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SPACE SIMULATION EXPERIMENTS ON REACTION CONTROL SYSTEM THRUSTER PLUMES

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SPACE SIMULATION EXPERIMENTS ON REACTION CONTROL SYSTEM THRUSTER PLUMES

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Abstract

A space simulation procedure was developed for studying rocket plume contamination effects using a 5-pound bipropellant Reaction Control System thruster. Vacuum chamber pressures of 3×10^{-5} torr (70 miles altitude) were achieved with the thruster firing in pulse trains consisting of eight pulses - 50 msec on - 100 msec off and seven minutes between pulse trains. The final vacuum was achieved by cooling all vacuum chamber surfaces to liquid helium temperature and by introducing a continuous argon leak of 48 std. cc/sec into the test chamber. An effort was made to simulate propellant system flow dynamics corresponding to actual spacecraft mission use. Fast time response liquid flow rate measurements showed that large variations occurred in the ratio of oxidizer to fuel flow for pulse-on times up to 120 msec. These variations could lead to poor combustion efficiency and the production of contamination.

Introduction

Reaction Control System (RCS) thruster plumes, interacting with various spacecraft surfaces, are a cause for concern in the design and performance of important thermal control systems. Pulse mode firing of these thrusters forms liquid particle contaminates in the combustion chamber that are transported by the plume and can be deposited on large portions of a spacecraft surface. Damage studies,[1-3] clearly demonstrate a loss in performance of thermal control coatings, transmitting and reflecting materials, and optical sensors due to contamination. Consequently, true space simulation experiments of RCS plume-surface interactions are required for performance predictions of various spacecraft thermal control and optical measurement systems.

To study plume contamination effects a space environment test must meet two basic requirements. First, no plume test is acceptable unless the zone of mixing between the ambient and exhaust gas is either properly simulated or is far removed from the contaminated region of interest. For tests dealing only with the continuum core of the plume this is a simple criterion to satisfy. However, most heating and contamination problems involve far field regions where the mixing zone between external and exhaust flows are important, and collapse of the plume must be avoided. Consequently, the space simulation chamber must provide and sustain pressures below 3×10^{-5} torr (70 miles altitude) during thruster firings. Furthermore, the recirculation of residual exhaust gases and contaminants, following a series of thruster firings, must be extremely low. Otherwise, false deposition rates of contamination and erroneous contamination flux distribution will occur.

The second condition for proper plume contamination tests is that the spacecraft's propulsion-propellant system must be modeled properly. Pulse duty cycles, involving fast opening and closing of the fire valves, generate large propellant transients which affect injector dynamics and the combustion process. Propellant flow oscillations create large variations in ratios of oxidizer to fuel flow (O/F) at fixed lengths from the injector surface. Also, nonuniform radial O/F ratios occur across the combustion chamber.[4] All of these factors have a significant effect on contamination formation and distribution through the plume.

Space Simulation Experiments

Simulation Chamber Characteristics

The space simulation experiments were performed in the Lewis Research Center's 6- by 12-foot liquid helium cooled space simulation chamber. This facility is uniquely suited for plume contamination studies. It has a large pumping capacity because the test chamber is completely enclosed by liquid helium cooled surfaces. Also, a solar simulator, giving a constant and uniform one solar constant (1400 W/m^2) beam, permits in situ thermal radiation studies on materials and surfaces of interest. Figure 1 is a schematic diagram of this facility. The test chamber and its associated liquid helium fill and return lines, vacuum pumps, vacuum jacket, and instrument feed through section may be separated at elevation - XX by removing the optics tower. The optics tower, contains a beam turning mirror, a lenticular lens system for beam uniformity, a 40-in. diameter collimating mirror, and is open to the pumping system during testing. With the optics tower removed test packages are positioned in the test section. Various operating and vacuum pumping modes using liquid helium, gaseous helium (5° to 40° K), liquid nitrogen, and two 32-in. diffusion pumps are available for developing a test chamber vacuum. The 6- by 12-foot test chamber extends to elevation - BB - and is formed from 3/8-in. thick 310 stainless steel. A liquid helium jacket, with a spacing of 1/2 in., surrounds the test section. Also, 22 elliptical cross-section tubes, equally spaced circumferentially, are positioned in this jacket and can be filled with liquid nitrogen to obtain 77° K wall temperatures. Liquid helium, from a 35 000 liter storage system, enters the facility at elevation - AA - and is piped to the bottom of the jacket (elevation CC). A dewar to helium exit pressure of 1.5 to 2.0 psi fills the liquid helium jacket in about one hour. The helium boil-off gas is collected in the exit manifold at elevation - AA - and is forced to flow through copper tubes welded to a copper radiation shield surrounding the exterior of the liquid helium jacket. An advantage of this return system is that the cold helium gas cryopumps the gas in this jacket which prevents conduction losses to the outer steel vessel. Finally, the boil-off gas returns to a gaseous helium storage system consisting of 400 000 std. ft^3 of low-pressure storage and 585 000 std. ft^3 of high-pressure storage. Dual liquid helium refrigerators, having a production capacity of 200 liquid liters per hour, liquefy the storage gas and complete the cycle by providing liquid helium to the storage dewars. If 5° by 40° K test chamber walls are desired as much as 30 000 cubic feet per hour of cold helium gas can be obtained from the outlet of the refrigerator's expansion engine. Also, the cold helium gas is used to precool the liquid helium jacket thereby conserving liquid helium.

Test section wall temperatures are measured with germanium resistors as reported in reference 5. However, an improved heat sink has been developed for cases of large incident thermal radiation. A heat sink consists of 0.003 in. thick mylar tape having a width of 1 in. and a length of 4 in.. Thin electrical conducting rails (0.01 in. wide) are sprayed on the mylar tape using a silver based paint. All lead wires are then attached to the heat sink with a silver based epoxy. Accuracies of $\pm 0.05^\circ$ K are obtained with a heat load of 1000 W into the liquid helium jacket.

Experimental Cryopumping Tests

The effectiveness of liquid helium cryopumping was investigated by emitting room temperature, missle grade hydrogen into the test section as a free jet. The free jet was formed from a 0.190 in. I. D. tube and was located 1.5 ft off of the vertical center line and 2 ft above the floor of the test section. The direction of the free jet was parallel to the vertical center line or perpendicular to the floor. Cold cathode gauges, with glass envelopes, were placed 4 ft and 6 ft above the exit plane of the free jet. The mouth of each vacuum gauge was pointed towards the vertical center line.

With the liquid helium jacket filled the test section pressure, as recorded by the three gauges, was below 1×10^{-12} torr. The volume flow rate was established by passing the hydrogen through an accurate rotometer and fine control leak valve. As the hydrogen flow rate was increased from 0.1 std. cc/sec to 0.2 std. cc/sec (fig. 2) the test section pressure increased to 4×10^{-6} torr which is the vapor pressure of hydrogen at the recorded wall temperatures of 4.7° K. Each data point in figure 2 represents a steady state point where the test section pressure remained constant for at least 15 minutes. Figure 2 shows that hydrogen flow rates of 2 std. liters/sec can be discharged into the test section without increasing the test section pressure above 4×10^{-5} torr. These pumping tests were repeated several times with vacuum gauges interchanged and also with new gauges replacing the original gauges. Also the same pumping results were obtained with the two 32-in. diffusion pumps turned off. For hydrogen flow rates of 0.2 to 350 std. cc/sec the test section pressure remains at 5.5×10^{-6} torr. This compares with a pressure of 1.7×10^{-4} torr, with similar hydrogen flow rates, in the Lewis Research Center's 25- by 82-foot vacuum chamber having liquid nitrogen baffles and twenty 32-in. diffusion pumps. RCS thruster plumes contain gases other than hydrogen. Consequently, pumping tests including combinations of hydrogen and nitrogen and hydrogen and sulphur hexafluoride were performed. The second gas-free jet was formed identical to that for hydrogen except that it was located 1.5 ft on the opposite side of the vertical center line. Thus, the free jets were separated by a distance of 3 ft. Figure 3 shows the resulting changes in test section pressure for various total gas flow rates. First, the hydrogen flow rate was set at a constant value of 0.28 std. cc/sec. The test section pressure increased from below 1×10^{-12} torr to 3.8×10^{-6} torr. The nitrogen flow was then started and increased to a total flow rate of 142 std. cc/sec. Figure 3 shows that the test section pressure falls continuously to a value almost 2.4 orders of magnitude lower than the initial hydrogen only pressure. This decrease illustrated the increased effectiveness in cryopumping hydrogen when a condensible gas is added. To establish the effect of initial hydrogen flow rate, this rate was increased to 16 std. cc/sec. Adding a nitrogen flow rate up to 122 std. cc/sec for a final total flow rate of 138 std. cc/sec produced only a very small lowering of the test chamber pressure. Sulphur hexafluoride (SF_6) was also used as a condensable gas source because its thermodynamic properties are similar to those of air. Again, a 16 std. cc/sec hydrogen flow rate was established, followed by an increasing sulphur hexafluoride flow rate, resulting in a final total flow rate of 110 std. cc/sec. Figure 3 shows that the sulphur hexafluoride flow reduced the test section pressure by almost a factor of 3 over the equivalent nitrogen flow. Further increases in SF_6 flow rate were limited by the facility flow meter.

Reference 6 reviews several theoretical models for enhanced cryopumping using condensible gases. The physical adsorption model postulates that hydrogen cryopumping is a function of the condensible melting temperature and the condensing surface temperature. The present results are consistent with this model, in that sulphur hexafluoride has a much higher melting temperature than nitrogen. Due to storage and gas flow transfer difficulties however, sulphur hexafluoride was not considered for the truster plume tests. Instead, argon, which has a melting temperature between nitrogen and sulphur hexafluoride was used to help provide low test section pressures during RCS thruster operation.

Thruster Plume Cryopumping Results

A 5-lb thrust monomethylhydrazine (MMH) - Nitrogen Tetroxide (N_2O_4) thruster with its performance instrumentation, propellant lines, and purge system was placed in the space simulation chamber. As illustrated in figure 1, the thruster was fired in towards the liquid helium cooled walls at an elevation of 6 ft above the test section floor. The nozzle exit plane was 4 ft from the test section walls. Figure 4 shows the test package containing the thruster and vacuum gauge systems.

The test chamber pressure was measured with a nude buried collector gauge.(7) Fast time response is achieved by connecting the collector directly to a fast response solid state electrometer. The output of the electrometer is a 3-V per decade change in test section pressure and was recorded on a dual beam oscilloscope. A remote range control system provided readings from

1×10^{-12} to 1×10^{-4} torr. The time response of the vacuum gauge and its electrometer was determined by discharging puffs of various gases from a fast acting valve. Results from these tests indicated a total system time response of 100 μsec per decade change in test section pressure. The vacuum gauge was located 6 in. below the thruster center line and 8 in. upstream of the nozzle exit plane.

Cryopumping results during rocket firing, with and without a continuous 48 std. cc/sec leak of argon and with test chamber walls between 4.2^{o} and 4.8^{o} K are shown in figure 5. The argon leak was located 2 ft above and parallel to the thruster center line with its discharge plane in the thruster's nozzle exit plane. The thruster pulse firing sequence consisted of eight 50-msec burns with 100 msec between firings. Seven minute intervals occurred between pulse sequences. Without the argon leak the test section pressure, during thruster operation, was 8×10^{-5} torr. During the seven minute interval between sequences, the pressure returned to 5×10^{-6} to 6×10^{-6} torr. The effect of the argon leak was a decrease in test section pressure to 3×10^{-5} torr during thrust operation. Also, the prerun test section pressure decreased to 6×10^{-7} torr. Figure 5 also shows that a total of 456 thruster firings have been made. This sequence required seven hours and a liquid helium consumption of 11 000 liquid liters.

Figure 6 is an oscilloscope trace of the buried collector gauge. Channel B indicates seven of the eight pulses during one firing. Channel A is the chamber pressure during the pulse sequence. These results indicate that the pressure, with the argon leak, remains near 3×10^{-5} torr for all eight pulses. Results from slower sweep speed traces showed that the chamber pressure decreased to 6×10^{-7} torr in 3 minutes after each pulse train.

RCS Thruster Experiments

The RCS thruster plume contamination study can be divided into three parts: formation, transport and effects. The formation study involves the production of contaminants by the engine and propellant system. The transport study includes contamination identifications and distributions throughout the plume. Finally, the effects of contamination on optical properties of typical spacecraft surfaces and materials are studied, and preliminary results were reported.[1,2] This report discusses the processes of formation of contaminants. A series of experiments were performed to determine thruster-propellant system performance with a variety of pulse duty cycles.

Table I lists the design and nominal steady state performance characteristics of the thruster system. Perhaps the two most important performance measurements for pulsed operation are the propellant flow rate and thrust measurements. Thrust measurements, with a 1 msec time response, were obtained with this thruster and recorded in reference 8. Also, the results of reference 8 include a correlation of chamber pressure against thrust. Hence thrust measurements were not included in these experiments.

The propellant flow rate measurement techniques were established in a separate experimental program where time response measurements and reliability tests were performed on three types of flow meters. These were turbine meters, hot film anemomenters, and an orifice with its companion pressure transducer. The orifice was sized according to the ASME power test codes and was constructed with a saphire plate in a 1/4 in. AN union. Reliability tests showed that turbine meters and hot film anemometers were not compatible in N_2O_4 as bearing and epoxy failures occurred on several commercial flow meters. Also, limited turbine meter tests indicated time responses of 10 msec which are too long for 50 msec firing times. A separate propellant system was constructed so that all dimensions, materials, and components would be identical to the system used in the liquid helium space simulation chamber. Kerosene was used as the working fluid which allowed time response comparisons between the hot film and orifice system. The orifice and hot film anemometer were separated by 3 in. and the flow was established by a spare thruster fire valve. Flow against time results obtained from both flow meters indicated that the orifice system followed the 1 msec time response of the hot film anemomenter. However, the similar results between flow meters occurred only for a propellant system void of trapped gas. Pumping the propellant system to vacuum conditions of one micron before filling produced a propellant system with no small gas pockets. The pressure transducer's time response and ability to operate in a vacuum environment are also important. A commercially available (Statham Model PM 180) transducer having a 1-msec time response and constructed from a single block of steel was used. Thus, gas leaks into the strain gauge assembly did not occur thereby giving zero offset and excellent vacuum reliability.

Other thruster instrumentation included the following items: coils wound on the fire valve bodies for valve openings and closing times, propellant manifold pressure, piezoelectric and strain gauge chamber pressure instrumentation, injector, nozzle, and propellant temperatures. Figure 7 shows the thruster with instrumentation.

As noted in figure 4, the thruster instrument and control shroud contained all performance instrumentation and purge valves. For safety reasons, both propellant tanks were located outside of the space simulation chamber. Thus, each propellant line, extending from the tank to the fire valve was heated and had a length of 40 ft. Missile grade argon was used to pressurize the propellant tanks and as a purge for each line. A hard propellant system was obtained by opening each line to the space simulation chamber's vacuum environment until the pressure at the tank and fire valve dropped to 1μ.

Thruster System Results

Thruster performance results for pulse duty cycles are presented in figures 8 and 9. "On time" is defined as the time that power is applied to the fire valves for any one pulse. These results are for a single pulse with an on time of 200 msec. The time scale starts when voltage is applied to the fire valves. Propellant flow rate and valve position measurements indicate an 8 msec valve opening time. Fast time response chamber pressure measurements showed than an ignition lag of 3 msec occurred. Thus, thrust buildup begins 11 msec after power is applied to the fire valves. For the first 120 msec sizable oxidizer and fuel flow oscillations occurred. Consequently, time variations in O/F ratio of between 1.0 and 3.0 occurred as compared with the nominal steady state value of 1.6. These oscillations combine with the radial O/F gradients, as noted in reference 4, to form large changes in the combustion process and may be partially responsible for the formation of contamination. Thrust and specific impulse results (fig. 9) also indicate similar oscillations, with the same basic frequency, occurred on the independent chamber pressure measurement. Hence, the thrust and specific impulse should follow the same oscillatory action. It is apparent from figures 8 and 9 that the magnitude of the flow oscillations is larger for shorter on times. For this thruster, pulse on times less than 35 msec would hardly be enough time for complete thrust buildup.

Except for some shifting of the maximum and minimum values, similar results occurred for shorter pulses but only on the first pulse of the duty cycle. If the duty cycle consists of multiple pulses, with time between pulses of 100 msec or less, then the manifold pressure oscillations, due to the fast valve shutdown, have not damped out and are part of the propellant driving force for the next pulse.

Figure 10 shows the contamination resulting from a 2000 pulse test. In situ television monitors were used to watch contamination buildup on the nozzle lip. Several times during the pulse operation large segments were blown from the nozzle lip and deposited throughout the space chamber. Figure 10 also shows a flow of contamination leaving the nozzle lip and moving forward along the exterior surface. Thus, contamination buildup probably effects nozzle boundary layer flow which in turn effects the distribution of contamination both inside and outside of the plume boundaries.

Conclusions and Recommendations

Two conclusions can be obtained from this test program. First, cryopumping methods, using liquid helium cooled walls and an injected condensable gas, (in the present case, argon) are effective in producing the very low test chamber pressures needed for RCS thruster plume contamination tests. Second, measurements obtained during pulse mode duty cycles indicates that large flow oscillations occur in the propellant lines. These oscillations, caused by the fast opening and closing of the fire valves, along with residual propellant contained within the dribble volume of each valve appear to be responsible for the formation of a large fraction of locally deposited contamination. These conclusions imply that fire valves, having slower opening and closing times, should be tested to see if they eliminate the large propellant flow oscillation. Also RCS thrusters with different dribble volumes should be tested so that these effects can be isolated from the flow oscillation effects.

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Table I. - Design and Performance Characteristics of 5-Pound Thruster

(a) Dribble volume oxidizer manifold = 0.00058 in.^{3}
(b) Dribble volume fuel manifold = 0.00113 in.^{3}
(c) Injector = single doublet
(d) Oxidizer injector orifice diameter = 0.0186 in.
(e) Fuel injector orifice diameter = 0.0158 in.
(f) L^* = 4.5
(g) Nozzle throat area = 0.0298 in.^{2}
(h) Nozzle area ratio = 39.2
(i) Nozzle contraction ratio = 4.2
(j) Oxidizer - fuel ratio = 1.6
(k) Fuel and oxidizer value opening time = 8 msec
(l) Fuel and oxidizer value closing time = 10 msec
(m) Ignition lag = 3 msec
(n) Thrust = 5-lb
(o) Specific impulse = 286 sec
(p) Total propellant flow rate = 0.0175 lb/sec
(q) Steady state nominal oxidizer flow = 0.011 lb/sec
(r) Steady state nominal fuel flow for, O/F ratio = 1.6, = 0.0065 lb/sec
(s) Combustion chamber pressure = 100 psi

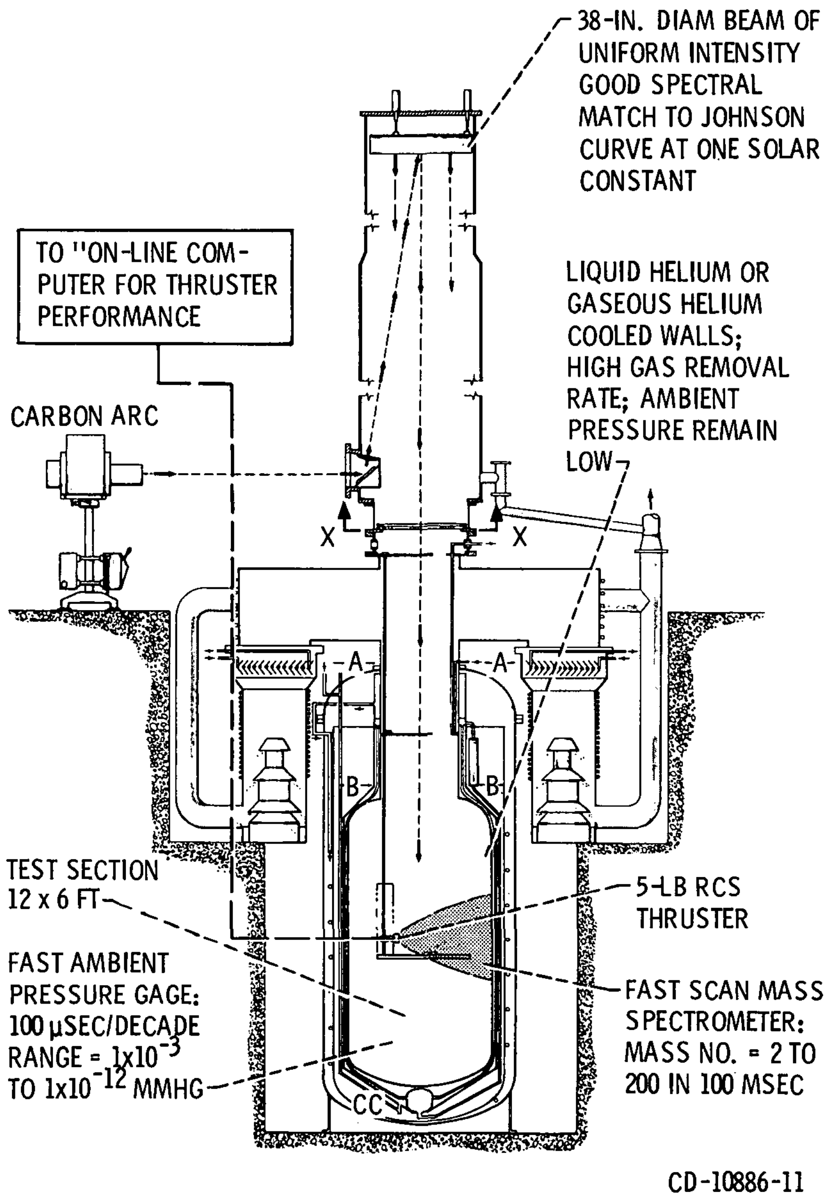

Figure 1. - In-situ rocket plume effects facility.

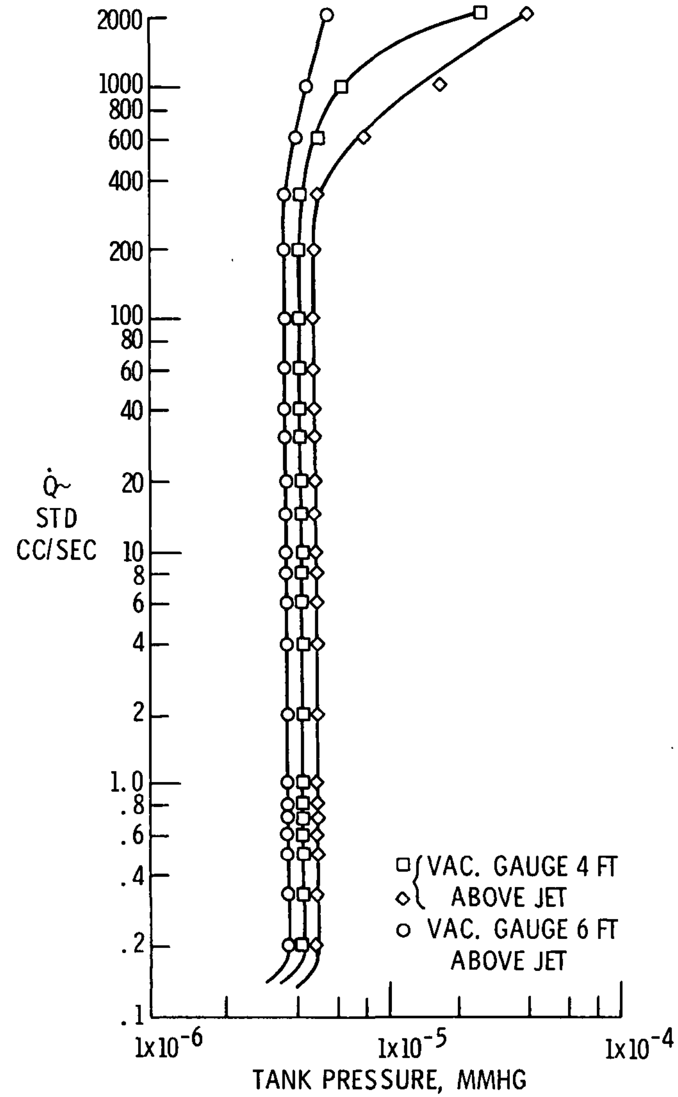

Figure 2. - Solar tank H_2 pumping tests. $\dot{Q}$ versus pressure.

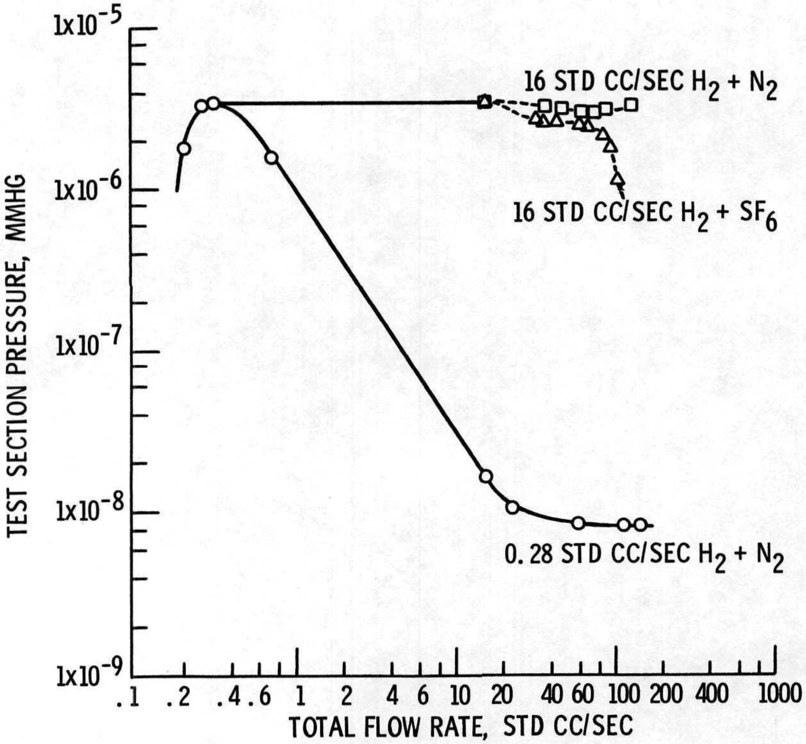

Figure 3. - Cryopumping test results.

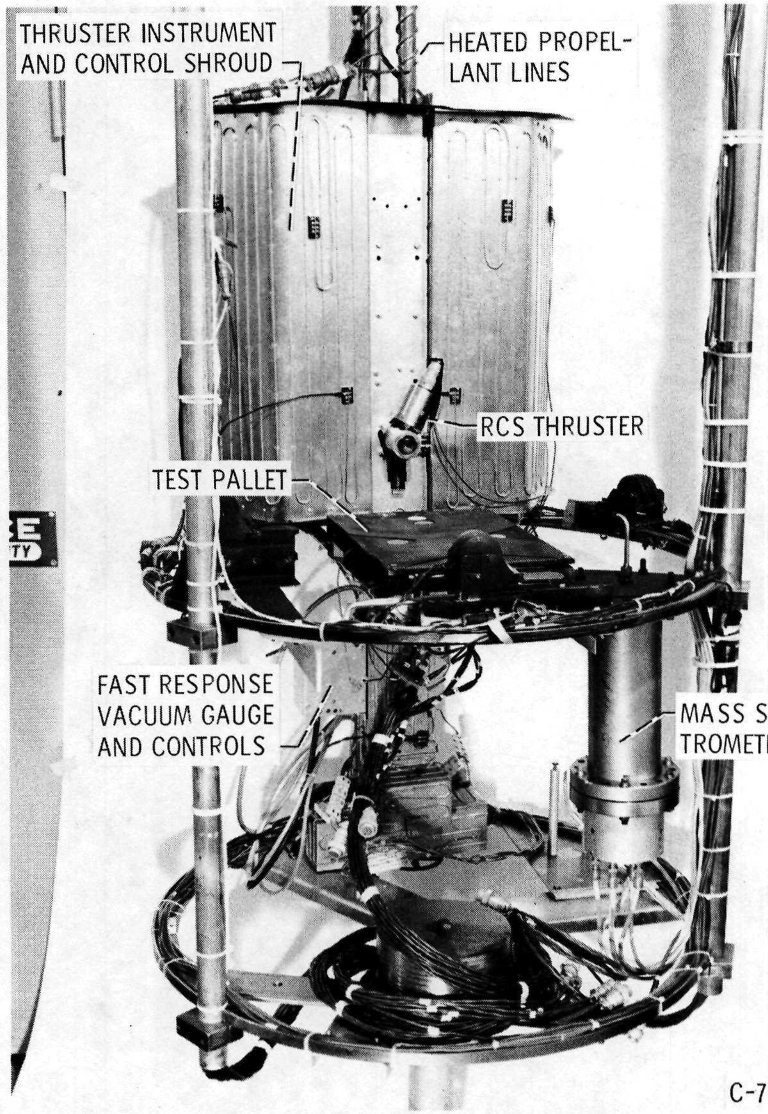

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Figure 4. - RCS thruster test package.

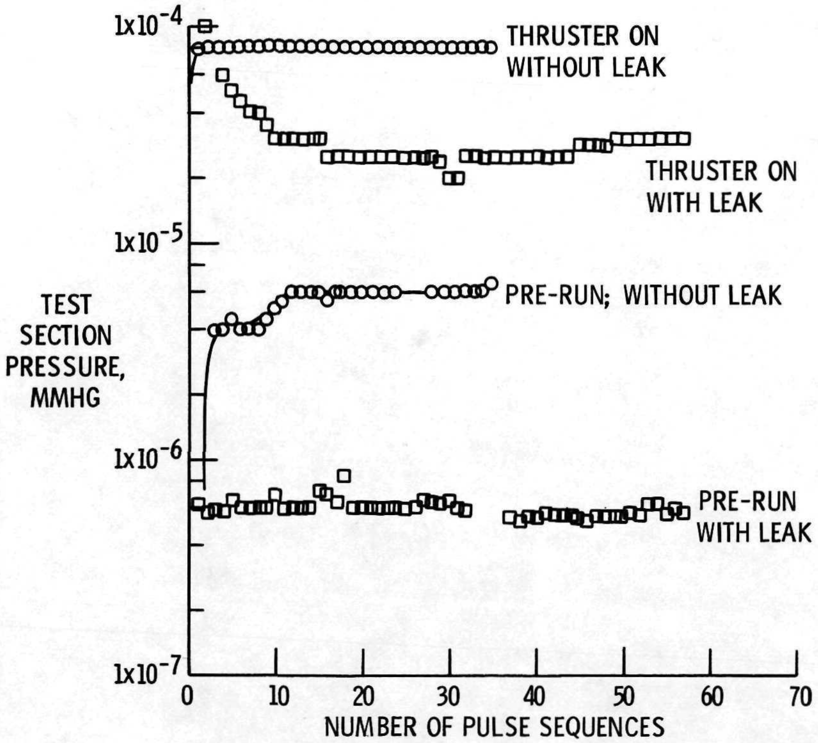

Figure 5. - Cryopumping effects.

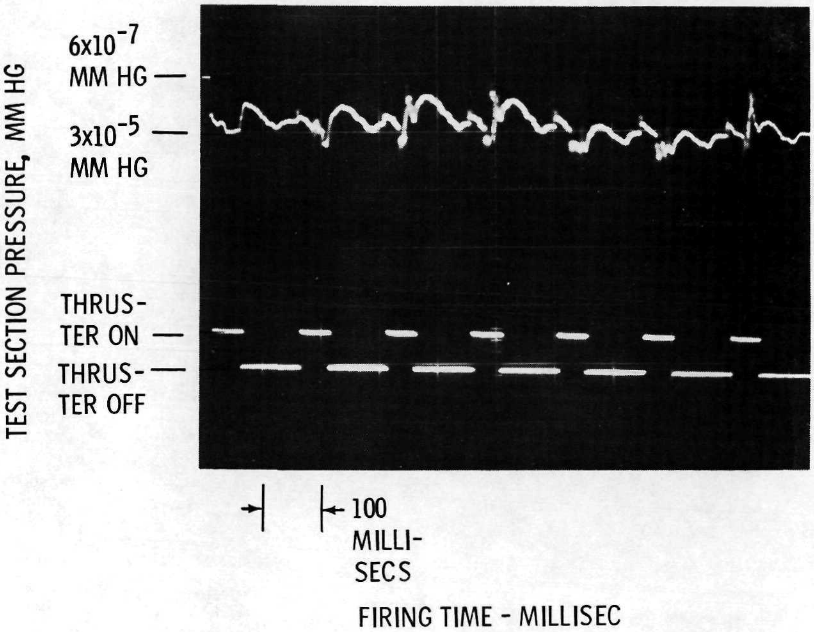

Figure 6. - Vacuum chamber pressure response.

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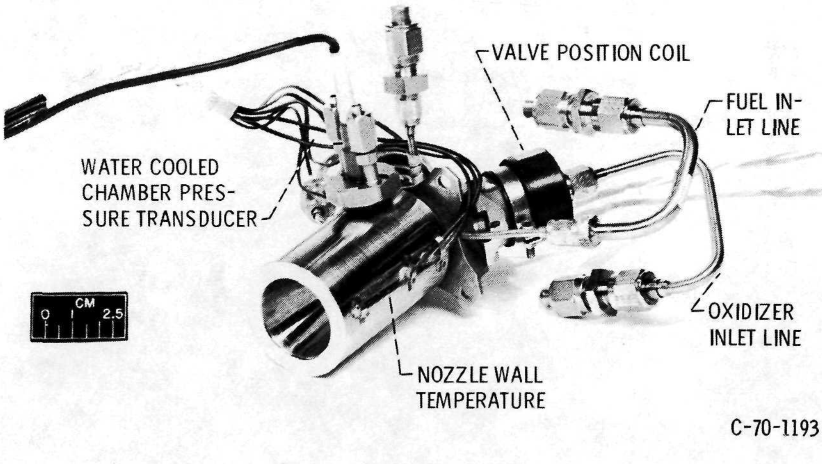

Figure 7. - RCS thruster.

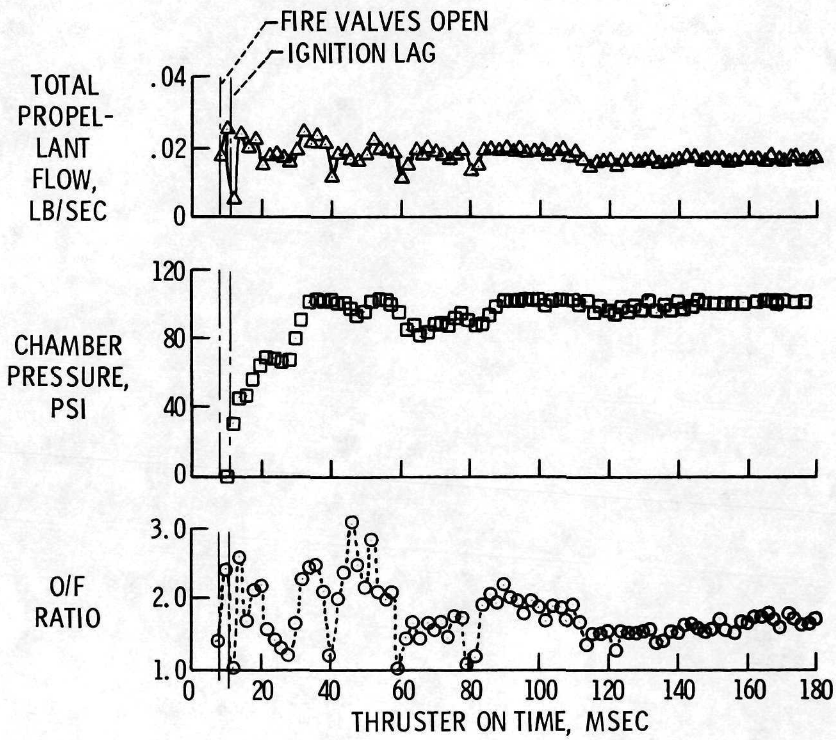

Figure 8. - RCS thruster performance.

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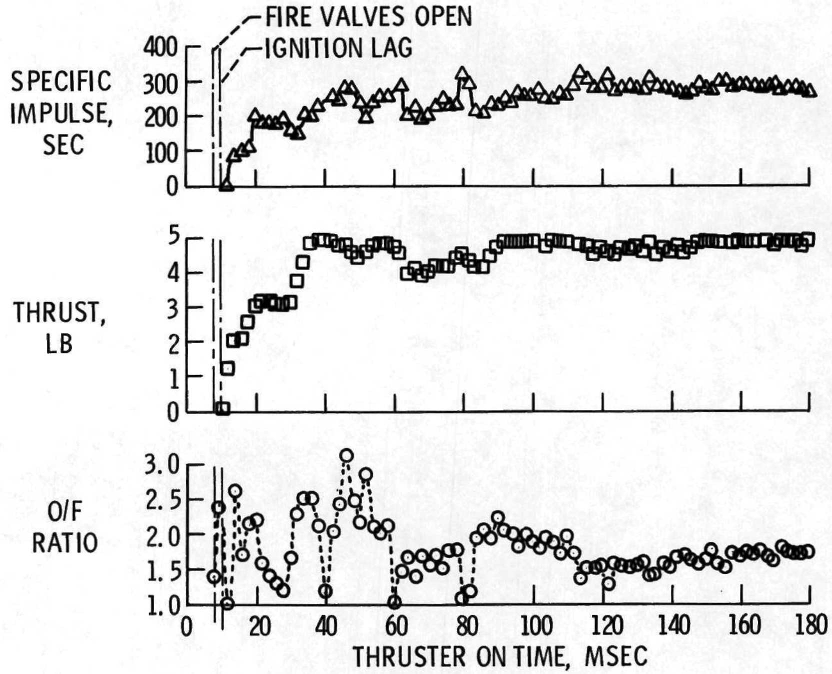

Figure 9. - RCS thruster performance.

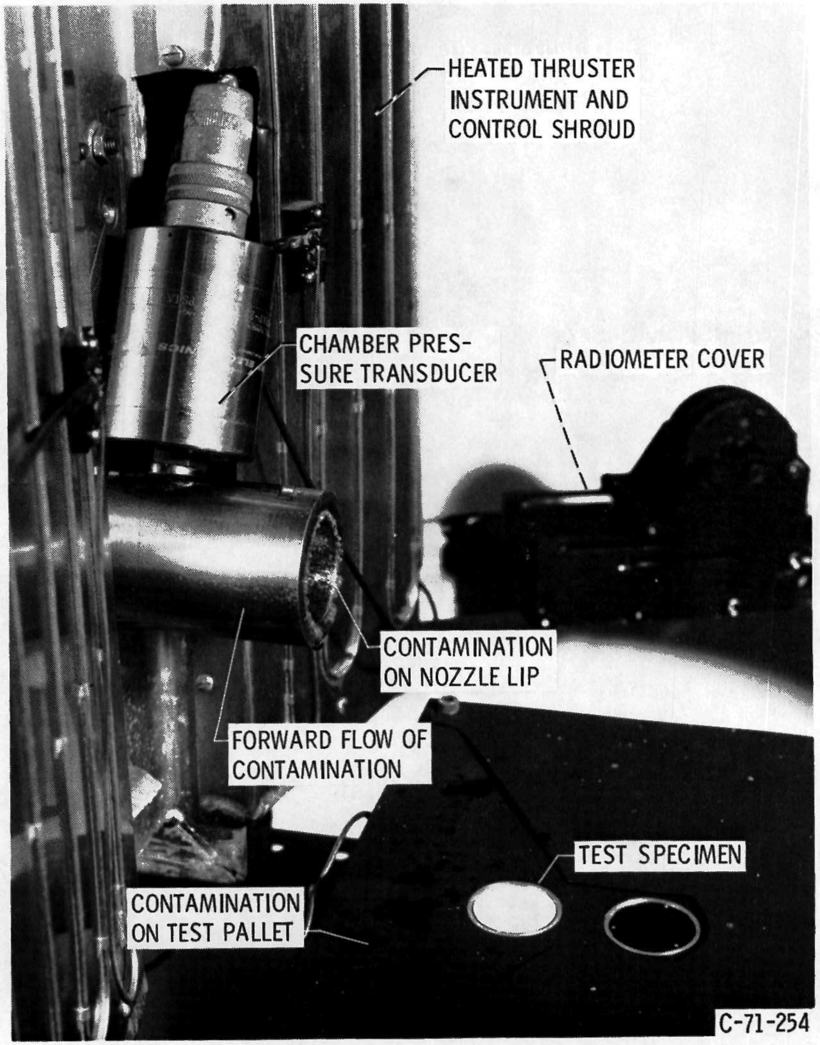

Figure 10. - RCS thruster contamination.

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