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DEMONSTRATION OF A PULSING LIQUID HYDROGEN/LIQUID OXYGEN THRUSTER

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Abstract

Successful operation of a pulsing liquid hydrogen/liquid oxygen attitude control propulsion system thruster (1250 lb_f) at cryogenic inlet conditions while maintaining high specific impulse and low impulse bit capability has been demonstrated under a technology contract. This demonstration is the result of a continuing search for a simple, lightweight and high performance reaction control system concept and is an advancement in the state-of-the-art of auxiliary engine technology. The use of cryogenic liquid propellants with pulse-mode rocket engines has heretofore only been possible with the aid of heavy and complex propellant conditioning equipment to convert the cryogenic liquids to gases.

Significant technical advances and departures from conventional injector design practices were necessary in order to achieve an operable thruster. These advancements were achieved through extensive analyses of heat transfer and injector manifold priming that established the baseline feasibility for an actual hardware design. Promising results from the thermal analysis, subscale injector chilldown tests, and ignition experiments at cryogenic propellant temperatures (150°R oxygen, 45°R hydrogen) led to the generation of two injector design concepts.

The primary subject of this paper is the result of the experimental evaluation of the 45°R hydrogen inlet temperature injector concept. The test matrix consisted of 66 hot firing tests in a heat sink thrust chamber.

The testing of a complete film cooled thruster assembly at simulated altitude conditions will complete the scheduled technology effort.

A summary of analytical and experimental phases of the liquid/liquid thruster technology efforts will be discussed in this paper.

Introduction

During the early phases of the Space Shuttle vehicle definition and propulsion system studies (1971), several candidate auxiliary propulsion system concepts were proposed and evaluated by both the NASA Centers and vehicle contractor specialists[1,2,3]. From these extensive study and vehicle optimization efforts, it was concluded that the lightest weight Attitude Control Propulsion System (ACPS) for the Space Shuttle application (1.5 to 2.3 million lb-sec total impulse) would be a liquid hydrogen/liquid oxygen (L/L) system of the type shown schematically in Figure 1. It was suggested that this could be a near ideal system if it could be made to work in a satisfactory manner[3]. However, major technical questions concerning the feasibility and operability of such a system had not been previously addressed and, therefore, many technical issues such as ignition and transient-flow characteristics of the cryogenic liquid propellants remained unresolved.

An extensive technology program was initiated by NASA-Lewis in June 1972 (NAS3-16775) to resolve several of the basic technical issues associated with a L/L Attitude Control Propulsion System thruster concept. Some of the critical technology issues to be investigated were; low temperature ignition (liquid propellant inlet conditions), pulse mode operation, delivered performance, combustion stability[4,5], and thruster heat rejection rates to the propellant feed lines.

The specific technical issues were combined into four broad technical areas for parametric analyses. These areas were (1) thruster thermal management, (2) ignition requirements and limitations, (3) performance and operational characteristics, and (4) thruster component and feed system interactions. The results of these parametric analyses provided design guidance in determining which key technology areas had to be demonstrated and aided in the formulation of preliminary design concepts.

In addition to the extensive analytical effort, two critical experimental activities were conducted in support of the parametric analyses prior to design concept selections. First, a series of ignition limit experiments was undertaken to verify the analytically predicted limits of ignitability of cryogenic hydrogen/oxygen mixtures. Other experiments investigated chilldown and priming characteristics of both prechilled and low thermal capacity manifold concepts.

Thruster design configurations, based on the results of the above analyses and experiments, were generated. Each configuration was analyzed in detail using an engine simulation model to predict fill, ignition and shutdown transients. The nominal design point, operating range, performance, and response goals selected by NASA for the demonstration engine are provided in Table I. The designs were tailored to 45°R hydrogen and 150°R oxygen at the propellant valve inlets. The 45°R hydrogen inlet temperature was selected for the nominal design point of the L/L injector because all system considerations analyzed[2,3] indicated 45°R would be a realistic temperature at which the hydrogen could be held in the vehicle supply system and, therefore, supplied to the thrusters. One injector/thruster design was fabricated and hot fired.

The following sections of this paper highlight the technical efforts which brought the advanced thruster technology into reality.

Thruster Configuration

Overall Thruster Analysis and Requirements

Analytical assessments of all major technical areas of concern associated with liquid hydrogen/liquid oxygen thruster concepts were conducted early in the investigation to narrow down the number of conceptual possibilities. Several independent analyses related specifically to ignition requirements, thermal management (propellants and hardware);

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**Leonard Schoenman, Project Manager, Aerojet Liquid Rocket Company injector design synthesis, thrust chamber cooling and feed system interactions were conducted to find an acceptable match of hardware with the overall thruster operating requirements (Table I).

Injector/thruster concepts were evaluated in which either propellant could be injected as received from the feed lines (i.e., as a cryogenic liquid), or could be converted from a liquid to gas prior to injection. This conversion to gas could be accomplished by using the heat rejected from a partial regeneratively-cooled thrust chamber during steady-state operation, and by using residual energy in the thruster during start transients. Other concepts evaluated included those having the propellant valves located upstream or downstream of the chamber cooling jackets and the use of an interpropellant heat exchanger integral to, or separate from, the injector.

The results of these early analyses indicated that liquid-phase oxidizer injection was most desirable in order to have fast response and low minimum impulse bits; this requirement would make it necessary to maintain the oxygen in a high density liquid phase through the injector. It was also determined that the internal volume and heat transfer surface area of the oxidizer manifold had to be minimized in order to produce the desired pulsing characteristics over the wide range of inlet temperatures specified in Table I. Results of these same analyses indicated the hydrogen side of the injector manifold would not be as critical in terms of volume because of the lower fuel density ($\cong$1/20 that of LO_2) and the higher effective thrust derived from residual hydrogen compared to residual oxygen in the injector manifold. Thus minimizing the oxygen manifold volume was a design driver. The LH_2 density change due to heat absorption in the injector manifold was more critical than that of the LO_2. These two considerations ultimately dictated the propellant valve and manifold masses, shapes and locations.

Results of this analysis also indicated that the physical state of the propellants to the igniter assembly could not be guaranteed because of the low flow rates and wide range of duty cycles. Thus, an igniter design which could operate in the L/L, G/L, or G/G mode was required.

Thrust chamber cooling analyses indicated a simple film-cooled thrust chamber could probably provide reasonable performance and life margins while still meeting the operational requirements of both pulsing and steady-state modes of firing.

Ignition Considerations

Igniter design considerations included energy level requirements, propellant flow and mixture ratio requirements, ignition sources, propellant inlet sequencing, overall total ignition energy requirements (torch vs. other igniter approaches) and inlet temperature and chamber pressure limitations.

Preliminary ignition system analyses and laboratory scale ignition experiments indicated that it would be extremely difficult to control the density of the small quantities of propellant reaching the igniter. It was, therefore, necessary to develop an igniter design that could function nearly independently of the physical state or quantity of the propellants being supplied to it. Two criteria were established for any igniter design: first, that the mixture ratio in the primary ignition zone should be centered in the broad band of the H-O ignitability range (2-90), which corresponds to a mixture ratio (MR) of about 40:1; and second, that the total energy level of the igniter torch should be sufficiently high, even in a low flow (vapor restricted) condition, to ensure reliable ignition of liquid phase propellants in the thruster.

An igniter design that could provide reliable and rapid thruster ignition while accepting H_2 and O_2 in a gas, liquid, or two-phase state was configured based on MR and flow requirements and the successful results of the high MR spark gap, capacitive discharge ignition system previously developed[7,11]. The only operational constraint assumed was that both fuel and oxidizer must be supplied to the igniter assembly at close to the same temperature. This could be accomplished easily by the use of tangent or coaxial feed lines or other forms of an interpropellant heat exchanger. Since the flow rates of propellants employed in the igniter are very small (O.1 lb/sec), such a device would also be very small.

Spark energy effects were investigated experimentally and it was concluded that 10 mJ of spark energy was sufficient to provide reliable ignition under all circumstances where an ignitable mixture was present in the spark gap area.

The complete ignition system consists of five major components: (1) a spark plug, (2) valves, (3) a body that forms or contains all manifolding and seals, propellant metering and injection orifices, a platform for mounting the spark plug and valves and all necessary instrumentation ports, (4) a hydrogen cooled nickel chamber, and (5) a high voltage capacitance discharge power supply.

In the selected igniter design concept, shown in Figure 2, the fuel flows from an annular manifold into parallel coolant and injection flow passages. A small portion of the hydrogen (10%) is injected into the igniter chamber where it impinges on the spark-excited oxygen, producing ignition within the igniter at a high mixture ratio ($\cong$40:1). The bulk of the fuel (90%) bypasses the primary reaction zone (kernel zone) and is used as igniter coolant. This fuel coolant is ducted down slotted passages formed between the igniter combustion chamber sleeve and the internal cavity of the injector (igniter port) into which the igniter assembly is inserted. The passage dimensions were selected to provide adequate convective cooling of the igniter combustion chamber that contains the high mixture ratio hot gas. The coaxial core (MR 40) and coolant streams partially mix when the coolant sleeve is terminated upstream of the igniter throat. The secondary fuel added to the oxidizer rich core raises the torch combustion temperature and film cools the igniter throat. The hot ($\cong$ 4500°R) gases from the igniter torch provide the energy source for thruster ignition. This oxidizer-augmented spark-torch igniter concept which was developed under contract NAS3-14348[7], was redesigned to: accept both gas and liquid phase propellants; provide proper cold flow pressure over the total propellant temperature range of interest (37°R-530°R); integrate the valves to reduce dribble volume; and interface with a liquid hydrogen/liquid oxygen injector concept. The flow characteristics of the oxidizer-gap, capacitive discharge spark, torch-igniter design were modified to provide an ignitable propellant mixture ratio in the spark gap (core) region that varied from a MR of 20 to 60 as the propellant temperatures varied within their specified ranges. The igniter was also designed to provide adequate total flow to ignite the thruster with corresponding variations in overall MR from 2 (3600°R) to MR 6 (6300°R).

The variations of igniter propellant flow, mixture ratio and P_C with propellant temperature are shown in curves A, B and C of Figure 3. Curve A shows that the total propellant flow through the igniter decreases as the temperature of the propellants increase. A large uncertainty in flow exists in the two-phase oxidizer region. The minimum flow, even in a full vapor restricted condition (gas-gas), is noted to be approximately an order of magnitude greater than the value required for thruster ignition. Significant variations in the igniter mixture ratio are also to be noted (curve B); all ratios are readily ignitable. A total MR of 2.5 (core = 25) is indicated when both propellants are in the liquidous state. The total MR rises with increasing supply temperature and reaches a maximum of 5.8 (core = 58) when the oxidizer is liquid and the fuel is supercritical gas. The MR then drops rapidly with increasing temperature as the oxidizer also becomes a gas to a MR of 2.0 (core = 20). Curve C of Figure 3 shows the relationship of P_C with propellant flowrate which also decreases as the flow rate was reduced, but still within the flame-quenching and spark standoff propellant cold flow limitations.

Injector Design Considerations

Detailed feasibility analyses were conducted on several candidate injector start concepts (i.e., dry start and prechilled) in conjunction with suitable thruster designs. The following items were evaluated: Injector element type, manifold design, propellant distribution techniques, propellant freezing within injector body, injector face temperature profiles, injector material selections (considering low vs. high conductivity materials), injector performance predictions, injector transient flow considerations, fill times, flow instability tendencies, propellant pressure drops, velocity ratios and propellant density effects- all at the operational conditions specified in Table I.

Any LH_2/LO_2 pulsing injector design concept must consider several potential thermal management problems: (1) freezing of the oxidizer by exposure to the temperature of the fuel (45°R), (2) warming or vaporization of the fuel by heating from the oxidizer, (3) warming or vaporization of both the oxidizer and fuel by heat contained in the injector body. For a dry start, nonprechilled manifold design, the last of the above three problems is the most difficult to solve. If not resolved satisfactorily, uncontrolled propellant density changes will occur that will seriously affect thrust level, mixture ratio, and pulse profiles. Uncontrollable propellant flows would result in the following detrimental thruster behavior:

a. Poor pulse repeatability and erratic startup and shutdown transients.
b. Erratic ignition.
c. Low pulse specific impulse.
d. Local injector and chamber overheating.

Satisfactory theoretical solutions to these problems were found by: (1) minimizing the thermal energy input from the combustor to the injector in both steady-state and transient operation, and (2) controlling the rate at which the energy stored in the injector hardware is transferred to the propellants during thruster startup. The generation of fabricable and durable design concepts that would satisfy these requirements required careful attention to propellant manifold design, injector face cooling, and injector/thruster interface. Several baseline injector concepts were evaluated in an engine simulation model that computed the startup, ignition, and shutdown transients. The results of this feasibility analysis indicated the desired objectives could probably be achieved by either of two different design approaches shown in Figure 4: (1) Prechilled Manifold (fig. 4a) - conventionally constructed and manifolded designs, prechilled by propellant recirculation to allow rapid response; and (2) Low Thermal Capacity Manifold (fig. 4b) - designs which do not require temperature conditioning and which are fabricated from thin walled tubular, platelet or honeycombed materials, or design that utilizes internally insulated manifold designs of materials having very low thermal conductivities.

The prechilled manifold approach maintains the valves, injector manifolding and igniter at cryogenic temperature such that the injector is chilled and ready to fire at all times. The temperature conditioning is obtained by low velocity recirculation of propellants through special injector manifolds and back through the vehicle feed system. In this design, each cooled injector manifold is thermally isolated from each other, the surrounding structure and as much as possible from the injector face by thermal standoffs. Propellants to be combusted are routed from the manifolds to the face through long thin-wall tubes, which reduce heat flow from the warm injector face back to the manifold. Calculated heat leaks for this concept, i.e., heat that would be transferred back to the feed system by recirculation, were found to be 20 to 30 BTU/hr (prefire condition) which is at least an order of magnitude greater than the systems study[2] considered acceptable.

The low thermal capacity manifold approach does not employ propellant recirculation to keep the injector cold. Instead, it utilizes a thermal standoff between the valve and thruster manifolds so that propellants may be maintained as liquids at the valves for long periods with minimal heating from the injector or thruster. It was analytically predicted that this design concept would have slightly poorer start and pulse characteristics than the chilled design. It does, however, significantly reduce the heat leak to the propellant feed system for the same manifold volume (≅2 BTU/hr/thruster). In order to obtain a fast and repeatable starting characteristic from a nonchilled injector with liquid propellants, it was deemed necessary to limit the heat input to the incoming propellants by coating the propellant flow passages with a thermally insulating material, such as sprayed on teflon[6], or devising a manifold structure of very low total heat capacity. Both approaches would accelerate the internal surface chilldown and reduce propellant vapor generation during the start transients. A review of the anticipated fabrication difficulties that could be encountered with the application and durability of a thermally insulating material made this approach less desirable and, therefore, it was dropped in favor of the low thermal capacity design using a double-walled manifold concept. Injector chilldown cold-flow experiments were conducted to aid in selection between the prechilled manifold concept and the low thermal capacity manifold concept.

For the low thermal capacity concept, a unique dual-wall manifold was devised, which employs locally supported 0.005-in. thick stainless steel manifold liners, as shown in Figure 5. The support spacing and diameter was based on structural analyses and permits safe manifold operation up to 1000 psia. Experimental data from the dual wall manifold chilldown tests are also shown in Figure 5, in which the initially ambient temperature dual wall manifold is exposed to sudden flows of LO_2 ($\cong$160°R) at pressures of 483, 750 and 1116 psi. These tests demonstrated the structural durability of the dual wall manifold design concept and showed that the actual chilldown rates were generally faster than the predicted values. Manifold surface temperatures reached the LO_2 liquidous temperature within 0.020 seconds from first indication of propellant flow. The predicted reduced heat input to the feed system and better structural characteristics of the low thermal capacity design using the dual wall manifold concept and its favorable quick chilldown characteristics resulted in a decision to employ this concept in the full-scale injector designs.

In order to select the injector element design, the following injector element types were analyzed: H-O-H noncircular orifice triplet (rectangular hydrogen and circular oxygen orifices), concentric tube, and like doublet. Each of these element types was considered capable of providing acceptable performance, chamber wall compatibility, injector face heat flux, and combustion stability. Elements having low thermal contact area with the propellants were considered more desirable. The noncircular triplet and concentric tube elements were found to be less favorable than the like doublet because they resulted in increased heated surface/flow area ratio of the fuel circuit and hence caused a greater tendency to thermally choke. The number of concentric tube elements was limited to 15 because of the high injection velocities required and the high fuel density resulting in a small annulus gap size ($\cong$.007 in.). The predicted Energy Release Efficiency (ERE)* of a 24 element like-on-like doublet was determined to be approximately 1.5 to 2.0% lower (96-97% ERE) than the H-O-H triplet element design, but was selected over the triplet on the basis of predicted better start transient tendencies, stability, more favorable chamber wall compatibility and ease of fabrication.

A major factor in injector face cooling design involved the proper selection of injection pattern that would recirculate unburned relatively cool propellant near the face. Each of the elements discussed earlier was configured to insure a fuel rich environment near the injector face. In the like doublet design, long oxidizer and short fuel impingement lengths were employed. In addition, a cooling circuit, which utilizes 7% of the fuel was incorporated within the injector design between the combustion gases and the propellant injection orifice plane to preclude heat penetration into the orifices and manifolds during periods of sustained firing. The deep-cup effect of each injector element shown in Figure 6 is the result of placing a multiple pass cooling circuit between the injection orifices and the combustion gases. The strategically located smaller holes are the face bleed cooling ports. Prediction of the temperature profile on the injector face showed that injector face temperature at any location should not exceed 350°F under steady-state firing conditions.

The final configuration of the like doublet injector designed to operate with LH_2/LO_2 is a 24-element design shown in Figure 7. It has the following features:

- An actively fuel-cooled face to preclude thermal penetration to the injection orifices and feed manifolds.
- Dual wall, low thermal capacity, low volume manifolding in both propellant circuits to allow rapid propellant bleed in and fill and thus good pulsing performance.
- Low volume, integral valve seat assemblies that are thermally isolated from the injector body.
- An injector/chamber interface that forms a cooled corner resonator cavity tuned as a quarter-wave length cavity to suppress the first tangential and first tangential plus first radial instability modes of 17 and 16 KHz, respectively.
- A central port that accommodates the spark torch igniter.

The 24-element like-on-like doublet element injector face pattern is configured so that the outer row of this configuration consists of 16 fuel pairs oriented 15° to the chamber radius. The successive inner rows consisted of 16 oxidizer pairs, 8 fuel pairs and another 8 oxidizer pairs all oriented 15° to the chamber radius. The inner row of oxidizer elements was designed to mix with the centrally located igniter which produces a fuel-rich torch.

Figure 7 also shows that the LO_2 discharges from the valve and passes through the thin wall thermal standoff (3.5 in. length of 0.008 in. wall, 0.354 in. dia. 321 stainless steel tube, formed as a bellows to an effective length of 0.5 in.) into a low volume tapered toroidal flow distribution manifold. The oxidizer then passes through 12 equally spaced flow distribution orifices discharging into a disk shaped, dual-wall plenum, which in turn supplies propellant to the 24 oxidizer doublet elements. The measured propellant volume between the propellant valve seal and discharge end of the propellant injection orifices is 0.45 in^3 for the oxidizer circuit. Flow from the fuel valve located on the opposite side divides and flows through two short thin wall lines prior to passing through two symmetrically located thermal standoffs. The fuel discharges directly into the dual wall fuel manifold from which the 24 doublet fuel elements and face cooling circuit are supplied. The measured propellant volume between the valve seal and discharge end of the propellant injection orifices for the fuel circuit is 0.64 in^3.

The propellant valves are structurally attached to the injector through laminated, low thermal conductivity glass/phenolic rings, as shown in Figure 8. Each propellant valve contains a bleed port at the upstream edge of the seal, which permits priming of the valves without passing propellants through the injector.

A full combustion stability analysis of all injector/combustor combinations was undertaken and the most likely modes of instability were identified. The first longitudinal mode was not considered troublesome because it could easily be eliminated by proper selection of chamber length. The first

*Based on JANNAF vaporization, mixing models and combustion models.

tangential and first tangential plus first radial modes were considered most likely to occur and potentially the most destructive (17 and 16 KHz, respectively). The energy level of higher modes was considered to be small. Helmholtz and 1/4 wave length resonators tuned to attentuate the first tangential and first tangential plus first radial modes were evaluated. A 1/4 wave length corner cavity providing an effective area equal to 20% of the chamber cross section, was incorporated into the injector design from inception, including the necessary provisions for cooling the cavity.

Low frequency (chugging modes) were evaluated by means of a conventional double dead time analysis with an appropriate range of assumptions made for vaporization and mixing times. The pressure drops for each of the above injection element types were selected on the basis of these analyses. The oxidizer $\Delta P/P_c$ for the selected 24 element like doublet injector design was 0.27 and fuel $\Delta P/P_c$ was 0.20, based on a chamber pressure of 500 psia and liquid flows at nominal design temperatures.

Thrust Chamber Considerations

There are normally several thrust chamber cooling schemes suitable for auxiliary engine designs (i.e., ablative, film, radiation, dump, barrier and partial regenerative or combinations of these); however, the selection is much more limited for a rocket engine that is intended to pulse hundreds of thousands of firings and still maintain good performance characteristics as well as exhibit repeatable, sharp start transients. Only film and dump cooled chamber concepts appeared viable for the L/L hydrogen/oxygen thruster application. Regenerative cooled chambers were ruled out on two counts: (1) poor startup and tailoff transients due to large coolant passage volumes, and (2) non-predictable propellant density within and discharging from the hydrogen coolant jacket. Ablative cooled chambers were not considered suitable because of possible oxygen attack and limited life. The results of the complete thruster thermal and cooling analyses indicated it would be very desirable to select a thruster design/material combination that could withstand momentary high mixture ratios without damage to the thrust chamber.

With all of the above evaluations and design considerations completed (including data from heat sink chamber tests), the thrust chamber cooling concept selected for the L/L thruster application was a film-barrier cooled design shown in Figure 9. The one disadvantage of the film-barrier cooled chamber is the performance losses associated with the coolant flow requirements. At a nominal MR of 4.5, the chamber is designed to operate at a maximum throat temperature of 1500°F; the equivalent of 44% fuel film cooling being derived from the outer row of injector fuel doublets. However, for this same thrust chamber design to survive a short duration (0.1 to 0.3 seconds) of high mixture ratio operation (i.e., 6.5 to 7.0), it had to be fabricated from a material that could operate at 3000°F in the throat and remain undamaged. To meet this operating requirement, columbium alloy FS85 with an oxidation-resistant fused-silicide coating was selected.

Experimental Results

Ignition Tests

In ignition feasibility tests using laboratory type igniter hardware, ignition delays of less than 0.020 sec. were consistently demonstrated with gaseous, two-phase and liquid oxygen/gaseous hydrogen propellants at temperatures ranging from 134 to 520°R, mixture ratios from one-half to two times the nominal values, and flow rates down to 25% of design values. Subsequent testing provided comparable data for liquid hydrogen inlet temperature down to 44°R. Exciter power levels of 10 mJ were used for reliable ignition; this power level has been demonstrated to be compatible with very long electrode life[7].

A series of full-scale igniter assembly tests with the unit shown in Figure 2 was conducted prior to the initiation of the injector/thruster assembly testing. Approximately 250 hot tests were conducted with this prototype igniter. Rapid and repeatable ignitions were again demonstrated with this unit over the following range of test conditions:

O_2 temperature (at valve)	134 to 520°R
H_2 temperature (at valve)	44 to 518°R
Pressure (O_2 valve)	330 to 910 psia
Pressure (H_2 valve)	309 to 924 psia
Flow rate	0.04 to 0.125 lb/sec
MR (overall)	2.5 to 7.5
Ambient pressure	less than 0.5 psia
Hardware temp	150 to 530°R

Satisfactory ignitions were demonstrated with LH_2/LO_2, GH_2/LO_2, GH_2/LO_2 plus GO_2 and GH_2/GO_2 supplied to the valves with spark plug power levels of 10 mJ. Ignition was detected by a rise in the igniter chamber pressure within .020 sec. from the time fuel or oxidizer pressure was sensed in the igniter manifolds. This was true for all propellant supply conditions listed above. Thermocouples located in the igniter throat and exhaust stream were also used to verify that ignition had occurred. Igniter durability and cooling was demonstrated by continuous firings of up to 10 sec. and pulse trains consisting of twenty 0.20 sec. firings in rapid succession (≅ 0.500 sec between firings). No cooling problems were encountered. Typical igniter oscillograph traces and ignition delay times are shown in Figure 10 for (a) cold propellants (150°R), warm hardware (540°R), (b) cold propellants (150°R), cold hardware (150°R), (c) warm propellants (530°R), warm hardware (540°R), and (d) liquid propellants (49°R H_2-141°R O_2) and cold hardware (260°R).

Injector Tests

Steady-State Performance: A series of 66 hot firings was made using the like doublet injector/thruster shown in Figure 8 with heat sink thrust chambers. During these hot firings, the following range of test conditions were covered:

P_c (psia)	237 - 490
MR	3 - 10
Fuel temp, °R	49 - 70
Ox temp, °R	166 - 184
Injector body temp, °R*	160 - 530
Injector face, °R*	500 - 600
Duration min/max sec	.076/7.700 sec

*At fire signal

The first start of each test series was a short pulse (0.100 sec.) which provided data for a warm-manifold dry-start. Longer burns (1.000 to 2.000 sec.) followed immediately and gave steady-state performance and thermal data. A subsequence 0.100 second pulse provided data for the delivered performance and start characteristics with cold manifolds and a warm injector face. The duration of the longest test firing (7.700 sec.) was limited by the thermal capacity of the heat sink thrust chamber.

The results of the steady-state performance tests, conducted in both 14 and 17 in. L* (5.5 and 7.0 in. length) chambers, are shown in Figure 11. The steady-state thrust based energy release efficiencies at the design mixture ratio of 4.5 were 91% at 5.5 in. length and 96% at 7.0 in. length. These results also show the effect on performance of variation in MR: thruster performance decreased with increasing MR. These tests also showed that steady-state performance levels were approached within 0.100 sec of thruster ignition.

The L/L injector/thruster was found to be stable with quarter wave length corner resonators tuned to 17,000 Hz. The injector was bombed with 2 gr. RDX charges and recovered satisfactorily from 100% overpressure within 1 msec. Throttling to 50% of the design flow was also demonstrated without encountering chugging.

Pulsing Tests: Figure 12 shows the first checkout pulse with a L/L ACPS size thruster. The main valve sequencing was 0.005 sec lead for oxidizer and 0.020 sec for full travel of both valves. The igniter was valved independently from the main injector propellant supply and started approximately 10 msec prior to main propellant valve being energized. The igniter was permitted to operate for 60 msec prior to thruster ignition in order to uncouple the sequence of events involved in the start transient and more easily observe first, the igniter ignition and then, the thruster ignition. In subsequent tests, the igniter lead was reduced and valve signals varied until the 0.075 sec (signal to 90% thrust) response goal was demonstrated. Comparison of this and subsequent pulses with the predictions and engine simulation showed the transient analyses to be quite accurate. Of the 0.075 sec response demonstrated, 0.050 sec was associated with response time of the main pilot valve solenoid and 0.025 sec with valve travel, ignition and thrust rise. Thrust rise, 0 to 90%, was accomplished in less than 0.010 sec when a slight oxidizer lead (0.003 to 0.005 sec) was employed. Thrust rise rates (signal to 90% thrust) with simultaneous flow or fuel leads were in the order of 0.010 to 0.020 sec. Figure 13 shows electrical signals and resulting sequence of events for the minimum pulse width demonstrated. The resultant valve position, igniter chamber pressure, thruster chamber pressure and thrust traces for 5 successive thruster pulses are also shown in Figure 13. The measured impulse (thrust-time integral) for each of these five, 76 msec electrical pulse width (EPW) pulses was 51.8 ± 1.5 lb-sec. A series of slightly longer pulses (105 EPW) and varying off times is shown in Figure 14. From this figure, it can be seen that excellent pulse repeatability can also be obtained for several pulsing off times (1.6 to 5.0 sec).

Using a data sample of 20 pulses, a plot of total impulse vs. electrical pulse width, Figure 15, shows good linearity for the L/L thruster design when operated down to electrical pulse widths of 0.075 seconds. The variation in impulse (lbF-sec) was approximately ± 3% for the entire test range shown. Pulses in which the injector was warm and unprimed and subsequent pulses in which the injector was cold and unprimed provided nearly the same total impulse values.

The effect of successive reductions in firing durations on the thruster performance is shown in Figure 16. Data for these 31 thruster firings at 250 and 500 psia are based on the integrals of measured thrust and total propellant flows, including those of the priming startup and shutdown transients. Each data point compares the full pulse specific impulse to that which could be attained at the same propellant supply pressures after all transient effects were damped (i.e., % of steady-state I_{sp}). These data thus represent the loss in performance due to valve sequencing, mixture ratio shifts (resulting from feed system flow oscillations from the valve openings), manifold priming and blowdown losses. The curve drawn through the upper limits of the data correspond to the loss predictions for the feed system oscillations and dribble mass losses. The remaining data scatter corresponds to thermal and priming shifts for warm dry starts and cold restarts. Relative losses were the same for chamber pressures of both 250 and 500 psia.

Thermal Results: Thruster/injector hot firing test durations ranged from 0.050 sec to 7.700 sec. All thruster components were instrumented during these tests to determine the maximum temperatures and time required to achieve steady-state thermal conditions. Temperature measurements indicated that the injector face was operating below 250°F and reached thermal equilibrium in approximately 1.5 sec. There was no evidence of heat soak problems or injector component overheating of any kind during the 66 hot test firings.

The adiabatic wall temperatures, computed from the measurement of transient temperatures on wall surfaces made on the heat sink copper chambers were found to be approximately 1500°F at the throat for nominal operating conditions.

The significance of these test results is that the outer row of fuel elements of the like doublet injector assembly were providing sufficient cooling to permit the use of a simple lightweight 40:1 film-barrier cooled chamber and that no further cooling would be required. The design point maximum wall temperature (MR = 4.5, P_c = 500 psia) of 1500°F is sufficiently conservative that state-of-the-art columbium materials with fused silicide coatings should provide many hours and hundreds of thousands of pulses of thruster operation based on design predictions. The wide design margins in the chamber design should also allow significant mixture ratio shifts to be tolerated for sustained firing and even full fuel vapor restricted for short periods using the heat sink capacity of the chamber.

Injector face temperature variation from initial condition to running temperature was found to be only 150°F by conducting a series of hot restart tests. The firing sequences conducted consisted of a short pulse (0.1 sec), a long burn (2.0 sec) and a repeated short pulse.

Thermal soakback data, to determine the thermal input to the propellant feed system by the thruster assembly, were not taken during these tests. Estimates of heat soakback based on injector and valve body temperature rises were made, however, and the original estimate of less than 2 BTU/hr/thruster assembly still appears to be a reasonable value.

Concluding Remarks

The experimental data acquired to date have demonstrated the feasibility of pulsing a liquid hydrogen/liquid oxygen thruster over the thrust and chamber pressure ranges of 625 to 1250 lbF and 250 to 500 psia, respectively. The L/L concept now appears to be a viable candidate for advanced spacecraft (e.g., Space Shuttle or Space Tug systems) where high performance and the development of advanced concepts is warranted.

The injector/thruster concept demonstrated under the subject technology program exhibited a 0.075 sec response from start signal to 90% thrust, does not require prechilling prior to firing, is thermally isolated from the valve and does not appear to exceed the permissible heat leak into the vehicle propellant feed system established by the initial vehicle studies[1,2]. The experimental data also show that ignition with low temperature propellants is rapid and repeatable and that bit impulse is repeatable within ≅3% at 50-lb-sec levels. The deliverable vacuum specific impulse of a 17 in L*, 40:1 area ratio nozzle, barrier cooled chamber is predicted to be 427 lb_f-secc/lb for a complete liquid/liquid thruster assembly when sea level injector test data are extrapolated to the 40:1 nozzle vacuum conditions. This 427 I_{sp} performance prediction is based on a 96% ERE demonstrated by the L/L 24 element injector in sea level hot firings in a 17 L* chamber. However, auxiliary propulsion system performance with L/L thrusters is predicted to be considerably higher than the system performance with either a gas/gas or gas/liquid thruster system where "propellant conditioning" is required[2].

Further activities in the optimization of the liquid/liquid injector pattern would probably result in a 1 to 2% improvement in delivered performance. The combination of injector pattern, element type and L* could possibly be changed to achieve higher performance.

All analysis and design activity associated with this technology program was directed, from the beginning, toward establishing the feasibility of a pulsing liquid hydrogen/liquid oxygen injector/thruster. The L/L thruster hardware designed and tested was not an optimized "fixed point" design (i.e., for a single chamber pressure) but was a "laboratory tool" designed to acquire experimental data over a wide range of operating conditions. Therefore, conservatism was employed in most areas of the hardware design selections. Now that feasibility has been demonstrated, refinement and further design iterations would very likely result in an improved performance level. Also faster response times could be obtained if better propellant valving were developed.

Further experimentation under the subject contract will include altitude testing of the complete cooled thruster assembly. Additional technology work should be undertaken which would include the following steps to fully demonstrate the technology for a flight-type L/L thruster and a L/L H_2-O_2 ACPS:

(1) Further optimization of the dual wall manifolding concept.

(2) Design/development of a lighter weight injector assembly including flightweight valves.

(3) Durability testing of the flightweight design.

(4) Insulation of the thrust chamber and evaluation of this effect on design margins, thruster life and thermal soakback.

(5) Demonstration of a complete propulsion system including small pumps, high pressure run tanks (liquid accumulators), lines and thrusters at simulated space conditions (breadboard system demonstration).

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TABLE I

L/L THRUSTER DESIGN AND OPERATING CONDITIONS

	Nominal Conditions	Experimental Operating Range
Thrust:	1250 lbF	—
Chamber Pressure:	500 psia	250 – 750 psia
Overall Mixture Ratio (O/F):	4.5	3.5 – 9.0
Nozzle Expansion Ratio:	40:1	
Characteristic Chamber Length:	—	14 – 17
Propellant Inlet Conditions:		
H_2 Temperature:	45°R	37°R to 75°R
O_2 Temperature:	150°R	100°R to 200°R
H_2 Pressure:	625 psia	As required
O_2 Pressure:	625 psia	As required
Specific Impulse Goal:		
Steady-state	435 lbF-sec/1bM	
Pulsing	400 lbF-sec/1bM	
Response:		
0 to 90% Thrust (from electrical signal)	75 milliseconds	
Minimum Impulse Bit (MIB)	50 – 75 lbF-sec (goal)	

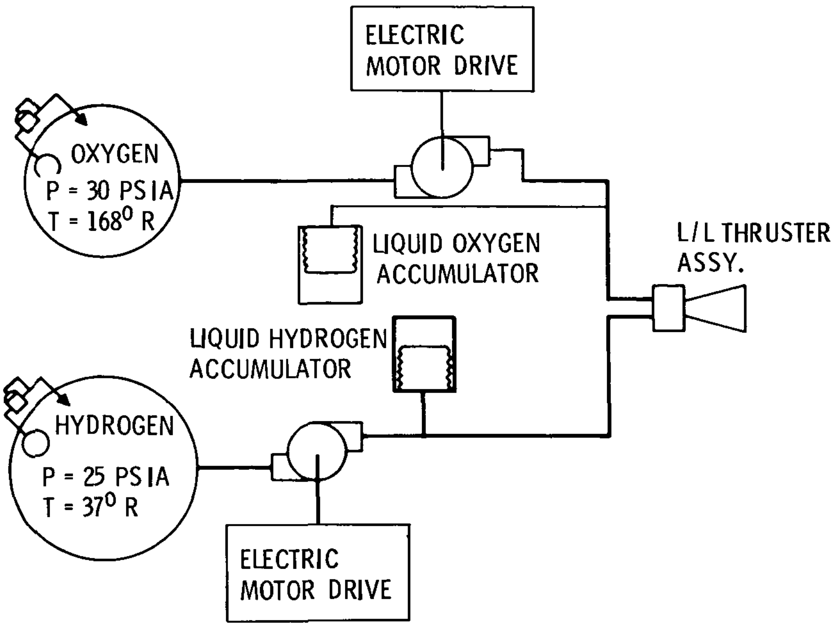

Figure 1. - Liquid hydrogen/liquid oxygen APS schematic.

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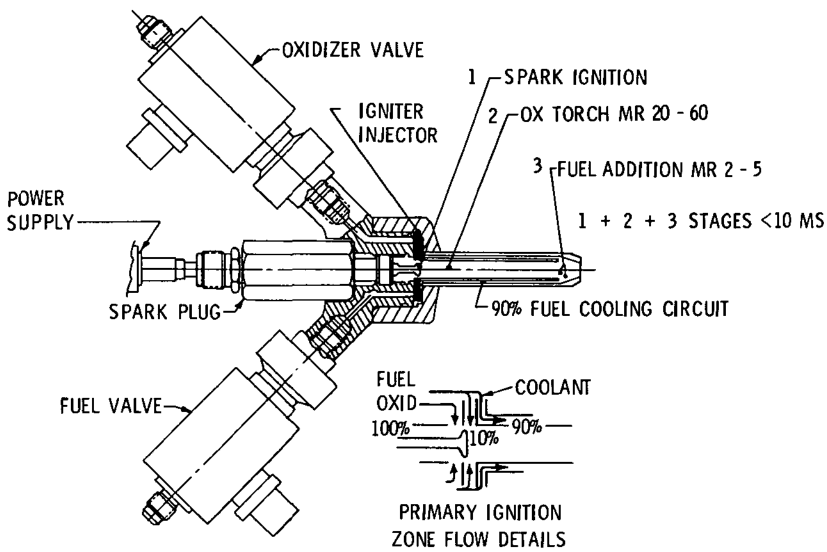

Figure 2. - Staged torch igniter.

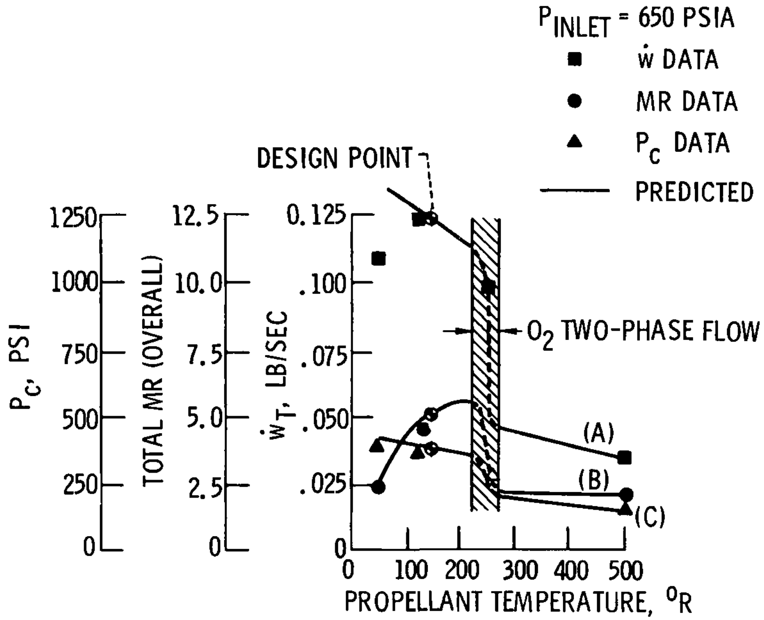

Figure 3. - Igniter flow characteristics versus propellant temperatures.

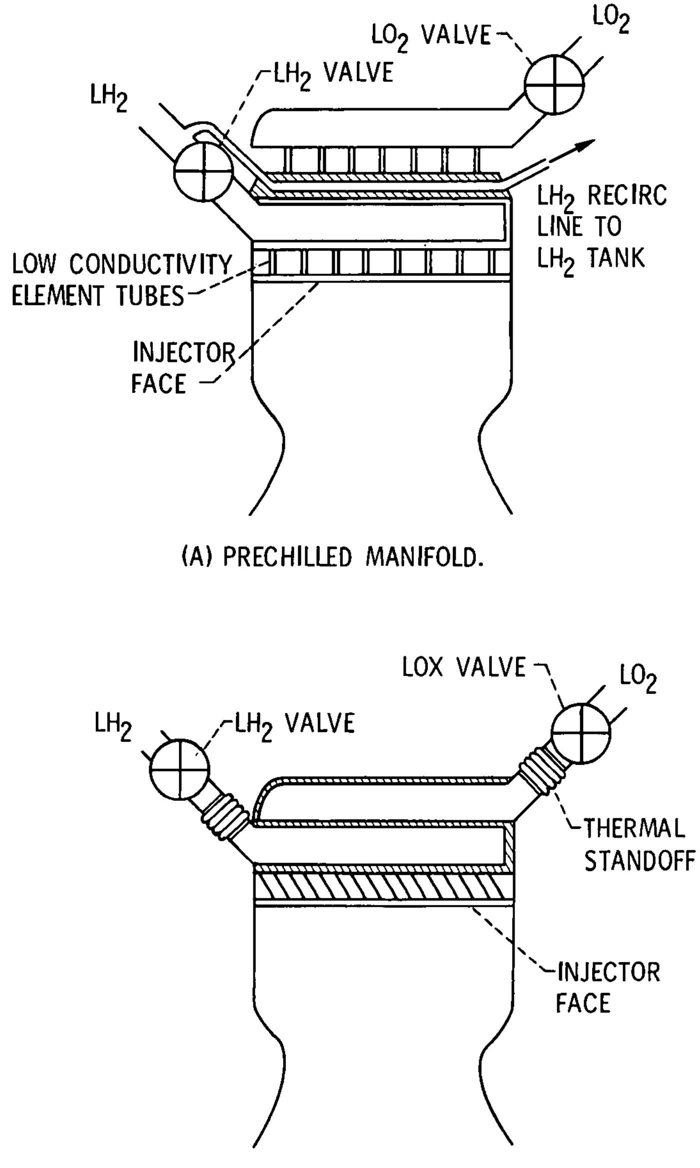

Figure 4. - Injector thermal management concepts.

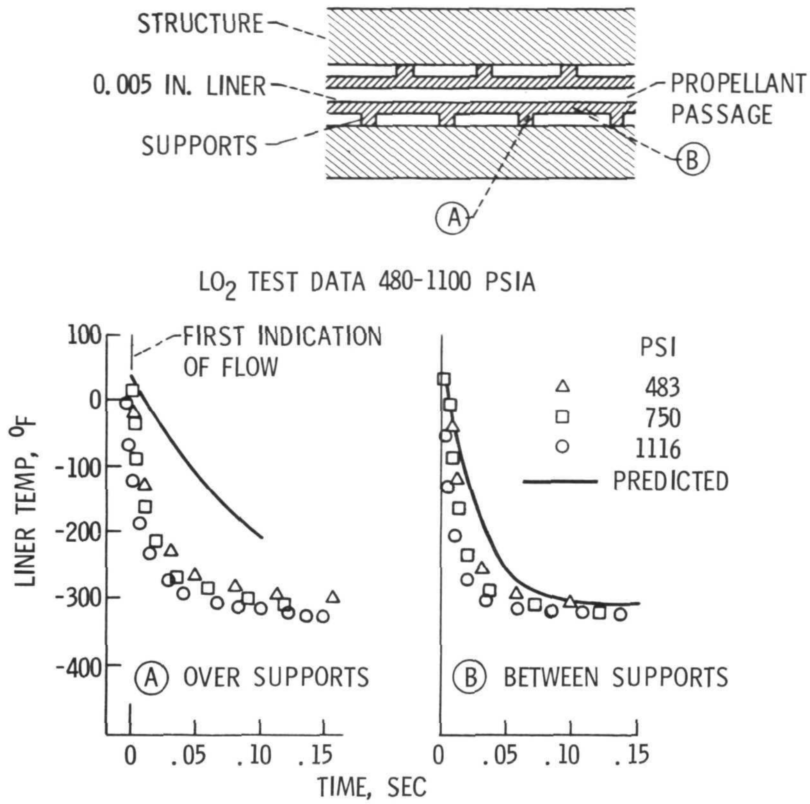

Figure 5. - Dual-wall low thermal capacity manifold concept.

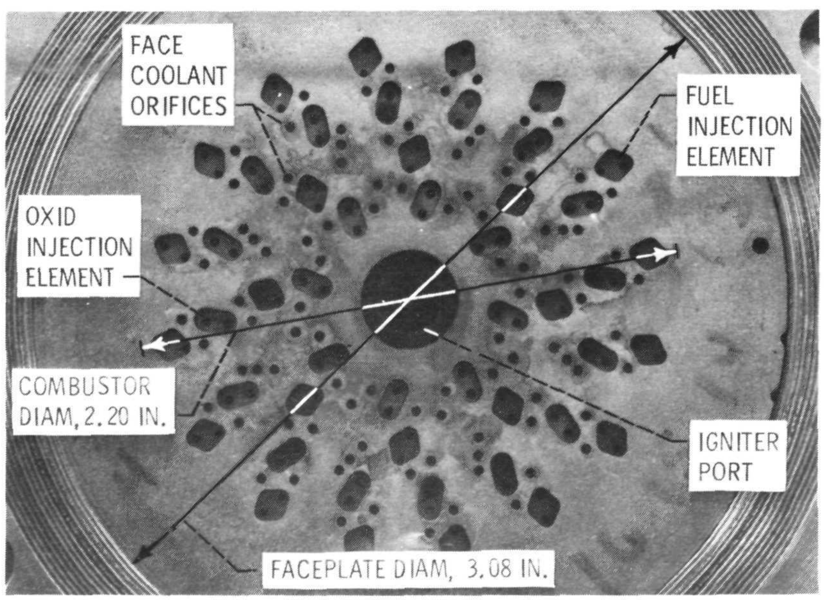

Figure 6. - Like doublet L/L injector face.

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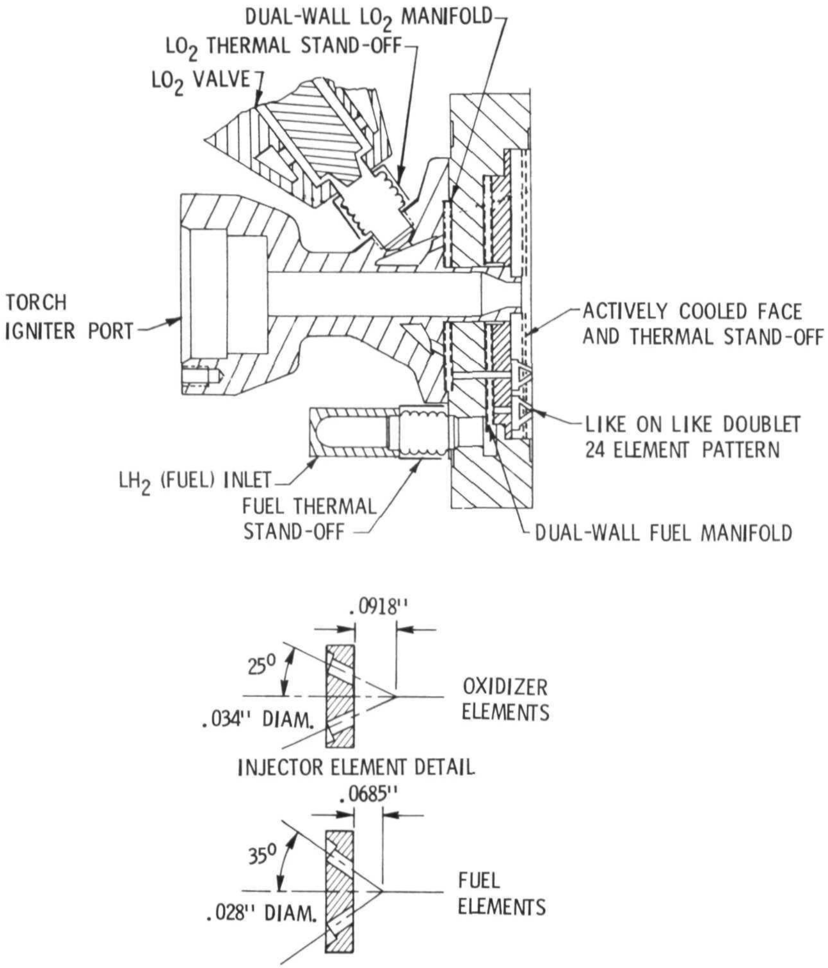


Figure 7. - LH_2-LO_2 injector assembly.

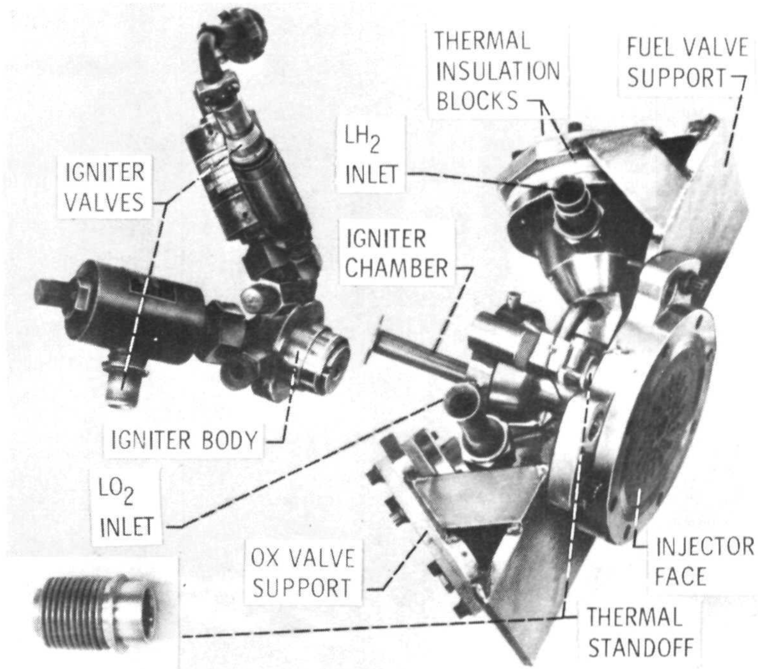


Figure 8. - LH_2/LO_2 injector assembly.

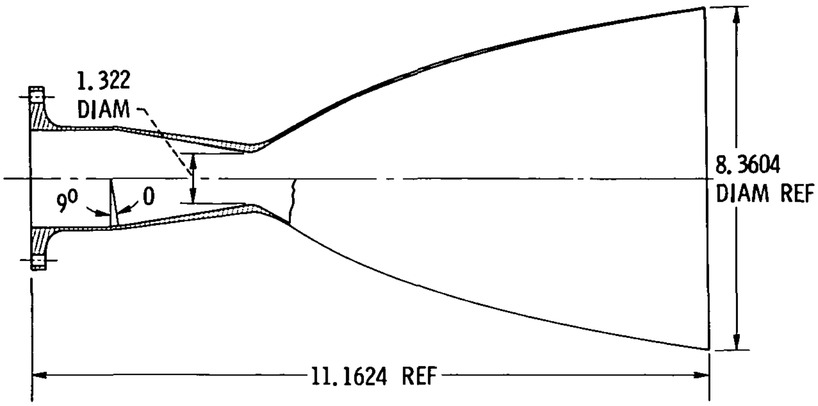
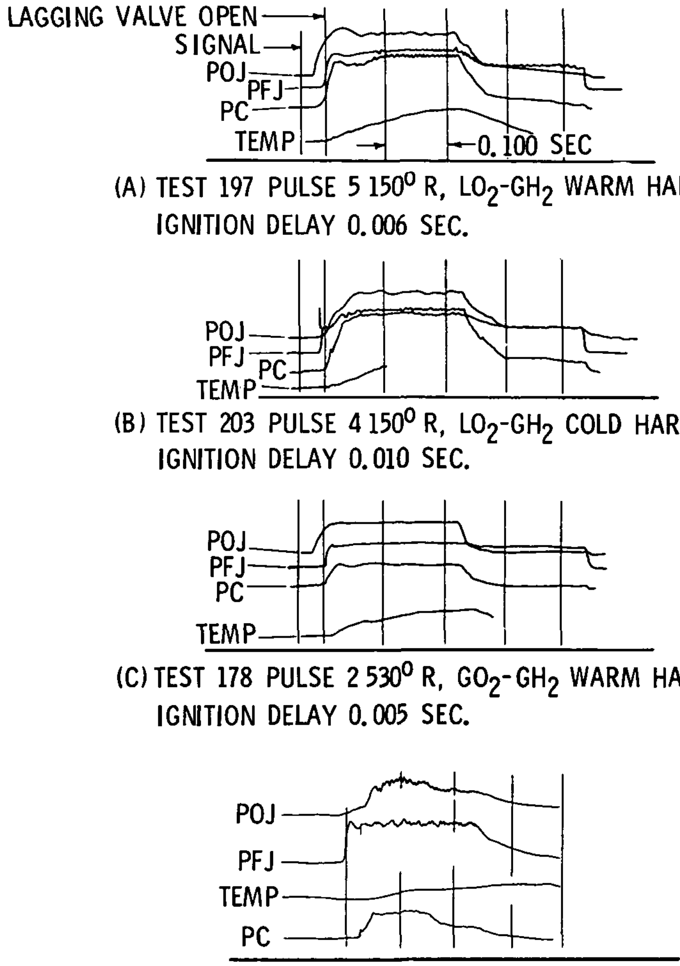


Figure 9. - Columbium thrust chamber design.



(A) TEST 197 PULSE 5 150° R, LO_2-GH_2 WARM HARDWARE, IGNITION DELAY 0.006 SEC.

(B) TEST 203 PULSE 4 150° R, LO_2-GH_2 COLD HARDWARE, IGNITION DELAY 0.010 SEC.

(C) TEST 178 PULSE 2 530° R, GO_2-GH_2 WARM HARDWARE, IGNITION DELAY 0.005 SEC.

(D) TEST 108 PULSE 4 150° R, LO_2-LH_2 COOL HARDWARE, IGNITION DELAY 0.010 SEC.

Figure 10. - Typical igniter start transients.

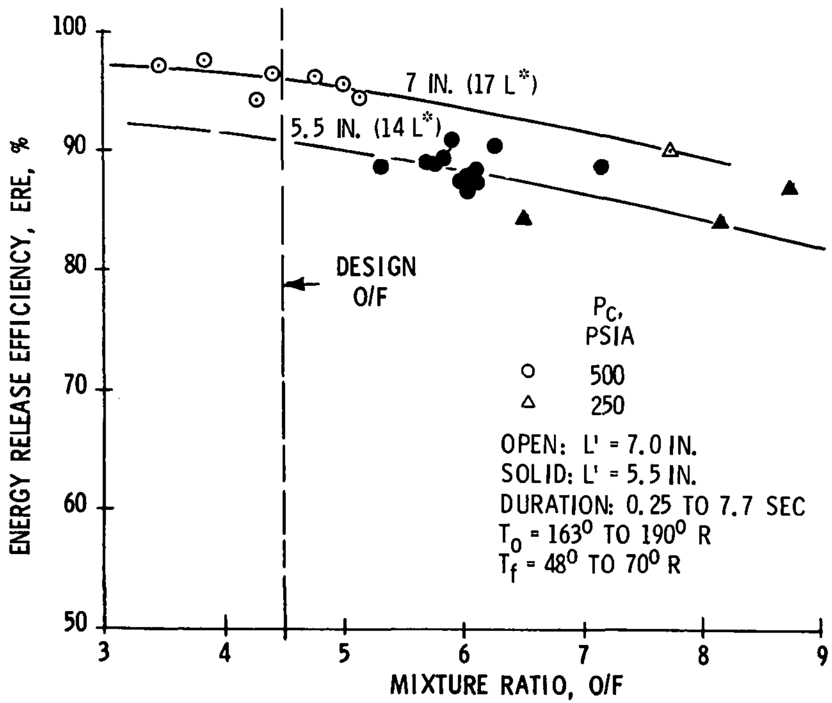


Figure 11. - Liquid H_2/liquid O_2 injector steady state performance.

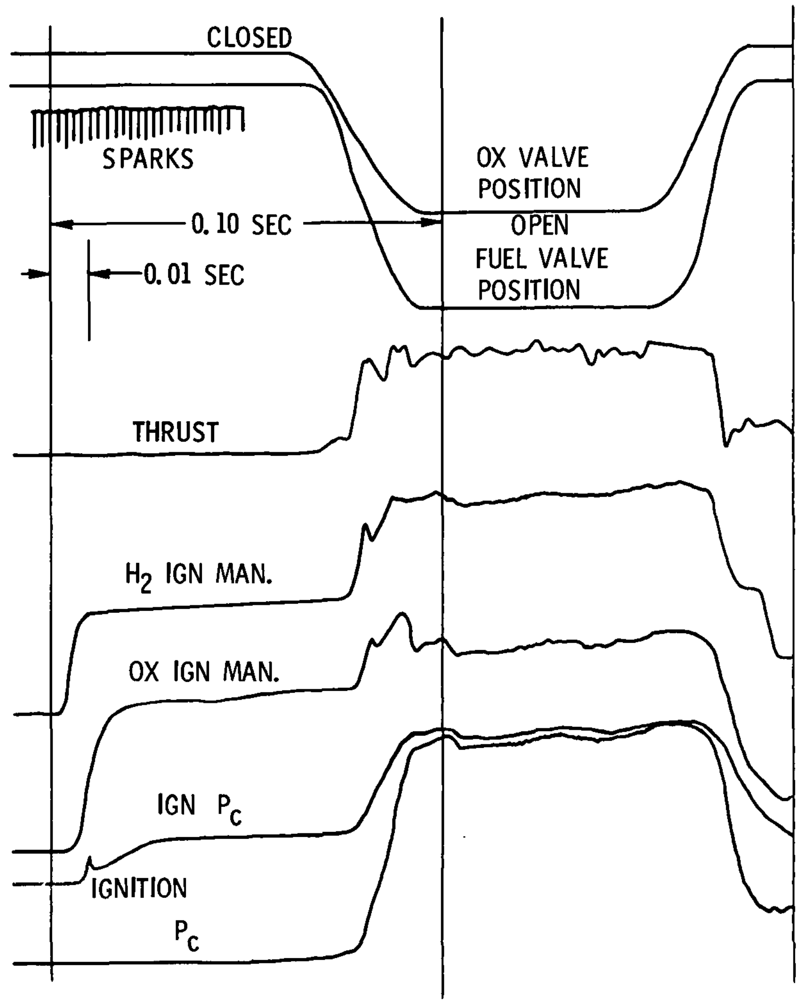


Figure 12. - First LH_2/LO_2 thruster firing traces.

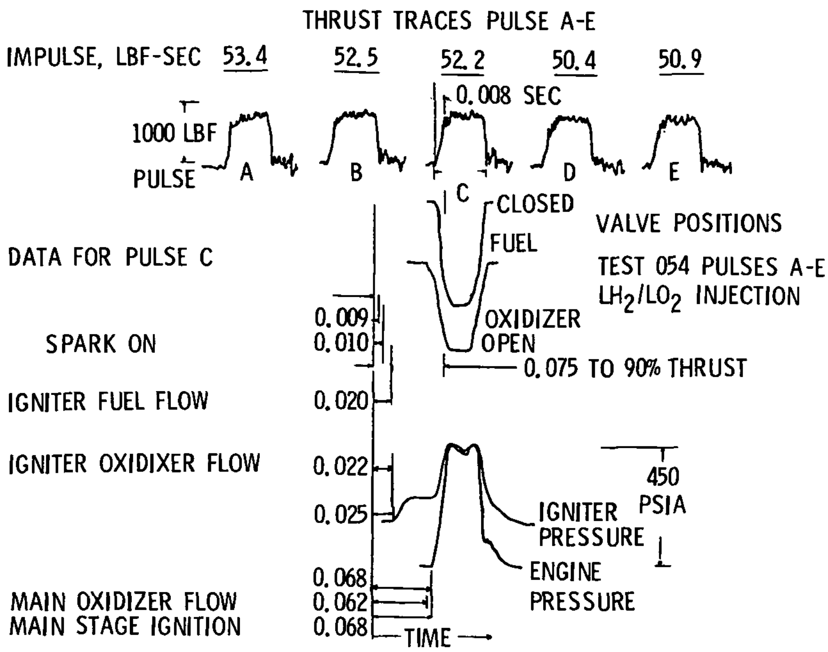


Figure 13. - LH_2/LO_2 thruster sequence, and response characteristics (pulse train 054).

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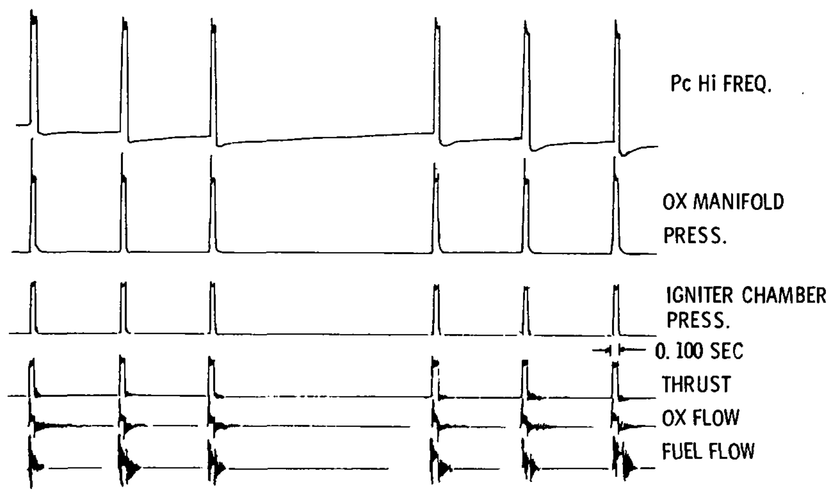

Figure 14. - L/L Thruster electrical pulse width (E. P. W.) train (varying off time).

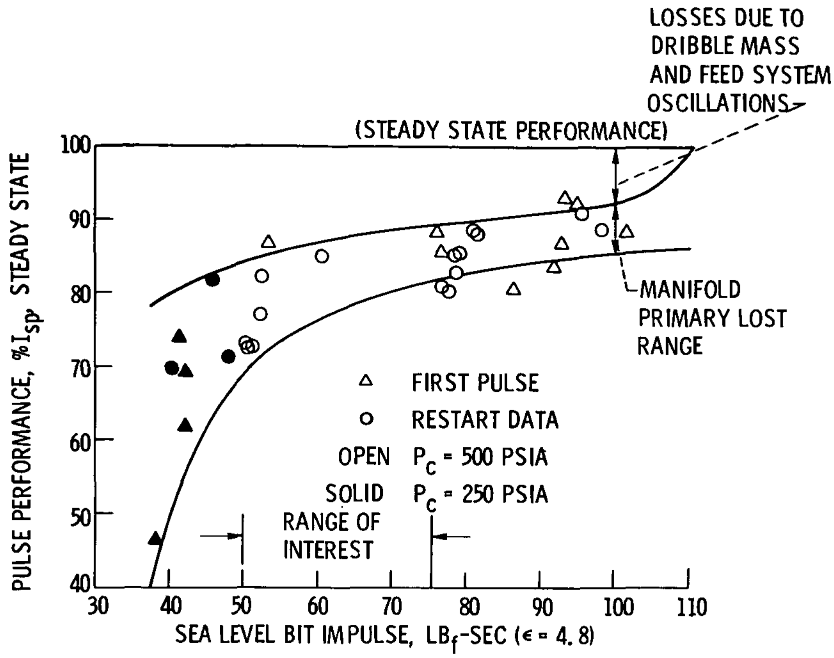

Figure 15. - Bit impulse pulse performance/Sample 31 tests.

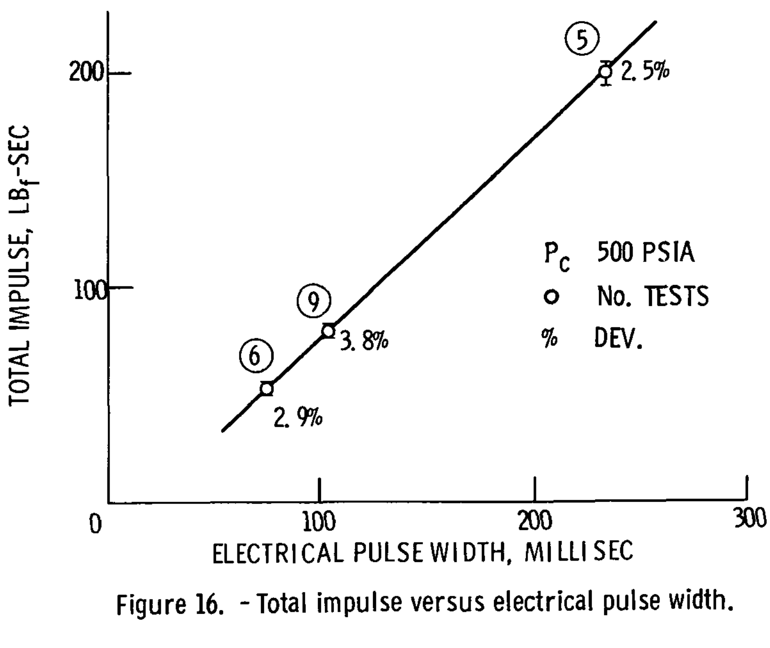

Figure 16. - Total impulse versus electrical pulse width.

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