α_0 depends upon: 1) alignment of the control mass path with the principal axes, 2) I_y/I_z at the start of the maneuver, and 3) timing of the mass movement. Misalignment data presented here are for the system of Fig. 1a, but the effects of misalignment in the other configurations are essentially the same. With perfect alignment; i.e., with the path of the control mass coincident with the **z** principal axis, the directions of the principal axes remain fixed within the spacecraft as the control mass is moved (the shift in the center of gravity has no significant effect), and θ (the angle between the principal axis of intermediate moment of inertia and the original spin axis) changes abruptly from 90° to 0° as I_y/I_z changes from a value greater than 1.0 to less than 1.0. Figures 2a and 2b show two types of misalignment that will be present to some degree in any real system; their effects are very similar. Figure 2c (from fundamental principal axis equations) shows the effects of various Δy offset errors on θ.

Misalignment in the **x-z** plane, offset or angular, will not in itself affect α_0. Movement of the mass with such misalignment rotates the **x** and **z** principal axes without affecting the direction of y.

Euler's equations are not valid for the period of mass movement. For rigorous computer simulation of this phase, equations have been developed using the method of Grubin.[2] I_i, I_j, I_k are defined as the moments of inertia of the main body about its principal axes **i**,**j**,**k** (corresponding, respectively, to **x**,**y**,**z** of the total vehicle, but not normally quite aligned with them), and the control mass is assumed to be a point mass. M is the mass of the main body and m that of the control mass. For the system of Fig. 1a, with the path of the control mass confined to the **j-k** plane, and r_k being the instantaneous distance of the mass from the **j** axis and r_j its instantaneous distance from the **k** axis, the equations of motion are

$$\dot{\omega}_i = \frac{-1}{I_i + m'(r_k^2 + r_j^2)} \{ [I_j - I_k + m'(r_k^2 - r_j^2)]\omega_j\omega_k + m'[2(r_k\dot{r}_k + r_j\dot{r}_j)\omega_i + r_jr_k(\omega_j^2 - \omega_k^2) + r_k\ddot{r}_j - r_j\ddot{r}_k]\}$$

$$\dot{\omega}_j = \frac{-[(I_k + m'r_j^2)A + m'r_jr_kB]}{I_jI_k + m'(r_j^2I_j + r_k^2I_k)}$$

$$\dot{\omega}_k = \frac{-[m'r_jr_kA + (I_j + m'r_k^2)B]}{I_jI_k + m'(r_j^2I_j + r_k^2I_k)}$$

$$A = (I_k - I_i)\omega_i\omega_k - m'(r_k^2\omega_i\omega_k + r_jr_k\omega_i\omega_j + 2r_k\dot{r}_j\omega_k - 2r_k\dot{r}_k\omega_j)$$

$$B = (I_i - I_j)\omega_i\omega_j + m'(r_j^2\omega_i\omega_j + r_jr_k\omega_i\omega_k - 2r_j\dot{r}_k\omega_j + 2r_j\dot{r}_j\omega_k)$$

$$m' = Mm/(M + m)$$

When the control mass is stopped, these become equivalent to Euler's equations, although not of the same form since **i**,**j**,**k** are not in general aligned with **x**,**y**,**z**.

Figure 3 shows simulated inversion for various I_y/I_z ratios and Δy offset errors. The total tilt angle is highly dependent upon the alignment error. With $\Delta y = 1$ in. and $I_y/I_z = 0.9990$, for example, the spacecraft tilts 147°, leaving a 33° nutation angle to be damped out. With I_y/I_z the same but $\Delta y = \frac{1}{16}$ in., however, the nutation angle is only 2°. Figure 3 is based on $I_x = 73.8$ slug-ft^2 during inversion. (With the configuration of Fig. 1a, such a spacecraft would have $I_x = I_z$ during normal operation.) A reduction in I_x would cause the maneuver to occur more rapidly.

For uniformity, Fig. 3 is based on the initial tilt angle α_0 being identical to the angle of rotation of **y** within the spacecraft (i.e., the value of θ in Fig. 2c corresponding to I_y/I_z at the end of the control mass movement). The rigorous simulations of the period of mass movement show that this is quite accurate if that movement occurs within a few seconds. If it takes longer, wobble builds up, causing inversion to occur somewhat more rapidly after the mass is stopped, but even a period of 30 sec does not greatly affect the data of Fig. 3.

A mechanically despun antenna could be allowed to spin up until it was fixed with respect to the basic spacecraft just before the maneuver, then be despun again after inversion. (It must rotate, with respect to the spacecraft, in the opposite direction after inversion.)

The smaller the I_y/I_z ratio in normal operation, the smaller the required control mass. Spin stability decreases as I_y/I_z approaches 1.00, however, so a tradeoff is necessary. The value 1.04 used for developing the data in this paper has proven to be satisfactory for the ATS-3 (Applications Technology Satellite) and it is likely that a smaller value would be practical for many missions.

References

[1] Beachley, N. H. and Uicker, J. J., Jr., "Reply by Authors to L. H. Grasshoff," *Journal of Spacecraft and Rockets*, Vol. 6, No. 10, Oct. 1969, pp. 1215–1216.

[2] Grubin, C., "Dynamics of a Vehicle Containing Moving Parts," *Journal of Applied Mechanics*, Vol. 29, Sept. 1962, pp. 486–488.

Hydrazine Azide as an Additive for Monopropellant Hydrazine

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Introduction

HYDRAZINE is used for various satellite propulsion applications because of its simplicity, its demonstrated near-theoretical performance, and its long-term stability. This Note summarizes results of an investigation of blends of hydrazine and hydrazine azide ($N_2H_4HN_3$, whose acronym is HA) offering significantly better theoretical performance. Based on liquid reactants at 298°K, Fig. 1 shows the standard performance parameters vs the computed fraction of ammonia dissociated. Adiabatic flame temperature decreases sharply as the fraction dissociated increases, and increases as the proportion of HA in the blend increases. Blends of 20–26% HA are particularly attractive because the freezing point is lower and performance is higher than with neat hydrazine.

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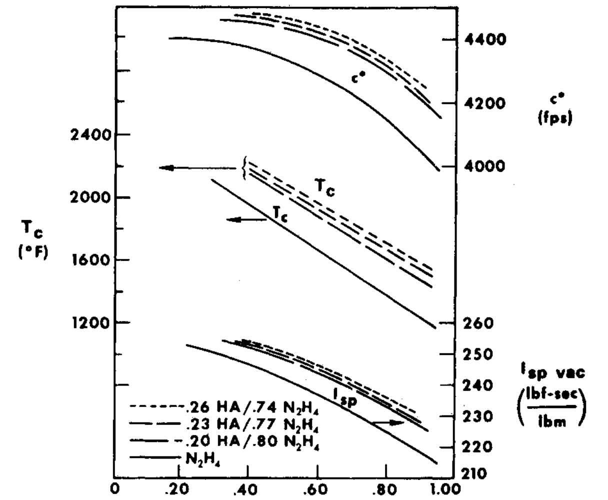

Fig. 1 Theoretical performance.

The decomposition of HA is analogous to that of hydrazine[1] which may be simplified to a two-step process. First, NH_3 and N_2 are formed and, second, some of the NH_3 is dissociated. With HA, the product species are formed in different proportions and with greater heat of reaction than with hydrazine; therefore, there is no reason to expect the degree of ammonia dissociation to be the same.

Properties of Hydrazine/Hydrazine Azide Blends

Freezing points of three HA/N_2H_4 blends were determined by using warming curves. A sample in a glass tube was frozen; then the tube was evacuated, sealed, and vibrated during warming. The melting point was taken as the break in the time temperature plot when the last trace of solid disappeared. The results did not agree precisely with earlier data[2] (Fig. 2) because impurities (e.g., water in the hydrazine) shifted the eutectic point. Near the eutectic point, the freezing point increased by ~5°C for a 1% increase in HA content, while the increase was only 1.3°C for a 1% decrease in HA content.

Densities were measured in a calibrated straight-tube pycnometer to an accuracy of 0.2%. Viscosities (Fig. 3b) were measured in a modified capillary similar to the viscometer described in Ref. 3; an accuracy of 1% was easily obtained. Results for 20%, 21.8%, and 27.6% are shown in Fig. 3.

Shock sensitivities were determined by using the standard Joint Army-Navy-NASA-Air Force (JANNAF) Card Gap Method. None of the blends detonated with zero cards; this indicated that the mixtures were not shock sensitive.

ICRPG Thermal Stability Tests[4] showed that exotherms started around 130°C for 21.8%, 24.6%, and 27.6% HA as compared to 250°C for pure hydrazine. In all cases once reactions were started, they continued until rupture of the 4700-psi burst diaphragm.

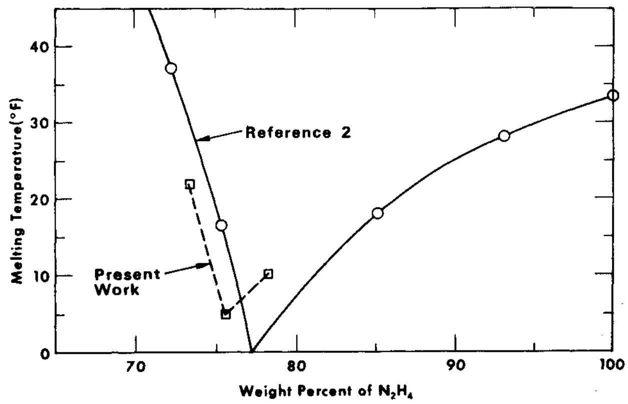

Fig. 2 Freezing point of HA blends.

Long-Term Thermal Stability and Metal Compatibility

The long-term thermal stability of two HA hydrazine blends and their compatibility with two metals were determined at 60°C in a glass reservoir connected to a mercury manometer. For the compatibility determinations, metal coupons of 6061-T6 aluminum and 6A1-4V titanium (which are candidate tank materials) were placed in the apparatus with the propellants. The metal coupons were sized to provide a gas volume to metal surface ratio of 1.7 in., which corresponds to a 10-in.-diam spherical tank with a 50% ullage and a 36-in.-diam tank with a 20% ullage. Runs were made in duplicate at a constant temperature of 60°C and a 20% ullage.

Pressures in the apparatus during the 100-day tests were plotted against time and the instantaneous pressurization rates ($\dot{P}$) were obtained from the curves. For the HA/hydrazine blends, both with and without the metals present, $\dot{P}$ became constant by the 20th day. In the case of neat hydrazine, which was run for comparative purposes, $\dot{P}$ became constant by the 60th day. These steady $\dot{P}$ values of the various fuels are summarized in Table 1. Rates of gas evolution (r), calculated from the steady pressurization rates, and pressure increases (ΔP_t) in propellant tanks after five years of storage are also included in this table.

The gas evolution rates in the HA/N_2H_4 system were compatible with data from other hydrazine-plus-additive systems. In fact, the data confirmed the finding that blends of hydrazine salts in hydrazine are less stable than either of the components, and it appeared that the $N_2H_5^+$ ion was the major source of the instability. Both metals increased gas evolution by 70%, and some surface attack by the blends was evident. Table 2 summarizes the material lost during the 100-day tests.

Table 1 Propellant long-term thermal stability and metal compatibility

HA, % by wt	Temp., °C	Metal coupon present	$\dot{P}$, mm Hg/day	Gassing rate, r, Scm³/lb-min	Predicted ΔP_t,[a] psi 20% ullage	50% ullage
0	60	None	0.6 ± 0.08	5.4×10^{-5}	4.0	2.5
0	38[b]	None	...	1.8×10^{-6}	...	...
20	38[b]	None	...	8×10^{-6}	...	...
21.8	60	None	27 ± 7	2.4×10^{-3}	190	120
21.8	60	Al	$47 + 7$	4.2×10^{-3}	330	210
21.8	60	Ti	48 ± 4	4.3×10^{-3}	340	215
27.6	60	None	40	3.5×10^{-3}	280	180
27.6	60	Al	68 ± 8	5.9×10^{-3}	480	300
27.6	60	Ti	$58 + 2$	5.1×10^{-3}	410	260

[a] After 5-yr storage at a constant temperature of 60°C with no outflow.
[b] 5 % ullage.

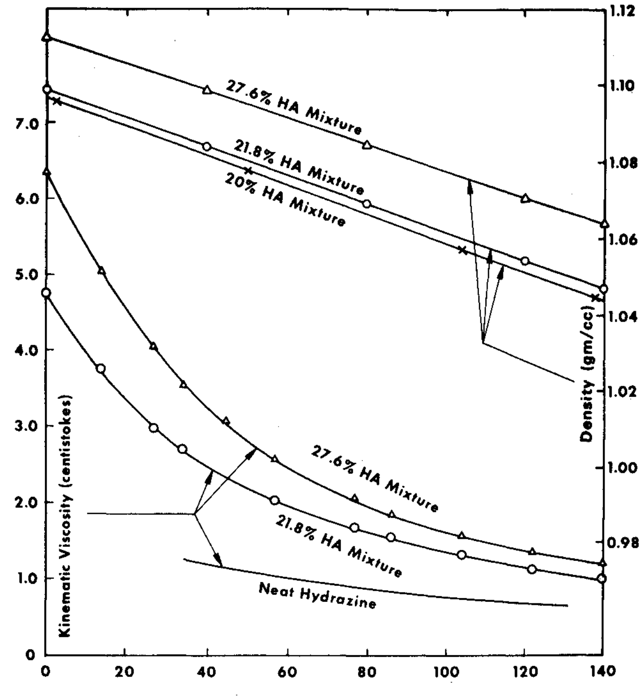

Fig. 3 Density and viscosity of HA blends.

Experimental Engine Testing

A workhorse engine (Fig. 4) consisting of a catalyst bed, nozzle, showerhead injector, flight-type monopropellant pulse valve, and instrumentation was used for experimental engine testing. Design of the catalyst bed and decomposition chamber was based primarily on existing correlations[1] and operating experience with hydrazine. Firings made prior to this program using pure hydrazine and a 20% HA blend indicated that the same fraction of ammonia was dissociated in each case, even though the more energetic azide provided a higher gas temperature. A mass velocity of 0.02 lbm/in^2-sec was selected after a number of factors had been considered (including the desire for long catalyst bed life and stable operation).

With a thrust level of 5 lbf, a chamber diameter of 1.21 in. gave the desired mass velocity. The design P_c was 150 psia.

Discussion of Test Results

Table 3 summarizes the steady-state results for hydrazine and the 20% HA blend. The tests with hydrazine provided a standard of catalyst bed behavior. In these tests, the percent of ammonia dissociation was estimated from the temperature downstream of the catalyst bed (T_c).

For hydrazine, thrust measurements (in one run) showed a delivered I_{sp} of 143 sec, 93% of the theoretical value for the

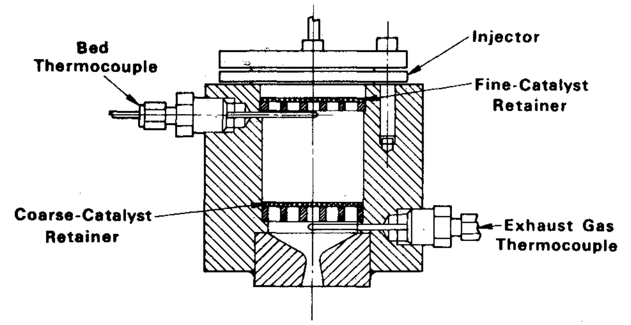

Fig. 4 Experimental thrust chamber.

expansion from 131 psia and 1540°F to sea-level through a nozzle with ϵ = 4.05. Similarly, for a 21% HA blend, a delivered I_{sp} of 149 sec, 92.5% of the theoretical value, was obtained. The 21% HA blend provided a 4% improvement in I_{sp} and C^* over hydrazine. The T_c's listed in Table 3 corresponded to 65% ammonia dissociation for both propellants. The temperatures measured in the middle of the catalyst bed corresponded to dissociation levels of 45–57%.

Catalyst Bed Experience

The catalyst bed consisted of $\frac{1}{4}$-in. of 20- 30-mesh Shell 405 catalyst held in place by retaining screens and plates, and 1.30 in. of the $\frac{1}{8}$-in. cylindrical Shell 405 pellets similarly retained. In some cases loss of catalyst from the bed was serious. To help retain the fine-mesh catalyst, a 100-mesh screen with 0.0055-in. openings was used for the remainder of the program.

The two largest loss rates of the $\frac{1}{8}$-in. pellets occurred in tests with neat hydrazine. The higher temperature of the HA blend did not seem to contribute to $\frac{1}{8}$-in. catalyst degradation. Fine-mesh catalyst was also lost through breaks in the retention screens which were caused partly by the high temperatures (2000°F) experienced with the hydrazine azide decomposition.

Ignition Delay and Engine Response

Ignition delay was defined as the time from propellant entry until a chamber pressure increase occurred. All new ignition-delay data which are applicable in the investigation are plotted in Fig. 5, along with the hydrazine data of Ref. 5. The data are reasonably consistent. There appear to be no significant differences between the published hydrazine data, the new hydrazine data, and the new HA blend data.

For vacuum ignition, bed condition and local temperatures become more significant because of propellant vaporization and cooling. When the bed temperature is significantly below the freezing point of the propellant and, in addition, the propellant is severely cooled by vaporization, ignition may not occur. Vacuum ignition tests in which the hydrazine froze in the catalyst bed have been reported. In the present pro-

Table 2 Metal erosion during 100-day compatibility tests at 60°C

(Percent) HA	Metal	Metal eroded on one side, in	Metal	Metal eroded on one side, in.
21.8	6061-T6-Al	7×10^{-5}	6A14V Ti	25×10^{-5}
27.6	6061-T6-Al	27×10^{-5}	6A14V Ti	44×10^{-5}

Table 3 Steady-state performance summary (expansion ratio ϵ = 4.05)

Propellant	c^*,[a] fps	I_{sp}, sec	C_F	T_c,[a] °F	Ammonia dissociated,[a] %
Hydrazine	3980 ± 30	143	1.155	1600 ± 50	65 ± 3
21% HA	4140 ± 60	149	1.158	1820 ± 70	65 ± 5

[a] Average and deviation for three tests.

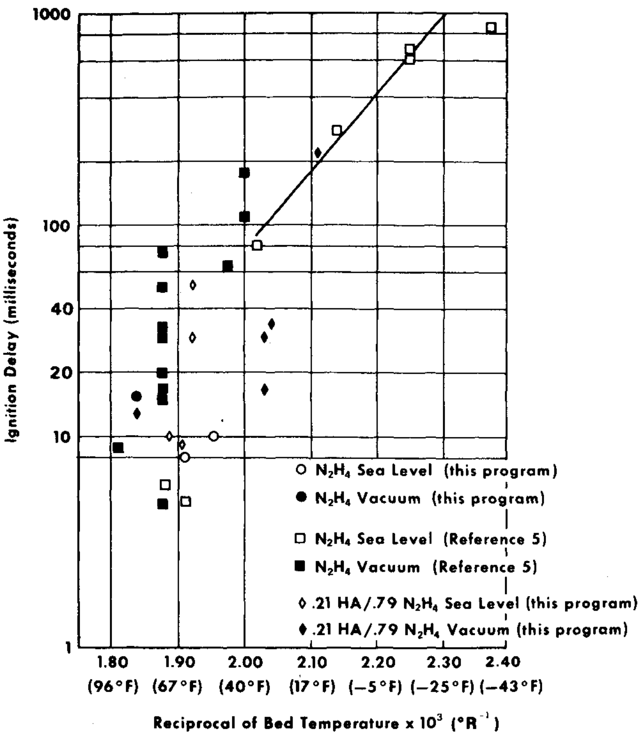

Fig. 5 Ignition of hydrazine and HA blends.

gram, ignition of the HA blend occurred at propellant temperatures as low as −7°C (bed temperature of −11°C) and bed temperatures as low as −29°C (propellant temperature of 1°C).

In the design of the engine, the response time to reach 80% P_c was calculated to be $51/T_{co}$ sec. For initial catalyst bed temperatures of 80°F (T_{co} = 540°R), this value would predict a 95-msec response time for hydrazine. In one test with N_2H_4, the measured response time was 106 msec. In tests with the 0.21 azide blend, the measured response times were 80 and 77 msec. It appears that the addition of azide shortens the response time of the catalyst bed.

Conclusions

Significant conclusions obtained from the program are as follows:

1) The freezing point of hydrazine can be reduced while its performance as a monopropellant is increased by adding up to 28% hydrazine azide to it.

2) Blends of 21.8%, 24.6%, and 27.6% HA in N_2H_4 were found to be insensitive to shock by the standard JANAF card-gap test. However, the three blends were found to be less thermally stable than neat hydrazine.

3) Both the density and the kinematic viscosity of the blends were higher than those of neat hydrazine. For example, at 77°F, the 21.8% HA blend had a density 6.5% higher and a kinematic viscosity 89% higher than neat N_2H_4.

4) In engine tests with the 20% HA blend, the average gas temperature was 240°F higher than the temperature of hydrazine decomposition products.

5) In a number of tests with a single 5-lb thrust engine, the average ammonia dissociation estimated from gas temperatures was 65% for both hydrazine and the 20% HA blend. (This result confirmed a similar conclusion previously obtained with a 25-lb thrust engine.)

6) Experimentally, performance improvements were realized with the 20% HA blend, even in an unoptimized catalytic thruster.

7) Satisfactory vacuum ignition occurred with the HA blend, even at propellant and bed temperatures significantly below the freezing point of neat hydrazine.

8) Engine response (to 80% chamber pressure) was slightly faster with the HA blend than with neat hydrazine.

References

[1] "Development of Design and Scaling Criteria for Monopropellant Hydrazine Reactors Employing Shell 405 Spontaneous Catalyst," RRC-66-R-76, Jan. 1967, Rocket Research Corp., Seattle, Wash.

[2] Dresser, A. L., Browne, A. W., and Mason, C. W., "Anhydrous Hydrazine, VI, Hydrazine Trinitride Monohydrazine, $N_2H^5N_3N_2H_4$," *Journal of the American Chemical Society*, Vol. 55, 1933, pp. 1963–1967.

[3] McGoury, T. E. and Mark, H., "Determination of Viscosity," *Physical Methods of Organic Chemistry, Technique of Organic Chemistry*, 2nd ed., Vol. I, Pt. I, Interscience, New York, 1949, Chap. VIII.

[4] *Liquid Propellant Test Methods Manual*, Chemical Propellant Information Agency, March 1967.

[5] Carlson, R. A. et al., "Space Environmental Operation of Experimental Hydrazine Reactors," Final Report 4712, Contract NAS 7-520, April 1967, TRW Systems Group, Redondo Beach, Calif.

Low-Cost Molded Plastic Sounding Rocket Motors

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THE major expense in the production of rocket motors is the labor cost. As indicated in Fig. 1, manufacturing a plastic motor with an integrally-molded case and nozzle requires fewer operations than a conventional motor.

The steps in Fig. 1 represent the basic operations for processing the two types of motors, and a comparison denotes the eliminated operations for molded motors. During fabrication, the plastic nozzle is molded integrally with the case and eliminates the separate operations of fabricating and installing a nozzle. In case preparation operations, the interior surface of both types of motor cases must be cleaned for adequate bonding of the adjacent material. Whereas the metal case requires insulation, the plastic case serves as its own insulator, and the propellant is cast directly to the case wall with adequate bonding being achieved during the feasibility demonstration tests. (Bonding tests using the ICRPG Joint-in-Tension test method showed that after sandblasting the phenolic case surface, the bond between the propellant and the roughened plastic material was stronger than the propellant itself.) Therefore, the insulating, lining, and associated liner-curing operations normally required for metal cases are eliminated for the plastic cases. Propellant casting and curing with the postcure mandrel-removal completes the propellant loading operations. The installation of a nozzle on the metal case and a headcap on the plastic case provides the end closure required for propulsive units. The headcap will be attached to the plastic case using threads which are integrally-molded in the case during the case and nozzle fabrication. Mating threads are either molded or machined on the headcap, depending on the method of fabricating the headcap. The description and advantages of these threads are discussed later.

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