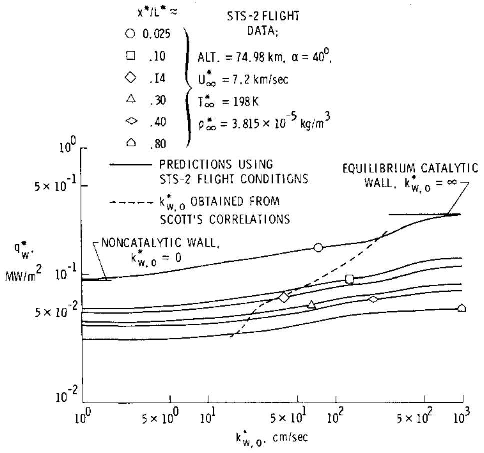

Fig. 2 Surface heat transfer as a function of oxygen recombination rate at various locations along the Orbiter windward centerline.

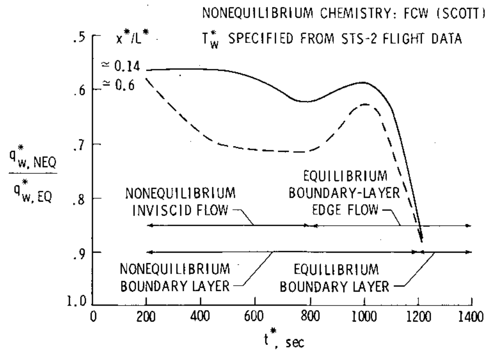

Fig. 3 Time history of heating reduction due to nonequilibrium chemistry at various locations along the Orbiter windward centerline for STS-2 flight.

comparable with the flight data for all locations along the Orbiter windward centerline for a given altitude and the values differ from altitude to altitude. The recombination rate $k^*_{w,O}$ obtained from the Scott's correlations, is also shown in Fig. 2.

At the lower altitude of 48 km (as shown in Fig. 3d of Ref. 7), q^*_w becomes relatively insensitive to the changes in $k^*_{w,O}$ in the range of 80-200 cm/s. For this altitude the flight data suggest the surface heating to be closer to the equilibrium predictions (with $k^*_{w,O} = \infty$). There does not appear to be any reduction in heating due to the nonequilibrium chemistry at 48-km altitude. A maximum reduction of 49% in heating due to the nonequilibrium chemistry is obtained at an altitude of about 75 km in the nose region ($x^*/L^* \approx 0.025$) when the flight data are compared against the equilibrium value. At other body locations along the windward centerline, as well as at higher and lower altitudes, the reduction in heating due to the nonequilibrium chemistry is much less.

Time History of Boundary-Layer Edge Quantities and Nonequilibrium Heating

The nonequilibrium effects at the boundary-layer edge appear to become less significant after the entry time of about 800 s (or below an altitude of 65 km) as shown in Fig. 9 of Ref. 7. The tangential velocity U^*_e, temperature T^*_e, and the constituent air species $(C^*_i)_e$ at the boundary-layer edge are predicted reasonably close from the equilibrium as well as nonequilibrium chemistry after this time. The boundary-layer flow, however, is still in chemical nonequilibrium as suggested by the time history of wall heat-transfer ratio, $q^*_{w,\mathrm{NEQ}}/q^*_{w,\mathrm{EQ}}$, of Fig. 3. Figure 3 also indicates that beyond 1200 s (or below 50 km), both the boundary-layer edge quantities and the boundary-layer flow appear to approach the equilibrium chemistry very rapidly. It may be mentioned here that the results of Fig. 3 are based on Scott's surface recombination rates[8] and may require some modification when more appropriate values of the recombination rates are available.

References

[1] Goulard, R., "On Catalytic Recombination Rates in Hypersonic Stagnation Heat Transfer," *Jet Propulsion*, Vol. 28, Nov. 1958, pp. 737-745.

[2] Rakich, J.V., Steward, D.A., and Lanfranco, M.J., "Results of a Flight Experiment on the Catalytic Efficiency of the Space Shuttle Heat Shield," AIAA Paper 82-0944, June 1982.

[3] Shinn, J.L., Moss, J.N., and Simmonds, A.L., "Viscous-Shock-Layer Heating Analysis for the Shuttle Windward Plane with Surface Finite Catalytic Recombination Rates," AIAA Paper 82-0842, June 1982.

[4] Zoby, E.V., "Approximate Heating Analysis for the Windward-Symmetry Plane of Shuttle-Like Bodies at Large Angle of Attack," *Progress in Astronautics and Aeronautics: Thermophysics of Atmospheric Entry*, Vol. 82, edited by T.E. Horton, AIAA, New York, 1982, pp. 229-247.

[5] Moss, J.N., "Reacting Viscous-Shock-Layer Solutions with Multicomponent Diffusion and Mass Injection," NASA TR R-411, 1974.

[6] Davis, R.T., "Numerical Solution of the Hypersonic Viscous Shock-Layer Equations," *AIAA Journal*, Vol. 8, May 1970, pp. 843-851.

[7] Gupta, R.N., Moss, J.N., Simmonds, A.L., Shinn, J.L., and Zoby, E.V., "Space Shuttle Heating Analysis with Variation in Angle of Attack and Surface Condition," AIAA Paper 83-0486, Jan. 1983.

[8] Scott, C.D., "Space Shuttle Laminar Heating with Finite Rate Catalytic Recombination," *Progress in Astronautics and Aeronautics: Thermophysics of Atmospheric Entry*, Vol. 82, edited by T.E. Horton, AIAA, New York, 1982, pp. 273-289.

Thermodynamic Considerations in Bipropellant Blowdown Systems

H. C. Hearn*
Lockheed Missiles & Space Co., Inc.
Sunnyvale, California

Nomenclature

A = surface area
c = specific heat at constant volume
D = propellant tank diameter
g = gravity constant
Gr = Grashof number
h = heat transfer coefficient
J = Joule proportionality factor
k = thermal conductivity
m = mass
Nu = Nusselt number
P = total tank pressure
Pr = Prandtl number
$\dot{Q}$ = heat transfer rate
R = gas constant
Re = Reynolds number
T = temperature
V = ullage volume

Received March 21, 1983. Copyright © American Institute of Aeronautics and Astronautics, Inc., 1983. All rights reserved.

*Research Specialist. Member AIAA.

Subscripts

c = condensation
g = gas
l = liquid
v = vapor
w = tank wall

Introduction

THE feasibility of bipropellant blowdown propulsion systems has been demonstrated,[1,2] and it is probable that this concept will see increasing use in spacecraft applications. Since accurate impulse delivery may be a requirement in some cases, it is necessary to understand the thermodynamic aspects of the blowdown and the effect on engine inlet pressure. The treatment of heat transfer and pressure blowdown for the fuel tank (using monomethylhydrazine) is straightforward and similar to that used for existing mono-propellant systems. However, the knowledge of oxidizer tank (using nitrogen tetroxide) effects is less defined due to uncertainties involving the relatively high oxidizer vapor pressure. For example, at 70F (294 K) the vapor pressure is 14.6 psia (101 kPa) and the vapor represents over 60% of the total ullage gas mass for a tank pressurized to 200 psia (1379 kPa) with helium.

A recent article dealt with blowdown pressure for a pump-fed system,[3] and this paper will address uncertainties in the thermodynamic aspects of a pure blowdown system. One area of concern regards the heat transfer characteristics of the helium/vapor mixture; the presence of the relatively large quantity of oxidizer vapor will influence the heat transfer rates from the tank wall and propellant surface to the ullage gas. Another concern is the uncertainty in the mass of vapor during the blowdown. As the ullage volume increases, there is the potential for evaporation at the liquid surface to maintain a vapor pressure consistent with the temperature of the liquid film. However, expansion cooling of the gas changes the saturation vapor pressure of the ullage such that condensation may actually occur, reducing the total ullage vapor mass during the blowdown. In this case the partial pressure of the vapor immediately above the liquid surface may differ significantly from the bulk gas partial pressure.[4] The heat of condensation released in the ullage will then influence the gas temperature and pressure.

Discussion

A computer program was developed to model a bipropellant blowdown system employing tanks which contain propellant in contact with helium pressurant. The system, including heat and mass transfer mechanisms, is illustrated in Fig. 1. Performance and flow characteristics of a 100 lb_f (445 N) thruster have been incorporated for the purpose of simulating blowdowns. Input variables include propellant tank volume and mass, initial propellant mass, temperature, pressure, and acceleration. The output includes propellant remaining, helium and vapor pressures, chamber pressure, mixture ratio, vapor mass, wall and gas temperature, and heat transfer rates.

The heat transfer from the tank wall and propellant surface is based on free convection, since the value of Gr/Re^2 is very large[5] (>1000). The appropriate thermodynamic equations are:

$$\dot{Q} = hA(\Delta T) \tag{1}$$

$$h = kNu/D \tag{2}$$

$$Nu = 5.90 \text{ for } GrPr < 10^4 \tag{3}$$

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

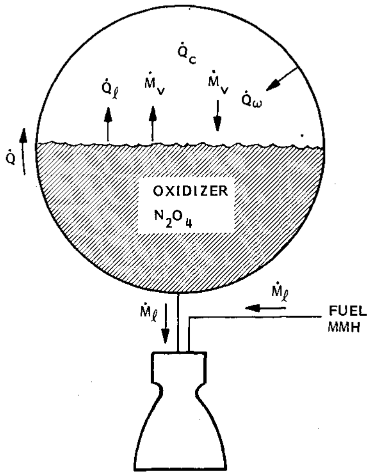

Fig. 1 Bipropellant blowdown schematic.

$$Nu = 0.13\,(GrPr)^{0.333} \text{ for } 10^9 < GrPr < 10^{12} \tag{5}$$

$$\dot{Q}_l + \dot{Q}_w + \dot{Q}_c + \dot{m}_v R_v T_g - P\dot{V}/J = (m_g c_g + m_v c_v)\,\dot{T}_g \tag{6}$$

The determination of the Nusselt number value is based on free convection inside spherical cavities of diameter D.[5] The program continuously computes the liquid surface area, wall area exposed to gas, and gas properties and heat transfer coefficients of the helium/vapor mixture. The liquid temperature is assumed to be constant and the wall temperature is based on heat loss to the gas, heat transfer along the wall, and the addition of wall area uncovered by the expelled propellant.

Since there is some uncertainty regarding the actual degree of evaporation or condensation, the computer program was written so that a particular blowdown application could be evaluated using each of three different cases.

A. Vapor Equilibrium

The vapor pressure is not allowed to be greater than the saturation value corresponding to the temperature of the ullage gas; this allows for vapor condensation as required to maintain this equilibrium.

B. Supersaturation

Vapor condensation is not allowed to occur even though the vapor pressure during the blowdown may exceed the saturation value; this case is conceivable if the number of available condensation sites is very low.

C. Constant Vapor Pressure

This represents the maximum evaporation case whereby enough vapor is released into the ullage to maintain a constant vapor pressure consistent with the temperature of the propellant.

A number of specific blowdowns were evaluated, but representative results are presented for one with titanium fuel and oxidizer tanks initially 80% full and starting at 350 psia (2413 kPa). The total tank volume is 10 ft^3 (0.28 m^3), and the acceleration assumed is 0.01 g. Figure 2 shows the tank pressure blowdown, and Fig. 3 shows the vapor pressure history in the oxidizer tank for a 550-s burn to a tank pressure of approximately 200 psia (1379 kPa). In Case 1, ullage gas expansion cooling is sufficient in the first 275 s to result in vapor condensation; therefore, the saturation curve is followed until the actual vapor pressure is less than the saturation pressure at the temperature of the gas. The heat transfer characteristics of Case 1 are shown in Fig. 4; heat transfer from the liquid surface and tank wall to the ullage gas

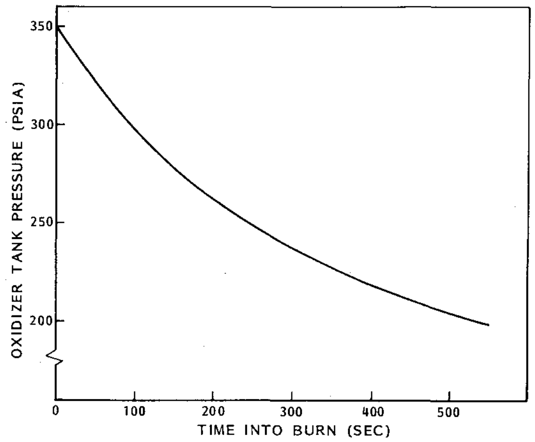

Fig. 2 Tank pressure blowdown.

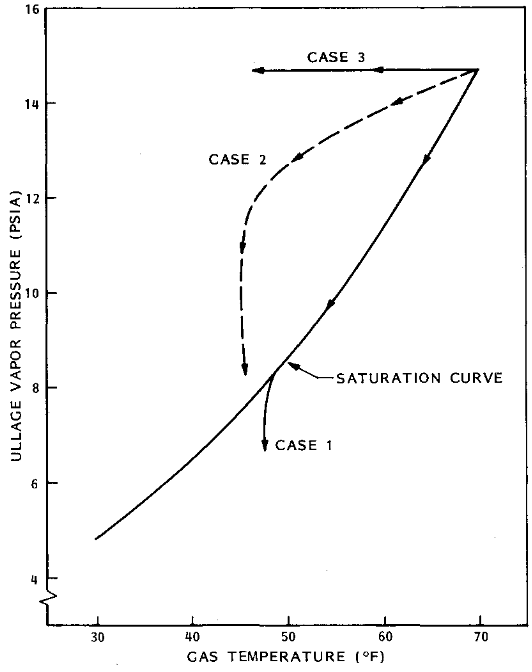

Fig. 3 Oxidizer tank vapor pressure.

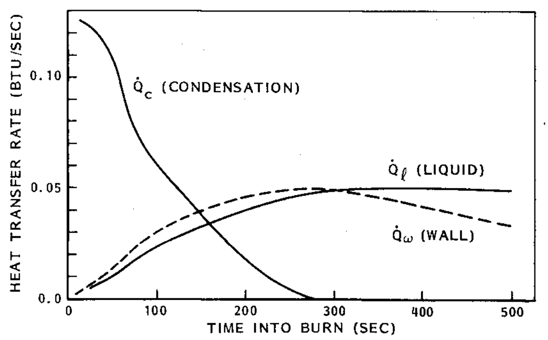

Fig. 4 Heat transfer for Case 1 blowdown.

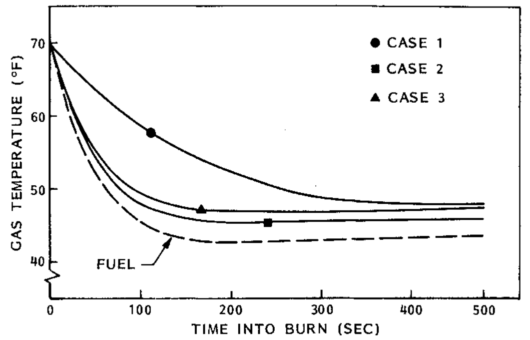

Fig. 5 Gas temperature during blowdown.

is overshadowed by vapor condensation heat release until the rate of condensation decreases later in the blowdown. In Case 2, condensation is not allowed, therefore, the vapor pressure decay shown on Fig. 3 is determined by changes is ullage volume and temperature. In Case 3, constant vapor pressure is assumed for the entire blowdown. Figure 5 displays the gas temperature history for the three oxidizer tank cases plus the fuel tank. The differences are minor, with the exception of the Case 1 oxidizer tank, where the effects of condensation maintain the temperature at a higher level.

In spite of the predicted 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 total pressure is less than 6 psi (41 kPa). The reason the Case 1 pressure is not significantly different is that the heat released by condensation is nearly balanced by the reduction in pressure caused by the loss of ullage vapor mass. The differences between fuel and oxidizer tanks are also not significant, and the maximum mixture ratio change during the blowdown is only 0.04. The primary reason for this is that, although the oxidizer tank gas heat transfer coefficient is about 25% larger than the fuel due to vapor effects, the higher value of $c_v m$ results in gas temperature changes which are very similar. The potential does exist for greater differences between the fuel and oxidizer tanks for situations where the blowdown is very rapid or where the oxidizer vapor pressure is a larger percentage of the total pressure.

Conclusions

For bipropellant blowdown systems operating in a pressure range compatible with existing thrusters, the presence of the relatively high oxidizer vapor pressure should not have an adverse effect on predicted performance and delivered impulse. Significantly, an exact knowledge of the actual physical processes (i.e., evaporation or condensation) is not required since the pressure blowdown during an engine firing is relatively insensitive to the assumptions regarding vapor mass.

References

[1] Hearn, H. C., "Feasibility of Simple Bipropellant Blowdown Systems," *Journal of Spacecraft and Rockets,* Vol. 17, March 1980, pp. 157-158.

[2] Hearn, H. C., "Evaluation of Bipropellant Pressurization Concepts for Spacecraft," *Journal of Spacecraft and Rockets,* Vol. 19, July 1982, pp. 320-325.

[3] Estey, P. N., Lewis, D. H. and Conner, M. "Prediction of a Propellant Tank Pressure History Using State Space Methods," *Journal of Spacecraft and Rockets,* Vol. 20, Jan.-Feb. 1983, pp. 49-54.

[4] Ring, E., *Rocket Propellant and Pressurization Systems,* Prentice-Hall, Englewood Cliffs, N.J., 1964, pp. 223-225.

[5] Kreith, F., *Principles of Heat Transfer,* Intext Educational Publishers, New York, N.Y., 1973, pp. 402-407.