AD 39981

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Aerojet-General CORPORATION

AZUSA, CALIFORNIA

INFORMAL REPORT OF PROGRESS

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23 June 1954

TO: Director
Naval Research Laboratory
Washington 25, D. C.

Attn: Mr. J. W. Townsend, Jr.
Scientific Officer

VIA: Bureau of Aeronautics Representative
Aerojet-General Corporation
6352 N. Irwindale Avenue
Azusa, California

SUBJECT: Research and Development on XRV-N-13
Aerobee-Hi Sounding Rocket

CONTRACT: Nonr-1265(00)

PERIOD COVERED: 1 May through 31 May 1954

This is the tenth in a series of informal letter reports submitted in partial fulfillment of the contract.

Approved by:

T B Walker
T. B. Walker
Senior Engineer
Liquid Engine Division

NOTE: The information contained herein is regarded as preliminary and subject to further checking, verification, and analysis.

Item 1. Overall Design of Sounding Rocket (Model XRV-N-13)

The preliminary layout of the lightweight tail section with the enlarged tail fin is continuing. Hypersonic wind tunnel data for determining the most desirable size of the fin are not yet available. The preliminary layout (Drawing No. 3-007862) of the entire vehicle is complete except for the revised tail fin. Drawings for the propellant tank assembly No. 3-007863 have been completed and are ready for release.

Item 2. Fabricate and Deliver Two Prototype Sounding Rockets

A. Production work on the vehicle will be deferred, pending completion of the overall design.

B. Material for the two booster rockets has been received. These will be fabricated in time for the tentative delivery date of November 1954 for the boosters.

Item 3. Adapt the Model AJ11-6 Design for Chemical Pressurization (Model XRV-N-13)

A. Two satisfactory water pumping tests have been completed utilizing a graphite nozzle. In the next test acid will be pumped from the acid tank and water from the fuel tank.

B. Mixed gas temperatures have been approximately 1000°F, with the gas transfer line being heated to about 575°F. The highest skin temperature recorded was 371°F at the top of the acid tank.

C. Some difficulty has been encountered in the use of old-style Aerobee regulators, in that sudden changes in regulated gas pressure (approximately 50 psi reduction in pressure) have taken place in several pumping tests. A design revision which prevents twisting of the regulator spring during adjustment is expected to eliminate this difficulty.

D. Considerable variation in burning rate has been observed with AN-2091 solid propellant. This variation in burning rate appears to be a function of the amount of oxidizer settling which takes place when the grain is cast. A preliminary analysis indicates that there is a relation between burning rate and the particular part of the casting from which the grain is taken. This would indicate that only the center of the grain casting should be used, for example. The variation in burning rates is of importance because it affects both the pressure developed within the generator case and the duration of burning of the grain. Efforts are being made to increase the reproducibility of the solid-propellant grain.

E. Tests thus far conducted have provided a fairly firm figure for the quantity of helium diluent required. The helium tank will be reexamined to determine whether a reduction in either the gas pressure or volume, with a consequent saving in tank weight, can be permitted.

F. The design of the lightweight gas generator is nearing completion.

Item 4. Test Vehicle Launching, Modified Model RTV-A-1a (RTV-N-10b) (Model AJ10-24)

A. The test vehicle engine assembly had been successfully fired in the vertical position for full duration utilizing RFNA containing 6.5% NO_2 and was ready for shipment. At NRL's request it was held at Aerojet until 27 May, when it was shipped to WSPG.

B. In discussions with the NRL Project Scientist, it was agreed to conduct full-duration firings on two chambers utilizing RFNA containing 22% NO_2, and that if these tests were satisfactory this acid would be used for the flight of the RTV-N-106 vehicle. The use of RFNA with 22% NO_2 should increase the expected zenith altitude 4 to 5 miles. The thrust chambers are being prepared for testing.

C. Because of difficulties with the strain gage system, encountered by both Aerojet and NRL, it has been agreed to eliminate this instrumentation and to obtain the data on a subsequent flight.

Item 6. Drawings

One set of reproducible drawings of the prototype vehicle will be supplied upon completion of the design.

Item 7. Thrust-Chamber Design Changes and Static Test-Firings

A. No static test-firings were made during this period. However, continued analysis of already available test data has disclosed a definite correlation between thrust-chamber performance and the fuel injection temperature -- the higher the fuel injection temperature (up to 223°F) the better is the performance of the 32756 Aerobee chamber. Since the only obvious effect on the chamber reaction that an increase in fuel injection temperature can produce is to decrease the time lag before combustion occurs, it appears that at low injection temperatures a portion of the propellant leaves the chamber not completely combusted, because of insufficient reaction time. At 223°F and above reaction appears to be complete since a higher injection temperature does not improve the performance significantly. The degree of completeness of combustion for any given injection temperature has been determined empirically and is plotted in Figure 1 as a "propellant flow rate correction factor," with 1.00 representing apparently complete combustion.

B. In Figure 2, which is useful in determining the performance of the 32756 Aerobee chamber, the actual propellant flow rate is multiplied by the correction factor obtained from Figure 1, and this product is plotted as the abscissa in Figure 2 against the measured mixture ratio.

Thrust, chamber pressure, and thrust coefficient are then read as ordinates. The accuracy of this correlation is indicated in the last column of Table I, which summarizes all test data. Sixty-seven percent of this measured data differs from the chart data by 0.5% or less, and only 8% differs by more than 1.0%. The maximum difference is 1.8%. When these charts are used in conjunction with the "high-" and "low-performance charts" included in the previous progress report, the effect of any design change or operating condition on the performance of the thrust chamber can be evaluated regardless of differences in thrust, mixture ratio, chamber pressure, or other parameters. With these charts, one or two tests are sufficient to determine the magnitude of any improvement. This should simplify and expedite the thrust chamber development program.

C. Five thrust chambers which had been damaged in testing were cut apart and the cooling passages examined. In four of the chambers, which were "high-performance," the cooling passages were coated with a very thin, dense, tightly adhering, light-brown film. The fifth chamber, which was "low-performance," was coated with a black, porous, relatively thick deposit which appeared to be carbon. The difference in the character of the deposits explains the lower rate of heat transfer in the "low-performance" chamber. The reason for the difference in the deposits in the cooling passages of the different chambers is being considered.

D. The thin-walled Type 410 thrust chamber is now undergoing final assembly.

E. Complete engine static-firings will be conducted for the prototype rockets upon completion of fabrication of the units.

Item 8. Aspect Control Study

A. The control systems under consideration have been reviewed in the light of the requirements set forth in the letter from NRL dated 19 March. One system has been tentatively decided upon and is now undergoing intensive study. The controller and compensating device have been designed, and the electronic circuits have been laid out. The primary problem remaining is to determine the optimum control jet forces. Equations have now been developed to determine these forces and are being prepared for solution on a computer.

B. In an attempt to evaluate the additional fin area required for the proper stability of the Aerobee-Hi vehicle, existing aerodynamic data have been extrapolated to Mach 8. The reliability, however, is seriously questioned, since the highest available test point is Mach 4.5.

General

The total contract funds expended up to 15 May 1954 amounted to $135,434, exclusive of fee. The balance of the funds remaining, exclusive of fee, amounts to $144,542.

TABLE I

SUMMARY OF DATA FOR AEROBEE THRUST CHAMBER 32756

Run No.	Chamber Pressure, psia	Thrust Coef., C_F	I_{sp}, lb-sec/lb	Mixture Ratio	Propellant Flow Rate, lb/sec	Fuel Injection Temperature, °F	Flow Rate Correction Factor (Fig. 1)	Corrected Propellant Flow Rate lb/sec	Calculated Thrust, lb (Fig. 2)	Measured Thrust, lb	Difference %
PG1407-40	309	1.371	197	2.75	19.84	202	0.988	19.60	3885	3908	0.6
-41	321	1.387	198	2.76	20.60	204	0.989	20.37	4070	4088	0.4
A27-LC-1	319	1.357	193	2.75	20.60	182	0.977	20.13	4003	3980	-0.5
-2	322	1.358	196	2.74	20.48	186	0.979	20.05	3995	4020	0.6
D30-LC-1	307	1.356	194	2.77	19.77	173	0.972	19.22	3795	3830	0.9
-2	320	1.353	198	2.59	20.17	182	0.977	19.71	4003	3980	-0.6
-3	314	1.363	190	2.91	20.74	176	0.974	20.20	3953	3940	-0.3
-4	313	1.359	188	2.90	20.78	111	0.960	19.95	3890	3910	0.5
-5	315	1.373	189	2.82	20.73	111	0.960	19.90	3920	3920	0.0
-6	319	1.363	191	2.92	20.94	183	0.977	20.46	4017	4000	-0.4
-7	339	1.379	198	2.90	21.65	195	0.984	21.30	4270	4300	0.7
-8	346	1.380	197	2.82	22.34	168	0.970	21.67	4404	4390	-0.3
-9	342	1.382	202	2.53	21.54	190	0.981	21.13	4360	4350	-0.2
-12	320	1.373	212	2.31	19.01	264	1.000	19.01	4039	4044	0.2
-13	327	1.370	199	2.47	20.67	183	0.977	20.19	4170	4120	-1.2
-20	322	1.367	197	2.56	20.52	159	0.966	19.82	4045	4048	0.1
-21	321	1.363	198	2.57	20.28	168	0.970	19.67	4010	4024	0.3
A34-LC-1	325	1.376	204	2.57	20.12	225	1.000	20.12	4100	4112	0.3
-2	326	1.360	200	2.51	20.48	201	0.988	20.21	4150	4076	-1.8
D30-LC-22	326	1.349	203	2.59	19.90	280	1.000	19.90	4040	4040	0.0
-25	325	1.389	202	2.72	20.55	230	1.000	20.55	4145	4152	0.2
-26	325	1.354	201	2.65	20.13	223	1.000	20.13	4060	4048	-0.3
-27	314	1.357	195	2.73	20.12	168	0.980	19.72	3920	3920	0.0
-28	333	1.357	197	2.66	21.17	188	0.980	20.75	4200	4160	-1.0

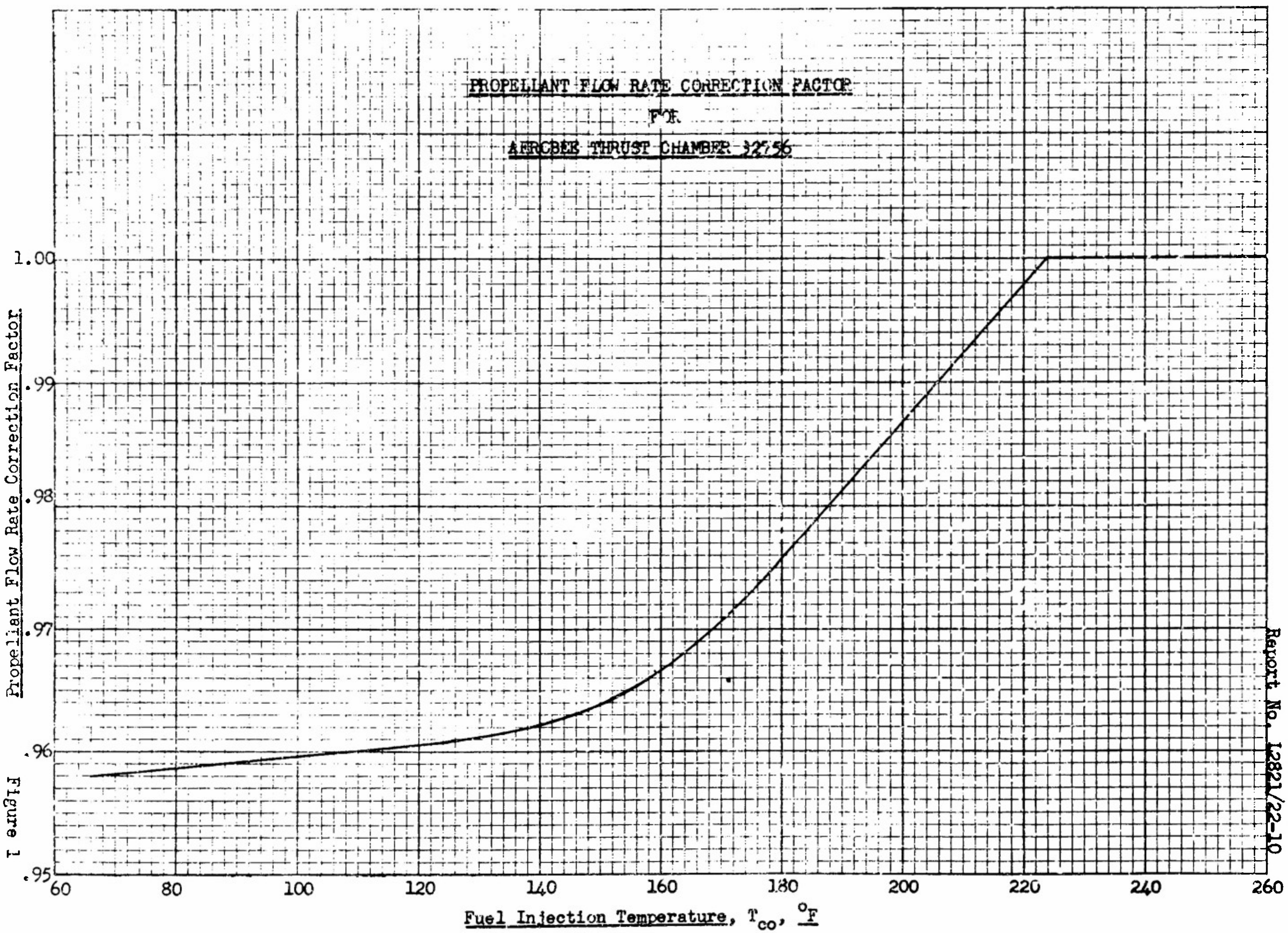
PROPELLANT FLOW RATE CORRECTION FACTOR
FOR
AEROBEE THRUST CHAMBER 32756
Propellant Flow Rate Correction Factor
1.00
.99
.98
.97
.96
.95
60
80
100
120
140
160
180
200
220
240
260
Fuel Injection Temperature, T_{co}, °F

Figure 1

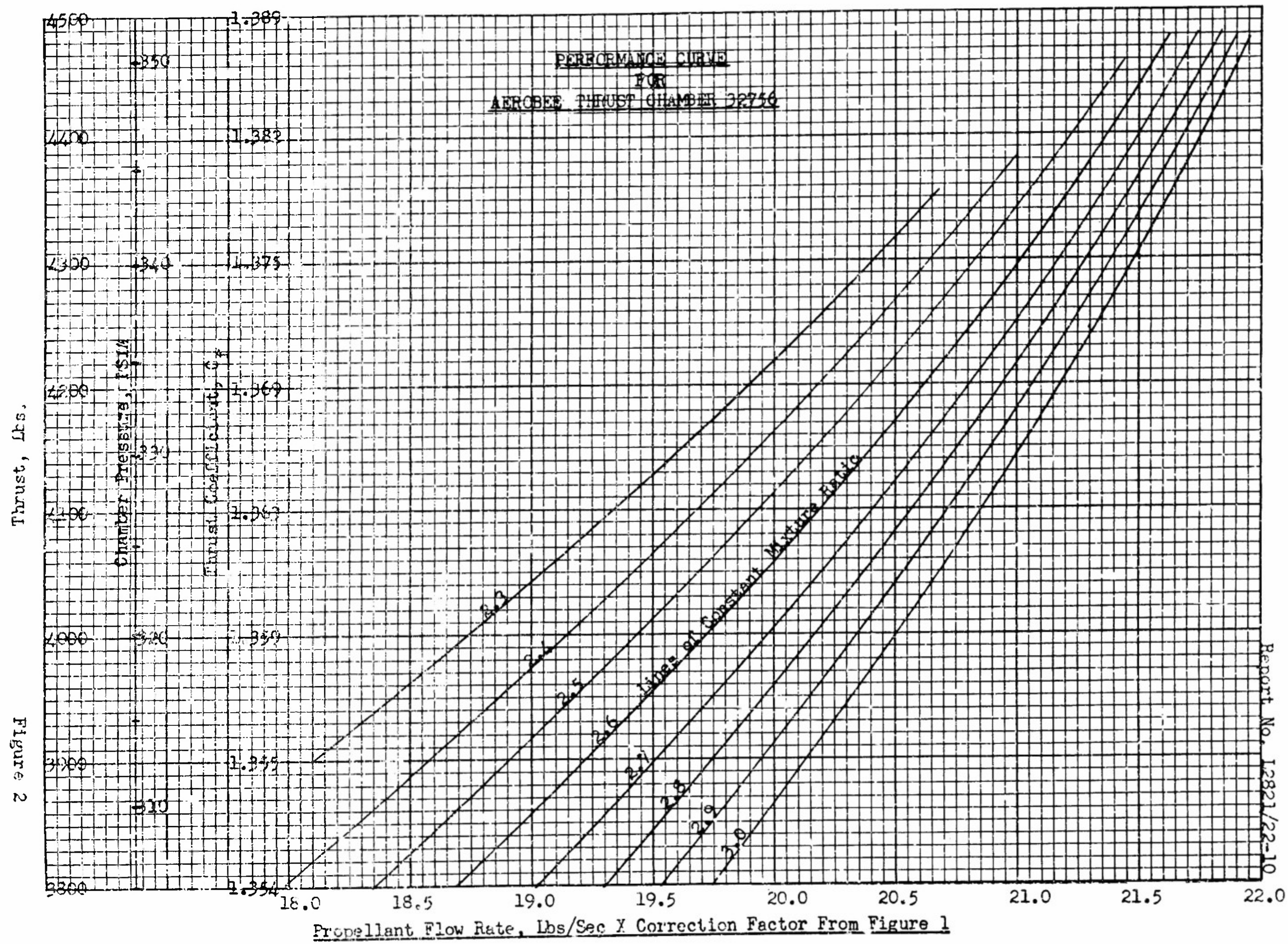
PERFORMANCE CURVE
FOR
AEROBEE THRUST CHAMBER 32756
Lines of Constant Mixture Ratio
2.3
2.4
2.5
2.6
2.7
2.8
2.9
3.0
Thrust, Lbs.
4500
4400
4300
4200
4100
4000
3900
3800
Chamber Pressure, PSIA
350
340
330
320
310
Thrust Coefficient, C_F
1.389
1.382
1.375
1.369
1.363
1.359
1.345
1.334
18.0
18.5
19.0
19.5
20.0
20.5
21.0
21.5
22.0
Propellant Flow Rate, Lbs/Sec X Correction Factor From Figure 1

Figure 2

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