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(6) PRELIMINARY SOLID ROCKET MOTOR
DESIGN TECHNIQUES

by

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FOREWORD

This report is intended for use at the working level and does not reflect the official view or final judgement of the Naval Weapons Center. It's main purpose is to insure that solid motor preliminary design studies maintain the sam level of technical expertise and reality which currently characterize ramjet preliminary design studies. This is particularly important when competitive rocket and ramjet designs are evaluated in the same mission analyses.

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General techniques to calculate solid rocket motor performance characteristics to determine if certain designs are feasible or to obtain more realistic preliminary design data are described in this report. There are many designs that can be used to produce specified motor performance characteristics and the possibility of using the "cookbook" approach to achieve "the" correct design is a non-existent distant dream.

Solid rocket motor design requires repetitive estimates and calculations where rough approximations are made and by repeated estimates and calculations, a refined motor is achieved that delivers the required performance. By this iterative process, an acceptable motor design is achievable.

This report can then serve as a practical primer for solid propellant rocket motor design. It may not solve all one's design problems but will help in many small ways.

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GUIDE TO INTERNATIONAL SYSTEM OF UNITS (SI)

The Metric Practice Guide[1] deals with the conversion of quantities in various systems of measurement to the International System of Units, which is officially abbreviated SI in all languages. Certain customary units, both U.S. and "metric," will gradually be phased out of existence, and this and other countries will use the common international measurement language, SI.

Therefore, since this memorandum was written with English Units, various base and derived units and conversion factors are included in this section to convert to the new SI.

Base Units

Quantity	Unit	Symbol
Length	Metre	M
Mass	Kilogram	kg
Time	Second	S
Temperature	Kelvin	K

Derived Units

Quantity	Unit	SI Symbol	Formula
Acceleration	Metre per second squared	-	M/S^2
Area	Square meter	-	M^2
Density	kilogram per cubic meter	-	kg/M^3
Force	newton	N	$kg\ M/S^2$
Pressure	pascal	Pa	N/M^2
Power	watt	W	J/S
Stress	pascal	Pa	N/M^2
Velocity	metre per second	-	M/S
Work	joule	J	N M

[1] ASTM E 380-74 of 24 February 1975.

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GUIDE TO INTERNATIONAL SYSTEM OF UNITS (SI) - Continued

Conversion Factors

From	To	Multiply by
in^2	M^2	6.451×10^{-4}
pound (F)/in^2	pascal	6.894×10^{3}
pound (F)avoir	newton	4.448
pound (F)/in^3	kilogram/$metre^3$	2.768×10^{4}
in/sec	metre/sec	2.540×10^{-2}
foot/sec^2	metre/sec^2	3.048×10^{-1}
BTU/ft^2-sec	watt/M^2	1.135×10^{4}
degree Fahrenheit	Kelvin (K)	$t°_K=(t°_F+459.67)/1.8$

INTRODUCTION

General techniques to calculate solid rocket motor performance characteristics to determine if certain designs are feasible or to obtain more realistic preliminary design data are described in this report. There are many designs that can be used to produce specified motor performance characteristics and the possibility of using the "cookbook" approach to achieve "the" correct design is a non-existent distant dream.

Solid rocket motor design requires repetitive estimates and calculations where rough approximations are made and by repeated estimates and calculations, a refined motor is achieved that delivers the required performance. By this iterative process, an acceptable motor design is achievable.

This report can then serve as a practical primer for solid propellant rocket motor design. It may not solve all one's design problems but will help in many small ways.

The theoretical design of a solid propellant rocket motor is, in general, simple but not necessarily straight forward. This paper will review and discuss the approach to the design philosophy required to engineer a solid propellant rocket motor.

Simplicity is the main feature of a solid motor. The lack of valves and plumbing serves to increase performance reliability, reduce inert component weight, and make them relatively easy to use. Along with the simplicity are the limitations. The main limitation is the relatively short burn time achievable; in general, the burn times vary from a few seconds up to a maximum burn time of about 180 seconds.

A solid rocket motor consists of a high energy propellant grain stored within an inert combustion chamber capable of withstanding high pressure and high temperature conditions. An igniter is positioned in the combustor to ignite the grain and at one end of the combustor is a nozzle to direct the discharge of the combustion gases. Insulation material lines the combustor from the high temperature gases, inhibitor material controls the propellant burning surfaces, and liner material insures a good bond between the propellant and insulation. Often, the liner and inhibitor are the same material. Once ignited, the propellant burns uniformly on the uninhibited surface and regresses in a direction perpendicular to the burning surface. Therefore, by proper grain design and inhibiting a wide variety of performance characteristics is achievable. Figure 1 represents a typical solid propellant rocket showing component arrangement and three bubbles magnify the specific points of interest.

Rigorous proofs and derivations of equations are beyond the scope of this memorandum.

PERFORMANCE REQUIREMENTS

Motor design begins when motor performance characteristics can be specified in specific terms. Specifically, the total impulse is defined and the way that impulse is to be delivered and the restraining limitations of the dimensional envelope.

Figure 2 shows three ideal ways that a constant 75,000 lb-sec total impulse can be delivered and each type of delivery requires a different design and different type of propellant formulation. The high thrust (15,000 lbs) short burn time (5 sec) shows the characteristics of a booster motor. Performance as shown will require a large exposed surface area for propellant burning and a high burn rate propellant. To achieve the desired performance, the motor will operate at a chamber pressure of 1500-2000 lbs/in^2. This performance is similar to that produced by booster motors.

The low thrust (2500 lbs) long burn time (30 sec) are the characteristics of a sustainer motor that would require a very small burning area, end-burn, and a slow burn rate propellant. This motor would probably operate at a chamber pressure of about 500-800 lb/in^2. The performance is similar to that produced by sustainer motors.

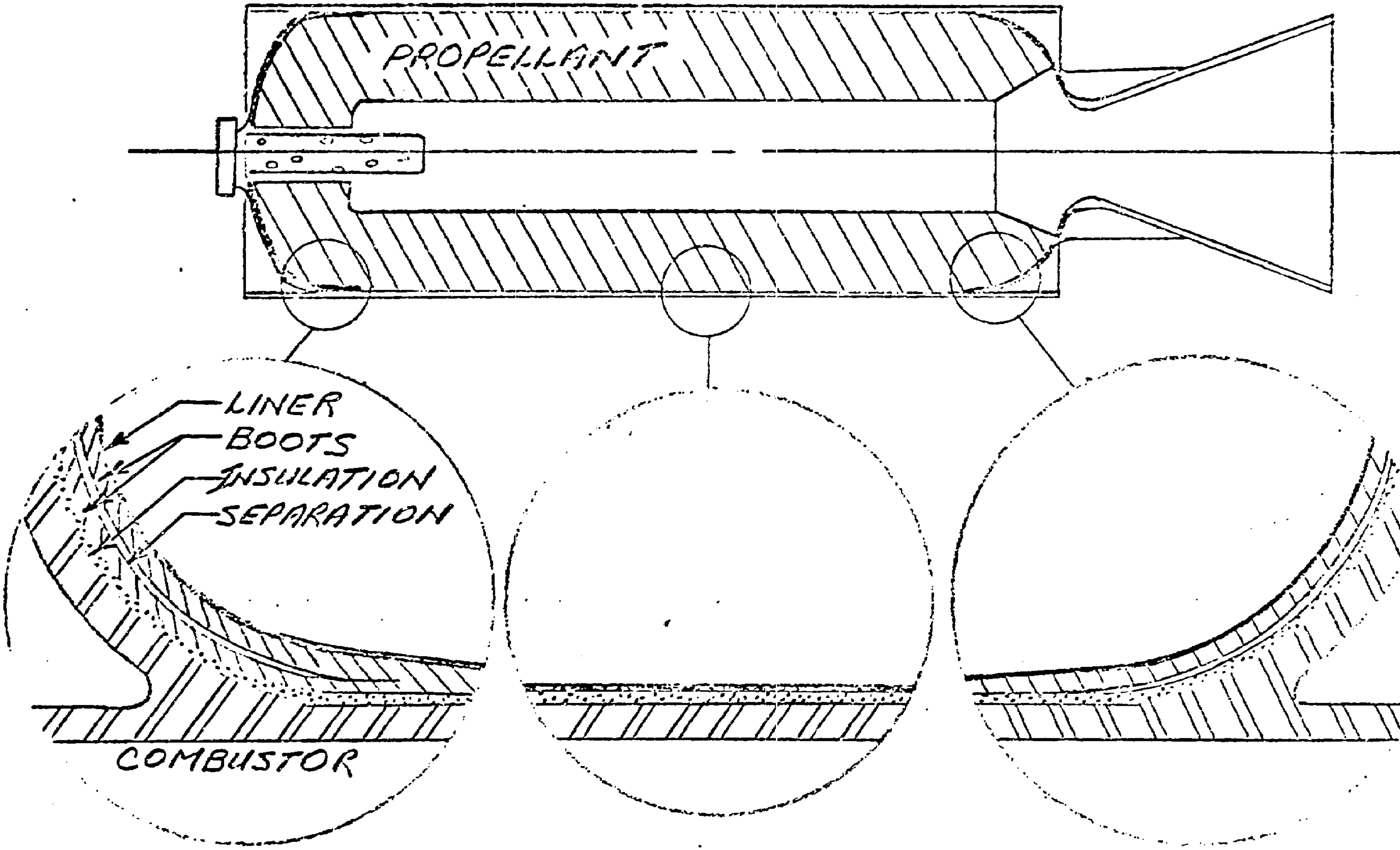

FIG. 1.
SOLID ROCKET MOTOR

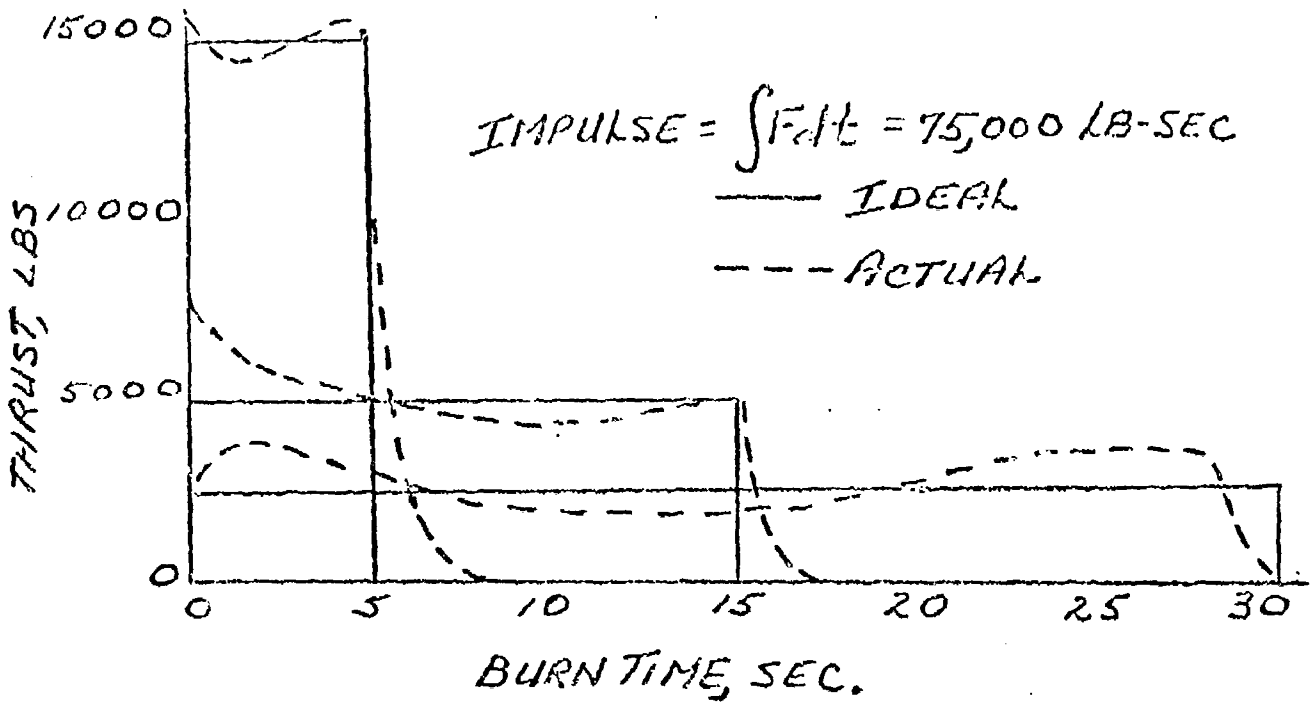

FIG. 2.
THRUST-TIME PERFORMANCE

An intermediate thrust-time performance would probably operate at a chamber pressure of about 800-1200 lbs/in^2, as a compromise between booster and sustainer rocket motor performance.

Though these impulse performance histories are shown as flat square ideal histories, in general they are never achieved but can be reasonably well estimated as shown by the broken lines on Figure 2. These types of curves are more typical of results obtainable.

Often, other specific irregular shaped performance histories may be required and some of these are reproduced in Figure 3.

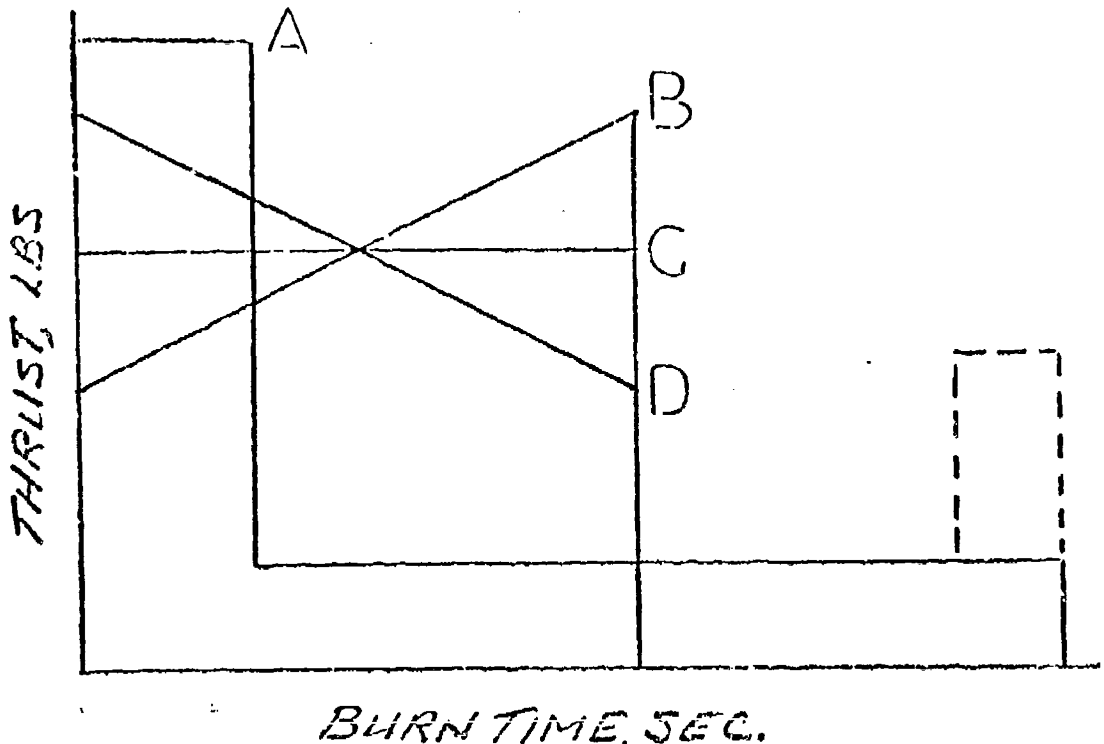

FIG. 3.

OTHER THRUST-TIME HISTORIES

Curve "A" shows a high thrust initial boost followed by a low thrust long duration sustain operation. At times, a high thrust at the end-of-burn may be required as shown by the broken lines. Curve "B" shows the progressive type performance that often occurs where the propellant loading is increased. This arrangement is not perticularly desirable since the maximum pressure is achieved at the end-of-burn when the case is hottest and the case strength reduced. This performance is generally undesirable since the highest thrust is achieved when the missile weight is minimum resulting in extremely great acceleration "g" loads. The neutral thrust-time performance as shown by curve "C" is quite desirable since this is the most effective use of the combustor. The combustor wall thickness is determined by the maximum pressure achieved, regardless of the length of time the load is applied. This design can, therefore, result in the thinnest combustor wall and a lower inert case weight. A definite advantage. A regressive thrust-time history is shown by curve "D". This performance can be required when a constant acceleration motor is needed. As can be noted, as the motor weight is reduced, by consuming propellant, the thrust is reduced maintaining a constant acceleration. The disadvantages are that the combustor wall thickness is calculated using the maximum pressure and this maximum pressure is achieved when igniter spikes develop to augment the high pressure.

In general, no matter how complicated the desired performance characteristics may be, a good approximation can be achieved by varying the grain geometry and the areas that are inhibited to control propellant burning. Designs are only limited by the ingenuity of the engineer. Of course, the problem may develop when the performance impulse requirements are incompatible with the dimensional envelope limitations. These conflicts do require compromises - or more ingenious designers.

How surface areas vary with web burnback for a few of the multitude of grain designs can be seen in Figure 4.

Though these graphs show the surface area variations as the propellant burns back, the pressure and thrust performances will follow the surface area variations closely. All these surface area graphs assume a constant length propellant grain as the surface recedes.

A more detailed diagram showing how the surface area variations with web burn-back are determined is shown in Figure 5 where the case diameter is assumed to be known.

These diagrams are carefully drawn and scaled and web burn-back surfaces drawn in as shown by the broken lines. These can be drawn in at any interval, usually varying from 0.1 to 0.5-inch, depending on the accuracy desired. But, it is essential that the distances be measured perpendicular to the burning surface. These surfaces can be accurately calculated, but they can be rapidly and accurately determined with the use of a map-reader. The little wheel of the

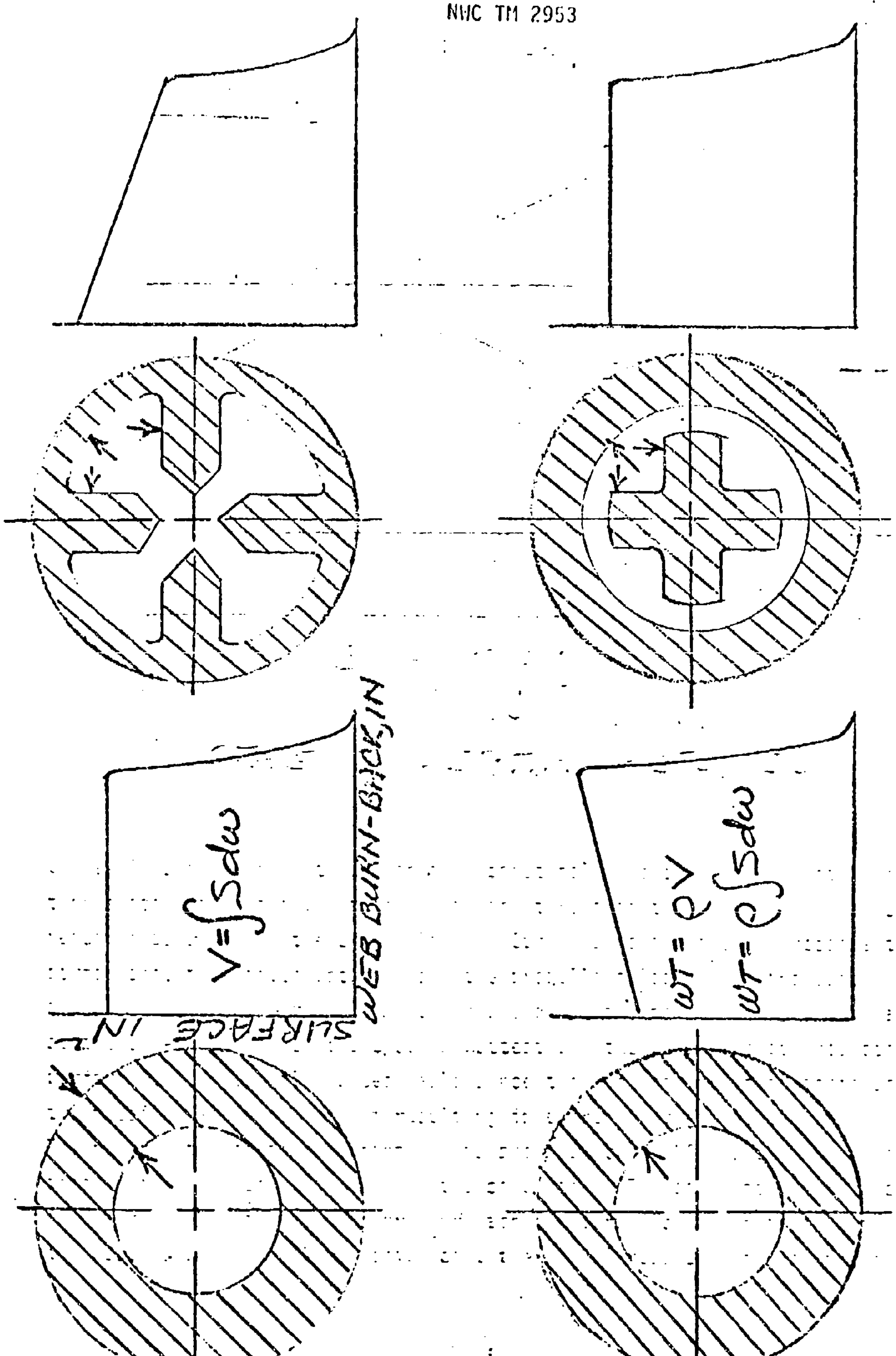

FIG. 4. PROPELLANT GRAIN GEOMETRIES

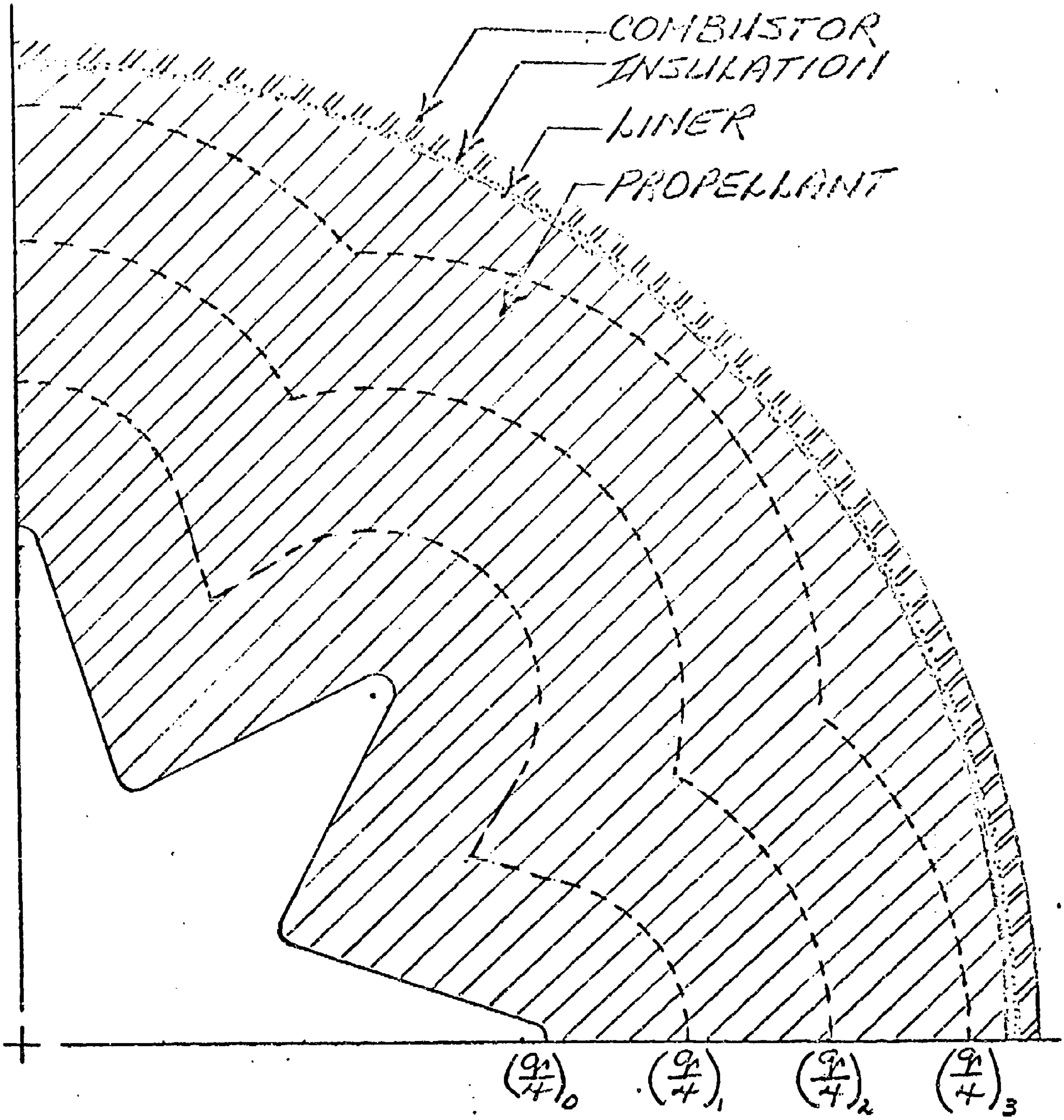

FIG. 5.
SURFACE AREA
BURN-BACK CONTOUR

map-reader is placed on the drawing and carefully pushed along each line and a dial gage records the distances that the wheel rolled. At this time, it is also important to planimeter the port area of the propellant grain.

After measuring these perimeter values quite accurately, the grain lengths at the same burn-back distances are determined from side view drawings of the motor. The product of the perimeter and grain length indicates the propellant surface areas at specific web burn-back intervals. Values determined for a typical booster motor are tabulated in Table 1 and these figures graphed in Figure 6. For this motor, a reasonably neutral performance was desired and a propellant weight of about 400 pounds.

TABLE 1. Surface Area and Web-Burn Back

dw, in	q/4, in	q, in	L, in	S, in^2
0	8.2	32.8	43.5	1427
0.5	8.5	34.0	43.4	1476
1.0	8.8	35.2	43.3	1524
1.5	9.0	36.0	43.3	1559
2.0	9.2	36.8	42.9	1579
2.5	9.8	39.2	42.2	1654
3.0	10.5	42.0	41.0	1722
3.65	11.5	46.0	39.0	1794
4.05	1.5	6.0	39.0	234

The surface variation appeared reasonably neutral and the area under the curve was planimetered to determine the propellant volume. The volume multiplied by the density indicates the propellant weight. Now, if the combustor is too short to contain this amount of propellant, or if the surface area variation with web burn-back is unacceptable, it will be necessary to begin redesign efforts on the propellant grain or selecting a candidate propellant formulation with a greater density. If the preliminary design appears acceptable, the propellant burn rate required can be determined by dividing the web thickness by the burn time, r_b = web,in/burn time, sec. This burn rate should be achievable at a chamber pressure of about 1000 lb/in^2.

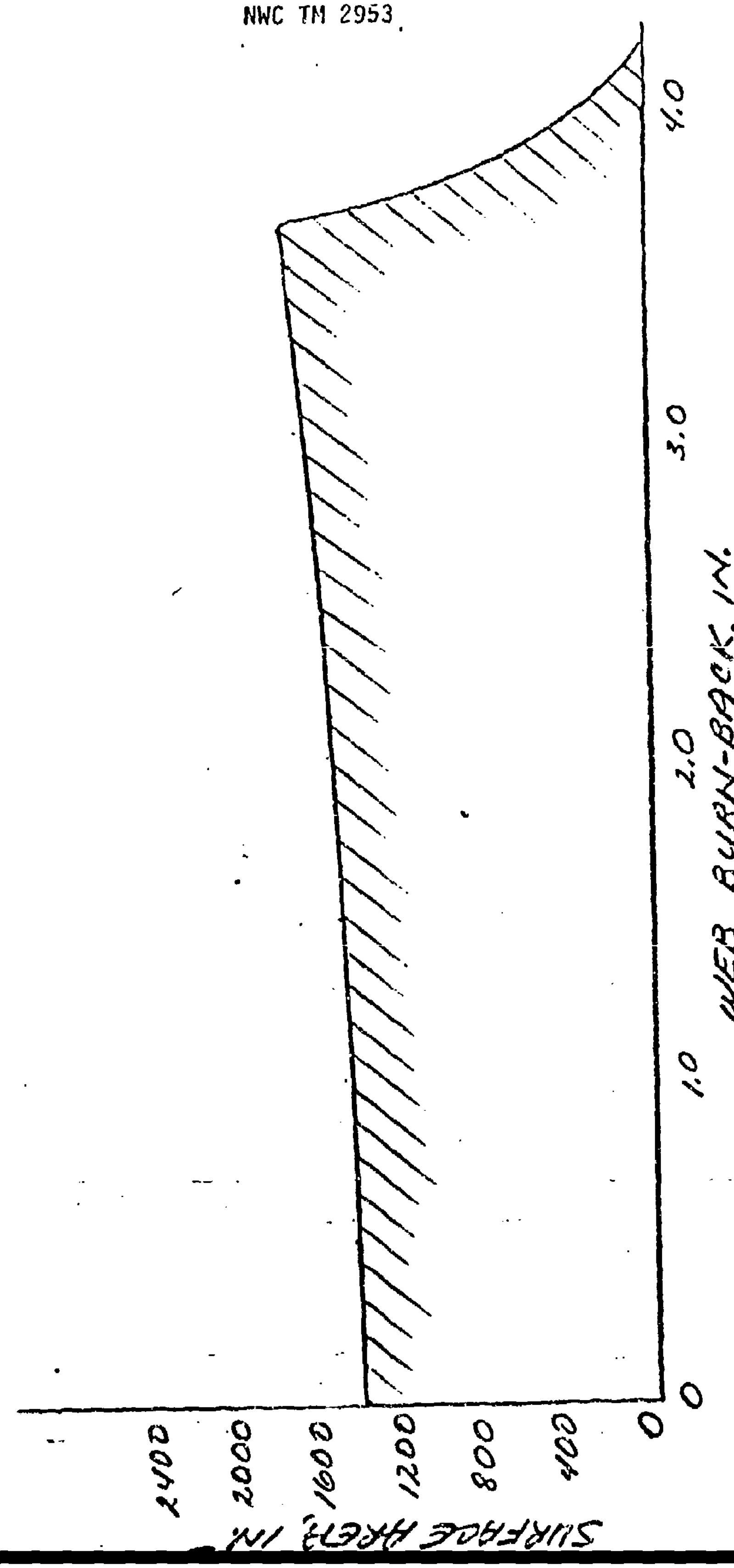

FIG. 6.
AREA VARIATION ESTIMATE

With both a burn rate and density known at this time, a literature search is necessary to determine which formulations appear acceptable for this specific application. Generally, there will be more than one formulation capable of producing these requirements. Therefore, a throat area can be calculated for each formulation by assuming equilibrium conditions from:

$$A_T = \frac{S \rho C\, C^*}{g\left[P_C\right]^{(1-n)}}$$

In this equation, the designer determined the surface area(s) and set the chamber pressure (P_C). By selecting a certain formulation, the density (ρ), temperature constant (C), and the characteristic exhaust velocity (C*), and the pressure sensitivity of the propellant (n) are all known from the propellant handbook. Therefore, the nozzle throat area necessary to achieve a chamber pressure of 1000 psi when the propellant surface area is maximum can be calculated. Once this throat area sets the maximum pressure, the equation is rearranged so that:

$$P_C = \left[\frac{S \rho\, C\, C^*}{g\, A_T}\right]^{\frac{1}{1-n}} = \left[K_N\, \rho \frac{C\, C^*}{g}\right]^{\frac{1}{1-n}}$$

and the remaining pressures can be calculated for the smaller surface areas. In this equation, S/A_T is designated as K_N, the area ratio, and is the only part of the equation that the designer has control over. Needless to say, none of these pressures can exceed the original 1000 psi pressure limit. With the pressures calculated at specified points on the web burn-back, these pressure values can be converted to burn rate information since:

$$r = CP^n$$

and since the burn distances are known, from the drawing the surface area-web burn-back graph can be converted to a pressure-time performance history quite readily.

At this time, both the port area (A_P) and throat area (A_T) are known and it is important to calculate the "J" factor which is the throat area to port area ratio:

$$J = A_T/A_P$$

The "J" factor controls the pressure drop along the propellant grain and this, in turn, determines the gas velocity as it flows towards the nozzle. Only the initial "J" factor is critical. A "J" factor of unity indicates that the throat and port areas are equal so that the propellant gases are flowing at sonic velocity at the end of the grain. The high velocity hot gases produce erosive or excessively high burn rates causing excessively high pressure peaks on ignition that can produce motor blow-ups. High "J" factors can be reduced by redesigning the grain by opening up the port area, by operating at higher pressures so that the throat area size is reduced, or a combination of both approaches.

The "J" factor is a good measure of how full the combustor is loaded with propellant. Values from 0.95 and above show high loading fractions but are generally unacceptable due to erosive burning complications; values from 0.70 to 0.95 generally indicate acceptable propellant loading characteristics; values less than 0.70 show poor loading characteristics. A good design approach would be to design for the lowest possible "J" factor consistent with achieving the required motor performance characteristics. This will minimize the possibility of encountering erosive burning problems while delivering the required performance.

After calculating the pressure-time performance of the motor, the pressure is readily converted to thrust-time by the following equation:

$$F = \eta \lambda C_{F_T} A_T P_C$$

At this time it is necessary to roughly design the rocket nozzle around the throat area, that was previously calculated, to obtain pressure and area ratios as well as a half-angle expansion factor. The η is an inefficiency factor due to incomplete combustion, two phase flow, heat losses, and friction effects due to non-ideal conditions. From past experience this inefficiency factor generally varies from about 0.93 to 0.95. The λ term results from the diverging portion of the rocket nozzle and corrects for the non-axial component of the exhaust gas direction. It is calculated from:

$$\lambda = 1/2\ (1 + \cos \alpha)$$

Here α is the half-angle of the expansion cone. In general, the nozzle half-angle will not exceed about 20° so that λ will be about 0.97. The theoretical coefficient of thrust, C_{F_T}, can be calculated from:

$$C_{F_T} = \sqrt{\frac{2k^2}{k-1}\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}\left[1-\left(\frac{P_E}{P_C}\right)^{\frac{k-1}{k}}\right]} + \frac{P_E-P}{P_C}\left(\frac{A_E}{A_T}\right)$$

where k is the specific heat ratio, P_C, the chamber pressure, P_E, the exit pressure, and P the atmospheric pressure. Though it can be calculated extremely accurately, the value is generally found on a graph where the values are plotted for specific values of k, (A_E/A_T) area ratios, and pressure ratios. A typical graph is reproduced in Figure 7.

Therefore, all the values are known from the thrust equation and the pressure-time history can be converted to a thrust-time history.

One other small refinement is generally applied and that is an estimation of the nozzle throat radius erosion rate. With graphite inserts in the throat, past experience indicates that the radius increases at a rate of about 0.001 in/sec.

Once the thrust-time values are calculated and graphed, the area is planimetered to determine the total impulse delivered by this specific motor design. The propellant weight was determined from planimetering the surface area-web burn-back graph so that the specific impulse can now be calculated for this propellant formulation in this motor design by:

$$I_{sp} = \frac{\int F dt}{\rho \int S dw} = \frac{IT}{WT_P}$$

Though transient operating conditions have not been investigated, sufficient information should now be available to determine if this grain and nozzle design and the general propellant properties are capble of fulfilling the motor performance requirements. If not, redesign efforts should be initiated.

PROPELLANTS

Propellant formulations mainly contain an oxidizer and a fuel capable of burning to completion once ignited. The properties can vary considerably depending on the type and particle size of the oxidizer, the type of fuel, and the minor

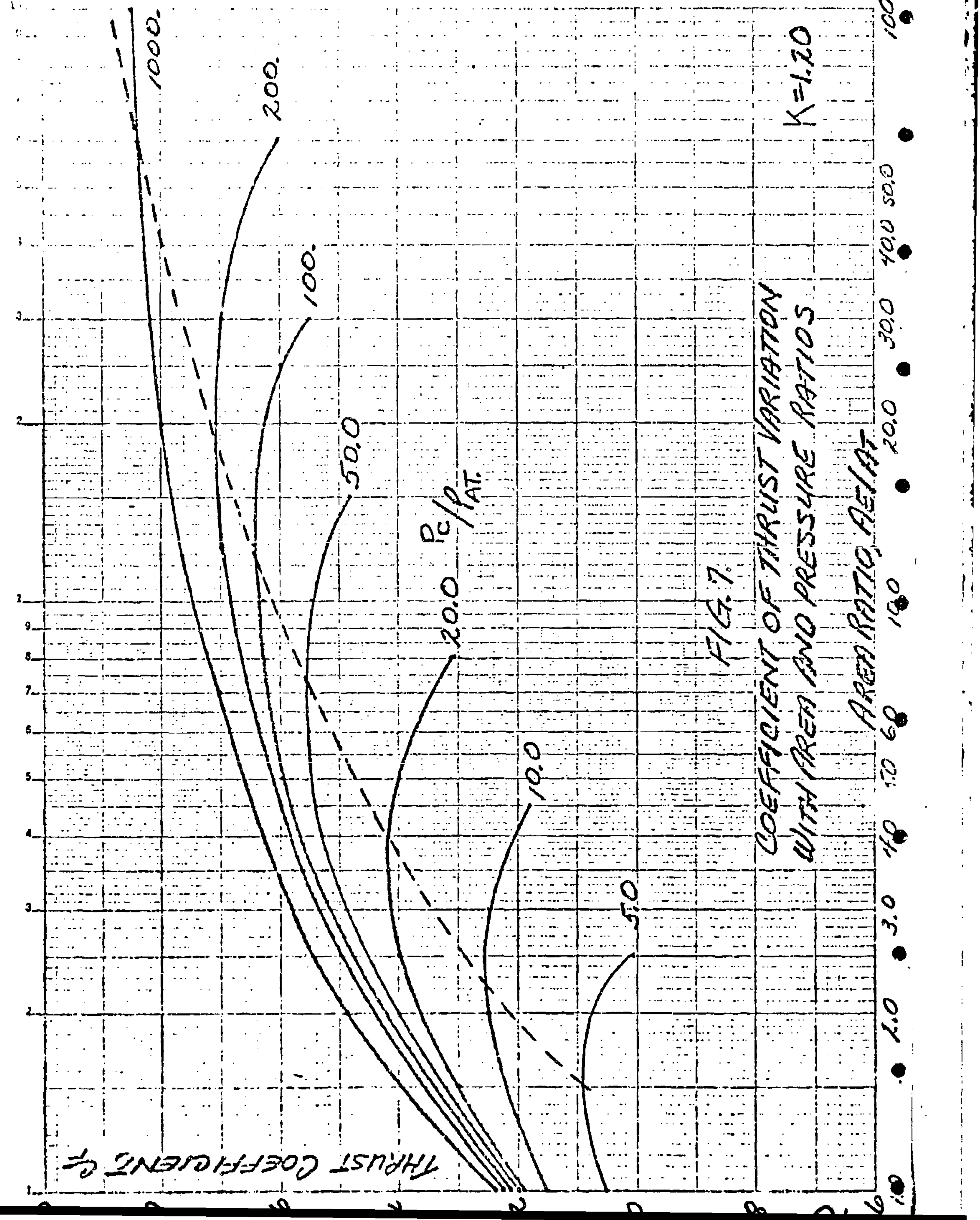

FIG. 7.
COEFFICIENT OF THRUST VARIATION WITH AREA AND PRESSURE RATIOS

ingredients used as ballistic modifiers. Often, propellant formulations can be tailored to specific requirements and motor applications when necessary. Table 2 shows the extent to which the propellant properties will vary:

TABLE 2. General Propellant Properties

	Type Motors		
	Gas Generators	Sustainers	Boosters
r_b1000 psi-70°F	0.005 - 0.10	0.20 - 0.40	0.50 - 1.20
n	.3 - .8	.3 - .8	.3 - .8
I_{sp}	160 - 180	180 - 210	210 - 250
C*	3000 - 4000	3500 - 4500	4000 - 5200
ρ	.057 - .059	.059 - .062	.062 - .065
K	.100 - .400	.100 - .400	.100 - .450

The pressure developed in a rocket motor depends on the burn rate of the propellant which in turn depends upon the temperature of the grain and the pressure in the combustor. Figure 8 shows the dependence of the burn rate and the K_N area ratio on the pressure and temperature for a specific propellant formulation.

This family of curves can be described by an exponential equation:

$$r = CP^n$$

$$\ln r = n \ln P + \ln C$$

Here ln C is the Y-intercept and is controlled by the temperature; and n is the slope and is controlled by the pressure. It is important to select propellant formulations with low pressure sensitivity constants that do not exceed about n = 0.7. The problem can be explained from:

$$P = \left[\frac{S \rho C\ C^*}{g\ A_T}\right]^{\frac{1}{1-n}}$$

so that,

$$\left(P_2/P_1\right) = \left(S_2/S_1\right)^{\frac{1}{1-n}}$$

$$P_2 = P_1 \bullet \left(S_2/S_1\right)^{\frac{1}{1-n}}$$

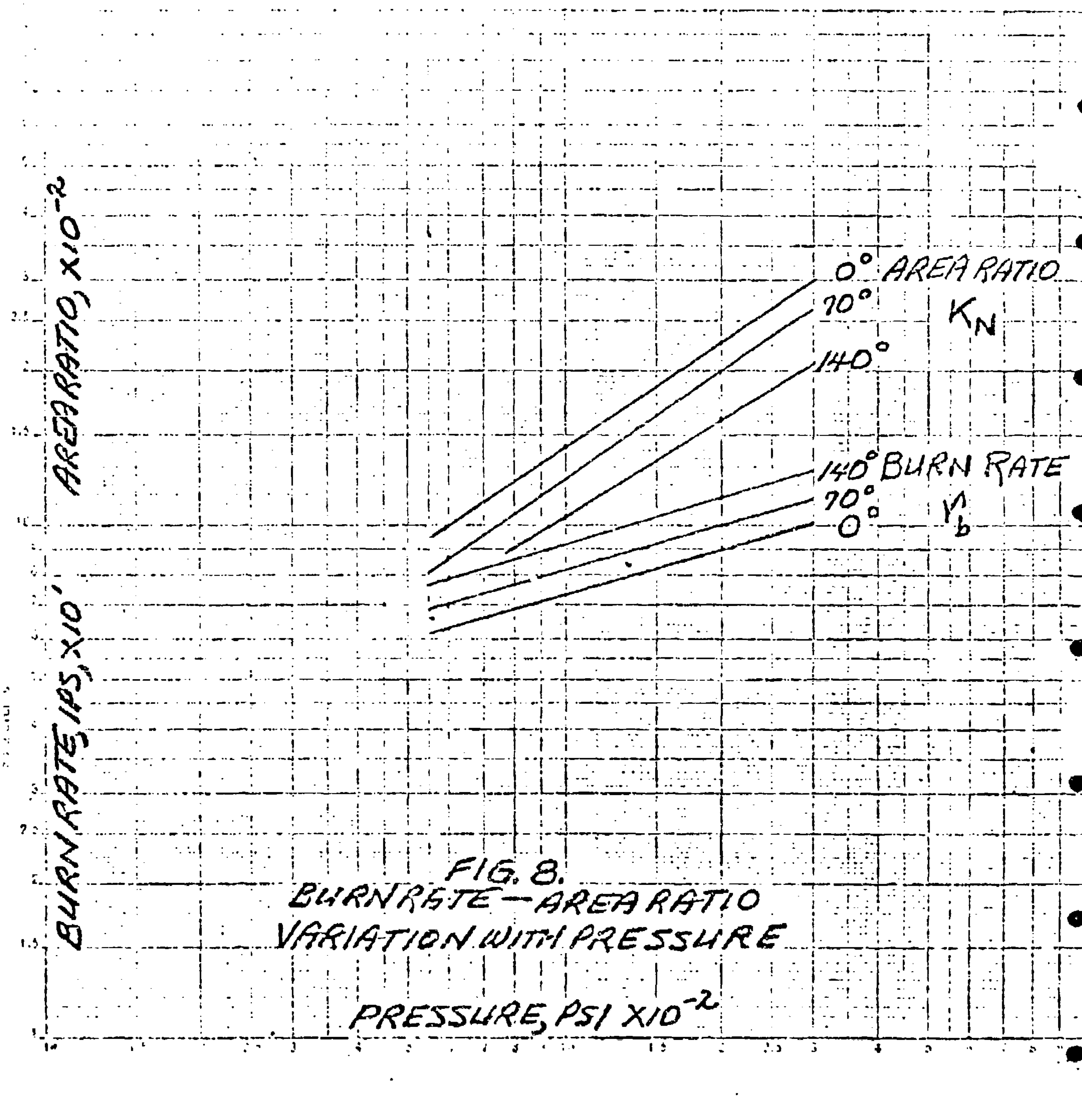

FIG. 8.
BURNRATE—AREA RATIO
VARIATION WITH PRESSURE

As can be noted, as the n term increases approaching unity, the 1/1-n amplification factor grows rapidly. Now for a propellant with n = 0.9 and a 10 percent increase of surface, the pressure is amplified 159 percent to insure a motor blow-up. If n = 0.8, and the burning area is increased by 10 percent, the pressure is increased by 61 percent. The increased burning surface area can result from a cracked propellant grain or possibly uninhibited areas that should have been coated.

Two other important parameters obtained from the K_N and r_b-P_C curves are the temperature sensitivity factors defined as:

$$\pi_k = \frac{1}{P}\left[\frac{\partial P}{\partial T}\right] \quad \text{and} \quad \sigma_p = \frac{1}{r}\left[\frac{\partial r}{\partial T}\right]$$

$$\text{where} \quad \pi_k = \frac{\sigma_p}{(1-n)}$$

so that:

$$P = P_o e^{\pi_k (T-T_o)} \quad \text{and} \quad r = r_o e^{\sigma_p (T-T_o)}$$

both the π_K and σ_p constants have units of % change per °F and show how pressure and burn rate vary with temperature, respectively.

Now if the maximum expected operating pressure, MEOP, for a motor is set at 1500 psi when fired at 165°F, the expected pressures at 70°F and -65°F can be calculated if the π_K term is known. For a π_K = 0.30:

$$P_{70} = P_{165}\, e^{\pi_K (T_{70} - T_{165})} = 1500\, e^{.003\,(70-165)}$$

$$P_{70} = 1500 \cdot .752 = 1128 \text{ psi}$$

and

$$P_{-65} = P_{165}\, e^{\pi_K (T_{-65} - T_{165})} = 1500\, e^{.003\,(-65\,-\,165)}$$

$$P_{-65} = 1500 \cdot .502 = 752 \text{ psi}$$

If the burn rate at 70°F and 1000 psi is known to be 0.5 and σ_p is 0.15 the burn rates at 165°F and -65°F can be readily determined by:

$$r_{165} = r_{70}\ e^{\sigma_p(T_{165}-T_{70})} = 0.5\ e^{.0015\ (165\ -\ 70)}$$

$$r_{165} = 0.5 \cdot 1.153 = 0.577 \text{ in/sec}$$

and

$$r_{-65} = r_{70}\ e^{\sigma_p(T_{-65}-T_{70})} = 0.5\ e^{.0015\ (-65\ -\ 70)}$$

$$r_{-65} = 0.5 \cdot .817 = 0.408 \text{ in/sec}$$

These temperature sensitivity terms are very important to determine the effect on motor performance. Low values are extremely desirable and values of zero for both π_K and σ_p would be ideal and show that temperature has no effect on the propellant performance.

Another important performance characteristic of the propellant formulation is the characteristic exhaust velocity, designated as C*. This term shows the inherent energy content of the formulation independent of the nozzle design and expansion characteristics. It is defined by this equation:

$$C^* = \frac{g\ A_T \int Pdt}{WT_P}\ .$$

For most high energy propellants the characteristic exhaust velocity will vary from about 4700 ft/sec to 5300 ft/sec. Lower energy gas generator propellants are generally cooler and have correspondingly lower values varying from about 3000 ft/sec to 4000 ft/sec. The term is generally measured by firing small test motors and planimetering the pressure-time integral and knowing the throat area, A_T, and the propellant weight, WT_P.

Once the nozzle design and operating characteristics are known so that a thrust coefficient, C_{F_T}, can be calculated, the characteristic exhaust velocity can be converted to the specific impulse, I_{sp}, by:

$$I_{sp} = \frac{C_F\ C^*}{g}$$

This term has the units of lb-sec/lb and can also be determined from:

$$I_{sp} = \frac{\int Fdt}{WT_P}$$

Example:

From the grain design shown in Figure 5 and tabulated in Table 1 and the propellant properties shown below, convert this information to pressure-time and thrust-time performance histories at 70°F. Assume the maximum chamber pressure at 70°F is set at 1100 psi.

Propellant Properties:

$C^* = 5060$ ft/sec	$r = CP^n$
$\rho = 0.063$ lb/in^3	$r = 0.116\, P^{.285}$
$\pi_K = 0.216$ %/°F	$r_{1000} = 0.831$ in/sec
$\sigma_P = 0.154$ %/°F	

First set the throat area so that the maximum surface will allow 1100 psi chamber to be developed. Therefore,

$$A_T = \frac{S \rho C C^*}{g P_C^{1-n}} = \frac{1794 \cdot .063 \cdot .116 \cdot 5060}{32.2 \cdot 1100^{1-.285}}$$

$$A_T = \frac{66339.4}{4813.4} = 13.782 \text{ in}^2$$

$$D_T = \sqrt{\frac{A_T}{.7854}} = 4.189 \text{ in}$$

Now that the throat area is set, the pressures to be developed by the remaining surfaces can be calculated and none can exceed the 1100 psi. From

$$P = \left[\frac{S \rho C C^*}{g A_T}\right]^{\frac{1}{1-n}}$$

only the surface, S, will be varying so that:

$$\left[\rho \frac{C C^*}{g A_T}\right] = \frac{.063 \cdot .116 \cdot 5060}{32.2 \cdot 13.782} = 0.0833$$

$$\text{and} \quad \frac{1}{1-n} = \frac{1}{1-.285} = \frac{1}{.715} = 1.40$$

$P_1 = [1427 \cdot .0833]^{1.4} = [118.9]^{1.4} = 804$ psi

$P_2 = [1476 \cdot .0833]^{1.4} = [123.0]^{1.4} = 843$ psi

$P_3 = [1524 \cdot .0833]^{1.4} = [127.0]^{1.4} = 882$ psi

$P_4 = [1559 \cdot .0833]^{1.4} = [130.0]^{1.4} = 911$ psi

$P_5 = [1579 \cdot .0833]^{1.4} = [131.5]^{1.4} = 926$ psi

$P_6 = [1654 \cdot .0833]^{1.4} = [137.8]^{1.4} = 988$ psi

$P_7 = [1722 \cdot .0833]^{1.4} = [143.4]^{1.4} = 1046$ psi

$P_8 = [1794 \cdot .0833]^{1.4} = [149.4]^{1.4} = 1100$ psi

$P_9 = [234 \cdot .0833]^{1.4} = [19.5]^{1.4} = 64$ psi

Now it is best to start tabulating these calculated figures so that:

dw, in	S, in^2	P, psi	r_b, IPS	(A) $\overline{r}_b$, IPS	t_b,sec	(B) $\sum t_b$, sec
0	1427	804	0.781	-	0	0
0.5	1476	843	0.791	0.786	0.64	0.64
1.0	1524	882	0.802	0.796	0.63	1.27
1.5	1559	911	0.809	0.805	0.62	1.89
2.0	1579	926	0.813	0.811	0.62	2.51
2.5	1654	988	0.828	0.820	0.61	3.12
3.0	1722	1046	0.841	0.834	0.60	3.72
3.65	1794	1100	0.854	0.847	0.77	4.49
4.05	234	64	0.380	0.617	0.65	5.14

The burn rate is calculated for these specific pressure points from:

$$r = 0.116P^{.285}$$
$$r_1 = 0.116 \cdot 804^{.285} = 0.781 \text{ in/sec}$$
$$r_2 = 0.116 \cdot 843^{.285} = 0.791 \text{ in/sec}$$
$$r_3 = 0.116 \cdot 882^{.285} = 0.802 \text{ in/sec}$$
$$r_4 = 0.116 \cdot 911^{.285} = 0.809 \text{ in/sec}$$
$$r_5 = 0.116 \cdot 926^{.285} = 0.813 \text{ in/sec}$$
$$r_6 = 0.116 \cdot 988^{.285} = 0.828 \text{ in/sec}$$
$$r_7 = 0.116 \cdot 1046^{.285} = 0.841 \text{ in/sec}$$
$$r_8 = 0.116 \cdot 1100^{.285} = 0.854 \text{ in/sec}$$
$$r_9 = 0.116 \cdot 64^{.285} = 0.380 \text{ in/sec}$$

(A) $\overline{r}_b$: Average burn rate, in/sec

(B) $\sum t_b$: Accumulated burn time, sec.

These burn rate data points are tabulated with the web burnback distances and averaged so that by knowing the average burn rate and distance burned, burn times can be calculated and accumulated burn times. Therefore, the pressure-time performance can be graphed as shown in Figure 9.

The thrust-time history can now be calculated from the pressure-time data points using the equation below:

$$F = \eta \lambda C_{F_T} A_T P_C$$

At this time additional nozzle design data can be determined. From the average pressure determined from Figure 9, the chamber to atmospheric pressure ratio can be determined as about 68 (1000/14.7) and from Figure 7 the coefficient of thrust and expansion ratio can be determined for optimum expansion. These values are found to be: $C_{FT} = 1.60$ and $A_E/A_T = 9.0$. Since the throat area was found to be 13.782 in^2 earlier, the exit area should be $9.0 \cdot 13.782 = 124.0$ in^2 or a diameter of 12.6 in. Now if the nozzle is designed with an 18° half-angle, the thrust can be estimated as:

$$F = 0.94 \cdot 0.976 \cdot 1.60 \cdot 13.782 \cdot P_C$$
$$F = 20.23\ P_C$$

and the data tabulated as:

dw, in	Σt, sec	P, psi	F, lbs
0	0	804	16265
0.5	0.64	843	17054
1.0	1.27	882	17843
1.5	1.89	911	18430
2.0	2.51	926	18734
2.5	3.12	988	19988
3.0	3.72	1046	21161
3.65	4.49	1100	22254
4.05	5.18	64	1295

These thrust-time data points can now be graphed as shown in Figure 10.

At this time all the important motor performance characteristics have been estimated for a 70°F motor test. These characteristics are summarized below:

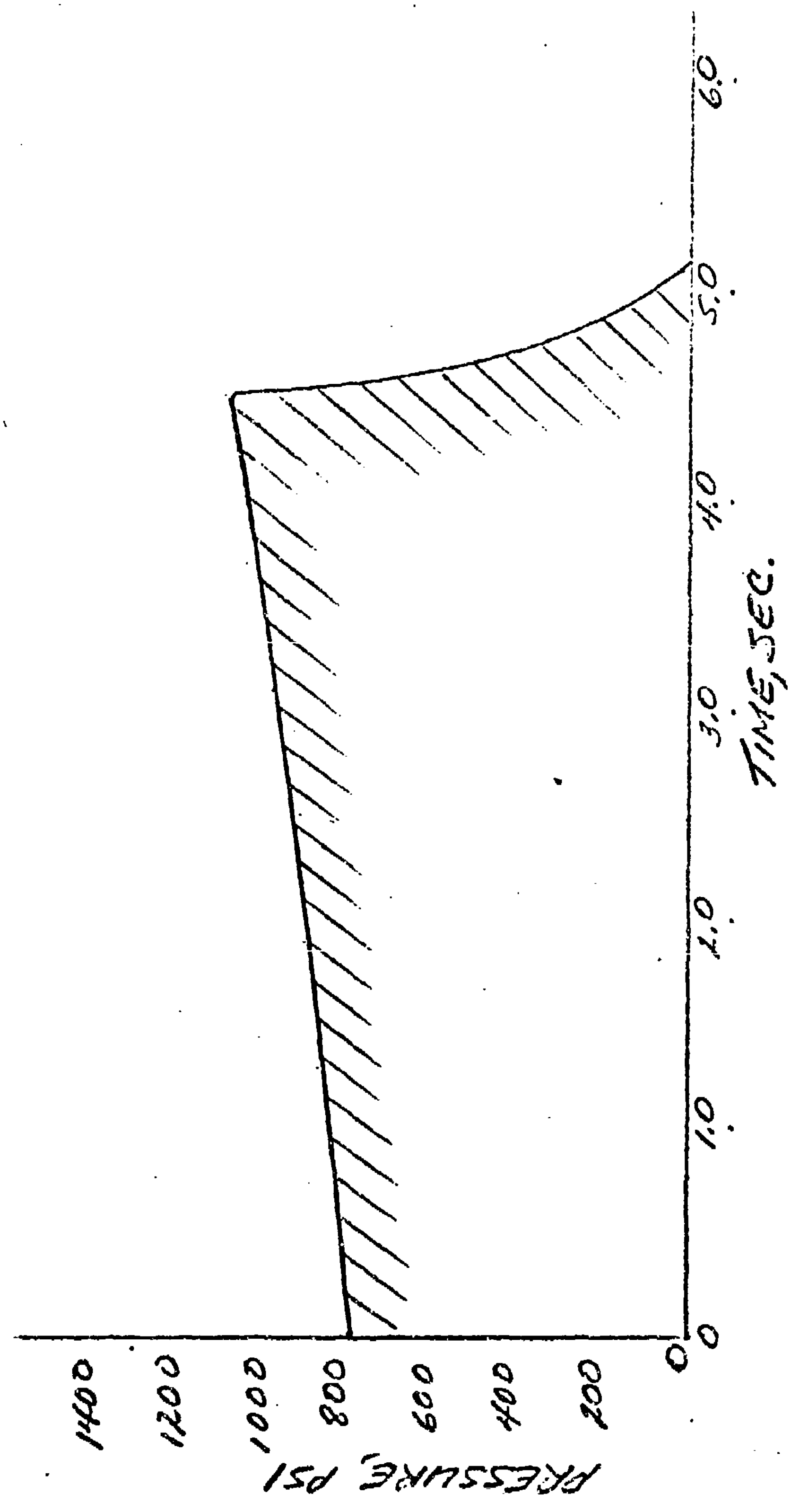

FIG. 9.
PREDICTED PRESSURE-TIME HISTORY

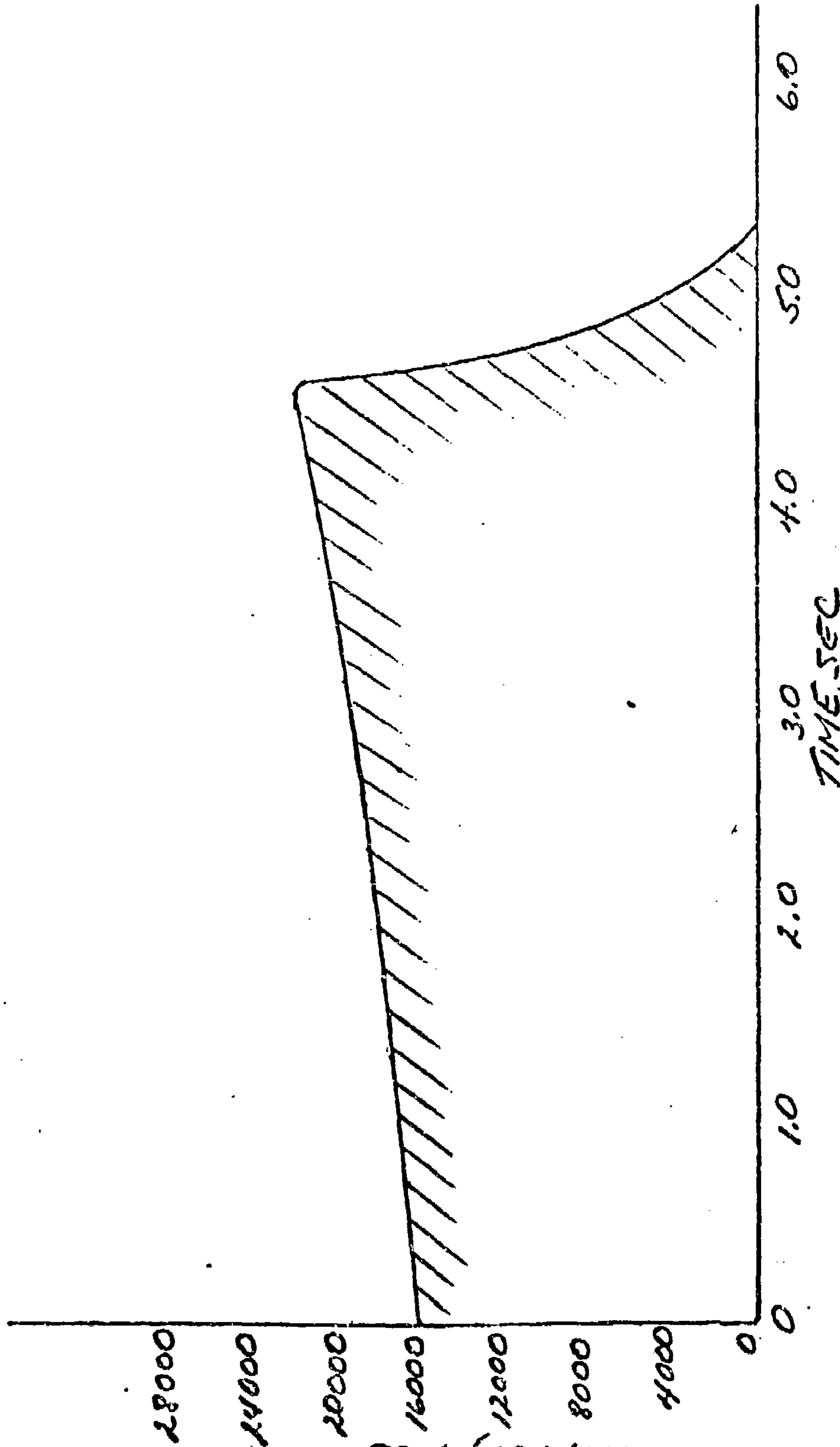

FIG. 10.
PREDICTED THRUST-TIME HISTORY

Rocket Motor Performance Data

Total impulse, lb-sec	95250
Integrated pressure, psi-sec	4615
Burn time, sec	4.37
Action time, sec	4.70
Maximum pressure, psi	1100
Average pressure, psi	1000
Maximum thrust, lb	22254
Average thrust, lb	21167
Specific impulse, lb-sec/lb	241.4
Propellant weight, lb	396.0

From the information supplied, it should be possible to calculate the pressure-time and thrust-time performance histories at both +165°F and -65°F.

Once the motor performance has been estimated, surface area data refined, and propellant characteristic data firm, the information can be programmed for future performance information.

A summarization of many of the motor performance equations is included in Table 3 and Figure 11 shows various performance characteristics.

MOTOR COMPONENTS

From earlier discussions and the general motor drawing as shown in Figure 1, it is obvious that a considerable number of inert components are necessary in the rocket motor. These components must be able to withstand high temperature and pressure loads and yet be lightweight and safe. Large safety factors can be utilized in many fields of engineering to increase safety, but for flightweight components the weight must be minimized. These inert components will account for 15-20 percent of the total motor weight and should be minimized. Extremely high performance motors will reduce the inert weight to about 5 percent. These high propellant loading fraction motors generally produce the highest burn-velocity missiles. A few of these components will be discussed:

ROCKET ENGINE FORMULAS

	A_t Throat area in.^{2}	c Exhaust velocity ft/sec	c^* Characteristic exhaust velocity ft/sec	C_F Thrust coefficient —	F Thrust lb	I_{sp} Specific impulse lbf-sec/lbm	P_c Chamber pressure lb/in.^{2}	$\dot{W}$ Weight flow rate lb/sec
$A_t =$	—	$c\frac{\dot{W}}{P_c C_F g}$	$c^*\frac{\dot{W}}{P_c g}$	$\frac{1}{C_F}\frac{F}{P_c}$	$F/C_F P_c$	$I_{sp}\frac{\dot{W}}{P_c C_F}$	$\frac{1}{P_c}\frac{F}{C_F}$	$\dot{W}\frac{c^*}{P_c g}$
$c =$	$A_t\frac{P_c C_F g}{\dot{W}}$	—	$c^* C_F$	$C_F c^*$	$F\frac{g}{\dot{W}}$	$I_{sp} \times g$	$P_c\frac{C_F A_t g}{\dot{W}}$	$\frac{1}{\dot{W}}(Fg)$
$c^* =$	$A_t\frac{P_c g}{\dot{W}}$	c/C_F	—	$\frac{1}{C_F}c$	$F\frac{g}{C_F \dot{W}}$	$I_{sp}\frac{g}{C_F}$	$P_c\frac{A_t g}{\dot{W}}$	$\frac{1}{\dot{W}}\frac{Fg}{C_F}$
$C_F =$	$\frac{1}{A_t}\frac{F}{P_c}$	c/c^*	$\frac{1}{c^*}c$	—	$F/A_t P_c$	$I_{sp}\frac{g}{c^*}$	$\frac{1}{P_c}\frac{F}{A_t}$	$\frac{1}{\dot{W}}\frac{Fg}{c^*}$
$F =$	$A_t P_c C_F$	$c\frac{\dot{W}}{g}$	$\frac{c^* C_F \dot{W}}{g}$	$C_F A_t P_c$	—	$I_{sp}\dot{W}$	$P_c C_F A_t$	$\dot{W} I_{sp}$
$I_{sp} =$	$A_t\frac{C_F P_c}{\dot{W}}$	c/g	$c^*\frac{C_F}{g}$	$C_F\frac{c^*}{g}$	$F/\dot{W}$	—	$P_c\frac{C_F A_t}{\dot{W}}$	$\frac{1}{\dot{W}}F$
$P_c =$	$\frac{1}{A_t}\frac{F}{C_F}$	$c\frac{\dot{W}}{C_F A_t g}$	$c^*\frac{\dot{W}}{A_t g}$	$\frac{1}{C_F}\frac{F}{A_t}$	$F/C_F A_t$	$I_{sp}\frac{\dot{W}}{C_F A_t}$	—	$\dot{W}\frac{c^*}{A_t g}$
$\dot{W} =$	$A_t\frac{P_c g}{c^*}$	$\frac{1}{c}\times(Fg)$	$\frac{1}{c^*}\frac{Fg}{C_F}$	$\frac{1}{C_F}\frac{Fg}{c^*}$	F/I_{sp}	$\frac{1}{I_{sp}}F$	$P_c\frac{A_t g}{c^*}$	—

TABLE 3

VOLUME/WEIGHT LIMITED

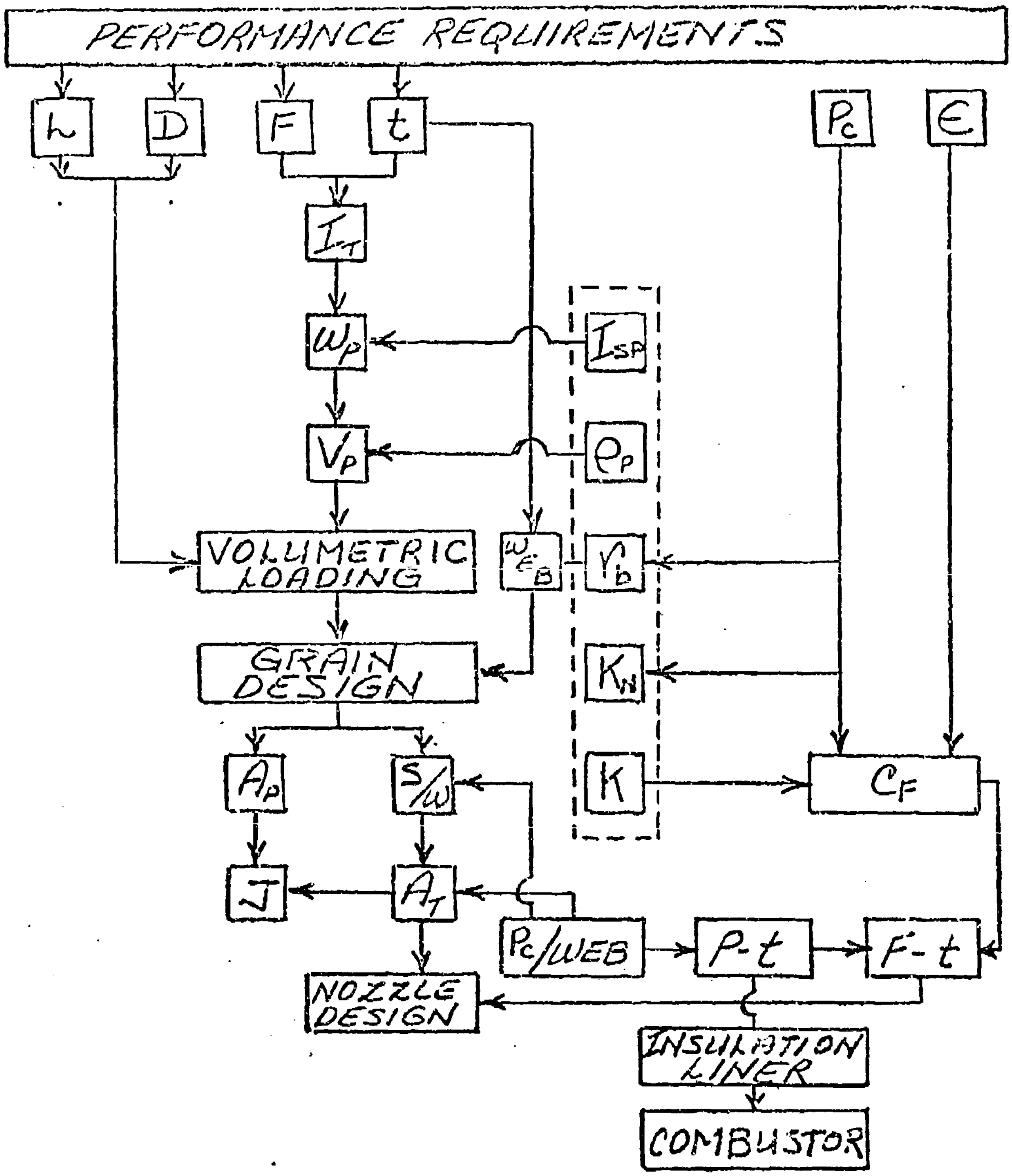

FIG. 11.
CALCULATION SEQUENCE

Combustor:

The combustor is a thin-walled cylindrical pressure vessel with a head closure at one end and a nozzle at the other. Case wall thickness must be sufficient to withstand the maximum expected operating pressures when the case is at the maximum expected temperature. The wall thickness can be estimated from the yield stress of the metal selected for fabrication of a case of radius R:

$$t_w = \frac{P_{yield}\, R}{\sigma_{y_{min}}}$$

in this equation the P_{yield} is about 20 percent greater than the nominal value calculated at the maximum operating temperature. This value can be estimated from:

$$P = \left[\frac{S\,\rho\,C\,C^*}{g\,A_T}\right]^{\frac{1}{1-n}}$$

where the average plus one standard deviation value is used for ρ, C, and C*. The average less one standard deviation is used for A_T. From typical experience this becomes:

$$P_{MEOP} = \left[\frac{1.\cdot 1.002\cdot 1.05\cdot 1.005}{1.\cdot .999}\right]^{\frac{1}{1-.6}} = [1.0584]^{2.5}$$

P_{MEOP} = 1.15 or 15% above nominal performance at the maximum temperature due to variations of the propellant properties and fabrication of the nozzle. A set of arbitrary pressure design factors used for booster motor design is summarized below:

P_{NOM} = 1224 psi	P_{NOM}/P_{NOM} = 1.00
P_{MEOP} = 1400	P_{MEOP}/P_{NOM} = 1.14
P_{HST} = 1500	P_{HST}/P_{NOM} = 1.22
P_{YIELD} = 1680	P_{YIELD}/P_{NON} = 1.37
P_{FAIL} = 2100	P_{FAIL}/P_{NOM} = 1.72

The metal case material, Inconel 718, has a minimum yield of 150,000 psi and a minimum ultimate of 190,000 psi. Therefore, for this case a wall thickness estimate is:

$$t_w = \frac{P_y R}{\sigma_{Y_{MIN}}} = \frac{1680\ 7.5}{150,000}$$

$$t_w = 0.083 \text{ in.}$$

Stress distributions in a combustor are extremely complex and this equation can only be expected to produce approximate results. It does not consider local loads at attachment points, bending loads, or thermal stresses. For these reasons, reasonable safety factors must be utilized supplemented by hydrostatic testing to demonstrate an acceptable design. While designing for safety, excessive safety factors should not be used since weight must be saved as the unit must fly.

Case Materials:

Since weight is a critical factor in motor design, steel is mainly used in construction due to the high ultimate and yield strengths with moderate densities. But, material selection is generally based on combustor design and motor application.

Aluminum can often be used as a material of construction for small diameter cases where the wall is not directly exposed to the propellant flames. Due to the differences in strength, the wall thickness for an aluminum case is three times that of a steel wall. Therefore, for a small motor case fabricated from steel may be 0.020 in. thick and an aluminum case would be 0.060 in. Fabrication of case walls having thicknesses less than 0.050 in. will be extremely difficult using ordinary techniques. Hence, combustors for large motors are often fabricated from steel and small diameter motors utilize aluminum since it's easier to maintain closer tolerances with thicker case walls. The main disadvantage to using aluminum is that the strength rapidly decreases with increasing temperature.

Occasionally pressure vessels are fabricated from fiberglass filaments bonded with epoxy resins to produce a high strength material with a very low density. Strength to density ratio may be as high as 1.3×10^6 in; whereas, for steel it may only be about 0.6×10^6 in. Limitation of the fiberglass is also the rapid loss of strength with increasing temperature.

For combustors that are required to operate for extended durations, high strength high temperature materials will be required. This situation may be encountered for ramjets operating for about 180-240 seconds. Two materials are often selected for this application: a nickel base alloy designated as INCONEL, or a cobalt base material. Both materials, I-718 nickel base and L-605 cobalt base, were evaluated for potential application in an integral rocket/ramjet application. Though both materials showed desirable high strength high temperature properties, the I-718 was selected due to the better machinability characteristics.

Table 4 shows some candidate case materials and the physical properties.

TABLE 4. Case Materials and Physical Properties

Material	Tensile Ultimate lb/in^2	Density lb/in^3	Tensile/Density in
4130 Steel	180,000	0.283	637,000
4140 Steel	220,000	0.283	778,000
7075-T6 Al	60,000	0.100	600,000
2024-T4 Al	50,000	0.100	500,000
I-718 Ni base	162,000	0.298	544,000
L-605 Co base	155,000	0.328	473,000
Fiberglass (filament/epoxy)	90,000	0.070	1,300,000

The extent to which the strength decreases as the temperature increases for some of these materials is shown in Figure 12.

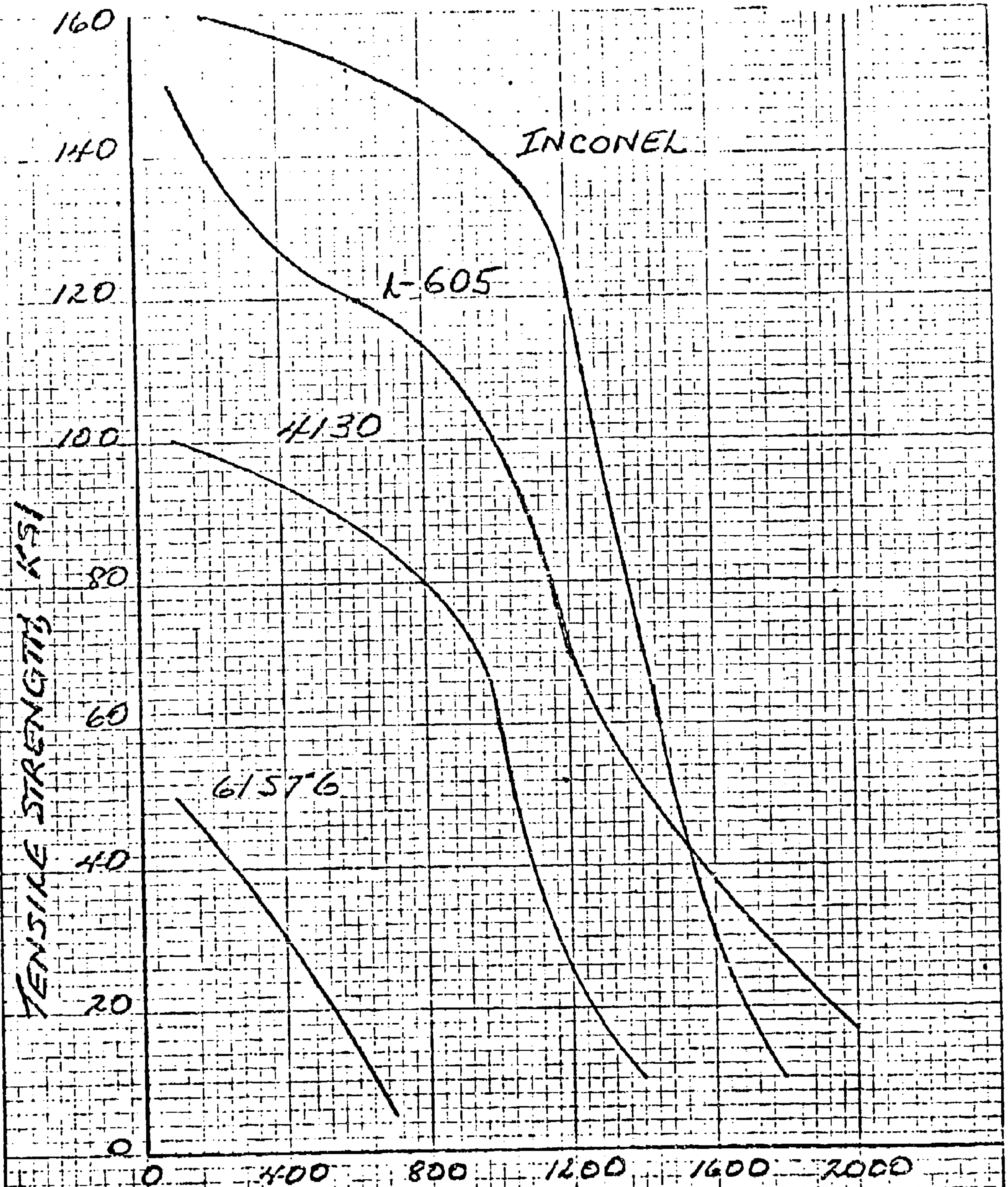

FIG. 12.

STRENGTH VARIATION WITH TEMPERATURE

Insulation Materials:

One of the more serious problems in rocket motor design is the high flame temperatures developed in the combustor from burning high-energy propellants. The problem of protecting metal parts from temperatures that exceed the melting point of the material has received considerable attention in order to prevent motor failure due to excessive heating of the combustor. This protective material is designated as insulation and often completely coats the surface of the component being protected. In Figure 1 it is shown as the material that completely lines the internal surface of the combustor. An ideal heat-resistant material, insulation, should be lightweight, occupy a minimum volume, resist chemical degradation, and resist thermal stresses.

Solutions to the high temperature problem are generally approached along one of two paths: (1) ablation or (2) heat-sink. A brief description of each technique and some candidate materials tabulated below.

Ablative materials:

Materials for ablative cooling either sublime, decompose, or melt after absorbing a certain amount of heat. Mass flow from the ablative surface reduces the amount of heat transferred to the motor wall. The heat of ablation is a function of heat-flux measured in BTU/FT^2-sec; each material has a minimum heat-flux value below which ablation will not occur. Above this value, the higher the stagnation temperature, the greater will be the rate of energy dissipation. Many insulating materials are available. In general, it has been determined that organic-fiber-reinforced plastics can withstand higher temperatures for longer durations than asbestos or glass reinforced plastics. A relatively new insulator is a silicone elastomer based compound, designated as DC 93-104. This material has been found effective in both a rocket and a ramjet motor environment. A partial list of useful ablative materials is summarized in Table 5.

Heat Sink Materials:

Heat sink materials may take one or a combination of the following forms:

TABLE 5. Ablative Material Summary

Material	Heat flux, Btu/sq ft/sec
Glass-phenolic fiber[1]	100
Nylon-phenolic fiber[1]	100
Pure silica-phenolic fiber[1]	1,000
Graphite[1]	2,000 (sublimes)
Quartz[1]	2,000 (sublimes)
Teflon[1]	100
Silicone elastomers[2]	760

[1] Oxy-acetylene flame torch.
[2] Oxy-kerosene flame torch.

1. Greater than normal thickness of structural components.

2. Lining layer of non-structural material with high heat capacity.

3. Liquid coolant circulating adjacent to the structural shell internal surface.

The usefulness of heat-sink devices is confined to a narrow heat-flux range of 0.1 to 10 BTU/Ft^2-sec. Table 6 is a partial list of various heat-sink materials in use.

An "In-Use" combustor is presently insulated with about 0.080 in. of zirconia applied by the plasma spray technique. Though a good insulator, the ceramic material is quite brittle and easily damaged. Therefore, the DC 93-104 silicone elastomer is being considered as a replacement. This material can be readily spun in place and is quite pliable; unfortunately, about 0.375 in. is required to insulate the combustor which reduces the propellant weight by about twenty

TABLE 6. Heat-Sink Materials Summary

Material	Specific heat, Btu/lb/°F	Temperature range, °F
Metals		
Beryllium	0.49 - 0.78	200 - 1,800
Magnesium	0.25 - 0.31	200 - 1,200
Aluminum	0.23 - 0.27	200 - 1,000
Chromium	0.12 - 0.18	200 - 2,000
Copper	0.09 - 0.12	200 - 1,800
Molybdenum	0.06 - 0.07	200 - 2,000
Ceramics		
Beryllia	0.30 - 0.52	200 - 1,800
Magnesia	0.25 - 0.29	400 - 1,800
Aluminia	0.22 - 0.30	200 - 2,000
Quartz	0.18 - 0.29	200 - 2,000
Zirconia	0.12 - 0.14	200 - 1,600
Reinforced Plastics		
Asbestos-phenolic	0.31 - 0.36	100 - 500
Glass-phenolic	0.[illegible]5 - 0.29	100 - 500

pounds. This represents about 5 percent of the total propellant weight. Asbestos-phenolic insulation was investigated but rejected since hand layup would require considerable time and expense.

In general, there are no accepted equations that can be used for selecting the most desirable insulating material. The heat distribution in a combustor is quite complex and often an expert's opinion is needed and this is followed by extensive testing with well thermocoupled units. In addition, not only thermal properties but physical properties and the ease of insulating the components must be considered.

One technique to reduce the insulation problem is to design the motor with a case bonded grain. This technique reduces the area of the combustor exposed to the propellant flame to minimize the possibility of case burn-through. Figure 1 shows this technique to keep the hot gases from the combustor wall.

Stress Release Boots:

The propellant grain is subjected to a number of stresses that must be identified and compensated for. The principle stresses are due to pressure gradients, flight acceleration, shrinkage during polymerization of the fuel-binder, and differential expansion, during thermal cycling, between the case and propellant bonded to it.

Propellant grains are cast at elevated temperatures and allowed to cool and polymerize. The resultant shrinkage develops residual stresses in the propellant charge that are not relieved by plastic deformation. These stresses can result in bond failure between propellant and liner or case to liner. If liner or the bond is high strength, the propellant charge will probably fail by cracking down the base of the star point. The result can be a catastropic failure due to the increased burning surface area.

A technique to overcome this weakness is shown in Figure 1 and is called the "stress release boot". The boot as shown is fabricated as two individual items and bonded out at the maximum diameter area. Each piece is about 0.100 in. thick and made from a rubbery flexible material. Present booster motors utilize nitrile-butadiene rubber filled with hydrated silica designated as GEN GUARD V-45. In addition, boots fabricated from DC 93-104, a silicone elastomer, have been successfully utilized.

The outside boot is bonded to the case or insulation and the inside boot is bonded to the propellant with the liner. When first cast, the split in the release boot is closed, but, as the propellant cools and polymerizes, the split opens to release these grain stresses and minimize the possibility of grain failure. The release boots also allow the rocket motor to be cycled over much greater temperature ranges than would otherwise be possible.

Though boots are desirable, they should be minimized in size since they reduce the volume inside the combustor allocated for propellant use. Often, the relief boot can be a single layer with the insulation serving as the outer boot. In addition, if the length-to-diameter is small enough, a boot at one end may be sufficient to reduce grain stresses to an acceptable level.

Liner Materials:

Liner materials serve two specific functions: (1) insures a good interface bond of propellant to boot and insulation and (2) inhibits burning so that propellant burning only occurs on the unlined surfaces. In general, the liner material is the bonding matrix or fuel material in the propellant. This insures a good propellant to liner bond.

It is desirable that the propellant to liner bond be stronger than any other bond. Therefore, if thermal or mechanical stresses are excessive, the bond failure will not occur between liner and propellant exposing new propellant surfaces where burning can occur. When new uncontrolled areas become exposed and burning is initiated, catastropic failures from motor blow-ups usually occur.

The liner material is generally applied to the cylindrical section of the combustor by pouring in a predetermined weight of material and spinning on rollers for uniform distribution. On each end, the boots are lined by painting the material on to a predetermined thickness. The spun section thickness can be as thin as 0.020 in. or as thick as 0.060 in.; these thicknesses can be controlled within about 0.005 in. In general, thinner liners produce greater bond strengths and increase the propellant loading fractions. Therefore, the thin liners are much preferred and it is extremely desirable to have both liner and propellant fuel from the same materials.

Motor Nozzle Design:

The nozzle is that part of a combustor which serves to direct and accelerate the combustion gases to high velocities. A maximum thrust coefficient consistent with minimum nozzle weight is the desired goal of the designer. A general view of a solid propellant motor is shown in Figure 1 and the graph to estimate the theoretical coefficient of thrust is included as Figure 7.

The nozzle throat area controls the chamber pressure; whereas, the pressure ratio (P_C/P_A), area ratio (A_E/A_T), and the nozzle divergent half angle (α) control the thrust amplification through the coefficient of thrust. Most nozzles are of conical contour since they can be easily designed and readily fabricated. Bell-shaped type nozzles have been developed as a technique to improve the thrust coefficient while reducing weight but the complexity of the curves for designing and fabricating reduce the desirability of this contour.

Most nozzles have half-angles (α) around 10° to 20° to achieve the desired expansion area ratio within a reasonable length. The convergent section can have almost any symmetrical well-rounded shape with a radius approximately equal to the throat diameter that faires into the expansion cone. The throat area should have a short flat section about 5 percent of the throat diameter to maintain the throat diameter during motor operation. Occasionally nozzles will be partially or totally submerged within the combustor in order to achieve the required expansion within a limited overall length.

Solid rocket nozzles are rarely cooled and, therefore, must retain shape and strength at extremely high propellant combustion temperatures. Because of the severe environment, very few materials function properly. Steel and copper have been successfully employed in many applications; steel for a short and copper for long burn times. In general copper is prohibitive due to the excessive weight. Steel is acceptable for short durations; but as the burn times are extended, thin coatings of zirconia (ZrO_2) or alumina (Al_2O_3) flame sprayed on the surface increase erosion resistance with high performance propellants. Coatings should not exceed 0.030 in. as excessively thick sections will flake, spall, or crack due to variations in the coefficients of thermal expansion.

The use of graphite as a small throat insert or as a full throat section is extremely advantageous especially where the heat transfer is greatest.

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Because of the high thermal conductivity and low thermal coefficient of expansion, graphite resists failure by thermal expansion. It has also been determined that the tensile strength of graphite increases up to 5000°F. At that temperature, and at 55,000 psi stress, the material displays an elongation of 53 percent. Two materials used on booster motor programs and found acceptable were H205 and ATJ graphite.

Figure 13 shows a booster motor nozzle which was designed for about 60 percent submergence within the combustor. After preliminary calculations were completed on this design, a detailed stress analysis was conducted using the finite element stress technique for axisymmetric solids with orthotropic temperature dependent material properties. The computer program, designated as SAAS II, is capable of predicting total stresses resulting from thermal and pressure loads. Results indicate marginal areas where design modifications may be necessary. Nozzle weight was found to be proportional to the total impulse and can be estimated from:

$$WT_N = 2.5 \times 10^{-4} \int Fdt$$

where $\int Fdt$, impulse, is given in lb-sec and WT_N is in lbs. Though these guidelines will help, nozzle design for solid propellant rocket motors is largely emperical.

INTERNAL BALLISTICS COMPUTER PROGRAM

The Internal Ballistics Computer Program (420004) is a proprietary program obtained from the Lockheed Propulsion Company. This program calculates the pressure-time and thrust-time histories of a motor including accurate theoretical calculations of both ignition and tail-off transients. The program tape number is 6369 and has been used on many local booster motor performance predictions and has been found completely satisfactory.

Main assumptions made to obtain this computer program were:

1. Combustion products are ideal gases.

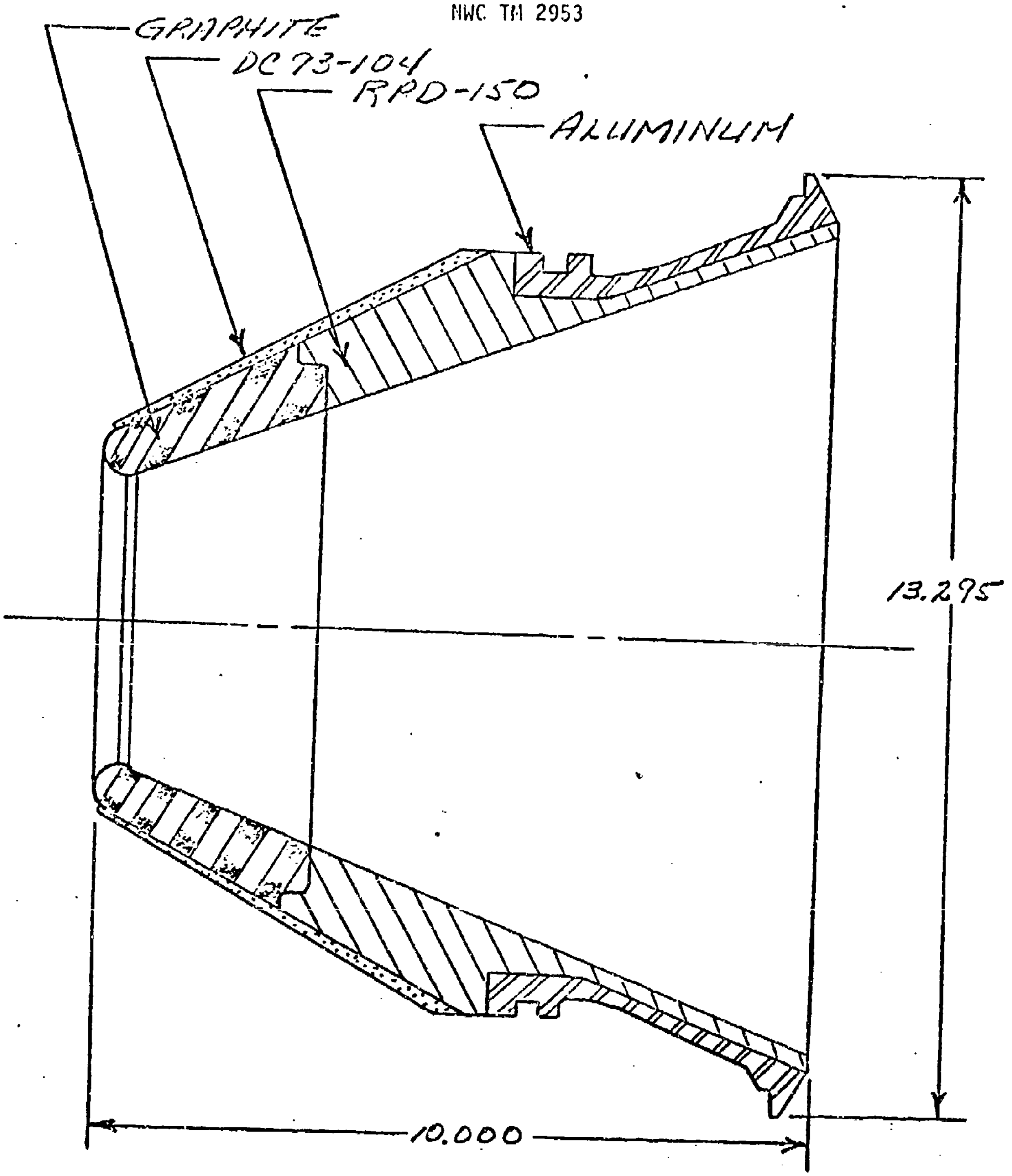

FIG. 13.
NOZZLE DESIGN

2. Burning rate follows Vieille's law, $r = CP^n$, from zero to maximum pressure.

3. Effects of mass addition and erosive burning can be ignored.

4. The combustion chamber gases have negligible inertia.

5. Variation of C* with chamber pressure follows the relationship:

$$C^* = C^*_{1000} \, (P/P_{1000})^x$$

6. Propellant burn rate can be corrected for ambient temperature (other than 70°F reference temperature) of the propellant by the relation:

$$r = r_{REF} e^{0.01 \, \sigma_p (T-70°)}$$

where

$$\sigma_p = (1-n)\pi_K$$

In order for the program to be used, the motor must be designed and the propellant properties known. The information supplied to the computer falls into two main categories:

I. Surface area and web burn-back in as many increments as desired.

II. Motor design and propellant properties:

1. Area, throat, in^2
2. Area, exit, in^2
3. Volume, motor, in^3
4. Temperature, ambient, °F
5. Pressure, ambient, psi
6. Pressure, blowout, psi
7. Lambda, λ
8. Discharge, coefficient, C_o
9. Gamma, K

10. Erosion, throat, in/sec
11. Burn rate, in/sec
12. Exponent, pressure, n
13. Temperature sensitivity, π_K, %/°F
14. Characteristic exhaust velocity, C*, ft/sec
15. Density, ρ, lb/in^3
16. Ignition time, sec
17. Burn reference pressure, psi

With this amount of information required as input data, it is obvious that the motor must be mainly designed and the propellant formulation selected. The program output is subdivided into three main tables of tabulated data as summarized below:

Table I:

1. Time
2. Pressure
3. Pressurization rate
4. Free volume
5. Propellant weight consumed
6. Propellant consumption rate
7. Insulation weight ablated
8. Insulation ablation rate
9. Current throat area values

Table II:

1. Time
2. Web burn-back
3. Propellant burning area

These values are greatly expanded by interpolating between the values supplied from the input data.

Table III:

1. Time
2. Pressure
3. Thrust
4. Integrated pressure
5. Integrated thrust
6. Weight consumed
7. Current coefficient of thrust values

In addition to these three tables of information, the program also handles nozzle flow separation effects and denotes the times at which the flow is separated by an "S" in the extreme right hand column. Concurrently with the tabulated data, pressure-time and thrust-time performance characteristics are graphed for rapid visual inspection.

How the pressure and thrust performance varies with variations of ballistic and fabrication components, in addition to temperature is included in Figure 14.

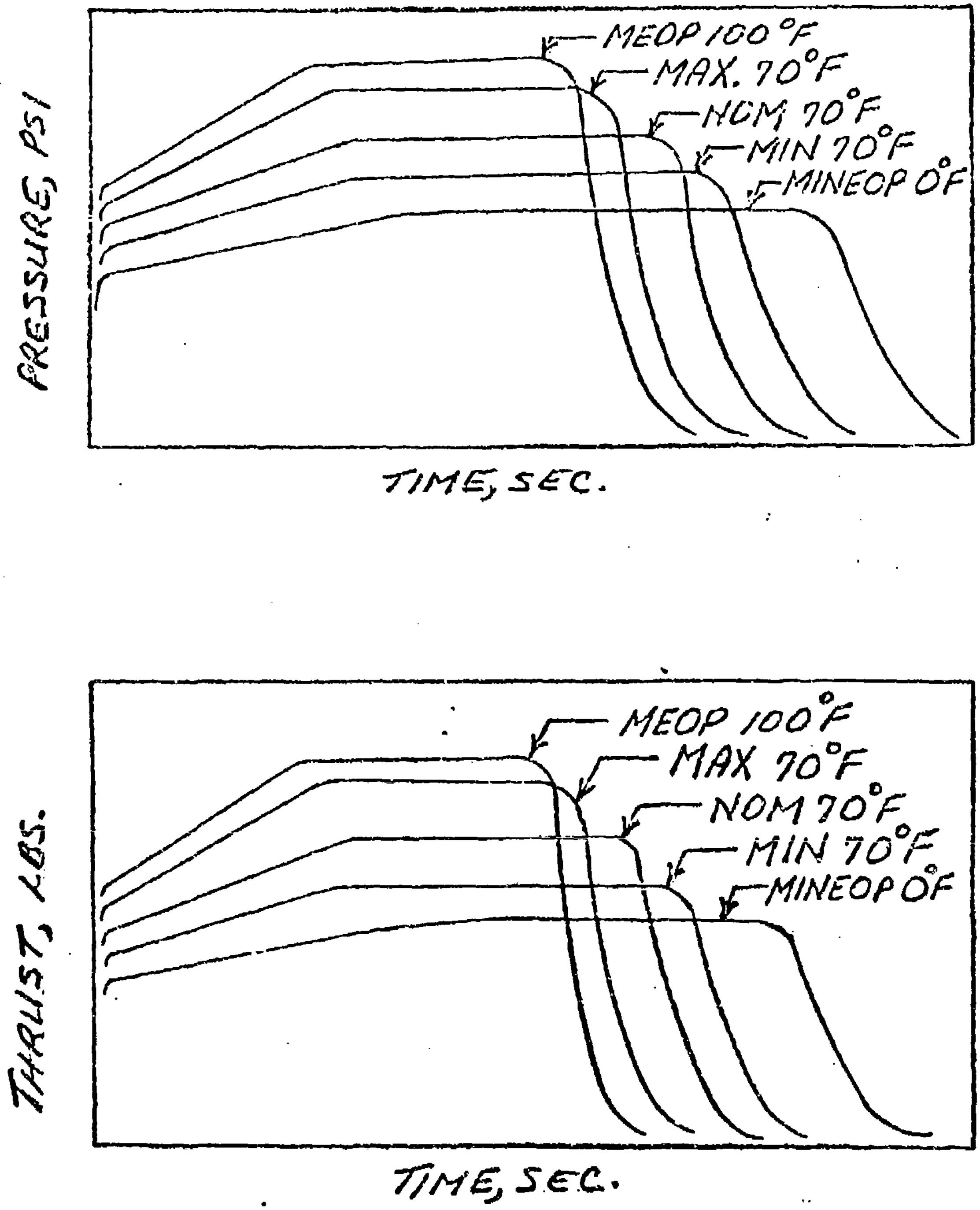

FIG. 14.
PERFORMANCE VARIATIONS

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