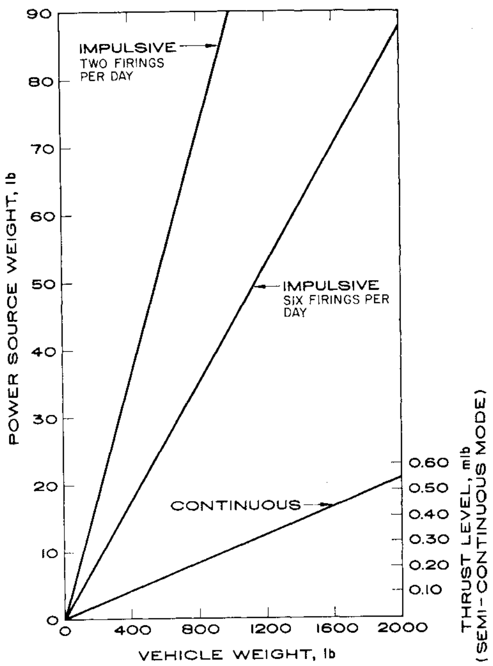

Fig. 4 Power supply weight and thrust level vs satellite weight.

result as firing twice daily at the nodal points. This represents a total daily thrust time of 104 min, corresponding to an impulse of 9.0 lb-sec.

Figure 3c illustrates the effectiveness of applying thrust 3 hr before and 3 hr after a node crossing as well as at the node, a total of six firings per day. The thrust time per firing required to control the vehicle was found to be 18 min for a total thrust time of 108 min per day and an impulse of 9.7 lb-sec.

It is apparent, therefore, that the number of firings executed each day can be increased and that the individual thrust periods will be shortened as a result. Another possible correction mode which was investigated was continuous thrusting at lower thrust levels. Figure 3d shows the result of a continuous 0.15-mlb thrust. This level is seen to be sufficient to control the satellite. However, thrust applied near and at the peaks of the oscillations is essentially useless due to the gyroscopic effect. Therefore, a correction curve again using 0.15 mlb continuous thrust, except for a 2-hr period when passing each peak, was plotted. Figure 3e illustrates the successful controlling of the satellite by this technique. The total impulse required, however, has increased to 10.8 lb-sec.

Conclusion

From the foregoing analyses, it is seen that the solar-lunar perturbations on a stationary satellite can be counteracted by thrusting alternately in the North and South (N-S) direction perpendicular to the satellite orbit. As shown, the most efficient N-S correction mode, in terms of total impulse, is thrust two times daily at the nodes. There are, however, other thrust sequences, such as multiple or continuous firing which can effectively control a stationary satellite.

The effect of thrust sequence on ion engine control system power supply weight is shown in Fig. 4. In both the pulsed and continuous correction modes, the weight of the satellite to be controlled determines the weight of the power supply (e.g., solar cells and batteries). In the latter case, the satellite weight also determines the required thrust level. Figure 4 shows, for the relatively high-thrust, impulsive-type control system the substantial weight savings ($\sim$50% for a 1000-lb satellite) afforded by correcting six rather than two times daily. The optimum control system operation, however, is seen to be the continuous (or semicontinuous) thrust mode. Since this mode of operation requires a continuous low thrust level (0.3 mlb for a 1000-lb satellite), the ion engines can be operated efficiently directly from the solar cells, thereby eliminating the need for batteries. In addition, the engine and power conditioning equipment will be somewhat reduced in weight because of the lower thrust level. For a 1000-lb vehicle, the power supply weight is reduced from 45 lb for the best pulsed mode operation to 10 lb for the continuous correction, representing an almost 100% reduction in total control system weight. As the satellite weight increases, the advantage of continuous correction becomes even more pronounced.

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Fin Temperature Measurements: Nike Cajun Sounding Rocket

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Introduction

THIS report summarizes results of tests conducted on stabilizing fins developed for the Nike Cajun sounding rocket. Since these tests were completed, there have been approximately 85 flights using these fins in the NASA Sounding Rocket Program (50 Nike Cajuns and 35 Nike Apaches) with no evidence of structural failure. Payload weights (gross) have varied between 50 and 95 lb, and the rockets have been launched with effective launch angles varying between 2° and 17° from the vertical.

Tests

Structural load tests based upon anticipated flight loads and temperatures indicated that a panel and shroud fin design was capable of withstanding 3890 lb per fin at destruction at room temperature; at elevated temperature, 210°F, the fin withstood 1800 lb load per fin without reaching the yield point.[1]

A Nike Cajun two-stage solid propellant sounding rocket was flight tested to determine the temperature of the second-stage panel and shroud design fins under flight conditions. A schematic of the Nike Cajun sounding rocket (see Ref. 2) is shown in Fig. 1. The warpage problem previously associated with panel and shroud fins[3] is avoided by the use of an ex-

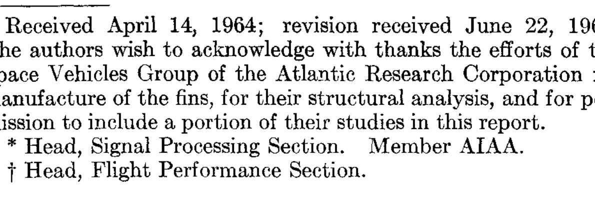

Fig. 1 Schematic of Nike Cajun sounding rocket.

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truded aluminum sheet. The panel and shroud fin design and location of temperature gages is shown in Fig. 2. Longitudinal fairing strips were used to protect the wires leading from the fin temperature gages to the telemetry section in the payload. Fin temperature gage no. 1 was mounted on a fin panel surface; no. 2 was recessed, being mounted in a region where bending loads were not critical.

Results

The results of the fin temperature tests are shown in Fig. 3. Fin temperature gage no. 1 ceased sensing temperature at approximately $19\frac{1}{2}$ sec after launch, presumably being stripped off the surface by the windstream. A comparison between predicted and actual temperatures to that time shows reasonably close agreement. Temperature gage no. 2 continued to sense temperature in excess of 50 sec with reasonable agreement except for approximately 7 or 8 sec immediately after second-stage burnout.

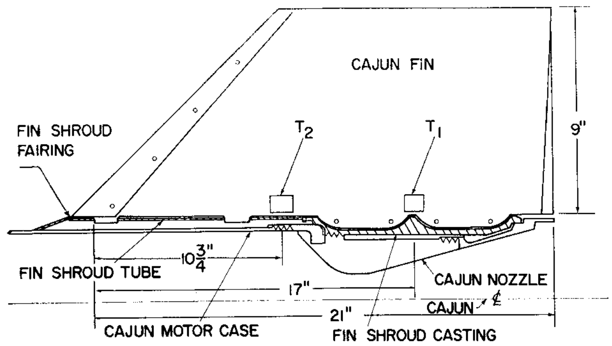

Fig. 2 Panel and shroud fin design; location of temperature gages.

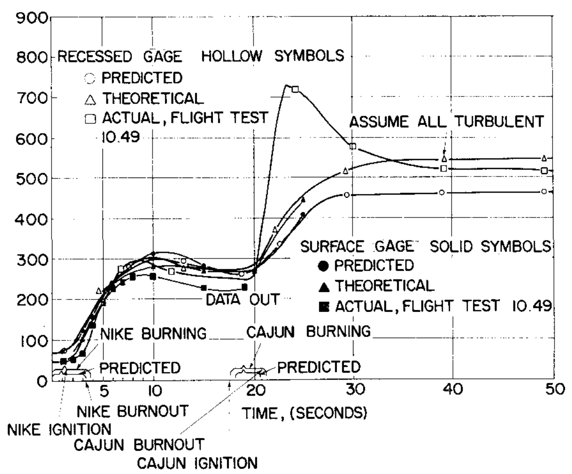

Fig. 3 Results of fin temperature tests.

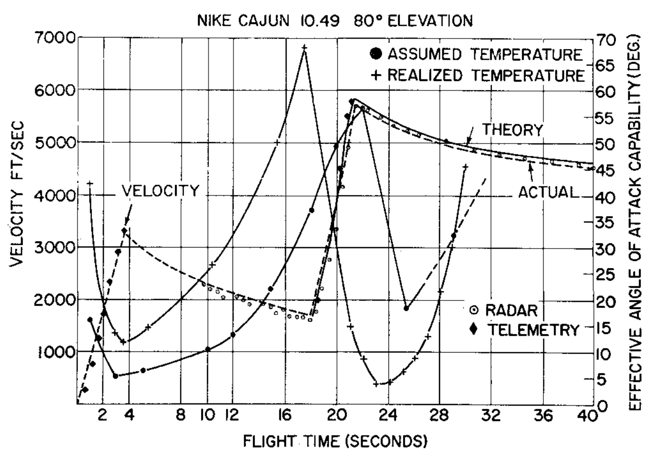

Fig. 4 Angle-of-attack capability.

Since wind gusts, stage separation, or thrust misalignment can cause disturbances in the flight path resulting in the rocket flying at some angle of attack until the disturbing force is overcome, calculations were made to determine the angle of attack capability based on the realized temperature. These calculations are compared with those based on design assumptions in Fig. 4. Two critical periods occur, at Nike burnout and at Cajun burnout. At both critical periods the fins can withstand loads approximating 4° angle of attack. In flight, the fin temperature appears to undergo a momentary surge at second-stage burnout which may be attributable either to 1) transient effect of stagnation temperature prior to heat conduction into the fin mass proper, or 2) heat conduction from the nozzle rather than frcm aerodynamic heating.

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Some Measurements of the Effects of Ring Baffles in Cylindrical Tanks

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Relatively large lateral forces can be developed by liquid sloshing in the propellant tanks of large rocket boosters. Liquid resonant frequencies, if uncontrolled, could also give dynamic coupling with structural components and with the rocket's control system. For a given tank, the first-mode liquid resonant frequency can be substantially altered and its motion damped by means of either tank compartmentation or ring baffling of various types.

The present paper compares the effects of several types of ring baffles in a partially filled cylindrical tank undergoing forced vibration. Experimental resonant frequencies and damping values are obtained in a cylindrical tank at various depths below the liquid-free surface. This study may be regarded as an extension of the preliminary data given in Ref. 1, with particular emphasis on the effects of perforations in flat ring baffles. The equipment and procedures used are similar to those in Ref. 2. A 1.2-ft-diam rigid-wall cylindrical tank is supported by four dynamometers. The ring baffles were 0.018 to 0.030 in. thick, with width-to-radius ratio fixed at $W/R = 0.157$‡, and with varying degrees of perforation. All types were tested at three amplitudes of translation excitation and several baffle depths.

Liquid Resonant Frequencies

Figure 1 shows the first-mode liquid natural frequencies in terms of the dimensionless frequency parameter $2\omega^2R/a$

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‡ The effects of baffle width are rather well delineated in Ref. 1.