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A FIVE-STAGE SOLID-FUEL SOUNDING-ROCKET SYSTEM

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SUMMARY

A five-stage solid-fuel sounding-rocket system which can boost a payload of 25 pounds to an altitude of 525 nautical miles and that of 100 pounds to 300 nautical miles is described. Data obtained from a typical flight test of the system are discussed.

INTRODUCTION

Knowledge of the structure and composition of the extreme upper atmosphere is still quite limited. Need continues for means of exploring this region. Artificial earth satellites, sounding rockets, and balloons have been and will continue to be used for this purpose for all of these tools have been found to be complementary rather than competitive in obtaining the information that they can supply.

Described in this report is a five-stage solid-fuel sounding-rocket system which is capable of carrying sounding probes to altitudes of several hundred miles. Calculations have been made which show that an instrument package weighing 25 pounds (complete with its external structure) can be propelled to an altitude of about 525 nautical miles and a 100-pound package can be propelled to about 300 nautical miles. In addition to the presentation of calculated performance parameters for this system, the results of a typical flight test are described. In this flight test an instrument package having a total weight of 49 pounds was boosted to a measured maximum altitude of 409.8 nautical miles ($\approx 2.5 \times 10^6$ ft).

This five-stage rocket system was originally designed by the Langley Pilotless Aircraft Research Division and the Langley Engineering Service Division for use as a hypersonic aerodynamic research vehicle. With appropriate modifications to the structure and firing sequence, it has been converted to the altitude sounding rocket described herein.

GENERAL CHARACTERISTICS OF SOUNDING-ROCKET SYSTEM

Test Vehicle

A specific test vehicle which has been flight tested is described L
as a typical example of the sounding-rocket configuration. Different 2
types of test packages could alter the size and shape of the last stage 7
of this configuration. Such changes, of course, must be made within 6
the limits imposed by structural loads, stability, and aerodynamic heating.

A sketch of this five-stage configuration is shown in figure 1. The first-stage rocket motor is an Honest John missile rocket motor, JATO unit M6. The second- and third-stage motors are Nike missile boosters, JATO unit X216A2. The fourth-stage rocket motor is a Recruit, JATO unit XM19, and the fifth-stage rocket motor is a JATO unit T55. The instrument package is fastened to the forward end of the fifth-stage rocket motor. A sketch of the fifth stage and instrument package of the vehicle flight tested is shown in figure 2.

The total launch weight of this test vehicle is about 7,100 pounds. Photographs of the rocket system mounted on a zero-length boom-type launcher are shown as figures 3 and 4. Figure 5 is a photograph of the system just after launch.

The first three stages of the rocket system are stabilized by fins. The fins are mounted inline (no interdigitation) and have thin double-wedge or hexagonal airfoil sections. The last two stages are stabilized by 10° half-angle conical frustums (or conical flared skirts). The fifth-stage flare also serves as a nozzle extension (single wall flare and nozzle).

The material of which the fifth stage of the vehicle is constructed is noted in figure 2. The fifth-stage rocket motor is enclosed in an Inconel shell; the steel cases of all other motors serve as the outside skin. The fourth-stage flare is fabricated from 1/4-inch-thick magnesium. Second- and third-stage fins are constructed of magnesium. First-stage fins are standard Honest John missile fins. Couplings between stages are constructed from magnesium except the coupling between stages four and five which is made of steel.

Flight Program

The rocket system, which is ground launched, is designed so that the burned-out first-stage motor will drag separate from the remaining stages which then coast for about 5 seconds before a delay igniter, actuated at launch, fires the second stage. Near burnout of the second-stage motor, a delay squib (also actuated at launch) operates to release a lock pin. This pin locks the second and third stages together to prevent drag separation of those stages at first-stage burnout. After burnout, the second stage (now unlocked) drag separates from the rest of the system. A sketch of the lock pin is shown in figure 6.

The last three stages then coast for about 20 seconds. A preset mechanical timer (started at time of launch by an inertia switch) then fires the third stage. Shortly after third-stage burnout, the fourth stage is fired by a delay igniter which is actuated by the mechanical timer at third-stage ignition.

In order to prevent premature separation, the fourth stage is locked to the third stage by a disk which is threaded to both the fourth-stage nozzle and an adaptor coupling mounted to the third-stage motor headcap. This thin disk is designed to collapse under pressure of the fourth-stage-motor exhaust gases; hence, the stages are unlocked at fourth-stage ignition. Firing of the fourth stage cocks a pressure switch which closes as the motor chamber pressure decreases at fourth-stage burnout. Closing of this switch fires the fifth-stage igniter. The fifth stage is locked to the fourth stage by a collapsible disk similar to that used between the third and fourth stages. A sketch of this locking arrangement is shown in figure 7.

After burnout, the empty fifth-stage motor and the instrument package coast together through apogee to impact.

Protection From Aerodynamic Heating

In order to protect the fins of the second and third stages from aerodynamic heating, the leading edges are capped with Inconel sheet. The second-stage fins are wrapped with a 1/32-inch-thick Inconel sheet which extends about 1 inch back from the leading edge. The third-stage fins have the 1/32-inch Inconel sheet extended 2 to 3 inches farther rearward on the fins to cover the forward wedge section and have an additional 1/16-inch-thick cap which extends about 1 inch back from the leading edge.

The fifth-stage rocket-motor case is enclosed in an Inconel shell with an air gap between motor and shell to prevent the motor grain from becoming too hot. The temperature rise on the motor cases of the other stages is sufficiently low that no protection is required.

CALCULATED PERFORMANCE OF SOUNDING-ROCKET SYSTEM

Trajectory

The nominal no-wind trajectory from launch to an altitude of 150,000 feet is shown in figure 8 for the rocket system shown in figure 1, launched at an elevation angle of 80°. The trajectory beyond this altitude is approximately ballistic. This trajectory represents a compromise to a theoretical optimum due to consideration of various conflicting factors. For example, since the vehicle contains no guidance or destruct system, 80° is considered to be the maximum launch angle consistent with safety from a launching site with adjacent populated land areas, such as the Langley Pilotless Aircraft Research Station at Wallops Island, Va., from which flight tests of the system were made. (Launch angles closer to vertical could, of course, be made under some circumstances; shipboard launch, for example.) The fourth and fifth stages have a low degree of static stability and it is desirable, therefore, to have them propulsive at low altitude to prevent large disturbances to the flight path due to various possible misalinements, gusts, and so fourth. On the other hand, aerodynamic-heating problems require higher altitude firings of the last stages. The trajectory shown is believed to represent a reasonable compromise for an optimum flight program when such factors are considered.

The velocities indicated in figure 8 are for the rocket system with a 49-pound instrument package mounted on the fifth stage. For the range of payloads likely to be flown with this configuration, only the velocities of the fourth and fifth stages change significantly with changes in payload.

Calculated Maximum Altitude and Velocity

Plotted in figure 9 is the maximum altitude attainable by the fifth stage (and payload) as a function of payload weight. Payload weight is considered to be the entire weight of the test package fastened to the fifth-stage motor and shell assembly and includes instruments, batteries, telemeter, structure, and other equipment. These maximum altitudes are for a system launched at 80°, following the flight program shown in figure 8, and having the same drag as the system shown in figure 1. Wind effects are not included in the data; these effects are discussed subsequently.

Also shown in figure 9 is the performance of a sounding-rocket system flying the same trajectory as the five-stage system except that no fifth (T55) stage is used and the payload is attached directly to the fourth (Recruit) stage. This system might have some advantages for the higher payload range since complexity is reduced and reliability is increased when only four stages are used. The fourth-stage drag was assumed to be that of the fourth and fifth stages of the test vehicle shown in figure 1.

Figure 10 shows the fifth-stage velocity at different altitudes for various payloads. Shown in figure 11 is a typical variation of altitude with flight time for a system with a payload of about 50 pounds and flying along a trajectory with no wind effects.

These data have been computed for a spherical, nonrotating earth with an inverse-square gravitational field. As mentioned previously, the drag assumed for all payload weights was that of the system and instrument package shown in figures 1 and 2. Drag is a small percentage of thrust for all stages for this configuration and drag-to-weight ratio is also fairly small. Stability considerations would preclude any gross changes in the configuration of a payload package from the configuration shown in figure 2. Therefore, drag-to-weight and thrust-to-drag ratios would not change sufficiently to make these data significantly in error for any test package that would likely be flown with this system. Stability, heating, drag, and so forth would, of course, have to be investigated for any unique configuration to determine the exact applicability of the performance data of figure 9 and the flight program of figure 8.

From these data, the five-stage system can be seen to be capable of boosting a 25-pound payload to an altitude of 525 nautical miles and a 100-pound payload to 300 nautical miles. The four-stage system can propel a 50-pound package to an altitude of about 300 nautical miles and a 150-pound package to about 240 nautical miles.

Effect of Wind

The calculated effect of one wind profile on the flight path of the system is shown in figure 12. The wind data tabulated are a series of straight-line approximations to a wind profile likely to be encountered in the eastern United States. The wind is assumed to be either a direct headwind or a direct tailwind and only a horizontal component is assumed. The flight paths are markedly different at the higher altitudes; however, most of this difference arises in the early portion of the flight. An initially steeper flight path results in less bending of the trajectory due to gravity and conversely with the initially lower flight path. There is, of course, a continuous integrated effect of the wind, but the main alteration to the flight path is due to wind effects early in the flight.

The flight-path angle at fifth-stage burnout for the wind effects presented is given in figure 12. The effect of change in flight-path angle at fifth-stage burnout on maximum altitude is shown in figure 13. In these data it is assumed that the early portion of the flight is similar to the trajectories shown in figure 12 and that the velocity data of figure 10 are applicable.

The calculated effect of various constant winds (horizontal component of direct headwind or tailwind) on the flight-path angle at first-stage burnout is shown in figure 14. The particular magnitude of wind velocity at launch that can be tolerated is a unique function of the launching site. However, it can be seen that realistic values of wind velocity can have noticeable effect on the flight path.

Also shown in figure 14 is the approximate effect a steady crosswind (at 90° to the original launch azimuth) would have on altering the flight azimuth from the original launch azimuth. High crosswinds can be seen to require adjustment in launch azimuth in order for the stages of the system to impact where desired.

These wind effects are calculated for winds with constant direction. Actual wind vectors must be integrated over the entire flight path to account properly for wind effects. Since the main wind effects seem to occur early in the flight, the magnitude of this task is not overwhelming for operational launchings.

Stage Impact Points

The nominal impact points for the no-wind trajectory for an 80° launch angle are approximately as follows:

Stage	Nautical miles
One	3
Two	5
Three	60
Four	200
Five	500

These impact points are, of course, different when wind or other disturbances alter the trajectory and are given as a nominal guide when test range requirements are considered.

Acceleration Limits

The solid-fuel rocket motors used in this missile system are short-burning, high-thrust motors. Axial accelerations from 70g to 120g (depending on package weight) are encountered. Transient high-frequency vibrations from 20g to 200g, depending on the structural characteristics of the test package, may occur at ignition of motors in both axial and lateral directions. Instrument packages must be able to function after exposure to loadings of these magnitudes.

ACTUAL FLIGHT-TEST PERFORMANCE

Test

Several successful operational firings have been made with the system described herein from the Langley Pilotless Aircraft Research Station at Wallops Island, Va. Photographs and sketches of one of these vehicles have been presented as a typical sounding-rocket configuration. This vehicle was launched at an 80° elevation angle and was programed to fly the nominal trajectory previously shown in figure 8. A 49-pound payload was carried.

Instrumentation

The test-vehicle instrumentation contained three accelerometers (x-, y-, and z-directions). Data from these instruments were transmitted and ground recorded continuously throughout the 15-minute flight by an FM telemetering system and ground receiving station. The transmitter in the vehicle had a nominal power output of 8 to 10 watts.

A ground-based CW Doppler radar measured vehicle velocity until shortly after second-stage burnout. Two ground-based space position radars tracked the system through fifth-stage burnout. The Millstone Hill experimental radar of the M.I.T. Lincoln Laboratory tracked the fifth stage on the high-altitude portion of the flight.

RESULTS

The velocity of the system was determined from the data obtained by CW Doppler radar for the time period these data were available; from then until third-stage ignition (or from about 20 to 35 seconds) velocity was determined from differentiation of the space-position-radar data. The velocity determined at third-stage ignition was about 50 feet per second higher than the machine calculated estimate of 1,700 fps given in figure 8; the estimated accuracy of the radar velocity data at that point is ±50 fps. After third-stage ignition, velocity was determined from integration of the axial accelerometer data. The accelerometer data gave a velocity at fifth-stage burnout about 150 fps greater than the 13,000 fps calculated. This difference is within the estimated accuracy of the accelerometer data. Differentiation of the space-position-radar data gave essentially the same maximum velocity, which may be somewhat fortuitous since, at the time of maximum velocity, the estimated accuracy of these data is about ±500 fps. The flight path determined by the space position radar was approximately that given in figure 8.

The Millstone Hill radar tracked the fifth stage to a measured apogee of 409.8 nautical miles ($\approx 2.49 \times 10^6$ ft) and a measured impact distance from the launcher of 480 nautical miles. This altitude was somewhat less (about 16 nautical miles) than the altitude predicted by the calculated value in figure 9. Evidently, deviations from the flight path due to wind and thrust misalinements caused the flight path to be slightly lower than the nominal value. The data from the space position radar at the Wallops Island testing station indicated that the flight-path angle at fifth-stage burnout was essentially the no-wind flight-path angle; however, the required deviation ($\approx 2^{\circ}$) is within the accuracy with which flight-path angle can be determined from the radar data at the time of fifth-stage burnout.

Misalinements, probably thrust misalinement, resulted in lateral oscillations with 5g to 6g peaks during fourth-stage burning and 8g to 10g peaks during fifth-stage burning. The magnitude of these accelerations decreased after fifth-stage burnout. There was probably some residual angle-of-attack oscillation since the damping of a flare-stabilized vehicle is quite low and there is very little dynamic pressure after fifth-stage burnout. Calculations indicate that thrust misalinements of the order of $1/10^{\circ}$ could cause the noted oscillations. Lateral disturbances were negligible prior to fourth-stage ignition.

The characteristics of the telemeter and radar signals indicate that the test vehicle flown began to tumble at an altitude of about 300,000 to 400,000 feet. It is possible that there was sufficient residual oscillation from the motion which occurred during fifth-stage burning to result in tumbling when the dynamic pressure approached zero. The test model contained a jettisonable nose cap, however, and the tumbling appeared to begin at the time the cap was removed. Tumbling at such high altitudes has, of course, no significant effect on the trajectory but it could be a handicap in some types of experiments.

CONCLUDING REMARKS

Several successful operational launchings of a five-stage solid-fuel sounding-rocket system have been made. Calculated performance data for this system for various payloads have been presented. This configuration is capable of propelling a 25-pound instrument package to an altitude of about 525 nautical miles and a 100-pound package to an altitude of about 300 nautical miles.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., December 23, 1958.

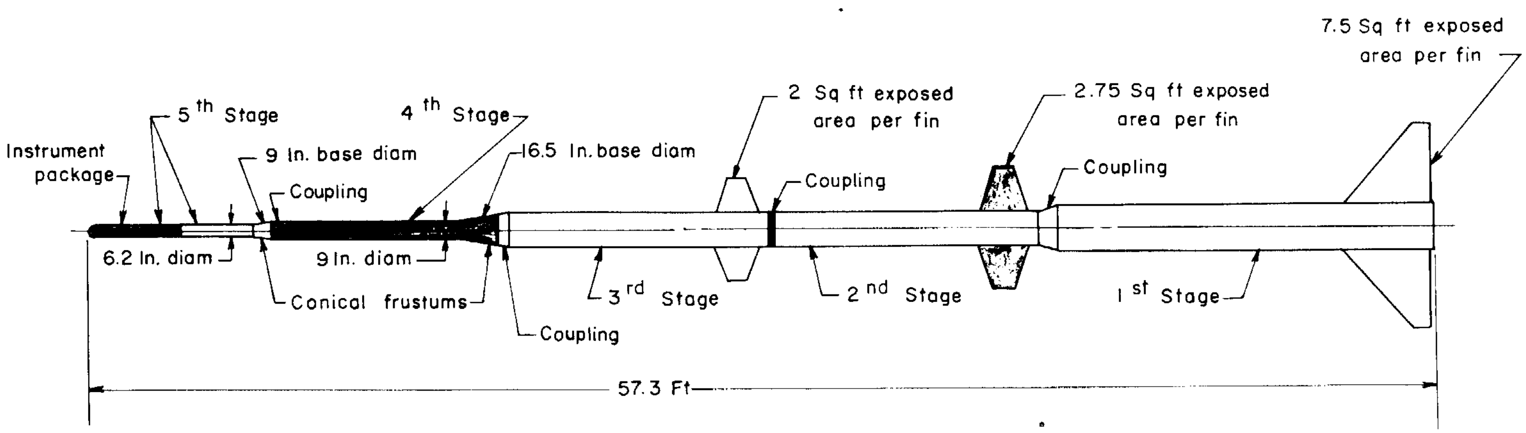


Figure 1.- Sketch of five-stage solid-fuel sounding-rocket system. Total launching weight approximately 7,100 pounds.

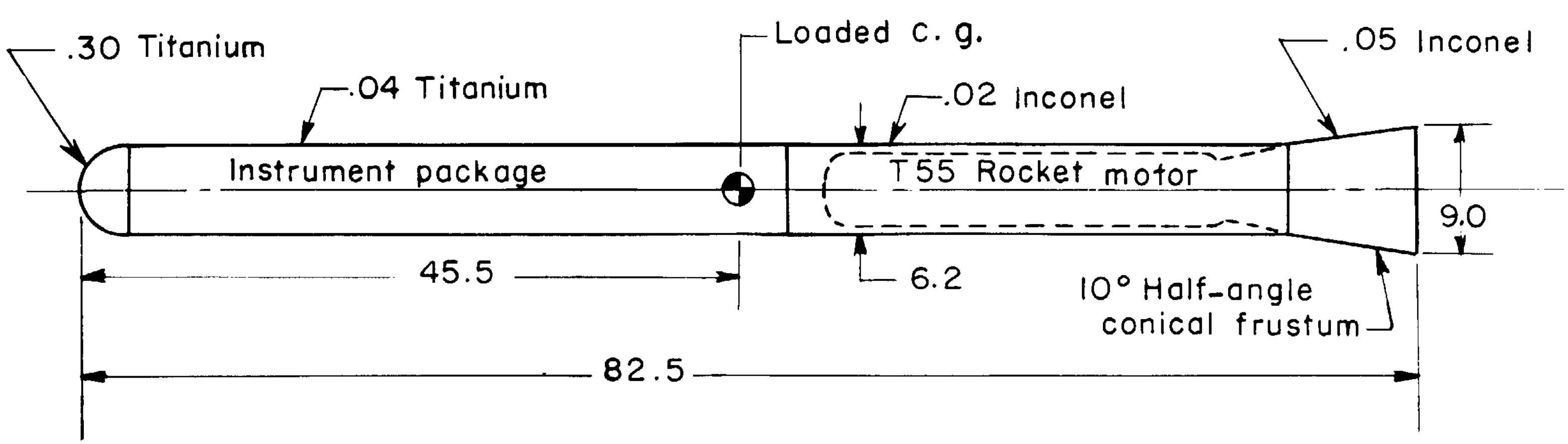


Figure 2.- Sketch of fifth stage of vehicle flight tested. Dimensions are in inches unless otherwise indicated.

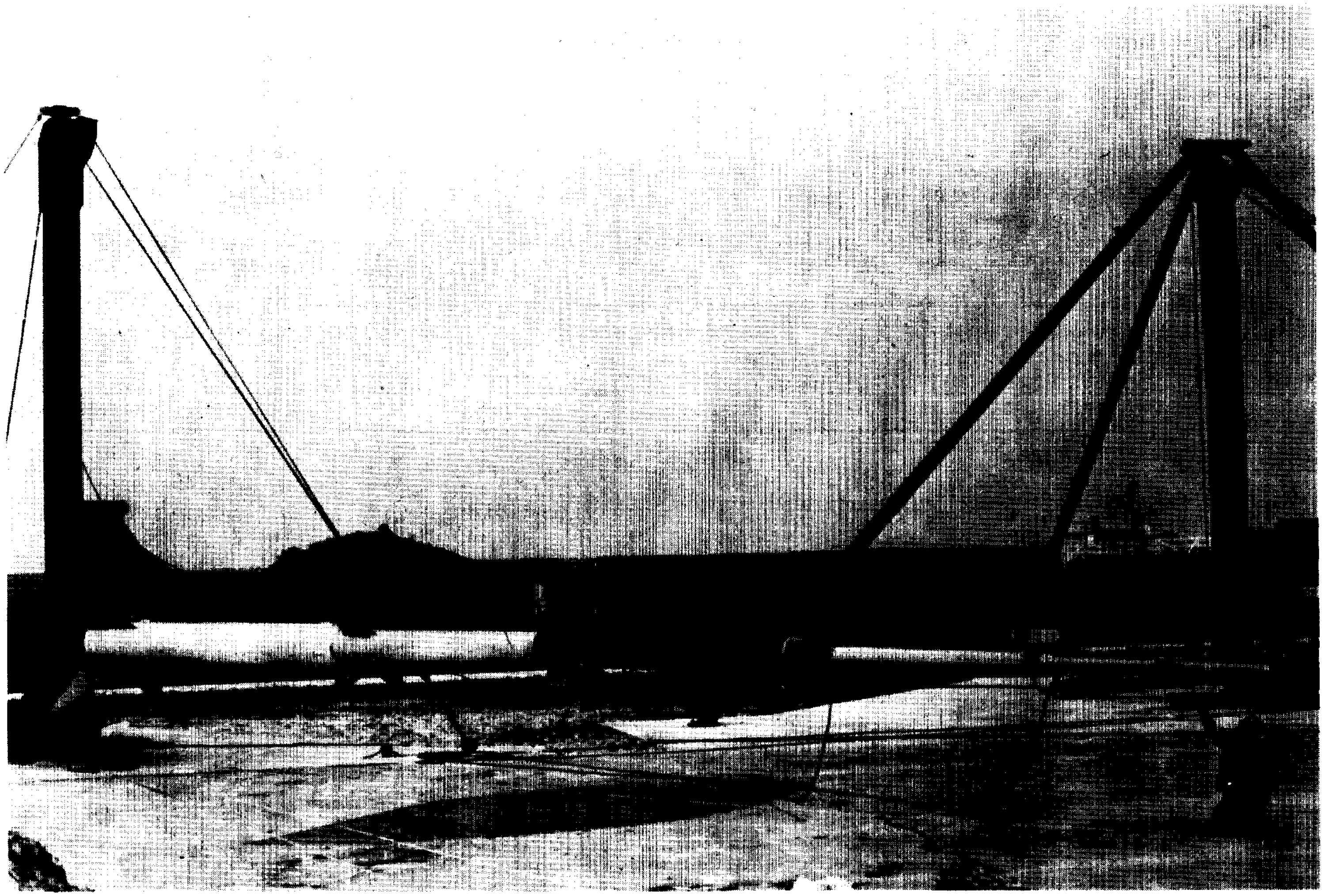
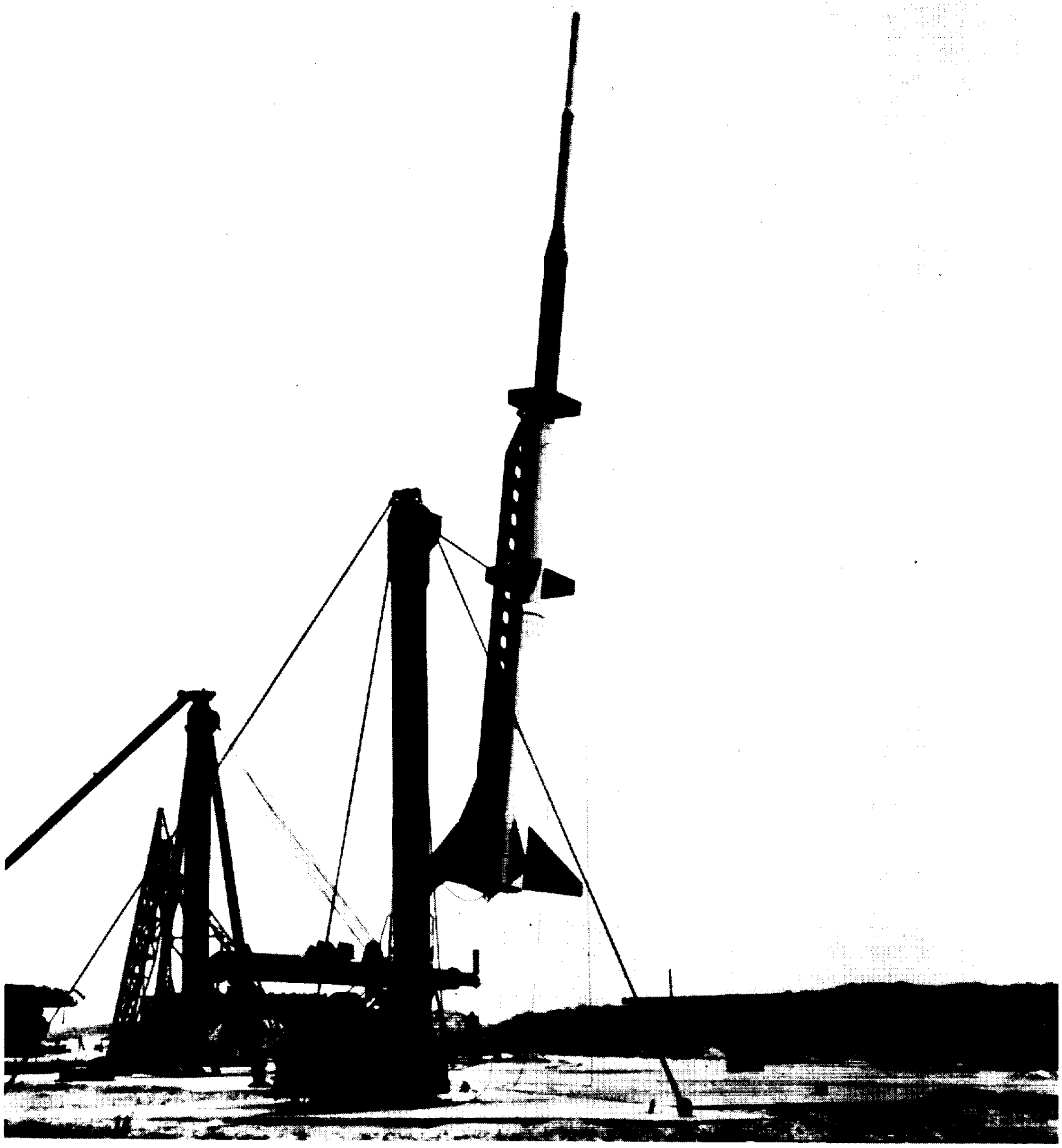


Figure 3.- Test vehicle on launcher. L-58-3414.1



L-58-3415.1
Figure 4.- Test vehicle with launcher elevated in firing position.

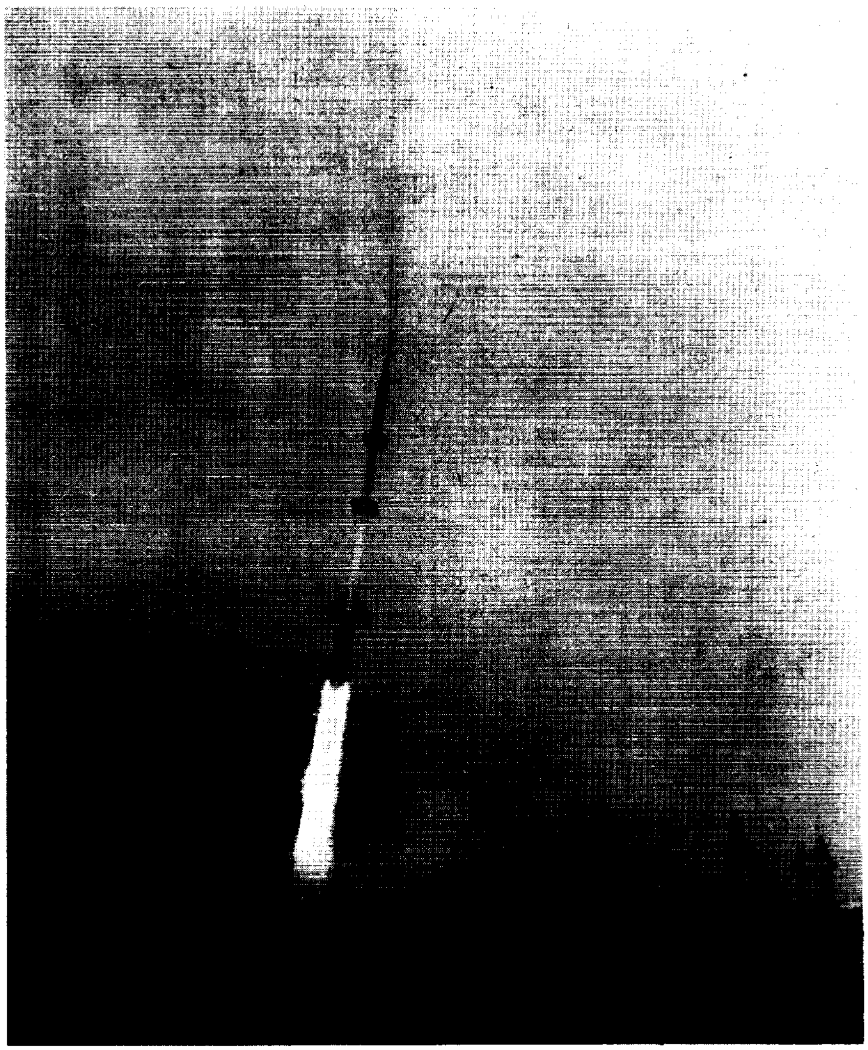


Figure 5.- Test vehicle just after launch. L-58-3423.1

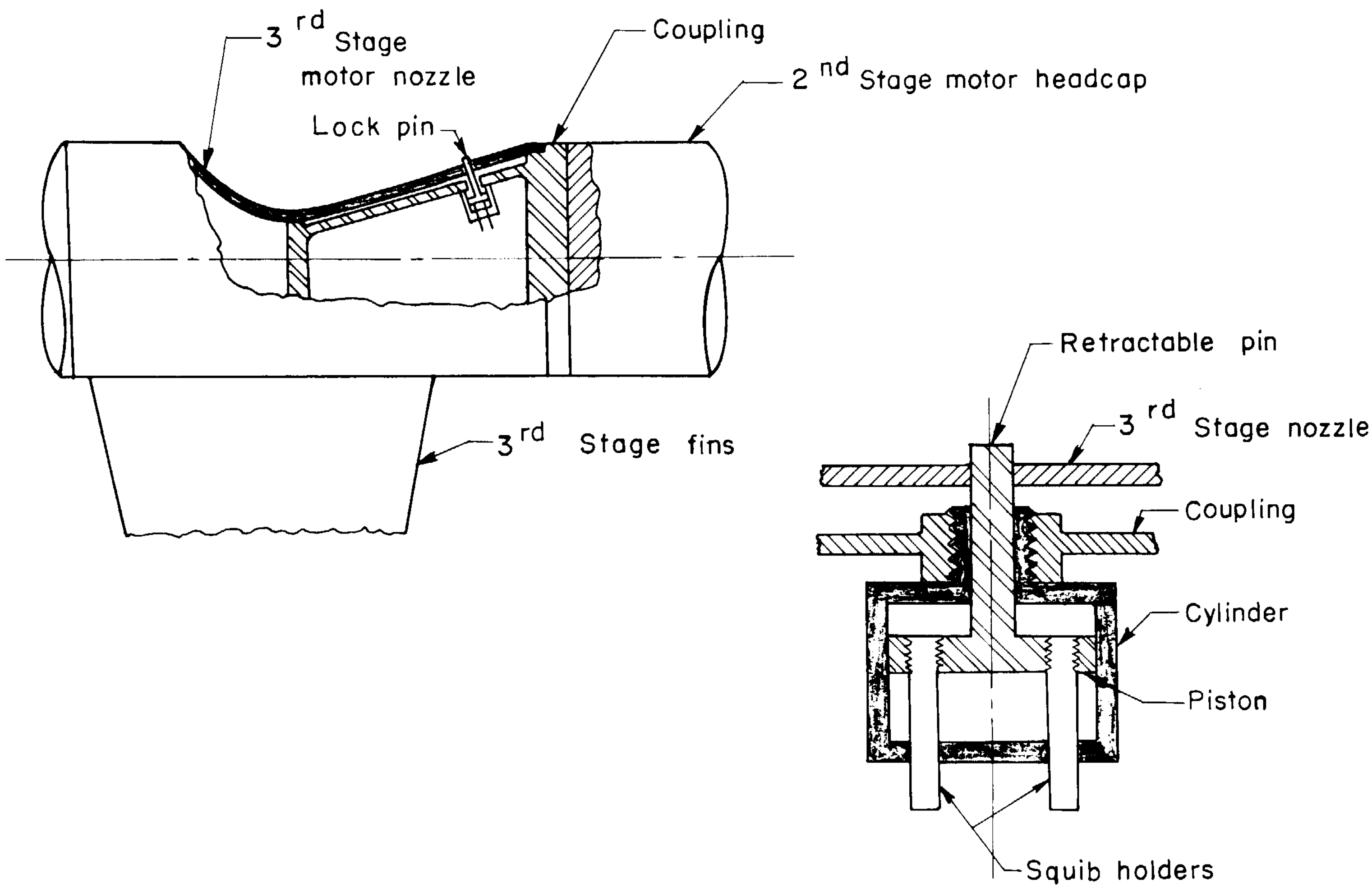


Figure 6.- Coupling and lock pin between second and third stages of test vehicle.

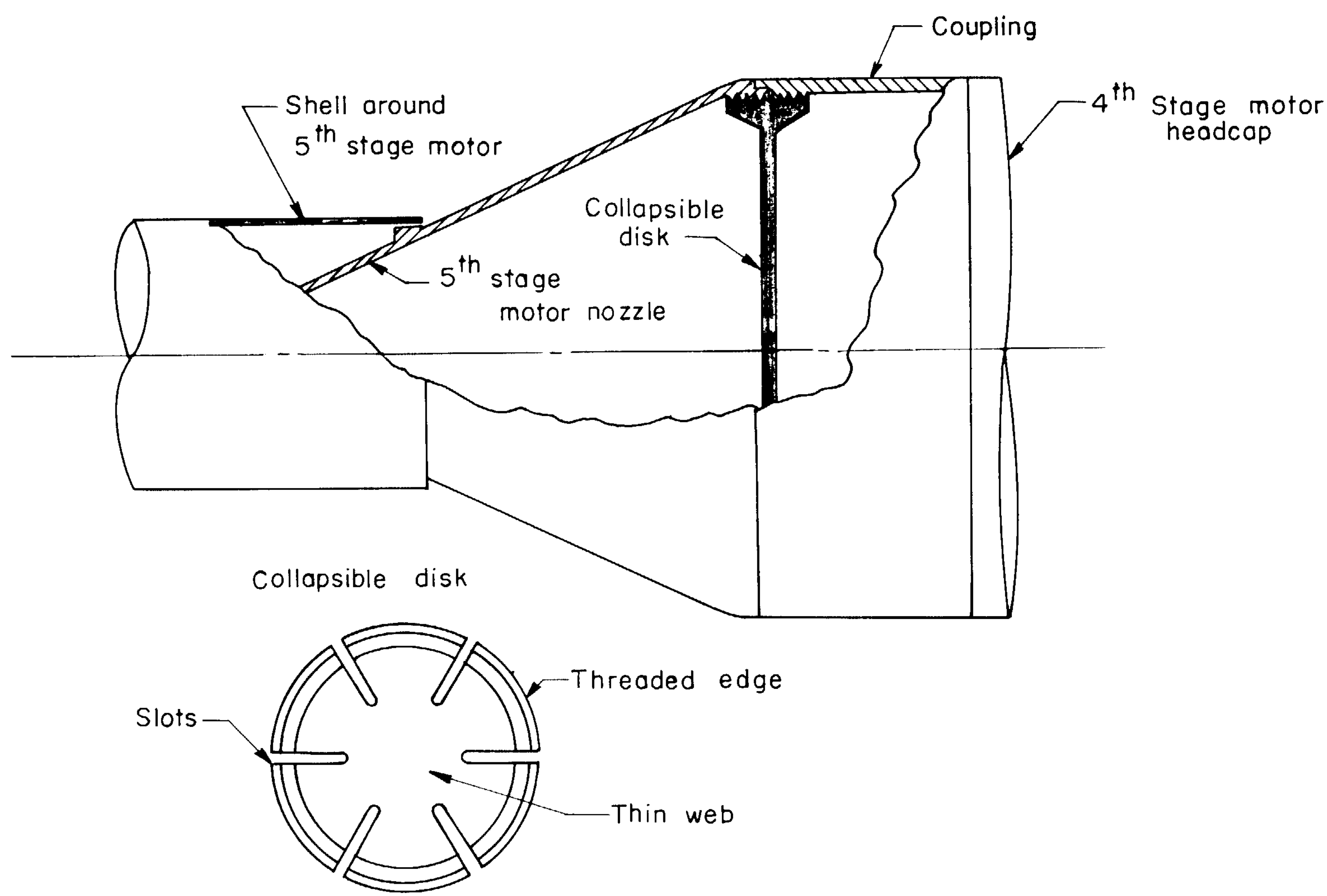


Figure 7.- Coupling and locking device between fourth and fifth stages of test vehicle.

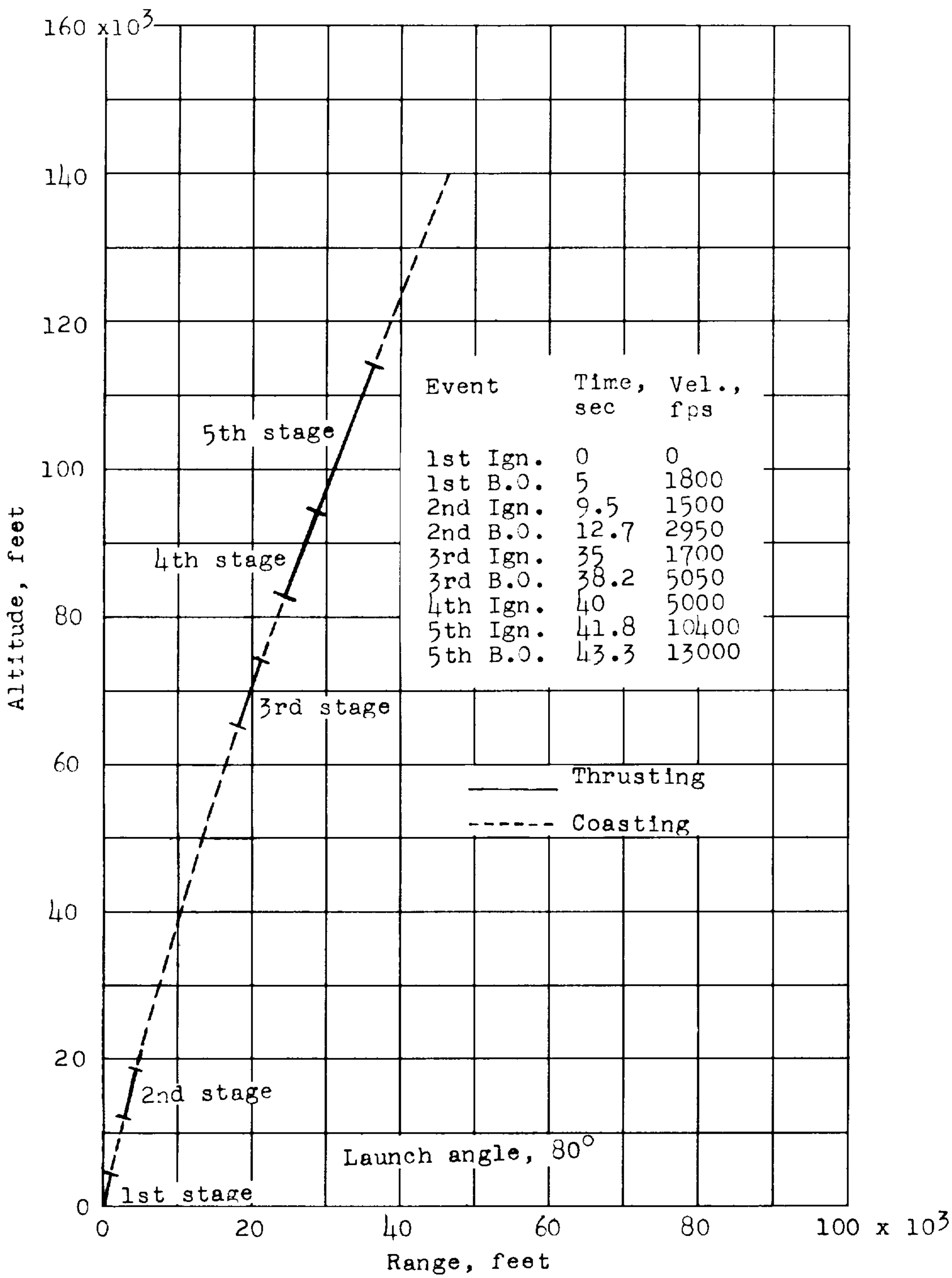


Figure 8.- Nominal no-wind trajectory for vehicle flight tested.

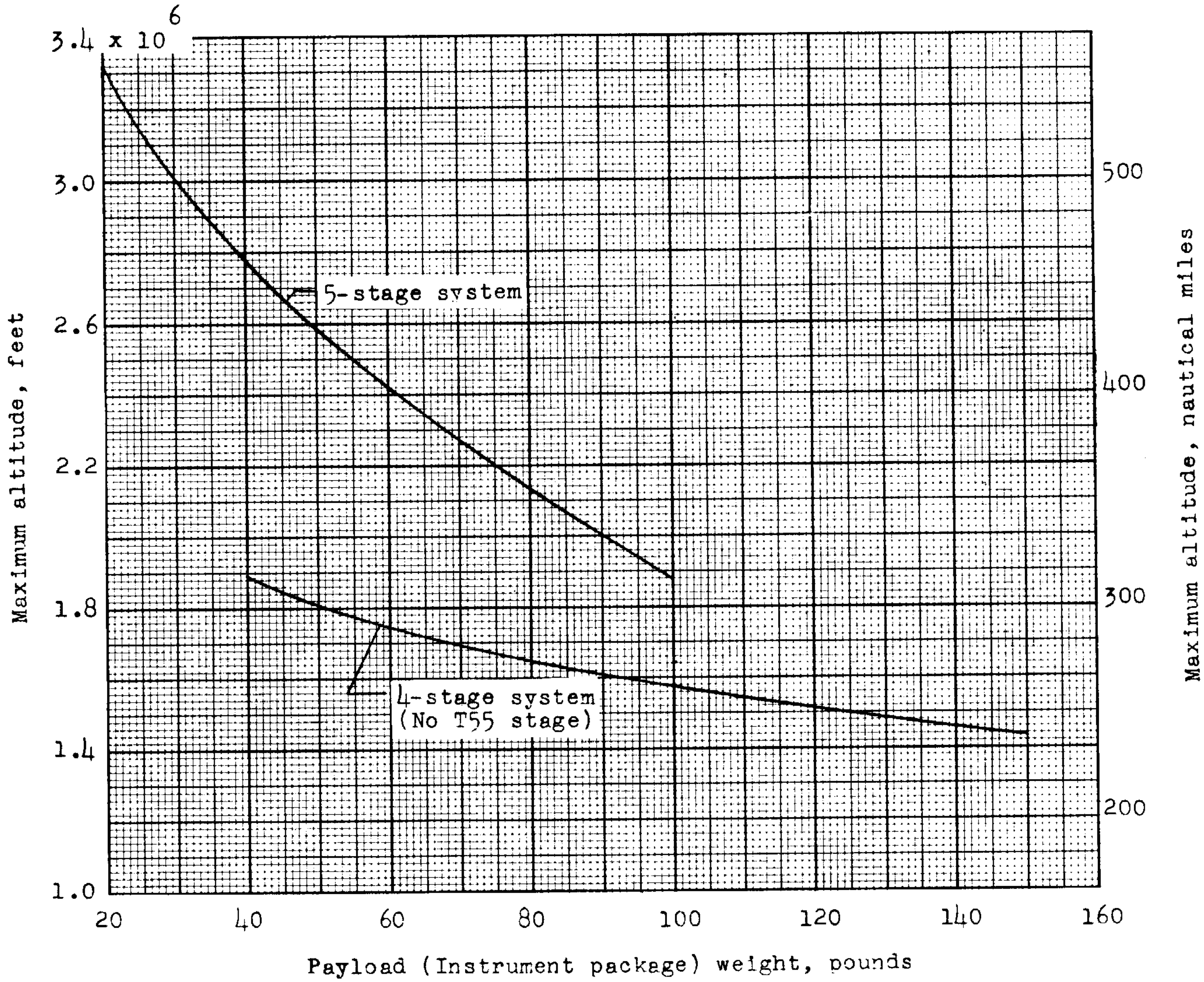


Figure 9.- Maximum altitude as a function of payload. 80° launch angle; no wind.

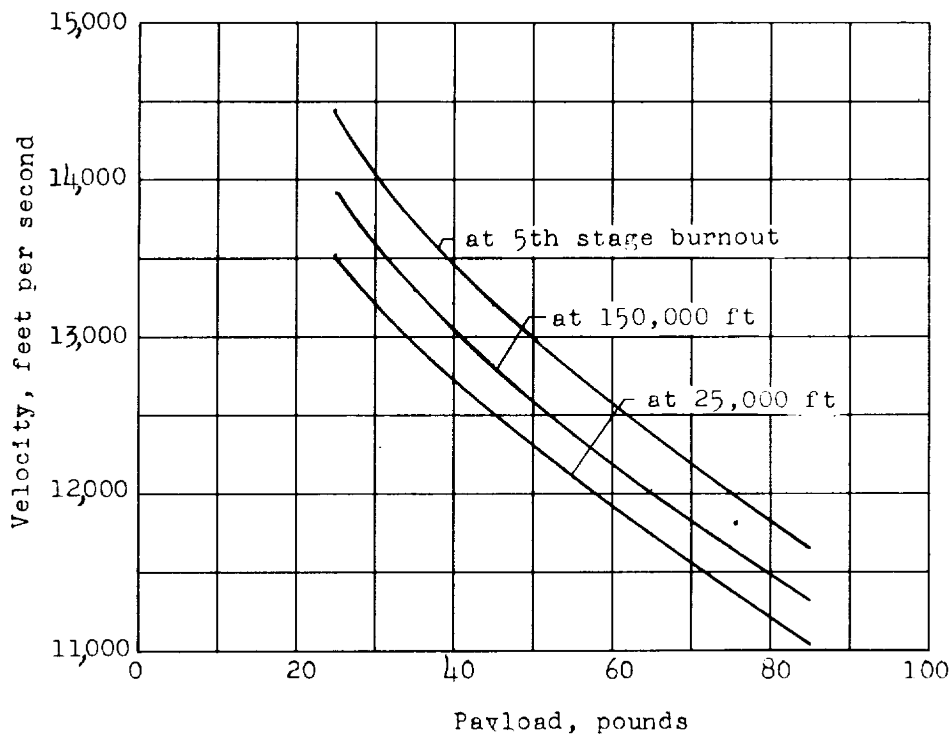


Figure 10.- Velocity of fifth stage as a function of payload.

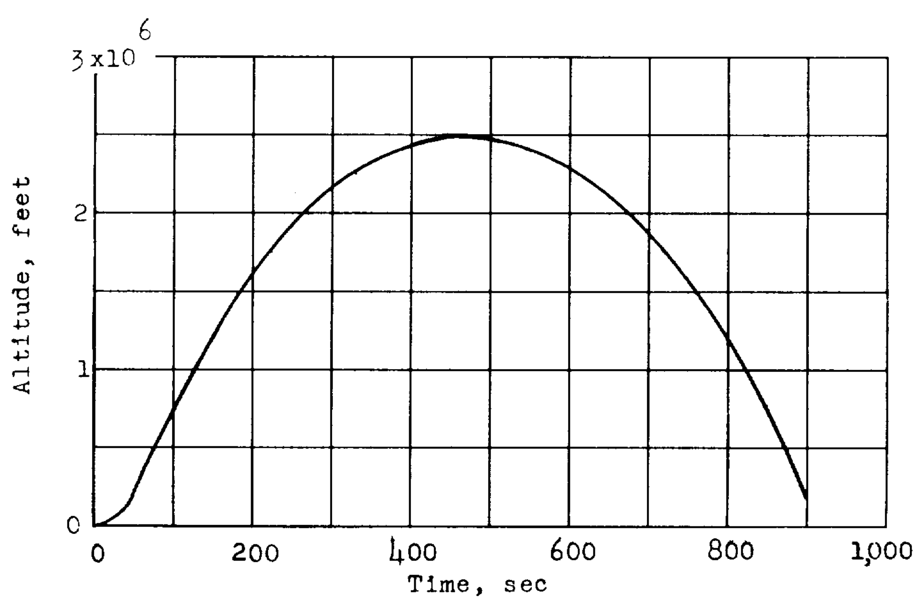


Figure 11.- Typical variation of altitude with time.
Payload ≈ 50 pounds.

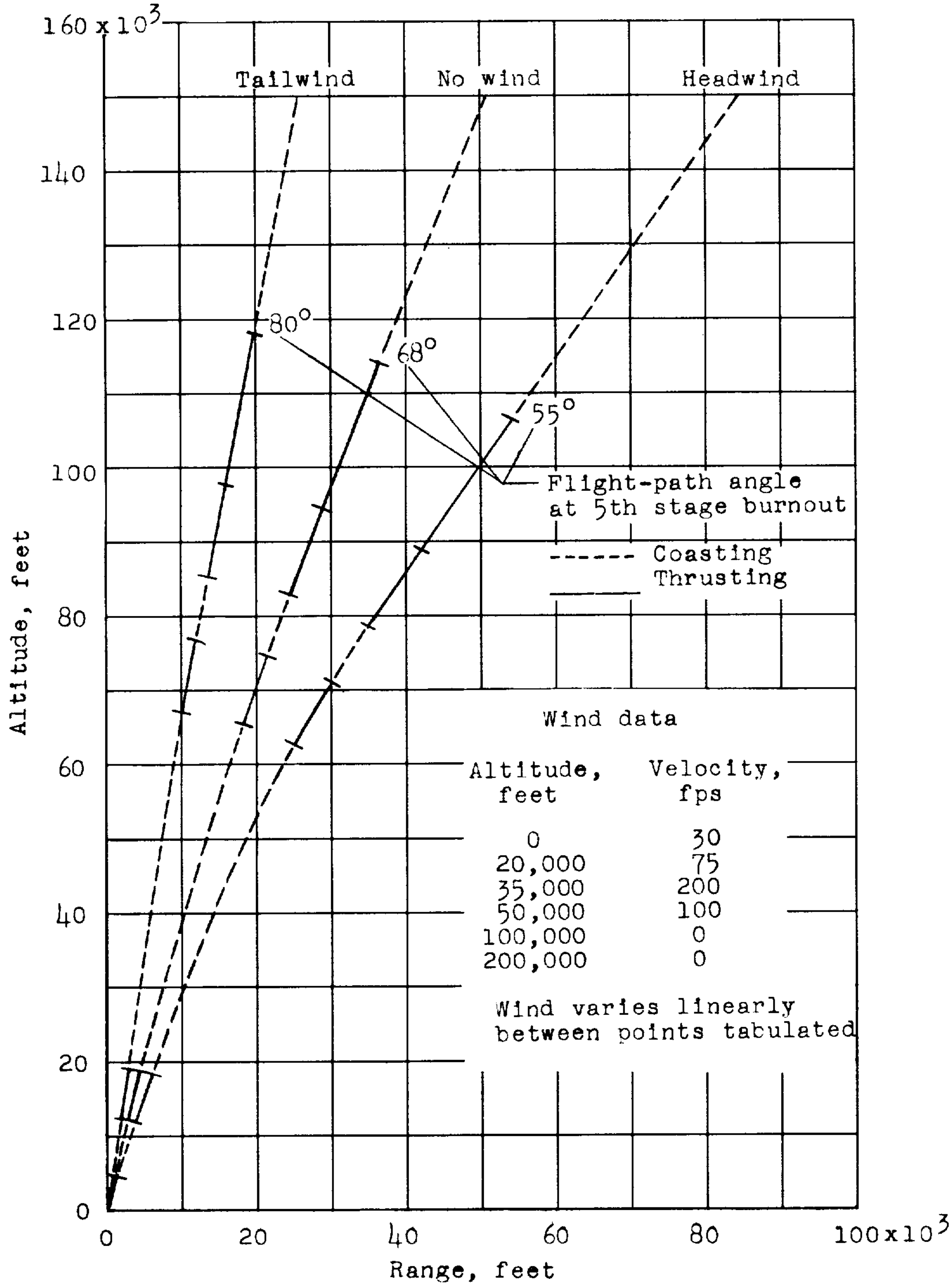


Figure 12.- Effect of constant-direction horizontal winds on flight path. 80° launch angle.

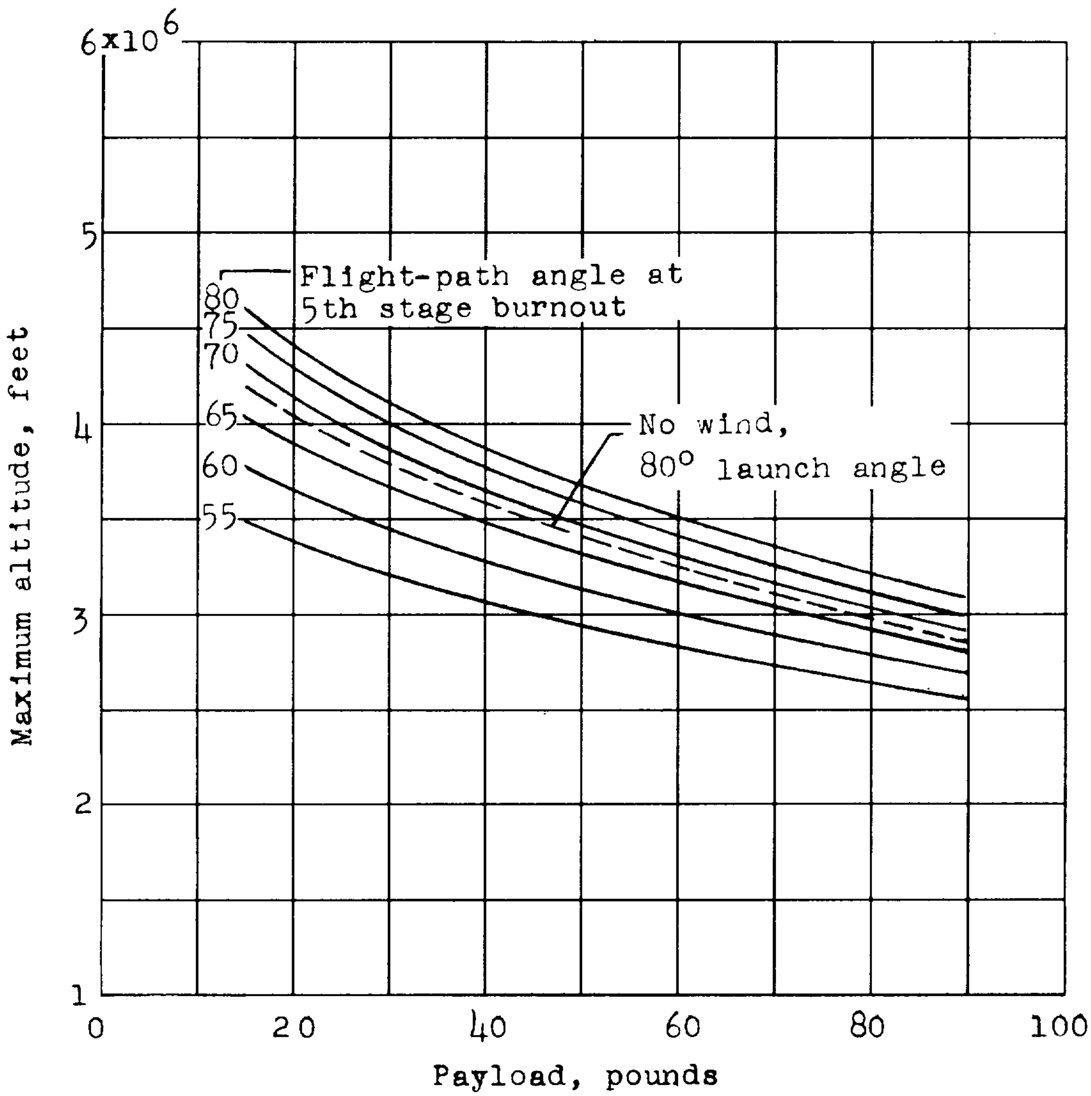


Figure 13.- Effect of flight-path angle on maximum altitude.

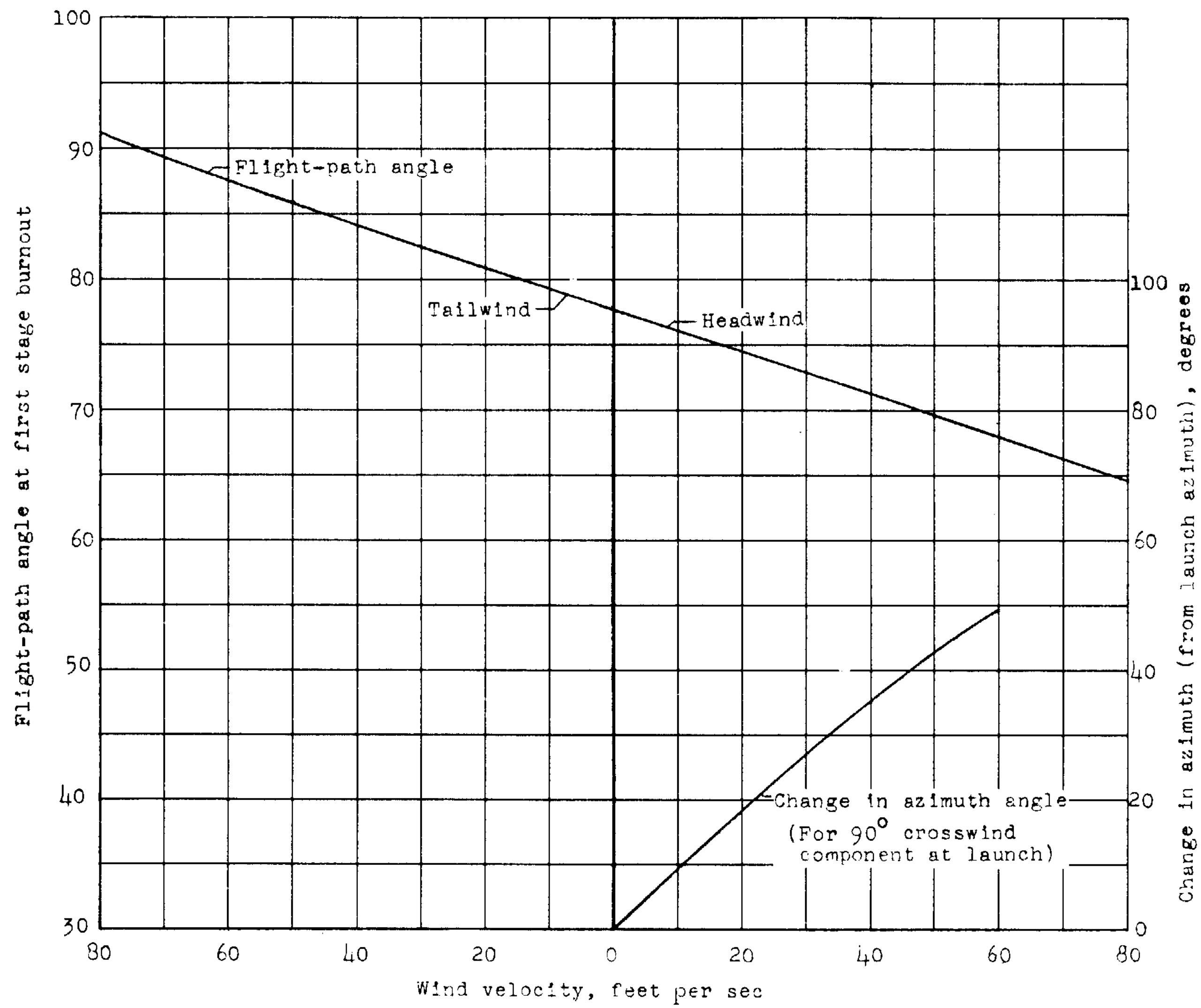


Figure 14.- Effect of horizontal wind on flight-path angle and azimuth angle. 80° launch angle.

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A FIVE-STAGE SOLID-FUEL SOUNDING-ROCKET SYSTEM. Andrew G. Swanson. March 1959. 22p. diagrs., photos. (NASA MEMORANDUM 3-6-59L)

A description is presented of a five-stage solid-fuel sounding-rocket system which can boost payloads of 25 and 100 pounds to altitudes of 525 and 300 nautical miles, respectively. Data obtained from a typical flight test of the system are discussed.

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