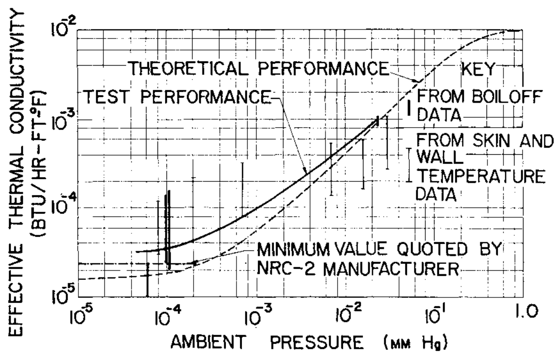

Fig. 4 Effective conductivity of NRC-2 insulation.

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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scope, this jet is positioned on the body at right angles to the telescope to precess the vehicle in the telescope plane. The system will thus null out the a.c. component of the telescope signal, leaving the vehicle spin axis vertical.

Because of the lack of damping in the space environment, a coning motion will be built up during the control phase, and the system may be dynamically unstable due to system time lags. To compensate for this, a stability augmentation signal is obtained from a rate gyro and summed with the telescope signal. The gyro is mounted with its input axis parallel to the axis about which the control jet produces torque. Since there is an energy interchange between the body axes each revolution, damping one axis is sufficient to damp the system, and only the single-rate gyro and reaction jet are required.

The horizontal version of the pitch control, shown in Fig. 2, uses two body-fixed horizon detectors with narrow fields of view mounted on opposite sides of the vehicle spin axis. With the spin axis horizontal and at the operating altitude, the telescopes are set at angles with the spin axis, so that they produce conical scans which just intersect the forward and aft horizons. If the vehicle is in a nose-up position, the forward scan will be completely off the earth, and the rear telescope scan will dip into the earth and see the earth for a portion of the vehicle revolution based on the amount of nose-up error. In a nose-down attitude the converse is true.

With the narrow fields of view of the telescopes, the output signals are essentially off-on signals—off when the telescope sees space, on when it sees the earth. These signals are suitably amplified and operate a switching amplifier that turns on the control jet whenever an earth signal is present from either telescope. Again, the control jet is at right angles to the telescopes, and the spinning vehicle is precessed to the horizontal position. Stability augmentation is provided by a rate gyro signal as before. However, since the system described here is an off-on signal system, the rate is limited only to some preset threshold limit, determined by the amount of residual coning which may be tolerated. A lower-rate threshold will cut down on the coning amplitude and make the system more stable but will require a longer time and use more reaction jet fuel to make a given correction.

The action of the damping system, as well as any system lags, causes the vehicle to build up an error in yaw, or azimuth, while undergoing a control maneuver in pitch. To compensate for this action, the two-axis control scheme shown in Fig. 3 has been devised. In this case an azimuth control has been added to the horizontal control just described. The azimuth error sensors consist of two simple body-mounted shaded photo cells and use the sun as the azimuth reference. The cells and shades are set in the vehicle so that, at the correct azimuth angle, both cells are completely shadowed by their respective shades. If the vehicle is in a nose-left attitude, cell *A* will be exposed to the sun for a portion of the

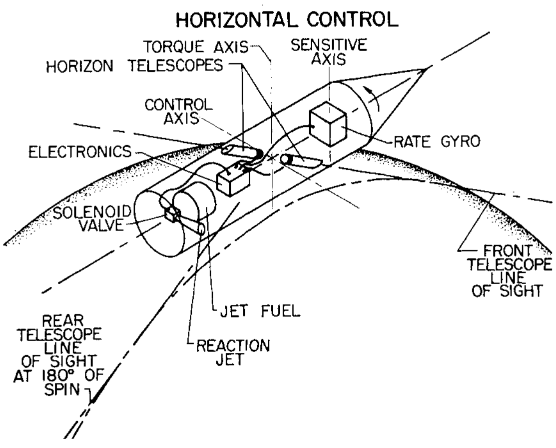

Fig. 2 Horizontal control.

vehicle revolution, while cell *B* will be shaded throughout the vehicle revolution. For a nose-right condition, the reverse would be true. The outputs of these cells feed into the same amplifier as the horizon detectors and operate the same jet. The jet is turned on whenever either detector sees the sun, using the same logic as the horizon detectors, and no increase in electronic complexity has been incurred over the single-axis system.

Computer studies show that the two-axis control will perform satisfactorily for sun angles as much as 45° off the optimum angle, where the sun lies in the azimuth plane. An increase in control gas usage is incurred by the addition of the second controlled axis, though this increase has not yet been fully evaluated.

Preliminary studies have indicated that it would be feasible to use this two-axis control scheme to inject a payload into orbit, using an aerodynamically stable multistage vehicle with attitude control of the orbit injection stage only. The vehicle could be launched from a simple rail launcher, using conventional wind-weighting techniques to set the launcher angle, and all the attitude control would take place during the coast period prior to ignition of the orbit injection motor.

A variation of the pitch-control miniguide scheme has been successfully flight tested. The tested system was to be used as a tipoff disturbance corrector, which provided the requirements for the system. The miniguide was to control the pitch attitude to 65° and damp out coning down to a body rate of 4°/sec, within the initial 12 sec of control-stage engine burning. The attitude at time of ignition was 235,000 ft, the pitch angle was 55°, and the spin rate 3 rps. Vehicle moments of inertia at ignition were 60 slug-ft^2 in pitch and yaw, and 6.5 slug-ft^2 in roll.

The high pitch angle and the free jet or plume from the main engine precluded aiming the aft telescope back at such

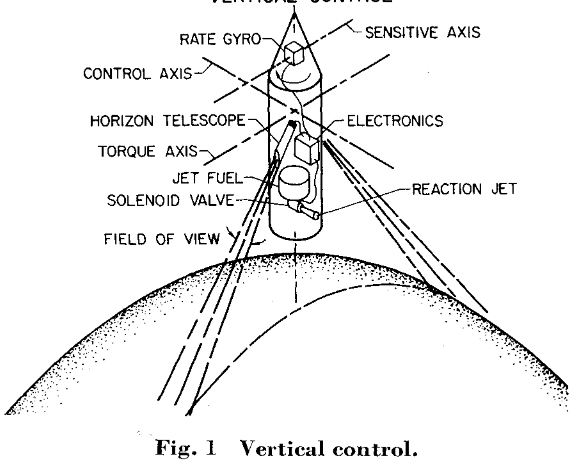

Fig. 1 Vertical control.

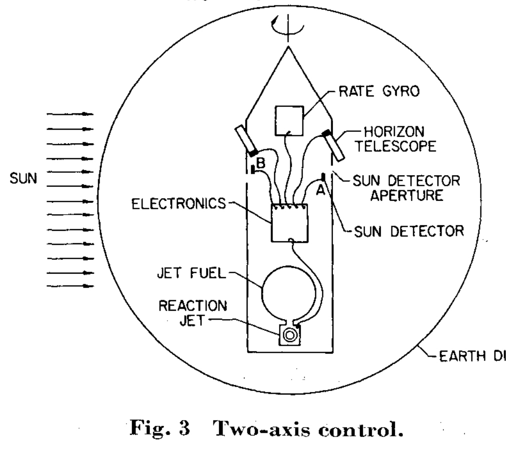

Fig. 3 Two-axis control.

an angle as to see the aft horizon. Accordingly, an alternate scheme was devised. The forward telescope was set at the correct control angle so that it just intersected the horizon at the 65° control attitude, whereas the aft telescope was aimed almost directly out of the side of the vehicle. At this angle, it would see the horizon for a sizeable portion of the vehicle revolution regardless of the pitch angle. In this configuration only the forward telescope is used as a control transducer as long as the forward telescope sees the earth once per vehicle revolution. If the forward telescope does not see the earth once per revolution, the signal from the aft telescope is used to torque the vehicle nose down to where the forward telescope can again obtain control. This function is accomplished by a time-delay, lockout circuit actuated by the forward telescope signal. In all other respects the system operates as the pitch control previously described.

The entire flight-tested system weighed under 18 lb, including tankage, $1\frac{1}{2}$ lb of nitrogen, and battery supply, and it used only 18 transistors and two moving parts—the rate gyro and the solenoid valve. With only minor mechanical redesign, the same system could be flown at a weight of under 12 lb.

The results of the flight test proved to be almost identical to the preflight computer study results. The coning was limited to the body rate of 4°/sec, which produced a $\pm 2°$ cone, and the mean of the coning angle was well within the $\pm\frac{1}{2}°$ of the command angle required, for the entire design flight time. The vehicle reached steady-state conditions within the first 5 sec of operation and remained there throughout the design control phase.

As a result of the successful flight test of the miniguide system and the excellent agreement of the actual flight results with the preflight computer studies, plans are under way to use the vertical control version of miniguide on three probes to be fired within the next two years in a horizon-definition study program.

Rocket Vehicle Weight Trades for Injection Missions

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Nomenclature

g = acceleration due to gravity at earth's surface, 32.2 ft/sec^2
I_{sp} = specific impulse
V = velocity
W = weight

Subscripts

1 = first stage
2 = second stage
BO = burnout
D = drop
E = empty
I = inert
LP = loaded propellant
M = margin
0 = initial
P = expended propellant
P/L = payload
R = residual propellant (trapped + allowance)

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Table 1 Estimated error in approximate trades

Stage and range of operation	Estimated error
Booster, liftoff to suborbital	5 to 20%
Upper, suborbital to injection	0 to 10%
Kick, orbital to superorbital	0 to 5%

Introduction

ANY change in a rocket vehicle parameter is reflected as a related change in either vehicle empty weight or propellant margin, if the total characteristic velocity is held fixed. Approximate evaluations of such "tradeoffs" are useful at almost every phase of system development. Among these uses are: 1) evaluation of preliminary design changes, 2) assessment of the effects of uncertainties and tolerances in vehicle parameters, 3) determination of performance gains from ejected weights, and 4) determination of performance margins.

Vehicle Definition

Once a set of terms that describe the makeup of the vehicle has been defined (as in the Nomenclature), the interrelations between the various parameters to be traded are established:

$$W_E = W_I + W_{P/L} \tag{1}$$

$$W_{BO} = W_E + W_M + W_R \tag{2}$$

$$W_0 = W_{BO} + W_P \qquad W_0 = W_E + W_{LP} \tag{3}$$

$$W_{LP} = W_P + W_M + W_R \tag{4}$$

$$W_{0_2} = W_{BO_1} - W_{D_1} \tag{5}$$

The differentials of these defining equations establish the interdependence of the trades. With one or two terms held constant, the partial differentials, or influence coefficients, for other parameters can then be obtained. For example, from Eq. (2),

$$dW_{BO} = dW_E + dW_M + dW_R \tag{6}$$

Let

$$dW_{BO} = (\partial W_{BO}/\partial W_E)dW_E \tag{7}$$

Then, with W_R held constant

$$(\partial W_{BO}/\partial W_E) = 1 + (\partial W_M/\partial W_E) \tag{8}$$

Approximate One- and Two-Stage Trades

Expressions can be written for empty weight and margin trades in terms of the vehicle parameters, W_0, W_{BO}, I_{sp}, and μ, where $\mu \equiv W_0/W_{BO}$. These expressions are helpful in gaining a further understanding of the effects of vehicle parameter changes on payload capability and propellant

Table 2 Single-stage trades; partials of W_E and W_O

With constant	For G =	$\partial W_E/\partial G$ =	$\partial W_{BO}/\partial G$ =
W_M	W_{LP}	β	β
W_M and W_{LP}	I_{sp}	$m \ln\mu$	$m \ln\mu$
	W_R	$-\mu\beta$	$-\beta$
	W_I	0	0

Table 3 Single-stage trades; partials of W_M and W_{BO}

With constant	For G =	$\partial W_M/\partial G$ =	$\partial W_{BO}/\partial G$ =
W_E	W_{LP}	μ^{-1}	μ^{-1}
W_E & W_{LP}	I_{sp}	$(\mu\beta)^{-1}\, m \ln\mu$	$(\mu\beta)^{-1}\, m \ln\mu$
	W_R	-1	0
W_{LP}	W_E	$-(\mu\beta)^{-1}$	μ^{-1}