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SOUNDING ROCKETS

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Goddard Space Flight Center
Greenbelt, Maryland

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SOUNDING ROCKETS

by

Eleanor C. Pressly

When Dr. Robert Goddard began his efforts to develop a rocket, he stated that his intended purpose was to find a means of transporting instruments into our outer atmosphere so that direct measurement of that atmosphere could be made. At that time balloons were being used and the maximum altitude that could be surveyed was the 20 miles or so balloon peak. Today the sounding rocket has fulfilled Dr. Goddard's dream and it is the only means available for direct measurement between balloon peak and satellite perigee. After the end of World War II, many countries began to sponsor projects in upper atmosphere research using rockets. "Soundings" of the atmosphere were to be undertaken and so we have the "sounding rocket." In this paper we discuss the sounding rocket and its place in space exploration. Several particular rocket systems are discussed in detail in the appendices.

Basically a sounding rocket is a relatively small vehicle which carries scientific instruments to altitudes of 50 miles and over (an artificial upper limit has been set as one earth's radius) in nearly vertical trajectories. It may use liquid or solid propellant and it may have one or more stages, i.e., individual motors burning consecutively. It does not contain a guidance system but rather is fin stabilized and during powered flight it weathercocks into the wind.

Immediately after the end of World War II a program of upper atmosphere research was begun in the United States using V-2's. It was recognized quickly that rockets designed expressly for research purposes were necessary if that research was to be done in an efficient manner. Too much compromise was required between the different experiments for each to be able to collect data during the desired portion of flight. Thus development of Aerobee and Nike Deacon began. The Aerobee was boosted liquid propellant rocket and Nike Deacon was two stages consisting of the Nike Ajax booster as first stage and Deacon as second stage. Both could go into the E region of the Ionosphere, the Aerobee carrying 150 pounds and the Nike-Deacon 50 pounds. More advanced versions of both rockets are in use today.

During the International Geophysical Year (July 1957-December 1958) there was international cooperation in upper air research by means of rockets with seven countries participating. Late in 1958 the National Aeronautics and Space

Administration was formed in the United States with one of its aims being to help other countries start programs in the scientific exploration of space. Already the number of countries participating has increased markedly and just recently a new international range has come into being on the southern tip of India. Now that we are in the International Quiet Solar Year, Wednesday has been chosen as the special day on which firings are to be made. This will permit scientists of many countries to coordinate their results and thus obtain a better scientific understanding.

UNIQUE USEFULNESS

The sounding rocket has a unique usefulness in space work. Some of its major attributes are: (1) It provides the capability of investigating a vertical cross section of the atmosphere. Such data provide us with knowledge of the changes in various atmospheric phenomena with altitude, which are important for themselves and for correlation with data obtained from the satellite orbit. In the summer of 1963 at Wallops Island, Virginia an attempt was made to have a particular sounding rocket in the air when a satellite passed. The two bodies actually came within 20 miles of each other. Thus vertical and orbit data of the same phenomena were obtained at the same time. (2) Available observing time provided is at least 5 minutes (with some rockets it is as much as 30 minutes). This is ample time for many types of experiments. (3) One experiment can be flown many times on a sounding rocket with results from one flight being used to improve the experiment on the next. Thus individual payloads are simpler, advantage can be taken of new data, and, overall, a more thorough experiment can be accomplished. (4) It provides a cheap tool for checking out an experiment which is destined for satellite use. Since one is doing basic research, it is appropriate to use the sounding rocket as a means of determining the best experimental approach to the satellite experiment, the areas of greatest interest, and the proper gains to set so that all data return from the satellite will be meaningful. In addition assurance of the capability of the equipment to withstand the extreme environment imposed on it by the rocket is obtained. (5) The sounding rockets smaller size makes it practicable to fly a single experiment. This is a very convenient solution to the old problem of lack of compatibility which sometimes forced experiments to operate only a fraction of the total flight time. (6) Logistics of firing are simple and ground support equipment and facilities required are at a minimum. Thus firings can be made at selected places and selected times. In this way special events can be covered in a manner which would not be practicable with a satellite. An excellent example of this occurred at Fort Churchill, Manitoba during the solar eclipse of July 20, 1963. The Churchhill Research Range, an established but small sounding rocket firing range, was within 130 miles of the center of totality of the eclipse. Within a period of 130 minutes centered at the maximum of the eclipse, eight rockets were fired from this range. Five of these rockets are shown in Figure 1. The picture was made during a "dress rehearsal" which explains the plastic protective covers on nose cones and fins. All of the range's launchers were used and additional temporary launchers were installed. With very careful planning, all firings were made as scheduled. (7) Normal preparation time for a sounding rocket payload is not more than six months. Thus discoveries made in one flight can be followed up quickly with another. (8) Similarly it is good for synoptic work in which the same experiment is repeated over a period of time. Frequently shots of this type are coordinated from various locations. (9) Most of the advantages listed above make the sounding rocket a good training ground for those who are going into rocket and satellite work. The relative simplicity of experiment, rocket, and firing range logistics all contribute to this end.

SCIENTIFIC DISCIPLINES COVERED

The sounding rocket is useful in most of the scientific disciplines with which scientists of many countries are concerned and the variety of experiments that can be performed is virtually unlimited.

In the Aeronomy field the smaller sounding rockets are used extensively around the world for synoptic studies of winds and diffusion and to a lesser extent for studies of the structure of the atmosphere as a function of altitude and of geographic location. Two techniques used in synoptic work are the sodium vapor cloud and grenades. Figure 2 shows one superimposed on the other. Frequently the two methods are used concurrently for data correlation purposes. Similarly the composition, neutral and ion, of the atmosphere is under study both by use of mass spectrometers and of sampling techniques. This latter method is being employed to investigate noctilucent clouds from Sweden. Both day and night airglow are under study. Micrometeoroid dust is being collected for later analysis in the laboratory.

With careful planning, biological experiments can be accomplished on sounding rockets. Normally recovery is required and the closeness to firing time that specimens can be installed in the rocket is a big factor. To the writer's knowledge, the highest such flight to date went to 1,260 miles altitude and was safely recovered.

Small rockets can be used to advantage to study particles arriving at the earth as a result of solar disturbances. The earth's magnetic field can be studied. Rockets fabricated of non-magnetic materials have been developed expressly for this purpose.

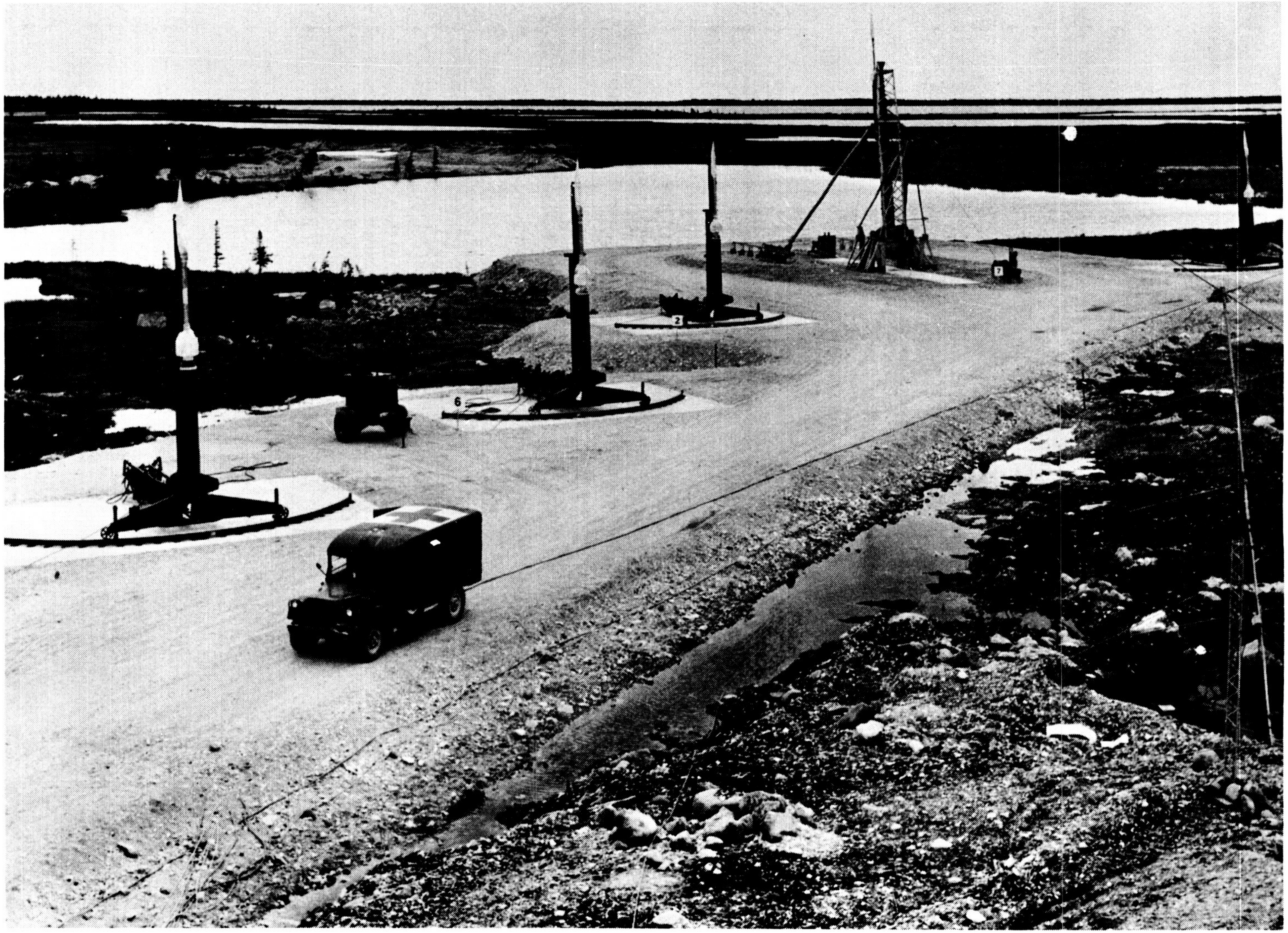


Figure 1–Solar Eclipse Experiment Being Readied at Fort Churchill, Manitoba

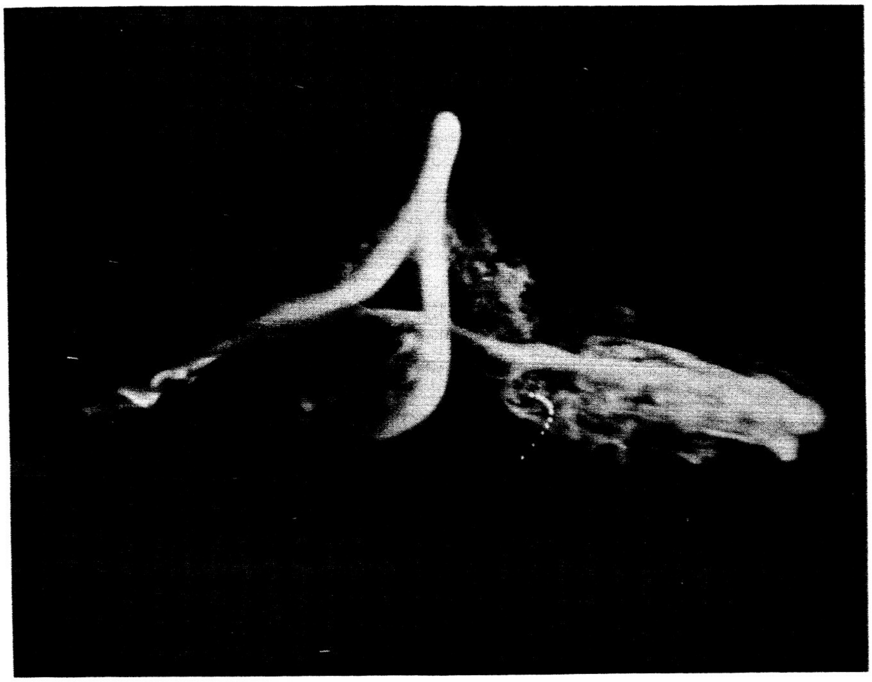


Figure 2–Sodium Vapor Cloud and Grenade Experiment

Properties of the ionosphere can be investigated from the "D" region to above the "F_2 maximum." A cooperative program between the United States and Canada called the "Topside Sounder" has been in operation for several years.

Sounding rockets can be used very effectively to carry telescopes to study the stars. In some cases the equipment is sufficiently delicate that liquid propellant rockets, which provide a softer ride, are required and a system for pointing the telescope accurately at the desired star, or stars, is needed. One such system is shown in Figure 3. The same may be said for experiments in solar physics. The higher altitude sounding rockets can be used for radio astronomy work.

Engineering questions which arise during the development of larger boosters can be investigated economically on the sounding rocket.

U.S. ROCKETS IN CURRENT USE

Since about 1947 one or another version of the Aerobee has been the principal sounding rocket workhorse. At the present time there are two, the Aerobee

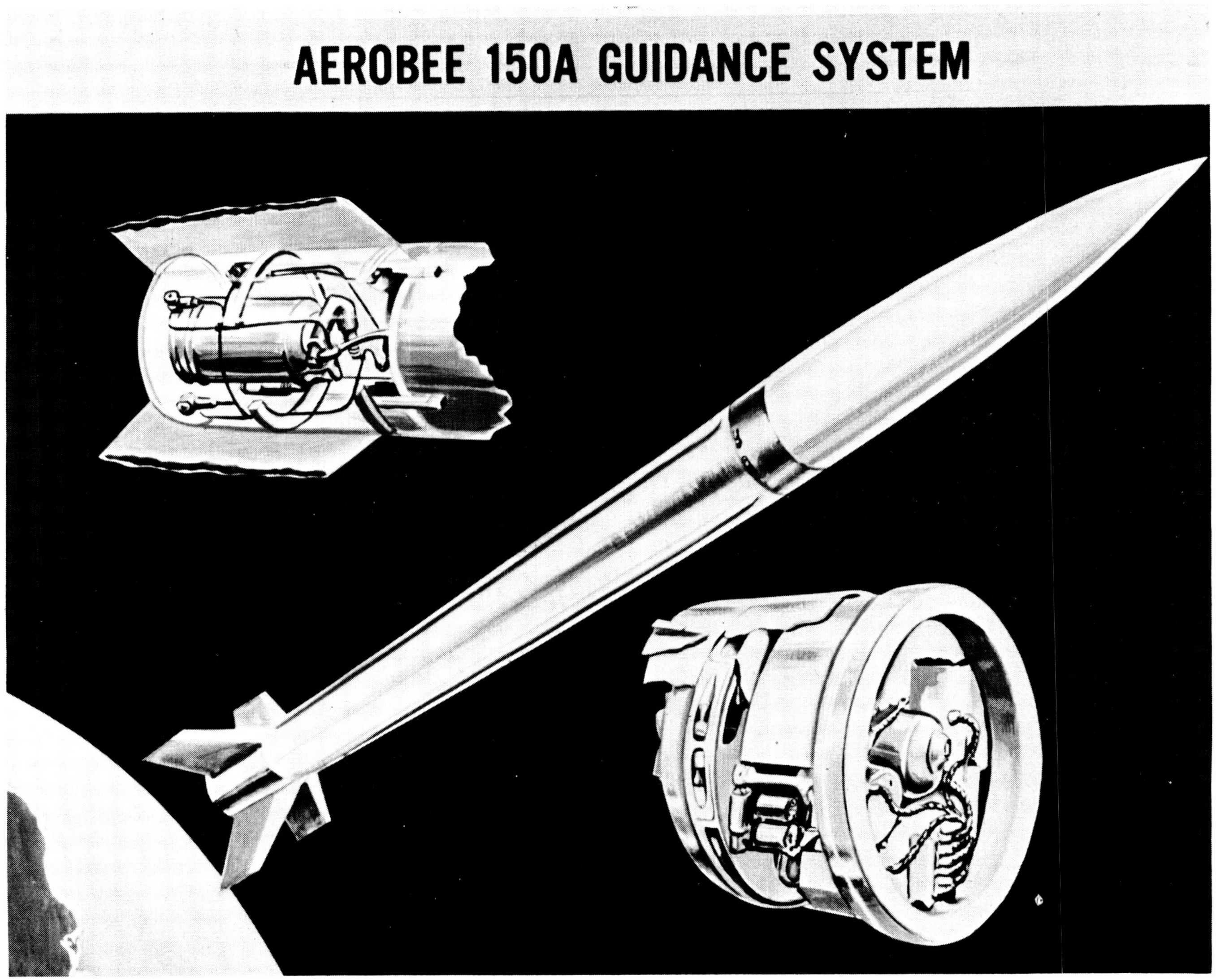


Figure 3–Aerobee 150A Guidance System

150 and the Aerobee 300, both of which come in 2 configurations. The Aerobee 150 technically is a two stage system consisting of a solid propellant booster and a liquid propellant sustainer stage. However both are burning for the duration of the booster and the booster merely provides assistance at takeoff. The rocket is 15 inches in diameter and provides a minimum of 4-5/8 cu. ft. of payload volume with up to 6 cu. ft. additional volume available as required. This rocket is discussed in detail in Appendix A. The Aerobee 300 is the Aerobee 150 propulsion system with the addition of a third stage, the Sparrow 1.9 KS 7800 motor. The standard Aerobee 150 is a three fin configuration and the alternate, called Aerobee 150A, is four finned. Aerobee 300A uses the 150A second stage.

Nike Apache and Nike Cajun are perhaps the most heavily used rockets in the United States sounding rocket program. They are identical in external appearance, the difference being that the Apache propellant provides more total impulse and thus takes a given payload to a higher altitude. The payload shell consists, normally, of an 11° cone and a 6-3/4 inches diameter cylinder of overall length not to exceed 80 inches. Details of the system may be found in Appendix D. Because of their simplicity and low cost, these rockets are commonly used for experiments which are synoptic or semi-synoptic in nature. For the same reasons and because logistics are relatively simple, they are frequently the first rockets new groups and new firing ranges use.

The Javelin is the largest sounding rocket in relatively heavy use. It is a staged system which employs rocket motors which were developed for the U.S. Department of Defense. Only the fourth (last) stage was developed exclusively for space use. This system is described in Appendix C.

Other sounding rockets in use are: (1) The Astrobee 200, a two stage solid propellant rocket which is similar to the Aerobee 150 but imposes a higher acceleration regime on the payload. (2) Journeyman, a four stage solid capable of reaching 1200 miles altitude. (3) Astrobee 1500, again a two stage solid, this time capable of reaching 1500 miles altitude (See Appendix B). (4) Black Brant II, built by Canadian Bristol Aerojet Limited. Line drawings to scale of four of these vehicles appear in Figure 4.

OTHER NATIONS' ROCKETS IN CURRENT USE

Canada entered the field of upper atmosphere research by means of rockets during the International Geophysical Year in cooperation with the United States and using U. S. rockets. At the same time they initiated a sounding rocket development program with the objective being a series of vehicles, called Black Brant, which would cover the altitude range up to about 600 miles. Black Brant I

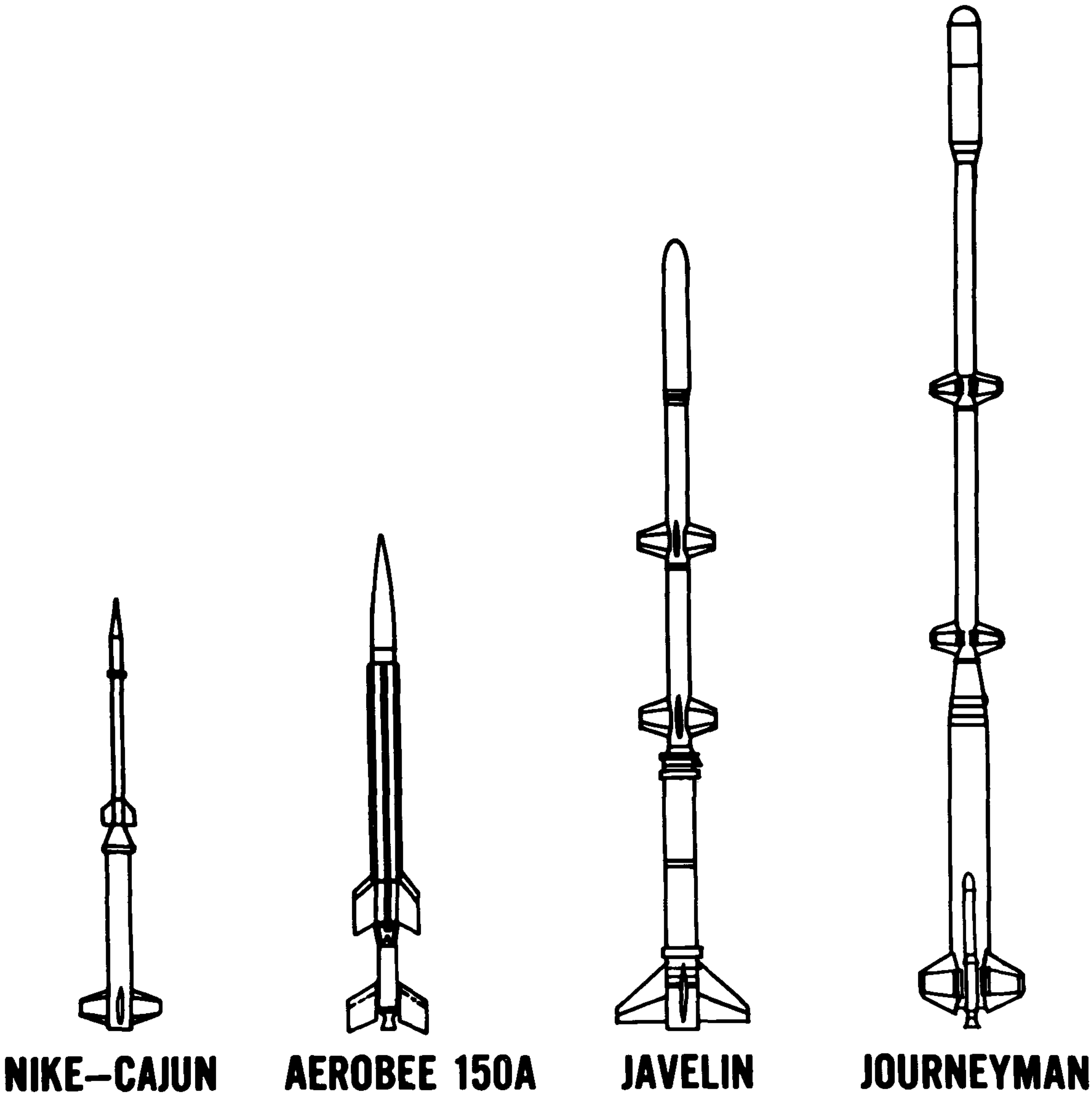


Figure 4—Four Sounding Rocket Systems

was the name given to the rocket during development of the propellant. This resulted, in 1960, in the Black Brant II, the first rocket for scientific use, see Figure 5. It is a single stage, 17 inch diameter, solid propellant rocket with a performance capability essentially equal to that of the Aerobee 150 and the Astrobee 200. The next rocket in the series, Black Brant III, is a 10 inch diameter vehicle which uses the same type of propellant. Development is nearly complete and the rocket will be ready for scientific use in the near future. Its

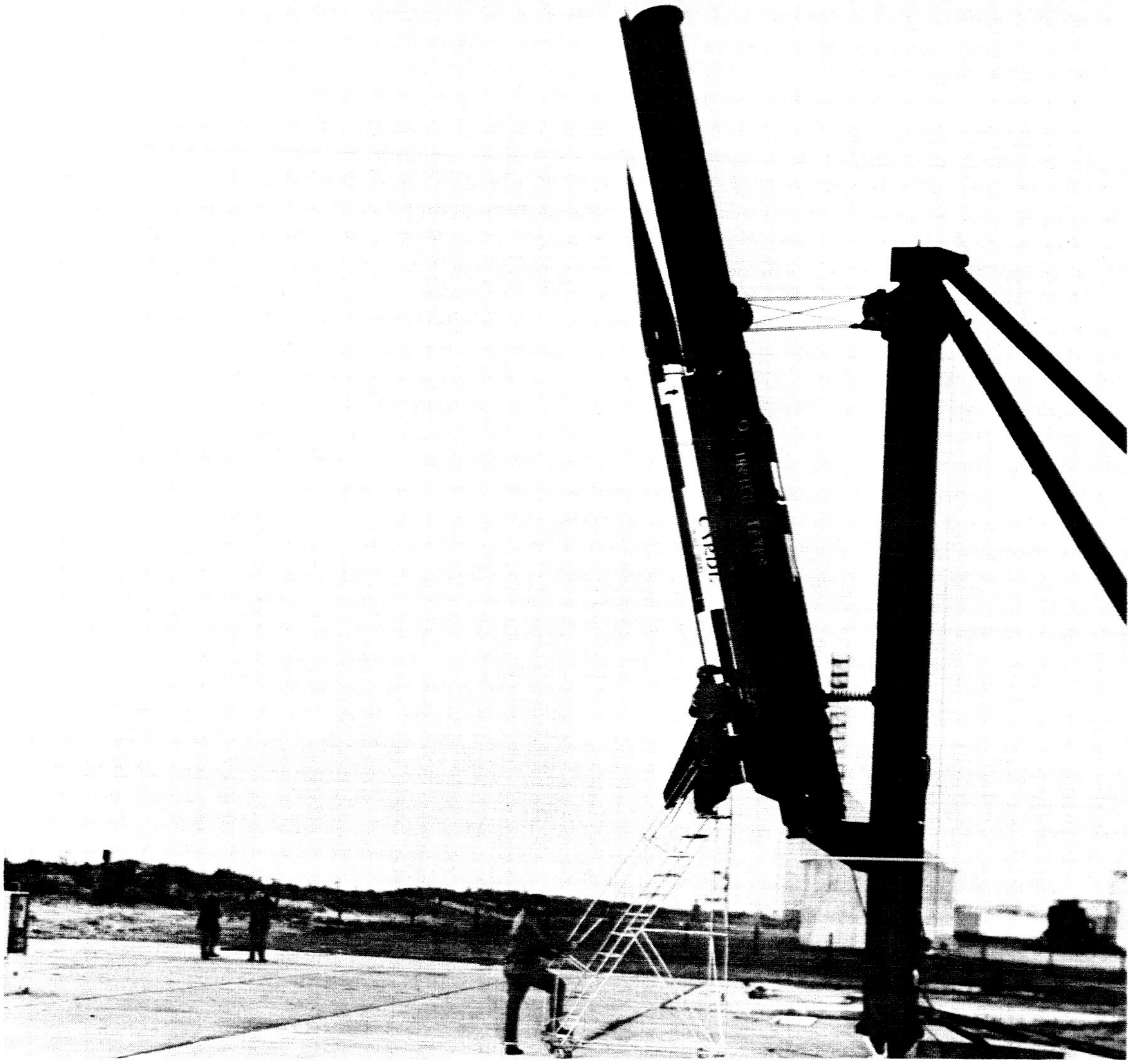


Figure 5—Black Brant II at Lift Off

performance capability is 24 pounds to about 100 miles. As originally envisioned the Black Brant IV was to be a two stage system, made up of the II and III. Black Brant V is planned as a higher performance system similar in appearance to the Black Brant II. This vehicle will probably be used as the first stage of the Black Brant IV and the system capability is expected to attain 600 miles altitude.

The first British Skylark was flown from Woomera, Australia in 1957. This is a solid propellant, 17 inch diameter, single stage vehicle which carries a typical payload to about 100 miles altitude. It has a slow initial acceleration and is fired from a tower.

One of the first rockets to be developed and used for scientific purposes is the French Veronique. Its first flight was made in 1952 from Hamaguir, North Africa. It is a single stage, liquid propellant vehicle and the present model is designed to carry a nominal payload to about 130 miles altitude. The Veronique employs a novel launching system. During the first 180 feet of flight the direction of the rocket axis is fixed by four stretched cables which are attached to the four fins and which pay out from a common drum.

SOUNDING ROCKET FEATURES

The prime, in fact the only, reason for the existence of the sounding rocket is its ability to carry a load of scientific instrumentation to some altitude above the surface of the earth. Particular attributes of different sounding rockets determine their usefulness in any given situation. Certainly no such thing as an "ideal" sounding rocket exists for all purposes. It then becomes the scientists job to determine which rocket most nearly fits his experiment requirement.

The first consideration is, of course, the altitude regime of interest. A rocket capable of 200 miles altitude, for example, would be quite unsuitable as a carrier for an ionospheres D region experiment simply because it would spend so little time in the D region. One of the payloads prepared for the ionosphere solar eclipse experiment is shown in Figure 6. The payload in its housing with antennas installed is shown on the right. Rockets spend 2 minutes of their total flight time within 10 miles of peak altitude. Thus the optimum peak altitude for any given experiment would be just above the region of major interest so that the greatest amount of time could be spent in this region.

Another prime consideration is availability of payload space. The several rockets described in Appendices A through D provide a good estimate of the variety of diameters and lengths within which an experiment must be confined. A payload for Journeyman, Figure 7, was flown from the Pacific Missile Range in February 1963. This rocket uses the same fourth stage as Javelin and payloads must be able to take the extreme vibration environment imposed by the X-248. Over the years, experimenters in the United States have found that it is best to fly a single experiment or group of related experiments. If too much space is available there is always the temptation to add more experiments and this always compromises all of them.

Next the acceleration and vibration environments which the rocket motor will impose on the payload must be considered. It is in this area, and only in this area, that all liquid propellant rockets have the advantage over solid propellant ones. The liquid type is characterized by much longer burning time and

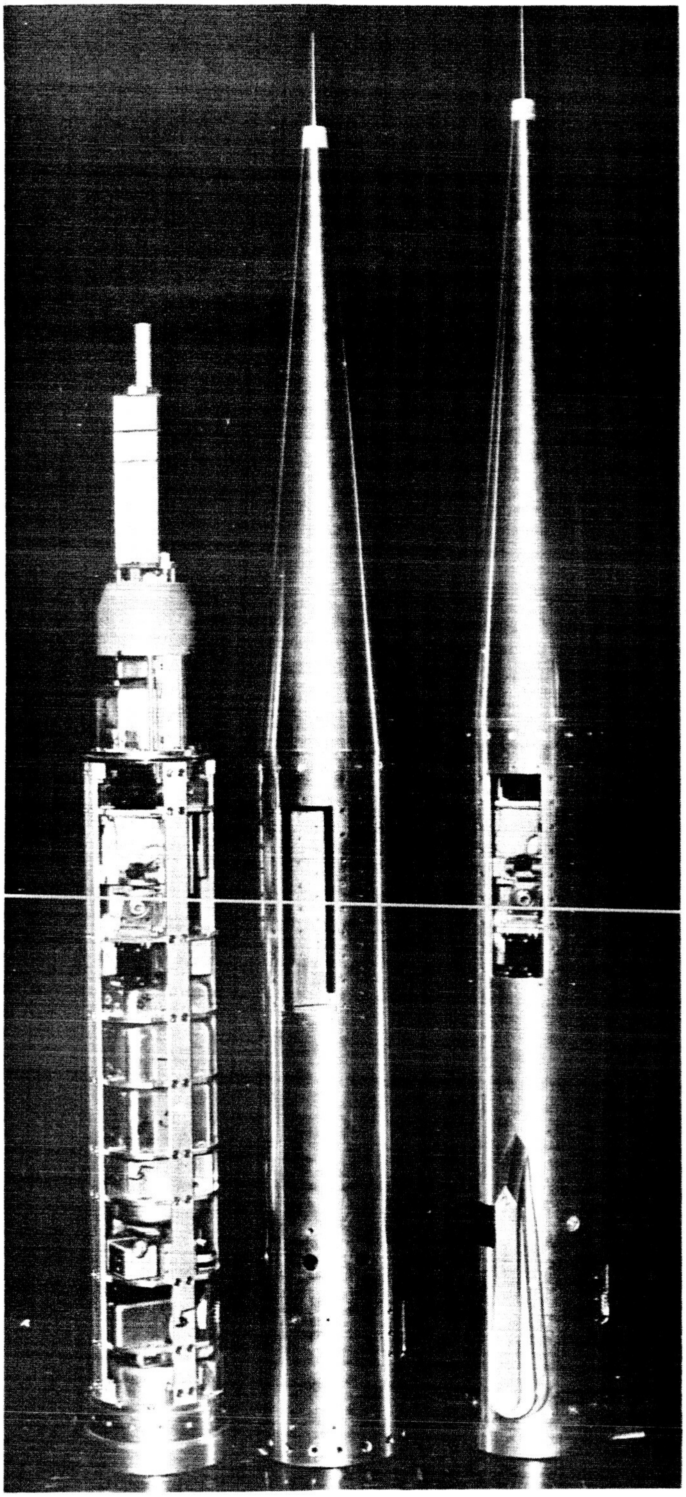


Figure 6—Ionospheres Experiment Prepared for Flight on Nike Apache

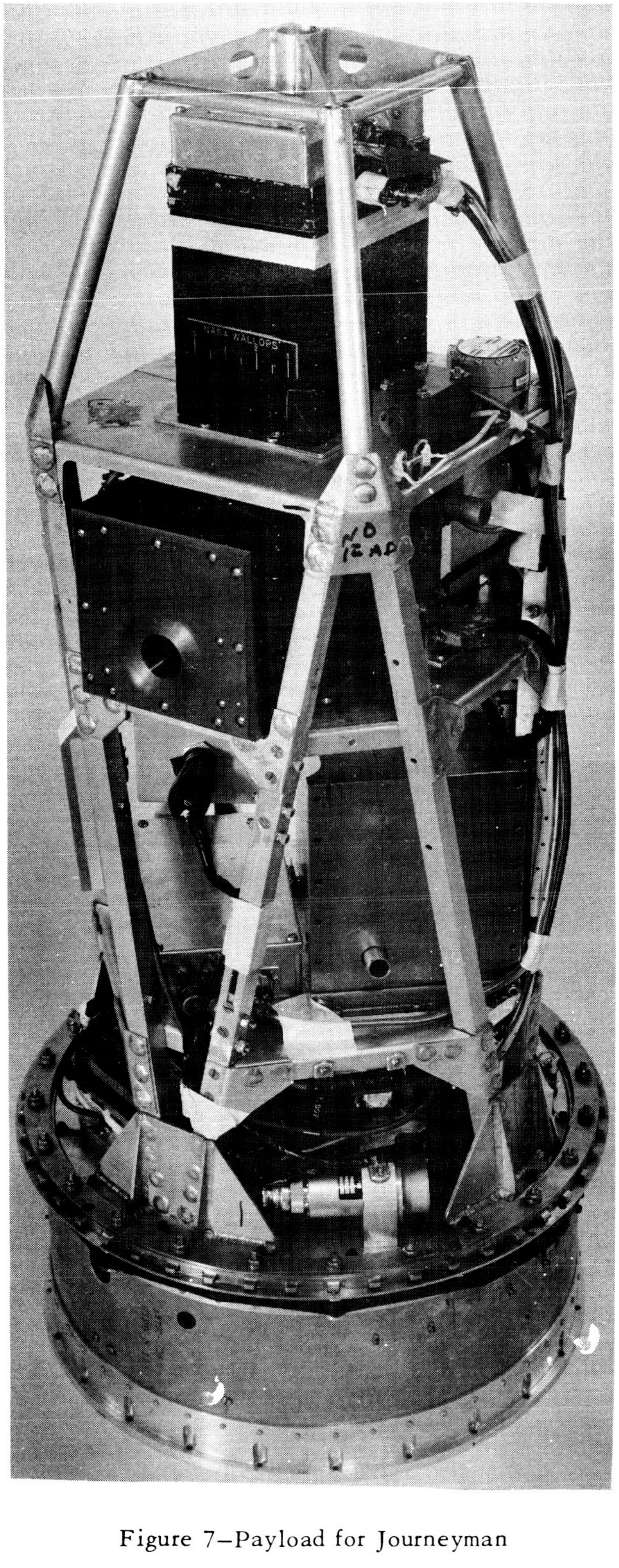


Figure 7—Payload for Journeyman

therefore by smaller longitudinal accelerations and in general vibrations caused by combustion are much less serious. Both the Aerobee and the Veronique have maximum accelerations on the order of 10 g's. On the solids it is not uncommon to encounter 75 g's or even higher accelerations. It is usually possible to design and build experiments which can successfully withstand the high accelerations, some experiments using optics being the major exception. Heavy penalties in sheer structural weight are required in order to provide the necessary strength. It would be nice if the vibration environment to be experienced were known so that a simple shake test of the payload could prove its flight worthiness. Except for a few isolated cases, these data are not available on most of the rocket motors in use today. However, there is available information arrived at by experience to the effect that a payload successfully tested to certain levels will fly successfully on a given rocket. It may be that these are proper levels. It may also be that they are higher than necessary and thus experiments are being penalized.

Temperature is another important item to consider. Payload compartment temperature is determined largely from the heat output of the instrumentation and normally the temperature rise is held to a relatively few degrees. Nose cone skin temperature is an entirely different matter. The higher the acceleration in the atmosphere, the greater the temperature rise will be. For each rocket system the material, thickness, etc. of the nose cone (or payload housing) must be chosen with due regard to the temperature history. For example, the standard cone for Aerobee is less than half as thick as the one for Nike Apache, both are aluminum alloys. The nose cone for Javelin is fiberglass with an ablative coat.

Payloads from sounding rockets may be recovered with relative ease and a minimum of ground support and equipment. With recovery a dual purpose can be served. Those experiments which do not lend themselves to recovery of data by telemetry but rather must be physically recovered (camera film, nuclear emulsions, air samples, micrometeorote collection to name a few) can be carried out with every expectation of a successful recovery. Also, and sometimes economically very important, expensive equipment can be saved and reflown with minimum rehabilitation.

Recovery systems normally consist of a parachute (a personnel type canopy is frequently used), separation devices to break the rocket apart and release the parachute, timers and power supplies, and perhaps recovery aids such as dye markers or beacons. One standard system provides for breaking the rocket at the proper altitude to "spoil" it aerodynamically. This is accomplished by primer cord initiated by a timer. Then when the section to be recovered returns to 20,000 feet, as sensed by baroswitches, the parachute is released. Ground impact will occur at a speed of about 25 feet per second. Not infrequently, recovery crews in the impact area are able to spot the parachute while it is still in the air.

Sea recovery presents many more problems than land, however it also can be accomplished quite successfully. An Aerobee payload as seen by the recovery helicopter is shown in Figure 8. In any event the slow descent allows time for a good radar fix on the location.

Figure 8—Sea Recovery of Aerobee Payload

One feature of the sounding rocket which is of interest to everyone concerned with space work is its relatively low cost. Sounding rocket systems vary from about $5,000 for the smaller ones to $150,000 for the most sophisticated.

It is important that outgassing either from the rocket motor or from air trapped in the payload be prevented in so far as possible that the atmosphere may not be contaminated. Such contamination interferes with the observation the experiment is attempting to make, and in extreme cases can negate the experiment entirely. Outgassing from the rocket motor can be controlled more easily on liquid propellant vehicles. Shutoff valves located just upstream of the combustion chamber can reduce the outgassing volume to a minimum. In the case of the solid propellant rocket, about the only solution available today is to separate the payload from the spent motor by some means which provides a differential velocity and thus remove the payload from the vicinity of the motor. Outgassing from the payload compartment can be prevented by the use of "O" ring seals at all joints.

PERFORMANCE

A rocket is a jet propelled device which is self inclusive, that is it carries along its own oxidizer and fuel. The propulsion system of a liquid propellant rocket in its simplest form consists of the two propellants, a combustion chamber, and a means of forcing the propellants into the chamber at the desired rates. Standard and well known propellants are used in sounding rockets. The oxidizer is usually some form of nitric acid. Fuels vary, in Aerobee the fuel is a mixture of aniline and furfuryl alcohol, in Veronique it is turpentine with a starting slug of furfuryl alcohol. Thus in both systems the propellants are hypergolic (i.e., ignite spontaneously when mixed). Pressurization systems to feed propellants into the combustion chamber vary from the storage tank of helium at high pressure which is fed to the propellant tanks through a regulator valve to systems which are chemically produced from propellant mixtures in a small combustion chamber. Solid propellant rocket motors contain mixtures of fuel and oxidizer particles which are pre-loaded into the motor case. The standard designation used for solid propellant motors is of the form 25 KS 20,000, in which the first number (here 25) is the number of seconds of burning time, the first letter (here K) indicates the type of propellant, the S that it is solid propellant, and the last number is the approximate thrust level. All rockets use the de Laval nozzle to generate the jet.

One important area in the design of a rocket system for good altitude performance out of the system is the mass ratio, defined as the ratio of the total loaded mass, i.e., the mass at ignition, to the mass at burnout. Obviously the more propellant that can be flown in the least amount of mechanical hardware, the greater will be the performance. Advantage can be gained by multi-staging several motors into a vehicle system. Thus one rocket motor fires and carries other stages to some altitude and velocity. When its propellant is consumed, the metal case, fins, etc. are dropped away, thus improving the mass ratio of the remaining stages of the system. The second stage burns, is dropped, and so on. Such a system, operating in a vacuum, would be most efficient if the several stages fired one immediately after the other with no delay between stages. This is obvious since maximum velocity would be obtained only if there were no decrease in velocity between stages. Of course in the actual case, atmospheric drag must be encountered and there is a trade off between drag and coast time. For any given vehicle system the optimum trajectory must be determined by theoretical calculation. Computer programs are available to meet almost any sounding rocket needs. However, a reasonably accurate altitude vs. time history can be obtained by hand calculation.

For a rocket moving in the earth's gravitational field, the basic differential equation of motion is

$$m \dot{v} = F - D - mg \tag{1}$$

where m = mass in slugs

$\dot{v}$ = acceleration in feet per second squared

F = Thrust in pounds

D = Drag in pounds

g = acceleration of gravity in feet per second squared at any instant of time at an associated altitude. Altitude at burnout is the double integral, over the burning time, of this equation.

The thrust of a rocket can be computed by use of the formula:

$$F = P_c A_t C_F + (P_e - P_o) A_e \tag{2}$$

where

F = thrust in pounds,

P_c = chamber pressure in pounds per square inch (absolute),

A_t = nozzle throat area in square inches,

C_F = thrust coefficient (dimensionless),

P_e = pressure of jet at nozzle exit in pounds per square inch,

P_o = pressure of the atmosphere at the nozzle exit in pounds per square inch, and

A_e = nozzle exit area in square inches

When the rocket is calibrated at rest at sea level,

$$P_e = P_o = P_0 \tag{3}$$

where P_0 is the atmospheric pressure at sea level in pounds per square inch. Equation (2) then reduces to

$$F \text{ sea level} = P_c A_t C_F \tag{4}$$

When the rocket is in motion, a first approximation is

$$F \text{ static} = P_c A_t C_F + (P_0 - P_a) A_e \tag{5}$$

where P_a is the ambient pressure at the altitude in question in pounds per square inch. A more exact expression for the actual thrust acting on the rocket is

$$F \text{ static} = P_c A_t C_F + (P_0 - P_b) A_e \tag{6}$$

where P_b is the base pressure at the nozzle exit at the altitude in question in pounds per square inch.

C_F can be gotten from equation (4). All other elements of this equation can be obtained from static tests.

Drag is the force on the rocket due to the presence of the atmosphere and it acts in the direction opposite to the direction of motion. The magnitude of the drag is expressed by

$$D = \frac{1}{2} \rho v^2 A C_D \tag{7}$$

where

D = drag in pounds

ρ = atmospheric density in slugs per cubic foot

v = velocity along the flight path in feet per second

A = maximum cross sectional area of the rocket is square feet, and

C_D = drag coefficient (dimensionless)

C_D is best obtained from measurement of a model in a wind tunnel. In actual practice this formula provides drag for the so-called clean rocket, i.e., no external protuberances such as antennas, shrouds, etc. Such items can increase the amount of the drag considerably.

g is the acceleration of gravity and is considered to be directed toward the center of the earth. In trajectory calculations, g at any altitude may be approximated by

$$g = g_o \left(\frac{R_o}{R_o + h} \right)^2 \tag{8}$$

where

g_o = acceleration of gravity at earth's sufrace

R_o = radius of earth

h = altitude above surface

Step integration should be employed in the solution of equation (1) and it should be continued to a point on the trajectory where drag has become negligible. From here peak altitude and peak time may be obtained from

$$h_p = h_1 + \frac{v_1^2 (R_o + h_1)^2}{2 g_o R_o^2 - v_1^2 (R_o + h_1)} \tag{9}$$

and

$$t_p = t_1 + \left(\frac{(R_o+h_p)(R_o+h_1)(h_p-h_1)}{2 g_o R_o^2} \right)^{1/2} + \left(\frac{(R_o+h_p)^3}{2 g_o R_o^2} \right)^{1/2} \tan^{-1} \left(\frac{h_p-h_1}{R_o+h_1} \right)^{1/2} \tag{10}$$

where

h_p and t_p = peak altitude and time respectively

h_1, v_1, and t_1 = altitude, velocity, and time at which step integration is stopped.

g_o = acceleration of gravity at sea level

R_o = radius of the earth

The above are obtained by integrating the expression for g given in equation (8) thus vertical flight from a non rotating earth is assumed. This assumption holds reasonably well down to launch elevation angles of 5° from the vertical. If one is concerned for extreme accuracy, it is virtually imperative that a computer program be utilized.

A rocket is statically stable when the center of pressure is aft of the center of gravity. For practical purposes in any particular case a minimum of at least one caliber of separation (or static margin) is considered necessary on a sounding rocket. A few rocket motors are of such a configuration as to make achievement of one caliber minimum impossible. On these, the rocket must be spun fast enough to obtain stability gyroscopically.

The center of gravity moves as propellant burns, the manner being dependent on the type and geometry of the propellant. It can and should be measured before flight. With this and a knowledge of propellant burning characteristics, the center of gravity history during burning can be obtained theoretically. Center of pressure can be determined from wind tunnel measurement and approximated theoretically for the specific aerodynamic configuration. It starts to move forward when the vehicles velocity becomes supersonic and continues forward with increasing mach number.

Other considerations make it desirable to spin all sounding rockets if such action will not jeopardize the success of the experiment. Spinning will reduce dispersion caused by thrust malalignment. It also makes for better vehicle attitude during the time the experiment is collecting data. Knowledge of the vehicle's natural frequency in pitch as a function of time is imperative in the choice of spin rate. If the spin rate couples with the pitching frequency it will be driven with the pitching frequency and excessive, perhaps catastrophic, yaw will result. If crossover points cannot be avoided, it is best that they occur as quickly as possible and at altitudes where the restoring moment is high. Several different methods of inducing spin are available, perhaps the most satisfactory being the canted fin technique in which spin up is proportional to velocity increase. Another method uses small rocket motors which fire for a short duration to rapidly increase (or decrease) the spin of the rocket vehicle.

FIRING RANGE

It was mentioned earlier that one of the reasons a sounding rocket is uniquely useful is that the logistics of firing it are simple and that ground support equipment and facilities required are at a minimum. A "minimum" permanent facility must include a launcher, blockhouse, area for assembling the rockets, telemetry receiving station, and radar tracking facility.

Different types of rockets do require different types of launchers. However, many of the sounding rockets in use around the world today can be fired from a Nike rail (See Figure 9) and most can be fired from zero length launchers. Whatever the launcher type, it is important that the capability to change azimuth and elevation angles remotely be present.

Good ground safety practices must be established and rigidly adhered to in the care and handling of the solid rocket motors and the liquid propellants. In order to prevent accidental ignition, all solid propellant motors should be grounded when in storage and when personnel are working around them and igniter wires should be shorted. Only safety tools should be used when working on or around a loaded motor, by this we mean non sparking metal tools, non-electrical power tools and drills. All lighting and switches should be of an explosion proof type. Pyrotechnics and igniters should be checked and tested only with test equipment especially designed for this purpose. All vehicles used in the vicinity of the

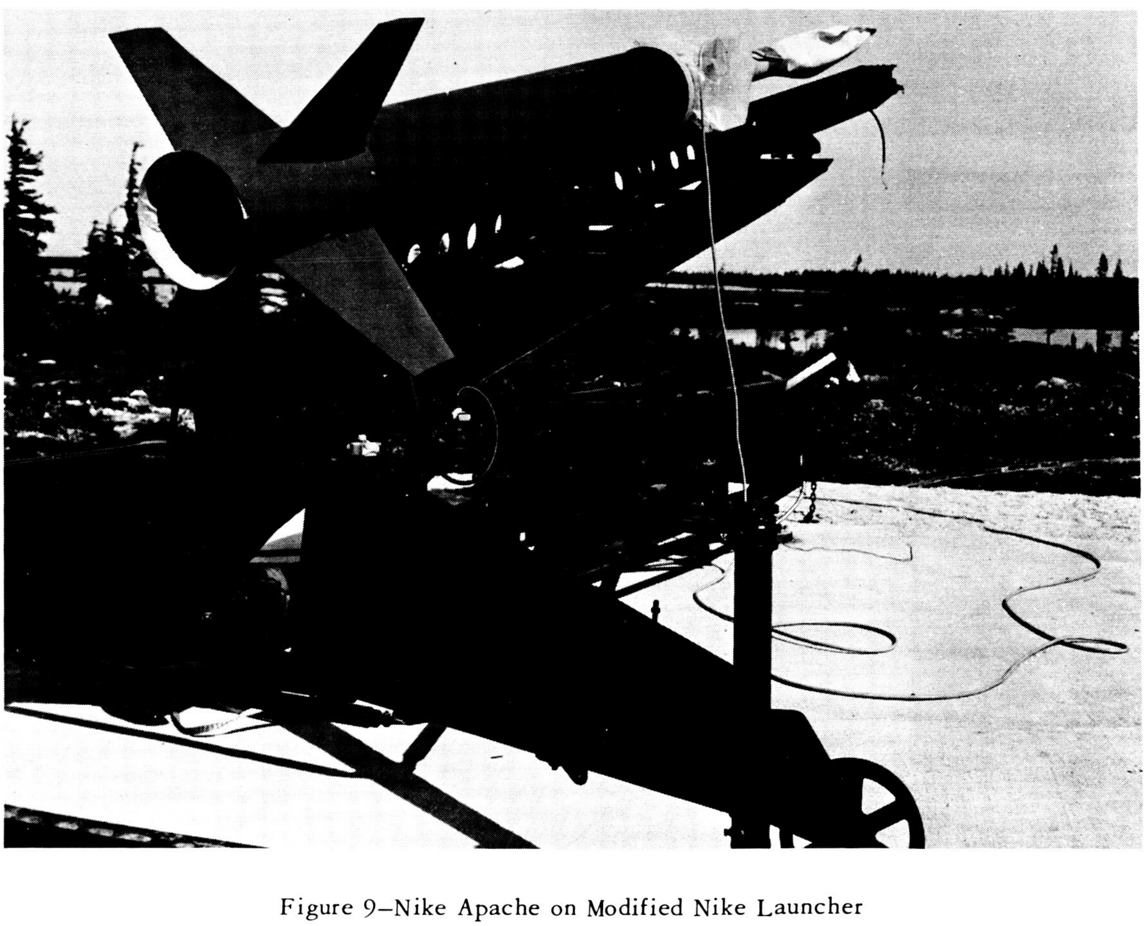


Figure 9—Nike Apache on Modified Nike Launcher

pyrotechnics and motors should be equipped with spark and flame arresters. Finally, there should be a minimum number of people in the vicinity of rocket motors and pyrotechnics at any time and open flames, matches, and smoking must be forbidden. Protective clothing should be worn when working with liquid propellants. The hazards, precautionary measures, and emergency procedures of each type of propellant used should be clearly understood and prepared for.

In Europe there are active sounding rocket firing ranges at Sardinia, Italy; Andoya, Norway; and Stockholm, Sweden. As mentioned earlier, French firings are conducted from Hamaguir in French North Africa and British firings from Woomera, Australia. There is a site at Karachi, Pakistan and a new international range has come into being recently at Thumba, India. Japanese sounding rockets are fired from Akita. The Churchill Research Range is in Manitoba, Canada. Sounding rockets in the United States are fired from a number of facilities: Wallops Island, Virginia; White Sands Missile Range, New Mexico; Eglin Air Force Base, Florida; Pacific Missile Range, California; and there is an Atlantic Missile Range site on Ascension Island.

IMPACT PREDICTIONS

Every firing range has the problem of assuring that the rocket will not land where it can do damage to people or property. Since the sounding rocket has no guidance system, extreme care must be taken that the rocket is aimed properly at launch to fly in the desired path. During the time that the rocket is under powered flight, the force of the wind on the rockets fins will cause the rocket to weathercock into the direction of the wind. The velocity vector of the rocket and the velocity vector of the wind are the major contributing factors. If the rocket were infinitely stable, it would instantaneously turn into the resultant vector and conversely if the rocket were neutrally stable there would be no effect of the wind. Since in practice the rocket does have a finite amount of stability, it responds somewhat less than the resultant vector. During unpowered, or coasting, flight, the rocket moves with the wind. The effect is usually quite small compared to that experienced while the rocket is under power.

Thus an accurate knowledge of the winds aloft at firing time is necessary so that the launcher may be pointed in a direction that will counter balance the effect the wind will have on the rockets flight and permit the rocket to follow the desired path. The first one or two thousand feet will be the most critical because the rocket's velocity is still relatively low but winds up to about 100,000 feet can affect the flight. It is standard practice at most ranges to begin taking wind measurements some four hours ahead of the desired launch time. Since upper winds are known to remain relatively constant over short periods of time, it is necessary to make only a couple of soundings above 10,000 feet but it is wise to make frequent checks to about 2,000 feet. If results of these checks are highly variable it will not be likely that a satisfactory impact prediction can be found. Usually measurements of the upper winds are made by rawinsonde or by radar balloon. Lower winds are determined by theodolite tracking of pilot balloons (pibals). No given pibal can be depended on to rise at the nominal rate, so it is best to use a double theodolite tracking system.

Since the amount the wind will displace the rocket is a function of the rocket's velocity, it is necessary to weight the actual winds, thus giving more importance to the first few thousand feet of altitude. In fact, if there were a constant wind from the ground up to the altitude where winds become negligible, on many vehicle systems 75% of the effect would have occurred by the time the rocket reached 2,000 feet. The result of this weighting, performed on the actual winds near firing time, gives the "ballistic wind." Each unit of ballistic wind will displace the rockets point of impact a given number of miles into the direction of the resultant wind. Similarly each unit of tilt in the launcher elevation will displace the point of impact a given number of miles in the launcher azimuth direction. Assuming that the desired impact location and the amount and direction of displacement due to winds are known, the required launcher azimuth and elevation angles can be determined by vector addition.

The theoretical work necessary to determine weighting factors, wind and tilt effects for any rocket system is considerably more complicated than is indicated above. The weighting factor does not vary much with payload weight and need be done only once for any particular system but wind and tilt effects are different for different payload weights (i.e., different maximum velocities). In fact there is an interaction between these two factors and there is a difference in the wind effect depending on whether there is a cross or range wind.

CONCLUSION

Sounding rockets provide an effective and economic means for the scientific investigation of space. There are many areas of usefulness with the unique attribute being that they collect data over a vertical cross section of the atmosphere. Many countries have programs or research using sounding rockets and others are planning such programs.

APPENDIX A

Aerobee

by

Jon R. Busse

VEHICLE DESIGN AND DESCRIPTION

Aerobee is the name given to a family of sounding rockets. Those used most extensively today are the Aerobee 150, 3 fin Aerobee 150A, 4 fin Aerobee 300, and Aerobee 300A. These rockets are described here with the Aerobee 150A being taken as the prime vehicle.

AEROBEE 150A

The Aerobee 150A (Figure 1 a, b, c) is a four-fin sounding rocket approximately 30 feet long and 15 inches in diameter, manufactured by Space-General Corporation and first flown in February, 1960. The rocket is capable of transporting payloads (100-300 lbs.) to high altitudes (100-150 statute miles) with a stable, near vertical trajectory. The rocket is a free-flight, fin stabilized, tower launched, vehicle consisting of a liquid propellant sustainer and a solid propellant booster. The rocket is rolled to decrease dispersion.

The sustainer portion of the rocket (Figure 2) basically consists of a nose cone and extensions, which house the payload; a forward skirt, which houses the pressurization system; integral pressurization and propellant tanks, which form the main rocket body; and an aft structure, which supports the fins and houses the thrust chamber. Four shrouds are mounted between the forward skirt and aft structure for propellant tank pressurization lines, instrumentation lines, and antenna cabling. Equidistant between the shrouds, fore and aft, are two sets of riding lugs which support the rocket between the rails in the launch tower.

The nose cone is an aluminum compartment, formed by spinning, circular in cross section, with a 31 caliber ogive, cylinder profile. The assembly is 87.8 inches in length, has a volume of 4.75 cu. ft., and is attached to the rocket, or payload extension, by means of 16 screws and is sealed by an O-ring immediately forward of the screws.

A two piece, cone-cylinder nose cone is used optionally on the Aerobee. The forward piece is a cone 42.7 inches long with a 34° vertex angle, and the aft piece is a 44 inch cylinder, 15 inches in diameter, made of aluminum.

The payload extensions come in lengths from 6 to 45 inches and are essentially rolled and welded magnesium cylinders 0.063 inch thick. It is attached and sealed in the same manner as the nose cone.

The space within the forward skirt (Figure 3) contains the pressure regulator valve, overboard bleed port, external instrumentation pullaway plugs, and internal instrumentation plugs which provide access to the shrouds. The forward skirt is a magnesium cylinder, riveted to the forward end of the tank assembly, and includes an access door. The forward end of the skirt contains tapped holes for attachment of the nose cone or extension by means of 16 screws, and 4 stud bolts for attachment of the payload assembly.

The tank assembly is a cylinder 161.5 inches long and 15 inches outside diameter. The assembly consists of three, welded tanks for helium, fuel, and oxidizer, arranged fore to aft. The tank walls form the external skin or the rocket and the aft helium bulkhead and forward fuel bulkhead are common; likewise the aft fuel bulkhead is common to the forward oxidizer bulkhead. The assembly is fabricated from 410 stainless steel, heat treated to a minimum tensile strength of 142,000 psi. The helium tank contains 3,450 psia of gas initially. The gas feed lines come out through the forward skirt, go down the shrouds and terminate in the ullage portion of the propellant tanks. The fuel feed line extends from the center of the aft fuel bulkhead through the oxidizer tank and out the oxidizer tank head. Baffles are provided at the outlet of each tank to stabilize the propellant flow into the feed lines.

The aft structure contains the thrust support structure, the modified Nike propellant start valve, the fuel and oxidizer shutoff valve, propellant flex lines, and the regeneratively cooled thrust chamber (Figure 4). The shutoff valves are poppet-type, normally open, squib actuated valves.

The shutoff valves may be closed by ground command, in the event of an emergency or erratic rocket performance, to terminate thrust. The valves are frequently closed after rocket burnout to conserve the helium for attitude control or despin functions or to prevent contamination of the experiment by the exhausting gas. The positions of the valves are monitored through microswitches. The thrust chamber is a welded assembly consisting of four major areas: the fuel coolant jacket, the fuel and oxidizer manifolds, the injector, and the combustion chamber and DeLaval nozzle. The aft skirt is a 31 inch length, rolled and welded, magnesium cylinder, attached to the tank assembly with 16 screws.

The fins (Figure 5) are a modified single wedge fin with magnesium skin and spars, an aluminum box structure attachment base, and a stainless steel (.010 inch thick) leading edge cuff. The total fin area is 14.88 sq. ft. Each fin weighs 7 pounds, and is attached to the aft skirt by means of two, half-inch bolts. The fins can be canted up to 20' of arc to induce rolling motion to the rocket. The propellants used in the 150A are: a fuel (ANFA) comprised of a mixture of 65 percent aniline and 35 percent furfuryl alcohol, by weight, and an oxidizer of inhibited red fuming nitric acid (IRFNA).

The fuel and oxidizer are hypergolic, i.e., they react violently and ignite immediately when they come in contact.

IRFNA is a corrosive, toxic, non-flammable liquid mixture of nitric acid (HNO_3), water, and dissolved nitrogen dioxide (NO_2). The liquid is colored from light orange to orange. A small percentage of hydrofluoric acid (HF) is added to inhibit corrosion in the oxidizer tank.

The significant characteristics of IRFNA are:

(1)	Water	less than 2%
(2)	NO_2	6-8%
(3)	HF_1	0.6%
(4)	HNO_3	Residual
(5)	Odor	Acrid
(6)	Hygroscopic	Yes
(7)	Specific Gravity	1.54 (68°F)
(8)	Freezing Point	-49°F
(9)	Normal Boiling Point	+152°F
(10)	Approximate Maximum Decomposition Pressure at 80°F and 10% Ullage	150 psia

Aniline ($C_6H_5NH_2$) is a flammable, non-corrosive, toxic, oily liquid which varies in color from colorless to brown. It has an amine odor, is low in volatility, and is hypergolic with nitric acid.

The significant characteristics of aniline are:

(1)	Odor	Amine
(2)	Boiling Point	364°F
(3)	Freezing Point	22°F
(4)	Flash Point (Closed Cup)	168°F
(5)	Ignition Temperature	1418°F
(6)	Specific Gravity	1.002 (68°F)

Furfuryl alcohol ($C_4H_3OCH_2OH$) is a flammable, non-corrosive, non-toxic liquid which varies in color from straw yellow to dark amber. It has a brine-like odor and is hypergolic with nitric acid. The significant characteristics of furfuryl alcohol are:

(1)	Specific Gravity	1.136 (68°F)
(2)	Hygroscopic	Slightly
(3)	Boiling Point	332.6°F-343.4°F
(4)	Flash Point	167°F

The booster is a 2.5 KS-18,000 motor (Figure 6) of short-duration and high thrust. The motor, which is occasionally used in sled testing, was designed during 1946 to 1948 and reflects state-of-the-art techniques of that period. Over 850 of these motors have been fired since 1947.

The thrust structure consists of three legs and a thrust ring which transmits the booster thrust to the aft structure of the sustainer. The thrust structure is of cast and weld construction and is attached to the booster by 12 screws. It mates with the aft skirt by a shoulder-keyed joint. A flight cone is attached above the igniter firing cap and diverts the liquid rocket flame away from the booster.

The chamber barrel is rolled and welded from 0.190-in. thick AISI-4130 sheet stock. An aft ring and forward closure are welded to the barrel section. The forward closure contains a boss for installing an adjusting adapter and igniter assembly. The external surface on the forward closure is machined for attaching a thrust skirt assembly. The aft ring is drilled and tapped for attaching the nozzle assembly. Centering clips are spot welded to the inside of the barrel section for centering the charge assembly. Fin attachment pods and guide lugs are welded to the outside of the chamber. The chamber, heat treated to 150,000 psia minimum ultimate tensile strength, is hydrostatically tested to 2400 psig after fabrication.

The nozzle assembly is primarily composed of an AISI-4130 steel closure and exit cone. The forward ring of the closure is drilled for attaching the assembly to the chamber; the aft end of the closure is threaded for attaching the exit cone. The closure is heat treated to 150,000 minimum ultimate tensile strength and is hydrostatically tested to 2400 psig. A 0.50-in. thick plastic closure is cemented to the entrance section of the nozzle closure to maintain adequate pressure for aiding propellant ignition. The plastic closure bursts between 800 and 1200 psig. The entire nozzle assembly is attached to the chamber with 24 1/2-in.-dia. bolts.

The booster fins are similar to the 150A fins in shape and construction. Each fin is attached to three pads on the booster chamber by means of bolts. The holes are so machined to provide a pre-set fin cant of 2-1/2°. The fin cant is not adjustable.

The charge assembly is composed of two internal-external burning grains, a center trap assembly, forward and aft traps, an adjusting adapter, and igniter guide post with lead wires. The grain assemblies consist of two 130-lb. cylindrical AK-14-Mod I propellant grains. Both ends of each grain are inhibited with glass laminates. An asbestos sheeting is cemented to the glass laminates to preclude radiant ignition, and discs are cemented to the asbestos sheeting to allow for compression during assembly. The grain assemblies are torqued to the center trap assembly, using the forward and aft traps as compression plates. The total impulse delivered by the 2 grains is between 44,375 to 50,000 lb. sec.

The igniter is designed to produce sufficient pressure and heat to ignite the propellant grain. The main charge, consisting of 180 gm of sodium nitrate black powder is contained in a circular frangible plastic container. Two squibs, filled with potassium nitrate black powder ignite the main charge. A 5-amp current is used for reliable ignition.

The sequence of firing an Aerobee 150A is accomplished remotely as follows (Figure 7 a,b):

(1) The helium overboard dump is closed by actuation of a squib operated mechanism.

(2) After a 200 millisecond delay, the booster igniter is fired.

(3) As the rocket moves vertically, the regulator is actuated by pulling a trip wire.

(4) This action opens the pressure regulator valve and feeds helium into the propellant tanks (the regulator reduces the gas from 3450 psia to approximately 500 psia).

(5) At 160 psig the oxidizer diaphragm in the propellant start valve breaks (Figure 8).

(6) At 325 psig the fuel diaphragm breaks. (The rate of tank pressurization is controlled, by orifices downstream of the regulator. The fuel tank is pressurized at a rate faster than the oxidizer tank to prevent implosion of the common bulkhead.

(7) The Nike valve cam holds the pintles in a position to meter the flow of the propellants in the combustion chamber.

(8) When the chamber pressure reaches 100 psig, the propellant start valve actuator releases the cam and pintles and the propellants are in a full flow condition (this is approximately 0.6 second after first booster motion).

(9) At approximately 2.5 seconds the booster burns out and separates from the sustainer.

A resume of nominal performance characteristics of the Aerobee 150A is as follows:

a. <u>Liquid-Propulsion System:</u>

Pressurizing Gas	Helium-Grade A
Oxidizer to Fuel Ratio	2.56 to 1
Thrust Chamber Pressure	324.0 psia
Sea Level Thrust	4100 pounds
Powered Duration	50.9 sec.
Propellant Flow Rate	20.71 pounds per sec.
Nozzle Expansion Ratio	4.6
Specific Impulse	198 lb./lb./sec.
Total Impulse at Sea Level	208,690 lb.-sec.

b. Solid-Propulsion System:

Sea Level Thrust	18,600 pounds
Powered Duration	2.5 sec.
Chamber Pressure	1340 psia avg.
Area Throat	8.50 in.^{2}
Expansion Ratio	7.9
Weight Flow Rate	104 #/sec.^{2}
Surface-to-Port Ratio	74
ISP	178 sec.
Flame Temperature	2960°F

c. Rocket Performance Characteristics. Performance characteristics are obtained using an ogive nose cone. The cone-cylinder results in loss of 4% altitude. (Launched at Sea Level):

Peak Altitude	179	statute miles with 100 pound payload
	101.7	statute miles with 300 pound payload
Burnout Velocity	7,094	ft./sec. with 100# payload
	5,260	ft./sec. with 300# payload
Burnout Altitude	138,584	ft. with 100# payload
	111,688	ft. with 300# payload
Burnout Acceleration	349.44	ft./sec.^{2} with 100# payload
	218.24	ft./sec.^{2} with 300# payload

d. Dispersion Factors (150-lb. Payload):

Tower Tilt Factor	94.0 statute miles for 4° tower tilt
Ballistic Factor	5.2 miles/mph ballistic wind

e. Weights (In Pounds):

Payload	100 to 300
Vehicle Gross Weight (with payload)	2097.5
Rocket (loaded)	1347.5
Rocket (empty)	279.1
Booster (loaded)	600.0
Booster (expended)	338.0
Fins (4)	28.0

f. Volumes (In Cubic Feet):

Pressure Tank	2.57
Fuel Tank	4.46
Oxidizer Tank	7.88
Ullage Space	0.14

g. Propellant Weights (In Pounds):

Oxidizer	758.2
Fuel	303.3
Helium	5.15

The 150A is launched from NASA Wallops Station, Wallops Island, Virginia only, because Wallops has the only tower that can accommodate a four-finned vehicle.

The launch tower and preparation facilities are shown in Figure 8. The launcher is 160 feet in height and can be adjusted in elevation, and in azimuth. The coordinates of Wallops Island are 37° 50' North 75° 29' West.

AEROBEE 300A

The Aerobee 300A is an Aerobee 150A with a third stage motor added. All 150A volumes, weights, and characteristics, except those pertaining to the payload, are applicable to the 300A; so discussion here will be limited to the third stage and its effects on increasing rocket performance.

The third stage (Figure 9) is composed of a 1.8 KS-7800 solid propellant motor, a high expansion ratio nozzle, a separation subsystem, and the nose cone, which has a maximum diameter of 8 inches, limited by the diameter of the solid motor.

The 1.8 KS-7800 motor (nominal 1.8 seconds burning time with 7800 lbs. thrust) is the same motor as the one used with the Sparrow weapon system.

The 1.8 KS-7800 motor has the following nominal propulsion system ratings:

Duration	2.15 sec.
Propellant flow rate	32.09 lb./sec.
Chamber pressure	1000 psia
Throat area	4.43 in.^{2}
Area ratio	30
Diameter	8 inches
<u>Vacuum Performance</u>	
Thrust coefficient	1.73
Thrust	7664 lb.
Total impulse	16,478 lb. sec.
Specific impulse	238.8 lb./lb./sec.

The high expansion ratio nozzle is used because the motor operates only at high altitudes; thus optimum expansion is closely approached.

Aerodynamic stability, during third stage burning, is provided by the 14.5 degree half-angle conical transition section, which houses the 1.8 KS-7800 nozzle.

The third stage ignition signal is given upon thrust termination of the 150A sustainer. The sustainer and third stage are attached by a blowout diaphragm (Figure 10). The diaphragm engages mating threads on the sustainer and third stage. When the third stage ignites, the energy of the motor exhaust causes the diaphragm to deflect, releasing its thread engagement and allowing the third stage to "fly-away" from the sustainer.

Payload volume is nominally 0.9 cu. ft., and the following total rocket system performance characteristics are expected with a 60# payload:

Peak Altitude	265 statute miles
Peak Time	350 sec.
Burnout Velocity	8770 fps
Burnout Altitude	144,000 ft.

Because of the requirement for a four-rail tower, the 300A, like the 150A, is launched only from Wallops Island.

As a footnote, an Aerobee 300 can be made by putting the 1, 8 KS-7800 motor on a 150. NASA has done this in the past for launches from the Churchill Research Range, Manitoba, Canada. None have been launched from WSMR because of range boundary limitations.

AEROBEE 150

The Aerobee 150 (Figure 11 a, b) is similar in many ways to the 150A. The main performance difference results from launching the 150 (the greatest portion of them) from a higher elevation (approximately 4000 feet) than the 150A.

The physical differences are: the booster and sustainer fins (both number of and configuration), the number of shrouds, the number of riding lugs, the location of the common oxidizer and fuel bulkhead, the propellant start slug, the orientation of the pressurization system in the aft structure. The discussion of the 150 will be limited to the performance and physical differences.

The 150 has three each sustainer and booster fins and is launched from a three rail tower. The fins on the 150 are biconvex with dimensions as shown in Figure 12. Eight bolts are used to attach each fin. The sustainer fins may be canted to induce roll, and the booster fins, like the 150A booster fins, are present at an angle of 2-1/2° to induce roll.

There are three shrouds on the 150.

The 150 sustainer used a start slug of 7 lbs. of 70% furfuryl alcohol, 30% aniline by weight which gives a slight improvement in ignition characteristics over the 65% aniline, 35% furfuryl alcohol by weight.

On the 150, the oxidizer is located forward of the fuel (Figure 13). This propellant positioning requires a change in regulated feed pressures; as the system hydraulic pressure drops and in flight acceleration heads are different from the 150A.

The fact that the 150 has only three shrouds and is launched from a three-rail tower necessitates a rearrangement of the pressurization system in the forward skirt (Figure 14). The components of the pressurization system are the same. There are several fewer instrumentation plugs on the 150 than the 150A. (These are noted on Figure 14).

The use of hard lines, flare fittings, and the arrangement of the hardware are different from the 150A (Figure 15).

The use of low pressure burst diaphragms (50 ±10 psi) provides the initial throttling capability afforded by the Nike valve in the 150A.

There are no differences in the booster case or grain arrangement.

Typical performance characteristics for the 150 at sea level with a 150 # payload are as follows:

Peak Altitude	165 statute miles
Burnout Altitude	134,000 ft.
Burnout Velocity	6778 fps
Thrust at Sea Level	4100#
Duration of Thrust	50.9 sec.

Aerobee 150 rockets are launched from White Sands Missile Range (WSMR), New Mexico (Figure 11 a) and Churchill Research Range, Manitoba, Canada (Figure 16).

The 150's at White Sands are launched from the Naval Ordnance Missile Test Facility with range services provided by the U.S. Army. The range is located approximately 40 miles north of El Paso, Texas. Coordinates of the WSMR facility are 32° 24' North and 106° 32' West.

The Churchill launch facility is located on the shore of the Hudson's Bay; 58° 44' North, 93° 49' West.

FLIGHT PERFORMANCE

Detailed information concerning flight performance will be given for the 150A only. These data are representative of those used on all Aerobee flig ts and thus eliminate much duplication and presentation of similar information.

AERODYNAMICS

The aerodynamic parameters used to predict the in-flight rocket performance were generated from data gathered in wind tunnel tests. The tests were performed in the Jet Propulsion Laboratory supersonic and hypersonic wind tunnels using a 1/10 scale rocket model.

Figures 1 through 6 represent drag, normal force, center of pressure, natural pitching frequency, dynamic pressure, roll rate and lift characteristics of the Aerobee 150A. The curves assume a total rocket length of 288.9 inches, a diameter of 15 inches, an ogival nose cone, and a 150 lb. payload unless otherwise noted on the curves.

Figure 7 presents envelops of required stability for the 150A. The curves assume an ogive nose cone and a go indication of rocket static margin if the point is located to the right of the curve. The gross payload weight is the net payload weight plus the nose cone weight (15.5 lb.).

LIQUID PROPULSION SYSTEM

Figures 8 through 11 represent the computed performance characteristics of the liquid sustainer engine. The data were derived from a propellant utilization computer program, developed by the rocket manufacturer, and have been substantiated by static and flight firings. Figure 12 represents a chamber pressure trace obtained from a 150A flight by using a potentiometer type pressure gage operating through a telemetry link.

ALTITUDE, RANGE CAPABILITIES

Figure 13 is representative of the input information used to compute figures 14 through 17 to obtain altitude, range, velocity, and acceleration parameters of the Aerobee 150A.

The input data are used in a computer program which assumes a 2-D point mass rocket and a spherical, non-rotating earth.

Figure 18 represents an acceleration trace from a 150A flight with a 262.7 lb. payload. The data were obtained via telemetry using a potentiometer type accelerometer.

LOADS DATA

Figure 19 through 22 show the results of load, shear, and bending moment calculations for the Aerobee with a cone-cylinder nose cone. The curves represent loads, shears, and bending moments for 1 degree angle-of-attack. The curves increase linearly with angle of attack to conservatively 5 or 6 degrees.

The curves were computed for the time at which there is a maximum load on the rocket.

MISCELLANEOUS

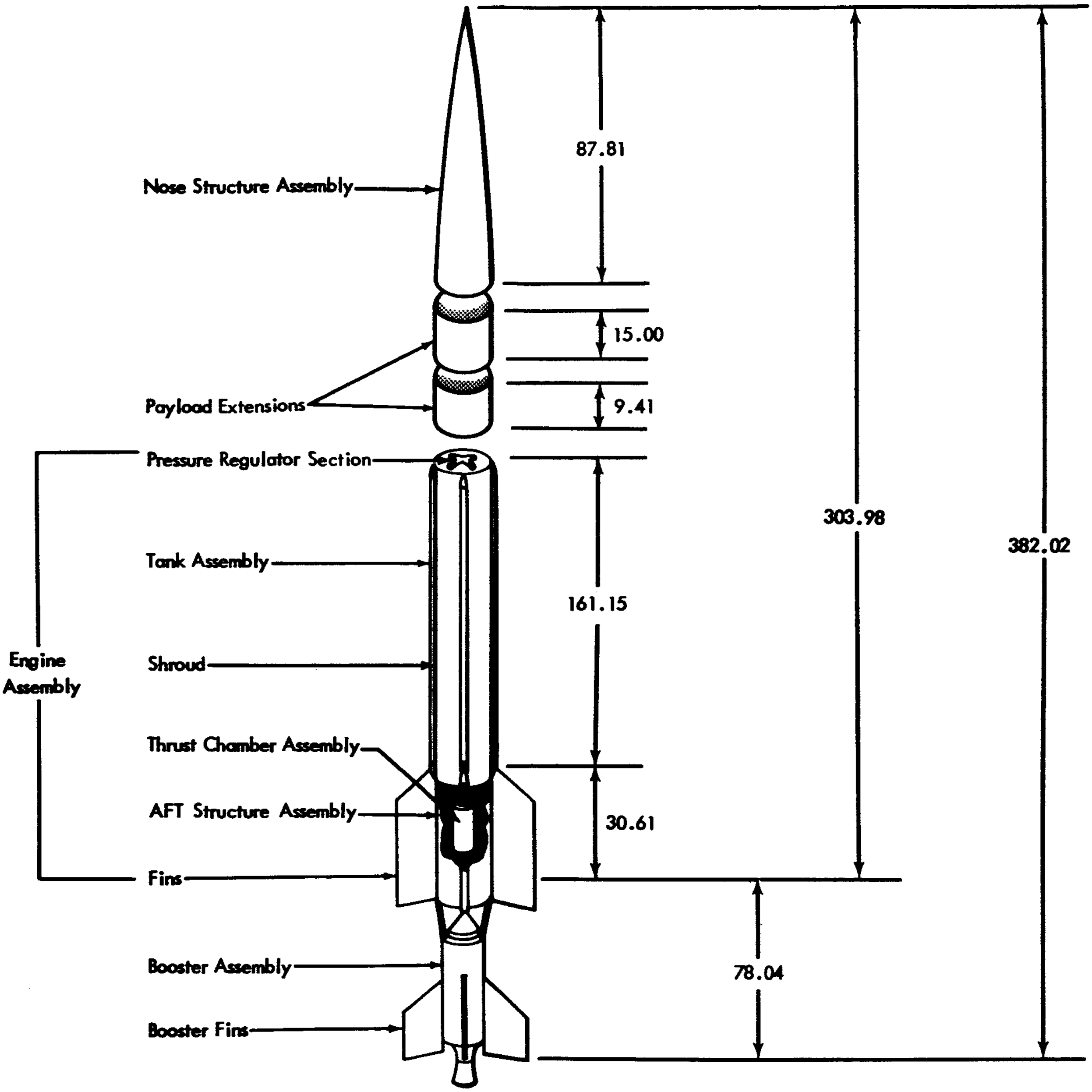
Figures 23 through 26 contain vehicle moment of inertia, center of gravity position versus time, weight versus time and weight breakdown data. These curves were generated using previously stated weights, flow rates, and dimensions.

Figures 27 and 28 are theoretical predictions of temperature time profiles 6 and 12 inches aft of the fin's leading edge. The inconel-X (curve 26) is a cuff that fits over the magnesium skin.

Curve 29 shows the amount of roll rate (rps) that can be obtained from 1 minute of fin cant (average for the four fins) for a burnout velocity (the burnout velocity being determined by the payload weight). The roll rate fin cant relationship is linear; so that the fin cant required to give a desired roll rate may be obtained by dividing the desired roll rate by the roll rate per minute of fin cant. The rocket is rolled to decrease the dispersion effects of thrust and structural misalignments.

APPENDIX A – PART I

LIST OF ILLUSTRATIONS



(All Dimensions Indicated in Inches)

Figure 1A–Major Components & Dimensions of Aerobee 150A

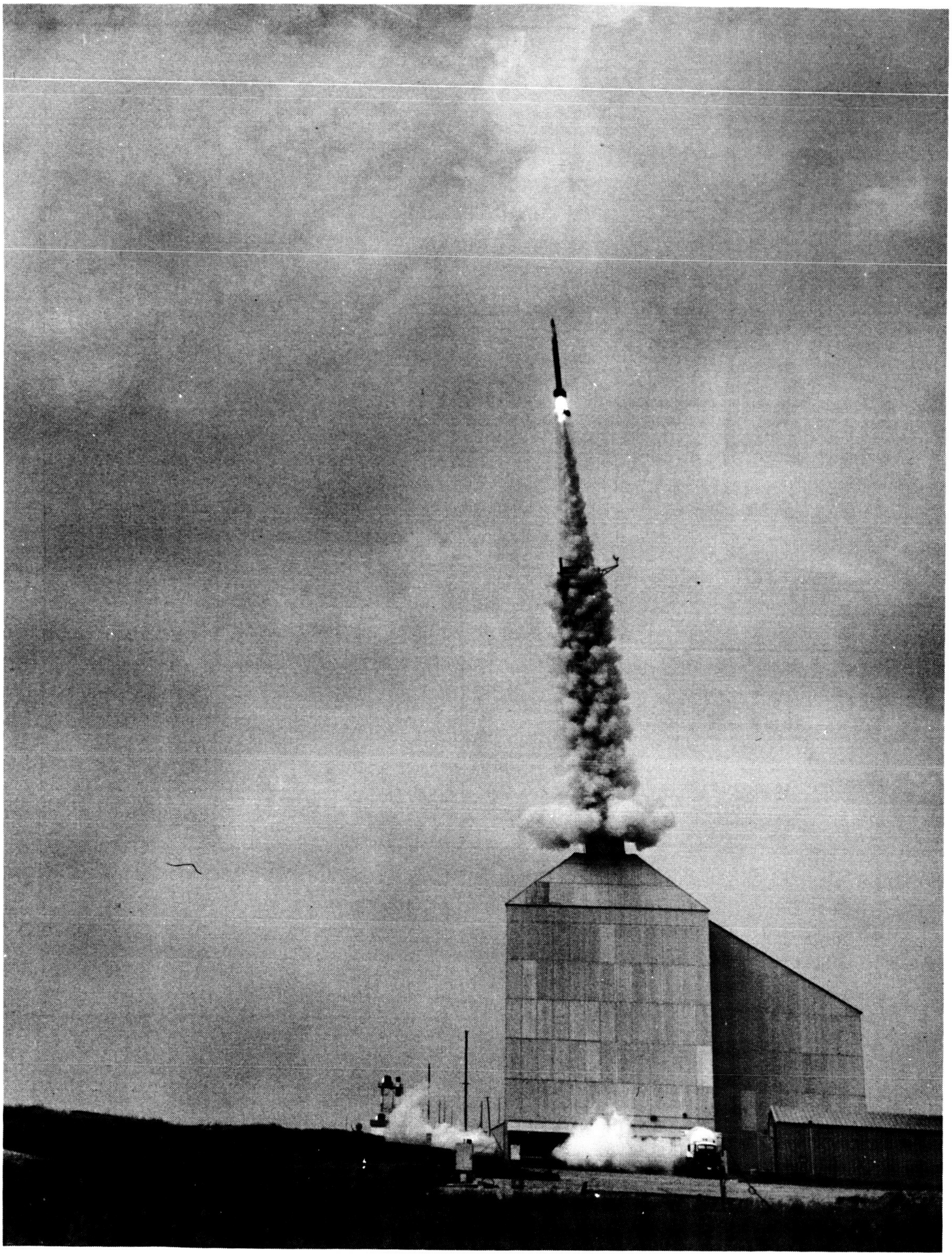


Figure 1B—Aerobee Launch

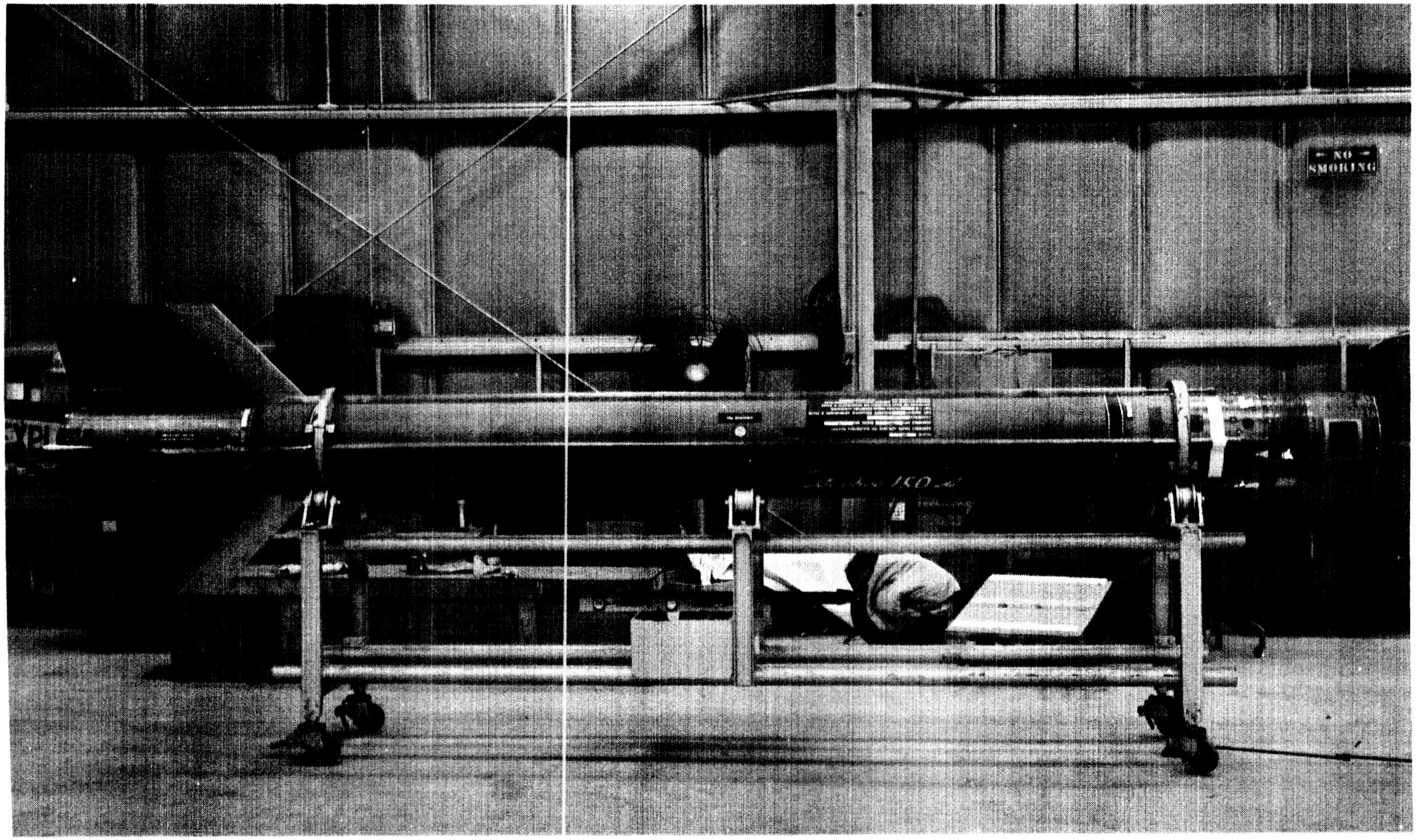


Figure 2–Aerobee 150A Sustainer

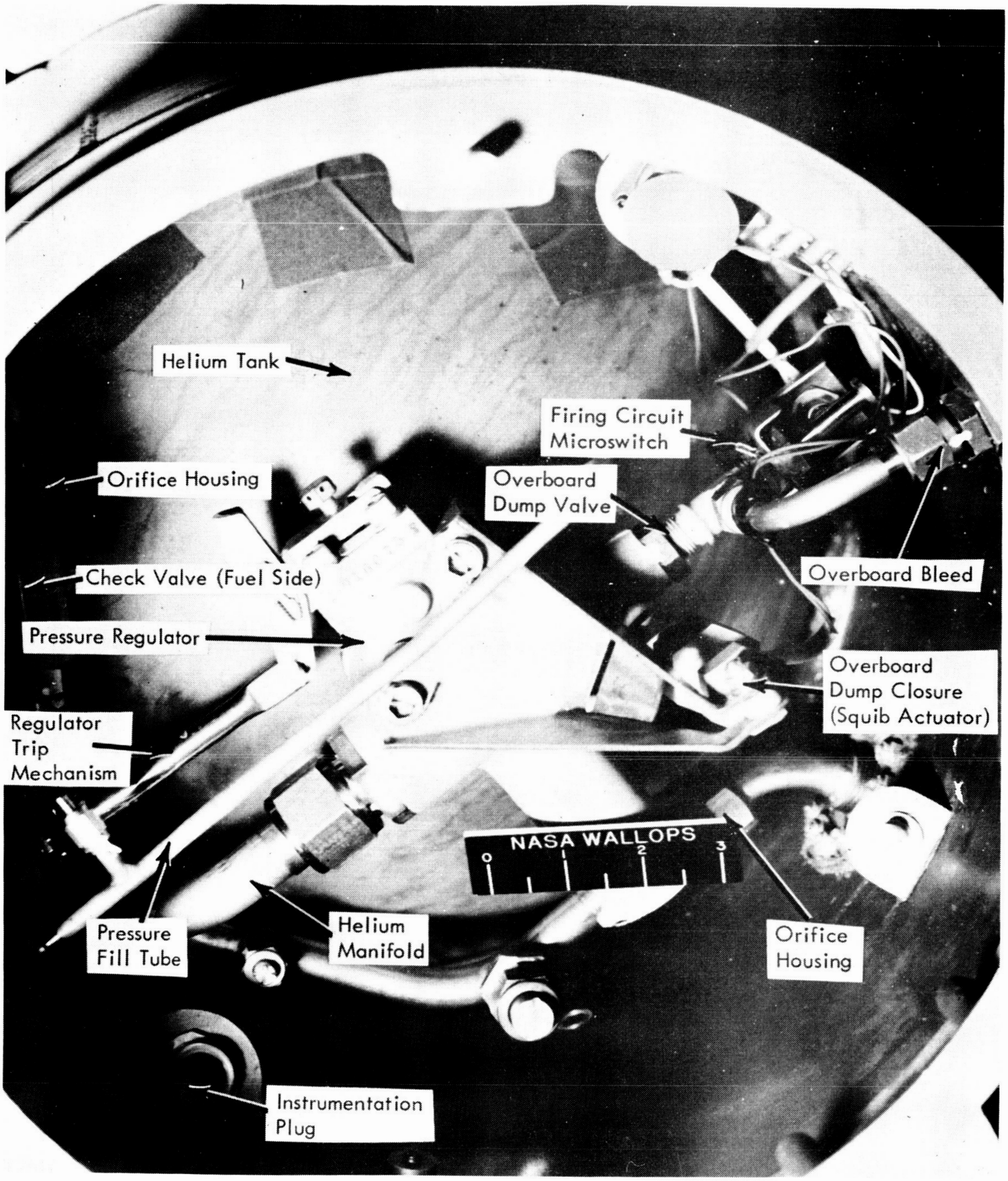


Figure 3—Aerobee 150A Forward Skirt

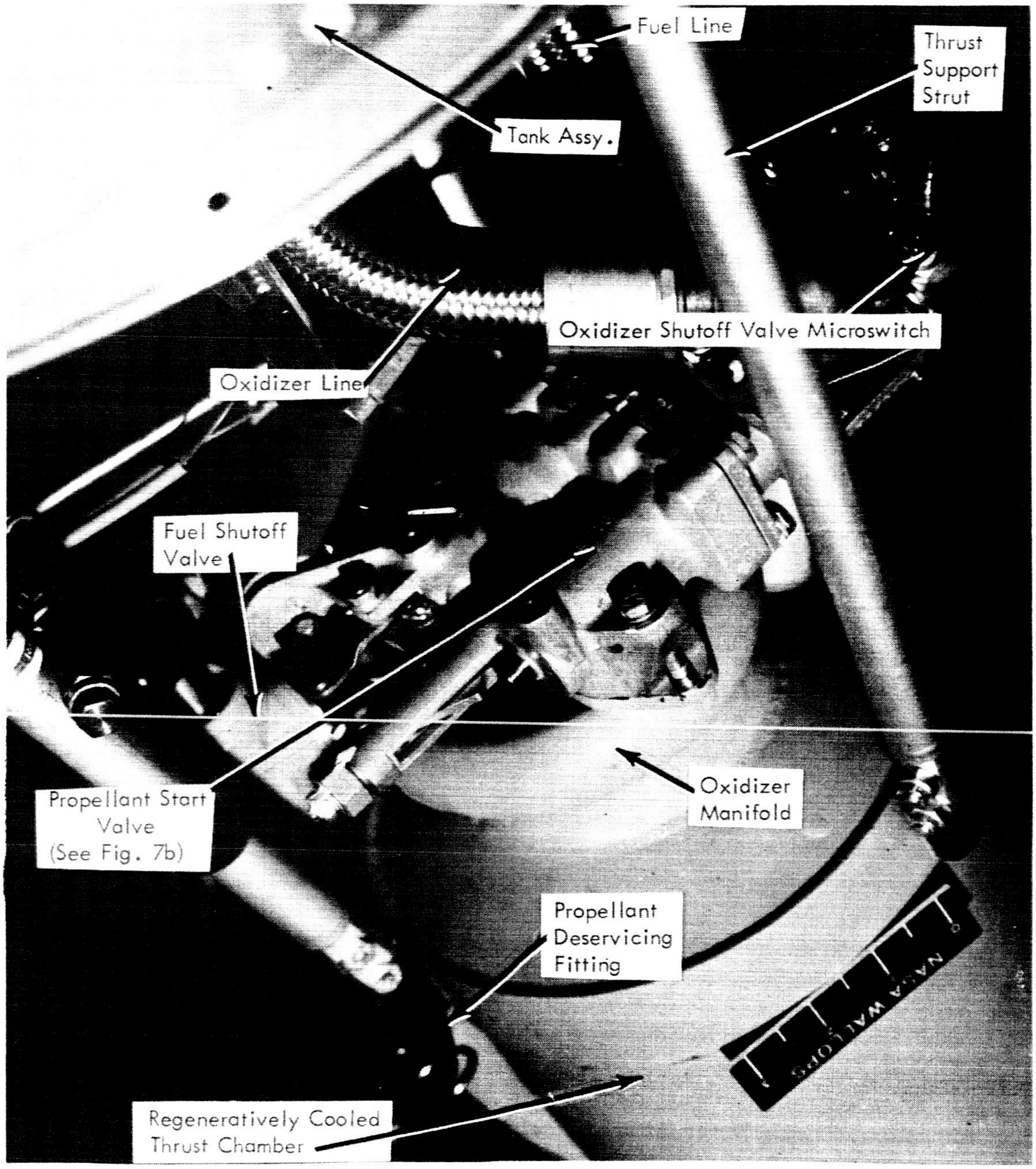


Figure 4—Aerobee 150A Tail Assembly

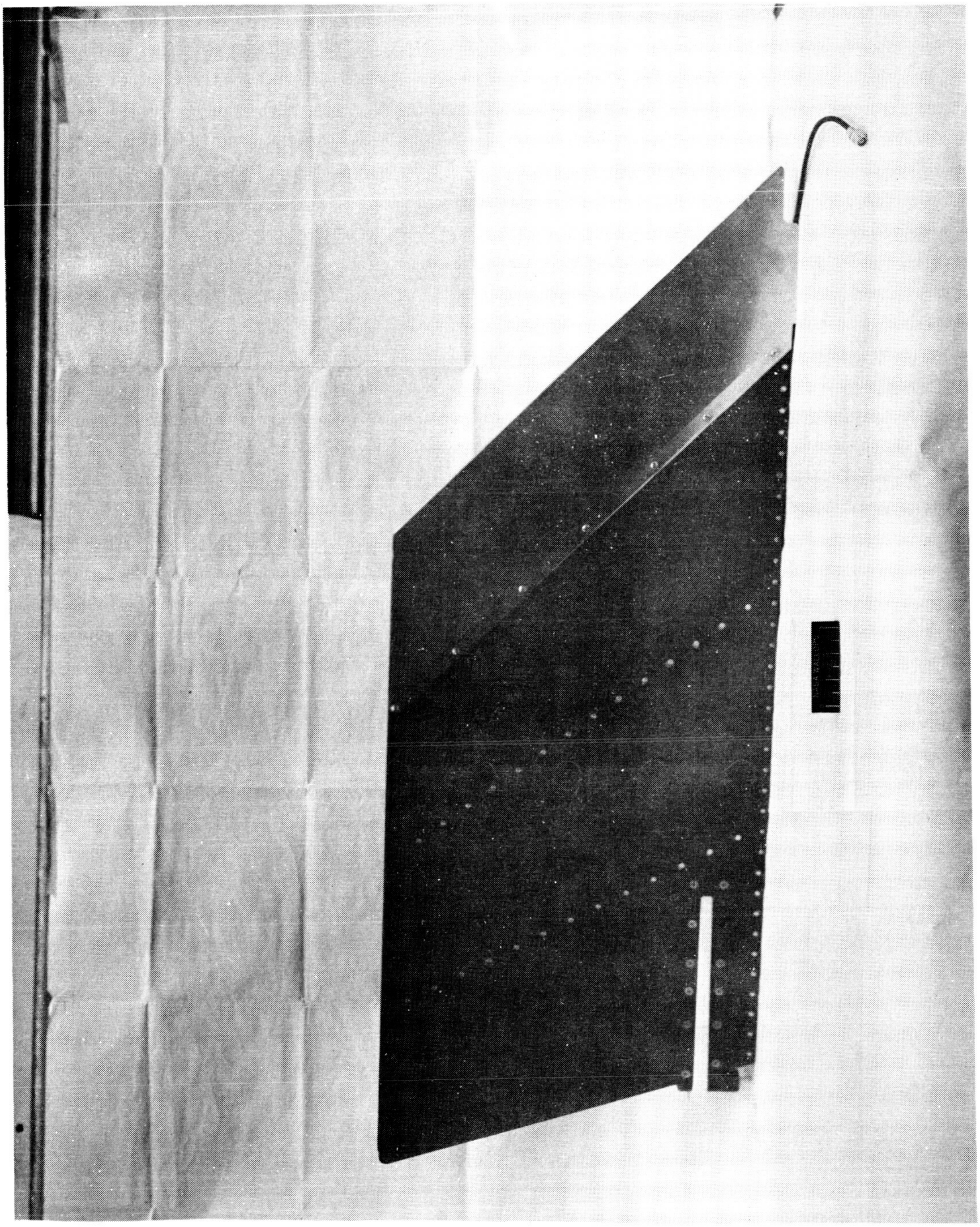


Figure 5–Aerobee 150A Sustainer Fin

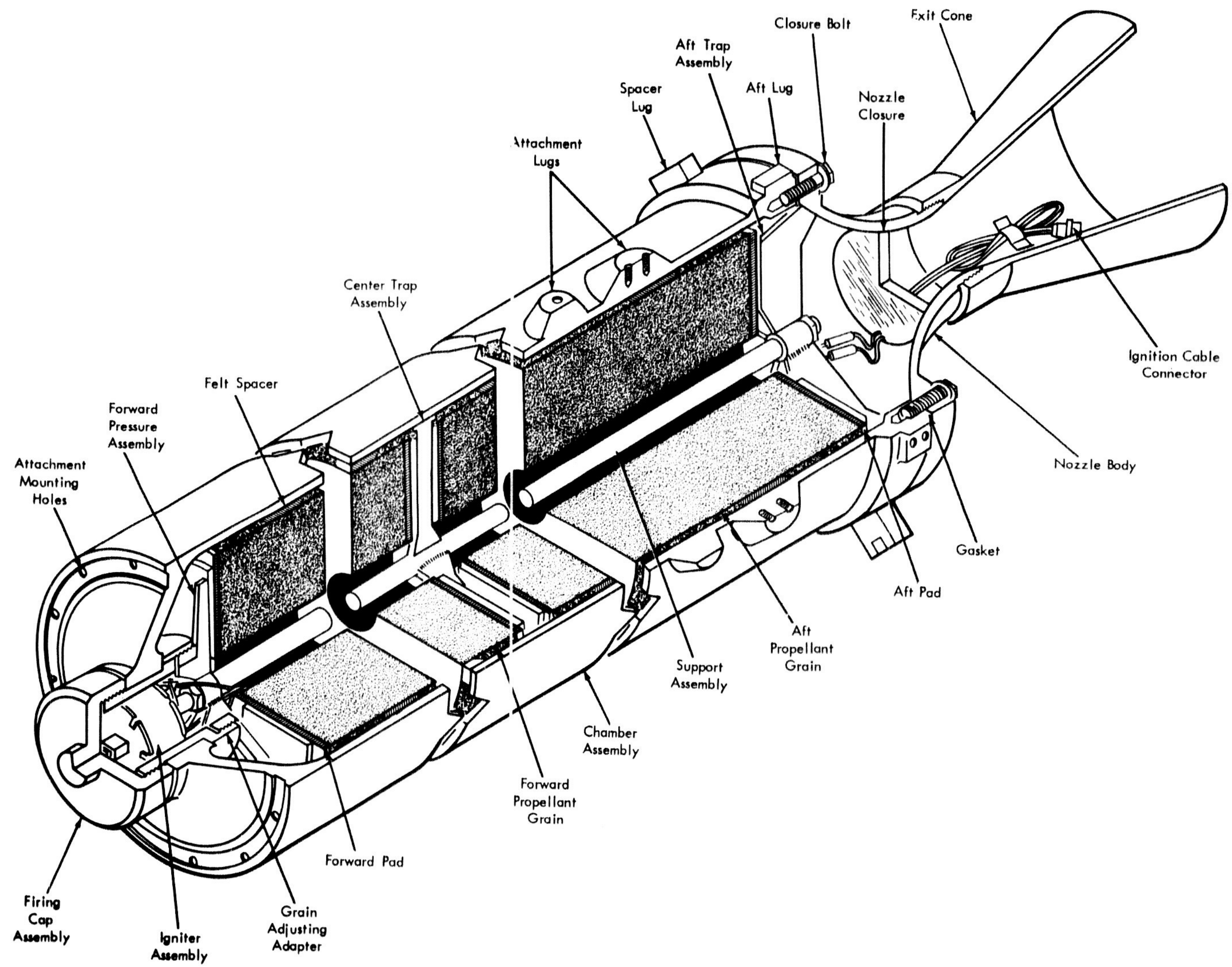
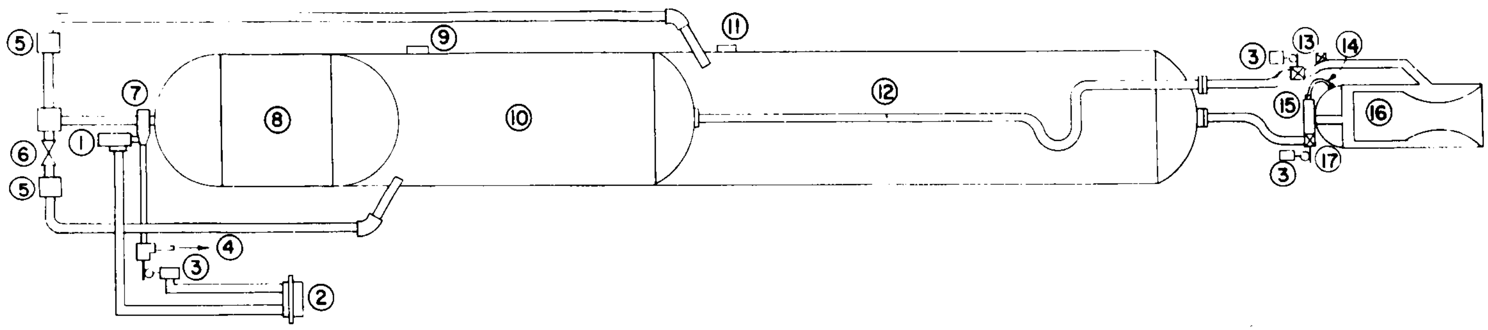


Figure 6–2.5KS – 18,000 Aerobee Booster



1. Overboard Dump Actuating Cylinder
2. Electrical Receptacle
3. Microswitch
4. Overboard Bleed Port
5. Orfice & Burst Diaphragm Assy.
6. Fuel Check Valve
7. Pressure Regulator & Dump
8. Pressure Tank
9. Fuel Fill Boss
10. Fuel Tank
11. Oxidizer Fill Boss
12. Oxidizer Tank
13. Fuel Shutoff & Drain Valve
14. Chamber Pressure Line
15. Propellant Start Valve
16. Thrust Chamber Assembly
17. Oxidizer Shutoff & Drain Valve

Figure 7A—Propulsion System, Schematic Diagram

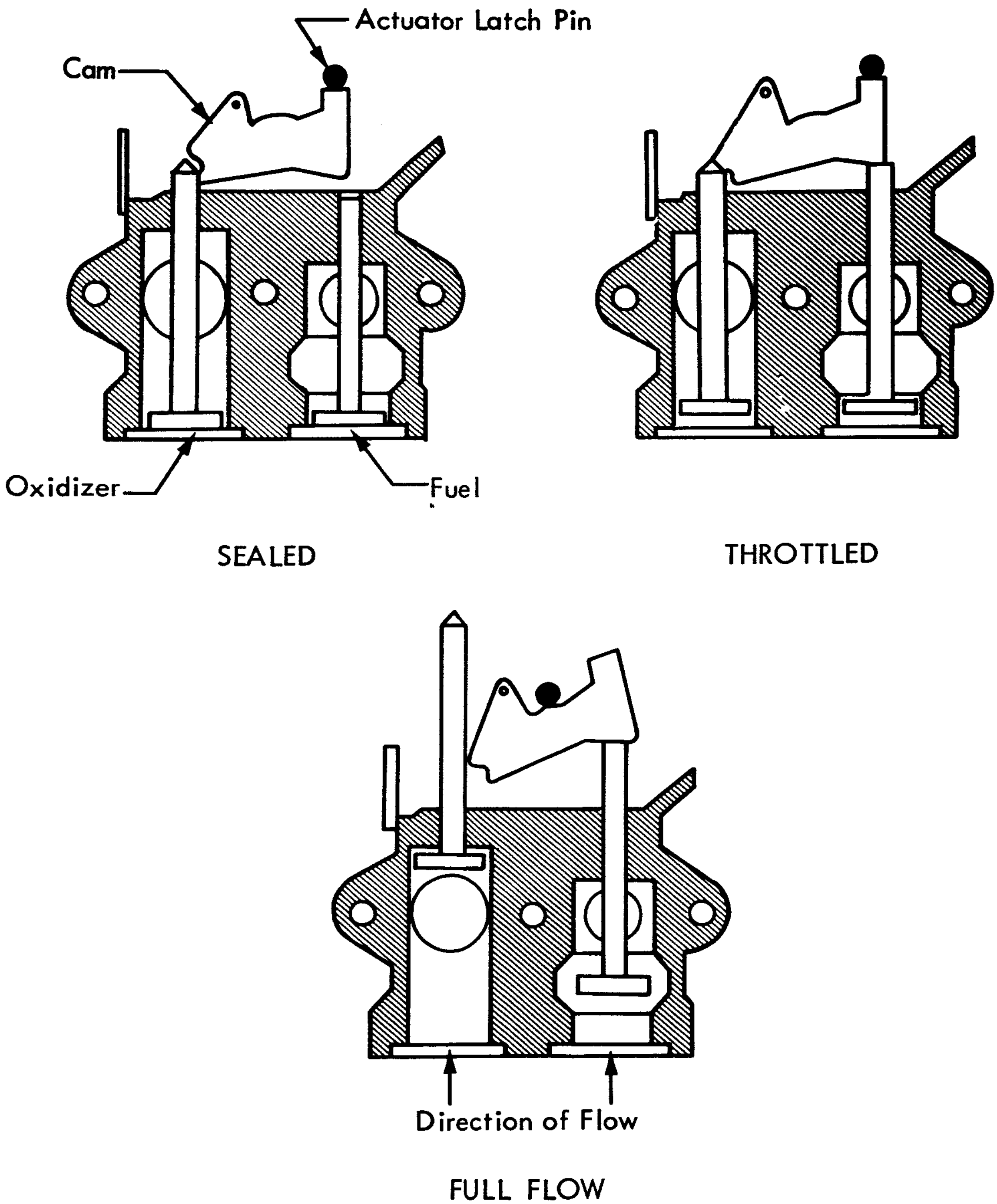


Figure 7B–Propellant Start Valve Positions

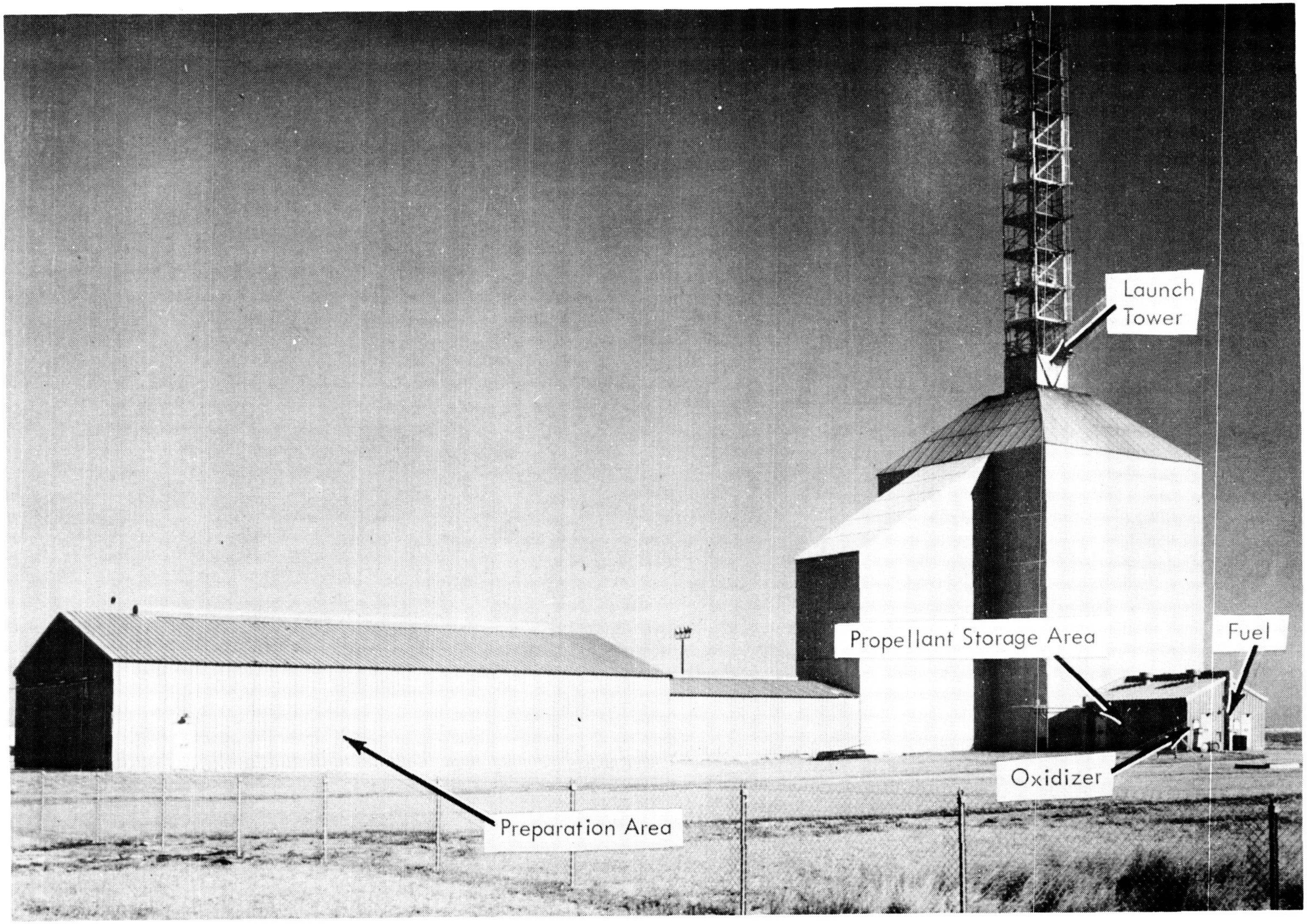


Figure 8–Aerobee Launch Facility, Wallops Island, Virginia

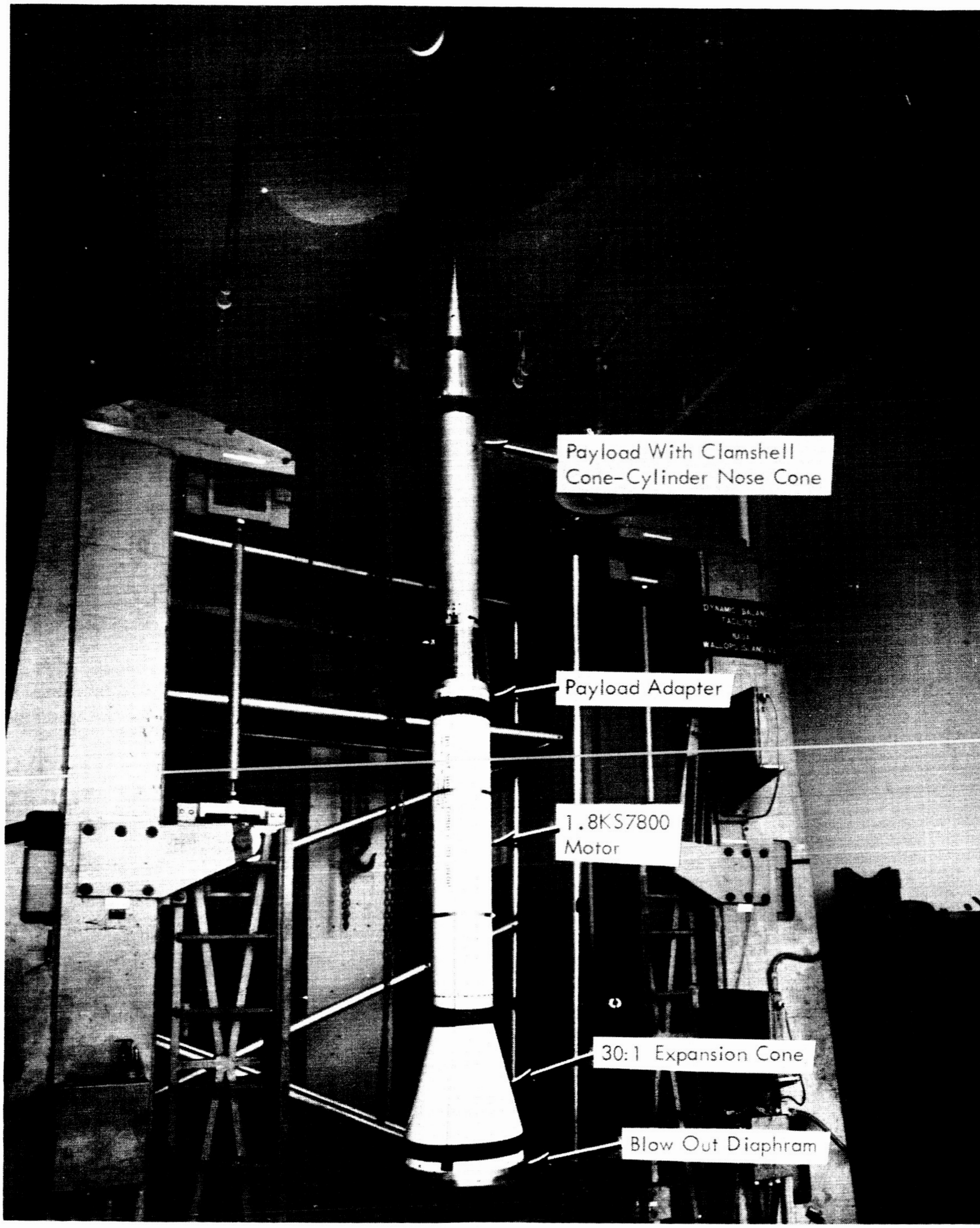


Figure 9–Aerobee 300A (Third Stage) in Dynamic Balance Facility

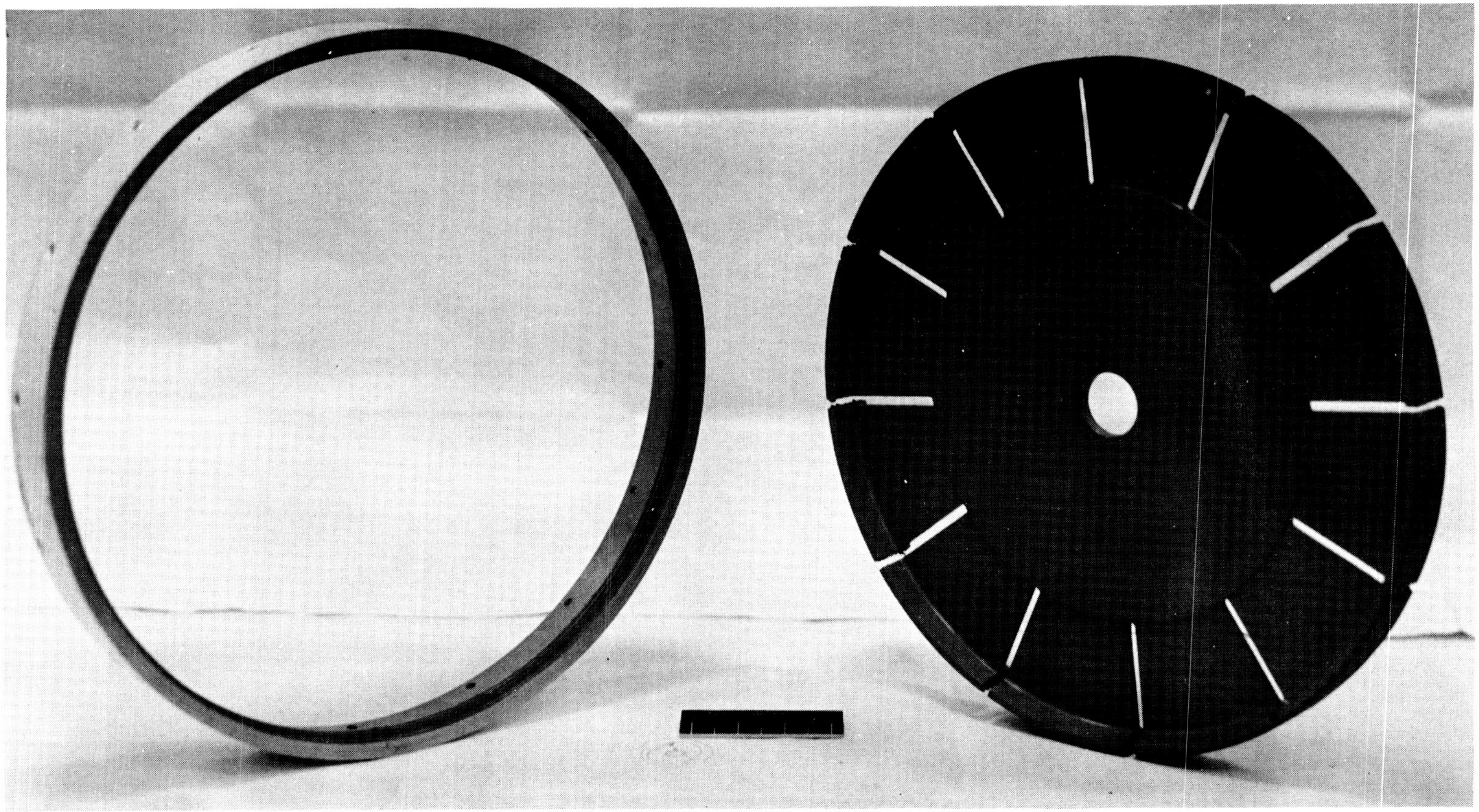


Figure 10–Aerobee 300A Blowout Diaphragm

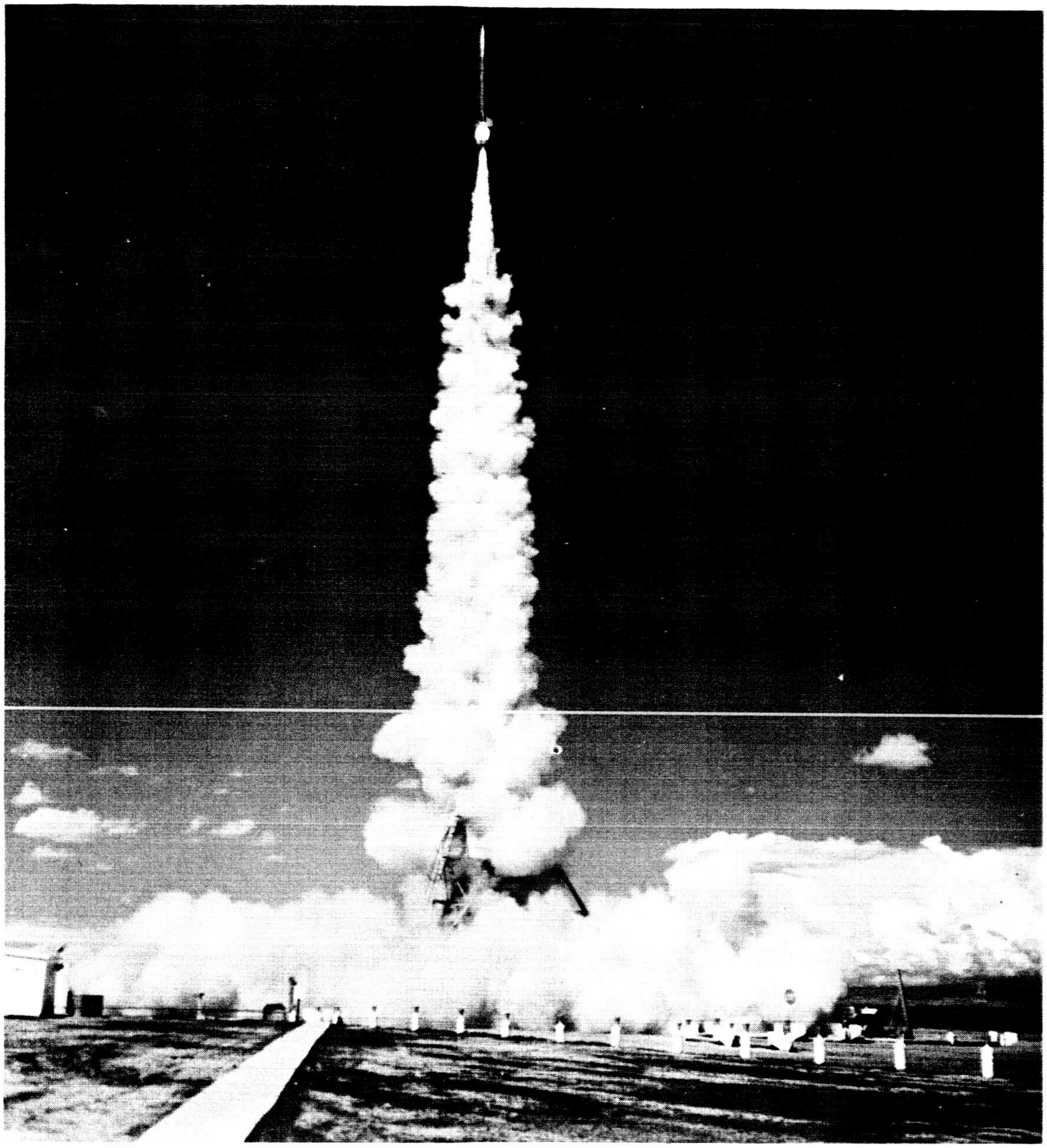


Figure 11A—WSMR Aerobee 150 Launch

Figure 11B—Aerobee Tower Installation, White Sands Missile Range

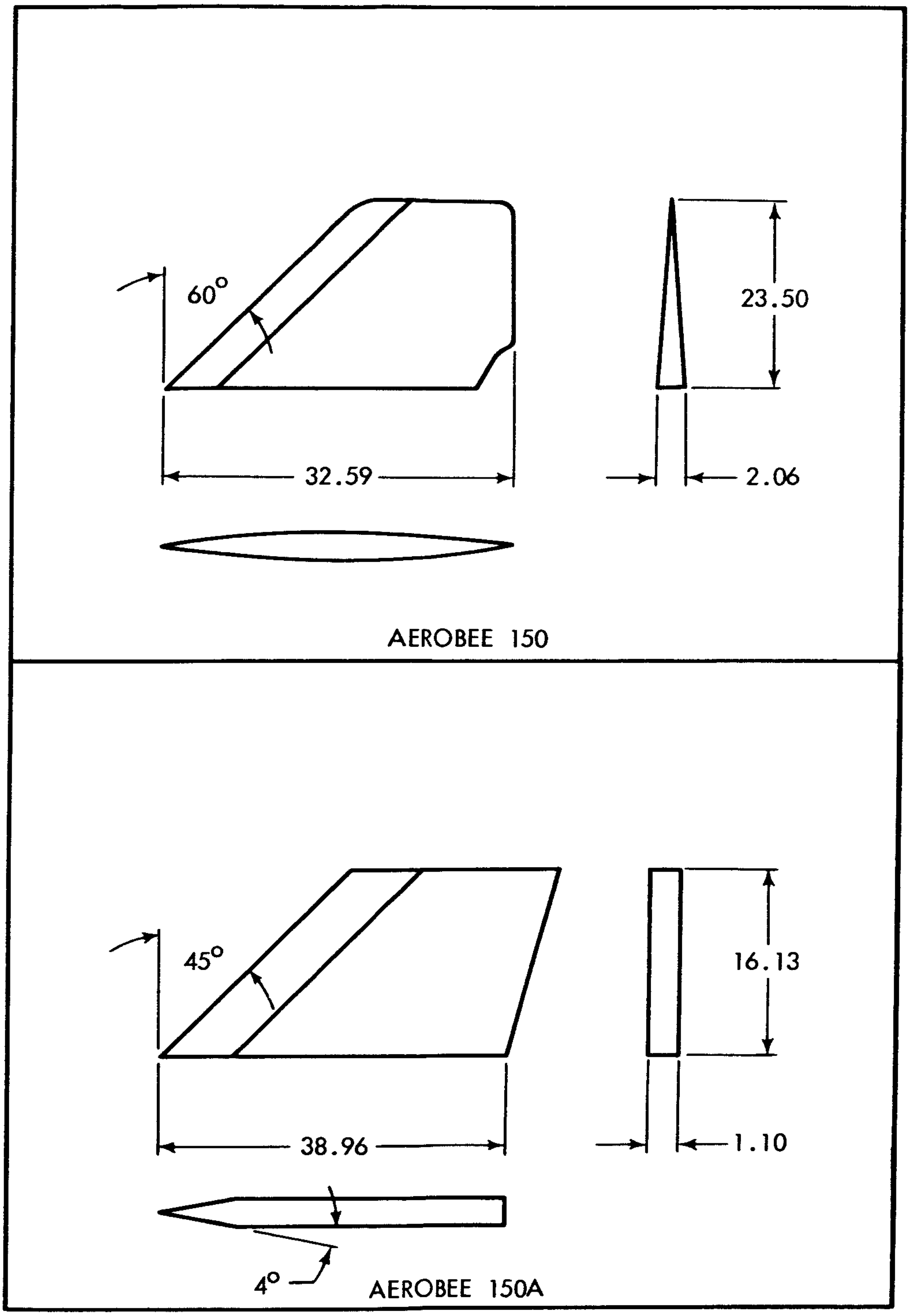


Figure 12–Aerobee Fin Configuration

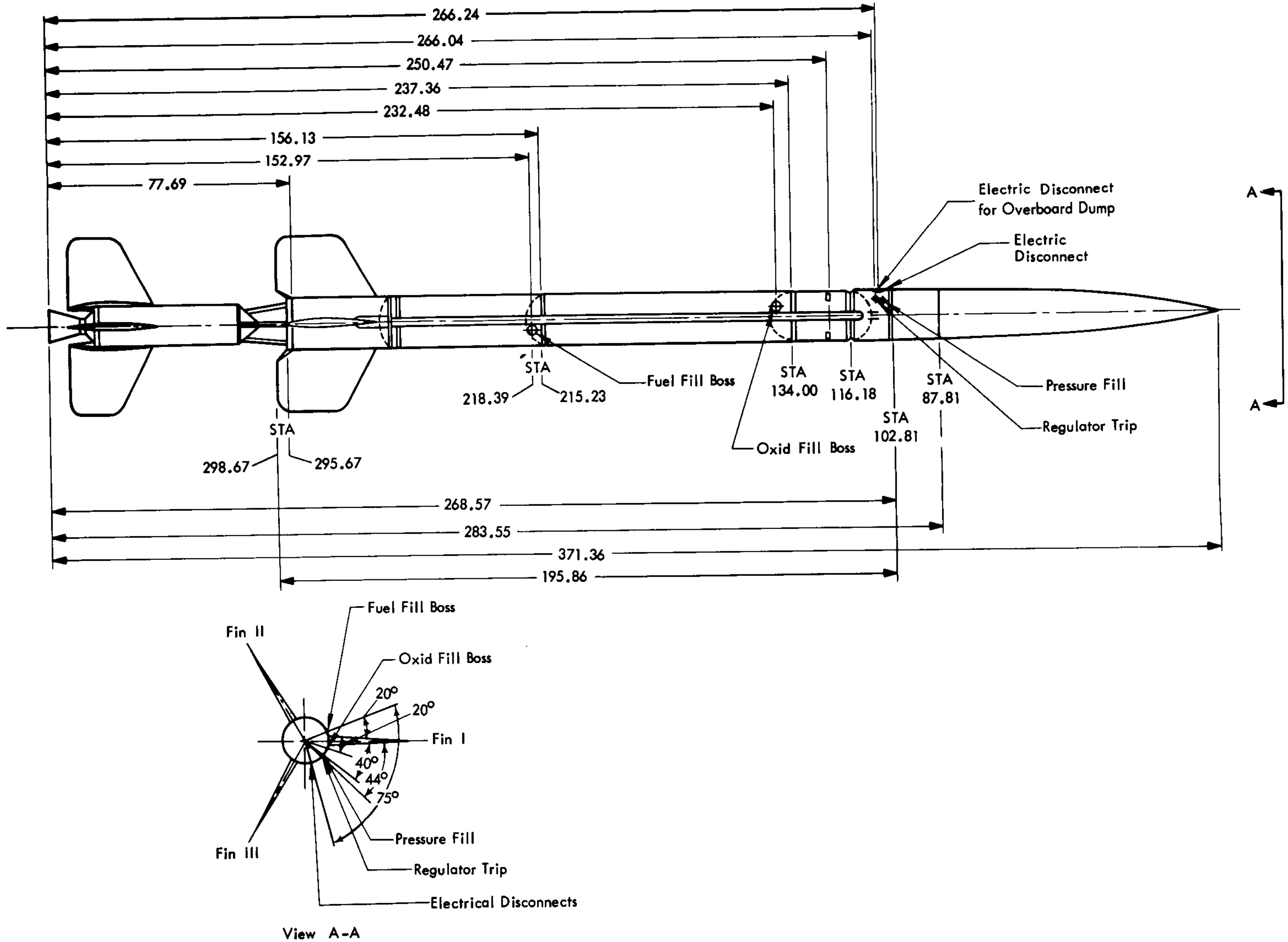


Figure 13—Aerobee 150 Outline Drawing

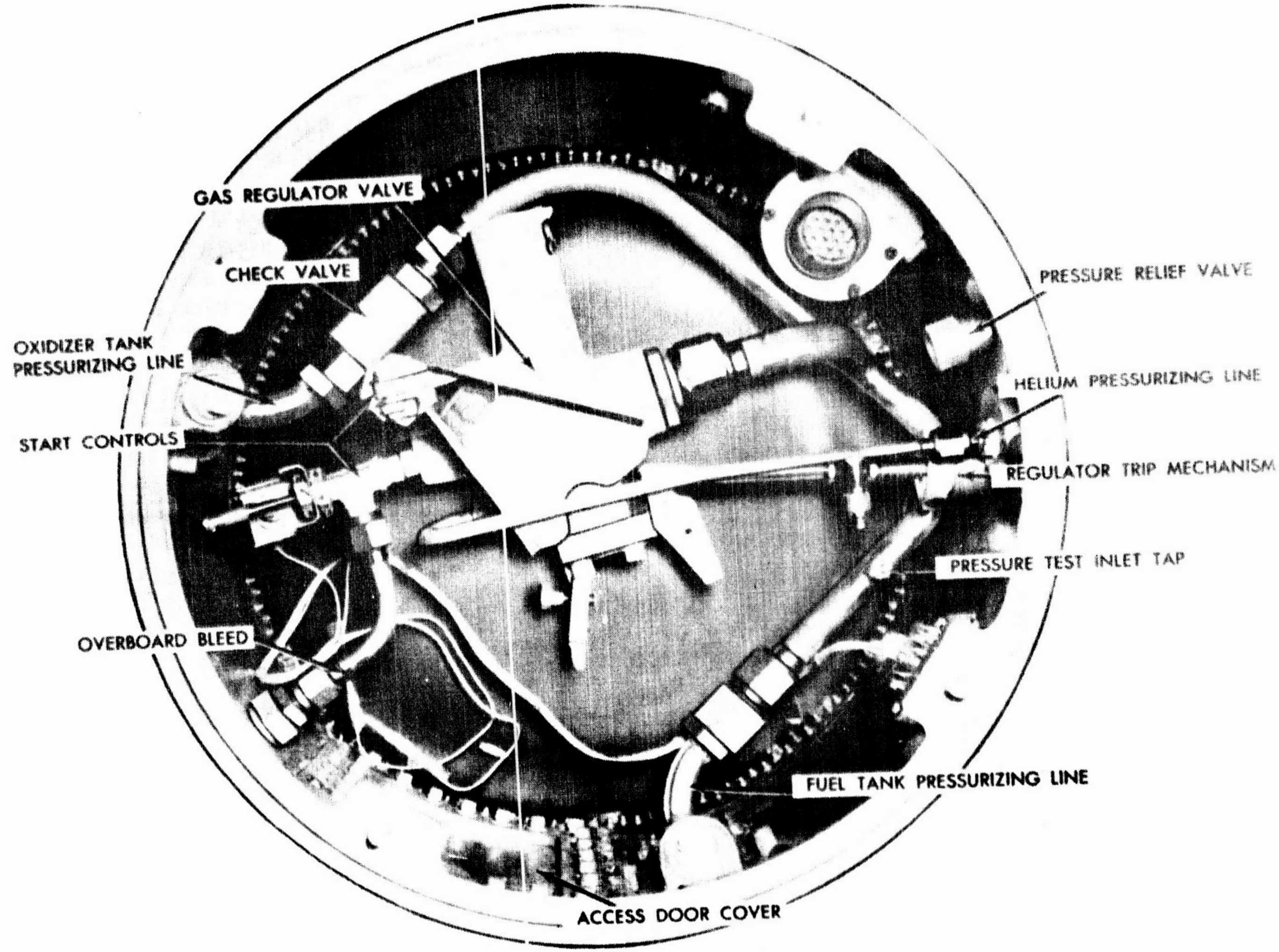


Figure 14—Aerobee 150 Forward Skirt

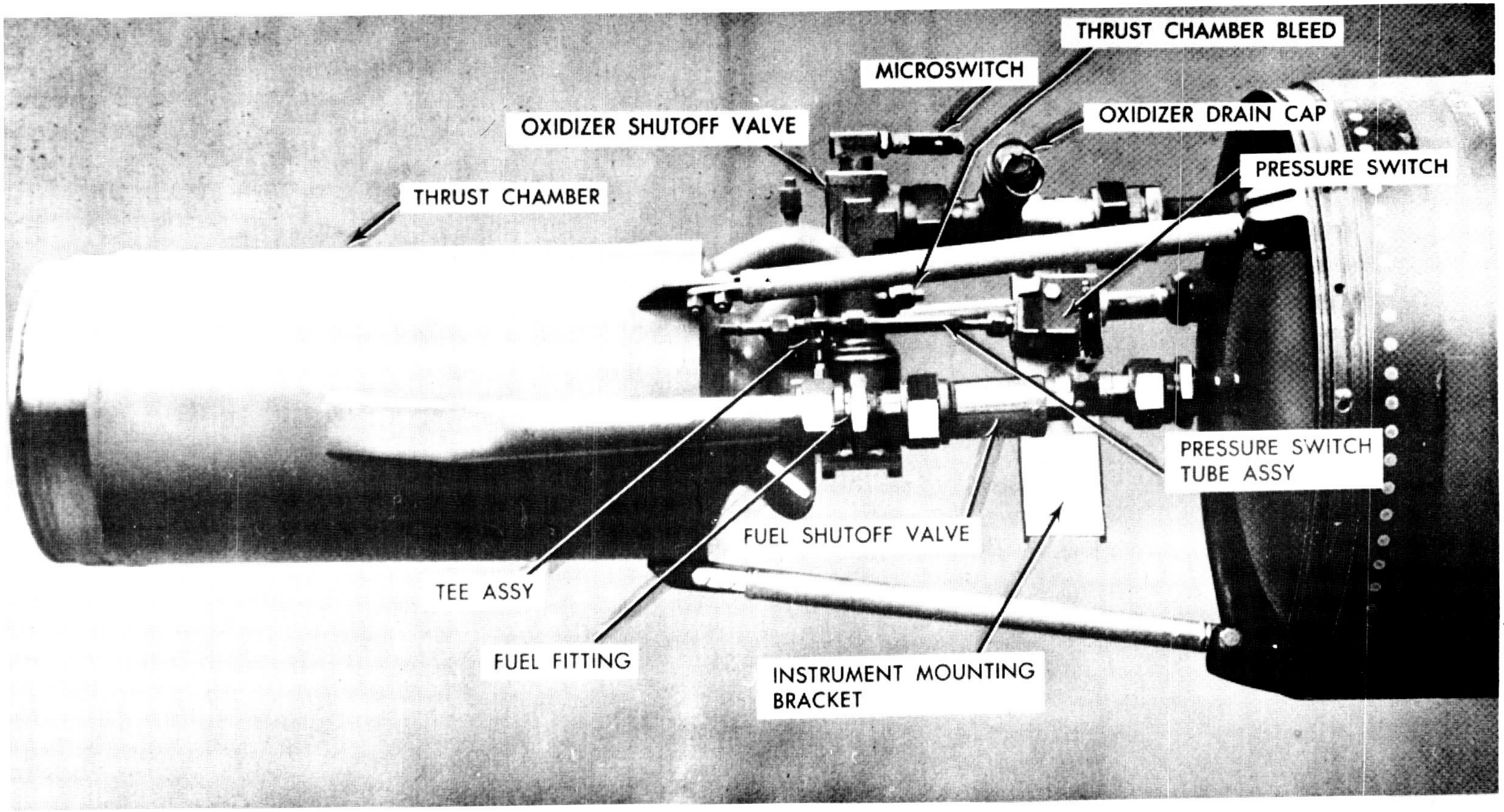


Figure 15–Aerobee 150 Tail Assembly

Figure 16–Aerial View, Fort Churchill Launch Facility

APPENDIX A – PART II

LIST OF ILLUSTRATIONS

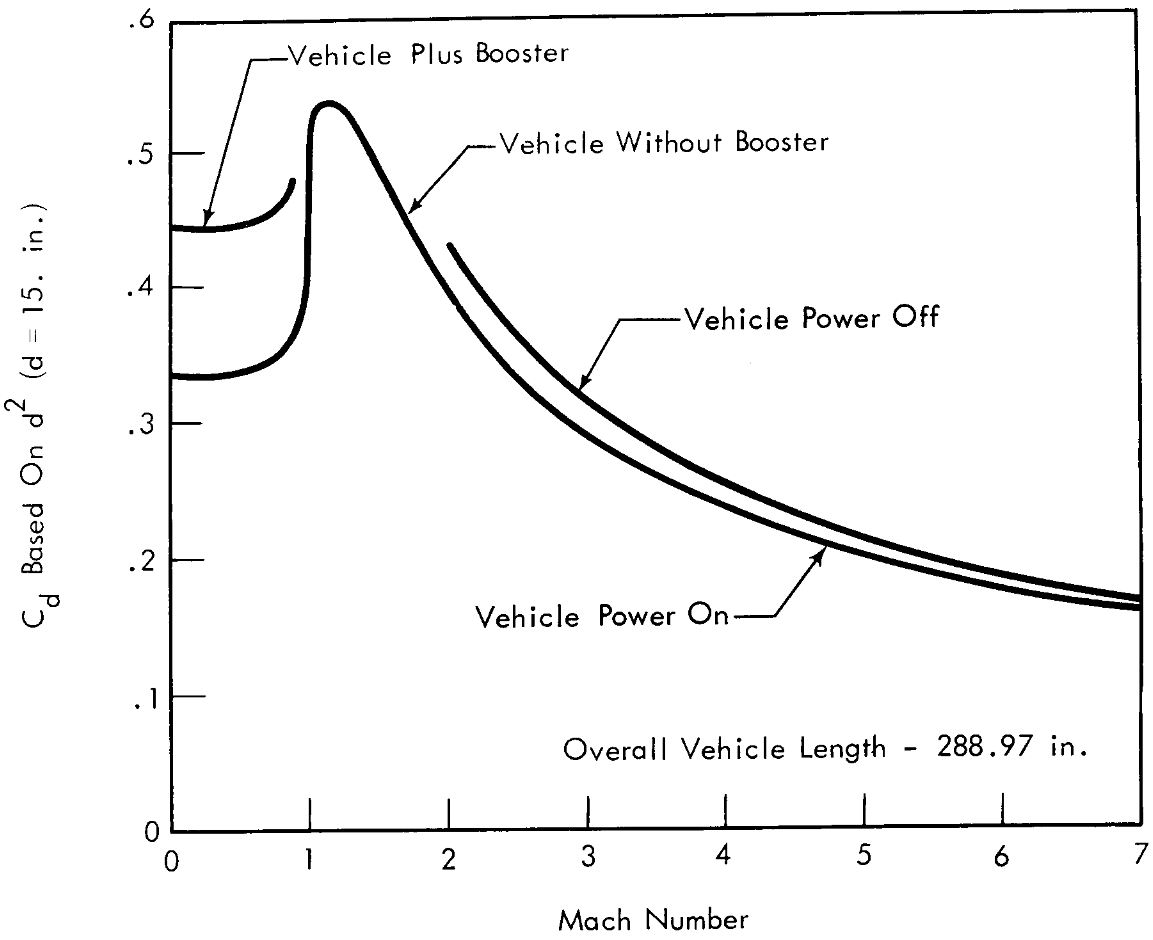


Figure 1—Aerobee 150A Drag Coefficient Vs. Mach Number

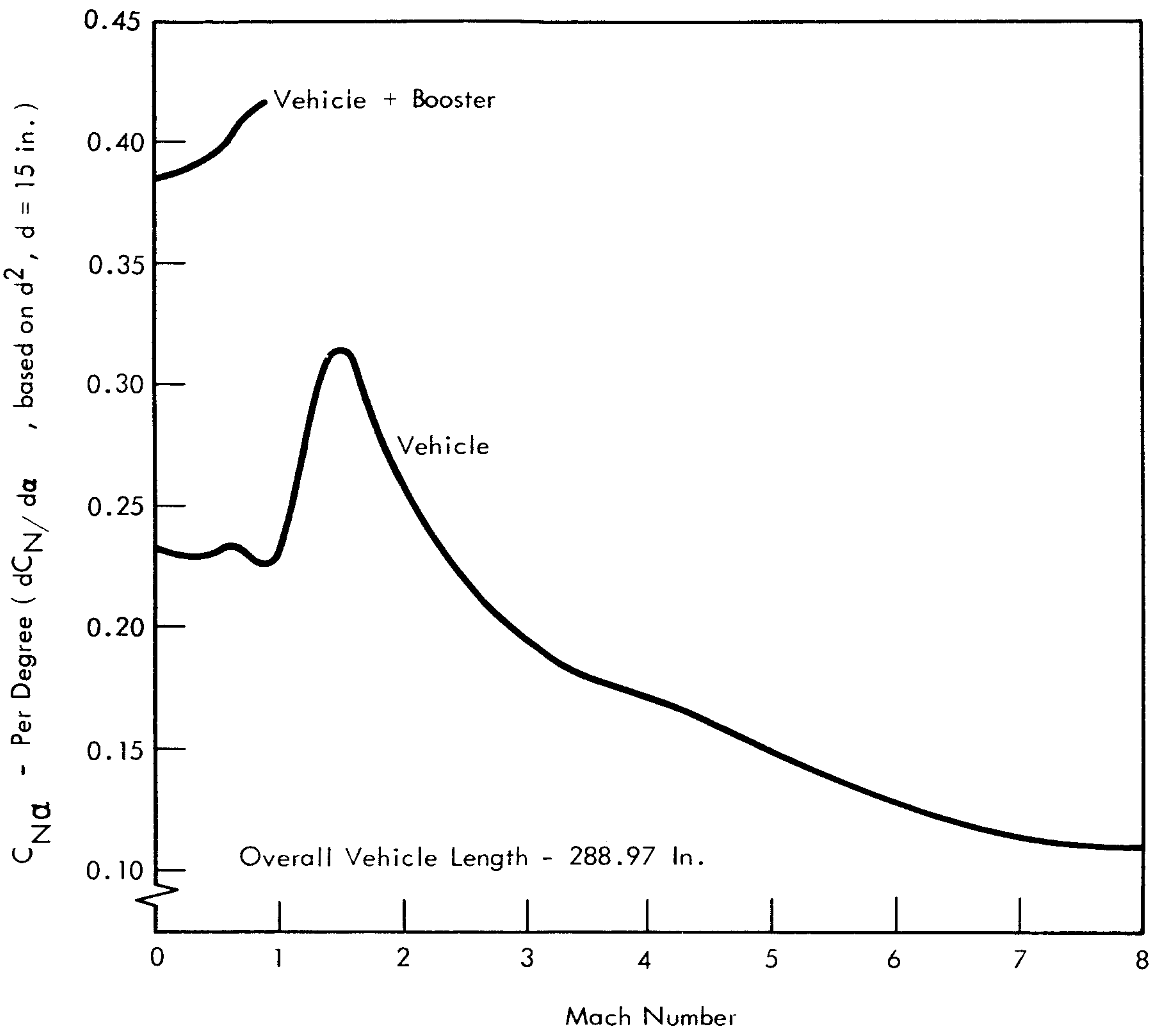


Figure 2–Aerobee 150A Normal Force Coefficient Slope Vs. Mach Number

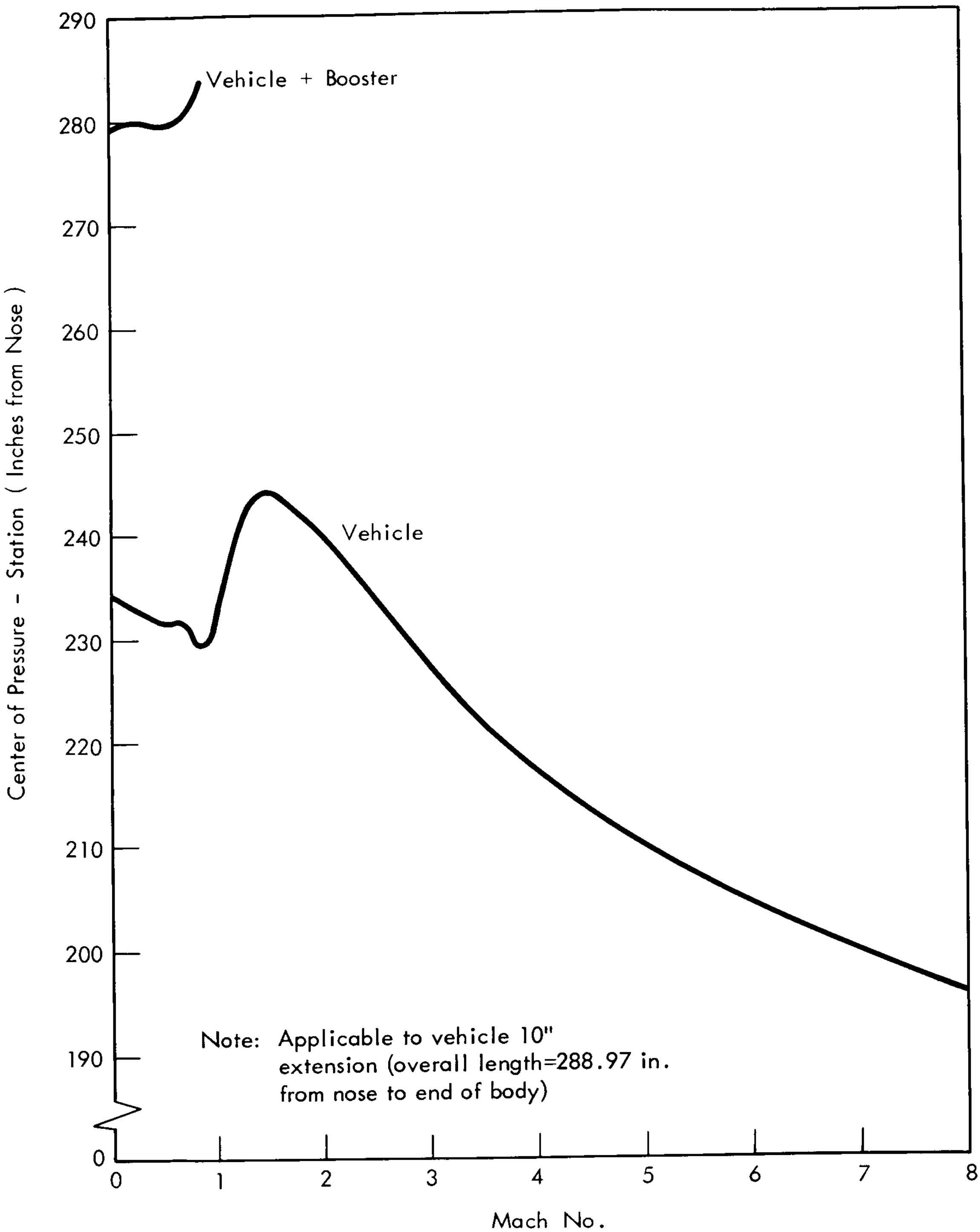


Figure 3—Aerobee 150A Center of Pressure Versus Mach Number

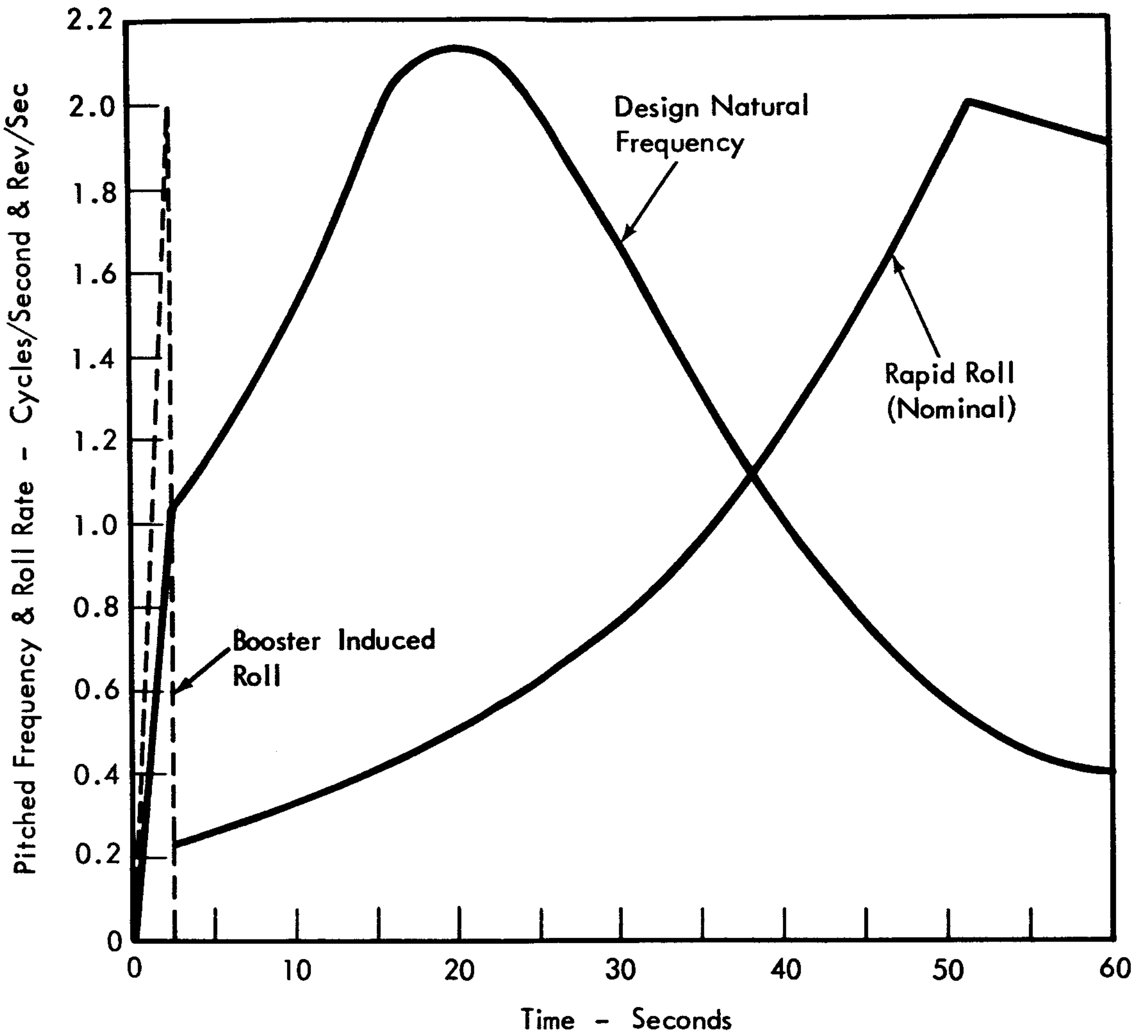


Figure 4–Aerobee 150A Pitching Frequency & Roll Rate Vs. Time

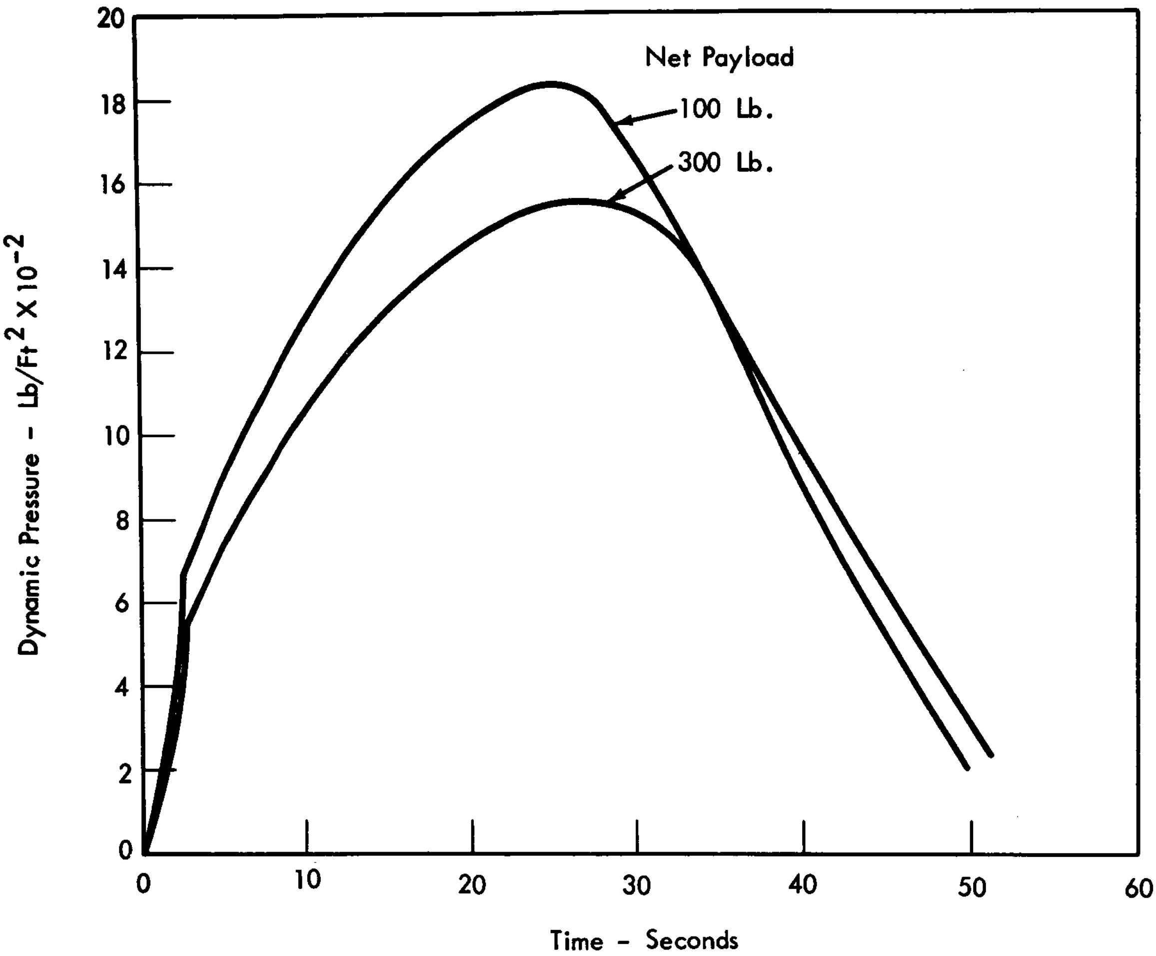


Figure 5—Aerobee 150A Dynamic Pressure (q) Vs. Time

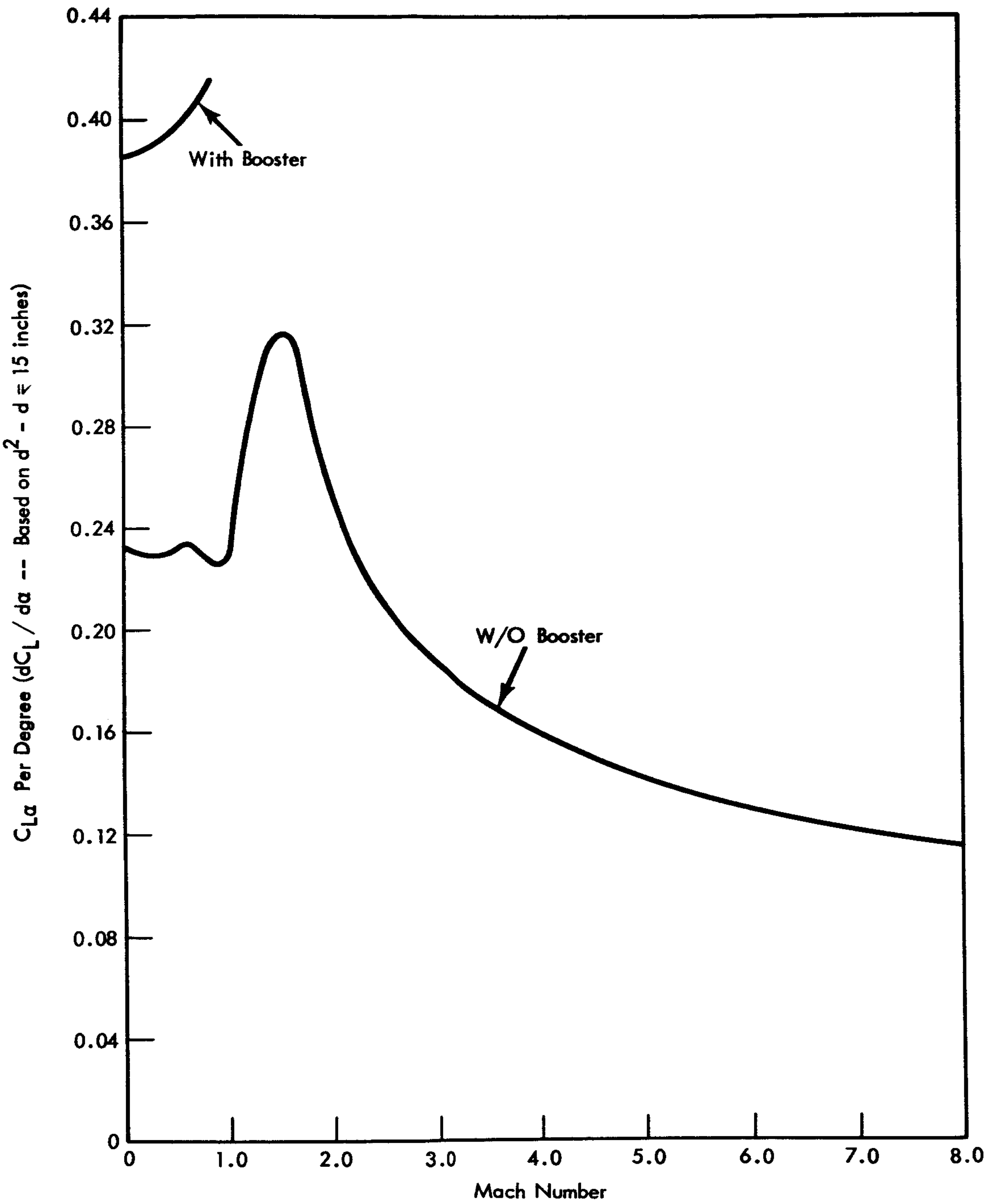


Figure 6—Aerobee 150A Normal Lift Coefficient Slope Vs. Mach Number

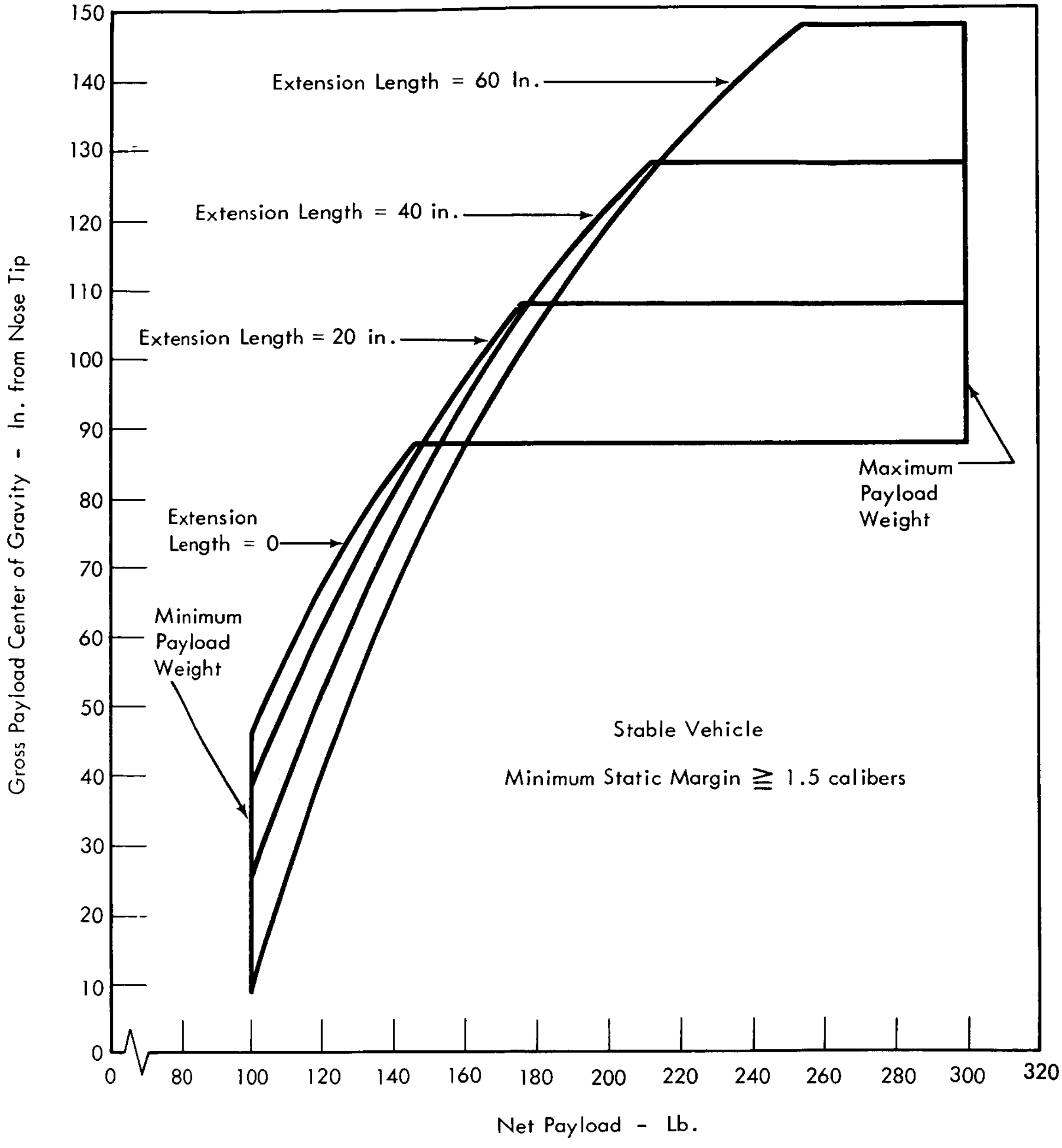


Figure 7—Aerobee 150A Payload Stability Requirements, Payload Structure Weight =15.5 Lb.

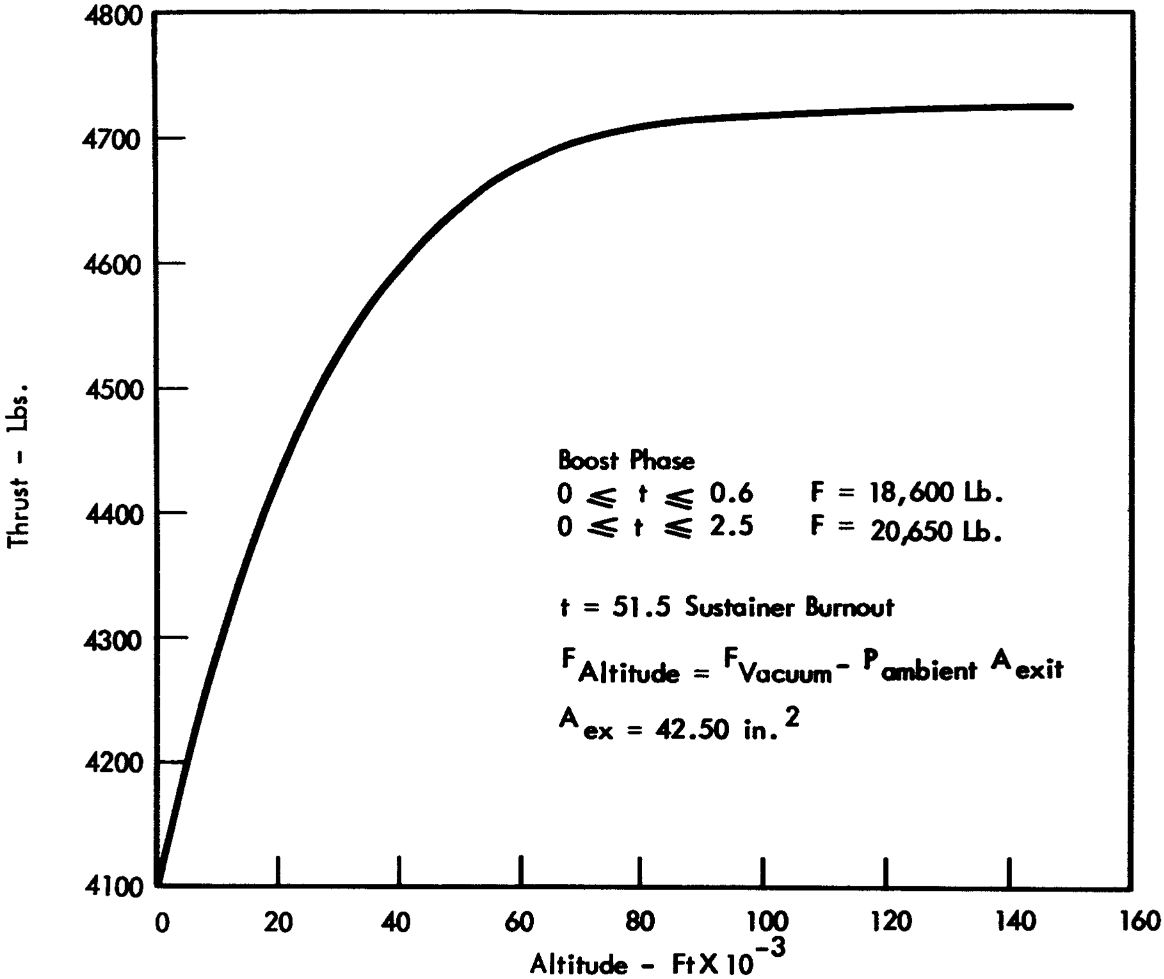


Figure 8—Aerobee 150A Sustainer Thrust Vs. Altitude

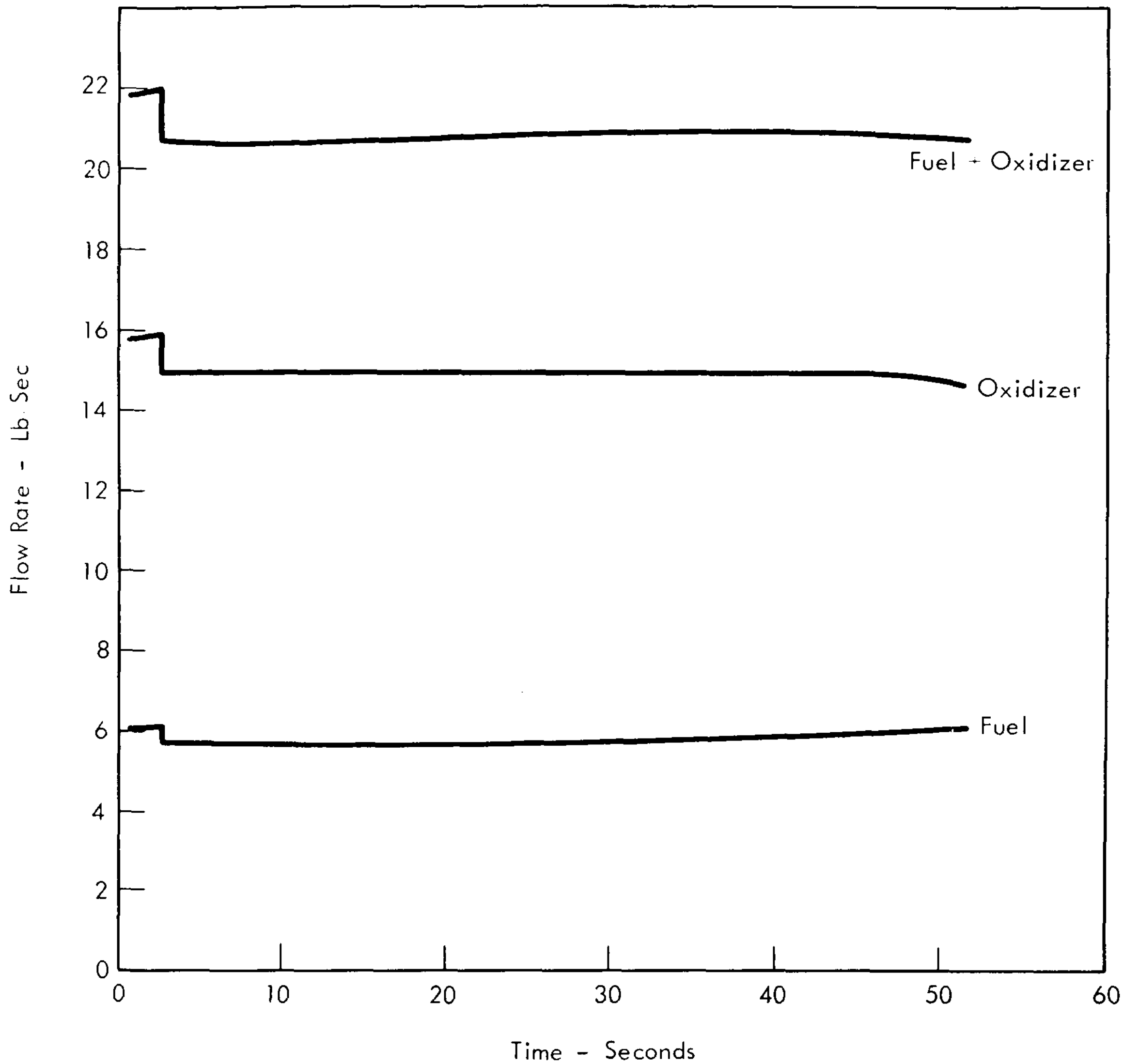


Figure 9—Aerobee 150A Flow Rates Vs. Time

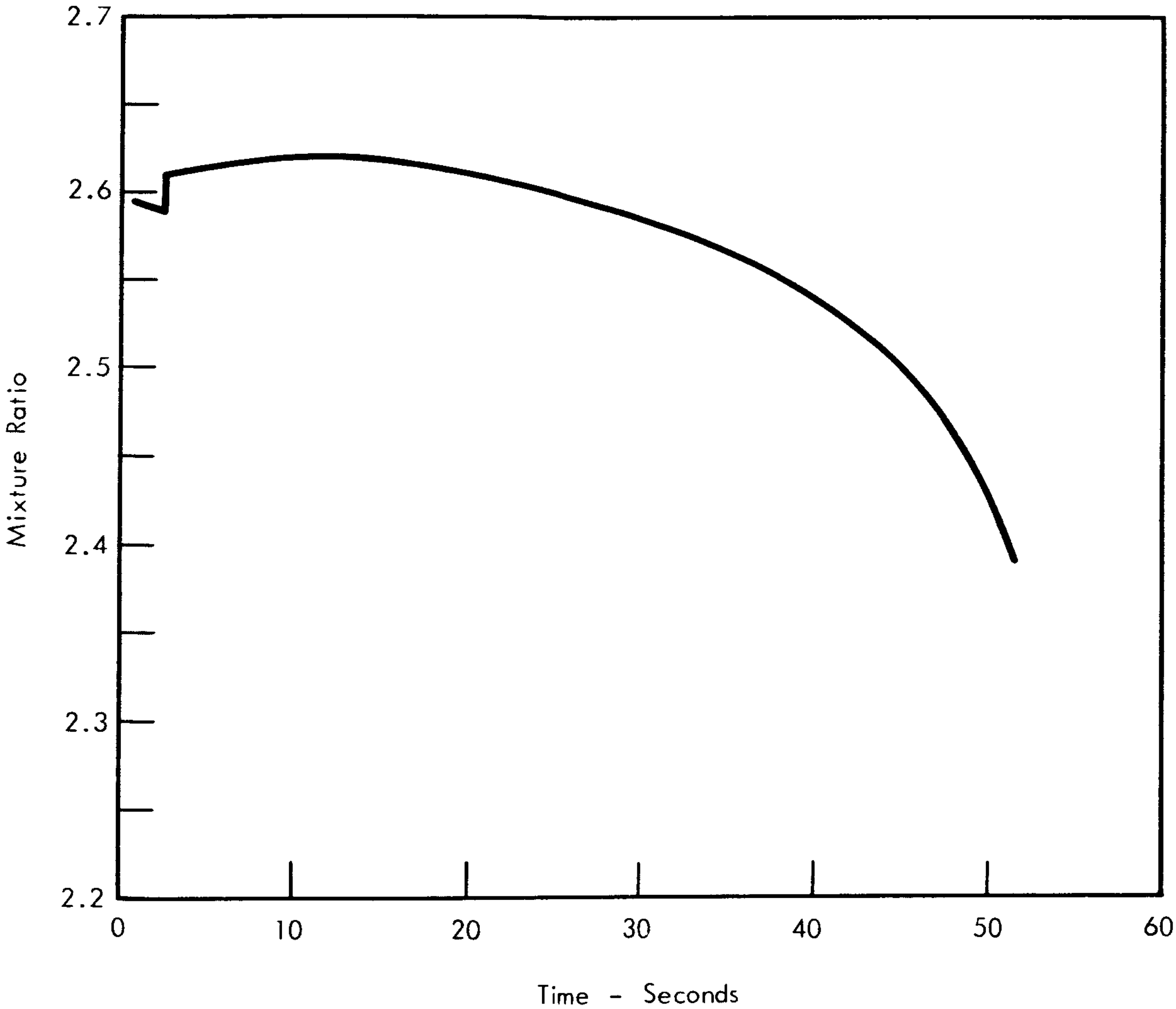


Figure 10—Aerobee 150A Mixture Ratio Vs. Time

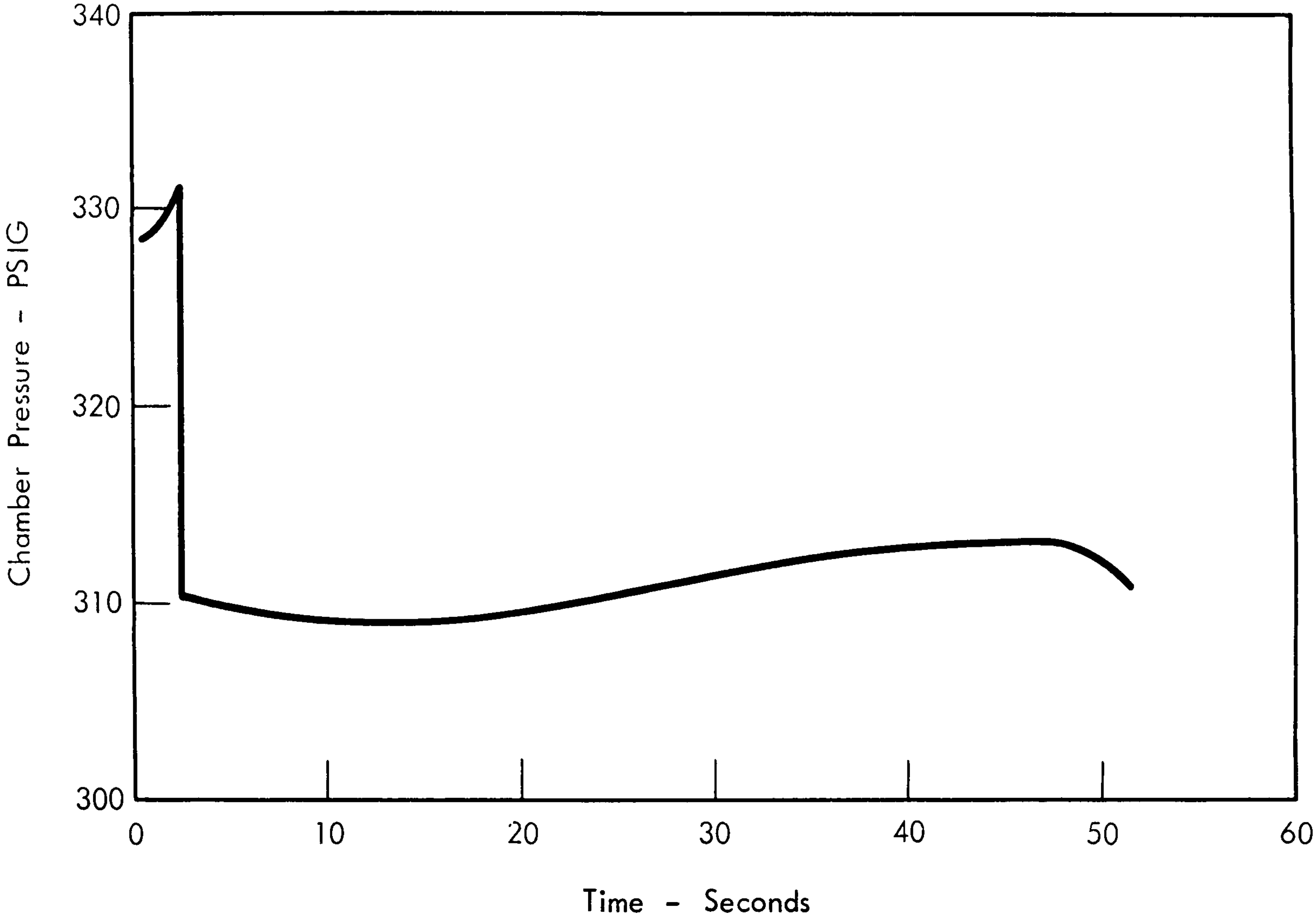


Figure 11—Aerobee 150A Chamber Pressure Vs. Time

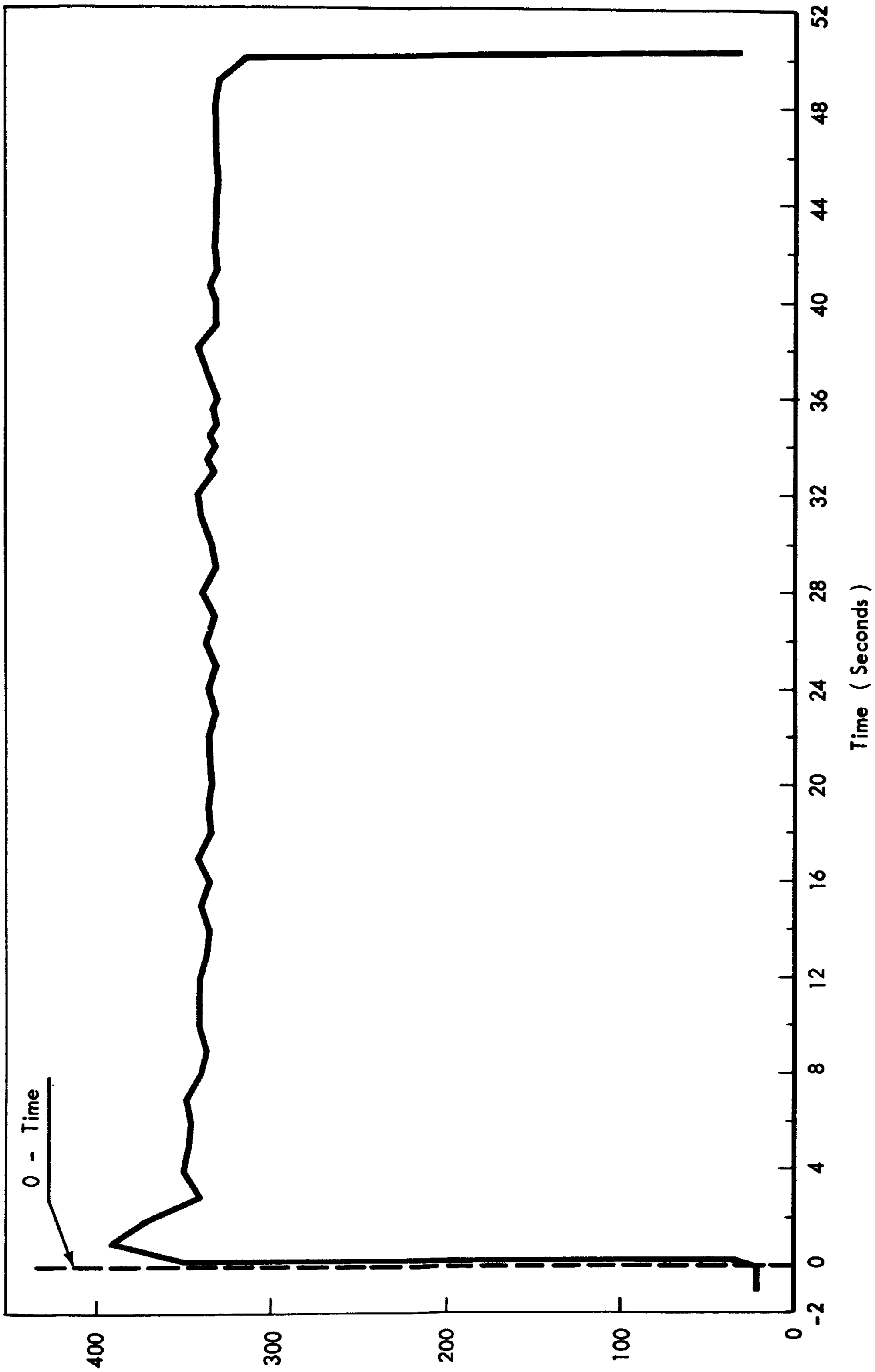


Figure 12—Aerobee 4.09 – 150A Chamber Pressure (P.S.I.'g.)

AEROBEE 150A
TRAJECTORY INPUT INFORMATION

Net Payload = 200 Lbs.

Weight at t = 0, 2144.8 lbs. (with 200 lbs. of NPL)

Launch Tower 160 ft.

BOOSTER PARAMETERS $0 \leqq t \leqq 0.6$

Isp (vac) = 188.3 lb.-sec./lbm

Ref. Area (S) = 1.5624 $ft.^2$

Nozzle Exit Area (Ae) = .466 $ft.^2$

Duration of Thrust = 0.6 sec.

Thrust (vac) = 19,585 lb.

BOOSTER + SUSTAINER PARAMETERS $.6 \leqq t \leqq 2.5$

Isp (vac) = 174.9 lb.-sec./lbm

Ref. Area (S) = 1.5625 $ft.^2$

Nozzle Exit Area (Ae) = .6145 $ft.^2$

Duration of Thrust = 1.9 sec.

Thrust (vac) = 21,949 lb.

SUSTAINER PARAMETERS $2.5 \leqq t \leqq 51.5$

Isp (vac) = 228.3 lb.-sec./lbm

Ref. Area (S) = 1.5625 $ft.^2$

Nozzle Exit Area (Ae) = .297 $ft.^2$

Duration of Thrust = 49.0 sec.

Thrust (vac) = 4728 lb.

Drop Weight (Booster) = 340.0 lbs.

Figure 13—Aerobee 150A Trajectory Input Information

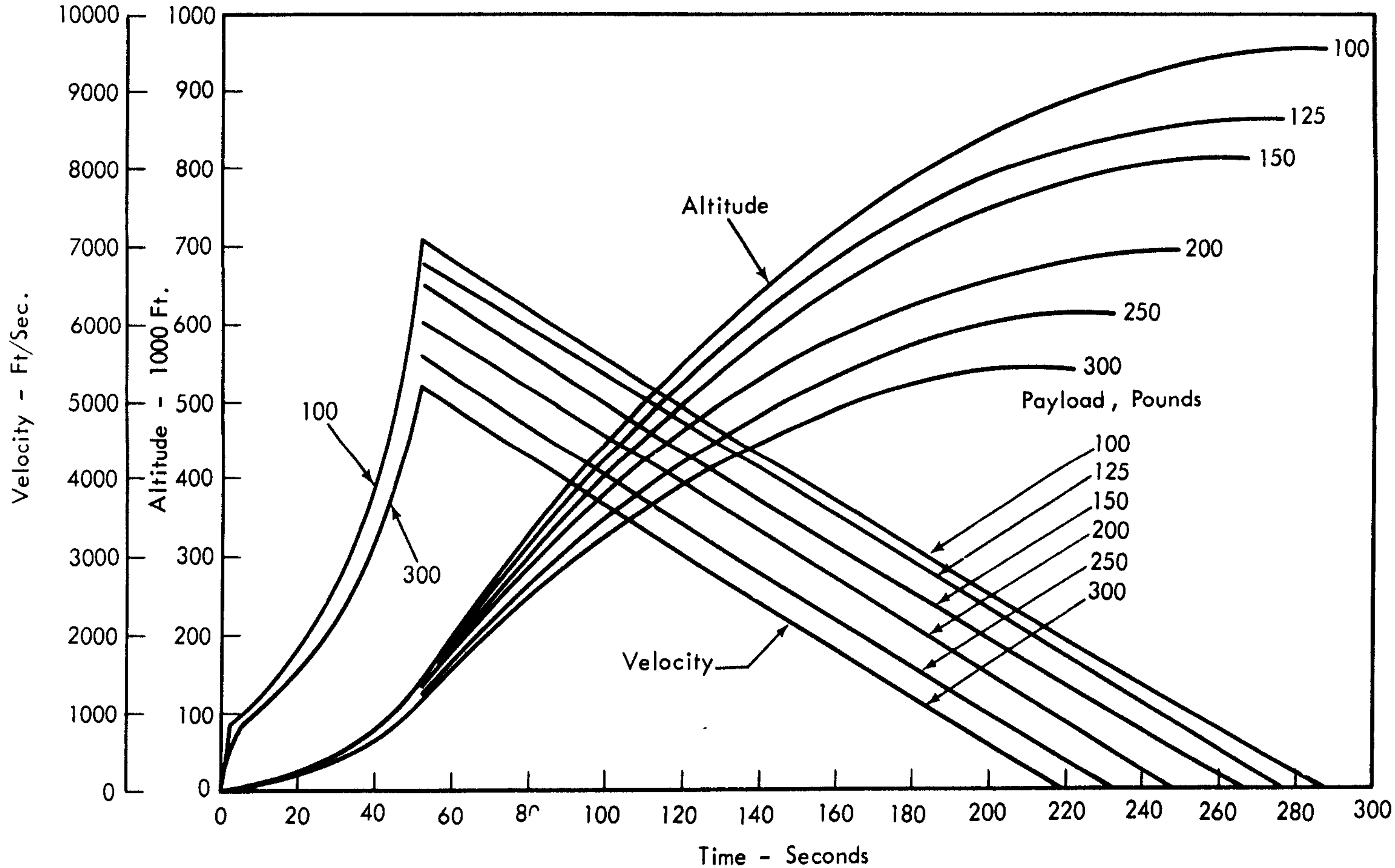
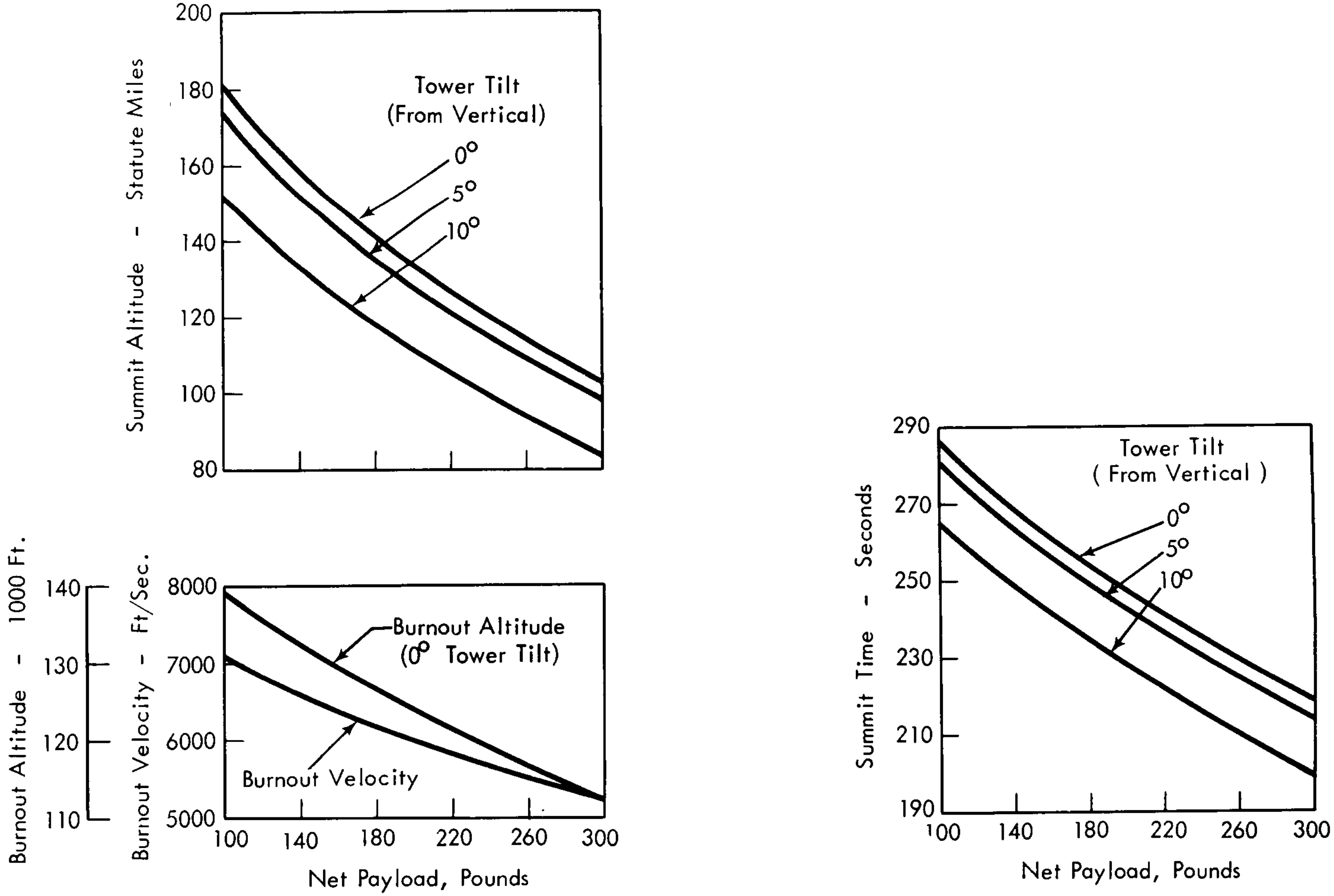
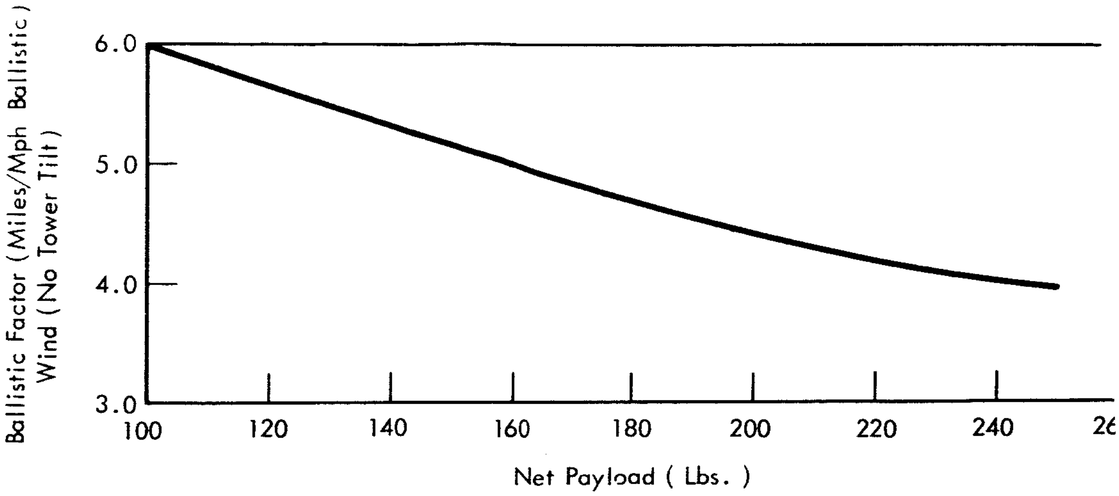


Figure -- 14 Aerobee 150A
Velocity & Altitude Vs. Time
For Various Payloads
Vertical Launch
Sea Level Launch

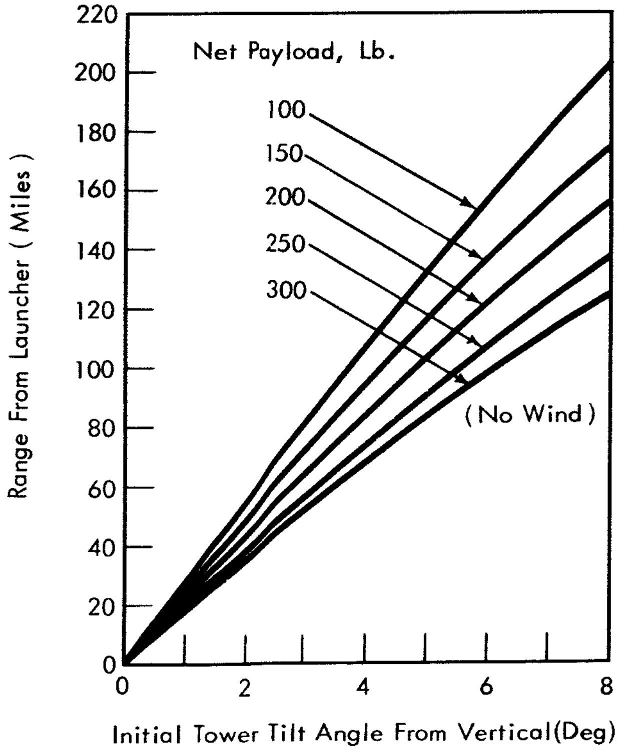


Note: Time to inspect
is approximately twice
the time to summit.

Figure 15 -- Aerobee 150A
Variation Of Performance With
Net Payload Weight
Sea Level Launch



Aerobee 150A
Ballistic Factor Vs. Payload
Launching Elev. - S. L.
ARDC Standard Atmosphere



gure 16 -- Range Versus Tower Tilt For Various Payloads
Launching Elev. - S. L.
ARDC Standard Atmosphere

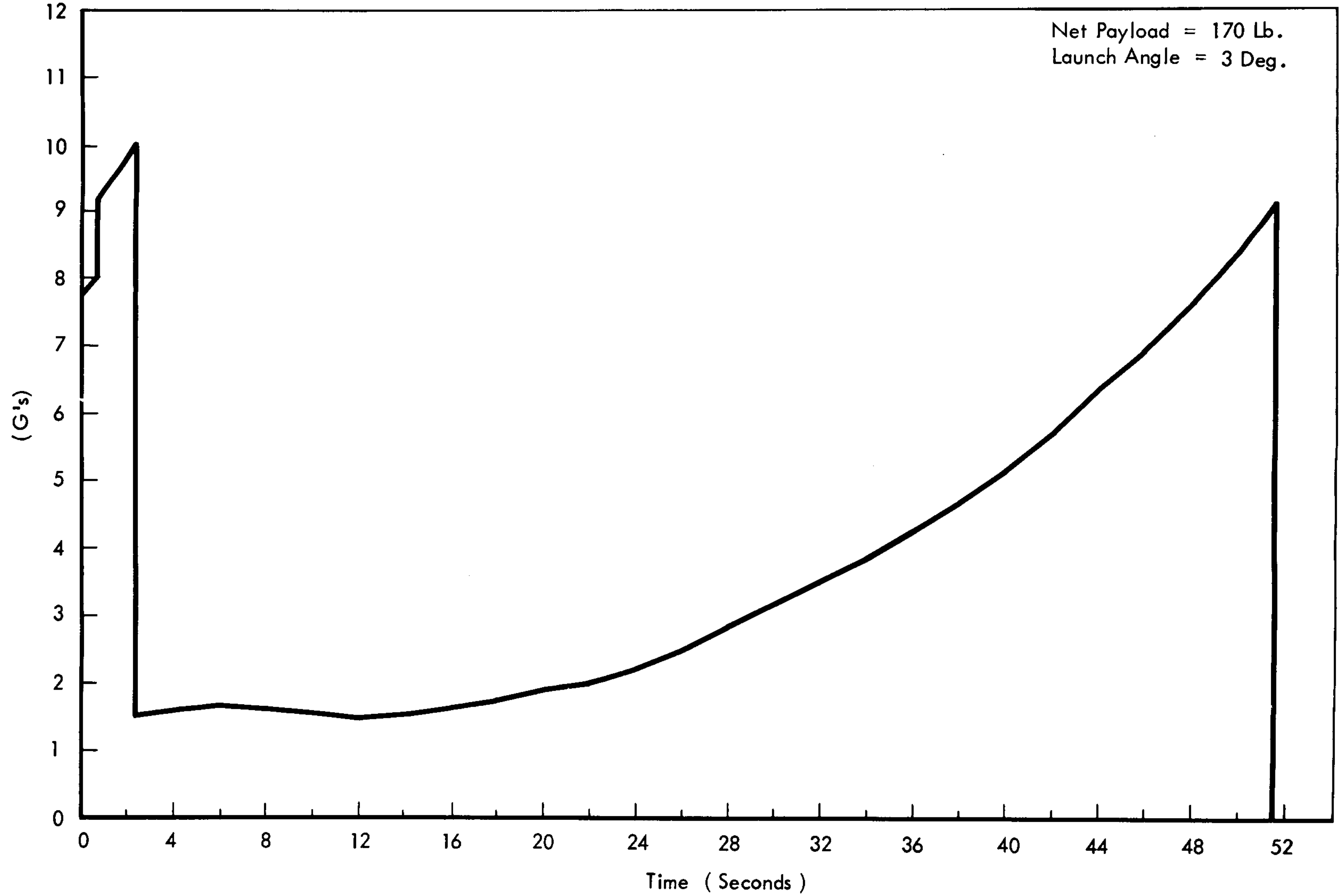


Figure 17–Aerobee 150A Acceleration Vs. Time

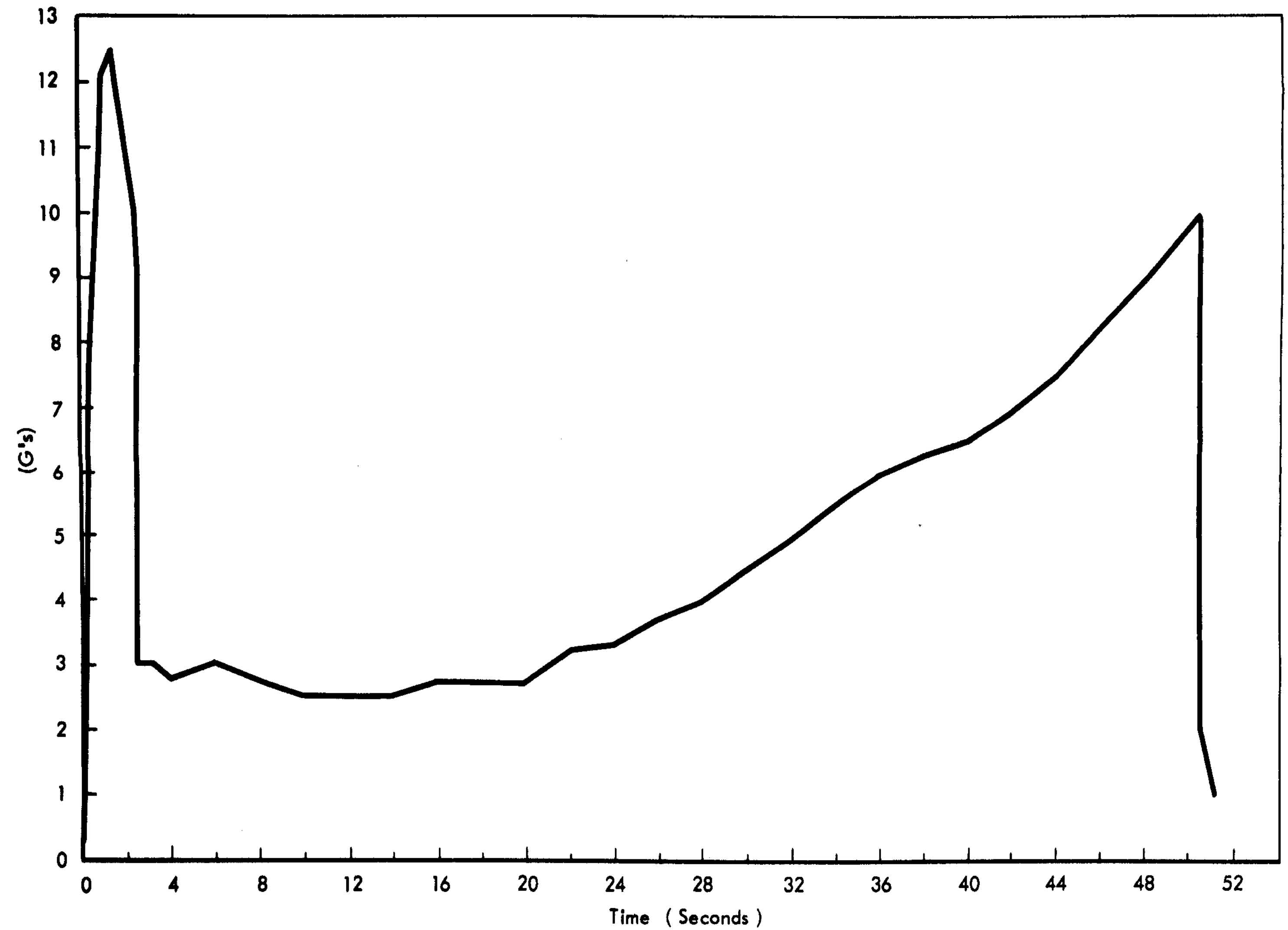


Figure 18–Aerobee 4.09 – 150A Longitudinal Accelerometer

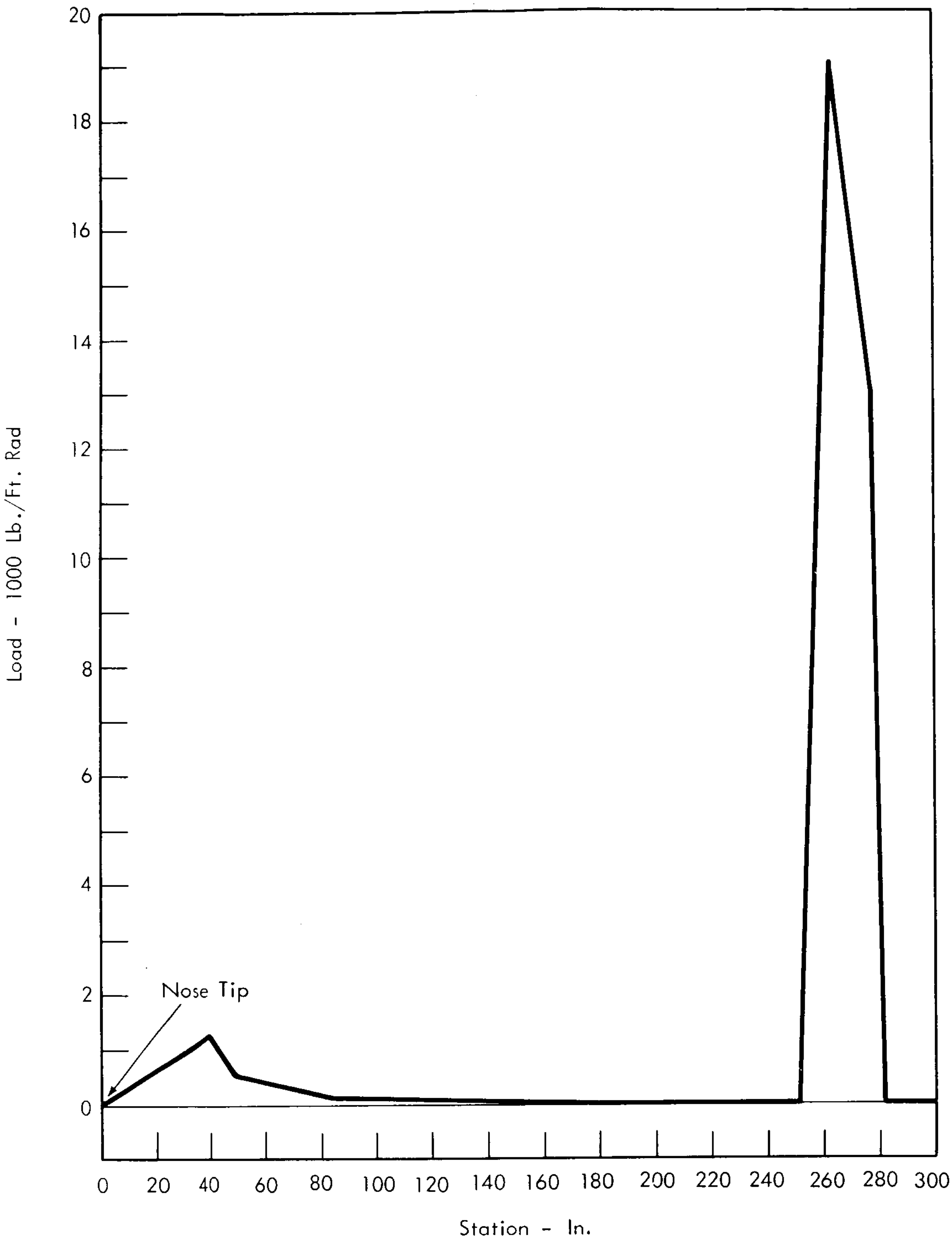


Figure 19 -- Aerobee 150A
Load Vs. Station

Net Payload = 200 Lb.
Extension Length = 0
t = 12.5 sec.

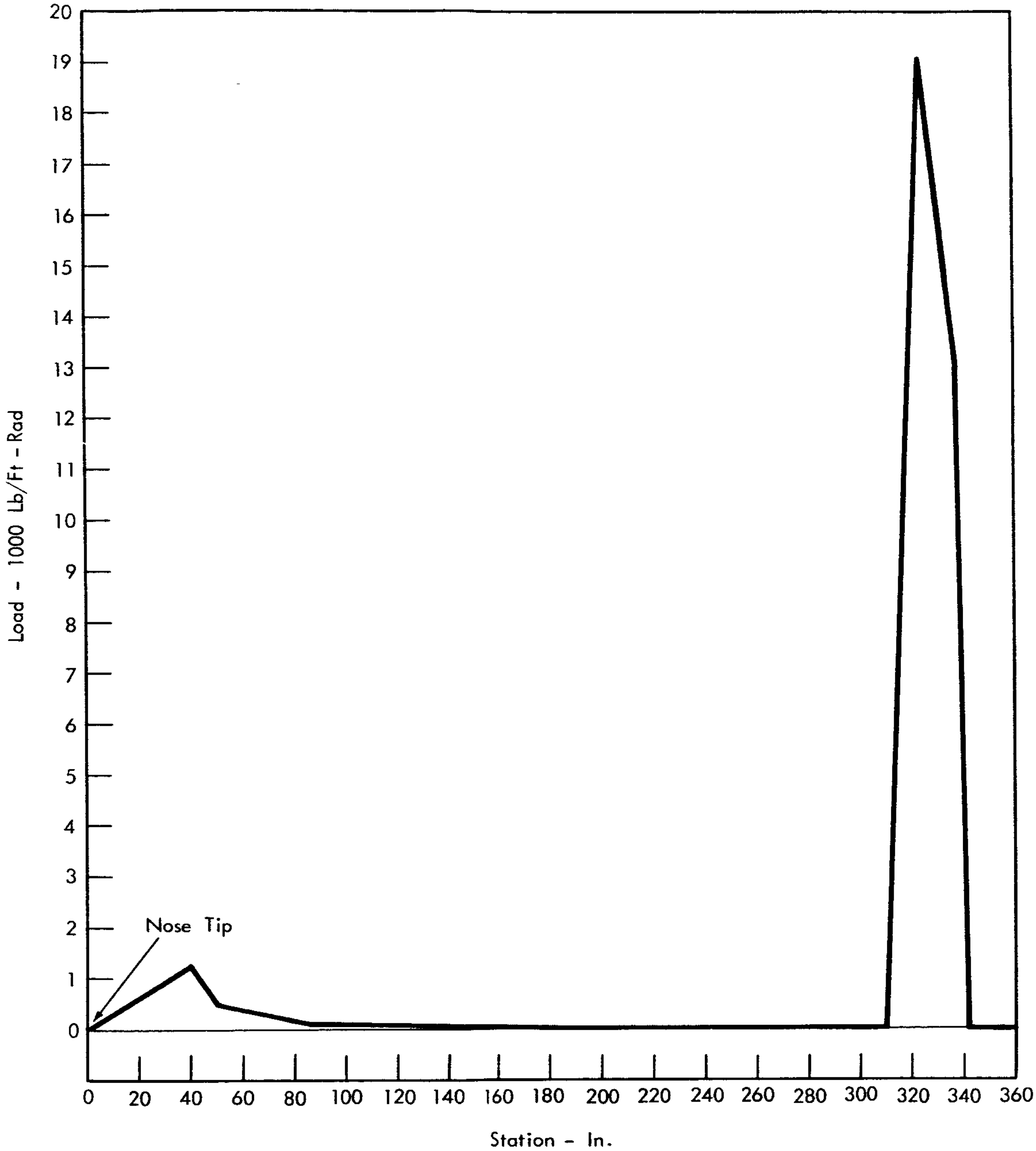


Figure 20 -- Aerobee 150A
Load Vs. Station
Net Payload = 200 Lb.
Extension Length = 60 In.
t = 12.5 Sec.

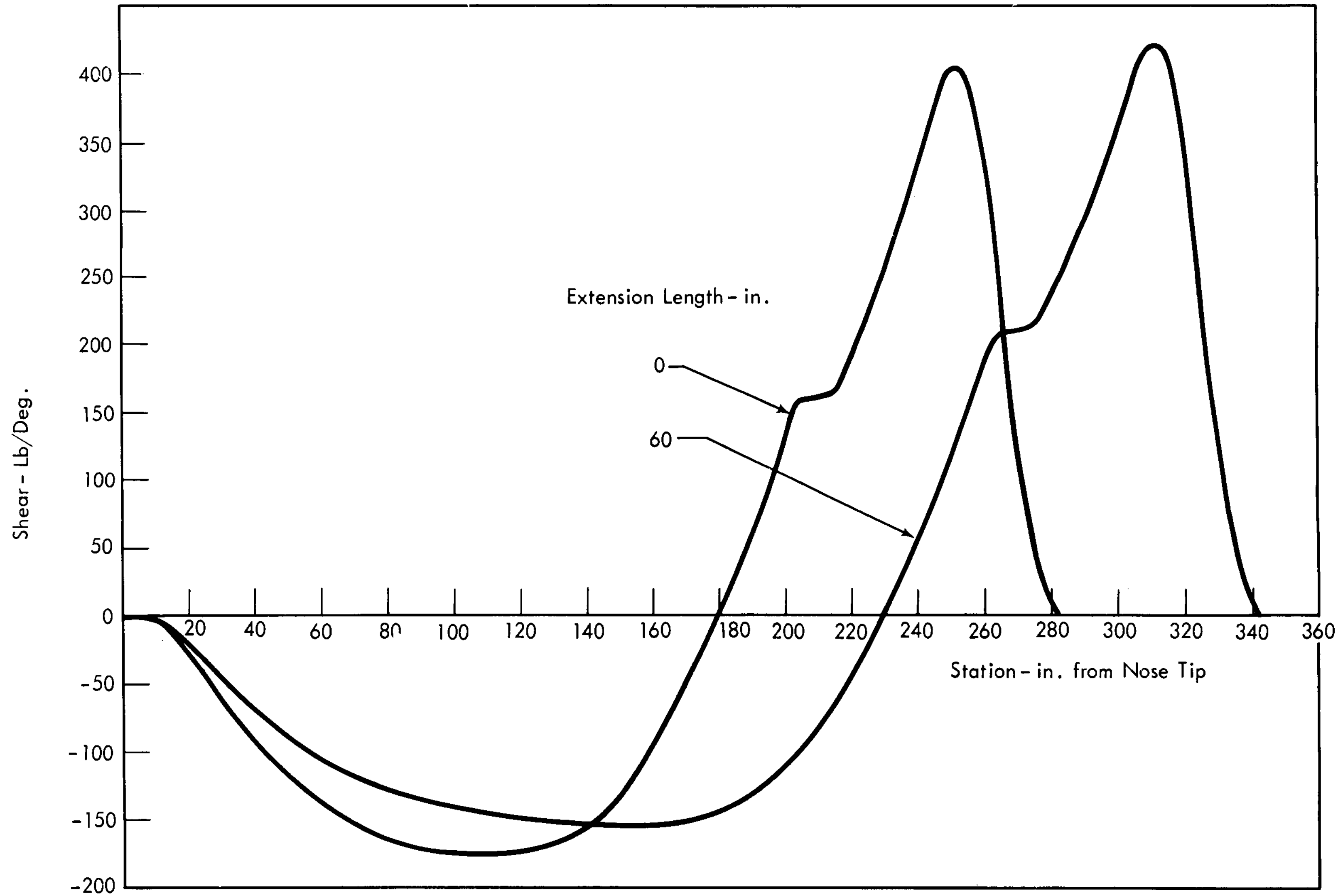


Figure 21 -- Aerobee 150A
Maximum Shear Vs. Station
Net Payload = 200 Lb.
t = 12.5 sec.

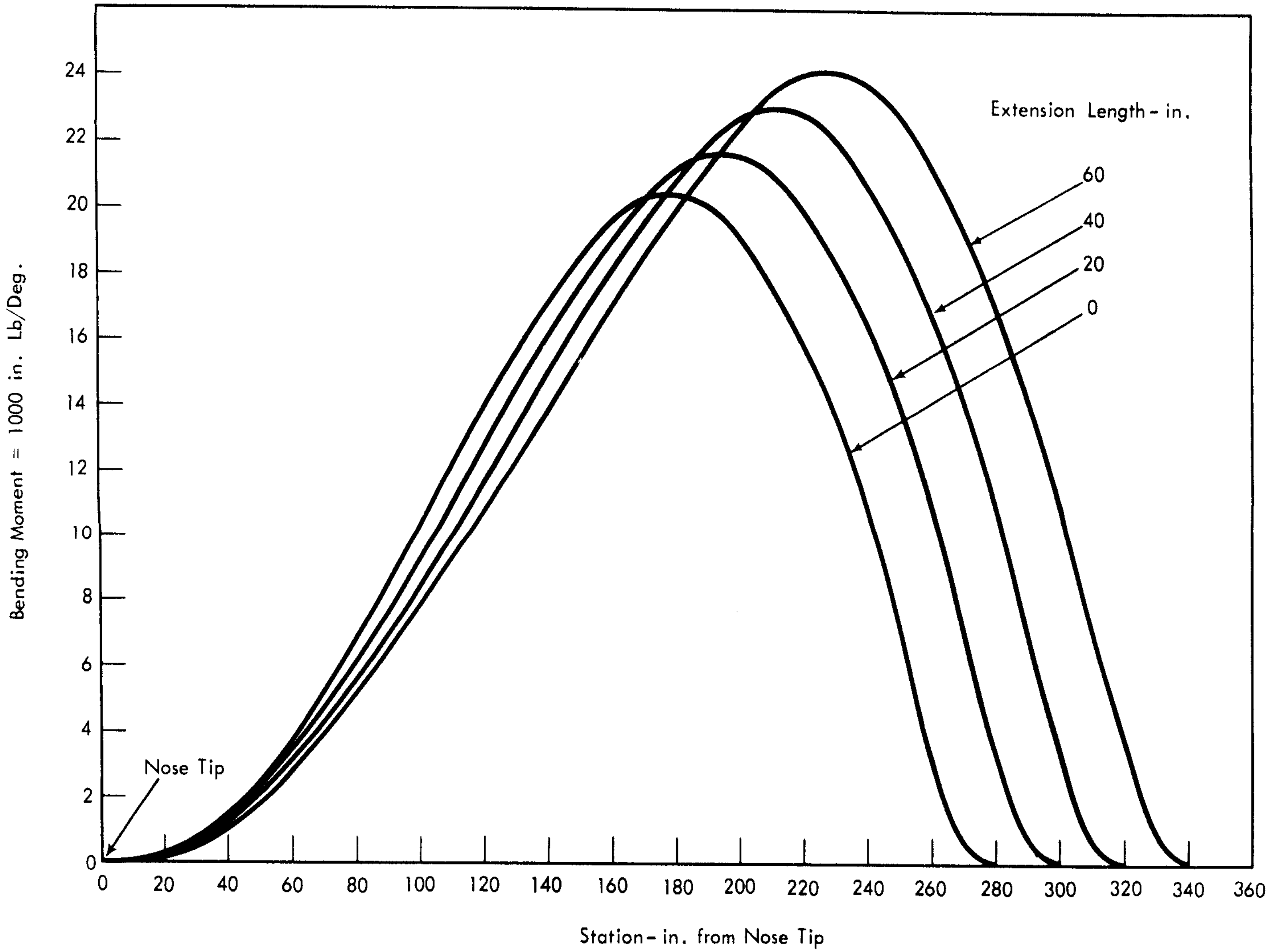


Figure 22 -- Aerobee 150A
Maximum Bending Moment Vs. Station
Net Payload = 200 Lb.
t = 12.5 sec.

DETAIL WEIGHT BREAKDOWN

SECOND STAGE

Inert Weight:	
Nose Cone	15.5
Tank Assembly	163.0
Shrouds	6.6
Fairings	.5
Aft Structure	15.5
Fins	28.0
Regulator Valve	3.0
Regulator Manifold	1.0
Thrust Chamber Assembly	(32.1)
Chamber	25.5
Nike Valve	2.9
Flex Lines	2.0
Shutoff Valves	1.7
Miscellaneous (lugs, etc.)	12.4
Helium	7.0
Unuseable Fuel	7.0
Total Inert	291.6
Consumed Weight:	
Fuel	296.2
Oxidizer	758.2
Total Loaded	1,346.0

Figure 23–Detail Weight Breakdown

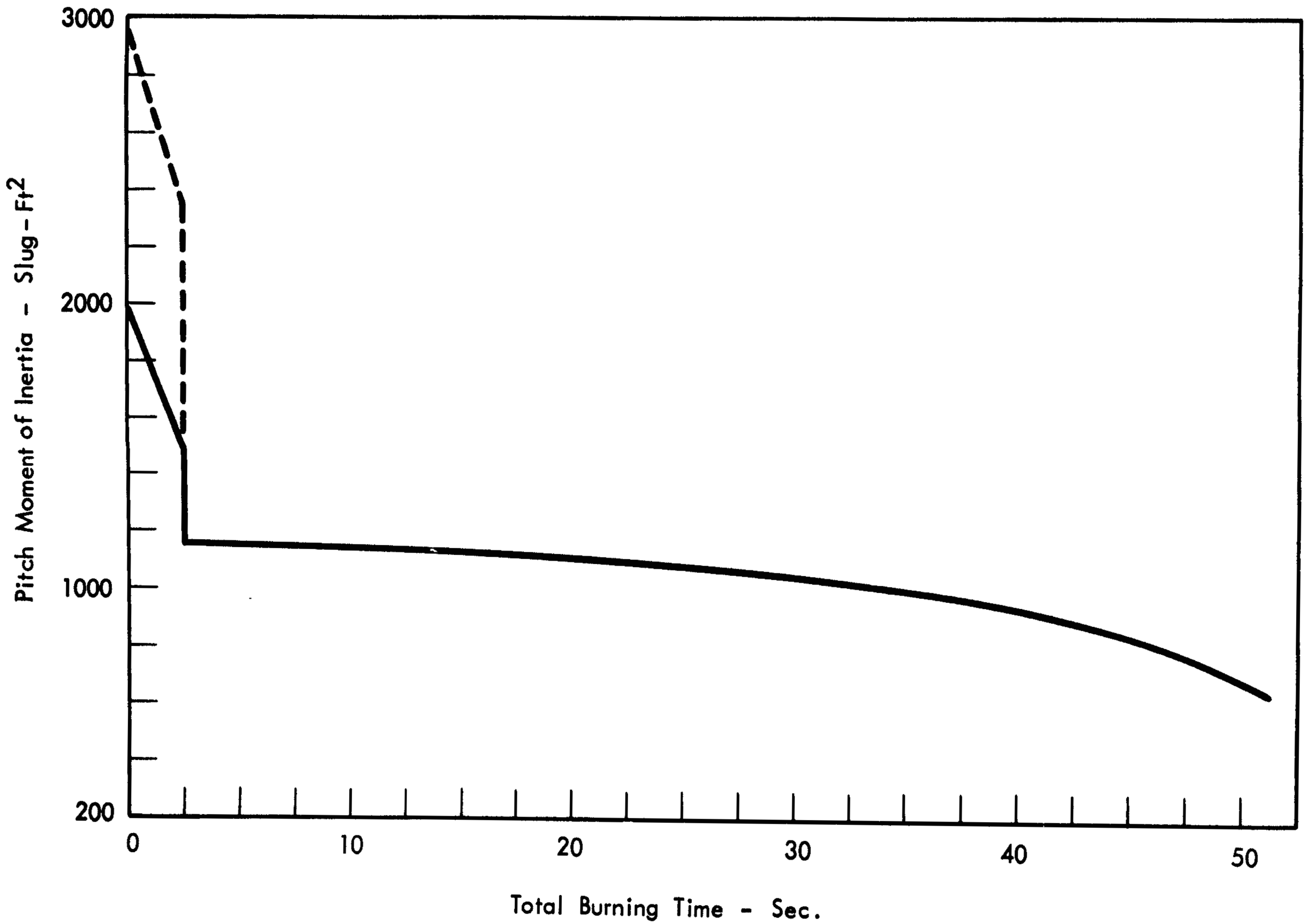


Figure 24—Aerobee 150A Moment of Inertia Vs. Time 150 Lb. Payload

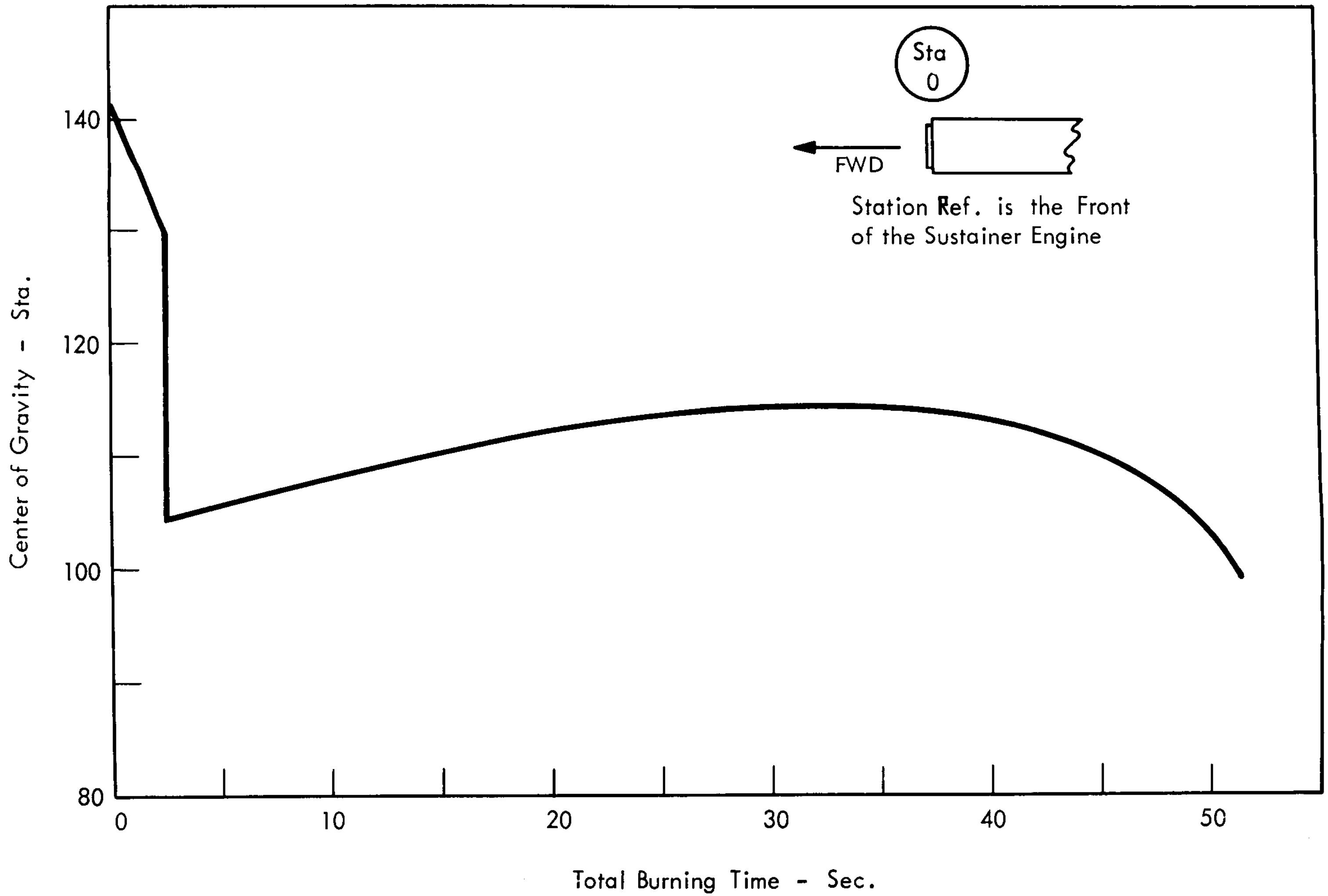


Figure 25–Aerobee 150A Center of Gravity Vs. Time

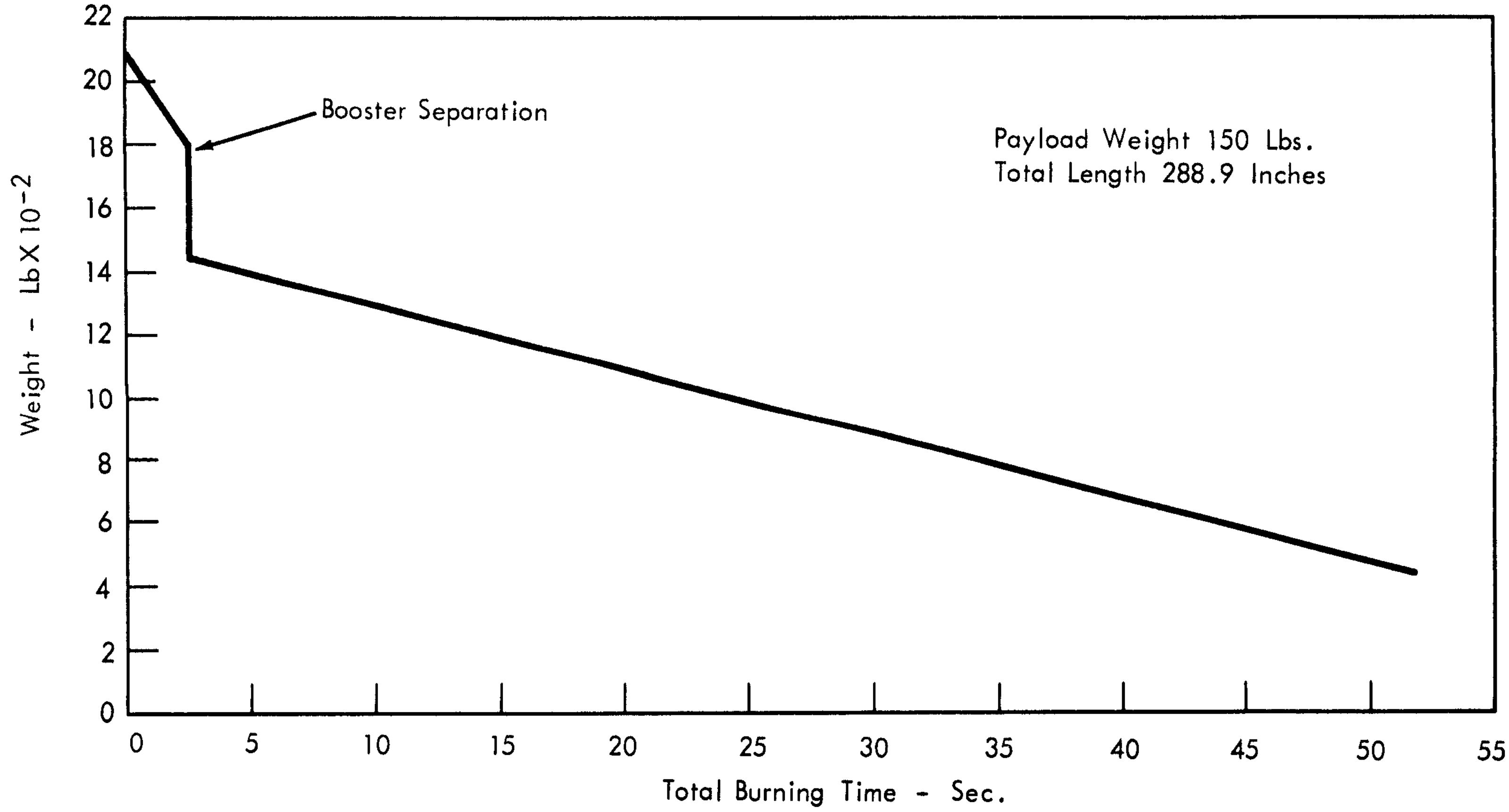


Figure 26—Aerobee 150A Weight Vs. Time

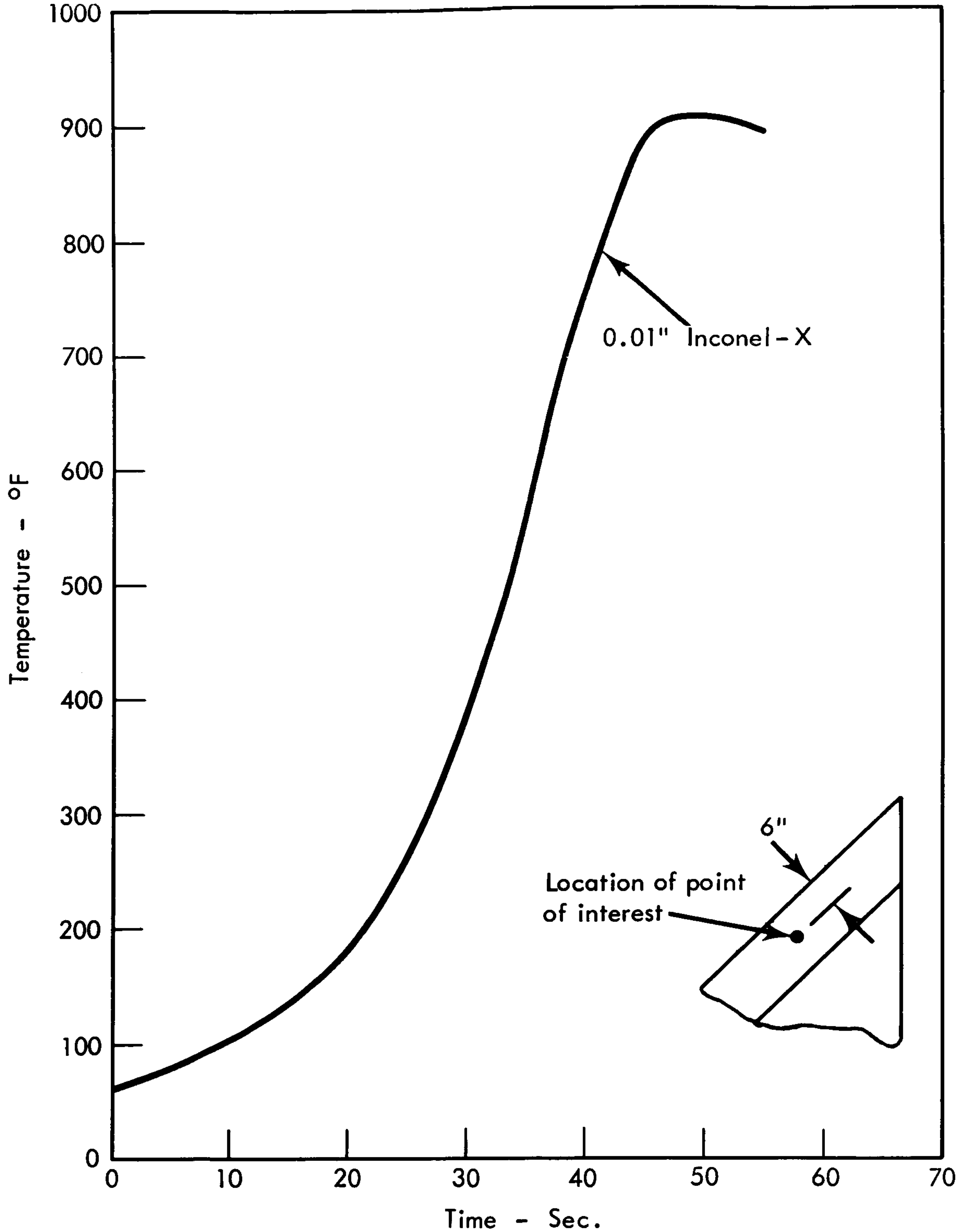


Figure 27 -- Aerobee 150A
Aerodynamic Heating of
Sustainer Cuffs (45° Sweep)

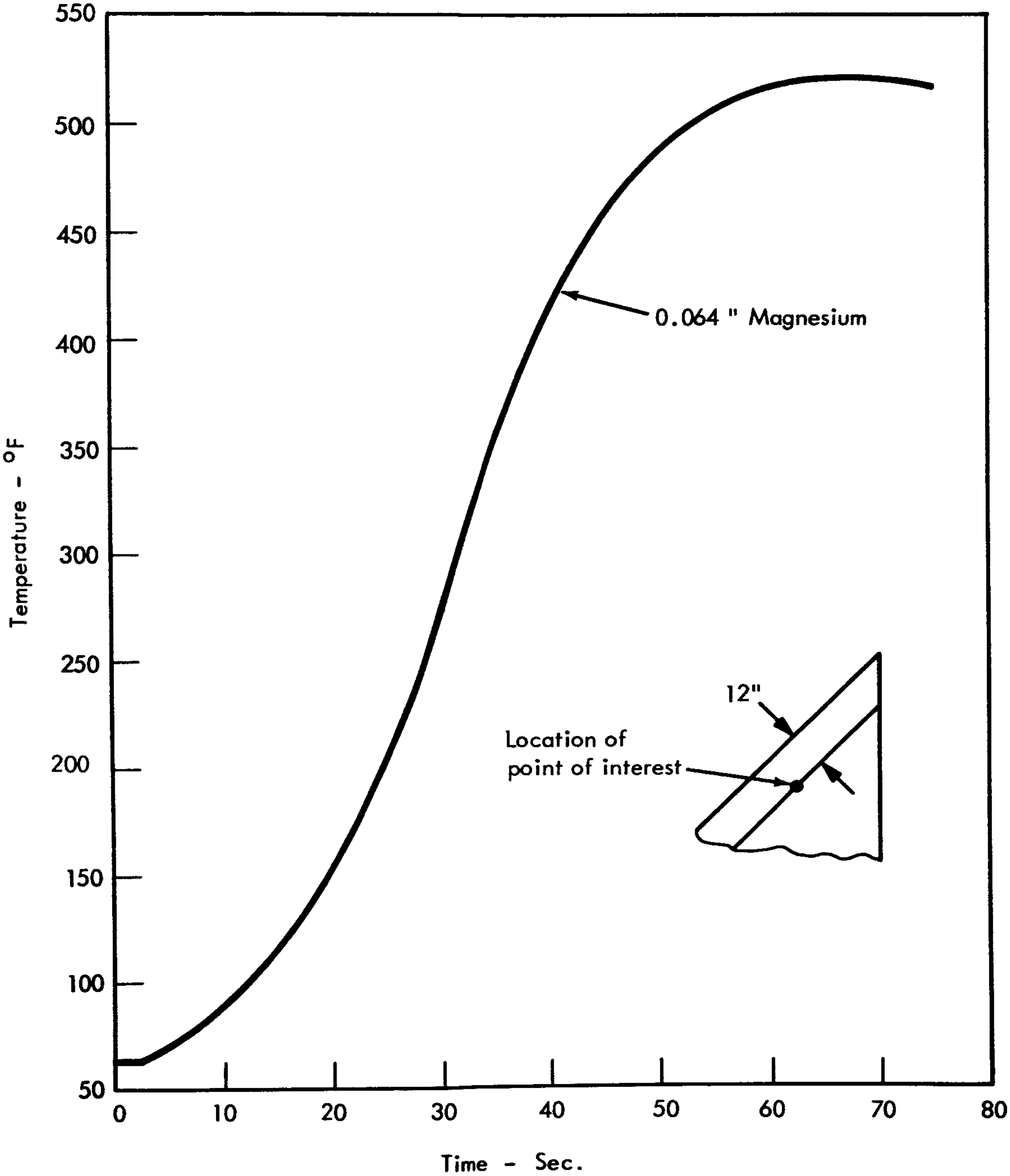


Figure 28 -- Aerobee 150A
Aerodynamic Heating of
Sustainer Fin Skin (45° Sweep)

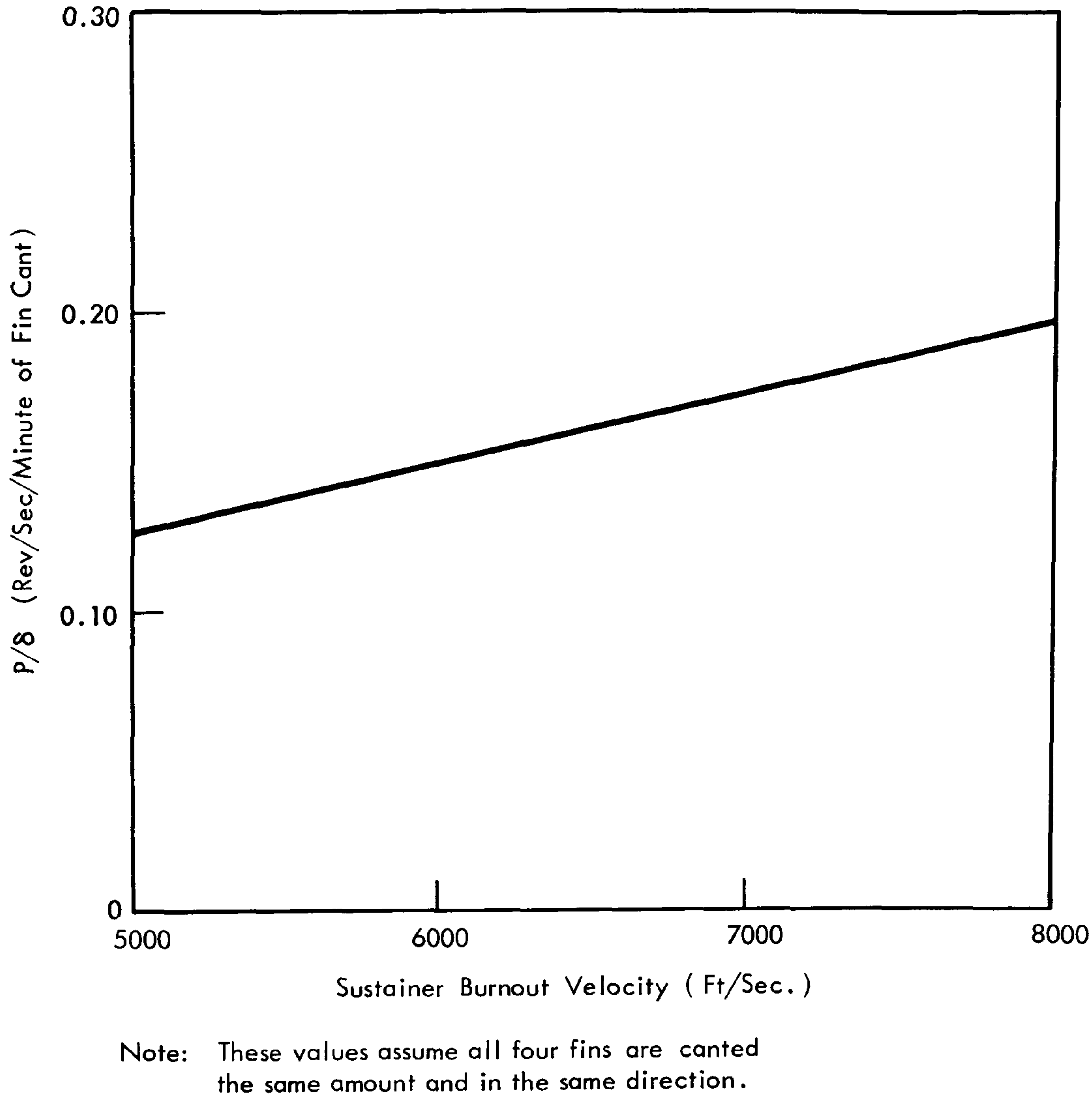


Figure 29—Aerobee 150A Sustainer Burnout Roll Rate Per Fin Cant Angle Vs. Burnout Velocity

APPENDIX B

Astrobee 1500

by

Norman E. Peterson, Jr.

VEHICLE DESCRIPTION

The Astrobee 1500 sounding rocket is a two stage, unguided, solid-propellant vehicle capable of taking a seventy-five (75) pound payload of scientific instrumentation to an altitude of approximately 1600 statute miles when launched at an angle of five (5) degrees from the vertical. The overall length of the vehicle is thirty-two (32) feet and the maximum body diameter is thirty-one (31) inches. The vehicle gross launch weight without payload is 11,500 pounds.

The first stage consists of one Aerojet Junior motor (28 KS 57,000) with a recruit booster (1.5 KS 35,000) attached to each side. The nozzles of the recruits are canted at an angle of nine (9) degrees to the vehicle centerline in order to permit their thrust vector to pass through the vehicle center of gravity. Since the vehicle is launched from a "zero length" launcher the recruit boosters are necessary to impart an initial high acceleration to the vehicle in order to minimize wind effects and to reduce impact dispersion. The recruits remain attached to the first stage motor after they have been expended, therefore the overall vehicle performance is increased only slightly.

Aerodynamic stability during first stage thrusting is provided by four (4) fins mounted at the aft end of the vehicle. The fins are of the swept, single wedge design and have a planform area of 12.5 square feet each. The fins are fixed in a canted position of one-half (1/2) degree to the vehicle centerline to produce a two (2) revolution per second roll during first stage thrusting. It is necessary to spin an unguided vehicle to minimize the effects of thrust misalignment which is one of the major contributors to the increase of impact dispersion distances.

At the forward end of the first stage motor an interstage structure is mounted. This interstage structure contains a spin table, four spin rocket motors, blowout diaphragm, sequencer programmer and pyrotechnic power supply in addition to providing a structural means of connecting the first and second stage motors together.

The second stage motor is attached to the interstage by a conical structure which is bolted to the aft flange of the second stage motor and connected to the spin table by means of a threaded blowout diaphragm. Separation of the two stages is effected when the second stage motor exhaust gases deflect the blowout diaphragm (See Figure 2).

The spin table contains four spin rocket motors (1 KS 210) which impart an incremental spin of ten (10) revolutions per second to the second stage, just prior to ignition, for dynamic stability.

The interstage structure also serves as an attachment for the clamshell nose fairing which covers the payload-second stage motor assembly to provide protection from aerodynamic loads and heating during the first stage thrusting. The nose fairings are held closed during first stage thrusting and the subsequent ten (10) second coast period by two marmon clamps. The marmon clamps are released by means of electrically actuated explosive bolts just after second stage spin up occurs. Centrifugal forces then open the clamshells on their hinges providing an opening at the front of the vehicle through which the second stage is fired.

The scientific payload package is attached to the forward end of the second stage motor and is protected from aerodynamic heatings by a conical plastic nose cone cover and two (2) thirteen (13) inch long cylindrical metal covers.

SEQUENCE OF EVENTS

The normal sequence of events occur as follows:

LAUNCH TIME – SECONDS	
T + 0.0	First stage motor is ignited by an electrical impulse from a remote source (usually located in a blockhouse)
T + 0.1	Recruit motors are ignited as soon as first motion is sensed (usually they are wired through a first motion switch mounted on the launcher). The source of the electrical impluse is the same as that used to fire the first stage.

Launch Time – Seconds (Continued)

T + 2.5	Recruit motors burnout and remain attached to the first stage.
T + 29	First stage motor thrust decay begins.
T + 35	Electronic sequencer programmer begins.
T + 40	First stage motor burns out. Vehicle begins coast period and is spinning at two (2) revolutions per second.
T + 48	Four rocket spin motors mounted on the spin table are fired, by an electrical impulse from the sequencer programmer, causing the second stage payload assembly to spin up to twelve (12) revolutions per second within the clamshell nose fairing.
T + 49.5	Four explosive bolts holding the marmon clamps are fired by an electrical impulse from the sequencer programmer permitting the clamshell nose fairing halves to open.
T + 50	Second stage motor is fired by an impulse from the sequencer programmer. The motor is released from the first stage by deflection of the blowout diaphragm caused by motor exhaust gases (See Figure 2).
T + 80	Second stage motor burns out, the payload with the expended second stage motor attached coasts under the influence of gravity and with a fixed orientation in inertial space to peak altitude.

AERODYNAMICS

Vehicle performance (See Figures 4, 5 and 6).

STABILITY

During first stage flight, aerodynamic stability is achieved by the use of fins which are canted to induce roll in order to minimize wind effects and thrust misalignment.

Dynamic stability is achieved during second stage operation by igniting the motor in the upper atmosphere where the air is less dense (dynamic pressure is at a maximum of 72 pounds per square foot). The high spin rate (12 revolutions per second) also serves to overcome any aerodynamic and dynamic forces which would tend to cause the vehicle to deviate from its normal flight path.

The use of the Ballisticians Gyroscopic Stability Factor is frequently made to determine if dynamic stability can be obtained for a spinning, statically unstable vehicle in the upper atmosphere.

$$S_G = \frac{I\left[\frac{I_R}{I}\right]^2 P^2}{4\,M\alpha}$$

where:

S_G - Gyroscopic Stability factor

I - Moment of Inertia in the pitch or yaw axis - slug - feet ^{2}
(I pitch = I yaw for a Symmetrical Vehicle)

I_R - Moment of Inertia in the roll axis slug - feet^2

P - Roll rate – Radians per second

$M\alpha$ - Aerodynamic overturning moment

By definition, when the Gyroscopic Stability factor is less than one the vehicle is dynamically unstable.

HEATING

Several methods are used to protect the vehicle and payload from heat damage.

The first stage fin bases are protected from first stage motor exhaust plume heating by the addition of a layer of ablative material.

The second stage is protected from aerodynamic heating during first stage operation by the clamshell nose fairings. The interstage components are protected from the first stage motor heat by the addition of a fiberglass insulation cover over the first stage motor dome. The payload is protected during second stage motor operation by means of a fiberglass insulation cover and a plastic nose cone cover.

PREPARATION FOR LAUNCH

The typical assembly and preparation for launch sequence occurs as follows:

1. The first stage motor is uncrated and installed on a handling trailer.
2. The tail shroud is installed over the first stage motor nozzle.
3. The four fins are installed on the tail shroud and checked for proper alignment.
4. The recruit motors are installed and canted nozzles aligned.
5. The first stage motor igniter is checked and installed in the first stage motor.
6. The interstage firing circuits are checked, the four spin motors installed, and the interstage assembly is installed on the first stage motor.
7. The first stage motor assembly and interstage assembly is mounted on the launcher.
8. The second stage motor is prepared, igniter installed and attached to interstage assembly.
9. The payload and payload covers are then installed on the second stage motor.
10. The nose fairings, marmon clamps, and explosive bolts are then installed and connected.
11. The payload is checked for proper operation.

12. The firing lines and battery charging cables installed and checked for continuity.
13. The vehicle is elevated to the nominal launch elevation angle and the launcher is set for the nominal azimuth angle.
14. The vehicle pyrotechnic batteries are then charged and tested for proper voltage.
15. The vehicle final azimuth and elevation angle adjustments are then made to the launcher to compensate for ballistic winds.
16. The vehicle is launched.

LAUNCHER

A boom type launcher is required for the Astrobee 1500 which has a capacity of 15,000 pounds. The vehicle is attached to the launcher by two sets of launch fittings. The forward set consists of two (2) tee lugs which are attached to the interstage structure and inserted in mating slots on a hanger attached to the underside of the launcher boom. The aft launch fitting consists of a pin 2.25 inches in diameter which is mounted in the aft face of the tail structure. This pin mates with a hanger mounted on the underside of the launcher boom.

The launcher is designated "zero length" as guidance is provided for only two (2) inches. (See Figure 1)

In addition to a launcher, handling trailers with a capacity of 12,000 pounds, battery charging equipment and a vehicle checkout console are required for launching the Astrobee 1500.

IMPACT PREDICTION

A computer trajectory simulation is used to determine the nominal impact point for each flight. Since this vehicle is, as are most sounding rockets, unguided, performance of the motors may vary and slight errors in aiming may cause the vehicle to impact at some distance from the predetermined nominal impact point. This impact deviation describes an area (dispersion area) around the nominal impact point where all normal flights will impact.

The aiming of the vehicle is accomplished by taking wind readings, velocity and direction versus altitude, at regular intervals for several hours before launch time. The wind measurements are then used to calculate the wind effects on the vehicle and the launcher is adjusted to compensate for them. Vehicles have been aimed more than thirty (30) degrees in azimuth from the pre-planned azimuth direction to compensate for the wind effect and have successfully corrected back to the nominal azimuth.

ANCILLARY EQUIPMENT

A recovery system could be made available for use on the Astrobee 1500 if required.

A spin control system of reducing the payload spin from the nominal twelve (12) revolutions per second down to any desired rate by the use of a yo-yo despin device is available.

GROUND SAFETY

The Astrobee 1500 vehicle imposes no more rigid ground safety requirements than any similar solid propellant rocket vehicle system.

APPENDIX B

LIST OF ILLUSTRATIONS

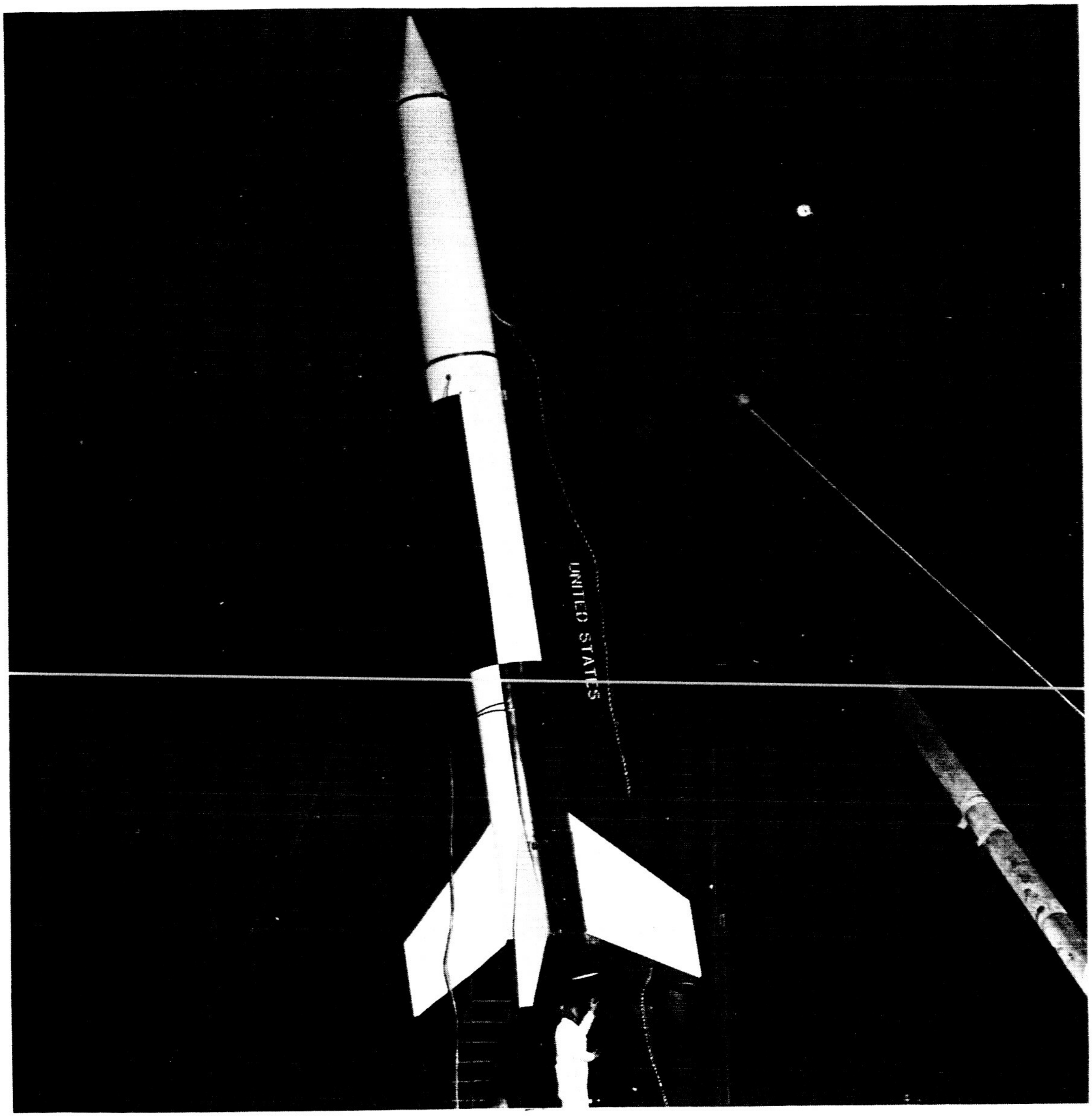


Figure 1—Astrobee 1500 on Launcher

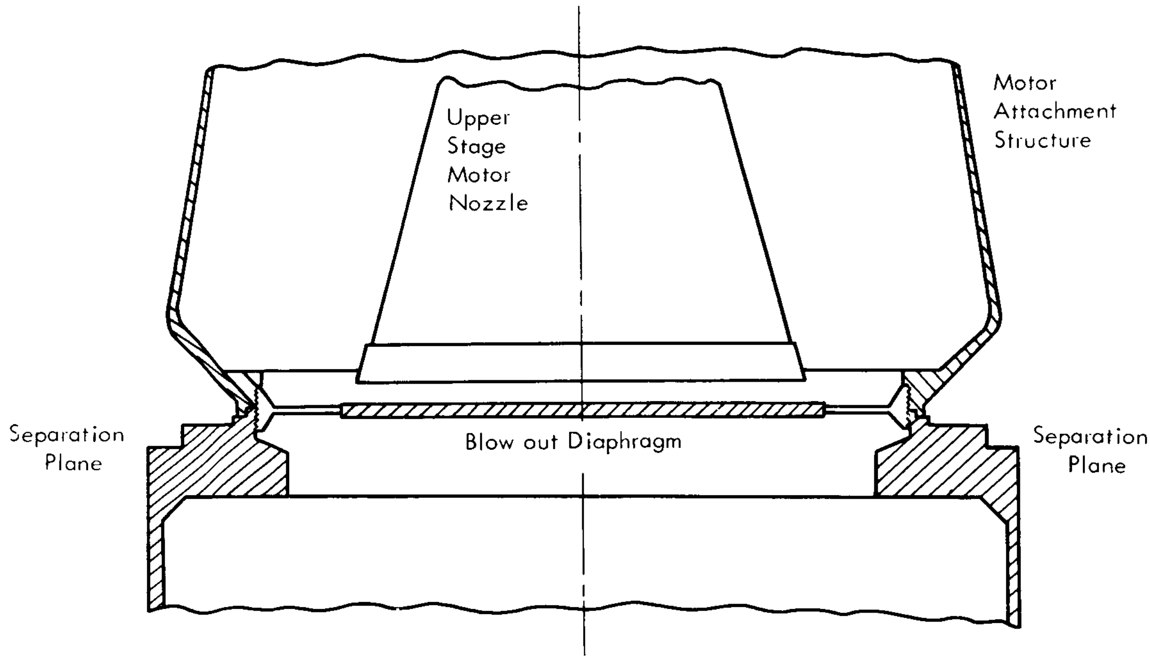


Figure 2a -- Blow Out Diaphragm
Used for Stage Attachment

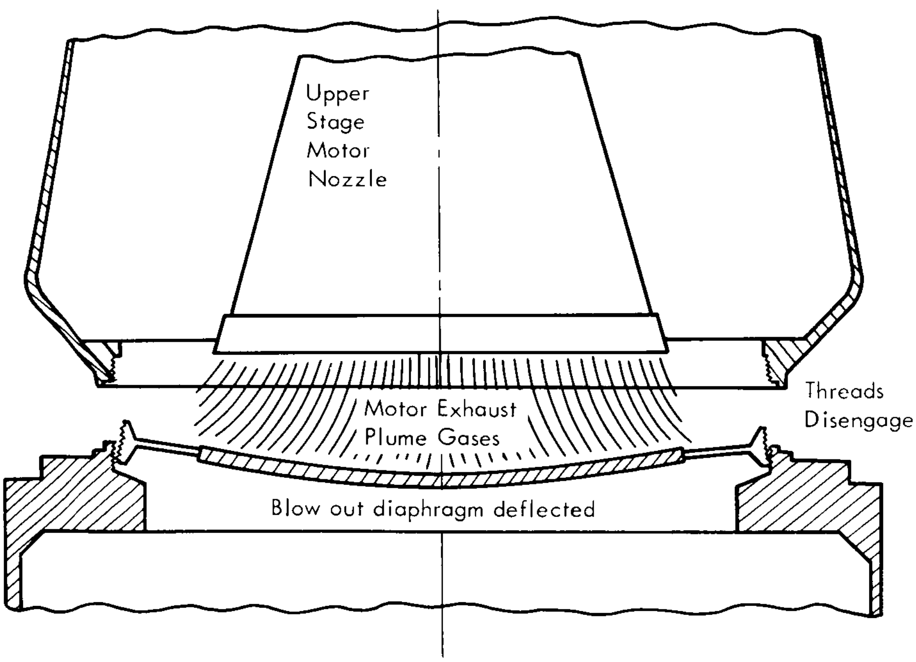


Figure 2–Blow out Diaphragm Operation

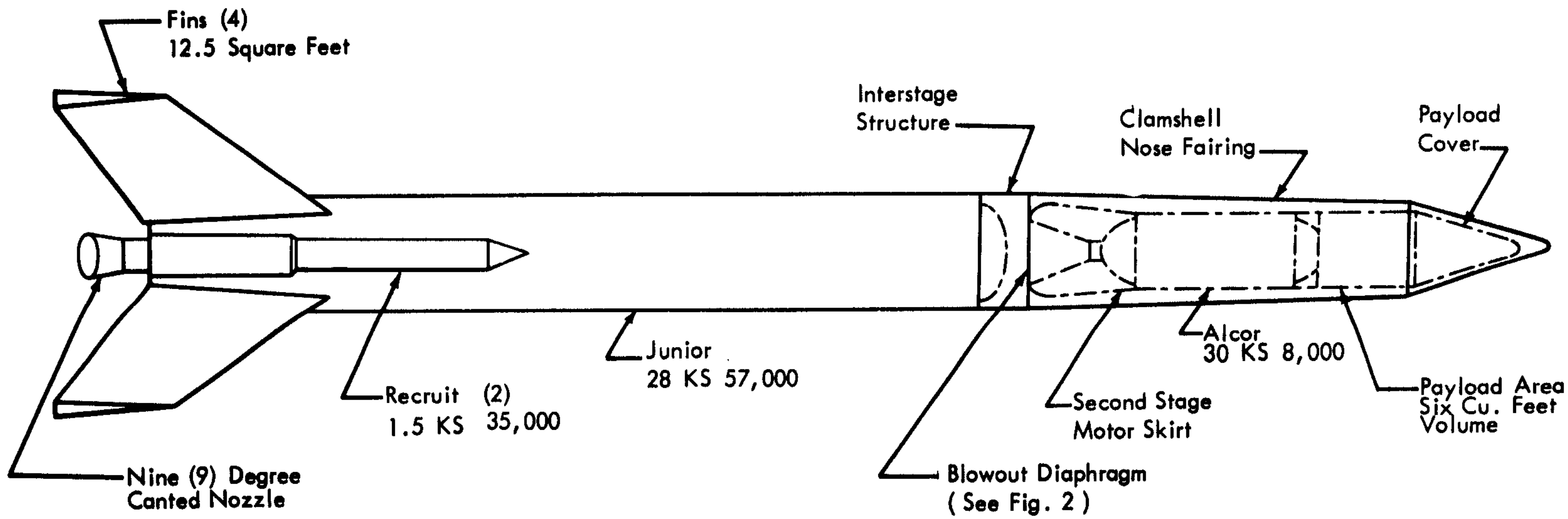


Figure 3—Astrobee 1500 Outline Drawing

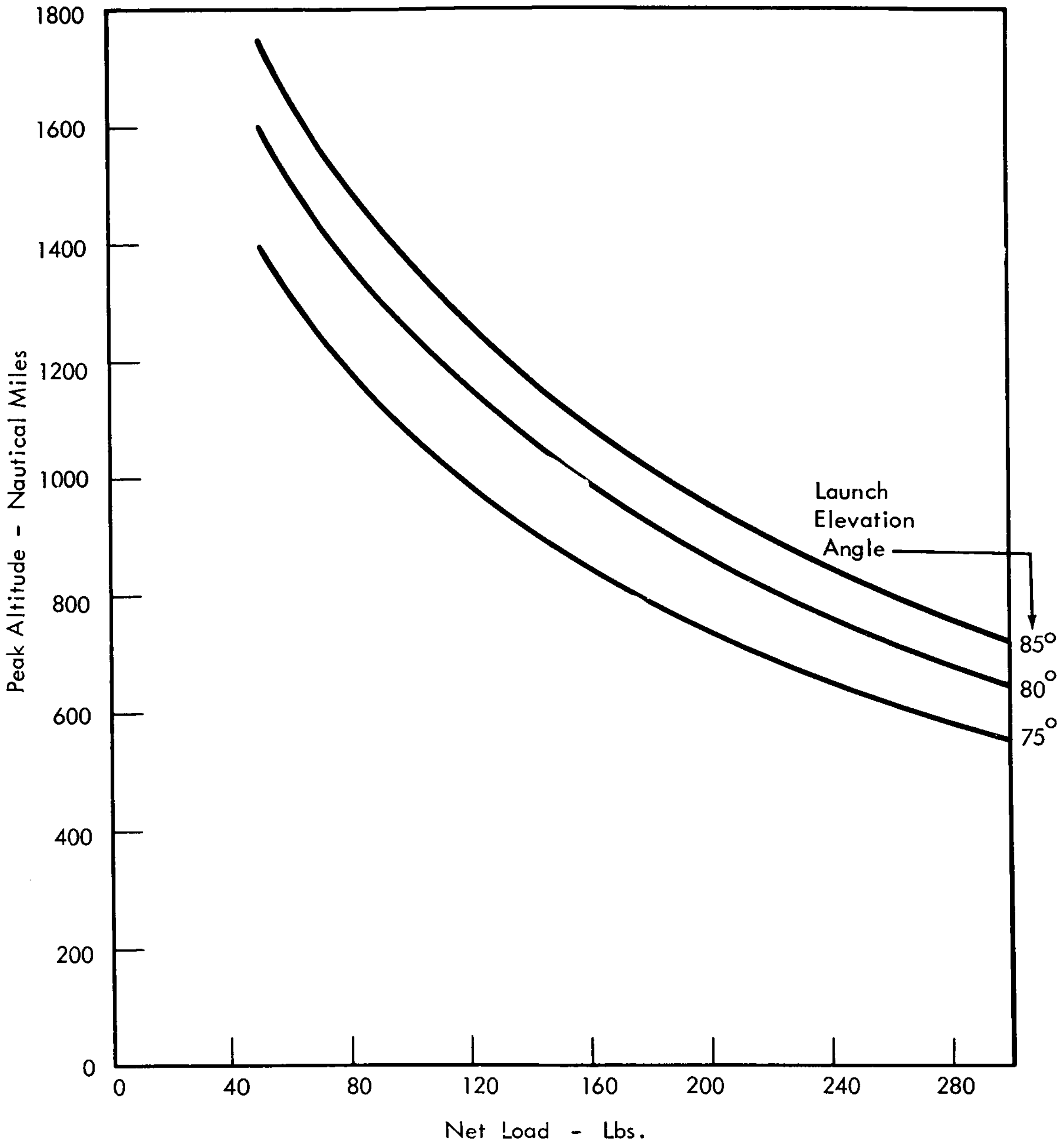


Figure 4 -- Astrobee 1500
Peak Altitude Vs. Net Payload
For Various Launch Angles

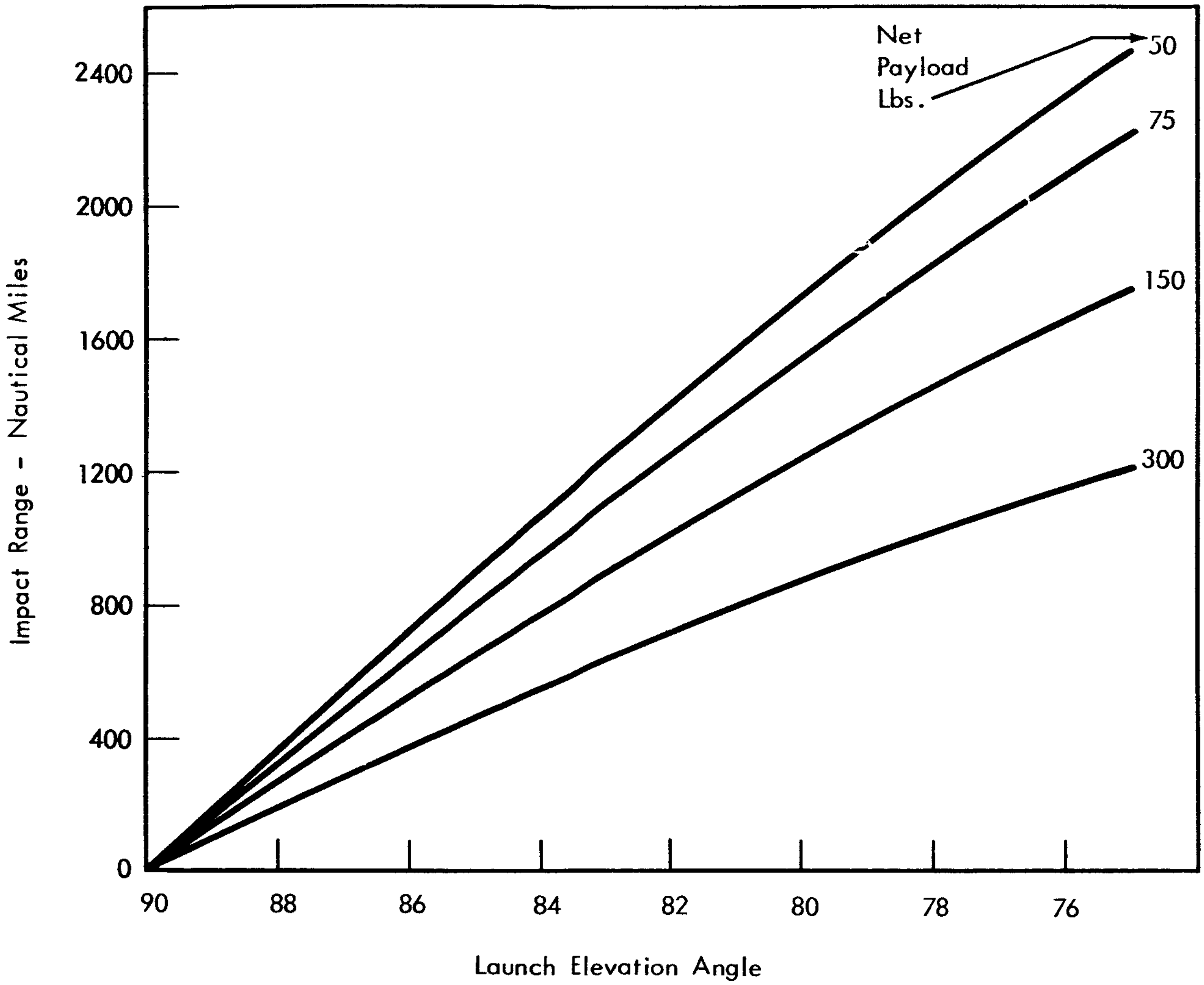


Figure 5—Astrobee 1500 Impact Range Vs. Launch Angle for Various Payload Weights

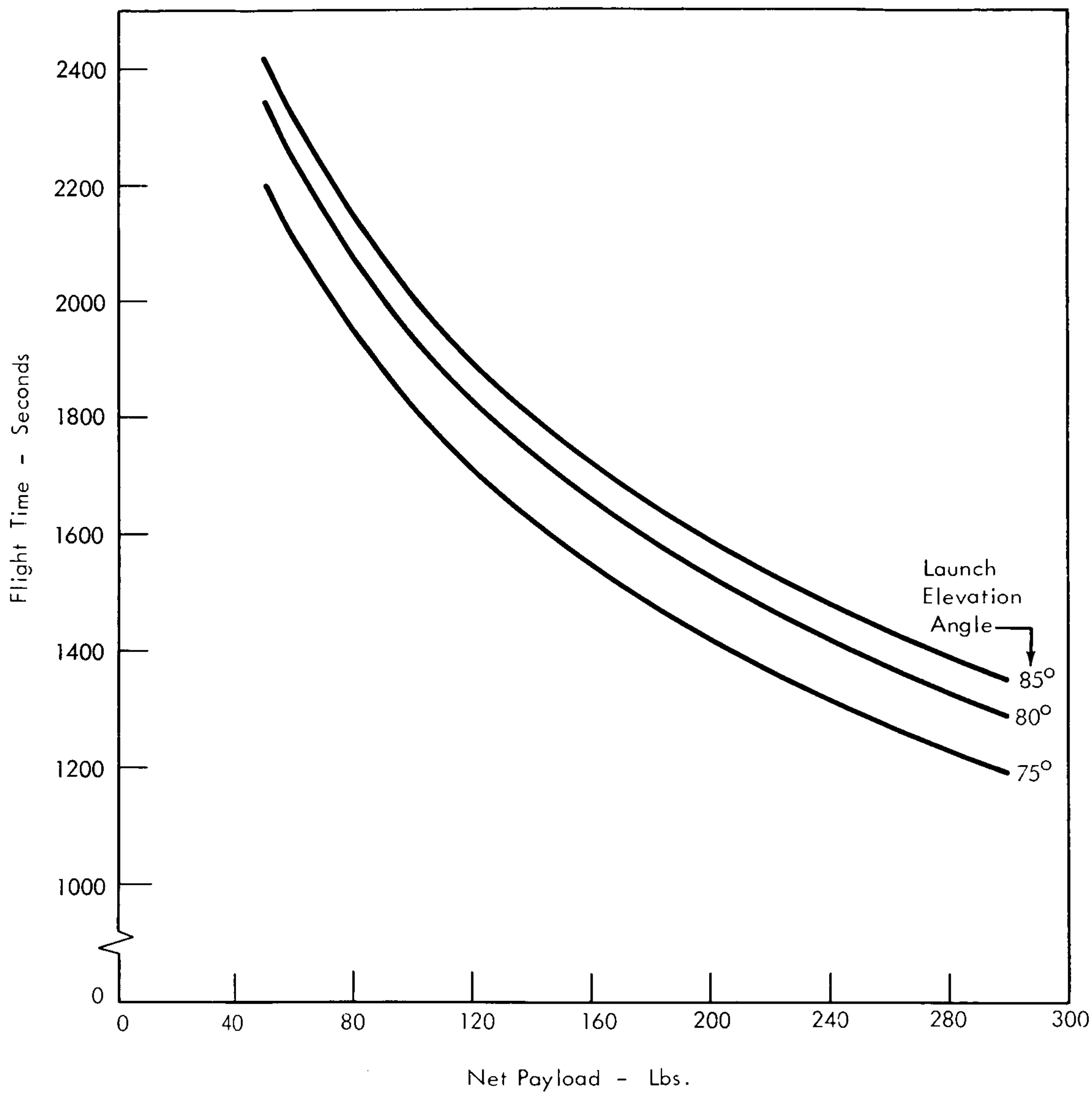


Figure 6—Astrobee 1500 Flight Time Vs. Net Payload

APPENDIX C

Javelin

by

John G. Guidotti

DESIGN DESCRIPTION

The Javelin sounding rocket, Figure 1, is a four stage solid fuel research rocket. The first three motors are standard military units (Honest John and two Nike Ajax boosters), the fourth is an X-248 Altair rocket motor originally developed for the Vanguard launch vehicle. The rocket system is built around these motors. Static stability is provided during flight of the first three stages by cruciform fins mounted on the aft end of each stage. The fourth stage is spin stabilized and has no fins.

Nominal event time history for the Javelin is as follows: The first stage burns out at T = 5.0 seconds. At burnout the first stage Honest John drag separates. The second – third – fourth stage configuration coasts until 9.7 seconds at which time the second stage Nike ignites. The second stage burns until T=13 seconds and is drag separated. The remaining stages coast until T = 25 seconds when the third stage Nike ignites. The third stage burns out at T = 28.3 seconds and remains attached to the fourth stage during a coasting period which lasts until fourth stage ignition at approximately 56 seconds. The third stage is blast separated through the action of a blowout diaphragm. Burning time for the fourth stage is 41 seconds. At fourth stage burnout the velocity is 12,600 feet per second and the altitude is 420,000 feet, based on a gross payload weight of 120 pounds launched at 80° elevation. The cylindrical heat shield around the fourth stage and support tube is released two seconds after ignition and the nose cone is ejected at T = 120 seconds nominally.

The first stage is built around a stripped down Honest John MA 6 motor. This stage includes the standard Honest John military fins on the aft end. The aft end is also supplemented by a rear launch fitting which is bolted to the rear of the MA 6 rocket motor case. The forward end of the Honest John motor is fitted with the largest of three magnesium adapters and a "T" launch fitting. The adapter, which is somewhat bell shaped, is bolted to the forward thrust flange of the rocket motor and is designed to fit snugly into the 2nd stage Nike nozzle. The tee launch fitting is bolted in place using six bolts.

The second stage Nike is fitted out on the aft end with four cast magnesium quadrants. These quadrants are precisely machined to fit the Nike nozzle and are retained in this position by a heavy shroud which is bolted to the Nike body and the four quadrants. The shroud gives continuity to the streamline of the vehicle as well as providing the structural attachment necessary to maintain both the longitudinal and radial position of the quadrants. The fins are attached one each to the four quadrants. The fins are 2-3/4 square foot fins with an inconel leading edge cuff for heat protection. These fins are accurately positioned by a cam to a predetermined incidence angle. The fins are held in place by 10 bolts which are torqued to 16-1/2 ft. lbs. to insure that the incidence angle is maintained. In addition to the bolts two dowel pins are installed for positive assurance against fin movement. The pinning operation is accomplished in the field and the equipment used for this operation consists of a 1/2" capacity air driven hand drill. The next step is to attach a launch tee to the forward end of the motor. The launch tee is attached via three large bolts in pre-drilled and tapped holes. The final operation is the bolting in place over the forward thrust flange of the second stage Nike rocket motor of the second of three adapters. This adapter slips into the nozzle of the third stage Nike which has been pre-drilled for an unlock mechanism which has been loosely installed in the second stage adapter. The second stage adapter also contains two large pins at 180° which slip into grooves of the third stage nozzle to prevent rotation of the stages.

The third stage Nike motor aft end is prepared similar to the second stage aft end except in this case two square foot fins are bolted and pinned in place. The third stage fins are thermally protected by inconel and fiberglass cuffs along the leading edge and along the flat plate fin surface. The forward end of the third stage Nike rocket motor is equipped with the last of the magnesium adapters, this adapter is a truncated cone shaped casting which allows for the transition between the 16.5" Nike rocket motor and the 19" X-248, 4th stage motor. In this adapter the necessary power pack and timing units for both 3rd and 4th stage ignition are installed. This adapter also houses the arming bars and the charging plug for battery charging. Finally the forward end of this adapter is threaded for the blowout diaphragm which holds the 4th stage motor and payload.

It must be pointed out at this time that all interstage adapters are provided with shear lips, these are accurately machined diameters used to retain stage to stage alignment. The mating surfaces of the forward and aft portions of the adapters are held parallel to the thrust axis within .005 T.I.R. at maximum radius.

The fourth stage of the Argo D-4 contains the most complex of all the structure discussed thus far. The X-248 rocket motor is fitted out on the aft attachment flange with a support tube release ring, a precise and complex machined part. The function served by the release ring is that of retaining the support tube sufficiently rigid so that the entire weight of the 4th stage rocket motor and payload can be cantilevered from this aluminum fabricated, rib reinforced tube. The support tube is held in position with 24 spring loaded segments, which are forcefully contained in pockets by tension bands. The segments are installed so that they wedge between flanges, the flanges are alternately on the support tube fingers and the support tube retaining ring, the latter has been bolted permanently to the X-248 rocket motor. There are guide bars installed to the X-248 nozzle and the support tube retaining ring for the purpose of assuring that the support tube will slide off without cocking in the necked down portion of the nozzle, these bars are permanently installed and clear the inside diameter of the support tube. The support tube surface is used for mounting the two pin pushers (Pyrotechnic devices) used to release the support tube, and the heat shield, retaining segments, battery brackets and delay squibs which will be discussed in the operational sequence section of this discussion.

Having a good grasp thus far we are now ready to install the fourth stage to the third stage and lest we forget the support tube contains an aft flange which is threaded to accept the blowout diaphragm which has been mentioned in connection with the third stage forward flange adapter. The diaphragm is installed in the third stage adapter and the fourth stage is now ready to be threaded to the protruding portion of this diaphragm. Mating in this instance is not via bolts, etc. but rather this blowout diaphragm. The alignment with the rest of the vehicle is achieved by means of accurately machined diameters, a shear lip rather than the screw threads provide the stage alignment. Upon completing this assembly, the two stages are pinned to prevent rotation. The pins are located along the thrust axis from the support tube into the third to fourth stage adapter thus preventing stage separation, which would occur if either stage threading itself off the diaphragm. The fourth stage is protected during flight by a cylindrical fiberglass heat shield. The heat shield is now slipped over the fourth stage motor and onto the 3/8" shoulder of the third to fourth stage adapter. While in this position the heat shield retaining blocks are center punched, drilled and tapped for ultimate screw attachment, if the extension tube has been installed the location for the umbilical is also marked so that an access hole for the umbilical cable and connector can be cut into the heat shield. The process of fitting the heat shield is time consuming and not one of the more precise operations. With the heat shield properly marked we now slip it off and prepare it for final installation.

In preparing the forward thrust face of the X-248 rocket motor, the extension tube has become standard on the Javelin. This tube, which is 7-1/2 inches long, contains a shear lip and 24 holes on its aft attaching flange. These holes are used to secure the extension tube to the X-248 rocket motor. The extension tube houses the nose cone ejection timers and "G" switches, the power pack for actuating the nose release pin puller (pyrotechnic mechanism) and the mounting surface for the payload umbilical connector. On the forward ring of the extension tube there is provision for installing the scientific payload and the nose cone lock and payload despin (yo-yo) mechanism.

Assuming the payload has been installed and that the heat shield has been fitted we are now ready to install the nose cone lock and despin mechanism. This is a precision mechanism which contains 23 heavy duty springs. These springs provide the force necessary to rotate the lock ring when the pin puller is retracted, exposing the lock fingers on the nose cone. This action enables the nose cone to be ejected forward of the fourth stage under the forward thrust of the 300 lb. nose cone ejection spring.

Installing the nose cone presupposes that the lock ring has been cocked, the pin puller positioned to prevent rotation and the retaining fingers of the nose cone properly positioned and keyed in this position under complementing fingers on the lock ring (the latter fingers move to allow nose cone ejection).

The nose cone ejection spring is installed thru the opening provided in the nose tip. A tripod tool compresses the 300 lb. spring 1-1/2 inches in order to develop the full spring force. This installation supposes that the payload rack has been provided with a surface against which the spring can be bottomed. The compressed spring is retained in this position by three flathead screws which thread into the nose cone. The final step is to install an inconel nose tip to round off the forward end of the nose cone. This is accomplished by the use of a standard 1/4 20 socket head bolt which is secured by a 1/8 NPT set screw which threads down over the 1/4" bolt to retain the former from coming loose. (See Figures 2 and 3).

LAUNCHER INSTALLATION

The Javelin is launched from several different types of launchers and the usual launch site is Wallops Island, Virginia. The launchers used are descriptively labeled tubular and "I" beam launchers. There is a third launcher which has been installed at Fort Churchill, Canada which is used for launching the Javelin. This launcher is known as a General Purpose Rocket Launcher which was designed and developed by the Aerolab Development Company.

The type of launch is termed zero length, actually the vehicle travels 2" prior to disengagement from the launcher.

The first two stages are fitted to the launcher. The first stage which is on an airlog is positioned with respect to the launcher and then eased into the launcher fittings. The second stage is positioned and installed similar to the first stage and once installed on the launcher a large clamp is used to retain the forward launch tee to the launcher fitting. This clamp is later removed. The pre-assembled third and fourth stages are the last to be installed. These two stages require that the lock mechanism be positioned thru the third stage nozzle to assure that premature separation does not occur. (See Figure 4).

OPERATIONAL SEQUENCE

The Javelin is now on the launcher and the ground circuits are prepared, the first two stages being ground fired. A two wire system fires the first stage on closing the power source which is a D.C. supply. The second stage is fired via a three wire system thru the same ignition system, however a "maypole" is used here. This system requires that the vehicle move a very short distance in order to fire the second stage. This positive movement results in closing the circuit to a 9.7 second delay squib which is in the second stage igniter and, simultaneously, the circuit to a 12 second delay squib which actuates the second to third stage unlatch mechanism. The third and fourth stages are fired from similar airborne power supplies. Therefore the firing circuit for these two stages will be discussed together. The airborne power supply for this circuit is six Yardney HR-1 cells in series which are rated at 1.5 volts per cell. The common lead divides the power supply so that 4.5 volts nominal is supplied for each side of the circuit; "G" actuated timer switches and paralled "G" switches which are actuated when the first stage is fired are located in each of the outside legs of the three wire system. The timers which are spring driven clock mechanisms are set to close and ignite the third stage 25 seconds after launch and to close and ignite the fourth stage 56 seconds after launch. The fourth stage ignition time is adjusted, when necessary to obtain the desired gyroscopic stability factor of 3.0. The fourth stage firing circuit also includes in parallel two delay squib switches which close the circuits to two pin pushers. The power for these pin pushers is provided by two independent power supplies, consisting of 2 Yardney HR 05 silvercell batteries each which are attached to the support tube. Two seconds after fourth stage ignition the pin pusher circuit is closed igniting the pin pushers which release the support tube and heat shield from the fourth stage motor. The fourth stage rocket motor, payload and nose cone fly out of this heat shield.

The nose cone ejection circuit embodies the same redundant scheme of "G" switches and "G" actuated timers and is housed in the 7-1/2" long extension tube.

The circuit is the same as the third stage ignition circuit, the difference being that at approximately 120 seconds a pin puller is fired, releasing the nose cone lock ring. When the lock ring releases the nose cone, the 300 pound ejection spring kicks the nose cone forward at a velocity of 7 ft. per second faster than the velocity of the burned out fourth stage.

AERODYNAMIC PERFORMANCE

The Javelin vehicle has been flown by NASA-GSFC on 18 different missions, 17 of the missions were successful. The one unseccessful launch was attributed to a ground handling error which results in no fire of the fourth stage X-248 rocket motor.

Many changes have been made since the first vehicles were launched. The changes included the hardware, the method of computing launcher corrections for wind, and fourth stage stability.

The present fourth stage configuration which has been standarized since 1962, consists of a fiberglass nose cone and 7-1/2" extension tube, forward of the X-248 rocket motor. This configuration has been launched repeatedly with a great deal of success. It occasionally tips off but the actual tipoff angle if measured from the launch pad has not exceeded 6°. There is a great deal of evidence that during the last 20 seconds of fourth stage thrusting all coning and other flight anomolies such as vehicle precession and nutation are significantly dampened, in many cases there is evidence of their complete disappearance. Future attempts will be made to determine with greater accuracy actual flight parameters affecting tipoff. The period of flight which of greatest concern is during the 3rd and 4th stage coast. It is felt that unbalance in the expended third stage initiates some of the flight anomolies which are later dampened out during the first 20 seconds of fourth stage thrusting. Again some of the anomolies seem to be magnified at the moment of fourth stage ignition, in particular tipoff seems to occur at this time. The tipoff anomoly has been variously ascribed to uneven diaphragm rupture, heat shield hang up and the unbalanced force caused by snapping the wire cable which initiates heat shield ejection at third stage separation. With so many suspected areas for possible trouble we have eliminated the cable problem on our newer vehicles. In the near future we plan to monitor the time of heat shield ejection, and in a year or so a test version of the Javelin may be flown less the blowout diaphragm. In concluding this section it must be pointed out that all Javelin payloads are dynamically balanced as payloads alone, and then balanced as a composite of the fourth stage motor. In the composite balance operation the nose cone, the heat shield and the support tube are all assembled in flight condition and balanced as a unit while spinning at 540 rpm. The balance criteria for the entire fourth stage is 2.0 ounce inches static and 60.0 ounce inches squared dynamic, this is always exceeded. Frequently the residual unbalance is less than 3 grams, the total weight of the assembled fourth stage with a 120 pounds gross payload is 683 pounds.

AERODYNAMIC STABILITY

The Javelin as mentioned elsewhere is a fin and spin stabilized vehicle. The stability imparted by the fins is based on the same theory as that used for any other fin stabilized vehicle.

GYROSCOPIC STABILITY

The fourth stage is spin stabilized and the theory in this case is analogous to the theory used in explaining the inherent stability of a gyroscope.

The Javelin must have a gyroscopic stability factor of 3.0 at fourth stage ignition. This is the reason why a great deal of caution is exercised in selecting the fourth stage ignition time. Fourth stage ignition time can vary depending on the vehicle payload weight and the roll rate at ignition. For purpose of brevity and roll rate cannot be less than 4 rps nor more than 9 rps. These limits are a direct function of pitch roll coupling if less than 4 rps and simply X-248 rocket motor damage at greater than 9 rps.

1. Stability factor is defined as:

$$S = \frac{\left[\frac{I_R}{I_Y}\right]^2 p^2}{4\, M\alpha}$$

2. Aerodynamic turning moment is defined as:

$$M\alpha = C_L\,\alpha \cdot q \cdot S\,(X_{CG} - X_{CP})$$

3. Dynamic pressure is defined as:

$$q = (\text{lb./ft.}^2) = 0.7\, P_o M^2$$

I_y = Mass moment of inertia (lb. - ft. - sec.^{2}) of the entire fourth stage in pitch and yaw

I_R = Mass moment of inertia (lb. - ft. - sec.^{2}) of the entire fourth stage in roll

P = The roll rate (radians/sec.) of the fourth stage

$C_L \alpha$ = Lift coefficient (1/radian) for Javelin vehicle = 2.01/radian

S = The reference area = 2.846 ft.^{2} for the Javelin

X_{CG} = The vehicle station at which is located the longitudinal center of gravity on the entire fourth stage alone.

X_{CP} = The vehicle station at which is located the aerodynamic center of pressure of the fourth stage vehicle alone = 2.3 ft. for the Javelin fourth stage

M = The mach number at fourth stage ignition

P_o = The ambient pressure existing at the altitude of the fourth stage vehicle at the instant of ignition

For each Javelin launch the Gyroscopic Stability factor is checked out by using the above formula. It is quite obvious that for each payload launched on the Javelin the following must be either calculated or measured: I_R, I_Y, I_P, CG and the weight of the payload. In all instances these parameters are measured.

The above moments are measured using a two wire suspension system called the bifilar system. The center of gravity is measured by means of load cells which sense the fine balance or lack thereof; and by knowing the magnitude of the unbalance, we simply resolve the forces indicated and locate the center of gravity or resultant force.

AERODYNAMIC HEATING

A complete analytical computer program is available to probe the heating problems at any station along the entire Javelin launch vehicle. For this discussion it is sufficient to state that using the fiberglass nose cone the payload sees less than a total of 52BTU of heat introduced in the entire 120 seconds prior to ejection. This is despite stagnation temperatures of 3,000° on the nose tip. The areas which are subject to high heat, are protected by thermolag, an ablative compound which protects the nose cone and inconel and fiberglass cuffs which protect the magnesium second and third stage fins.

USERS INSTRUCTIONS

1. Since gyroscopic stability of the fourth stage of the Javelin vehicle depends a great deal upon the mass characteristics of the payload, engineering studies have been made of this vehicle and they indicate that it will be necessary to compute the gyroscopic stability factor at ignition of the fourth stage for each flight. In order to keep this factor above 3.0, it will be necessary to consider the mass characteristics of the payload versus the gyroscopic stability factor, and make necessary tradeoffs of apogee (extend the coast time before ignition of the fourth stage) versus the mass characteristics. The following mass parameters of the payload must be made available for the above work. Approximately three months before a scheduled launch date: Weight of the payload, location of the c.g. from the main surface of the base plate with the extension tube, the moment of inertia about the c.g. in pitch, the roll moment of inertia about the longitudinal axis, and the dynamic and static balance expected to be obtained for the payload.

2. All payloads will be balanced to the following criteria: Dynamic balance–20 ounce inch squared, static balance–one ounce inch. Other less restrictive balance criteria are possible if requirements are closely integrated by the concerned parties.

3. The following design criteria for Javelin payloads should be followed for a smooth transition thru design and launch.

   a. Establish special requirements of the payload versus the standard vehicle capabilities as early as possible.

   b. Design for the c.g. as close to the base plate as possible.

   c. Keep the ratio of the moment of inertia in roll to the moment of inertia in pitch as large as possible.

   d. Design the outside configuration of the payload for smooth ejection of the nose cone in the forward direction.

e. Center the payload by using two taper pins on the same bolt circle as the attaching screws. The pins should be located so that the payload can only be installed to the extension tube in one way. The payload base plate must be a snug fit with the 16.550 ± .005 dia. of the extension tube, i.e. 16,540 ± .000/.002.

f. A circuit diagram must be provided three months before flight.

g. At least two separate discontinuities must be provided in every circuit for squibs or explosive actuated devices. The following can be used for arming circuits in flight: A "g" actuated timer, nose cone ejection action to operate a switch, etc. With close coordination a signal from the circuit used for ejection of the nose cone can be provided.

4. In order to provide experimenters with up to date criteria on rocket performance, it is very desirable to carry performance gages on each rocket flight. The two most important performance gages are magnetometers and longitudinal accelerometers. Highly desirable gages would be lateral acceleration gages, pitch-yaw indicators, roll rate indicator, aspect measurement devices, and high frequency vibration transudcers. The latter instruments would greatly assist in determining the environmental test criteria.

TYPICAL PERFORMANCE CURVES

The following curves have been included so that the reader can better visualize the actual performance of the Javelin rocket vehicle. With the curves chosen one can plan a typical Javelin launch if he knows the weight of the payload:

Figure 5 – Acceleration vs. Time for 83° launch

Figure 6 – Apogee vs. Gross Payload (Kilometers) (Statute miles)

Figure 7 – Time above 200 statute miles

SPECIAL EQUIPMENT

The Javelin is equipped with a yo-yo despin mechanism for payload despin after nose cone ejection. A discussion of this system is contained in NASA TN D-708 authored by Mr. J.V. Fedor in August 1961.

The despin system releases known weights at 9-1/2 ft. arms. The despin is accomplished in a manner similar to a skater opening his arms to stop himself from spinning. These weights having accomplished the despin function fly off tangent to the spinning rocket. Another definition for the despin phenomenon is that the release of these weights tends to discard momentum thus reducing the spin. Both the above definitions are quite inadequate to do justice to a rather complex problem.

The following data sheet and Figure 8 which are based on the NASA TN D-708 is applicable for the Javelin. It is to be noted that we need to know the I_R of the payload and expended X-248 rocket motor case.

An outline drawing of the vehicle which gives individual weights and center of gravity locations appears in Figure 9.

ARGO D-4 YO-YO DE-SPIN CALCULATION
(BASED ON NASA TN D-708)

Moment of Inertia Roll For

X-248 Motor Case Assy	=	.6101	Slug-Ft ^{2}
Extension Tube	=	.1750	Slug-Ft ^{2}
Lock Ring	=	.0575	Slug-Ft ^{2}
Payload (Measure)	=	____	Slug-Ft ^{2}
MI Total			

(1) Record

I =		Slug-Ft ^{2}		(Total MI Roll)
a =	.762	Ft		(Radius of Rocket)
l =	9.420	Ft		(Length of One Wire)
ω_o =		Rev/sec $\times 2\pi$ =	Rad/sec	(Initial Spin Rate)
ω_f =		Rev/sec $\times 2\pi$ =	Rad/sec	(Final Spin Rate)

(2) Compute $r = \frac{\omega_f}{\omega_o}$ = __________ = ____________

(3) From Curve Read Value of B = __________________

(4) Compute $W = Mg = \frac{Ig}{B(1+a)^2} = \frac{(\qquad)(32.2)}{B(10.148)^2} = \frac{(\qquad\qquad)}{(\qquad)(103)} =$

W = __________ Lbs. (Weight of Both Weights + Wires)

(5) Subtract Weight of Wires
W - 1/3 Wt. of Wires = __________ - .0502 = _________ Lbs.

(6) Convert to Grams
_______________ Lbs. $\times$ 453.6 = _________ Grams

(7) Weight of One Spin Weight

$\frac{\underline{\qquad\qquad}\text{ Grams}}{2}$ = _____________ Grams

APPENDIX C

LIST OF ILLUSTRATIONS

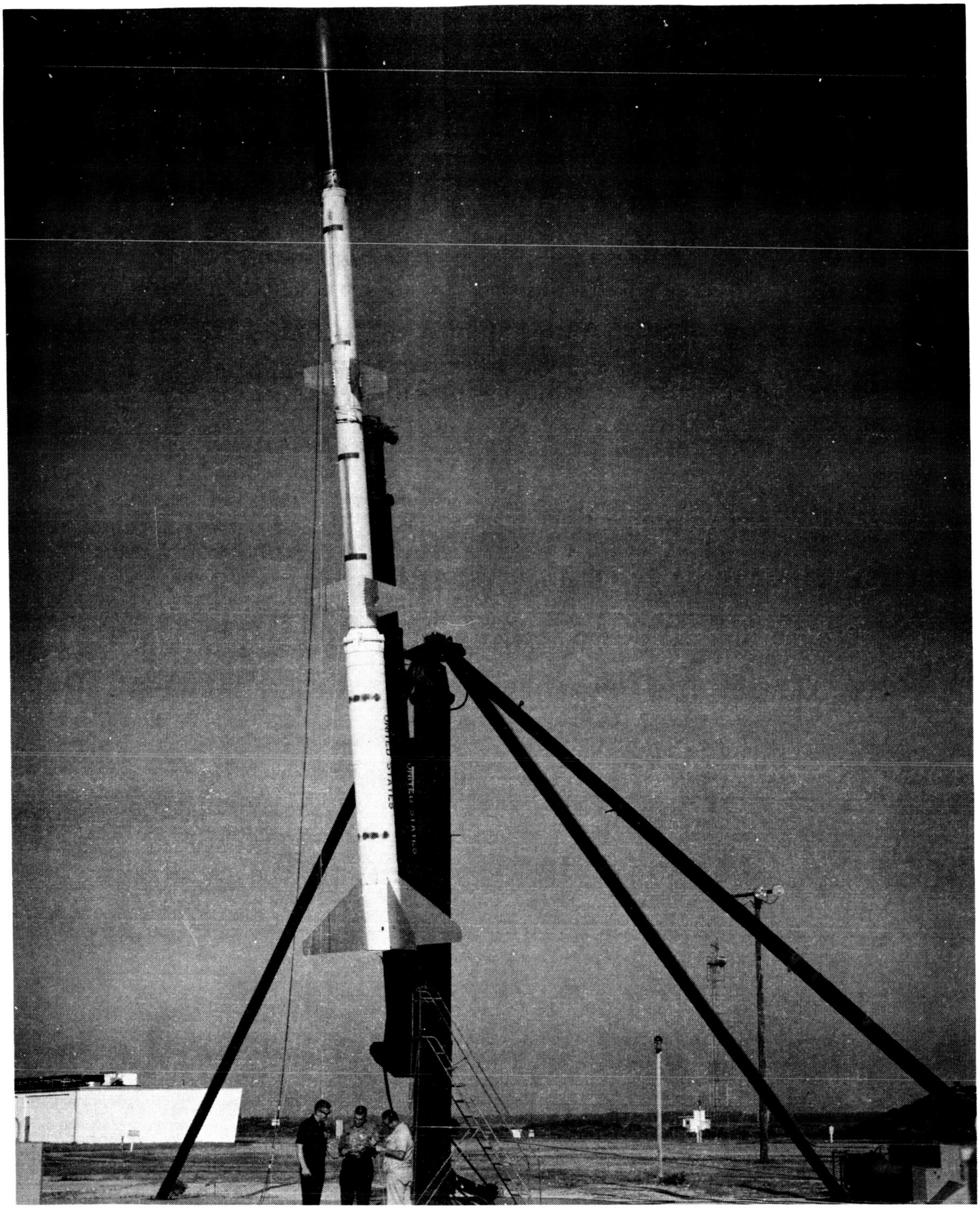


Figure 1–Javelin on Launcher

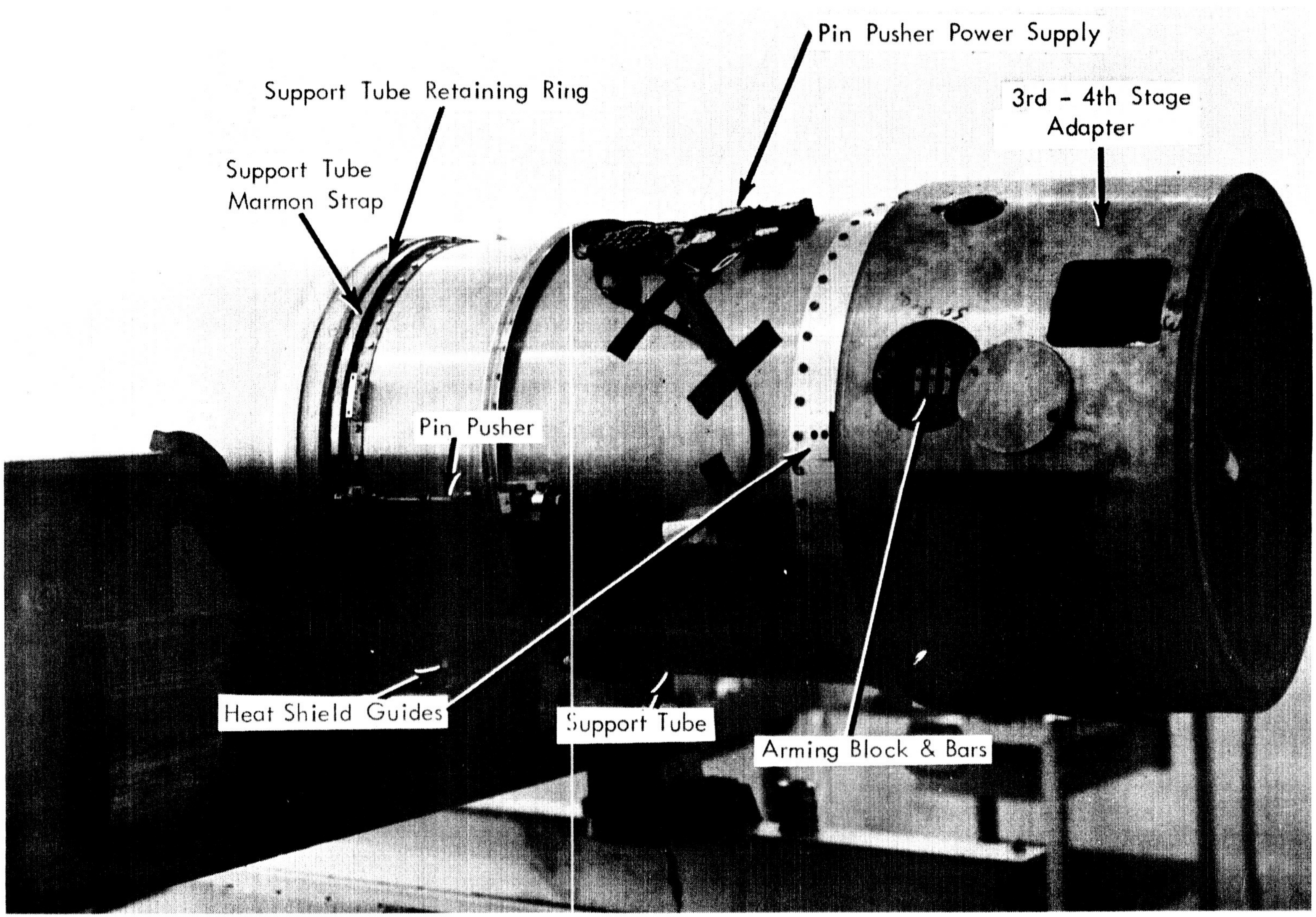


Figure 2–Javelin Fourth Stage Aft Installation

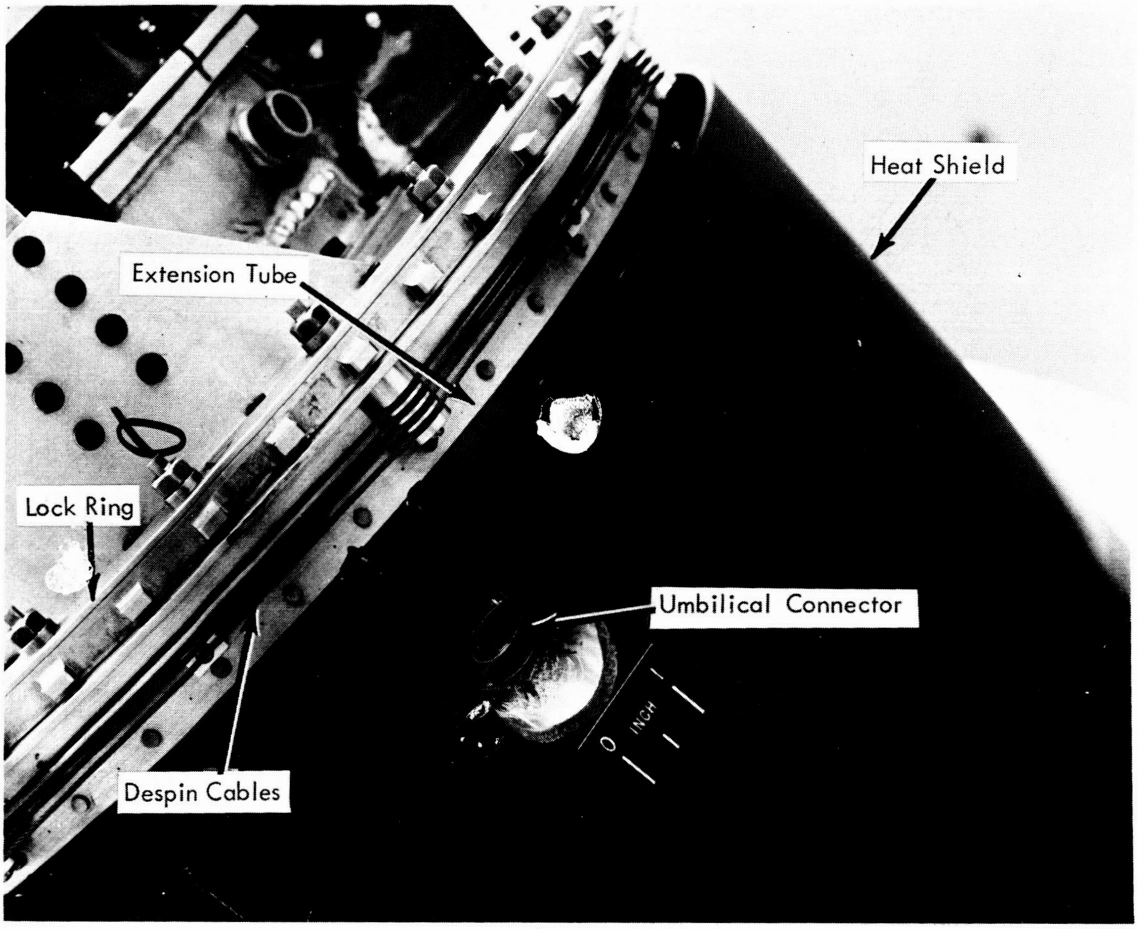


Figure 3—Javelin Lock Ring – Despin and Umbilical Installation on Launcher

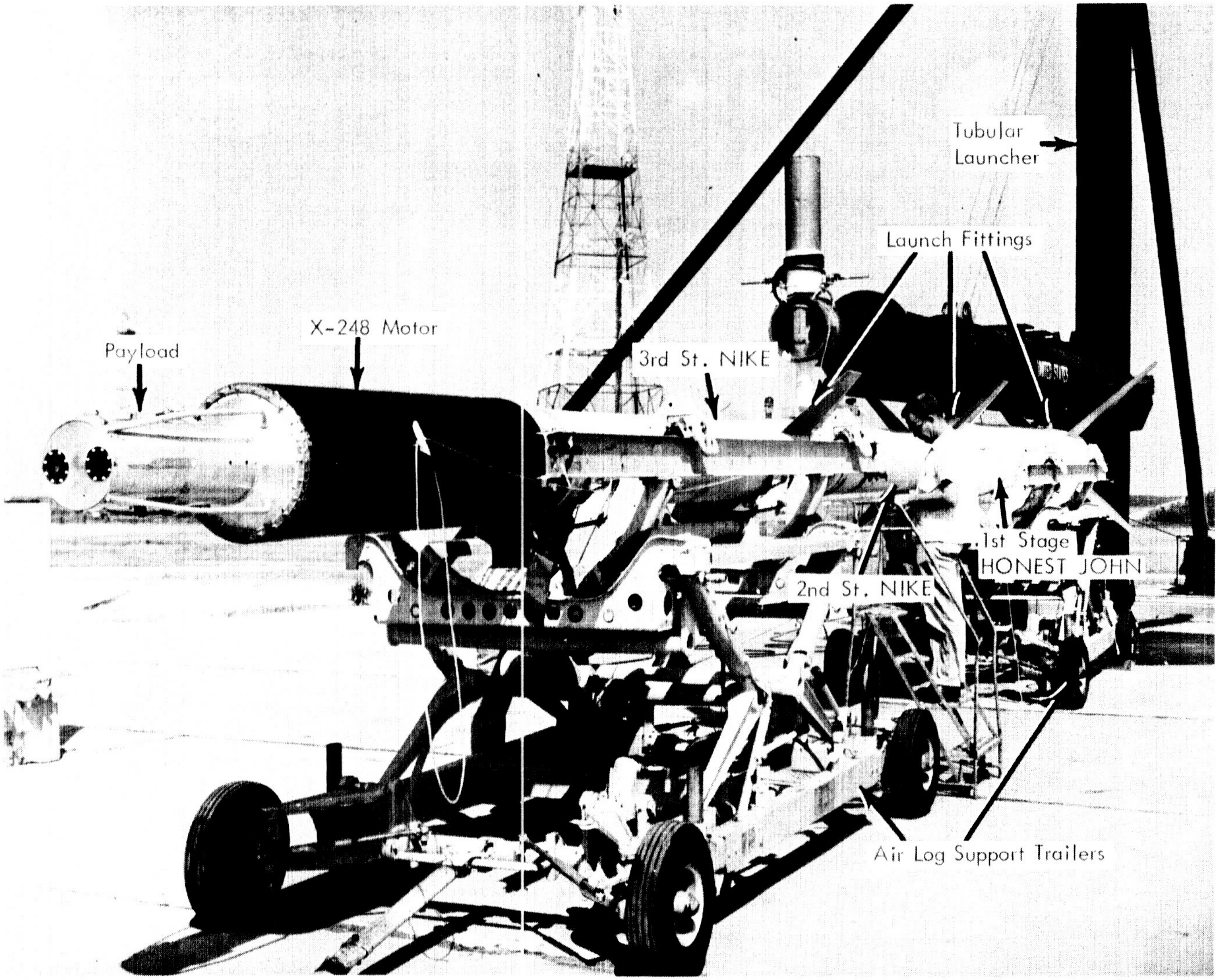


Figure 4—Javelin Installation on Launcher

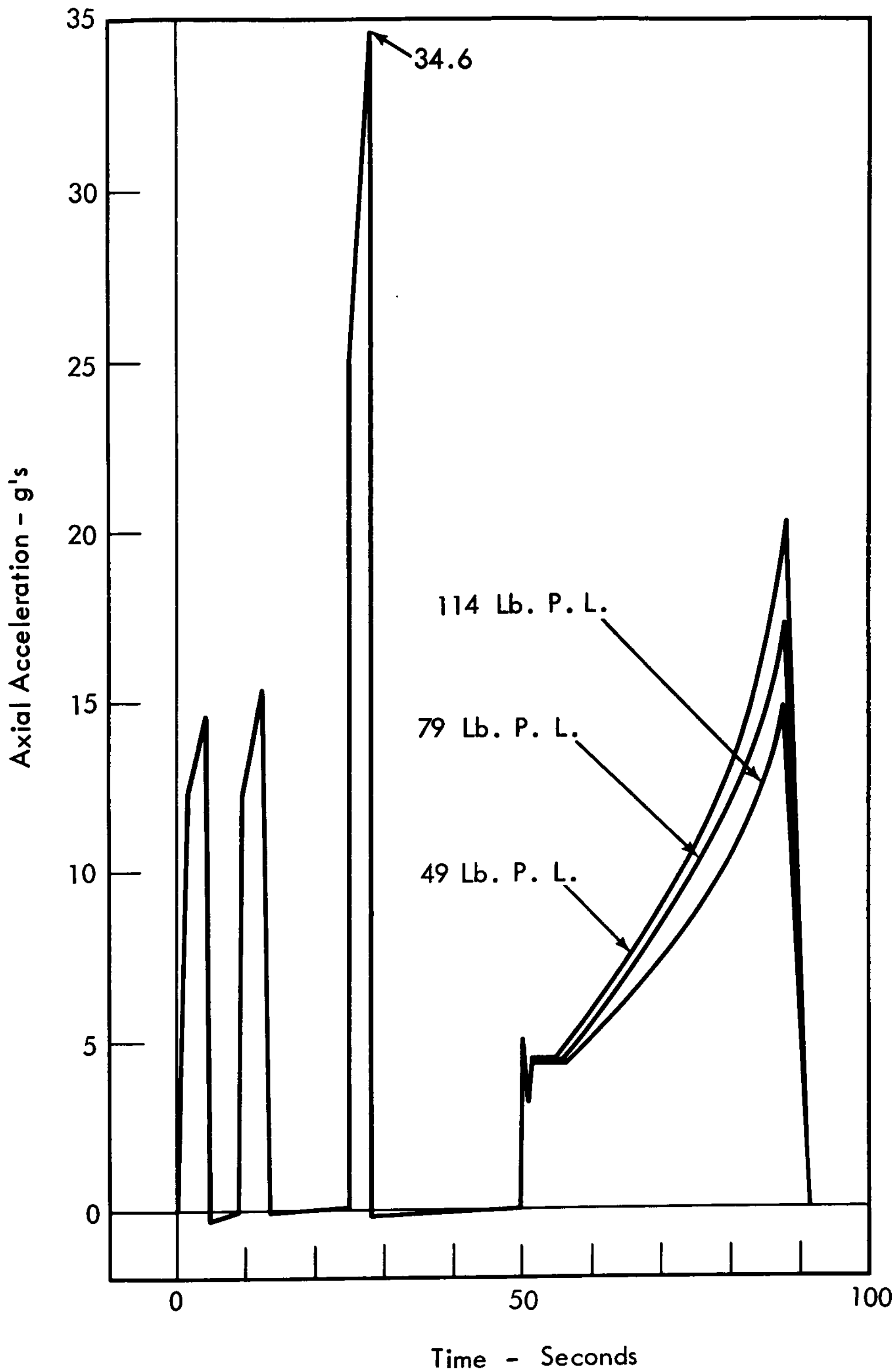


Figure 5—Javelin Axial Acceleration

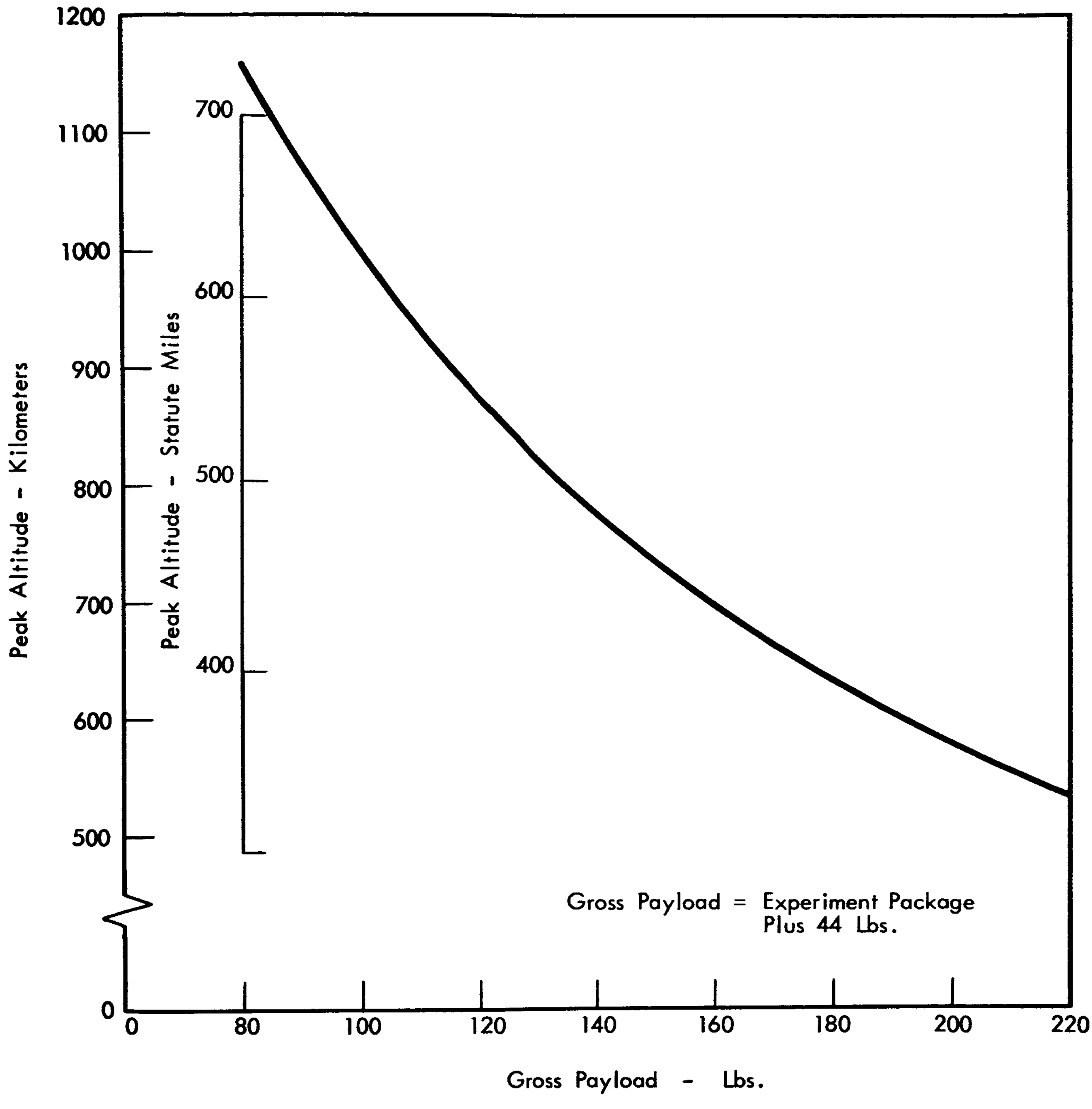


Figure 6 -- Javelin
Peak Altitude (km) Vs.
Gross Payload (Lbs.)
80° QE & 90° Azimuth

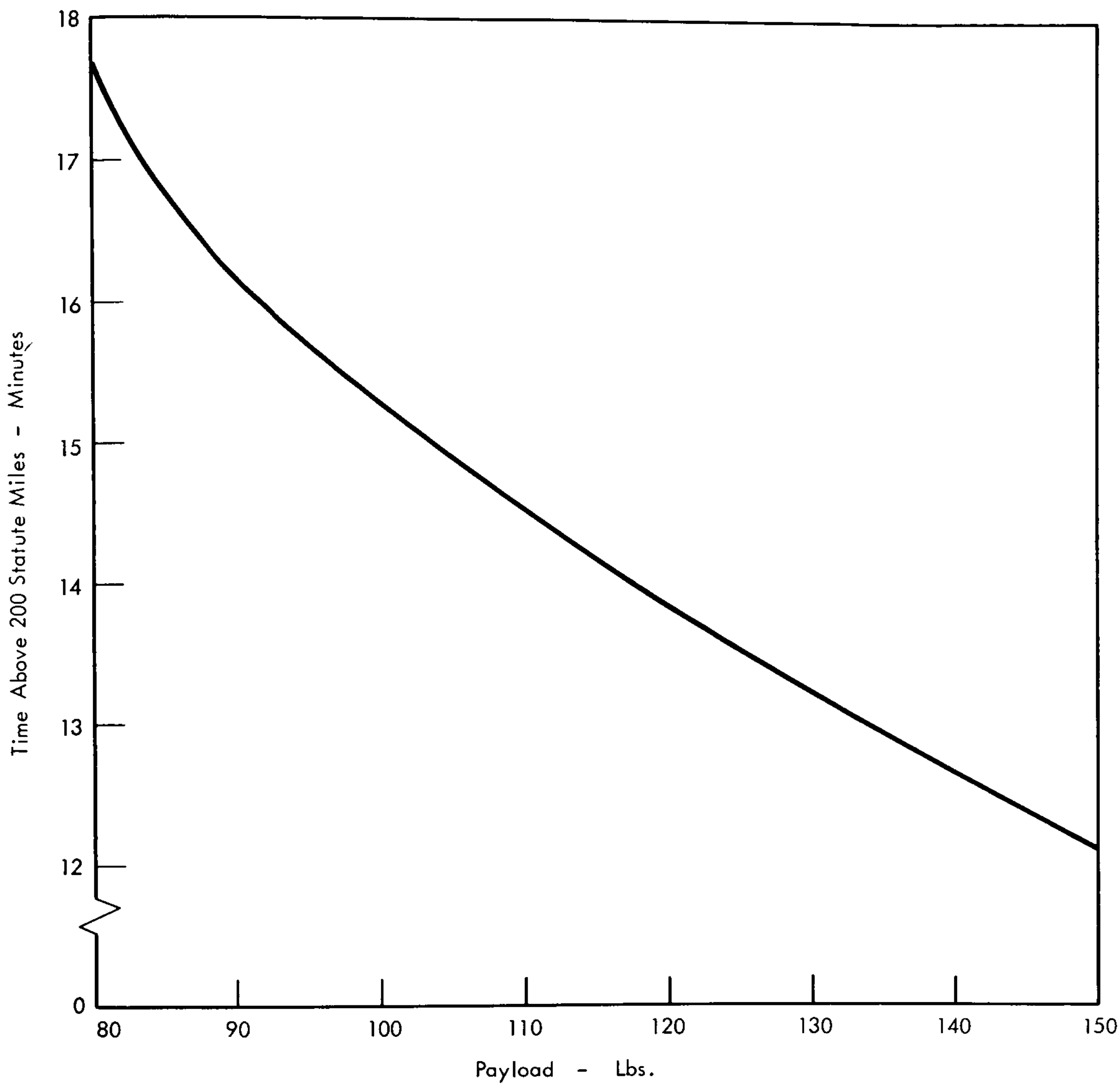


Figure 7—Javelin Time Above 200 Miles Vs. Payload

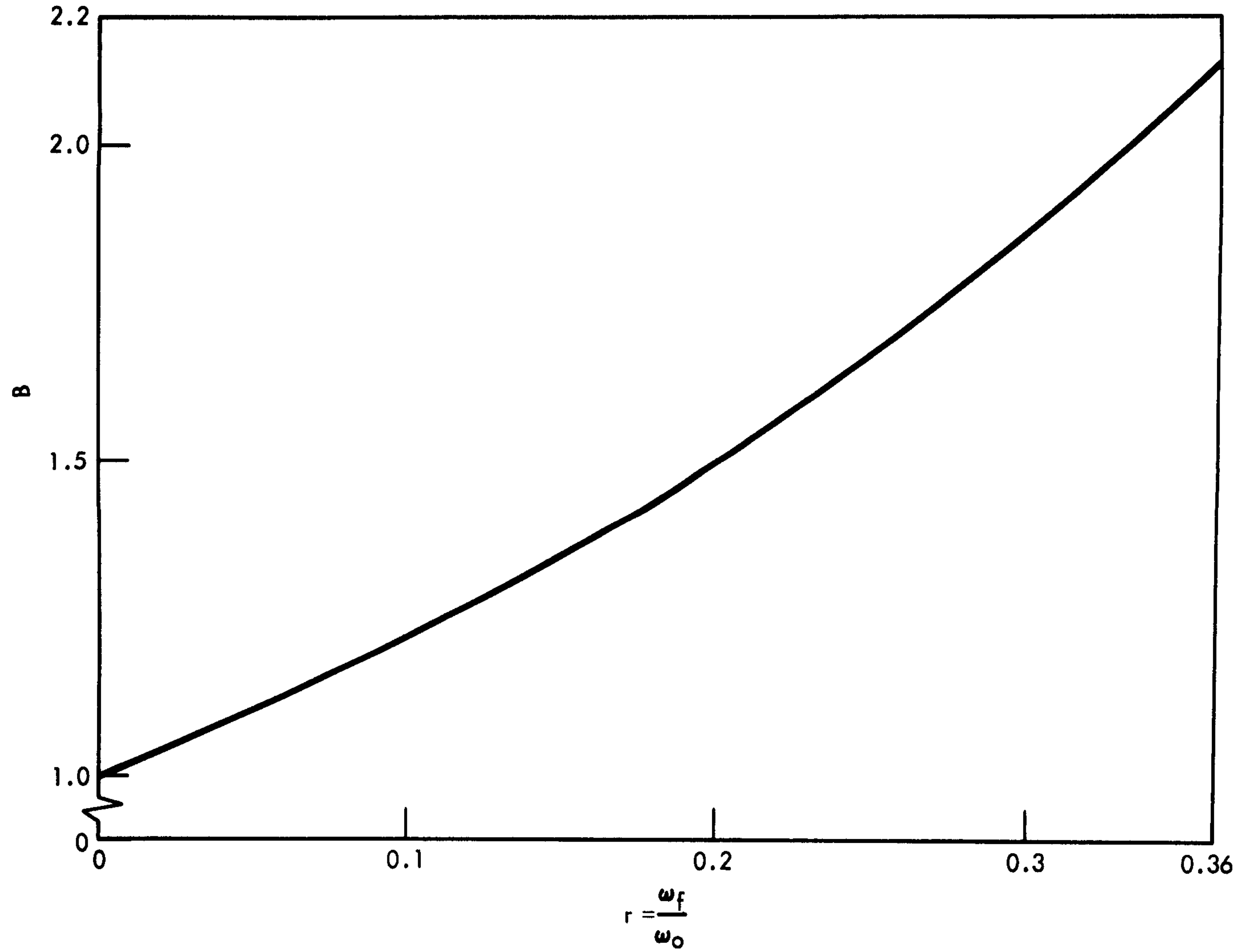
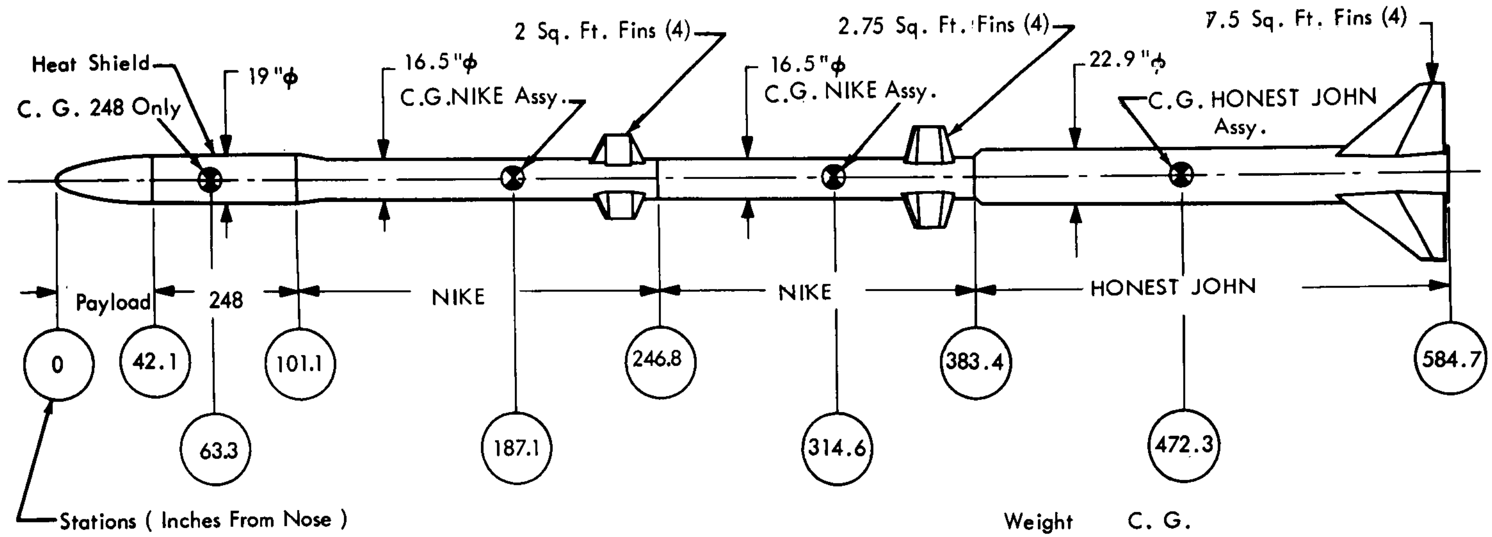


Figure 8 -- Yo-Yo Despin Calculation Curve Javelin (Based on NASA TND-708)



	Weight Lbs.	C. G. Station
Gross Payload (with Nose Cone, Ext. Tube & Lock Ring) (8.23 GA)	110.	
248 4th. Stage Motor Only	514.	63.3
248 Heat Shield, Support Tube, Guide Rods & Hdw.	45.	
NIKE 3rd. Stage Motor (with Fins & Adapter)	1,322.	187.1
NIKE 2nd. Stage Motor (with Fins & Adapter)	1,344.	314.6
HONEST JOHN 1st. Stage Motor (with Fins & Adapter)	4,156.	472.3
VEHICLE AT LIFTOFF	7,491.	

Figure 9–Javelin Outline Drawing

APPENDIX D

Nike Apache

by

C. M. Hendricks

GENERAL DESCRIPTION

The Nike Apache Vehicle System is a two stage, solid fueled, unguided rocket, see Figure 1, used primarily for upper atmosphere research. The basic system is a simple, reliable one which is adaptable to use by many different experimenters, even in the most remote locations.

The first stage of the system is a standard M5E1 Nike, used by the U. S. Army as the booster for the Nike-Ajax. This motor is coupled to the second stage by an adapter which slip fits into the Apache nozzle.

The second stage is the Thiokol TE 307-2 Apache. It is 6-1/2 inches in diameter, 107 inches in length and contains 131 pounds of aluminized polyurethane propellant.

The size of the payload varies but is usually 6.5 inches in diameter and between 60" and 80" in length. Payload weights fall into the range from 50 to 90 pounds. Desired altitude is necessarily a function of payload weight, since each added pound imposes a penalty in terms of peak altitude. Typical performance for the Nike Apache is shown in Figure 2. Other payload size considerations, such as stability, will be discussed later.

The payload volume is variable, dependent on length and shape of the payload. This volume is on the order of 2,000 cubic inches for average payloads.

The flight time of the payload is nominally about seven minutes. Of more interest to the experimenter is time above some useful altitude, say 70 kilometers. This time is approximately 300 seconds.

POWER PLANT PERFORMANCE

The first stage rocket motor is a U.S. Army M5E1 Nike originally designed to be used as the first stage of the Nike Ajax missile system. It is also used in clusters of four in the Nike Hercules missile system. This motor develops approximately 47,000 pounds of thrust and has a nominal burning time of 3.5 seconds.

The environment imposed on the vehicle system by the Nike is extremely severe and must always be accounted for in payload design. The initial longitudinal acceleration during ignition of the Nike may exceed 100 g's for a period of several milliseconds. The quasi-steady state acceleration during boost builds up to a value near 45 g's prior to burnout. A typical acceleration trace during Nike burning is shown in Figure 3.

The Apache motor is produced by the Thiokol Chemical Corporation, Elkton, Maryland. This motor has an average thrust of approximately 5,000 pounds and a nominal burning time of 6.0 seconds. Detailed dimensions and weights are given in Table 1. The difference between this motor and the Cajun lies in the propellant. Externally they are similar, except for the nozzle structure. Detailed formulation of both the Apache and Cajun propellant is classified. Comparison of figures in Table 1 will show the difference between the motors.

Ignition of the Apache motor is accomplished by means of a pyrogen igniter. This igniter is initiated by delay squibs which are ignited on the ground by the same pulse that ignites the Nike. The ignition delay is used in order to allow the second stage to coast through the more dense portion of the atmosphere at a low velocity, thereby keeping the dynamic pressure and heating rate within tolerable limits. Since drag is a function of dynamic pressure, a delay time can be chosen which minimizes total drag on the vehicle and allows the maximum altitude to be reached. This time for the Apache is approximately 20 seconds. For the Cajun the time delay has been chosen as 17 seconds. The Cajun igniter is different from the Apache, in that it uses an explosive charge rather than the pyrogen.

STRUCTURE

The Nike structure consists of a steel cylinder with nozzle welded on the rear. The headcap is sealed with an "O" ring and held in place by a snap ring in a groove.

The interstage adapter is a casting which bolts to the front of the Nike after the igniter is screwed into the Nike headcap. This adapter is a slip fit into the nozzle of the second stage. Separation is accomplished by differential drag forces after first stage burnout.

The Apache structure is an aluminum tube with an aluminum, steel, phenolic and graphite nozzle. The forward end of the Apache is closed with a forged headcap with igniter attached. This headcap is aluminum with male threads for payload attachement.

The payload structure is of necessity a variable, but normally falls within certain limits. These limits are imposed by aerodynamic drag and stability considerations for the most part, but structural integrity also is involved.

A standard payload housing has been developed by GSFC and is used in many cases. This housing is made in two forms, one aluminum and one fiberglass. The fiberglass housing is used in cases where the experiment requires a non-metallic structure.

Both housings are conservatively designed to withstand the environment imposed by the flight profile.

Variations in payload configuration appear due to experimental requirements, such as the need to eject a nose cone or expose various sensors during flight. Such equipment as despin mechanisms and recovery packages are occasionally used.

AERODYNAMICS

As in most missile systems, the experimental requirements are not always compatible with the optimum aerodynamic configuration. In the case of the Apache or Cajun a reasonable amount of agreement has been reached. The most common payload consists of an 11° cone attached to a cylinder. The antenna configuration is four tapered rods, about one foot long, swept back at a 45° angle. Figure 4 shows this typical configuration. Other nose cone configurations are used occasionally and shroud type antennas are frequently attached to the Apache motor for use with Doppler tracking equipment.

The payload gross weight varies from 50 to 90 pounds, with occasional excursions beyond this range. Nominal length falls in the range 60-80 inches, also with variations beyond.

Since the payload structure weighs on the order of forty pounds, the experimenter is left with a net weight of 15-45 pounds for his experiments.

Packaging of equipment demands careful consideration, since the available volume is rather small.

STABILITY

The rocket system is unguided, and stability is achieved through the use of fins on both stages. The Nike fins are canted to achieve a low (2 rps) roll rate at Nike burnout. The purpose of this practice is to minimize dispersion of the rocket.

The second stage is spun to about five revolutions per second at burnout. The purpose of this is to assure stability during the portion of the flight above the atmosphere. In doing this, good stabilization in the lower atmosphere is also achieved, since the roll rate is kept well above the point where pitch-roll coupling might occur. The second stage spin is obtained by the use of wedges on the trailing edge of the fins. The size of these wedges is determined from predicted second stage burnout velocity prior to flight.

LAUNCH PREPARATION

The Nike-Apache system is a simple and reliable one that requires a minimum of effort for preparation prior to launch. After uncrating of the motors, the necessary preparations consist of installing and aligning the fins and wiring the igniters.

LAUNCHERS

The mobility of the system is due in large part to the type launcher required. The most common, and most readily available is the standard Nike Ajax launcher. This is a rail type launcher with wheels which may be towed to its final location by truck. For scientific applications a small reinforced concrete pad is recommended for launcher installation. Once installed the launcher may be trained in azimuth and elevation. Elevation adjustment is accomplished electrically but may be done by hand if necessary.

Other launchers in use are zero length types in which the rocket is free of restraint after one to two inches of travel.

IMPACT PREDICTION

Impact predictions may be carried out by hand or by computer if available. Rather simple methods are available for these computations. Winds are obtained by theodolite sightings of balloons released prior to flight time. These data are combined in a form which allows launcher setting to impact at a certain point, or conversely to predict the point of impact for a given launcher setting.

RECOVERY

On occasions a recovery system is used for the Nike Apache system. Unless there is a definite scientific requirement for recovery, however, such as in a photographic mission or air sampling experiment, the recovery system usually imposes too great a penalty in terms of weight and size. Having once flown, there are many payload components which would not be considered reliable for use again so recovery would be of little value for this reason.

SAFETY

The Nike Apache system is a safe one, but like all explosives must be handled with care. In particular the rockets must be kept away from open flame and grounded at all times as protection against static charges. In general, the safety procedures are the same for these units as for other items in the same hazard class. In launching the vehicle care must be taken to assure that no one is near the rocket when power is applied or radio frequency energy is radiated for tests. By strictly adhering to these practices at all times it is possible to avoid injury to personnel involved with the rockets.

In this brief description an attempt has been made to give the reader a feel for the problems involved and advantages of using such a sounding rocket system as the Nike-Apache or Nike-Cajun. The safety, reliability and ease of handling make it a system worth consideration by any experimenter interested in the region between 80 and 240 kilometers.

TABLE 1

WEIGHTS AND DIMENSIONS

NIKE

(All figures include igniter and adapter)

Weight	1317	Pounds
Length	149	Inches
Diameter	16.5	Inches
Center of Gravity (from nozzle exit plane)	75	Inches

APACHE

Weight	217.5	Pounds
Length	107.94	Inches
Diameter	6.5	Inches
Center of Gravity (from nozzle exit plane)	50	Inches

CAJUN

Weight	202	Pounds
Length	107.94	Inches
Diameter	6.5	Inches
Center of Gravity (from nozzle exit plane)		

APPENDIX D

LIST OF ILLUSTRATIONS

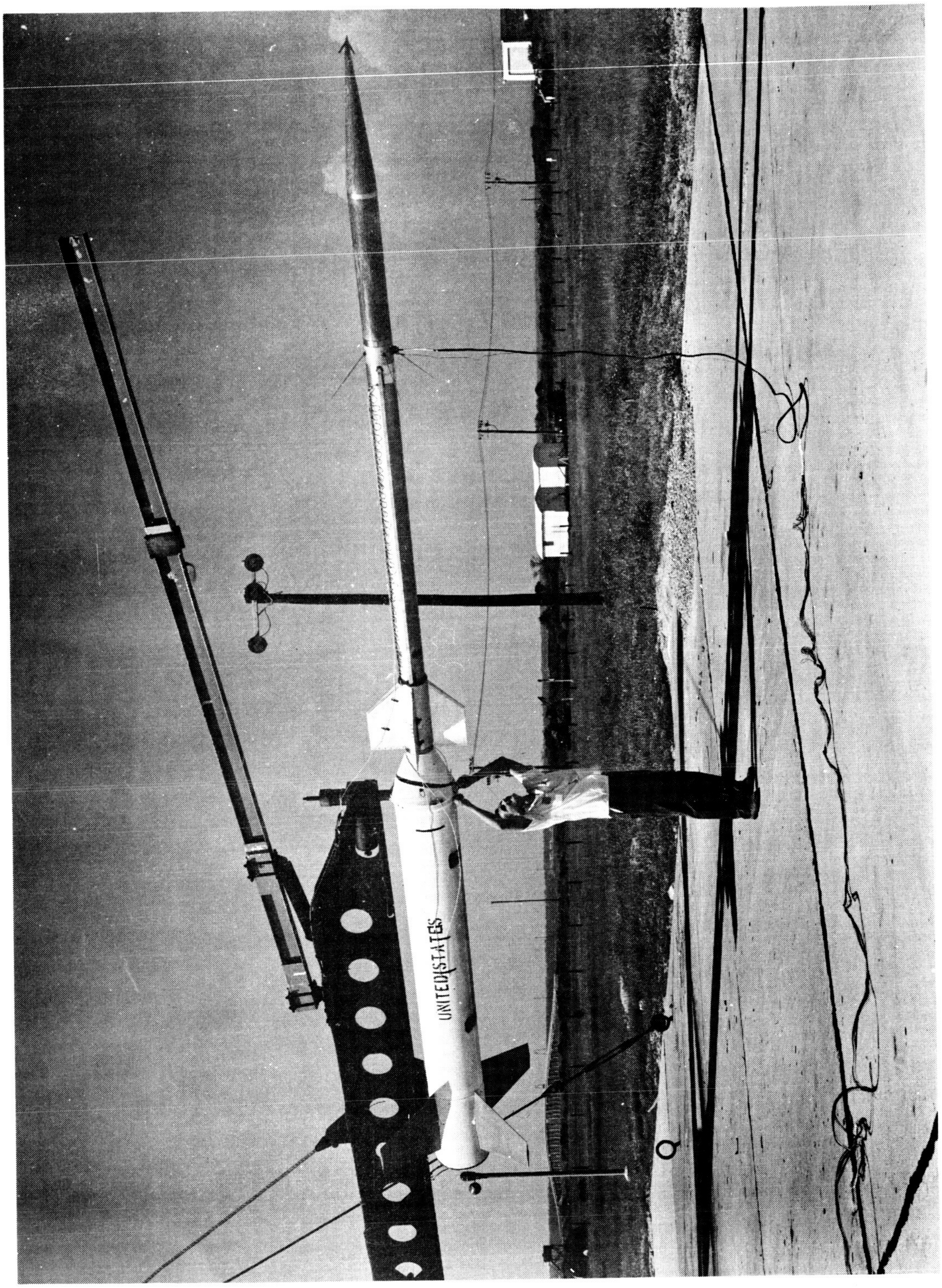


Figure 1–Nike Apache Launch Preparation

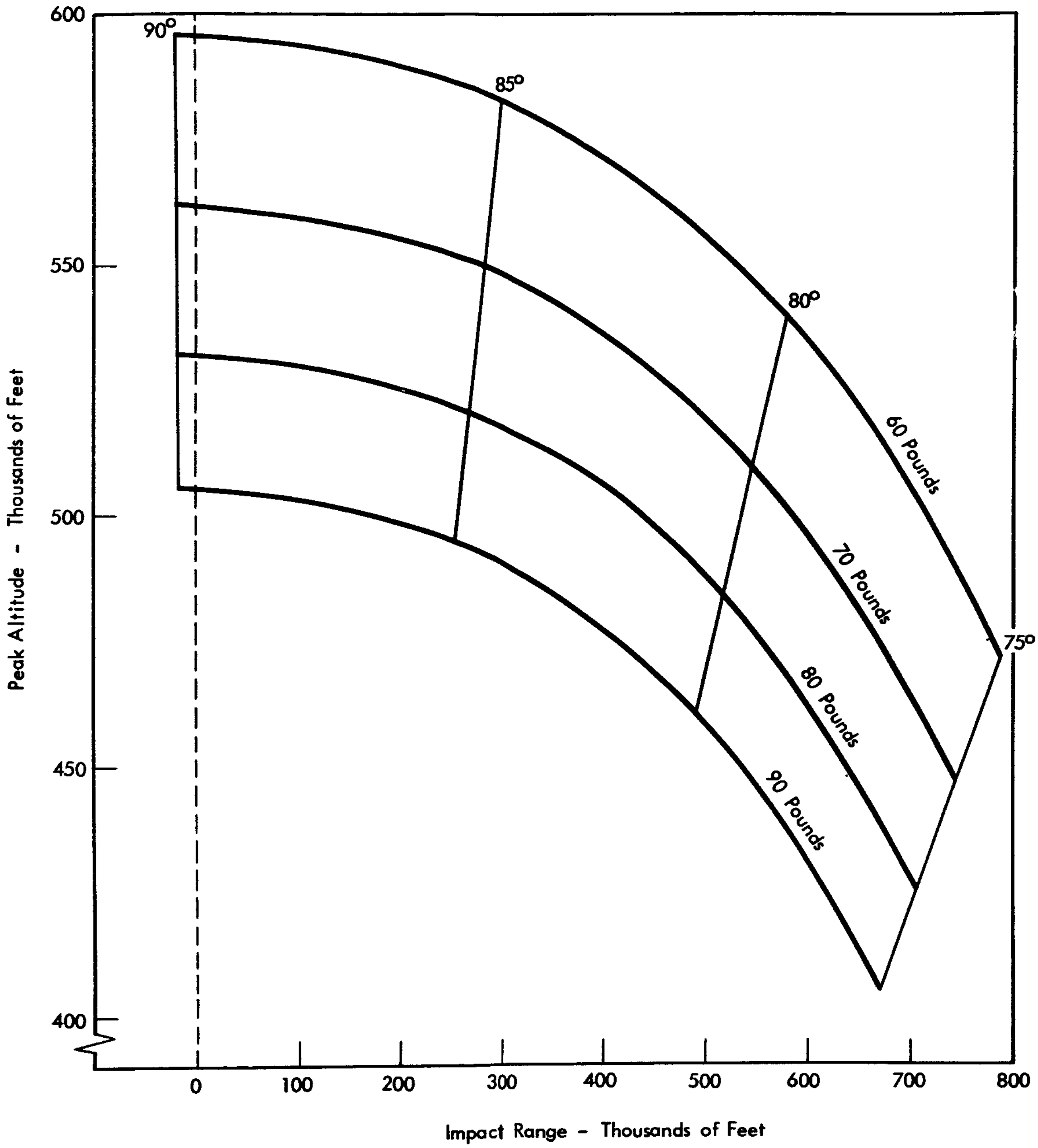


Figure 2—Nike Apache Performance as Function of Payload Weight and Launch Angle

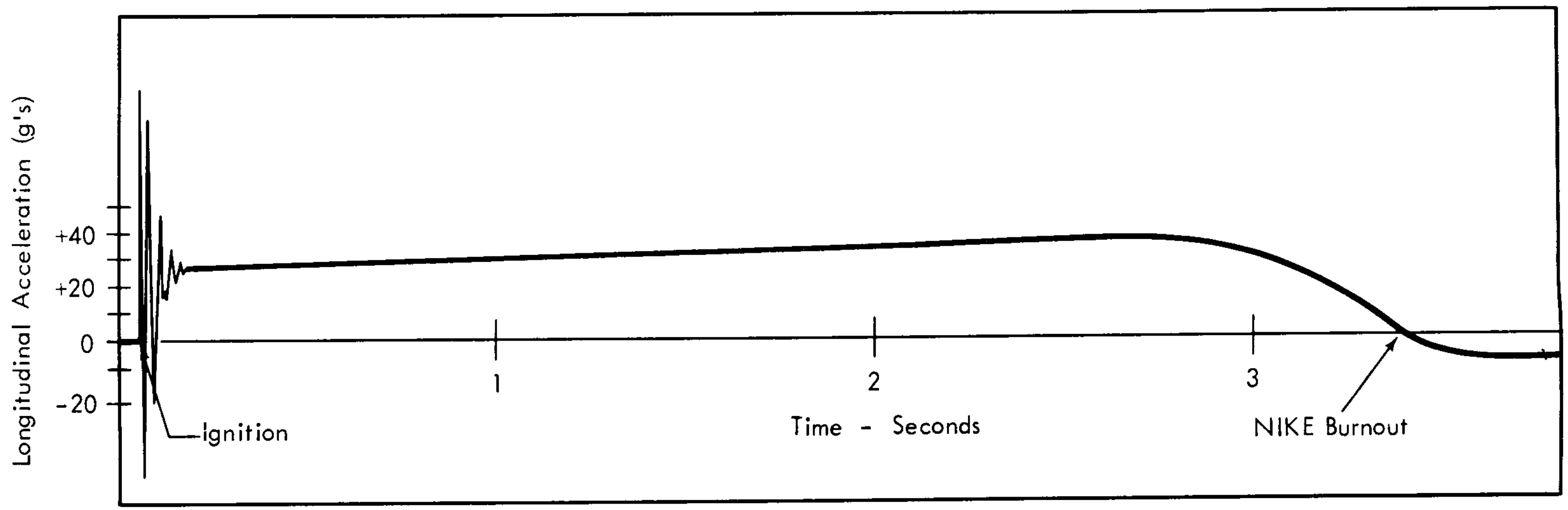


Figure 3–Typical Acceleration Along Thrust Axis

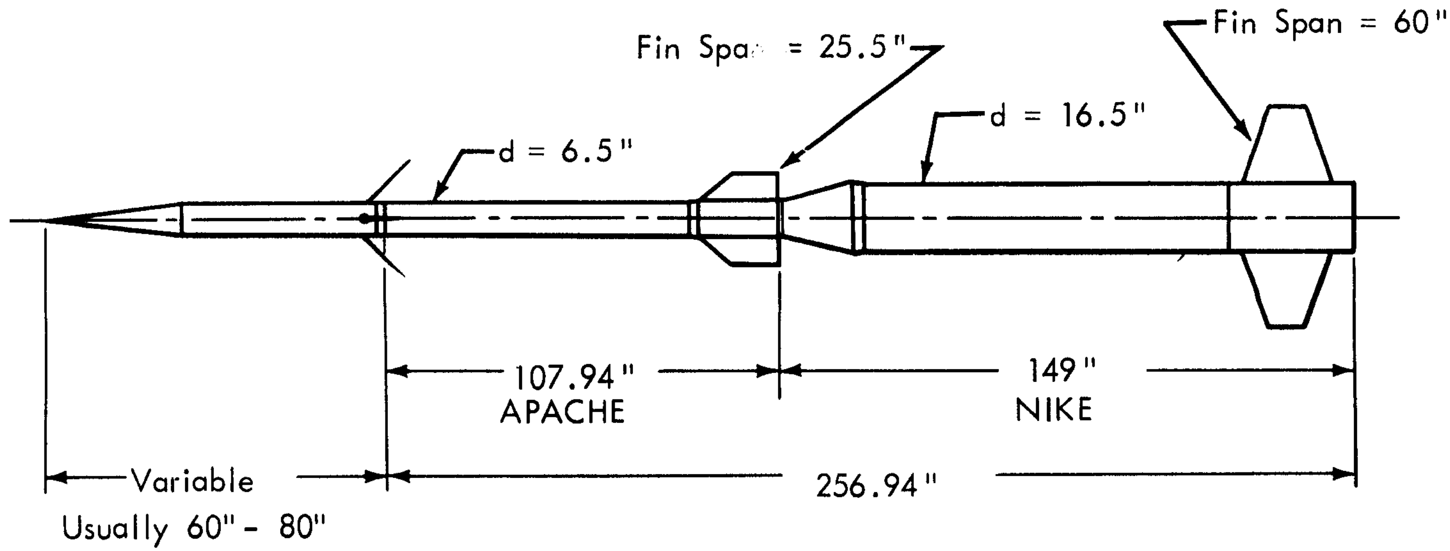


Figure 4–Nike Apache Outline