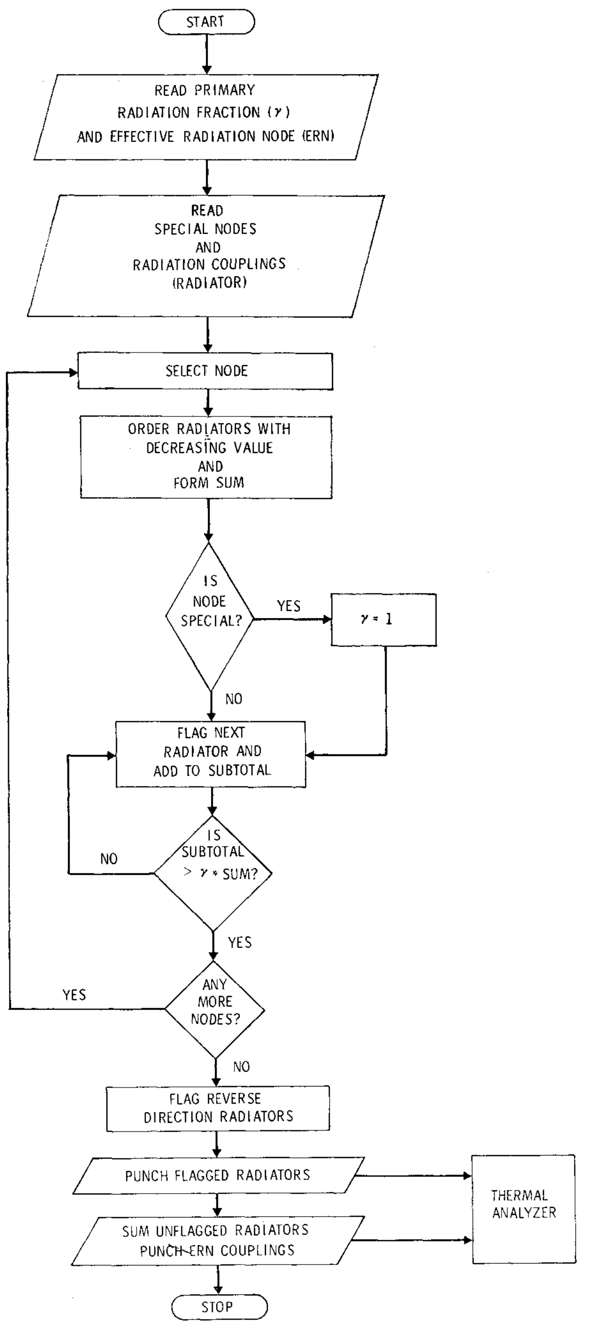

Fig. 1 Simplified flow diagram for the RC program.

rounding coldplates. These nodes were defined as RC special nodes and their couplings were left intact. Application of the RC program reduced the number of enclosure radiators from 2450 to 620, with a γ of 0.7. Comparative thermal analysis showed that the maximum temperature error in using the ERN node was 0.5°F. Similar results have been obtained for other network enclosures including the Skylab cluster vehicle compartments.

For a constant γ, experience has shown that the greatest percentage reduction in radiators is for large enclosures (greater than 75 nodes) with significant shadowing and low emittance surfaces. The smallest radiator reduction is for enclosures with a low number of nodes (less than 30) with symmetrical geometry. A γ value of 0.7 has been found to be a good compromise value that results in significant reduction in couplings with acceptable error.

The ERN technique is general and can be used to reduce the number of radiative paths required to model typical spacecraft enclosures. The percentage reduction in enclosure radiators and subsequent network error is controlled by the analyst selection of a γ value consistent with known accuracy of problem parameters (enclosure geometry, surface properties).

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Optimization of Stored Pressurant Supply for Liquid Propulsion Systems

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Nomenclature

A = surface area
C = specific heat
c_v = specific heat, constant volume
g = acceleration of gravity
Gr = Grashof number
H,h = heat-transfer coefficient and enthalpy, respectively
M,m = tank and fluid masses, respectively
Nu = Nusselt number
p = pressure
Pr = Prandtl number
Q = heat content
R = pressurant gas constant
ρ = fluid density
T,t = temperature and time, respectively
u = internal energy
V,v = fluid velocity and specific volume, respectively

Subscripts

0,1,2,3 = pressurant tank, regulator inlet, propellant tank ullage, and propellant feed line locations, respectively

Introduction

A THERMODYNAMIC study of a particular pressurization system leads to a set of complex differential equations which analysts have attempted to simplify by approximating the actual thermodynamic processes.[1,2] These approximations allow a closed-form solution but introduce errors too severe for final design purposes. This Note describes a mathematical model of the pressurization system used on the Surveyor lunar-landing spacecraft that was formulated with a minimum of simplifying assumptions. The predicted performance is compared with test data.

Discussion

The Surveyor spacecraft was designed and built by Hughes Aircraft Company under the direction of the California Insti-

Received June 4, 1970; revision received August 28, 1970.

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tute of Technology Jet Propulsion Laboratory (JPL) for NASA. The basic spacecraft objectives and the design of the liquid propellant vernier propulsion system (VPS) have been described in Refs. 3 and 4. The facts that 1) the regulator is mounted on a flange on the pressurant supply tank, and 2) the lines from the regular to the propellant tanks were relatively long (as shown in Fig. 2 of Ref. 4) strongly influence the pressurant temperature variation between the regulator and tanks.

The schematic (Fig. 1) shows that the system under study is typical of many regulated pressurant supply systems. The pressurant leaves its tank at the instantaneous rate of $\dot{m}_1$, and heat is transferred to it from the tank wall at an instantaneous rate $\dot{Q}$. The instantaneous rate of change of energy within the pressurant tank is

$$\dot{U}_0 = \dot{Q} - \dot{m}_1[u_1 + V_1^2/2g + p_1v_1] \tag{1}$$

The kinetic energy term, $V_1^2/2g$, may be ignored, since a significant velocity at station 1 would result in a significant pressure head loss, to be avoided in good system design. Also, assuming that the internal energy of the pressurant is a function of temperature only, and noting that $(u_1 + p_1v_1) = h_1$, Eq. (1) becomes

$$(\delta U_0/\delta m)\dot{m}_1 + (\delta U_0/\delta T_0)\dot{T}_0 = \dot{Q} - h_1\dot{m}_1 \tag{2}$$

Assuming a perfect gas and integrating,

$$-c_vT_0 \int_0^{\tau} \dot{m}_1 dt + mc_v \int_0^t T_0 dt = \int_0^t \dot{Q}dt - h_1 \int_0^t \dot{m}_1 dt \tag{3}$$

If, for a small time increment, all other parameters are assumed to be invariant, the pressurant temperature change (ΔT_0) may be found.[5] The new pressurant temperature, in conjunction with the new parameters derived from the auxiliary equations, allow the computation of a new temperature change for the next time interval. The accuracy of this approach increases as the time increment approaches zero; thus, a computer solution is desirable.

The pressurant flow rate $\dot{m}_1 = \dot{m}_2$ is a function of the propellant flowrate $\dot{m}_3$ which is determined by the controlling orifices of the rocket engine when operating at the regulated tank pressure, i.e.,

$$\dot{m}_2 = (\rho_2/\rho_3)\dot{m}_3 = (p/R_2T_2\rho_3)\dot{m}_3 \tag{4}$$

The heat flux from the pressurant tank wall to the pressurant is given by

$$\dot{Q} = HA(T_w - T_0) \tag{5}$$

The determination of the heat-transfer coefficient H is the key to the development of a general set of equations which can be used to describe pressurant behavior under varying environments. Although data on free-convection heat transfer to a gas in a sphere are unavailable, Ref. 6 presents a relationship derived from the results of "experiments on free-convection heat transfer from the surface of a sphere to a fluid filling the interior. Alcohol, glycol, and water were used as fluids, and a range of 3×10^8 to 5×10^{11} of the product Gr_dPr was covered." The equation is

$$Nu_d = 0.098(Gr_dPr)^{0.345} \tag{6}$$

Since the Grashof number Gr_d is directly dependent on the gravity field g, the reduced heat transfer resulting from the decrease in free-convective fluid flow at low accelerations is accounted for in the results. This fact makes the equations applicable during the variable accelerations which accompany the various spacecraft maneuvers.

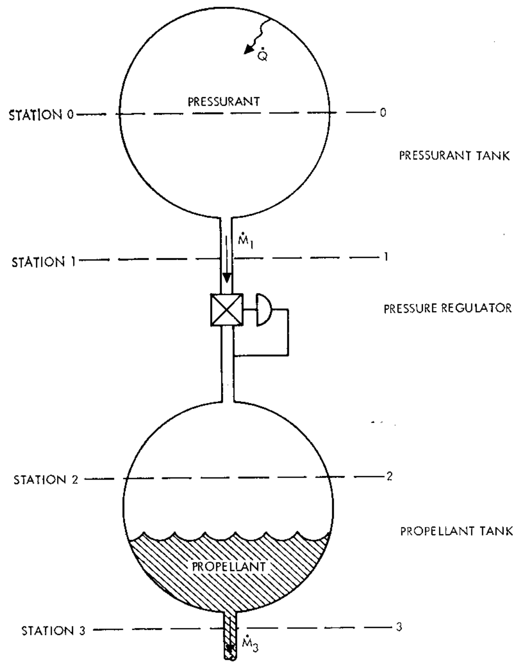

Fig. 1 Pressurization system schematic.

Finally, for an adiabatic external surface, the pressurant tank wall temperature change is found to be

$$(T_2 - T_1)_w = \frac{1}{(MC)_w} \int_0^t \dot{Q}dt \tag{7}$$

Since Eq. (6) includes fluid properties which are temperature and/or pressure-dependent, the property variations over the expected pressure and temperature range were investigated. The variation of C for helium was found to be less than 1%, justifying the perfect gas assumption incorporated into Eq. (3), and was neglected; however, thermal conductivity and viscosity varied by about 3%, and straight-line approximations were used to relate them to the gas temperature. (Pressure effects were found to be second-order.)

The bulk gas temperature was used to determine the viscosity and conductivity of the pressurant. The nonideal

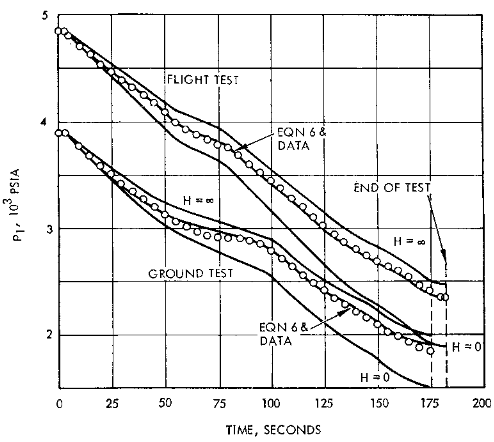

Fig. 2 Terminal descent.

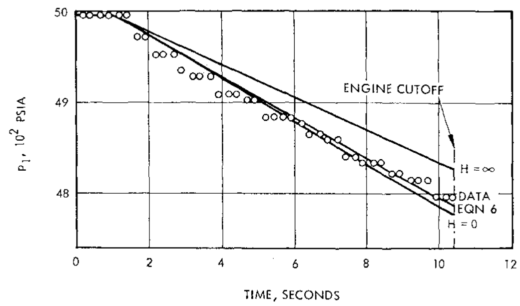

Fig. 3 Midcourse maneuver, Surveyor 6.

behavior of helium at high pressures was accounted for by the use of the compressibility factor[7] which was programed to vary with pressure alone, since its variation with temperature was of the second-order. The external surface of the pressurant tank was assumed to be adiabatic in all cases, since the ground testing was done with well-insulated tank, and orbital tank wall temperature changes during operation were small relative to the initial equilibrium temperature. This assumption is inherent in Eq. (7). The gas entering the propellant tanks was assumed to be instantaneously heated to the propellant temperature. This approximation circumvents a determination of changes in pressurant temperature during its transit from the pressurant tank to the propellant tank and, subsequently, its rate of heat exchange with the propellant and tank walls. Since heat transfer causes the gas temperature in the propellant tank to approach the propellant temperature, the validity of the approximation [Eq. (4)] improves as the expulsion period lengthens.

Since the gas velocity at the pressurant tank exit was assumed to be low, h_1 in Eq. (3) is equal to h_0.

The solution of these equations is substantially simplified if H can be assumed constant. References 1 and 2 discuss the restrictions imposed on the analysis when H is either zero or infinity. The $H = 0$ case corresponds physically to an adiabatic gas and is satisfactory for short expulsion periods. The $H = \infty$ case requires that the pressurant storage vessel and the gas be at the same temperature at all times and is approached physically during long expulsion periods.

Results

To facilitate comparison with test data, predicted pressurant utilization is described in terms of p_0 rather than $\dot{m}_1$. Since T_0 is computed as an integral part of the solution, $\dot{m}_1$ could be easily obtained if desired. Also, putting the gas utilization in terms of p_0 allows the penalty of the "minimum allowable" regulator inlet pressure (which is reflected in an unusuable pressurant residual) to be readily evaluated.

Data from three VPS tests were plotted: ground simulation of the lunar descent phase,[8] inflight midcourse correction,[9] and inflight lunar descent.[9] The primary independent test

Table 1 Independent test parameters

	Midcourse flight test	Terminal descent: Ground test	Terminal descent: Flight test
Initial He pressure, psia	5003	3900	4853
Regulated pressure, psia	720	730	725
Initial He tank temperature, °F	83	−3	73
Gravity field, g	0.1	1.0	0.1–10
Expulsion period, sec	10.4	175	182

parameters are presented in Table 1. The pressure histories for three tests are shown in Figs. 2 and 3. They exhibit nonlinearities due to the vernier rocket throttling. Solid lines represent the analytical predictions, whereas the data points are circled. The hysteresis inherent in the potentiometer-type inflight pressure transducer causes a discrete "step" pressure reading which lags the actual gas pressure (Fig. 3). These steps are not seen in Fig. 2 inflight case (lower set) since the data have been time-smoothed.

The coefficient limits, $H = 0$ and $H = \infty$, bound the nominal case. The zero case predicts the initial data in Fig. 2 (or the brief maneuver in Fig. 3) well, since the temperature difference between the tank wall and pressurant is small initially. After 30–40 sec in the descent maneuver, the infinite coefficient produces a better approximation.

Concluding Remarks

The heat transfer coefficient H defined by Eq. (6) allows the pressurant usage to be predicted to a significantly greater accuracy than either of the approximations ($H = 0$ or $H = \infty$). In comparison, on the one hand, the $H = \infty$ assumption would have resulted in a design which would not have provided adequate pressurant. On the other hand, additional gas and tank weight required to provide the same terminal pressure as the nominal inflight case with $H = 0$ showed that a combined weight increase of $\sim$8%, a significant percentage on the Surveyor spacecraft, would be required.

The independent parameters required to solve the equations are fluid properties (gas and propellant), initial pressures and temperatures, propellant flowrate history, and pressurant tank geometric parameters. If the finite difference method of solution is used, and the pressurant tank is assumed to be in initial thermal equilibrium with the pressurant, the driving temperature difference between the gas and tank [Eq. (8)] is zero, making $H = 0$ during the first time interval. This fact allows the initial conditions for the second time interval to be calculated. The resulting temperature drop in the gas creates a temperature gradient and allows the subsequent H to be calculated. This process is repeated until the minimum allowable p_0 is reached simultaneously with propellant depletion. Obviously, the accuracy of the method improves as the time interval is decreased. A time interval of $\frac{1}{16}$ sec was adequate for the computations presented here.

Pressurant system optimization can be achieved by varying either the initial tank pressure at fixed volume or tank volume at a fixed pressure until the desired terminal pressure is found.

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