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DEVELOPMENT OF THE TITAN STAGE III

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Summary

This paper presents the design and the field test program associated with the development of the propulsion module for the third stage of the Titan III Standard Space Launch Vehicle. The primary objective of this vehicle development is to provide a launch vehicle with the capability of performing a wide variety of space missions. The vehicle configuration(s) evolved to satisfy this objective consists of a modified Titan II equipped with a third stage designated as Configuration "A" or Core, and the addition of two segmented solid propellant rocket motors, one on either side of the Core designated as Configuration "C".

The third stage of this vehicle is called the transstage. This stage is composed of two modules: the control module, which contains the guidance, flight controls, and attitude control subsystems; and the propulsion module which is identified as stage III.

Stage III consists of a pressure-fed, re-startable, space propulsion system which utilizes storable hypergolic propellants. Propellants are contained in two titanium tanks which are secured to the vehicle airframe in a parallel arrangement. The engine assembly consists of dual thrust units which are individually gimbaled to provide vehicle attitude control during stage operation. Each thrust unit consists of an ablative cooled combustion chamber with a radiation cooled nozzle exit section. The pressurant is stored gaseous helium and propellant tank pressure is maintained by use of pressure switch controlled solenoid valves. The stage is designed to have a useful life in space of at least 6.5 hours.

A review of the stage III design requirements, the mechanization used to fulfill these requirements, and the development test program used to verify the mechanization is presented. Stage III subsystem and design confirmation testing includes mockup and full scale (battleship) tests of the pressurization subsystem, model and full scale (battleship) test of the propellant subsystem, and captive tests using battleship tankage of the complete propulsion system. Stage design confirmation tests include a series of altitude simulation firings at the Arnold Engineering Development Center Rocket Test Facility and System Compatibility Firings at the Martin Denver Static Test Facility. The flight test program at the Eastern Test Range consists of several "A" configuration launches from a reworked Titan I launch pad followed by "C" configuration launches from the Titan III Integrated Transfer and Launch (ITL) facility. In addition, the prelaunch and countdown preparations and utilization of checkout and flight test results in system design improvements are presented.
The Titan III Standard Space Launch Vehicle

The Titan III or Program 624A is the U. S. Air Force's booster system for varied space projects. The intent of the program is to provide the Air Force and the country with a standardized (multiple payload capability with no modifications to the booster), economical, and reliable launch vehicle with mission flexibility. Payload capability ranges from 5,000 to 25,000 pounds and mission capabilities include:

1. Low altitude elliptical orbits by direct injection
2. Low altitude circular orbits
3. Low altitude circular orbits with Hohmann transfer to another orbit
4. Synchronous orbit
5. Deep space trajectory to escape
6. Orbits with plane changes.

Payload configurations may include:

1. Lifting body vehicles
2. Bulbous loads
3. Various payloads which can be enclosed in a ten foot diameter standard payload fairing.

In order to provide this greater mission flexibility, the launch vehicle has been designed such that it can be flown in one of two configurations. These two configurations are Configuration "A" and Configuration "C". Figure 1 illustrates Configuration "C".

Configuration "A"

Configuration "A" or Core vehicle consists of a modified Titan II ICBM, a new liquid propellant stage referred to as transtage, and a standard payload fairing. Modifications to stage I consist of structural strengthening, mounting provisions for two solid propellant rocket motors, and an engine altitude start capability. The stage II modifications include the shortening of the space between propellant tanks by increasing tank size and the addition of autogenous pressurization to the oxidizer tank.

The transtage is composed of two modules: a propulsion module (Stage III), and a guidance and control module. The guidance and control module, located atop the propulsion module, contains the major elements of the flight control, electrical, instrumentation and guidance subsystems. This module also contains an attitude control propulsion system which is used for propellant settling prior to stage III engine starts and attitude control during transtage coast. The propulsion module contains two titanium propellant tanks, two 8,000 pound-thrust, re-startable engines, and a helium gas pressurization system. The transtage is illustrated in Figure 2.
Configuration "C"

Configuration "C" consists of Configuration "A" plus two identical ten foot diameter, 1,000,000 pound-thrust solid propellant rocket motors (stage 0) attached in parallel to stage I of the core vehicle.

Transtage

The key to the Titan III mission flexibility is the transtage. By coasting and restarting techniques, the following maneuvers may be performed:

1. Orbital altitude change
2. Orbital plane change
3. Circularization of orbit
4. Precision escape trajectory.

The propulsion systems which provide these capabilities consist of an attitude control system and the main stage III propulsion system. Both systems utilize nitrogen titroxide oxidizer, and a blend of unsymmetrical demethylhydrazine and hydrazine fuel.

Attitude Control System

The attitude control system provides impulse for bottoming the propellants in the main transtage tanks, and provides pitch, yaw, and roll control impulse for the transtage and payload. The system is a pressure-fed, storable, hypergolic, liquid-propellants, pulse width modulated rocket engine control system. The system is subdivided into packages, namely, the pneumatic, ordnance, regulator, safety, and engine modules plus propellant tanks, lines, and burst disc assemblies. Reference Figure 3.

Gaseous nitrogen is stored at a pressure of 3100 ± 100 psig in a spherical pressurant tank. The tank is made of titanium and has a nominal capacity of 3.55 pounds of GN₂. Included in the pneumatic module is a manual fill valve.

The ordnance package which isolates the high pressure nitrogen from the entire system until the launch countdown consists of a squib fired two-way isolation valve plus a squib fired three-way valve. The three-way valve is part of the redundant regulator system and is only fired upon overpressurization in the primary regulator system.

Components included in the regulator package are the primary regulator, secondary regulator, two ground test ports, and two filters. Pressurant upon release by the isolation valve flows through a two micron nominal and ten micron absolute filter into the regulator. The regulator is an all stainless steel regulator with a capacity up to 12 scfm while maintaining a regulated pressure of 290 ± 10 psig at inlet pressures varying from 3100 to 500 psig. Ground test valves are installed in the module for ground checkout of the regulators before the isolation (start) valve is actuated. If overpressurization occurs, the over pressure switch trips when 345 ± 10 psia is reached, which in turn actuates the three-way valve, bringing the standby regulator into operation, and isolates the primary regulator from the system.
A filter, two check valves, two relief valves, and two ground checkout valves are incorporated into the safety module. The function of the check valves is to prevent any propellant or propellant vapors from leaking back into the pneumatic module. The relief valves are sized to relieve maximum flow (12 scfm) of the regulator and will relieve at 370 psig and reseat at 355 psig. Checkout valves are used for checkout and during cleaning and filling procedures.

Propellant tanks, one oxidizer and one fuel, are made from 6066 aluminum and have capacities of 73.64 pounds of oxidizer and 47.26 pounds of fuel at 130°F. The volume of the tanks are 1515 in.³ including 10.23 in.³ and 20.86 in.³ ullage for fuel and oxidizer respectively. The tanks are designed for a 98% expulsion efficiency. Teflon bladders are used to give positive expulsion under zero "g" conditions. The oxidizer tank has two 0.007 inch ballders and the fuel tank has three .004 inch baldders. The bladder life is 20 cycles minimum.

The attitude control system utilizes four engine modules. Two of the modules (pitch) contain a 45 pound thrust chamber. The other two modules (roll - yaw) each contain two 25 pound thrust chambers and one 45 pound thrust chamber. The chambers are made of 90° oriented astrolite ablative material with an external glass wrap and a non-ablating silicon carbide throat insert. The rocket engine modules incorporate burst diaphragms upstream of the propellant valves. The diaphragms rupture when the isolation valve is actuated. The rocket engine assemblies utilize a splash plate injector and the nozzles have a 60 to 1 expansion ratio. The propellant flowrates are 0.0942 lb/sec oxidizer and 0.0604 lb/sec fuel for the 45 pound engine and 0.0524 lb/sec oxidizer and 0.0335 lb/sec fuel for the 25 pound engines. The mixture ratio is 1.56 ± 2%. Propellant control valves are fast acting, solenoid operated with a filter incorporated at the inlet. Each engine is balanced by the use of calibration orifices at the entrance to each propellant valve.

The attitude control system employs both passive and active thermal control systems. The passive system consists of high conductance, low emittance fairings which encloses the engine modules, and low emittance coatings on the engine propellant lines and fittings. Thermal insulators minimize conduction of heat from component packages to the vehicle structure.

The propellant lines are surrounded with nine water filled polyethylene tubes held in place with teflon string. These line and tube assemblies are covered with three layers each of coated FEP film (shield) and type I heat cleaned fiberglass cloth (spacer). These layers are tacked in place with small pieces of tape. At hanger clamp locations, a split teflon bushing isolates the line from the structural mounting clamp.

The two propellant tanks are each covered with three layers each of heat cleaned fiberglass (spacer) and coated FEP film (shield) attached with tape. Each tank is isolated from its supporting truss by thermal standoff washers.

The pneumatic module incorporates three water cans (approximately one pound of water) to provide heat to compensate for the gas temperature losses in the nitrogen tank and conduction and radiation losses from the module. To prevent unnecessary heat losses to the mounting
brackets, thermal standoffs have been incorporated.

The active system is composed of electrical heaters which are imbedded in the engine mount. These heaters are sized such that with a nominal 28 volt D.C. input, the heaters will draw about 2.1 amps of current. For a 6.5 hour coast, this corresponds to an energy expenditure of 382 watt-hours.

The flight instrumentation consists of pressure transducers to monitor nitrogen tank pressure, regulated pressure, and thrust chamber pressure on all eight engines. Ground instrumentation consists of a nitrogen tank low limit switch which makes at 2920 psig and breaks at 2800 psig, and high and low limit switches on regulated pressure which break at 308 \(\pm 8\) psig and makes at 270 \(\pm 10\) psig respectively.

Prelaunch checkout of the attitude control system at the Eastern Test Range consists of engine functional checks, relief valve functional and leak checks, overpressure and launch limit switch checks, pneumatic module functional and leak checks, burst disc assembly and tank bladder leak checks, and high pressure gas system leak checks. Engine functional checks consist of valve response checks (both opening and closing), heater element resistance check, and engine leak checks. An attempt was made to conduct injector blow-down checks to evaluate injector contamination, but a suitable criteria could not be established. Stated another way, test results would not permit the determination of a blocked or partially blocked injector hole(s) and the test was deleted.

The main problem uncovered at ETR on the ACS system is one of regulator failure. It has been determined that these failures were caused by a ball in the control section of the regulator being susceptible to oxidizer vapor. A change is currently in existence proposing the use of a ball material more compatible with \(\text{N}_2\text{O}_4\) vapors.

The only other major problem associated with ACS checkout at ETR has been one of establishing an acceptable procedure and criteria for bladder leak checks. The technique currently used is to pressurize the propellant side of the bladder to 45 psia for 15 to 20 minutes, decrease pressure to 15 psig and observe bubbles from the gas side of the bladder for three successive five minute intervals. If the bubble rate does not decrease or drop to zero during the successive intervals, the bladder is repressurized to 45 psig for another 15 to 20 minutes and is then decreased to 2 \(\pm 1\) psig. After 20 minutes at this reduced pressure, no leakage (bubbles) is allowed in the next five minutes.

**Stage III**

Stage III provides the propulsive power necessary for orbit injection, orbit plane changes, orbit transfers, or providing escape velocity to the payload. This stage is composed of three subsystems: the engines, the propellant system, and the pressurization system. Two engines capable of multiple burns are mounted side by side through flexural pivots in the engine mount to the thrust structure of the airframe. The flexural pivots make it possible to gimbal both engines for roll, pitch, and yaw control. An electrically driven hydraulic pump, controlled by signals from the guidance and control system, provides the hydraulic pressure to the gimbal actuators. The engines are capable of burning times up to 500 seconds and produce a thrust of 8,000 pounds each. The propellant tanks are mounted side by side in the airframe.
and are constructed of titanium. The tanks are designed to contain, nominally, 15,320 pounds of oxidizer and 7,690 pounds of fuel. The propellants are forced from the propellant tanks to the engines by a stored gas (helium) pressurization system. A pressure regulating system consisting, primarily, of solenoid controlled valves and pressure switches, maintain correct propellant tank pressures. The stage is designed to operate after being exposed to the environment of outer space for six and one-half hours.

The AJ 10-138 Engine. Two of the AJ 10-138 engines, manufactured by the Aerojet General Corporation, are used on stage III. Each of these liquid propellant rocket engines (Figure 4) deliver 8,000 pounds of thrust when operated in the hard vacuum of outer space. The engines utilize the same liquid propellants (nitrogen tetroxide oxidizer and a blend of hydrazine and unsymmetrical dimethylhydrazine fuel) as used in the first and second stages of the Titan III launch vehicle. The major components of the engine are the thrust chamber assembly, propellant lines, electrical controls harness, and the propellant valve assembly.

The thrust chamber assembly consists of an injector, thrust chamber, and nozzle extension. An ablative lining protects the thrust chamber from excessive temperatures. The nozzle extension is made of high strength metals and operates at temperatures below their yield strength. The injector and injector baffles are regeneratively cooled by the fuel.

The injector is a welded aluminum unit consisting of a faceplate, fuel manifold, oxidizer manifold, forged dome and baffles. The injection pattern is made up of unlike doublets or like-on-like-on-unlike quadlets. An outer ring of holes spray fuel to provide film cooling of the thrust chamber walls. Regeneratively cooled baffles are utilized to inhibit high-frequency combustion instability caused by circumferential pressure fluctuations.

The thrust chamber and nozzle extension assembly form a bell shaped DeLaval nozzle designed to impart optimum velocity to the exhaust gases. The thrust chamber is composed of an edge wrapped Refrasil glass strands impregnated with a char forming rubberized phenolic resin. This inner liner is insulated with a layer of phenolic impregnated asbestos felt. This assembly is then wrapped with a layer of epoxy impregnated glass cloth. The metallic hardware is then installed and secured by another layer of glass cloth and a layer of filament roving. When the injector and nozzle extension are bolted to the thrust chamber, the mating surfaces are sealed by O-rings and zinc chromate putty. During a full duration engine firing, the throat area growth is approximately 6%.

The nozzle extension consists of a forward skirt and an aft skirt. The temperature encountered in the forward skirt area exceeds limits established for titanium; therefore, the forward skirt is fabricated from .030 inch thick type C103 Columbium sheet. The aft skirt is constructed from .025 inch thick titanium sheet. The forward and aft skirts are then welded together. The nozzle extension provides the engine with an expansion ratio of near optimum for vacuum conditions.

Propellant flow to the engine is controlled by a bi-propellant valve mounted on the injector dome. Simultaneous flow of both
propellants is assured by mounting the fuel and oxidizer poppets on a common shaft. Propellant flow is controlled by electrical signals to a solenoid operated pilot valve. When the solenoid is energized, the pilot valve directs fuel pressure to the power cavity on the opening side of the piston which shuttles the shaft and opens the fuel and oxidizer poppets. When the signal is removed, the pilot valve vents the fuel pressure behind the piston and springs return the valve to the closed position.

Propellant System. Stage III propellants are stored in cylindrical pressure vessels (tanks) with conical bottoms and elliptical tops. The tanks are fabricated from titanium. Each tank contains slosh baffles, a gas diffuser, a propellant trap, outlet antivortex baffle, an outlet screen, a manhole, and internal braces. The tanks are mounted in parallel and secured to the vehicle airframe by struts. The oxidizer tank has nominal volume of 177.4 cubic feet and a maximum loadable volume of 170.9 cubic feet. This tank is approximately 63 inches in diameter and 143 inches long. The fuel tank nominal volume is 142.2 cubic feet and maximum loadable volume of 137.0 cubic feet. It is approximately 50 inches in diameter and 167 inches long.

The gas diffuser, a screen mounted in the tank top, serves two functions: first, to slow and disperse the high velocity pressurization system gases entering the tank and second, to provide a large amount of surface area for the propellant to wet and inhibit the possible flow of the propellant into the pressurization line under zero "g" start.

The propellant trap, in the tank bottom, contains two 4 inch check valves; one permits flow into, and one permits flow out of the tank. These check valves unseat at a pressure of approximately 2 psi and are fully open at 4 psi. This trap ensures propellant feed to the engines for a zero "g" start. The purpose of the outlet screen is to break up any possible gas bubbles in the propellant feed system prior to their reaching the engines.

The propellant system is required to supply oxidizer at flow rates between 32.7 and 35.5 pounds per second and fuel between 16.3 and 17.7 pounds per second to the engines at tank pressures between 160 and 166 psia. A design objective for this system is to have a maximum outage of 1.0% and a mean outage of 0.318% with propellant temperatures between 45° F. and 90° F.

Pressurization System. The pressurization system is required to maintain propellant tank ullage pressures between the minimum pressures required for propellant feed and the maximum allowable structural pressure. This is accomplished by controlling the flow of helium gas from high pressure storage vessels to the propellant tanks. The system flight tested to date is shown schematically in figure 5a. Gaseous helium for pressurization is stored in two interconnected spherical containers at a prelaunch pressure of 3600 ± 100 psig. The storage spheres are sized such that a minimum gas pressure of 300 psig will exist at the end of a 440 second engine run at a temperature of 400° R. The nominal helium load is 46.8 pounds at launch.
On vehicles launched to date, the flow of gas from the inter-connected storage spheres is through two identical but separate pressure control systems. Each of these systems contain a filter, a solenoid controlled pressure operated shutoff (bang-bang) valve, a flight pressure switch, an accumulator, a relief valve, and a check valve. The bang-bang valves are connected electrically to the flight pressure switches. These switches are normally closed at pressures below 163 psia and open above this pressure. The flight pressure switches sense pressure in the accumulator. The accumulator function is to damp out pressure transients that exist in the main pressurization line. This, in turn, causes the pressure switch to respond only to average system pressure and accordingly, limit pressure switch cycles to only those required by system demand. Check valves are utilized to minimize propellant vapor diffusion into the pressurization system.

System safety is provided by an electro-mechanical relief valve. Prior to launch, the valve is electrically connected to the upper launch limit switch mounted on the propellant tank. The upper launch limit is established at 120 psi to insure that a safety factor of two is always maintained while the vehicle is on the launch pad. To maintain tank pressures below this limit, prior to launch, a checkout pressure switch is utilized to control tank pressures. The checkout switch is tank mounted and is set to operate at 92 psia. It is electrically connected to the bang-bang valve and when energized, controls the tank pressure to this limit. This switch is electrically disabled at launch as is the electrical section of the relief valves. The relief valve is designed to relieve mechanically at pressures in excess of 180 psia. The flight pressure switches are electrically enabled at stage II ignition and tank pressures increase to 163 psia at this time.

Following the failure of the pressurization system to perform its intended function during the first flight of the Titan III, Configuration "A", a modification was made to the pressurization system to provide protection against the possibility of a fail close bang-bang valve. This modification consisted of an ordnance operated crossover valve installed below the bang-bang valves, which when operated, permits either of the separate pressurization system to supply pressurizing gas to both propellant tanks. Pressure switches are utilized to provide a signal to the crossover valve. These switches also sense accumulator pressure. The under pressure switch is electrically armed at stage III ignition and disarmed at shutdown. It will provide an electrical signal to the crossover valve if either tank pressure drops to 153 psia during stage operation.

A completely redesigned pressurization system shown schematically in figure 5b is currently under development and will be utilized on future launch vehicles. This system is designed such that no single failure in any electrical element will result in loss of tank pressure control. The functional operation of this system is similar to that already discussed except that the bang-bang valves are arranged in series - parallel and a single pressurization line tees to supply pressure to both propellant tanks. Electrical relays have been added between the flight pressure switches and the valve solenoids to minimize switch contact arcing. Two check valves in series in each tank top are utilized to preclude the possibility of fuel and oxidizer vapors entering the pressurization system. Since the bang-bang valve arrangement now provides protection against a valve fail open situation, the relief valve has been replaced with an ordnance operated vent valve. This valve is to provide only launch pad protection and is disabled at launch.
Environmental Control. During missions requiring operation for extended periods in space, transtage component temperatures are maintained within allowable limits by utilizing the heat sink potential of the entire stage. In addition, special surface finishes and insulation are used to maintain heat balance over the wide range of orbital thermal environments. The stage III helium spheres and the portion of the propellant tanks internal to the airframe are insulated with one layer of fiberglass cloth covered by another layer of gold plated Viton. A Viton shield is suspended across the staging plane (II/III) with cutouts for the propellant tanks. Below the staging plane, the propellant tanks are protected by alternate layers of course mesh fiberglass and gold plated stainless steel foil (two layers each starting with fiberglass) and an outer layer of stainless steel wire mesh. The trap areas of the tanks, the feed lines, and the engine truss are covered with three layers of fiberglass and steel foil with a protective covering of wire mesh. Special preformed insulation is also used to protect the engine injector and bi-propellant valve.

Design Confirmation Tests. In addition to the normal component development and design assurance testing and engineering evaluation tests, stage III underwent a series of design confirmation tests. A brief description of these tests and test objectives are presented below:

1. Cold flow tests of the propellant and pressurization subsystem, using battleship propellant tanks and prototype pressurization system components were conducted at the Martin Denver Cold Flow facility. The objectives of this test series were:
   a. Confirm full-scale tankage configuration
   b. Confirm full-scale propellant trap, screen and outlet configuration
   c. Confirm propellant outflow characteristics and feed line dynamics
   d. Confirm propellant loading methods and procedures
   e. Confirm the compatibility of the propellant and pressurization subsystems with actual propellants
   f. Confirm the pressurization subsystem performance as determined by pressure histories, flow capacities, response, lack of cyclic and pressure transients, and the thermodynamics of the helium loading process
   g. Confirm system mechanical integrity
   h. Confirm helium usage characteristics
   i. To accumulate reliability data and to evaluate subsystem reactions to imposed malfunctions.

2. Captive (static firing) tests using battleship tankage and prototype hardware were performed at the Aerojet General Corporation. The purpose of this test series was to demonstrate the objectives listed under 1.a through 1.h in an engine firing environment as well as confirm the propulsion system analytical model and engine subsystem performance.
3. A series of altitude simulation firing tests were performed in the J-3 satellite rocket cell at the Arnold Engineering Development Center. The purpose of this test series was to confirm the design of the transtage propulsion system and to determine the heating rates of transtage components in the actual airborne configuration while operating at a simulated altitude. Propulsion test objectives were:

a. Determine engine radiative heating of tanks, lines, and vehicle structure
b. Determine the engine radiative cooled skirt and external ablative chamber temperatures
c. Determine skin heating effects of the Attitude Control System engine exhaust
d. Determine propellant and pressurization system dynamic response during vacuum start and shutdown transients
e. Verify that engine performance during start, steady-state, and shutdown is within specification limits
f. Collect data defining operating characteristics of the complete pressurization system in a simulated altitude environment.

4. A series of environmental tests were performed in the high vacuum orbital simulator of the Lockheed Missiles and Space Company. This test series followed the altitude simulation firings and utilized the same hardware. These tests were to confirm the ability of the passive temperature control subsystem to maintain temperatures within the predicted range during extreme thermal environments encountered in outer space.

Design Verification Tests. Prior to the first Titan III flight test at ETR, a complete production flight type transtage was subjected to a captive compatibility firing at Martin Denver. The purpose of these firings was to demonstrate compatible operation of the altitude control and propulsion subsystems with the control module and to determine subsystem steady-state and dynamic performance characteristics. Additional objectives were to demonstrate the functional compatibility of the propellant subsystem with the ground equipment and to verify operating procedures.

The final verification of a propulsion system design is the flight test program. This program consists of 17 vehicles to be launched from the Eastern Test Range. The stage III propulsion test objectives in flight test programs are to verify that:

1. Tank top pressures and helium usage remain within predicted levels
2. Stage III engine operation conforms to specification requirements
3. Propellant system insulation methods are adequate for extended orbital coast times
4. The stage III engine system is compatible with the transtage
5. All interface pressures fall within specified limits
6. That the complete system, including airborne equipment, ground equipment, and checkout procedures will fulfill their intended purpose when the launch vehicle reaches operational status.

Necessary for the accomplishment of a successful flight test program are adequate prelaunch subsystem checkout and test procedures to accomplish this checkout. Stage III propulsion system checkout utilizes methods and techniques developed during the Titan I and Titan II programs. All operating subsystems components are functionally checked prior to the accomplishment of leak checks. Engine checks consist of bi-propellant valve functional, electrical harness, and instrumentation checks followed by a complete leak check of the thrust chamber and propellant lines. The calibration of all pressure switches utilized in the propellant and pressurization subsystems is functionally verified and complete end-to-end checks performed on the electrical circuitry.

Procedural changes were made to the original pressurization functional tests to expand the scope of the test to include an overall system check and a water vapor content check of the pressurizing gas.

In the accomplishment of the functional check, the pressurization system is required to perform a simulated flight. Tanks with volumes representing propellant tank ullages and bleed orifices to simulate inflight usage of gas, are connected to the outlets of the pressurization subsystem. With gas in the helium sphere, the pressurization subsystem is activated and the system operation is monitored. Typical results of this test are presented in figure 6. Additionally, the propellant and pressurization subsystems are completely leak checked. These checks include internal component leakage as well as the more conventional checks.

**Conclusion**

The intent of the Titan III design, development and field test programs is to permit delivery of "Zero Defect" launch vehicles to the customer. The success of this intent is born out by the flight test results. To date, three Configuration "A" vehicles have been launched. The first vehicle achieved 95% of the flight test objectives. The last two achieved 100%. The third vehicle flew a highly complex flight program in that an orbit transfer was accomplished requiring multiple (three) burns of stage III. The final burn occurred 4½ hours after launch and injected the payload into a 1500 nautical mile circular orbit.

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Titan III, Configuration "C"
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ATTITUDE CONTROL SYSTEM SCHEMATIC
BI-PROPELLANT VALVE
INJECTOR
THRUST STRUT SUPPORT
GIMBAL RING ASSEMBLY
VEHICLE FRAME (REF)
OXIDIZER LINE
FUEL LINE
SHIPPING ARM
ENGINE SUBASSEMBLY #3
NOZZLE EXTENSION
ENGINE SUBASSEMBLY #4
AJ10-138 ROCKET ENGINE - TRANSTAGE
STAGE III PRESSURIZATION SYSTEM SCHEMATIC (INTERIM)
STAGE III REDUNDANT PRESSURIZATION SYSTEM
Typical pressurization system functional test results (fuel)