Development of Monopropellant Propulsion System for Low Earth Orbit Observation Satellite

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Abstract

The currently developed propulsion system(PS) is composed of propellant tank, valves, thrusters, interconnecting line assembly and thermal hardwares to prevent propellant freezing in the space environment. Comprehensive engineering analyses in the structure, thermal, flow and plume fields are performed to evaluate main design parameters and to verify their suitabilities concurrently at the design phase. The integrated PS has undergone a series of acceptance tests to verify workmanship, performance, and functionality prior to spacecraft level integration. After all the processes of assembly, integration and test are completed, the PS is integrated with the satellite bus system successfully. At present, the severe environmental tests have been carried out to evaluate functionality performances of satellite bus system. This paper summarizes an overall development process of monopropellant propulsion system for the attitude and orbit control of LEO(Low Earth Orbit) observation satellite from the design engineering up to the integration and test.

Key Word : Propulsion System, Monopropellant, Low Earth Orbit(LEO), Satellite

Introduction

Generally, the purpose of propulsion system of the spacecraft is to provides the required impulse for initial orbit correction of nominal launch vehicle dispersions to insert the spacecraft into the final target orbit, as well as orbit maintenance and attitude control.

Especially, the monopropellant propulsion system is a widespread used type due to its great advantages of low cost, technical simplicity, high reliability and relative stability. Therefore, most of light weight satellites widely located on low earth orbit (LEO) have accepted this system for their missions over decades[1]. Typically in this system, the high pressure pressurant provides a pressure force to drive the liquid propellant, such as hydrazine (N_2H_4), flow moving from the tank to the thrusters where hot gases are created by chemical decomposition and exhausted through a nozzle.

The currently developed propulsion system (PS) in Korea Aerospace Research Institute (KARI) is an all-welded, monopropellant hydrazine system for the LEO observation satellite, KOMPSAT (Korea Multi-purpose Satellite), as shown in Fig. 1. It includes all the components and assemblies associated with storing, conditioning, routing, controlling, and expelling propellant required to meet the satellite mission requirements.

This paper summarizes an overall development process of the monopropellant propulsion system in Korea from the design engineering up to the integration and test.

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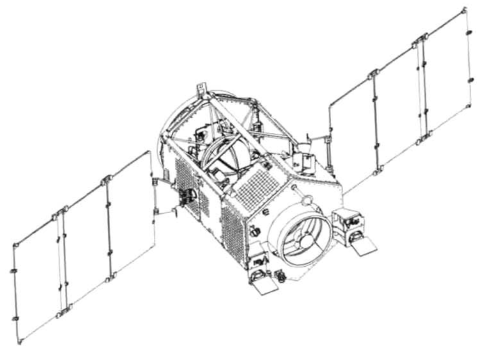


Fig. 1. Schematic of KOMPSAT Satellite

System Design Overview

The PS is designed to load a maximum 72.5 kg propellant budget of monopropellant grade hydrazine per MIL-P-26536 and to operate in a pressure blowdown mode from 2410 kPa maximum at 38 ℃ to 410 kPa at 4 ℃. It consists of the subassemblies and components such as thrusters, propellant tank, pressure transducer, filter, latching isolation valves, fill/drain valves, and interconnecting propellant line assembly. Also, thermal control circuits with heaters, thermostats, and temperature sensors are installed to prevent propellant from freezing in the space environment[2]. To guarantee the functional system reliabilities and minimize risks, those components which have flight heritage and have been flight qualified on other space programs are selected carefully and used in the PS. A design schematic of the PS is shown in Fig. 2.

The propellant tank is made of titanium forgings that are vacuum-welded with a flexible diaphragm inside to maintain pressure on the hydrazine throughout the mission. Its inlet and outlet ports are connected to fill/drain valves for loading or unloading the liquid propellant and the gaseous pressurant. The pressurant gas is separated from the propellant by an expulsion diaphragm which ensures a steady supply of gas-free propellant under the adverse acceleration environment. As the liquid propellant is expelled from the tank, the inlet pressure supplied to

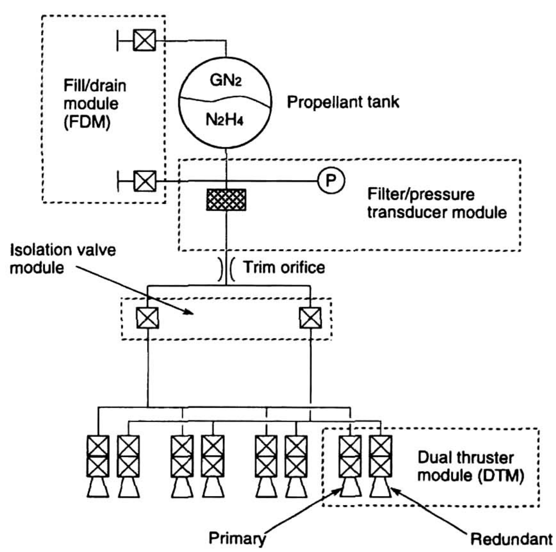


Fig. 2. Design Schematics of Propulsion System

thruster valves drops throughout the lifetime of the propellant usage. Downstream of the propellant tank is connected with a filter and a pressure transducer together. The filter prevents contamination from the tank or propellant from entering the downstream components and thruster feed lines while a pressure transducer is used to measure propellant supply pressures and provide telemetry signal to the ground during mission life. At further downstream, the latching isolation valves are installed to protect the system against failed open thruster valves or excessive leakage conditions by providing an isolation capability between the propellant tank and the thrusters. All the propellant lines in the PS are made of stainless steel tubes to prevent from chemical reactions such as corrosion, erosion and dissolution due to toxic hydrazine propellant. A flow restriction orifice located in front of isolation valves minimizes pressure transients such as water hammer phenomenon in the thruster feed lines. Eight 4.45 N thrusters with dual seat thruster valve on each are used for orbit transfer and on-orbit functions. The thrusters are packaged into four dual thruster modules (DTMs), with one primary and one redundant thruster per DTM. Each DTM is attached to a single mounting plate and associated thermal control hardware and electrical circuits. The thruster chamber consists of a head-end assembly, catalyst bed, and a chamber/nozzle assembly. The catalyst bed contains two different sizes of Solvay KC12GA catalyst. The chamber/nozzle assembly is a 30° canted nozzle with an expansion ratio of 50:1.

When an electrical signal for thruster operation is enabled, liquid hydrazine is supplied to the thruster chambers as a spray through injector from a propellant tank in a pressure blowdown mode. As soon as hydrazine contacts the iridium catalysts, the highly pressurized hot gases generated by the chemical decomposition of hydrazine are expelled from the chamber and are accelerated through a high expansion ratio nozzle. These supersonic exhaust gases produce a reaction force called a thrust and it enables a satellite to control its maneuver in the vacuum environment. During thruster firing, two kinds of blowdown modes, a steady state or a pulse mode, are generally employed either. In a steady state mode, thrusters burn for an extended time period whereas thrusters are successively fired on-off to produce a series of impulse bits in a pulse mode operation.

There are a number of performance requirements for the thruster, e.g., thrust level, thrust duration in steady state firing mode, and impulse bit, impulse repeatability in pulse mode firing, as well as total impulse depending on a deliverable propellant quantity. The nominal thrust level of each thruster flowed down from satellite system requirement is 4.23 N at beginning-of-life (BOL) for an inlet pressure of 2413 kPa when a thruster is fired continuously under normal operating conditions.

Once the performance requirements of thruster are decided, the requirement of all the components comprised in the PS is subsequently specified. Trade-off study under the system design constraints is necessitated while selecting and sizing the components. The physical

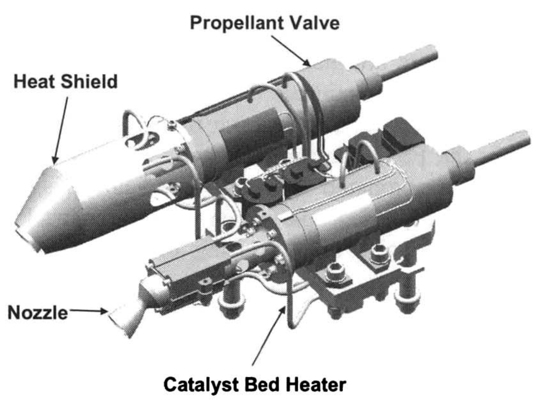


Fig. 3. Configuration of Dual Thruster Module

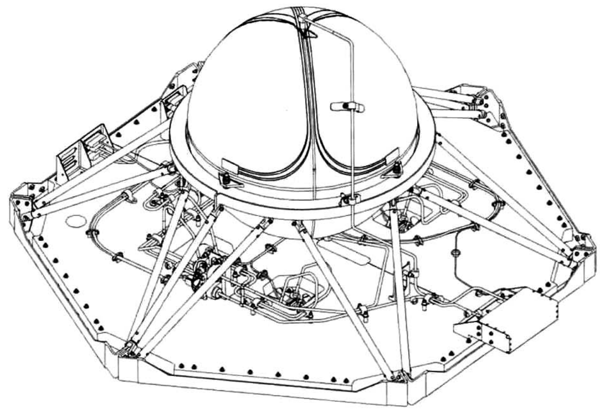


Fig. 4. Final Design Configuration of Propulsion System

characteristics including mechanical and electrical configuration of each component is fed back to the PS design. A compromise and modification of the PS configuration is made repetitively through tradeoffs with upper-system level of constraints and with manufacturing practicality of the PS itself.

A final design configuration of the PS is shown in Fig. 4. The PS is furnished as a submodule that is readily integrated into a satellite to a broad range of requirements with lightweight construction. The design also reflects all the mechanical, electrical, and thermal interface definition as well as GSE (Ground Support Equipment) interface to spacecraft bus. Furthermore, the PS is designed to comply with the safety requirements required to fill, drain, maintain, and service the satellite during all ground operations. Hazard analysis results to date indicate that the PS is two-fault tolerant to Class I hazards. For ground and flight safety during the launch phase, three mechanical inhibits for the highly toxic hydrazine are provided by the isolation valves upstream of the thrusters and the dual seat thruster valve. Similarly, three mechanical seals are in place at the fill/drain valves.

System Engineering Analysis

The PS is designed to load a maximum 72.5 kg propellant budget of monopropellant grade hydrazine per MIL-P-26536 and to operate in a pressure blowdown mode from 2410 kPa maximum at 38 ℃ to 410 kPa at 4 ℃. It consists of the subassemblies and components such as thrusters, propellant tank, pressure transducer, filter, latching isolation valves, fill/drain valves, and interconnecting propellant line assembly. Also, thermal control circuits with heaters, thermostats, and temperature sensors are installed to prevent propellant from freezing in the space environment[2]. To guarantee the functional system reliabilities and minimize risks, those components which have flight heritage and have been flight qualified on other space programs are selected carefully and used in the PS. A design schematic of the PS is shown in Fig. 2. Primary design parameters in the PS are originated from structural analysis, thermal analysis, propellant flow analysis, and plume analysis, besides bus system requirements. Propellant budget estimation based on satellite mission scenario also affects sizing of propellant tank at the early phase of major component selection.

Structural analysis is performed for the verification of structural integrity of the PS under both launch and operating environments. Methods of classical mechanics as well as computer simulation using FEM are employed for the analysis. A typical analysis result of the DTM bracket is depicted in Fig. 5 in terms of stress distribution. Fundamental frequencies of all brackets resulted from vibration analysis reveal that they are well above the minimum required frequency of 100 Hz (125 to 328 Hz). Analysis results show that all brackets have positive margins of safety under limit design load conditions which are much greater than acceptance test levels that is the highest loads expected during the launch (0.35 to 0.88)[3].

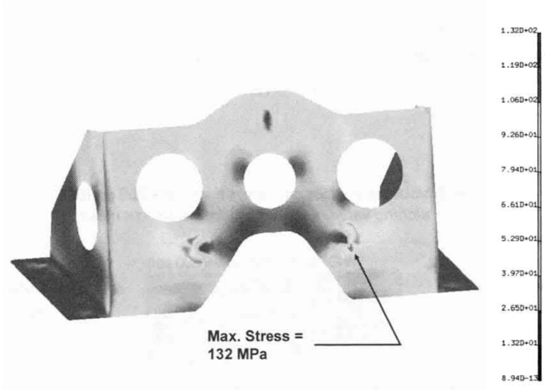


Fig. 5. Stress Distribution on DTM Bracket

In order to protect propellant against freezing and over-temperature, components of the PS which contain liquid propellant hydrazine shall be maintained at all times above 7 ℃ and below 49 ℃, except for the thruster valves for which the maximum allowable temperature resulting from thruster firing is 116 C and the propellant tank for which the maximum allowable temperature shall not exceed 38 ℃. Both a primary and a redundant heater circuit, each with two thermostats placed in series, are attached on each hydrazine wetted components, all of which are enclosed by MLI (Multi Layer Insulation), to meet the thermal management criteria. A lumped parameter modeling using TAS (Thermal Analysis System) is employed for the prediction of thermal behavior of each propulsion component. A typical thermal behavior of propellant tank is shown in Fig. 6. Propulsion heaters are sized based on the condition of worst cold case and over the worst case average voltage (25 volts) of spacecraft bus. All heaters assigned to propulsion components are proven to operate meeting the thermal requirements endowed by spacecraft bus specification[4].

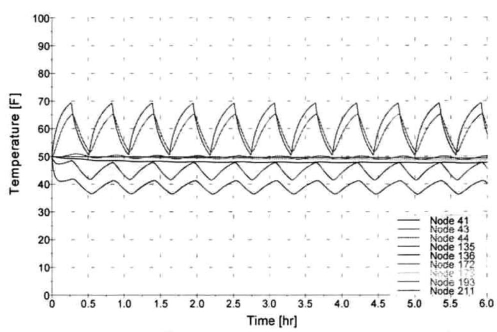


Fig. 6. Thermal Behavior of Propellant Tank

Both a minimization of steady-state pressure drop and a prediction of transient flow behavior resulting from valve operations in fuel feeding system are essential to flow passage design and for mitigation of the unwanted fluid dynamic behavior such as water hammering. Loss coefficient of propellant lines and tees are calculated based on Moody Chart characterized by Reynolds number, line curvature, and relative roughness, whereas the pressure loss in major equipment components is directly interpolated based on vendor's empirical data. Transient pressure behavior is predicted utilizing MOC (Method of Characteristics) network model under the assumption of uniform, isothermal, and single-phase (liquid) transport of propellant. Fig. 7 presents the predicted peak pressures accompanied by opening isolation valve in unprimed-propellant condition which must be avoided in practical propellant filling configuration. Maximum pressure of pressure transducer is higher than the required MOP, when the isolation valve is opened with its real opening time (13-14 msec) under the condition of unprimed propellant line[5].

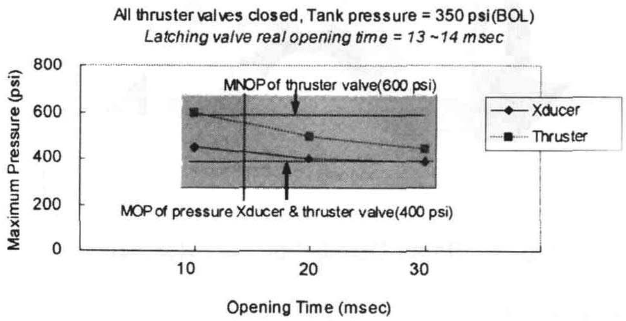
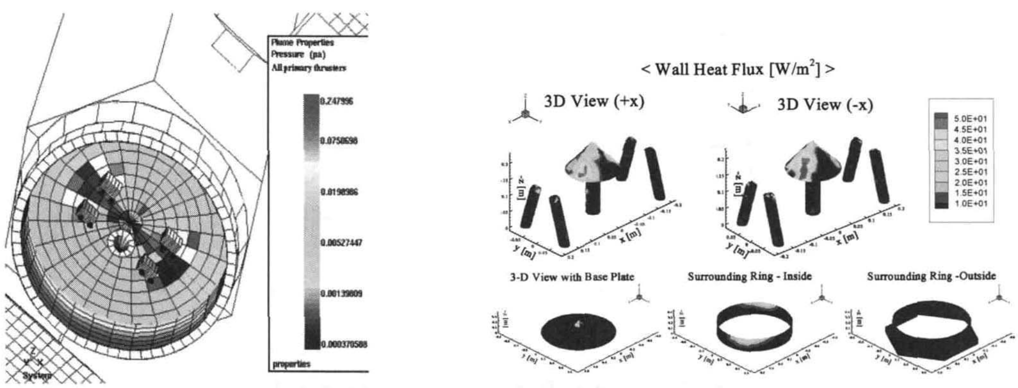


Fig. 7. Max. Pressure Estimations of Propellant Flow

The catalytic decomposition of hydrazine (N_2H_4) can be ideally described as a two-step process while ignoring the other steps and intermediate products. Firstly, the hydrazine decomposes into gaseous ammonia (NH_3) and nitrogen (N_2). Secondly the ammonia decomposes further into nitrogen and hydrogen (H_2) gases. Consequently, the exhaust mixture consists of NH_3, N_2, and H_2. Thruster exhausted plume causes impingement forces and torques on spacecraft, heating on thruster-nearby components, as well as particle deposition on some contamination -critical satellite parts. Navier-Stokes or Euler Solver with Source Pointing Method is introduced as a way of simplified prediction of plume effects whereas DSMC (Direct Simulation Monte-Carlo) Method is adopted for a comprehensive understanding of the plume behavior.



(a) Surface Pressure Distribution **(b) Wall Heat Flux**

Fig. 8. Thruster Exhausted Plume Analysis Results

A typical result of plume-induced dynamic effects is depicted in Fig. 8 in terms of surface pressure and wall heat flux distributions for the case of all thrusters firing. Prediction of the plume-induced parasitic torques is significant to agility augmentation of spacecraft attitude control as well as the realistic estimation of propellant consumption. Plume-induced heating also affects the positional design of thruster-nearby components. Since the given operating condition in which all four thrusters are burned simultaneously is the most severe case for thermal loading, it is expected that the thermal loading would be no problem for any operating conditions of satellite[6].

Parts, Materials & Processes Engineering

The PS is made of over 170 kinds of parts and material and process specifications utilized for the buildup of the PS constituents amount to over 150 as shown Fig. 9. PM&P (Parts, Materials, and Processes) engineering sets up those specifications and controls manufacturing processes in order to obtain the highest reliability of a satellite during the mission lifetime. Mainly it generates a parts control plan/specification, defines the parts requirements, resolves any problems during procurement, and provides reviews or decisions promptly to prevent any critical schedule delays.

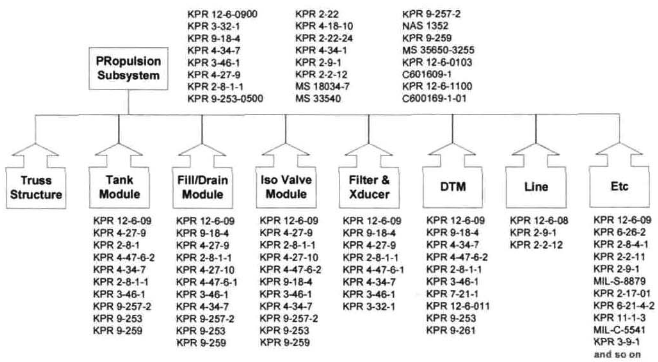


Fig. 9. List of PS Manufacturing Processes

All the used PM&P in the PS satisfy the satellite mission's functional constraints such as resistance to corrosion, radiation, thermal cycling and vacuum with the specified or with sufficient margins. The most important requirement is that all the components of the PS shall be compatible with the hazardous propellant. PM&P engineering also deals with specifying component requirements endowed by system requisites, describing the statement of work for the contract with component suppliers, and following up major engineering activities which are performed by each manufacturer. Component specifications commonly include functional description, physical characteristics, tests methods, acceptance criteria, and documentation requirement, etc. The prepared specifications shall be explicitly agreed between design engineer and manufacturer to avoid any conflicts. Flight-proven and low-cost hardware ensuring minimum program risk is the principal philosophy for the selection of equipment components.

Another critical item in PM&P engineering is the provision control of shelf-life materials just in time not to give a setback in program schedule and budgeting. As the shelf-life is a period of time during which materials can be processed to produce final properties with consistently stable parameters, the chemical products such as adhesive, paint, tape shall be clearly identified with the shelf-life and the date of the beginning of life or the date of manufacture. In addition, quantities which are split from a batch shall be fully traceable to it and bear the same date and life indications. These shelf-life materials shall be stored in a clean area at room temperature of 22 ℃ with a relative humidity of 55% RH unless specified by manufacturer. In case of re-certification of shelf-life period, it shall be achieved by retesting the material to verify that its properties are still within limits taking into account tolerances[7].

All the delivered parts and materials are submitted to a detail incoming inspection with procurement documents prior to using in the PS. In necessary case, an inspection procedure defined the inspections and tests to be carried out, particularly for materials which are known to be variable in their final properties. Any discrepancies or failures shall be detected in this inspection phase otherwise additional cost and time may be required to fix them during assembly phase. Also each supplier verified that all critical materials have been validated before being used in the manufacture of qualification and/or flight products. When any modifications, changes of condition or configuration of application are occurred, the use or the additional validation testing of the materials are determined through MRB (Material Review Board).

Assembly, Integration & Test

The PS is modular-designed for the easiest repair, rework, and replacement while manufacturing build-up. Fabricated parts are assembled into submodules and subsequently into the PS module level. The PS submodules can be divided into DTM (Dual Thruster Module), IVM (Isolation Valve Module), FXM (Filter/Pressure Transducer Module) and FDM (Fill/Drain Valve Module) as shown Fig. 10.

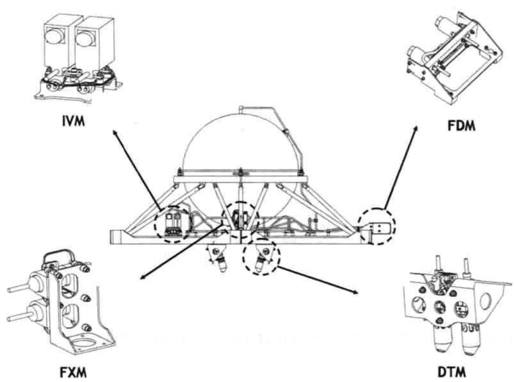
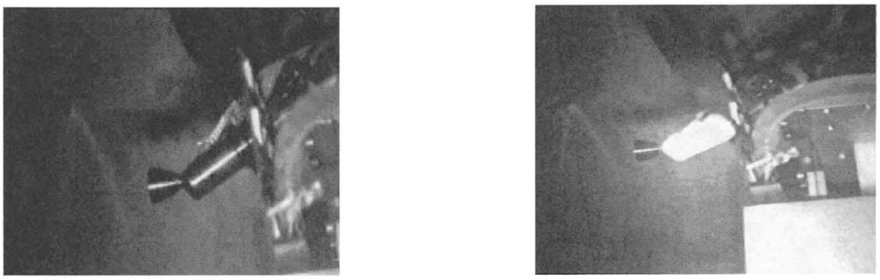


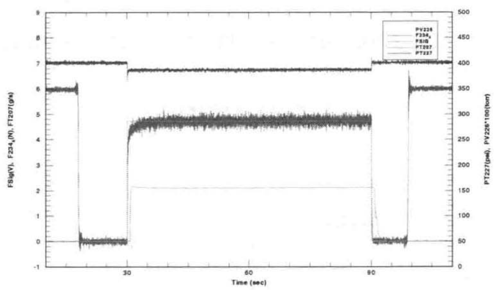
Fig. 10. Configuration of PS Submodules

In case any conflicts are found during integration, engineering orders are promptly issued and FA&I (Fabrication, Assembly & Integration) engineering reflects them directly on the spot with concurrent configuration control accompanied by. Safety is the first priority to be considered in testing of the PS of high pressure and hazardous propellant. It is designed and operated based on MIL-STD-1522A to preclude or limit hazards to personnel and equipment throughout all processes. Tests are performed at the level of parts, equipment components, submodules, and complete the PS module.

A majority of engineering activity in the PS is dedicated to the test and performance evaluation of thrusters. Thruster performance test needs hot-firing test (HFT) facilities and equipments. Simulation of high altitude vacuum environment under which thrusters are functioning is made by vacuum chamber. An acceptance test program is performed on each thruster to ensure the performance characteristic. The acceptance program is comprised of thruster weight measurement, gas flow impedance test, electrical test with solenoid valve, pre-HFT inspection, alignment with test stand, acceptance hot fire test, post-HFT inspection, and re-weighing of thruster, etc. All HFT is conducted under the simulated altitude of 30.5 km minimum (1.103 kPa maximum). Each thrust chamber assembly is hot-fire tested as per the specific test matrix to verify compliance with the performance requirements. Fig. 11 shows a steady state thruster firing heated up to 1000 ℃ at 2758 kPa of propellant injection pressure due to hydrazine decomposition and measured test results of thruster performance.



(a) Thruster Operation during Hot Firing



(b) Measured Results of Thruster Performance

Fig. 11. Steady State Hot Firing Test of Thruster

The integrated PS has undergone a series of acceptance tests which are electrical functional, cleanliness, proof pressure, internal and external leakage, gas flow and polarity to verify workmanship, performance, and functionality. These tests are performed at the subsystem level prior to satellite integration. Functional verification of equipment components is also repeated as far as test configuration is allowed. The final flight model configuration of the PS is presented in Fig. 12.

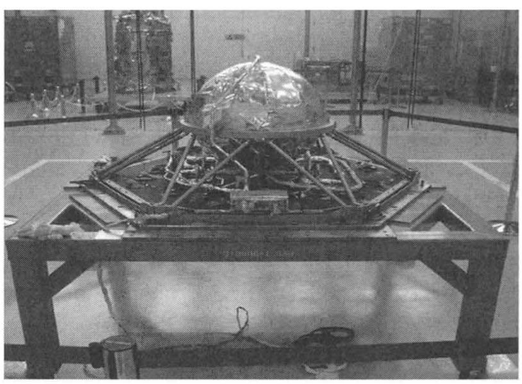
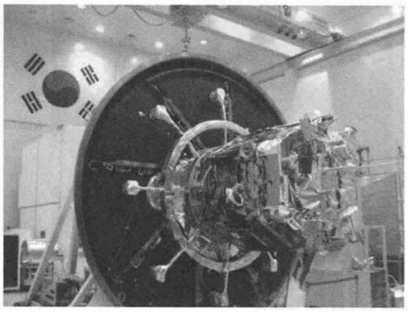
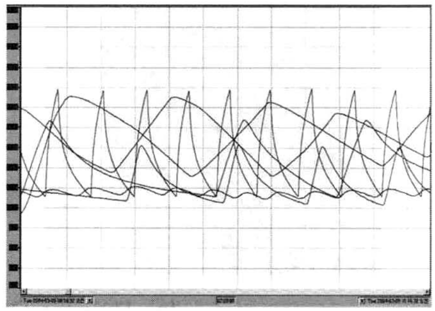


Fig. 12. Final Configuration of PS Assembly

The completed PS assembly has been delivered to KARI and finished the integration with the KOMPSAT bus system successfully. Also, the performance of the PS has been verified perfectly throughout the system level environmental test as shown in Fig. 13.



(a) Satellite on the Thermal Vacuum Chamber



(b) Thermal Behavior Data of PS Heaters

Fig. 13. Thermal Vacuum Test Configuration

Conclusion

In this paper, an overall development process of monopropellant propulsion system is presented which includes the design engineering, analysis, component and PM&P engineering, FA&I, and up to the test and evaluation. The final PS has been successfully integrated, performance-tested, and delivered to satellite system, recently.

From this extensive research experience, the complete development process of the monopropellant propulsion system has been systemized, and finally it is believed that this paper will play an important role as a guideline in the future Korean space program.

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