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Development and Validation of Fluid/Thermodynamic Models for Spacecraft Propulsion Systems

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Models and simulations have been developed and applied to a wide variety of monopropellant and bipropellant spacecraft applications, including regulated and blowdown systems. These models and analytical techniques have been validated through the use of actual flight and test data from several programs, and a better understanding of the physical processes affecting internal thermodynamics has been achieved. The influence of propellant vapor effects for both regulated and blowdown systems has been assessed and quantified. It has been determined that one relationship can be used with acceptable accuracy for all convective heat-transfer processes, simplifying the modeling and avoiding excessive complexity. In addition, the modeling approach has been validated over a range of accelerations from near zero *g* to one *g*, providing confidence in understanding the differences between ground testing and the flight environment.

Nomenclature

A	=	surface area
C, K	=	engine coefficients
c_p, c_v	=	specific heat at constant pressure, volume
D	=	diameter
Gr	=	Grashof number
h	=	specific enthalpy
h_c	=	convective heat-transfer coefficient
k	=	thermal conductivity
M	=	molecular weight
m	=	mass
Nu	=	Nusselt number
P	=	pressure
Pr	=	Prandtl number
$\dot{Q}$	=	heat-transfer rate
R	=	gas constant
T	=	temperature
u	=	specific internal energy
V	=	gas volume
Y, y	=	weight fraction, mole fraction
μ	=	absolute viscosity
v	=	specific volume
ρ	=	density
σ	=	Stefan–Boltzman constant

Subscripts

a	=	ambient or outside
c	=	chamber
e	=	entering or exiting
f, o	=	fuel, oxidizer
g	=	pressurant gas
t	=	tank
v	=	propellant vapor
w	=	wall

Introduction

FLUID and thermodynamic modeling of propulsion system operation is conducted for performance prediction, sizing of tanks and fluid components, and determination of component flow and temperature requirements. The achievement of realistic simulation models can preclude overly conservative design solutions in these areas. Previous modeling of regulated systems[1] has included internal thermodynamics and heat transfer, regulator characteristics, and flow models. Other references[2,3] also deal with thermodynamic and fluid relationships involving pressure regulated applications. For applications involving propellant tank blowdown, previous studies have addressed thermodynamic modeling[4–6] and the particular influence of oxidizer vapor on blowdown prediction accuracy.[4,6]

The primary goals of this investigation are to improve the modeling capability for a variety of applications and to develop a better understanding of the relevant physical processes. The reporting of the comparison of analytical predictions with actual flight or test data has been limited, and the significance of some of the assumptions regarding physical processes was not always addressed. In addition, applicability of the models over a range of accelerations has not been discussed. The development of simulations in support of a wide variety of propulsion system applications now presents the opportunity to not only validate the accuracy of the models but also search for commonality in modeling methodology. One of the study objectives is to determine if common analytical techniques, such as heat-transfer relationships, can be used for a variety of applications. Another goal is a better understanding of some of the relevant physical processes associated with system operation. The fluid and thermodynamic methods discussed in this paper have been applied to regulated bipropellant systems, blowdown monopropellant systems, blowdown bipropellant systems, and tank pressurizations.

Issues and Methodology

All of the models involve thermodynamics and heat transfer, and the challenges include 1) selecting methods that provide adequate accuracy, 2) identifying and describing all of the relevant physical processes, and 3) avoiding excessive complexity. A good understanding of the physical processes is important; for example, it would not be sensible to overcomplicate a model by including detailed analytical models of processes that are relatively insignificant to the overall results and that are overwhelmed by system uncertainties in other areas. The treatment of the physical processes can be illustrated by expressing the first law of thermodynamics as applied to perhaps the most complicated control volume, the oxidizer tank ullage of a regulated system, pictured schematically in Fig. 1.

$$\frac{\mathrm{d}}{\mathrm{d}t}(m_g u_g + m_v u_v) = \dot{Q} - P_t \dot{V} + \dot{m}_g h_e + \dot{m}_v h_v \tag{1}$$

On the right side of the equation, the terms in order represent convective heat transfer to the ullage, the work done on the propellant as it is expelled from the tank, the energy of entering pressurant gas,

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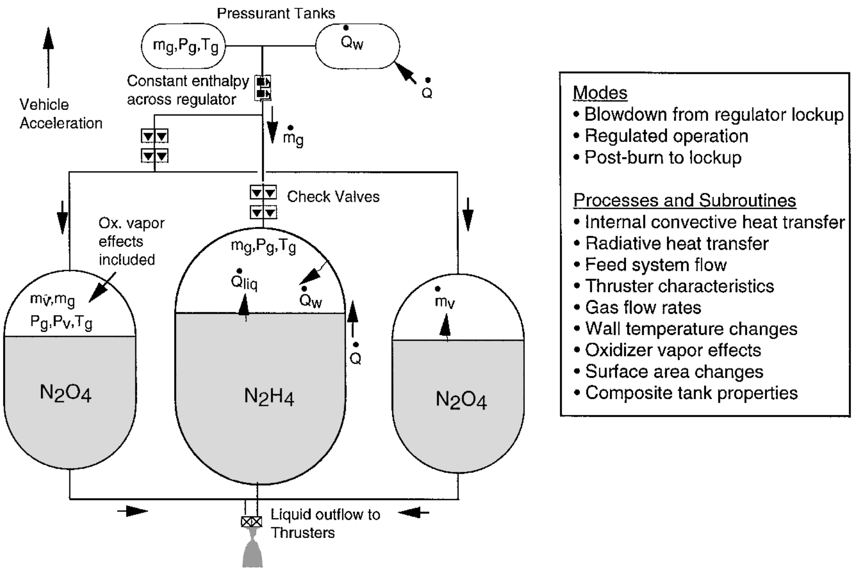

Fig. 1 Regulated system representation.

and the energy of entering vapor. Assuming constant vapor pressure, the vapor mass flow can be written in terms of volume and temperature changes, thus eliminating the vapor mass-flow term. After appropriate manipulation and substitution in the equations, an expression can be written for temperature change:

$$\dot{T} = \frac{\dot{Q} + \dot{m}_g(h_e - c_{vg}T) - \dot{V}P_g}{m_g c_{vg} + m_v c_{vv} + P_v V/T} \tag{2}$$

It is assumed that $u = c_v T$ for both helium and vapor, recognizing that there is greater uncertainty for the vapor. However, the inclusion of vapor effects in the propellant tank ullage does not significantly influence the results, especially for a regulated pressure application where ullage temperature changes are small. This finding was confirmed by running cases with constant vapor mass as well as zero vapor mass in the oxidizer tank. The change in pressurant gas mass for the volume is expressed as

$$\dot{m}_g = \frac{\mathrm{d}}{\mathrm{d}t}\left(\frac{P_g V}{RT}\right) = \frac{V}{RT}\dot{P}_g + \frac{P_g}{RT}\dot{V} - \frac{P_g V}{RT^2}\dot{T} \tag{3}$$

where P_g is assumed to be constant for a regulated system. After substituting Eq. (2), an expression can be written for gas flow into the ullage volume:

$$\dot{m}_g = \frac{(m_g c_{vg} + m_v c_{vv} + P_v V/T)P_g T\dot{V} - P_g V(\dot{Q} - P_g\dot{V})}{(m_g c_{vg} + m_v c_{vv} + P_v V/T)RT^2 + P_g V(h_e - c_{vg}T)} \tag{4}$$

Equation (4) represents the gas flow required to maintain a given regulated pressure in the propellant tank. The derivation for the fuel tank is similar, except that vapor effects are deleted. For the pressurant tank vapor effects and volume changes are deleted. For pressurization of a tank with no propellant outflow, the gas-flow rate into the tank is governed by the flow characteristics of a regulator or orifice and may be sonic or subsonic depending on the pressures. Blowdown systems are treated similarly, except there is no gas inflow to the propellant tank.

The selection of appropriate heat-transfer mechanisms is critical to internal thermodynamic modeling. Heat transfer inside propellant and pressurization tanks is generally characterized as free convection because of the relatively low fluid velocities and mixing intensity, although the flow itself may be laminar or turbulent. Free convection dominates over forced convection if the Grashof number is larger than the square of the Reynolds number,[7] and especially if the ratio of the two values is greater than 10. This relationship was evaluated for several spacecraft applications ranging from orbital burns to ground testing, with the result that the ratio of interest exceeds 10,000 for most of the time. Therefore, the selection of free convection for all propellant and pressurant tank convective heat transfer appears valid and can be expressed as

$$\dot{Q} = h_c A(\Delta T) \tag{5}$$

where ΔT applies to the temperature difference between wall and gas or between liquid and gas. Modeling is based on free convection inside spherical cavities[7] with the following relationships:

$$h_c = kNu/D \tag{6}$$

where

$$Nu = 5.90 \quad \text{for} \quad GrPr < 10^4$$

$$Nu = 0.59(GrPr)^{0.250} \quad \text{for} \quad 10^4 < GrPr < 10^9$$

$$Nu = 0.13(GrPr)^{0.333} \quad \text{for} \quad 10^9 < GrPr < 10^{12}$$

The heat-transfer coefficient is influenced by a number of parameters, including vehicle acceleration and gas properties. The change in the temperature of the wall exposed to the ullage is calculated using the convective heat transfer in conjunction with wall heat capacity. The effects of liquid outflow are accounted for by uncovering new wall area and calculating a mass-averaged wall temperature assuming the added wall area has the same temperature as the propellant. For the pressurant tank at each time increment the convective heat transfer is calculated, and the new gas mass and temperature are used in the determination of pressure with the Beattie–Bridgeman equation of state, which for helium takes the form

$$P = RT/v + a/v^2 + b/v^3 \tag{7}$$

where a and b are functions of temperature.

Radiation heat transfer between a tank and its surroundings is expressed as

$$\dot{Q} = A\varepsilon\sigma\left(T_a^4 - T_w^4\right) \tag{8}$$

In this case, ε is the effective emissivity and takes into account emissivities and surface areas of the tank and its surroundings. The pressurant tank in particular can undergo significant cooling, with the result that tank wall temperature change caused by radiation heat transfer has an impact on internal gas temperature. However, analysis runs using a range of emissivity values show that this effect is small compared to internal convective heat transfer; therefore, the overall contribution to required helium mass is minor.

The throttling across the pressure regulator is treated as a constant-enthalpy process. This assumption was confirmed by a blowdown test using the A2100 (satellite bus used for commercial geosynchronous applications) regulator to characterize the outlet temperature as a function of inlet pressure and temperature. The recorded outlet temperatures were within approximately 2°C of the value predicted by constant enthalpy, with the total temperature range during the blowdown covering about 22°C. One simplifying assumption is that the enthalpy of the gas leaving the pressurant tank is unchanged up to the propellant tank entrance. Factors that could adversely affect this assumption include long pressurization manifolds, low gas-flow rates, and operation of component and line heaters. The increasing use of composite instead of all-metallic tanks potentially adds complexity to the modeling. For composite tanks a mass-average specific heat capacity is determined using the separate masses and specific heats of the liner and overwrap. The entire mass is assumed to be at the same temperature; this assumption has been validated by the results of Cassini ground testing, which shows a very small temperature gradient between liner and overwrap.

Bipropellant systems introduce additional variables involving the oxidizer, which is typically nitrogen tetroxide. The relatively high vapor pressure means that vapor properties must be included in the oxidizer tank model; in addition, for bipropellant blowdown systems the possibility of vapor condensation or evaporation must be evaluated. In the calculation of ullage gas properties, the incorporation of vapor effects is accomplished as follows[8]:

$$c_p = c_{pv} Y_v + c_{pg} Y_g \tag{9}$$

$$\mu = \frac{\mu_v y_v M_v^{0.5} + \mu_g y_g M_g^{0.5}}{y_v M_v^{0.5} + y_g M_g^{0.5}} \tag{10}$$

$$k = k_v y_v + k_g y_g \tag{11}$$

Bipropellant thruster operation can be modeled as

$$P_c = C(\dot{m}_f + \dot{m}_o) \tag{12}$$

$$\dot{m}_f = K_f \sqrt{\rho_f (P_f - P_c)}, \qquad \dot{m}_o = K_o \sqrt{\rho_o (P_o - P_c)} \tag{13}$$

The fuel and oxidizer pressures are at the engine inlet, and the fuel and oxidizer engine coefficients (K_f, K_o) account for flow area and fluid resistance. The preceding engine coefficients can be modeled as functions of flow rate or inlet pressure, depending on the ranges covered.

Fluid/Thermodynamic Modeling and Validation

Analytical models and simulations based on the preceding methodology have been developed for various spacecraft applications. Some of these applications, including comparisons with actual flight or test data, are discussed next. The computer programs use a spreadsheet application with a built-in programming language; a finite difference approach is used, with a time step small enough to ensure accuracy and stability.

Regulated Systems

The computer model accommodates the phases 1) blowdown from regulator lockup, 2) regulated operation, and 3) postburn pressure increase to regulator lockup, at which point the regulator flow goes to zero. Each phase is handled separately for the fuel and oxidizer sides. Thruster flow is handled with an iteration subroutine, which takes into account the fuel and oxidizer tank pressures plus feed system characteristics. In the initial blowdown phase involving liquid outflow and gas expansion, the temperature of the ullage gas decreases; heat transfer takes place between the gas and wall and between the gas and liquid surface. When the pressure in a tank reaches the prescribed value, the regulated phase begins, and the gas-flow rate is given in Eq. (4). In the lockup phase the liquid outflow is zero, and the gas flow into the tank ullage is governed by the regulator characteristics and changing pressures; this phase ends when the check valve reseats.

The regulated system simulation has been used to support the design of bipropellant propulsion systems for Space-Based Infrared System (SBIRS) and the upgraded Athena Orbit Adjust Module. Specific uses have included pressurant tank sizing, determination of component flow rate and temperature requirements, and assessment of engine mixture ratio excursions. The accuracy of the computer model has been evaluated using actual flight data from A2100 and Mars Global Surveyor (MGS). The A2100 system is operationally similar to Fig. 1, except that the pressurization manifold has latching solenoid valves instead of passive check valves. Oxidizer tanks are titanium, and the fuel and pressurant tanks are composite. Figure 2 shows the pressurant tank pressure for a representative orbit-transfer apogee burn involving a single bipropellant engine at approximately 650 N, with a regulated tank pressure of approximately 251 psia. The 82-liter pressurant tank is composite overwrap with titanium liner, and the initial temperature was 18.5°C. Average acceleration during the burn was 0.032 g. Predicted and actual pressures are in close agreement; the predicted isothermal and adiabatic cases, representing the upper and lower bounds, are shown for reference. Using telemetry data to calculate pressurant gas used for the burns, the model prediction has been within 2% of the calculated value for most of the burns evaluated.

Another check of modeling accuracy was obtained using data from the Mars orbit insertion (MOI) burn performed by MGS. The propulsion system, shown in the simplified schematic of Fig. 3, is a regulated system with titanium propellant tanks, a 43.4-liter composite pressurant tank with aluminum liner, and a 655-N bipropellant

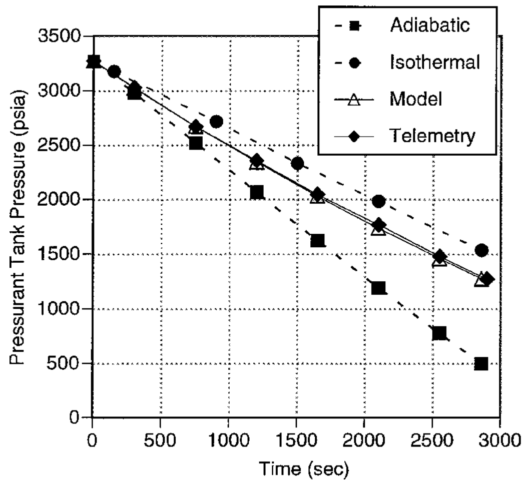

Fig. 2 Simulation of A2100 apogee burn.

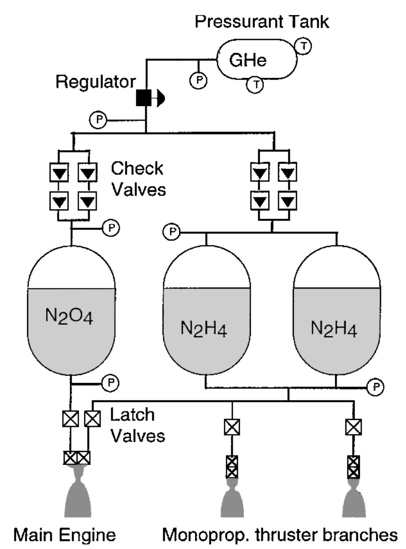

Fig. 3 Simplified MGS operating schematic.

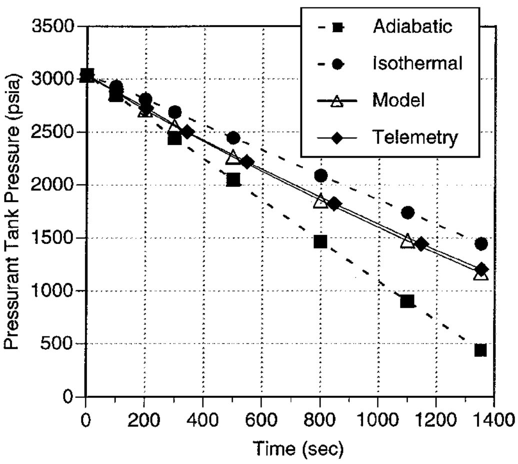

Fig. 4 Simulation of MOI burn.

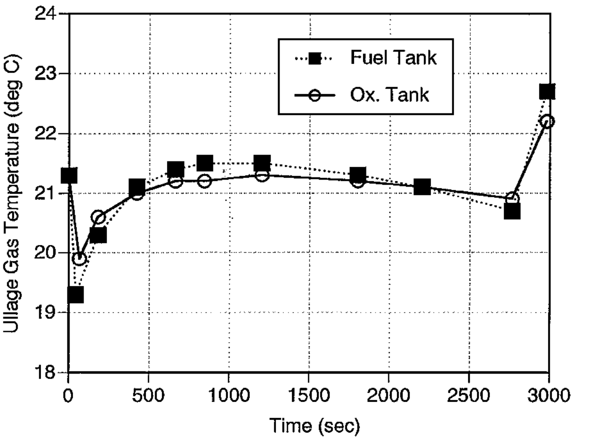

Fig. 6 Temperature prediction for regulated system.

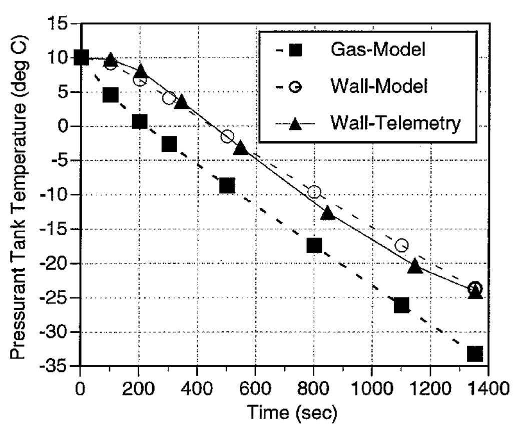

Fig. 5 Temperature comparison—MGS burn.

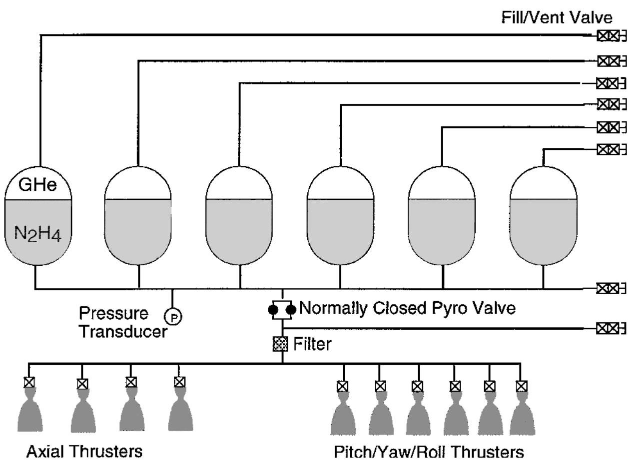

Fig. 7 Athena Orbit Adjust Module schematic.

engine. Characteristics of the MGS configuration, including pressurization/feed system and tank properties, were incorporated in the computer model to simulate the 1352-s MOI burn, which used nearly 282 kg of propellant and involved an average acceleration of 0.075 g. Figure 4 shows that the simulation provided a very close prediction of pressurant tank pressures; the isothermal and adiabatic predictions are shown for reference. Figure 5 shows the model prediction for pressurant tank gas and wall temperatures, as well as the telemetry average for two temperature sensors on the tank wall; consistent with the pressure predictions, the predicted temperatures are in close agreement with actual telemetry values.

One reason for predicting propellant tank ullage temperatures is to assess the potential for postburn fuel/oxidizer pressure differentials after establishment of thermal equilibrium. This effect is a potential contributor to mixture ratio excursions for bipropellant thrusters. Representative temperature predictions for the SBIRS regulated system, operating at an average acceleration of 0.025 g, are shown in Fig. 6; the three phases of initial blowdown, regulated operation (setpoint 270 psia), and regulator lockup (273 psia) are clearly seen. The initial ullage volumes for the fuel and oxidizer tanks are 24 and 38%, respectively. The simulation assumes that vapor pressure in the ullage remains constant at the value consistent with the propellant temperature; this implies sufficient vaporization during the burn to maintain that condition and the entering vapor influences vapor mass as well as heat transfer. Simulations were made for both constant vapor pressure and constant vapor mass to investigate the sensitivity of the assumption, and the results show that the assumption of constant vapor pressure does not have a significant impact on thermodynamics during the burn.

Although the simulations do not predict significant overall temperature changes or differentials between fuel and oxidizer ullage, observations typically show an increase in oxidizer tank pressure in the minutes and hours after substantial burns. This can be attributed to reestablishment of oxidizer tank vapor pressure as a result of vaporization not keeping up as the ullage volume increases during the burn. An analysis of flight data from long burns on A2100, Galileo, and MGS indicates that the vaporization efficiency is roughly 50%. This means that the postburn effects can be estimated by calculating the vapor pressure decay caused by ullage volume changes during the burn (assuming constant vapor mass) and taking half that value as the potential increase in postburn oxidizer tank pressure. Systems operating at significantly different flow rates and burn durations can experience differences in the degree of vaporization. The impact on engine operation is an increased mixture ratio at the start of the next burn, and the duration of this higher than normal mixture ratio will depend upon ullage volumes and component characteristics such as check valve variability. This phenomenon may be important for engines with a very tight operating envelope.

Blowdown Systems

The monopropellant blowdown model accounts for pressure and temperature changes as propellant is expelled, and thruster flow rates are calculated with an iterative subroutine. Free convection heat transfer is assumed in the propellant tank, using the same relationship as for the regulated system model, and the tank wall and propellant surface areas are continually updated in similar fashion.

The accuracy of the simulation has been assessed using data from the monopropellant systems on the Athena launch vehicle and the Lunar Prospector spacecraft. The Athena Orbit Adjust Module system, shown schematically in Fig. 7, involves six titanium/bladder tanks and four 220-N axial thrusters in a blowdown mode for orbit injection. Figure 8 shows the comparison between flight and simulation for the portion of the burn on the Lunar Prospector mission

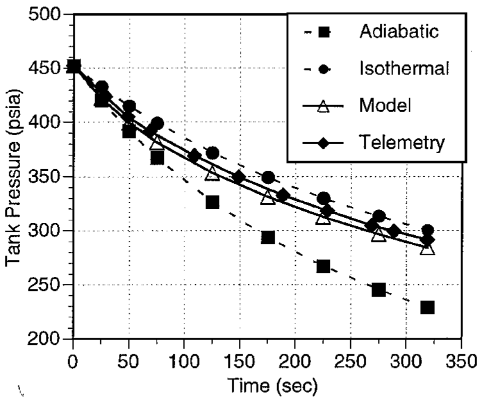

Fig. 8 Simulation of Athena OAM burn.

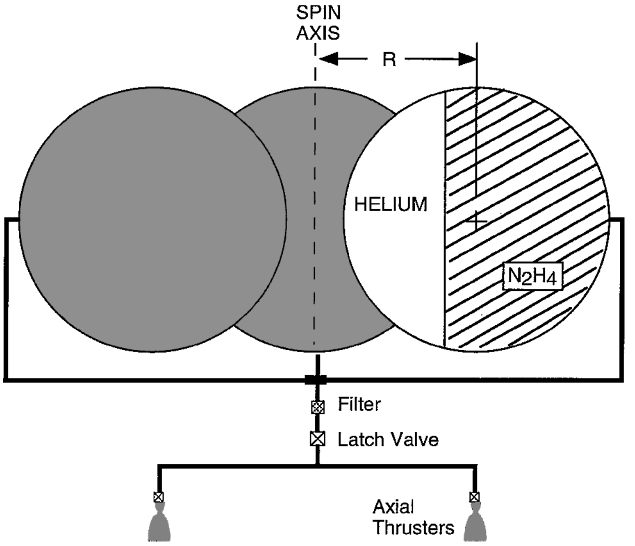

Fig. 9 Lunar Prospector propulsion configuration.

where telemetry was available. The initial temperature was 14°C, and the ullage changed from 46 to 69%; the burn had an average acceleration of 0.035 g. The blowdown process trends toward isothermal as the burn proceeds, and the model predicts pressures that are close to but slightly lower than the actual values. The simulation was also used in a preflight analysis to assess Lunar Prospector spacecraft performance predictability for the lunar orbit insertion (LOI) burns, which were accomplished using two axial 22-N thrusters firing steady state while the spacecraft was spinning at approximately 12 rpm. Figure 9 illustrates the configuration during the burn with a cutaway of one of the three propellant tanks. For convective heat transfer the effects of spinning plus the acceleration caused by thruster operation were taken into account. Postburn evaluation of LOI#1, which had a duration of about 1930 s, showed that predicted tank pressure was within approximately 4 psi of the actual over a blowdown range of 361–214 psia. Because of the low thrust, the process approached isothermal near the end of the burn; the predicted 4.4°C decrease in ullage temperature compares favorably with the calculated actual value of 4.9°C.

The blowdown model for bipropellant systems is more complex and takes into account properties of the helium/vapor mixture. Free convection heat transfer is again used, with thermal conductivity affected by the oxidizer vapor. The simulation model was used extensively in the design and development of the Bus 1 system,[9] which required the capability for long blowdown burns and the development of thrusters with a very wide operating range. The investigation and understanding of the thermodynamic processes was important in demonstrating the viability of this departure from the customary regulated systems. Because there is some uncertainty regarding vapor effects, especially the degree of vaporization or condensation,

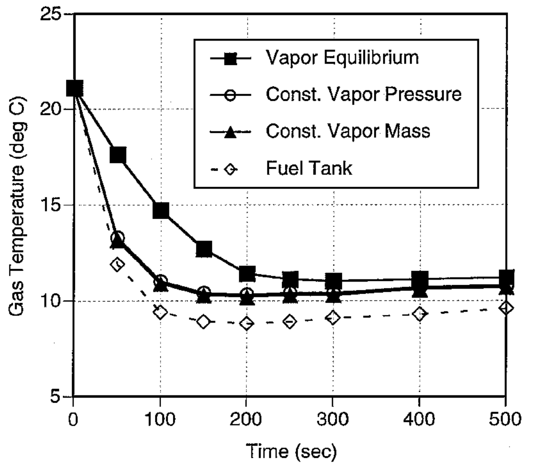

Fig. 10 Effect of different vapor assumptions in oxidizer tank during blowdown burn.

the simulation incorporates three options for dealing with oxidizer vapor: 1) vapor pressure is not allowed to be greater than the saturation value at the ullage temperature, which allows for vapor condensation as required; 2) constant vapor mass, with no condensation or vaporization allowed; and 3) constant vapor pressure, where the vapor pressure is consistent with the propellant temperature at the start of the burn.

The effect of the various vapor assumptions can be seen in Fig. 10, which shows the predicted ullage temperature history for a hypothetical 500-s blowdown burn with a 445-N engine involving an average acceleration of 0.03 g and a 350-200 psia tank pressure change; the fuel tank gas temperature is shown for comparison. The 283-liter titanium propellant tank is pressurized with helium and starts with 20% ullage. In spite of the gas temperature differences, the difference in propellant expended among the three cases at any time during the blowdown is less than 1%, and the difference in oxidizer tank pressure is less than 7.2 psi. The higher temperature of case 1 (vapor equilibrium) is primarily a result of condensation heat release, but this effect is balanced by the reduction in pressure caused by the loss of ullage vapor mass. In general, the higher heat-transfer coefficient in the oxidizer tank caused by incorporation of vapor effects is balanced by the higher value for $c_v m$, resulting in temperature changes that are similar to the fuel tank. The overall conclusion is that the relatively high oxidizer vapor pressure does not adversely affect performance prediction. An important finding is that an exact knowledge of the vapor physical processes is not required because the pressure blowdown is relatively insensitive to the assumption used.

Discussion of Techniques and Parameters

The free convection heat-transfer relationship described in this paper, valid for $10^4 < GrPr < 10^{12}$, is not the only candidate. Another expression,[3] said to be valid for $3 \times 10^8 < GrPr < 5 \times 10^{11}$, is

$$Nu = 0.098(GrPr)^{0.345}$$

It was expected that there would not be significant differences between the two methods because the calculated values for the Nusselt number are relatively close over the range of interest. The preceding alternative relationship was used in the model simulations for MGS, A2100, Cassini ground test, Lunar Prospector, Athena, and the simulated bipropellant blowdown. Results for the two methods were compared, and the differences in the predicted pressures and temperatures were very small. Both relationships provided acceptable accuracy in simulating flight characteristics, and given the minor differences between the two methods, it is not possible to conclude that one is better than the other.

Because the current model assumes use of the same heat-transfer relationship for all surfaces, a relevant question is whether incorporation of additional fidelity and complexity results in any significant increase in modeling accuracy. For example, textbooks provide relationships for both vertical and horizontal surfaces in acceleration fields. In addition, a previous analysis[4] has modeled the heat transfer associated with the thin-film layer between the liquid and gas/vapor ullage. A review of the various free convection relationships reveals that the greatest difference between the methods at any given value of *GrPr* is only about 30%. In addition, the highest heat-transfer values are for the flat (propellant) surfaces, which in a propellant tank are only a portion of the total heat-transfer surface area. In an effort to bound the heat-transfer sensitivity, various model simulation runs were made with the heat-transfer rate increased by 20% at all times; comparisons were then made with baseline runs for MGS, A2100, Athena, Cassini, and bipropellant blowdown. Differences in predicted pressure and temperature were small, and pressurant tank gas usage differed by less than 0.20%. It was concluded that a common heat-transfer relationship can be applied to all surfaces, and there is little benefit to increasing the complexity of the modeling.

Considering the significance of acceleration to the free convection process, an important question is whether ground testing can provide adequate simulation of the flight environment. An assessment of modeling accuracy for the 1-*g* environment was obtained by simulating one of the long regulated burns conducted as part of Cassini ground testing. The pressurant tank is composite overwrapped with aluminum liner and has a volume of 239 liters. Tank and gas-flow characteristics for the Deep Space Maneuver burn involving a single 490-N engine were modeled, and Figs. 11 and 12 show that prediction accuracy is very good for this 1-*g* test. The test was conducted in a vacuum chamber to provide a realistic thermal simulation external to the hardware. The predicted pressurant tank gas and wall temperatures are in close agreement with the measured values at all times during the long pressurant tank blowdown. A comparison was made between model runs at 1 *g* (actual test) and 0.04 *g* (representative of a spacecraft case); the differences in final pressurant tank pressure and gas temperature were only 12 psi and 2.6°C, respectively. For a run at 0 *g* (which might represent a tank repressurization), the differences increase significantly; the parameters were lower by 170 psi and 36.2°C. Next, the MGS model simulation was run for 1 *g* to predict the outcome of a hypothetical ground test. For the same parameters as just stated, the differences were only 25 psi and 6.1°C. These two examples illustrate the conclusions reached: 1) for typical flight accelerations of >0.01 *g*, ground testing in a controlled environment (preferably near vacuum) can provide an adequate simulation of flight thermodynamics, and 2) for lower acceleration events such as a tank repressurization with no thrust, ground testing significantly understates the internal gas temperature changes.

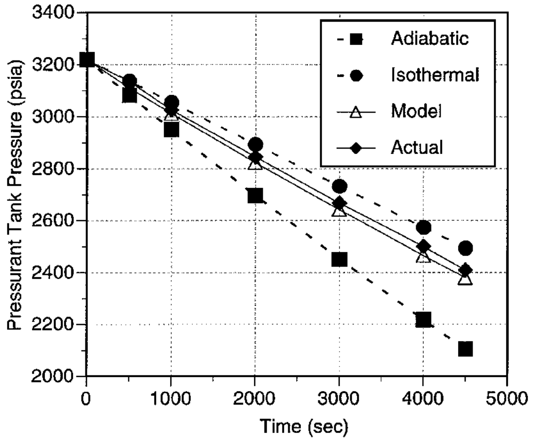

Fig. 11 Simulation of Cassini ground test.

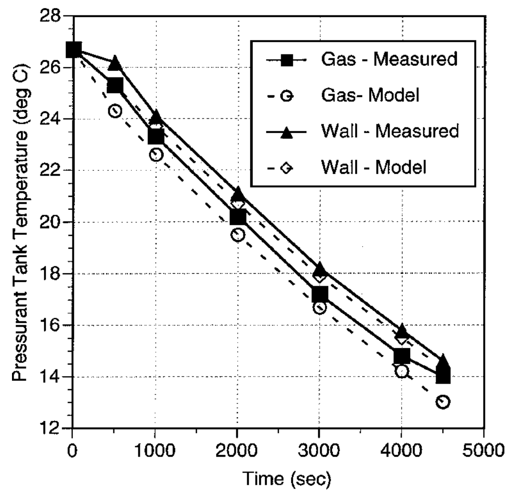

Fig. 12 Pressurant tank temperature comparison for Cassini ground test.

Mission Simulation

This paper is primarily concerned with the realistic modeling of individual propulsion system burns or events. However, many missions involve multiple burns where each period of thruster usage is a small portion of the total. In this case, although it would be feasible to include complex thermodynamic models in an all-up simulation, it is simpler and sufficiently accurate to assume an isothermal process for the mission. These mission simulations are conducted to support propellant budgeting, determine residuals, evaluate uncertainties and dispersions, and define thruster operating envelopes. The simulations incorporate system operating characteristics and thruster flow models to provide an output of propellant remaining and thruster performance as a function of time. Sensitivity studies evaluate the impact of certain variables on propellant residuals; variables include propellant temperature, fuel/oxidizer temperature differentials, inlet pressure differentials, and thruster mixture ratio variability.

An example of an isothermal mission simulation of this type is the model used extensively in the development of the Bus 1 bipropellant blowdown system.[9] In addition to system and thruster characteristics, the model accounts for vapor pressure and the effects of pressurant solubility. The gas saturation level at loading is an input, and the model accounts for expulsion of dissolved gas as well as gas returned to the ullage during the blowdown. Figure 13 shows the thruster operating envelope developed for the Bus 1 application by plotting the results of individual mission simulation cases using a range of system variables, including solubility uncertainties,

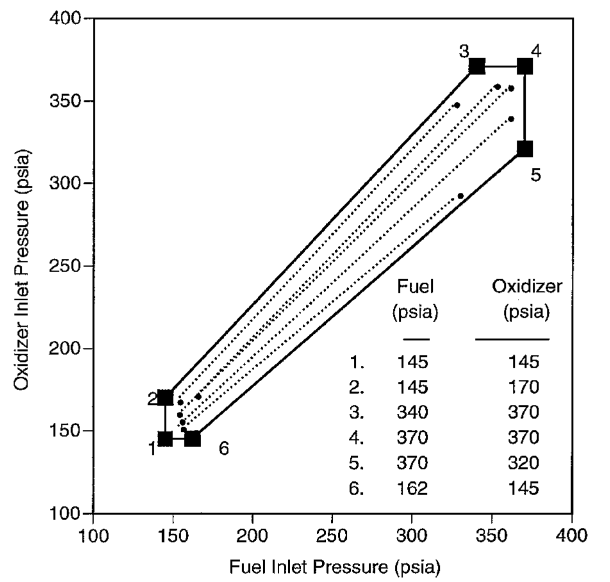

Fig. 13 Thruster operating envelope derived from bipropellant blowdown simulations.

pad-holds, temperature excursions, mixture ratio variability, and the effects of tank repressurization. The dotted lines represent a few of the many cases that were used in constructing the overall envelope. The envelope was drawn to encompass the possible operating conditions and is used as a criteria for thruster selection as well as in defining test requirements. The size of the envelope illustrates the thruster robustness required for blowdown applications.

Conclusions

Analytical models and simulations have been developed and used for a wide variety of monopropellant and bipropellant system applications; the models have been validated through the use of actual flight- and ground-test data from several programs, with both regulated and blowdown systems included in the validation. The goal of demonstrating significant commonality in modeling the various applications has been achieved, and an important finding is that use of a single free convection heat-transfer relationship yields very acceptable accuracy for all applications. Either of the two methods discussed in the paper can be used. In addition, good accuracy has been achieved over a range of accelerations from near-zero g to 1 g, providing confidence in the ability to define differences between ground testing and the flight environment. Investigations have led to a better understanding of physical processes affecting internal gas temperatures, including the significance of propellant vapor in the tank ullage thermodynamics.

For bipropellant systems oxidizer vapor can represent a significant fraction of the ullage gas mass. It has been shown that the assumption of constant vapor pressure is a valid approach from the standpoint of thermodynamics and tank pressure predictions. In fact, for bipropellant blowdown systems an exact knowledge of the vapor physical processes is not required for the accurate prediction of tank pressures during a blowdown burn.

References

[1] Purohit, G. P., "Modeling of the Intelsat VI Bipropellant Propulsion System," AIAA Paper 93-2518, June 1993.

[2] Ricciardi, A., and Pieragostini, E., "Prediction of the Performance and the Thermodynamic Conditions of a Bipropellant Propulsion System During Its Lifetime," AIAA Paper 87-1771, June 1987.

[3] Pasley, G. F., "Optimization of Stored Pressurant Supply for Liquid Propulsion Systems," *Journal of Spacecraft and Rockets*, Vol. 7, No. 12, 1970, pp. 1478–1480.

[4] Estey, P. N., Lewis, D. H., and Connor, M., "Prediction of a Propellant Tank Pressure History Using State Space Methods," *Journal of Spacecraft and Rockets*, Vol. 20, No. 1, 1983, pp. 49–54.

[5] Pasley, G. F., "Prediction of Tank Pressure History in a Blowdown Propellant Feed System," *Journal of Spacecraft and Rockets*, Vol. 9, No. 6, 1972, pp. 473, 474.

[6] Hearn, H. C., "Thermodynamic Considerations in Bipropellant Blowdown Systems," *Journal of Spacecraft and Rockets*, Vol. 21, No. 2, 1984, pp. 219–221.

[7] Kreith, F., *Principles of Heat Transfer*, Intext Educational Publishers, New York, 1973, pp. 402–407.

[8] Ring, E., *Rocket Propellant and Pressurization Systems*, Prentice–Hall, Upper Saddle River, NJ, 1964, pp. 223–225.

[9] Hearn, H. C., "Design and Development of a Large Bipropellant Blowdown Propulsion System," *Journal of Propulsion and Power*, Vol. 11, No. 5, 1995, pp. 986–991.