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ISTRACON HANDBOOK

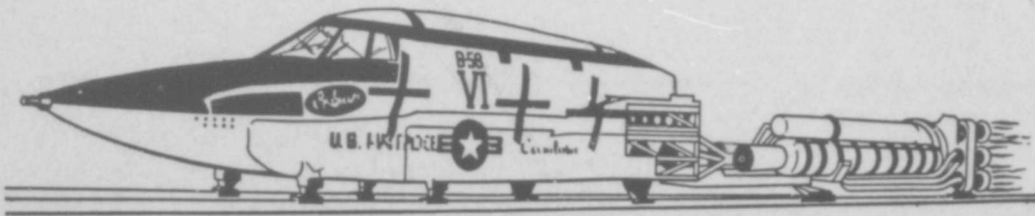
**INTERSTATION SUPERSONIC TRACK CONFERENCE
PROPULSION WORKING GROUP**

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**AIR FORCE MISSILE DEVELOPMENT CENTER
U.S. NAVAL ORNANCE TEST STATION
U.S. NAVAL AIR TEST FACILITY (SI)**

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FOREWORD

This handbook, published by members of the Propulsion Working Group of the Interstation Supersonic Track Conference (ISTRACON), is intended to supplement the Rocket Sled Design Handbook (Report #60-1/Rev Dec 1961) produced by the ISTRACON Structures Working Group.

The regular member facilities of ISTRACON include the Air Force Missile Development Center, Holloman AFB, New Mexico; the Naval Air Facility (SI), Lakehurst, New Jersey; and the U. S. Naval Ordnance Test Station, China Lake, California; the Sandia Corporation, Albuquerque, New Mexico, is an associate member. The U. S. Naval Propellant Plant, Indian Head, Maryland, and Code RMMP of the U. S. Bureau of Naval Weapons act as advisors to the Propulsion Working Group.

The Sandia Corporation, the Naval Propellant Plant, and Code RMMP have all contributed to the material presented in this handbook, which was prepared for publication by the Test Department Editorial Branch of the U. S. Naval Ordnance Test Station.

The manual is intended to provide sled designers with background information on the design of solid and liquid propulsion units and their use as track motors. It includes information on rocket motors, mounting provisions, electrical initiation of rocket motors, and a description of liquid propulsion systems currently in use. A glossary of track propulsion terms appears on page 24. A reproduction of informal report, "Staged Ignition of Sleds at SNORT," (5 Sep 1957), by Howard S. Olson, NOTS, is also appended.

ISTRACON encourages all users of the handbook to present new ideas which may advance the state of the art of track testing. It is suggested that such ideas be brought to the attention of the Propulsion Working Group for possible incorporation in future revisions of the handbook.

CONTENTS

History of Track Testing	1
Development of Solid-Propellant Systems	3
Use of Tracks	3
General Characteristics of a Track Propulsor	3
General Operating Principles for a Solid-Propellant Unit	4
Design Concepts for Solid-Propulsion Units	5
Propellant Grain	5
Motor Case	9
Exhaust Nozzle	9
Igniter	10
Design Conclusions	10
Ignition Characteristics	10
Ignition of Solid Propellants	12
Igniter Types	12
Ignition Techniques Used in Track Testing	14
Rocket Motor Electrical Distribution Systems	14
Fabrication of a Sled-Mounted Electrical Distribution System	14
Design of Pusher Sled for Use With Solid-Propellant Units	17
Mounting Provisions	17
Igniter Considerations	17
Wiring Provisions	18
Development of Liquid Propulsion Systems	18
Design of the Liquid Propellant System	19
Three-Chamber Liquid Engine Test Preparation and Firing Sequence	20
Liquid Propulsion System Performance	21
Future Liquid Propulsion Systems	23
Glossary	24
Appendix: Staged Ignition of Sleds at SNORT	31
Bibliography	51

Figures:

1. Thrust Development in a Rocket Motor	5
2. A Type of Igniter Assembly	11
3. Various Types of Igniter Assemblies for Solid Pro- pellant Rocket Systems	13
4. Single-Stage Knife-Blade Mounting	16
5. Track-Mounted Screen Box	16
6. Track-Mounted Screen Box for High-Velocity Applications	16
7. Thrust Profile Using 4,800-lb Forebody	22
8. Typical Thrust Profile Showing Results of Staggered Termination Technique	22

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HISTORY OF TRACK TESTING

With the advent of high-speed aircraft and weapon systems, it became necessary to devise some method for testing entire systems or their component parts in such a way that test items could be recovered for study. Wind tunnels verified many design factors; however, it was very difficult to simulate the exact environmental conditions encountered in free flight because they are constantly changing. Free-flight tests, themselves, seldom produce completely adequate results since the test item cannot be recovered intact. Therefore, supersonic track testing has been developed to breach the gap between wind-tunnel and free-flight testing.

During and following WW II, scientists and engineers of the U. S. Naval Ordnance Test Station (NOTS) sought better ways of testing rockets and guided missiles. Aircraft fire-control systems required a moving target whose position could be predicted at any specific time, and closely controlled with respect to the position of the firing aircraft. It also became necessary to develop a method of checking aircraft rocket launchers in such a way that neither the aircraft nor the pilot would be endangered.

In 1946, a railroad track was laid at NOTS, China Lake, to attempt to meet these requirements. A small railroad car, powered by a gasoline engine, was used in the first successful track runs. As a result of these runs, it became apparent that higher speed and accelerations were necessary to more closely approximate the actual environment of weapon systems. To attain these higher accelerations, a sled configuration was developed by replacing the wheels of the railroad car with slippers which wrapped around the rail heads to prevent the vehicle from leaving the track, and the gasoline engine was replaced by aircraft rockets. This was the beginning of track testing as it is known today.

The original moving target driven by a gasoline-powered sled at relatively low velocities up to 35 mph has been extended and improved so that vehicles weighing 25 tons have been successfully recovered at speeds up to 300 mph, and comparatively light sleds weighing 300 lb have been recovered at speeds of 3,100 mph. To obtain these performances, it became necessary to use rocket motors which were both reliable and inexpensive. The first rocket motors used were the 11"75 Tiny Tim and the 5"00 high-velocity aircraft rockets, both of which are still in use today. As newer and higher-performance rockets were developed for weapon systems, they were scrutinized for use in sled propulsion applications.

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The various acceleration and velocity profiles needed to meet test specifications of different components have generated the need for many different types of rockets. For example, testing of guidance systems and associated components normally requires constant accelerations of from 4 to 10 g; whereas aircraft seat-ejection systems require velocities from 120 to 1,200 mph with no acceleration at ejection. Some test specifications include velocity requirements and maximum acceleration limitations, while others have acceleration requirements and velocity limitations.

To meet these criteria, rockets are staged (or ignited) at different times during the sled run. This is done by delivering the necessary power to the rocket igniter at the exact point during the sled run where the added thrust is needed. However, in some types of sled tests, staging is not suitable since it produces an abrupt change in acceleration. This is most evident in guidance testing.

Because of these requirements and the necessity for conducting a large number of runs at relatively low-propulsion cost per run, a cold gas, pressurized-liquid, bipropellant system was adapted for track use. The two propellant systems in current use at the tracks are acid--JPX (a mixture of 40% unsymmetrical dimethylhydrazin and 60% JP-4) and liquid oxygen-alcohol. Liquid oxygen and inhibited red fuming nitric acid act as the oxidizers in each system; the alcohol and JPX act as the fuels. Nominal specific impulses in the order of 210 lb sec/lb of propellants have been realized. In most cases, however, these liquid-propellant sleds deliver a constant thrust and are therefore limited in application. While the initial cost is high, a significant monetary saving can be realized in a long series of tests.

With the development of numerous small missile systems, solid-propellant boosters were also developed to meet the varying demands of each system. When the basic parameters of the systems changed, solid-propellant boosters became obsolete, and track range personnel were asked to evaluate the solid-propellant boosters for use in track testing. It was found that, in most cases, the boosters could be adapted to test sleds and thereby reduce propulsion costs to the agencies using the tracks. In addition to saving money, the boosters have increased the acceleration and velocity profile capabilities of recoverable sleds.

There are some track tests that require certain profiles for which neither current liquid-propulsion systems nor obsolete solid-propulsion systems prove entirely adequate. Private companies have developed solid-propulsion rockets and prepackaged liquid-propulsion rockets to meet many different thrust-time profile requirements. These are readily available from the manufacturers for use in tests needing special propulsion environments.

During the evolution of track testing from the gasoline-powered railroad car to some of the present high-performance sled propulsion systems, many performance parameters have been established. However, future requirements for higher velocities and accelerations will necessitate the development of new and higher performance propulsion systems. Track testing is in a state of continual change; many new applications of this type of testing are being established to evaluate the progressing defense programs of the Nation.

DEVELOPMENT OF SOLID-PROPELLANT SYSTEMS

The increasing demands made on the test tracks, and the extensive use by these tracks of solid-propellant units as boosters have made it necessary to study the concepts of solid-propulsion unit design. The characteristics of a solid-propulsion unit desirable for track use, and a definition of the parameters which affect design decisions are presented below.

USE OF TRACKS

The philosophy underlying track testing is to simulate acceleration and thrust conditions which the test item would be subjected to in use. Representative testing of this type depends upon programming the acceleration and velocity of the test sled. With solid-propulsion units, the program is developed by 'clustering' (or the simultaneous firing of several units) by igniting units or clusters of units in a pretimed sequence, or by using combinations of different propulsion units. The accuracy of the results of any test depends upon the control which can be exercised over the propulsion program. With solid-propellant units, this control is determined by the reproducibility and reliability of the thrust-time performance.

Reproducibility requires that there be little variation in performance among the individual units of one type.

Reliability implies a high degree of confidence in the successful operation of the unit from its ignition through burnout.

GENERAL CHARACTERISTICS OF A TRACK PROPULSOR

The general solid-propellant unit characteristics desirable for track use are: high thrust, low frontal area, short action time, neutral thrust-time operation, high systems impulse, and low cost.

High thrust and low frontal area characteristics determine the thrust range that can be obtained by clustering the units within a given area. Thus the obtainable acceleration and velocity variations can be determined.

Short action time (approximately 2 sec in most instances) permits greater flexibility in staging units to achieve the desired velocity-time program.

Neutral thrust-time operation (within $\pm 10\%$ deviation from average thrust) simplifies acceleration and velocity programming and aids in attaining realistic environments.

High systems impulse (the ratio of total impulse to loaded weight) implies low inert-parts weight which generally goes hand-in-hand with increased cost. Since several units may be expended in one test, it is important that the unit cost be as low as possible. In addition, the unit should be very smooth burning so as not to further increase the vibration inherent in track testing.

GENERAL OPERATING PRINCIPLES FOR A SOLID-PROPELLANT UNIT

A solid-propellant rocket motor is comprised of a solid-propellant grain enclosed in a pressure-confining chamber, a DeLaval-type exhaust nozzle, an igniter to initiate grain burning, and electrical leads for supplying the electrical energy to fire the igniter.

Figure 1 is a schematic of a solid-propellant motor with an internal burning grain. The combustion of the solid grain creates high-temperature gases in the motor chamber. The rate at which these gases can escape is restricted by the nozzle to maintain the pressure within the chamber. As the gases flow through the nozzle, the internal energy of pressure and heat are converted into kinetic energy of moving gas molecules. The momentum of the high-velocity gases ejected through the nozzle creates an oppositely directed force on the rocket motor. The gas momentum expressed in mass-flow per unit time is thus balanced by the force acting on the motor.* Excluding other changes, as the internal energy of the combustion gases is increased, the specific impulse of the propellant increases, and, consequently, the resulting thrust increases. DeLaval-type nozzles are used to obtain maximum thrust from the expanding exhaust gases.

*Mathematical expressions for the flow of compressible fluids through high-velocity nozzles are presented in many reports already published. See Bibliography, page 31.

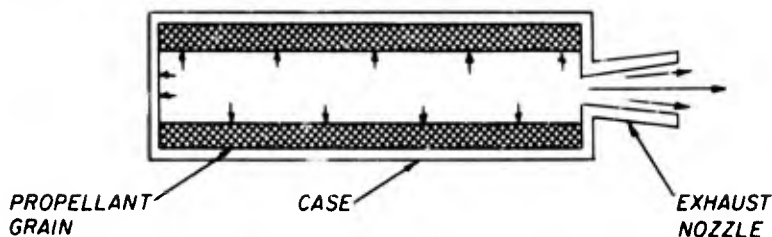


FIG. 1. Thrust Development
in a Rocket Motor.

DESIGN CONCEPTS FOR SOLID- PROPULSION UNITS

Usually the design of a solid-propulsion unit is limited by the need for satisfying dimensional, weight, burning time, and performance specifications. The designer chooses the propellant formulation, the method of supporting it in the motor case, and the configuration of the propellant grain. The first two choices lend a great deal of flexibility to the design of the propellant grain. The designer combines these variables to obtain maximum performance. The items to be designed for a unit include: (1) the solid propellant grain, (2) the motor case, (3) the exhaust nozzle, and (4) the igniter.

Propellant Grain

In selecting the propellant, primary consideration must be made of the burning rate, specific impulse, temperature sensitivity, pressure sensitivity, physical properties, and manufacturing processes. In addition, the chemical stability with age and the hazards involved in handling must also be considered.

During operation, the propellant surface is consumed in parallel layers. The velocity at which the propellant is consumed is called the linear burning rate, r , and is measured in inches per second perpendicular to the propellant surface. The linear burning rate in any one propellant type can be varied extensively by changing the number or types of ballistic modifiers or by varying the particle size of the constituents. By proper selection of the propellant, the desired thrust and impulse values can be obtained without using a complicated grain design. The control exercised over the burning rate determines, to a large extent, the reproducibility of performance of the unit.

Choice of Propellant. Propellant selection is often the result of a compromise in which specific impulse, the designer's experience with the formulation, and propellant cost per pound are factors. Generally, a propellant with high specific impulse is selected so that the maximum energy can be contained in the limiting volume of the casing. However, the density of the propellant should also be considered. For instance, a propellant with a lower specific impulse but with a higher density would yield a larger total impulse for the volume involved. Light metals such as aluminum and lithium are added to both composite and double-base propellant formulations to increase specific impulse. The maximum obtainable specific impulses of presently available propellants are in the range of 240 to 250 sec.

Propellant Grain Temperature. The initial temperature sensitivity of the propellant grain materially affects the physical properties and performance of the grain. As the temperature of the propellant rises, the operating pressure and thrust increase. Experience has shown that a higher initial temperature increases the burning rate of the propellant. The temperature sensitivity of solid propellants is expressed as:

The percent change in thrust per unit-temperature-change per pound of original thrust.

A low temperature sensitivity is desirable since it reduces the corrections which must be made to obtain the desired thrust at the ambient temperature of the test track. Operating temperature limits are dependent upon the physical properties of the propellant. At low temperatures, the plastic properties of many solid propellants become poor, and the propellant may become so brittle that cracking will occur from mechanical or thermal shock. Differences in thermal expansion of the casing and the propellant grain may cause cracking, especially in case-bonded grains. At temperatures above 140°F, some propellants become plastic and cannot withstand pressure and acceleration loads. Serious attention must be given to both storage- and operating-temperature limits to provide a unit which will perform properly under proposed service conditions.

Pressure sensitivity, or propellant burning-rate variation with pressure at any constant temperature, is approximated in many cases by

$$r = cP^n$$

where r = the burning rate

c = an experimental constant dependent upon the propellant formulation

P = the chamber pressure

n = the experimentally determined pressure exponent

The value of c varies between 0.05 and 0.002 while that of n varies between 0.1 and 0.85 in general. There are some double-base propellants for which n is zero or negative over a usable pressure range. It is desirable that n be small to decrease the variation in performance due to minor variation in the propellant surface-to-nozzle-throat area ratio, K .

Binder Characteristics. The physical properties and fluidity during casting of composite propellants are functions of the oxidizer-binder ratio and oxidizer particle size. In double-base propellants, physical strength is related to the properties of the nitrocellulose or other binders used. A desirable propellant would retain good physical properties over the temperature range of -65 to $+165^{\circ}\text{F}$ and be able to withstand cycling between those limits.

Process-Control Factors. The propellant processing methods which will be used in manufacture have to be considered in designing a unit. Use of expensive or critical supply materials should be minimal. Process-control methods should be incorporated into the manufacturing processes to assure product uniformity. Attention to process-control requirements during the design phase will result in a more reliable unit.

Propellant Storage and Handling Characteristics. The propellant should be chemically stable and relatively nonhazardous. Chemical stability is necessary for good aging characteristics so that performance will not be adversely affected by long-term storage. The propellant should be safe to handle using normal procedures and should not detonate due to mechanical or thermal shock. It should not ignite readily at low or atmospheric pressures; the exhaust products should be nontoxic and nonlingering. For most applications, there should be no smoke nor solid particles in the exhaust products.

Propellant Grain Dimensions and Types. The exact dimensions of the propellant grain depend on the configuration selected; i.e., internal burning star, rod and tube, etc. The thrust of a solid propellant unit varies with the area burning surface, A_b . By planned variation in the area of the burning surface, programming of thrust as a function of time can be achieved. Thrust programming can also be achieved by using propellants having different burning rates in the same grain.

There are three general types of propellant grain: neutral, progressive, and regressive burning. A grain designed to maintain constant burning surface during operation produces a relatively constant thrust and is termed a neutral-burning grain. If the burning area (and consequently the thrust) increases with the burning time, the grain is termed progressive; if it decreases with the burning time, the grain is termed regressive. The different types of burning can be achieved by varying the geometric proportions of the grain. In addition to geometric control, inhibiting coatings which keep certain areas of the propellant surface from burning can be used in obtaining the desired thrust-time program.

To determine the burning area necessary to produce a desired thrust:

$$F = I_{sp} \dot{w} \quad \text{and} \quad A_b = \frac{\dot{w}}{r\rho}$$

where I_{sp} , r , and ρ are determined experimentally, w = mass flow rate in lb/sec.

The nozzle throat area can be determined by

$$A_t = \frac{F}{C_F P_c} \quad \text{or} \quad A_t = \frac{\dot{w}}{C_D P_c}$$

where C_F = thrust coefficient

C_D = weight flow coefficient

Note: Exit area of the nozzle depends on the expansion ratio for the exhaust cone of the nozzle and the specific heat ratio.*

Erosive Burning. In designing the propellant grain to achieve the necessary thrust-time program, care must be exercised to avoid erosive burning, especially in internal-burning grains. Erosive burning is a result of high gas velocity and increased heat transfer along the propellant surface. This effect generally occurs near the nozzle or port end of the grain and is more pronounced in a restricted flow channel such as that of an internal burning grain during the early stages of combustion. Erosive burning requires increased consideration as the thrust-to-weight ratio is increased. High burning rate propellants are affected less by erosion than those with low burning rates. Erosion, however, effectively limits the minimum grain port area and thus determines the maximum loading density of the solid-propellant unit.

Resonant Burning. Resonant burning, or combustion instability, has been observed in internal-burning solid-propellant grains. The condition is always accompanied by dangerous high-frequency, large-amplitude oscillation in the chamber pressure and, consequently, in the thrust. During resonant burning the propellant burning rate may increase to two and one-half times its steady-state value. There are indications that resonant burning is related to the grain geometry and the physical properties of the propellant. Experiments have shown that the addition of aluminum or aluminum-oxide to double-base and composite propellants effectively reduces or eliminates resonant burning.

*The theory of gas flow through nozzles is included in some of the publications listed in the bibliography.

Motor Case

The motor case is comprised of a forward closure, a motor tube and an aft closure, and nozzle. Recent improvements in the performance of solid-propellant units have been due largely to the achievements in decreasing the weight of the motor case. The modern concept of case-bonded propellant grains permits the use of light-weight motor tubes since the propellant insulates these parts from the hot gases. Other designs require use of an insulating material or an increase in thickness of the case walls. This latter method greatly reduces performance and is used only in short-burning-time units where the wall temperatures reached are not detrimental to the unit. The interior of the aft closure is often insulated to keep weight to a minimum.

Material Stress Characteristics. In general, the cold strength of the material is used in calculating stress conditions. For design purposes, the design condition is selected wherein the stress in the material is equal to the yield stress at the maximum chamber pressure which may be expected, multiplied by a suitable factor of safety.

The criterion for material selection is the strength-to-weight ratio of the material. Steel is most often used in contemporary units, although other high strength-to-weight ratio materials such as aluminum and titanium alloys, and spun fiberglass bonded with epoxy resin have been used. By good design and improvements in materials, it is possible to achieve propellant-to-motor weight ratios or loading densities in excess of 0.90. Decreases in motor weight and increases in the loading density are usually accompanied by increasing costs. Since the nature of track tests requires relatively inexpensive propulsion units, motor weight and loading density must be equated with cost.

Exhaust Nozzle

The nozzle in current designs usually accounts for a major part of the inert weight; hence, any design that reduces the weight of the nozzle will substantially improve performance. Most contemporary solid-propellant units operate with an uncooled exhaust nozzle, the interior surfaces of which become hot from the exhaust gases. The effects of heating are most severe in the nozzle throat. In addition, the nozzle and nozzle throat are also subjected to erosion by the rapidly moving gases of many of the high-performance metallized propellants.

To counteract heating and erosion and decrease nozzle weight, a composite construction can be used in many designs. This construction features a nozzle insert or coating of carbon, or ceramic backed by an insulating material and enclosed in a thin metal casing. This composite construction is relatively expensive, however, and ideal performance characteristics may have to be sacrificed for economic reasons.

Igniter

The igniter for a solid-propellant unit is usually comprised of an electrically fired squib or initiator, the main igniter charge, and the case. The squib consists of two lead wires connected by a fine, high-resistance wire around which is a heat-sensitive combustible material called the primary charge. This charge, when ignited, must release sufficient heat to ignite the main igniter charge. A more detailed discussion of ignition parameters is presented under Ignition Characteristics, below.

The igniter case should be designed so that it will fit the allowed volume and in such a way that it can be loaded easily into the motor case. Current designs feature slanted holes in the main charge container that direct the igniter gases onto the propellant surface.

DESIGN CONCLUSIONS

The design details of a solid-propellant unit depend upon the performance requirements, the storage temperature and temperature cycling conditions it must withstand, and the field handling conditions it will encounter. For track use, reproducibility of performance and cost must be considered.

Most requirements are of a practical nature and are not subject to an accurate analytical study. The designer must balance the chemical, physical, and ballistic characteristics of the propellant with the design of the grain configuration and inert parts to attain the desired performance.

IGNITION CHARACTERISTICS

The effectiveness of a rocket depends upon the steady-state burning of the contained propellant system. The steady-state of propellant burning is always preceded by one or more transient events designated collectively as the ignition process.

The igniter of a rocket motor must perform two functions: (1) heat the propellant grain to ignition temperature and (2) increase the combustion chamber pressure to a point where the reaction of the propellant responds satisfactorily. An igniter should accomplish these functions with only milliseconds of delay and without creating undue high pressures that will subject the propellant grain to excessive forces.

Primary initiation of an igniter is electrical in most rocket motors used today because the location of the igniter in the motor usually prohibits the use of a percussion type ignition system.

One type of igniter assembly (see Fig. 2) consists of one or more electric squibs (electric matches or initiators) and an igniting charge of black powder or pyrotechnic composition housed in a container. Leads (electrical wires) from the squibs are passed from the igniter housing to an external point where they can be attached to the electrical circuit. The squib contains a heat-sensitive composition that is ignited by the heat generated by the electrical impulse that passes through the electric lead wires. The heat of the squib ignites the black powder which then ignites the propellant. The black powder may contain a booster charge of other explosive materials. When more than one squib is provided in an igniter, the squibs are wired in parallel to insure against misfires. For safety during handling, storage, and shipment, the leads are shorted to prevent accidental ignition by stray or induced current or other sources of electrical energy.

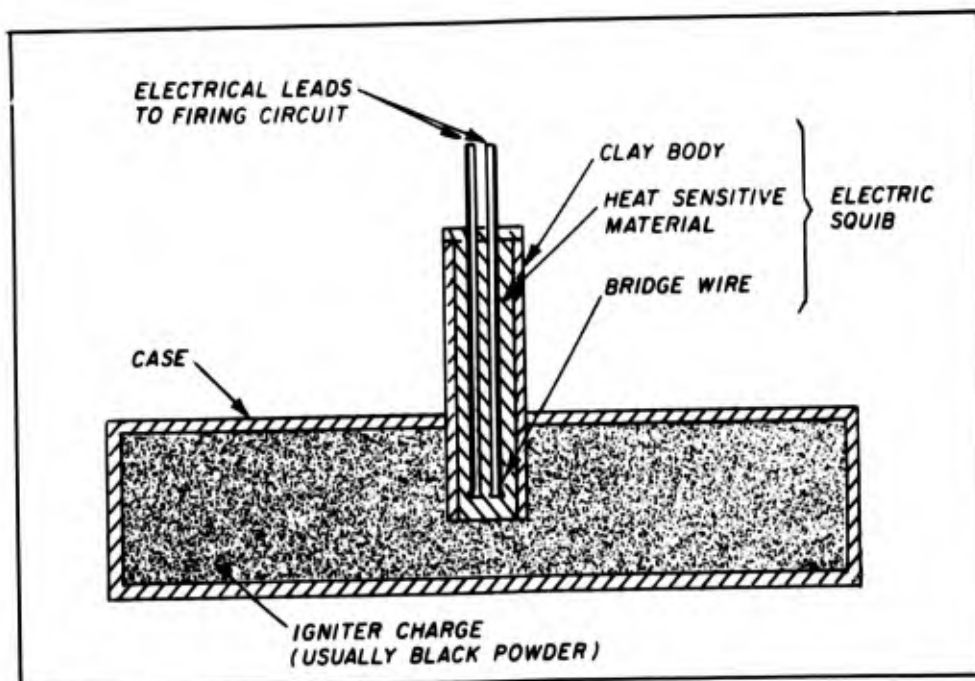


FIG. 2. A Type of Igniter Assembly.

Ignition of Solid Propellants

The igniter for many solid propellants consists of an electric match and black powder encased in a container. An electric current applied to the lead wires of the squib produces sufficient heat to ignite the sensitive electric match or squib. The flame from the electric match spreads rapidly through the powder charge in the igniter, increasing the internal pressure until the igniter case is broken open. The products of decomposition of the igniter material transfer energy to the propellant by the three mechanisms of energy transfer--conduction, convection, and radiation. The impingement of the hot, solid particles of the igniter material causes the propellant to ignite and burn in parallel layers. This burning produces hot gases and an increase in pressure that causes the propulsion of the rocket.

Igniter Types

Igniter design for use with solid-propellant rocket motors (see Fig. 3) is frequently predicated upon the shape of the grain configuration. However, their specific shapes are governed more by their location in the rocket chamber. This location is dictated by the particular requirements imposed. Most rocket designs place the igniter in the head end of the rocket so that the igniter reaction products will sweep over the entire length of the propellant grain. A few rockets use nozzle-end or internal igniters.

Head-End Igniters. Rockets using fin stabilizers have a relatively large length-to-diameter ratio. Consequently, space for an igniter is usually available at the head end of the rocket. Igniters for small caliber rockets will have a smaller outer diameter than the inner diameter of the rocket chamber. In larger rockets, the igniter may be contained in a large can located in the center of the head closure or it may be in the form of a torus or a ring. The larger units have several squibs connected in parallel to insure functioning of the igniter and to increase the effectiveness of the burning of the large quantities of igniter materials.

Nozzle-End Igniters. Rockets of the spin-stabilizer type have severe limitations on overall length; any increase in length to accommodate an igniter is prohibitive. Therefore, a nozzle-end igniter, ordinarily placed in the center of the nozzle plate with the multinozzles located in a ring around it, is generally used. By building the igniter into the nozzle plate, the need for a container, and the possibility of container fragments plugging the nozzles, are eliminated.

Nozzle-end igniters can be used in both single and multinozzled rockets.

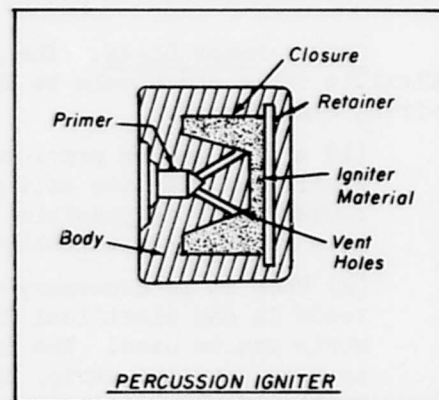
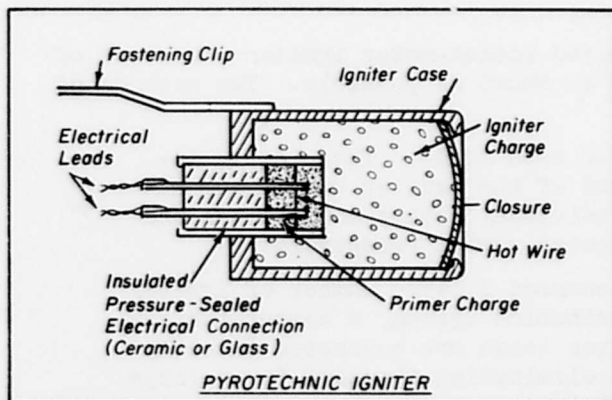
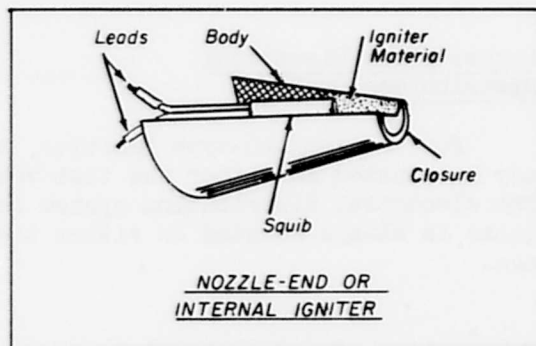
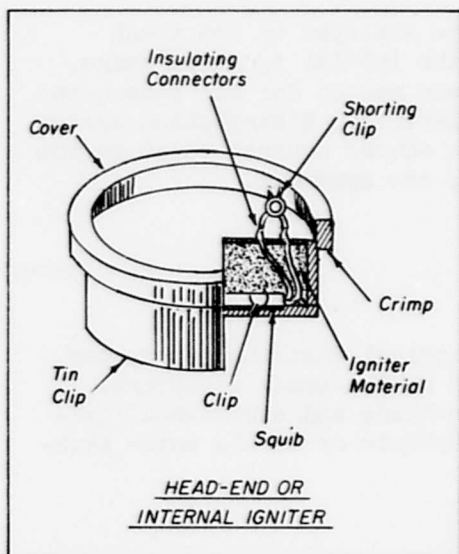
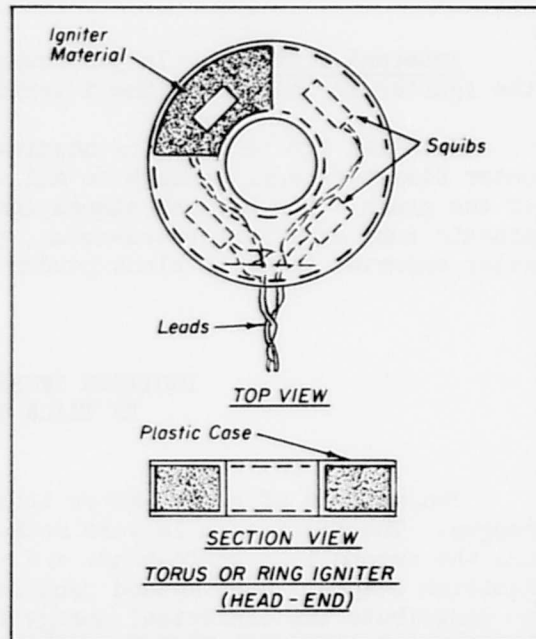
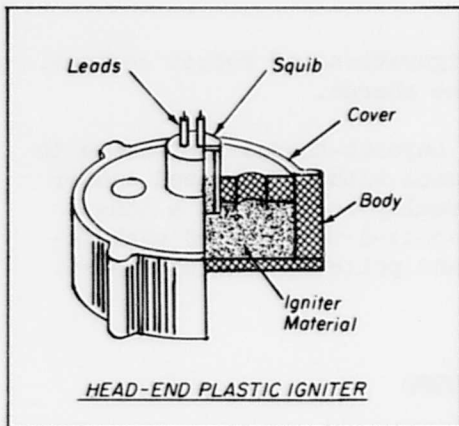


FIG. 3. Various Types of Igniter Assemblies for Solid Propellant Rocket Systems.

Internal Igniters. In particular configurations of rocket charges, the igniter is located in the interior of the charge.

Internal igniters can be nozzle-end or bayonet-shaped igniters with outer diameters small enough to allow placement within the cored center of the grain. The bayonet-shaped igniter usually consists of a long plastic tube or a perforated-metal, plastic-coated tube filled with igniter material (such as black powder or borane pellets) and initiators.

IGNITION TECHNIQUES USED IN TRACK TESTING

Two methods of rocket-motor ignition are employed on the track ranges. The first is a lanyard method for the initial ignition stage, and the second is a knife-blade and screen-box method for all subsequent ignition stages. Each method requires an electrical distribution system to distribute the electrical energy from the source contact point to the igniters of each rocket motor to be ignited, see appendix.

Rocket Motor Electrical Distribution Systems

For the lanyard-type ignition, the electrical distribution system may be mounted on either the test vehicle or on the track structure. The electrical distribution system for knife-blade and screen-box ignitions is always mounted on either the test vehicle or on the motor package.

Fabrication of a Sled-Mounted Electrical Distribution System

Rocket Motor Leads. The exposed rocket-motor igniter wires are of flexible cable and should be kept as short as possible. Two methods of wiring can be used:

- (1) a plug can be provided for each motor. The plug is installed as near the exit point of the igniter wires from the rocket motor as possible. Individual rocket-motor plugs are then routed to a suitably located common receptacle.
- (2) When it is necessary to connect a large number of igniter leads in one electrical distribution system, a common terminal strip can be used. The igniter leads are connected directly to this terminal strip, thus eliminating the need for a large number of plugs and receptacles.

The type of wiring system used in a particular vehicle will depend both on the type of rocket motors being used and on the configuration of the test vehicle.

Circuit Receptacles and Terminal Strips. Circuit receptacles and terminal strips should be fabricated to provide for connecting all the rocket motors into various firing stages. Also, provision should be made for connecting all of the motor igniters for one stage in a parallel circuit.

Knife-Blade and Screen-Box Assemblies. A pair of knife blades for each ignition stage supplies the energy source contact points for the firing of all ignition stages except the first. One end of a wire should be connected to each knife blade and the other end should be connected to the circuit receptacle or terminal strip. Rocket motor igniter leads and the knife blades for each stage should be wired as a complete parallel circuit with the two knife blades wired as the power source.

Knife-blade assembly construction is more critical than that of other ignition components since it must meet certain dimensional tolerances to enable the use of standard mounting brackets for the screen boxes. In addition, it must provide electrical insulation of the knife blades to withstand a maximum of 600 volts to ground and between the blades. The knife-blade assembly should also provide RF shielding for blade wiring and maximum blade wiring protection from windstream damage. Typical examples of a single-stage knife-blade mounting and the associated track-mounted screen boxes are shown in Figs. 4 and 5. Figure 6 shows a pair of screen boxes used for stage ignition of a high-speed monorail sled. The greater screen box length is required to insure adequate ignition current for stage ignition at sled velocities of 2,000 fps.

Lanyard Receptacle. When an initial ignition stage that requires a vehicle-mounted electrical distribution system is to be fired by using a lanyard, a lanyard receptacle must be provided on the sled. This receptacle can be designed in several ways; however, it should be located at the rear of the vehicle and it should be protected from wind and blast damage.

Conduit System. Since the wiring between the component parts must be protected from windstream damage, shielded from RF energy, and (when near the nozzle end of the rocket motors) must be protected from damage due to afterburn, it should be installed in conduit which will be grounded, and it should be adequately secured to the test vehicle system structure.

In past designs, copper or aluminum tubing, with flare type nuts for terminations, has been used for the conduit system.

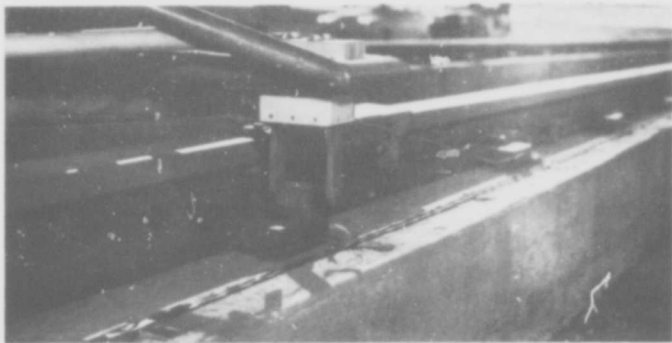


FIG. 4. Single-Stage
Knife-Blade Mounting.

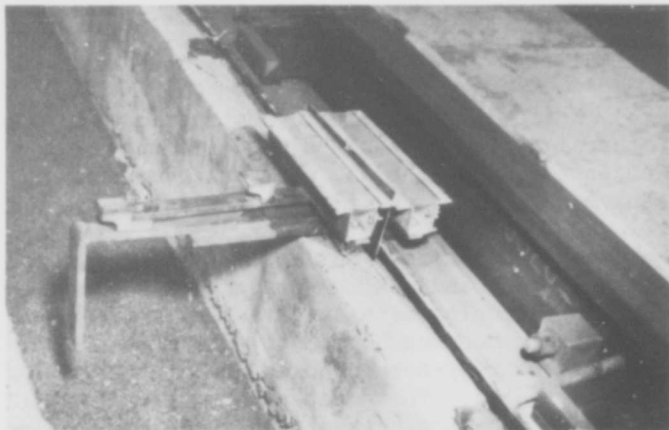


FIG. 5. Track-Mounted
Screen Box.

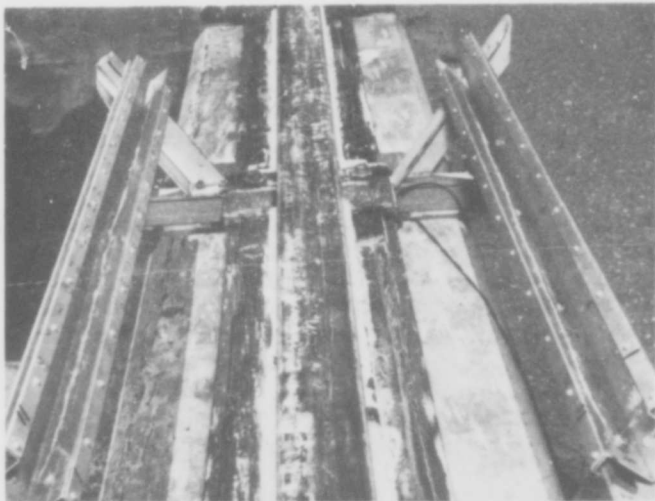


FIG. 6. Track-Mounted
Screen Box for High-
Velocity Applications.

DESIGN OF PUSHER SLED FOR USE WITH SOLID-PROPELLANT UNITS

MOUNTING PROVISIONS

Most solid-propulsion units used on the tracks today were designed initially for uses that did not subject the motor grain and case to the severe dynamic conditions encountered on test tracks. Since the units, themselves, cannot be readily modified for track use, it has been necessary to design test vehicles to accommodate the propulsion units as they already exist.

In most instances, this adaptation has entailed development of mounting fixtures to fit the mounting provisions already on the motor. Usually a bolted fitting is installed at the forward end of the motor case and an additional tie-down support is installed at the aft end. On some of the larger units it has been found necessary to provide additional support at intermediate points on the case.

Mountings on solid propellant motors are usually an integral part of the motor casing. Therefore, thrust beam and mounting fixtures must be constructed so that no undue stresses are transmitted back into the motor casing during the thrust phase of the test.

To save manhours expended and time during which test vehicles are restricted due to rocket motor loading operations, the vehicle design should allow for easy, expeditious loading. This can best be accomplished by making use (whenever consistent with structural integrity) of an open structure with detachable end fittings, using pinned rather than bolted connections.

IGNITER CONSIDERATIONS

Generally, solid-propulsion units are loaded on the test vehicle just before a test. In the interest of personnel safety, it is highly desirable to have the motors nonpropulsive until just before firing. This can best be accomplished by keeping the igniter out of the motor until all the prerun instrumentation checks have been completed, and until all personnel have been cleared from the track and vehicle area. Vehicle structural members should be so designed that the igniters can be inserted and tightened after the motor has been fastened into position.

On units such as the HVAR and LOKI, the igniter is an integral part of the motor. When handling these units, additional care must be exercised to eliminate all extraneous inputs which could fire the motor. The vehicle designer can help insure personnel safety by designing the motor supports so that it takes a minimum of time to load such units.

WIRING PROVISIONS

When building a pusher vehicle to accommodate solid propellant motors, the wiring should be so designed that it can be checked for continuity and high-voltage leaks from the knife blades to terminal boards, etc. before the motors are loaded. Terminal blocks should be located so that wiring can be done with a minimum length of lead to motors.

It is frequently necessary to fire several motors simultaneously at some downrange point after the vehicle has attained considerable velocity. Provisions for completing a firing circuit through a screen-box, pylon-knife pair are generally limited in number so that a single knife pair must be made to serve several motors through a parallel circuit. This must be well balanced to preclude the possibility of the igniter from one motor draining the current from other units in the circuit, thus causing some motors to fail to fire.

Most pusher type vehicles using solid-propellant motors are of an open-beam construction. Wiring from knife pylons to terminal boards and motors should be routed through conduit and be well anchored so that wind blast or foreign objects will not damage the wiring.

On some units, such as the LOKI, HVAR, and Vipers, the wiring leads from the igniters are routed out the aft end of the motor. When these motors are to be fired at a downrange station, the wiring leads must be protected so that hot exhaust gases from motors operating in close proximity do not damage the leads. When such units are used for first stage ignition, the igniter leads should be carefully secured to protect them against any outside forces that might break the connection and preclude firing of the motor.

DEVELOPMENT OF LIQUID PROPULSION SYSTEMS

Liquid propulsion was considered for use in track testing with the idea of lowering operating costs. Existing systems were studied for possible track test adaptation. The North American liquid oxygen-alcohol system was the first successful liquid-propelled rocket engine used in this application. It had a burning time of 4.5 sec and a thrust of 50,000 lb. A lox-alcohol engine having a thrust of 150,000 lb and suf-

efficient tankage for a 10-sec burning time was also adapted for track usage. A third (and most used) system is the Bomarc "A" launch engine using fuming nitric acid as the oxidizer and a mixture of unsymmetrical dimethylhydrazine and JP-4 jet fuel. This system can be used in sleds with one to three thrust chambers. Each chamber normally delivers a 35,000-lb thrust and some have been modified to provide a 42,000-lb thrust. The three-chamber sled, with a maximum burning time of 7.5 sec, has been used in the greatest number of tests.

In their seven years of use, liquid propulsion systems have proved to be very useful in track testing although several problems have been encountered. The severe environment of acceleration, deceleration, and track-induced vibration requires extensive maintenance. Converting engines designed for a single firing to be used in repeated firings is also difficult. In addition, more extensive ground-support equipment was needed for maintenance and firing than was anticipated. However, the liquid-propelled engine has proved its worth, especially in guidance system testing. Over 250 tests have been completed at the AFMDC track in which payloads of 3,000 to 5,000 lb have been accelerated to an average of 1,600 fps with variations in thrust to simulate the free-flight environments encountered by missile guidance systems.

There has been a definite cost advantage in the use of liquid-propelled sleds for some track tests; however, solid-propulsion systems are still more satisfactory for use in certain other types of track testing. Besides cost, there are other advantages in using liquid propulsion. The three-chamber sleds using Bomarc engines can be adjusted to provide a total thrust of 96,000 to 111,000 lb. The three chambers can be shut down individually to give a very low acceleration rate or to provide the sustain portion in a test profile. (One limitation is that the thrust chambers will not operate under negative g forces for longer than 0.3 sec.) Another advantage is that the engine acceleration at ignition is approximately 7 to 8 g . This is lower than can be obtained with most solid-propulsion systems pushing the same payload to a final velocity of 1,500 to 1,600 fps. This lower acceleration rate and steady thrust are very useful in testing components used in large missiles that also have a relatively slow acceleration rate. Staging of solid-propellant rocket motors is undesirable in this type of testing since high-acceleration spikes are encountered during staged firings of the solid-propulsion motors.

DESIGN OF THE LIQUID PROPELLANT SYSTEM

Liquid-propelled rocket engine thrust chambers of conventional design, except for the addition of drain taps to remove propellants trapped in the propellant manifold due to the horizontal position, are used for track testing. Valves and other propellant flow-control devices are similar to those used in missile systems. The primary difference in the

overall configuration is that propellant transfer is achieved by cold gas rather than by turbine pump systems. This entails using rather large, heavy sledborne tanks to store the gas used in propellant transfer from tankage to thrust chamber. The tankage is constructed to form a structure between the front and rear sled support beams. No other beams are used to supplement the tank structure. In addition to gas tanks, propellant tanks, thrust-chamber and control systems, a momentum-exchange water brake is incorporated in the sled.

The liquid-propelled rocket engine (or pusher sled) seldom carries instrumentation or test items. These items are usually housed in a forward sled called a forebody. The forebody has two slipper beams, aircraft-type box construction, and is wedge-shaped to reduce air drag. The forebody and liquid rocket engine are fastened together with a universal-type coupling placed 7.79 in. above the top of the rail. Explosive bolts are often used to separate the forebody and pusher during the braking phase.

Basic liquid propulsion system design is described in publications listed in the Bibliography, page 31 of this report. The following paragraphs cover the use of liquid propulsion systems as they apply to track testing.

Three-Chamber Liquid Engine Test Preparation and Firing Sequence

1. Component Check and Fueling. The engine is placed on the track and component checks are made. Component checkout is concerned mainly with the gas regulation, firing circuit, and other electrical systems. Fueling usually requires an hour. The tank is filled with the oxidizer (fuming nitric acid) to a point where overflow is obtained. This overflow point is determined by use of a preselected ullage tube which allows the tank to be filled to a selected volume. The fuel (JFX) is transferred in a similar manner. Pressure transfer has proved to be the most reliable fueling method. With the three-chamber system, a special starter fuel--a mixture of aniline and furfuryl alcohol--is used. It is placed in the propellant lines on the engine and held in place by rupture disks. Used with the oxidizer, the starter fuel provides a smoother takeoff.

2. Engine Pressurization. At 20 minutes before firing, the launch pad is cleared and engine pressurization with nitrogen gas is begun. The pressurization rate is 200 psi/min and is continued until a sled pressure of 2,400 psi is indicated. The flow rate and tank pressures are monitored in the blockhouse.

3. Firing Switches. After pressurization is complete, the engine is ready to fire. The firing panel has two main switches used in firing--the load-run switch and the fire switch. The load-run switch is thrown

at firing time minus 10 sec. This switch arms the fire switch and traps the gas on board the sled by closing the gas vent valves.

4. Gas Regulator. When the fire switch is thrown, the main gas regulator begins operation and allows the gas stored at 2,400 psi to flow into the propellant tanks. Regulator settings provide the desired thrust. Pressures can be set to provide a 32,000- to 37,000-lb thrust per chamber. If all pressure and electrical systems function normally, the engine will fire from 5 to 6 sec after the fire switch is thrown. This time delay is required to allow adequate pressure to build up in the main propellant tanks.

5. Safety Factors. Even after a satisfactory firing sequence, the engine will still shut down if there is an electrical failure or if thrust-chamber or tank pressures fall out of operating limits. This safety factor, plus the prefiring sequence, has been instrumental in preventing catastrophic failures. In over 250 firings at the AFMDC track no damage due to engine failure has been sustained by the test item.

6. Shutdown Operation. When the engine has fired and moved down-track to a selected shutdown point (there may be more than one shutdown point during a test), electrical circuits on the sled are interrupted to cause shutdown of one, two, or three chambers. These interruptions are obtained by physically cutting exposed wires mounted on the sled. The wires are cut by steel knives mounted on brackets and attached to the track rail. After the wires are cut, electrical power is removed from the sled, propellant valves close, and the gas vent valves open. This sequence also takes place at any point where a malfunction occurs during the firing.

7. Engine Recovery. Engine recovery operations include removal of the gas pressurization line common to both propellant tanks, and defueling before removal from the track. Engine operation is monitored by on-board tape records which record the sequence of electrical events, and propellant tank and chamber pressures. With this information, it is possible to check actual engine performance against predicted performance. In the event of a malfunction, this information is invaluable in locating the cause of failure.

LIQUID PROPULSION SYSTEM PERFORMANCE

Figures 7 and 8 are graphs showing actual liquid engine performance. In Fig. 7, a 4,800 lb forebody was used. A profile of near zero acceleration for approximately 2 sec was required. This was accomplished by accelerating with all three thrust chambers, terminating thrust on two, and sustaining with one for 2 sec. Results of this thrust profile are shown on the graph in two plateaus--one for velocity and one for acceleration. The second drop in acceleration is due to thrust termination of

the third chamber. The third deflection in the acceleration curve at approximately 36 sec is due to water-brake entry.

Figure 8 is a more typical thrust profile. In this profile the engine provides thrust with all three chambers terminated at 0.2-sec intervals. This staggered termination technique is used to temper the rapid changes in acceleration. Again the velocity and acceleration are changed rapidly during water braking.

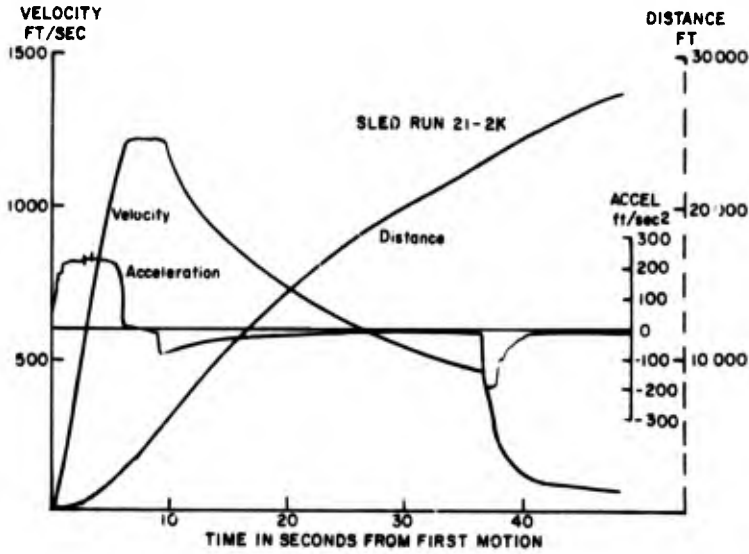


FIG. 7. Thrust Profile Using 4,800-lb Forebody.

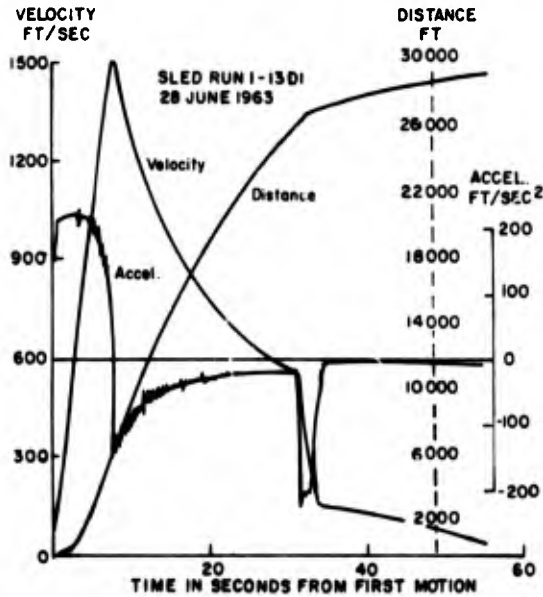


FIG. 8. Typical Thrust Profile Showing Results of Staggered Termination Technique.

FUTURE LIQUID PROPULSION SYSTEMS

A liquid propelled rocket engine with a maximum thrust of 300,000 lb is presently under construction. It consists of two thrust chambers using nitrogen tetroxide and unsymmetrical dimethylhydrazine as propellants. The basic design is essentially the same as that of previously used liquid-propelled rocket engines, only on a larger scale. The engines will have a variable-thrust control with in-flight ignition capabilities, and it will be possible to program thrust varying from 50,000 to 300,000 lb during the sled flight. Burning time under maximum thrust is 10 sec. Longer burning time can be obtained by lowering the thrust settings. Maximum velocity available is predicted to range from 2,200 to 2,500 fps depending upon payload.

Future plans for liquid propulsion systems using existing track facilities include an attempt to improve the fuel feed system. The thrust and burning time of sleds that can be used on present tracks do not warrant conversion to turbine pump systems. Pressurization of propellant tanks by internal mixing of propellants and use of resulting gas for pressurization is being considered.

Another area of upgrading track propulsion capabilities would be to duct-ram air to the hot gases from the thrust chamber to achieve higher efficiency. Structures, tankage, and propulsion capabilities available at the present time point toward an upper velocity limit of approximately 3,000 fps. Performance can also be improved with the use of more exotic propellants such as liquid oxygen and liquid hydrogen, and lighter propulsion systems.

GLOSSARY

- Action Time*** Particular time interval that starts at a point corresponding with 10% of the maximum rocket thrust and ends at a corresponding point where the rocket thrust has decreased to 10% of the same maximum value.
- Afterburn*** A characteristic burning that takes place in a rocket thrust chamber of certain solid propellant or liquid propellant rocket engines. It is a low grade combustion that takes place after the main propellant burning is completed.
- Air Brake** A mechanism on a rocket sled which utilizes the atmosphere as a braking source.
- Auto-Ignition** Spontaneous ignition.
- Auto-Ignition Temperature** That temperature at which spontaneous ignition will occur strictly because of temperature environment.
- Average Thrust*** The value of thrust developed by a rocket engine between a specified starting and a specified shut down point of the total operational duration. Average thrust equals the area between the designated thrust ordinates under the thrust duration curve divided by the time increment between the same ordinates.
- Blow-Out Disc*** A disc installed in the wall of a solid propellant-rocket thrust chamber for the purpose of releasing excessive pressure that might occur during combustion of the propellant charge.
- Burning Time*** The length of time from the instant the thrust of a solid propellant rocket engine has risen to 10% of its maximum value to a point where the thrust begins to drop sharply near the end of the firing operation.
- Burn-Out*** The point at which the chamber pressure of an ignited rocket drops because all of the usable propellant has been consumed.

*Herrick, John W., Eric Burgess, and Wayne Langford. Rocket Encyclopedia Illustrated, Los Angeles, California, Aero Publishers, Inc., 1959.

Burst Diaphragm	A disc installed across the propellant flow lines to serve as a closed valve until pressure increases sufficiently during the starting phase to burst the diaphragm.
Burst Diaphragm Assembly	A pressure container accessory which houses a burst diaphragm.
Captive Pusher	A pusher vehicle which is fastened to a test vehicle.
Chuffing or Chugging	Pulsating thrust resulting from uneven burning or consumption of residual slivers in a rocket combustion chamber.
Cluster*	Popular name for a combination of rocket engines that are mounted together in a closely assembled unit.
Combustion	Exothermic chemical reaction.
Compatibility	Acceptability for adjacent storage.
Cryogenic	Pertaining to temperatures lower than those normally associated with refrigeration.
Detonation	<ol style="list-style-type: none"> 1. The explosive destruction of the rocket engine and other rocket combustion components. 2. A hard start during the ignition of a liquid propellant rocket engine. 3. A hard shut down during the termination of engine operation.
Drag	The force retarding the sled, usually measured in pounds.
Drag Curve	A plot showing the magnitude of drag throughout the motion of the rocket sled.
Engine	A sled unit or propulsion type which propels the sled system.
Forebody	The structure that contains the test item as opposed to structure containing the propulsion.

*Herrick, John W., Eric Burgess, and Wayne Langford. Rocket Encyclopedia Illustrated, Los Angeles, California, Aero Publishers, Inc., 1959.

Frangible Dam	A frangible piece of material installed in the trough of the track to contain or retard the water used for the rocket sled braking system.
Free Pusher	A pusher vehicle which is not fastened to vehicles which it is pushing and which can separate when the boost phase is completed.
Frontal Area	A representative area presented to the air by the rocket sled.
Fuel*	That material which, in a reaction with an oxidizer, yields thermal energy in a rocket thrust chamber.
Hanger	The structure attached to the rocket sled to which the shoe is fastened.
Head Cap	The adapter on the forward end of the rocket motor attaching it to the rocket sled. Also known as closure cap.
Horizontal Momentum Exchange Brake	A piece of hardware hanging from the rocket sled which picks up water in a trough and changes its direction with a resulting change of momentum causing the sled to stop.
Hygroscopic	Ability to absorb moisture from air.
Hypergolic*	Characteristic of substances being mutually chemically reactive. The ability of a liquid fuel and liquid oxidizer, which react when brought in contact with each other, to achieve ignition temperature without any outside assistance.
Igniter*	The complete device for establishing ignition in the thrust chamber assembly, the liquid-reactant gas generator, the solid-reactant gas generator, and any other device of a typical rocket power plant.
Ignition Interval or Ignition Time	The increment of time, following initiation, required for the squib of the rocket igniter to function.

*Herrick, John W., Eric Burgess, and Wayne Langford. Rocket Encyclopedia Illustrated, Los Angeles, California, Aero Publishers, Inc., 1959.

Impulse (Specific)*	The numerical ratio obtained by dividing the total thrust by the total flow rate of all the propellants consumed in producing this thrust.
Impulse (Total)	A characteristic of a rocket motor determined by integrating the thrust time curve, usually expressed in lb-seconds.
Inert Rocket	A rocket which in itself contains no propulsive or explosive elements.
Injector*	The component of a rocket propulsion unit that injects one or more of the propellants or a monopropellant into a combustion chamber or thrust chamber.
Intermediate Pusher	A pusher vehicle staged second (or later) in a two or more pusher vehicle train.
JATO	Jet Assisted Take-Off, commonly applied to all solid propellant rockets in supply nomenclature.
JPX	A fuel used in propulsion systems of some liquid engines, consisting of a mixture of JP-4 jet fuel and unsymmetrical dimethylhydrazine.
Knife Blades	The hardware on the sled that passes through the screen box to pick up electrical energy.
LOX	Liquid oxygen used as an oxidizer for propulsion of some types of liquid engines.
Magnet Bracket	A structure on the rocket sled upon which the magnet is mounted.
Mass Ratio*	A term applied to a complete rocket vehicle; the numerical ratio between the launching mass fully loaded with propellants and the final mass when all the propellants have been consumed or ejected.
Motor	Term sometimes heard in rocket work when reference is made to a rocket motor.
Nitric Acid	Oxidizer used in combustion of JPX for propulsion of some types of liquid engines.

* Herrick, John W., Eric Burgess, and Wayne Langford. Rocket Encyclopedia Illustrated, Los Angeles, California, Aero Publishers, Inc., 1959.

Onset	The slope of the acceleration curve. Usually expressed in g's per second.
Oxidizer	Solid, liquid or gas which contains oxygen in molecular or a combination form that can react with a fuel to liberate thermo chemical energy by an exothermic chemical reaction.
Probe Brake	A structure attached to the underside of the rocket sled that drags through water or sand, thereby decelerating the rocket sled.
Pusher Pad	The part of a pusher vehicle which makes contact with the pushed vehicle.
Pusher Vehicle	A vehicle which imparts momentum to another vehicle or group of vehicles.
Push Receiver Pad	The part of a vehicle which is designed to receive the applied load from a pusher.
Residual Propellant*	The liquid or the solid propellant remaining in a rocket engine or rocket power plant after the firing operation has terminated.
Retro Rocket*	A rocket which applies thrust in a direction opposite to motion causing deceleration of a rocket sled.
Rise Time	The increment of time required for the rocket motor to reach its maximum thrust level. Also used to describe the period of time during which a rate of onset occurs.
Rocket Motor Index	Ratio of total impulse to the total rocket firing weight for a solid propellant rocket. (This is also referred to as Rocket Performance Index (RPI)).
Rocket Sled	A land vehicle that rides or slides on a specially constructed railroad type track under the thrust of a rocket power plant.
Rupture	Pressure damage that occurs at a rate or magnitude below what is generally considered the explosive level.

* Herrick, John W., Eric Burgess. and Wayne Langford. Rocket Encyclopedia Illustrated, Los Angeles, California, Aero Publishers, Inc., 1959.

Screen Box	A source of capacitive electric power located at trackside for energizing a passing rocket sled electrical system. Can be used to ignite down range propulsion.
Shoe, Slipper, Skid*	A mounting device for connecting a rocket sled to the tracks on rails of a sled track so that the vehicle will hold to these rails during a test run.
Shut-Down Knife	A static knife installation located so that it will cut an ignition lead on the liquid propellant engine of a rocket sled so that engine shut-down is initiated. Usually used as a back-up or safety device.
Shut-Down Wire	An electrical lead on a liquid propellant rocket engine, which, if severed, will initiate shut-down. (See Shut-Down Knife).
Slug*	A coined name in rocketry, used in connection with the starting of a liquid propellant rocket engine by means of injecting a small volume of an auxiliary or third propellant ahead of the main propellant of the same category; a fuel with a fuel or an oxidizer with an oxidizer.
Specific Impulse	The ratio of the total impulse of the system to the total weight of propellant.
Squib Igniter*	A pyrotechnic igniter that is used to ignite a solid or liquid propellant rocket engine.
Stage	A group of rockets fired simultaneously from a common ignition harness.
Staging	The act of igniting rockets downrange.
Test Vehicle	A rocket propelled vehicle which is used to test rocket engines, test components of a system or a system configuration.
Thrust	The lb force exerted on the thrust bulkhead of the rocket sled by a rocket motor.
Thrust Bulkhead	The structure upon which the thrust of the rocket motors is applied.

*Herrick, John W., Eric Burgess, and Wayne Langford. Rocket Encyclopedia Illustrated, Los Angeles, California, Aero Publishers, Inc., 1959.

Thrust Chamber	That part of a rocket engine in which combustions take place. The bolted, welded, or otherwise joined single assembly of a combustion chamber, divergent expansion nozzle, liquid propellant injector assembly, thrust chamber propellant valves and any other attached part that is required for injecting bi-propellants of a mono-propellant into the combustion chamber.
Thrust-Curve	The profile of the thrust magnitude throughout the burn time.
Towing Ring	An attachment to permit the vehicle to be pulled along the track by a truck on a trackside road.
Track Gauge	The distance between rails measured from the inside of the railheads.
Train	A test vehicle and one or more pusher vehicles lined up for firing.
Trough	That configuration of the track supporting structure between the rails.
Ullage	The volume of a container which remains empty when its capacity is less than its maximum volume.
Vacuum Velocity	The velocity the rocket or system would achieve in a vacuum normally calculated by the formula velocity - total impulse of the system X the gravity constant divided by the average weight of the system.
Wheelbarrow	A single axled vehicle whose aft vertical support is derived from attachment to a captive pusher vehicle. This term is also used synonymously with some types of forebodies.

* Herrick, John W., Eric Burgess, and Wayne Langford. Rocket Encyclopedia Illustrated, Los Angeles, California, Aero Publishers, Inc., 1959.

Appendix

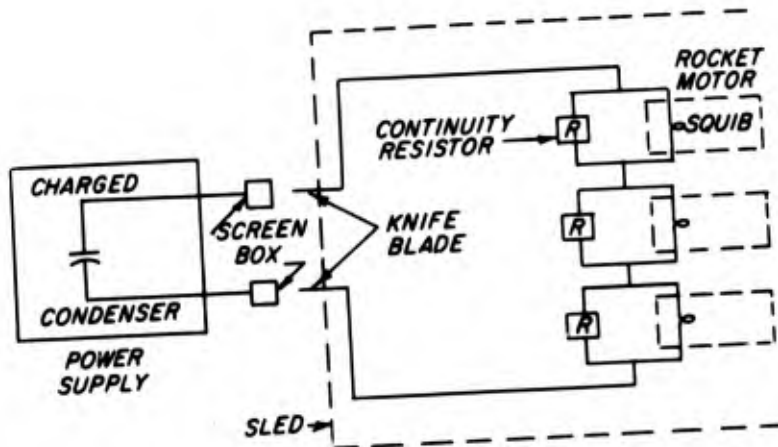
STAGED IGNITION OF SLEDS AT SNORT*

by

Howard S. Olson

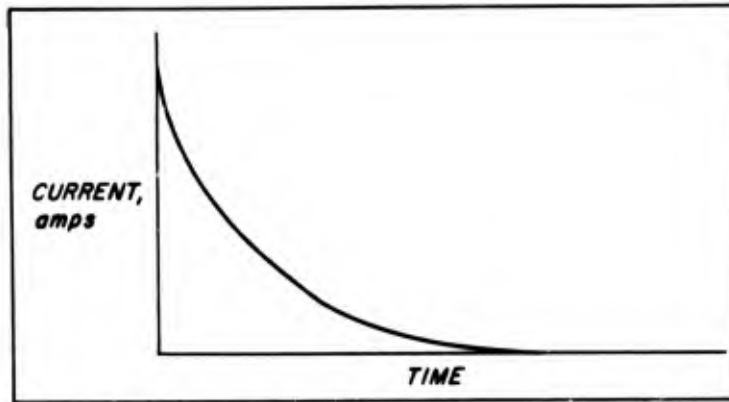
Firing rocket motors on a track sled in successive stages is a standard means of achieving sled velocity increases. Once the first stage has been fired and the sled is underway, the next stage is fired at some predetermined spot along the track at which it is known that the sled will have reached the maximum velocity obtainable from the previous stage. Our technique for firing the stage of motors is to connect a power supply (consisting of a charged condenser) to a pair of screen boxes fastened beside the track in such a way that knife-blade contacts on the sled complete the firing circuit to ignite the motors as the blades pass through the boxes.

A typical circuit setup for firing a stage consisting of three rocket motors is shown below. The rocket motor squibs are connected in series between the knife blades, with a "continuity" resistor in parallel with each squib. The "continuity" resistor provides a continuous circuit should one or more rocket motor squibs fire faster than the others. As the knife blades pass through the screen boxes, the condenser discharges into the resistance network and fires the rocket motor.

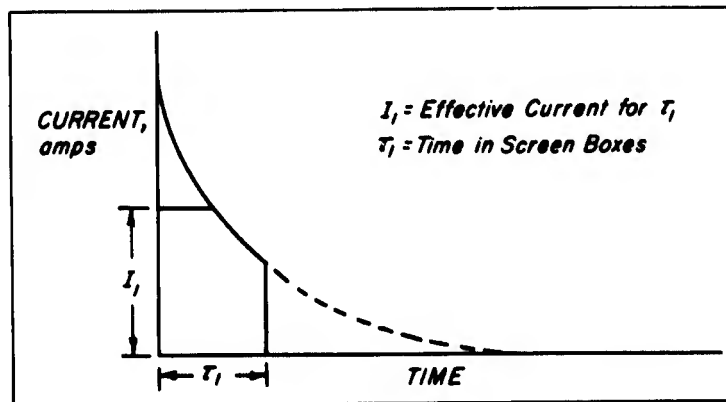


*The material under this title is a reproduction of an informal U. S. Naval Ordnance Test Station report dated 5 September 1957.

Following is a typical curve of the discharge of a condenser into a resistance network.



The discharge curve during the time the knife blades are in the screen boxes is:



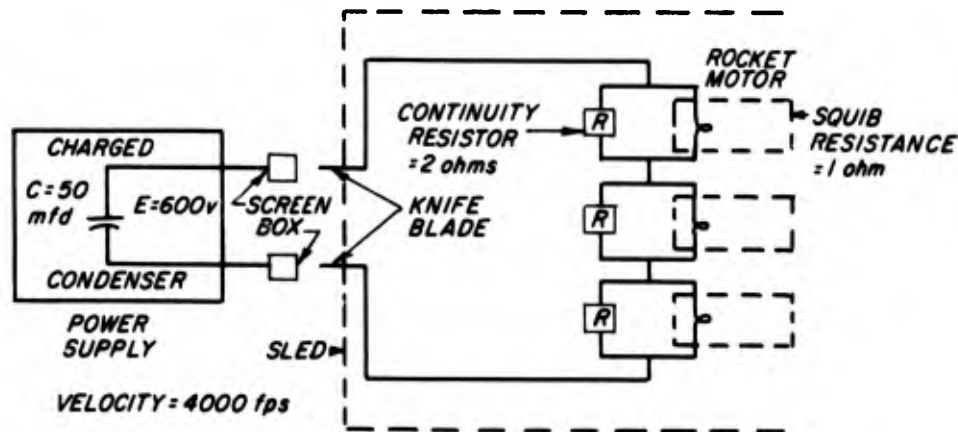
The effective current is the current which, if applied to the resistance network for the same time as the actual current, will produce the same heat in the resistance network. (It is similar to an "average" value.)

After applying current to a squib, it takes a definite time for the squib to become hot enough to ignite the surrounding powder. During this time, a certain amount of energy is used by the squib. Therefore, in firing staged rocket motors, we must make sure that each rocket motor squib receives enough energy to ignite the surrounding powder.

Our problem is

- a. How much energy do we need for each rocket motor squib?
- b. How much energy is wasted in each continuity resistor?
- c. How big a power supply will be needed to furnish enough energy for all rocket motor squibs and continuity resistors during the time the knife blades are in the screen boxes?

The solution to our problem is best illustrated by an example. Assume we have the staged rocket motor setup as shown below. The sled velocity (when the knife blades are in the screen boxes) is 4000 feet per second.



The first step is to calculate the energy needed for each rocket motor. For any rocket motor the following information is always given:

- a. Minimum firing current (I)
- b. Squib resistance (R_s)
- c. Ignition delay time (t)

For our example assume:

$$I = 3 \text{ amps, } R_s = 1 \text{ ohm, and } t = .030 \text{ seconds}$$

The energy required to fire this rocket motor squib is

$$W = I^2 R_s t$$

$$\text{Energy} = (\text{Current})^2 \times \text{Squib Resistance} \times \text{Time}$$

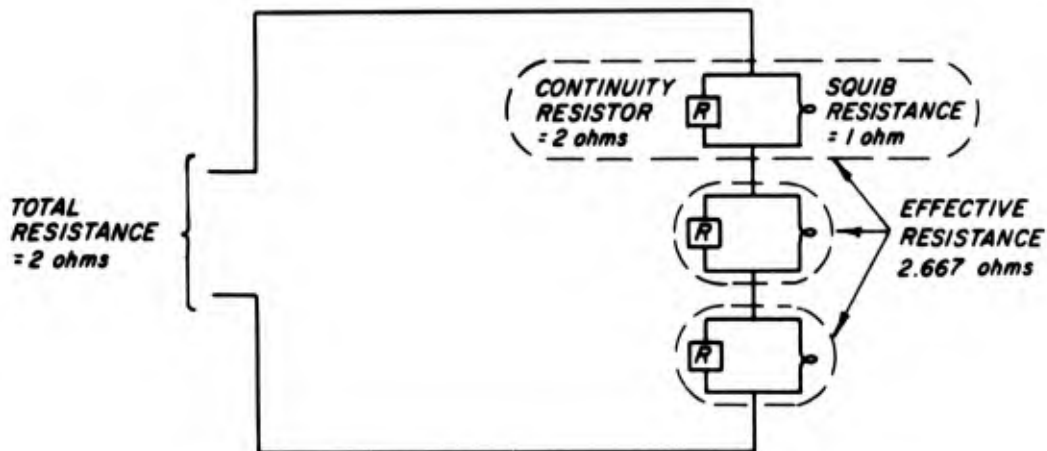
$$\text{watt-seconds} = (\text{amperes})^2 \times \text{ohms} \times \text{seconds}$$

Substituting we have

$$W = 3^2 \times 1 \times .03$$

$$W = .27 \text{ watt-seconds}^*$$

Now that we know how much energy we need to fire each rocket motor, our next step is to calculate the total resistance of the circuit between the knife blades into which the condenser is discharging (see sketch).



The effective resistance of each rocket motor and its continuity resistor can be found from the formula:

$$R_{\text{eff}} = \frac{R \times R_s}{R + R_s}$$

*This conservative value is explained in Addendum A.

effective resistance =

$$\frac{\text{resistance of continuity resistor X squib resistance}}{\text{resistance of continuity resistor + squib resistance}}$$

$$\text{ohms} = \frac{\text{ohms X ohms}}{\text{ohms} + \text{ohms}}$$

$$R_{\text{eff}} = \frac{2 \times 1}{2 + 1} = \frac{2}{3}$$

$$R_{\text{eff}} = .667 \text{ ohms}$$

Since each rocket motor squib and its continuity resistor is in series with the others, the total resistance between the knife blades is the sum of the effective resistance for each rocket motor squib and its continuity resistor.

$$R_T = R_{\text{eff}_1} + R_{\text{eff}_2} + R_{\text{eff}_3}$$

total resistance =
 effective resistance of rocket motor 1 and its continuity resistor + effective resistance of rocket motor 2 and its continuity resistor, etc. Thus
 ohms = ohms + ohms + ohms.

The total resistance for our problem

$$R_T = .667 + .667 + .667$$

$$R_T = 2 \text{ ohms}$$

The time the knife blades on the sled are in the screen boxes can be determined from the velocity of the sled (when passing through the screen boxes) and the length of screen boxes.

$$t = \frac{L}{V}$$

$$\text{Time} = \frac{\text{length of screen box}}{\text{velocity}}$$

$$\text{Seconds} = \frac{\text{inches}}{\text{inches per second}}$$

Our screen boxes are 5.5 inches long, therefore

$$t = \frac{5.5}{4000 \times 12}$$

$$t = .000115 \text{ seconds}$$

To find the total effective current of the condenser discharge into the rocket motor squibs and continuity resistors we substitute values in this formula:*

$$I_{\text{eff}} = \sqrt{\frac{E^2 C}{2Rt} \left[1 - e^{-\frac{2t}{RC}} \right]}$$

Effective current =

$$\sqrt{\frac{(\text{Voltage})^2 \times \text{Capacitance}}{2 \times \text{Total Resistance} \times \text{Time}} \left(1 - 2.72 \frac{-2 \times \text{Time}}{\text{Total Resistance} \times \text{Capacitance}} \right)}$$

$$\text{Amperes} = \sqrt{\frac{(\text{Volts})^2 \times \text{Farads}}{2 \times \text{Ohms} \times \text{Seconds}} \left(1 - 2.72 \frac{-2 \times \text{Seconds}}{\text{Ohms} \times \text{Farads}} \right)}$$

$$I_{\text{eff}} = \sqrt{\frac{(600)^2 \times .00005}{2 \times 2 \times .000115} \left(1 - 2.72 \frac{-2 \times .000115}{2 \times .00005} \right)}$$

$$I_{\text{eff}} = \sqrt{39,100 (1-.1)} = \sqrt{39,100 \times .9} = \sqrt{35,200}$$

$$I_{\text{eff}} = 188 \text{ amperes}$$

Because the rocket motor squib and the continuity resistor are in parallel this current divides between them. The current through the rocket motor squib is calculated from

* See derivation of formula in Addendum B.

$$I_s = I_t \times \frac{R_{\text{eff}}}{R_s}$$

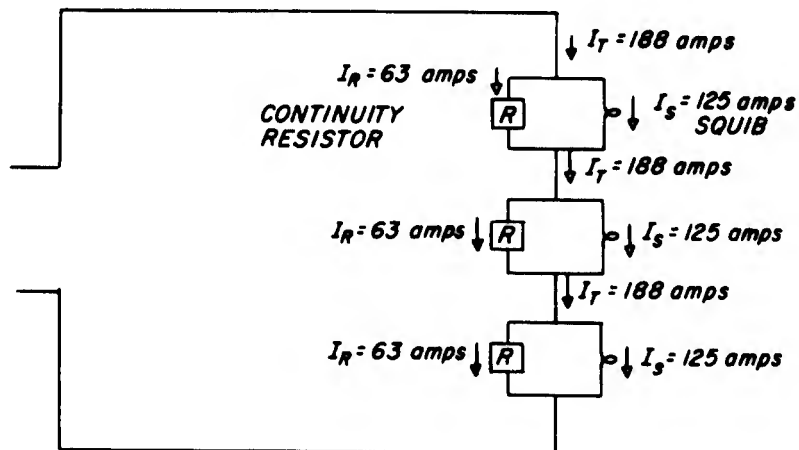
$$\text{squib current} = \text{total current} \times \frac{\text{effective resistance}}{\text{squib resistance}}$$

$$\text{amperes} = \text{amperes} \times \frac{\text{ohms}}{\text{ohms}}$$

$$I_s = 188 \times \frac{.667}{1}$$

$$I = 125 \text{ amperes}$$

The rest of the current must go through the continuity resistor as shown below (188-125 = 63 amperes).



We can now calculate the energy used in the rocket motor squib using the formula mentioned previously.

$$W = I^2 R_s t$$

$$W = 125^2 \times 1 \times .000115$$

$$W = 1.8 \text{ watt-seconds}$$

We then have .27 watt-seconds required to fire each rocket motor and we are supplying 1.8 watt-seconds for each rocket motor. Safety factor = $\frac{1.8}{.27} = 6.6$. We always have a safety factor of 3 or more. If the safety factor is less than 3 we can vary either one or all of the items below and repeat the calculations:

Increase time in the screen boxes by increasing the length of the screen boxes.

Increase condenser capacity (C) by putting another power supply in parallel.

Increase condenser voltage (E) by charging the condenser to a higher value.

As you can see these calculations are very time consuming. Therefore this paper includes several nomographs that greatly reduce the time of calculations.

To solve this same problem by the nomographs, find the time in the screen boxes from graph (1). For 4000 feet per second the time is 115 microseconds. Since we have tentatively selected one 600 volt screen box power supply, we look at nomograph #3 to see how much energy is transferred to the resistance network. The resistance in ohms is the total resistance between the knife blades (i.e., 2 ohms). From this the total energy used by the resistance network is 8.1 watt-seconds. Since we have three identical setups (rocket motor squibs and continuity resistors) each setup takes 1/3 of the total energy, or 2.7 watt-seconds.

Using nomograph #5 we can find out what percent of this energy is available for the rocket motor squib. Sixty-six percent is available for the squib or $.66 \times 2.7 = 1.78$ watt-seconds which is very close to what we got by the long method.

The procedure for determining the power supply for a stage of rocket motors is summarized below.

A. Long Method

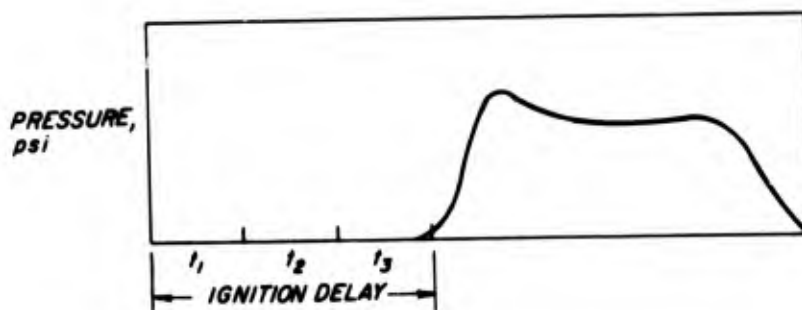
1. Determine energy needed to fire each rocket motor.
2. Determine total resistance connected between knife blades.
3. Find the time the sled knife blades are in the screen boxes.
4. Find the effective (heating) value of current.
5. Find how the current divides between the squib and its continuity resistor.
6. Find the energy used in the rocket motor squib; be sure it is at least 3 times what is needed.

B. Using Nomographs

1. Determine energy needed to fire each rocket motor.
2. Determine total resistance connected between knife blades.
3. Find the time the sled knife blades are in the screen boxes from graph #1.
4. Select a power supply (nomograph 1, 2, 3, or 4) and see how much energy is transferred to the resistance network.
5. Find how much energy is used by each rocket motor setup by dividing the energy transferred to the resistance network by the number of rocket motor setups.
6. Find what percent of the energy is used in the rocket motor squib from nomograph #5. Multiplying this by value of energy found in (5) above we have energy used in rocket motor squib. Be sure it is at least 3 times what is needed.

Addendum A
IGNITION DELAY

The sketch below shows a typical pressure graph for a rocket motor. Ignition delay is defined as the time delay between the closing of the firing circuit and the start of pressure rise in the rocket chamber.



t_1 = Squib functioning time

t_2 = Ignitor functioning time

t_3 = Grain ignition time

$$\text{Ignition Delay} = t_1 + t_2 + t_3$$

As shown ignition delay consists of:

Squib functioning time
Ignitor functioning time
Grain ignition time

Squib functioning time is the time we need to make an exact calculation of the energy required to fire the rocket motor. This information is rarely known. However, by using ignition delay time, which is always known, we can calculate a value of energy that will always be on the safe side since ignition delay time is greater than the squib functioning time.

Addendum B

Derivation of Effective Current Formula

By definition the effective current, in amperes, is

$$I_{\text{eff}} = \sqrt{\frac{\int_{t_1}^{t_2} i^2 dt}{t_2 - t_1}}$$

Where i is instantaneous current, in amperes, at any time: t_1 and t_2 are time, in seconds, at instant 1 and instant 2, respectively.

In a circuit with a condenser discharging into a resistor or resistor network the formula for instantaneous current is:

$$i = \frac{E}{R} e^{-\frac{t}{RC}}$$

Where:

i = instantaneous current in amperes

E = condenser voltage in volts

R = resistance in ohms

e = Napierian base (2.72)

t = time in seconds

c = condenser capacity in farads

Current squared is

$$i^2 = \left(\frac{E}{R} e^{-\frac{t}{RC}} \right)^2 = \frac{E^2}{R^2} e^{-\frac{2t}{RC}}$$

Integral of current squared is

$$\int_{t_1}^{t_2} i^2 dt = \int_{t_1}^{t_2} \frac{E^2}{R^2} e^{-\frac{2t}{RC}} dt$$

$$= \frac{E^2}{R^2} \int_{t_1}^{t_2} e^{-\frac{2t}{RC}} dt$$

$$= \frac{E^2}{R^2} \left[\left(e^{-\frac{2t}{RC}} \right) \left(\frac{1}{-2/RC} \right) \right]_{t_1}^{t_2}$$

$$= \frac{E^2}{R^2} \left[\left(e^{-\frac{2t}{RC}} \right) \left(-\frac{RC}{2} \right) \right]_{t_1}^{t_2}$$

$$= -\frac{E^2 RC}{2 R^2} \left[e^{-\frac{2t}{RC}} \right]_{t_1}^{t_2}$$

$$= -\frac{E^2 C}{2R} \left[e^{-\frac{2t}{RC}} \right]_{t_1=0}^{t_2=t}$$

$$= -\frac{E^2 C}{2R} \left[e^{-\frac{2t}{RC}} - \left(e^{-0} \right) \right]$$

$$= -\frac{E^2 C}{2R} \left[e^{-\frac{2t}{RC}} - 1 \right]$$

Bring minus sign inside bracket

$$= \frac{E^2 C}{2R} \left[1 - e^{-\frac{2t}{RC}} \right]$$

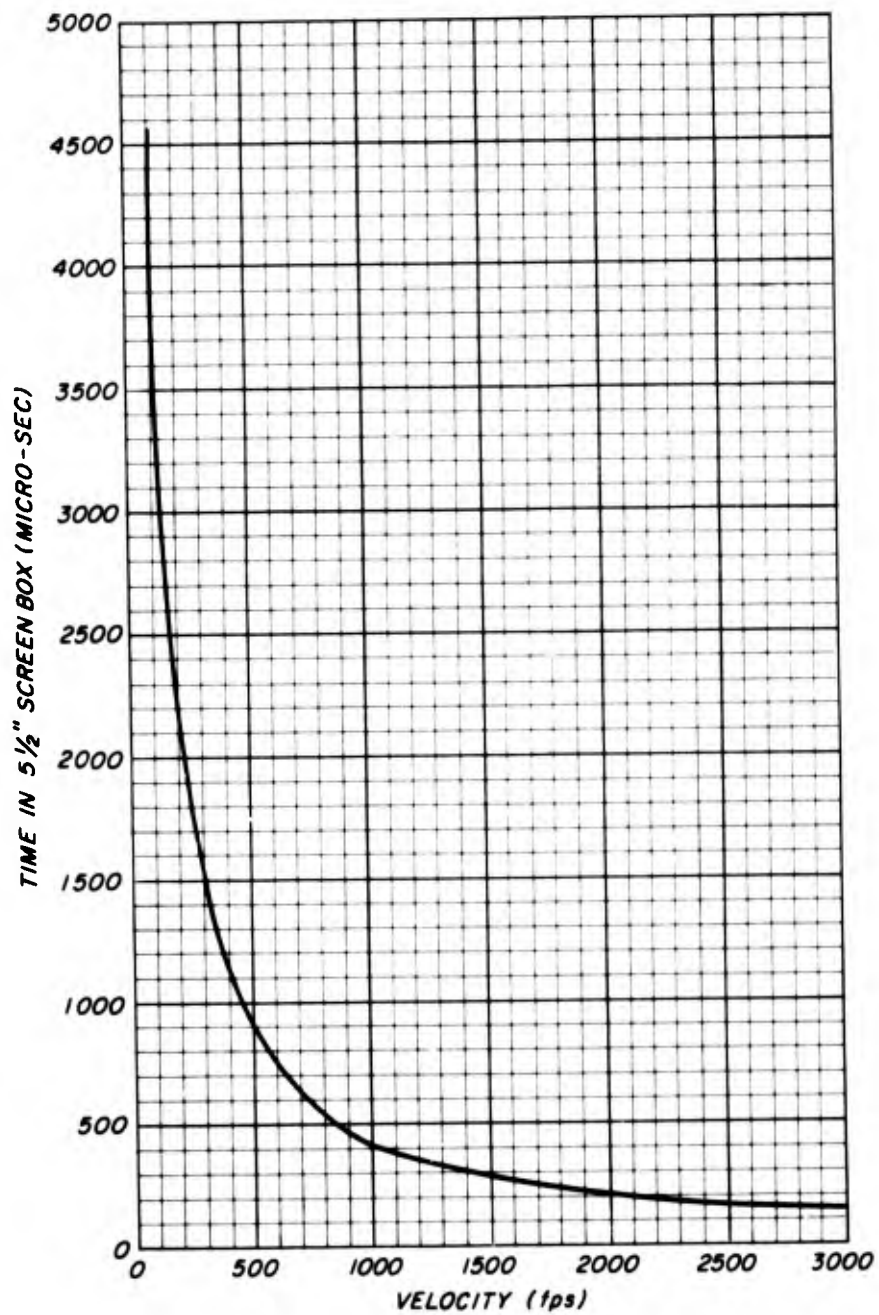
Dividing integral $i^2 dt$ by t gives

$$= \frac{E^2 C}{2Rt} \left[1 - e^{-\frac{2t}{RC}} \right]$$

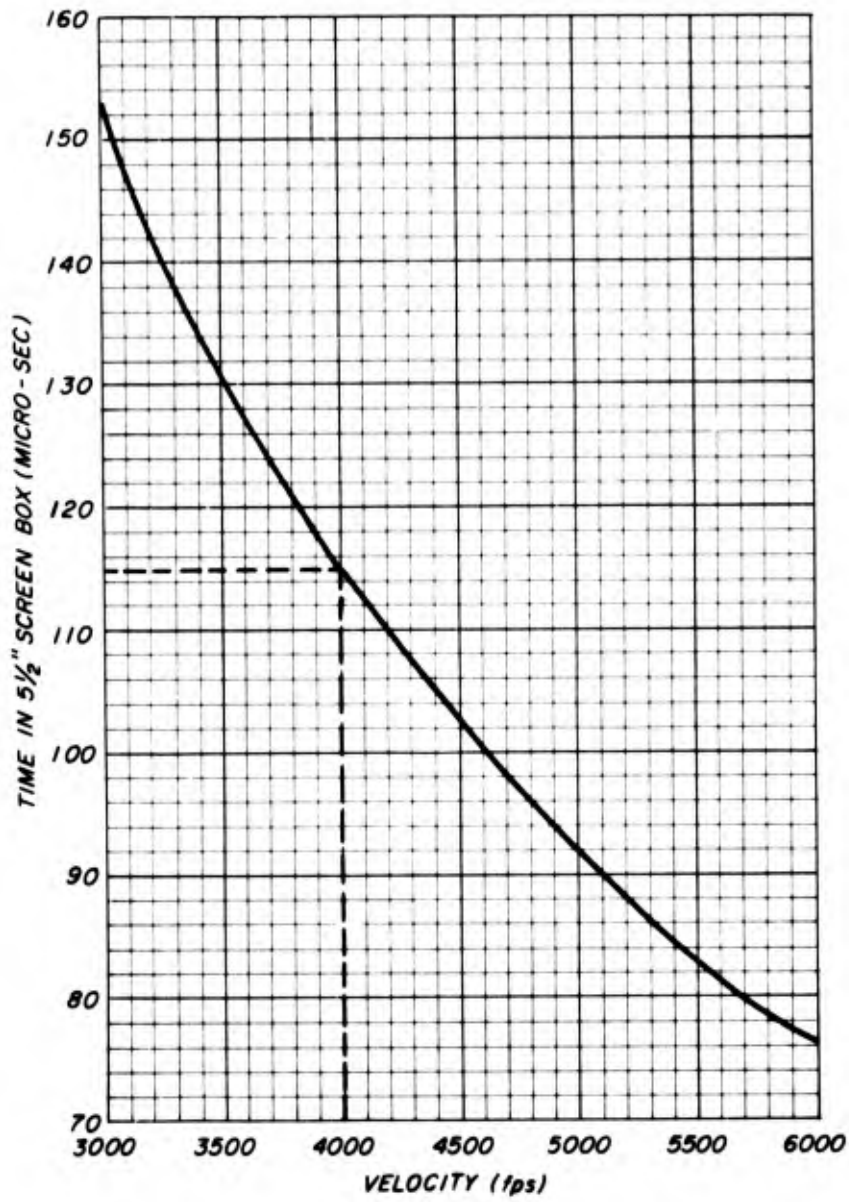
Taking the square root of the above gives effective value of the current

$$I_{\text{eff}} = \sqrt{\frac{E^2 C}{2Rt} \left[1 - e^{-\frac{2t}{RC}} \right]}$$

GRAPH NO. 1



GRAPH NO. 1 (CONT.)

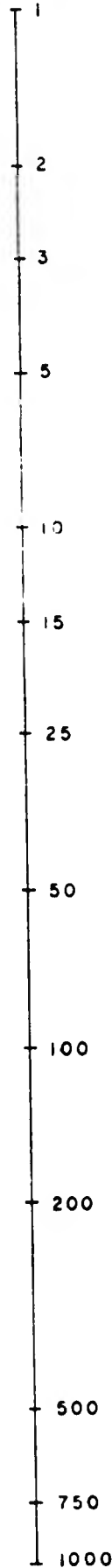
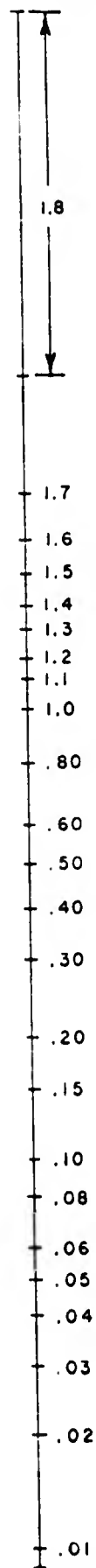
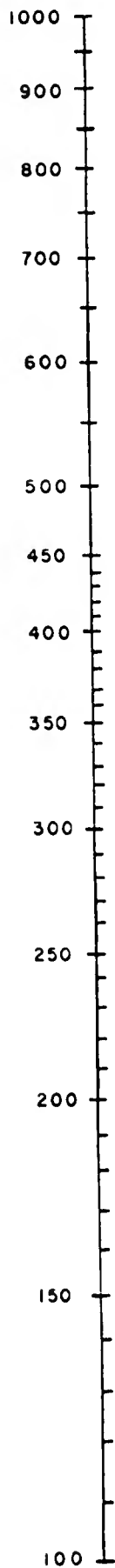


DASHED LINES ARE FOR PROBLEM OF TEXT

Time in
Screen Box
(Microseconds)

Energy
(Watt-Seconds)
(Joules)

Resistance
(Ohms)



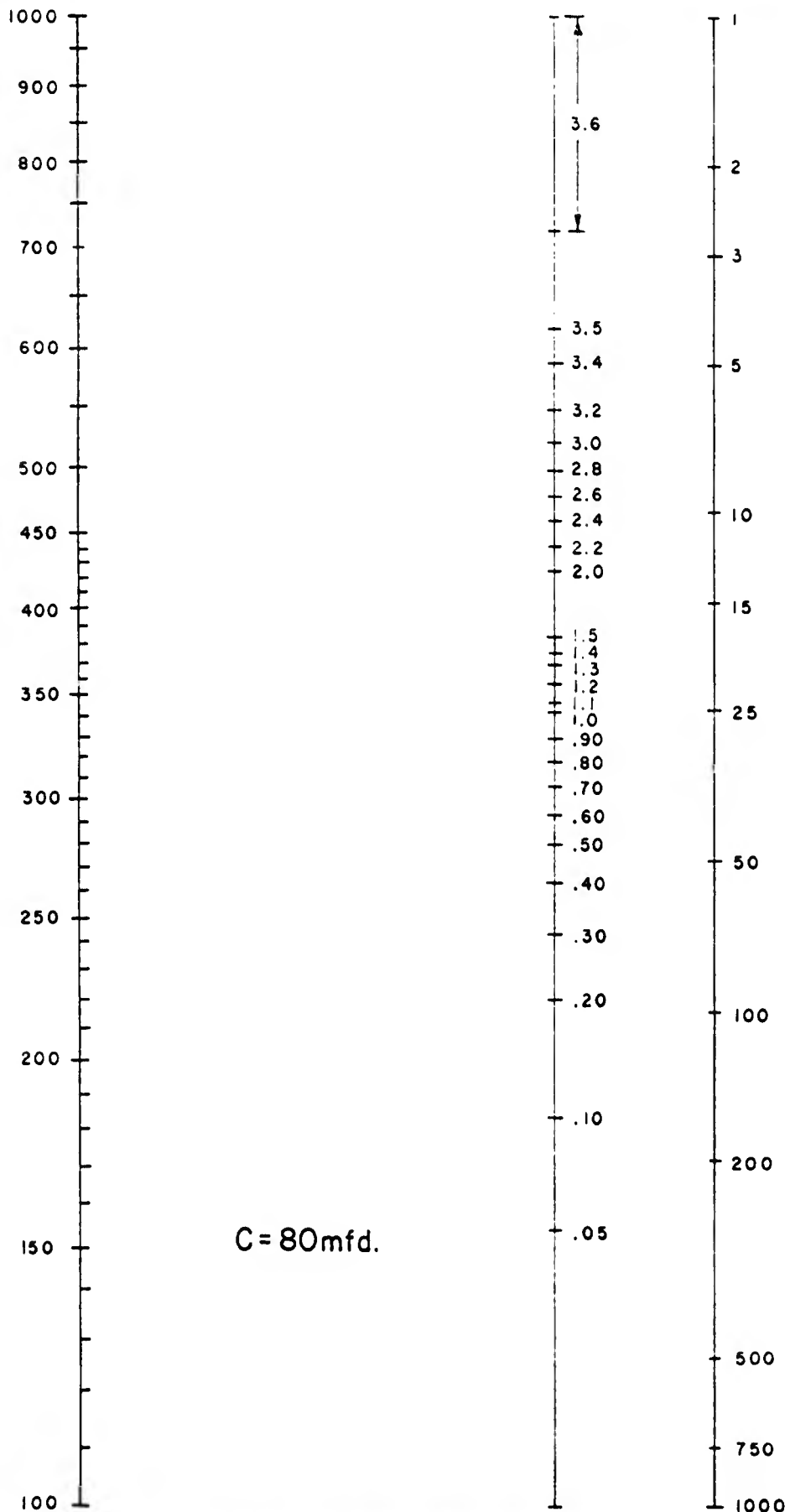
C = 40 mfd.

Energy Transfer Nomograph #1
Screen Boxes → Knife Blades
(For 1-300V Screen Box Supply)

Time in
Screen Box
(Microseconds)

Energy
(Watt-Seconds)
(Joules)

Resistance
(Ohms)

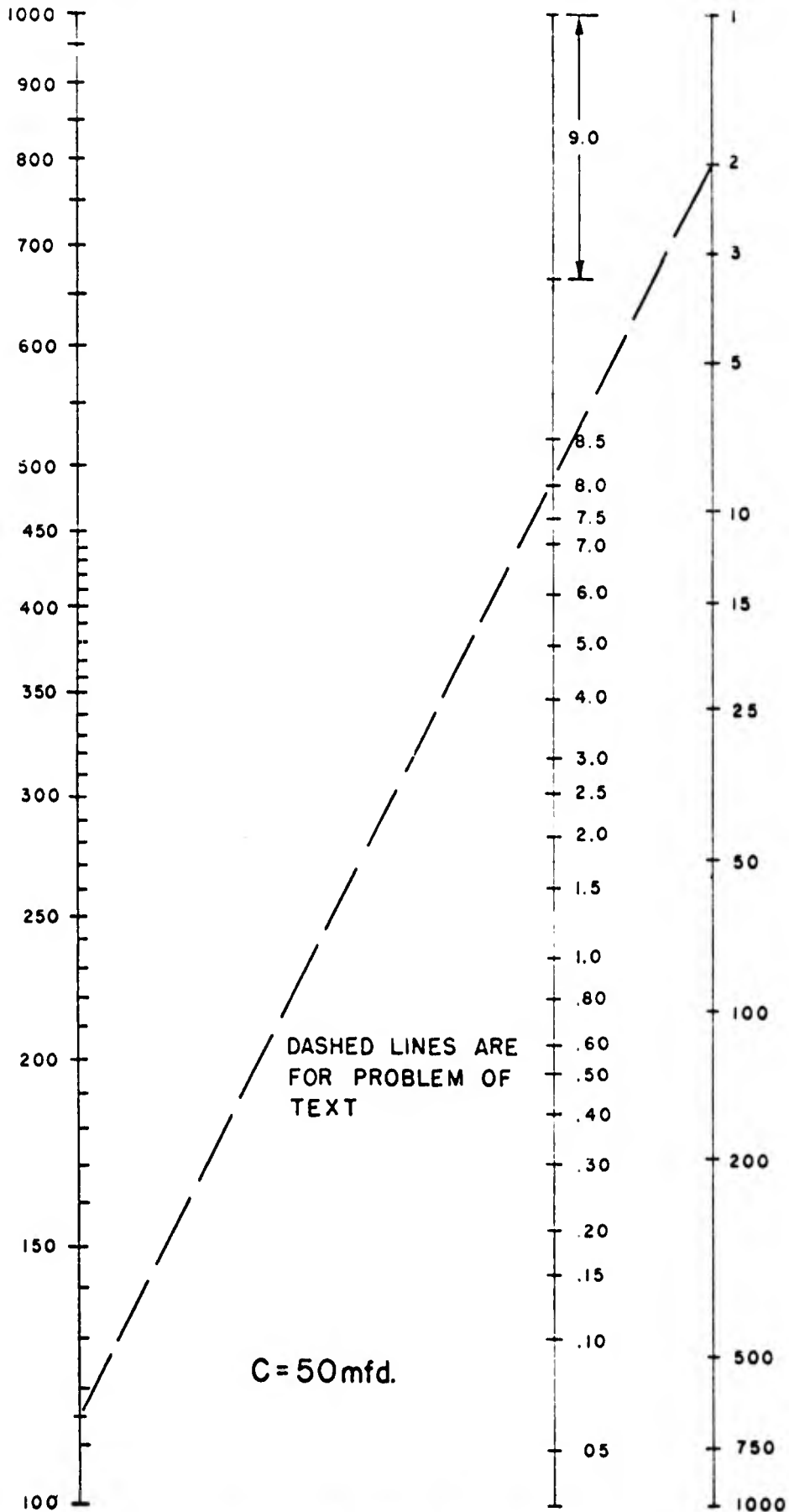


C = 80mfd.

Energy Transfer Nomograph #2
Screen Boxes → Knife Blades
(For 2-300V Screen Box Supplies in Parallel)

Time in
Screen Box
(Microseconds)

Energy
(Watt-Seconds)
(Joules) Resistance
(Ohms)

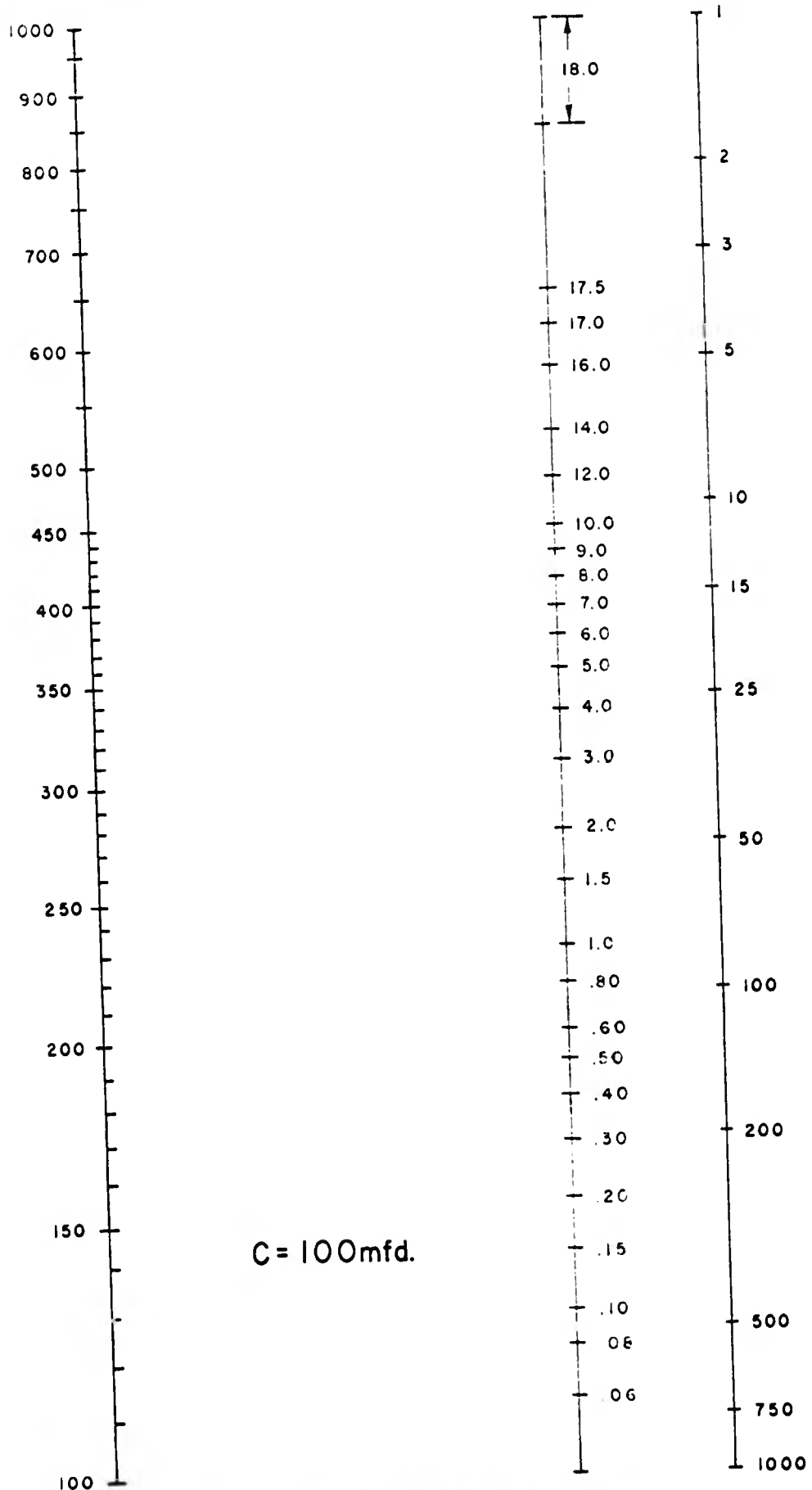


Energy Transfer Nomograph #3
Screen Boxes → Knife Blades
(For 1-600V Screen Box Supply)

Time in
Screen Box
(Microseconds)

Energy
(Watt-Seconds)
(Joules)

Resistance
(Ohms)



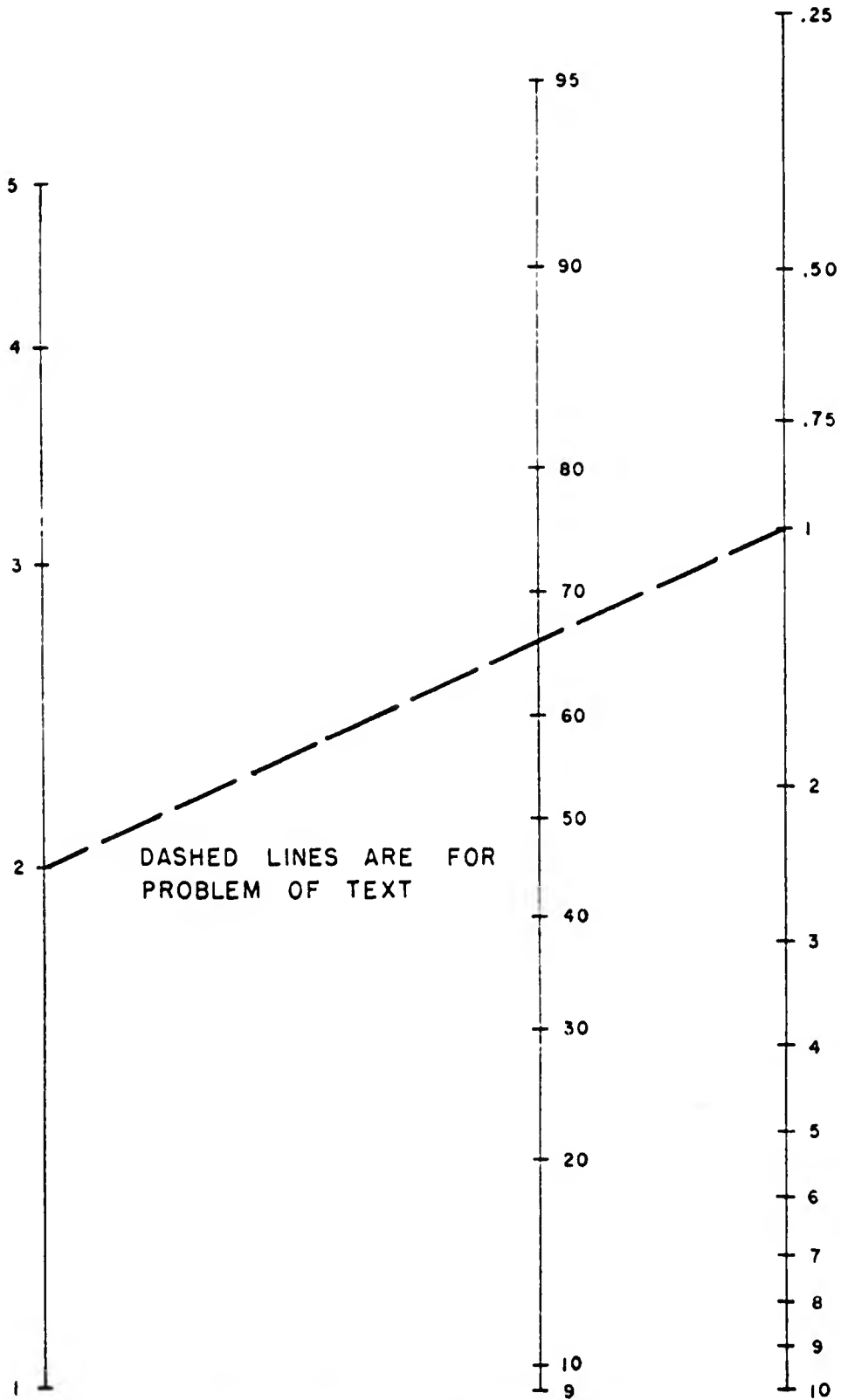
C = 100mfd.

Energy Transfer Nomograph #4
Screen Boxes → Knife Blades
(For 2-600V Screen Box Supplies in Parallel)

Continuity
Resistor
(Ohms)

% of Total
Energy Available
To Squib

Squib
Resistance
(Ohms)



DASHED LINES ARE FOR
PROBLEM OF TEXT

Energy Distribution Nomograph #5
Squib ←→ Continuity Resistor

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