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RESEARCH MEMORANDUM

ROCKET THRUST VARIATION WITH FOAMED LIQUID PROPELLANTS

By G. Morrell

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS
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SUMMARY

Flow theory of liquid foams was applied to the rocket-engine injection process. By foaming the propellants and thereby changing their bulk densities it is possible, in theory, to vary rocket thrust continuously. An analysis of the method, assuming constant orifice flow coefficients, is presented and discussed.

Data from preliminary experiments in a 1000-pound-thrust ammonia - nitric acid rocket engine agreed only qualitatively with theory; two to six times the theoretical gas-flow rate was required in the experiments. It was demonstrated, however, that the "foam-flow" method of thrust variation is feasible.

INTRODUCTION

Continuous thrust variation in rocket engines (throttling) is especially desirable for piloted aircraft applications. In guided missiles the use of a single powerplant for both boost and sustainer operations would be possible; vernier thrust control near cut-off is another possible field of application. Currently, rocket-thrust variation is accomplished by using multiple fixed-thrust cylinders.

Mechanical methods for varying thrust by injection orifice control are described in references 1 and 2. This report describes a method for controlling thrust by varying propellant density. Continuous density variation is achieved by foaming the propellants with an inert, insoluble gas. To maintain high propulsive efficiency, the exhaust-nozzle area must also be varied. This complementary problem of exhaust-nozzle-area control is treated in reference 3.

Foam-flow theory is discussed in reference 4, and has been applied to the hydroduct, an underwater propulsion device, as reported in references 5 and 6. Reference 6 also reports experimental data which are in substantial agreement with theory.

In this report the foam-flow theory is applied to the injector orifice of a rocket engine to calculate the reduction in the propellant flow as a function of the ratio of the flow rates of the inert gas and the propellant. Other parameters involved are the initial injection pressure ratio, the gas specific heat ratio, and a term including the fluid properties.

Several preliminary experiments were conducted in a nominal 1000-pound-thrust ammonia - nitric acid rocket to measure the reduction in liquid flow over a small range of gas flows. The deviation of the results from calculated values is discussed as well as the feasibility of this method of thrust control.

SYMBOLS

The following symbols are used in this report:

A cross-sectional area

C_D orifice discharge coefficient

g conversion constant

p pressure

R gas constant

T absolute temperature

u velocity

V volume

W weight

w weight-flow rate

α ratio of injection pressure to chamber pressure for full flow, $p_l/p_{c,o}$, dimensionless (see subscript list)

β $RT\rho_l/p_{c,o}$, dimensionless

γ ratio of specific heats, dimensionless

δ ratio of part-thrust flow rate to full-thrust flow rate, $w_l/w_{l,o}$, dimensionless

ε ratio of gas-flow rate to liquid-flow rate, w_g/w_l, dimensionless

ρ propellant density

τ ratio of part-thrust burning time to total burning time, dimensionless

ω ratio of discharge coefficients, $C_{D,l}/C_{D,f}$, dimensionless

Subscripts:

c combustion chamber

f foam

g gas

l liquid

o full-thrust condition

th theoretical

1 upstream of injection orifice

2 injection orifice discharge station

ANALYSIS

In the following analysis, expressions are derived for the reduction in liquid flow as a function of the gas to liquid ratio ε for several values of α and β and for a value of γ of 1.67. Two flow cases are treated: compressible, isothermal gas flow, assuming thermal equilibrium between the gas and the liquid and essentially constant internal energy; and compressible adiabatic flow, assuming no interchange of energy between the gas and the liquid. In both cases, any heat transfer between the fluid and the surroundings is ignored.

Derivation of Flow Equations

The foam-flow equations are derived on the basis that the gas is uniformly dispersed and is insoluble in the liquid and that the flow is one-dimensional. The continuity equation is

$$w_g + w_l = w_l(1 + \varepsilon) = \rho A u \qquad (1)$$

Gas and liquid volumes may be considered additive, so that

$$V = V_g + V_l$$

or since

$$V = \frac{W}{\rho}$$

$$\frac{W_g + W_l}{\rho} = \frac{W_g}{\rho_g} + \frac{W_l}{\rho_l}$$

Substituting the gas-liquid ratio

$$\varepsilon = \frac{w_g}{w_l} = \frac{W_g}{W_l}$$

gives the following equation for foam density:

$$\rho = \frac{\varepsilon + 1}{\frac{\varepsilon}{\rho_g} + \frac{1}{\rho_l}} \tag{2}$$

The simplified Euler equation for one-dimensional compressible-flow is written as

$$\frac{u\,du}{g} + \frac{dp}{\rho} = 0 \tag{3}$$

In solving equation (3), the following assumptions were made: (1) flow velocity at station 1 is negligible compared with that at station 2; (2) at station 2 the pressure is equal to the combustion-chamber static pressure, that is, $p_2 = p_c$; (3) the characteristic velocity based on liquid flow ($p_c A_t g/w_l$, where A_t is nozzle-throat area) remains constant, or $p_c \propto w_l$; (4) the liquid flow is isothermal; (5) the gas behaves ideally, that is, $\rho_g = p/RT$; and (6) the propellant mixture ratio remains constant.

Integration of equation (3) for the case of isothermal gas flow after substituting equation (2) yields

$$u_2 = \sqrt{\frac{2g}{1 + \varepsilon}\left(\varepsilon RT \ln \frac{p_1}{p_c} + \frac{p_1 - p_c}{\rho_l}\right)} \tag{4}$$

and substitution in equation (1) gives

$$\frac{w_l}{A_2} = \frac{C_{D,f}\rho_2}{1+\varepsilon}\sqrt{\frac{2g}{1+\varepsilon}\left(\varepsilon RT \ln \frac{p_1}{p_c} + \frac{p_1 - p_c}{\rho_l}\right)}$$

4258

$$= \frac{C_{D,f}\sqrt{\frac{2g}{1+\varepsilon}\left(\varepsilon RT \ln \frac{p_1}{p_c} + \frac{p_1 - p_c}{\rho_l}\right)}}{\frac{\varepsilon RT}{p_c} + \frac{1}{\rho_l}} \tag{5}$$

where the discharge coefficient has been introduced. For no gas flow, $\varepsilon = 0$, $w_l = w_{l,o}$, $p_c = p_{c,o}$, and

$$\frac{w_{l,o}}{A_2} = C_{D,l}\sqrt{2g\,\rho_l(p_1 - p_{c,o})} \tag{6}$$

Dividing equation (5) by (6) gives the liquid-flow ratio,

$$\frac{w_l}{w_{l,o}} = \frac{\sqrt{\frac{1}{1+\varepsilon}\left[\frac{\varepsilon RT\rho_l \ln p_1/p_c + (p_1 - p_c)}{p_1 - p_{c,o}}\right]}}{\omega\left(\frac{\varepsilon RT\rho_l}{p_c} + 1\right)} \tag{7}$$

By letting $p_1/p_{c,o} = \alpha$, and $RT\rho_l/p_{c,o} = \beta$, and since $\delta \equiv w_l/w_{l,o} = p_c/p_{c,o}$, equation (7) may be reduced to

$$\delta = \frac{\sqrt{\frac{1}{1+\varepsilon}\left(\frac{\varepsilon\beta}{\alpha - 1}\ln\frac{\alpha}{\delta} + \frac{\alpha - \delta}{\alpha - 1}\right)}}{\omega\left(\frac{\varepsilon\beta}{\delta} + 1\right)}$$

or

$$\varepsilon^2\beta^2 + 2\varepsilon\beta\delta + \delta^2 = \frac{\varepsilon\beta \ln\frac{\alpha}{\delta} + \alpha - \delta}{\omega^2(1+\varepsilon)(\alpha - 1)} \tag{8}$$

4258

If on the other hand the gas-phase flow through the injector is adiabatic, the gas density derived from the ideal gas law and the adiabatic equation of state is

$$\rho_g = \frac{p^{1/\gamma} p_1^{(\gamma-1)/\gamma}}{RT_1}$$

which upon substitution with equation (2) in equation (3) followed by integration yields

$$u_2 = \sqrt{\frac{2g}{1 + \varepsilon}\left[\frac{\varepsilon\gamma}{\gamma - 1}\,\frac{RT_1}{p_1^{\frac{\gamma-1}{\gamma}}}\left(p_1^{\frac{\gamma-1}{\gamma}} - p_c^{\frac{\gamma-1}{\gamma}}\right) + \frac{p_1 - p_c}{\rho_l}\right]} \tag{9}$$

Following the same procedure as above and substituting equation (9) in equation (1) gives the liquid flow per unit area

$$\frac{w_l}{A_2} = \frac{C_{D,f}\sqrt{\frac{2g}{1 + \varepsilon}\left[\frac{\varepsilon\gamma}{\gamma - 1}\,\frac{RT_1}{p_1^{\frac{\gamma-1}{\gamma}}}\left(p_1^{\frac{\gamma-1}{\gamma}} - p_c^{\frac{\gamma-1}{\gamma}}\right) + \frac{p_1 - p_c}{\rho_l}\right]}}{\frac{\varepsilon RT_1}{p_c^{\frac{1}{\gamma}}\, p_1^{\frac{\gamma-1}{\gamma}}} + \frac{1}{\rho_l}} \tag{10}$$

Division of equation (10) by $w_{l,o}/A_2$ from equation (6) yields the liquid ratio

$$\frac{w_l}{w_{l,o}} = \frac{\sqrt{\frac{\rho_l^2}{1 + \varepsilon}\left\{\frac{\varepsilon\gamma}{\gamma - 1}\,\frac{RT_1}{p_1^{\frac{\gamma-1}{\gamma}}}\left[\frac{p_1^{\frac{\gamma-1}{\gamma}} - p_c^{\frac{\gamma-1}{\gamma}}}{\rho_l(p_1 - p_{c,o})}\right] + \frac{p_1 - p_c}{\rho_l^2(p_1 - p_{c,o})}\right\}}}{\omega\left(\frac{\varepsilon RT_1 \rho_l}{p_c^{\frac{1}{\gamma}}\, p_1^{\frac{\gamma-1}{\gamma}}} + 1\right)} \tag{11}$$

Again, by letting $\delta \equiv w_l/w_{l,o} = p_c/p_{c,o}$, $\alpha = p_1/p_{c,o}$, and $\beta = RT_1\rho_l/p_{c,o}$, equation (11) becomes

4258

$$\delta = \frac{\sqrt{\frac{1}{1+\varepsilon}\left[\frac{\varepsilon\gamma}{\gamma-1}\,\frac{\beta}{\alpha^{\frac{\gamma-1}{\gamma}}}\left(\frac{\alpha^{\frac{\gamma-1}{\gamma}} - \delta^{\frac{\gamma-1}{\gamma}}}{\alpha-1}\right) + \frac{\alpha-\delta}{\alpha-1}\right]}}{\omega\left(\frac{\varepsilon\beta}{\delta^{\frac{1}{\gamma}}\alpha^{\frac{\gamma-1}{\gamma}}} + 1\right)}$$

which reduces to

$$\beta^2\left(\frac{\delta}{\alpha}\right)^{2\left(\frac{\gamma-1}{\gamma}\right)}(\varepsilon^2) + \frac{2\beta}{\alpha^{\left(\frac{\gamma-1}{\gamma}\right)}}\delta^{\frac{2\gamma-1}{\gamma}}\varepsilon + \delta^2 = \frac{1}{\omega^2(1+\varepsilon)}\left[\frac{\varepsilon\gamma\beta}{(\gamma-1)\alpha^{\frac{\gamma-1}{\gamma}}}\left(\frac{\alpha^{\frac{\gamma-1}{\gamma}} - \delta^{\frac{\gamma-1}{\gamma}}}{\alpha-1}\right) + \left(\frac{\alpha-\delta}{\alpha-1}\right)\right] \tag{12}$$

Equations (8) and (12) were solved on a card-programmed calculator for several values of α and β, for a single value of γ (helium gas assumed; γ, 1.67), and for a constant discharge coefficient, that is, when ω equals 1. The last condition was assumed in order to simplify the calculation. There is no experimental evidence either to support or negate this assumption. Figures 1 and 2 show the relations between the liquid ratio δ and the gas-to-liquid ratio ε for compressible, isothermal flow and compressible, adiabatic flow, respectively. The dependence of ε on α is shown in figure 3, and the dependence of ε on β is shown in figure 4 both for a δ value of 0.5.

Discussion

In terms of gas requirements, adiabatic gas flow is less economical than isothermal flow. From the nature of the flow process, especially the short stay time in the injector, it appears probable that the amount of energy interchange between liquid and gas will be small, and the gas-phase flow will be more nearly adiabatic than isothermal.

For the α range normally encountered in rocket practice (1.1 to 1.3), injection pressure has a large effect on gas economy. From figure 3 it appears that engines using the foam-flow technique should be designed with injection pressure ratios in the range of 1.5 to 2.0, if possible.

The parameter β also has a large influence on gas economy; the larger the value of β, the lower is the value of ε for a given value of δ, as shown in figure 4. In effect, this means that for a given gas and chamber pressure, foam-flow thrust variation is most practical for high-density propellants. In the following table are listed values of β for several propellants for the case where R is 386 foot pounds per pound per $^{\circ}R$ (helium gas) and $p_{c,o}$ is 450 pounds per square inch:

4258

Propellant	T_l, $^{\circ}R$	ρ_l lb/cu ft	β
Hydrogen	37	4.4	0.97
Oxygen	167	71	70.7
Ammonia	460	41.3	113.0
Jet fuel	530	47	150.7
Nitric acid	530	93	290.1

It is apparent immediately that fuel - nitric acid systems are best suited to the foam-flow method of thrust variation, and that the method would be impractical for hydrogen systems.

To illustrate the application of the analysis, the following example is presented. The model chosen is a rocket missile using jet fuel and nitric acid at a mixture ratio of 4. Full-thrust conditions were assumed to be as follows: thrust, 20,000 pounds; net specific impulse, 220 pounds per second per pound; chamber pressure, 500 pounds per square inch; injection pressure, 650 pounds per square inch; and injection temperature, 530° R. Powered flight duration was held constant at 2 minutes, and helium gas was used, which was stored at 3000 pounds per square inch and 530° R. For maximum total impulse the volume of propellants required is 140 cubic feet. Helium density at storage conditions, 1.93 pounds per cubic foot, was calculated from the data of reference 7. The average density of the propellants is 78 pounds per cubic foot, equivalent to a β value of 222. When compressible, adiabatic flow is assumed, the relation of δ and ε from figure 2 is

δ	0.75	0.5	0.25
ε	0.0085	0.0235	0.065

These data and assumptions were used to calculate the required gas and liquid volumes as a function of the fraction of total powered flight time for one-quarter, one-half, and three-quarters thrust operation. The results of the calculations are shown in figure 5. To provide half thrust for half the flight duration would require a gas volume equal to about 30 percent of the liquid volume. Total volume would be only slightly less than that for full-thrust, full-duration flight. Since high-pressure gas tankage will be heavier than liquid tankage on a unit volume basis, it is clear that the foam-flow technique for thrust variation imposes a weight penalty on the vehicle structure, even though the gross vehicle weight is less than that for full-thrust flight.

How this weight penalty compares with the added weight due to multiple thrust cylinders and the gear for varying injection orifice area will depend on the flight program and vehicle design and cannot be stated in generalized terms. Qualitatively, however, it is apparent from figure 5 that the foam-flow technique is most advantageous for smaller values of τ and larger values of δ. For large values of δ, precise control of gas flow may be a problem.

EXPERIMENTAL APPARATUS

A short series of experiments were run in a nominal 1000-pound-thrust water-cooled rocket to check the analysis. The propellant system was ammonia - white fuming nitric acid (WFNA) with helium used as the pressurizing and foaming gas. Each propellant was foamed separately before entering the injection manifold. Flow-line addition of lithium metal to the ammonia caused spontaneous ignition in the combustor. A schematic drawing of the flow system is shown in figure 6.

The thrust cylinder had an inside diameter of 4 inches and an over-all length of $14\frac{3}{8}$ inches. The cylindrical portion of the combustor was $8\frac{5}{8}$ inches long; nozzle-throat diameter, $1\frac{13}{16}$ inches; and nozzle-exit diameter, $3\frac{3}{8}$ inches. The characteristic length of the rocket was 47 inches.

The doublet-type injector consisted of 24 pairs of fuel and oxidant orifices arranged in a circle with a mean diameter of 2.5 inches. The fuel-orifice diameter was 0.0515 inch, and the oxidant-orifice diameter, 0.072 inch.

Gas-Injection Device

Several gas-injection devices were qualitatively tested with water in a transparent tube to check the degree of gas dispersion and flow stability. The final design, shown in the insert of figure 6, consisted of a 2-inch length of 5/16-diameter tubing with 220 holes, each with a 0.0135-inch diameter, arranged in 11 circles of 20 holes each. The tube was fitted into a T so that the tube axis coincided with the through axis of the T.

Calibration of the gas-injection device with water as the liquid in the propulsion system indicated that flow was free from surges when the gas-injection pressure was not more than 100 pounds per square inch greater than the liquid-injection pressure.

Instrumentation

Flow rates of the propellants were measured by rotating-vane-type meters with an error of about 1 percent. Gas-flow rates were measured by orifices with an error of about 2 percent. Pressures were measured with strain-gage-type transducers having an error of about 2 percent. Thrust was measured with a strain-gage load cell having an accuracy of about 1 percent. Temperatures were measured with chromel-alumel thermocouples.

Operating Procedure

The engine was started and operated at full-flow conditions for about 10 to 15 seconds, then helium was admitted and the engine operated at part thrust for an additional 10 to 15 seconds. Operation was finally shifted to full-flow conditions for approximately 3 to 5 seconds prior to shut down in order to check the initial data.

RESULTS AND DISCUSSION

Data for the four experiments performed are listed in table I. Table II shows the values of the analytic parameters and a comparison of the experimental and theoretical gas-liquid ratios. The assumption of constant characteristic velocity (chamber pressure proportional to flow) was nearly fulfilled in the experiments as shown in table I. The largest deviation occurred in the second run. The assumption of a constant mixture ratio was not fulfilled in the first two runs, (i.e., δ for the fuel and the oxidant were not the same (table II)), which may account for some of the variation in the characteristic velocity.

Comparison of Gas-Liquid Ratios

Although the experiments definitely show that substantial thrust variation can be obtained by foaming the propellants to alter their bulk density, the experimental gas-liquid ratios were about twice the theoretical values for the fuel and three to six times the theoretical values for the oxidant.

There are a number of probable reasons for the discrepancy between the experimental and predicted results: (1) the analysis assumed the same discharge coefficient for foam and liquid, and the experiments departed seriously from this condition, as is shown in table III (discharge coefficients were calculated by eqs. (6) and (10)); (2) the gas-injection device may not have produced a uniform gas dispersion since the design was selected on the basis of gross visual observations; and (3) the injector manifold design tended to produce a centrifugal component of velocity in the fluid which may have caused gas-liquid separations. The anomaly of discharge coefficients greater than unity may have been caused by experimental error.

More research is needed to improve the foam-flow method of rocket-thrust variation. Studies are needed on the methods for producing uniform, stable foams and for the design of flow passages which will minimize gas-liquid separation. The flow of foams through orifices should also be studied to establish discharge coefficients as a function of the flow conditions and the fluid properties. Since one of the propellants will be used as a coolant, studies should be conducted on the heat-transfer characteristics of foamed liquids. It might be expected that for moderate gas-liquid ratios, the heat-transfer characteristics of foams will approach those for liquids under nucleate boiling conditions. It was found in water calibrations of the flow systems that large pressure differences between liquid and gas produced intermittent flow of liquid. Flow of this type would certainly result in low-frequency combustion oscillations in a rocket engine. When applying the foam-flow technique to a powerplant, therefore, special attention will be required in the design of the flow system to avoid this type of flow.

Operating Characteristics

The transition from full-thrust to part-thrust and back to full-thrust was free from surge in all runs. Since the pressure difference between liquid and gas was maintained at a low value, the combustion was also free from oscillations.

CONCLUDING REMARKS

This work has demonstrated that it is technically feasible to vary rocket-engine thrust by the foam-flow method. Research on several problems is needed in order to improve the efficiency of the process, particularly in view of the fact that two to six times the theoretical gas-flow rates were required in the few experiments performed. Further than this, detailed analyses are required to establish the relative merits of foam-flow thrust variation against mechanical techniques such as orifice-area variation or pump-speed variation. It is quite probable that no one method will be best, but rather that the application and propellants chosen will determine the optimum method.

SUMMARY OF RESULTS

An analysis is presented on a method for continuously varying rocket thrust by foaming the propellants to change their bulk density. For a given thrust ratio, the gas-liquid ratio depends on the full-thrust-injector pressure ratio, the molecular weight and temperature of the gas, the liquid density, the normal combustion pressure, and the gas expansion process.

From a series of four experiments in a nominal 1000-pound-thrust 4258
rocket using ammonia - nitric acid propellants with helium as the foaming gas, the following results were obtained:

1. The technical feasibility of the foam-flow method for thrust variation was shown.

2. The experimental gas-liquid ratios were two to six times the predicted values.

3. The characteristic velocity for part-thrust operation was within 10 percent of that for full-thrust operation, indicating that foaming of the propellants does not impair combustion efficiency.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, November 27, 1956

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4258

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TABLE I. - EXPERIMENTAL DATA ON ROCKET THRUST VARIATION

Run	Thrust, lb	Combustion pressure, lb/sq in.	Oxidant flow, lb/sec	Fuel flow, lb/sec	Characteristic velocity, ft/sec	Oxidant-helium flow, lb/sec	Fuel-helium flow, lb/sec	Fuel temperature, °F	Oxidant temperature, °F	Oxidant-helium temperature, °F	Fuel-helium temperature, °F	Oxidant-injection pressure, lb/sq in.	Fuel-injection pressure, lb/sq in.	Fuel-helium pressure, lb/sq in.	Oxidant-helium pressure, lb/sq in.
1	760	241	3.42	2.20	3500	------	------	34	86	---	---	350	550	---	---
1a[a]	340	144	2.46	1.21	3200	0.024	0.021	34	86	45	48	350	550	550	350
2	860	251	3.17	2.21	3810	------	------	45	52	---	---	350	550	---	---
2a	530	157	2.63	1.09	3440	.032	.026	45	52	36	42	350	550	550	350
3	815	240	3.08	2.35	3610	------	------	40	52	---	---	350	550	---	---
3a	390	129	1.71	1.33	3460	.059	.021	40	52	30	36	350	550	550	400
4	890	255	3.76	2.25	3460	------	------	18	52	---	---	400	550	---	---
4a	425	136	1.94	1.34	3380	.071	.023	18	52	28	32	400	550	550	450

[a]The letter a indicates part-thrust portion of run.

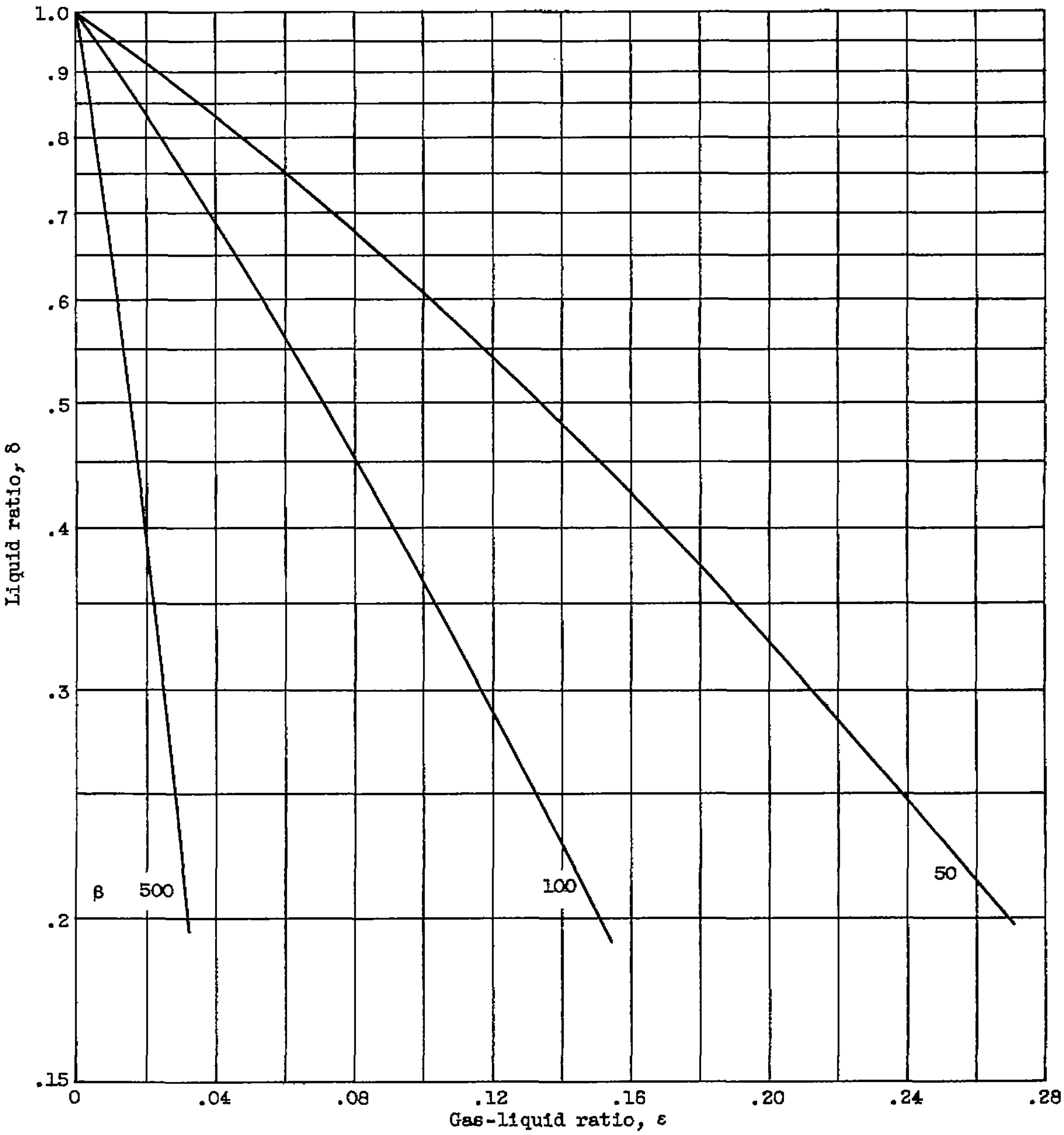
4258

TABLE II. - COMPARISON OF EXPERIMENTAL AND THEORETICAL GAS-LIQUID RATIOS

Run	δ	$p_c/p_{c,o}$	ε	α	β	Theoretical gas-liquid ratio for compressible, adiabatic flow	$\varepsilon/\varepsilon_{th}$
Oxidant system							
1	0.720	0.598	0.00983	1.45	518	0.0033	2.98
2	.830	.625	.0122	1.39	500	.0019	6.42
3	.555	.537	.0344	1.56	515	.0057	6.04
4	.516	.534	.0367	1.66	484	.0065	5.65
Fuel system							
1	0.550	0.598	0.0174	2.28	226	0.0097	1.79
2	.494	.625	.0241	2.19	211	.0131	1.84
3	.565	.537	.0161	2.29	219	.0094	1.72
4	.595	.534	.0170	2.16	210	.0092	1.85

TABLE III. - COMPARISON OF INJECTOR DISCHARGE COEFFICIENTS

Run	Full-thrust coefficient, $C_{D,l}$	Part-thrust coefficient, $C_{D,f}$	Coefficient ratio, ω
Oxidant system			
1	0.507	0.910	0.558
2	.489	1.02	.479
3	.451	1.08	.418
4	.479	1.1	.435
Fuel system			
1	0.590	0.886	0.666
2	.611	.875	.699
3	.637	.975	.653
4	.617	.968	.638



(a) Ratio of injection pressure to chamber pressure for full flow, 1.1.

Figure 1. - Variation of liquid ratio for isothermal gas flow.

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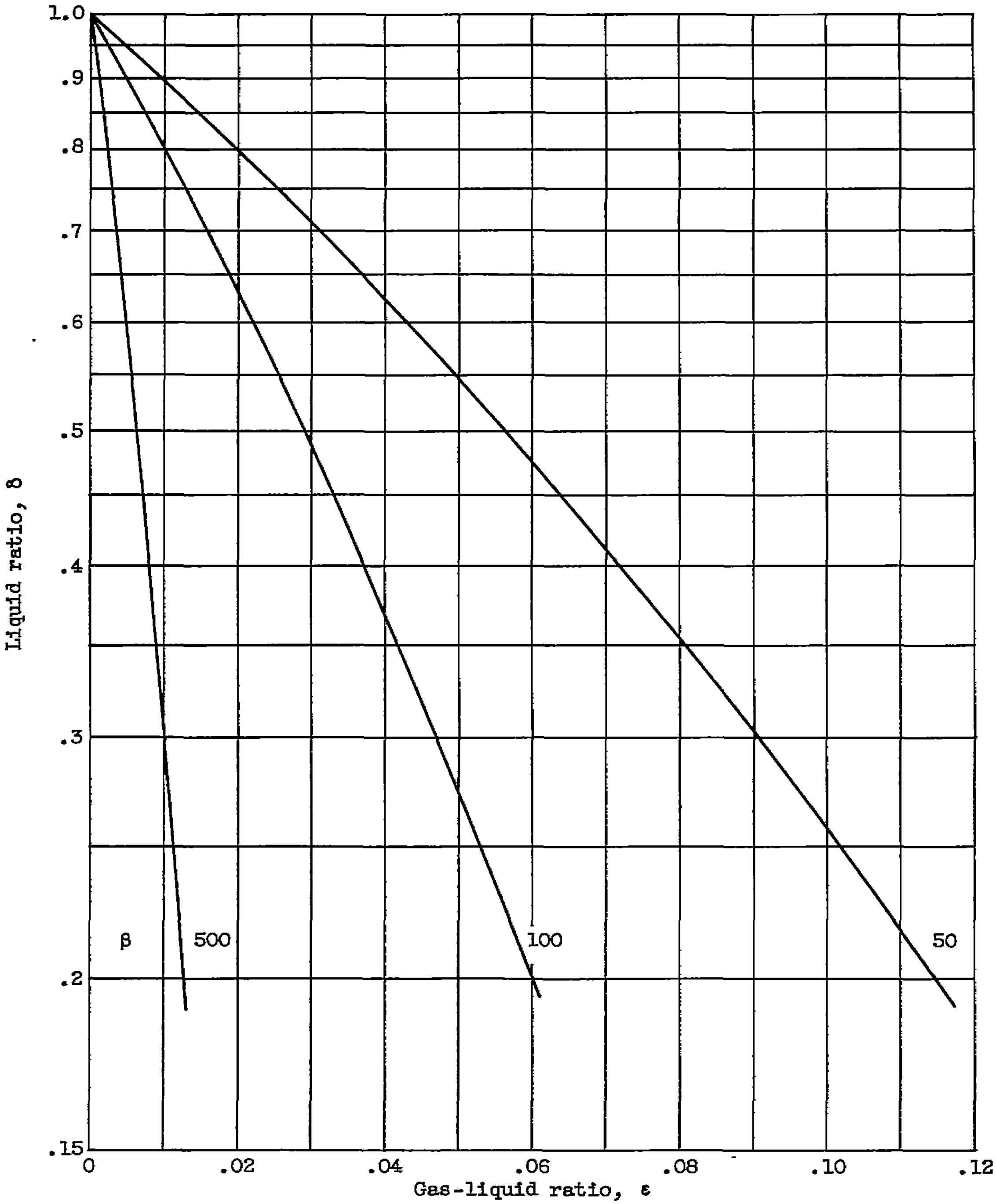
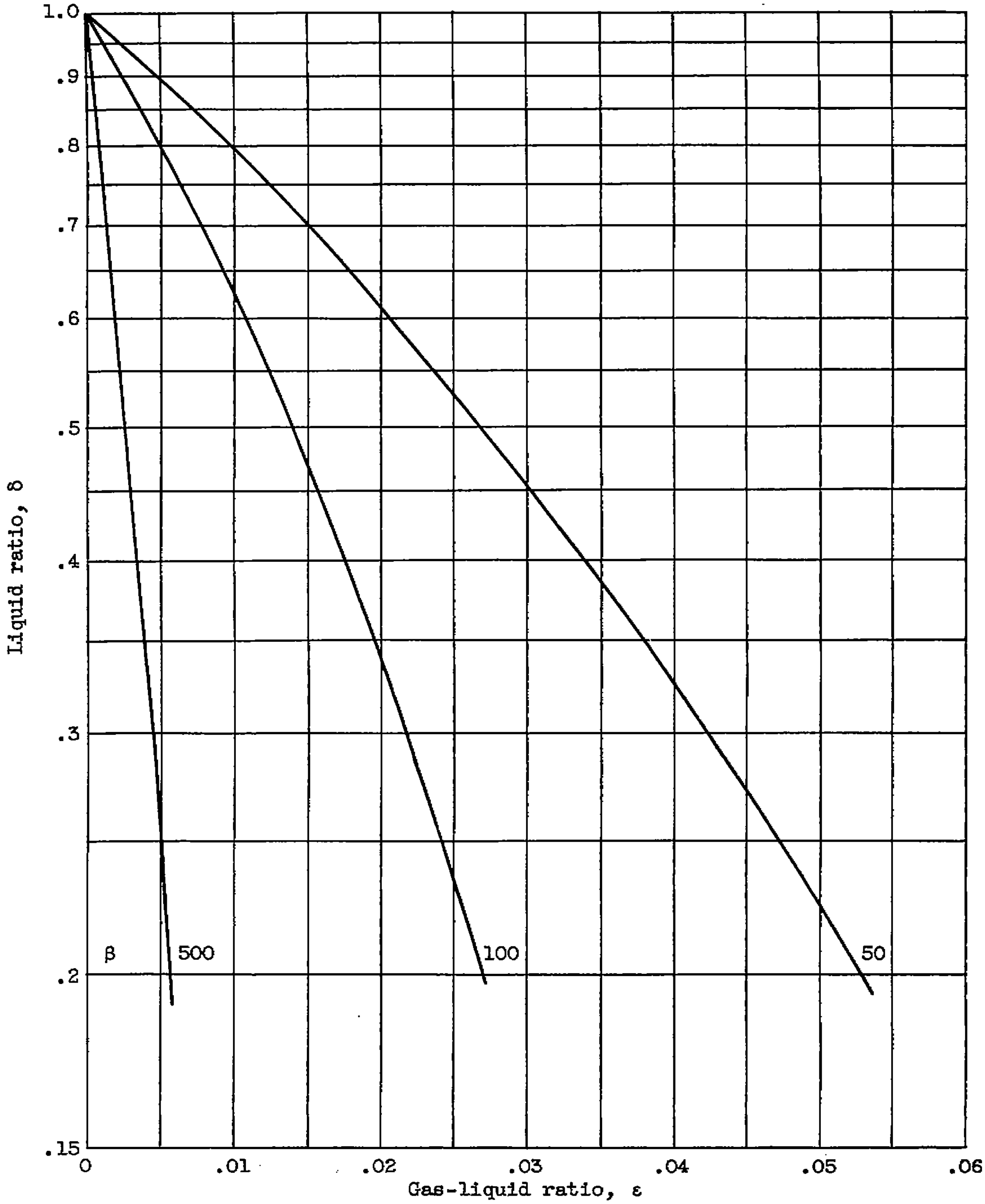


(b) Ratio of injection pressure to chamber pressure for full flow, 1.3.

Figure 1. - Continued. Variation of liquid ratio for isothermal gas flow.

4258



(c) Ratio of injection pressure to chamber pressure for full flow, 1.9.

Figure 1. - Concluded. Variation of liquid ratio for isothermal gas flow.

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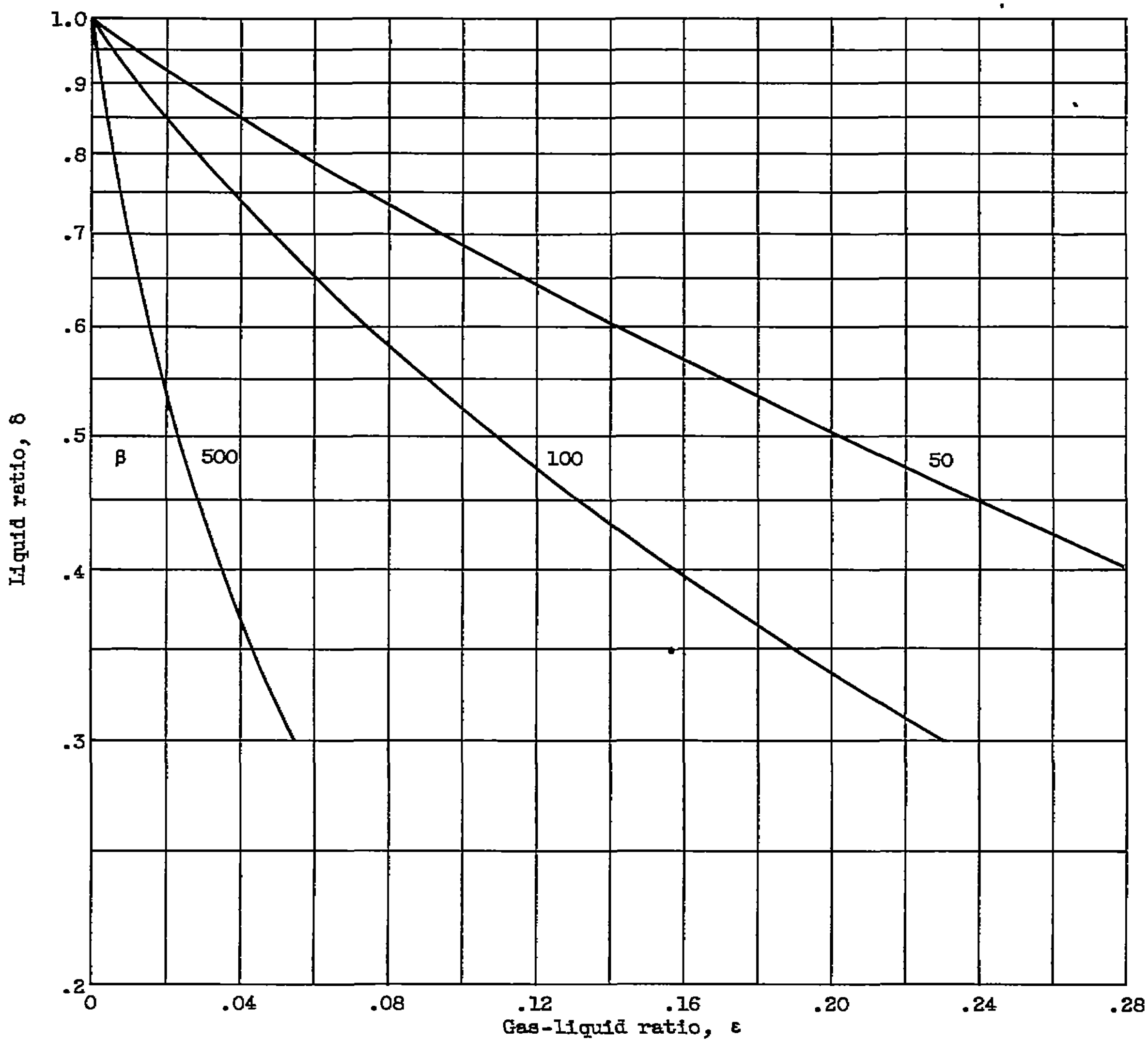
CN-3 back



(a) Ratio of injection pressure to chamber pressure for full flow, 1.1.

Figure 2. - Variation of liquid ratio for adiabatic gas flow with a specific heat ratio of 1.67.

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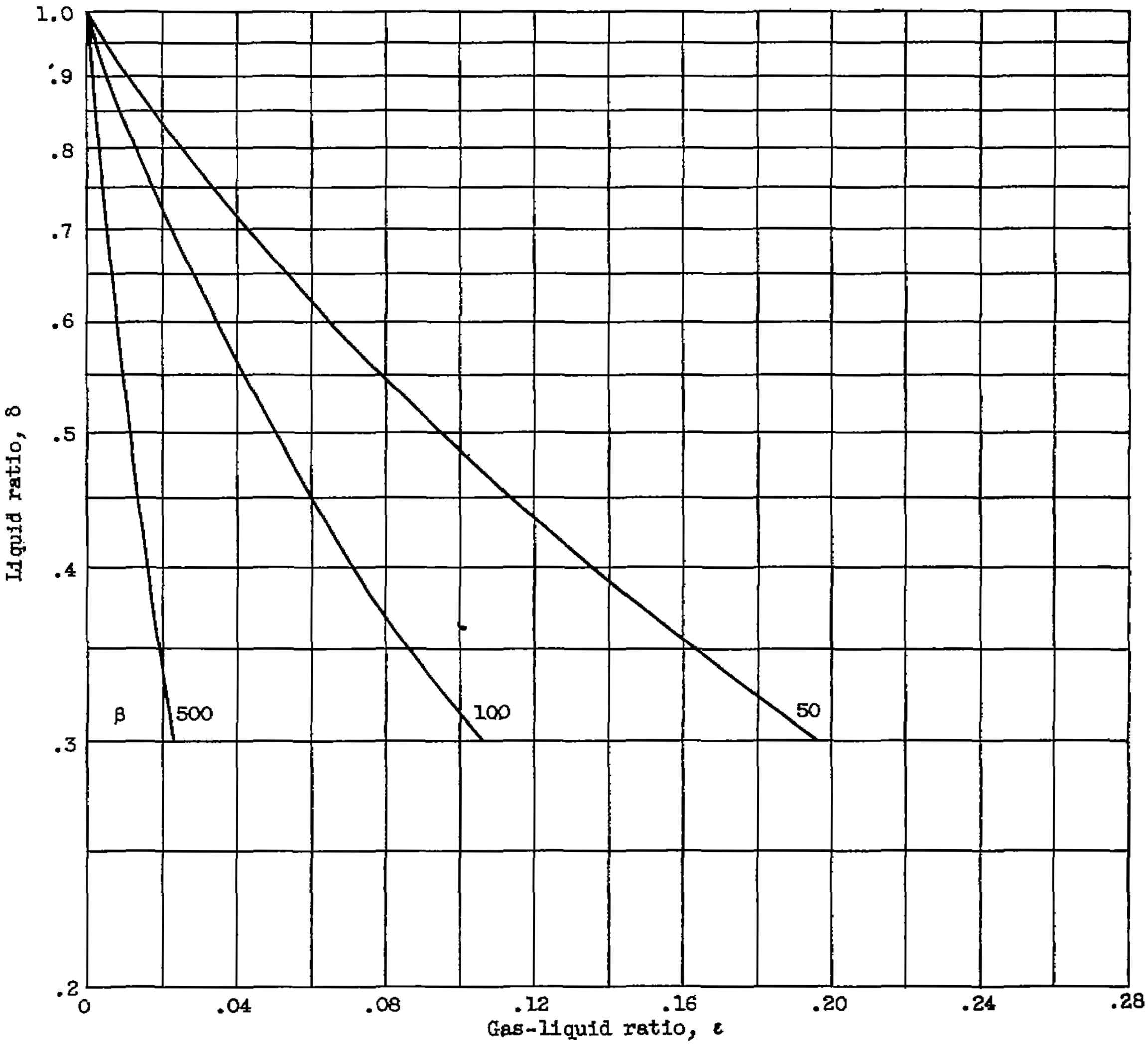

(b) Ratio of injection pressure to chamber pressure for full flow, 1.3.

Figure 2. - Continued. Variation of liquid ratio for adiabatic gas flow with a specific heat ratio of 1.67.

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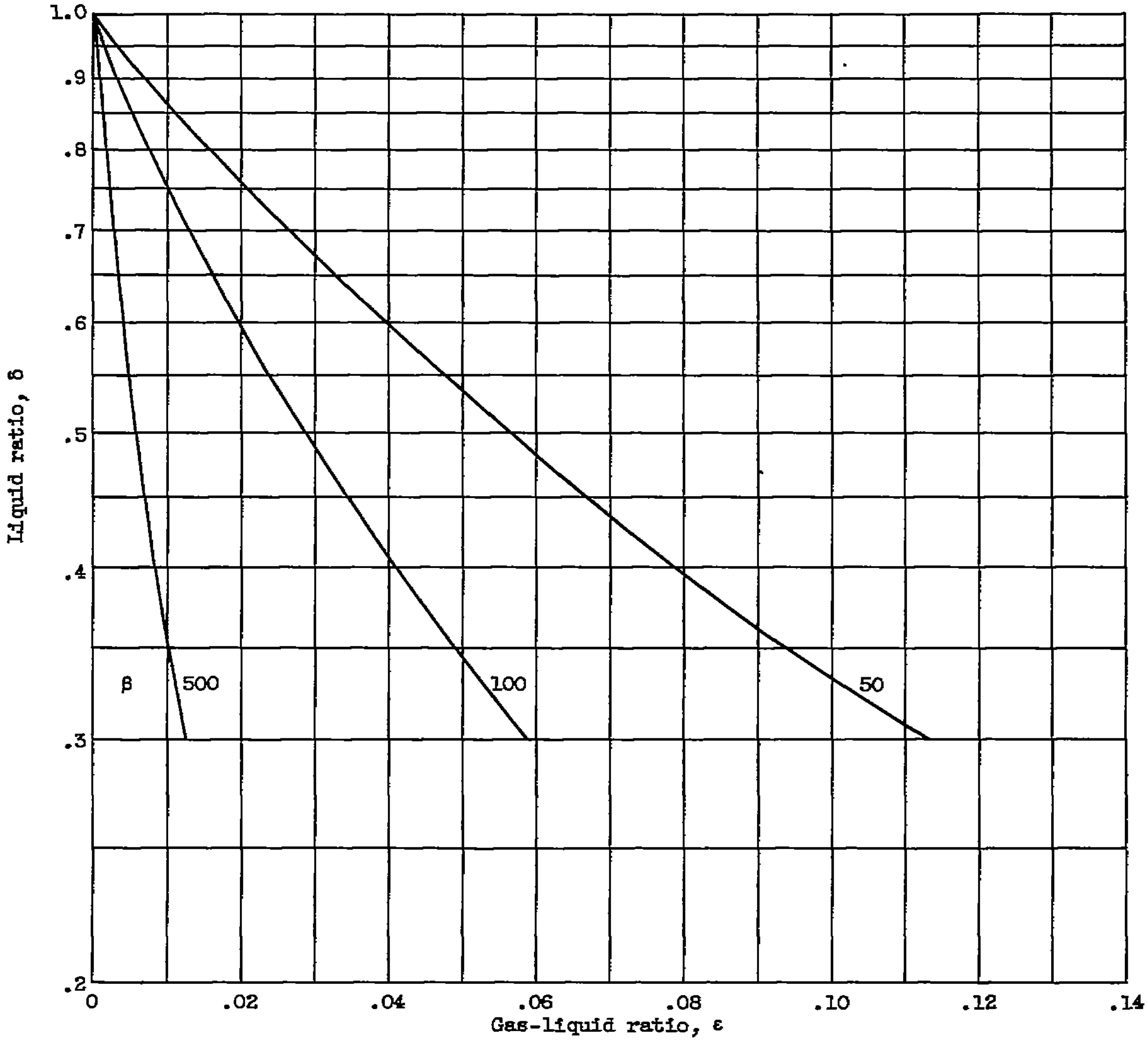

(c) Ratio of injection pressure to chamber pressure for full flow, 1.9.

Figure 2. - Concluded. Variation of liquid ratio for adiabatic gas flow with a specific heat ratio of 1.67.

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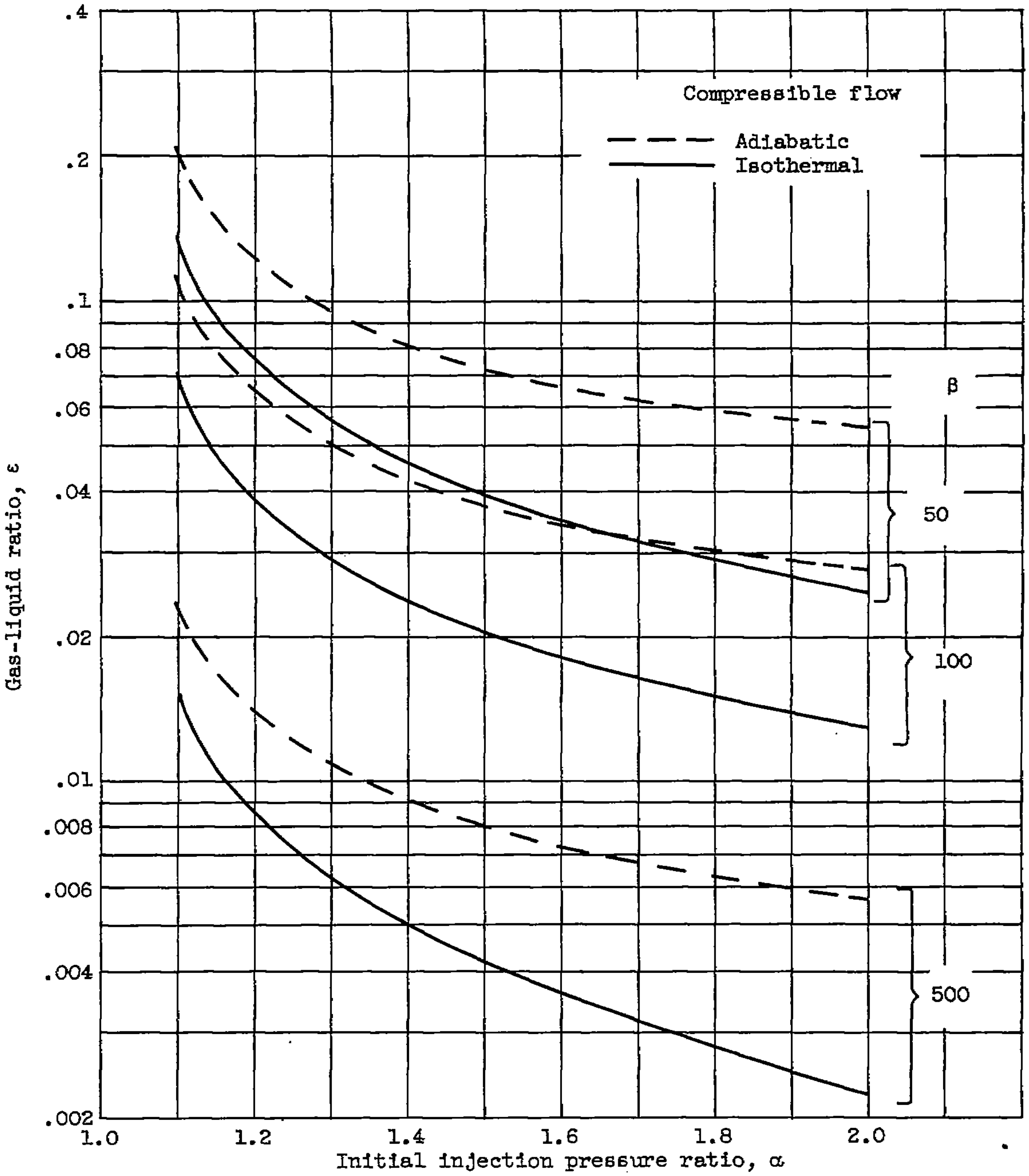

Figure 3. - Variation of gas-liquid ratio with initial injection pressure ratio; liquid ratio of 0.5.

[redacted]

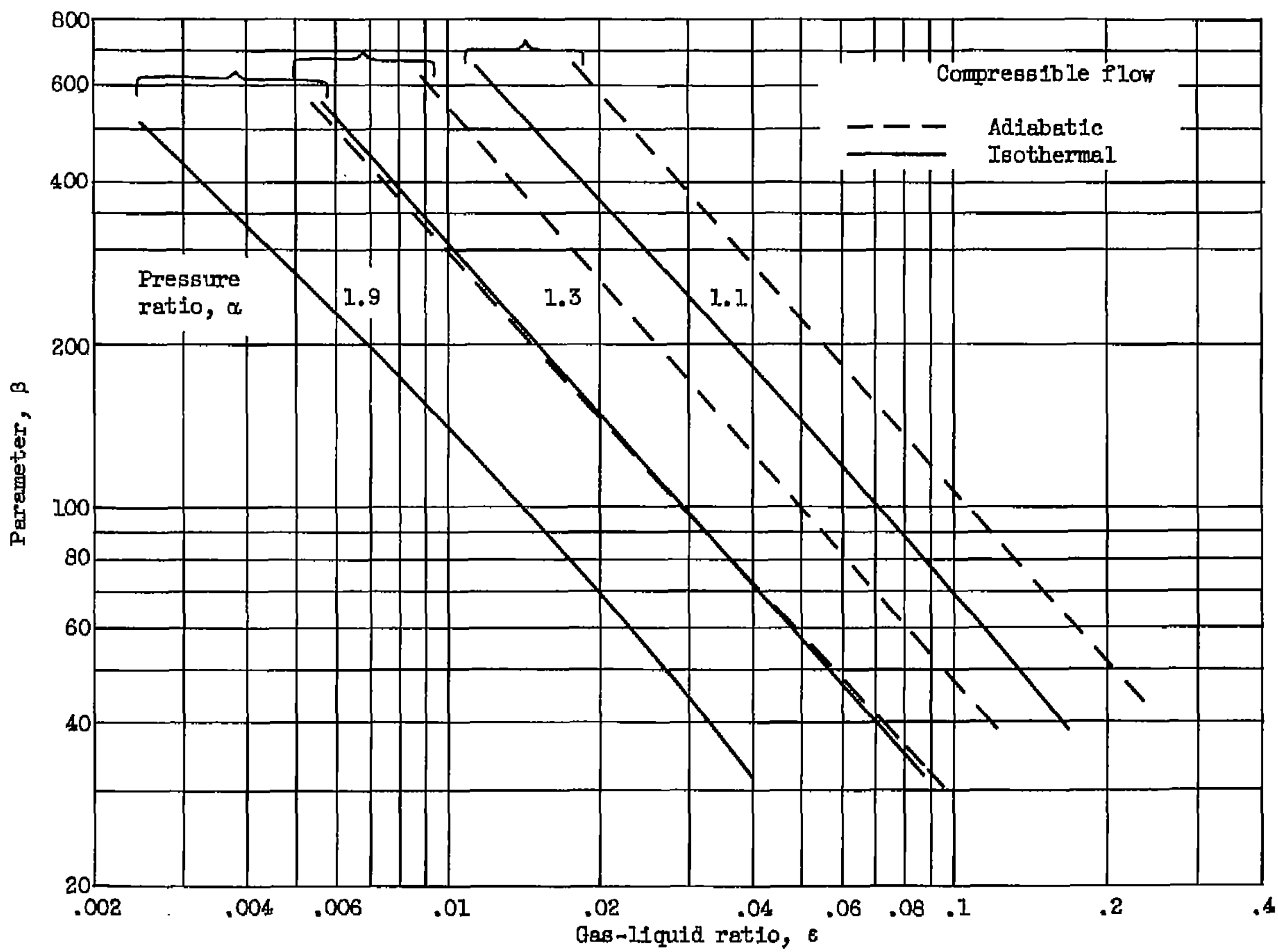

Figure 4. - Variation of gas-liquid ratio with the parameter β; liquid ratio of 0.5.

[redacted]

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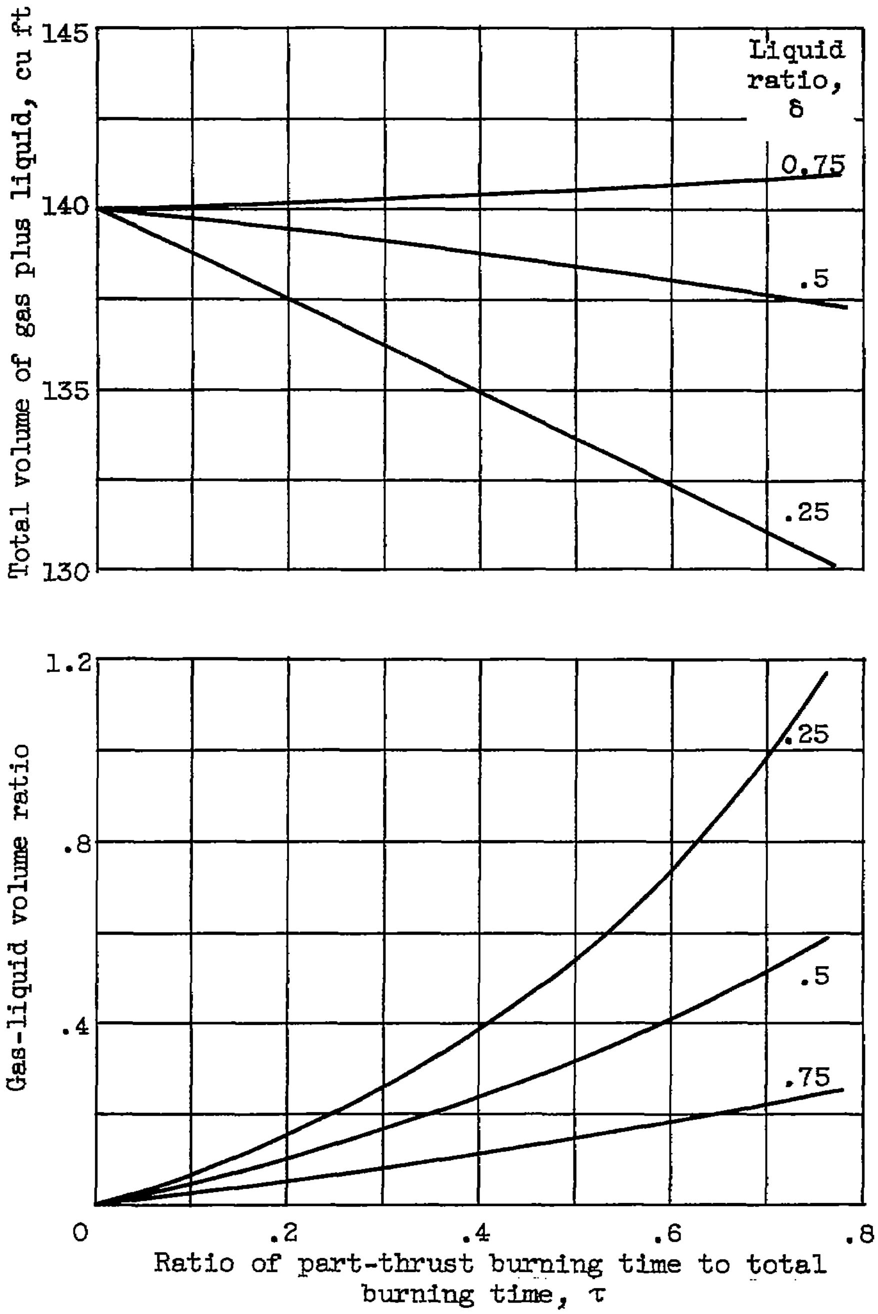

Figure 5. - Gas and liquid requirements for part-thrust flight.

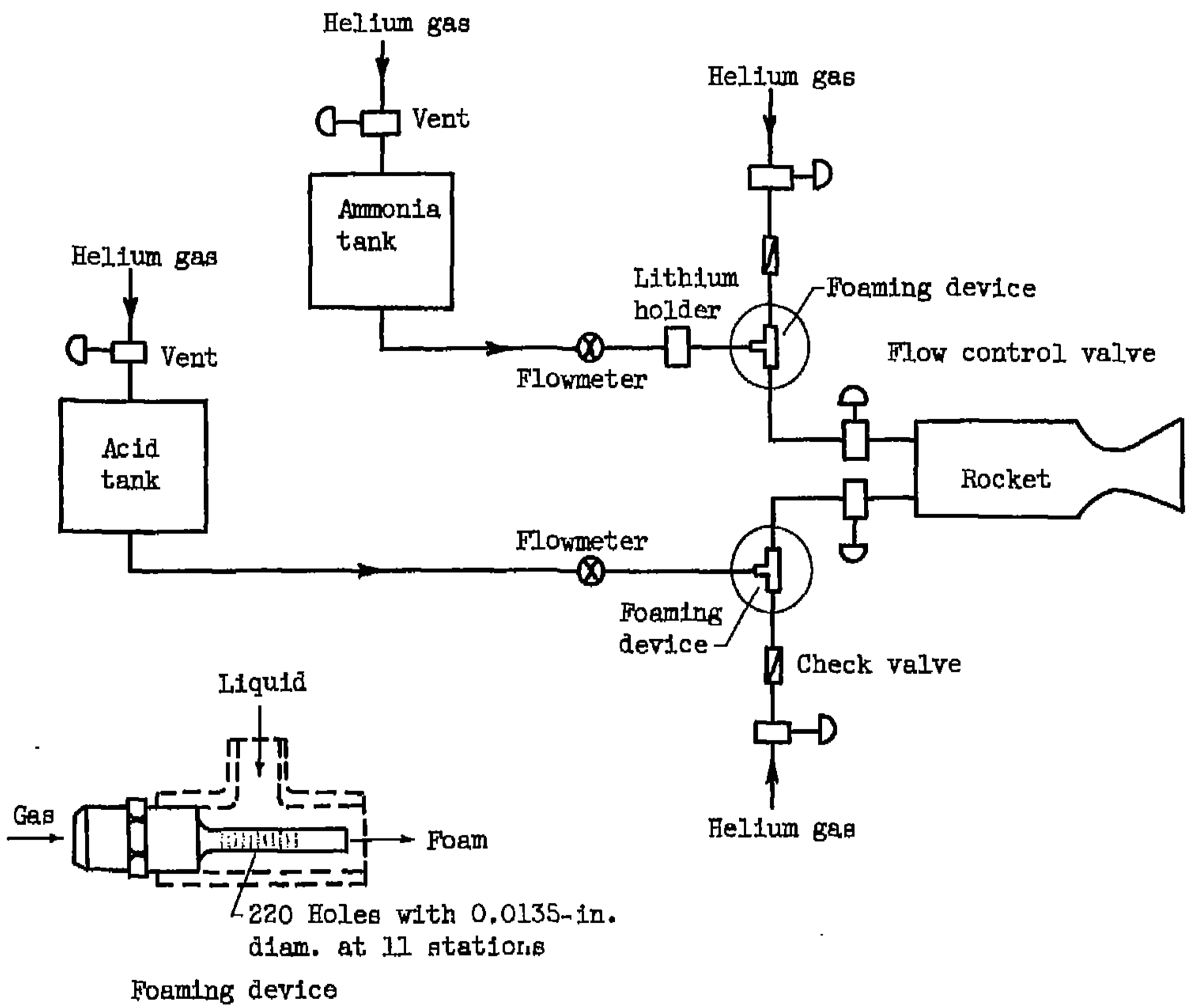

Figure 6. - Schematic diagram of experimental rocket system.