A99-31253

AIAA-99-2469
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35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit
June 20-23, 1999 / Los Angeles, CA

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DESIGN AND ANALYSIS OF A LOW-COST REACTION CONTROL SYSTEM FOR GPS IIF

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Abstract

This paper presents the design and analysis of the Reaction Control System Module (RCSM) for the GPS IIF satellite. This modular system uses monopropellant hydrazine propulsion to provide satellite delta-V for final orbit adjustment, orbital maintenance, and subsequent readjustment of the orbit, backup attitude control, and desaturation of momentum wheels. There are two modules, one each on opposite sides of the spacecraft. Each module has six (6) 1 lbf thrusters and two (2) 5 lbf thrusters, its own bladder type propellant tank and propellant management system (fill valve, pressurant valve, filter and latch valve). A crossover line assembly, installed at the spacecraft level connects the two modules. This allows sharing of propellant between the modules, as required, to maintain balance and provide for availability of propellant, where needed, in the case of unequal demand. The structural design of the module allows all loads to be carried directly to spacecraft framework and facilitates stability during module manufacture. The use of previously flight-qualified components and standard analysis approaches results in a low-cost, high reliability, spacecraft propulsion system.

Objective

To design a modular spacecraft Reaction Control System for the GPS IIF satellite that provides low-cost and ease of integration while fully maintaining the capability and flexibility of the GPS IIA Reaction Control System (which required a dedicated propulsion system manufacturing team). To maintain aggressive spacecraft cost targets, it must also be producible at a significantly lower cost than its predecessors.

Approach

The system was baselined as a monopropellant hydrazine system based on cost, performance, safety, reliability, and heritage to previous GPS satellites. Monopropellant hydrazine designs are simpler, lower cost, safer and have higher reliability than competing bipropellant systems since monopropellants require storage tanks, plumbing, valving, and components for one propellant rather than two. With the long mission life (approximately 15 years) and thrust range (using primarily 1.0 lbf thrusters) hydrazine meets all the mission requirements.

In order to meet the required performance flexibility, while providing for optimum packaging within the GPS IIF envelope, the system is designed as a module which incorporates an outer panel of the spacecraft as 1) the primary mounting location for all system components and plumbing (except the propellant tank), 2) the structural mounting interface to the vehicle, 3) the rocket engine module (REM) mounting interface, and 4) a primary support for the propellant tank mounting ring.

The system structure, see isometric in Figure 1, includes the exterior panel tank support ring and tank support panels. By employing the formed aluminum tank support panels the module becomes self-supporting when mounted at the vehicle interface. These panels also provide a location for mounting the electrical panels, Figure 2, such that system connectors are supported and can be accessed easily during vehicle integration. The machined aluminum exterior panel, which utilizes an isogrid design, is as light and strong as a honeycomb design while costing significantly less. The tank support ring is machined from titanium (6Al-4V) plate and provides the needed strength and stiffness for the fully loaded propellant tank. It also provides thermal isolation from conductive losses.

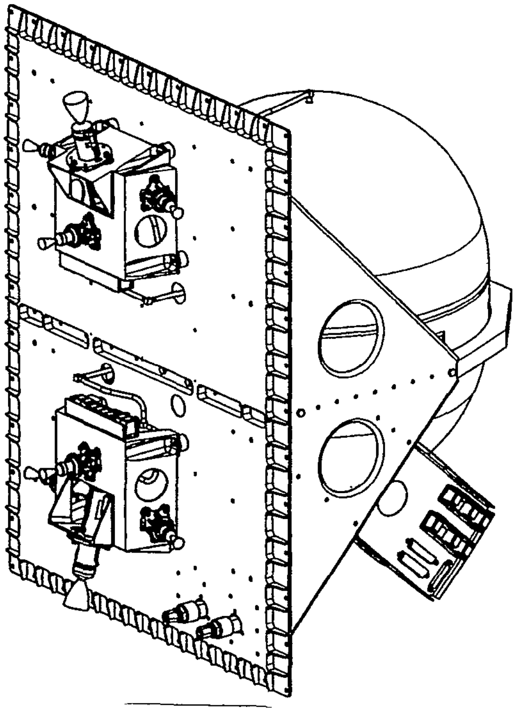


Figure 1: GPS IIF Low-Cost Propulsion Module - Outboard View

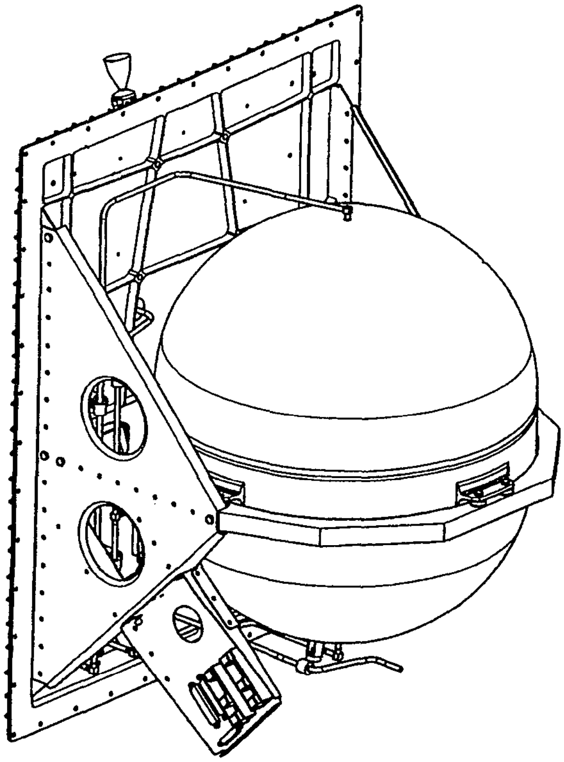


Figure 2: GPS IIF Low-Cost Propulsion Module - Inboard View

Design Description

A schematic overview of the reaction control system is shown in Figure 3. The system is composed of two identical modules connected by a crossover line allowing them to share fuel from their separate fuel systems. The modules (shown in isometric view, Figure 4) are installed on opposite sides of the spacecraft. Each module consists of a fuel storage and pressurization subsystem, fuel management subsystem, two rocket engine modules, thermal management components, and supporting structural elements.

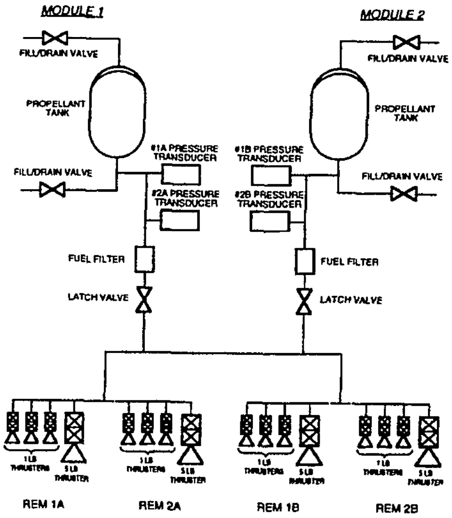


Figure 3: GPS IIF Low-Cost Propulsion Module - Mechanical Schematic

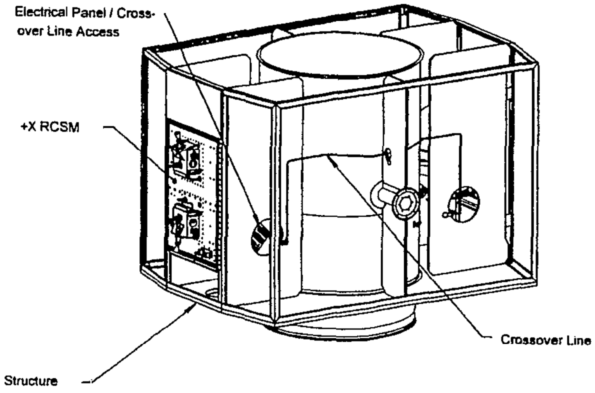


Figure 4: GPS IIF Low-Cost Propulsion Module - Integrated Into Spacecraft Structure

The fuel storage subsystem include the propellant tank, fill/drain valve for fuel and fill/vent valve for pressurant. These service valves are located on the exterior panel at the low end, providing for gravity drain in the event of emergency detanking is required at the launch site. (Figure 5)

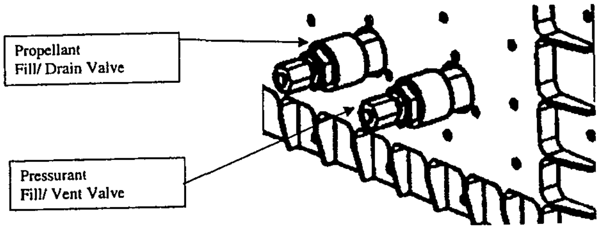


Figure 5: GPS IIF Low-Cost Propulsion Module - Service Panel

The fuel management subsystem includes the pressure transducers, system filter, and latch valve. Either tank can provide fuel to any of the thrusters.

Each of the identical rocket engine modules (REM's) has a single 5.0 lbf (MR-106E) thruster and three 1.0 lbf (MR-111C) thrusters. The 5.0 lbf thruster and the two 1.0 lbf thrusters, which are oriented tangentially to the spacecraft panel are canted outward to reduce plume impingement on the spacecraft. The outward facing 1.0 lbf thrusters are oriented normally to the spacecraft Z-axis. To prevent EMI originating within the RCSM from affecting other external spacecraft systems, EMI shrouds are installed on the REM's and over the service valves (Figure 6).

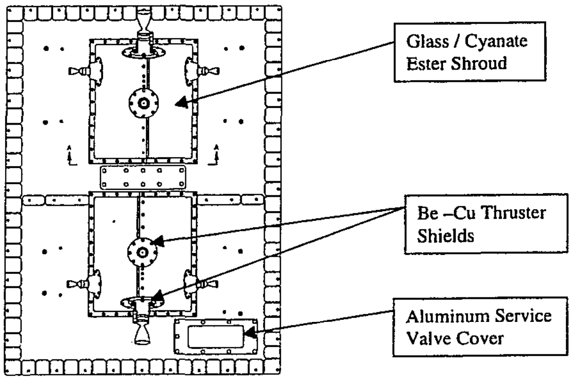


Figure 6: GPS IIF Low-Cost Propulsion Module- EMI Shields Installed

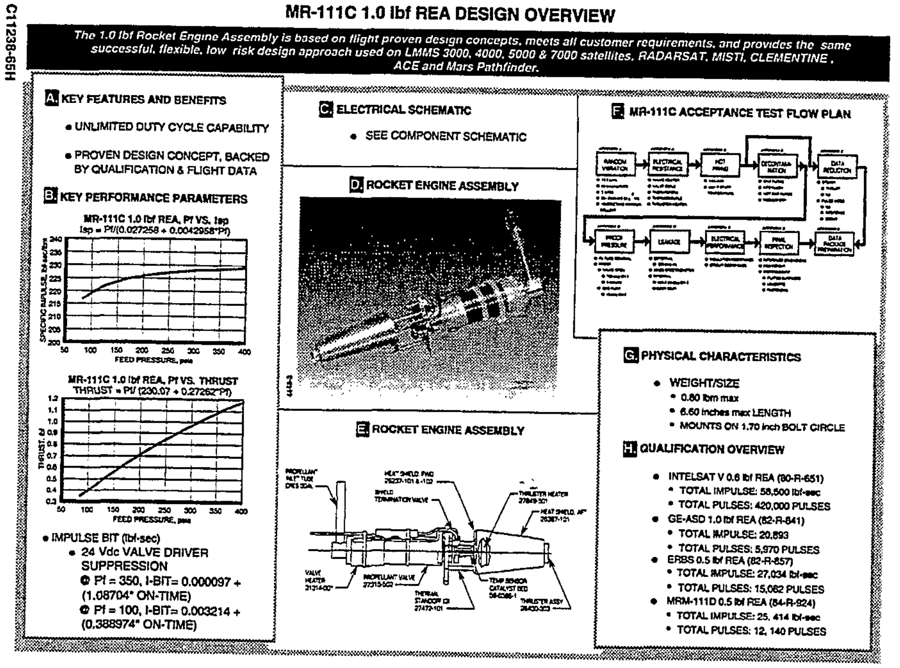
The thermal control system consists of line, tank, component, and thruster heaters controlled by an onboard controller. The controller is provided inputs from sensors integrated in the propulsion system.

Component Descriptions

Thrusters

BNA selected PAC's MR-111C 1.0 lbf thrusters for fine attitude control adjustment, station adjustments, and de-saturation of the momentum wheels. PAC's MR-106E 5.0 lbf thrusters were selected for spacecraft orientation during spin mode in the transfer orbit, final orbit insertion trim , and coarse adjustments. Each of these thrusters is well qualified beyond the GPS IIF total impulse and pulse requirements, and was able to contribute a strong heritage, low-cost, and a continuous production line to the GPS IIF program. PAC (then Rocket Research Company) originally qualified the 1.0 lbf thruster in its 0.5 lbf version for Intelsat V, where engines saw service lives of up to thirteen years with little or no degradation. The 5.0 lbf thruster was originally qualified for the Centaur upper stage of the Atlas and Titan launch vehicles and is also used on the GPS IIR spacecraft. Recent launches of spacecraft with the MR-111C 1.0 lbf thruster have included Mars Climate Orbiter and Mars Polar Lander, Mars Global Surveyor, ACE, NEAR, Clementine, and many others. PAC has produced over 700 of the MR-111 0.5/1.0 lbf thrusters. Recent launches of spacecraft with the MR-106E 5.0 lbf thrusters include A2100™ communications spacecraft, Lunar Prospector, NEAR, and GPS IIR. PAC has produced over 2500 of the MR-106 5-9 lbf thrusters.

Each of the thrusters has a catalytic decomposition chamber, a dual seat, dual coil, sliding fit propellant valve, a dual element catalyst bed heater (for pre-heat of the decomposition chamber), a dual element propellant valve heater (to maintain the hydrazine above freezing), and a Chromel/Alumel thermocouple (for catalytic decomposition chamber temperature telemetry). Figure 7 provides details of the 1.0 lbf thruster: Figure 8 provides details of the 5.0 lbf thruster. Because the thrusters had a strong heritage, and met or exceeded all of the GPS performance and environmental requirements, no additional qualification testing was required.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MR-111C 1.0 lbf REA DESIGN OVERVIEW

The 1.0 lbf Rocket Engine Assembly is based on flight proven design concepts, meets all customer requirements, and provides the same successful, flexible, low risk design approach used on LMMS 3000, 4000, 5000 & 7000 satellites, RADARSAT, MISTI, CLEMENTINE, ACE and Mars Pathfinder.

A. KEY FEATURES AND BENEFITS

- UNLIMITED DUTY CYCLE CAPABILITY
- PROVEN DESIGN CONCEPT, BACKED BY QUALIFICATION & FLIGHT DATA

B. KEY PERFORMANCE PARAMETERS

MR-111C 1.0 lbf REA, Pf VS. Isp
Isp = Pf/(0.027258 + 0.0042958*Pf)

MR-111C 1.0 lbf REA, Pf VS. THRUST
THRUST = Pf/ (230.07 + 0.27262*Pf)

- IMPULSE BIT (lbf-sec)
  - 24 Vdc VALVE DRIVER SUPPRESSION
    - @ Pf = 350, I-BIT= 0.000097 + (1.08704* ON-TIME)
    - @ Pf = 100, I-BIT= 0.003214 + (0.388974* ON-TIME)

C. ELECTRICAL SCHEMATIC

- SEE COMPONENT SCHEMATIC

D. ROCKET ENGINE ASSEMBLY

E. ROCKET ENGINE ASSEMBLY

F. MR-111C ACCEPTANCE TEST FLOW PLAN

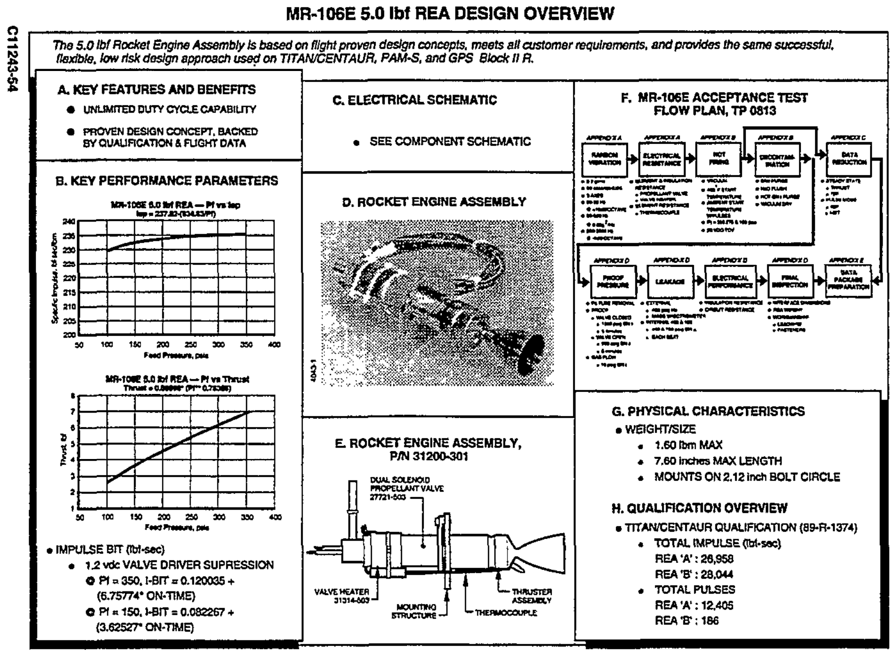
G. PHYSICAL CHARACTERISTICS

- WEIGHT/SIZE
  - 0.80 lbm max
  - 6.60 inches max LENGTH
  - MOUNTS ON 1.70 inch BOLT CIRCLE

H. QUALIFICATION OVERVIEW

- INTELSAT V 0.6 lbf REA (80-R-651)
  - TOTAL IMPULSE: 58,500 lbf-sec
  - TOTAL PULSES: 420,000 PULSES
- GE-ASD 1.0 lbf REA (82-R-841)
  - TOTAL IMPULSE: 20,893
  - TOTAL PULSES: 5,970 PULSES
- ERBS 0.5 lbf REA (82-R-857)
  - TOTAL IMPULSE: 27,034 lbf-sec
  - TOTAL PULSES: 15,082 PULSES
- MRM-111D 0.5 lbf REA (84-R-924)
  - TOTAL IMPULSE: 25, 414 lbf-sec
  - TOTAL PULSES: 12, 140 PULSES

Figure 7: MR-111C Thruster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MR-106E 5.0 lbf REA DESIGN OVERVIEW

The 5.0 lbf Rocket Engine Assembly is based on flight proven design concepts, meets all customer requirements, and provides the same successful, flexible, low risk design approach used on TITAN/CENTAUR, PAM-S, and GPS Block II R.

A. KEY FEATURES AND BENEFITS

- UNLIMITED DUTY CYCLE CAPABILITY
- PROVEN DESIGN CONCEPT, BACKED BY QUALIFICATION & FLIGHT DATA

B. KEY PERFORMANCE PARAMETERS

MR-106E 5.0 lbf REA — Pf vs Isp
Isp = 237.82-(934.83/Pf)

MR-106E 5.0 lbf REA — Pf vs Thrust

- IMPULSE BIT (lbf-sec)
  - 1.2 vdc VALVE DRIVER SUPRESSION
    - @ Pf = 350, I-BIT = 0.120035 + (6.75774* ON-TIME)
    - @ Pf = 150, I-BIT = 0.082267 + (3.62527* ON-TIME)

C. ELECTRICAL SCHEMATIC

- SEE COMPONENT SCHEMATIC

D. ROCKET ENGINE ASSEMBLY

E. ROCKET ENGINE ASSEMBLY, P/N 31200-301

F. MR-106E ACCEPTANCE TEST FLOW PLAN, TP 0813

G. PHYSICAL CHARACTERISTICS

- WEIGHT/SIZE
  - 1.60 lbm MAX
  - 7.60 inches MAX LENGTH
  - MOUNTS ON 2.12 inch BOLT CIRCLE

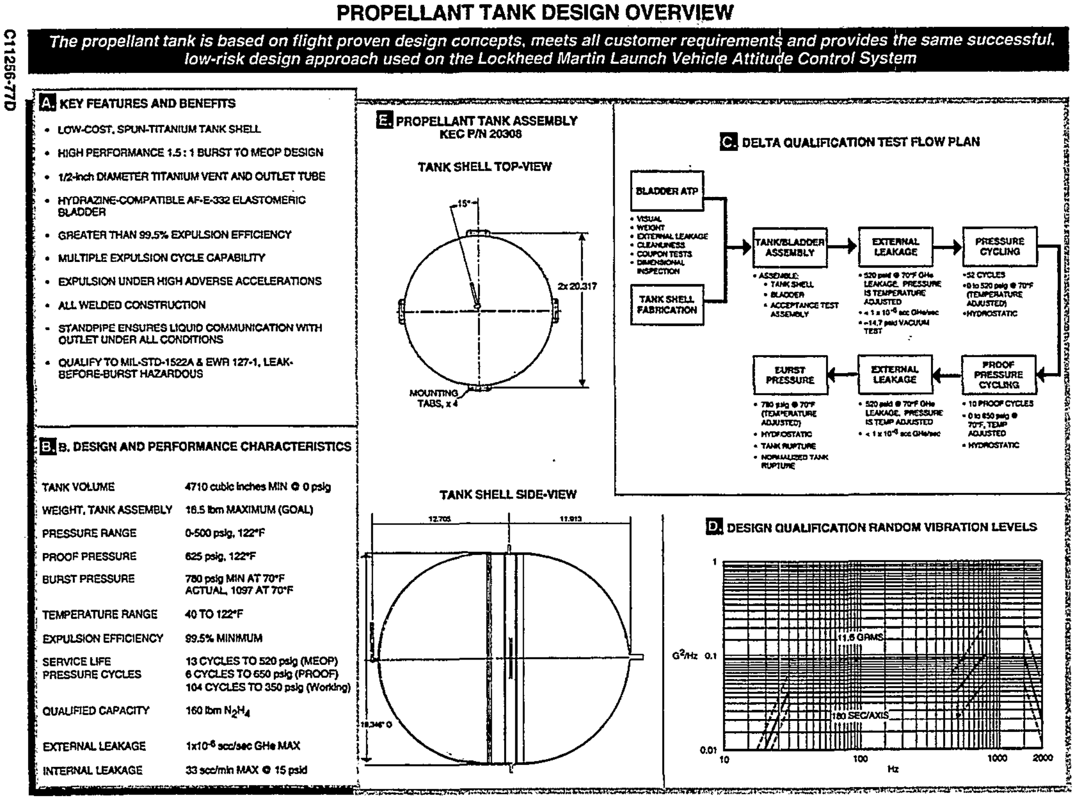
H. QUALIFICATION OVERVIEW

- TITAN/CENTAUR QUALIFICATION (89-R-1374)
  - TOTAL IMPULSE (lbf-sec)
    REA 'A' : 26,958
    REA 'B' : 28,044
  - TOTAL PULSES
    REA 'A' : 12,405
    REA 'B' : 186

Figure 8: MR-106E Thruster

Propellant Tank

A design overview of the propellant tank is shown in Figure 9. The tank assembly consists of a titanium shell, a bladder retention assembly, and a hydrazine compatible positive expulsion bladder. The shell has two spin formed domes and a forged girth ring. Each component is made from 6AL-4V titanium and is machined to final wall thickness and length. The bladder retention/outlet assembly consists of a number of machined elements which capture the bead of the bladder and prevent cross contamination of the fuel and pressurant. The GPS IIF propellant tank evolved from earlier designs where the bladder assembly was bolted into the tank at a flange ring. By replacing the bolted interface with an all welded assembly the weight was reduced while eliminating an o-ring type seal to the outside. This was appropriate to the extended mission life of GPS. The bladder, was developed and qualified for the earlier, bolted tank design. The AFE-332, an ethylene-propylene terpolymer, from which the bladder is made, was specifically developed for use in hydrazine compatible positive expulsion devices. The bladders are molded on breakout plaster molds using a vacuum bagging technique that results in a very low-cost finished product. The standpipe, which is part of the bladder retention/outlet assembly, prevents the bladder from prematurely closing off the outlet during expulsion. The result is 99.5% expulsion efficiency. Bolted and welded versions of this tank have flown LMLV and the Pegasus HAPS stage.



PROPELLANT TANK DESIGN OVERVIEW

The propellant tank is based on flight proven design concepts, meets all customer requirements and provides the same successful, low-risk design approach used on the Lockheed Martin Launch Vehicle Attitude Control System

A. KEY FEATURES AND BENEFITS

- LOW-COST, SPUN-TITANIUM TANK SHELL
- HIGH PERFORMANCE 1.5 : 1 BURST TO MEOP DESIGN
- 1/2-inch DIAMETER TITANIUM VENT AND OUTLET TUBE
- HYDRAZINE-COMPATIBLE AF-E-332 ELASTOMERIC BLADDER
- GREATER THAN 99.5% EXPULSION EFFICIENCY
- MULTIPLE EXPULSION CYCLE CAPABILITY
- EXPULSION UNDER HIGH ADVERSE ACCELERATIONS
- ALL WELDED CONSTRUCTION
- STANDPIPE ENSURES LIQUID COMMUNICATION WITH OUTLET UNDER ALL CONDITIONS
- QUALIFY TO MIL-STD-1522A & EWR 127-1, LEAK-BEFORE-BURST HAZARDOUS

B. B. DESIGN AND PERFORMANCE CHARACTERISTICS

TANK VOLUME	4710 cubic inches MIN @ 0 psig
WEIGHT, TANK ASSEMBLY	16.5 lbm MAXIMUM (GOAL)
PRESSURE RANGE	0-500 psig, 122°F
PROOF PRESSURE	625 psig, 122°F
BURST PRESSURE	780 psig MIN AT 70°F ACTUAL, 1097 AT 70°F
TEMPERATURE RANGE	40 TO 122°F
EXPULSION EFFICIENCY	99.5% MINIMUM
SERVICE LIFE PRESSURE CYCLES	13 CYCLES TO 520 psig (MEOP) 6 CYCLES TO 650 psig (PROOF) 104 CYCLES TO 350 psig (Working)
QUALIFIED CAPACITY	160 lbm N_2H_4
EXTERNAL LEAKAGE	$1x10^{-6}$ scc/sec GHe MAX
INTERNAL LEAKAGE	33 scc/min MAX @ 15 psid

Figure 9: Low-Cost Propellant Tank

Latch Valve

The selected latching valve, shown in Figure 10, was previously used by PAC on the ACE Spacecraft and by Hughes on the HS-376 series of spacecraft. The isolation valve is used in orbit to allow controlled use of fuel from one module at a time. This protects the spacecraft from total loss of propulsion in the event of a leak from a single tank or manifold. The latch valve back-pressure relief is set at 400 – 600 psid to assure that fuel pressure isn't lost from both sides of the system in the event of loss of pressure on one side. The linear force motor design uses a separate back-pressure relief valve, which can be set independently of the motor forces needed to open and close the valve. The requirement for a fairly high, and stable, reverse cracking pressure, is therefore met while maintaining a low pressure drop and high flow area through the main seat. Although the valve is protected from contaminants by the system filter, there is additional protection provided by the 30 micron filter integral to the valve inlet.

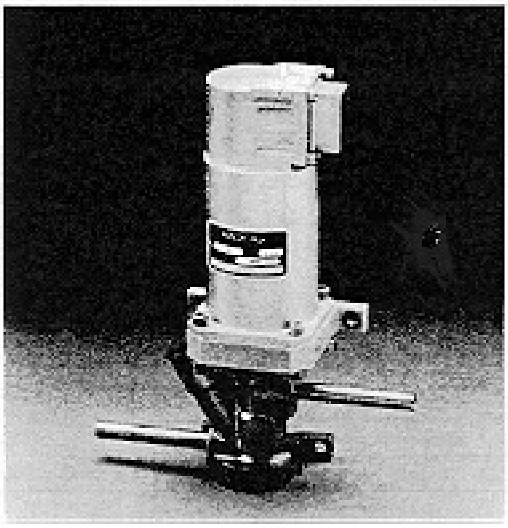


Figure 10: Linear Force Motor Isolation Valve

Service Valves

The selected service valves are shown in Fig. 11, and been previously used successfully on MILSTAR and CRSS. It is also being used currently on EO-1. The service valves are constructed of stainless steel and provide redundant seals with metal-to-metal primary seal and a backup o-ring or cap seal. While the service valves are identical internally different size inlet fittings are used for the pressurant and propellant to eliminate the possibility of misconnection. Service valve adapters, with filters, are used to protect the valves during assembly and test operations.

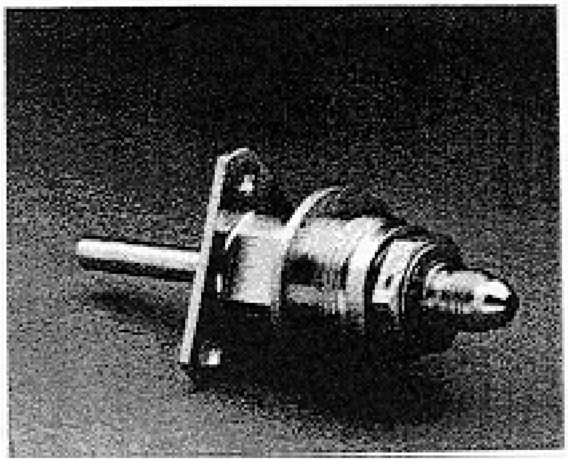


Figure 11: Service Valve

Pressure Transducer

The selected transducer, used on numerous PAC programs, is an amplified 0 to 5 volt output with a 0 to 500 psia pressure range and an accuracy of +/- 1.5 % of full scale. The design utilizes foil strain gages bonded to the backside of an integral beam pressure diaphragm. The transducer is used by the vehicle telemetry system, and transmitted to ground control, to monitor fuel pressure. This data is used both for gauging remaining fuel and to calculate available thrust levels.

Filter

A high capacity, 10 micron, in-line filter is used to remove any contaminants that may have entered the system or been generated in the tank. The design utilizes a CRES wire mesh screen filter and is capable of handling up to 500 mg. of a.c. fine dust. The filter has successful heritage on IUS and Athena.

Analysis Results

System analyses have been conducted in the areas of performance, thermal, structural, plume, reliability, safety, and fluid dynamics to ensure that the design is capable of providing the required mission performance throughout the mission life and over the range of mission environments. The results of these analyses are summarized below.

Performance Analysis

Performance modeling of the system was conducted to provide: 1) tank pressure during blowdown, 2) pressure drops throughout the system, and 3) nominal thruster performance as a function of system variables (including tank temperature and duty cycle).

Structural Analysis

The approach to determining the structural acceptability of the RCSM involved: i) verifying that the qualified component environmental levels exceed the predicted vehicle levels; ii) conduct system level finite element modeling of the tank, plumbing, and structure; and iii) perform propellant tank stress and fracture mechanics analysis. The finite element analysis of the system structure, tank, and plumbing was used to estimate overall margins and system frequency response. Subtier models of the REMŌs were used to evaluate the input into the thrusters themselves from the overall response of the exterior panel. The requirement to maintain a fundamental frequency for major system structural elements above 42 Hz was demonstrated analytically and the model subsequently verified by modal test. Figure 12 shows the rendering of the system finite element model. The propellant tank fracture mechanics analysis demonstrated the tank's ability to meet the requirements of MIL-STD-1522A.

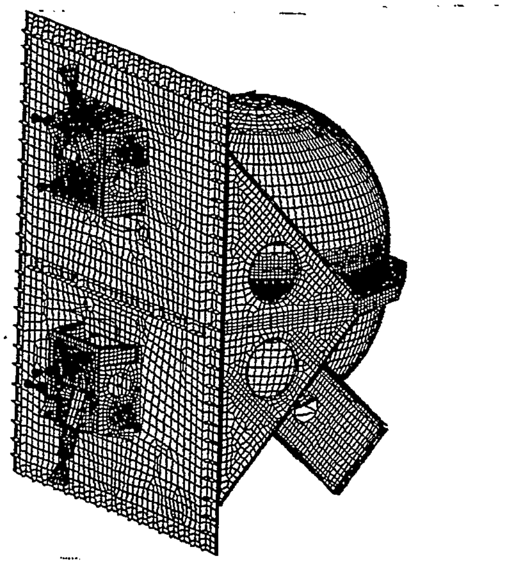


Figure 12: GPS IIF Low-Cost Propulsion Module-Rendering Of The System FEM

Thermal Analysis

Thermal analysis was conducted to assure that neither overheating nor freezing of propellant occurred during any phase of the mission. Worst case minimum and maximum environmental temperatures and thruster component temperature bias the system and feed back to the spacecraft environment. As a result of the spacecraft level thermal analysis; line, tank, and component heater sizes were selected. Each RCSM is divided into four control zones: one for each REM, one for the tank, and one for the lines and components. Thermal control is provided through a proportional controller, which gets feedback from RTD's placed at predicted cold points within each thermal control zone. Low emissivity tape covers lines and components to reduce radiative losses and components are attached to the structure through insulating standoffs. The tank is covered with a multi-layer insulation (MLI) blanket. All heater circuits are redundant and within each circuit the heaters are wired in series to assure that an open circuit in a heater or leadwire will cause the temperature at the control sensor to drop, activating the backup circuit. There are also incorporated into the design a number of thermistors for telemetry. These provide for overall health monitoring by ground controllers.

Models of the thrusters are used to evaluate heat soak-back to the REM and spacecraft structures and determine required times for catalyst bed heating, prior to firing spacecraft maneuvers. The post firing thermal soak becomes part of the operating environment in which the heaters for the REM must maintain all wetted surface temperatures above freezing. Because the REM is a single control zone, soakback must not bias the control sensors to keep heaters turned off while a line or valve further from the heat source (recently fired thruster) drops below the freezing point of hydrazine. A simplified thruster thermal model is shown in Figure 13. Output from this model, firing at specified duty cycles becomes an input to the REM modeling.

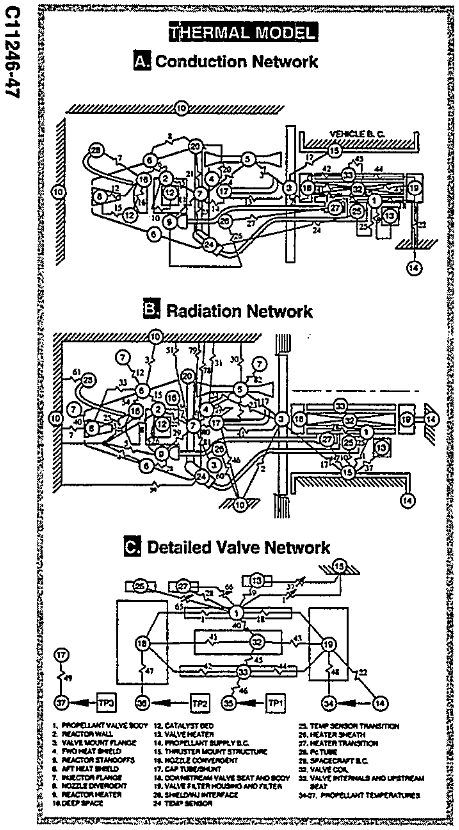


Figure 13: MR-111C Simplified Thruster Thermal Model

Plume Analysis

Near field plume modeling of the 1.0 and 5.0 lbf thrusters was conducted to determine plume impingement forces and heating on the vehicle structure. The nozzle flow fields were solved using a fully coupled Navier Stokes solution which was then used to obtain the plume flow field via direct simulation Monte Carlo (DSMC) technique. The nozzle plume fields are shown in Figure 14. The plume analysis showed that the thruster alignment results in acceptable impingement forces and heat loads to the vehicle.

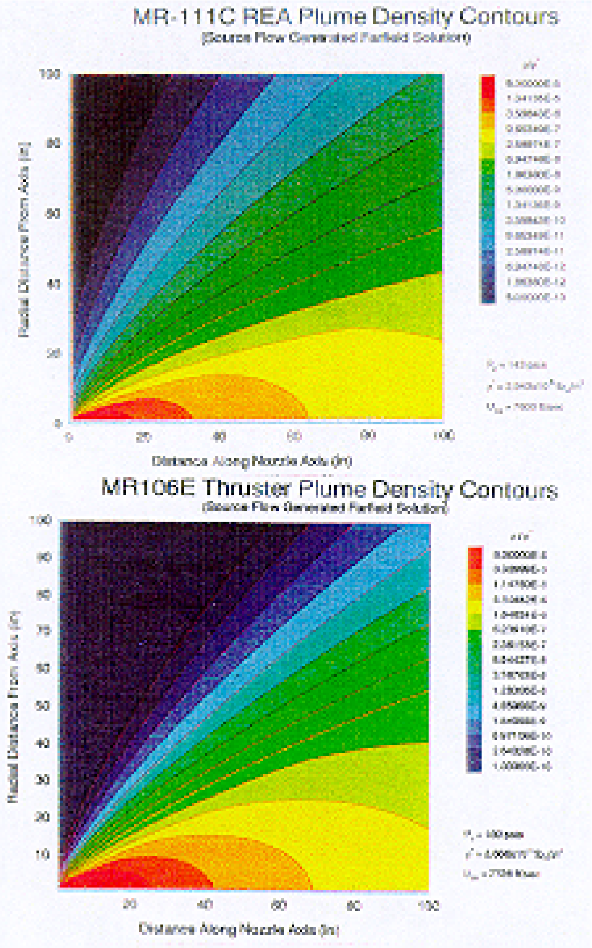


Figure 14: Thruster Plume Field Analysis

Reliability Analysis

A reliability prediction was determined with the use of reliability logic diagrams, mathematical models and complete calculations progressing from the detailed listing of generic failure rates, the modification factors used to account for different environmental stresses and operating conditions, applicable time/cycle data, and the step-by-step use of this data in the reliability mathematical models. The reliability analysis takes into account the redundancy of the thruster valve seats, service valve caps, propellant tank redundancy (once orbit is achieved) and redundant thrusters (on station). The overall predicted reliability is .99762.

In accordance with Range Safety requirements, a complete failure modes, effects, and criticality analysis was completed. A single point failure summary and critical items list was also prepared.

Fluid Transient Analysis

Water hammer analyses were conducted using detailed simulation model. All significant feed system elements were incorporated including: latch valve, filter, line viscous losses, and propellant valves. Equivalent line lengths were used to account for losses associated with bends elbows and tees. Transient predictions were generated via a method of characteristics solution. Peak system feed pressure for the analysis used 425 psia, including margin. The worst shutdown case was assumed to be simultaneous firing of one MR-106E (5.0 lbf thruster) and two MR-111C (1.0 lbf thruster). The peak predicted transient are summarized in Table I. Plots of pressure transients at the MR-106E thruster, latch valve and pressure transducer are shown in Figure 15.

Table I. GPS IIF Low-Cost Propulsion Module Peak Predicted Pressure Transients

Component	Peak Pressure (psia)	Proof Pressure (psia)
Latch Valve	627	900
Drain Valve	511	900 (w/Δqual)
Filter	545	1000
Pressure Transducer	542	1000
Propellant Valve		
MR-106C	736	1500
MR-111C	714	1500

Test and Integration

The propulsion system acceptance test plan is shown in Figure 16. All thruster and RCSM level testing is conducted at Primex Aerospace Co. facilities. The modules are then sent to Boeing North American (Seal Beach) for integration into the GPS spacecraft. Along with each pair of modules a crossover line kit; including lines, heaters, thermal sensors, and low emissivity thermal tape is provided. For each module a separate EMI close-out kit is also provided. After integration into the vehicle, including connecting the crossover line; complete electrical and leak checks are performed, thruster alignment is verified (and adjusted to spacecraft datums if required) and the EMI shield kits are installed. Spacecraft level environmental testing is performed and the satellite is shipped to the launch site, where final checkout and fueling is performed prior to launch.

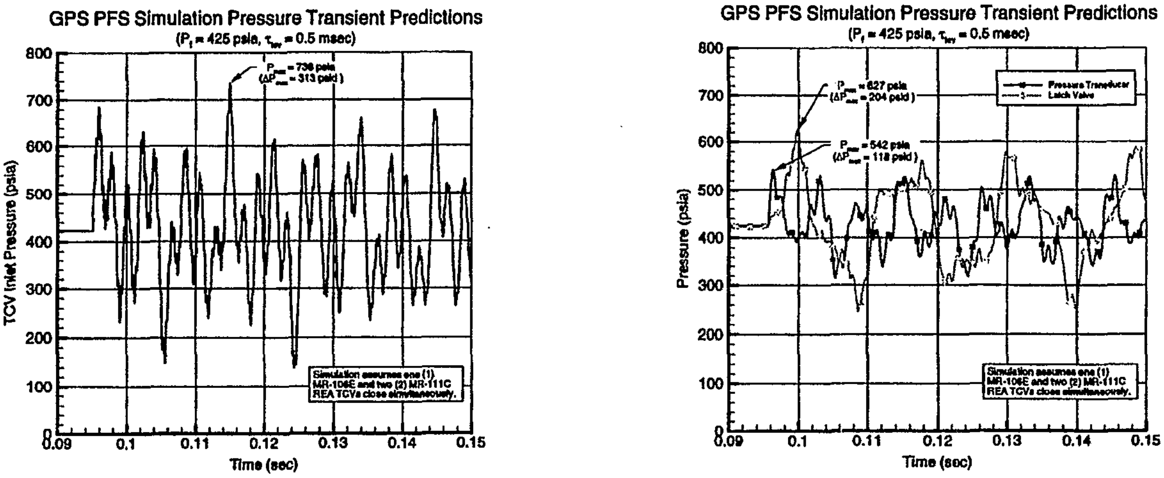


Figure 15: GPS IIF Low-Cost Propulsion Module Pressure Transients

Operational Description

Mission

The Navstar Global Positioning System (GPS) is a space-based radio-positioning system consisting of a constellation of more than 24 orbiting satellites that provides global navigation and timing information to users. In addition to the satellites, the system consists of a worldwide satellite control network and GPS receiver units that translate satellite signals into position information.

The system is operated and controlled by the 50th Space Wing located at Falcon Air Force Base, Colo.

RCS Usage Related to GPS Mission

The GPS RCS is serviced with propellant and pressurant prior to integration with the Launch Vehicle. The launch vehicle places the spinning satellite into a highly elliptical transfer orbit. A solid rocket motor, as part of the Orbital Insertion Subsystem (OIS), provides the instantaneous thrust and total impulse required to accomplish an inclination change, as well as circularize to the final orbit. This rocket firing is at the apogee of the transfer orbit, allowing the OIS motor to be also named the Apogee Kick Motor (AKM). The spacecraft is spinning at a high rate (2-axis stabilized) during early mission operations to be as stable as possible during the AKM burn. The RCS compliments the OIS to position the satellite prior to the OIS burn, as well as provide velocity and control changes required placing the spacecraft in the final GPS orbit. The RCS is also used for stationkeeping, disposal and reaction wheel momentum unloading.

Inversion Maneuver

The MR-106E thrusters are used in a coupled, pulsed mode, correlating with a reference spacecraft angle to torque the vehicle to the desired final orbit inclination. Spacecraft attitude sensors precisely measure the timing of the thruster pulses, to accomplish the inversion maneuvers in a timely manner.

OIS Burn and Trim, Drift Rate Maneuvers

The balanced OIS SRM provides the majority of the total impulse required positioning the GPS spacecraft in the final circular orbit. Typically, due to launch timing and relative positions of other GPS satellites in the constellation, an orbital period different than that used by operational GPS satellites is used during early mission operations. This period difference allows the spacecraft to 'drift' into the optimal slot relative to other GPS satellites. If drift rates are not within mission timeline requirements, the MR-106 thrusters are steady state fired to provide the proper drift, moving the spacecraft towards final orbital slot.

Horizontal Maneuver

The Horizontal Maneuver orients the satellite for earth capture by repositioning the spacecraft so the antennas point towards earth. This is accomplished by pulse firing of the MR-111C thrusters, timed in accordance with spacecraft spin rate, attitude control logic and ground station coverage.

Stop Drift Orbit Correction

The MR-106 thrusters are fired steady state to stop the spacecraft drift of in the final orbital slot.

Orbital period will be in phase with the GPS constellation.

Despin Operations and Earth Capture Operations

MR-111 thrusters are fired steady state, opposite the spin direction to stop the spacecraft from spinning. Upon completion of all despin operations, the GPS spacecraft is 3-axis stabilized, ground communication is validated.

Final Station Maneuvers

The RCS MR-111C thrusters are used to control spacecraft disturbances imparted by solar wing deployment and reaction wheel startup.

Stationkeeping

The RCS MR-111C thrusters are used for providing the velocity increments necessary to maintain the orbital period and eccentricity limits in relationship with the GPS constellation. The MR-106E thrusters are available for coarse velocity changes, as well as redundant steering control.

Disposal

The RCS MR-111C thrusters are used for providing the velocity increments necessary to move the GPS spacecraft to a safe orbit, away from the GPS operational constellation at end of it's useful life.

Conclusions

This paper presents the low-cost design approach for the Reaction Control System Module for the GPS IIF satellites. The engineering design phase of the program is complete and the first shipsets of hardware are complete.

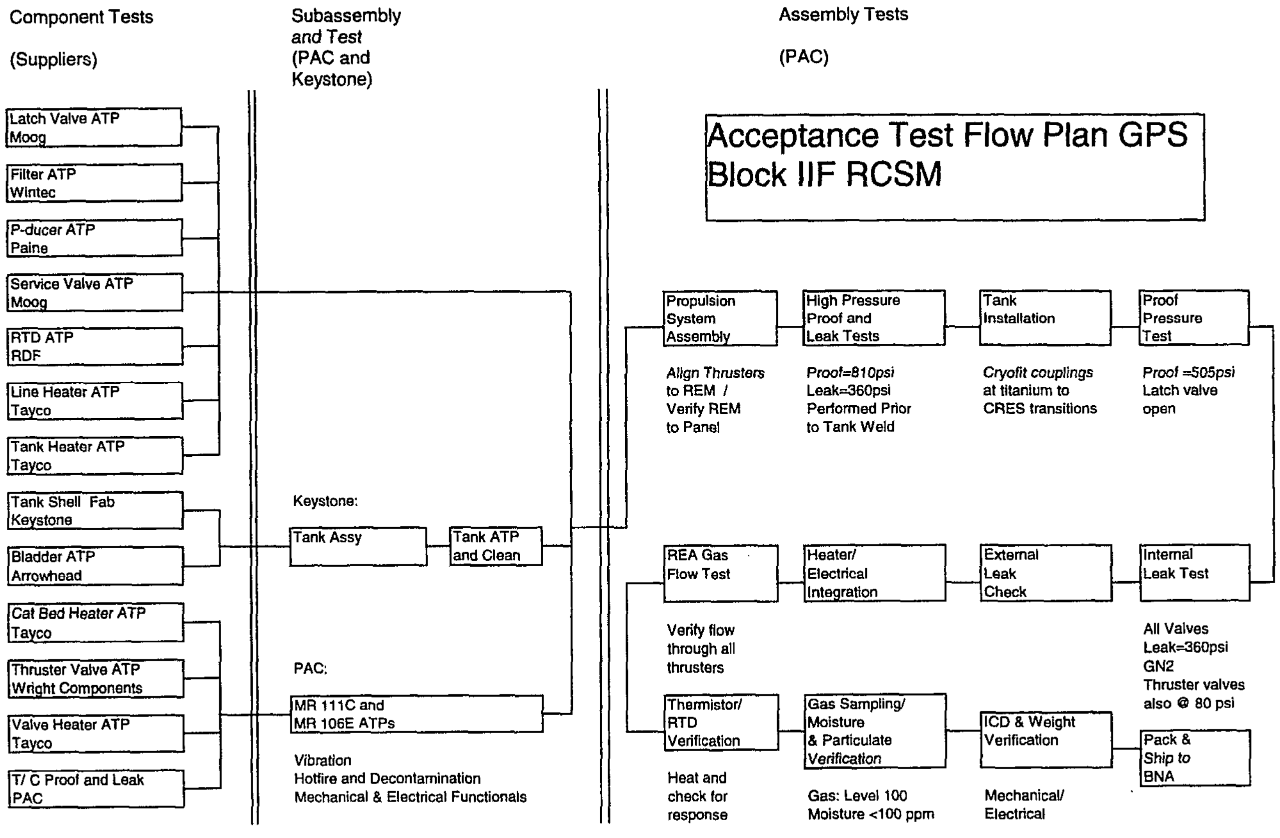


Figure 16: GPS Low-Cost Propulsion Module Acceptance Test Flow Plan