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# Design and Development of a Large Bipropellant Blowdown Propulsion System

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**The development and qualification of a large bipropellant propulsion system incorporating blowdown pressurization represents a departure from current systems employing pressure regulation. The wide range of operating conditions inherent in this approach presented technical challenges from the standpoint of thruster design and performance as well as system configuration. The resulting design requirements were addressed in a comprehensive thruster development program, and system-level verification was conducted using actual propellants and thruster firings. The viability of the blowdown approach for simple, long-life propulsion systems was demonstrated.**

## Introduction

BIPROPELLANT propulsion systems currently in use for spacecraft are pressure regulated to assure a high propellant loading efficiency and constant performance. The applications typically are for communications satellites where the large usage initially for orbit insertion is followed by early isolation of the regulation components, avoiding the need for regulator operation over long periods of time. A shallow pressure blowdown is allowed for on-orbit operations. However, other applications may call for significant propellant usage spanning the entire design life. In this case the regulated system entails failure modes and complexities that are undesirable. For the Bus 1 application, the decision was made to pursue the design and development of a large propulsion system with blowdown pressurization to take advantage of the inherent simplicity and reliability. Requirements for Bus 1 included multiyear lifetime, periodic propellant usage, three-axis control, and compatibility with both Space Transportation System (STS) and expendable launch vehicles.

The selected propulsion system for Bus 1 required the development and qualification of new hardware with operating and performance requirements generally exceeding that for previous bipropellant applications. The greatest technical challenges and program risks were in the thruster areas, which involved the development of both orbit adjust and reaction control thrusters. The system hot-fire test program conducted at the NASA White Sands Test Facility was extremely important to the program, not only for system verification, but also in uncovering some problems not identified previously.

## Requirements and System Selection

### Key Requirements

The primary design requirements called for a propulsion system that could provide over 3,000,000 lbf-s (13,344,000 N-s) for orbit adjust and reaction control functions over a multiyear life using monomethylhydrazine (MMH) and nitrogen tetroxide (NTO). The required thrust level for orbit adjust was determined to be approximately 200 lbf (890 N), and for the reaction control system (RCS) it was about 15 lbf (67 N). The thrust level for the RCS was driven by the requirement to maintain control during orbit adjust firings; in addition, the capability of providing three-axis control functions meant that a variety of duty cycles might be encountered. The required compatibility with both STS and expendable launch vehicles also affected system design.

### System Selection

The Bus 1 application required significant propellant usage over the entire design life, therefore serious consideration was given to finding an alternative to a relatively complex regulated system involving a large number and variety of components. Several factors were important in the determination that a blowdown system was feasible and appropriate for this application. First, considerable experience had been accumulated on large monopropellant blowdown systems involving propellant loads in excess of 5000 lbm (2270 kg) and making use of tank repressurization to increase loading efficiency. These three-axis applications resulted in many "lessons learned" in terms of operations, propellant management, and thruster duty cycle effects.[1] Secondly, the bipropellant blowdown option had been studied extensively, and a sound analytical basis was established.[2] Simulations and analyses showed that propellant residuals would not be excessive, thruster operating requirements could be confidently predicted, and thermodynamic processes during blowdown operation were understood. Finally, the development and testing of bipropellant thrusters had progressed to the point where the risk associated with new development would be acceptable.

## System Description

Figure 1 shows a schematic of the Bus 1 propulsion system that consists of two separate systems with the capability to

**Table 1 Propulsion system components**

| Component | Heritage |
|---|---|
| Propellant tank | New development |
| Pressurant tanks | Similar to existing tanks |
| Main engine | Scale-up existing engine |
| RCS thruster module | New development |
| Isolation valve | New development |
| Flex hose | New development |
| Fill/drain valves | Mod. existing design to titanium |
| Pyrotechnic valve | Existing design |
| Pressure transducers | Existing design |
| Propellant filter | New development |

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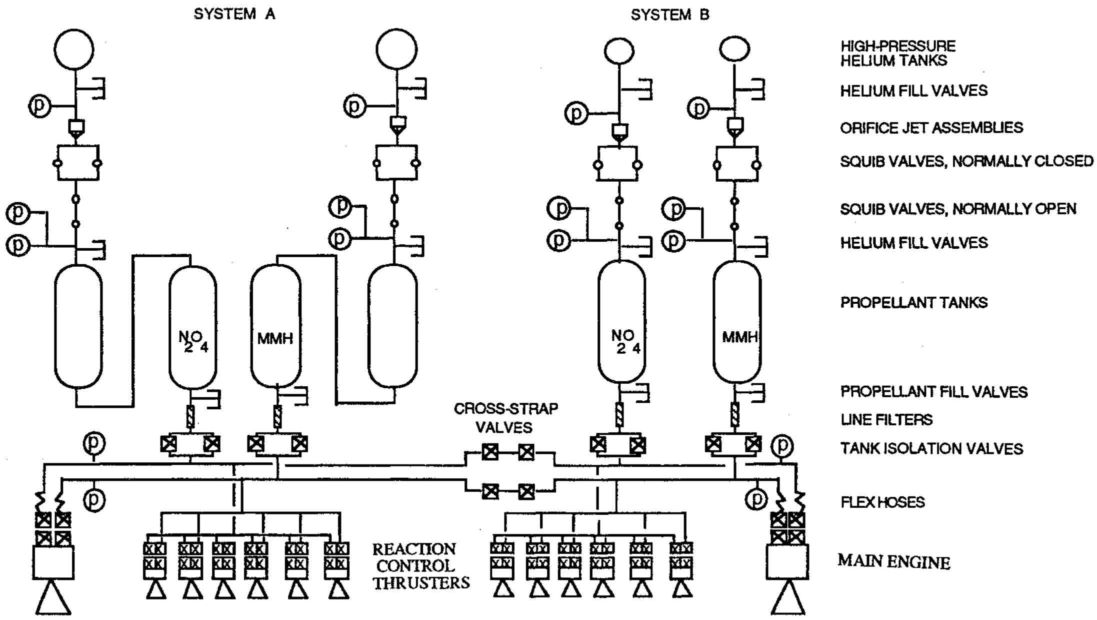


Fig. 1 Bus 1 propulsion system schematic.

cross-strap between tanks and thrusters. Table 1 lists the major system components and heritage. A propellant capacity of 11,500 lbm (5221 kg) is provided with an 80% fill fraction and a single repressurization. The titanium propellant tanks operate over a pressure range of 145–370 psia (1.00–2.55 MPa) and incorporate a surface-tension-type propellant management device to assure gas-free delivery of propellant to the engines under all acceleration conditions. The use of titanium tanks and lines assures oxidizer compatibility for the long-life application. The feed system solenoid isolation valves provide propellant isolation, leak protection, parallel flow redundancy from the tanks, and cross-strap capability between tanks and thrusters. The absolute back-pressure relief feature incorporated in the valves protects against overpressurization of lines and components due to thermal effects. The cross-strap valves are in series since cross-strap operation is not a normal activity and the consequences of leakage between system halves are potentially serious in terms of affecting mixture ratio and residuals for the system. The flexible propellant hose, also titanium, at each main engine allows for engine adjustment and alignment to accommodate changes in spacecraft c.g.

The high-pressure spheres upstream of the propellant tanks provide tank repressurization by firing pyrovalves when the tank pressure decays to a prescribed value. Figure 2 shows the pressure blowdown profile and illustrates the use of repressurization to increase propellant capability. Following tank repressurization, normally-open pyrotechnic valves are then fired closed to preclude migration of either liquid or vapor out of the propellant tanks. Thermal control using redundant heaters and thermostats is provided for tanks, fluid components, and thrusters. Extensive pressure and temperature instrumentation are built into the system to provide health monitoring as well as propellant mass statusing.

The R-42SR main engine shown in Fig. 3 is qualified over a 140–300 lbf (623–1334 N) thrust range and has a total impulse capability in excess of 3,000,000 lbf-s (13,344,000 N-s). An average specific impulse of approximately 300 s is provided with the 164:1 area ratio nozzle, and a maximum burn duration of 3600 s can be accommodated. During engine firing, cooling is achieved by fuel film cooling and chamber radiation; on-orbit thermal control is achieved through the use of heaters and thermostats within an aluminum thermal control cover that surrounds the front end of the engine. Series redundant fuel and oxidizer valves provide the desired

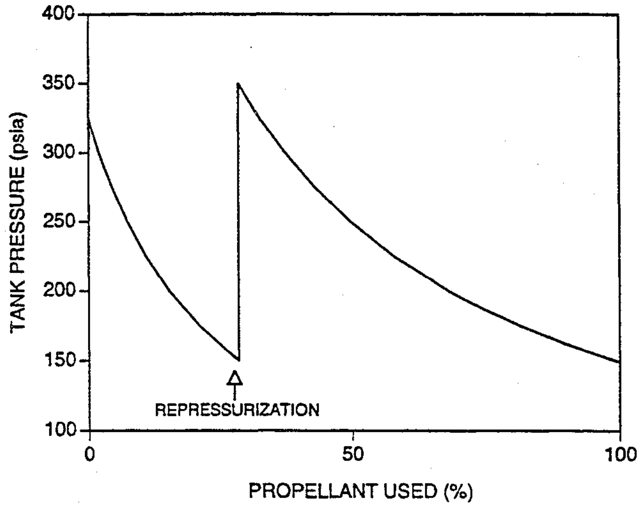


Fig. 2 Pressure blowdown profile.

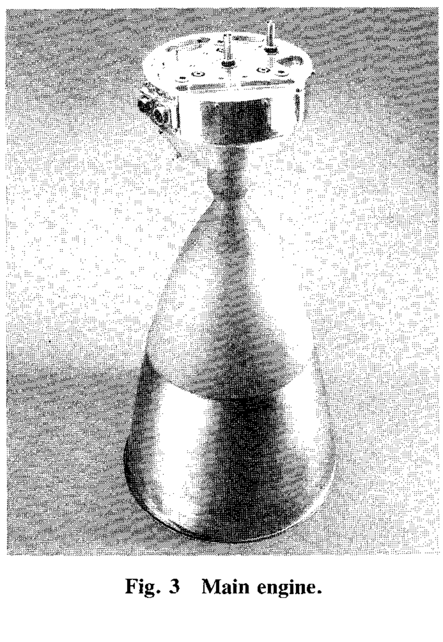

Fig. 3 Main engine.

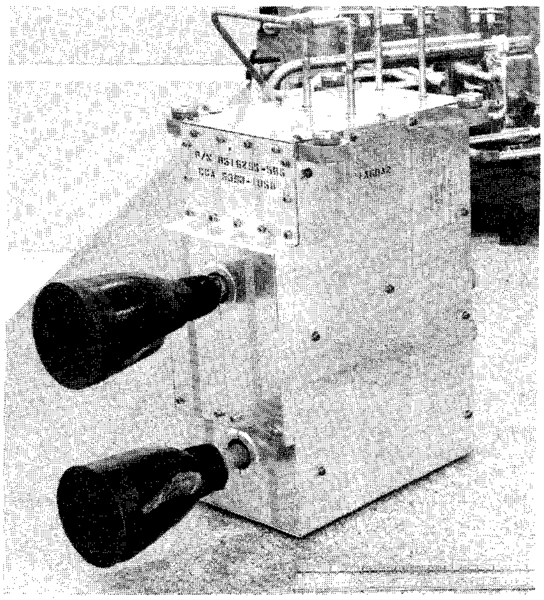

Fig. 4 Reaction control thruster module.

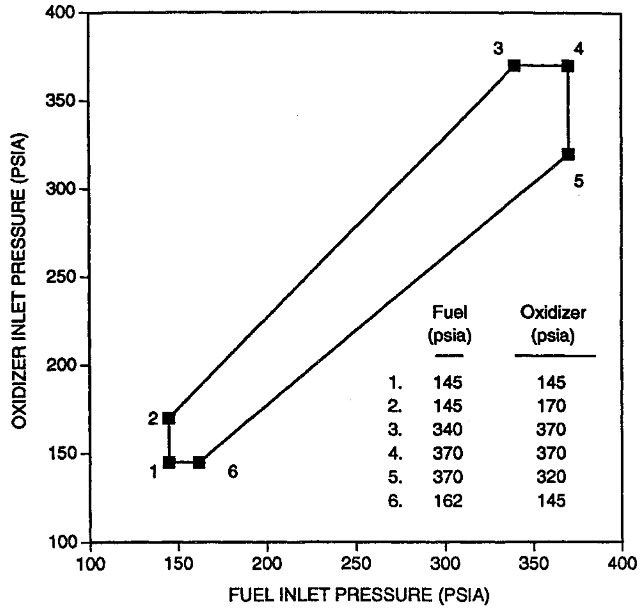


Fig. 5 Thruster operating envelope.

leak protection redundancy. The RCS thruster module shown in Fig. 4 packages two AJ10-220 thrusters and provides structural support and thermal control; heaters, thermostats, temperature monitors, and pressure transducers are contained within the module. The thrusters are qualified over a 10–22 lbf (44.5–97.9 N) thrust range, and have a demonstrated chamber life of greater than 40 h of operation. At the 15 lbf (67 N) thrust level, the specific impulse is approximately 287 s, and the minimum pulse width is 0.020 s. The RCS thruster also incorporates fuel film cooling and series redundant valves provide leak protection redundancy. The propulsion system has six thruster modules, each containing one system A and one system B thruster.

## Design and Performance Issues

The choice of a blowdown pressurization approach resulted in a number of additional requirements and issues that affected the system design and especially the development of the two thrusters.

### Thruster Issues

The major development issue for the Bus 1 propulsion system concerned the qualification of the main engine and reaction control thrusters to operate over the range of conditions resulting from the blowdown mode. A computer model incorporating pressurization characteristics, feed system design, and thruster performance was used to determine the thruster operating envelope that encompassed all conceivable variables, including loading uncertainties, helium solubility effects, propellant temperature, and thruster mixture ratio uncertainties. The resulting envelope is shown in Fig. 5. The challenge was to maintain good performance over the pressure and mixture ratio range while avoiding excessive chamber temperature and duty cycle thermal sensitivities. The thrusters would have to demonstrate a degree of resilience and flexibility not required for regulated operation, since the combination of blowdown operation plus three-axis control presented a very wide variety of possible operating conditions.

The adoption of series-redundant valves created the potential for fuel or oxidizer leads in the event downstream seat leakage occurred on one side, requiring verification that damaging ignition spikes would not occur. Thruster stability was another parameter requiring verification, and the wide pressure range in conjunction with mixture ratio and helium saturation variations increased the level of concern. Although helium saturation of the propellant was not unique to the program, the blowdown mode of operation meant that the degree of saturation could vary from very low at launch, to fully saturated several weeks after launch, to supersaturated during long firings with pressure blowdown. In addition, due to the two-system configuration, it was required that the thrusters be able to accommodate helium gas ingestion during propellant depletion of one system and then normal operation after cross-strap to the other propellant supply.

### System Issues

The maximum tank pressure, in this case up to 370 psia (2.55 MPa), means that system components must be qualified to higher than normal values. For the Bus 1 configuration, operating and proof pressure requirements are influenced by the presence of feed system isolation valves as well as the thruster valve design. For example, the highest static pressure on the thruster valves would occur when the system is at maximum pressure and isolation valves are closed; the thruster could experience the tank pressure plus an additional value due to back-pressure relief characteristics in the isolation valves. The specification of an absolute value rather than a delta pressure for back-pressure relief minimized the impact. The worst case pressure is in the intercavity region of the series-redundant thruster valves, since the upstream seat also has a back-pressure relief feature built into the design. In addition to static pressure requirements, the system must tolerate waterhammer effects over the entire pressure range.

Another significant issue involves feed system priming for the STS launch case. Whereas a regulated system can accomplish feedline priming while the propellant tanks are still at low pressure, the blowdown system with its initially high tank pressure entails pressure surge concerns when the isolation valves are opened, especially if the feedline is evacuated or at low pressure. Finally, there was some concern over the predictability of pressure and thermal characteristics during long burns since no experience existed in this area. Previous analyses[3] provided insight into the phenomena involved and confidence that performance and delivered impulse could be accurately predicted.

## Thruster Development Experience

This discussion highlights some of the technical problems and solutions. A comprehensive treatment of thruster issues and testing for bipropellant blowdown systems was previously addressed[4] and additional information pertaining to the main engine can be found in a previous paper.[5]

**Main Engine**

The starting point for the development of the main engine was the existing 110-lbf (489-N) design (model R-4D-11), and the approach taken was to scale-up the existing design, maintaining the same injector design and using the same engine valves. The existing thruster had shown an increasing chamber temperature as inlet pressure was increased, and the new 200-lbf (890 N) design exhibited the same trends. The conclusion was that the temperatures were too high at the high end of the blowdown pressure range, raising concerns over chamber coating life. Through a trial and error process of injector flow changes and testing, a solution was achieved that involved increases in fuel film cooling; the resulting performance penalty for accommodating blowdown operation was a derating of approximately 7 s in specific impulse. The resulting performance characteristics are shown in Fig. 6.

Thruster stability was another area of concern. However, testing over a wide range of inlet pressures, mixture ratio, gas ingestion, and degree of helium saturation verified satisfactory performance. High-altitude ignition testing at test cell pressures below the triple point of MMH investigated the potential for chamber overpressures; test variables included inlet pressure, propellant temperature, injector temperature, chamber temperature, and fuel or oxidizer leads. Figure 7 shows some of the test results. The maximum recorded chamber pressures were well below the static burst capability of the chamber, and the testing demonstrated that downstream valve seat leakage would not create hazardous conditions associated with fuel or oxidizer leads.

Use of the existing engine valve in a series-redundant configuration and at a higher flow rate did pose some challenging issues and required design modifications. Internal contour changes were made to reduce pressure loss, and the valve poppet stroke was increased to increase the flow area. The latter change did require an increase in power consumption. System testing, discussed in a later section, revealed an oxidizer flow problem necessitating some additional design changes.

**Reaction Control Thruster Module**

Development of the 15-lbf (67-N) thruster used as a point of departure an existing 5-lbf (22-N) design involving a platelet injector. Injector design and test iterations were made to achieve the desired performance characteristics over the operating range while avoiding excessive chamber temperatures. The high inlet pressures necessitated a relatively high percentage of fuel film cooling, resulting in a slightly derated design. Figure 8 shows the performance characteristics. One case of high-frequency instability was encountered after the injector acoustic cavities were deliberately blocked in an attempt to reduce upstream heat transfer during pulsing duty cycles. It was concluded that the acoustic cavities were required for stability margin, especially at the higher inlet pressures. The high-altitude ignition testing involving fuel and oxidizer leads was similar to the one conducted on the main engine, and the testing showed that downstream seat leakage would not result in damaging ignition spikes.

The most challenging technical issue was achieving satisfactory thermal control of the thruster and module during pulsing operation over the wide range of operating conditions associated with the blowdown mode. Testing showed that thermal control during pulsing was significantly influenced by inlet pressure, mixture ratio, pulse width, and duty cycle. Figure 9 illustrates a duty cycle sensitivity discovered in testing

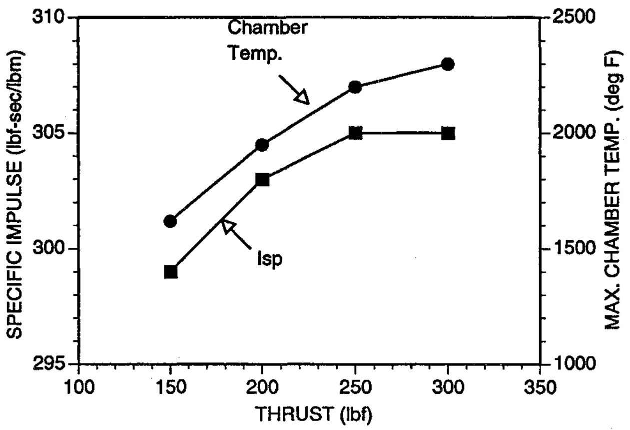


Fig. 6 Main engine performance.

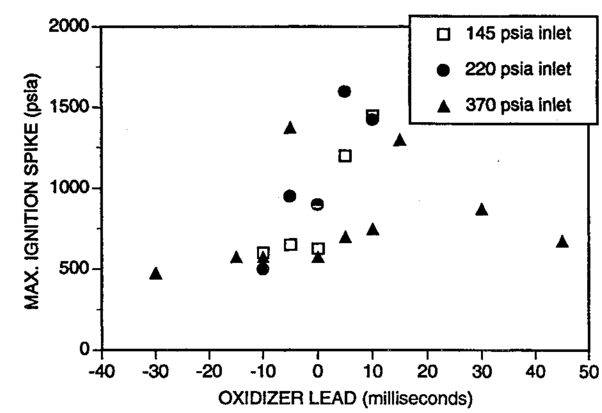


Fig. 7 Main engine ignition spikes.

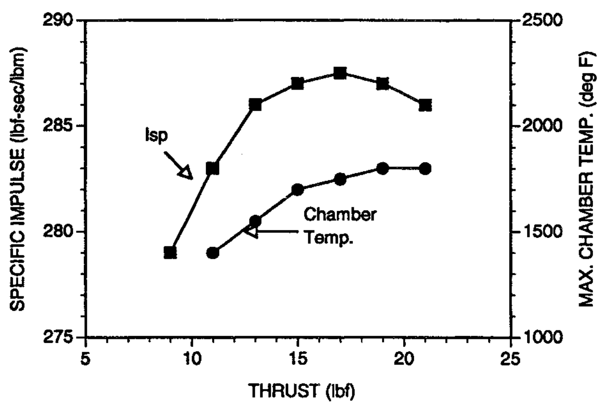


Fig. 8 Reaction control thruster performance.

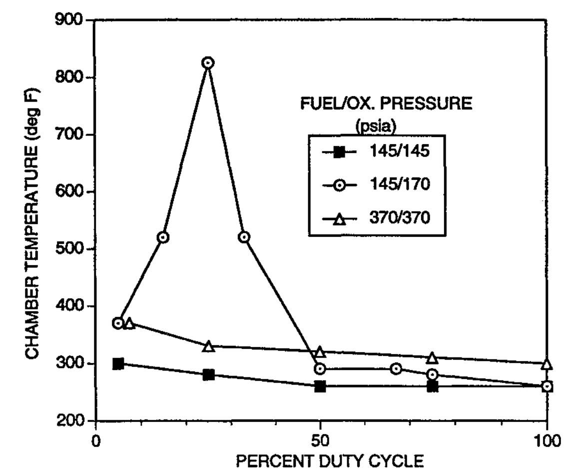


Fig. 9 Example of duty cycle sensitivity.

at low inlet pressures and high mixture ratio. A very extensive test program was required to investigate the various duty cycles and to check out proposed solutions. Injector fuel film cooling was increased and the chamber wall thickness was decreased to impede the heat soakback toward the head end of the thruster. More efficient shunts were incorporated to transfer heat away from the thruster head end into the module structure. In addition, since testing had shown that longer pulse widths were more effective in maintaining film cooling efficiency, the pulse width was baselined at the highest value consistent with other control requirements.

## System Testing

The decision to conduct a system test, involving actual propellants and thruster firings, was driven by the departure from existing experience plus the new developments in tank, thruster, and component areas. A blowdown system of this nature and scope had not been previously developed nor flown, and there were a number of new issues that made verification at the system level desirable. Specific objectives included characterization of propellant loading, pressurization, pressure transients during priming and thruster operation, helium saturation effects, and gas ingestion. An important objective was to establish an acceptable feed system priming configuration and procedure.

### Test Article Description

The test article is a flight-quality, all-welded titanium propulsion system using flight-type components, tubing, and plumbing fittings. The fuel tank contained a full propellant management device (PMD), whereas the oxidizer tank was an empty shell. This approach saved on cost while offering an opportunity to evaluate thermodynamic effects resulting from the presence of a PMD. The feed system plumbing closely simulated a flight layout and included the propellant filter, propellant isolation valves, and titanium flex hose upstream of the main engine. Instrumentation fittings were installed at various points in the feed system, and high response transducers were used to monitor transient pressure characteristics. The main engine and reaction control thrusters were supplied with truncated nozzles for sea level operation, since test objectives did not require a vacuum test. Thruster instrumentation included chamber pressure transducers and temperature sensors.

### Test Results

System testing was conducted over a 7 month span at the NASA White Sands Test Facility. The initial helium absorption tests consisted of pressure and temperature monitoring for the purpose of determining expected operational trends. Pressure decay characteristics are shown in Fig. 10, illustrating the greater oxidizer solubility. The test was later repeated

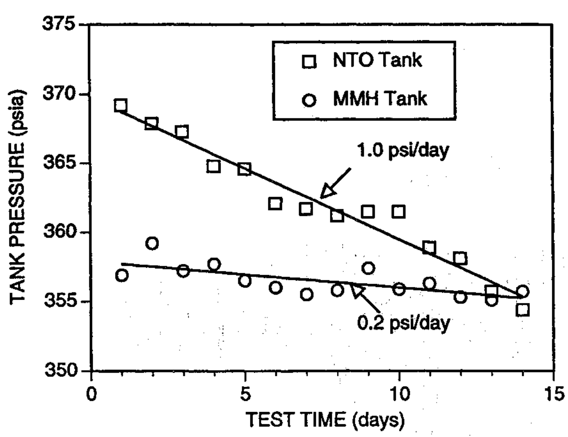


**Fig. 10 Helium absorption test results.**

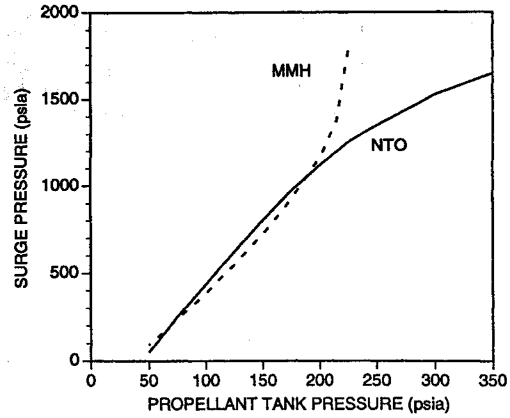


**Fig. 11 Surge into evacuated lines.**

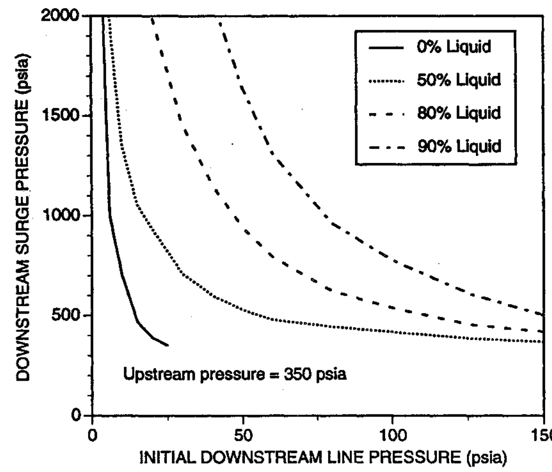


**Fig. 12 Fuel priming characteristics.**

with oxidizer in the tank with the PMD, and the results indicated that the presence of the PMD did slightly inhibit the helium absorption rate.

The thrusters were fired to obtain system dynamics data as well as to achieve tank pressure decreases for the testing. The main engine was fired a total of 146 times for 5118 s, whereas the three reaction control thrusters were fired a total of 1198 times for 2027 s. Investigation of waterhammer effects over a wide range of tank pressures and degrees of helium saturation showed that transient pressures were within the expected range and did not present operational hazards. As predicted, the highest pressures were in the smaller diameter lines near the thrusters, and peak pressures were about 25% lower than analytical predictions.

An important part of the test program was the determination of pressure transients resulting from isolation valve actuation to prime the feed system. The data from the pressure surge tests were evaluated to determine the initial conditions required to satisfy system safety requirements while precluding damaging pressure surges upon feed system priming. As expected, opening the valves into evacuated lines was found to be unacceptable, and Fig. 11 shows peak pressures experienced as upstream pressure was increased from test to test. It was decided not to proceed with increased upstream pressure on the fuel side due to concerns over hardware damage. A number of tests were conducted with the downstream line configuration varying from all-gas to various percentages of gas–liquid mixtures. Figure 12 shows the trends for the fuel side, and oxidizer trends were similar. Significant findings from the surge pressure testing were 1) the evacuated downstream condition resulted in the highest surge pressures, with the oxidizer side being less severe due to the higher vapor pressure and lower fluid velocity; 2) surge pressures were highest in the small-diameter RCS lines; 3) for a given downstream pressure, the all-gas condition results in the lowest surge pressures, and increasing liquid fill percentages results in increased pressures; and 4) the all-liquid condition can result in very high surge pressures, especially since the initial downstream pressure is highly sensitive to thermal changes in the line.

Based on the surge test results, it was decided to prescribe a 75% liquid fill with 240-psia (1.65-MPa) helium pressure pad as the initial condition for flight operations. While an all-gas situation might have been the best choice, a failure mode exists whereby gas leakage can result in line evacuation and potentially damaging surge pressures. The presence of liquid in the lines provides sealing against gas leakage from the feed system. The selection of this approach allowed use of the existing design without having to introduce additional components such as special bleed valves.

System propellant compatibility and cleanliness were verified by posttest measurements and inspections. A total of 1950-lbm (885-kg) oxidizer and 1310-lbm (594-kg) fuel were put through the filters without evidence of plugging, and visual examination of the system, main engine, and isolation valve filters showed no contamination. Within sampling accuracies, there was no change in propellant chemical analysis or particulate count. An unexpected hardware finding was that the relief pressure of the isolation valve was affected by changes in inlet pressure, and the 450–480 psia (3.10–3.31 MPa) requirement was not being met. This resulted in corrective action to provide close control of the poppet area to bellows relationship, assuring that relief pressure will be independent of inlet pressure.

The most serious problem noted during the testing was a departure from the predicted propellant usage, with the oxidizer usage being 11% low and fuel usage 4% greater than expected. The problem was attributed to an anomaly in the main engine that manifested itself in a gradually degrading chamber pressure. In addition, leak testing conducted immediately after the test showed that upstream oxidizer seat leakage was well above specification requirements. An investigation concluded that Teflon® seat swelling and distortion was occurring as a result of long-term oxidizer exposure and intercavity pressure increases during heat soakback. The combination of flow area reduction and saturated propellant led to flow cavitation and associated flow reduction in the upstream oxidizer valve. The design fix involved an increase in valve stroke, modifications to the upstream valve seat design, and use of a higher temperature in teflon processing during valve buildup.

## Conclusions

Development and qualification of the Bus 1 propulsion system was successfully concluded, demonstrating the viability of bipropellant blowdown operation for spacecraft. A number of new technical and design issues were encountered and resolved, and new thrusters were developed that could operate over a wide range of conditions. The importance of system testing using flight-type components and operating conditions was demonstrated, and test results influenced both system design and procedures.

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