N85-16939

CRYOGENIC PROPELLANT MANAGEMENT - INTEGRATION OF DESIGN, PERFORMANCE AND OPERATIONAL REQUIREMENTS

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ABSTRACT

The integration of some of the unique design features of the Shuttle elements into a cryogenic propellant management system is described. The implementation and verification of the design/operational changes resulting from design deficiencies and/or element incompatibilities encountered subsequent to the critical design reviews are emphasized. Major topics include subsystem designs to provide liquid oxygen (LO_2) tank pressure stabilization, LO_2 facility vent for ice prevention, liquid hydrogen (LH_2) feedline high point bleed, pogo suppression on the Space Shuttle Main Engine (SSME), LO_2 low level cutoff, Orbiter/engine propellant dump, and LO_2 main feedline helium injection for geyser prevention.

INTRODUCTION

America's Space Shuttle program challenged the cryogenic propulsion disciplines to extend the single launch Saturn-Apollo technology into a multilaunch space vehicle. Some of the parametric studies that were conducted to define the features of a reusable cryogenic propulsion system are summarized in reference 1. The design of each Space Shuttle element (SSME, Orbiter, External Tank (ET), and ground support facilities) was influenced by the reusability requirements and by the program goal of low cost per flight. The design and development of the ET cryogenic components and subsystems are presented in references 2, 3, and 4. The challenge to the engineers/designers of the Cryogenic Propellant Management System was to assure functional and operational compatibility of the interfacing elements. A primary emphasis was to resolve design deficiencies or to implement requirement changes encountered in the development process. The implementation of many of the changes encountered was accomplished by (1) utilizing software and control functions in lieu of hardware redesign and (2) extending the function of existing components. These approaches were selected in order to minimize impact to the program schedule and the cost objectives.

CRYOGENIC PROPELLANT MANAGEMENT OVERVIEW

The design and operation of the Cryogenic Propellant Management System are to provide LO_2 and LH_2 propellants at conditions that are compatible with the requirements/capabilities of the interfacing subsystem. The facility and vehicle propellant conditions are controlled during the prelaunch operations to preclude undefined loads from being imposed on the elements and to assure that the engine prestart requirements are achieved. Ascent performance requires that (1) the nominal usable propellant mass be 1,345,000 pounds ± 0.5% of LO_2 and 225,000 pounds ± 0.65% of LH_2 and (2) the residual propellants in the Orbiter and engine be dumped after SSME cutoff.

The LH_2 propellant delivery system shown in Figure 1 consists of main tankage with level control sensors and dual-function vent/relief valve, an internal "siphon" feedline, ET/Orbiter disconnect, Orbiter manifold and feedlines to the three SSME's, fill and drain line, LH_2 recirculation and high point bleed subsystem, and the SSME's. Propellant is loaded at high flow rates through the fill and drain line which connects the LH_2 ground servicing facility to the manifold. The low flow rates for topping and replenish are routed from the fill and drain line through the recirculation return line to the LH_2 tank, bypassing the Orbiter/ET feedline. The SSME LH_2 preconditioning is accomplished by recirculation pumps mounted on the Orbiter manifold. Each pump forces LH_2 around the prevalve, through the feedline and engine. Downstream of the engine, the recirculation flow joins the replenish flow in the recirculation return line. The recirculation pumps are powered by electric motors operated with ground power, since their function is completed prior to lift-off.

LH_2 liquid level control during replenish utilizes the duty cycle (percent-wet) of the ET point (warm wire) sensors as the input to the facility Launch Processor System (LPS). The LPS adjusts the position of a ground control valve to provide makeup fluid to compensate for boiloff losses. This system maintains the propellant level within ± 2 inches (± 0.1%) of the desired level. The balance of the loading accuracy error budget is allocated to propellant density and ET dimensional uncertainties.

The LO_2 propellant delivery system shown in Figure 2 consists of main tankage with level control sensors and dual-function vent/relief valve, ET feedline (with accompanying antigeyser line on vehicles 1-4), ET/Orbiter disconnect, Orbiter manifold and feedlines to each of the three SSME's, and the SSME's which contain the LO_2 bleed system and pogo suppressor system. LO_2 is loaded through the fill and drain line which connects the manifold to the LO_2 ground servicing facility. During periods of low flow rate into the ET (slow fill to 2%, topping from 98 to 100%, and replenish at 100%), subcooled liquid is maintained in the LO_2 main feedline by using the thermal pumping of the 4-inch diameter antigeyser line to circulate liquid from the tank down the main feedline and back up the antigeyser line. Subcooled liquid in the ET's main feedline is essential to preclude LO_2 geysering (the formation of gaseous oxygen (GO_2) vapor pockets in the feedline which expand rapidly, expelling liquid from the feedline into the tank, leading to a sudden and damaging refill "water-hammer"). Helium is injected into the aft elbow of the antigeyser line to assure flow circulation. Throughout the loading and replenish operations, LO_2 is bled through the SSME turbopumps and then overboard through the engine/Orbiter bleed system.

LO_2 liquid level control during replenish is similar to LH_2 level control. The LO_2 replenish system must make up the boiloff losses and provide the bleed flow required for SSME thermal conditioning. The LO_2 level is maintained within ± 3 inches (± 0.15%) of the desired level. The balance of the loading accuracy error budget is allocated to propellant density and ET dimensional uncertainties.

DEVELOPMENT AND INTEGRATION OF DESIGN

An integrated system approach to the development and operation of the Cryogenic Propellant Management System is illustrated by Figure 3, which contains subsystem requirements that were changed subsequent to element critical design reviews (CDR's). These requirement changes were the most cost effective/timely solutions to problems or element incompatibilities encountered during the development process. The subsystem designs to provide LO_2 tank pressure stabilization, LO_2 facility vent for ice prevention, LH_2 feedline high point bleed, pogo suppression on the SSME, LO_2 low level cutoff, Orbiter/engine propellant dump, and LO_2 main feedline helium injection for geyser prevention will be discussed.

LO_2 TANK PRESSURE STABILIZATION

A design goal of the ET program was a free-standing unpressurized structure. However, the forward ogive of the LO_2 tank of the ground vibration test article buckled while being filled with liquid. Subsequent analyses and testing of the structural test article defined a pressure requirement of 1.7 psig to preclude buckling of the forward ogive for liquid levels above 2%. The methods to satisfy the pressure requirement without major impact on other subsystems were developed on the Main Propulsion Test Article (MPTA), a flight hardware test facility used to verify the performance of the integrated cryogenic propulsion system. The initial approach to meet the pressure requirement was based on increasing the vent flow resistance with a corresponding increase in ullage pressure and liquid saturation temperature. A 2.75-inch diameter orifice in each of the two 5.5-inch diameter vent ducts (downstream of the vent/relief valve) maintained the ullage pressure above 1.7 psig when the tank was loaded at approximately 5000 gallons-per-minute. The orifices would not maintain adequate tank pressure during replenish or fill at KSC flow rates of 1400 gpm. Therefore, the ET vent valve capability to control ullage pressure during loading and replenish was evaluated.

The use of the vent/relief valve for pressure control was complicated by the design that placed the valve control functions on the facility. However, tank pressure control by vent valve cycling was accomplished by experimentally determining acceptable control limits and modifying the LPS software to provide the valve control functions. Narrow range pressure transducers were added to the tank and used by LPS to determine when to close the valve. During loading, LPS closes the valve at 2.2 psig and opens the valve at 8 psig. The 2.2 psig limit accounts for system reaction time and instrumentation errors such that the minimum ullage pressure is $\geq$ 1.7 psig. The 8 psig upper limit allows time for the LPS to identify a valve failed closed and then perform alternate procedures which will maintain the ullage pressure below 17 psig, the maximum pressure that would not put a pressure cycle on the tank. During replenish, the valve is closed at 2.2 psig and opened 3 minutes later. The time limit ensures consistency of cycles for any replenish flow rate. The lightweight tank (LWT) design (ET-7 and subs) includes additional structure (250 pounds) to eliminate the pressure stabilization requirements for levels above 98%.

The vent valve cycling to provide tank pressure stabilization makes the ground control function critical for structural integrity. The LPS monitoring, control logic, and corrective actions must be able to compensate for component failures and off-nominal operating conditions. In addition to increasing the quantity and criticality of the software, two design changes were required. Specifically, the helium vent valve closure actuation flow was separated from the helium injection flow to the anti-geyser line (originally the design had helium inject off whenever the vent valve was actuated closed), and the gaseous nitrogen (GN_2) auxiliary pressurization flow rate was increased so that the auxiliary system could maintain the ullage pressure during an emergency drain with the vent valve failed open.

LO_2 FACILITY VENT FOR ICE PREVENTION

The ET LO_2 tank and protuberances were uninsulated in the initial design concept. During the development of the Orbiter reentry thermal protection system, it became apparent that ice/frost falling off the ET due to lift-off vibrations could damage the tiles and endanger the Orbiter during reentry. The requirement to eliminate ice/frost formation on the ET was imposed just prior to the CDR. The design changes incorporated to satisfy this requirement included: (1) ground controlled, heated purges for the intertank compartment, nose cone cavity, and pressurization lines; (2) insulation on the ET acreage and small protuberances; and (3) electrical heaters under insulation on large protuberances. The addition of insulation to the LO_2 tank reduced the heat input to the cryogen such that one of the two vent valves was eliminated from each tank.

A test program to assess the effectiveness of ET design changes for ice/frost prevention determined that the GO_2 vent louvers would accumulate ice. The resulting requirement to preclude ice/frost on vent louvers was unique, i.e., prior launch vehicles exhausted GO_2 directly into the atmosphere. Modifications to provide a hard disconnect umbilical for the GO_2 vent similar to that used for gaseous hydrogen (GH_2) would have been extensive. Therefore, a facility GO_2 vent hood was selected to remove the vent gases from the tank with an inflatable dock seal (Figure 4) to provide a soft "footprint" on the ET insulation surface.

The vent hood has a dock seal for each of the vent louvers. These seals are attached in a retractable vent hood tip assembly mounted on a service arm off the launch pad fixed service structure. The dock seals are inflatable Herculite cloth with an LO_2 compatible beta cloth liner. The dock seals are inflated to 0.5 psig and are used to duct the vent gases from the louvers to a pair of exhaust ducts that remove the gases from the immediate vicinity of the LO_2 tank. A GN_2 purge in the vent hood volume, external to the dock seals, eliminates the accumulation of hazardous gases. A separate GN_2 flow purge (25 lb/min) through the dock seals provides thermal conditioning of the flexible material, thus ensuring no leakage under the dock seal "footprint" on the tank surface.

The design verification testing of this system, presented in reference 5, utilized a complete LO_2 tank vent system and nose cone assembly to assure realistic system performance. The test conditions for tank pressures, vent temperatures, and flow rates were derived from the MPTA. The LO_2 tank pressure range during operation varied from 22 to 2.2 psig with vent temperatures ranging from ambient to -220°F. Subsequent to a dock seal failure during the initial test series, testing revealed velocity pressures resulting from the nonsymmetrical flow downstream of the 2.75-inch orifices in the vent manifold. Experimental evaluation of the tank vent system showed that the 2.75-inch orifices in the two ET vent ducts should be removed and the vent valve stroke changed from 2.6 to 1.1 inches. These changes substantially reduced velocity pressures at the louvers as shown in Figure 5. Testing with two pairs of service arm ducts (12-in OD X 62-ft long and 24-in OD X 27-ft long) showed that duct size was a significant factor in reducing the pressure spike in the dock seal plenum when the vent valve was opened, i.e., the forces required for acceleration of the residual gases in the facility ducts produced substantial pressures in the dock seal plenums. The 24-inch diameter by 27-foot long ducts were selected for the Pad 39A launch facility.

The changes to the tank vent system and facility ducting, in conjunction with component optimizations, i.e., bungie sizes, dock seal sizes, and internal pressure, resulted in a functional LO_2 vent hood system that prevents ice accumulation on the LO_2 tank due to GO_2 venting. The design also provides reconnect capability to the tank if required by a countdown recycle and minimizes possible damage to the tank insulation. The dock seals are deflated prior to service arm retraction and the bungie cords retract the seals from the tank surface approximately 2 1/2 minutes prior to launch. If reconnect of the vent hood to the tank is required, the service arm is moved back over the tank, the vent hood is lowered, and the dock seals are inflated.

LH_2 FEEDLINE HIGH POINT BLEED SYSTEM

The LH_2 propellant feed system of the Shuttle elements (Figure 1) resulted in an inverted U-tube design that traps the warmer propellant flowing upwards from the tank bottom and collects it at the high point of the Orbiter 17-inch feedline. Integrated system analysis showed that the stratified fuel would vaporize, forming a large pocket of hydrogen vapor in the feedline during tank replenish. This vapor volume would grow, cavitating the recirculation pumps and resulting in cessation of engine thermal conditioning and violation of the SSME start requirement. Analysis indicated that the hydrogen bubble in the feedline would not recondense during tank prepressurization, resulting in bubble ingestion by the SSME's during start or mainstage operation with potentially catastrophic results. Therefore, a high point bleed system was added to prevent vapor accumulation in the hydrogen feed manifold.

The system consists of a 3/4-inch insulated line connected to the Orbiter disconnect with a high point bleed valve approximately 2 feet downstream of the inlet and a disconnect valve at the Orbiter/facility interface. An orifice in the facility line limited the bleed flow rate. Extensive high point bleed system testing on MPTA resulted in removal of the flow-limiting orifice in the facility. Bleedline performance was sensitive to the facility vent line back-pressure due to the low pressure head available (2.8 psi) to expel the hydrogen vapor. Therefore, a separate facility vent line was provided for the high point bleed system. The MPTA data showed that the high point bleed system should be chilled and operational prior to the start of the recirculation pump in order to ensure normal pump performance. The prelaunch operation of the bleed system is continued until the recirculation pumps are turned off to assure bubble free operation at engine start. System performance is monitored by the LH_2 feed manifold disconnect temperature and the high point bleed temperatures in the Orbiter and facility line. A manifold disconnect temperature less than 45°R indicates a vapor free feedline when the tank is unpressurized.

The high point bleed system also helps to reprime the recirculation pumps after a recirculation flow interrupt resulting from power failure to the pumps or test sequence recycling. A further use of the high point bleed system being considered is the removal of trapped LH_2 feedline residuals (70 pounds) following an aborted mission with a return to launch site (RTLS). This function could be accomplished by connecting a facility line to the bleedline disconnect, pressurizing the feedline through the on-board feedline repressurization system, and allowing the liquid residual to be expelled through the high point bleed system.

POGO SUPPRESSION ON SSME

Longitudinal vehicle instability due to closed-loop coupling of the structural, propellant delivery, and engine subsystems (commonly called pogo) was encountered on Thor, Titan II, and Saturn vehicles during development flights. The remedial solutions that provided vehicle stability were feedline accumulators. Spring/piston and contained gas accumulators were used on the Titan II program and overflow gas (helium) accumulators were utilized on the Saturn S-IC and S-II stages. The emphasis from early in the design phase of the Space Shuttle program was to ensure vehicle stability by the inclusion of an engine mounted accumulator in the liquid oxygen system. The primary concern with an engine accumulator mounted upstream of the high pressure oxidizer turbopump (HPOTP) was the overflow of a non-condensible gas (helium) from the accumulator, resulting in pump cavitation and overspeed. The design goal was an accumulator that could be pressurized with oxygen, a condensible gas. To make this goal a reality required (1) a solution to the problem of accumulator ullage collapse caused by heat and mass transfer at the liquid/gas interface and (2) the integration of the pogo system with the engine helium and oxygen pressurization subsystems and the Orbiter propellant feed subsystem.

The initial engine mounted pogo accumulator used a blanket of floating Teflon balls to separate the liquid/gas interface, and a pleated Dutch twill screen in the neck of the accumulator prevented the Teflon balls from entering the HPOTP. The structural integrity and reliability of a pleated Dutch twill screen and the problems with batch testing Teflon material for LO_2 compatibility necessitated a design improvement. Tests with turning vanes to inhibit the turbulent flow in place of the pleated Dutch twill screen and the Teflon balls were marginally successful, i.e., some accumulators collapsed during engine tests due to spraying of LO_2 into the GO_2 ullage. The addition of parallel perforated splash plates above the turning vane, shown in Figure 6, resulted in a semiquiescient liquid/gas surface and successful accumulator performance at all SSME power levels. Accumulator ullage collapse during the SSME start transient was precluded by helium charges prior to engine start and at 2.4 seconds after start command. Ullage collapse subsequent to SSME shutdown command was prevented by a helium post-charge initiated at cutoff command. The post-charge was subsequently extended and used in the in-flight shutdown sequence as described in the LO_2 low level cutoff section of this paper.

The pogo suppression system is shown schematically in Figure 7. The LO_2 normally closed bleed valve and the normally open recirculation isolation valve (RIV) are powered from a common pneumatic source to assure that the RIV is closed during the prestart period when the SSME is being thermally conditioned with LO_2 flow through the bleed system. This also assures that the valves are in the proper positions for the SSME start, i.e., bleed valve closed and RIV open. The Orbiter pogo recirculation valves change the flow path from overboard LO_2 bleed to GO_2 recirculation at SSME start. The Orbiter pogo recirculation valves and SSME bleed valves are launch commit criteria monitored by the LPS. The accumulator precharge pressure is an SSME parameter verified by the SSME controller during start. During engine operation, the accumulator GO_2/LO_2 overflow is routed to the Orbiter through the engine bleedline. This dual use of the bleedline minimized engine weight and interface connections. In the Orbiter, the overflow is routed to the feed manifold near the ET/Orbiter disconnect to maximize the time available for the GO_2 to collapse before entering the low pressure oxidizer turbopump (LPOTP).

The SSME mounted pogo suppressor has been tested extensively to verify functional and dynamic characteristics (reference 6). Pump subsystem tests defined the accumulator diffuser and thermal barrier configuration. Single engine tests defined: (1) the helium precharge and post-cutoff charging times; (2) pressurant flow rate/engine power level relationship; and (3) the baffle configuration to assure an adequate thermal barrier for all operating conditions. The integrated system tests refined the precharge and post-charge times and verified the overall vehicle performance.

LO_2 LOW LEVEL CUTOFF

A low level cutoff (LLCO) system is needed to satisfy the minimum SSME LO_2 net positive suction pressure (NPSP) requirement and to preclude the catastrophic consequences of an LO_2 depletion shutdown. The uniqueness of the LO_2 feedline design, which is over 100 feet long containing 15,000 pounds of LO_2, resulted in the engine cutoff (ECO) sensors being mounted in the Orbiter feedline to reduce the LO_2 residual dispersion at LLCO, and to make the ECO sensors reusable along with the signal conditioner electronics. ECO sensors mounted in the Orbiter feedline had to be reliable and able to quickly respond to the fast-moving liquid interface. A warm wire ECO sensor design was selected because of fast response, simple electronic design, light weight, and similarity to the ET liquid level control sensors.

The generation and propagation of cavitation bubbles within the feed system and their effect on ECO sensor performance had to be determined experimentally because of the complex routing of the LO_2 feedline. A series of full scale LO_2 flow tests were performed. Although the ECO sensors performed normally in the tests, a pressure dropout recorded prior to ECO dry indication showed NPSP requirements would not be satisfied. The presence of a large concentration of bubbles was also photographed at the simulated SSME inlet. The concern relative to the pressure dropout and the vapor volume in conjunction with Orbiter location inability to support the MPTA tests due to 1-G limitations resulted in the ECO sensors being moved to the vertical portion of the ET feedline. The pressure dropout phenomenon was later identified as a facility data problem, and the bubble concentration was determined by single SSME tests to be acceptable. However, the ET location for ECO sensors was retained for the development flights because of increased LO_2 NPSP requirements and to obtain flight performance data.

The ECO system currently incorporates three timers that are entered into the Orbiter General Purpose Computer in order to minimize the LO_2 residual. The timer values correspond to a normal mission shutdown of three engines from minimum power level (MPL), two engines from MPL for an RTLS abort, and two engines from full power level for an abort once-around. The timer values were determined from terminal drain tests with correction for flight acceleration rate and predicted thrust angle. For STS-12 and subsequent flights, the ECO sensors will be mounted in the original Orbiter design location.

The original SSME NPSP requirement was only defined for mainstage operation with the engines accepting the self-generated shutdown NPSP transient. During engine development, tests and analyses indicated a potentially catastrophic overspeeding of the oxidizer turbopumps due to inadequate NPSP during an in-flight shutdown. This condition is a result of the vehicle acceleration transient and the SSME fuel flow transient. The SSME staged combustion cycle (Figure 8) routes the LH_2 propellant flows through the preburners and turbines prior to entering the main combustion chamber. The fuel-rich engine shutdown (to prevent turbine and main injector damage due to high combustion temperatures) is accomplished by a main fuel valve closure profile that allows fuel flow to continue to the preburners for approximately 5.5 seconds. A helium purge initiated 1.8 seconds into the cutoff transient to purge the oxygen trapped downstream of the preburner oxidizer valves results in power being reapplied to the high pressure pumps as this oxygen combusts with the incoming fuel. The rapid decay in available NPSP due to loss of vehicle acceleration could result in pump cavitation as power is reapplied to the HPOTP.

Three potential solutions to this problem were: (1) a preburner fuel valve added to the engine to stop fuel flow to the preburners, precluding combustion when the oxygen is purged out; (2) a purge pressure increase from 750 psi to 2000 psi to allow the oxygen to be purged earlier in the cutoff sequence when NPSP is higher; or (3) the NPSP level could be increased during shutdown by pressurizing the engine inlet. The third option was selected as the most cost effective and timely solution. The SSME LO_2 inlet pressurization would be accomplished by closing the existing prevalves located in the Orbiter feedline earlier during engine shutdown, and pressurizing the engine and feedline with the pogo accumulator helium supply.

The Orbiter pneumatically operated prevalves were not designed to close rapidly. For this concep. to work, the prevalves had to be closed prior to the start of the preburner purge. If the prevalves closed too soon, the prevalves would starve the engine of LO_2 flow, resulting in a more severe pump cavitation and overspeed problem. If the prevalves closed too late, feedline pressurization would be delayed, resulting in possible damage to the HPOTP.

A ground test program was utilized to demonstrate the use of the prevalves and the pogo suppressor pressurization system to prevent turbopump overspeed during cutoff. The tests were devised to simulate worst case flight NPSP and prevalve closing response ranges. A single engine stand was modified by installing a long vertical feedline from the LO_2 tank to the horizontal plane of the engine inlet. The LPOTP was rotated 90 degrees on its discharge flange and a flight Orbiter feedline and prevalve were installed horizontally between facility feedline and the LPOTP inlet. This configuration allowed the LO_2 liquid level to be drained very low in the vertical feedline, simulating the engine NPSP decay in-flight, without starving the engine. The ability of the engine to cut off safely for all prevalve closure tolerances was demonstrated. The first test series qualified the engine cutoff sequence for STS-1 by demonstrating the engine's ability to shut down safely with a transient NPSP of 10 psi at ECO to 2.0 psi at prevalve closure. After the STS-1 flight, a second test series was conducted to qualify the shutdown sequence for low level shutdown and shutdown from rated power level. The engine test stand was reconfigured for this series of tests by adding a separate feedline vent system. This allowed the vertical feedline to be drained much lower before engine cutoff while maintaining the required main-stage NPSP with the tank pressurization system. The valve at the tank bottom and the feedline vent valve were sequenced during the shutdown transient to simulate the in-flight NPSP decay because the feedline volume was much smaller than the tank. The incorporation of prevalve sequencing and an extended pogo accumulator post-charge into the SSME shutdown were effective in preventing HPOTP over-speed for worst case flight conditions. The test also demonstrated that the engine could shut down safely in-flight with a minimum of 80 pounds of LO_2 upstream of the LPOTP inlet.

ORBITER/ENGINE PROPELLANT DUMP

The cryogenic subsystems of the reusable Space Shuttle Orbiter are a fixed part of the orbital and reentry vehicle. The liquid propellants trapped in the SSME's and feedlines at main engine cutoff (MECO) must be dumped to (1) reduce system weight for on-orbit and reentry operations and (2) minimize the hazards associated with venting combustible propellants during post-landing operations. The original concept was to dump both propellants through the SSME's with helium pressurization provided to accelerate the dump. The propellants were to be dumped in series. The LO_2 residual (4600 pounds) was to be dumped first because of its higher temperature and greater mass, followed by the LH_2 residual (300 pounds) dump. The 300-second dump was to be accomplished during OMS-1 burn to provide impulse to the Orbiter, reducing the Orbital Maneuvering System propellant requirement by approximately 130 pounds.

The concept was changed because of a potential HPOTP overspeed problem during LH_2 dump. The potential overspeed results from the SSME staged combustion design where all the LH_2 propellant flows through both preburners and turbines before going into the main combustion chamber. With the engine LO_2 system empty, the hydrogen dump flow would accelerate the unloaded oxygen pump to catastrophic speeds. Since an alternate LH_2 dump path requirement was identified late in the program (May 1980), the solution was to use existing Orbiter components to preclude impacting initial Shuttle launch date. Both the LH_2 fill and drain system and the recirculation/replenish system, shown in Figure 9, were used with only software changes to perform the on-orbit dump function. This modified dump concept allowed a shorter dump sequence by simultaneously dumping the LO_2 and LH_2. The LH_2 dump time is minimized by a short (6-second) dump through the 8-inch inboard and outboard fill and drain valve. This is followed by a 114-second dump that allows the LH_2 residual to be expelled from the 12-inch Orbiter feedlines, and the SSME's through the LH_2 replenish valve and the outboard fill and drain valve. The LH_2 component of impulse was no longer usable for vehicle delta V since the LH_2 dump flow is routed out the side of the Orbiter.

Extensive analytical modeling of LH_2 two-phase flow to vacuum was required to define the dump period and to determine if solid hydrogen formation could inhibit the dump system capability. The performance analysis of the LO_2 and LH_2 dump systems was important because the flight characteristics could not be determined by sea level tests due to 1-G and ambient pressure limitations. Sea level tests of the LO_2 and LH_2 dump systems were performed on a single SSME test stand and on the MPTA to verify the software sequences and component responses.

The LO_2 and LH_2 dump systems performance analysis, using the Lockhart-Martinelli correlation for two-phase flow, agrees with flight data. The correlation between the reconstructed LO_2 dump thrust and flow rate histories and analytical predictions is presented in Figure 10. The LO_2 dump thrust was determined from accelerometer data by converting the measured acceleration rates to total vehicle thrust and subtracting the effects of the Orbital Maneuvering Engine. The LO_2 dump flow rate was verified two ways: (1) the reconstructed dump thrust was divided by the calculated Isp and (2) dump flow was calculated from the helium pressurization flow rate. The LO_2 dump flow rate reconstruction from flight data indicated approximately 1100 pounds remained at the end of dump. The high LO_2 residual, due to loss of helium pressurant which tunnels through the liquid core under the low-G environment, is vented to space as a result of normal leakage through the engine HPOTP seals and during feedline vacuum inerting. Table 1 summarizes the predicted and reconstructed LO_2 dump performances for the development flights.

The LH_2 dump prediction also agrees well with the flight data. Flight data analysis was based on using temperature data to indicate when the liquid interface passed the transducer locations. The predicted and measured dump times for the first SSME to complete dump (engine number 3) are shown below:

	Predicted	Measured
STS-1	71 seconds	60 seconds
STS-2	52 seconds	47 seconds
STS-3	52 seconds	56 seconds
STS-4	52 seconds	52 seconds

Following the STS-1 flight, a 30-second vent of the LH_2 feedline was added to the ET separation sequence (MECO + 10 seconds) in order to protect the system if a relief valve failed. This vent resulted in a reduced post-shutdown pressure rise in the Orbiter feedlines due to heat soakback. The 30-second vent reduced the predicted LH_2 dump time by 19 seconds. The only change to the LO_2 and LH_2 dump sequences, subsequent to the development flights, was to reduce the feedline pressurization period by 18 seconds to save 5 pounds of helium.

The current on-orbit dump sequence presented in Figure 11 cannot be used during an RTLS abort because aerodynamic drag on the Orbiter settles the liquid away from the SSME's and from the LH_2 fill and drain line. A separate LH_2 dump system, from the Orbiter feedline disconnect to the vehicle exterior, dumps the residuals during an RTLS abort. The LH_2 dump, initiated 15 seconds after ET separation with the opening of the RTLS dump valves and manifold repressurization valve, removes approximately 230 pounds of hydrogen. Seventy pounds of the LH_2 in the Orbiter feedlines cannot be dumped due to vehicle attitude. An Orbiter LO_2 RTLS dump system is not required since the LO_2 residual is not hazardous and its effect on the vehicle center of gravity is acceptable. For an RTLS abort, the SSME main oxidizer valves are opened to remove 1280 pounds of the LO_2 residuals from the SSME.

LO_2 MAIN FEEDLINE HELIUM INJECTION

Concurrent with the development program for the Shuttle elements, there was continuous emphasis on performance improvement. One objective was a weight and cost reduction of the ET. The changes implemented on the LWT included deletion of the LO_2 antigeyser line, which resulted in weight and cost reductions of 700 pounds and $113,000 per flight, respectively. The major problems with antigeyser line deletion were geyser prevention, increased SSME LO_2 prestart temperature, and tank liquid level control.

The development activities for the main feedline helium injection system, summarized in Figure 12, were necessary to resolve design incompatibilities with the Shuttle elements. Geyser prevention was accomplished by using main feedline helium injection and facility flow control. Successful geyser prevention depends on the LPS to monitor the feedline conditions and to take corrective action. Feedline temperature redlines are established to assure subcooled propellant for each phase of loading. For a redline exceedence, the LPS initiates a stop flow and changes the facility flow direction to remove the warm propellant from the vehicle. Extensive testing on the MPTA: (1) defined the LPS control requirements and redlines; (2) evaluated procedural and design changes; and (3) demonstrated adequacy of corrective actions. Experience has shown that the characteristics of all components must be well defined for proper operation of this configuration. For example, STS-5 loading was satisfactorily accomplished with Mobile Launch Platform No. 1 (MLP-1). STS-6 loading with the identical LPS sequence using MLP-2 encountered two stop flows and a nondamaging geyser during the slow fill to 2% operation. This condition resulted from a difference in the flow characteristics of two facility replenish control valves for the same position setting. Subsequent to STS-6, additional parameters are being controlled at the facility/Orbiter interface to preclude a reoccurrence of this situation.

Deletion of the antigeyser line resulted in increased LO_2 feed system temperatures that were incompatible with the SSME prestart requirement. An SSME test facility was modified to experimentally evaluate the impact of the higher temperature. The test results allowed the SSME preburner pump discharge temperature requirement to be changed from 178°R to 183.5°R, eliminating the temperature incompatibility.

The use of main feedline helium injection changed the flow profiles in the tank, resulting in the inability of the level control sensors to define the liquid level. A special test series on MPTA defined the reorientation and baffling of the sensors necessary to regain the level control function.

CONCLUSION

The government/contractor team has met the challenge to develop a cryogenic propellant management system that integrated the design features of the Shuttle elements. The flights of the Space Shuttle Columbia and Challenger portray the success of these efforts. Future emphasis will be on the automation of the prelaunch operations, integration/activation of the Western Test Range, and performance improvements that increase vehicle payload. These performance improvements include: (1) removing the LO_2 tank pressure stabilization requirement for liquid levels above 98% (for increased propellant density); (2) reducing the ullage volume (for higher loading levels and shorter drain-back time); and (3) reducing the liquid residuals at engine cutoff. These improvements can result in an additional increase in the Shuttle capability by about 1500 pounds.

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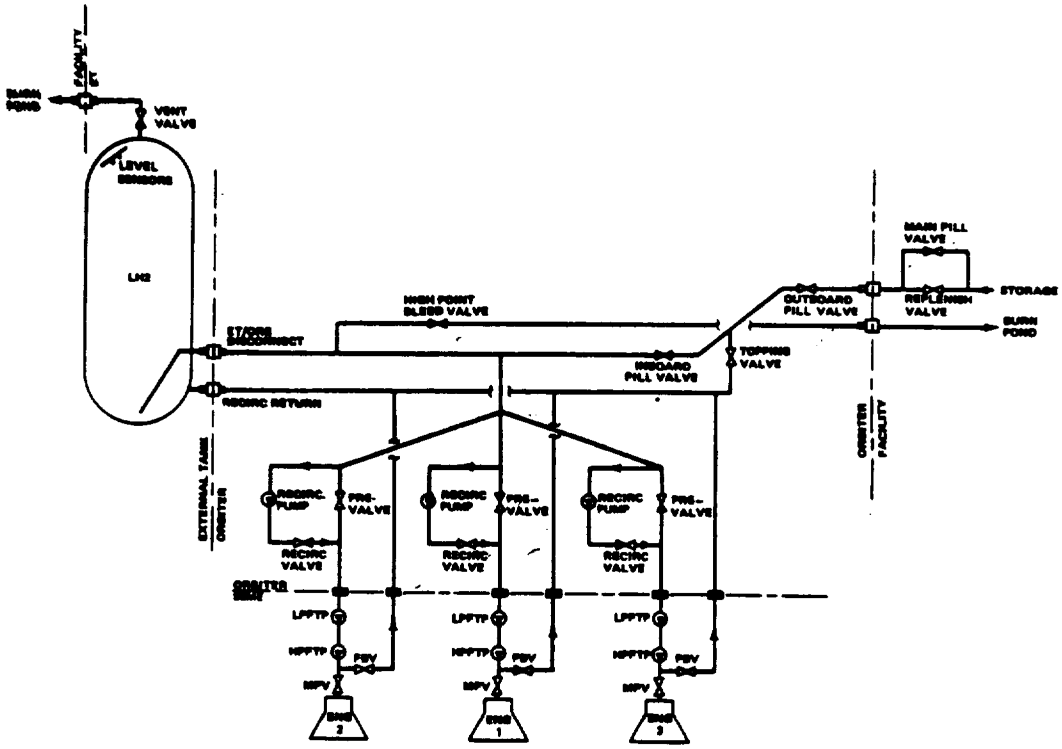

FIGURE 1. LH_2 PROPELLANT DELIVERY SYSTEM

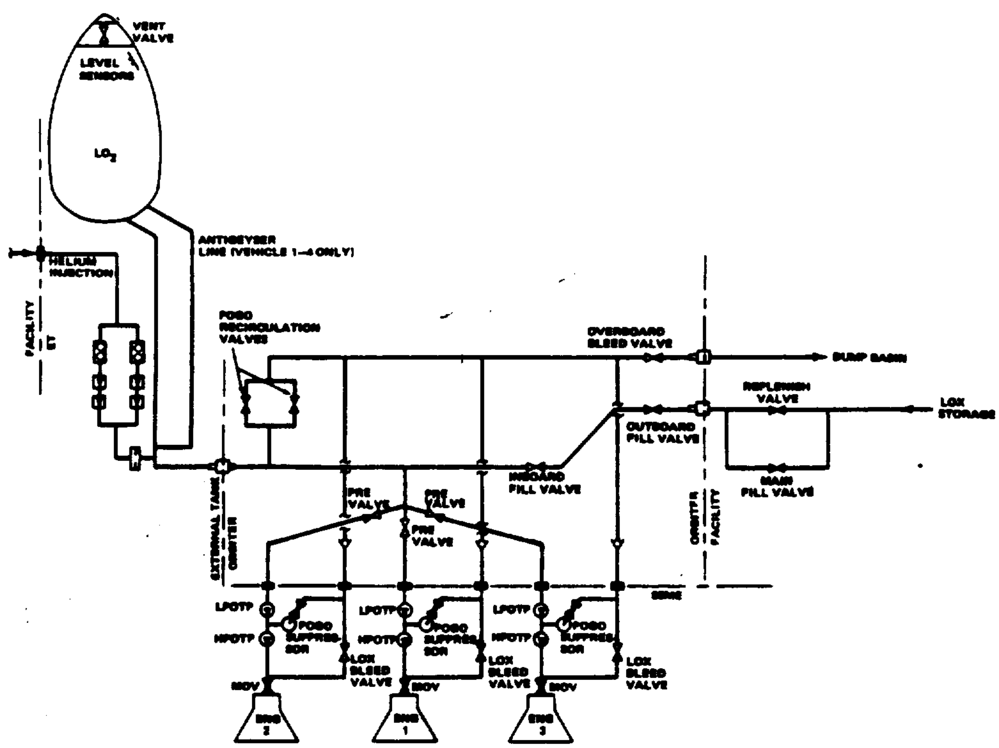

FIGURE 2. LO_2 PROPELLANT DELIVERY SYSTEM

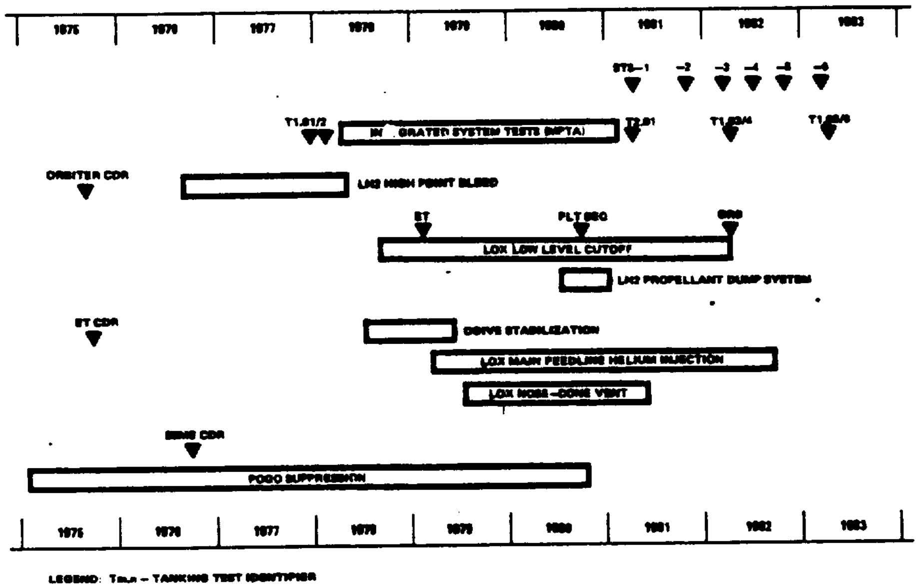

FIGURE 3. CRYOGENIC PROPELLANT MANAGEMENT SYSTEM CHRONOLOGY

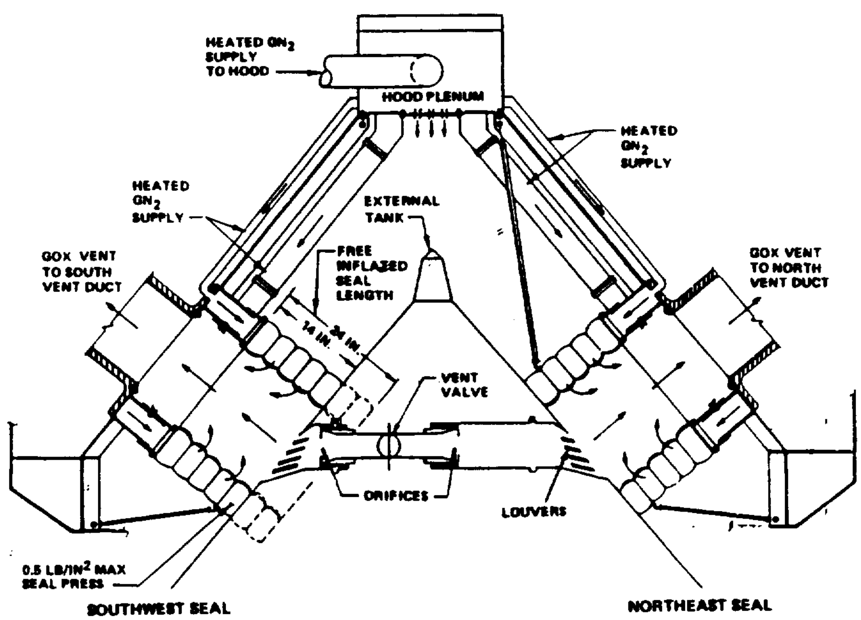

FIGURE 4. VENT HOOD CUTAWAY WITH DOCK SEALS AND EXTERNAL TANK COMPONENTS

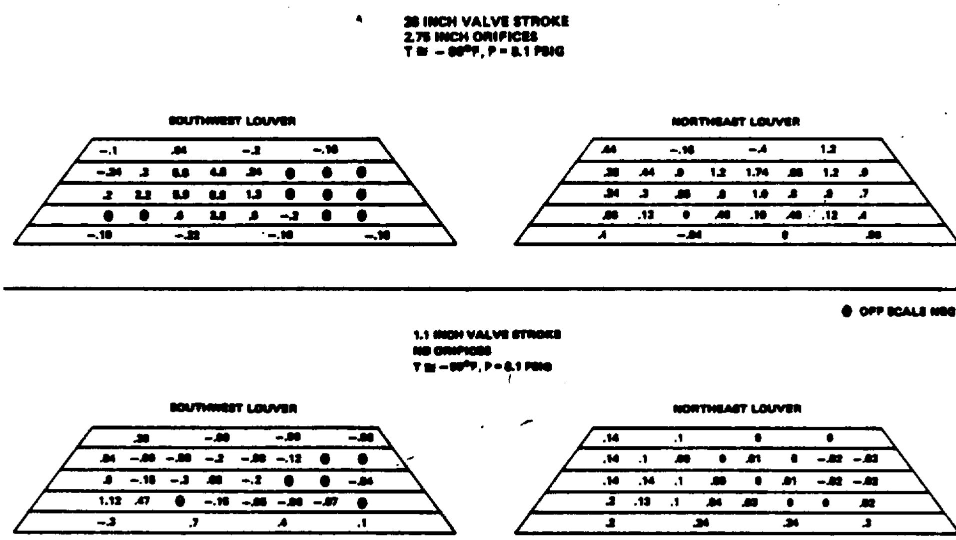

FIGURE 5. GO_2 VENT SYSTEM PRESSURE DISTRIBUTION 3/4" INSIDE LOUVERS

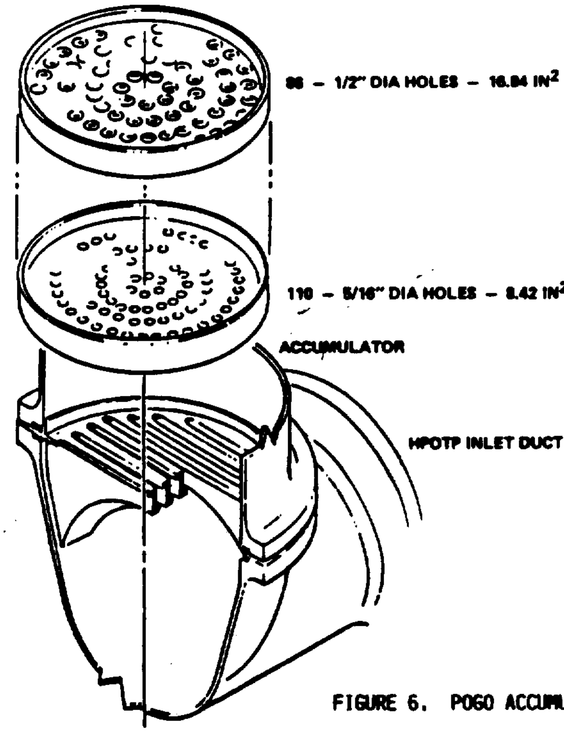

FIGURE 6. POGO ACCUMULATOR SPLASH PLATE

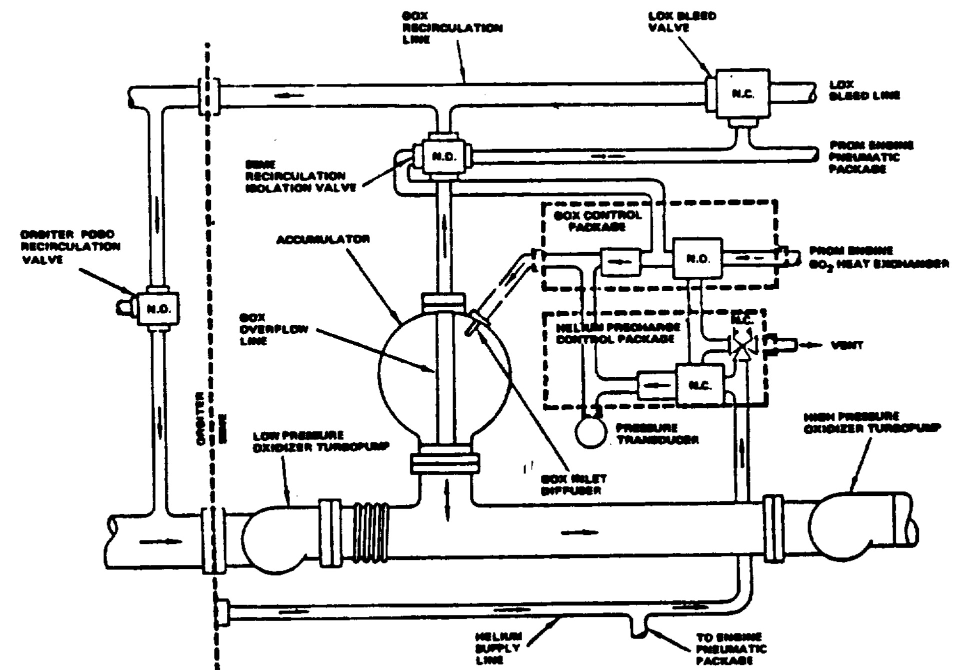

FIGURE 7. POGO SUPPRESSION SYSTEM SCHEMATIC

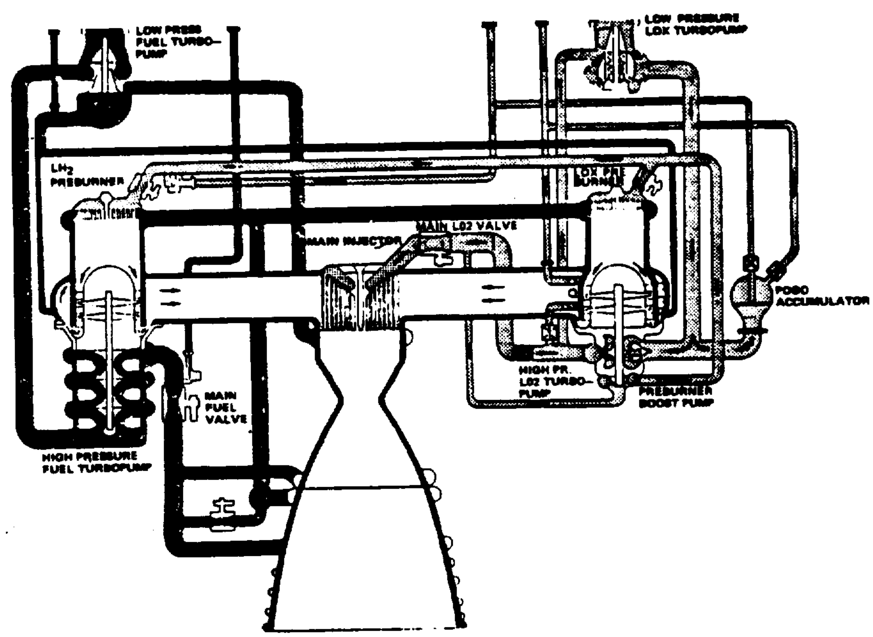

FIGURE 8. SSME PROPELLANT FLOW SCHEMATIC

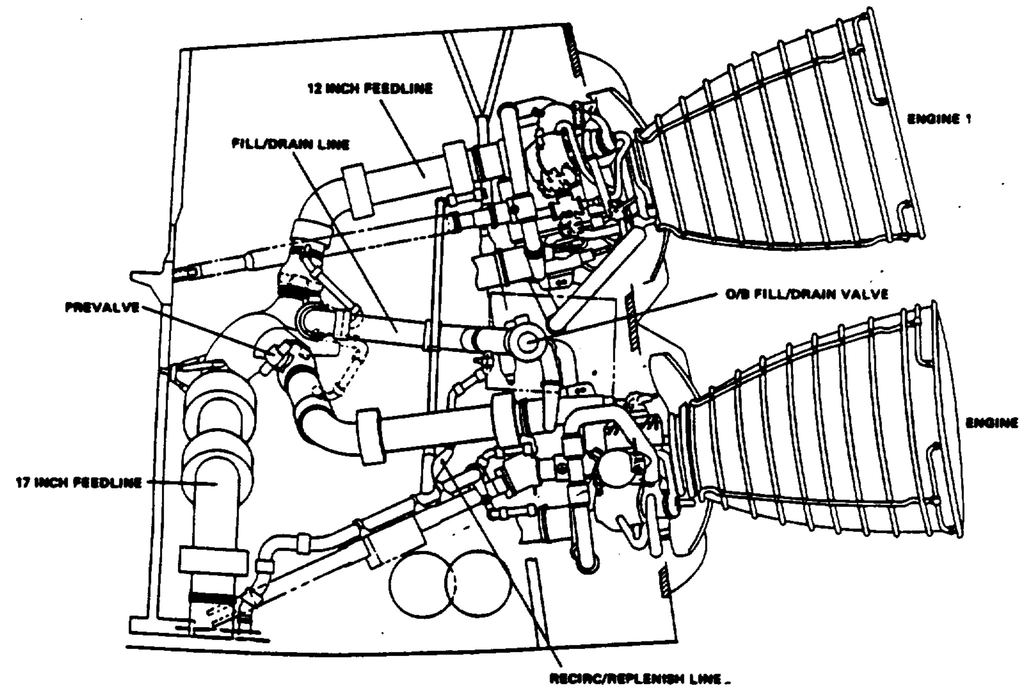

FIGURE 9. LH_2 MPS PROPELLANT DUMP

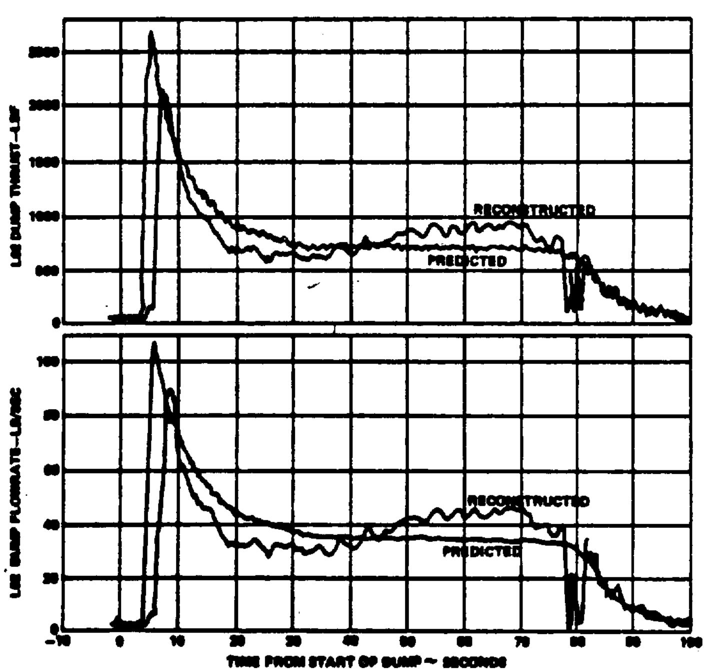

FIGURE 10. LO2 PROPELLANT DUMP MODEL CORRELATION (STS-2)

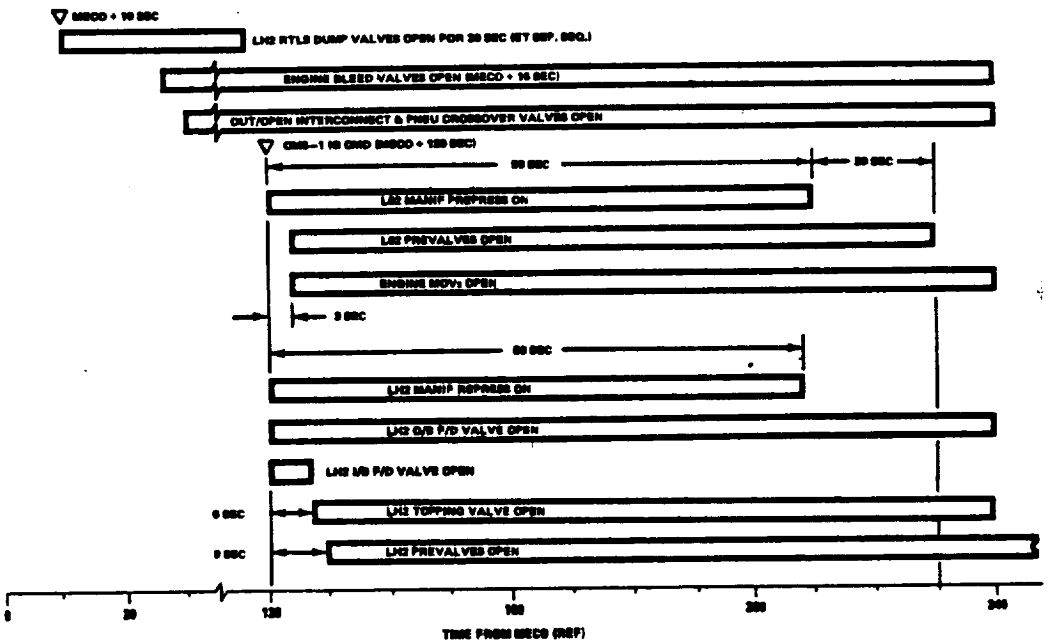

FIGURE 11. MPS DUMP SEQUENCE - NOMINAL (STS-5 AND SUBS)

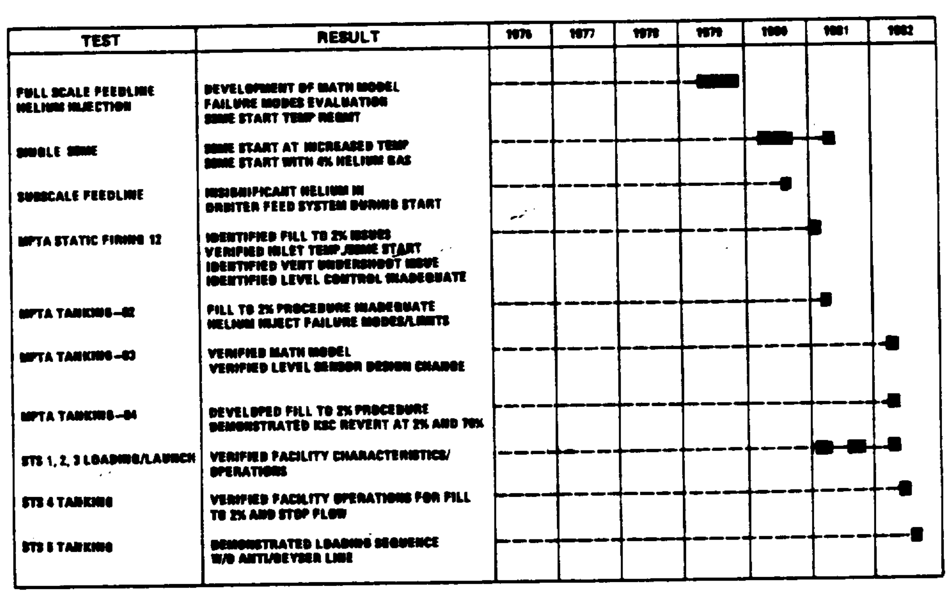

FIGURE 12. DEVELOPMENT/IMPLEMENTATION OF LO_2 MAIN FEEDLINE HELIUM INJECTION

TABLE 1. MPS LO2 PROPELLANT DUMP PERFORMANCE SUMMARY FOR STS-1 THRU STS-4

STS	ANALYSIS BASED ON ACCEL. DATA				ANALYSIS BASED ON FEED PRESS DATA				AVERAGE	
	IMPULSE (LBF-SEC)	DELTA V [3] (FT/SEC)	LO2 DUMPED [4] (LBM)	RESIDUAL (LBM)	IMPULSE (LBF-SEC)	DELTA V [3] (FT/SEC)	LO2 DUMPED [4] (LBM)	RESIDUAL (LBM)	IMPULSE (LBF-SEC)	LO2 DUMPED [4] (LBM)
1	ACCELEROMETER DATA NOT AVAILABLE				72347	10.7	[illegible]	707	72347	[illegible]
2	65291	9.4	3213	1080	70144	10.0	3412	901	67718	3313
3 [1]	44613 [2]	8.2 [2]	2100 [2]	2194 [2]	55330	7.8	2040	1054	49926	2379
4	66700	9.4	3400	805	67920	9.2	3348	906	68316	3378

(1) DUMP THROUGH SSME NO. 1 & NO. 2 ONLY (EARLY APU NO. 3 S/D).
(2) BASED ON QUICK-LOOK ACCELEROMETER DATA
(3) DELTA V GENERATED IS A FUNCTION OF ORBITER MASS AT OMS-1 IGNITION.
(4) 270 LB LO2 LEAKED THROUGH HPOTP SEALS PRIOR TO DUMP INITIATION.