of $\mathbf{V}_{S2}$ is the coefficient of the unit vector $\hat{e}$ of Eq. (1). With $|\mathbf{V}_{S2}|$, $\hat{e}$, and $\mathbf{V}_P$ known, for a first guess at flight time, $\mathbf{V}_{H2}$ is found from Eq. (1). Then for a given launch date, an iterative technique, e.g., Newton-Raphson, is used until a trip time is found for which $|\mathbf{V}_{H1}| = |\mathbf{V}_{H2}|$, and the problem is solved.

This iterative procedure was applied to a patched conic trajectory sequence in which a spacecraft is launched from the Earth, flys by the planet Jupiter, and leaves Jupiter in a heliocentric trajectory whose plane is normal to the ecliptic, and whose perihelion is fixed at 0.1 a.u. It is also assumed here that aphelion is fixed at the Jupiter orbital distance, although this assumption is not generally required for normal-to-the-ecliptic trajectories. This merely simplifies the choice of direction of $\hat{e}$, i.e., $\hat{e}$ itself is normal to the ecliptic rather than being located arbitrarily in a plane which is normal to the ecliptic. If $\mathbf{r}_a$ is Jupiter's radius vector at the time of the flyby, then the required magnitude of the heliocentric velocity, $|\mathbf{V}_{S2}|$, is given by

$$|V_{S2}|^2 = \mu_S[(2/|\mathbf{r}_a|) - (1/a)]$$

where

$$a = |\mathbf{r}_a| + r_p/2$$

μ_S is the sun's gravitational parameter, $1.327 \times 10^{11}\text{km}^3/\text{sec}^2$, and $r_p = 1.49599 \times 10^7\text{km}$.

Six launch opportunities were examined, spaced apart in time at approximately the mean synodic period of Earth and Jupiter ($\sim$400 days), and were timed to reach their perihelia of 0.1 a.u. at intervals over one-half cycle of solar activity (minimum in 1981, maximum in 1986). During each launch period, the required Earth injection $C3$ drops to well within the launch capability of, for example, the Titan III-E/Centaur/Burner II, and has a launch window of at least 20 days. This $C3$, of the order of 110 to 140 km²/sec² (corresponding to injection velocities of approximately 14.75–15.75 km/sec), contrasts sharply with the $C3$ of 1000 km²/sec² required for direct Earth injection into a normal-to-the-ecliptic trajectory mentioned earlier; and, in fact, is comparable with the situation in which the spacecraft would be launched from Earth and would recede to a very great distance from the sun (velocity $\sim$ zero). At this point, a very small velocity would suffice to affect a 90° plane change and reverse the direction of the spacecraft in that it would fall back to the sun with a perihelion of 0.1 a.u.

The Earth/Jupiter trip times are of the order of 500 days, with an additional 800 days required to get from Jupiter to perihelion, giving a total mission time from launch to perihelion passage of about 1300 days. All trajectories take the spacecraft about ⅞ of an a.u. out of the ecliptic, at a point essentially over the middle of the asteroid belt, permitting a fairly substantial sampling of the interplanetary medium from Jupiter into the sun. For comparison purposes, if Jupiter is not used to deflect the trajectory, a velocity of 7–8 km/sec, in addition to the injection velocities just quoted, would be needed in a multiburn sequence resulting in the same trajectory. Jupiter swingbys thus result in a savings of this ΔV over more conventional or direct trajectory segmenting.

All of the Jupiter-centered hyperbolas approach Jupiter from below and to the left, as seen by the spacecraft as it approaches the planet. The hyperbolic periapses are all much greater than Jupiter's radius (8–9 planet radii), and motion is retrograde relative to Jupiter.

These trajectories present many interesting and optional opportunities for a large number of scientific investigations. For example, they pass through the asteroid belt on the way out of Jupiter, affording an opportunity to fly near enough to one or more asteroids for optical or sensor measurements to be made. The hyperbolic flyby itself presents a possibility of flying close to one or more of Jupiter's moons, and could be used to determine their masses, and possibly some information on the gravitational field anomalies (mascons) of Jupiter itself. The heliocentric portion of the solar leg of the trajectory would provide an opportunity to measure the properties of the interplanetary medium from a distance of 5 a.u. down to 0.1 a.u. or less. A normal-to-the-ecliptic mission, as described here, coupled with an essentially simultaneously launched vehicle whose trajectory segments remain in the ecliptic, could provide a wealth of scientific measurements in two orthogonal planes at practically the same measurement schedule, thereby greatly augmenting our present knowledge of the sun, its activities, its dynamic processes, and its origin.

References

[1] Flandro, G. A., "Fast Reconnaissance Missions to the Outer Solar System Utilizing Energy Derived from the Gravitational Field of Jupiter," *Astronautica Acta*, Vol. 12, No. 4, 1966, p. 329.

[2] Minovitch, M. A., "The Determination and Characteristics of Ballistic Interplanetary Trajectories Under the Influence of Multiple Planetary Attractions," TR 32-464, 1963, Jet Propulsion Lab., Pasadena, Calif.

[3] Minovitch, M. A., "Utilizing Large Planetary Perturbations for the Design of Deep-Space, Solar-Probe, and Out-of-Ecliptic Trajectories," TR 32-849, 1965, Jet Propulsion Lab., Pasadena, Calif.

[4] Deerwester, J. M., "Jupiter Swingby Missions to Nonspecific Locations in Interplanetary Space," TN D-5271, June 1969, NASA.

Determination of Rocket Ignition Induced Silo Transient Pressures Using an Expansion Tube

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Nomenclature

A = area
$B = \gamma RKA_2{}^*(To_2)^{1/2}/V$, see Eq. (2)
$K = \{(\gamma/R)[2/(\gamma+1)]^{\gamma+1/\gamma-1}\}^{1/2}$
P = pressure
R = gas constant
t = time
T = temperature
V = volume
γ = ratio of specific heats

Superscripts
* choked conditions

Subscripts
cv control volume
o stagnation condition
1 supply tube
2 combustion chamber

Introduction

TRANSIENT pressure pulses caused by rocket motor ignition in a silo create overpressures which must be considered in booster and payload designs. Analytical studies of silo overpressures were first made by Broadwell and Tsu.[1]

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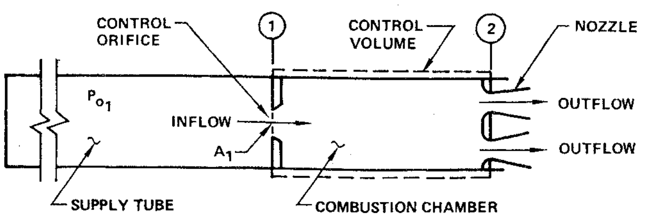

Fig. 1 Ignition transient analysis model.

Their approach has been extended to include exhaust gas-air mixing, heat-transfer and small missile/silo annular area changes due to missile contour changes or a missile support system. However, due to the complex nature of the flow, the technique cannot predict base pressures or adequately predict the effect of large flow area discontinuities on silo pressures. These data must be obtained experimentally. This Note describes the test technique used to obtain data required to design a proposed missile support system.

Several candidate test techniques were reviewed, including small solid rocket grains, reflected pressure region of a shock tube, shock tube with combustion driver and an expansion tube. Rocket grains were expensive and ignition transients nonrepeatable. The reflected region of a shock tube produced very rapid transient pressures which limited model size flexibility (due to time scaling) and also presented repeatability problems. A shock tube combustion driver had been successfully used to simulate the ignition transient from Saturn S1-C.[2] However, this technique was dropped because of cost and schedule constraints.

The expansion tube technique was selected because it closely duplicated three important parameters required for transient pressure simulation: motor ignition pressure transient, product of silo pressure and gas velocity, and sound speed. Furthermore, the expansion tube provided a means for obtaining the required data on a timely and relatively inexpensive basis.

Rocket Motor Ignition Transient Simulation

Simulation of the rocket motor ignition transient required a controlled chamber pressure increase followed by a period of constant pressure. This was accomplished by using the expansion tube device sketched in Fig. 1. By selecting a high supply tube pressure, the pressure ratios across the flow areas at stations 1 and 2 exceeded critical ratios almost immediately. This simplified the analysis since choked flow relationships could be used to obtain a control volume energy balance equation of the form[3]

$$\gamma K Po_1 A_1^*(To_1)^{1/2} = \gamma K Po_2 A_2^*(To_2)^{1/2} + V/R(\mathrm{d}P_{cv}/\mathrm{d}t) \quad (1)$$

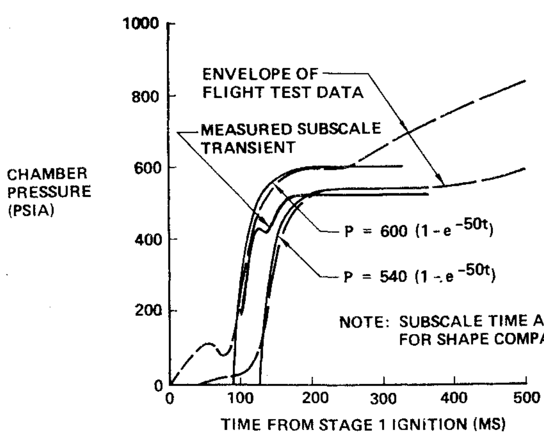

Fig. 2 Comparison of ignition transients.

Since $Po_2 \approx P_{cv}$ and $To_1 = To_2$ (adiabatic flow), Eq. (1) may be rearranged and integrated to yield

$$P_{cv} = Po_1 A_1^*[1 - \exp(-Bt)]/A_2^* \quad (2)$$

Note from Eq. (2) that the rise time could be set to the desired value by varying B. In this case B was most easily changed by adjusting the chamber volume. Note also that, following the transient, P_{cv} remains constant as long as Po_1 is constant. Since Po_1 was constant until the first rarefaction wave reflected from the supply tube end wall and returned, the desired steady state duration of P_{cv} was achieved by selecting the appropriate supply tube length.[4]

In Fig. 2, Eq. (2) with appropriately selected constants is compared with measured flight test ignition transients. Also shown is the measured transient (suitably time-scaled) from the subscale tests discussed below. The results demonstrate the ability of the expansion tube device to produce the required simulation.

The full scale chamber sound speed is over 3500 fps. However, as exhaust products mix with the air and transfer heat to the silo, the sound speed decreases to approximately 3200 fps. Excellent simulation of the silo sound speed was achieved by using room temperature helium in the subscale tests.

The product of silo gas pressure and velocity must be duplicated to achieve transient pressure duplication. By equating the test gas pressure-velocity product to that of the full scale motor and solving for mass flow, it was determined that the required model flow rate was 2.3 times the scaled full scale mass flow at the same chamber pressure. This was verified in the test.

Test Facility and Instrumentation

The test scale factor was 0.048. Items scaled included the silo, missile support system and missile. The simulated Stage I motor consisted of a plenum, supply tube, control orifice, double diaphragm assembly and helium system for charging the supply tube prior to each run. The double diaphragm technique was used to provide a reliable and accurate means of controlling diaphragm burst pressure, and steady-state plenum chamber pressure. The chamber pressure rise time was controlled by plenum volume. The missile model was anchored to a test stand. A false floor, supported by pedestals, was raised or lowered to simulate different silo depths. Fast response pressure transducers were installed in the missile plenum, missile base and silo wall. Oscillogram data were reduced by manual measurements of analog pressure amplitudes. Real time correlation was acomplished by recording a precision 1 kHz time signal on one channel of the tape recorder.

Results and Conclusions

Data from a typical test run are shown on Fig. 3. Time zero is the time when missile combustion chamber pressure first begins to rise. Initial spikes were caused by compression

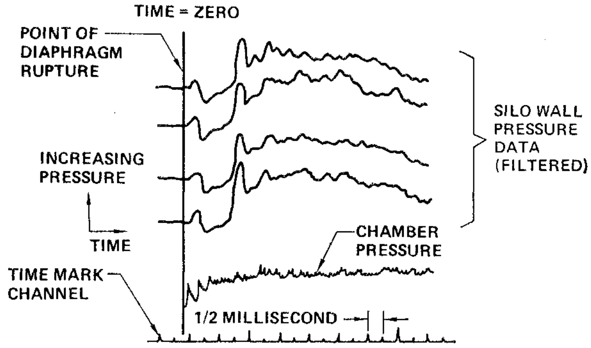

Fig. 3 Typical test data.

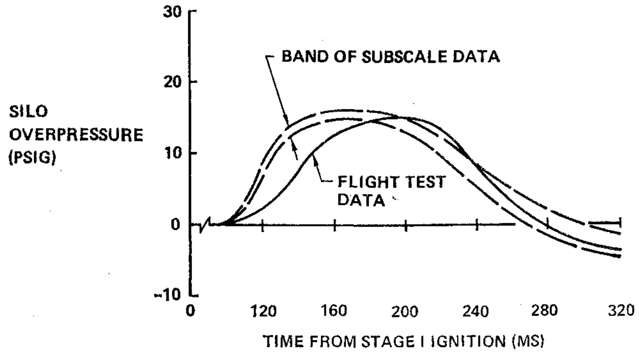

Fig. 4 Silo wall pressure comparisons.

waves originating at the diaphragm station. The balance of the transient is related to motor chamber filling (ignition transient) and in-silo transient gas dynamics. Comparison of a full scale silo wall transient pressure with a faired model pressure transient envelope is shown in Fig. 4. The comparison is considered good. In Fig. 4 (and Fig. 5 below) subscale times were converted to full scale values to allow relevant comparisons of subscale and full scale data.

Missile base pressures obtained with two existing support systems are compared with flight test results in Fig. 5. Since the model ignition transients were steeper than the full scale transients, the subscale pressure were expected to be slightly higher. This was confirmed by the data which shows model pressures to be 2 psig higher than the full scale data for both support systems.

The results demonstrate that the full-scale chamber pressure transient, sound speed and product of silo pressure and gas velocity were closely duplicated in the subscale tests. Thus, it was concluded that the model results formed a valid basis for predicting overpressures required for full scale hardware designs. Actual costs were approximately 20% of estimated costs for either the small rocket grain or shock tube combustion driver techniques.

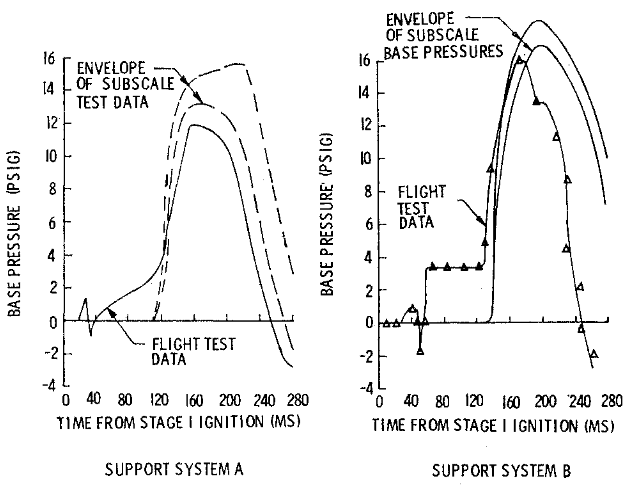

Fig. 5 Base pressures comparisons.

References

[1] Broadwell, J. E. and Tsu, C. N., "Transient Pressures Caused by Rocket Start and Shutdown in Ducted Launchers," *Journal of Spacecraft and Rockets*, Vol. 4, No. 10, Oct. 1967, pp. 1323–1328.

[2] Shrout, L. C. and Green, L. J., "Simulation of Rocket Engine Starting Transients Using a Shock Tube Combustion Driver," AIAA Paper 66-761, Los Angeles, Calif., 1966.

[3] Shapiro, A. H., *Dynamics and Thermodynamics of Compressible Fluid Flow*, 1st ed., Vol. I, Ronald Press, New York, 1953, p. 38.

[4] Wayte, M. J., Dsida, J. A., and Sheeran, W. J., "A Study of Rocket Exhaust Plumes at High Altitudes," AIAA Paper 69-575, U.S. Air Force Academy, Colo., 1969.

Missile Liquid Rocket Propulsion Unit VR35 and Some of Its Development Problems

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THE propulsion unit developed for the air-to-ground missile carried by the Swedish strike aircraft Viggen (AJ 37), Fig. 1, is prepacked, hot gas pressurized and has positive expulsion of the storable and hypergolic propellants—inhibited red fuming nitric acid (IRFNA) and Hydyne. Smokeless operation allows the pilot to guide the missile on the line of sight to the target.

The major elements comprising the engine are concentric propellant tanks, a solid propellant gas generator, an electric safe-arm igniter, a pressure relief valve, an ablative combustion chamber, and the injection system.

The oxidizer is stored within a collapsible annular aluminum bladder and the fuel in a central tank equipped with a welded break away piston for the expulsion. Both propellant tanks are placed within a 1.85 mm thick (0.073 in.) maraging steel case with a burst pressure of 2×10^7 N/m^2 (3000 psi) to take the pressure load and compatability with the missile hull. The principal specifications for the propulsion unit (VR35) are given in Table 1.

The thrust-time curve is characterized by a boost phase with almost constant thrust as long as the gas generator is burning. After this period, when roughly half of the propellants are expelled, the expulsion proceeds during the sustain phase with a successively decreasing tank pressure and

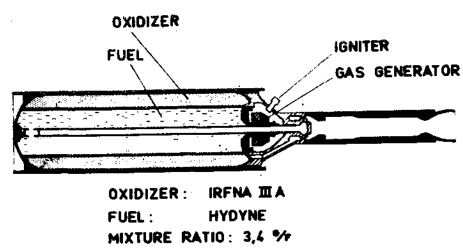

Fig. 1 Schematic view of VR35.

Table 1 Specifications on VR35

Diameter	0.3 m (12 in.)
Length	1.77 m (70 in.)
Total weight	127 kg (280 lb)
Propellant mass fraction	0.59
Operating temperature range	−50°C (−58°F) to +65°C (+149°F)

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