the measured time of arrival of both propellant species at the catalyst active sites until reaching 90% of steady-state chamber pressure) appeared to vary drastically with variations in both initial propellant temperature and reactor operating pressure (chamber pressure). Also, the ignition response was never faster than approximately 200-250 milliseconds. Additional investigation was, therefore, undertaken to reduce the ignition delay time at low propellant temperatures and to find possible means of significantly improving ignition response times under all conditions.(18,19,20) Catalytic igniters were designed to minimize thermal and pneumatic lag and tests were made to evaluate the relative importance of several factors, such as pneumatic lag, bed thermal mass, reactor body thermal mass, and initial O/F entering the reactor, in causing long ignition delay times.

In the search for faster response of the thermal reactor bed type igniter, high initial O/F's up to 10:1 were attempted, which only created flashbacks. Reducing the thermal inertia of the reactor body by means of insulation and isolating the catalyst bed from the reactor body brought about no appreciable improvement in response time. A special catalytic igniter design finally evolved (fig. 30) which used downstream injection of pure oxygen into the effluent from the catalytic reactor. This approach produced significant ignition response improvement. Figure 31 shows the ignition response of the same type of H_2/O_2 reactor with and without downstream oxygen injection. This technique brought about an order of magnitude change in the ignition response - from 250 milliseconds to approximately 25-30 milliseconds just by the use of downstream O_2 injection. The 25-30 millisecond response of this igniter still fell short of the 10 millisecond goal set forth in Table II however. With cold temperature propellants (-250° F), the overall reactor response remained very good - approximately 40-50 milliseconds.

Durability tests, including both pulsing and steady-state firings, were also conducted(18) to establish the cyclic and steady-state life potential of two key noble metal based catalysts - Shell 405 and Engelhard MFSA. It was clearly established that both of these catalysts were very durable for steady-state firings up to 4000 seconds duration and continuous pulse mode operation up to 5000 thermal cycles without significant physical or chemical degradation. Ultimate durability of catalyst beds has not been firmly established.

(5) Autoignition Torch (Compression-Resonance Tube). - The second passive ignition system concept evaluated for potential Space Shuttle ACPS thruster applications was the autoignition or compression-resonance tube igniter.(14) This concept requires minimal electrical power and no high voltage but does require a gaseous propellant. Hydrogen gas was used because of its availability and because it provides rapid heating to tempera-

tures above the autoignition temperature of hydrogen/oxygen mixtures.

The physical principle upon which the resonance tube igniter operates is the use of a resonating gas column to heat the hydrogen gas to the H_2/O_2 autoignition temperature. As shown in Figure 32, the hydrogen was injected at 350 psia through a sonic nozzle and impinges directly upon the resonance tube entrance. Gas trapped in the resonance tube is quickly heated by repeated adiabatic compression to temperatures of 1500° F (Mode 1). The hydrogen is introduced about 5-10 milliseconds before the oxygen to allow time for the resonant heating to occur. Oxygen is then injected into this hot hydrogen and ignites (Mode 2). Combustion is sustained in the larger diameter tube, from which it flows out into the thruster, thus providing a torch for thruster ignition. The concept requires no external power other than for valve operation and, therefore, is a passive system.

During the autoignition technology program, several resonance cavity configurations and other igniter/resonance tube variables (gap ratio and pressure ratio) were evaluated. Ignition responses, from electrical signal to 90% of igniter chamber pressure, of 20 to 30 milliseconds were obtained. The basic feasibility of the concept was demonstrated with work horse hardware and relatively rapid ignition response appears to be directly related to the high igniter operating pressure (350 psia).

Several hundred hot tests were conducted with an optimized resonance tube igniter assembly, shown in Figure 33, of which 22 were igniter/thruster ignitions in a 1500 lb. ACPS thruster assembly.[14] The overall response of this ignition system in the thruster assembly was determined to be approximately 50 milliseconds from valve signal to 90% thruster chamber pressure. The resonance tube igniter generally performed well but needs further investigation to optimize the design for low propellant temperatures and reduce its sensitivity to pressure ratio and valve sequencing. From the amount of testing performed with this igniter concept, the only durability limiting device is the propellant valve assembly since the basic igniter assembly contains no moving parts.

From the array of ignition data produced and the variety of ignition schemes, it is evident that a number of viable ignition system candidates exist for the ACPS thrusters. Final selection of the ignition system for flight use will depend to some extent upon the relative emphasis placed upon various requirements, such as power required, redundancy, reliability level, life, etc. At this point in time, the electric spark/torch system appears most attractive because of its high experience level, reliability, positiveness, fast response, light weight, low power drain, and low RFI. The preferred spark/torch system is the air/gap plug, high O/F core, capacitance discharge unit. Other candidates may be developed into superior systems with additional work, but would have difficulty competing with the high experience level of a spark plug system. In the selection of a Spark Plug torch or Direct Spark Plug igniter system, the spark rate (spark per second) and the thruster response requirements must be closely matched in order to provide spark availability overlap during rapid pulse mode thruster operation to preclude unduly delayed thruster ignition or excessive propellant accumulations. A summary of all five ignition system design and operational characteristics is shown in Table IV.

Propellant Valves

Another critical ACPS thruster component area that needed technology improvement was the thruster propellant valves. Existing propellant valve designs of the required size were not capable of meeting the long life, low leakage, and rapid response requirements of the ACPS thruster application.

In order to provide propellant valve design and materials selection criteria for the ACPS thruster propellant valves, two parallel contracts (21 and 22) were sponsored by NASA-LeRC in 1970. These programs included analysis and conceptual design of propellant valve subcomponents, including fluid shutoff devices, actuators, linkages, and seals. The design requirements for the propellant shutoff valves, which applied to both contracts, are listed in Table V. Following the analytical phase, selections were made of the more promising sealing techniques, materials, and actuation methods, and screening tests were performed. Based on this data, preliminary designs were completed for complete valves and more extensive cycle life tests performed on valve test fixture assemblies. The leakage specification set forth in Table V was based upon values of total propellant loss for the entire one-week Shuttle mission considered acceptable, considering that leakage would be occurring from 30-40 thruster fuel and oxidizer valves. The operating temperature range specified assumes that valve temperatures may reach 850° R due to heat soakback from the thruster after a firing. This requirement necessitated consideration of all metallic valve seats or the use of thermal isolation to allow safe use of nonmetallics. Both metallic and nonmetallic seals were studied in the valve programs.

The types of fluid shutoff devices included in the studies were: ball valves, butterfly, poppet, blade, and diaphragm valving concepts. Conceptual designs were completed for each type to establish sizing and seat load requirements. Both impact and sliding seal closures, singly and in combination, were included in the conceptual designs. The results of these analyses were compared and rated in terms of pressure drop, leakage, response time, cycle life, contamination sensitivity, actuation forces, weight, envelope, degree of fabrication difficulty, assembly and inspection difficulties, and cleaning and handling requirements.

The valve technology contracts also included study of various methods of actuating the valve, including pneumatic, hydraulic, and electrical. These methods were compared on a systems basis as to the source of actuation energy and the weight of the source system as well as the size, weight, response, and operating characteristics of the valve actuators themselves.

Based on these studies, both contractors selected poppet type valves and pneumatic actuation using the hydrogen or oxygen (propellant) gas as the actuating media. The poppet type sealing closure was selected because this type of closure requires the least amount of sliding between the sealing surfaces. This reduces the effects of wear and enhances the cyclic life of the seal closure.

The poppet-type seal is sensitive to misalinement, however. The amount of misalinement between the sealing surfaces at closure results in varying degrees of clamshelling and scrubbing actions, which produce sliding wear. The amount of wear for a given length of sliding varies greatly for different seal materials. The selection of the proper seal material is, therefore, equally as important as the actual seal design configuration.

Pneumatic actuation was selected for the valves because gaseous propellants would normally be available for use as an actuation medium and this approach offers system simplicity and low weight while meeting the valve's fast response requirement.[21,22]

In order to select the internal sealing closure for the valve, screening tests of many seal closure configurations and seal materials were conducted for impact and sliding concepts. Rocketdyne[22] tested the following seal closures in a specially designed seal test fixture:

1. Flat 440c (stainless steel) seat on flat 440c poppet.

2. Grooved gold seats on flat 440c and tungsten carbide poppets.

3. Hard sharp tungsten carbide seat on flat tungsten carbide poppet.

4. Captive plastic seat on flat 440c poppet.

5. Elgiloy disc seat on 440c poppet.

The hard sharp tungsten carbide and the captive plastic seal configurations, shown in Figure 34, were selected for testing in a valve test fixture. The captive plastic seal successfully achieved more than 100,000 cycles under all test conditions and the leakage remained well below specifications. The other seal closure/material configuration tested failed by leaking excessively.

The preferred propellant valve design concept generated by Rocketdyne, shown in Figure 35, is a hydrogen gas actuated poppet valve employing the captive plastic seal arrangement. The valve incorporates a solenoid actuated pilot valve. The pressure unbalanced poppet is designed so that the solenoid valve vents pressure behind the actuator piston to open the valve and pressurizes this cavity to close the valve using a return spring for assistance.

The objectives of the propellant valve technology effort conducted by Marquardt[21] were identical to the previously discussed Rocketdyne effort but the approach and design configurations differed.

In order to select the internal sealing closure for the valve, screening tests of many seal closure configurations and seal materials were again conducted for impact and sliding concepts. The following seal closures were tested:

1. Spherical and flat polyimide seats on tungsten carbide poppets.

2. Spherical and flat teflon seats on tungsten carbide poppets.

3. Flat tungsten carbide seat and poppet.

4. Flat teflon coated and gold plated lip seal seats on tungsten carbide poppets.

Based on these test results, the flat polyimide (Vespel 21) seat on tungsten carbide poppet and the gold plated lip seal on tungsten carbide poppet were selected. The flat polyimide seat configuration successfully met the leakage specification in hot, cold and ambient temperature tests after 100,000 opening/closing cycles. The gold lip seal seat configuration only slightly exceeded the original leakage specification, therefore, it is still considered a viable candidate for a final valve design. The other seal closure configurations tested exhibited excessive leakage and were deemed unacceptable. Based on valve test fixture cycle testing with these seal concepts, a preferred flightweight valve concept was designed that is shown in Figure 36. This is a coaxial poppet valve with a single solenoid actuated pilot valve and uses the flat polyimide seat and an Inconel 718 poppet. It is a vent-to-open, pressurize-to-close design.

The valve concept is based upon a previous design which was built as a valve test fixture and met the leakage, response, and cycle life requirements of previous technology efforts.

In summary, the NASA-LeRC sponsored propellant valve technology effort indicates the poppet type, propellant actuated valve to be the most promising concept to meet the Space Shuttle ACPS thruster shutoff valve requirements, which include: fast response, low leakage, high cyclic life, minimum weight and envelope, simplicity, and reliability. The valve test fixture tests have produced data for two sealing design configurations (flat polyimide and captive plastic seal) which appear capable of meeting the stringent APS life and leakage requirements.

Thruster Assembly Experimental Evaluation

Within the thruster contract programs, specific components (i.e., injectors, thrust chambers, igniter assemblies, and propellant valves) were selected from the many component designs evaluated for use in testing of complete ACPS thruster assemblies. Four of these specific thruster assemblies will be discussed in this section. Except as noted below, these assemblies used spark/torch igniters and modified off-the-shelf valves that were readily available. The thrusters were subjected to extensive hot firing tests to establish the actual delivered performance, chamber cooling, and ignition behavior in both steady-state and pulse mode operation under simulated altitude conditions using 40:1 area ratio nozzles. Propellant inlet temperatures (250° R to 800° R), chamber pressure (100 psia to 500 psia), overall thruster mixture ratio (3.0 to 5.5), coolant flow rates, and pulsing duty cycles (10% to 90% were the primary operating parameters that were varied during these tests. A comparison chart of the four selected thruster assemblies showing typical operating and design characteristics is shown in Table VI.

The four thruster assemblies discussed below are: (1) Regenerative/Dump Cooled, (2) Film/Dump Cooled, and (4) Reverse Flow.

(1) Regenerative/Dump Cooled Thruster Assembly. - The regenerative/dump cooled thruster assembly, shown in Figure 37, consisted of a concentric tube injector assembly (fig. 8), a channel wall thrust chamber assembly (fig. 14), an electrical spark-plug/torch igniter assembly (fig. 25), and modified, facility-mounted, butterfly-type main propellant valves. It was first tested at nominal steady-state design point conditions: 300 psia chamber pressure, mixture ratio of 4.0, with ambient temperature propellants. The fuel coolant flow rates and thruster mixture ratio were varied to determine the effects on delivered performance and chamber wall temperatures. Also, the chamber pressure, overall mixture ratio, and propellant inlet temperatures were varied with coolant flowrate held constant at a selected fixed value. One 500-second duration test was conducted with the thruster assembly at nominal design point conditions with no evidence of chamber degradation.

The steady-state performance obtained with this thruster is shown as a function of mixture ratio and propellant inlet temperature in Figure 38. Thruster performance decreased as mixture ratio was increased; performance also decreased as propellant inlet temperature decreased. Chamber pressure effects on performance were very small over the range of 100-500 psia. This factor leaves an option open to the vehicle designer of operating this thruster at a lower chamber pressure to gain increased chamber life (see fig. 24). The performance effects of mixture ratio and propellant temperature variations show the importance of a well controlled propellant feed system.

A vacuum specific impulse of 447 lb_f sec/lb_m was consistently demonstrated at the nominal design point (ambient temperature propellants) operating conditions with approximately 17% of fuel film cooling for the nozzle. Tests were also conducted with this same thruster (not reoptimized for cold propellants) to obtain the effects on delivered performance by the use of low temperature propellants (320° R O_2 and 283° R H_2). A vacuum specific impulse value of approximately 425 lb_f-sec/lb_m was obtained at the nominal operation condition of P_c = 300 and M.R. of 4.25.

Several series of pulsing tests were conducted with the regenerative/dump cooled thruster assembly to determine its delivered pulse mode performance, ignition transients, thrust transients and thermal characteristics. This design was selected for pulse testing because very little experimental data existed on the effects of coolant jacket volume, on pulse mode performance, and start-up and shutdown transients (response). The pulse mode tests covered various duty cycles, ranging from 30 milliseconds (ms) "on"/1 sec "off" to 1 sec "on"/30 sec "off". Fuel and oxidizer leads and lags were also varied to determine the effects of propellant sequencing on thruster response and pulsing performance. These tests covered a range of oxidizer leads to simultaneous propellant entry to a fuel lead, and it was concluded from these tests that a fuel or oxidizer lead of 4 ms or less was the most desirable mode of operation for this thruster design.

A definite performance (I_{sp}) trend with impulse bit was established during the duty cycle test series. The delivered specific impulse was reduced as the thruster on time was decreased as shown in Figure 39. At the minimum impulse bit (MIB) of less than 50 lb-sec, the I_{sp} was ≅ 412 lb_f-sec/lb_m.

A thruster durability test series of 2547 pulses at a duty cycle of 20% (100 ms "on" and 400 ms "off") was also conducted with no noticeable physical degradation of the thruster and with excellent pulse repeatability. Considerably more pulse testing would be required to verify the 500,000 cycle life of the chamber; the 2500 pulse test was made primarily to evaluate pulse repeatability.

(2) Film/Regenerative Cooled Thruster Assembly. - The film/regenerative cooled thruster assembly shown in Figure 40 consisted of an "I" triplet premix injector assembly (fig. 9), a film-cooled thrust chamber assembly(fig. 18), and a spark-plug/torch igniter (fig. 27). Pneumatically actuated propellant valves that were poppet type and facility mounted, were used for propellant flow control during all thruster tests. Variations in propellant inlet temperatures, chamber pressure, and overall thruster mixture ratio were made during these tests. All test data were acquired under simulated altitude conditions using a 40:1 nozzle.

Several series of steady-state hot firing tests were conducted to establish the amount of film coolant required to satisfy the desired equilibrium wall temperatures of the chamber. Fuel film coolant flow was varied from 13.9% to 30% of the total fuel flow. It was determined that 20% fuel film cooling was required to meet the limiting wall temperature at the throat of 500° F during the startup transient at 0.05-0.10 seconds after start-up. This limit was dictated by chamber thermal fatigue life requirements. At nominal conditions of chamber pressure, O/F, and propellant temperature, with 20% film cooling, the steady-state throat wall temperature was maintained at 800° F and the maximum skirt temperature at 1400° F, which were well below the limiting values for 50 hour creep life of 1250° F and 1800° F, respectively.

The effects of chamber pressure, mixture ratio, and percent film cooling on delivered vacuum specific impulse are shown in Figure 41 for the film/regenerative thruster. All of the performance curves have a similar trend of decreasing performance with increasing thruster mixture ratio indicating the gains in thruster specific impulse available by operation at O/F's of 3-4. Comparison of the two upper curves indicates that the performance gain obtained by increasing P_c from 300 to 500 psia is small (about 3 lb_f-sec/lb_m). Comparison of the two curves for 300 psia chamber shows that performance is degraded from

444 lb_f-sec/lb_m with 20% film cooling to 434 lb_f-sec/lb_m with 30% film cooling. The curve for performance obtained with cold propellants shows 435 lb_f-sec/lb_m I_{sp} at an O/F of 4 due to use of cold propellants. The performance value obtained at the nominal design conditions of P_c = 300 psia, 20% film cooling, O/F of 4 and cold propellants meets the contract goal with the thruster reoptimized for low temperature propellants.

The same film/regenerative cooled thruster assembly as discussed above was also evaluated in a series of pulse mode tests. These tests had the objectives of evaluating the transient behavior of the film/regenerative cooled thruster on start-up and shutdown, as well as the thermal soak-back, minimum impulse bit (MIB) capability and specific impulse variation with impulse bit. A durability demonstration of this chamber, under pulse mode operating conditions, was also demonstrated by accumulating 2813 total pulses on the thruster hardware with varying pulse widths. Other operating variables such as chamber pressure, mixture ratio, valve sequencing and film coolant flow rate remained fixed. After the 2813 pulse test series was completed using ambient temperature propellants, approximately 100 pulses of varying pulse widths were conducted using low propellant inlet temperatures, 170° R hydrogen and 280° R oxygen.

The bit specific impulse obtained with the film/regenerative chamber is shown as a function of bit impulse in Figure 42. The minimum impulse bit (MIB) obtained was about 50 lb-sec which occurred at an electrical signal "on" time of about 50 milliseconds. The limit on MIB was caused by the slow response of the valves (approximately 35-45 milliseconds from signal to full open or closed) and is not truly a characteristic of the thruster design. The actual ignition delay time of the thruster was less than 2 milliseconds. The delivered specific impulse decreased as impulse bit was reduced. At an impulse bit of 100 lb-sec, the specific impulse was about 390 lb_f-sec/lb_m.

(3) Film/Dump Cooled Thruster Assembly. - The film/dump cooled thruster assembly shown in Figure 43 consisted of a raised-post triplet injector (fig. 10), a film/dump cooled thrust chamber (fig. 19), two interchangeable igniter assemblies (electrical spark plug or catalytic reactor) (fig. 29), and two Marquardt propellant valve assemblies (similar to fig. 36). Evaluation of this thruster assembly consisted of both steady-state and pulse mode tests to thoroughly evaluate the concept over a range of typical operating conditions.

Approximately 40 steady-state firings of the film/dump cooled thruster assembly were conducted to establish delivered performance and chamber cooling characteristics. After the first few firings to establish the optimum main propellant valve sequencing, the electrical spark plug igniter assembly was replaced by the catalytic igniter assembly, which was used in all subsequent thruster testing. During the steady-state thruster test series, overall thruster mixture ratio was varied from 3.6 to 4.8, film coolant flowrate was varied from 25% to 37% of the fuel, and propellant inlet temperatures were varied from ambient down to 250° R H_2 and 350° R O_2. Figure 44 shows the performance obtained for various mixture ratios at 300 psia chamber pressure, ambient propellants and 32% film cooling. The delivered specific impulse at nominal O/F of 4.0 was 432 seconds. These tests showed dramatically the performance penalty incurred with this chamber design because of the high percentage film coolant needed to keep wall temperatures within the limits specified by chamber life requirements. Also shown on this figure is the delivered impulse for this thruster with propellant inlet temperatures of 250° R for the H_2 and 300° R for the O_2, using 25% film cooling. Results of these tests indicated that the use of lower temperature propellants actually raised the delivered impulse level of this thruster design by allowing use of lower hydrogen film coolant flow rate. The measured throat wall temperature was 1200° F with a film coolant flow of 25% which met the life requirements. The delivered impulse of the chamber meets the performance goal of 435 lb_f-sec/lb_m at an O/F of 4.0 with cold propellants. This thruster hardware was not reoptimized for low temperature operation.

Long duration tests (up to 290 seconds) were completed with 32% film cooling at nominal chamber pressure and mixture ratio wherein the chamber wall temperatures were successfully maintained at 1200° F at the throat and a maximum of 1550° F in the nozzle at an expansion ratio of 10.

Pulse mode testing on the film/dump cooled thruster assembly consisted of duty cycle variations only. Propellant temperatures were ambient and all other thruster operating parameters (P_c, O/F, and film coolant flow) were held at nominal values. The range of thruster "on" and "off" times (duty cycle percentages) covered "on" times of 50, 75 and 100 ms and "off" times of 100 and 200 ms. These values of "on" and "off" times were selected in various combinations to evaluate thruster duty cycles of 25%, 50% and 75%. The MIB achievable with this design was found to be 33 lb-sec. All pulse traces (thrust vs. time curves) were repeatable. The response of the catalytic igniter was about 25 ms during each startup.

A series of igniter-only thruster tests were conducted to establish the MIB capability of the thruster assembly when only the igniter is firing and to determine whether supplemental coolant flow is necessary for the thrust chamber, duct, and nozzle when only the igniter is operating. In these igniter-only tests, no propellant was flowing in either the main injector assembly or the "duct" coolant passages during the time the igniter was firing.

This type of thruster operation could possibly be used for spacecraft reaction control maneuvers that require very low thrust or MIB levels (e.g., 10 lb thrust or 1.0 lb-sec impulse bits).

The catalytic igniter assembly used in the previous pulse mode tests was used for the igniter-only test series with a copper thrust chamber assembly. Tests were made both with and without the use of film coolant in the coolant passages and with both ambient and low temperature propellants.

The results of the "igniter-only" tests clearly indicate that extermely low thrust and

MIB values (as low as 10 lbs and 2 lb-sec, respectively) can be obtained with a 1500 lb thruster assembly using cold propellants and no supplemental film cooling is required in the main thruster coolant passages.

(4) Reverse Flow Thruster Assembly. - The reverse flow thruster assembly, shown in Figure 45, was tested in both pulsing and steady-state altitude tests to verify several design features unique to this particular concept and to determine its steady-state and pulse mode performance. This thruster configuration consisted of the reverse flow combustor/nozzle assembly (fig. 20), and electric spark plug (surface mounted) ignition system (fig. 31), and two ball-type main propellant valves that were thruster mounted.

Steady-state tests were made on the reverse flow thruster to determine the effect on performance and chamber wall temperatures of variations in mixture ratio, chamber pressure, propellant inlet temperatures, and nozzle film coolant flow rates. Figure 46 presents the data obtained at 300 psia chamber pressure with ambient temperature propellants. The performance trend of decreasing I_s with increasing O/F is similar to the other thrusters tested. The reverse flow thruster delivered approximately 440 lb_f-sec/lb_m impulse with up to 6% nozzle film cooling, which exceeds the contract goal performance level. When 9% nozzle film cooling was employed, however, the I_s dropped significantly, as shown in Fig. 46. A nominal design condition of 7.5% film cooling was selected based on the wall temperature required for chamber design life time. Chamber pressure variations from 100 to 500 psia affected the delivered I_s by less than 1%. However, the use of low propellant temperatures (250° R H_2 - 350° R O_2) reduced performance by approximately 30%, indicating that the reverse flow chamber designed for ambient propellants needed to be optimized for operation with cold propellants.

The durability of the reverse flow thruster was demonstrated by one 500-second duration firing at nominal design point conditions. No evidence of thrust chamber or nozzle skirt overheating or other degradation was detected.

Several series of pulse mode firings were conducted with the reverse flow thruster assembly. Pulse width variation sequences were conducted in which thruster "on" times were varied from 30 ms to 1 second and "off" times from 100 ms to 30 seconds. As shown in Figure 47, this thruster design exhibited typical specific impulse vs. electrical "on" time (pulse width) characteristics, i.e., longer "on" times produced higher specific impulse. The MIB achieved with this thruster design was 65 lb-sec at 30 ms "on" time, which was limited by slow valve response. A "pulse train" of 2500 pulses completed the pulse mode test activity for this thruster design. Data from these tests revealed the pulses to be very repeatable in terms of shape and total impulse bit.

Integrated Thruster Assembly

The integrated thruster assembly (ITA), shown in Figure 48, is a new design under investigation by Aerojet[23] in a contract placed in 1972. The ITA represents the culmination of all of the component technology for gas/gas hydrogen-oxygen thrusters into an integrated, flightweight design. This thruster will be fabricated and extensively tested during the next year to verify its ability to fully meet the design requirements set forth in Table I. In the ITA concept, greater emphasis will be placed upon meeting vehicle interface and Shuttle operational requirements than heretofore. For example, the thruster will not use copper material downstream of the throat in order to assure its ability to meet the stringent reentry heating and life requirements. Also, lightweight materials and design features, such as a welded-on injector, will be employed to assure a truly flightweight design. The test program will emphasize evaluation of chamber life and operational problems such as heat soakback from the thruster to the igniter and valves. A series of 50,000 pulse firings are planned plus a minimum of 5,000 full thermal cycles, which will exercise the thruster to 10-20% of its design life.

Completion of the ITA program will provide a strong technology background for gas/gas hydrogen-oxygen thrusters so that their application to the Space Shuttle or other vehicles can proceed with a minimum of development risk.

Concluding Remarks

The NASA-Lewis sponsored technology program on hydrogen-oxygen thrusters for Space Shuttle ACPS has included extensive work on injectors, thrust chambers, ignition systems, valves, and thruster assemblies. A summary of the technology status in each area is given below. Also provided are comments pertaining to the application of hydrogen-oxygen auxiliary propulsion systems.

1. Injectors - Of the injector types tested, the extent to which each concept was tested, and the cooling concept each was tested with, the concentric tube and premix types best demonstrated the attributes of performance, durability, and acceptable heat transfer to the thrust chamber wall. However, acceptable combustion efficiency and stable combustion were obtained with all of the gas/gas injectors tested.

2. Thrust Chambers - A variety of thrust chamber designs were evaluated which used combinations of regenerative, film, and dump cooling. All of these designs were found to be capable of meeting the performance requirements at operating conditions consistent with the predicted life requirements. The film cooled designs provide a lighter weight, potentially lower cost chamber but are also somewhat lower in delivered specific impulse than a regenerative design.

3. Ignition Systems - The spark/torch igniter with air gap plug, high mixture ratio core, and capacitance discharge exciter was most successful in providing rapid, reliable ignition at all test conditions. The plasma/torch and resonance tube igniters are also promising ignition methods, but require further testing and optimization before application. The catalytic igniter with downstream oxygen injection meets the overall thruster response requirements but is limited in its ability to ignite low temperature propellants (below 250° R).

4. Valves - For the requirements of the gas/gas thruster, the poppet type, propellant gas actuated valve concepts were perferred. Both the captive plastic seal and flat polyimide seat on tungsten carbide poppet appear capable of meeting the cycle life, leakage, and response specifications. These concepts are capable of providing lightweight, compact, and reliable valves suitable for flight application.

5. Thruster Assemblies - The thruster assemblies tested provided delivered specific impulse ranging from 432 to 447 lb_f-sec/lb_m at 300 psia chamber pressure and an O/F of 4 with ambient temperature propellants. However, if vehicle volume and weight allocations could permit the use of slightly larger hydrogen tanks, operation of any of the thruster assemblies at mixture ratios (O/F) at values lower than 4.0 would increase the delivered specific impulse values significantly (see figs. 38, 41, 44 and 46). The regenerative chamber delivered highest performance and the film cooled chambers the lowest for the same design life. The best pulsing performance of 412 lb_f-sec/lb_m at 50 lb-sec impulse bit was also obtained with a regenerative chamber.[10]

6. System Considerations - There are presently no critical gas/gas H_2-O_2 thruster technology problem areas outstanding that would prevent the use of this system. After assessment of the above results, it was concluded that further work was required to optimize thruster performance with cold (250° R H_2 - 375° R O_2) propellants and demonstrate the long life required of the thrusters has yet to be experimentally proven.[23] Some of the components of the gas/gas feed system, such as the turbopumps and gas generator/heat exchangers are being investigated in technology contracts sponsored by MSC and MSFC, as previously mentioned, and this work has not yet been completed. The liquid/liquid H_2-O_2 system has the potential of providing a simpler, lighter weight system and deserves further consideration. Liquid/liquid thruster technology problems, such as ignition and pulse mode operation, are presently under investigation.[6] Additional technology work is needed on the system components, such as liquid accumulators, vacuum jacketed lines, and pumps.

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TABLE I. - THRUSTER OPERATING CONDITIONS AND DESIGN REQUIREMENTS

	Nominal	Testing Range
Thrust	1500 lb	
Chamber pressure	300 psia	100-500 psia
Mixture ratio (O/F)	4.0	3.0-5.0
Nozzle expansion ratio	40:1	
Propellant inlet temperatures	250° R-hydrogen 375° R-oxygen	200-600° R Sat.-600° R
Propellant inlet pressure (to valve)	400 psia - both propellants	
Total life capability	50 hr	
Total number of firings	500,000 pulses, plus 25,000 full thermal cycles	
Minimum impulse bit (MIB)	50 lb-sec	
Response (from signal to 90% thrust)	50 milliseconds	
Specific impulse:		
Steady-state	435 sec	
Pulsing (at MIB)	400 sec	
Reentry heating:	30 minutes exposure/mission to the following temperatures	
At nozzle	1800° F	
At chamber throat	1200° F	
Weight (of thruster assembly) including valves	25 lb	

TABLE II. - IGNITION SYSTEM DESIGN REQUIREMENTS

Propellant inlet temperatures		
Hydrogen	250° R nom.	150-600° R range
Oxygen	375° R nom.	Saturated to - 600° R range
Propellant inlet pressures		375-425 psia (both propellants)
Igniter body temperature		200-600° R
Igniter response		10 millisec (first propellant flow to 90% igniter P_c)
Environmental pressure		1×10^{-8} mm Hg to 14.7 psia
Energy input		Minimize
Radio frequency interference (RFI)		Conform to MIL-STD-461
Life		
Operating life		50 hr total, 30 min/mission for 100 missions
Overall life		10 years
Cycle life		1,000,000 cycles with maintenance, 10,000 cycles/mission
Input voltage		28±4 volts DC

TABLE III. - SPARK/TORCH IGNITER/EXCITER NOMINAL CONDITIONS(14)

Igniter chamber pressure (before thruster ignition)	200 psia
Igniter mixture ratio (O/F)	1.3
Igniter flowrate	0.04 lb/sec
Power input	
Operating	8 W (max.)
Control	1 W (max.)
Spark energy	22 millijoules/spark
Spark voltage	10 kV
Spark rate	200/sec
Response, signal to first spark	0.015 sec
Spark plug type	Dual surface gap

TABLE IV. - COMPARISON OF IGNITION SYSTEM (CONCEPT) PERFORMANCE

Igniter type	Characteristics				
	Spark rate	Igniter response (overall)	Power requirement	R.F.I. level	Stage of development
Spark plug(15) torch	500 SPS* at 20 kV	≅10 ms	28 V.D.C. at 3 amps	Meets mil-STD-461A	Operational
Pulsed plasma(15) torch	250 SPS at 5 kV	≅10 ms	28 V.D.C. at 3 amps	Meets mil-STD-461A	Advanced R&D
Catalytic(18) torch	------	≅25 ms	---------	--------	Advanced R&D
Direct S.P.(13) type	250 SPS at 15-20 kV	≅10 ms	28 V.D.C.	Not tested	Operational
Resonance(14) torch type	------	≅25 ms	---------	--------	R&D

*SPS - sparks per second.
Operating conditions: Ambient temperature propellants 350 psia chamber pressure.

TABLE V. - PROPELLANT SHUTOFF VALVE DESIGN REQUIREMENTS

Valve type	Single or bipropellant
Propellants	Hydrogen (gaseous) Oxygen (gaseous)
Operating temperature range	200° R to 850° R
Propellant temperature range	
Hydrogen	200° R to 600° R
Oxygen	250° R to 600° R
Propellant pressures at valve inlet	400±50 psia
Pressure drop (maximums)	
Fuel	15 psi at 0.69 pps* and 540° R
Oxidizer	15 psi at 2.76 pps and 540° R
Actuation types	a. Pneumatic b. Hydraulic c. Electrical
Opening and closing Response	10-15 milliseconds, total response time from signal to open or closed no greater than 30 milliseconds
Internal leakage	100 scc/hr with gaseous helium at operating pressure and temperature
External leakage	1×10^{-6} scc/sec with gaseous helium at operating pressures and temperature
Operating life (goal)	1,000,000 cycles
Size and weight	Design to minimize
Proof pressure	1.5 times operating pressure
Burst pressure	1.33 times proof pressure

*pps - mass flow in pounds per second.

TABLE VI. - THRUSTER ASSEMBLY DESIGN AND PERFORMANCE COMPARISON

Thruster types	Operating characteristics					
	Steady-state delivered performance	Response in milliseconds (0-90% thrust)	M.I.B. lb-sec	Performance at M.I.B. (I_{sp})	Design life (a) critical location (b) life (pulses)	Demonstrated life (a) total pulses (b) total time
Regenerative/ dump(10) (with coaxial injector)	447 lb-sec/lb	40 ms	48 lb-sec	≅412 lb-sec/lb	(a) Throat-wall transition (b) 8.0×10^6 pulses	(a) 2762 pulses (b) 325 sec
Film/ regenerative (11) (with premix triplet injector)	444 lb-sec/lb	≅50 ms	50 lb-sec	≅350 lb-sec/lb	(a) Throat (b) 10^6 pulses	(a) approx. 2813 pulses (b) approx. 250 sec
Film/ dump(12) (with triplet injector)	432 lb-sec/lb	≅45 ms	33 lb-sec	≅390 lb-sec/lb	(a) Nozzle Div. Section (b) 10^6 pulses	(a) 275 pulses (b) approx. 300 sec
Reverse/ flow(13) (with vortex-cup injector)	440 lb-sec/lb	≅30 ms	≅65 lb-sec	N/A	(a) Nozzle liner (b) 900,000 pulses	(a) 3625 pulses (b) N/A

Operation conditions: Amb. temp. propellants - 540° R
Thrust - 1500 lb
P_c - 300 psia
O/F - 4.0
ϵ - 40:1

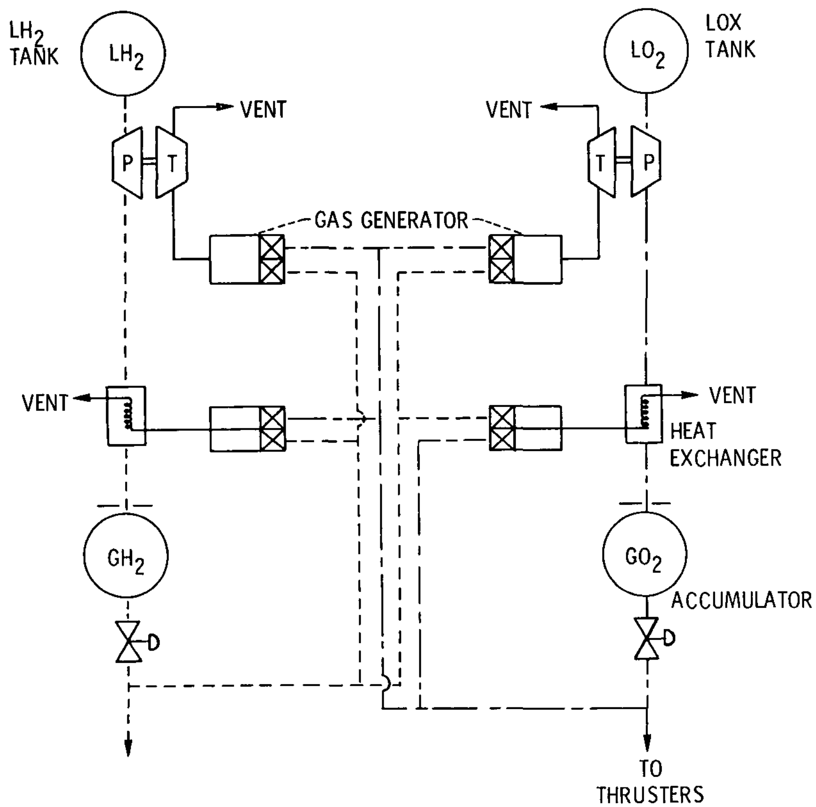

Figure 1. - GO_2/GH_2 system.

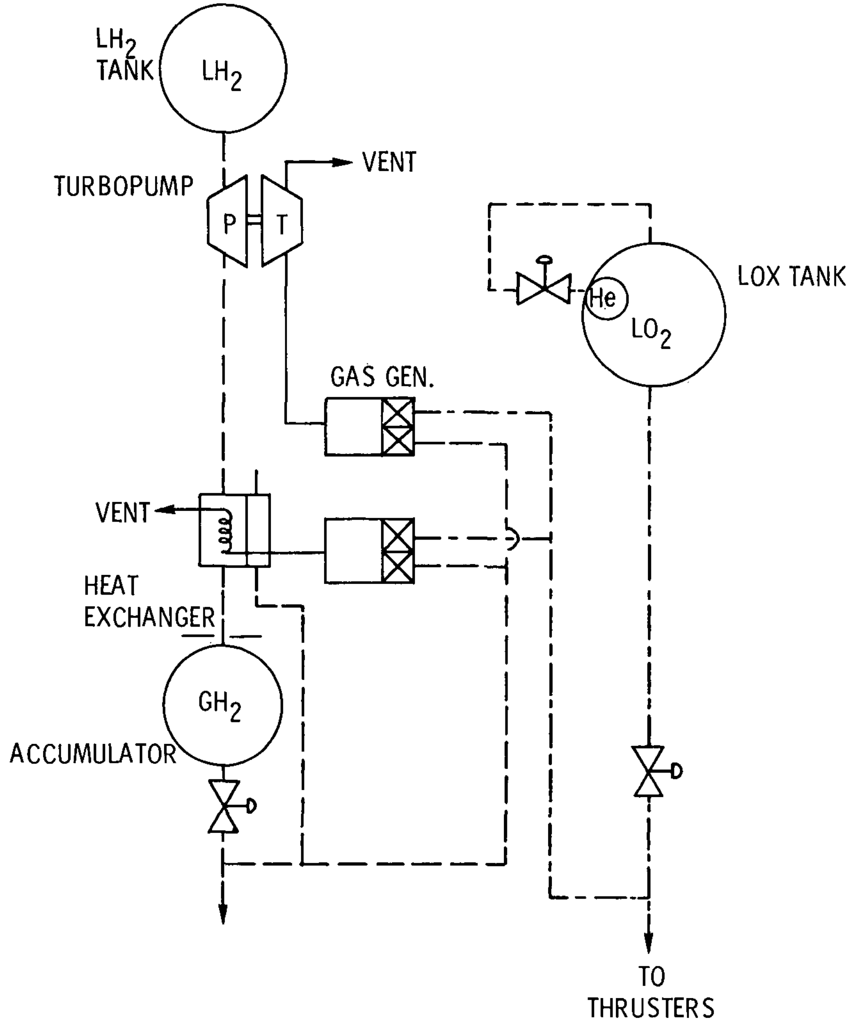

Figure 2. - LO_2/GH_2 system.

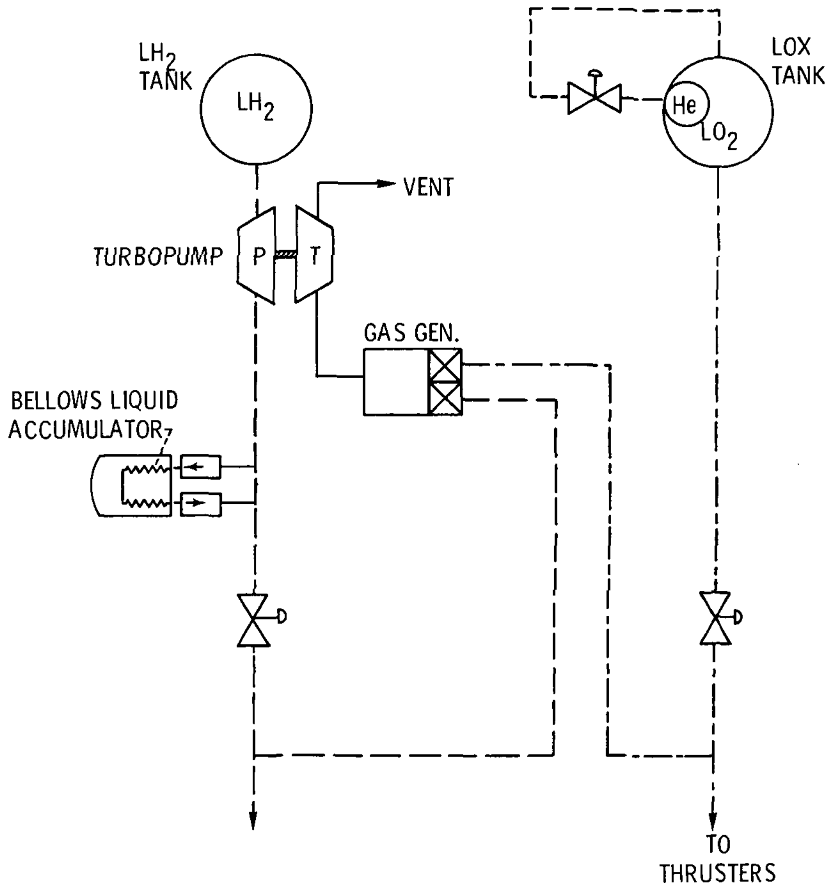

Figure 3. - LO_2/LH_2 system.

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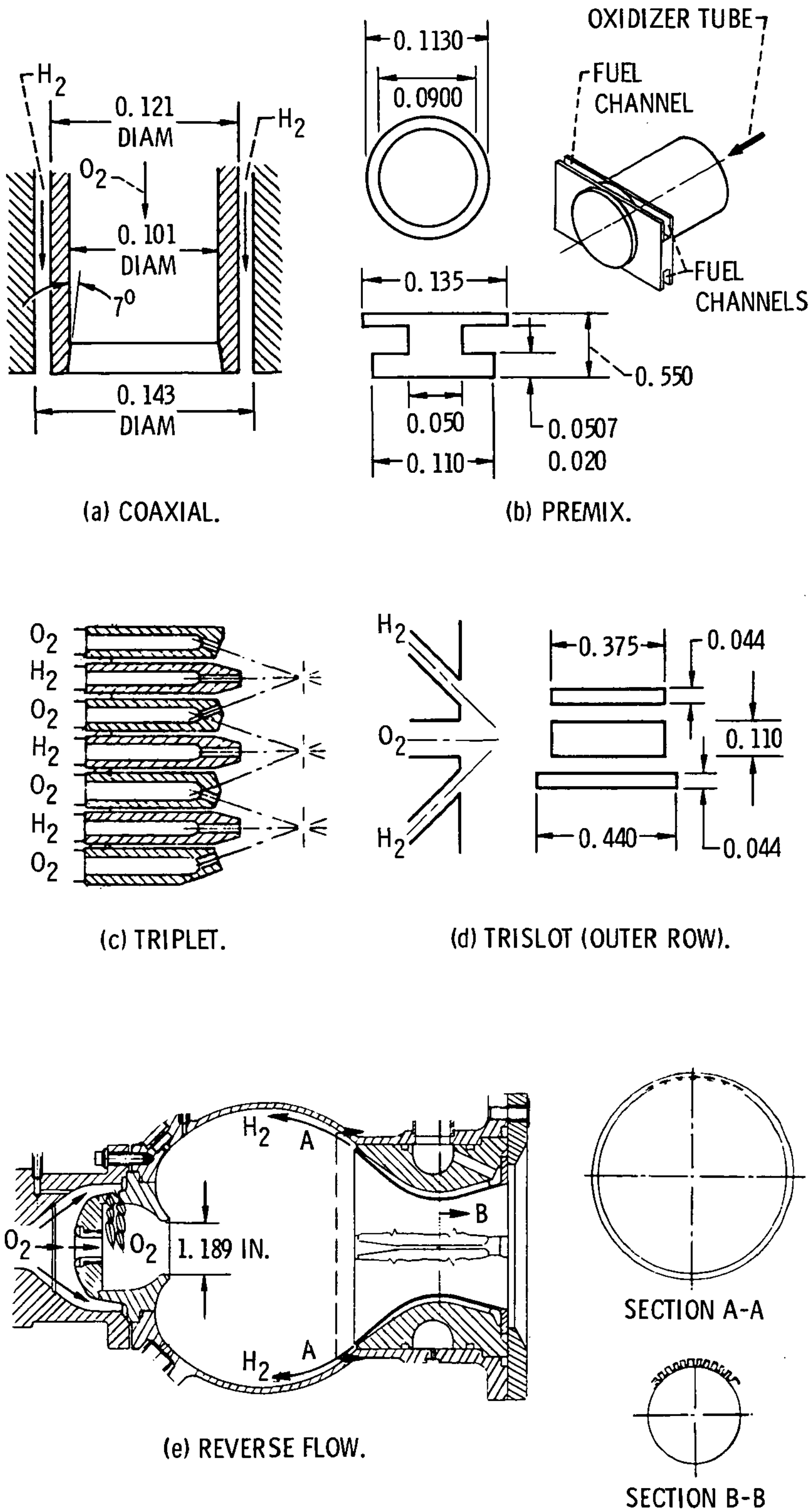

Figure 4. - Injector element details.

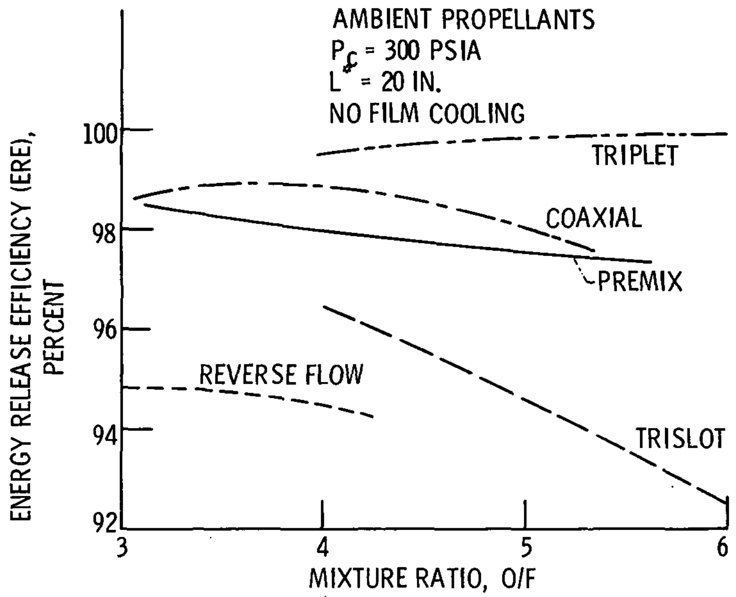

Figure 5. - Comparison of injector performance characteristics.

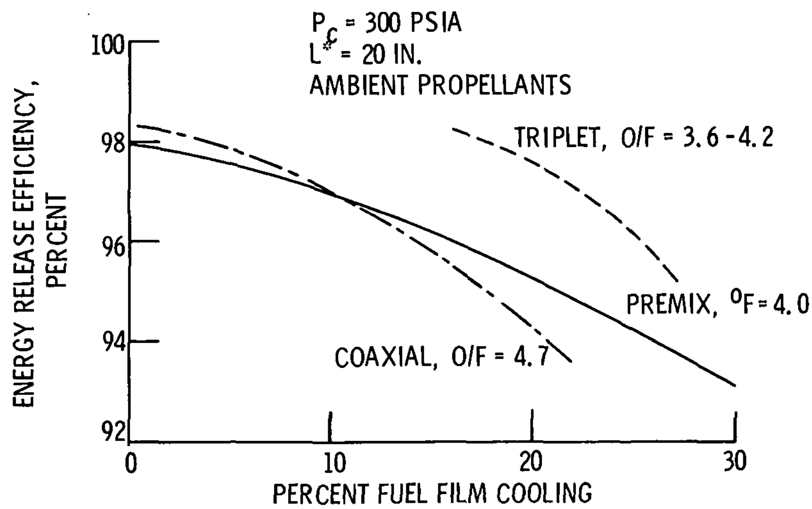

Figure 6. - Effect of film cooling on injector performance.

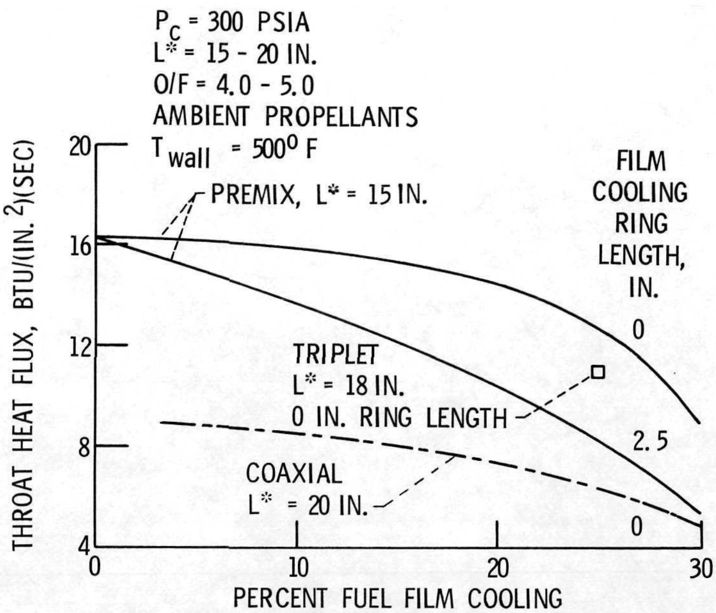

Figure 7. - Effect of film cooling on throat heat flux.

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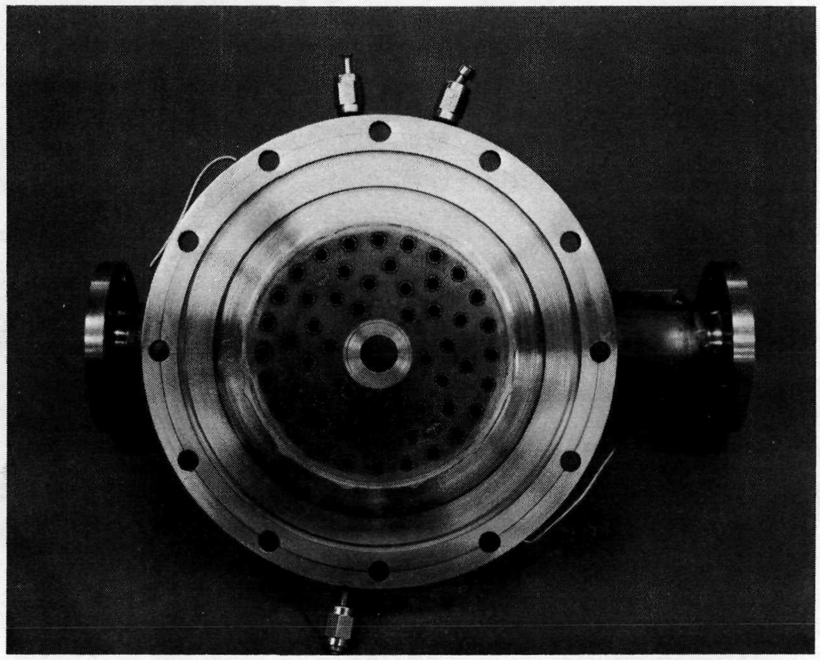

Figure 8. - Concentric tube (coaxial) injector.

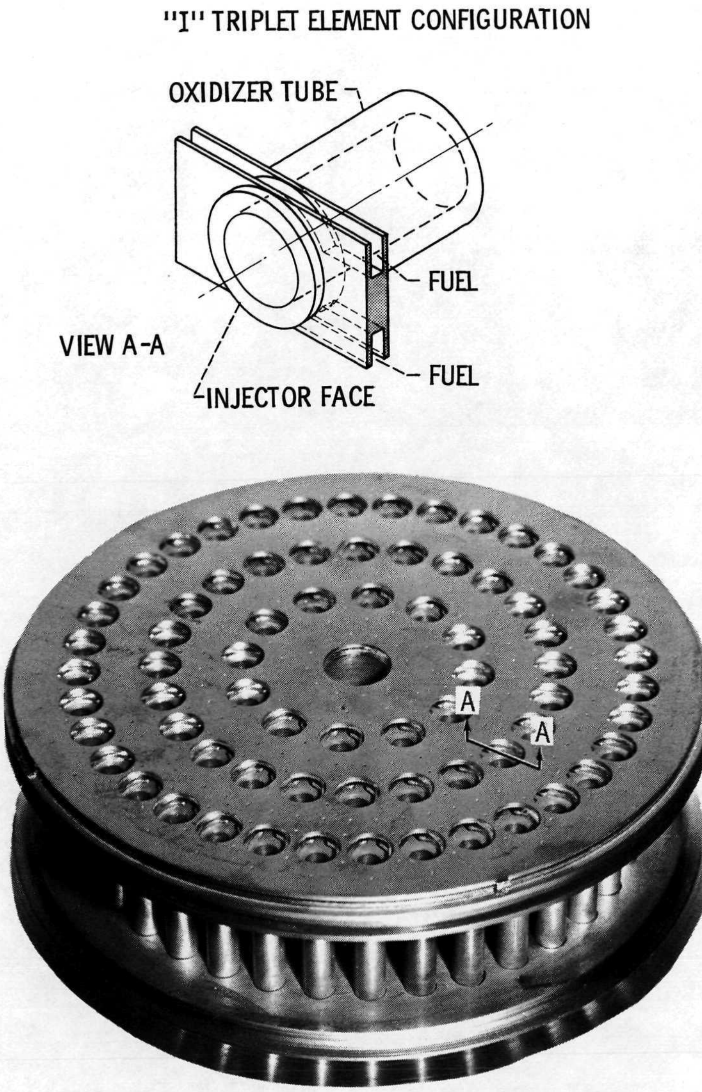

Figure 9. - Premix Injector assembly.

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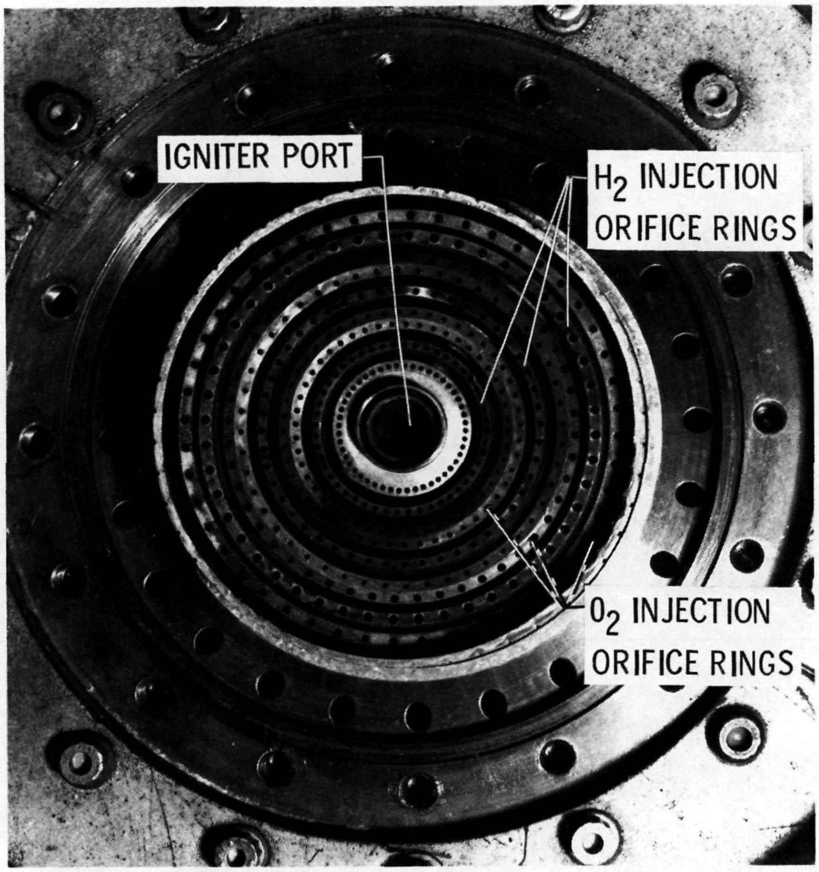

Figure 10. - Raised post triplet injector assembly.

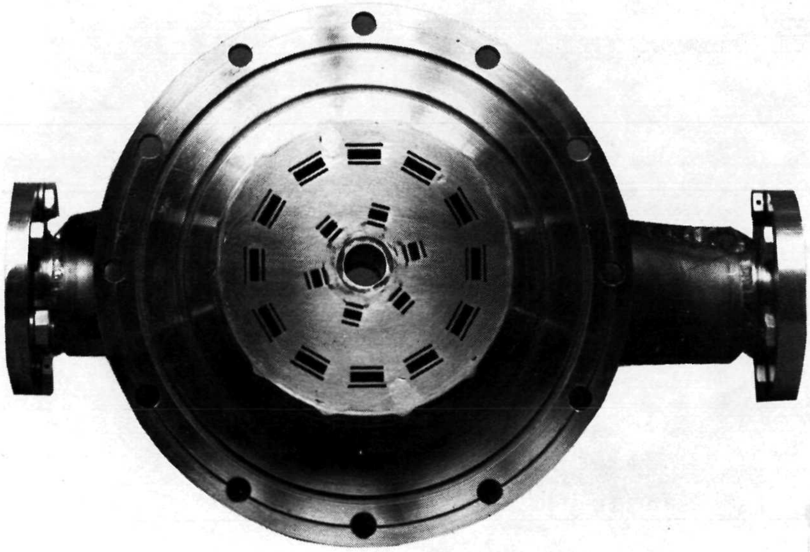

Figure 11. - Trislot injector assembly.

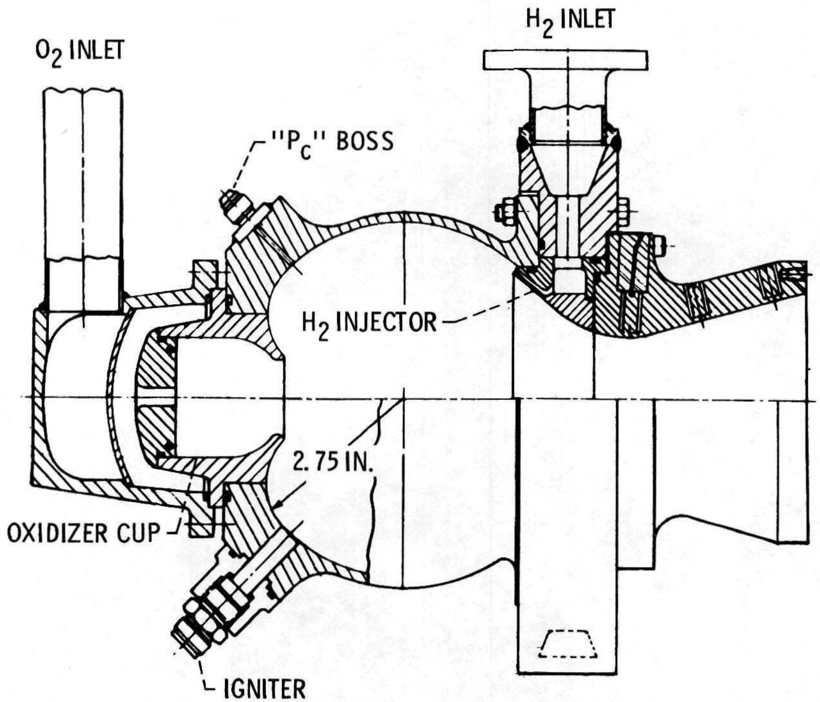

Figure 12. - Reverse Flow injector/thrust chamber assembly.

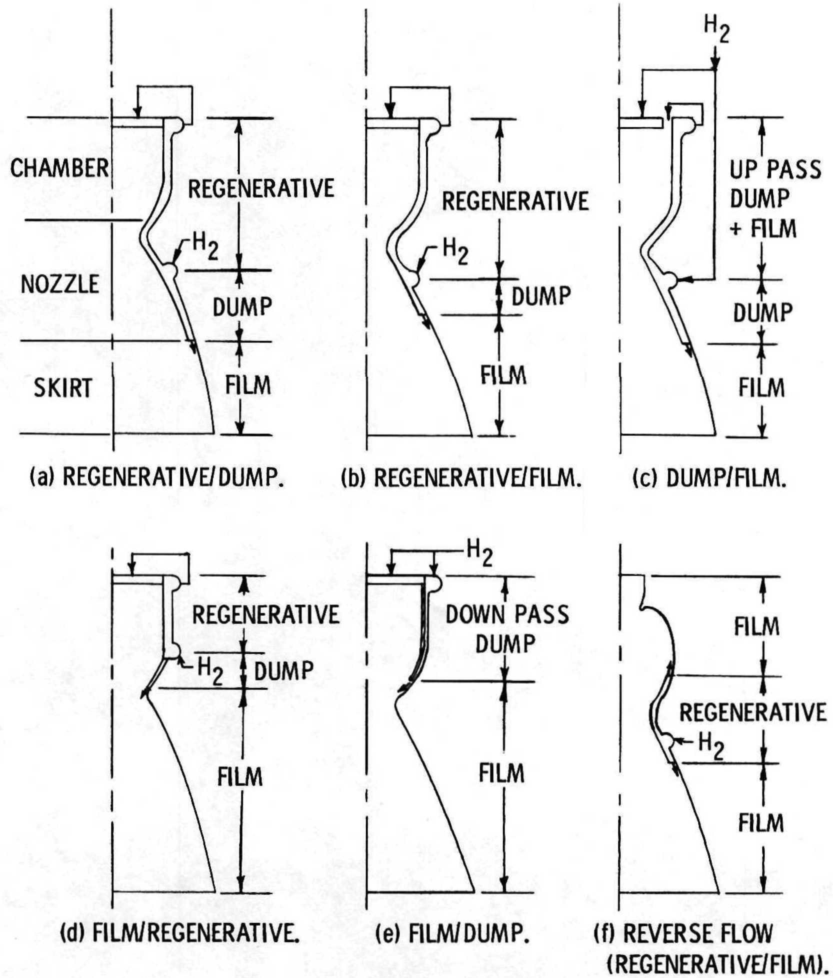

Figure 13. - Thrust Chamber Cooling methods.

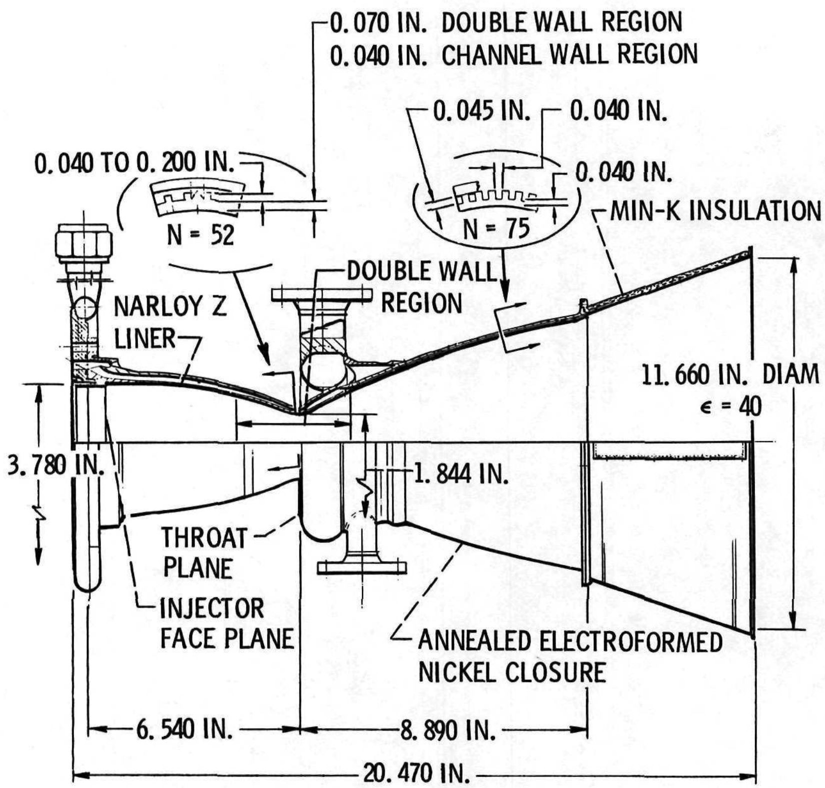

Figure 14. - Rocketdyne Regenerative/Dump Cooled thrust chamber.

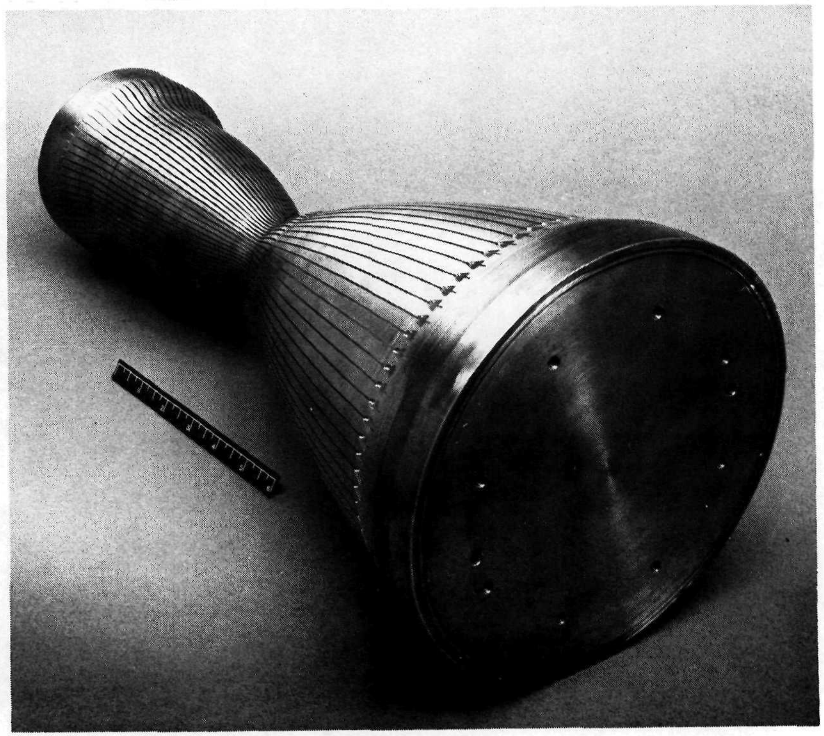

Figure 15. - Rocketdyne regenerative/dump cooled chamber Narloy-z-liner.

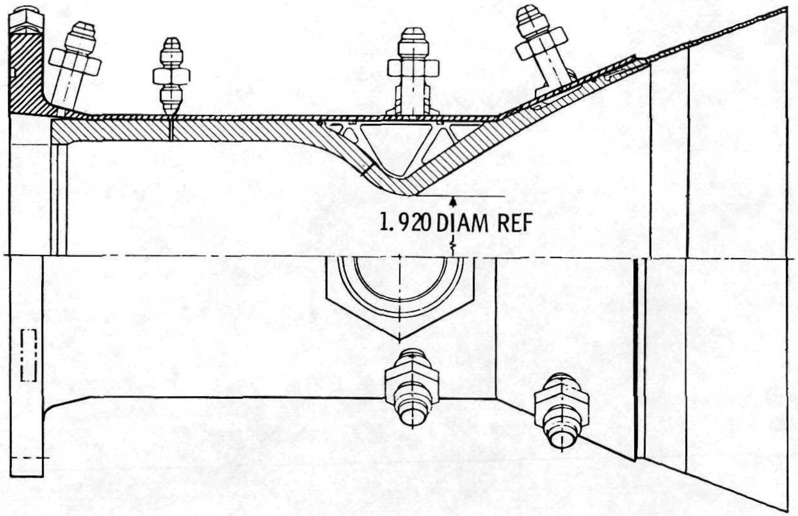

Figure 16. - Regenerative/Film cooled chamber section.

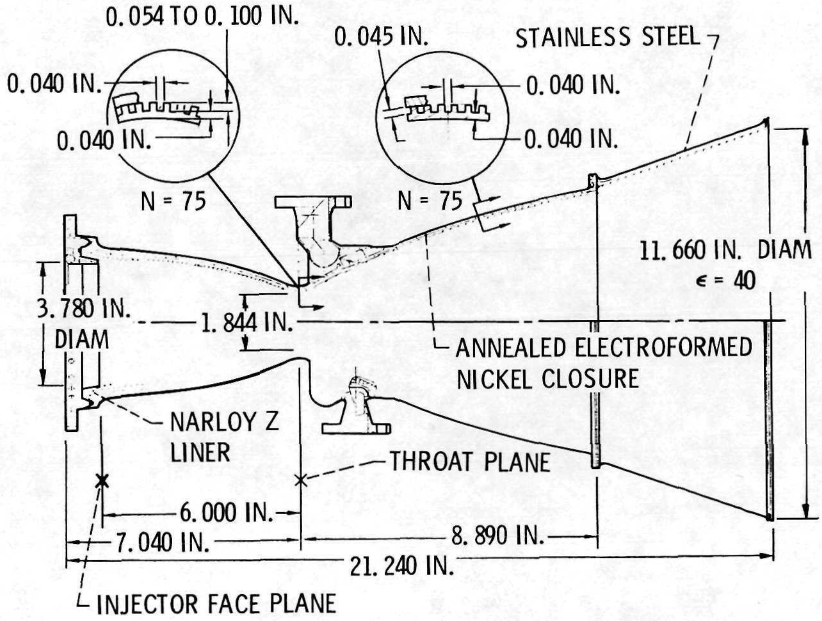

Figure 17. - Rocketdyne Dump/Film Cooled thrust chamber.

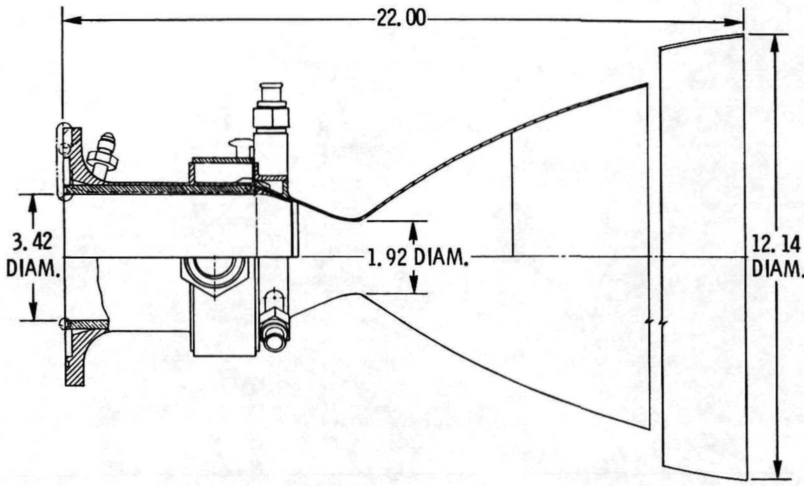

Figure 18. - Aerojet Film/Regenerative cooled thrust chamber.

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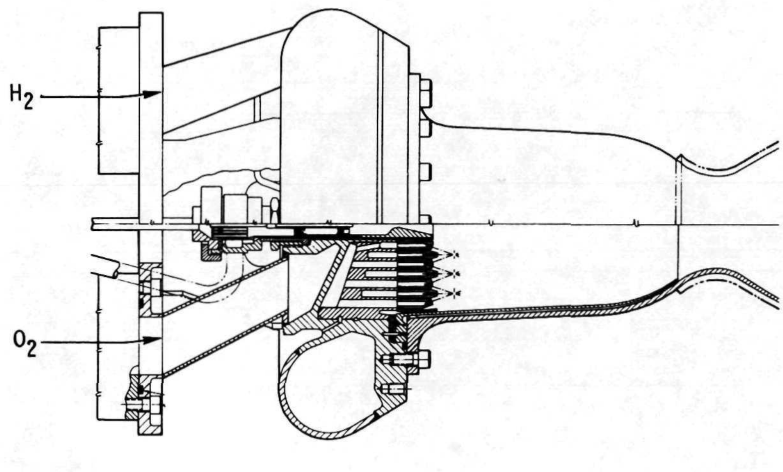

Figure 19. - TRW Film/Dump (Duct) cooled thrust chamber assembly.

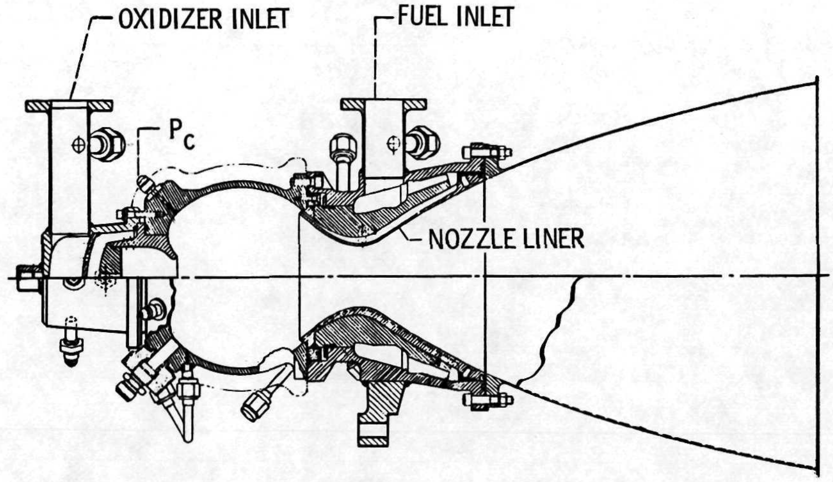

Figure 20. - Bell Reverse Flow thrust chamber assembly.

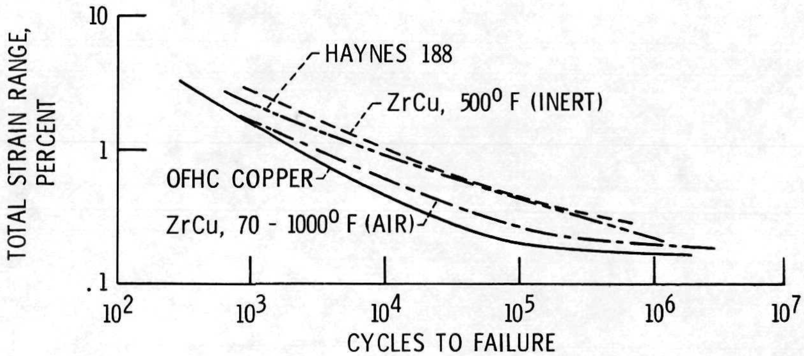

Figure 21. - Total strain versus fatigue life.

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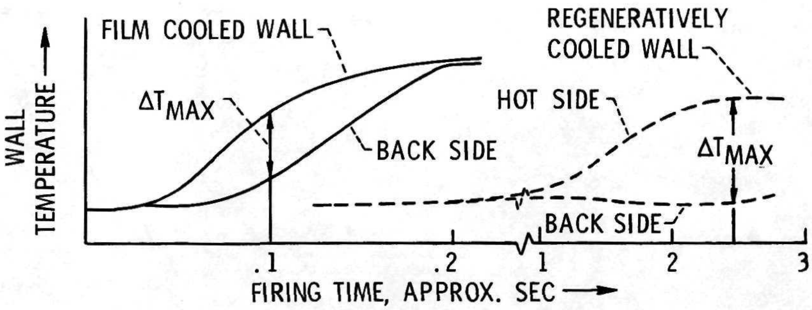

Figure 22. - Transient wall temperature behavior of film and regeneratively cooled chambers.

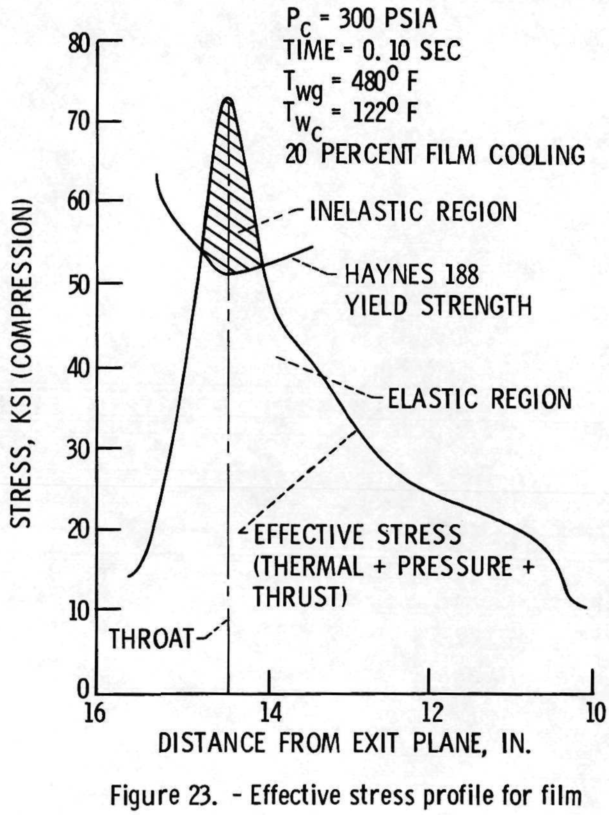

Figure 23. - Effective stress profile for film cooled chamber.

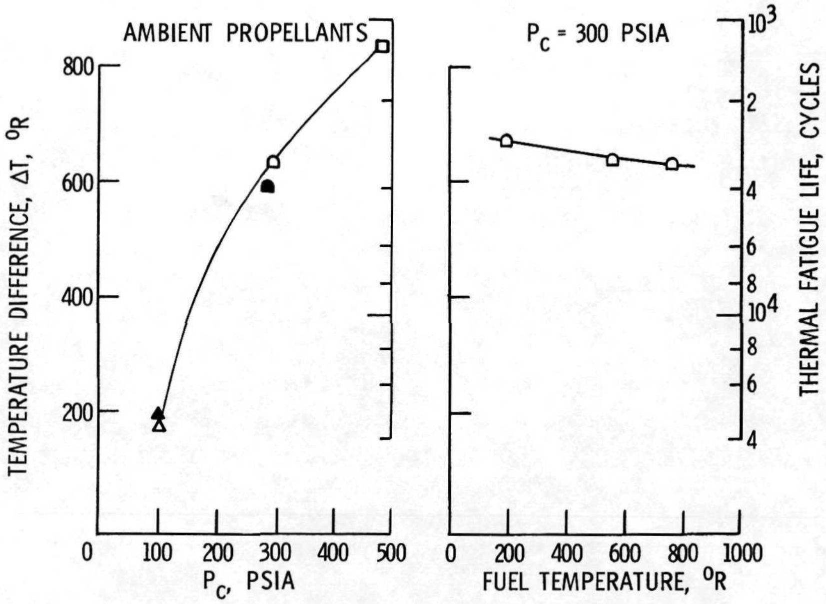

Figure 24. - Regenerative chamber life and wall temperature difference. Maximum temperature 0.6 inches upstream of throat.

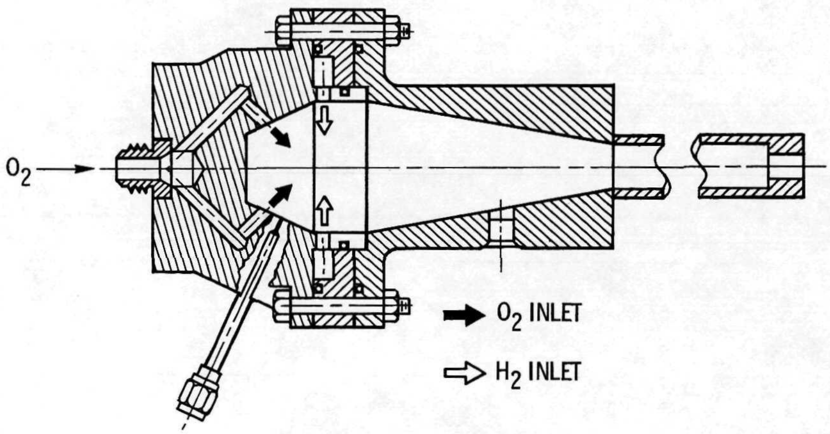

Figure 25. - Rocketdyne Spark/Torch igniter.

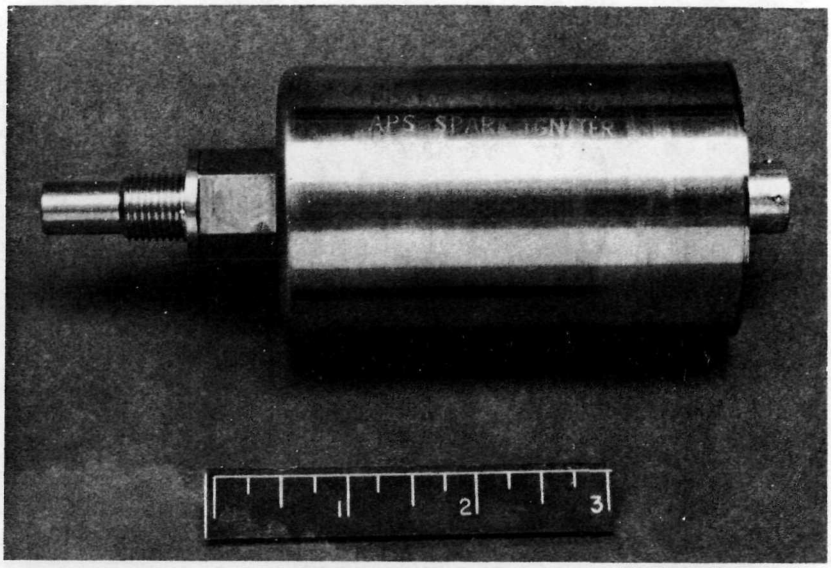

Figure 26. - Prototype integral sparkplug/exciter igniter assembly.

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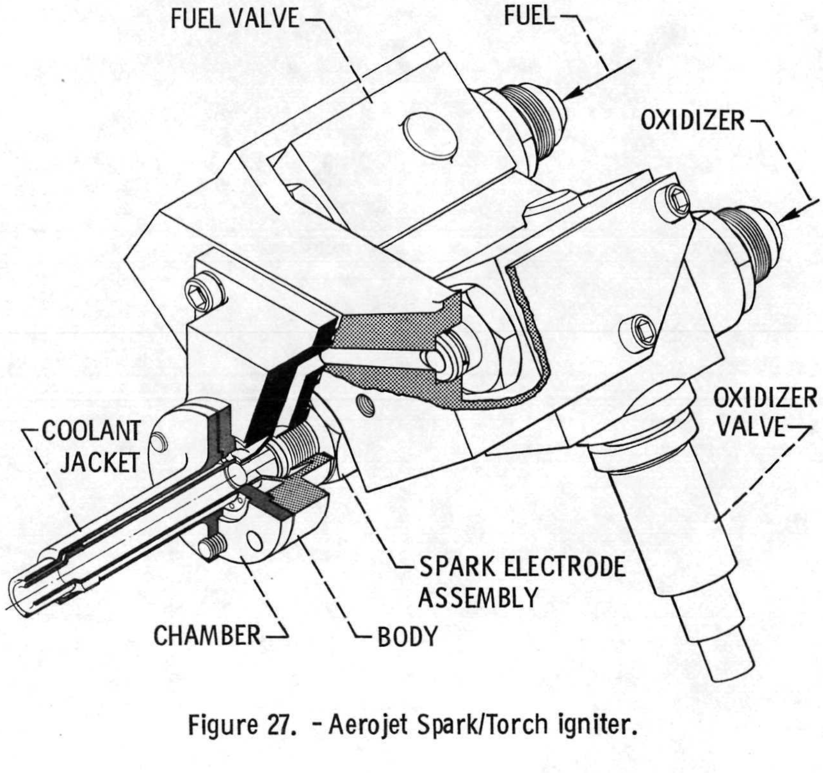

Figure 27. - Aerojet Spark/Torch igniter.

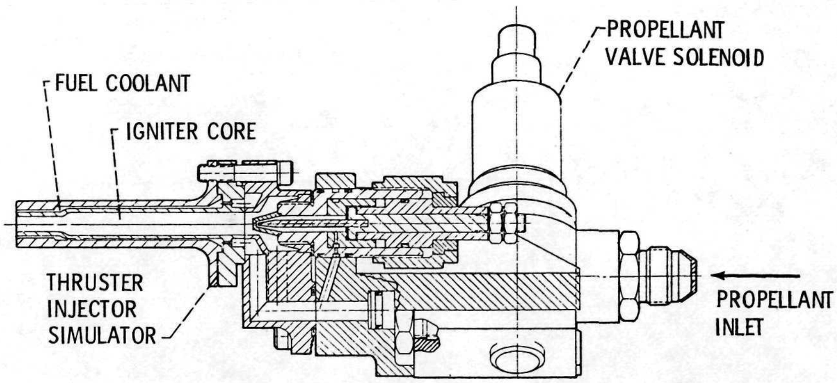

Figure 28. - Aerojet Plasma/Torch igniter.

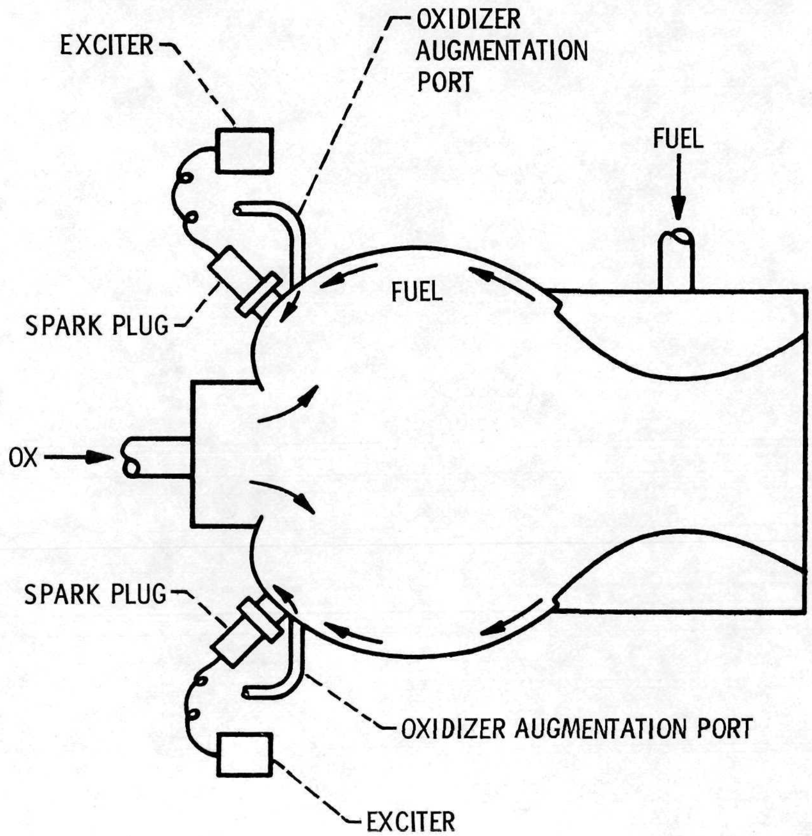

Figure 29. - Direct Spark Plug ignition.

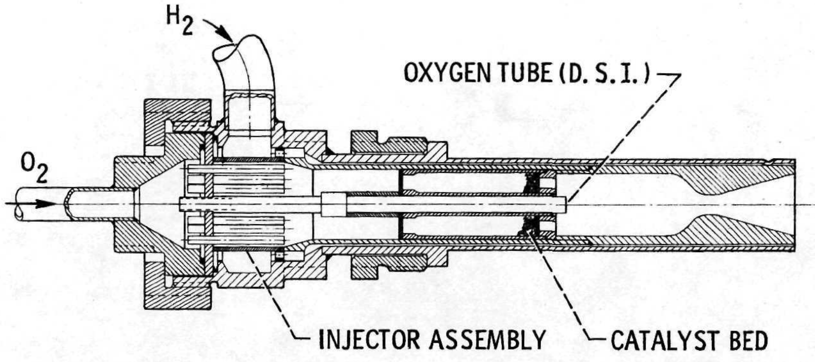

Figure 30. - TRW Catalytic igniter with downstream injection.

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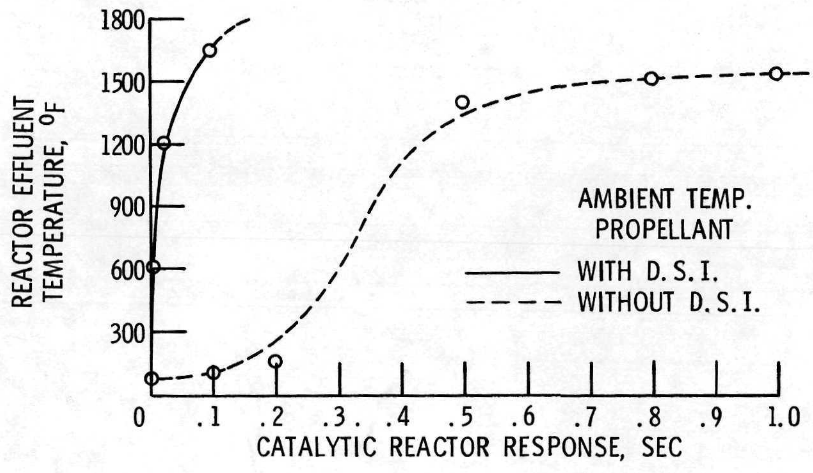

Figure 31. - Catalytic igniter response comparison.

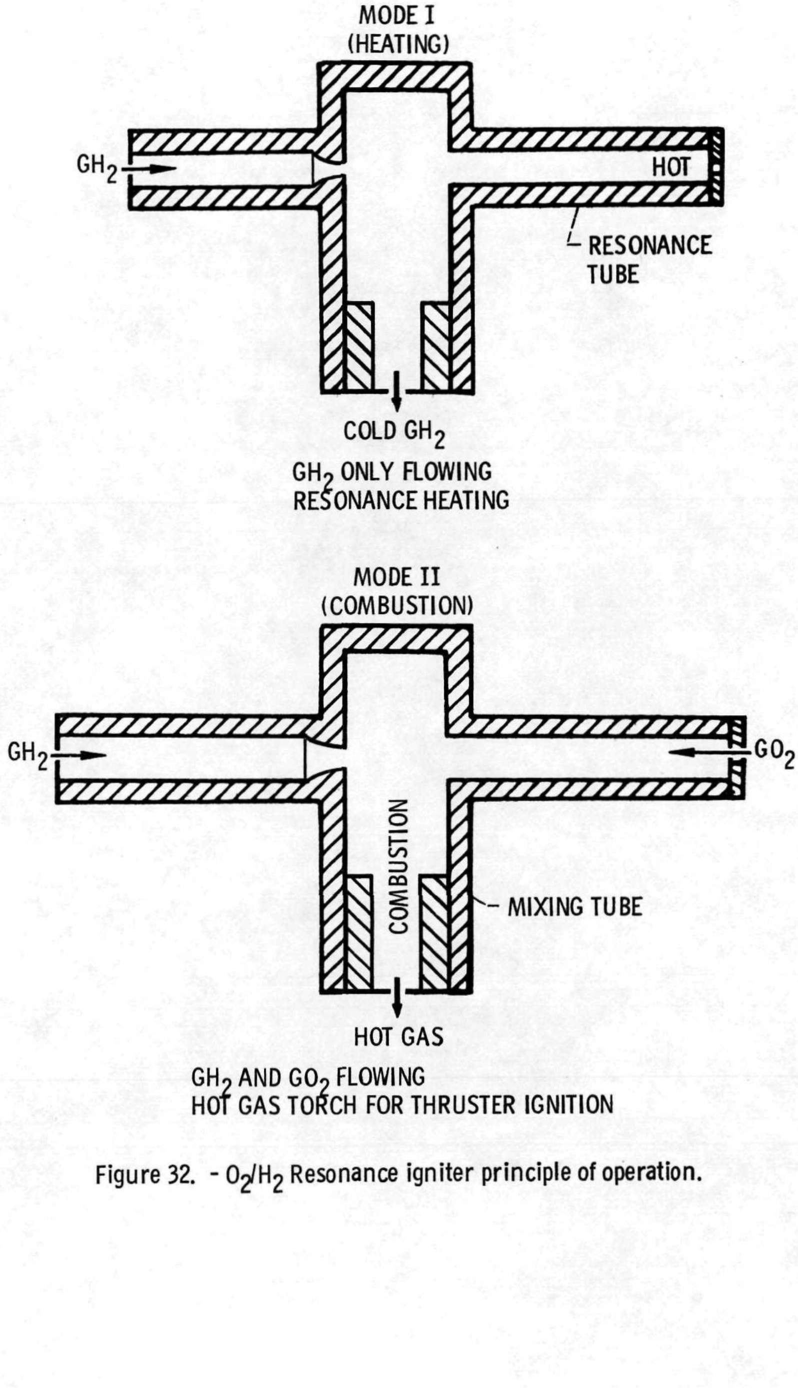

Figure 32. - O_2/H_2 Resonance igniter principle of operation.

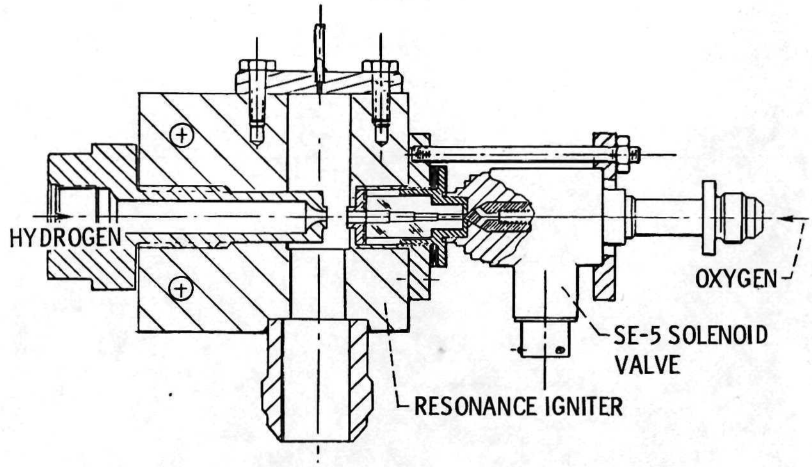

Figure 33. - Optimized Auto igniter and oxygen valve assembly.

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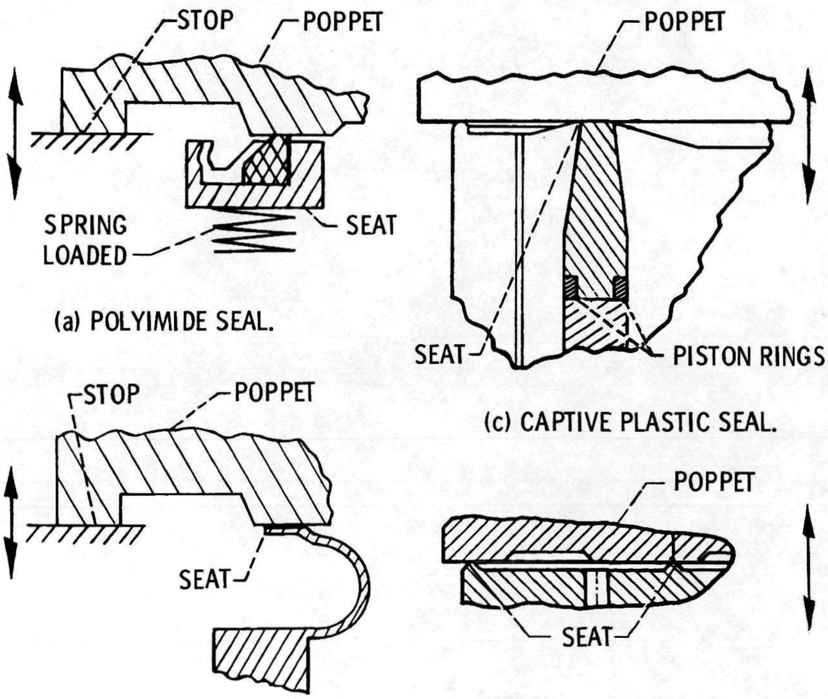

Figure 34. - Selected Valve Seal concepts.

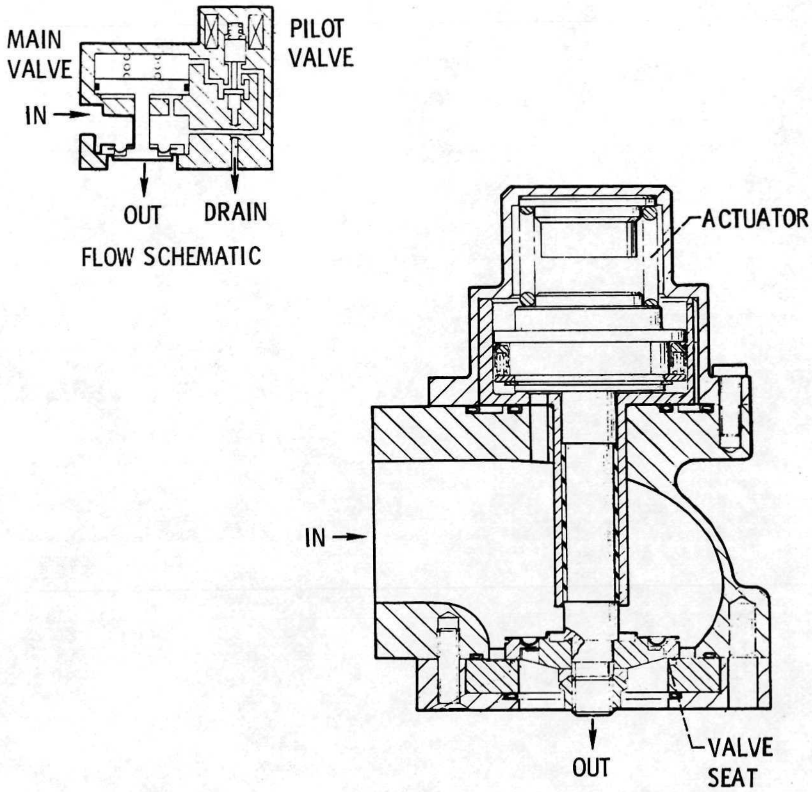

Figure 35. - Rocketdyne ACPS valve design.

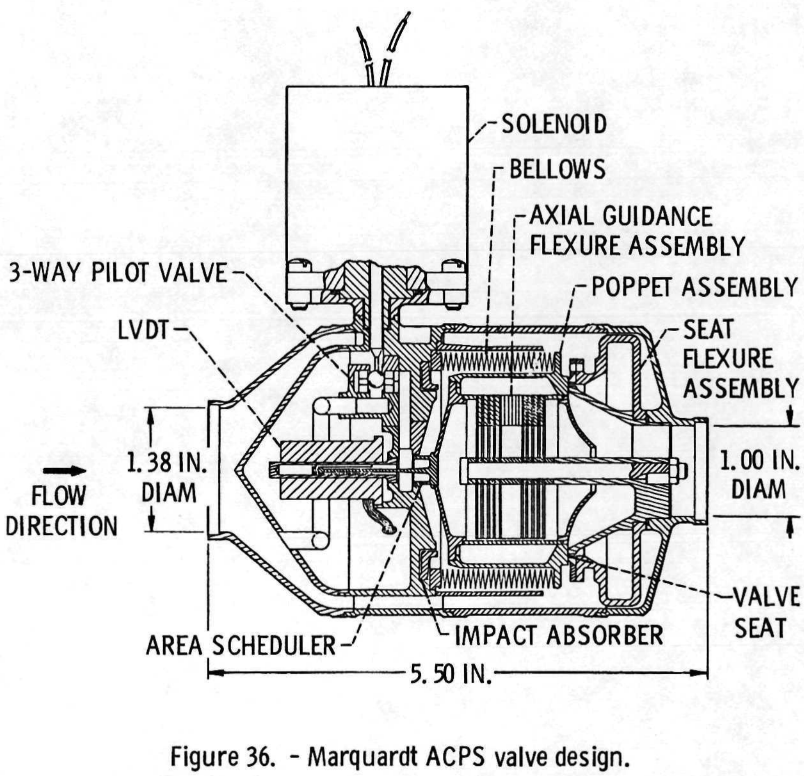

Figure 36. - Marquardt ACPS valve design.

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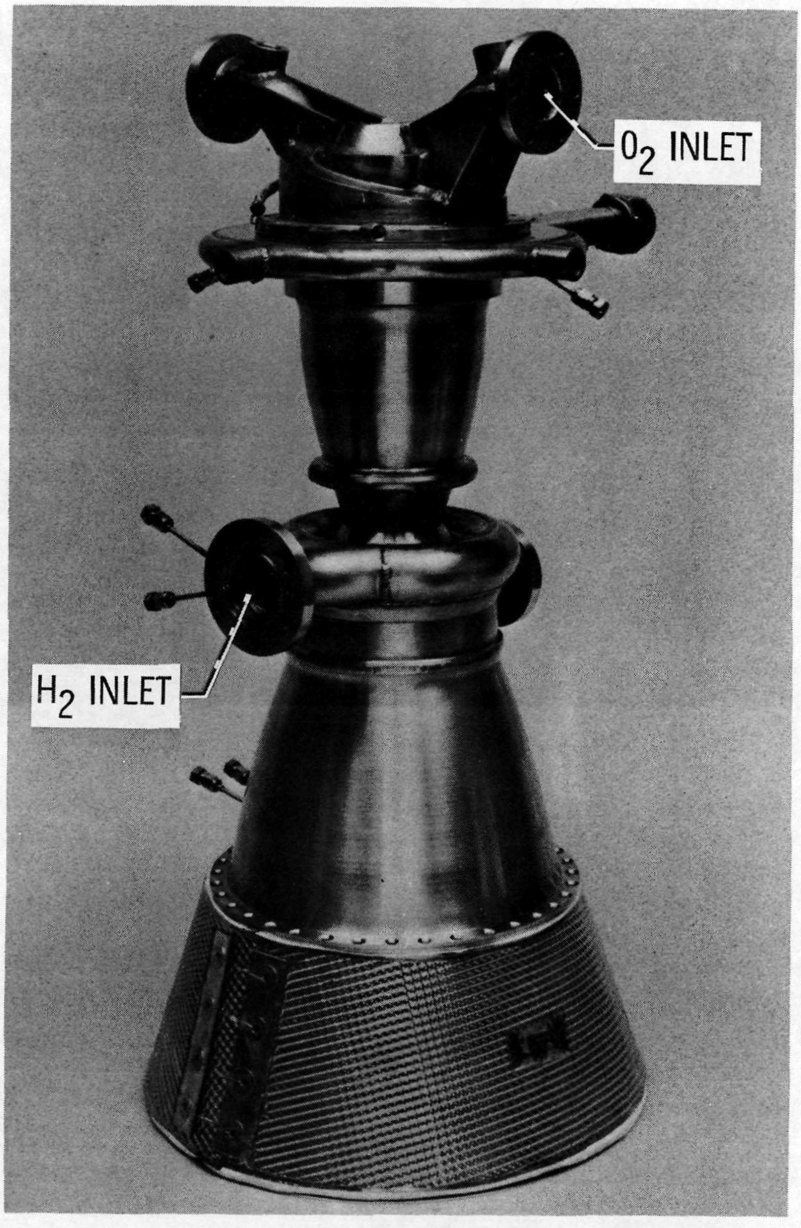

Figure 37. - Regenerative/dump cooled thruster assembly.

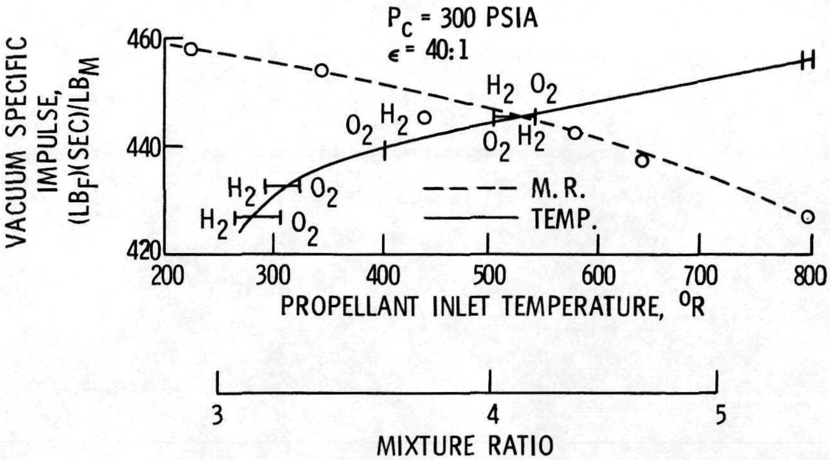

Figure 38. - Regenerative/dump cooled thruster performance characteristics.

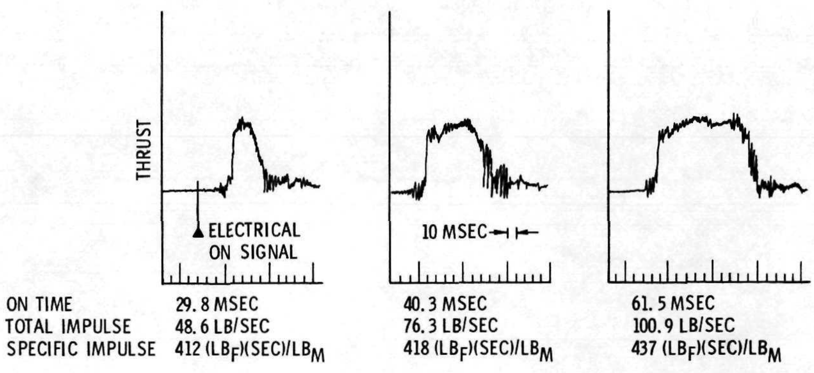

Figure 39. - Pulse performance data for regenerative/dump cooled thruster.

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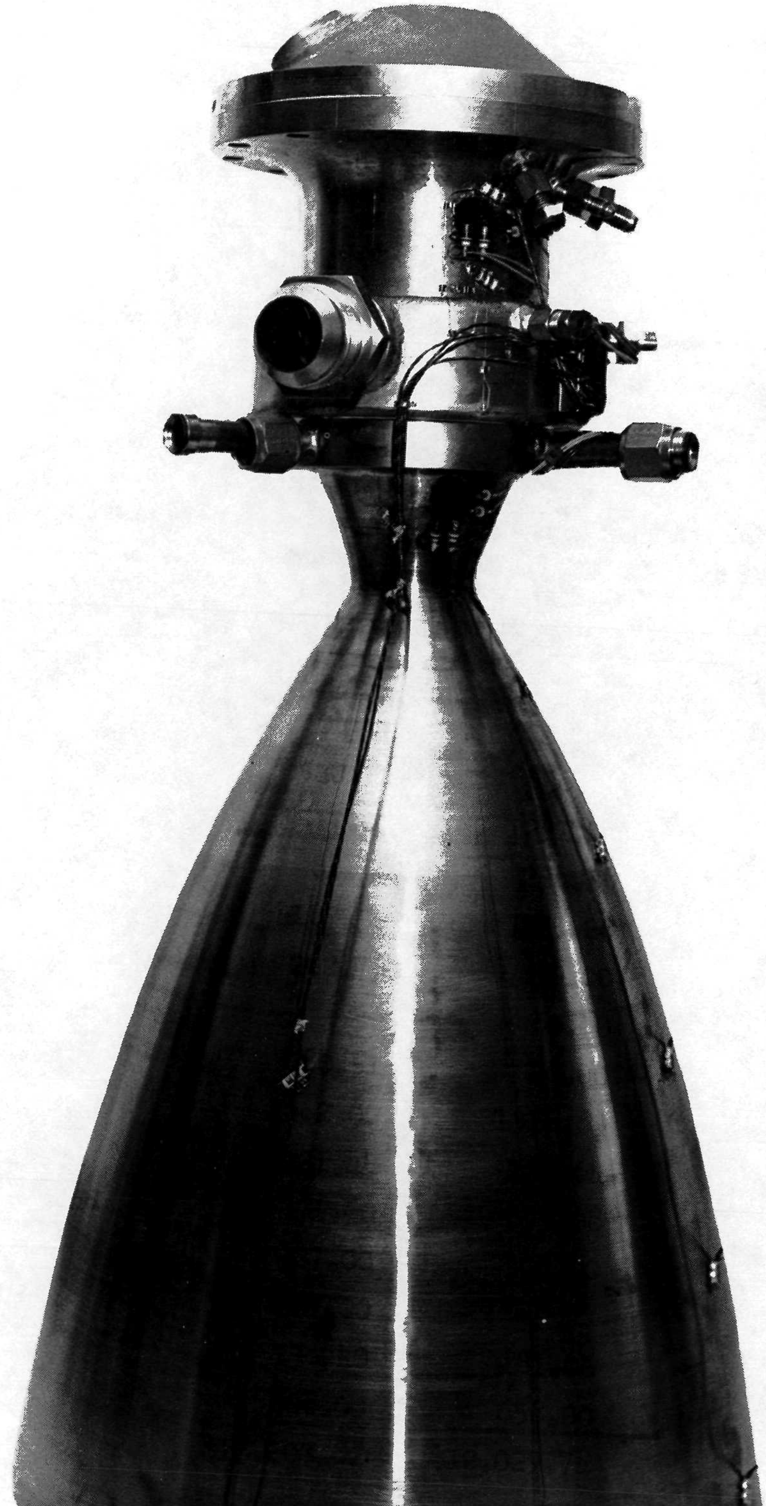

Figure 40. - Film/regenerative cooled thrust chamber.

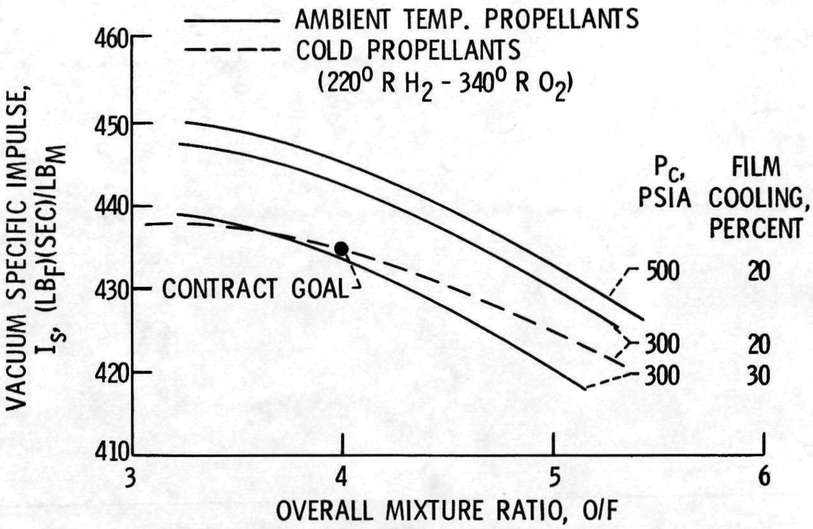

Figure 41. - Vacuum specific impulse for film/regenerative thruster.

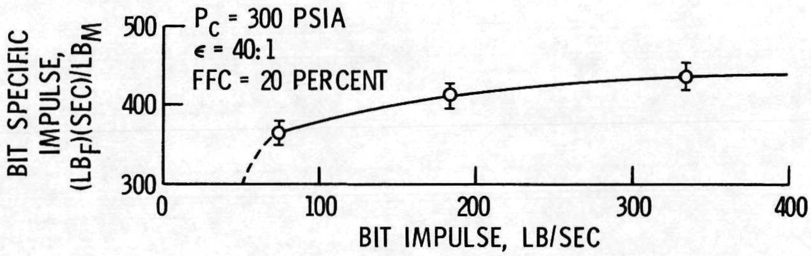

Figure 42. - Bit specific impulse versus pulse width, film/regenerative cooled chamber, ambient temperature propellants.

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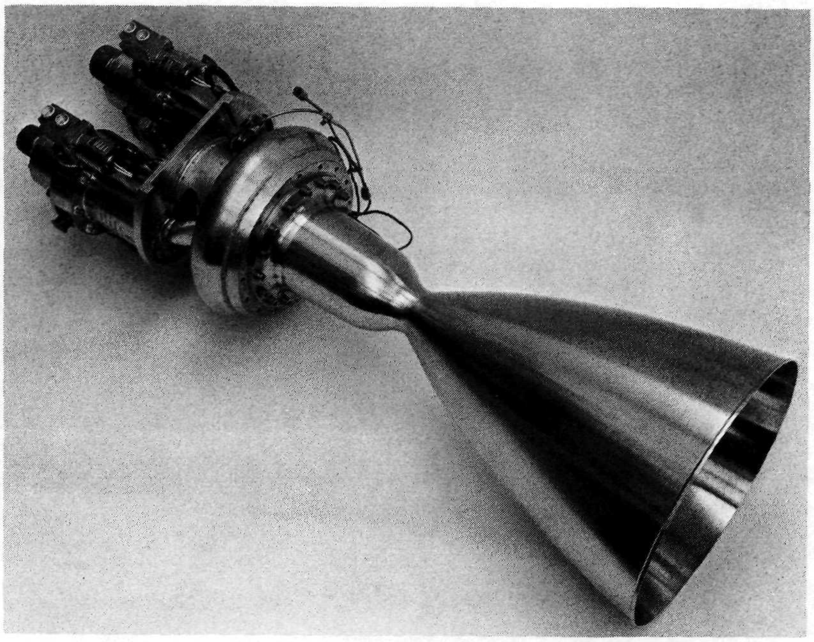

Figure 43. - TRW film/dump cooled thruster assembly.

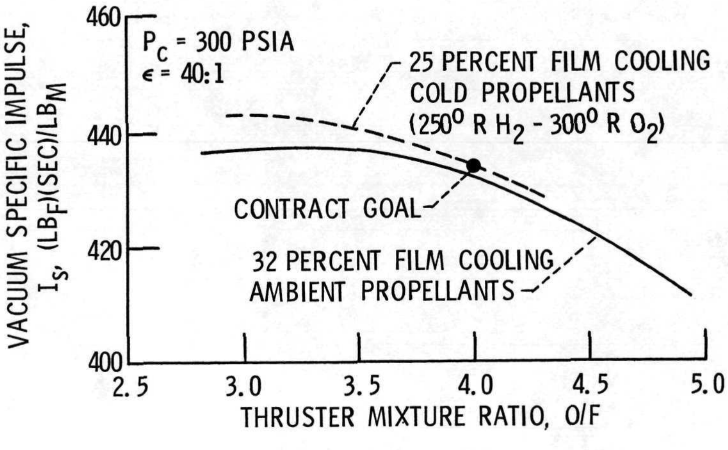

Figure 44. - Film/dump cooled thruster performance.

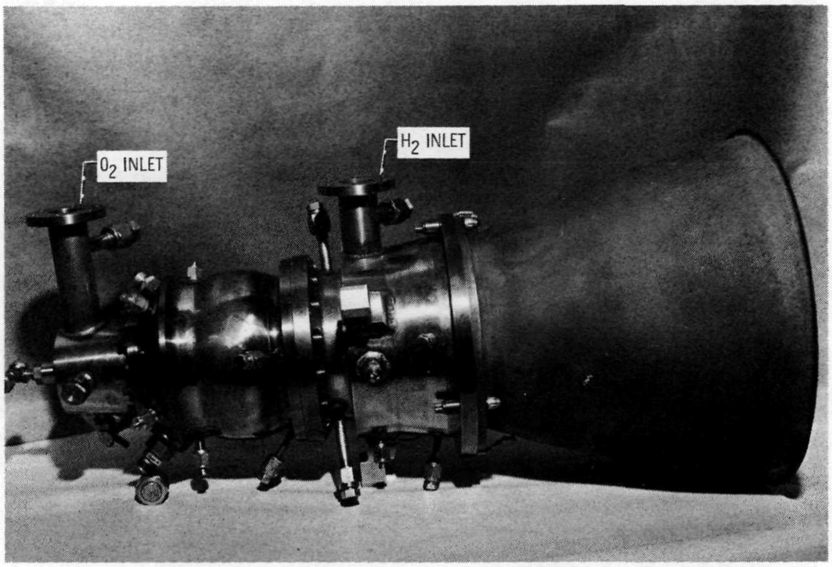

Figure 45. - Bell reverse flow thruster assembly.

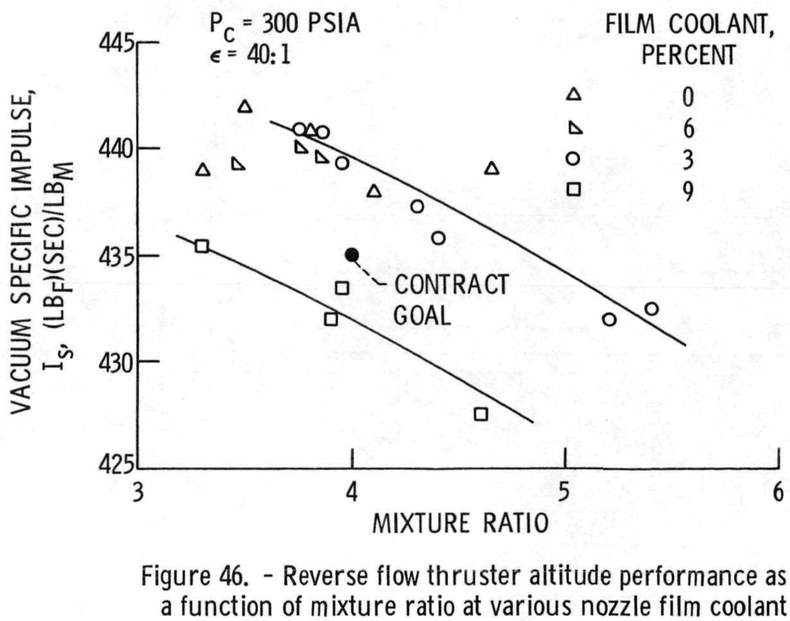

Figure 46. - Reverse flow thruster altitude performance as a function of mixture ratio at various nozzle film coolant flowrates.

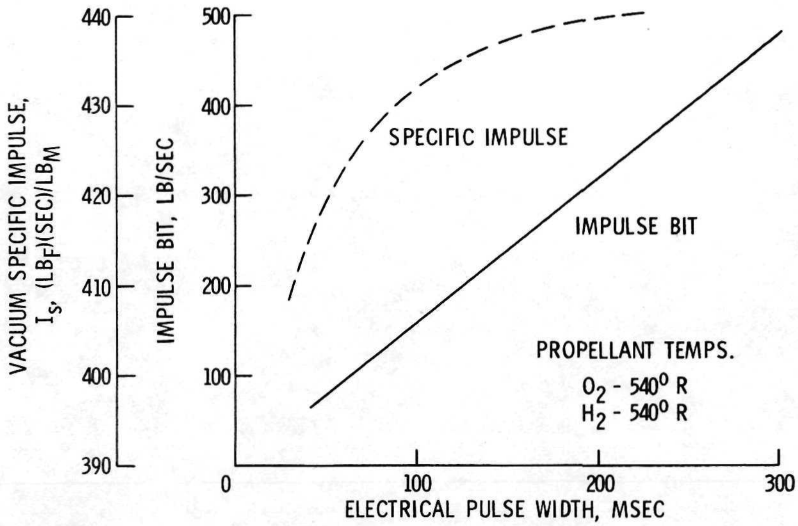

Figure 47. - Reverse flow thruster pulse mode performance.

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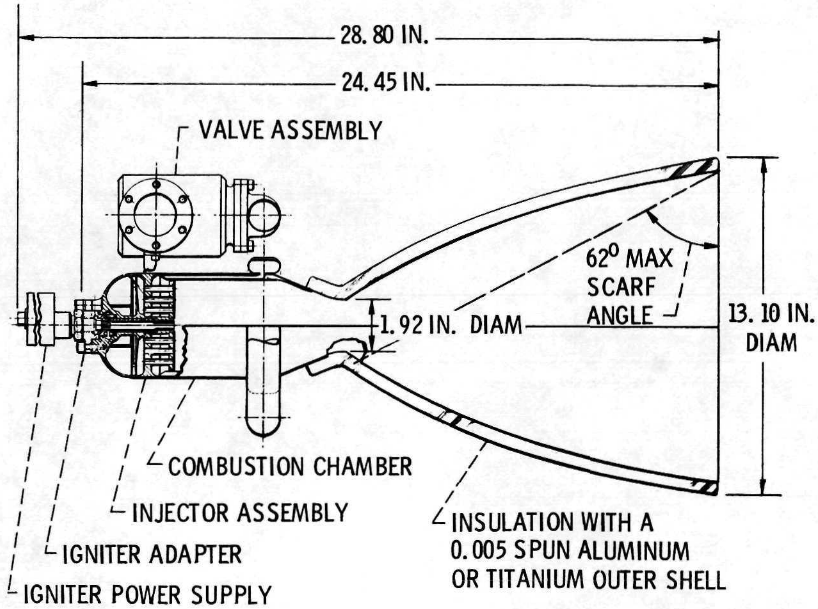

Figure 48. - Integrated Thruster Assembly.