J O U R N A L

OF THE

AMERICAN ROCKET SOCIETY

C O N T E N T S

Thermochemistry of Rocket Propellants*

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**To be presented at the annual American Rocket Society — ASME Convention,
Atlantic City, December, 1947**

*Based on Chapter IV of the author's forthcoming book, "Rocket Propulsion Elements", J. Wiley & Sons, 1948.

Rocket motors utilize the heat liberated in the combustion of a fuel with an oxidizing agent as a source of energy The nature of this combustion process determines the performance of rocket motors and the following discussion will examine the general thermochemical aspects.

The method commonly accepted for determining the theoretical performance of rocket motors wtih different propellants will be outlined. Thereafter the computed results for various propellant combinations will be given and discussed.

Experiments have shown that theoretical values are usually less than ten percent above the actual performance of rocket motors. This excellent agreement is a justification for these rather elaborate thermochemical computations.

Although thermochemical calculations permit the evaluation of the performance of rocket propellants and the selection of an optimum propellant mixture ratio, it is necessary to consider carefully other significant propellant properties and design parameters when choosing a propellant combination for a particular rocket application.

Figure 1 shows a schematic diagram of a typical rocket motor. The combustion occurs between the plane of injection (1) and the entrance to the nozzle (2). The thermal energy of the hot combustion products is converted into kinetic energy by expanding the gases from the chamber pressure to the surrounding pressure in a supersonic nozzle.

PERFORMANCE PARAMETER

The rocket motor exhaust velocity has been selected as a performance parameter. It is defined as

$$V = \sqrt{\frac{2gkR}{k-1} \frac{T}{M}\left[1-\left(\frac{p_3}{p_1}\right)\cdot\frac{k-1}{k}\right]}$$

$V =$ rocket motor exhaust velocity (ft/sec)

$g =$ acceleration of gravity (32.2 ft/sec)

$k =$ specific heat ratio

$T =$ combustion chamber temperature (°R)

$M =$ mean molecular weight (lb/mole)

$p_3 =$ nozzle exit pressure $=$ external pressure for optimum expansion (lb/sq. in.)

$p_1 =$ chamber pressure (lb/sq. in.)

$R =$ universal gas constant (1544 ft. lb./°F mole)

The theoretical exhaust velocity can only be caluculated if the following thermochemical quantities are previously determined: combustion temperature, molecular weight of combustion products and specific heat ratio.

ASSUMPTIONS

The basic principles and assumptions in calculating rocket combustion characteristics are essentially the same as those involved in determining the combustion equilibrium of a conventional fuel with air.

1. The perfect gas laws are sufficiently accurate to describe the actual combustion gas conditions.

2. A gas mixture can be treated as a homogeneous perfect gas.

3. The combustion process proceeds at constant pressure, namely, the rocket motor chamber pressure (usually between 200 and 800 psi).

4. There is no heat transfer between the hot gases and the rocket motor chamber walls.

5. The injection and mixing of the oxidizing agent and the fuel is perfect and the mixture ratio is constant across the rocket motor chamber section.

6. Complete chemical equilibrium is attained in the short time during which the individual propellant molecules remain within the combustion chamber.

LIQUID PROPELLANT ROCKET MOTOR
SCHEMATIC DIAGRAM

FIG. (1)

METHOD OF CALCULATION

Before explaining the method of calculation, the following basic quantities such as heat of formation, heat of reaction, and chemical equilibrium, are defined.

The heat of formation is defined as the change in enthalpy which results when a compound is formed from its elements isothermally and at constant pressure.

The heat of reaction is defined as the change in enthalpy which occurs when products are formed from reactants at a constant reference temperature and pressure. This enthalpy change may either be positive or negative depending on whether the reaction is exothermic or endothermic. Combustion reactions are usually exothermic, i.e. heat is released during the process.

The heat of reaction can be computed from the difference between the sum of the heats of formation of the products minus the sum of the heats of formation of the reactant compounds.

A chemical reaction is said to be in equilibrium when the rate of formation of products is equal to the rate of formation of reactants in a reversible reaction. A reversible reaction may be forced in either direction by changes in temperature, concentration or pressure. Many substances, when mixed in definite proportions combine almost completely. This type of reaction is called an irreversible reaction. It is the former type of reaction with which equilibrium conditions are particularly concerned. For example, the water gas equation is as follows:
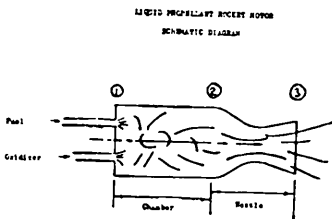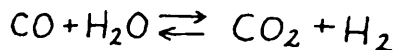
$$CO + H_2O \rightleftarrows CO_2 + H_2$$

The equilibrium constant K for this relation is expressed as

$$K = \frac{(CO_2)\ (H_2)}{(H_2O)\ (CO)}$$

where the chemical symbols in the parentheses represent the partial pressures of the gases. The numerical value of K for any given reaction depends only on the temperature and is independent of all other physical conditions.

$$(IV\text{-}13) \quad K_n = \frac{(n_c)^c\ (n_d)^d}{(n_a)^a\ (n_a)^b}$$

where

K_n=equilibrium constant, when concentration is expressed in volume or molar percentage.

V_p=equilibrium constant, when concentration is expressed in terms of pressure units (atmospheres).

a, b, c, and d=the number of moles of substances A, B, C and D.

(p_A), (p_B), (p_C), (p_D)=the partial pressures of the substances A, B, C and D.

(n_A), (n_B), (n_C), (n_D)=the respective molar concentrations or molar percentages.

Figure 2 shows the value of the logarithms of the equilibrium constant for various chemical reactions.

Although it is not possible to write a single equilibrium equation for the complex reaction within a rocket motor, equilibrium relations of several of the product gas components can be determined. The watergas equilibrium equation, for instance, can be used to relate the proportions of carbon dioxide, carbon monoxide, water vapor and hydrogen within the rocket exhaust gases.

The exact nature of the chemical reaction of rocket propellants is not fully understood although reaction rates and ignition time delays are known to vary for different propellant combina-

Therefore, at any given temperature, the composition of the products and the reactants is fixed by equilibrium. The individual partial pressures may be expressed as concentrations, e.g., as moles of each gas per unit volume of gas. (The partial pressures are proportional to the volumetric or molar composition of a gas mixture.) The units of K will depend on the units selected for the partial pressures.

A more general form of the equilibrium equation, is

$$K_p = \frac{(P_c)^c\ (P_d)^d}{(P_a)^a\ (P_b)^b}$$

tions. Their influence on the combustion equilibrium appears to be of minor importance. The principle of conservation of energy gives a satisfactory means of solution, provided chemical equilibrium is established within the combustion chamber.

Calculations of the combustion temperature and the product gas composition are based on equating the heat of reaction of the propellant combination and the heat capacity of the product gases.

$$[Q_R]_{T_o} = \sum n_p \int_{T_o}^{T} c_p\, dT$$

$[Q_R]_{T_o}$ = Heat of reaction of propellant combination at reference temperature T_o. (To be computed from heats of formation).

$\int_{T_o}^{T} c_p\, dT$ = Enthalpy change necessary to heat one mole of each product gas from the reference temperature T_o to the flame temperature T.

n_p = moles of product gas.

c_p = molar specific heat at constant pressure for a particular product gas.
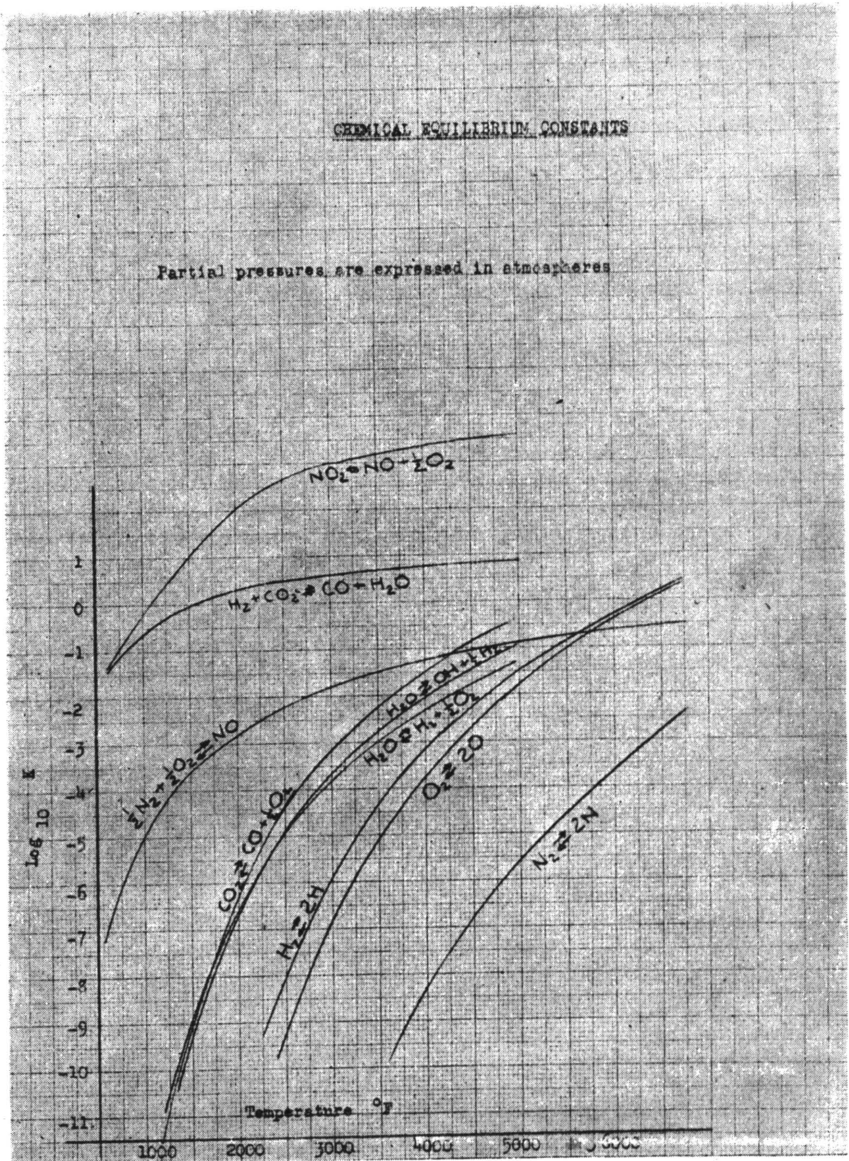
T = Reaction temperature.

FIG. (2)

The above equation, solved for T will give the combustion temperature. The heat of reaction at reference temperature T_o is determined from the heats of formation as explained previously; this value is corrected for the heat necessary to bring the reactants up to the reference temperature.

To evaluate the equilibrium temperature T, the number of moles of each product gas, i.e., the composition, must be evaluated using the equilibrium relations described above and the principle of conservation of matter. The amount of each element in combined and free form must be the same before and after the reaction. If a large number of possible products are formed, the calculation becomes quite tedious because of the large number of unknowns which must be determined before the combustion temperature can be computed.

Propellant combinations containing only carbon, oxygen and hydrogen have at least eight possible combustion products. Typical combinations are: liquid oxygen — gasoline, liquid oxygen — alcohol, and hydrogen peroxide — alcohol. The eight possible products are: carbon dioxide, carbon monoxide, hydrogen, water vapor, oxygen and dissociation products such as monatomic hydrogen, monatomic oxygen and hydroxyl. In solving for the equilibrium composition and temperature it is necessary to determine the relative proportion of each of these constituents. This involves eight equations with eight unknowns. Three of these equations are material balance equations which equate the amount of each element (carbon, oxygen and hydrogen) in the reactants to that in the products. The five other equations are equilibrium equations. Since the final combustion temperature must be known before the

equilibrium constants for the equilibrium equation can be determined, a long, tedious, but relatively straightforward, trial and error solution is indicated.

Reaction gases which contain only hydrogen, carbon, oxygen and nitrogen are obtained from the following common propellant combinations: nitric acid — aniline, nitric acid — furfural alcohol, and hydrogen peroxide — German "C" Stoff (mixture of hydrazine hydrate, water and alcohol). At least eleven possible products may be formed; they are: carbon monoxide, carbon dioxide, water vapor, hydrogen gas, oxygen gas, nitrogen gas, monatomic hydrogen, monatomic oxygen, monatomic nitrogen, hydroxyl and nitrous oxide. Other possible reaction products such as solid carbon, ozone, nitric oxide and various hydro-carbons are discounted, because experience has shown that very little of these products is formed. The method of a trial and error solution with eleven unknowns is necessarily more difficult than the previously mentioned calculations involving eight.

For a given propellant combination, chamber pressure and mixture ratio it normally requires from two to four working days to perform one such computation. To obtain a curve at various mixture ratios is even more tedious.

In a simplified method, such as that devised by Satterfield, Hottel and Williams[1], charts and curves are constructed for a fixed chamber pressure which permit the ready determination of the equilibrium composition and the chamber temperature for any given propellant combination containing these elements.

RESULTS

These thermochemical calculations may be used in obtaining the performance parameters of rocket motors.

CALCULATED PERFORMANCE OF VARIOUS LIQUID PROPELLANT COMBINATIONS

(Specific impulse and exhaust velocities were calculated for an expansion to 1 atmosphere)

Propellant Combination (Oxidizer—Fuel)	Chamber Pressure (psi)	Mixture Ratio (Oxidizer to fuel)	Exhaust Velocity (ft/sec)	Specific Impulse (sec)	Chamber Temperature (°F)	Mean Molecular Weight (lb/mole)	Specific Heat Ratio
Liquid oxygen - gasoline	*300	2.5	7780	242	5470	22.7	1.22
Liquid oxygen - 75% ethyl alcohol, 25% water	300	1.3	7700	239	5060	22	1.22
Liquid oxygen - 100% ethyl alcohol	300	1.5	7810	243	5250	-	-
Liquid oxygen - ammonia	300	1.4	8220	255	4951	-	-
Liquid oxygen - hydrazine	300	.33	7920	246	3632	-	-
Liquid oxygen - liquid hydrogen	300	.33	11050	456	4290	-	-
Hydrogen peroxide (100% pure)	300	-	4700	145	1790	23	1.25
Hydrogen peroxide (90% pure, 10% water)	300		4250	131	1360	22	1.25
Hydrogen peroxide (87%)- C Stoff(57% alcohol, 13% water, 30% hydrazine hydrate)	300	2.5	6920	215	4945	-	-
Nitromethane	300		7010	218	3960	20	1.25
Gaseous oxygen - nitromethane	270	.05	7300	227	4500	21	1.23
Red fuming nitric acid - aniline	300	3.0	7090	221	5020	25	1.22
White fuming nitric acid - furfural alcohol	300	1.9	6890	214	5020	-	-

TABLE (1)

Results of thermochemical computations are given in Table I for a series of representative types of liquid propellant combinations. It can be seen that the exhaust velocities range, with one or two exceptions, between 6800 to 8000 feet per second.

A high exhaust velocity can be obtained by increasing the chemical energy per unit of propellant weight, which in turn increases the combustion temperature, or by lowering the molecular weight of the combustion products. A combustion temperature above approximately 6000° F. does not appear feasable because of dissociation effects and the relatively low chemical energy content of existing propellants. The dissociation of combustion gases into monatomic constituents and radicals lowers the combustion temperature, because energy is consumed, which otherwise would be available for raising the temperature of the gases and as yet there are no known propellants which have a chemical energy content appreciably higher than that of existing rocket fuels and oxidizers.

A decrease in molecular weight can best be effected by using propellants which are rich in combined light weight atoms and molecules. However, the mean molecular weight of the exhaust gases cannot be made smaller than that of hydrogen.

In Table I only liquid hydrogen together with liquid oxygen shows a markedly improved performance (11,500 ft/sec.). A few propellant combinations rich in hydrogen such as liquid fluorine reacting with liquid hydrogen (exhaust velocity of over 10,900 feet per second), may be added in the high performance range. Hydrogen is advantageous as a fuel for two reasons: it has desirable thermochemical properties, such as a high heat of reaction and a high specific heat, and it has a low molecular weight.

In general thermochemical calculations will show that one definite mixture ratio gives an optimum rocket motor performance and that an excess of fuel or an excess of oxidizer will reduce this performance. This optimum mixture ratio shifts slightly with chamber pressure. It is usually a richer mixture than

[1]Satterfield, C. N., Hottel, H. C., & Williams, G. C., "Generalized Thermodynamics of High Temperature Combustion" Mass. Institute of Technology, Division of Industrial Cooperation, Report DIC 6351, May 15, 1947.
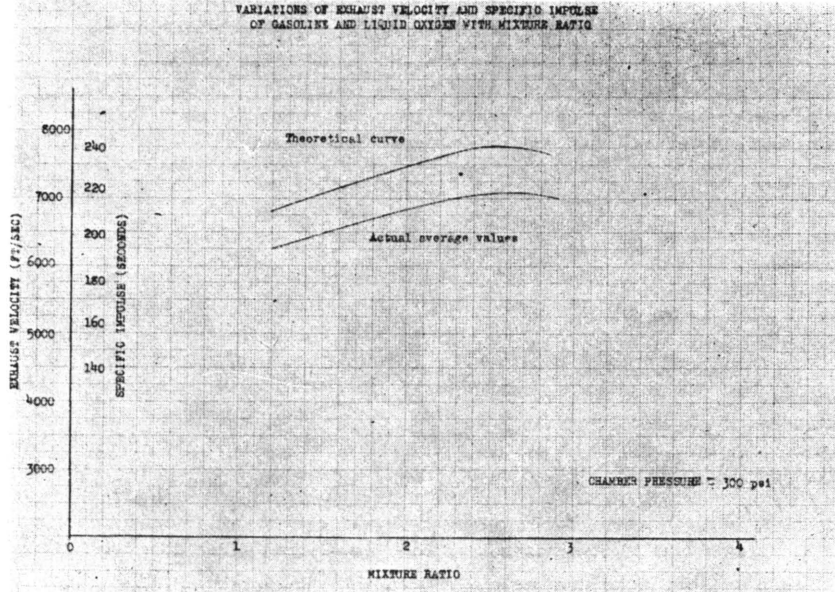
FIG. (3)

the stoichiometric ratio at which completely oxidized products can theoretically be obtained. Rich mixtures have a lower molecular weight and a lower combustion temperature. A slight excess of fules containing combined hydrogen is particularly desirable. From Figure 3 it can be seen that a mixture ratio of about 2.5 gives an optimum performance for the liquid oxygen-gasoline propellant combination.

Propellant mixtures with low combustion temperatures but relatively high exhaust velocities are particularly desirable, since the design of rocket motors is thereby simplified. For example, thermochemical calculations show that liquid oxygen hydrazine or nitromethane fall into this classification.

The variation of performance with chamber pressure, as calculated by thermochemical analysis, approaches a finite maximum at infinite chamber pressure (see Figure 4). In fact, for many propellants any appreciable increase in chamber pressure above 300 psi results

in only a small increase in performance. Since the design chamber pressure for any particular application depends on other considerations besides performance, it is not possible to make a general conclusion regarding its optimum value.

Propellant combinations with exhaust velocities above 9000 feet per second, have, to the best of the author's knowledge, not yet outgrown the research stage and are as yet not suitable for extensive flight applications. Existing and proven propellant combinations are low in performance and therefore do not permit the design of efficient long range or interplanetary rocket vehicles.

The possible increase in performance of 40 to 60 percent is well worth working for. A radical and large increase in rocket exhaust velocity by improved chemical propellants is, however, not to be expected. The choice of rocket propellants will therefore be decided by practical utilization factors rather than by performance considerations.

VARIATIONAL EXHAUST VELOCITY AND SPECIFIC IMPULSE
OF NITRIC ACID-ANILINE WITH CHAMBER PRESSURE

Theoretical curve

Actual Average Values

Mixture ratio = 2.75

FIG. 4
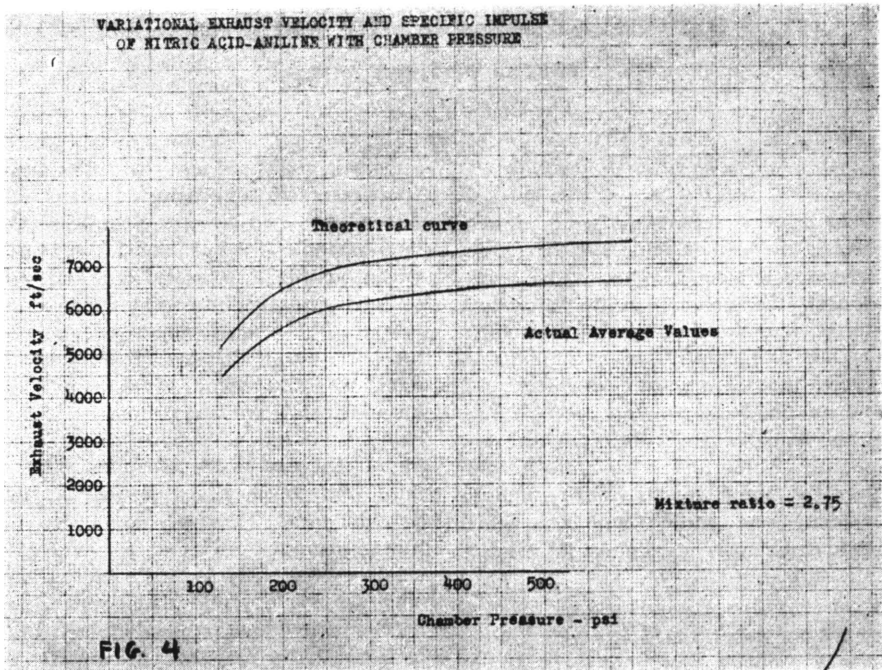
Nuclear energy permits a much higher kinetic energy per pound of propellants than is possible by chemical reaction. This method of propulsion relies on a suitable propellant which is to be heated by a nuclear pile and then ejected at high velocity through a nozzle. If hydrogen is the working fluid and if it is heated in a pile to 7000° F. and adiabatically expanded from 300 psi chamber pressure, the theoretical exhaust velocity will be approximately 23,500 feet per second. If higher temperatures can be withstood by the pile materials, then a corresponding increase in exhaust velocity is to be expected. If H_2 could be dissociated into monatomic hydrogen, than an exhaust velocity of 32,200 feet per second may be possible. The problem of applying nuclear energy to rocket propulsion is, however, difficult. It will require years of development and research effort and will not become practical in the near future.

Some Possibilities For Rocket Propellants

PART III OF THREE PARTS

By ARTHUR S. LEONARD

SUMMARY

In Part I equations were derived which gave the final velocity of the rocket in terms of the mass of the empiv rocket, the volume of the propellant tanks, and the density and jet velocity of the propellant. These equations indicate that the highest final velocity will be obtained through the use of a variable propellant, starting with the combination which gives the highest value for the product of its density and jet velocity and ending with one which produces nearly the highest jet velocity.

In Part II a discussion was given of the more important factors which affect the jet velocity. Equations were derived by which the jet velocity of various propellant combinations may be estimated. Tables of thermochemical properties of some of the compounds which may appear in exhaust gases were presented.

In this installment is a discussion of the use of compounds for liquid rocket propellants. Tables of some of the physical properties of representative types of compounds and pure elements which might be employed as liquid oxidizers or fuels are included. Calculated jet velocities and bulk densities for over 300 combinations of these oxidizers and fuels are presented graphically. An analysis is made of the relationship between the location of an element in the periodic table and its performance as a rocket fuel. Tables are given in which the elements are arranged in the form of the periodic table and in which each element has assigned to it a figure-of-merit which is indicative of its performance as a rocket fuel.

From a study of these and other data (not presented) the conclusion is reached that oxygen and fluorine are by far the best oxidants. The best oxidizer appears to be fluorine monoxide, followed by fluorine, nitrogen trifluoride, ozone and oxygen. The best elements for use as fuels appear to be hydrogen, lithium, beryllium and boron. These are followed closely by magnesium and calcium, in the second periodic group, and all of the rest of the elements in the third and fourth main groups in the periodic table. These elements form a rather compact group in a relatively small region of the periodic table. The most promising liquid fuels appear to be boron hydrides, molten lithium and hydrazine. The propellant combinations of reasonably high density which might be expected to produce the highest jet velocities are fluorine monoxide used with a mixture of beryllium and hydrazine or with boron hydrides.

THE USE OF COMPOUNDS AS PROPELLANTS

Many elements, which, from the standpoint of density and reaction energy, appear attractive for use as propellants, possess such very low boiling or high melting point temperatures that they would be difficult to handle as liquids in the rocket. However, by combining two or more elements to form compounds or solutions, some of the objectionable characteristics may be overcome or at least moderated, while the desirable ones are retained. Listed in Tables 3 and 4 are a few elements and representative compounds which might be used as oxidizers or fuels. Elements

and compounds which cannot be handled in the liquid state because of a high melting point or chemical instability at temperatures below the melting point may be handled as pulverized solids suspended in a liquid. A liquid, in order to be suitable for this purpose, should not react with the solid at the temperatures involved, and, in order to prevent the solid particles from separating out, its density should be as near as possible to that of the solid. A high viscosity is also helpful in preventing the solid from settling out.

In computing the effective enthalpy of formation when compounds are employed, the heats of formation of the compounds must be taken into account. Also, since the propellant will be stored in the rocket at some temperature other than absolute zero, allowance for its heat content must be made. The effective enthalpy of the reaction products in the combustion chamber, when using compounds as propellants, may be computed with the aid of Eqs.* (27) and (28) as follows:

$$N_r \left(H^{\circ}_{erc} - \Delta H^{\circ}_{oer} \right) = N_o \left(H^{\circ}_{To} - \Delta H^{\circ}_{oo} \right) + N_F \left(H^{\circ}_{Tf} - \Delta H^{\circ}_{of} \right)$$

$$(27)$$

Combining Eq. (27) and (12a) we get:

$$h_{erc} = \frac{1}{N_r M_r} \left[N_r \, \Delta H^{\circ}_{oer} + N_o \left(H^{\circ}_{ro} - \Delta H^{\circ}_{oo} \right) + N_f \left(H^{\circ}_{rf} - \Delta H^{\circ}_{of} \right) \right]$$

For most compounds the heat of formation ΔH:), as has been defined, is positive. From Eq. (28) it can be seen that any positive value for the heat of formation of the oxidizer or fuel lowers the value for the effective enthalpy of the reaction products. Therefore, in selecting compounds to be used as propellants we should give considerable weight to a low value for the heat of formation. Endothermic compounds (those having a negative value for their heat of formation) will give even higher reaction energies than the pure elements, but they are inherently explosive and can only be used with some sacrifice in safety. In some cases endothermic compounds can be made less dangerous by mixing with them sufficient amounts of exothermic compounds or even pure elements.

Another thing which might be done to increase the enthalpy of the reaction products is to heat the oxidizer and fuel as hot as possible before firing. This has the effect of making H°_{To} and H°_{Tf} as large as possible. Unfortunately, we may not be able to realize much of a net increase in the final velocity of the rocket by so doing because (1) we will probably need to use either the oxidizer or the fuel as a coolant for the motor, thus putting a rather low upper limit on the temperature to which we may heat one of the components of the propellant and (2) preheating these substances will lower their densities, thus reducing the total weight of propellant that can be stored in the tanks.

OXIDIZERS AND FUELS

Listed in Tables 3 and 4 are some possible oxidizers and fuels. These lists are not intended to be complete, but to give examples of some of the types of compounds which might be employed as rocket propellants. Although there are many other substances which might be used for this purpose, most of them will be found to give no better performance and to possess physical properties which fall within the range of those listed.

The oxidizers given in Table 3 were found by searching chemistry handbooks for substances which were rich in oxygen or fluorine, had negative or at least low positive values for their heats of formation, and which were reason-ably stable and could be handled as liquids at some feasible temperature. The fuels listed in Table 4 were selected in a similar manner; but, instead of looking for oxygen or fluorine, substances were sought out which were composed of elements which, from Table 2A and 2B, appeared to be promising for use in rocket fuels.

In order to compare the performance of the various elements and compounds listed, the bulk density and theoretical jet velocity was calculated for over 300 oxidizer-fuel combinations. The results were plotted and are presented in Figs. 1A and 1B. The calculations were made as follows:

1. The chemical equation for the reaction between the fuel and oxidizer was written assuming that

TABLE 3.   SOME OXIDIZERS AND THEIR PHYSICAL PROPERTIES

Identi-fying Letter	Chemical Formula	Mol. Weight, M_o	Liquid Density, ρ_o	Liquid Temp., t_o	Melting Point Temp., t_{ro}	Boiling Point Temp., t_{vo}	Heat of Form-ation, ΔH_{fo}^o	Heat Con-tent at Liq. Temp., H_{Lo}^o
A	F_2O	54.00	1.90	-224	-224	-145	$-7x10^3$	$0x10^3$
B	IF_5	221.92	3.5	20	-8	97	[30]	9
C	IF_7	259.92	2.8	, 6	5	5	[30]	11
D	BrF_5	174.92	2.47	25	-61	40	[20]	9
E	F_2	38.00	1.14	-200	-223	-187	0	.1
F	NF_3	71.01	1.54	-129	-217	-120	24	3
G	ClF_3	92.46	1.77	13	-83	11	[30]	7
H	O_3	48.00	1.71	-183	-251	-112	-35	1
J	CF_4	88.01	1.96	-184	-184	-128	164	1
K	NO_3F	81.01	[1.62]	-150	-175	-46	[-2]	3
L	NO_6Cl	145.47	[1.65]	0	[-20]	[80]	[-10]	11.
M	N_2O_4	92.02	1.49	0	-9	21	2	9
N	ClO_2	67.46	[1.70]	0	-59	12	-20	5
P	$HClO_4$	100.47	1.76	22	-112	dec.	26	8
Q	$C(NO_2)_4$	196.04	1.65	13	13	126	-16	16
R	O_2	32.00	1.14	-183	-218	-183	0	1
S	HNO_3	63.02	1.50	20	-42	86	38	5
T	$LiClO_4$	106.40	[2.0]	236	236	dec.	90	19
U	H_2O_2	34.02	1.46	0	-2	152	45	5
V	$LiClO_3$	90.40	[2.2]	129	129	dec.	73	14
W	$LiNO_3$	68.95	[2.2]	255	255	dec.	111	14
X	HIO_3	175.93	[4.0]	110	110	-	55	10

the fuel elements would be oxidized to the oxides or fluorides listed in Tables 2A and 2B. It was also assumed that in the reaction products nitrogen would exist in the molecular form and iodine, bromine, chlorine, or mercury would be in the atomic state. From these equations the number of moles of oxidizer (N_o), of fuel (N_f), and of reaction products (N_r) were obtained.

1.. The bulk density was computed with the aid of Eq. (29), as follows:

$$\rho_p = \frac{N_o M_o + N_f M_f}{\frac{N_o M_o}{\rho_o} + \frac{N_f M_f}{\rho_f}} \quad \dots (29)$$

3. For the combinations in which more than one compound or free element was present in the reaction products, the value for the product $(N_r M_r)$ was evaluated by means of Eq. (30), as follows:

$$N_r M_r = (\Sigma N M)_r \quad \dots (30)$$

4. Values for the effective heat of formation (ΔH_{ir}) for each compound in the reaction products were obtained from Tables 2A* and 2B, and for the elements in the atomic form, from handbooks and other sources. Values for H''_{to}, Δ_{io}, H''_{Tf}, and $\Delta H''_{of}$ were taken from Tables 3 and 4. With these values and Eq. (28), h_{rre} was computed.

5. Values for C_p for each compound and free element in the reaction products was obtained from Table 1. With these values and Eqs. (31)

$$C_{pr} = \frac{(\Sigma N C_p)_r}{(\Sigma N)_r} \quad \dots (31)$$

and (32), the ratio $\dfrac{\gamma - 1}{\gamma}$ was calculated:

$$\frac{\gamma - 1}{\gamma} = \frac{R}{J C_{pr}} \quad \dots (32)$$

6. Using the numerical value for $\dfrac{\gamma - 1}{\gamma}$ thus obtained, a value of 25 for the pressure ratio (R_r), and Eq. (13)*, the nozzle efficiency (N_n) was calculated.

7. The theoretical jet velocity (U_j) was then calculated by means of Eq. (14).

A study of Fig. 1A shows that over a rather wide range of bulk density, the propellant combinations giving the highest jet velocity for any given density fall approximately on a straight line having a slope of — 0.5. This corresponds to a constant value for the product $(U_j P_t^{0.5})$. If we were to make use of the principle of the variable propellant** and were free to choose any of the combinations plotted on Figs. 1A or 1B, we would employ propellants all of which had approximately the same value for the product $(U_j P_t^{0.5})$. For this reason the decision was made to use this product as a figure-of-merit in rating the various propellant combinations.

The order of listing of the oxidizers and fuels presented in Tables 3 and 4 was determined by the following procedure: A table was made up in which the oxidizers were listed vertically and the fuels horizontally. The value for the product $(U_j P_t^{0.5})$ was tabulated in the appropriate space for each combination for which computations had been made. For each oxidizer and each fuel the average of the three highest values appearing in the corresponding line or column was taken as a figure-of-merit for the oxidizer or fuel. Tables 3 and 4 were then made up by arranging the individual oxidizers and fuels in the order of decreasing values for this figure-of-merit.

TABLE 4. SOME FUELS AND THEIR PHYSICAL PROPERTIES

Identi-fying Number	Chemical Formula	Mol. Weight, M_f	Liquid Density, ρ_f	Liquid Temp., t_f	Melting Point Temp., t_{ff}	Boiling Point Temp., t_{vf}	Heat of Formation, ΔH_{of}^o	Heat Content at Liq. Temp., H_{fr}^o
1	Hf+Hg	379.21	13.46	0	-39	360	0x10³	4x10³
2	SiLi₄	55.82	1.04	630	630	-	0	40
3	La	138.92	5.45	826	826	1800	0	10
4	LiH	7.95	.65	680	680	dec.	21	10
5	Al	26.97	2.38	660	659	1800	0	7
6	Be(C₂H₅)₂	67.14	.70	12	12	>200	35	10
7	Mg	24.32	1.57	651	651	1110	0	8
8	BeB₂H₈	38.72	.60	123	123	91	15	10
9	AlB₃H₁₂	71.53	.70	-60	-64	44	20	10
10	B₁₀H₁₄	122.31	.78	100	100	213	50	24
11	Li	6.94	.500	186	186	1336	0	3
12	C₁₅H₁₂	192.25	1.08	85	80	350	-28	19
13	C₄H₂	50.06	.75	-30	-36	10	-112	5
14	C₈H₁₈	114.26	.76	-50	-57	125	63	13
15	C₆H₅NH₂	93.12	1.05	-6	-6	184	-8	7
16	C₂H₅OH	46.07	.89	-110	-115	79	65	3
17	B₂H₇N	42.70	.70	-65	-66	76	10	4
18	CaN₂	42.10	1.3	815	815	dec.	45	17
19	Si₃H₈	92.24	.88	-100	-117	53	20	3
20	LiNH₂	22.96	1.00	375	375	430	24	8
21	(SiH₃)₃N	107.26	.895	-106	-106	52	-10	8
22	Ge₃H₈	225.86	2.2	20	-106	110	20	6
23	Ga	69.72	6.09	32	30	2000	0	2
24	Zn	65.38	6.92	419	419	907	0	6
25	Ba	137.36	3.34	850	850	1140	0	12
26	HCN	27.03	.70	20	-14	26	-24	4
27	NH₃	17.03	.73	-73	-78	-33	16	3
28	C₂N	52.04	.87	15	-34	-20	-63	5
29	Sn	118.70	6.99	232	232	2270	0	4
30	In	114.76	7.1	155	155	1450	0	3
31	C₃S₂	100.15	1.27	20	-1	-	-20	5
32	CuCN	89.59	2.2	475	475	dec.	-28	15
33	H₂	2.02	.070	-252	-259	-253	0	0
34	P	30.98	1.70	44	44	280	0	2
35	Na	23.00	.929	98	98	880	0	3
36	Pb	207.21	10.66	331	327	1613	0	5
37	S	32.06	1.81	113	113	445	0	4

FUELS AND THE PERIODIC TABLE

If we were to plot as ordinates the values of the effective ethalpy of formation of the oxides of the elements, and as abscisas the atomic number or atomic weight of the elements, we would obtain a rough curve having several maxima and minima.

The major peaks would be found to correspond roughly to boron, aluminum, titanium, zirconium, lanthanum, and thorium. Minor peaks would be found to correspond to lithium, phosphorus, and germanium. If we were to plot a curve for the fluorides in a similar manner, we would find similar peaks which correspond to approximately the same elements.

It should be noted that all six elements to which the major peaks correspond fall in the third and fourth groups in the periodic table of the elements. This suggests that there may be some correlation between the position in the periodic table occupied by an element and its performance as a rocket fuel.

A study of Table 4 shows that the first 8 fuels contain the following fuel-elements (mercury is assumed to be an inert diluent): hydrogen, lithium, beryllium, boron, carbon, magnesium, aluminum, silicon, lanthanum, and hafnium. With the exception of lanthanum and hafnium, these elements form a rather compact group in the periodic table (see Table 6). Although this points very strongly to the location in the periodic table occupied by the best fuel elements, it is not conclusive because many elements, for lack of sufficient data for making calculations, do not appear anywhere in Table 4.

In order to get a more complete, and at the same time a more quantitative, correlation between the positions in the periodic table occupied by the elements and their performance as rocket fuels. the following method was devised: Find a radical which, when combined with any element in the periodic table, would enable the element to be handled as a liquid, and base all performance calculations on the use of such compounds as fuels, with a single oxidizer. After examining available data on several types of compounds, the decision was made to use the methyl radical (CH_3) as the radical which would be combined with each fuel-element to form the liquid fuel, and fluorine monoxide as the oxidizer.

In Table 5 are listed the compounds (methylides) for which performance calculations were made. The results of these calculations are included in Figs. 1A and 1B. Density data were found for 10 of the compounds. The densities of the rest of the methylides

were estimated from the densities of the pure elements and the effective volume of the methyl radical. Heats-of formation and sensible heats of the liquids had to be estimated in most cases. The performance calculations were based on the assumption that the oxygen of the oxidizer (F_2O) would combine with the carbon of the methyl radical to form carbon dioxide while the fluorine would combine with the hydrogen of the methyl radical and with the fuel-element. Propellant densities and jet velocities were calculated by the procedure previously described.

A study of the calculated bulk densities and theoretical jet velocities presented in Table 5 show a rather small range of variation. This is to be expected because the fuel-element, in most cases, contributes only a small fraction of the total mass and total energy of the propellant combination. This fact, however, makes small differences in calculated performance rather significant.

In order to compare one fuel-element with another, the value for the product ($U_jP_r^{0.5}$) was calculated for each combination, and is tabulated in the next to the last column of Table 5. Since, for the latter part of the burning period of a very long range single step rocket, or for all but the first step of a multistep rocket, propellant density is, in comparison with jet velocity, relatively less important than is indicated by the product ($U_jP_r^{0.5}$), another figure-of-merit ($U_jP_r^{0.5}$) was calculated for each fuel-element, and is tabulated in the last column.

In order to show the correlation between the positions in the periodic table occupied by the elements and their performance as rocket fuels, the values of these two figures-of-merit were arranged in the form of the periodic table and are presented in Tables 7A and 7B. The key to the arrangement

of the elements in these tables is to be found in Table 6. When we look for the 12 highest values presented in Table 7A (3.92 and higher), we find that they correspond to lithium in the first periodic group, beryllium, magnesium and calcium in the second group, boron, aluminum and lanthanum in the third group (sufficient data were not available for computations on scandium and yttrium), and silicon, titanium, zirconium, hafnium and thorium in the fourth group. When we look for the 12 highest values in Table 7B (3.50 and higher), we find the same group with the exception that hydrogen has been added and thorium dropped. It should be noted that, in both cases, the 12 highest values form single rather compact groups which, with the exception of scandium and yttrium, for which no values are presented, have no omissions.

It should be noted that the results presented in Tables 7A and 7B are strictly applicable only when fluorine or a fluorine-rich compound is employed as the oxidizer. The reason for this is that, even though the oxidizer used in making the calculations (fluorine monoxide) contains oxygen, the fuel-element, in each case, was assumed to combine only with fluorine. However, if a similar analysis is made, based on the use of an oxidizer which would react to form the oxide of the fuel element in each case, a somewhat similar result would be obtained, in that the greatest values of the figures-of-merit would fall in the same groups of the periodic table. The principle difference would be that the values would all be lower than those in Tables 7A and 7B, and the highest values would not extend so far down the table.

DISCUSSION

From the standpoint of rocket performance, the most important characteristics of the propellant appear to be jet velocity and bulk density. When the

problems involved in the application of the propellant, and in particular the cooling of the motor, are considered, reaction temperature must be considered. At the present time it is difficult to evaluate rocket performance in terms of reaction temperature, but it may be safe to say that in general the difficulties associated with the application of a propellant will increase with reaction temperature. It should be pointed out that propellant combinations in which fluorine is employed as the oxidizer will in most cases produce considerably higher reaction temperatures than those in which oxygen is employed as the oxidizing element. This fact should be kept in mind when comparing propellants and particularly when comparing oxidizers since the data and results of calculations presented in this paper do not take this property into account.

Since, from the standpoint of rocket performance, the two most important properties of the propellant are jet velocity and bulk density, it seems logical to plot jet velocity against bulk density for the purpose of graphical comparison of various propellant combinations. This has been done in Figs. 1A and 1B. Where the range of jet velocities or densities is large, log-log coordinates appear to give a better presentation than linear coordinates. Where the range of jet velocities and densities is small, the pictures obtained by the two methods are so nearly alike that the log-log coordinates may not be justified. Because of the extremely wide range of jet velocities and propellant densities covered by this investigation, the log-log coordinate system was used in plotting the results.

When the results of the calculations were plotted to the scale of Fig. 1A, severe crowding of the points occured in the central region of the sheet. In order to make the individual points distinguishable, only a few of the more

TABLE 5. SOME INORGANIC METHYLIDES, THEIR PHYSICAL PROPERTIES, AND
THEORETICAL PERFORMANCE AS FUELS WHEN USED WITH FLUORINE MONOXIDE

Chemical Formula	Mol. Weight, M_f	Liquid Density, ρ_f	Heat of Formation, ΔH_{of}^o	Heat Content at Liq. Temp., H_{Tf}	Propellant Density, ρ_p	Theor. Jet Vel., $U_j - \frac{km}{sec}$	Figure-of-Merit $U_j \rho_f^{0.5}$	$U_j \rho_f^{a25}$
HCH$_3$	16.04	.415	21x10^3	2x10^3	1.299	3.316	3.78	3.54
LiCH$_3$	21.97	[.516]	[10]	3	1.308	3.445	3.93	3.67
Be(CH$_3$)$_2$	39.09	[.610]	[19]	4	1.435	3.381	4.05	3.70
B(CH$_3$)$_3$	55.92	.625	[29]	4	1.461	3.311	4.00	3.64
C(CH$_3$)$_4$	72.15	.613	39	8	1.461	3.103	3.75	3.41
NaCH$_3$	38.03	[.714]	[9]	3	1.327	3.177	3.36	3.41
Mg(CH$_3$)$_2$	54.39	[.743]	[19]	5	1.448	3.294	3.96	3.61
Al(CH$_3$)$_3$	72.07	[.729]	[28]	6	1.472	3.288	3.99	3.62
Si(CH$_3$)$_4$	88.20	.651	[39]	8	1.433	3.277	3.92	3.59
P(CH$_3$)$_5$	106.15	[.665]	[48]	10	1.456	3.181	3.84	3.49
S(CH$_3$)$_6$	122.26	[.633]	[58]	12	1.443	3.067	3.68	3.36
KCH$_3$	54.13	[.720]	[9]	3	1.229	3.043	3.37	3.20
Ca(CH$_3$)$_2$	70.15	[.823]	[19]	5	1.440	3.269	3.92	3.63
Ti(CH$_3$)$_4$	108.04	[.837]	[38]	8	1.515	3.239	3.99	3.69
Cr(CH$_3$)$_3$	97.11	[1.008]	[29]	6	1.578	3.096	3.89	3.47
Fe(CH$_3$)$_2$	85.92	[1.296]	[20]	5	1.677	2.981	3.86	3.39
Ni(CH$_3$)$_2$	88.76	[1.349]	[20]	5	1.698	2.950	3.84	3.37
CuCH$_3$	78.60	[2.14]	[10]	3	1.996	2.608	3.68	3.10
Zn(CH$_3$)$_2$	95.45	[1.386]	[19]	5	1.705	2.948	3.85	3.37
Ga(CH$_3$)$_3$	114.82	[1.141]	[29]	7	1.619	3.052	3.88	3.44
Ge(CH$_3$)$_4$	132.74	1.006	[38]	8	1.571	3.103	3.89	3.47
As(CH$_3$)$_5$	150.08	[.905]	[48]	10	1.533	3.043	3.77	3.39
Se(CH$_3$)$_6$	169.16	[.862]	[58]	12	1.522	2.966	3.66	3.30
Sr(CH$_3$)$_2$	117.70	[1.266]	[19]	5	1.615	3.031	3.85	3.42
Zr(CH$_3$)$_4$	151.36	[1.142]	[38]	8	1.620	3.178	4.04	3.59
Mo(CH$_3$)$_6$	186.15	[.996]	[58]	12	1.579	3.082	3.87	3.46
AgCH$_3$	122.91	[3.08]	[10]	3	2.388	2.291	3.54	2.84
Cd(CH$_3$)$_2$	142.48	1.985	[19]	5	1.933	2.694	3.75	3.18
In(CH$_3$)$_3$	159.86	[1.530]	[29]	7	1.759	2.890	3.83	3.33
Sn(CH$_3$)$_4$	178.84	1.314	[38]	9	1.681	2.958	3.84	3.37
Sb(CH$_3$)$_3$	166.86	1.523	[29]	7	1.754	2.843	3.77	3.27
Te(CH$_3$)$_6$	217.81	[1.100]	[58]	12	1.607	3.038	3.85	3.42
Ba(CH$_3$)$_2$	167.43	[1.702]	[20]	6	1.807	2.834	3.81	3.29
La(CH$_3$)$_3$	184.02	[1.652]	[29]	7	1.802	3.017	4.05	3.50
Hf(CH$_3$)$_4$	238.74	[1.812]	[39]	9	1.868	2.992	4.09	3.50
Hg(CH$_3$)$_2$	230.68	3.069	[19]	6	2.366	2.373	3.65	2.94
TlCH$_3$	219.42	[4.68]	[9]	3	3.157	2.068	3.67	2.76
Pb(CH$_3$)$_2$	237.28	[3.17]	[19]	5	2.405	2.434	3.76	3.02
Bi(CH$_3$)$_3$	254.10	2.30	[29]	7	2.057	2.600	3.73	3.11
Th(CH$_3$)$_4$	292.26	[2.10]	[39]	9	1.976	2.902	4.08	3.44

promising propellant combinations falling in this region were plotted. A second sheet (Fig. 1B), covering only the central region of Fig. 1A, was made up to a larger scale; and all of the points falling on that part of the sheet were plotted.

When we study Figs. 1A and 1B for oxidizer performance, we find fluorine monoxide to be outstanding. The exceptional performance of this oxidizer may be attributed partly to its high density and partly to its heat of formation, which is negative. Another oxidizer which is prominent among the propellant combinations giving high jet velocity is elemental fluorine. The highest performing oxygen-rich oxidizer is found to be ozone. This is due to its high density and heat of formation, which is negative. By itself, liquid ozone is very dangerous to handle because of its liability to explode. However, it might

TABLE 6. PERIODIC TABLE*

Group\Period	I	II	III	IV	V'	VI'	VII'	VIII'			I'	II'	III	IV'	V	VI	VII
	H																
1	Li	Be	B	C										C	N	O	F
2	Na	Mg	Al	Si										Si	P	S	Cl
3	K	Ca	Sc	Ti	V	Cr	Mn	Fe	Co	Ni	Cu	Zn	Ga	Ge	As	Se	Br
4	Rb	Sr	Y	Zr	Cb	Mo	Ma	Ru	Rh	Pd	Ag	Cd	In	Sn	Sb	Te	I
5	Cs	Ba	La	Hf	Ta	W	Re	Os	Ir	Pt	Au	Hg	Tl	Pb	Bi	Po	Ab
6	Vi	Ra	Ac	Th	Pa	U											

* For the sake of brevity, the inert gases and the rare earth metals with the exception of Lanthanum have been omitted.

TABLE 7A. NUMERICAL VALUES FOR THE FIGURE-OF-MERIT, $u_i\rho_i^{2.5}$, ARRANGED IN THE FORM OF THE PERIODIC TABLE FOR FLUORINE MONOXIDE-METHYLIDE COMBINATIONS

Group\Period	I	II	III	IV	V'	VI'	VII'	VIII'			I'	II'	III'	IV'	V	VI	VII
	3.78																
1	3.93	4.05	4.00	3.75										3.75			
2	3.66	3.96	3.99	3.92										3.92	3.84	3.68	
3	3.37	3.92		3.99		3.89		3.86			3.84	3.68	3.85	3.88	3.89	3.77	3.66
4		3.05		4.04		3.87		3.54			3.75	3.83	3.84	3.77	3.85		
5		3.81	4.05	4.09							3.65	3.67	3.76	3.73			
6				4.08													

TABLE 7B. NUMVERICAL VALUES FOR THE FIGURE-OF-MERIT, $u_i\rho_i^{0.25}$, ARRANGED IN THE FORM OF THE PERIODIC TABLE FOR FLUORINE MONOXIDE-METHYLIDE COMBINATIONS

Group\Period	I	II	III	IV	V'	VI'	VII'	VIII'			I'	II'	III'	IV'	V	VI	VII
	3.54																
1	3.67	3.70	3.64	3.41										3.41			
2	3.41	3.61	3.62	3.59										3.59	3.49	3.36	
3	3.20	3.58		3.59		3.47		3.39			3.37	3.10	3.37	3.44	3.47	3.39	3.30
4		3.42		3.59		3.46		2.84			3.18	3.33	3.37	3.27	3.42		
5		3.29	3.50	3.50							2.94	2.76	3.02	3.11			
6				3.44													

added to liquid oxygen in reasonable amounts in order to give the propellant increased density and increased jet velocity without too much danger from explosion.

Although a theoretical analysis of rocket performance leads to the conclusion that, for a given payload and range, the use of a rather dense propellant at the start of firing will permit the size of the rocket to be reduced to a minimum, practical considerations may dictate the use of some other propellant during this part of the firing period. The reason for this is that a sacrifice in the performance of the propellant used during the first part of firing will make necessary only a relatively small increase in the size and, therefore, the cost of the rocket. If the lower performing propellant is much cheaper, a reduction in the total cost of the rocket and propellant may be realized.

In a multistep rocket, the dense propellant would be used only in the first step. Since this step will probably contain more propellant than all the rest put together, the use of a less costly substance here may reduce very greatly the total propellant cost for the rocket. Since the substitution of a lower performing propellant in the first step will make necessary only a rather small increase in the size of this step only, a net saving may be realized by so doing.

Since zirconium, lanthanum, Hafnium, and thorium would be applicable only to very dense propellants, and since they are all very costly and there seems to be little prospect that increased demand would result in the reduction of their price to reasonably low levels, we may safely drop these elements from further consideration. The rest of the high-performing fuel-elements with the exception of hydrogen and lithium give propellants of so nearly the same bulk density that none of them appears to be outstanding for use during the first part of the firing period.

When we consider the effect on the rocket of a sacrifice in the performance of the propellant used in the second and higher steps, we find that the result is an increase in size which is cascaded with each step. The net result is that even a small reduction in performance of the propellant used in this part of the rocket will make necessary a very great increase in total size and weight. Therefore, in selecting propellants to be used for this purpose, performance should receive much more consideration than cost. Since the most important criterion of performance of a propellant which is to be used in the latter part of the firing period, or in the second and higher steps of a multistep rocket, is jet velocity, we should examine the remaining elements of the high-performing group for their behavior in this respect.

From the jet velocities shown in Table 5, there appears to be relatively little preference between the various remaining elements. However, when we attempt to estimate the reaction temperatures that must exist in order to obtain the calculated values for jet velocity, we find that the heavier elements must produce considerably higher reaction temperatures than the first few. When account is taken of the effects of high temperature dissociation on jet velocity, the heavier elements may be expected to suffer a greater loss in performance than the lighter elements of this group. This leads to the conclusion that, from the standpoint of high performance, the most promising fuel-elements are hydrogen, lithium, beryllium and boron. Incidentally, the same results might have been obtained from a study of the values for effective enthalpy of formation presented in Tables 2A and 2B.
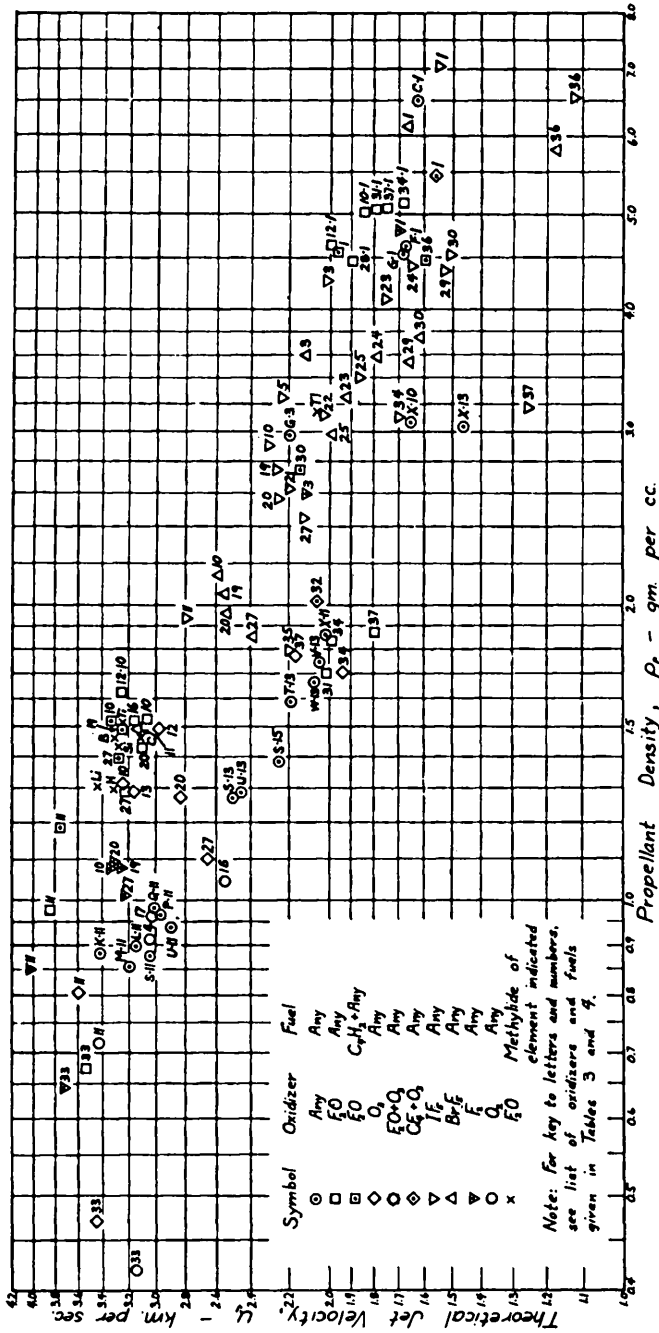
FIG. IA. THEORETICAL JET VELOCITY vs. PROPELLANT DENSITY FOR VARIOUS OXIDIZER-FUEL COMB.

Propellant Density, ρ_o – gm. per cc.

Theoretical Jet Velocity, U_j – km. per sec.

Symbol	Oxidizer	Fuel
⊙	Any	Any
□	F_2O	Any
⊡	F_2O	C_pH_2+Any
◇	O_2	Any
⬦	F_2O+O_2	Any
◈	CF_4+O_2	Any
▷	IF_5	Any
◁	BrF_5	Any
▶	F_2	Any
○	O_2	Any
×	F_2O	Methylide of element indicated

Note: For key to letters and numbers, see list of oxidizers and fuels given in Tables 3 and 4.

The data for beryllium oxide presented in Table 2A tends to indicate that as a fuel, beryllium is not particularly good when used with oxygen. This, however, is not true because the condensation temperature for beryllium oxide is so high that most of it can be expected to condense to a finely divided solid, or to at least a finely divided liquid, in the nozzle, and thus make a large fraction of the heat of condensation available for the production of jet velocity. When this is taken into account, all four of these elements may be expected to give high performance, using either fluorine or oxygen as the oxidizing element.

When we investigate the problem of handling these elements as fluids in the rocket, there appear to be relatively few feasible solutions. Boron and hydrogen may be combined to form boron hydrides, some of which are liquids at reasonable temperatures. Hydrogen may be liquified by cooling to extremely low temperatures, but its density is so low, even in this form, that its field of application appears to rather limited. Lithium may be liquified by heating to above its melting point, 186°C. Hydrogen may be combined with nitrogen to form hydrazine or ammonia. Hydrazine, because of its negative heat of formation, may be expected to give better performance than ammonia. Beryllium or boron might be finely pulverized and carried as a suspension in hydrazine or a hydrocarbon. In order to prevent settling out, an alloy of lithium and beryllium, lithium and boron, or of all three elements might be made to have a density approximately equal to that of hydrazine. If a suspension of such an alloy would not cake or react with hydrazine, it could be handled this way. Although the elements lithium, beryllium and boron are relatively scarce, it seems probable that quantities of these elements sufficient to fuel any rocket which might be sent to the moon

or to the nearer planets in the forseeable future could be obtained.

Another propellant combination of high performance, but which is not brought out by the data presented, is fluorine monoxide and paraffin or olefinic hydrocarbons. The performance of this combination is very nearly equal to that of fluorine monoxide and hydrazine.

CONCLUSIONS

From the preceding discussion and the data and results of calculations which have been presented, the following general conclusions may be formulated:

1. Of all the chemical elements, oxygen and fluorine are by far the best oxidants. Fluorine generally gives a little higher performance than oxygen. The use of oxygen generally results in a lower reaction temperature.

2. The highest performing liquid oxidizer appears to be liquid fluorine monoxide. This is followed in order by liquid fluorine, nitrogen trifluoride, ozone, and a mixture of liquid ozone and oxygen.

3. The highest performing fuel-elements appear to be hydrogen, lithium, beryllium, and boron. These are followed rather closely by magnesium, aluminum and silicon.

4. The highest performing liquid fuels of reasonable density appears to be the boron hydrides. These are followed by molten lithium and hydrazine.

5. For propellant combinations of reasonably high bulk density, those which might be expected to yield the highest jet velocities are fluorine monoxide used with a mixture of beryllium and hydrazine or with boron hydrides.
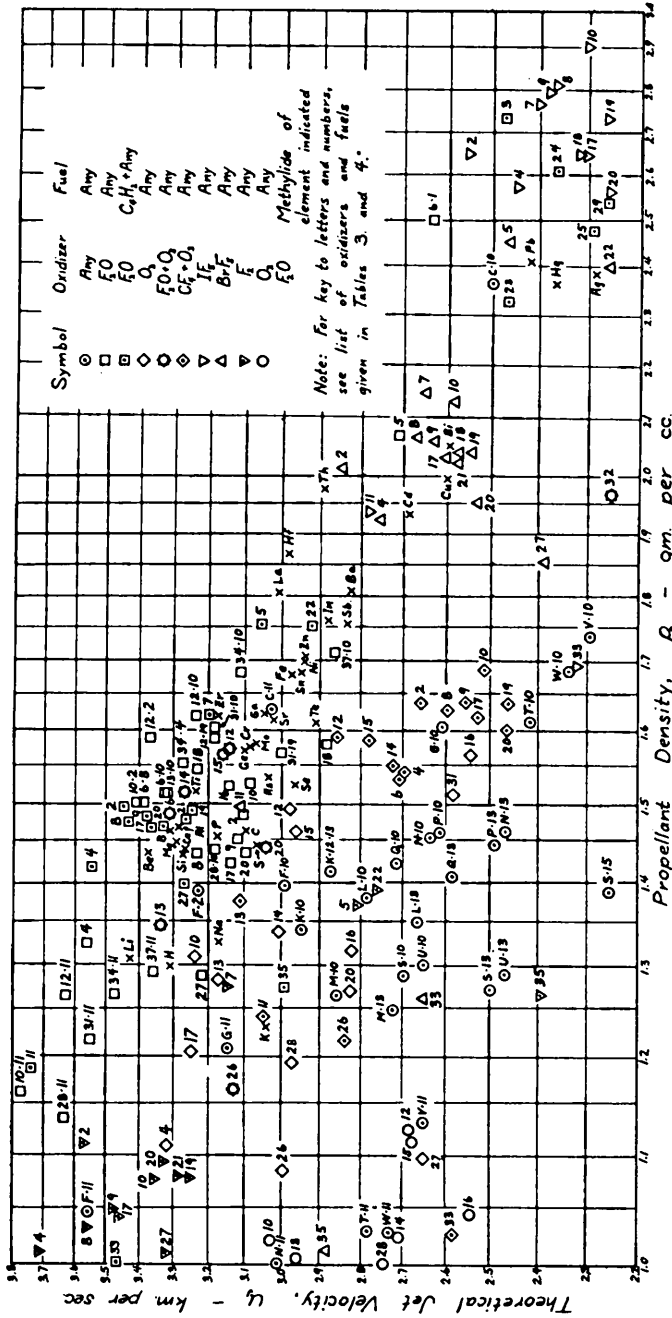
FIG. IB. THEORETICAL JET VELOCITY vs. PROPELLANT DENSITY FOR VARIOUS OXIDIZER-FUEL COMB.

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NEW JET-AND-ROCKET PLANE
DESIGNED FOR SPEEDS FROM
650 TO 750 MILES PER HOUR

Construction of the Navy's needle-nosed new jet-and-rocket airplane, the Skyrocket, designed to advance the assaults on the sonic barrier now being made by its sister-plane, the Skystreak, has been completed at the El Segundo, California plant of Douglas Aircraft Corporation.

Radically different in appearance from the conventionally designed Skystreak, the sleek, white-painted Skyrocket has swept-back wings and tail, and an increased thrust which will enable it to approach considerably closer to the speed of sound.

Powered by a turbo-jet engine, plus the extra thrust of a liquid-fuel rocket it is expected to perform in the 650-750 mile an hour zone. The Skystreak currently holds the official world speed record with a mark of 650.6 miles an hour.

The Skyrocket, designated the D-558-2, is the second phase of the D-558 research airplane project, a joint undertaking of the Navy, the National Advisory Committee for Aeronautics and Douglas Aircraft. It will undergo a test-flight program at the Air Force Test Center, Muroc Dry Lake, California, and then be turned over by the Navy to NACA for an intensive research program.

The powerplants of the Skyrocket are a Westinghouse 24C turbo-jet engine, developed by the Navy, and a liquid rocket system built by Reaction Motors, Inc. It is equipped to take off and land under its own power, with the rocket furnishing additional thrust in high-speed test runs. The turbo-jet engine burns aviation gasoline.

Almost twice as long as it is wide, the tiny plane is tipped by a needle-like "lance" in the nose, containing the pitot static tube for the airspeed indicator system. The fuselage is 45 feet long and the back-swept wings measure only 25 feet from tip to tip, giving the aircraft an over-all resemblance to a swordfish.

Incorporation of fuel and landing gear accommodations in the fuselage instead of in the wings gives the new model a considerably larger fuselage diameter than the Skystreak. The entire nose section, which contains the pilot's compartment, can be jettisoned for high-speed escape. The cockpit is fitted with refrigeration, heating and pressurization equipment. Aerodynamic brakes on the after portion of the fuselage furnish control of drag and speed. Handley-Page Automatic slots in the leading edge of the wings are designed to give improved low-speed lift characteristics.

Most of the fuselage skin is made of magnesium alloy; the wing and tail surfaces are of 75S aluminum alloy. It was originally planned to paint the Skyrocket the same bright red shade as the Skystreak and other experimental aircraft, but the color was changed to white at the request of NACA, for possible greater visibility at a distance.

Special instrumentation in the Sky-rocket will permit measurement of air pressure at 400 points on the wing and tail surfaces. Control forces and stresses in the structure will be measured and recorded by more than 900 strain gauges and oscillographs. Readings of the multitude of standard flight instruments will be recorded on motion picture film during the first flights and control demonstration, by means of a photo-graphic flight recorder.

AEROJET ENGINEERING CORP. MAKES PROGRESS

Television was used for the first time anywhere to observe the testing of high thrust rocket motors at the Aerojet Proving Grounds, Azusa, Calif., recently, before several score of top government officials. Providing views as close as two feet to observers seated comfortably in the conference room far removed from the test pits, this new method of viewing hazardous tests by television received the highest commendation from Army, Navy, and other technical leaders.

Developed by the Aerojet Engineering Corporation, subsidiary of The General Tire & Rubber Company, to provide safer and more adequate test viewing facilities, this method of televising test details was successfully demonstrated with the cooperation of electronic engineers of the General Electric Company who furnished the television equipment. Television observation of tests allows the only safe method for complete safety of the observers with the added advantages of better lighting and close-ups never before provided — safety being guaranteed by the remoteness allowed by television, picture light intensity and definition are far superior to direct viewing through glass, and shock-proofed cameras can be mounted adjacent to the rocket unit for viewing intimate details.

The limitations of present methods of test viewing are manifold. Observation blockhouses restrict viewing to either the direct method through laminated safety glass which becomes clouded from close-range effects of propellant fumes, or the indirect method using mirrors which, in addition to becoming clouded, limit the range of vision and often include distortion. Both of these observation systems require apertures through heavy safety walls of the blockhouse, thereby weakening the structure. They also limit two or three persons per aperture and, obviously, require the observers to be present within the hazard area of the test pits. The disadvantages of a remotely located blockhouse are apparent because of the limitations on vision due to the great distances. Additional advantages to the test engineers by the use of this television monitoring system are also important. A close-up view provided by the television camera allows the engineers to detect, in time to stop the test firing, any evidence of fuel leaks or other malfunctioning of the system which could result in an explosion and major damage to the rocket motor and its entire setup. Continual observation of the rocket and exhaust flame during the firing period also enables the test engineers to note any irregularities in mixture ratio.

The demonstration is the culmination of a requirement of the Naval Air Missile Test Center at Point Mugu, and is the first time that observers have been able to witness at extremely close range details of rocket engine operation during firing and be provided with comfort and freedom from hazard. Aerojet engineer Ernest Vogt supervised the development of this new test observation technique and was aided in the initial demonstration by C. G. Pierce, electronics engineer, General Electric, and B. L. Dorman, Chief Test Engineer of Aerojet.

The General Tire & Rubber subsidiary, Aerojet, is the largest manufacturer of rocket motors in the world and the outstanding organization of its kind in the research and development of solid propellant and liquid propellant jet propulsion motors.

Signing of a contract with the U. S. Navy for rocket test facilities was made known today by Dan A. Kimball, Executive Vice President of the Aerojet Engineering Corporation, simultaneously with the announcement of a joint venture with The Ralph M. Parsons Company of Los Angeles and the Aerojet Engineering Corporation for the design and construction of technical test facilities for Army and Navy Guided Missile and Pilotless Aircraft test stations.

First of the joint contracts which the Parsons-Aerojet Company anticipate was signed yesterday at the Naval Air Missile Test Center at Point Mugu, California with Commander Deane E. Carberry, CEC, USN, Officer-in-Charge of Construction at Point Mugu, representing the U. S. Navy. Signers for the Parsons-Aerojet Company were Mr. Ralph M. Parsons of The Ralph M. Parsons Company and Mr. W. E. Zisch of the Aerojet Engineering Corporation. This contract — "Notice to Proceed" — for the architectural and engineering studies for the technical facilities of a permanent Naval Air Missile Test Center was awarded to the Parsons-Aerojet Company after extensive consideration of 15 groups by the Chief of the Bureau of Yards and Docks who appointed Comdr. Carberry to administer the contract and coordinate with Captain Robert S. Hatcher, USN, Commanding Officer at Point Mugu, on the technical requirements. The study will be conducted at Point Mugu, however, the results will be applicable to any Missile Test Center. Incorporated in this contract are studies to include only technical facilities which do not incorporate barrack buildings, mess halls or any buildings considered standard for Naval Bases. It is anticipated that the contract will be completed in approximately six months.

Liquid Propellant Rocket Power Plants

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Contributed by the Aviation Division for presentation at the Semi-Annual Meeting, Chicago, Illinois, June 16—19, 1947, of The American Society of Mechanical Engineers.

TABLE OF CONTENTS

(1) INTRODUCTION

This paper discusses some of the general features of liquid propellant rocket power plants, and the propellant systems that have been developed. The object of the discussion is to call attention to problems requiring solution in order to improve the reliability and performance of liquid propellant rocket power plants.

(2) PRINCIPAL ELEMENTS OF A BIPROPELLANT ROCKET MOTOR

Fig. 1 illustrates the principal elements of a bipropellant rocket motor utilizing a liquid oxidizer and a liquid fuel. It comprises a combustion chamber, an exhaust nozzle, a propellant injection system, a cooling system (not shown), and propellant control valves for regu-lating the flow of oxidizer and fuel. No ignition system is shown, because suitable propellants are available that react spontaneously on impingement of their liquid streams. Examples of spontaneously reacting liquids are nitric acid and aniline or furfuryl alcohol derivatives, and concentrated hydrogen peroxide and fuels containing hydrazine hydrate. The liquid oxygen-gasoline propellant system, the oldest combination that has been used, requires an auxiliary ignition system to initiate combustion. This is also the case for the nitro-methane mono-propellant system.

The rate of flow of the propellants into the combustion chamber and the mixture ratio (oxidizer weight rate of flow/fuel weight rate of flow) are governed by the areas of the respective injection orifices and the differential pressures acting across them. Spray type injectors have also been used but to a more limited extent.

(3) PUMP PRESSURIZED LIQUID PROPELLANT ROCKET JET PROPULSION SYSTEMS

There are two general methods of transporting the propellants: (1) the gas pressurization system, which utilizes an inert gas pressure on the storage tanks to force the propellants to the combustion chamber; (2) the pumping system which employs a pump in the flow path between the storage tanks and the combustion chamber.

The former system suffers from the disadvantages that the propellant tanks are heavy, because they must withstand the pressures involved, and the heavy inert gas pressure tank occupies considerable space. These disadvantages make inert gas pressurization impractical for a rocket of any appreciable range.
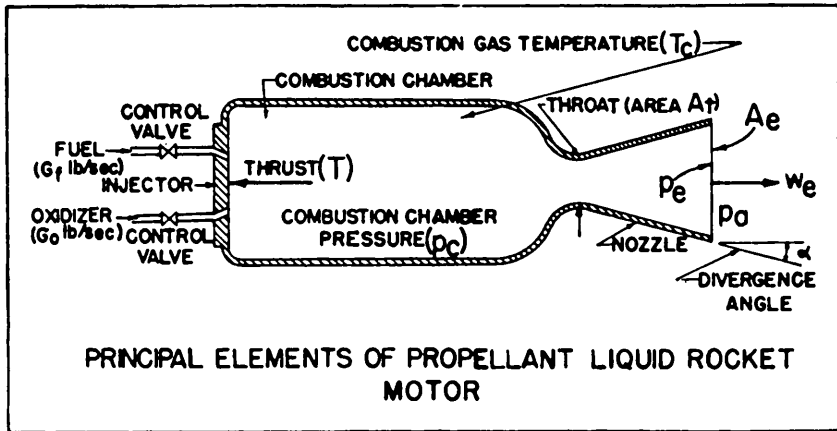
PRINCIPAL ELEMENTS OF PROPELLANT LIQUID ROCKET
MOTOR

FIGURE 1

The limitations imposed by gas pressurizing, stimulated the development of pumping systems for feeding the propellants to the rocket motor. Fig. 2 illustrates the principal elements of such a system. It should be realized, however, that the design features of an actual pumping plant are governed by the specific propellants employed and the number of rocket motors to be fed.

Modern pumping systems utilize a gas turbine to drive the propellant pumps; but reciprocating engine-driven pumping systems have been investigated experimentally. The high-pressure high-temperature gases for driving the turbine are produced in a gas generator by reacting suitable propellants. These need not be the same ones that are fed to the rocket motors. In those cases where the gases produced by reacting propellants in the gas generator are at too high a temperature, a diluent is mixed with them in the gas generator to reduce the temperature to a safe value. The diluent is usually a mixture of water and alcohol.

Turbine-driven pumping units of this type have been built for the acid-aniline propellant system, the liquid oxygen-

methyl alcohol system, the hydrogen peroxide-hydrazine hydrate methyl alcohol system, and for the nitromethane mono-propellant system.

The best known rocket power plants are the liquid oxygen-methyl alcohol unit used in the German V-2 missile, and the Walter power plant, Model 109-509-A1, used to propel the ME-163 airplane.

The diagrammatic arrangement for the V-2 power plant with its pumping units is illustrated in Fig. 3. (1) In this unit the gases for the turbine are produced by reacting concentrated hydrogen peroxide with a solution of calcium permanganate. These two chemicals are fed to the gas generator under pressure supplied by the nitrogen bottles shown in the figure. The advantage of the hydrogen peroxide-calcium permanganate reaction, — is that the gases produced, steam and oxygen, are at the low temperature of 790 F so that no diluent is required. The pressure of the gases entering the single stage partial admission turbine is approximately 350 psia. The turbine rotates at 4000 rpm and develops 500 to 600 hp.

ELEMENTS OF THE LIQUID PROPELLANT
PUMPING PLANT

FIG. 1
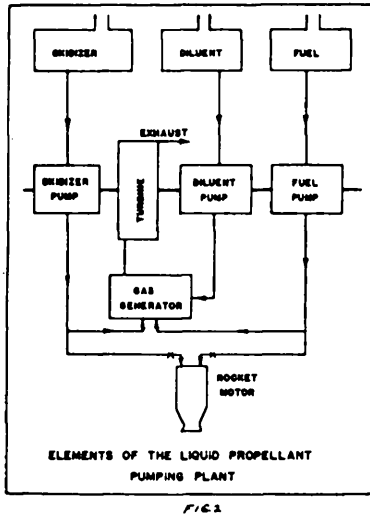
The operating chamber pressure for the rocket motor was 300 psia and the temperature of the chamber gases approximately 5400 R.

The Walter power plant, Model 109-509-A-1, used for propelling the ME 163 airplane used concentrated hydrogen peroxide and a solid catalyst, consisting of stones impregnated with calcium or sodium permanganate, to produce the gases for operating the turbine. This was the first successful rocket power plant to be employed as the sole propulsion means for a piloted airplane. It is of interest to note in passing that the ratio of the landing weight (4,620. lb.) to the take-off weight (9,020 lb.) for the ME 163 airplane is 0.51.
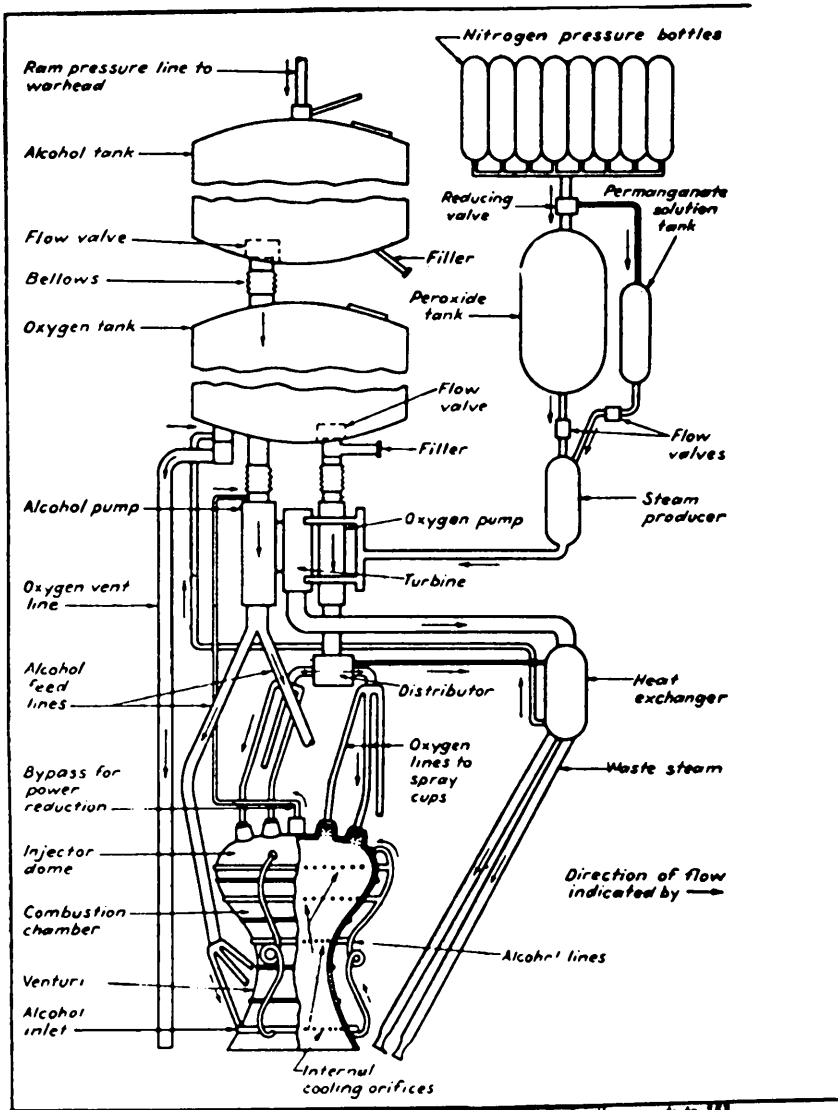
Fig. 4 (3) is a schematic flow diagram of the power plant showing the turbine pumping unit, which has a maximum operating speed of 16,500 rpm. The question arises as to how much propellant is required to operate the pumping unit compared to the total utilized for rocket power generation. Fig. 5 shows the total propellant consumption

and the ratio of the pumping plant consumption to the total consumption for various thrusts at sea level. Fig. 6 presents data on the propellant consumption distribution between the total and the pumping power requirements for variation of turbine speeds. It is seen that at full thrust the pumping plant requires approximately 4 per cent of the total propellant flow.

(4) PROBLEMS IN TURBINE PUMPING PLANTS

The brief descriptions of the rocket power plants presented demonstrate their potentialities as low weight propulsion means. Their ultimate development must be directed to securing greater reliability, still lower weight, and greater economy. At present the problem of sealing high speed pumps is largely unsolved, especially where leakage of the pumped fluid cannot be tolerated. This is particularly true of pumps for such propellants as liquid hydrogen and nitric acid. These same remarks are applicable to the seats for control valves which come in contact with corrosive oxidizers. The problem of cavitation at the pump entrance is not solved satisfactorily in most instances and there is a need for low weight auxiliary booster pumps for combatting this phenomena. The weight of the gas generating fluids consumed by turbine driven pumping plant ranges in most units from three to seven per cent of the total weight of the propellants furnished the rocket motors. Its absolute magnitude is, therefore, sufficiently large to warrant effort being expended to reduce it.

Some designs have been approached from the point of view that the goal is to use the same propellants for generating the gases for the turbine as are supplied to the rocket motors. It would appear that this is placing emphasis on

Nitrogen pressure bottles

Ram pressure line to warhead

Alcohol tank

Flow valve

Bellows

Oxygen tank

Reducing valve

Permanganate solution tank

Filler

Peroxide tank

Flow valve

Filler

Flow valves

Alcohol pump

Oxygen pump

Steam producer

Oxygen vent line

Turbine

Alcohol feed lines

Distributor

Heat exchanger

Bypass for power reduction

Oxygen lines to spray cups

Waste steam

Injector dome

Combustion chamber

Direction of flow indicated by ⟶

Alcohol lines

Venturi

Alcohol inlet

Internal cooling orifices

Schematic diagram of V-2's power plant plots sequence flow of fluids. Fuel ratio amounts to 183 lb./sec. of oxygen to 123 lb. of alcohol-water solution.

FIGURE 3

the wrong objectives. While this achievement reduces the number of fluids to be handled, it overlooks what should constitute the main objectives of this type of development; ease of control, safety, reliability, and lowest possible weight.

(5) LIQUID PROPELLANTS FOR ROCKET MOTORS

The ideal propellant (fuel plus oxidizer) for a rocket motor is one which satisfies the following principal requirements:

(a) Its calorific value per pound should be as large as possible;

(b) Its density should be high to keep the space requirements for containers at a low value;

(c) It should be easily stored and require simple handling equipment;

(d) Its corrosiveness should be low;

(e) Its toxidity should be low;

(f) Its performance should not be affected appreciably by ambient temperatures;

(g) Its ignition should be smooth and reliable;

(h) It should be stable chemically and not deteriorate appreciably over reasonable storage periods; and

(i) Its viscosity change with temperature should be small so that the pumping work at low operating temperatures will not be excessive.

These requirements are not satisfied completely by any of the rocket propellants in current use.

In general, liquid propellants may be divided into three principal groups:

(a) mono-propellants, (b) fuels, and (c) oxidizers.

(a) MONO-PROPELLANTS

A mono-propellants is a substance that requires no auxiliary material, such as an oxidizer, for the release of its thermochemical energy. To this class of materials belong such explosives as nitro-glycerine $C_3H_5(ONO_2)_3$, picric acid $C_6H_2(NO_2)_3OH$ and its derivatives, trinitro-toluene $C_6H_2(CH_3)(NO_2)_3$, ethylene glycol dinitrate $C_2H_4(ONO_2)_2$, nitromethane CH_3NO_2, and others.

For a liquid to be a satisfactory mono-propellant, it should satisfy the practical considerations discussed above. In addition, it must be so constituted that it is stable under all storage and handling conditions, yet it must decompose completely on injection into the combustion chamber. These two requirements are, in general, conflicting ones and greatly restricts the choice of materials for possible liquid mono-propellants.

To this same class of materials might be added such undeveloped sources of energy as that produced by the association of monatomic hydrogen, and atomic energy.

(b) FUELS

These are materials which do not liberate their thermochemical energy with great rapidity unless reacted with some auxiliary material, usually an oxidizer. The energy release is then an exothermic oxidation process.

There are a number of suitable materials that can serve as a liquid rocket motor fuel. Except for liquid hydrogen, hydrazine derivatives and the like, most of them are either alcohols or hydrocarbons. Since these fuels are used with an oxidizer, it is the energy liberation per pound of propellant mixture (oxidizer + fuel) that is of importance.

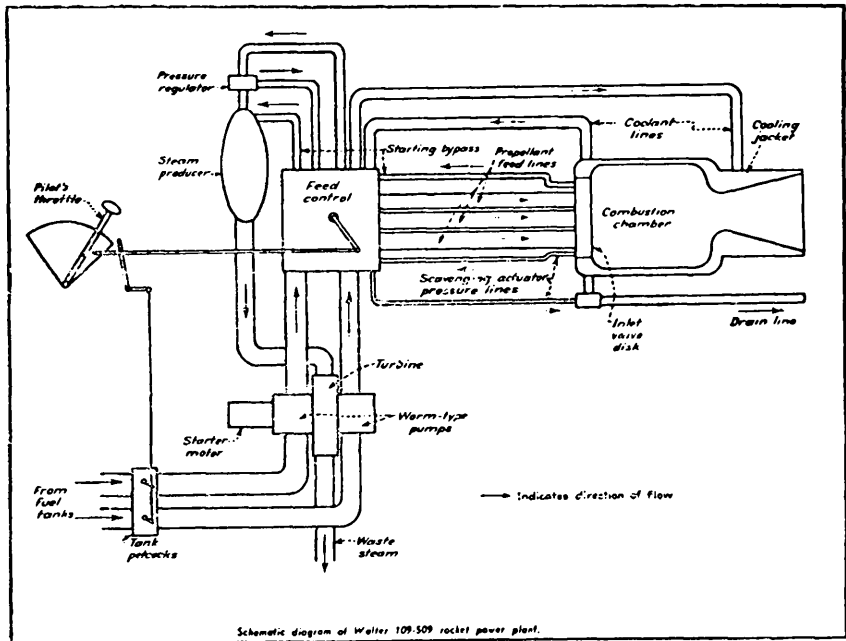Schematic diagram of Walter 109-509 rocket power plant.

FIGURE 4

(c) OXIDIZERS

The liquid oxidizer should contain a large percentage of oxygen in its composition. Of course, liquid oxygen and liquid ozone are the best from this point of view. During World War II the Germans developed a concentrated hydrogen peroxide and 90% hydrogen peroxide is currently being produced in this country. Another class of oxidizers which has been tested extensively is concentrated nitric acid and its modifications. Many data have been accumulated regarding the reaction between nitric acid oxidants and a variety of fuels.

In any rocket power plant it is the oxidizer that largely determines the major characteristics of the system and its design features. None of the principal oxidizers; liquid oxygen, nitric acid, and hydrogen peroxide satisfy the requirements for an ideal oxidizer. All of them introduce perplexing practical problems.

The three oxidizers discussed above have been investigated with a variety of fuels. The principal oxidizer-fuel combinations are presented in Table 1.

Table 1.   LIQUID BIPROPELLANT SYSTEMS

Liquid Oxygen (O_2)
Liquid Hydrogen (H_2)
Gasoline (C_8H_{18})
Ethanol (C_2H_5OH)
Methanol (CH_3OH)
Liquid Ammonia (NH_3)
Nitric Acid (HNO_3)
Aniline ($C_6H_5NH_2$)
Furfuryl Alcohol
    ($C_4H_3OCH_2OH$)
Monoethylaniline
    ($C_2H_5C_6H_4NH_2$)
Hydrazine (N_2H_4)
Hydrogen Peroxide (H_2O_2)
Ethanol (C_2H_5OH)
Methanol (C_2H_3OH)
Hydrazine (N_2H_4)
Ethylene Diamine
    ($C_2H_4N_2H_4$)

FIG. 5
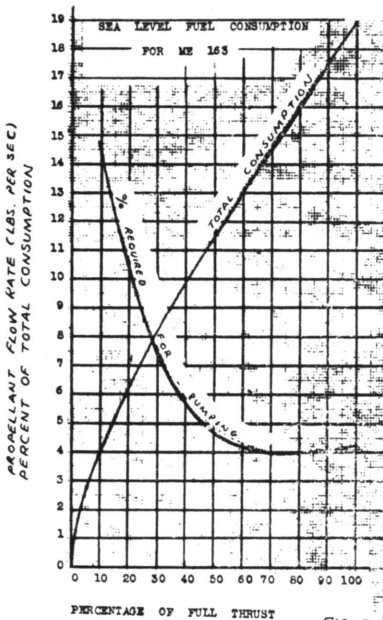
The liquid oxygen system was used in the earliest experiments with rocket motors, usually with either gasoline or methyl alcohol and many successful motors have been developed. As already mentioned the German V-2 missile employs liquid oxygen and methyl alcohol. Those familiar with the handling problems of liquid oxygen advocate its use, despite its disadvantages from other points of view.

Nitric acid has been investigated extensively during the war; and also its modifications such as, red fuming nitric acid containing up to 13 per cent nitric oxide; white fuming nitric acid; the mixed acid, a mixture of nitric acid with sulphuric acid containing sulphur trioxide. Those who have worked with acid oxidizers and have developed techniques for handling them without danger to personal are its proponents.

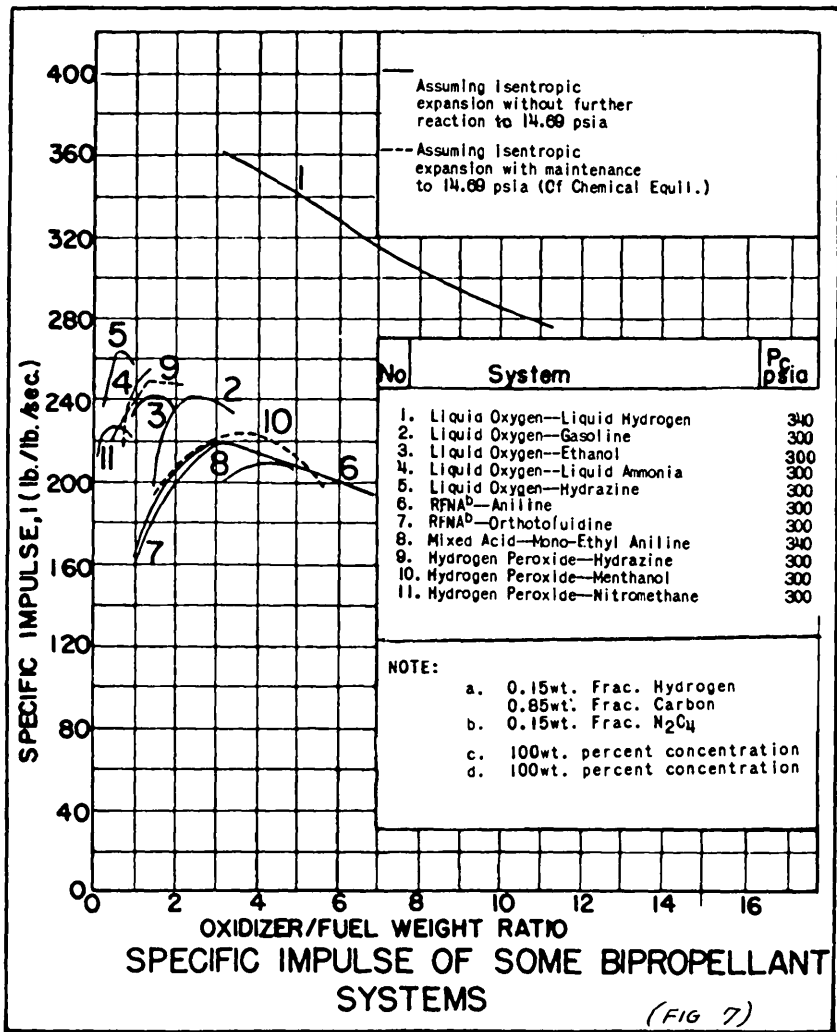TOTAL   CONSUMPTION   (LBS.   PER   SEC)   FIGURE 6
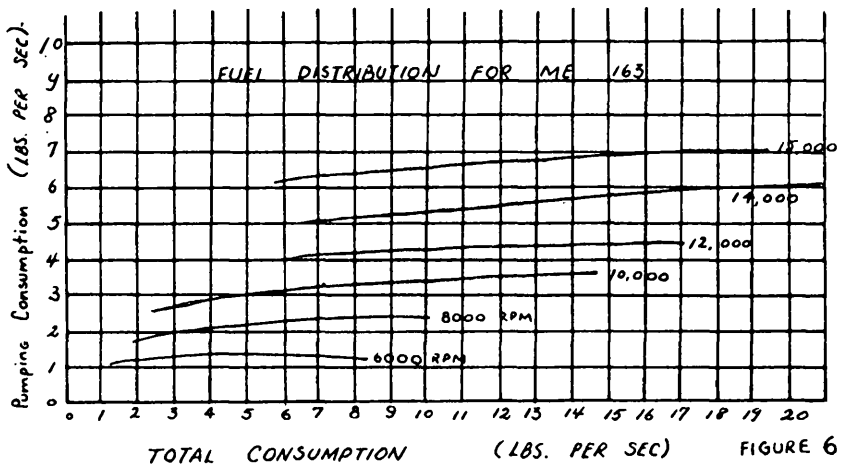
Nitric acid introduces fewer storage problems than either liquid oxygen or hydrogen peroxide. It is extremely corrosive and the containers must be of stainless steel. It introduces problems of materials for valve seats, packing, etc. Most of these have, however, been solved to the point where usable units using nitric acid have been developed.

Concentrated hydrogen peroxide, in concentrations of 80 to 90 per cent, was used on a large scale by the Germans. It has also been used experimentally in the United States. Rocket experience with concentrated hydrogen peroxide in this country is more limited than it is for either liquid oxygen or nitric acid. Data on its properties have been published recently. The more widespread use of hydrogen peroxide will naturally be hampered by the lack of well delevoped handling techniques and less familiarity with its properties. This oxidizer has much to recommend it and doubtless its application will increase despite the face that its handling and storage may involve certain dangers unless precautionary methods are instituted.

The aforementioned oxidizers have an undesirable property common to all of them. Because of their great chemical activity, they introduce problems of material selection. The most corrosive ones also make it necessary to replace certain parts at frequent intervals and to service equipment at regular periods. these are not, however, insurmountable problems, and serve as the stimuli for developing more adequate materials, and for investigating other promising oxidizers.

(6) PERFORMANCE OF
LIQUID PROPELLANTS

The performance of rocket fuels is stated in terms of either the specific impulse developed — the impulse obtained from the consumption of one pound of propellant — or the specific thrust — developed with a consumption of one pound per second. Both of these criteria have the same numerical magnitude. The specific impulse is related to the effective exhaust velocity w, by the relation $gI_{sp} = w$; where g is the acceleration due to gravity.

400

360

320

280

240

200

160

120

80

40

0

SPECIFIC IMPULSE, I (lb./lb./sec.)

OXIDIZER/FUEL WEIGHT RATIO

0   2   4   6   8   10   12   14   16

— Assuming Isentropic expansion without further reaction to 14.69 psia

--- Assuming Isentropic expansion with maintenance to 14.69 psia (Cf Chemical Equil.)

No	System	Pc psia
1.	Liquid Oxygen—Liquid Hydrogen	340
2.	Liquid Oxygen—Gasoline	300
3.	Liquid Oxygen—Ethanol	300
4.	Liquid Oxygen—Liquid Ammonia	300
5.	Liquid Oxygen—Hydrazine	300
6.	RFNA^b—Aniline	300
7.	RFNA^b—Orthotoluidine	300
8.	Mixed Acid—Mono-Ethyl Aniline	340
9.	Hydrogen Peroxide—Hydrazine	300
10.	Hydrogen Peroxide—Menthanol	300
11.	Hydrogen Peroxide—Nitromethane	300

NOTE:

a. 0.15wt. Frac. Hydrogen
    0.85wt. Frac. Carbon
b. 0.15wt. Frac. N_2C_4

c. 100wt. percent concentration
d. 100wt. percent concentration

SPECIFIC IMPULSE OF SOME BIPROPELLANT SYSTEMS

(FIG 7)

Examination of the equations for the effective exhaust velocity and the specific impulse shows that the magnitudes of these criteria depend directly upon the parameter $\sqrt{T_c/m}$, and to a smaller extent upon k and pe/pc. It is of interest, therefore, to examine the available pertinent information for some of the propellants that are being used.

Fig. 7 compares the calculated values of the specific impulse for several bipropellant systems operating at the same chamber pressure. It is apparent from the curves that excepting the liquid-oxygen liquid-hydrogen combination, the maximum values for all the others range from $I_{sp} = 210$ to 260 lb-sec/lb.

Fig. 8 compares the calculated combustion chamber temperatures for these same propellant systems, and Fig. 9 compares the calculated mean molecular weights of the combustion gases. Fig. 10 presents the corresponding values of the mean specific heat ratio.

All curves are based on thermochemical calculations and the basis of the thermochemical calculations are indicated on the curve sheets.
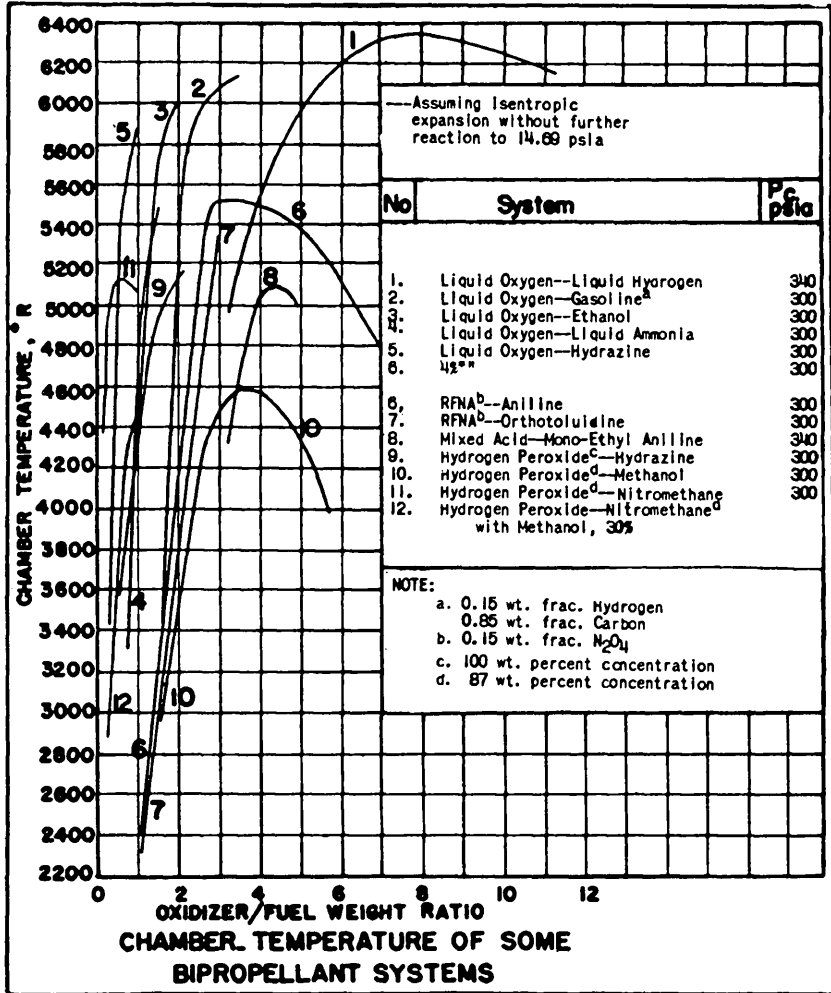
CHAMBER TEMPERATURE, °R

OXIDIZER/FUEL WEIGHT RATIO

CHAMBER TEMPERATURE OF SOME BIPROPELLANT SYSTEMS

—Assuming isentropic expansion without further reaction to 14.69 psia

No	System	Pc psia
1.	Liquid Oxygen--Liquid Hydrogen	340
2.	Liquid Oxygen--Gasoline[a]	300
3.	Liquid Oxygen--Ethanol	300
4.	Liquid Oxygen--Liquid Ammonia	300
5.	Liquid Oxygen--Hydrazine	300
6.	4½°"	300
6.	RFNA[b]--Aniline	300
7.	RFNA[b]--Orthotoluidine	300
8.	Mixed Acid--Mono-Ethyl Aniline	340
9.	Hydrogen Peroxide[c]--Hydrazine	300
10.	Hydrogen Peroxide[d]--Methanol	300
11.	Hydrogen Peroxide[d]--Nitromethane	300
12.	Hydrogen Peroxide--Nitromethane[d] with Methanol, 30%	

NOTE:
a. 0.15 wt. frac. Hydrogen
   0.85 wt. frac. Carbon
b. 0.15 wt. frac. N_2O_4
c. 100 wt. percent concentration
d. 87 wt. percent concentration

FIGURE 8

From these data it appears that since the maximum temperature in the combustion chamber is limited by dissociation reactions and, from a practical standpoint, by the materials or cooling methods available for constructing rocket motors, the main approach to increasing the specific impulse is to lower the molecular weight of the combustion gases.

The molecular weight of the gases can be lowered by using fuels rich in hydrogen, but this approach is not a panacea. Fuels rich in hydrogen have low specific weights and lower the average specific weight for the propellants. Consequently the tanks for the propellant supply must be of large volume, thereby greatly increasing their weight. Another guide to the suitability
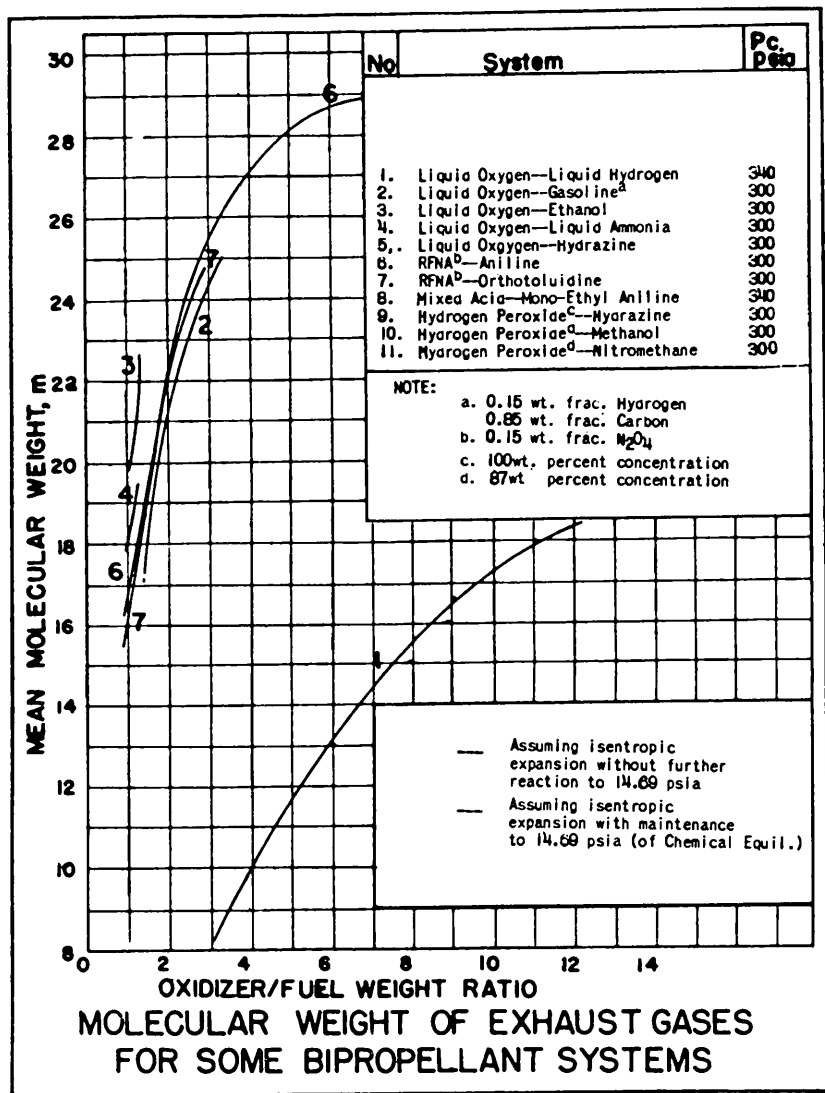
No	System	Pc, psia
1.	Liquid Oxygen--Liquid Hydrogen[a]	340
2.	Liquid Oxygen--Gasoline[a]	300
3.	Liquid Oxygen--Ethanol	300
4.	Liquid Oxygen--Liquid Ammonia	300
5,.	Liquid Oxygen--Hydrazine	300
6.	RFNA[b]--Aniline	300
7.	RFNA[b]--Orthotoluidine	300
8.	Mixed Acid--Mono-Ethyl Aniline	340
9.	Hydrogen Peroxide[c]--Hydrazine	300
10.	Hydrogen Peroxide[d]--Methanol	300
11.	Hydrogen Peroxide[d]--Nitromethane	300

NOTE:

a. 0.15 wt. frac. Hydrogen
   0.85 wt. frac. Carbon
b. 0.15 wt. frac. N_2O_4
c. 100 wt. percent concentration
d. 87 wt percent concentration

— Assuming isentropic expansion without further reaction to 14.69 psia

— Assuming isentropic expansion with maintenance to 14.69 psia (of Chemical Equil.)

MEAN MOLECULAR WEIGHT, m

OXIDIZER/FUEL WEIGHT RATIO

MOLECULAR WEIGHT OF EXHAUST GASES FOR SOME BIPROPELLANT SYSTEMS

FIGURE 9

of propellants is the product (specific weight x specific impulse), which is called the density impulse. A comparison of different propellant systems on this basis is presented in Fig. 11. This indicates that from a performance standpoint and considering a one-step rocket, there is little to choose between the liquid-oxygen, nitric acid, and hydrogen peroxide systems. The liquid-oxygen system appears less favorable. It should be realized that a direct comparison on this basis may not be fully justified since it neglects the improvement of tank design and rocket operation (stepped rocket) which influence the overall performance.

No	System	Pc psia
1.	Liquid Oxygen..Liquid Hydrogen	340
2.	Liquid Oxygen--Gasoline[a]	300
3.	Liquid Oxygen--Ethanol	300
4.	Liquid Oxygen--Liquid Ammonia	300
5.	Liquid Oxygen--Hydrazine	300
6.	RFNA[b]--Aniline	300
7.	RFNA[b]--Orthotoluidine	300
8.	Mixed Acid--Mono-Ethyl Aniline	340
9.	Hydrogen Peroxide[c]--Hydrazine	300
10.	Hydrogen Peroxide--Methanol	300

NOTE:
a. 0.15 wt. frac Hydrogen
   0.85 wt. frac. Carbon
b. 0.15 wt. frac. N₂O₄
c. 100 wt. percent concentration
d. 87 wt. percent concentration

—Assuming isentropic expansion without further reaction to 14.69 psia

—Assuming isentropic expansion with maintenance to 14.69 psia (of Chemical Equil.)

RATIO OF SPECIFIC HEATS FOR SOME BIPROPELLANT SYSTEMS

FIGURE 10

Before looking ahead to the future possibilities of rocket performance, consideration should be given to performance characteristics of present rockets. The foremost two are the V-2 missile and the ME 163 aircraft.

Professor V. von Braun who was the main instigator of the V-2 recognized that his rocket was still undeveloped at the close of the war, and has since stated its stage of development may

be compared to the stage of development of the airplane at the close of World War I. (2) However, in spite of its recent development the V-2 rocket presents extraordinary performance figures. These are tabulated in Table 2.

The rocket powered ME 163 airplane, a later development than the V-2 rocket gave a performance that is noteworthy. Fig. 12 shows two views of the ME 163, and Table 3 presents certain performance data.

No	System	Pc psia
1.	Liquid Oxygen--Liquid Hydrogen	340
2.	Liquid Oxygen--Gasoline^a	300
3.	Liquid Oxygen--Ethanol	300
4.	Liquid Oxygen--Liquid Ammonia	300
5.	Liquid Oxygen--Hydrazine	300
6.	RFNA^b--Aniline	300
7.	RFNA^b--Orthotoluidine	300
8.	Mixed Acid--Mono-Ethyl Aniline	340
9.	Hydrogen Peroxide^c--Hydrazine	300
10.	Hydrogen Peroxide^d--Methanol	300
11.	Hydrogen Peroxide^d--Nitromethane	300

NOTE:
a. 0.15wt. frac. Hydrogen
   0.85wt. frac. Carbon
b. 0.15wt. frac. N_2O_4
c. 100. wt. percent concentration
d. 87 wt. percent concentration

--- Assuming isentropic
    expansion without further
    reaction to 14.69 psia

— Assuming isentropic
  expansion with maintenance
  to 14.69 psia (of Chemical Equil.)

DENSITY IMPULSE, I_d (I_{sp} X Specific Gravity)

OXIDIZER/FUEL WEIGHT RATIO

DENSITY IMPULSE OF SOME BIPROPELLANT SYSTEMS

FIG. II

Table 2. V-2 PERFORMANCE CHARACTERISTICS (2)

Maximum Speed .... 3,400 m.p.h
Maximum Thrust .... 69,000 lb.
Maximum Horse-power (near end of powered flight)........................ 625,000 h.p.
Maximum Acceleration .... 8 g.
Ceiling .... 350,000 ft.
Range .... 220 mi.
Endurance .... 5 min.
Fuel Consumption at near end of powered flight.............................. 1.6 lb/hp-hr.
Payload .... 2,150 lb.
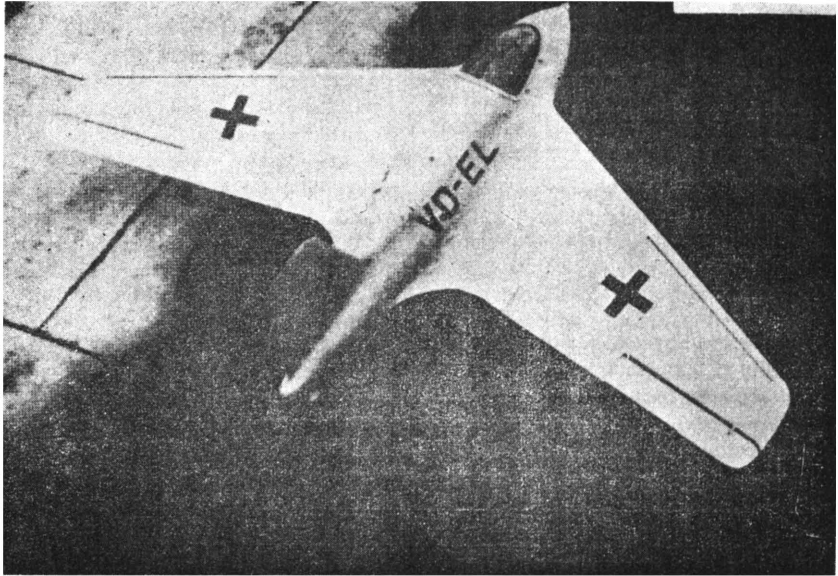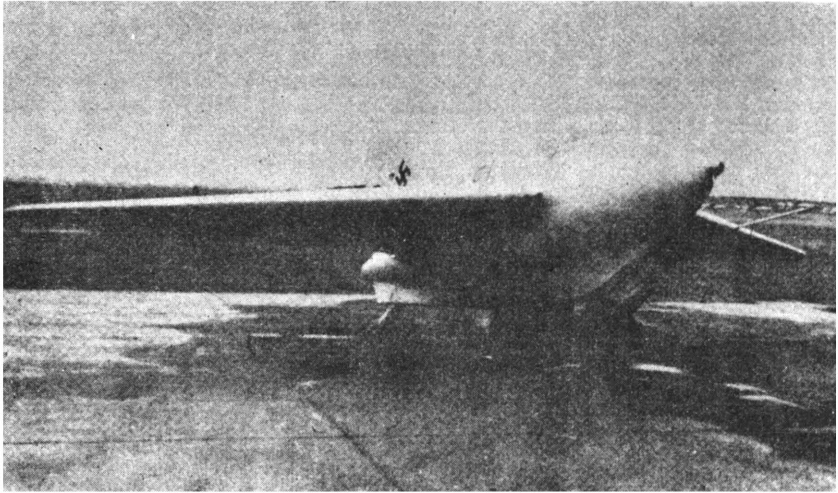
FIGURE 12

Table 3.   ME 163 PERFORMANCE CHARACTERISTICS

Maximum Speed	550 m.p.h.
Maximum Thrust	3,300 lb.
Maximum Horse-power	4,850 h.p.
Maximum Rate of Climb	20,000 ft/min. at 35,000 ft.
Time to reach 40,000 feet	3 min.
Normal Powered Endurance	12 min.
Propellant Consumption at Maximum Thrust........................	14.1 lbs./h.p.-hr.
Payload	1,470 lb.

DEPENDENCE OF ESCAPE VELOCITY ON DISTANCE FROM THE EARTH

$$V_e = h_e \sqrt{\frac{2 g_o}{h_e + h}}$$

V_e = escape velocity fps
h_e = radius of earth, ft.
h = altitude, ft.
g_o = acceleration due to gravity at sea level.

DISTANCE FROM EARTH'S SURFACE

VELOCITY (10³ FT / SEC)

| 0 | 10 | 20 | 30 (10^6 FT) | 40 |
| 0 | 1.9 | 3.8 | 5.7 (10^3 MI) | 7.6 |
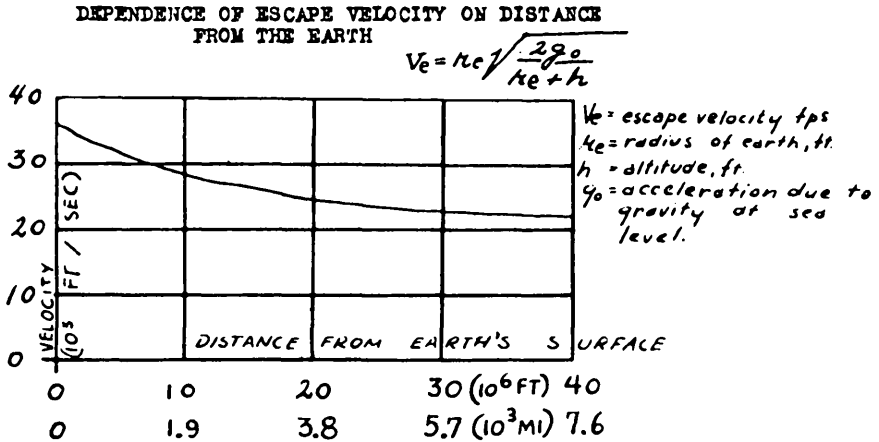
FIGURE   13

It is reasonable to expect that future development and research will eventually improve the propellant consumption rates.

Many men have speculated on the possibility of escape from the earth — and it appears that with the available propellants and appropriate design for the rocket this possibility can be realized. The primary problem is to attain a velocity great enough to escape the earth's gravitational field. The escape velocity at sea level is 36,700 feet per second. Fig. 13 (4) shows the variation of escape velocity with altitude. The calculated values neglect the air resistance forces, the earth's peripheral velocity, and the mass of the moon, because they have negligible effect on the rocket flight. A gun-fired projectile would require an initial velocity of 36,700 feet per second to escape from the earth; today this is not possible of attainment.

The problem of escaping rockets fired vertically from the earth has been studied recently by F. J. Malina and M. Summerfield. (4) The study concludes that escape is possible with engineering materials and available propellants. However, the rocket must be of the multiple-step type; that is, the rocket is composed of steps or sections which as they exhaust their propellants are progressively jettisoned from the rocket system. The authors conclude that escape with a single step rocket is impossible with the available liquid propellants.

Figure 14 presents the overall mass ratio required to attain 35,000 feet per second velocity as a function of the number of steps. The overall mass ratio is defined as the ratio of the initial total mass, at take-off, to the mass of the payload. Three different propellant combinations are presented; nitric acid-aniline, liquid oxygen-alcohol, and liquid oxygen hydrogen. In making the calculations the same structural factors, which allows for the weight of the propellant tanks, that were used by Malina and Summerfield were applied to the nitric acid-aniline and the liquid oxygen-alcohol systems. The structural factor for the liquid oxygen-liquid hydrogen system was increased 30 per cent above that used by the aforementioned authors, thereby making the curve for liquid oxygen-liquid hydrogen more conservative.
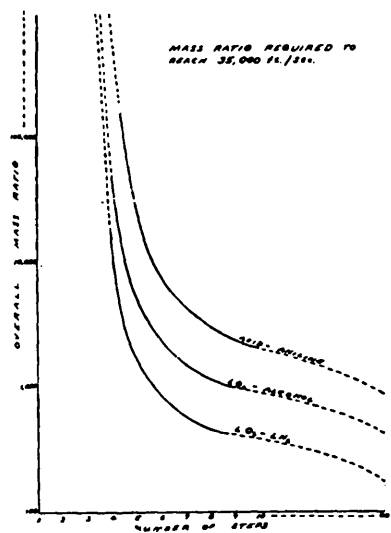
It is seen from Fig. 14, that if a multiple stepped rocket is used the best performances, as judged by payload carried, is given by liquid oxygen and liquid hydrogen. It is of interest to note that the 10-step liquid oxygen-liquid hydrogen rocket which can carry a 100 pound payload is approximately twice the weight of the present V-2 rocket.

(7) COOLING OF ROCKET MOTOR

The adequacy of a rocket motor depends almost entirely upon its ability to perform without damage at the high temperatures encountered. Of the total heat liberated by the reaction of the propellants approximately 5 per cent is transferred to the motor and nozzle walls; this amounts to 120 to 200 Btu per second. There are at present four major methods for protecting the motor walls from the high temperatures and this transferred heat; (1) the motor walls and nozzle can be constructed of suitable materials and heavy enough to absorb the heat during the operating period; (2) one of the propellants in its

FIGURE 14

passage to the injection system can be circulated around the heat absorbing surfaces to keep them cool; (3) high temperature-resistant (refractories) materials can be used for lining the heated surfaces; and (4) film cooling.

Method (1) employs the principles of supplying a heat reservoir capable of receiving the total quantity of heat to be absorbed without raising the metal temperatures to dangerous values. The most suitable materials for this type of cooling are those for which the product (specific heat x thermal conductivity x density) has high values. The best material from this standpoint is copper. The foregoing criterion is not, however, a unique guide to motor construction, since considerations of weight and strength are of greater importance. In any case, as the required operating duration for an uncooled motor is raised, the requisite motor weight becomes excessive for practical use. Consequently, this type of motor construction is adaptable only to rocket motors which are

to operate for short durations and employ low oxidizer-fuel ratios to reduce the combustion temperature. Satisfactory motors of this type have been built for the durations up to 35 seconds. They weigh more than rocket motors of equal thrust output that employ cooling Method (2).

Method (2), regenerative cooling, appears currently to be the best approach to the solution of the heat problem. It has the advantage that once the cooling system has been developed correctly, the motor can be operated for long durations (several minutes at a time) without damage. Furthermore, regeneratively cooled motors can be made extremely light in weight, the thrust-weight ratio increasing markedly with the larger thrusts.

Little advance has been made using Method (3), but it seems probably that good results should be obtainable by combining it with Method (2).

Method (4), film cooling, appears to offer a positive means for combatting high temperatures in the rocket motor. It is based on forming a complete film of liquid over the inside walls of the chamber and nozzle, then its evaporation keeps these surfaces cool. The potential effectiveness of this type of cooling can be judged by the results obtained in the German V-2 rocket motor which used a crude version of this form of cooling. By permitting approximately 7 per cent of the total fuel (alcohol) consumption to enter the motor through a large number of holes and provide a form of film cooling, temperatures of 5400 R could be withstood without damage in a motor constructed from plain carbon steel.

(8) OPERATION AT HIGH CHAMBER PRESSURES

The likelihood of achieving a significant increase in specific impulse with chemical propellants of high specific weight is not too promising. It appears, therefore, that development efforts in rocketry will have to be directed along those lines which can lead to the accomplishment of small but useful improvements in performance. In this connection the possibilities of higher chamber pressure, its advantages and disadvantages deserve more thorough study. Particularly for applications employing turbine driven pumping units.

Fig. 15 presents calculated curves of the thrust coefficient C_T versus chamber pressure for gases with $k = 1.2$. The upper curve represents ideal values, and the lower curve, marked $V = 0.936$, values that are probably attainable in a well designed rocket motor.

At the present time most rocket jet propulsion systems except those for nitro-methane, have been designed for operation with a chamber pressure of 300 psia approximately. Higher chamber pressures, or expansion ratios increase the value of C_T as seen from the figure. It is true that the rate of increase in C_T is smaller as chamber pressure is increased above 300 psia. Nevertheless, a gain of approximately 14 per cent can be achieved by raising the chamber pressure to 1500 psia. Some of this gain is offset by the weight of the added stages required for the pumps, propellant control valves, and piping from the pumps to the rocket motors.

For operation at higher chamber pressures more and heavier stages would be required for the last stages of the pumps, and some increase in the weight of the valving and discharge piping. No analysis has been made to determine the magnitude of this weight increase, but from a cursory examination it does not appear prohibitive, especially when it is realized that any reduction in propellant consumption makes the unused propellant available for use at altitudes where its effectiveness is substantial.
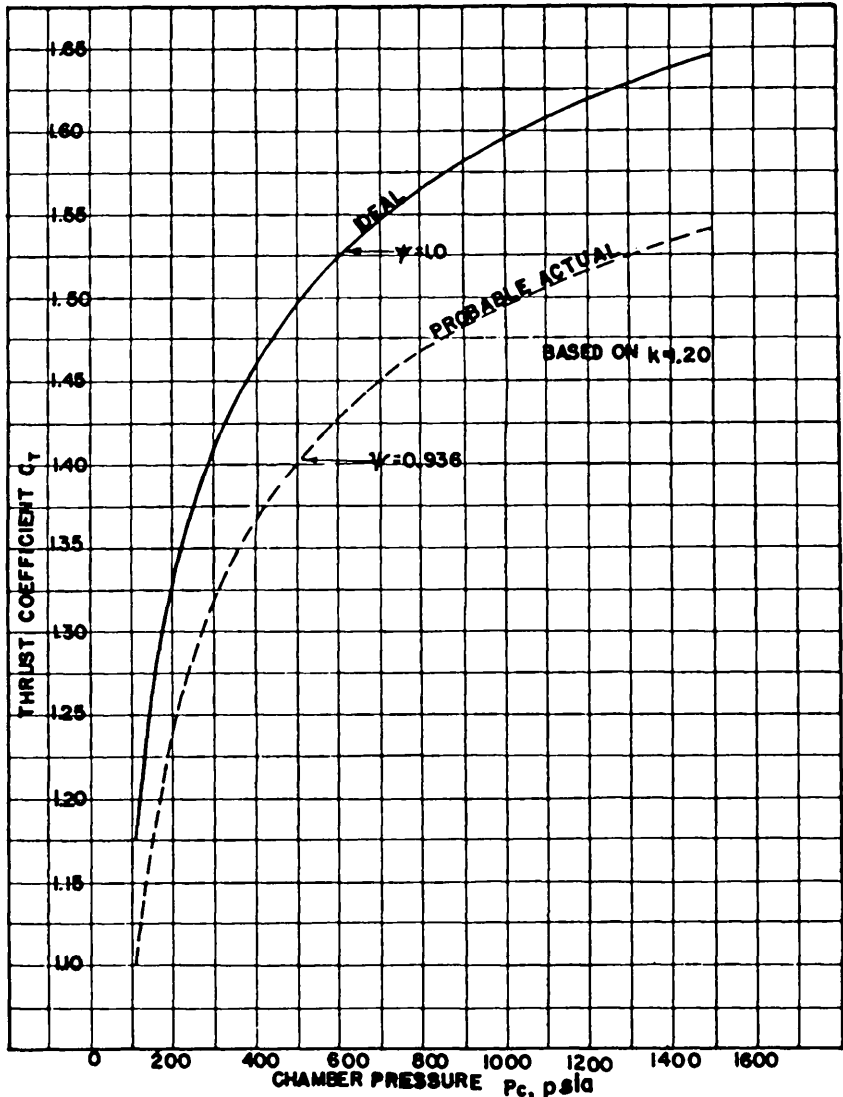
FIGURE 15

There are, of course, numerous problems that must be solved to make high chamber pressure operation realizeable. Perhaps, the most serious one is that related to cooling the motor. Consequently, the research concerning methods to improve the cooling of rocket motors so that they can be operated under the most severe conditions should be prosecuted with vigor. One of the most promising approaches appears to be film cooling which is controlled so that the amount of liquid for cooling purposes will not exceed 1 or 2 per cent of the total propellant consumption.

The Analysis Division, Intelligence T-2 Rocket Unit, Propulsion Section, Wright Field has conducted studies regarding film cooling for rocket motors to operate at 1500 psia chamber pressure. This study is in connection with the development of a high thrust rocket which is to operate for extended burning periods. The data provided by that agency indicates that three insulating characteristics are derived from film cooling, namely: (1) part of the cooling film breaks down on the motor walls into a layer of carbon about 0.03 in. thick; (2) a liquid layer of fuel passes over the carbon deposit; and (3) a vapor layer forms over the liquid layer. These insulating properties plus the heat of vaporization have made it possible to operate a Dural combusion chamber at 600 psia for 3 and 4 hours. It appears that with film cooling, combustion pressures of 1500 psia will be attainable.

(9) CONCLUSIONS

In conclusion it can be said that the rocket power plant is at that stage of development, where it has demonstrated its potentialities for making supersonic flight possible. To obtain the ultimate from this mode of propulsion, cooling methods must be developed to permit operating existing propellants with higher chamber pressures, and research directed to raising values of $\sqrt{T_c}/m$ is needed. In addition several practical problems must be solved to improve reliability: Control systems must be improved and the development of more adequate materials for valve seats, packings, and pump seals is of importance. Control means must be developed for operating rocket motors at the most favorable chamber pressure regardless of the thrust requirement.

It appears that really significant improvement in specific impulse of liquid propellants such as two or three times present values must await either the development of more powerful chemical propellants, or the application of other forms of energy.

(10) ACKNOWLEDGEMENTS

The author wishes to express his appreciation to Mr. W. J. Hesse, Purdue University, for his assistance in the preparation of the paper.

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LIST OF FIGURES

LIQUID ROCKET MOTOR TESTING

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INTRODUCTION

The liquid rocket power plant has attained a position of importance which was hardly imaginable a few years ago to most of the engineering profession and the public. Its continued advancement can be expected as long as man persists in wanting to go higher, farther and faster. One aspect of rocket development which is of fundamental importance is, of course, testing. The major problems of rocket testing — problems which differentiate it from other engineering test — arise from the presence, in combination, of high pressures, high temperatures, high velocities, high noise levels, high flow rates, high heat transfer rates, high forces, high design stresses, short times, and novel and dangerous working fluids. It is clear that the old adage "One test is worth a thousand expert opinions" could hardly find a better application.

The basic purpose of the present paper is to review the field as a subject of general and continuing interest to engineers. The paper first delineates the general types of rocket testing and the general apparatus under test. The following are then discussed in more detail: (a) apparatus under test; (b) purposes of test; (c) test facilities; (d) test structures; (e) equipment; (f) instrumentation; and (g) methods and results of tests. These topics are considered primarily with regard to research and development "hot-testing" or actual firing of rocket motors.

APPARATUS AND TEST PURPOSES — GENERAL

Liquid rocket power plants may be classified both by application and by the forms of their major components. The important applications today are as aircraft JATO units, as speed-booster units, and as main power plants for piloted aircraft, pilotless aircraft, missiles and sounding rockets. Each of these applications calls for a variety of designs. The power plant may be almost rudimentary in construction and performance or may be more complex and refined than the German A-4 power-plant. Its thrust may be 100 pounds or 100,000 pounds.

The evolution of all these various power plants requires research testing, basic and applied development testing, production testing and pre-flight testing. Production and preflight testing will usually employ the techniques and equipment evolved in the research and development testing. In this paper, therefore, attention will be focused more on the experimental phases of testing.

APPARATUS UNDER TEST

In experimental work the apparatus under test may range from a complete power plant to a new combustion chamber material. Let us examine the elements which comprise liquid rocket systems.

Rocket systems, both in powerplants and in test apparatus and equipment, have four basic groups of elements (Figs. 1 and 2). These groups are: (a) the propellant pressurizing system; (b) the propellant tanks, piping and control valves; (c) the propellants; and (d) the motor (consisting of combustion chamber, propellant injector and in some cases an igniter). The combination of motor plus propellant control valves is frequently called the engine.

Pressurizing systems can be classified as compressed-gas systems, chemically-generated-gas systems, and pump-pressurized systems. There are many variations and combinations of each. Propellant tanks, piping and control valves might almost be considered part of the pressurizing system.

Propellants (they should not be called "fuels") are classified as mono-, bi-, and tri-propellants. The bi-propellant combinations are the most widely used. They consist of an oxidizer, such as liquid oxygen, various nitric acids, or hydrogen peroxide, and a fuel, such as alcohol, gasoline or aniline. Substances such as aniline are used with nitric acid because the two ignite on contact, thus eliminating the need for an igniter squib or burner. The propellants are frequently as much the subject of test as the motor, since their properties have a major influence on motor design.

The elements of the motor take numerous forms. The chamber may be cooled or uncooled. It may be cooled by circulating water or one of the propellants through a jacket. It may be cooled by blanketing the walls with vapor introduced in any of several ways. If uncooled, it may depend on a large mass of heat-absorbent metal. Or it might be lined with a ceramic or a refractory metal. It may embody several of these principles together. Furthermore, its nozzle throat may be small

or large as compared with the combustion space. The combustion space may take a variety of shapes and the nozzle angle can vary.

The injector can also take a wide variety of forms and arrangements. It may employ multiple holes which produce impinging propellant streams. It may employ hollow cone and whirl sprays. It may employ sheet sprays. The openings may be large or small, few or many, and their orientations relative to each other and the motor can vary widely.

PURPOSE OF TESTING

Rocket development requires a great deal of testing on components of pressurizing systems — turbopump tests, tankage material research, and so on. Testing of a similar nature is also required on the engine components — pressure-drop calibrations of injectors and chamber jackets and valve flowrate determinations, for example. This body of testing, often called "cold-testing", is of great importance.

However, it is "hot-testing" of propellants and motor components which involves unique problems and techniques. It is the motor which is the business-end of a rocket system, and which is of most interest in a review of rocket testing.

The liquid rocket motor may appear to be elementally simple in theory and design. In actuality it harbors an unbelievable number of important variables. Testing is essential because the variables are at once so widely separate in subject and yet so closely intermeshed in action.

Testing of course is a means, not an end. The general goal is three-fold: to produce apparatus which works; to evolve apparatus which works better; and to ascertain the laws and relationships existing in the apparatus.
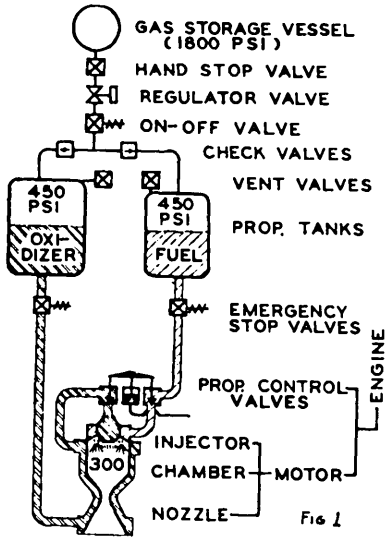
Figure 1. Elements of compressed-gas-pressurized liquid rocket system for motor or propellant testing (numbers denote typical operating pressures in psi).

The primary purposes of hot-testing are to obtain qualitative information and quantitative data. In research testing the emphasis is on data; in applied development it is more on information. This balance is a shifting one. For example, the designer of a new motor for a production JATO unit first seeks answers to such starkly qualitative questions as: "Did it blow up?" "Did it collapse?" "Did it burn out?" When these can be answered favorably, his interest turns to more quantitative data such as whether or not the motor produces rated thrust and specific impulse at the rated chamber pressure.

The researcher is concerned more with studies of the interrelations between isolated variables. He may be studying the effect of high chamber pressures on combustion efficiency in the motor. Of course, some studies which appear in prospect to be beautifully integrated may bog down simply because the apparatus used proves incapable of standing up under the conditions encountered. The objective then reverts to developing a more durable apparatus.

It should be mentioned that the isolating of some variables is rather difficult. In such cases a compromise must be made if the testing program is to avoid unwieldiness.

TEST FACILITY

The location of the test facility must be considered relative to the locations of (a) neighboring communities, and (b) the main plant and shops. The initial location of the company or agency undertaking testing is of course a major determinant. Equally important are the neighbors.

During the war one Navy contractor was haled into court by an old resident living about a mile from the test facility. The elderly gentleman complained the operations were shaking his house and his nerves. He was reminded of course that there was a war on. He knew that, he said, but "Do they have to fight it in my back yard?"

The location of the rocket test facility relative to neighboring communities is determined mainly by the size or scale of the motor being tested. It is obvious that from the points of view of safety to the surrounding areas and peace of mind to inhabitants, a 20,000 pound thrust motor requires more elbow room than a 200-pound motor.

Regarding noise, it is difficult to say what the relation between distance-to-neighbors and thrust should be. The type of terrain is important — the distance should be greater on flat bare ground or over water than in hilly, wooded areas. In addition, a noise muffler might be used, although relatively little investigation has been carried out to date on this difficult problem.
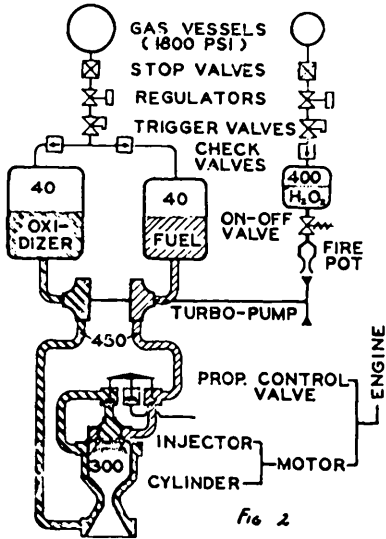
Figure 2. Elements of turbopump-pres-
surized power plant (num-
bers denote typical operating
pressures in psi).

Another problem of location which must be considered in parallel with noise is that of harmful fumes. Such fumes may be evolved in the combustion of leaded gasoline or special propellants. Fumes can also be released by the various nitric acids. The failure of an acid line or explosion of a motor can create a volume of irritating and corrosive fumes which may remain in the vicinity for several minutes before rising. One solution used at the Annapolis project was to make runs only when the wind was blowing out over Chesapeake Bay. Testing was at times severely limited, of course.

The hazards to neighbors attendant upon a motor explosion, apart from fumes, are twofold. First is the pressure wave generated, which may shatter windows and shake buildings. Second is the danger from flying fragments of motor or test stand. The degree of these hazards depends on the type of test structure employed.

All the above problems apply, in a modified way, to the location of the test facility with respect to the main plant and shops.

The test facility may be in a test stand area isolated from the shops and main plant, in an area including both stands and shops, or in an area embracing all three. The preferred practice from an operational view is to have small-motor test stands, below 6,000 pounds thrust, within a few hundred feet of the shops and main plant, and any large-motor test stands perhaps a half-mile or so away, the latter with auxiliary work shop and office. The scope of the shop and office increases with increasing distance from the main plant, and problems of liaison can also increase to a formidable degree.

TEST STRUCTURES

While fixed test installations are now the rule, much early or pre-war rocket motor testing was done on small portable or semi-portable test stands. These consisted of propellant tanks, gas-pressure system, valves, and gages as well as the motor and a thrust-measuring means, all mounted on a frame. The unprotected type was transported to an open space in the country, staked down, and fired by pulling a lanyard. The operators and observers crouched behind convenient trees and rocks. The protected type was equipped with a shield of steel or wood behind which the operator manipulated individual hand-operated valves. The protection was more psychological than physical. It was with such stands as these, though, that many remarkable advances were made in the rocket art.

The modern test facility as a whole may consist of a number of fixed structures, for small apparatus, for cold-testing, and for large motors or power-plants. The area is usually surfaced

with concrete, gravel or asphalt, and each structure has a concrete apron. Propellant storage areas, well-separated from each other and from the test structures, are also surfaced.

The test stand structures have five elements:

1. The stand or stands proper, on which the motor or powerplant is mounted.
2. Protection (a) against weather, (b) for adjacent zones.
3. The propellant tank and pressure gas cells.
4. The control room.
5. The instrumentation room.

These elements may be separate or combined in various ways. Their arrangement and design is influenced, as we shall see, by the following factors: desired direction of jet; scale of motor or powerplant; visibility for observation; safety to operating personnel, other personnel and buildings and neighbors; working convenience; terrain; climate; and cost and permanence. Let us examine the arrangement and features of the elements of two composite hot-test arrangements. Fig. 3 shows the first. It is a small horizontal firing stand for development and production testing of motors and power plants up to, say, 6,000 pounds thrust. The actual firing angle of the stand proper is often made a few degrees below horizontal, to permit drainage of any propellants away from the injector. Propellants which seep into the injector, due, for example, to leaky valves, have a tendency to explode when the run is started.

Both working protection from weather and explosion protection for adjacent and distant zones are provided by the walls and roof which form the test cell or pit. The best wall construction is of five to fourteen-inch reinforced concrete. The roof can be of similar construction but unless it is also thick there are the possibilities of cracking in an explosion

and of pieces of concrete being dislodged and acting as missiles. A corrugated iron roof can be used but does not offer protection against flying fragments unless it is sandbagged on top. The pit should be shallow and wide for working space and ventilation. The motor should be fairly near the open end. It is obvious that the area outside the open end of the pit must be cleared for some distance, or better, the open end located three or four feet from water. The danger of someone's accidentally walking into the line of fire just as a run is begun must be eliminated. A land area should be fenced and should also be cleared of all underbrush or woods.

Propellant tank cells, in the arrangement shown on the left, are not used. The tanks are either located in opposite corners of the cell or are located on the test stand, as for testing assembled missile powerplants. Steel plates, either just forwards of the motor or by the tanks, serve to protect them.

The control room is shown located between two test pits. Side-observation is preferable to end-observation, since the fullest possible visibility of all the apparatus is desirable. As to the safest means for providing close-up visibility, the author has a very definite opinion, shown in Fig. 3. The opening (or openings) in the wall is equipped with nine-ply safety glass. A thick steel plate is firmly mounted to the pit-side of the wall at 45 degrees, with a mirror on its inside face. A second mirror, on a very light backing, is located above the first so as to give a view of the pit and the area outside it. The use of safety glass alone is not enough.

The instrumentation room in the layout shown is one with the control room. The instruments may be located either alongside the observation ports or at the end of the room. The former is perhaps more convenient for the operator. Access
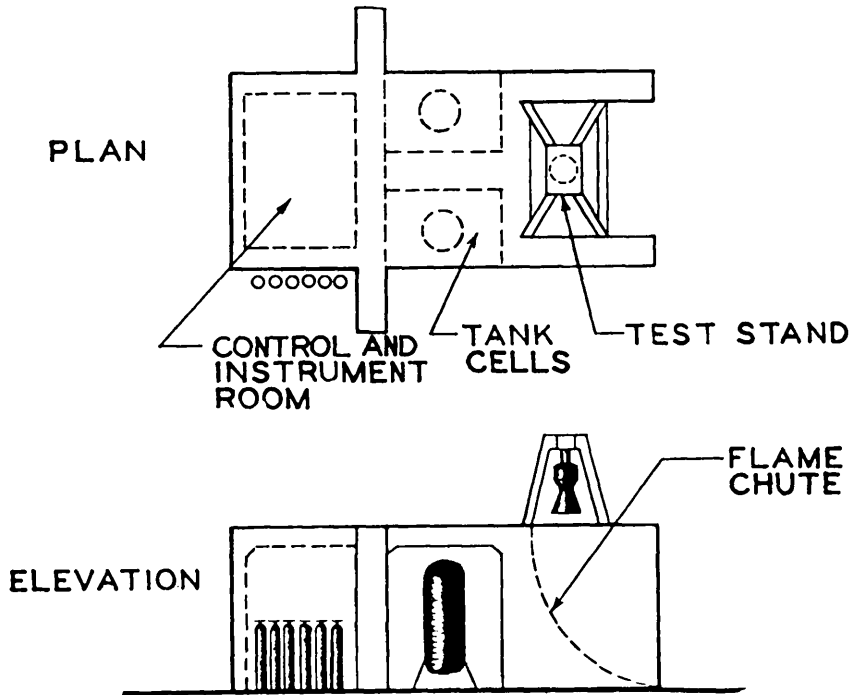
PLAN

CONTROL AND
INSTRUMENT
ROOM

TANK
CELLS

TEST STAND

FLAME
CHUTE

ELEVATION

Figure 3.   Typical arrangements of test
structure for small-motor test-
ing (0 to 6000 lb. thrust).

to the pit from the control room is desir-
able If the door opens to the rest of
the facility, a shielding wall is used.

Fig. 4 shows a larger scale, vertical-
firing test arrangement. Here the test
stand proper is a steel framework
securely anchored in reinforced con-
crete structure. Part of the structure is
a flume or flame deflector, usually
cooled. The rest comprises the open
propellant tank cells, gas bottle area,
and working surface. The motor may
have no weather protection or may be
covered by a rolling roof. Explosion
protection for neighboring areas is
achieved either by sheer distance or by
earthworks. The control room shown
is separate from the stand. It can be a
part of the stand if its construction is
sturdy and if it is placed so as to

avoid accidental flooding by propellants.
The instrument room is either a part of
the control room or is located very near
the stand and remotely operated. In
the latter case, some of the readings
are also indicated in the control room.

EQUIPMENT

The equipment, as was shown in Fig.
1, consists of tanks, piping gas pressure
source, regulators and valves, and elec-
trical and pneumatic control systems.
Emergency stop valves are usually in-
stalled near the tanks. Vent valves and
gas on-off valves are usually pneu-
matically actuated. The use of hand
valves in the test stand is kept to a
minimum.

TEST PITS

STANDS

TANK
CELLS

PLAN VIEW

CONTROL AND
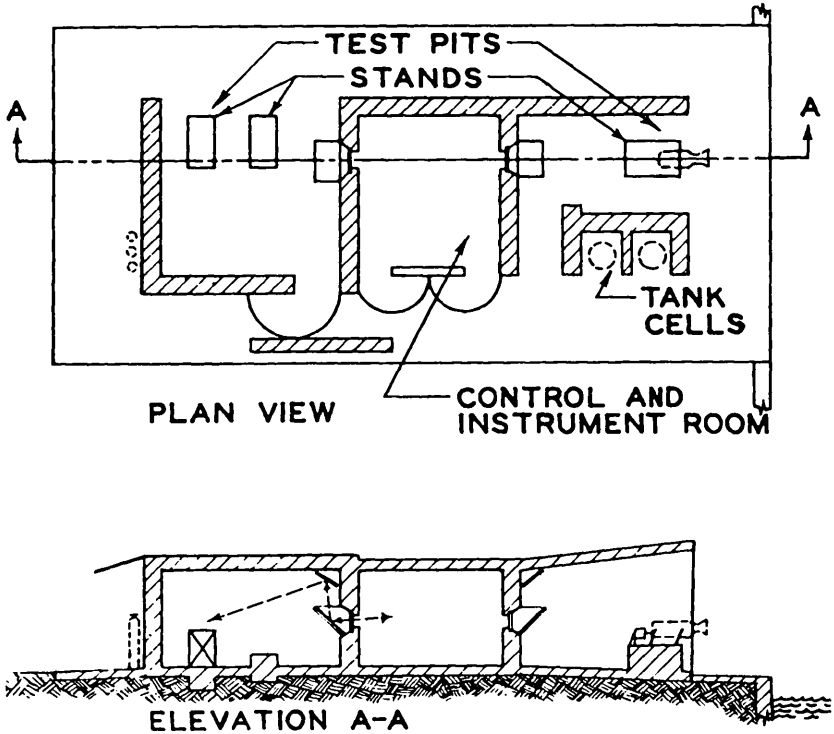INSTRUMENT ROOM

ELEVATION A-A

Figure 4. Arrangement of test structure
for large-motor testing.

The various elements of equipment are often developed by the test agency since commercial equipment frequently is not available for the required conditions.

Materials are important. Brass has a tendency to season-crack in the presence of acid. Steel becomes brittle in liquid oxygen. Hydrogen peroxide is susceptible to violent decomposition in the presence of copper, copper alloys and certain other materials. Only a few plastics are satisfactory in nitric acid.

In general, test equipment can take many forms and arrangements. Its development consumes an appreciable portion of total design and test effort.

Fire fighting apparatus, sprayer installations and test warning bells or horns are all essential safety equipment.

INSTRUMENTATION

Test stand instrumentation tends to fall into two categories — special and more or less routine. At the present time, "special instrumentation" might embrace such items as gas sampling arrangements or apparatus for measuring jet temperatures. Such instrumentation is developed and set up usually for research or basic development testing. Routine instrumentation includes equipment more or less commonly used in rocketry for measurement of pres-

sures, flow rates and thrust. The classification is a fluid one. Methods of flow rate measurement which were strictly special a few years ago are now routine. Perhaps special instrumentation can be defined as equipment which itself is more a subject of development than the apparatus to which it is being applied.

Of more general interest are the various forms of instruments used in routine work. Of first importance are the means used for recording data during a run. A run may last from perhaps three minutes down to two seconds, and usually its transient conditions are of interest. The reading and jotting down of indicated data is therefore of only limited or auxiliary usefulness, especially if the observer is nervous.

The four recording methods current today, and often used side-by-side, are (a) camera, (b) mechanical pen, (c) electromechanical pen, and (d) electrical or oscillographic. In the author's opinion, born of experience, camera recording of quantitative data is not particularly satisfactory and should be avoided wherever possible. The camera sticks, or runs down, floodlights burn out at critical moments, lens settings are wrong, the camera gets dropped, part of the gage board is cut off, and so on. In addition, the developing of the film takes time and is subject to mistakes. Finally, the reading of the film and plotting of data are tedious jobs and are subjec to error.

Mechanical pen chart recording is very satisfactory for pressure measurements when high fidelity response is not required, and when the lengths of gage line are over, say, 150 feet. The instruments properly used are remarkably accurate. Circular chart and strip chart types are available with synchronous motor drive stepped up in speed.

The circular type has the advantage of carrying 3 or 4 pens, and the circular charts are ready to file and can be planimetered with a special apparatus. For general recording the mechanical pen recorder is excellent.

Electromechanical pen type chart recorders are used when the measuring element puts out an electrical signal, as in thermocouples and high pressure flowmeters. The type is also useful if the instrument room is remote from the test stand.

The oscillographic type is used when accurate response to transients is required, as in starting-tests. It is sometimes also used to record general test data. The question of the proper types of pickups and circuits is controversial and best left to a specialist.

When direct pressure measurements of the propellants are taken, the instruments should either be encased or located somewhere other than in the control room, to avoid danger from failure of a connection or recording element. Alternatively, a snubber or hydraulic fuse can be used in the line.

Turning now to specific measurements, the gathering of data on thrust is nearly always important. Accurate thrust data requires first a method of mounting the motor or powerplant so it can move slightly but with minimum friction and binding. The common mounting methods are the rollergram, the parallelogram, and the pivoted beam. The suitability of each is a direct function of care in design, fabrication, and installation.

The second requirement for accuracy is the minimizing of forces due to piping reaction between the fixed test equipment and motor or power plant. Such forces occur as a result of rigidity in propellant piping, and of thermal contractions. One solution is to mount the

propellant tanks with the motor. Another is to employ flexibility high pressure metal hose. To expect rigid piping to hold a fixed "calibration" is usually unsatisfactory because its characteristics are affected when it is bumped or jarred or when connections are made and broken.

The third requirement for accuracy is a thrust sensing system which entails minimum motion, which is not subject to drift, and which has low hysteresis. Four general classes are used: (a) mechanical spring, (b) hydraulic, (e) pneumatic and (d) electrical.

The mechanical spring is inconvenient since it involves motion and since special apparatus is required to give remote reading. Hydraulic systems are fairly widely used. One common system for small motors employs a diaphragm or metal bellows as the thrust pickup or "jack", connected by a run of tubing to a receiving element. The element is a bourdon tube, helical coil, or spring loaded bellows mechanism of proper range actuating a mechanical pen. The system must be designed and installed in a way which permits all air to be bled out easily. Otherwise the system is spongy. Leaks must be eliminated; it is customary to pre-fill the system to a few psi so that any leakage shows up as a decrease in the "zero" reading. In cases where high accuracy is desired, and where the rest of the system has low hysteresis, a suppressed zero may be used.

A similar hydraulic system may employ a piston instead of a bellows type jack. Leakage past the piston is difficult to keep within limits. Balanced-piston or balanced-diaphragm types, however, are satisfactory. In these the travel is very small — less than 0.010 inches. A small motor driven pump is required.

One pneumatic system used is also of the balanced diaphragm type. Its response depends upon the distance between test stand and recording instrument. It does not require bleeding, of course.

Electrical jacks may employ strain gages on a bending beam or in a weak leg of the parallelogram mount. The electrical types entail problems common to such systems; they can be very good or very bad.

The fourth requirement for accuracy is proper thrust-calibration. Dead weights are very desirable for smaller stands. They may be applied directly or through a lever or linkage. In the latter case, care in design and use is essential. Dead weights on a floating piston plus linkage can be used. On larger stands, a Morehouse ring loaded by a hydraulic piston is convenient.

The measurement of propellant weight flow rate is of fundamental importance. There are two general approaches. One is to determine the total amount of each propellant (in pounds) consumed between beginning and end and dividing by effective time of firing (in seconds). The other is to measure the instantaneous rate of flow throughout the run.

Total consumption can be measured very satisfactorily by "sticking the tank," that is, noting the liquid levels in the calibrated tank by inserting a metal rod before and after a run and converting these levels to pounds. The method is difficult with liquid oxygen, but with care can be made very accurate with other propellants. Another method employs a sight gage on the tank. This method is good unless the gage valves are accidentally left open. With liquid oxygen, a modified form of this method, employing a second indicating liquid, has been used. The determination of time of firing is best done by planimetering the thrust-time curve for the run and calculating the "mean time" as the ratio of total impulse to operating thrust.

It is obvious that the above methods do not permit the gathering of data on a run which covers a range of mixture ratios, flow rates and thrusts. Hence research and development testing can be greatly speeded up and improved by the measurement of instantaneous flow rates. One early method employd continuously weighed tanks, the tanks resting on diaphragms connected to manometer tubes. Continuous reading of sight-glass levels has also been used. The first method involves problems of line and fluid reaction; the second involves the use of high pressure sight glasses and a camera, and there is a time lag in the readings.

Flow rate meters have proven satisfactory. One is the variable area type or rotameter, either indicating or recording. For flow rates up to 5 or 10 lb./sec. the rotameter may be used direct. For high flow rates the rotameter is connected as a bypass across a calibrated orifice.

Measurement of pressure drop across calibrated orifices is also useful. The chief problem is accurate measurement of low differentials at high pressure. There are numerous other possible and attractive methods of measuring flow rate. It can be stated almost categorically, however, that any method for measuring flow rates of such materials as nitric acid and liquid oxygen at 400 to 1000 psi requires considerable experimentation before its peculiarities are worked out. The problem of calibration alone is a difficult one when flow rates of 50 to 150 pounds per second are involved.

METHODS AND RESULTS OF TEST

Since methods and results of test vary considerably with the purpose and equipment, only a few general comments can be made. Of first importance is a carefully worked out and rigorously enforced safety procedure. It should apply to handling of propellants and to the conduct of tests.

Second is the following of a definite routine in checking out and testing. The test crew should have clearly defined individual assignments. It is also vitally important to have a highly competent test crew. It is depressing to burn out a motor because someone failed to tighten a chamber pressure tap.

Third is the use of data sheets which have labelled spaces for all data and observations of interest. Usually it is best to have a special form for each test stand and frequently for each test series, since the apparatus, equipment and methods may be widely different in each case.

Among the more important general desired results of testing, the first is reliable data. The second is inspired observation and deduction. A seemingly small happening can be the clue to new understanding of a problem. In this regard, close collaboration between the designer and the tester is essential. It is also the author's opinion that a man should have considerable experience on the test stand before he undertakes design. An ideal arrangement, perhaps possible only in smaller projects, is for the designer to be his own test engineer.

CONCLUSION

The present paper has attempted to cover one part of one aspect of rocket development, and that in almost outline form. The design of test structures, the subjects of thrust and flow rate measurement, the standardization of methods for propellant testing and numerous other important aspects require more detailed treatment. They also require a wider exchange of information within security limits. Exchange is desirable not only between rocket agencies but between these agencies and other scientific and engineering groups which possess backgrounds of experience on similar problems. A committee to undertake such liaison can be of great service.

Another observation is that the field of rocket powerplant design and testing should not be entered into lightly. This is said not because of the physical requirements in facilities, equipment, and so forth, but because of the extent of testing involved in even a seemingly straightforward motor development task. Rocket work is and will remain for some time as much an empirical art as a science. Ignorance of this fact can result in considerable cost in money and self-assurance before the fact is learned the hard way. A corollary word of caution should be uttered regarding construction of extensive test facilities. If design of such facilities is not based on a fair background of experience they are liable to be more hindrance than help.

Regarding future trends, the several unique features of the liquid rocket powerplant are not generally recognized. Such recognition was long overdue in this country. It can be expected that we will not delay in exploiting these features.

In its initial evolution and in its initial applications, the liquid rocket powerplant has proved its worth. Testing has been of great importance in these accomplishments. It must not be assumed that the testing phase is complete. Our goals — larger thrusts, lighter weight, increased dependability, greater flexibility and higher efficiency— will be attained primarily through extensive testing.