External Tank for the Space Shuttle Main Propulsion System

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A description and development status of the external tank (ET) propulsion system is described. The ET provides 1.5×10^6 lbm of cryogenic propellant for the Orbiter-mounted main engines. The subsystem includes a pressurization, vent and relief, propellant feed, propellant management, antigeyser, and environmental control systems, as well as interfaces with the Orbiter and ground systems. Detail design requirements, performance analysis, and component description of these systems are presented. Specific mention is made of trade studies and design approaches taken to achieve design-to-cost goals.

Introduction

THE Space Shuttle system is comprised of an orbiting vehicle that is a fixed-wing aircraft boosted into Earth orbit by two solid rocket boosters (SRB) combined with thrust from the main propulsion system contained in the Orbiter. The external tank (ET) is the core of this sytem, with the Orbiter and SRB's attached to the ET for launch and ascent. The ET provides 1.5×10^6 lbm of cryogenic propellant for the Orbiter-mounted main engines (SSME). The ET portion of the main propulsion subsystem includes a pressurization, vent and relief, propellant feed, propellant management, antigeyser, and environmental control system, as well as interfaces with the Orbiter and ground systems.

In a typical launch sequence, the Space Shuttle main engines (SSME's) contained in the Orbiter are started, and after thrust is achieved (in approximately 3.7 sec) the SRB's are ignited and liftoff occurs. At approximately 110 sec into flight, the SRB's are separated from the ET and, utilizing a parachute system, are returned safely to the water. The SRB's are recovered and then refurbished for another launch. The ET and Orbiter continue into near orbit, and as orbital velocity is approached main engine cutoff (MECO) occurs. After MECO, the Orbiter and ET are separated, and the ET ocean impacts in a predetermined impact zone.

Present NASA plans are to accommodate over 400 flights of the Space Shuttle in the 1980's, with the first manned Orbiter flight occurring in 1979. The concept of the shuttle is to provide a low-cost method of placing payloads into orbit. For this reason, the Orbiter is designed for reuse, as are the SRB's. However, the ET is expended on each launch; hence the major challenge is to achieve an ET design that satisfies a weight constraint for a low recurring cost.

The challenge of developing the ET proplusion system is to achieve a reliable, high-performance, lightweight system for low cost per flight. This paper will discuss the design approaches utilized in the ET main propulsion system (MPS) in context of these challenges.

MPS Description

The three 470,000-lbf thrust SSME's are fed from the ET through disconnects mounted at the aft end of the ET. Forward in the ET is the liquid-oxygen tank with a capacity of 1,337,558 lbm of LO_2, and aft is the liquid-hydrogen tank containing 224,458 lbm of LH_2. The ET is 28 ft in diameter and 154 ft in length (Fig. 1). On top of the LO_2 tank is the nose cap, which houses the vent and relief valve, ullage transducers, and tumble valve. Between the LO_2 and LH_2 tanks is the intertank. On the skin of the intertank is an umbilical carrier plate, which provides prelaunch ground services for the ET.

MPS Operations

The Shuttle MPS is shown in simplified schematic form in Fig. 2. The LO_2 and LH_2 tanks are pre-pressurized 3 min prior to launch with helium supplied from the facility through the Orbiter into the ET. After SSME start, an autogenous pressurization system maintains the tank pressure.

As depicted on the schematic diagram, LO_2 is passed through a heat exchanger on each engine, into a unit containing parallel fixed orifices with an on-off solenoid valve controlling the flow through one orifice, and then the vaporized GO_2 is manifolded into the single line to the ET LO_2 tank. The solenoid valves in the Orbiter are controlled by ullage pressure transducers mounted in the ET. Autogenous GO_2 is ducted from the Orbiter into the tank through a diffuser mounted on top of the LO_2 tank. Autogenous hydrogen from the engine heat exchanger is ducted from the Orbiter into the top of the LH_2 tank through a tee-type diffuser. Control of LH_2 tank pressure is similar in principle to the LO_2 system.

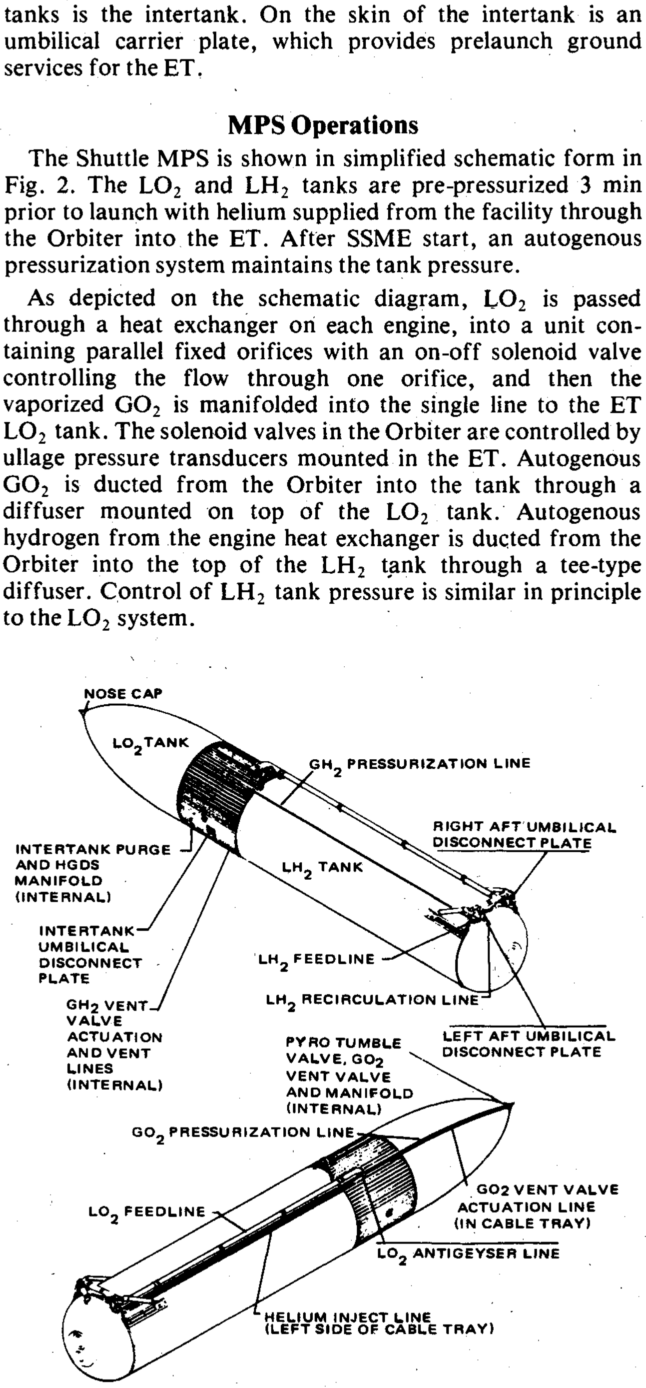

Fig. 1 External tank (ET) configuration.

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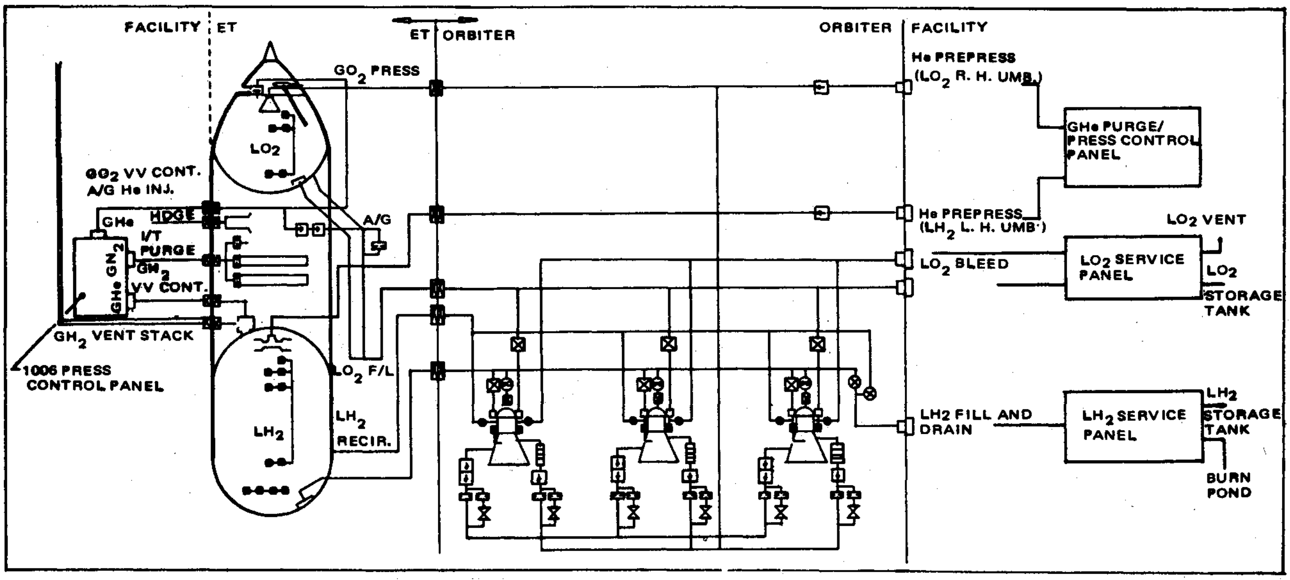

Fig. 2 ET-MPS schematic diagram.

The challenge of low cost, high reliability, and performance is achieved in the MPS pressurization system with the concept shown. High-cost components such as the heat exchanger, control valves, logic control system, and check valves are contained in the reusable Orbiter. The expendable ET contains the passive and less expensive components.

The LO_2 feed system ducts LO_2 from the tank outlet to the three SSME's as shown in the schematic. In order to preclude damaging effects to the ET from LO_2 geysering during loading or off-loading operations, a 4-in. diam line is provided for constant circulation of LO_2 from the aft portion of the LO_2 feedline back into the tank. Helium, ducted from the facility into the intertank umbilical carrier plate through series check valves and an orifice, is injected into the 4-in. antigeyser line. The helium insures that LO_2 will rise constantly in the 4-in. line, thus allowing a constant LO_2 flow downward in the 17-in. feedline to maintain the LO_2 in a subcooled state and prevent formation of a geyser bubble. A simple system was selected to meet the challenge of a low-cost ET. The system utilizes helium ducted from the GO_2 vent and relief valve actuation line and a simple off-the-shelf filter and two check valves.

To satisfy safety concerns associated with the close proximity of the LH_2 and LO_2 in the intertank area, a purge system and a hazardous gas detection system were developed. The GN_2 purge system consists of a GN_2 manifold around the intertank internal periphery to carry away GO_2 or GH_2 that may be leaking into the intertank from tank joints or other sources and to prevent moisture accumulation inside the intertank. The GN_2 is facility-supplied through a disconnect in the intertank umbilical carrier plate. Gas concentrations in the intertank are ducted to a facility-mounted hazardous gas detector. If hazardous levels are indicated, corrective actions would take place to preclude the possibility of fire or other hazardous conditions in the intertank.

The schematic-depicted GH_2 vent system consists of the LH_2 vent and relief valve, mounted on the dome of the tank, the GH_2 vent line connecting the valve to a disconnect on the carrier plate, and the helium actuation line to operate the vent and relief valve. Vented GH_2 is ducted through the disconnect to a facility line for safe disposal at the burn pond.

The LH_2 feed system depicted at the aft of the LH_2 tank feeds the three SSME's through the Orbiter ET disconnect into the manifold assembly. This feed system utilizes the "siphon" approach, which is unique for a booster vehicle. This system, described later in detail, utilizes a unique internal duct design concept to meet the low-cost design goal without sacrifice to reliability or performance. Rather than the conventional external feed system ducting, requiring extensive structural support and thermal protection, the simplified inside tank ducting approach minimized this extra cost.

Propellants are loaded or off-loaded from the facility loading system through the Orbiter plumbing as depicted in the schematic. To obtain proper pump inlet temperatures prior to engine start, an LO_2 bleed system returns LO_2 to the facility from the Orbiter, and an LH_2 recirculation system bleeds LH_2 back to the tank. Level sensors depicted in the schematic in each tank are used for propellant loading control and engine shutdown control.

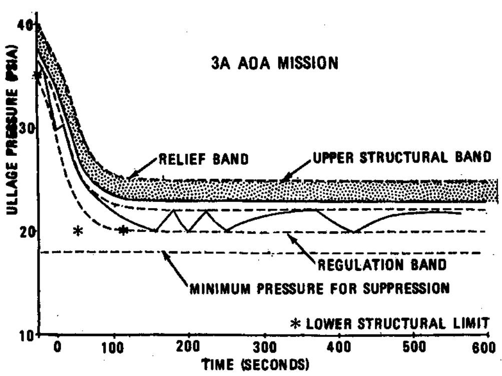

Fig. 3 LO_2 tank ullage pressure history.

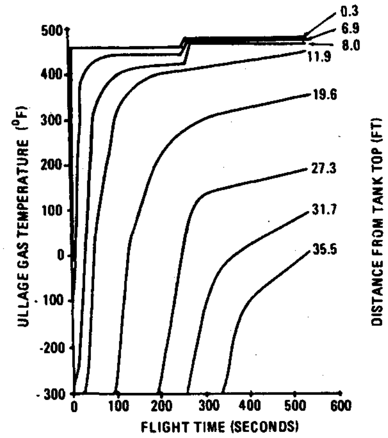

Fig. 4 LO_2 tank ullage temperature.

ET Performance

Usable propellants are the primary MPS performance objective of the ET. Predictions of usable propellants are made for the various missions and trajectories to be flown by the Shuttle, but the one significant mission that influences usables is mission 3A-A0A. For this mission, which is an abort-once-around, all usable propellant is consumed. For most missions, all available propellant is not consumed, leaving margin for an abort case.

Low cost per flight is achieved by proper propellant management control. Again, the expensive portion of the system is within the reusable Orbiter with minimum components in the ET. The basic concept is a constant mixture ratio control system comprised of flowmeters and ratio control. A mixture ratio (MR) of 6:1 oxidizer/fuel is achieved by throttle control of the engine valves based on flowmeter indications. Propellant is loaded in each ET to 100% utilizing a simple level sensor control system. With constant MR control, all predicted usable propellants may be consumed.

A flight performance reserve of 3346 lbm is established to account for error sources that affect usable propellants. There are also variations from launch to launch which affect propellant loading accuracy, such as variation in tank volume, variation in propellant density at liftoff, wave action of the liquid surface due to sloshing, and vertical alignment of the ET. Considering a statistical average of all error sources affecting loading accuracy, the present analysis indicates an expected accuracy of $\pm 0.44\%$ for LO_2 and $\pm 0.30\%$ for LH_2. Errors in mixture ratio control are averaged with loading accuracy for determination of the final flight performance requirements.

Minimal unusable propellants were achieved at a low cost per flight by employing special design and analysis considerations for the tank outlet design. To examine methods to minimize or eliminate propellant dropout during terminal drainage, scale tank testing was conducted utilizing water. Several design approaches were tested, and the optimum concept, a contoured outlet, identical in geometry for LO_2 tank and LH_2 siphon, was selected. The scale tests indicated no dropout for the LO_2 side at all of the expected burnout angles.

On the LH_2 side, the dropout was minimized by proper location of the siphon with respect to the tank bottom, as well as with selection of the contoured siphon. The test results were scaled to flight conditions by utilizing a dimensionless relationship Froude number ($FN = V^2/gd$). Dropout height, in inches above the siphon lip, was determined at various Froude numbers and for two different canted positions of the outlet. The test data matched the analytical predictions reasonably well. Based on the test results, 424 lbm of trapped LH_2 is unusable because of dropout.

The scale tests not only provided an outlet design that resulted in minimum residuals, but the design also resulted in low-cost fabrication because the contours for the LO_2 outlet and LH_2 siphon are identical. Quantity procurement efficiencies also are realized, resulting in cost per flight reduction of approximately $2,000 for these components.

The requirement to minimize retained autogenous vapor (unusable propellant) was especially unique and interesting. Many conflicting design requirements were balanced to achieve the optimum conditions considering performance, weight, and cost. Tank pressure control and profiles influence many design aspects of the ET as well as the MPS. Adequate ullage pressure not only is required to satisfy net pump suction pressure (NPSP) requirements but also provide structural stability for the LO_2 and LH_2 tank at liftoff, when the asymmetrical thrust vectors from the SSME's and SRB's produce a compression shear load in each tank. In addition, upper pressure limits are established to minimize the tank structure requirements for pressure-carrying capability; this factor strongly influences ET weight. High ullage gas temperatures are desired to minimize gas residuals; however, the high temperature degrades the structural properties of the tank shell as well as the operating components, such as the vent and relief valves, level sensors, and instrumentation.

The balance between all factors for the LO_2 tanks was achieved by selecting an ullage pressure profile as depicted in Fig. 3 and by establishing an ullage temperature profile as shown in Fig. 4. A gage relief valve, with a 3-psi control bandwidth, maintains ullage pressure below the upper structural limit during all portions of the flight, even if there is a failure in the pressurization system. This bandwidth is consistent with the state-of-the-art for low-cost relief valves. The cost trade between tank structural weight, when compared to narrower bandwidth, favored the 3-psi bandwidth. Cost analysis indicates that the net cost increase to narrow the bandwidth 1 psi is approximately $2,800/vehicle. The initial pre-pressure is 37 psia, which provides sufficient NPSP for engine start and adequate tank shell pressure at liftoff. Ullage pressure is controlled by use of gage transducers to a 20- to 22-psig bandwidth. After approximately 120 sec of flight time, the transducers and relief valve reference to vacuum; hence a constant absolute pressure control band is maintained. A minimum of 20 psia ullage pressure is required to suppress bulk boiling of the LO_2 liquid surface. (LO_2 vapor pressure is 18 psia.) Bulk boiling would increase vapor residuals significantly.

The ullage temperature profile (Fig. 4) shows that forward ullage gas temperatures reach 485°F after 300 sec flight time. A thermal analysis of the tank skin, considering tank shell thermal capacitance and the gas temperatures, indicated a maximum skin temperature of 300°F, which is the upper limit to maintain tank shell integrity with adequate margin. The diffuser designed for the LO_2 tank is expected to yield slightly lower temperatures at the forward end of the tank than those shown in Fig. 4.

A proper ullage pressure and tank structural integrity balance considering all factors was also achieved in the LH_2 tank. The ullage pressure profile is shown in Fig. 5. A gage relief valve with a 3-psi control bandwidth maintains pressure below the upper structural limit even if there is a pressurization system malfunction. Ullage prssure is maintained in a 2-psi bandwidth by absolute pressure transducers. Initial pre-pressurization to 44 psia is required for NPSP for engine start and for adequate pressure for tank structural stability at liftoff. Since the control transducers are absolute, ullage pressure is essentially in a blowdown mode until approximately 30 sec into flight, at which time ullage pressure control authority is established. Pump NPSP is a driving requirement for ullage pressure control, as can be seen in Fig. 5 by the lower curve (feed system interface in Fig. 5 by the lower curve (feed system interface requirements). A

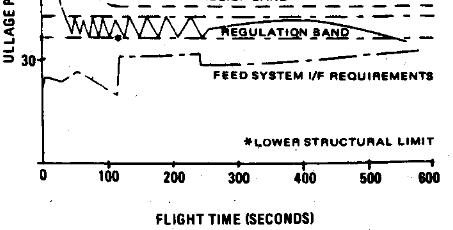

Fig. 5 LH_2 tank ullage pressure history.

regulation band of 32 to 34 psia provides adequate pressure for NPSP and satisfies all other requirements.

LH_2 ullage temperature profiles are shown in Fig. 6. Here, the upper temperature of 150°F is reached at MECO. The forward dome of the LH_2 tank is designed with adequate margin for this temperature. As with the LO_2 tank, a balance to achieve minimum LH_2 tank vapor residuals was reached considering the effects of pressure and temperature on engine operation and tank structural requirements. The optimum compromise to achieve these requirements as well as the design-to-cost goal resulted in the present design approach to the ET pressurization system.

LO_2 Feed System

The ET LO_2 feed system design presented special challenges to satisfy performance requirements, low weight, and low cost. In the contoured outlet design (previously discussed), an 800 μ screen is installed. The purpose of the screen is to preclude contamination within the tanks from clogging small passages within the SSME. In addition, the screen minimizes the need for stringent cleanliness control of the tanks, reducing the ET assembly costs. The screen consists of four separate sections, which, when bolted to the outlet and baffles, makes a 60-in.-diam screen, providing a large contaimination capacity margin. Integral to the outlet are vortex baffles, which preclude early propellant dropout because of sloshing or vortex effects. A square geyser splash plate is mounted on top of the baffles. This plate serves a redundant function, that is, if a geyser should develop in the LO_2 feed system, the plate will disperse the geyser and minimize the possibility of an ullage pressure collapse with resultant tank implosion.

The LO_2 feedline (Fig. 7) was designed specifically for low cost per flight. Commonality and similarity of design was the basic approach taken. Layout studies indicated the need for flexible joints to account for tank cryogenic shrinkage and motion changes due to pressurization effects. The 84-ft line consists of eight assemblies: three assemblies with five flexible joints. There are three spin-formed 2219 aluminum sections of straight line. The design of each straight line is identical except for length, resulting in reduced development costs and production efficiencies. At the aft end of the line is a 90° elbow, which was designed as a casting resulting in a significant unit cost advantage as compared to the cost of a machined forging. Mated to the elbow casting is the Orbiter to ET disconnect. This component is a visor-type valve that is actuated by Orbiter-contained pneumatics. After MECO, both valves are closed (one in the Orbiter and one in the ET), and then the disconnects are separated prior to Orbiter separation from the ET. The disconnects are provided with a mechanical backup system so that, if the primary pneumatics fail to close the valves, the separation mechanism will close both valves. Insulation is applied to the LO_2 feedline straight sections to minimize ice formation. A series of swing arm assemblies supports the line, with the key structural support at the aft elbow. Commonality of the slide mounts and flexible joints and the elbow casting design resulted in an estimated lower cost per vehicle of over $22,000.

Another requirement of the ET design is to preclude damaging geyser conditions, a phenomenon that has been experienced on many cryogenic launch vehicles. Because the ET LO_2 line is very long, the effects from a geyser are extremely damaging. In this case, rather than design for low cost and high risk, a design was selected to provide strong assurance that a geyser will not occur. This approach is very cost-effective when the damaging effects of a geyser are considered. The antigeyser system consists of a 4-in. line connecting the aft elbow with the tank outlet. After LO_2 is filled above the screens and during a hold or low-flow-rate mode, circulation always exists down the main feedline and up the antigeyser line because of the lighter density of the warmer LO_2 in the smaller diameter antigeyser line. In order to provide redundancy and air circulation, a helium-inject system into the antigeyser line is used, as shown in Fig. 7. Helium is provided through a filter, series check valves, and an orifice for flow control. The helium is provided any time the vent and relief valve is opened, but, as stated earlier, the helium system is redundant, since circulation exists without the helium.

The design of the antigeyser line is similar to the LO_2 main feedline in terms of commonality of parts. Six flexible joints were required, and each is identical, resulting in production efficiencies and a cost reduction of approximately $5,000/vehicle.

Another design approach toward meeting the low-cost design goal involves the LO_2 feedline and antigeyser line. A trade study to determine optimum line lengths was performed considering handling costs, transportation costs, fabrication costs, and weight consideration. It was determined that 20-ft line lengths were optimum; in this case, a weight penalty of 50 lbm was taken in order to reduce the overall program cost of the lines by approximately $100,000.

Performance requirements for the LO_2 feed system did not present any difficulties, primarily because the LO_2 tank is forward, resulting in reasonably high head conditions throughout a large portion of the flight region. A pressure temperature requirement is established at the Orbiter-ET interface to satisfy pump NPSP requirements, as shown in Fig. 8. The profile shows adequate pressure from engine start, through all phases of flight, until MECO. At MECO, as the LO_2 drains in the LO_2 feedline, the 20-psia limit is approached; however, since minimum ullage pressure is 20 psia, all of the LO_2 in the feedline may be consumed.

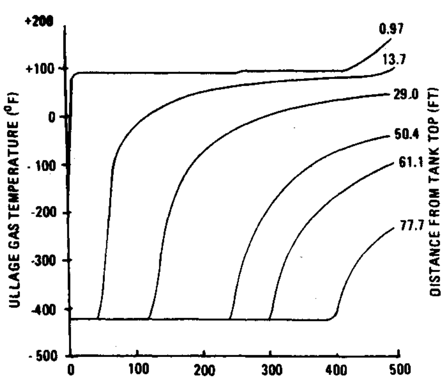

Fig. 6 LH_2 tank ullage temperature.

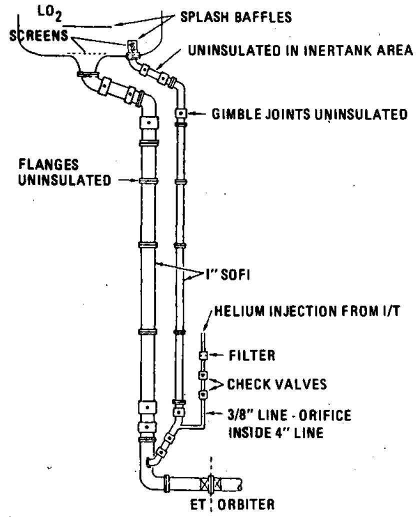

Fig. 7 LO_2 feed system.

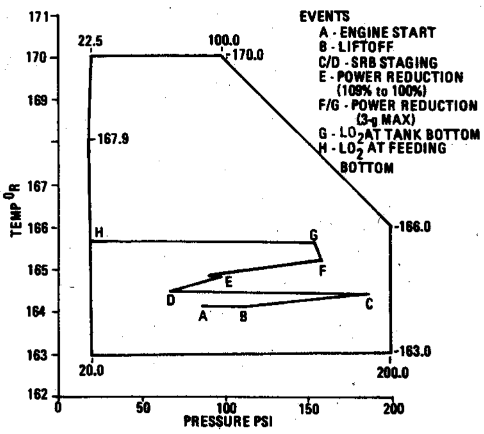

Fig. 8 LO_2 tank/ET to Orbiter interface pressures.

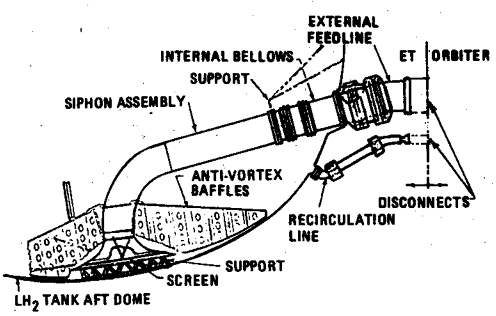

Fig. 9 LH_2 feed system.

LH_2 Feed System

Many significant design problems were encountered in designing the LH_2 feed system because of 1) aft location of the tank resulting in lower head, 2) large volume of the tank, and 3) lower temperature of the propellant. The internal siphon concept (previously mentioned) used in the LH_2 feed system resulted in a significant cost-effective design when compared to a conventional external feed system. The siphon concept is shown in Fig. 9. The contoured siphon (same contour as the LO_2 tank outlet) narrows to a 17-in.-diam line, which mates with an articulated bellows assembly inside the tank; the bellows assembly provides for relative motions during cryogenic shrinkage of the tank. The siphon is attached to the aft dome of the LH_2 tank.

Another LH_2 feed system design requirement involved the 44-in. diam 400 μ screen mounted in the siphon. It was important to minimize pressure losses across the screen, because a loss greater than 0.5 psid would result in the need to increase ullage pressure to compensate for the loss. This would result in a significantly heavier and more costly tank. It was determined that a screen was required to minimize clogging the small passages in the SSME, especially since the engine must be reused for multiple launches. Since the LH_2 tank has a large volume with resultant large amounts of potential contaminate, the screen was designed to achieve adequate contaminant capacity. Careful design of the screen coupled with extensive tests and analysis was required to minimize pressure loss across the screen. In this case, investment in development effort resulted in obtaining an efficient low cost per flight design. The screen design approach resulted in a cost and weight savings of \$5,600 and 350 lbm, respectively.

Low-temperature (-423°F) liquid hydrogen has been the classic driver for using vacuum-jacketed lines for all external ducting to reduce heat transfer and to minimize liquid air formation. The challenge on the ET program was to establish a line design that was not as expensive as Saturn experience indicated. To this end, a trade study was performed to evaluate several design concepts and the Saturn program cost-factor experience. It was determined that a permanently welded argon back-filled jacket was the most cost-effective approach, especially considering the high maintenance costs experienced with the functional components on a vacuum-jacketed line. This decision resulted in a total program savings of \$350K. The back-filled line concept is utilized for the LH_2 external feedline and the recirculation line (Fig. 9). Further cost reduction was realized on the recirculation line by jacketing only the bellows assembly and applying a polyurethane form on the straight section. This foam is the same foam that is used on the ET tank shell and on the LO_2 feedline. The Orbiter to ET disconnect on the LH_2 feedline is identical to the disconnect used on the LO_2 feed system; hence another advantage is gained to reduce ET unit cost. The disconnect on the recirculation line functions in principle as the LO_2 and LH_2 feed system disconnects.

Satisfying the performance requirements for the LH_2 feed system offered the most difficulty in meeting the constraint of low cost per flight. The Orbiter to ET interface pressure/temperature relationship required to satisfy pump NPSP is shown in Fig. 10, as well as the predicted operation conditions for maximum and minimum ullage pressure. As can be seen, after power reductions (points F-G and H), minimum pressures approach the lower limit; hence there is little margin for design changes. In order to minimize feed system pressure losses; the siphon assembly was located carefully in the tank with respect to distance from the aft dome. This separation distance was determined based on scale outflow tests that achieved the best balance between pressure drop and unusable LH_2. As previously mentioned, the LH_2 feed system was designed for minimum pressure drop. Design of the flexible lines was conducted with the specific intent to minimize pressure loss. Finally, the entire LH_2 tank is insulated to achieve a low propellant temperature, with particular emphasis on minimum heat transfer through the aft dome. Scale testing was accomplished to confirm the pressure-loss analysis. Scale testing also was conducted to verify the insulated tank heat-transfer analysis.

LO_2 and LH_2 Vent System

Commonality where possible, simplification of design, and reliance on experiences from previous programs were all keynotes in meeting the low-cost-pre-flight goals for the ET pressurization and vent systems. Design of the vent and relief valves was driven by commonality and previous experience. The valves for the LO_2 and LH_2 are essentially the same design except for relief settings and for flange differences to preclude improper installation. This commonality reduces design and development costs, as well as realizing production efficiencies for many of the valve piece parts. This approach realized a cost savings of \$400,000 for design and development and \$3,000/flight. In addition, the valve is based on a design satisfactorily used on many similar space programs, such as Titan and Saturn.

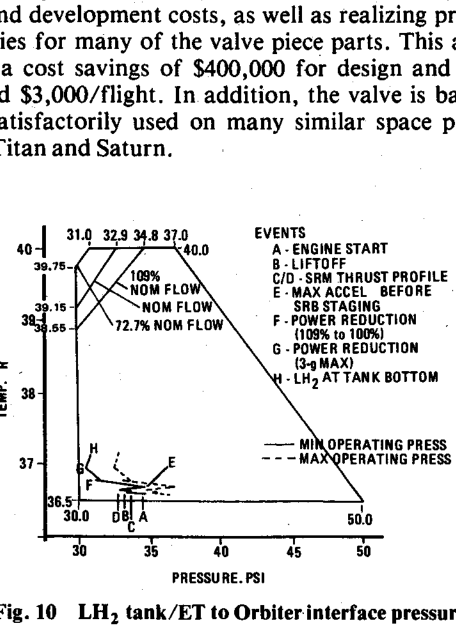

Fig. 10 LH_2 tank/ET to Orbiter interface pressure.

Simplicity was also important in the fundamental system-relief concept of the valves. The vent and relief functions are combined in one valve. However, a unique difference exists for vent actuation in the ET compared to previous launch vehicles. The command function for actuating the vent feature is located in the facility rather than on the ET (Fig. 2). To actuate the vent valve open, 750 psi is applied to the actuation system to open the valve, and for valve closure the pressure is removed. This concept eliminated the need for costly electrical disconnects, wiring, and solenoid valves on each ET, estimated at $10,000/flight. An acceptable penalty was incurred in the extra time required in the launch count sequence to accommodate the valve slow actuation time. This condition has been accommodated for in the launch count sequence.

A major step taken toward the low-cost design goal involved the interplay between the LO_2 vent system nose cap design and satisfying an ET tumbling system. A conical-shaped nose cap was determined to be the best concept to minimize aeroheating and insulation ablation during ascent. To accommodate the conical concept, a duct and louver approach to allow the discharge of vented gases was developed (Fig. 11). This approach significantly reduced the amount of insulation required on the ET to withstand the aeroheating effect, thereby realizing $91,000/flight cost avoidance.

The need for a tumble system was determined based on aerodynamic studies of ET trajectory after Orbiter separation. It was determined that there was a possibility that the ET could "skip" upon re-entry into the atmosphere rather than break up and impact into the predicted impact zone. Tumbling the tank will preclude this possibility. Rather than utilizing retrorockets, the stored energy of pressurized GO_2 is used by actuating the tumble valve at Orbiter-ET separation. The "lance" ducts GO_2 through the tumble valve. The need for a lance is based on the requirement for controlled thrust, since, shortly after tumbling is initiated, unused LO_2 will migrate toward the nose of the LO_2 tank; the lance insures that GO_2 rather than LO_2 is vented through the tumble valve and tumble nozzle. This tumble system is simple and provides a low-cost system, approximately $9.5M less than a solid-rocket retro system.

Another interesting design challenge involved the pressurization and relief system. During initial design of the ET, redundant relief valves were considered necessary for reliability. Detailed reliability studies and assessments of risk were made to eliminate the redundance requirement, leaving each tank with one vent and relief valve. It was determined that there was acceptable risk in eliminating one of the two relief valves, since more than two failures would be required to create a hazard condition. This decision reduced the total ET program cost approximately $2.1M.

Previous program and development experience was utilized to design the ET pressurant diffusers, minimizing the design and development costs for these diffusers; the diffuser was selected based on Saturn IV diffuser design concepts. This concept, as stated earlier, was selected to promote controlled mixing of the hot incoming autogenous gas, minimize forward ullage temperatures, and minimize pressurant residuals. The compromising risk, which was determined to be acceptable based on previous analysis and testing, was the gas impingement on the liquid surface during the start transient. The challenge here was to achieve a design that would satisfy conflicting requirements. For the LH_2 tank, the S-IVB diffuser concept was selected. S-IVB diffuser design and test experience was available, minimizing the effort required for the ET.

Commonality was established as the method to meet the low-cost-per-flight goal in design of the pressurant lines. These lines interconnect the Orbiter gas supply with the ET diffusers. Common line diameter, common line supports, and identical flexible joints were selected for both the GO_2 and GH_2 lines. Reduced fabrication costs and procurement of parts in larger quantities reduced the cost per flight by $2,700.

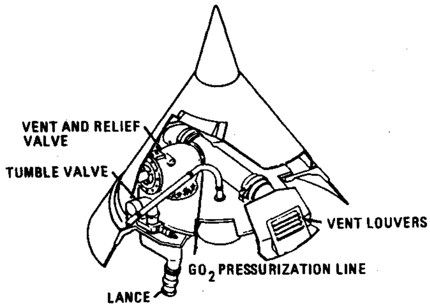

Fig. 11 LO_2 tank nose cap.

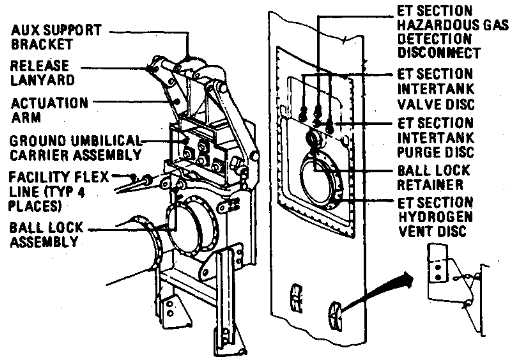

Fig. 12 ET to ground umbilical carrier plate.

ET-Facility Umbilical Carrier Plate

Previous program disconnect experience and Skylab experience bore heavily on the design of the intertank umbilical carrier plate. The design to low-cost goal was satisfied by relying on this past successful experience. The carrier plate is a pyro- actuated system, similar in principle to the one used on Skylab. The carrier plate is attached to the ET with a frangible bolt so that, if the actuation mechanism should fail, the bolt will fail and allow carrier plate separation without damage to the ET. The carrier plate assembly is shown in Fig. 12. Integral to the ground and ET side of the carrier plates are four 3/8-in. disconnects and one 7-in. diam disconnect. The small disconnects are identical to each other and service the vent valve actuation system, helium inject system, intertank purge, and hazardous-gas-detection system. The commonality of these disconnects allows procurement efficiencies. The 7-in. disconnect is the hydrogen vent disconnect, the design being based on a disconnect utilized on the Saturn program; again this resulted in reduced design and development costs without sacrificing confidence. The overall approach is estimated to save $40,000 in design and development costs.

Other Design-to-Cost Features

Another design step toward low cost and low weight involves all propulsion ducting on the ET. In the feed systems, pressurization systems, and vent systems, there are 34 flexible assemblies that are bellows devices. A design parameter for these assemblies is flow-induced vibration, an environment that has caused catastrophic failures on previous programs. The challenge for the ET was to utilize past experience and research[1] to the greatest extent possible, with the specific aim of attempting to eliminate the requirement for flow liners in the bellows assemblies. Elimination of the liners reduced fabrication costs, cleaning costs, and weight, with the risk of bellows failure due to flow-induced vibration. However, this risk is minimized by proper selection of materials and bellows design, as well as gas velocity limiting orifices in the pressurization system. It was possible to eliminate flow liners in all but four of the bellows assemblies, resulting in a total ET program savings of approximately $1.6M and a weight savings per ET of approximately 62 lb when compared to the more classical approach of using liners.

Additionally, a low-cost design approach is used for the seals of the mechanical joints. The classic approach was to use the highly sophisticated Naflex seal for the 4- to 17-in. diam lines. However, an acceptable leakage rate was determined, and a less expensive RACO seal, relieving tolerance on flanges, was proved to be adequate. The RACO seal approach is $10M less expensive.

Conclusion

Many steps have been taken to develop a low-cost ET portion of the main propulsion system and still achieve performance, low weight, and reliability. Emphasis was placed on locating expensive components in the reusable Orbiter, for example, ullage pressure controls and propellant-management controls. Commonality of design, use of proven design approaches from previous programs, careful selection of ullage pressure parameters, a balance with structural design requirements, and location of sophisticated controls on the facility rather than in the ET all have led to low cost, high confidence in reliability, and optimum ET weight. Judicial use of testing led to cost-effective screen designs and insulation designs. Reliability for safe loading has been achieved with splash-plate and antigeyser system. Significant weight saving was achieved with the "siphon" approach, conical nose cap, and tumble concept.

The ET propulsion subsystem has been designed to achieve its performance goals at the lowest practical cost. The design is complete, critical design review (CDR) is complete, and all components are now in initial fabrication and development testing. The first complete ET, main propulsion test article (MPTA), is being fabricated. A full-scale hot-firing test program of the Shuttle MPS will be conducted in 1977 on the MPTA to verify performance, operation, and functional requirements of the Shuttle main propulsion system. Successful completion of the MPTA test will verify that the ET MPS design truly has satisfied the challenge of low cost, without undue compromise to performance, operation, or weight.

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References

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