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Low-Gravity Propellant Gauging System for Accurate Predictions of Spacecraft End-of-Life

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This paper describes a low-gravity propellant gauging system (PGS) developed, tested, and patented at Hughes Aircraft Company. The PGS predicts spacecraft end of life (EOL) within ±2 months or better at midlife for a 15-year nominal mission and is used in the Hughes HS601 geosynchronous communications satellites. The PGS is conceptually simple and strives to minimize hardware required for implementation. It incorporates high-resolution pressure transducers and interconnect valves between existing pressurant and propellant tanks. The PGS employs periodic repressurization of propellant tanks by momentary opening of the interconnect valve between the relatively higher pressure pressurant tank and the propellant tank. Measurements of pressures before and after repressurization allow the determination of gas volume in the propellant tank through gas-law calculations, which in turn yields liquid volume, and hence propellant mass within the tank. The use of the gas law also requires temperature measurements. For accurate temperature measurement, multiple temperature sensors are employed in each tank, and to enhance further the accuracy, extremely accurate and high-resolution pressure transducers are used as gas thermometers. Prototype testing of the proposed measurement scheme has established confidence in the system's ability to predict spacecraft EOL accurately due to propellant depletion. The PGS is failure-tolerant and does not interfere with normal operation of the propulsion subsystem. The repressurization capability offers an additional advantage of closely controlling the oxidizer-to-fuel-mixture ratio in a bipropellant propulsion system. This, in turn, increases the stationkeeping life by minimizing the propellant residuals associated with depletion of one propellant species before the other.

Nomenclature

A	= surface area
c_p	= specific heat at constant pressure
m	= mass
m_{He}	= mass of helium
n	= polytropic gas constant
P	= gas pressure
P_p	= pressurant tank (gas) pressure
P_u	= propellant tank (ullage) pressure
$\dot{Q}_{rad}$	= radiation heat transfer
R_0	= radius of spherical propellant tank
T	= absolute temperature
T_p	= pressurant tank gas temperature
T_u	= propellant tank gas (ullage) temperature
T_W	= wall temperature
T_∞	= environmental temperature
t	= time
v	= specific volume of pressurant gas
V_L	= liquid (propellant) volume
V_p	= pressurant tank volume (unstretched tank)
V_T	= propellant tank volume (unstretched tank)
V_u	= propellant tank ullage volume
Z	= pressurant gas compressibility
α	= thermal diffusivity of pressurant gas
γ_p	= specific heat ratio of pressurant tank gas
γ_u	= specific heat ratio of pressurant gas in propellant tank ullage
ΔP_p	= pressurant tank pressure decrease due to repressurization
ΔP_u	= propellant tank pressure increase due to repressurization
ΔT	= temperature difference ($T_W - T_\infty$)
ϵ	= emissivity
σ	= Stefan-Boltzmann constant
τ	= heat transfer time constant

Subscripts

f	= final (post-repressurization) quantity
i	= initial (pre-repressurization) quantity

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I. Introduction

THE effective management of a communications satellite network requires accurate predictions of remaining spacecraft orbital life. Techniques presently used for low-gravity propellant remaining estimation and end-of-life (EOL) predictions typically rely on bookkeeping of propellant consumption and/or inferences based on propellant tank pressure/temperature measurements. The uncertainty in end-of-mission life projections employing such techniques can be in excess of one year, often far exceeding customer requirements.

Considerable effort has been expended in the field of low-gravity liquid propellant mass gauging for the purpose of accurately predicting spacecraft orbital life. Particular emphasis has been given to the gauging of storable liquid propellants that are employed for stationkeeping and orbital maneuvers by communications, meteorological, and remote sensing spacecraft. Payload performance of geosynchronous communications satellites is becoming increasingly dependent on orbital position maintenance through stationkeeping maneuvers as higher gain, more focused antenna patterns are employed. Such stationkeeping maneuvers utilize onboard propellant to generate impulses to compensate for orbit perturbations caused by solar and lunar gravitation, Earth nonsphericity, and solar radiation pressure. Increasing payload power requirements for applications such as direct broadcasting promote the use of body-stabilized spacecraft platforms, and the demands for improved mission performance have popularized the use of bipropellant systems that integrate transfer orbit and stationkeeping functions.

These spacecraft systems can realize significant economic and logistic benefits from accurate propellant gauging capability. Since such spacecraft systems are becoming increasingly more costly and complex, their efficient management requires more and more accurate propellant mass status information. This information is particularly useful for spacecraft replacement planning and orbit slot management via spacecraft station changes and deorbit maneuvers at the end of mission life. Replacement spacecraft program costs may be incurred one to two years earlier than would be required should accurate propellant mass status information be available. As a result, stringent requirements for end-of-mission life prediction capability of the order of ± 2 months uncertainty are now commonplace.

Presently available techniques for propellant gauging, unfortunately, cannot readily comply with such demanding EOL prediction requirements. This has motivated the development of the propellant gauging system (PGS) discussed in this paper. This method has been developed, tested, and patented at the Hughes Aircraft Company[1] and flown on the Hughes HS601 series communication satellites. While providing accurate measurement capability and maintaining high overall system reliability, the PGS design strives to minimize the use of critical resources such as spacecraft mass, power, and volume.

II. Existing Gauging Methods

Conventional techniques employed for onboard propellant remaining estimation and end-of-mission-life predictions typically include the following: 1) bookkeeping (flow rate integration) of predicted propellant consumption[2] based on thruster ground test data, 2) gas law methods based on measured propellant tank pressure and temperature readings, and 3) propellant hydrostatic head measurement using differential pressure transducers.

Other developmental approaches include flow meters, and ultrasonic, nucleonic, capacitive, and variable-volume methods. Although some such methods have undergone microgravity (KC-135 parabolic trajectory) testing,[3] they can suffer from fluid geometry dependence and sensor intrusiveness. Pressure amplitude and resonant frequency measurement methods using variable-volume mechanisms require compliant seals not readily compatible with propellants such as nitrogen tetroxide for extended mission durations of up to 15 years. Such mechanisms can be complex and may reduce overall system reliability.

High-power communications satellites with payload capabilities in excess of 2 kW typically employ solar tracking solar cell arrays and a body-stabilized spacecraft platform. For such spacecraft, the absence of a liquid-settling accelerational field (such as that present in spin-stabilized spacecraft) precludes the use of hydrostatic propellant pressure head measurement or other techniques relying on knowledge of propellant location. Consequently, bookkeeping and gas law methods are the most frequently used techniques for propellant remaining estimation in a microgravity environment. The bookkeeping method relies on the integration of propellant flow rates through thrusters as determined from system feed pressures, temperatures, and ground test data defining the flow characteristics of propulsion hardware. Continuous subtraction of predicted cumulative propellant throughput from the initial propellant load provides an estimate of the amount of remaining propellant. The gas law method relies on propellant tank pressure and temperature measurements to compute tank ullage volumes based on initially loaded quantities of pressurant gas. These techniques are simple and can be successfully employed for dedicated stationkeeping propulsion systems in which nearly all of the loaded propellant is consumed during stationkeeping life. However, the advent of integrated bipropellant propulsion systems, using a high percentage of propellant for transfer orbit operations, has increased the relative uncertainties of these methods due to the large ullage volumes present at the beginning of stationkeeping operations.

Bookkeeping Method

Integrated systems perform both orbit insertion (apogee firings) and stationkeeping maneuvers, with a majority (about 75 to 90%) of the propellant expended during orbit insertion. This leaves only a small fraction (about 10 to 25%) of the initial propellant load at the beginning of stationkeeping life. At the commencement of stationkeeping life, a relatively large uncertainty in propellant remaining as induced by liquid apogee motors (LAMs) already exists since a large quantity of propellant is consumed by these orbit insertion engines. Even small uncertainties in the bookkeeping of orbit insertion propellant consumption equate to large uncertainties in on-station propellant remaining at the beginning of life (BOL). Hence, the accuracy of the bookkeeping technique is markedly degraded for integrated bipropellant systems. Typically, these errors translate into stationkeeping life uncertainties of as large as ± 1 year at the onset of a nominal 10- to 15-year mission.

Since the bookkeeping technique integrates predicted propellant flow rates over mission life, the errors associated with the propellant remaining estimates also accumulate over mission life. Thus, bookkeeping provides progressively degrading accuracies of remaining mission-life predictions. Near the end of mission life, uncertainties of remaining mission life can exceed 1 to 2 years for a spacecraft life of 10 to 15 years. Customer needs in terms of remaining mission-life prediction accuracy, however, become more acute as mission life progresses.

It should also be noted that attitude control functions of body-stabilized spacecraft require the spacecraft thrusters to consume relatively large amounts of propellant in transient or pulsed-mode operation. This causes further degradation in the accuracy of the bookkeeping method for propellant mass status determination. Flow rate integration, for example, relies on ground test data of flight thrusters to generate estimates of propellant consumption based on telemetered thruster on-times and system parameters such as tank pressures and temperatures. The ground test data alone, however, contribute errors in propellant flow rate knowledge of the order of 1% for steady-state thruster operation and even greater errors for pulsed-mode operation. Since the great majority of the initial propellant load is consumed prior to commencement of on-orbit operations, these flow rate errors are amplified relative to stationkeeping life propellant requirements and contribute to life uncertainties of the order of 10% of mission life.

Bookkeeping accuracy is also affected by the uncertainty in the initially loaded propellant mass. Initial propellant loading uncertainties alone can exceed allowable on-orbit propellant remaining uncertainties. Load cells used for initial propellant loading measurement, for example, can provide accuracies of the order of ± 5 lbm propellant species. These uncertainties alone contribute about ± 2 months to the total stationkeeping life uncertainty of 1 to 2 years.

Gas Law Methods

The characteristics of the integrated bipropellant systems just described also significantly impact the accuracy of gas law methods. Specifically, the large ullage volumes within the propellant tanks during on-station operations undergo relatively small changes over mission life. Consequently, relatively small tank pressure changes occur as propellant is consumed over mission life. In addition, the accuracies achieved by gas law methods are strongly dependent on uncertainties in initial loading conditions, pressurant gas solubility in propellant, and thermal conditions, particularly as related to oxidizer vapor pressure in bipropellant systems. Uncertainties in end-of-mission-life predictions caused by these effects typically exceed those generated by bookkeeping techniques and fall far short of system-level requirements, particularly for spacecraft employing integrated bipropellant propulsion systems.

The PGS discussed in this paper represents a significant improvement over currently used methods such as bookkeeping.

III. PGS Concept

The PGS is conceptually simple and strives to minimize the hardware required for implementation. A functional schematic of the PGS is illustrated in Fig. 1. The system employs two high-resolution pressure transducers (one for each tank) and an interconnect latching valve between the two tanks, which represent existing propellant and pressurant gas (helium) tanks within a spacecraft propulsion system.

To perform a propellant measurement, the propellant tank is repressurized by briefly opening the interconnect valve between the relatively higher-pressure pressurant tank and the propellant tank. The interconnect valve is closed prior to pressure equalization between the pressurant and propellant tanks to allow multiple measurements to be performed over the course of mission life. Measurements of the tanks' pressures and temperatures before and after repressurization allow for the determination of the ullage gas volume within the propellant tank through gas law calculations. This, in turn, allows the calculation of the propellant volume and mass within the propellant tank.

Since helium pressurant mass is conserved during the repressurization process, the ideal gas law is used to describe the pre- and post-repressurization helium masses in both the pressurant and propellant tanks under isothermal conditions. Because $m_{He} \sim PV/T$, the helium mass balance applied to the PGS process gives

$$\left(\frac{P_p V_p}{T_p} + \frac{P_u V_u}{T_u}\right)_i = \left(\frac{P_p V_p}{T_p} + \frac{P_u V_u}{T_u}\right)_f \tag{1}$$

$$\left(\frac{P_p V_p}{T_p}\right)_i - \left(\frac{P_p V_p}{T_p}\right)_f = \left(\frac{P_u V_u}{T_u}\right)_f - \left(\frac{P_u V_u}{T_u}\right)_i \tag{2}$$

For an isothermal process $(T_p)_i = (T_p)_f = T_p$ and $(T_u)_i = (T_u)_f = T_u$, and Eq. (2) becomes

$$\frac{V_p}{T_p}[(P_p)_i - (P_p)_f] = \frac{V_u}{T_u}[(P_u)_f - (P_u)_i] \tag{3}$$

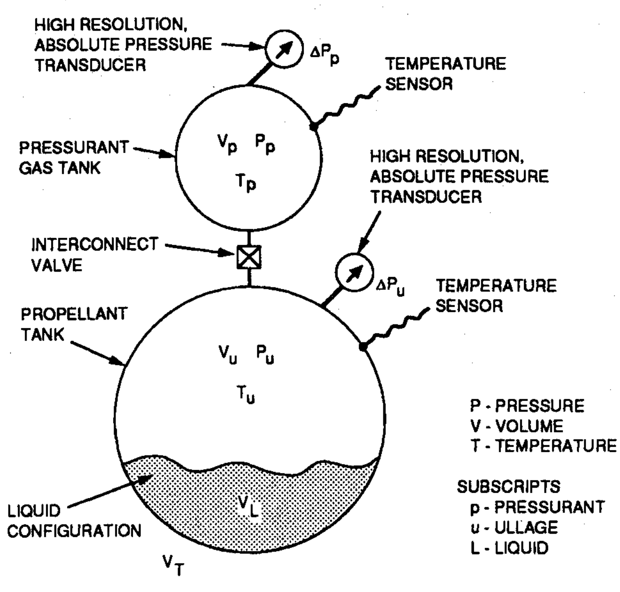

Fig. 1 Propellant gauging system functional schematic.

By denoting $(P_p)_i - (P_p)_f = \Delta P_p$ and $(P_u)_f - (P_u)_i = \Delta P_u$, Eq. (3) becomes

$$\frac{\Delta P_p V_p}{T_p} = \frac{\Delta P_u V_u}{T_u} \tag{4}$$

or

$$V_u = V_p\left(\frac{T_u}{T_p}\right)\left(\frac{\Delta P_p}{\Delta P_u}\right) \tag{5}$$

and finally

$$V_L = V_T - V_u = V_T - V_p\left(\frac{T_u}{T_p}\right)\left(\frac{\Delta P_p}{\Delta P_u}\right) \tag{6}$$

where (T_u/T_p) is the temperature ratio and $(\Delta P_p/\Delta P_u)$ the pressure change ratio. These ratios are measured quantities, whereas tank volumes V_T and V_p (as functions of pressures) are known from ground acceptance test data. Equation (6) thus represents a simple relationship between the liquid volume within the tank and the tank volumes, pressure change ratio, and temperature ratio. This equation is adequate for illustrative purposes, but must be modified to include real system effects such as pressurant gas compressibility, pressurant gas solubility in propellant, tank elasticity, and heat transfer effects. These effects and detailed thermodynamic analysis and modeling techniques are discussed in Sec. IV and VII.

To examine the feasibility of the PGS concept, ground tests have been conducted. The results of these tests have demonstrated the validity of the PGS and have provided confidence in its ability to satisfy spacecraft life-prediction requirements. These test results are discussed in Sec. V, whereas specific spacecraft implementation considerations are discussed in Sec. VI. PGS system modeling is described in Sec. VII, and an error analysis is presented in Sec. VIII.

IV. PGS Thermodynamic Model

The propellant measurement system employs propellant tank ullage volume repressurizations combined with gas law calculations (including real gas and system effects) to generate propellant volume measurements over the course of spacecraft mission life. The gas law calculations are based on conservation of helium mass before and after repressurization, as illustrated in Eq. (2), and are based on an isothermal assumption that manifests itself in the use of identical pressurant and ullage gas temperatures T_p and T_u before and after repressurization, i.e., $(T_p)_i = (T_p)_f = T_p$ and $(T_u)_i = (T_u)_f = T_u$.

In practice, however, the gas temperatures are perturbed by the repressurization process. These temperature perturbations due to the repressurization process can ideally be modeled adiabatically. Applying conservation of helium mass to the thermodynamic system comprised of only the gases within the tanks in this case yields the following result:

$$V_u = V_p\left(\frac{\gamma_u}{\gamma_p}\right)\left(\frac{T_u}{T_p}\right)_i\left(\frac{\Delta P_p}{\Delta P_u}\right) \tag{7}$$

The actual expansion and compression processes can be described by this adiabatic relationship only if the process occurs instantaneously, and only if the gas exchange and temperature uniformity within the gases were achieved instantaneously. Subsequently, as time progresses, the process approaches the asymptotic regime described by the isothermal relationship of Eq. (4). This transient process is thus not completely characterized by either the adiabatic or isothermal process, but must be modeled using the polytropic relationship $Pv^n = \text{const}$, where the exponent n varies from $n = \gamma$ (adiabatic limit) to $n = 1$ (isothermal limit) as time and heat transfer progress.

Since the polytropic model described by $Pv^n = \text{const}$ is transient (time-dependent) in nature, an asymptotic, time-independent (adiabatic or isothermal) model is desirable for actual applications. To enable the selection of the modeling approach, the heat transfer within an appropriate thermodynamic system and between the system and its environment must be examined.

During the propellant gauging process, there is heat transfer due to convection, conduction, and radiation. During repressurization, forced convection takes place due to the penetrating turbulent jet of pressurant within the propellant tanks, and free convection is present during 1-*g* ground testing in both the pressurant and propellant tanks. Conduction and radiation also transfer heat from the environment to pressurant and propellant tanks. Each of these heat transfer modes is examined and their effects on the system are included.

The definition of the thermodynamic system used for modeling purposes is clearly dependent on the modeling approach selected and is constrained by these heat transfer processes. For example, adiabatic modeling of the gauging process with only the gases comprising the thermodynamic system is valid only for times much shorter than the time constants associated with the heat transfer mechanisms present. Unfortunately, the time constants of the convective processes within the tanks during and following repressurization have been estimated and demonstrated to be not long compared to the time required to perform the repressurization process and the time period readily achieved for spacecraft pressure and temperature telemetry updates. Hence, adiabatic modeling is not practical.

If the thermodynamic system just described is modified to include the thermal capacities of the propellant and the tanks, in addition to the gases within the tanks, it can be adiabatically modeled. This is possible because the pressurant and propellant tanks are thermally insulated from their environment in spacecraft applications. Typical spacecraft tank installations include point mounting using low thermal conductivity titanium and also employ radiative insulation through the use of multilayer thermal insulation (MLI) blankets.

The effective emissivity and absorptivity across this MLI are typically about 0.025. Thus, for radiative heat transfer between the tanks and their spacecraft environment, the time constant is of the order of 3 h, as calculated next by assuming representative physical parameters:

$$\dot{Q}_{\text{rad}} = A\,\epsilon\sigma(T_W^4 - T_\infty^4) \tag{8}$$

since $T_W - T_\infty/T_W \ll 1$, Eq. (8) becomes

$$\dot{Q}_{\text{rad}} = A\,\epsilon\sigma[(T_\infty + \Delta T)^4 - T_\infty^4] \approx A\,\epsilon\sigma(4T_\infty^3\Delta T) \tag{9}$$

where $\Delta T = T_w - T_\infty$. Note that

$$\Delta T = \Delta T_i - \frac{\int_0^t \dot{Q}(t)\,\mathrm{d}t}{c_p m} = \Delta T_i - \int_0^t \frac{4A\,\epsilon\sigma T_\infty^3 \Delta T\,\mathrm{d}t}{c_p m} \tag{10}$$

Now by assuming the solution of the type

$$\Delta T = \Delta T_i(e^{-t/\tau}) \tag{11}$$

and

$$\frac{\mathrm{d}}{\mathrm{d}t}\,\Delta T = \Delta T_i \exp\left(\frac{4A\,\epsilon\sigma T_\infty^3}{c_p m}\,t\right) \tag{12}$$

thus,

$$\tau = \left(\frac{4A\,\epsilon\sigma T_\infty^3}{c_p m}\right)^{-1} \tag{13}$$

$$= \left\{\frac{4(3\ \text{m}^2)(0.025)(5.67\times10^{-8}\,W/m^2K^4)(300\text{K})^3}{[0.56(kJ/kg\text{K})(10\ \text{kg})(1000\ J/kJ)}\right\}^{-1}$$

$$\approx 12{,}000\ \text{s} \approx 3\ \text{h} \tag{14}$$

Since this radiative time constant is long relative to the time required for repressurization and the data sampling period, and conduction through the tank point mounting is negligible, an adiabatic model of such a system is plausible. Sufficient time must transpire, however, to allow thermal equilibration of the gases and their containing tanks and/or propellants. This time is dictated by the conduction time constant of the spherical gas volumes, which is estimated to be of the order of 0.1 h as follows:

$$\alpha\tau/R_0^2 \approx 0.02 \tag{15}$$

For $\alpha \approx 1.3\times10^{-5}$ m²/s for helium gas and $R_0 = 0.5$ m, Eq. (15) gives

$$\tau \approx 0.02\,\frac{\alpha}{R_0^2} = \frac{(0.02)(0.5\ \text{m})^2}{1.3\times10^{-5}\ \text{m}^2/\text{s}} = 400\ \text{s} \approx 0.1\ \text{h} \tag{16}$$

This time constant is estimated from the temperature time history of a sphere of radius R_0 with constant initial temperature and constant but different surface temperature.[4]

Alternatively, for times much greater than the radiative heat transfer time constant, the system can be modeled isothermally since the tank system will be in thermal equilibrium with its environment.

Thus, the physical process asymptotically converges to the isothermal model. The adiabatic model, on the other hand, is valid for relatively short times, and consequently the physical process diverges from the model with increasing time. In addition, the value of γ for the gas within the pressurant tank will differ from that of the ullage gas within the propellant tank due to the presence of the propellant vapor phase within the ullage volume. This introduces additional complications into the implementation of an adiabatic model. Hence, the isothermal approach has been adopted for the PGS and will be primarily addressed in the balance of the discussion here. Both the adiabatic and isothermal modeling approaches were examined experimentally, however, and are described in Sec. V.

To apply the isothermal model to the repressurization process, a means of establishing a thermal reference must be used to allow measurement of system pressures at essentially identical thermal conditions prior to and following the repressurization process. Since sufficient time must transpire following repressurization to allow adiabatic expansion and compression effects to subside within the two tanks involved in the measurement process, the constancy of the spacecraft thermal environment is not assured. Fortunately, this problem is solved by the pressure correlation technique as explained next.

Pressure Correlation Technique

The PGS employs high-resolution pressure transducers and temperature sensors for pressure and temperature measurements. The bipropellant propulsion subsystem employs multiple propellant and pressurant tanks. Since repressurization is performed on one propellant tank at a time, the pressure of that particular tank is perturbed. This perturbation is primarily due to the transfer of helium into the propellant tank, but the varying thermal environment of the spacecraft may also cause some temperature-induced pressure changes. The remaining tanks not involved in the measurement process (unperturbed tanks) may also experience small pressure changes—solely due to changes in the spacecraft thermal environment. The unperturbed tanks thus serve effectively as "gas thermometers." The pressures of these tanks change only as do their bulk gas temperature. Temperature/pressure correlations obtained from their high-resolution pressure transducers are very accurate because of the extremely fine resolution and high accuracy of the transducers. Thus, the pressure correlation technique is a method that uses "unperturbed" tanks' pressure measurements as indications of the spacecraft thermal environment.

V. PGS Concept Validation Tests

Ground tests have been conducted to establish the validity of the PGS concept. These tests were performed using flight spare tanks, off-the-shelf tubing and valves, and commercial grade high-resolution pressure transducers, and thermocouples. The test setup schematic is given in Fig. 2. Pertinent hardware details are summarized in Table 1.

An initial series of tank stretch (elasticity) measurements was performed to obtain tank volume as a function of pressure for both the pressurant and propellant tanks. The measured tank stretch values are summarized in Table 1. The first phase of system testing employed only gases (helium and nitrogen) within the tanks. These tests were performed to establish baseline system performance without the presence of liquid within the propellant tank, and to examine the effects of gases with differing thermophysical properties such as density, specific heats, and thermal diffusivity. The results of these tests allowed the assessment of the merits and limitations of the adiabatic and isothermal modeling approaches. The second phase of testing simulated the gauging of a volatile liquid propellant within the propellant tank. Freon was employed for these tests since its vapor pressure characteristics are similar to those of nitrogen tetroxide (N_2O_4), a commonly used oxidizer in bipropellant systems.

The pressurant tank contained helium at pressures in the range of 45 to 265 psia, whereas the propellant tank pressures ranged from 25 to 240 psia. The pressurant and propellant tanks were enclosed in fiberglass insulation to simulate the relatively weak radiative coupling to the spacecraft environment experienced by flight pressurant and propellant tanks. High-resolution quartz-resonant pressure transducers similar in design to those used in flight application (the flight transducers are even more accurate) were employed for tank absolute pressure measurements. The transducers provided as output the frequency of a quartz beam oscillator under tensile stress by the action of a Bourdon tube and achieved accuracies far exceeding those required. (The flight transducer design exhibits much less than 0.1% full-scale error.) These high-resolution transducers allowed for the accurate calculation of small pressure changes from the measured absolute pressures. Each pressure transducer also contained a nonpressure stressed quartz beam used for pressure-measurement temperature compensation and also to simulate reference tank pressure data since no unperturbed reference tanks were used in the test apparatus. Thermocouples bonded to the tank exteriors provided tank wall temperature-measurement capability.

The tanks were pressurized to the pretest levels and allowed to achieve thermal (and where applicable, pressurant solubility) equilibrium. The needle valve connecting the tanks (Fig. 2) was opened until about a 1.0 to 2.0 psid pressure increase was achieved in the propellant tank. This typically required a total flow time in the range of 5 to 10 s. The corresponding pressure decrease in the pressurant tank varied from about 10 to 20 psid. The ratio of these pressurant and propellant tank pressure changes, appearing explicitly in the thermodynamic modeling of the system [Eq. (5)], was calculated and plotted as a function of time after completion of repressurization.

Representative results of the first phase of tests are presented in Fig. 3. The figure illustrates an adiabatic system model plotted together with temperature/pressure change ratio data as functions of propellant tank pressure. The ordinate corresponds to the product of the temperature ratio and pressure change ratio, $(T_u/T_p)(\Delta P_p/\Delta P_u)$, observed in a quasi-steady sense. This adiabatic model considers the heat capacity of the tank/gas system and assumes no heat transfer between the system and its environment. Since none of the tests represented in Fig. 3 included liquid within the propellant tank, the absolute pressure within the propellant tank was representative of the heat capacity of the system and hence appears as the independent variable in the figure. The ordinate corresponds to the product of the temperature ratio and the pressure change ratio.

The quasisteady temperature change ratio $(T_u/T_p)(\Delta P_p/\Delta P_u)$ gradually increased to the "isothermal value" illustrated in Fig. 4. The measured quantity $(T_u/T_p)(\Delta P_p/\Delta P_u)$ for the isothermal limit is simply the ratio V_u/V_p, as can be seen from Eq. (5). For the gas-only case (i.e., zero fill fraction or no liquid in propellant tank), $V_u = V_T$; hence, the measured quantity $(T_u/T_p)(\Delta P_p/\Delta P_u)$ is essentially the propellant tank to pressurant tank volume ratio at corresponding tank pres-

Table 1 PGS test hardware characteristics

Tanks	Pressurant tank	Propellant tank
Heritage	Surveyor program	Syncom IV program
Shape	Cylindrical with hemi-spherical heads	Conispherical
Material	Titanium 6Al-4V	Titanium 6Al-4V
Burst pressure, psia	1200 psia	900 psia
Internal volume, in.^{3}, (unstretched tank)	1,487.58	19,151.4
Tank stretch, in.^{3}/psi	0.02535	0.5783
No. tanks used in test	2	1
Absolute pressure transducer		
Manufacturer	Paroscientific, Model 2300 AT	
Output	Digital	
Range	0–300 psia	
Accuracy	0.01% of full scale	

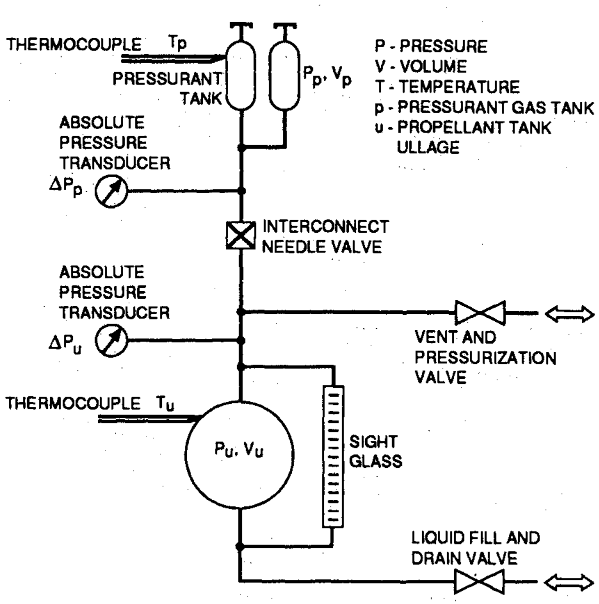

Fig. 2 PGS test setup schematic.

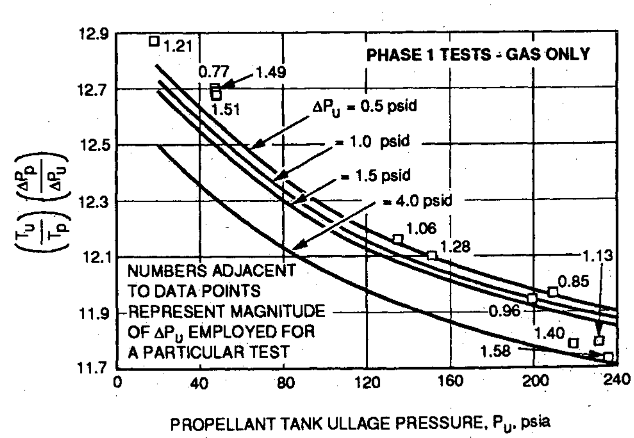

Fig. 3 Temperature-modified pressure change ratio (adiabatic case).

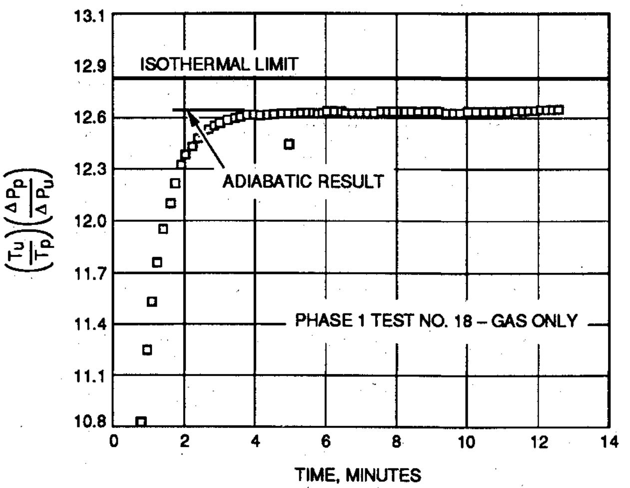

Fig. 4 Transient history of temperature-modified pressure change ratio.

Table 2 PGS test results[a]

Parameter	Test 42[b]	Test 39[c]
Freon fill fraction, %	40.8	0
Initial pressure (before pressurization), psia		
Propellant tank (P_p)	241.03	242.68
Pressurant tank (P_u)	212.71	223.76
Pressure change		
Propellant tank (ΔP_p), psi	−13.8	17.18
Pressurant tank (ΔP_u), psid	+1.8	+1.25
$(V_u/V_p)(T_u/T_p)(\Delta P_p/\Delta P_u)$		
Measured	7.674	13.069
Calculated	7.706	12.96
Error,[d]%	0.4	0.8

[a]Based on isothermal modeling. [b]Gas-liquid test; representative of BOL measurement. [c]Gas-only test; representative of EOL measurement. [d](measured value − calculated value/calculated value) × 100.

sures. The "adiabatic" pressure change ratio in Fig. 4 exhibited continued increase as heat transfer between the system and its environment slowly continued. A period of about 6 h was required to achieve the isothermal value for gas-only tests, whereas shorter times were required to approach the isothermal value for tests employing nonzero liquid fill fractions (phase 2 tests) within the propellant tank. This reduction in equilibration time with increasing liquid fill fraction is a result of the increased thermal mass of the insulated system. This allows the propellant tank gas to approach thermal equilibration with less heat transfer to the external environment.

Results of gas-only and gas-liquid tests obtained during the first phase of testing are summarized in Table 2. The table presents the isothermal pressure change ratio of two tests, one of which employed a fill fraction of about 40.8% Freon within the propellant tank, whereas the other employed only gas within the propellant tank. These isothermal pressure change ratios were obtained as described in Sec. IV, and represent the average of a large number of measured values during a 1- to 2-day monitoring period following the repressurization process. The pressure correlation technique was used to calculate the pressure change ratio each time the reference pressure (temperature) reached its corresponding prerepressurization value. A randomly distributed set of values was thus obtained, with the mean representing the true measured value.

Figure 5 illustrates the normal (Gaussian) distribution of pressure change ratios achieved in test 42 (Freon). The isothermal limit for this test as calculated from known pressurant tank and propellant tank ullage volumes is 7.705. The error in the temperature/pressure change ratio obtained in this test is about 0.4%. Note that a percentage error in the temperature/pressure change ratio of below 0.5% is required to achieve a system-level accuracy of propellant remaining equivalent to ± 2 months of stationkeeping life. Tests using nonzero liquid fill fractions within the propellant tank exhibited greater thermal stability and hence achieved greater accuracy, due to their increased thermal mass. Test 39, for example, employed only gas within the propellant tank and produced a temperature/pressure change ratio measurement error of 0.8%, greater than the 0.4% error achieved by the Freon test described by Fig. 5.

In summary, the ground test results validated the PGS concept and encouraged its implementation in spacecraft propulsion systems. In particular, sufficient measurement accuracies were achieved during ground tests in spite of the presence of more severe adverse thermal effects (as discussed in Sec. VI) than would be encountered in spacecraft applications.

VI. PGS Implementation in Spacecraft

The functional schematic of the concept (Fig. 1) contained a single pressurant tank and a single propellant tank for illustrative purposes. Actual spacecraft propulsion subsystems, however, include multiple pressurant and propellant tanks. In such cases, the propellant volumes within each tank are measured separately. This will require individual repressurization and pressure measurement capability for each propellant tank. Hence, the system will require additional interconnect valves and high-resolution pressure transducers. High-resolution pressure transducers are employed to allow relatively small (0.5 to 2 psid) pressure changes to be accurately measured for the propellant tank, and to also circumvent analog-to-digital conversion quantization errors and inaccuracies associated with conventional telemetry processing techniques.

The PGS is flight proven into the HS601 body-stabilized spacecraft propulsion system. A representative configuration of the PGS within a bipropellant propulsion system is illustrated schematically in Fig. 6. As seen in the figure, there are two pressurant tanks and four propellant tanks. Although Fig. 6 shows only one temperature sensor per tank, each propellant tank employs three temperature sensors and each pressurant tank uses two temperature sensors. Multiple temperature sensors are employed to minimize the errors introduced by tank thermal gradients and to minimize random errors.

Since the propellant volume within each propellant tank will be measured separately by the repressurization of one propellant tank at a time by the pressurant tank, the remaining three propellant tanks will be "unperturbed," or passive, during each measurement process. These three unperturbed tanks constitute thermal environment references to allow pressure changes to be calculated under nearly identical thermal condi-

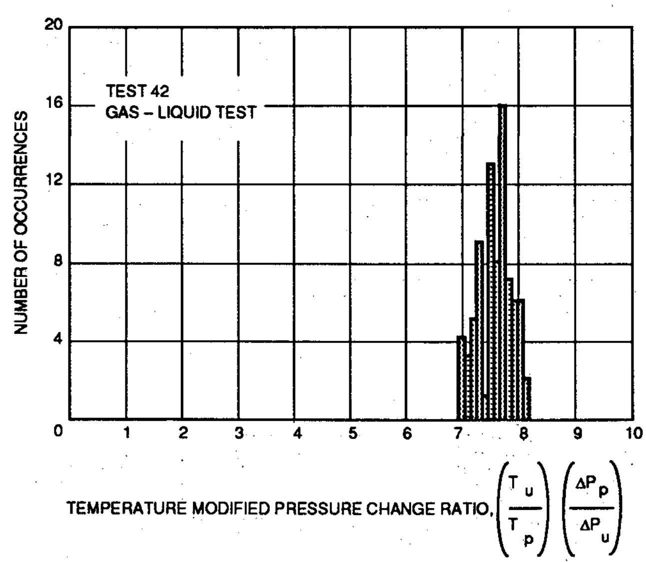

Fig. 5 Pressure change ratio histogram.

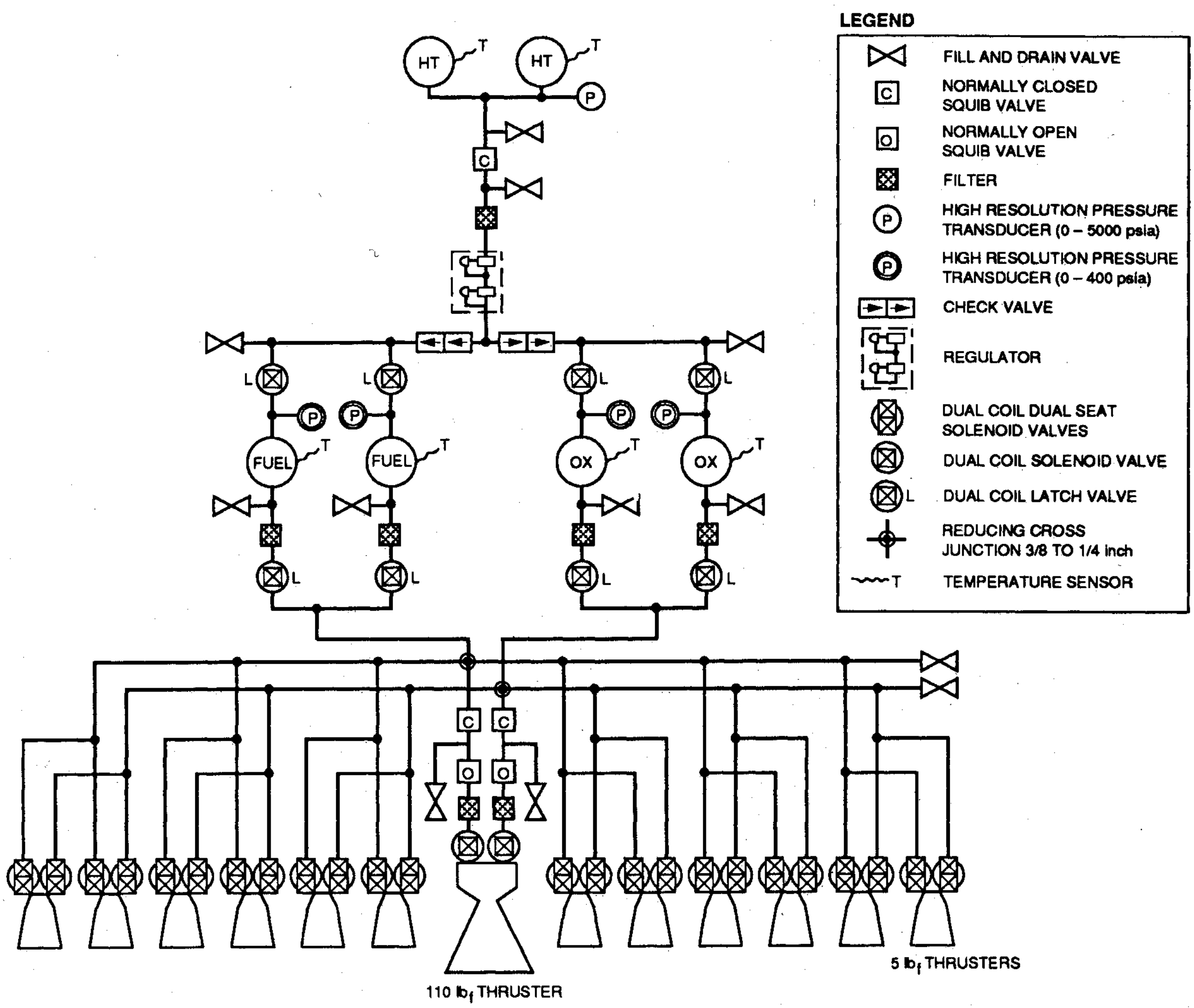

Fig. 6 Integrated bipropellant system with PGS.

tions pre- and postrepressurization. The high-resolution pressure transducers in these tanks act as "gas thermometers" for extremely precise temperature measurement.

Typical regulated/blowdown propulsion systems employ pressurant tanks for pressurant supply during the regulated phase of operation, and permanently isolate the pressurant tanks from the propellant tanks via pyrotechnic squib valves at the onset of blowdown operation once on-station. The proposed gauging scheme, however, utilizes the pressurant tanks during on-station blowdown operation for repressurization of the propellant tanks. The scheme calls for termination of regulation when the pressurant tank pressures have fallen to about 600 psia (at which point the propellant tank pressures are approximately 250 psia), thus maintaining a pressurant reserve in the pressurant tanks to allow periodic (annual) repressurization of the propellant tanks through the regulation gas flow paths as shown in Fig. 6.

The PGS accuracy corresponds to a ± 2 months mission-life uncertainty for a single (annual) pressurization measurement of propellant tanks prior to the end of a 15-year nominal mission. Statistical regression techniques incorporating the yearly propellant mass status data obtained from these measurements will be employed to generate end-of-mission-life projections (and corresponding uncertainties) that will be more accurate than single-point measurements.

VII. PGS System-Level Modeling

The PGS system model includes effects of tank stretch due to pressure, non-ideal gas effects (compressibility), helium gas solubility in liquid propellants, and thermal environment influences on the overall system accuracy. These effects are addressed next.

Tank Stretch

For lightweight, thin-wall tankage employed in spacecraft propulsion systems, volumetric changes with pressure can be significant. The pressure vessels employed in the spacecraft propulsion system are designed to a safety factor of 1.5:1. [Safety factor is the ratio of design burst-to-maximum expected operating pressure (MEOP).] The volumetric changes of the pressurant and propellant tanks during the repressurization process, as well as the tanks' volumes as functions of system pressures, are accounted for in the system modeling. Tank volume changes with temperature are not significant and are hence not considered. Typical values of tank stretch dV/dP, for the pressurant and propellant tanks are given in Table 3. Equation (6) is easily modified as follows to account for tank elasticity:

$$V_L = V_T - \frac{T_u}{T_p}\left[\frac{\Delta P_p}{\Delta P_u} V_p + \frac{\Delta P_p}{\Delta P_u} P_p\left(\frac{dV_p}{dP_p}\right)\right] - P_u\left(\frac{dV_u}{dP_u}\right) \quad (17)$$

where dV_p/dP_p and dV_u/dP_u represent tank elasticity for the pressurant and propellant tank, respectively. The magnitude of the correction due to tank stretch is about $0.9\% V_u$, as compared to Eq. (6) for representative parameter values.

Pressurant Compressibility

Pressurant gas compressibility Z is also accounted for in system modeling. Although the magnitudes of system pressures and pressure changes are relatively small compared to

Table 3 PGS flight hardware characteristics

Tanks	Pressurant tank	Propellant tank
Heritage	Hughes HS 601	Hughes HS 601
Shape	Cylindrical	Spherical
Material	Composite graphite	Titanium 6Al4V
Maximum operating pressure, psia	4200	260
Burst pressure, psia	6300	390
Internal volume (unstretched tank), in.^{3}	2,650 (spec. min.)	22,450 (spec. min.)
Tank stretch, in.^{3}/psi	0.0135	0.88
No. of tanks in propulsion system	2	4 (2 oxidizer, 2 fuel)
Absolute pressure transducer		
Manufacturer	Aeroquartz	
Output	$Vp-p$ square wave 36–40 kHz	
Range	400 psia for propellant tank transducer 5000 psia for pressurant tank transducer	
Accuracy	0.04% of full scale	

the critical pressure of helium, a correction to the idealized isothermal system model is performed with a simplistic assumption that after repressurization, pressure equalization occurs, i.e., both the propellant and pressurant tank reach the same pressure. Then, the helium mass balance before and after repressurization gives

$$\frac{P_{p_i}V_p}{Z_{p_i}T_p}+\frac{P_{u_i}V_u}{Z_{u_i}T_u}=\frac{P_fV_p}{Z_fT_p}+\frac{P_fV_u}{Z_fT_u} \tag{18}$$

where P_f is the equalization pressure following repressurization and Z_f is the compressibility at P_f. Rearranging Eq. (18), we get

$$\begin{aligned} V &= \frac{T_u}{T_p}\left[\frac{(P_f/Z_f)-(P_{p_i}/Z_{p_i})}{(P_{u_i}/Z_{u_i})-(P_f/Z_f)}\right]V_p \\ &= \frac{T_u}{T_p}\left[\frac{P_f-(Z_f/Z_{p_i})P_{p_i}}{(Z_f/Z_{u_i})P_{u_i}-P_f}\right]V_p \end{aligned} \tag{19}$$

Noting that $V_L = V_T - V_u$, we obtain

$$V_L = V_T - \frac{T_u}{T_p}\left[\frac{(Z_f/Z_{p_i})P_{p_i}-P_f)}{P_f-(Z_f/Z_{u_i})P_{u_i}}\right]V_p \tag{20}$$

In practice, pressure equalization is not performed, but this simplification does not introduce measurable errors for the pressures encountered in PGS. The magnitude of this correction is about 0.1% V_u as compared to Eq. (6) for representative parameter values.

Pressurant Solubility

Pressurant solubility within the liquid propellants can decrease the magnitude of the pressure changes in the propellant tanks and hence must be addressed. Since the spacecraft employs helium-saturated propellants at launch and operates in a blowdown mode on-station, with a relatively small decrease in tank pressures from BOL to EOL, the propellants continue to be nearly saturated during mission life. The time constant for helium dissolution (or desaturation) in propellants is also long compared to the measurement process time (days), particularly if we consider the absence of agitation and also the relatively small fill fraction of liquid within the propellant tanks on-station. Thus, the pressurant solubility effect is neglible.

Thermal Effects

The most significant effect on system accuracy is that due to thermal environment variations. Although spacecraft bulk diurnal temperature changes are less than 1 to 3°C, such changes must still be considered. During ground testing, diurnal temperature changes in the range of 2 to 5°C were encountered. These changes during the course of the measurement period will alter the system pressure by amounts not small compared to the changes caused by repressurization, particularly within the propellant tank. Since the magnitude of isothermal pressure change within the propellant tank is typically 1.5 psid, and the propellant tank pressure is of the order of 200 psi, thermal-variation-induced pressure changes are clearly significant. Since high-accuracy and high-resolution temperature measurements on propellant tank walls are not practical and will not correlate directly with gas temperatures, the propellant high-resolution pressure measurements are employed in a pressure correlation technique (discussed earlier in Sec. IV) essential to the accurate determination of isothermal pressures changes. By effectively using the system's remaining three "unperturbed" propellant tanks as gas thermometers, a pre- and postrepressurization datum can be established for the calculation of the pressure changes. For example, following repressurization, when the sum (or weighted sum) of unperturbed tank pressures reaches a prerepressurization value, a pressure change within a perturbed tank can be calculated. This reference sum will fluctuate over the time period of measurement due to thermal environment changes and provide a large number of pressure change calculation opportunities. The average value of pressure change ratio $\Delta P_p/\Delta P_u$ obtained in this manner is employed in system modeling, and the distribution of pressure change ratios calculated in this manner has been shown during testing to be essentially normal (refer to Sec. V). This correlation method minimizes the effect of thermal variations on the calculated pressure changes. The coherence and variance of the normal distribution of calculated pressure change ratios are indicative of the integrity of the thermal environment variation compensation.

Thermally related constraints are adopted during each spacecraft propellant-measurement process to minimize errors due to thermal environmental fluctuations. By performing the series of measurements on each propellant tank once per year during solstice, the spacecraft will provide a thermal environment with a minimum daily rate of change of bulk temperature dT/dt as illustrated in Fig. 7. Since geostationary body-stabilized spacecraft typically reject dissipative payload heat through north- and south-facing radiative surfaces, the bulk spacecraft temperature will reach seasonal maximums at solstices due to the incidence of solar radiation on these radiative surfaces during these periods. The bulk temperature rate of change will hence be at a minimum during solstice. Conversely, the bulk spacecraft temperature will reach seasonal minimums during periods of equinox, but dT/dt will fluctuate significantly during equinox as eclipses occur during these periods. Thus, periods of equinox do not provide a favorable thermal environment for propellant measurements, and so, the PGS measurements should be made during solstice.

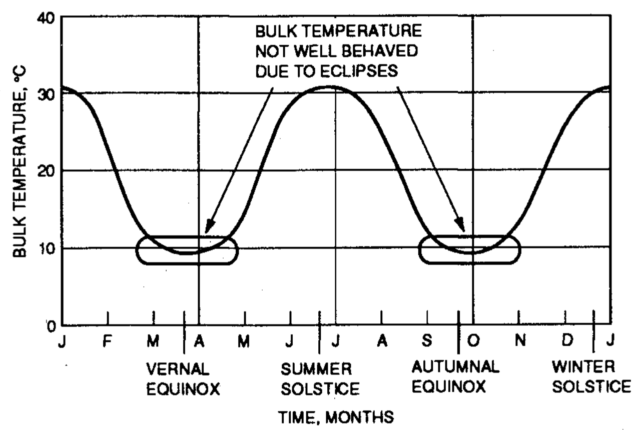

Fig. 7 Typical spacecraft bulk temperature history.

Spacecraft Power Dissipation

Major payload reconfiguration or other power dissipation changes within the spacecraft will also perturb the thermal environment. These changes, however, occur infrequently, and their effects can be observed in reference tank pressure measurements. Appropriate modifications to the reference pressure correlation can be made should such events occur during a measurement sequence. Significant spacecraft bulk temperature shifts over the measurement time period can similarly be compensated for by the pressure correlation technique.

Thruster Firings

No PGS measurement will be made during thruster firings. In other words, propellant usage periods will be avoided since propellant expulsion will perturb measured and reference tank pressures. The measurement process for each propellant tank will require approximately 2 days [long compared to the thermal time constant of the system, which is ~ 3 h per Eq. (14)], and thus a period of about 8 days is required for measurement of all four propellant tank ullage volumes of the system illustrated in Fig. 6. Periods between propellant usages are typically about 2 weeks or longer, as dictated by stationkeeping drift tolerances. Very small propellant usages (e.g., for attitude control) are not significant and may be allowed during the measurement process.

Frequency of Propellant Measurement

The thermal sensitivity of the measurement scheme is driven by the relatively small magnitude of propellant tank pressure increases created by the periodic repressurizations. The magnitudes of these pressure increases are established based on tank design limitations (pressure and volume) and through optimization of life-prediction accuracy as a function of single-measurement accuracy and measurement frequency. The use of a larger repressurization amplitude will increase the accuracy of a given ullage volume measurement, but will necessarily reduce the number of measurements that can be made over mission life for a given pressurant reserve available for gauging. The fewer measurements, in turn, could somewhat degrade the accuracy of life prediction due to statistical regression analysis.

Pressurant Leakage Across Latch Valve

Transfer of some pressurant gas from the pressurant tank to the propellant tank due to leakage across the PGS latch valve (Fig. 6) will not effect measurement acuracy so long as leakage-induced pressure changes across the measurement period (typically 2 days) are small compared to PGS pressure changes. This condition is satisfied for components meeting typical leakage specifications (e.g., 5 scc/h internal leakage for latch valves). Continuous leakage, however, may reduce the number of propellant measurements that can be made over the mission life.

Mixture Ratio Control

An additional benefit offered by the PGS is the increase in mission life that can be attained through the oxidizer/fuel mixture ratio control achievable by selectively represssurizing either the oxidizer or fuel tanks. Accurate propellant-mass-measurement capability allows spacecraft users to correct the mixture ratio during mission life. Hence, near-simultaneous depletion of both the oxidizer and fuel can be targeted to minimize propellant residuals and increase usable propellant, which, in turn, extends mission life and defers spacecraft deorbit. The PGS can be used in this capacity to the extent that a significant overall system mass benefit can be achieved by the incorporation of the PGS hardware into the spacecraft design. Variations from nominal repressurization amplitudes and sequences will be employed as required to alter the system mixture ratio to minimize propellant residuals due to premature single-propellant species depletion. Thus, the mixture ratio control capability allows for more efficient spacecraft utilization and replacement planning.

VIII. Simplified Error Analysis

This section presents a quantitative error analysis of the effects on system accuracy described in the preceding section. All errors are taken to be of the three sigma confidence level based on manufacturers' best estimates.

The overall system requirement of ± 2 month EOL predic-accuracy for the specific spacecraft implementation described in the preceding section corresponds to an amount of propellant equivalent to about $\pm 1.2\%$ of the propellant onboard at BOL. Since the propellant mass at BOL represents an average fill fraction of only about 20%, the overall accuracy requirement in terms of equivalent total propellant tank volume is 0.24% ($1.2\% \times 0.2 = 0.24\%$). Since stationkeeping life is limited by the amount of either oxidizer or fuel remaining in abipropellant system, the root sum square (rss) of measurement errors of two of the four tank (two oxidizer or two fuel tanks) propellant volumes yields the total uncertainty in remaining propellant life (if we assume independent measurement errors). Thus, the "single-tank" allowable propellant-measurement error in this case is 0.24% of total tank volume.

The propellant liquid volume as a fraction of total tank volume, in terms of the measured parameters, can be written by rearranging Eq. (6) as:

$$\frac{V_L}{V_T} = 1 - \frac{V_p}{V_T}\left(\frac{T_u}{T_p}\right)\left(\frac{\Delta P_p}{\Delta P_u}\right) \tag{21}$$

Here, the volume of propellant V_L is the unknown quantity to be calculated and the remaining terms are derived from ground measurements (V_T, V_p) and telemetered onboard measurements (T_u, T_p, ΔP_p, ΔP_u). The propellant tank stretch and pressurant compressibility terms of Eqs. (17) and (20) have been omitted for clarity; their corrections are described in Sec. VII. Errors in their terms contribute second-order system errors and are, hence, neglected, without significantly affecting the results of the error analysis. Quantitative errors associated with each of the terms in Eq. (21) are presented here.

Errors in Tank Volume Measurements (V_T, V_P)

The unit-level tank volume measurement error is typically $\pm 0.1\%$ (three sigma). Tank stretch due to pressure and temperature effects is included in the unit-level data. Errors in compensating for these effects are negligible. Because the errors in propellant and pressurant tank volume measurement are independent of each other, the rss three sigma error in tank volumes is $\pm 0.14\%$.

Errors in Temperature Measurements (T_u, T_p)

The temperature sensor or thermistors unit level three sigma errors are composed of a bias error of $\pm 0.7°$C and random error of $\pm 0.55°$C. The bias is consistent for all sensors, since it is a characteristic of a current source that multiplexes to all of the sensors. Since the temperature ratio in Eq. (21) is close to unity in value, this bias term does not contribute significant error in the calculation of the temperature ratio and can be neglected. The random error represents approximately 0.18% of the typical temperature value of 300 K. The use of two sensors per pressurant tank and three per propellant tank allows for the reduction of this random error to 0.13% (rss) for the pressurant tank temperature reading and 0.10% (rss) for the propellant tank temperature. The overall three sigma error in the measured value of the temperature ratio is then calculated to be $\pm 0.16\%$ (rss).

The temperature sensor outputs are multiplexed onboard to an analog-to-digital (A/D) converter for telemetry purposes. The bias error of the converter is negligible, since the temperature ratio is near unity, while the converter zero bias is similarly of no consequence. The A/D conversion random error and quantitization error (8-bit conversion is used) are negated through the use of multiple (hundreds) readings over the course of the 2-day (per propellant tank) measurement process.

Since the PGS employs isothermal modeling and the measurements of temperature and pressure begin after the tanks are allowed to reach thermal equilibrium with their surroundings, the differences between the tank exterior and the gas bulk temperature will be negligible. The data sampling period will also encompass an integral number (2) of diurnal temperature variation cycles; hence, the use of tank surface temperatures will allow for bulk gas temperature measurements. Finite-element thermal modeling of the spacecraft/tank system is employed to appropriately distribute the temperature sensors over the tank surfaces to accurately represent the mean gas temperature within the tank. Performance of measurements during periods of minimal seasonal thermal gradients also minimizes the effects of temperature distribution errors.

Errors in Pressure Measurements (ΔP_p, ΔP_u)

The high-resolution absolute pressure transducers have a three sigma unit-level error of $\pm 0.042\%$ (full-scale) and hence a sensitivity error of $\pm 0.042\%$ over the range of pressure changes (ΔP_p and ΔP_u). Since the pressure measurements for the pressurant tank and propellant tank are independent of each other, the overall error in pressure measurement is 0.059% (rss). The representative ratio of pressure changes ($\Delta P_p / \Delta P_u$) for the system illustrated in Fig. 6 is 4:1 (the volume ratio of two pressurant tanks to one propellant tank involved in the measurement process is approximately 0.23).

The frequency outputs of the high-resolution pressure transducers are processed by digital counters such that their outputs are converted into 16-bit words. Additional resolution can be obtained by overflowing the counters such that the three most significant bits of an effectively 19-bit conversion are lost. The resulting pressure ambiguity is not significant, since approximate values of pressure (to within 100th and 1000th psia for the propellant and pressurant tanks, respectively, are always known). The bias of the converter is of the order of the least significant bit and is, hence, negligible. The quantitization error of this conversion scheme is an order of magnitude less than the unit-level sensor error, and the use of multiple measurements further negates its significance.

The rss single-tank measurement error in propellant volume can then be expressed using the foregoing three sigma errors and Eq. (19) as

$$\begin{aligned}(V_L)\text{ error} &= \pm\sqrt{(0.14\%)^2 + (0.16\%)^2 + (0.059\%)^2}(V_T)\\ &= \pm 0.22\%\, V_T \qquad (22)\end{aligned}$$

Thus, the single-tank three signma measurement error is within that required ($\pm 0.24\% V_T$) for an equivalent stationkeeping life uncertainty of ± 2 months.

IV. Conclusions

The results of this development effort led to the following conclusions:

1) The PGS predicts spacecraft EOL within ± 2 months at midlife for a 15-year nominal mission of a body-stabilized spacecraft employing an integrated bipropellant propulsion subsystem.

2) Accurate propellant remaining information, together with the mixture ratio control capability of the PGS, allows the spacecraft user to extend useful mission life and deferred spacecraft deorbit through mixture ratio corrections.

3) The PGS does not require considerable spacecraft resources for implementation.

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