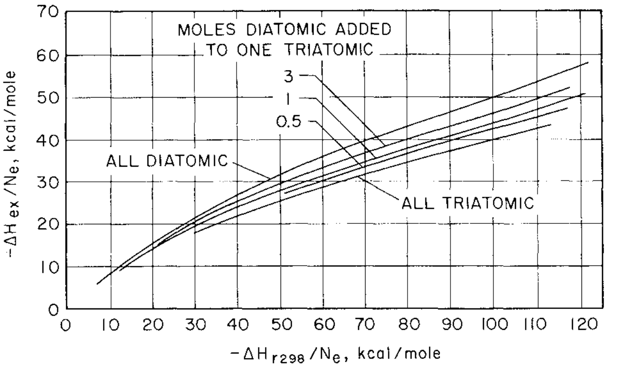

Fig. 3 Loci of energy conversion of unambiguous triatomic species with added diatomics (1000 psia/1 atm, shifting expansion).

For example, nitrogen and hydrogen (both diatomic) can be interchanged in the curves, although the latter dissociates much more than the former. The practical corollary is that the expansion energy of rocket formulations whose effective species are gaseous and predictable (as explained) can be estimated from the curves.

Theoretical specific impulse I_s is equal to 295 $(-\Delta H_{ex}/W)^{1/2}$ where $-\Delta H_{ex}$ is the theoretical expansion energy in kcal for W grams of propellant, and the constant 295 is the value of $(2J/g)^{1/2}$ for these units, J being the mechanical equivalent of heat and g the gravity constant. In using Fig. 4 to estimate I_s, the propellant is designated by its effective species through the applicable chemical equation and its $-\Delta H_{r298}$ to these species then determined by customary means. Dividing the latter by N_e, the moles of effective species, provides a value for direct entry into Fig. 4 on the proper N_A/N_e curve to obtain $-\Delta H_{ex}/N_e$. The latter, multiplied by N_e and divided by W, provides the necessary parameter for computing I_s.

An example for the nitrogen tetroxide-hydrazine system, formulated with 43 parts by weight of oxidizer (0.62 moles/100 g of propellant) and 57 parts by weight of fuel (1.34 moles), is given below for expansion from 1000 psia to 1 atm with shifting equilibrium:

$$0.62\,N_2O_4(1) + 1.34\,N_2H_4\,(1) = 2.48\,H_2O(g) + 1.96\,N_2(g) + 0.20\,H_2(g)$$

$$-\Delta H_{r298} = -[0.62\,(-5.4) + 1.34\,(12.0)] + [2.48\,(-57.8) + 0 + 0] = 156.1 \text{ kcal}$$

$$N_e = 2.48 + 1.96 + 0.20 = 4.64 \text{ moles effective species}$$

$$-\Delta H_{r298}/N_e = 33.7 \text{ kcal/mole}$$

$$N_A/N_e = [3\,(2.48) + 2\,(1.96) + 2\,(0.20)]/4.64 = 2.51 \text{ atoms/mole}$$

From Fig. 4, at this $-\Delta H_{r298}/N_e$ value and N_A/N_e curve

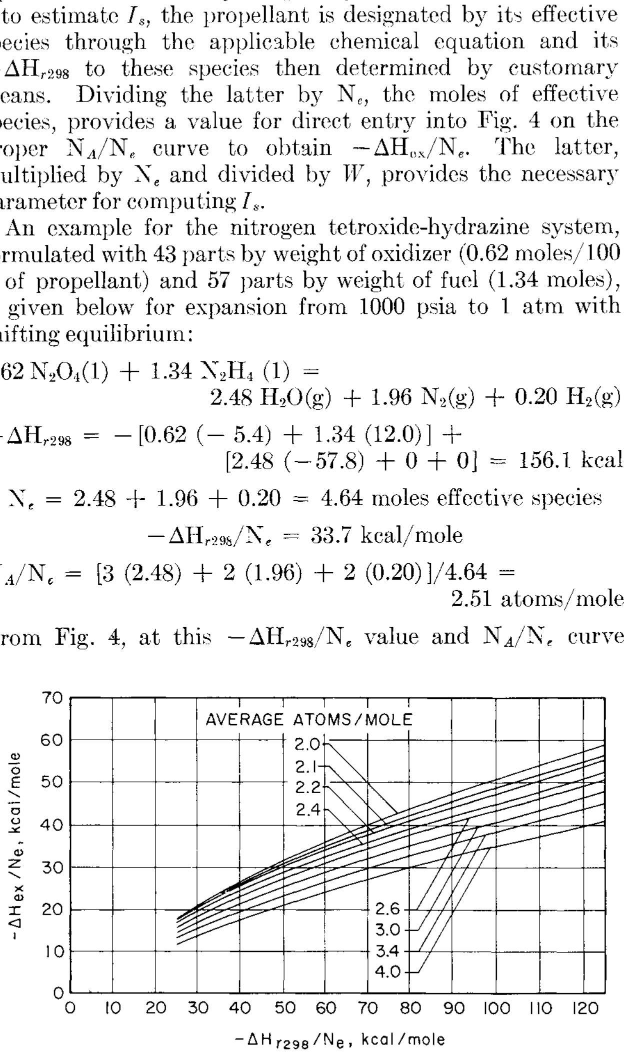

Fig. 4 Loci of energy conversion of unambiguous gaseous species of various average atoms/mole (1000 psia/1 atm, shifting expansion).

(interpolating for the latter),

$$-\Delta H_{ex}/N_e = 20.7 \text{ kcal/mole}$$

$$-\Delta H_{ex} = 20.7\,(4.64) = 102.0 \text{ kcal}$$

Then, $I_s = 295\,(-\Delta H_{ex}/W)^{1/2} = 295\,[(102.0)/100.0]^{1/2} =$ 290 sec, compared to I_s by machine computation = 291 sec.

Good agreement between theoretical specific impulses estimated by Fig. 4 and computed directly are typical, so the empirical method can be used with confidence to an accuracy better than 2% whenever the effective species can be unambiguously predicted. An example of a formulation which, however, cannot be handled by Fig. 4 is $B_5H_9 + 7.5\,F_2$, since its effective species are not the readily predictable $5\,BF_3 + 4.5\,H_2$, because interaction between these species will occur with the formation of some HF and partial dissociation of BF_3 to other effective species—i.e., there is insufficient oxidizer to combine with all the fuel, so that the effective species cannot be unambiguously predicted. On the other hand, the effective species of the formulation $B_5H_9 + 12\,F_2$ are $5\,BF_3 + 9\,HF$, as can be readily predicted, and the system fits the atoms/mole curve that applies to this assumption, i.e., sufficient oxidizer is now present, so that the effective species can be unambiguously predicted.

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Storability Design Criteria for Space Propulsion

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The storability of propellants in space is of great current interest because the higher-performing liquid propellants are cryogenic in nature and are thus inherently more difficult to store in space. The recent development of novel insulations that perform well in a hard vacuum makes the storage in space of cryogenic propellants attractive for periods up to two years. It will be shown that efficiently stored cryogenic propellants offer performance superior to that of many conventionally storable propellants.

Analysis

An analytical technique has been developed to permit evaluation of a wide variety of propellant combinations stored in space for time periods of up to two years. The rating criterion is the maximum velocity increment that each combination is able to impart, after space storage, to a "standard" upper-stage vehicle that is assumed to have constant gross and payload weights. A computer program has also been developed to determine tradeoffs between the weight penalties

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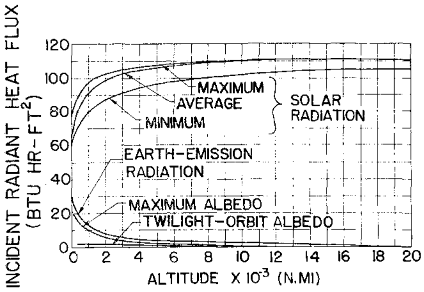

Fig. 1 Heat flux incident on an orbiting sphere.

associated with insulation propellant loss through boiloff and the penalties of increased tank size.

The three significant thermal sources in space near the earth are radiation from the sun, reflection from the earth, and radiation from the earth. The heat fluxes from these sources for orbital altitudes up to that of a 24-hr satellite (19,600 naut miles) are shown in Fig. 1. For this figure, it was assumed that the skin temperature was uniform, because the satellite probably would be spinning and because some multilayer insulations have high conductivity in the lateral direction. Calculated curves for the maximum and minimum solar radiation received (orbits perpendicular to and in the ecliptic plane, respectively) and a weighted average solar radiation are given. On the basis of a yearly average, earth-reflected radiation (albedo) is at a maximum when the solar radiation is at a minimum (orbit in the ecliptic plane), and it is at a minimum when the solar radiation is at a maximum (orbit initially perpendicular to the ecliptic plane).

The steady-state skin temperature is a function of the radiant heat, vehicle configuration, and the skin characteristics of the insulation (Fig. 2). The parameter F/S is the ratio of the fraction of the tank exposed to space to the fraction exposed to the thermal source. For normal configurations, this ratio will be greater than 1. Figure 2 is valid only when the quantity of heat entering the propellant is negligible compared to that which is reradiated (this is true for multilayer insulations). When the skin temperature and the thermal conductivities and thicknesses of the insulation layers are known, the heat-leak rate through the insulation for any given propellant temperature can be determined.

The heat conducted to the propellants through the tank supports and pipe connections was approximated from the experimental data on storability discussed below, employing the conservative assumption that this heat flow is proportional to the propellant weight and the temperature difference between skin surface and the liquid. The propellants were assumed to be at their boiling points as they arrived in orbit. Thus, propellant subcooling is available to balance the higher

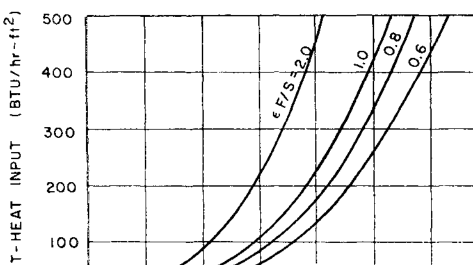

Fig. 2 Skin temperature vs radiant-heat input to propellant tanks: $S(\alpha Q_s + JQ_a + \epsilon Q_e) = \sigma F \epsilon T_S{}^4$, where α = absorptivity, ϵ = emissivity, F = tank-to-space shape factor, S = unshaded fraction of tank, and σ = Stefan - Boltzmann constant.

heat input expected during the prelaunch and boost phases. The theoretical specific impulses used to calculate effective exhaust velocities were reduced by 5% to account for engine inefficiencies.

Calculations based on the information in Figs. 1 and 2 show that proper coating of tank surfaces will maintain the maximum skin temperature at well below 500°F. Since the saturation temperatures of Aerozine-50, ClF_3, N_2H_4, N_2O_4, and B_5H_9 are all higher than 500°R at pressures greater than 1 atm, these propellants can be allowed to reach equilibrium with the skin temperature without boiloff.

Figure 3 shows computed maximum velocity increments for a hypothetical system, with a skin temperature of 425°F, as a function of the time in orbit before the rockets fire. Propellant boiloff causes the velocity increment of the nonstorable systems to decrease with orbit time.

Experimental Evaluation of Storability

A 3-ft-diam spherical tank was selected as the best shape and size for testing. It was made of 321-series stainless steel, with a nominal wall thickness of 0.125 in. Tension supports appear to provide the best available thermal isolation; $\frac{1}{8}$-in. Fiberglas cord was selected for the tank suspension members.

Insulating materials were investigated both for their insulation properties and adaptability to the spherical tank. The NRC-2 insulation used consists of 100 layers of metallized Mylar. The quarter-mil plastic is crumpled in order to insure that the layers touch only at points and have no large areas of planar contact, so that each layer can reach its own equilibrium temperature. The aluminum coating is so thin (about 1 μ) that the lateral conductivity of the metallized plastic is hundreds of times less than the lateral conductivity of an equally thin layer of aluminum foil. Accordingly, it is not necessary to provide spacers to separate the reflecting barriers; this greatly facilitates installation. The insulation is covered with Scotchcal, a pressure-sensitive film with a low absorptivity-to-emissivity ratio. This film also serves as a protection for the insulation during installation and handling.

Commercial Monel bellows-type hoses (0.005-in. wall thickness, 4 ft long, wrapped with glass braid) were used for the two 1.5-in. propellant lines, because they were found to have a lower thermal conductivity than the thinnest practical Fiberglas or Kel-F hose assembly.

The tank was tested with liquid nitrogen in an 8-ft-diam vacuum chamber. The tank was suspended from a load cell and two compound flexures. After calibration, this arrange-

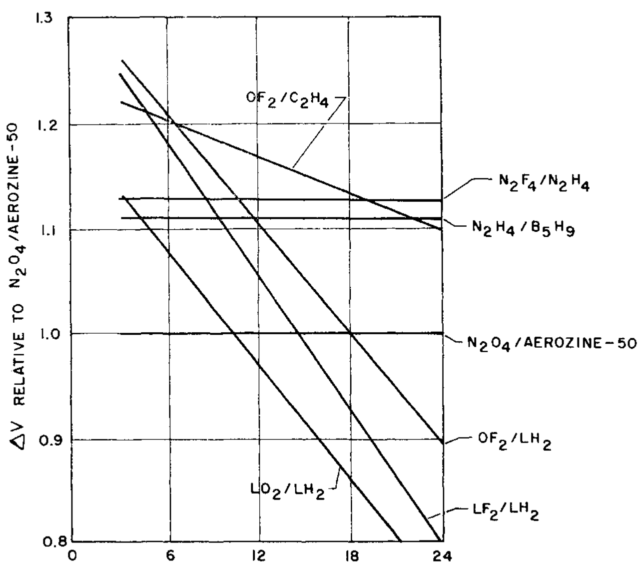

Fig. 3 ΔV vs space-storage time.

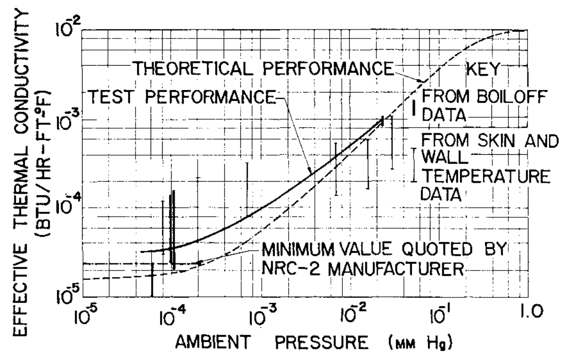

Fig. 4 Effective conductivity of NRC-2 insulation.

ment permitted continuous measurements of liquid N_2 boiloff. Thermal analysis showed that the vacuum chamber walls radiated enough heat to the test tank to maintain the outer layer of insulation at about 70°F. The vacuum-chamber walls radiated relatively uniformly to the tank; for this reason, the temperature of the outer layer of tank insulation was approximately constant throughout its circumference. (This simulated an orbit condition with vehicle rotation of about 0.1 rpm or more.)

A 13-hr checkout test and a 171-hr storability test were performed. Evaluation of test data indicated that the effective conductivity of the insulation at high-vacuum conditions was approximately 3×10^{-5} Btu/hr-ft-°F, compared with a minimum value for "ideal" application techniques of 2.5×10^{-5} Btu/hr-ft-°F.

Data were evaluated by two different analytical methods. In the first, temperature and weight-loss data were used to make an energy balance on the entire test tank, and the insulation conductivity was calculated. Calculations were made over three different time intervals, as shown by the vertical black bars on Fig. 4; an additional calculation, made by the same method from data taken during the checkout tests, is shown by a shorter vertical black bar to the right. The vertical height of the black bars represents the uncertainty introduced by the errors inheren tin the instrumentation and experimental technique. In the second method, heat radiated from the vacuum chamber wall to the insulation outer skin was balanced against the heat conducted through the skin into the propellant. Effective conductivities obtained by this analysis are shown as light lines on Fig. 4; the vertical heights of the lines indicate the estimated precision of the data. The data obtained by both methods show considerable scatter, but a definite trend (indicated by the solid line labeled "test performance") is established which is qualitatively similar to the theoretical curve.

The temperature of the outer insulation surface (tank skin) remained at approximately 70°F throughout the test period. Calculations indicate that the insulation surface would stabilize at a temperature of about −35°F in the low-altitude twilight orbit (the most severe thermal condition). Storability of propellants in space in this tank would therefore be about 20 to 25% better than that exhibited during the test performed.

The average total heat input to the tank from both conduction and radiation was approximately 20 Btu/hr under vacuum of 10^{-4} torr. This corresponds to a boiloff rate of 0.24 lb/hr of liquid nitrogen or 0.13 lb/hr of liquid hydrogen. If heat input continued at this rate, approximately 3000 hr would be required to boil off all of the liquid nitrogen in the tank. Because of the lower density of liquid hydrogen, the tank would contain fewer pounds of it; about 500 hr would be required to boil off a tankful of liquid hydrogen at the 20-Btu/hr heat-input rate.

Various storable propellants have been exposed to gamma radiation, simulating space dosages of gamma rays and x rays, and then analyzed for decomposition and other deleterious effects. The tests showed that, as long as provision is made for venting, little hazard is likely from this source.

Conclusions

This work has shown that cryogenic and, more especially, mild-cryogenic propellants can be stored in space for substantially longer periods than has been generally believed. The use of advanced techniques to minimize heat transfer to the propellants (such as tension-wire suspension of tanks and superinsulations) will probably be necessary to achieve these long-storage times with reasonable weight penalties. An analysis showed that the complex problem of optimizating upper-stage vehicle parameters can be accomplished by means of an appropriate computer program, which can provide large dividends in accurate selection of vehicle configuration. Chemical analyses of storable propellants that had been exposed to ionizing radiation indicated that, as long as provisions are made for venting, little hazard to storability is likely from this source.

Miniguide—A Simplified Attitude Control for Spin-Stabilized Vehicles

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MINIGUIDE is an extremely simple attitude-control concept for spin-stabilized vehicles. It uses body-fixed horizon detectors as an absolute attitude reference and a body-fixed reaction jet to provide control torque. The spin, which stabilizes the vehicle in space, also provides the scan for the attitude reference, and it allows only a single control jet to provide all the control torque for two axes, along with the required stability augmentation or damping torque. Use of an absolute attitude reference allows control of the final stage of a multistage vehicle without requiring control of the previous stages. This note will describe the following: 1) the two basic forms of miniguide—vertical attitude control and pitch attitude control; 2) a two-axis control scheme suitable for orbit injection; 3) flight test of a variation of the pitch control; and 4) future proposed usage of the miniguide system.

The vertical control, shown diagrammatically in Fig. 1, uses a horizon-detecting telescope with a field of view several degrees wide. This telescope is set at an angle with respect to the spin axis which, when the spin axis is vertical and the vehicle is at the correct altitude, allows the field of view to sweep around the edge of the apparent earth disk. In this condition, the percentage of the field of view occupied by the earth remains the same throughout the revolution of the vehicle. If the spin axis of the vehicle is not vertical, the percentage of the field of view occupied by the earth will change as the vehicle spins, providing an essentially sinusoidal signal at the spin frequency. This signal is suitably amplified and fed to an electronic switch, which is activated during the half-cycle when the signal indicates that the telescope sees the most earth.

The output of the electronic switch operates a body-fixed reaction jet positioned on the body such that, when actuated, it torques the vehicle so as to move the telescopic field of view off the earth. Since the spinning vehicle is acting as a gyro-

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