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Advanced Oxygen-Hydrocarbon Rocket Engine Study

Contract NAS 8-33452
Bi-Monthly Progress Report 33452M-3
April 1980

Prepared For:
National Aeronautics And Space Administration
George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

By:
C. J. O'Brien

Aerojet
Liquid Rocket
Company

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ENGINE STUDY

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Prepared by:

C. J. O'Brien
C. J. O'Brien
Project Engineer
ALRC Engineering

Approved by:

D. Rosenberg for
J. W. Salmon
Program Manager
ALRC Programs

AEROJET LIQUID ROCKET COMPANY
P.O. Box 13222
Sacramento, California 95813

FOREWORD

This is the third bi-monthly progress report submitted for the Advanced Oxygen - Hydrocarbon Rocket Engine Study per the requirements of Contract NAS 8-33452. The work is being performed by the Aerojet Liquid Rocket Company for the NASA-Marshall Space Flight Center. The contract was issued on 15 October 1979. The program inclusive dates for period of performance are 15 October 1979 through 15 February 1981. This report covers the period from 1 February 1980 to 31 March 1980.

The program consists of parametric analysis and design to provide a consistent engine system data base for defining advantages and disadvantages, system performance and operating limits, engine parametric data, and technology requirements for candidate high pressure LO_2/Hydrocarbon engine systems.

The NASA-MSFC Project Manager is Mr. R. J. Richmond. The ALRC Program Manager is Mr. J. W. Salmon and the Project Engineer is Mr. C. J. O'Brien.

Contributors to this bi-monthly report are:

R. L. Ewen - Heat Transfer Analysis
S. Kent - Engine Weight and Envelope Analysis
G. M. Meagher - Performance Analysis

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I. INTRODUCTION

In the decade of the 1980's and beyond, the nation's expanding space operations may require an improved surface-to-orbit transportation system using advanced booster vehicles which have increased performance and capability compared to the current space shuttle concept. The mixed-mode propulsion principle clearly indicates the potential performance advantages of using high density-impulse rocket propellants in such large ΔV applications. For this reason, hydrocarbon fuels exhibiting increased density relative to liquid hydrogen (LH_2), at the penalty of lower specific impulse, are being considered for the booster propulsion system of space shuttle improvements and derivatives as well as for single-stage-to-orbit and two-stage-to-orbit heavy-payload vehicles.

Preliminary identification and evaluation of promising liquid oxygen/hydrocarbon (LO_2/HC) rocket engine cycles is desirable to produce a consistent and reliable data base for vehicle optimization and design studies, to demonstrate the significance of propulsion system improvements, and to select the critical technology areas necessary to realize such advances.

It is the purpose of this study to generate a consistent engine system data base for defining advantages and disadvantages, system performance and operating limits, engine parametric data, and technology requirements for candidate high pressure LO_2/HC engine systems. The study will also synthesize optimum LO_2/HC engine power cycles and generate representative conceptual engine designs for a specified advanced surface-to-orbit transportation system.

To accomplish the program objectives, the study is composed of four major technical tasks and a reporting task. These tasks and summarized objectives are:

A. TASK I - ENGINE CYCLE CONFIGURATION DEFINITION

Formulate and assess families of high chamber pressure LO_2/HC engine cycles.

I. Introduction (cont.)

B. TASK II - ENGINE PARAMETRIC ANALYSIS

Generate performance, weight, and envelope parametric data for viable concepts based upon historical data and conceptual evaluations.

C. TASK III - ENGINE/VEHICLE TRAJECTORY PERFORMANCE ASSESSMENT (ENGINE SCREENING)

Conduct a preliminary comparison of selected engine cycles utilizing a simplified vehicle trajectory performance model.

D. TASK IV - BASELINE ENGINE SYSTEMS DEFINITION

Prepare preliminary designs of two baseline engine configurations. Conduct heat transfer, turbomachinery, combustion stability, structural, and controls analysis of the baseline engines and components. Conduct a parametric sensitivity analysis including the effects of turbine temperature and number of usable life cycles. Provide the appropriate data in a format suitable for use in vehicle application analyses.

E. TASK V - REPORTING

Provide informal bi-monthly technical and fiscal progress reports, hold program reviews at NASA/MSFC and prepare a final report.

II. TECHNICAL PROGRESS SUMMARY

The overall progress on the program is indicated in Figure 1.

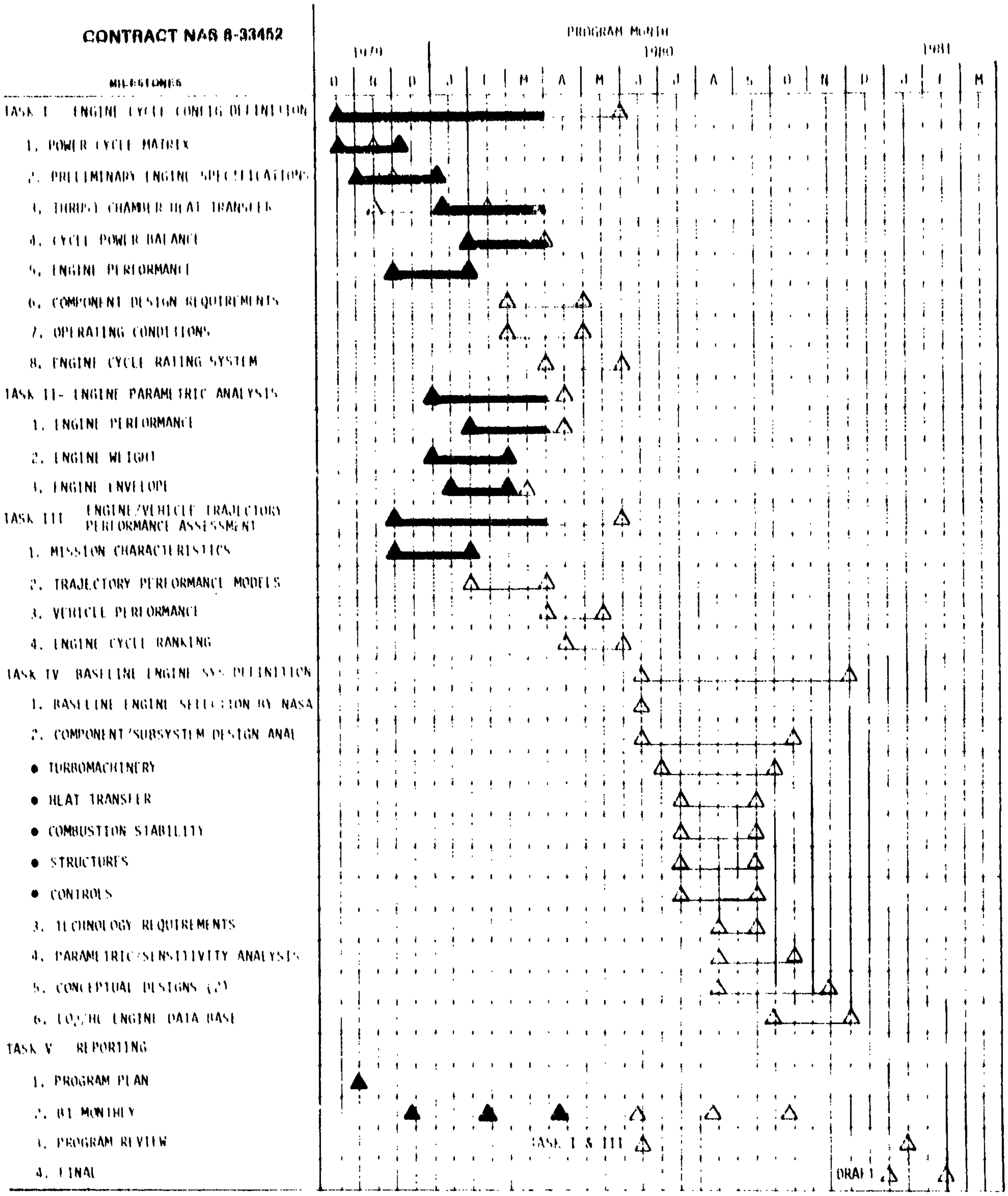

Figure 1. Major Milestone Schedule

A. TASK I - ENGINE CYCLE CONFIGURATION DEFINITION

1. Thrust Chamber Heat Transfer

The parametric chamber and nozzle, (ϵ = 8 to 40) cooling analysis is complete for the four potential coolants: RP-1, LCH_4, LO_2 and LH_2. Documentation of the results is underway. A summary of the cooling capability of each propellant is given in Tables I through VI.

RP-1 was found to be an unsatisfactory coolant when a coolant-side wall temperature of 550°F (the accepted coking temperature for this fuel) is imposed. If a purified version of RP-1 (e.g., JP-7) with a coolant-side wall temperature coking limit of 800°F is used, the purified fuel is seen in Table I to be capable of cooling engines operating at chamber pressures up to 2500 psia.

All of the heat transfer analyses are based upon a clean zirconium-copper hot gas-side wall. Early LO_2/RP-1 engines, such as the Titan I, the H-1 and the F-1 engines benefited from a carbon layer buildup on the thrust chamber hot gas-side wall. These engines operated at chamber pressures from 600 to 1000 psia and at fuel-rich mixture ratios from 2.23 to 2.27 (equivalence ratios from 1.53 to 1.51), and were RP-1 cooled to area ratios from 8:1 to 10:1. Since the carbon layer buildup on such engines has been known to flake off, advanced engine parametric studies (such as on Contract NAS 3-19727) were directed by NASA/LeRC to be based on uncoated walls. This assumption is seen to rule out conventional RP-1 as a coolant at chamber pressures of 1000 psia or greater.

An investigation of the sensitivity of a carbon coating is being conducted on another program (NAS 9-15958). The data and approach for that effort will be reviewed on this program, and the coating sensitivity

TABLE I

PURIFIED RP-1 COOLING SUMMARY

COKING WALL TEMPERATURE LIMIT = 800°F

Thrust 10^6 lbf	P_c psia	Coolant Fraction	ΔP psi	ΔT_b °F	L' in.
0.2	1000	0.9	192	191	14.14
	1500		641	203	12.88
	2000	1.0	1210	195	12.05
	2500		2322	200	11.45
0.6	1000	0.9	237	165	22.38
	1500		614	165	18.27
	2000		1210	168	15.83
	2500		2063	169	14.74
2.0	1000		332	147	40.86
	1500		794	148	33.36
	2000		1466	150	28.89
	2500	1.0	2517	135	25.84

TABLE II

METHANE COOLING SUMMARY

COOLANT FRACTION = 1.0

Thrust 10^6 lbf	P_c psia	Δp psi	ΔT_b °F
0.2	1000	45	147
	2000	258	176
	3000	840	193
	4000	2195	197
0.6	1000	62	129
	2000	295	138
	3000	833	147
	4000	2088	153
	5000	4885	156
2.0	1000	84	117
	2000	388	123
	3000	1220	124
	4000	2818	124

TABLE III

OXYGEN COOLING SUMMARY

Thrust 10^6 lbf	P_c psia	Coolant Fraction	Δp psi	ΔT_b °F	L' in.
0.2	1000	1.0	78	82	14.14
	2000		376	95	12.05
	3000		1122	103	10.98
	4000		2889	107	10.28
0.6	1000		108	72	22.38
	2000		450	73	[illegible]
	3000		1129	[illegible]	14.14
	4000		28[illegible]4	79	13.23
2.0	1000	0.87	125	73	40.86
	2000		553	74	28.89
	3000		1672	73	23.59
	4000		4424	73	20.43

TABLE IV

HYDROGEN COOLING SUMMARY

LOX/RP-1 ENGINE

Thrust 10^6lbf	P_c psia	Coolant Flow, lb/sec	Δp psi	ΔT_b °F
0.2	1000	7.5	53	655
	4000	15	440	445
	5000	20	799	355
0.6	1000	15	137	787
	4000	25	405	491
	5000	30	680	429
1.0	1000	20	256	946
	4000	30	429	559
	5000	35	770	498
2.0	1000	40	607	879
	4000	50	715	594
	5000	60	850	478

TABLE V

HYDROGEN COOLING SUMMARY

LOX/CH_4 ENGINE

Thrust 10^6 lbf	Pc psia	Coolar Flow, lb/sec	ΔP psi	ΔT_b °F
0.2	1000	7.5	72	740
	4000	14.6	611	495
	5000	19.4	1105	395
0.6	1000	15.0	199	900
	4000	24.7	563	558
	5000	29.5	1041	484
1.0	1000	25.0	334	852
	4000	29.7	767	642
	5000	34.6	1369	568
2.0	1000	40.0	893	1004
	4000	49.6	983	684
	5000	59.4	1122	551

TABLE VI

NOZZLE COOLING SUMMARY (ϵ = 8 to 40)

1. Baseline Designs (LOX/RP-1)

F = 600,000 lb $\quad$ Pc = 4000 psia

	Coolant	ΔP psi	ΔT_b °F
Purified	RP-1	440	106
	O_2	77	56

2. Scaling

$$\Delta P \sim Pc^{3.5}\ F^{0.25}$$

$$\Delta T_b \sim P_c^{0.05}/F^{0.075}$$

determined for select cases at a thrust level of 600K lbf. Care must be taken, however, in applying data from lower pressure (Pc $\approx$ 1000 psia) chambers. The higher pressures of this program require a mixture shift toward stoichiometric (e.g., Pc = 4000 psia, MR = 2.9, ER = 1.18), and the less fuel-rich environment may not result in as great a carbon deposit.

Methane is capable of cooling an LO_2/LCH_4 thrust chamber to chamber pressures of 4000 psia. Above this pressure level the coolant channel pressure drop becomes excessive as shown in Table II for the 600K lbf engine at a chamber pressure of 5000 psia (ΔP = 4885 psia).

Table III shows that liquid oxygen can cool an LO_2/RP-1 or LCH_4 engine to chamber pressures of 4000 psia. The high pressure drop shown for the large thrust (2M lbf) engine can probably be reduced through chamber geometry optimization. Scaling of the baseline 600K engine results in a relatively long chamber at the high thrust level, which results in a high coolant channel ΔP.

Liquid hydrogen cooling summaries are given in Tables IV and V, for LO_2/RP-1 and LO_2/LCH_4 engines, respectively. It is seen that small amounts of hydrogen are required, these amounts being commensurate with the power balance requirements for these engines.

The nozzle cooling summary is given in Table VI for both purified RP-1 and LO_2 coolants. The pressure drop and bulk temperature rise in both cases appear to be satisfactory for the baseline case given in the table. Scaling equations for different chamber pressures and thrust levels are also given in the table.

2. Cycle Power Balance

No additional work was conducted on this subtask pending completion of the heat transfer effort.

B. TASK II - ENGINE PARAMETRIC ANALYSIS

1. Engine Performance

Parametric engine performance data are summarized in Figures 2 through 9 for the LO_2/RP-1 and LO_2/LCH_4 engines. The simplified JANNAF methodology was used to derive the delivered performance for each point design engine. A more detailed description of the methodology and results will be given when this effort is completed.

2. Engine Weight and Envelope

These subtasks have been completed. An existing ALRC parametric weight/envelope computer program was modified to provide the weight and envelope data for the advanced oxygen-hydrocarbon engine cycles considered in this study. A sample output for the baseline case (LO_2/RP-1, fuel-rich gas generator, RP-1 cooled) is illustrated in Figure 10. Parametric engine weight and envelope data are presented for LO_2/RP-1 engines in Figures 11 through 20. Table VII gives the baseline weights used throughout the analysis. (This table is an updated version of Table V of Bimonthly Progress Report 33452 M-2, February 1980.)

Because most of the cycles studied are very similar, parametric information is presented only for a typical baseline cycle. Differences in these cycles leading to weight differences will be reported separately when a comparison is made between the cycles.

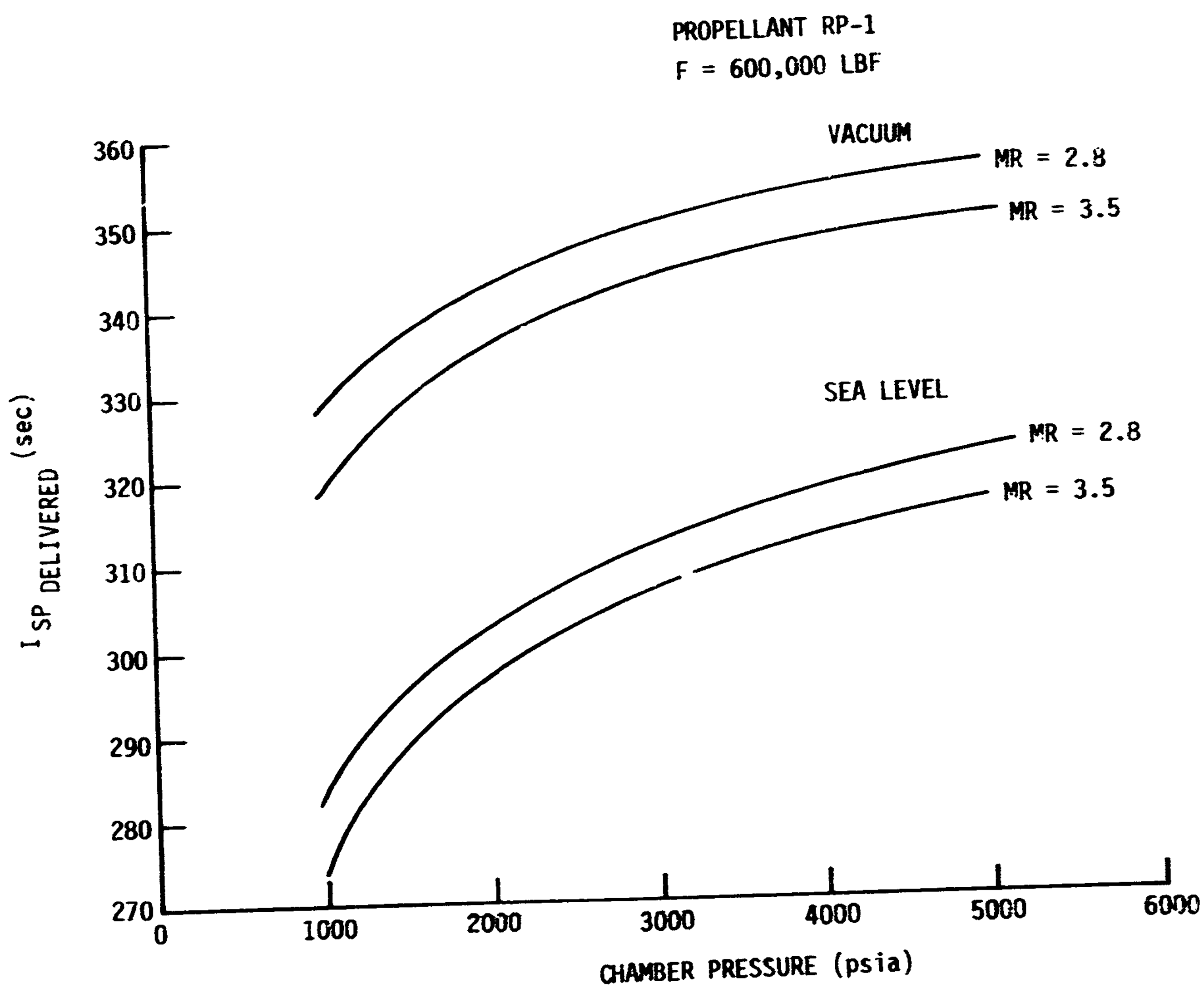

Figure 2. Delivered LOX/RP-1 Engine Performance Versus Chamber Pressure

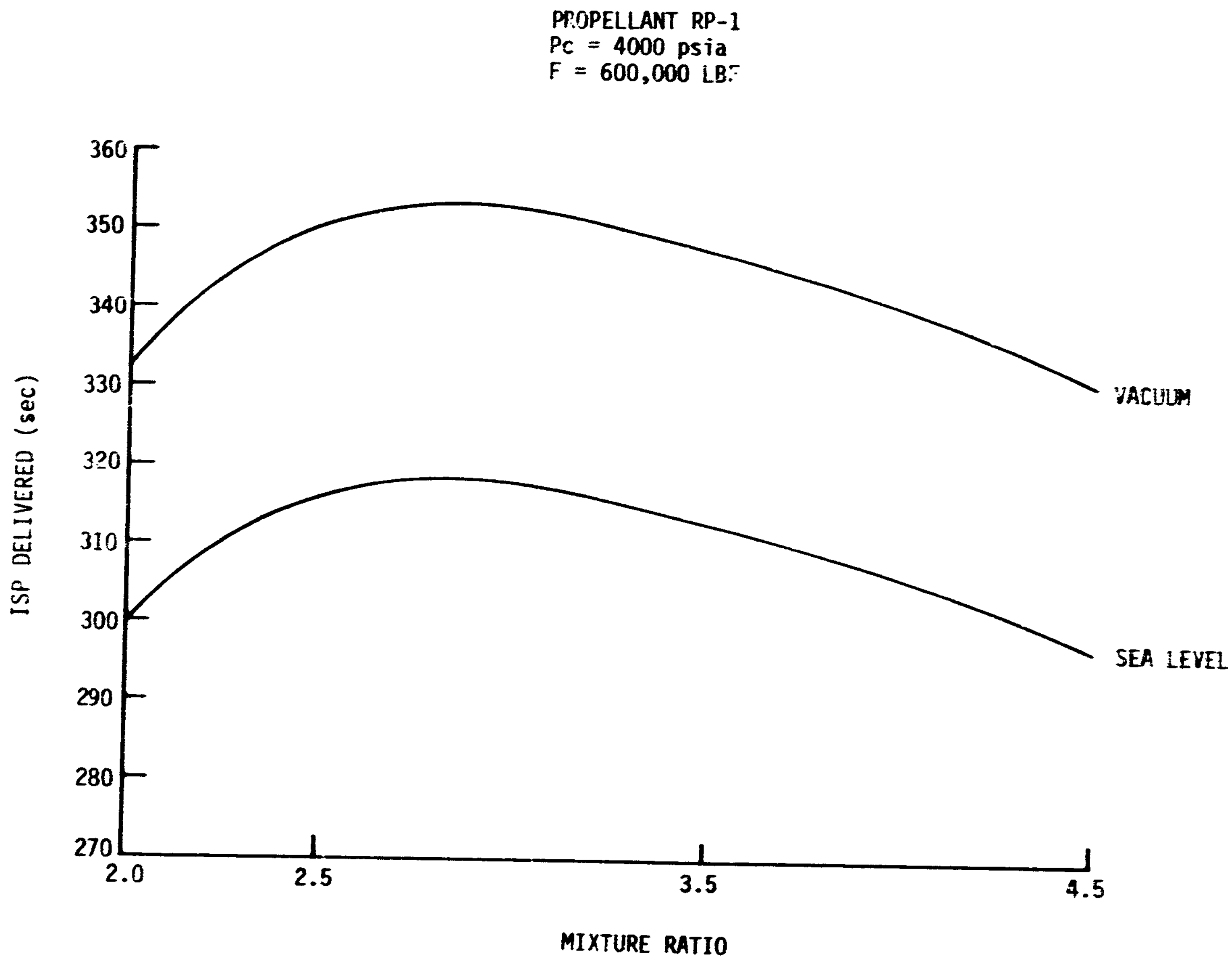

Figure 3. Delivered LOX/RP-1 Engine Performance Versus Mixture Ratio

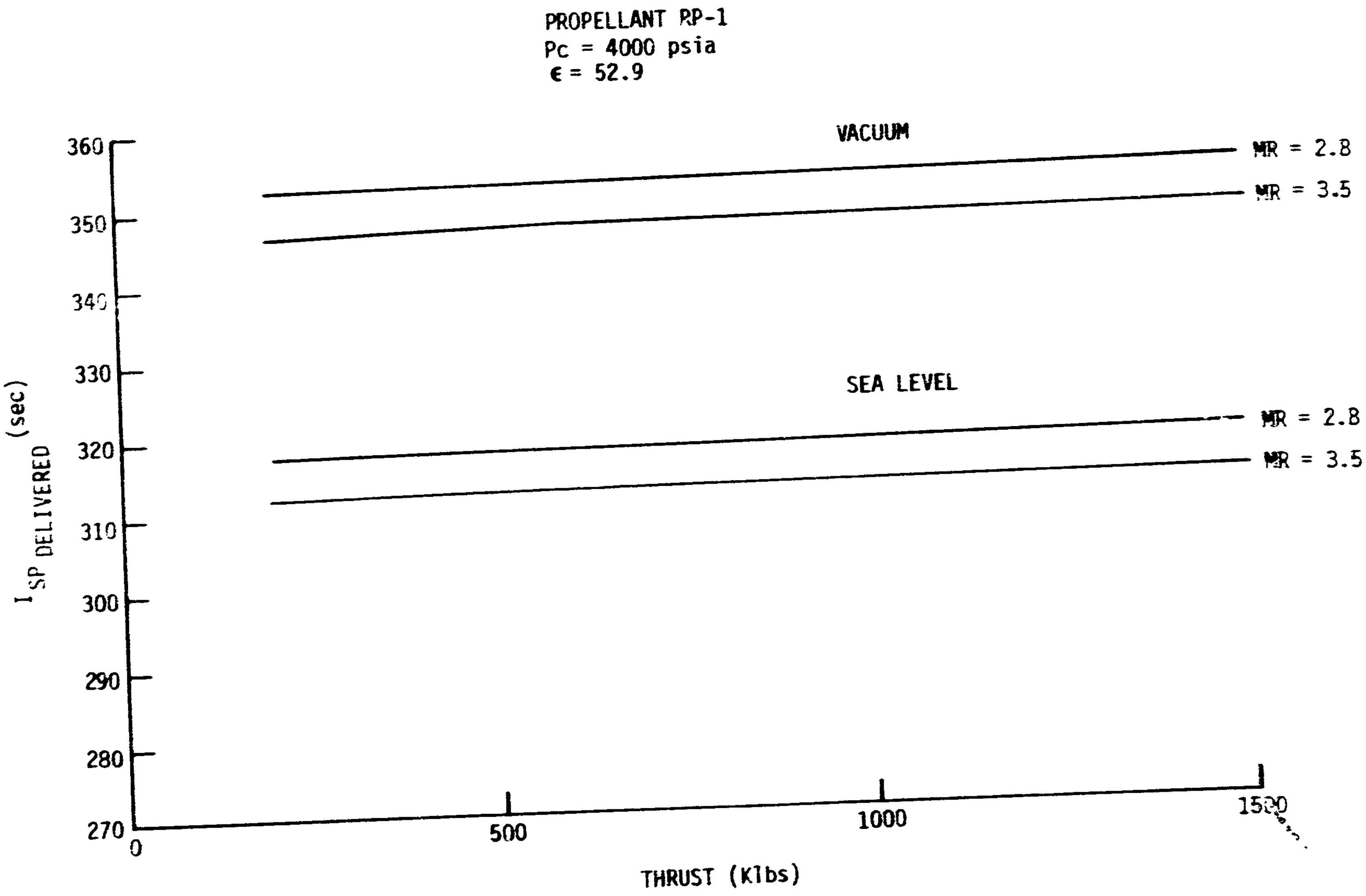

Figure 4. Delivered LOX/RP-1 Engine Performance Versus Thrust

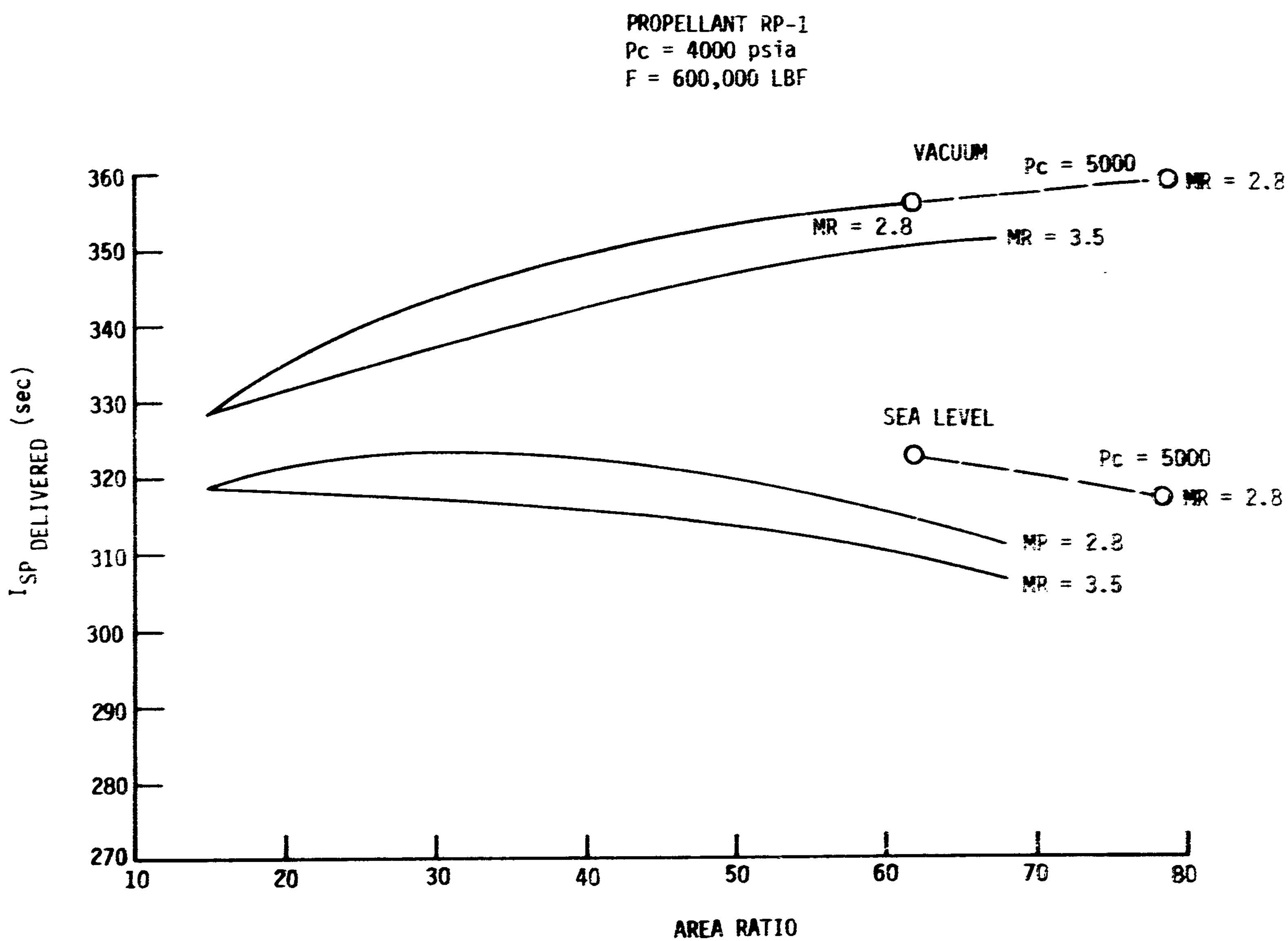

Figure 5. Delivered LOX/RP-1 Engine Performance Versus Area Ratio

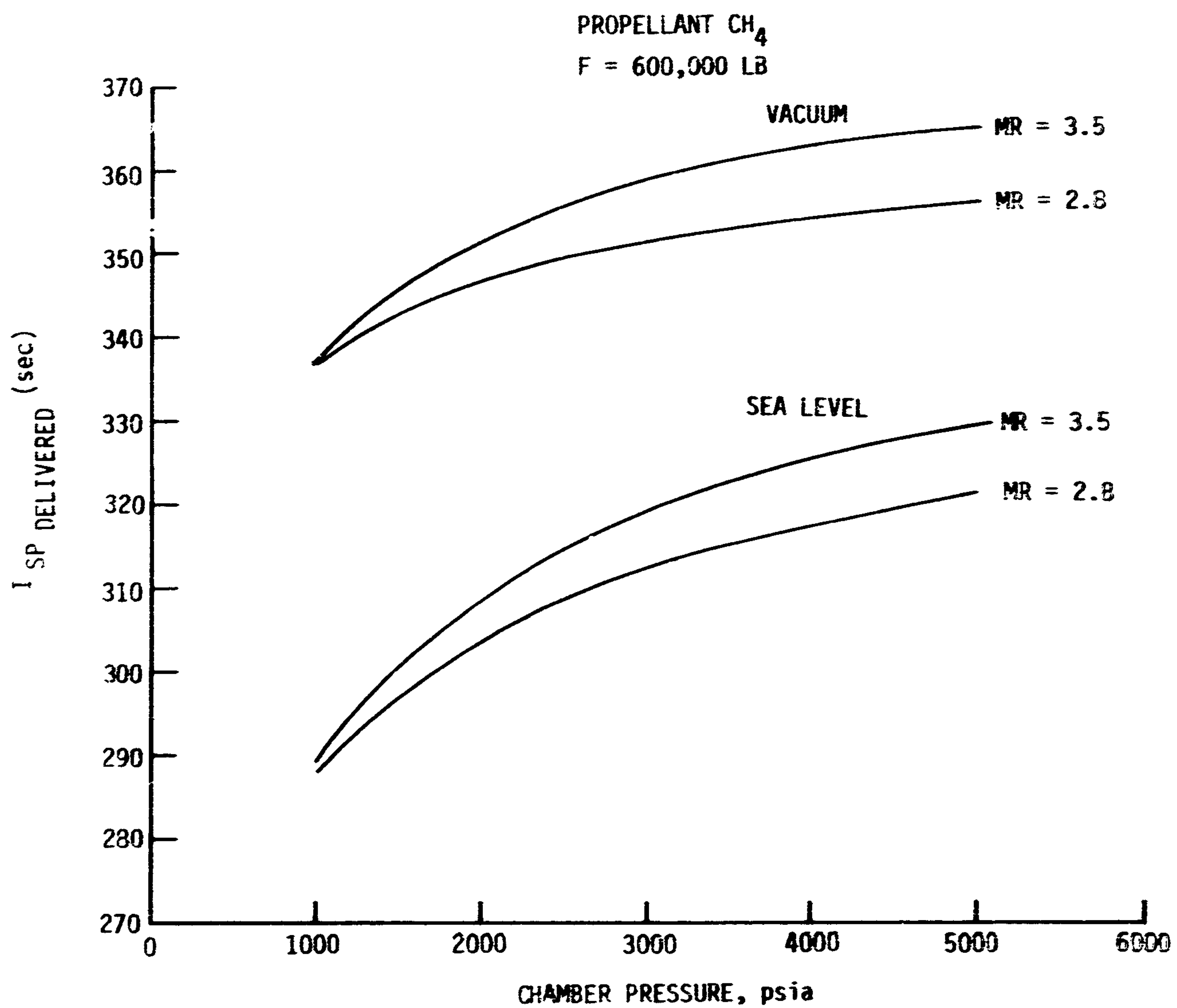

Figure 6. Delivered LOX/CH_4 Engine Performance Versus Chamber Pressure

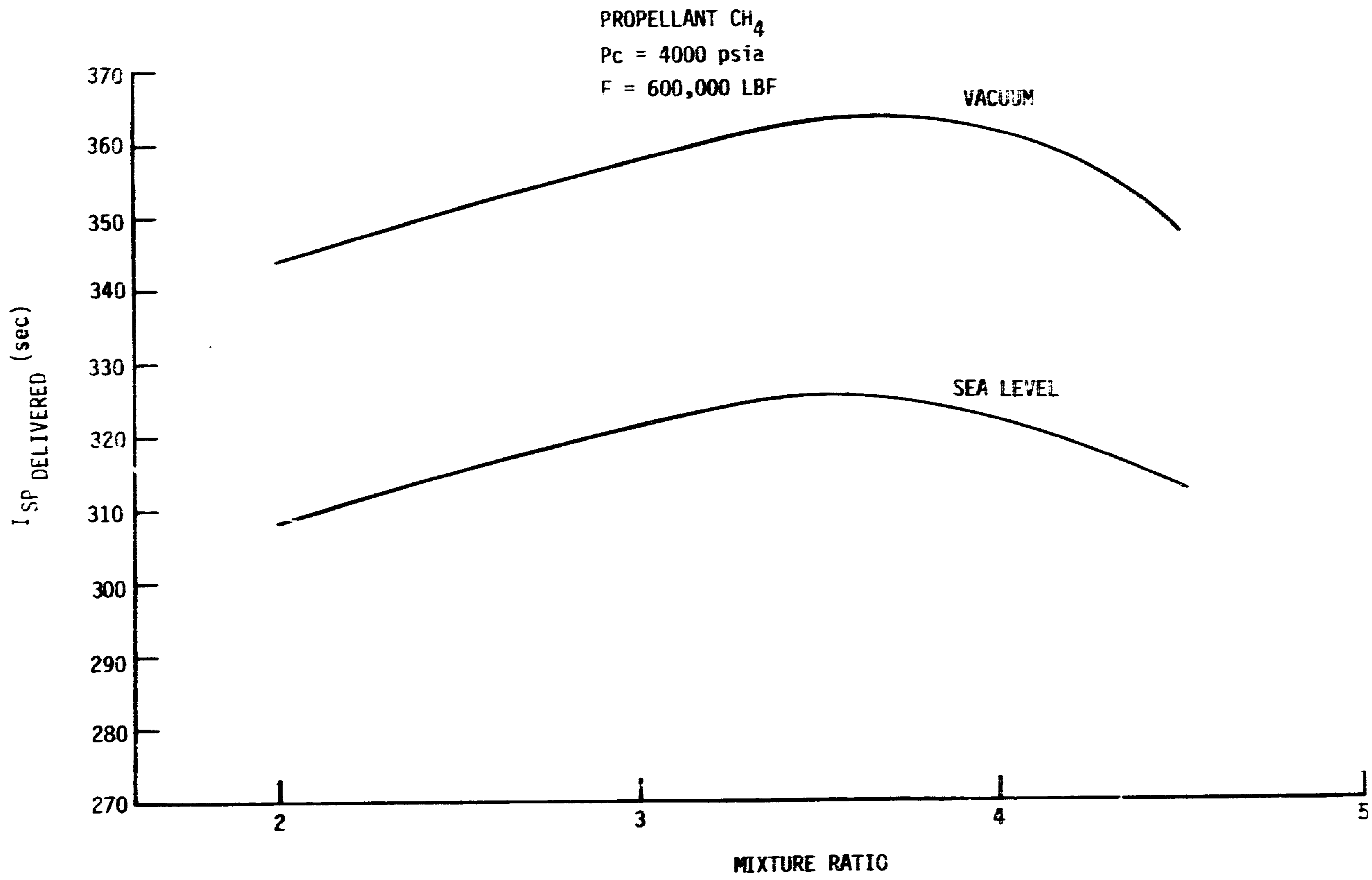

Figure 7. Delivered LOX/CH_4 Engine Performance Versus Mixture Ratio

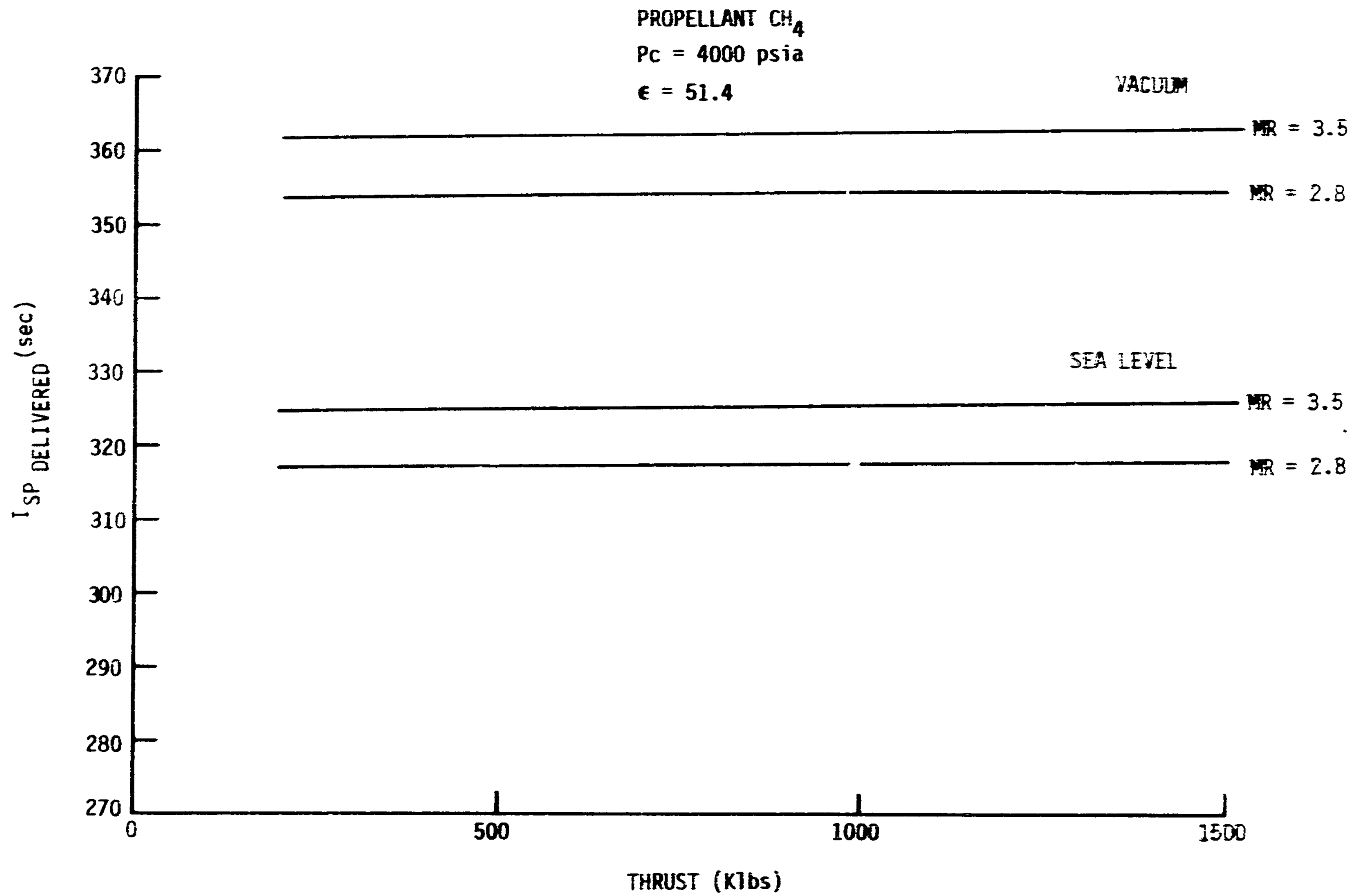

Figure 8. Delivered LOX/CH_4 Engine Performance Versus Thrust

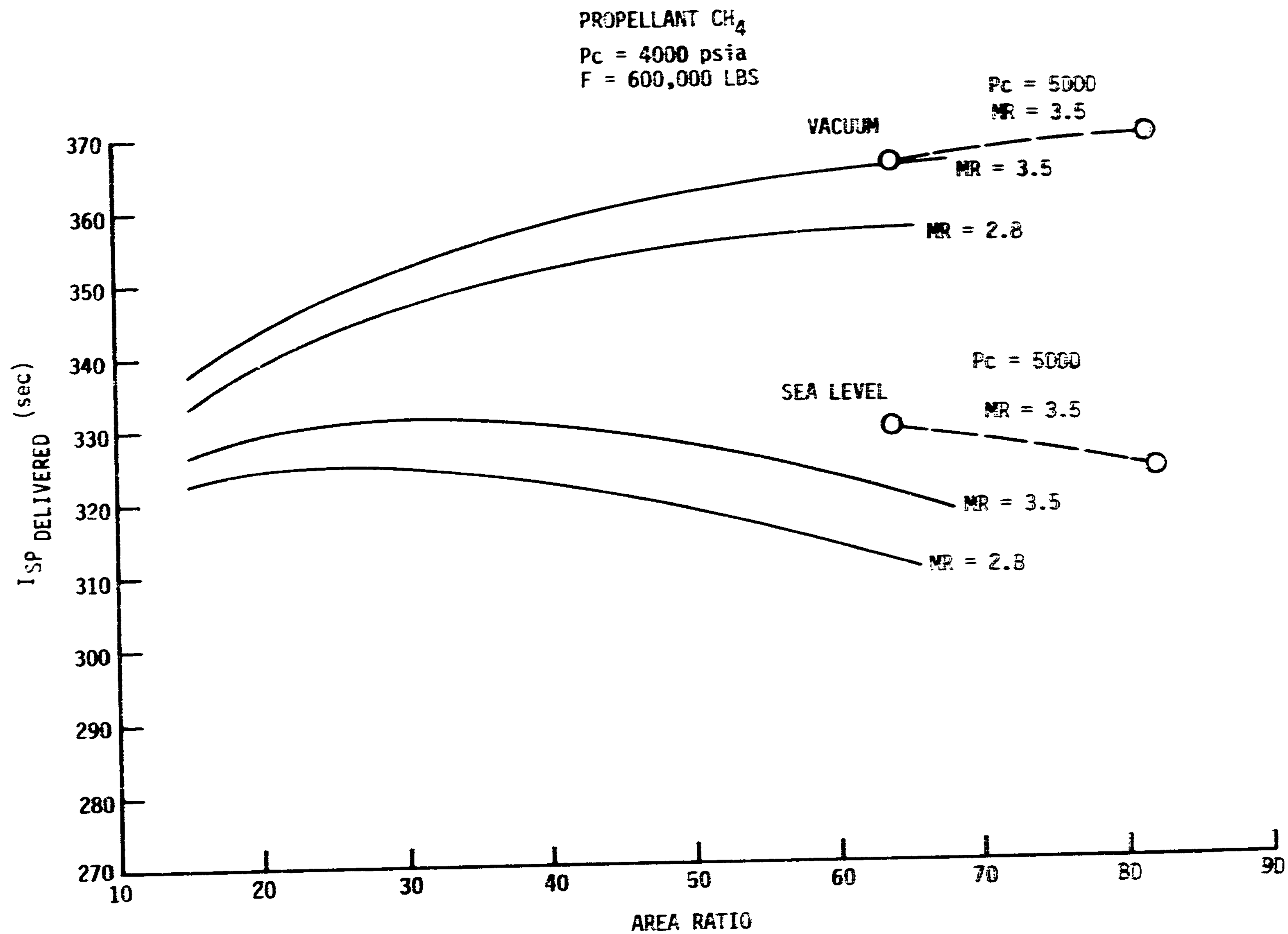

Figure 9. Delivered LOX/CH_4 Engine Performance Versus Area Ratio

ADVANCED OXYGEN HYDROCARBON BOOSTER ENGINE WEIGHT AND ENVELOPE

THRUST	(LBF)	[illegible]	(NEWTONS)	266[illegible]0.00
CHAMBER PRESSURE	(PSIA)	4000.00	(NEWTONS/M**2)	27580000.00
AREA RATIO			53.30	

PROPELLANT COMBINATION 1
[illegible]

COOLANT TYPE 1
ICOOL=1 RP-1 COOLED
ICOOL=2 [illegible] COOLED
ICOOL=3 [illegible] COOLED
ICOOL=4 CH4 COOLED

CYCLE TYPE 1
ICYCLE=1 FUEL RICH GAS GENERATOR (RP-1 OR CH4)
ICYCLE=2 FUEL RICH GAS GENERATOR (LNG)
ICYCLE=3 FUEL RICH STAGED COMBUSTION
ICYCLE=4 OXIDIZER RICH STAGED COMBUSTION
ICYCLE=5 EXPANDER CYCLE
ICYCLE=6 MIXED - OX RICH-FUEL RICH PREBURNERS
ICYCLE=7 STAGED COMBUSTION - 1 PREBURNER
ICYCLE=8 STAGED COMBUSTION - [illegible] PREBURNERS
ICYCLE=9 STAGED COMBUSTION - [illegible] PREBURNERS
ICYCLE=10 STAGED COMBUSTION [illegible]

	(LB)	(KG)
GIMBAL	[illegible]	93.9
INJECTOR	[illegible]	297.0
COMBUSTION CHAMBER	[illegible]	102.4
THRUST CHAMBER NOZZLE	[illegible]	190.4
CONTROLLER	[illegible]	89.8
OX RICH PREBURNER	.0	.0
FUEL RICH PREBURNER	[illegible]	22.7
OX VALVE	[illegible]	147.4
FUEL VALVE	[illegible]	37.2
FUEL BOOST PUMP	[illegible]	22.5
OXIDIZER BOOST PUMP	307.0	139.3
FUEL MAIN PUMP	[illegible]	166.0
OX MAIN PUMP	[illegible]	282.6
LOW PRESSURE LINE	[illegible]	91.7
HIGH PRESSURE LINE	[illegible]	121.5
IGNITER	[illegible]	27.2
HOT GAS MANIFOLD	[illegible]	93.9
MISCELLANEOUS WEIGHTS	[illegible]	134.3
PRESSURE SYSTEM	[illegible]	60.7
TOTAL ENGINE WEIGHT	[illegible]	2090.5

ENGINE THRUST TO WEIGHT RATIO 130.2

	(INCHES)	(METERS)
NOZZLE LENGTH	[illegible]	2.7
GIMBAL LENGTH	[illegible]	.3
CHAMBER LENGTH	[illegible]	.2
INJECTOR LENGTH	[illegible]	.2
IGNITER LENGTH	4.0	.1
TOTAL ENGINE LENGTH	[illegible]	3.5
NOZZLE EXIT DIAMETER	[illegible]	1.9

Figure 10. Typical AOHCWT Weight and Envelope Computer Program Printout

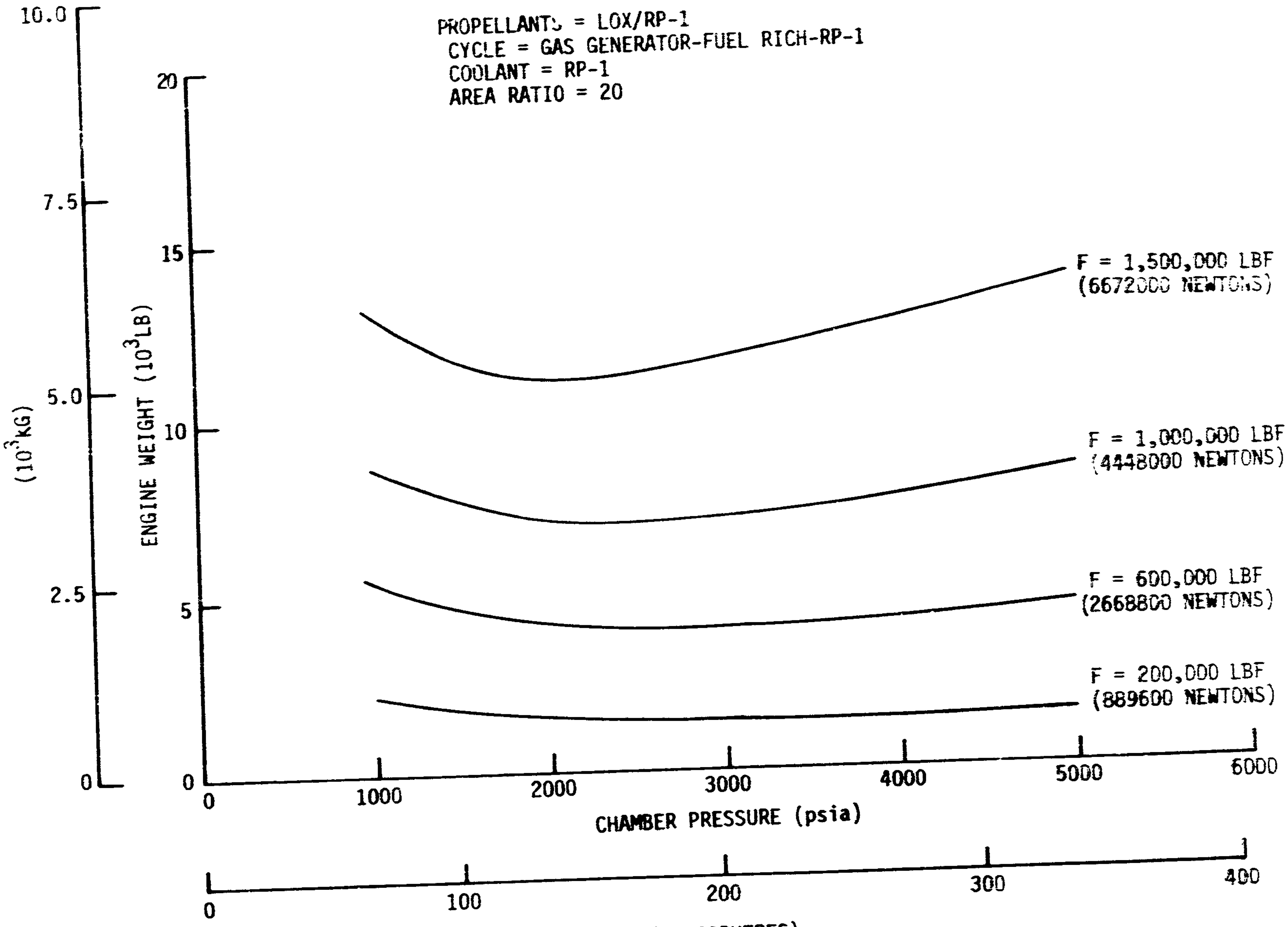

Figure 11. LOX/RP-1 Engine Weight Versus Chamber Pressure (ε = 20)

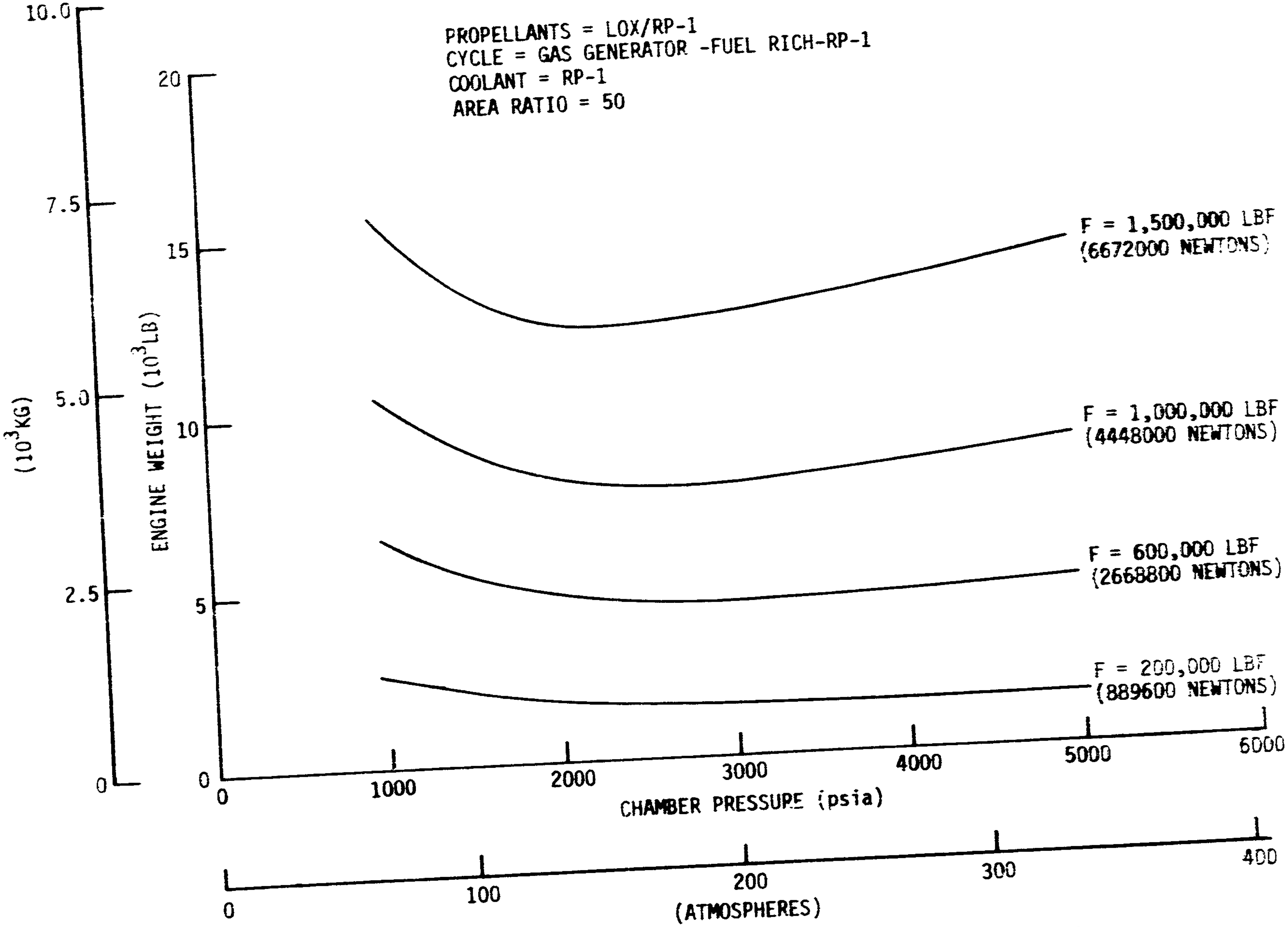

Figure 12. LOX/RP-1 Engine Weight Versus Chamber Pressure (ε = 50)

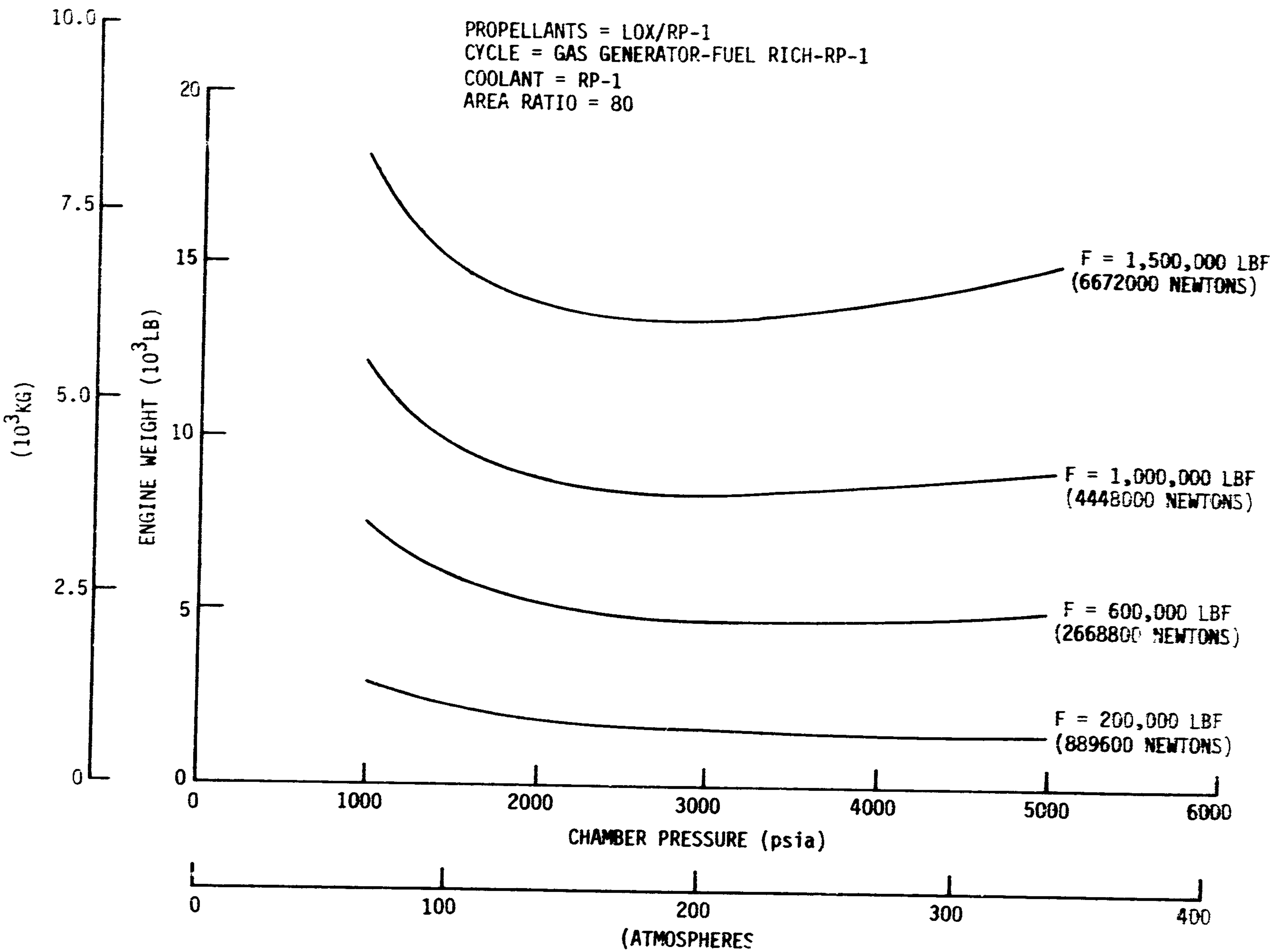

Figure 13. LOX/RP-1 Engine Weight Versus Chamber Pressure (ε = 80)

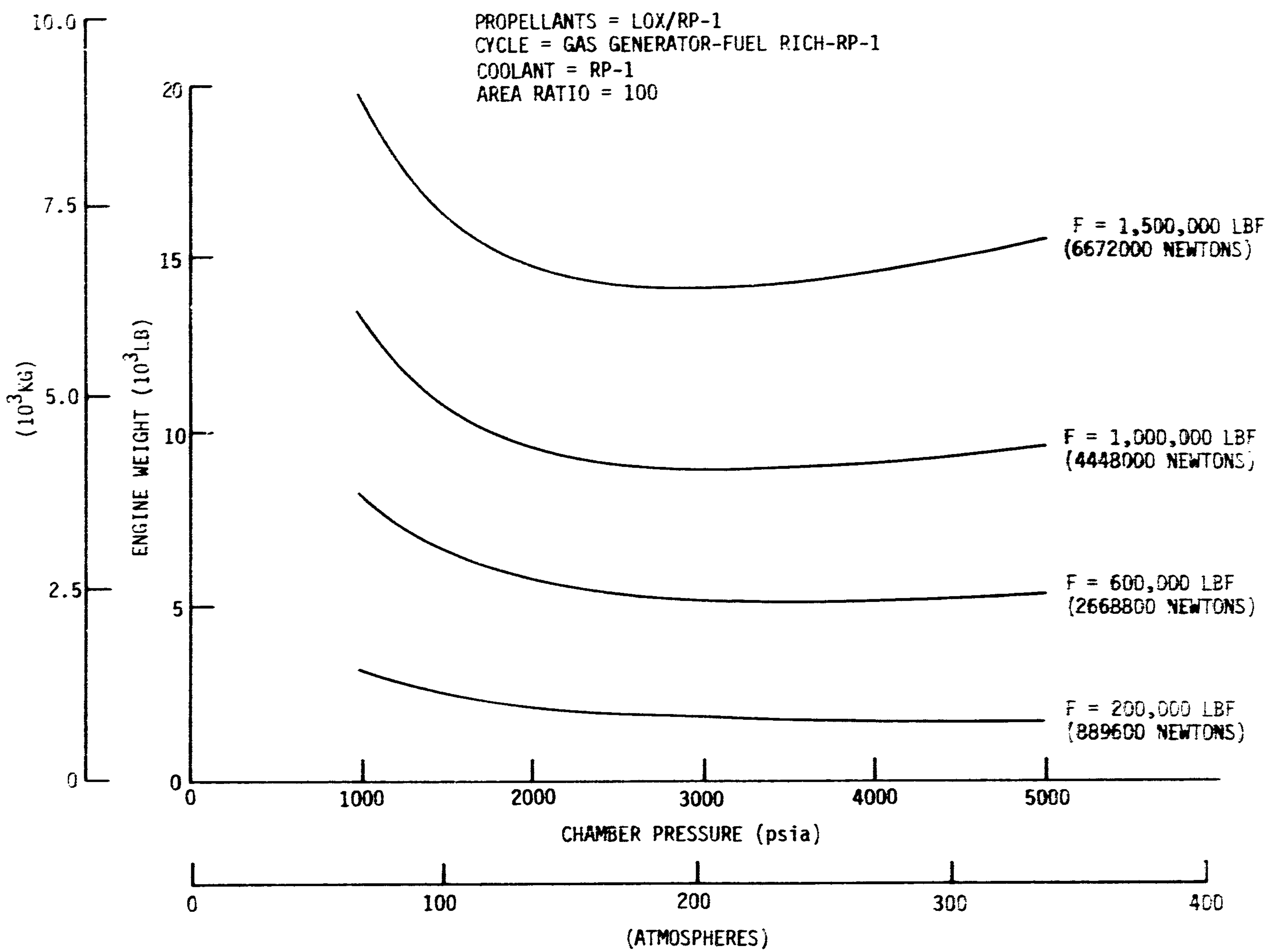

Figure 14. LOX/RP-1 Engine Weight Versus Chamber Pressure (ε = 100)

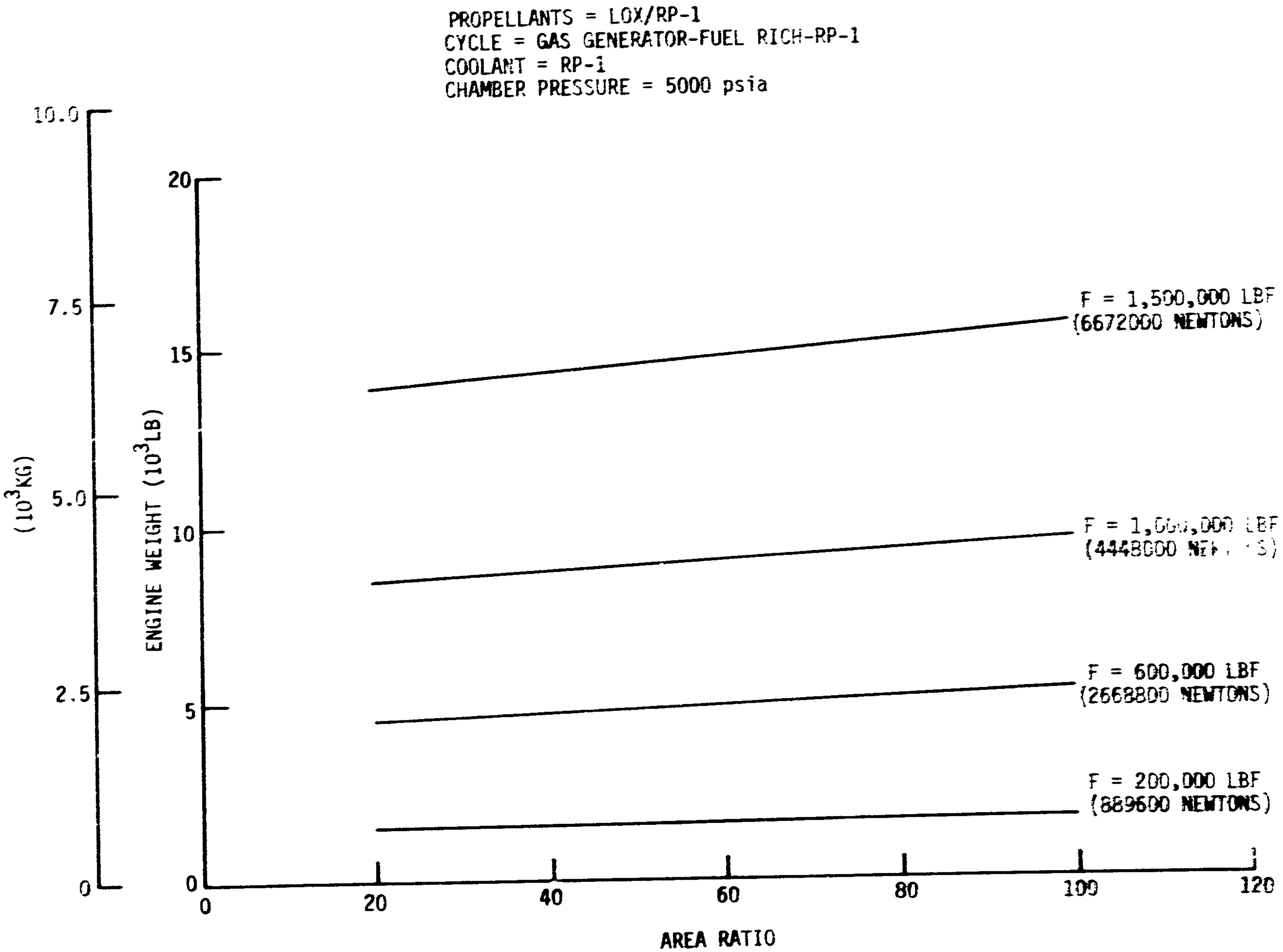

Figure 15. LOX/RP-1 Engine Weight Versus Area Ratio (Pc = 5000)

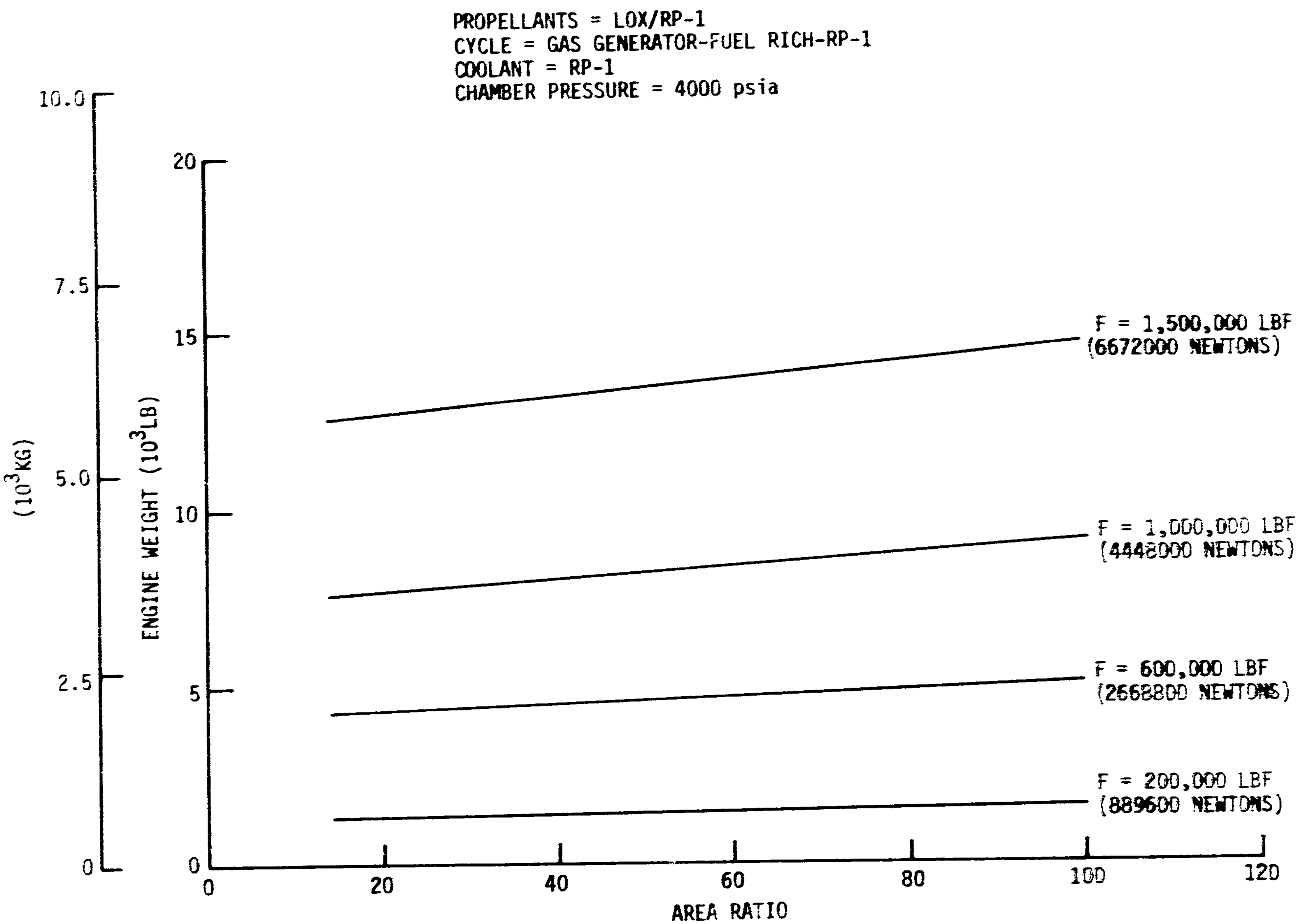

Figure 16. LOX/RP-1 Engine Weight Versus Area Ratio (Pc = 4000)

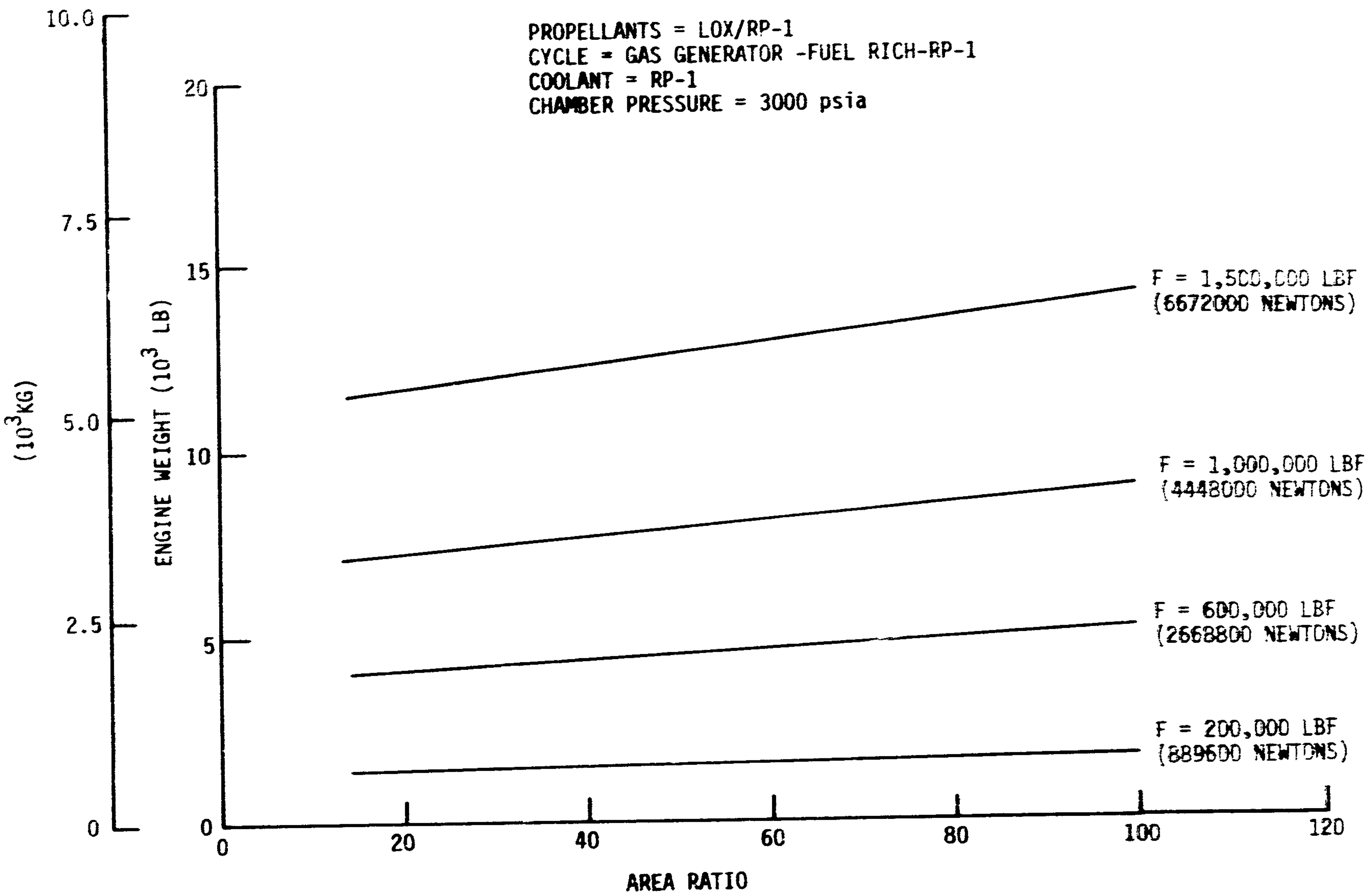

Figure 17. LOX/RP-1 Engine Weight Versus Area Ratio (Pc - 3000)

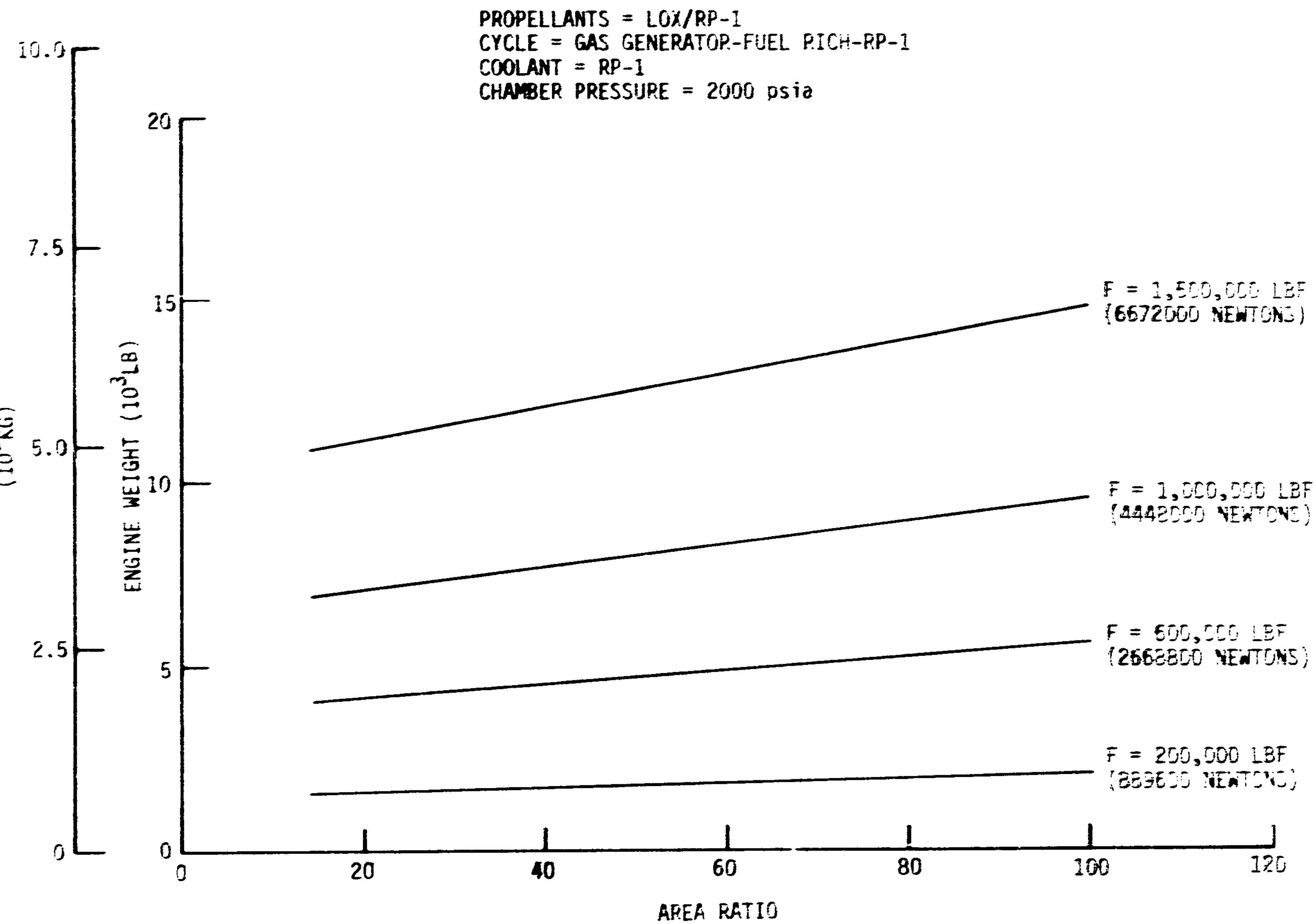

Figure 18. LOX/RP-1 Engine Weight Versus Area Ratio (Pc = 2000)

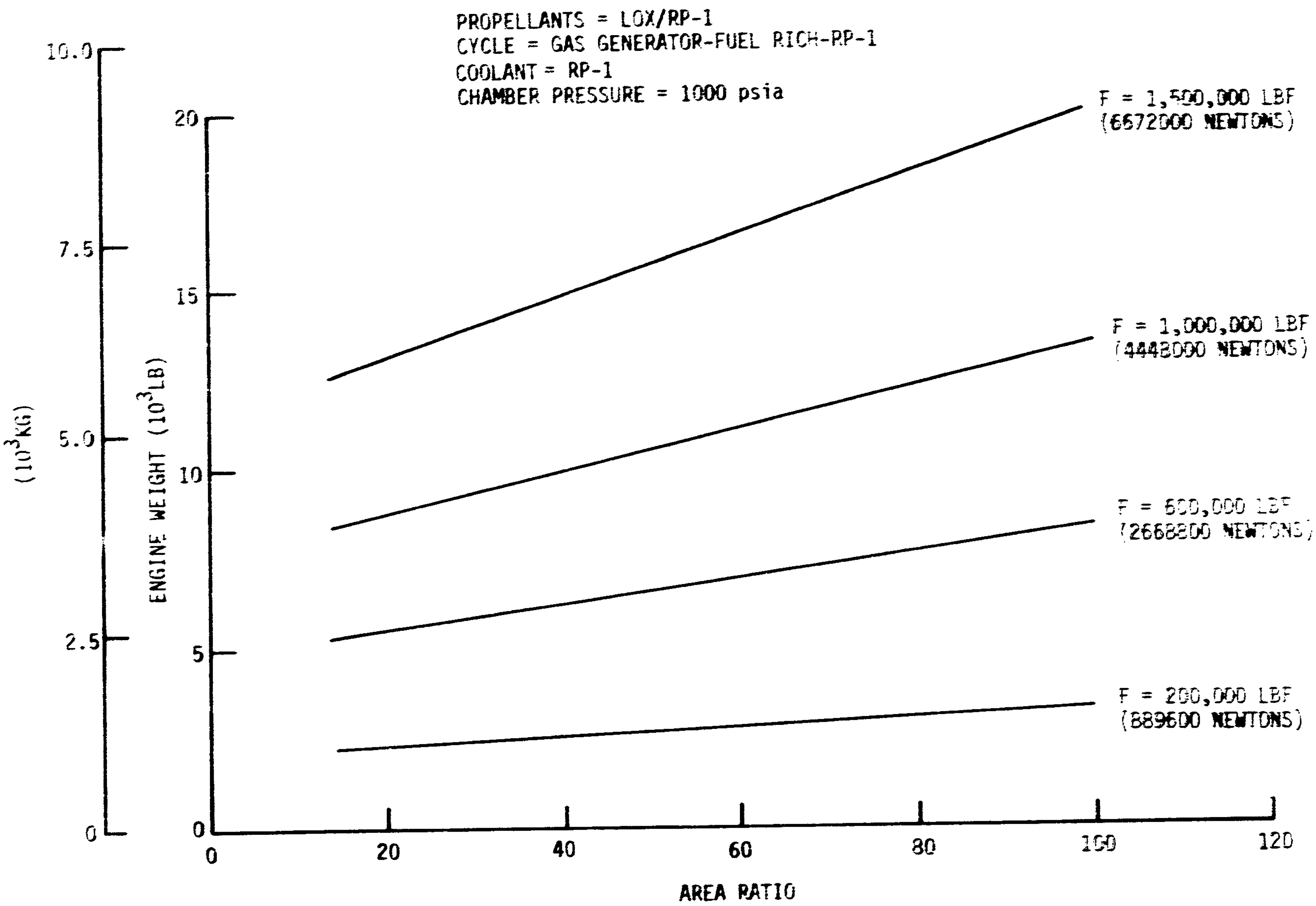

Figure 19. LOX/RP-1 Engine Weight Versus Area Ratio (Pc = 1000)

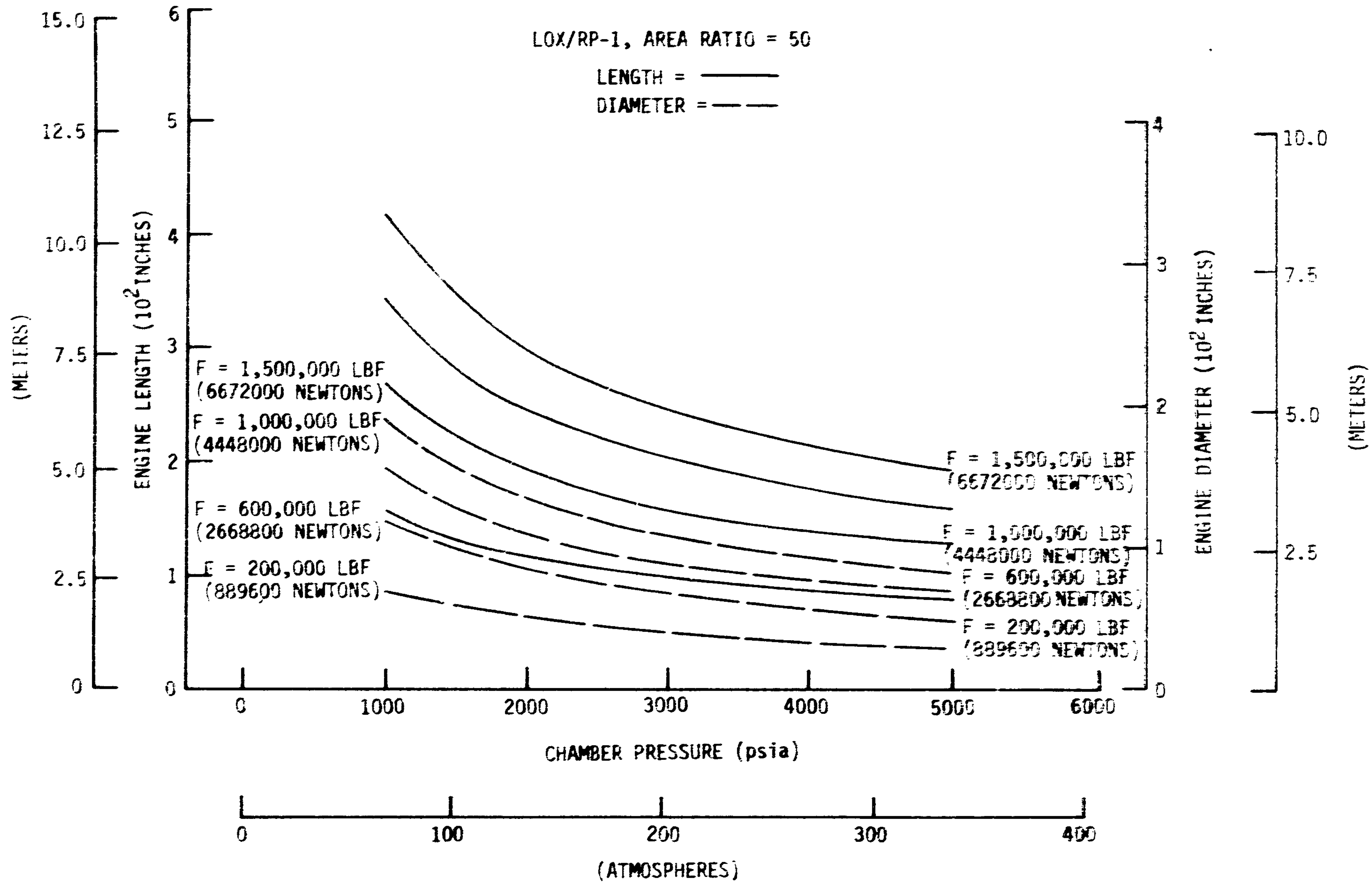

Figure 20. LOX/RP-1 Engine Envelope Parameters Versus Chamber Pressure

Baseline engine weight and envelope data for the LO_2/LCH_4 gas generator, fuel-rich gas generator, LCH_4 cooled cycle are presented in Figures 21 through 30.

Parametric envelope information, as generated by the AOHCWT computer program, is dependent only on thrust, area ratio, and propellant combination. These data are presented graphically in Figures 20 and 30 for the baseline area ratio of 50:1, but other area ratio envelope data are included in Tables VIII and IX.

C. TASK III - ENGINE/VEHICLE TRAJECTORY PERFORMANCE ASSESSMENT

1. Mission Characteristics

The characteristics data presented in Bimonthly Progress Report 33452M-2 were approved by the NASA Project Manager.

2. Trajectory Performance Models

This task was delayed until definitive engine data were generated in Task I.

D. TASK IV - BASELINE ENGINE SYSTEM DEFINITION

No scheduled activity.

E. TASK V - REPORTING

An informal program review was held 7 March with the NASA Project Manager. It was concluded from the discussion that such items as turbine

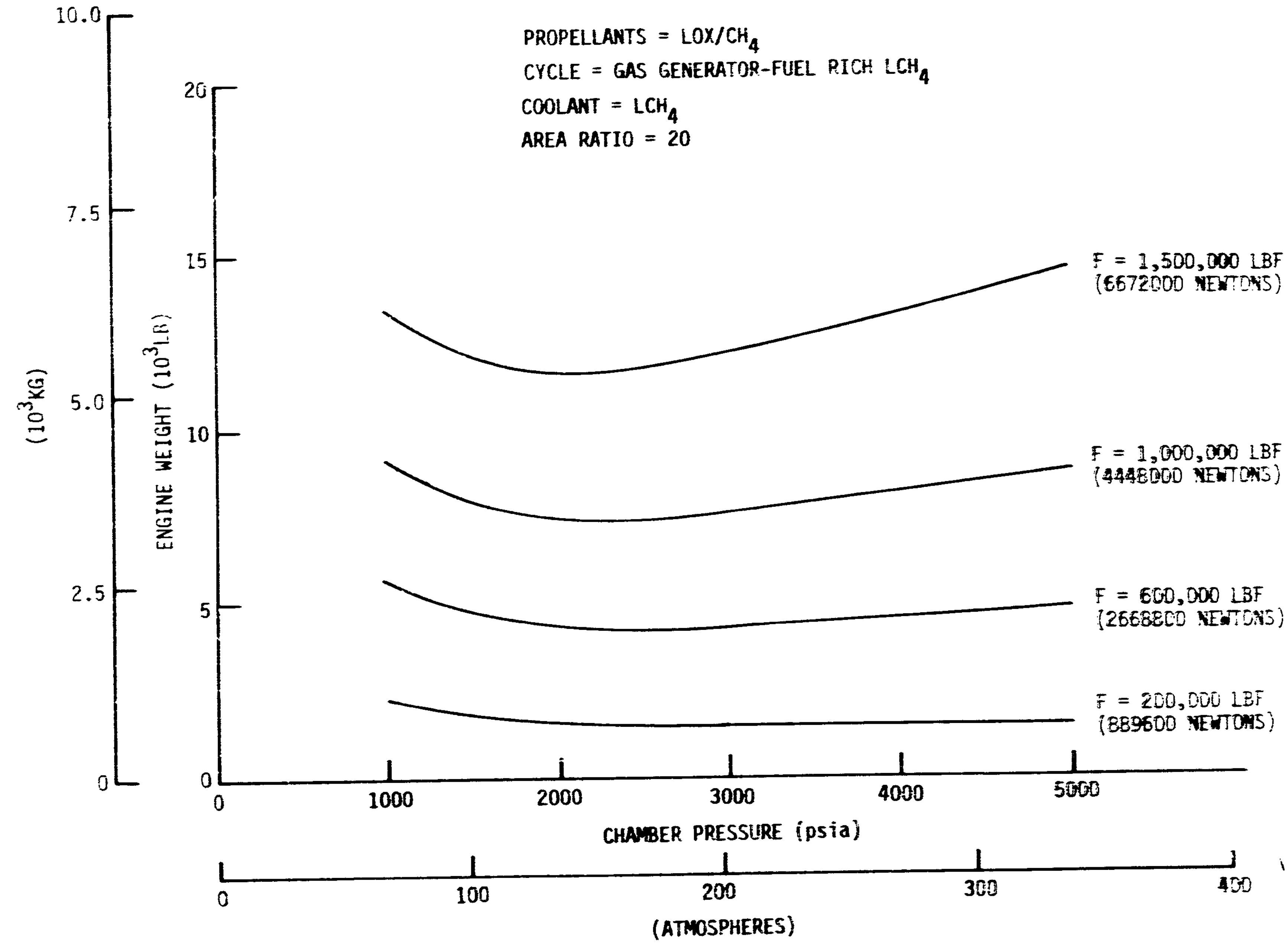

Figure 21. LOX/CH_4 Engine Weight Versus Chamber Pressure (ε = 20)

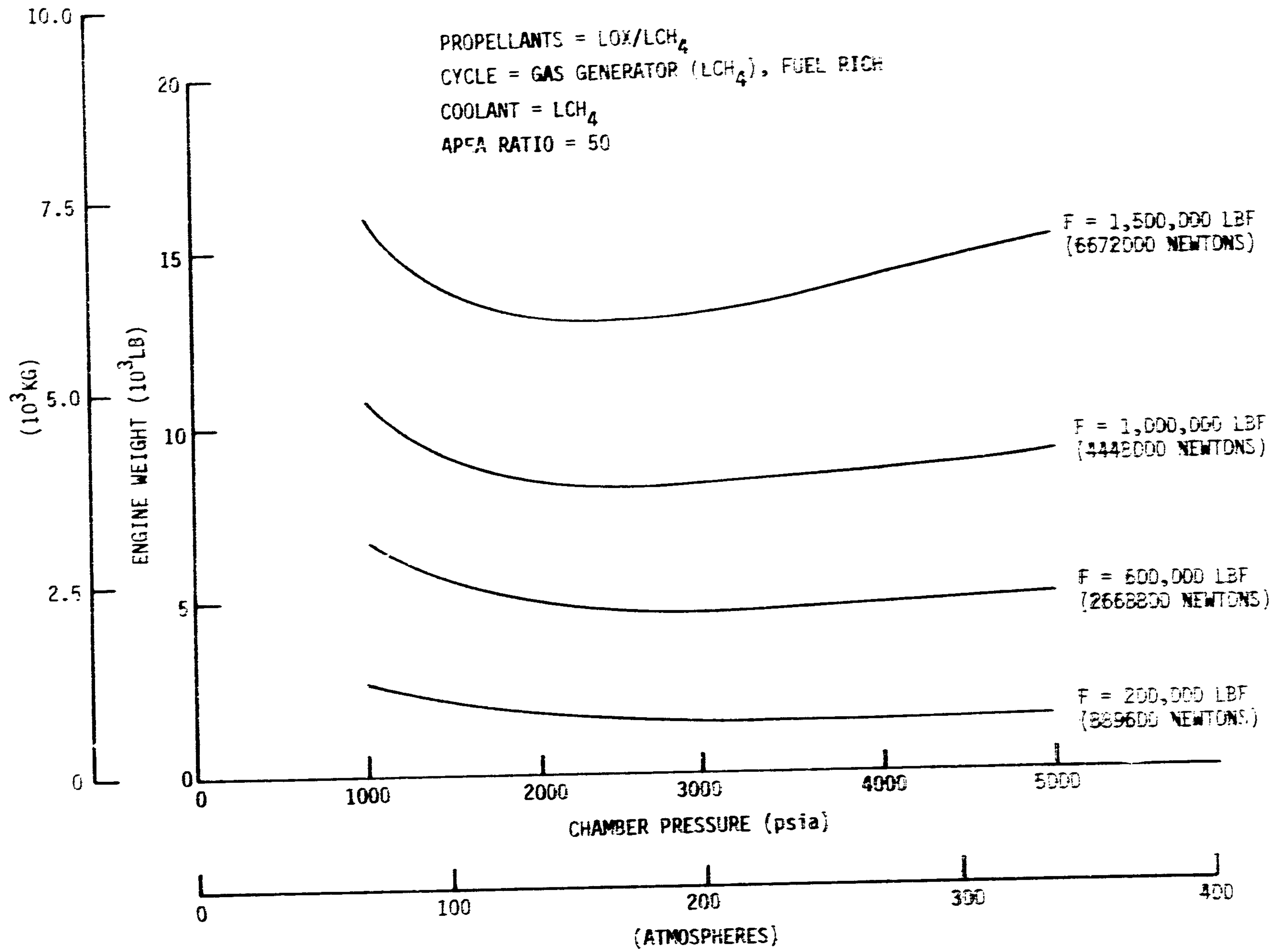

Figure 22. LOX/CH_4 Engine Weight Versus Chamber Pressure (ε = 50)

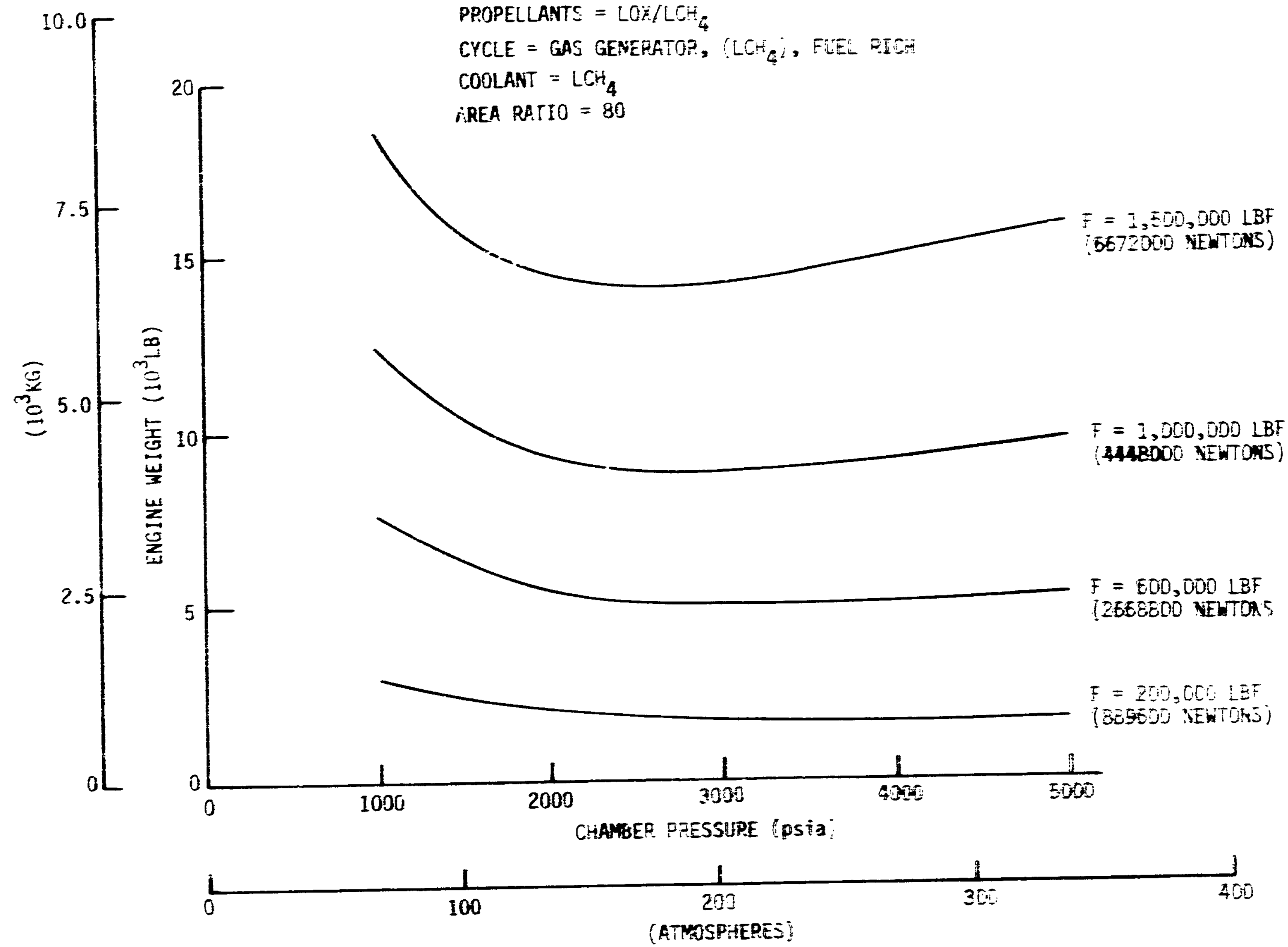

Figure 23. LOX/CH_4 Engine Weight Versus Chamber Pressure (ϵ = 80)

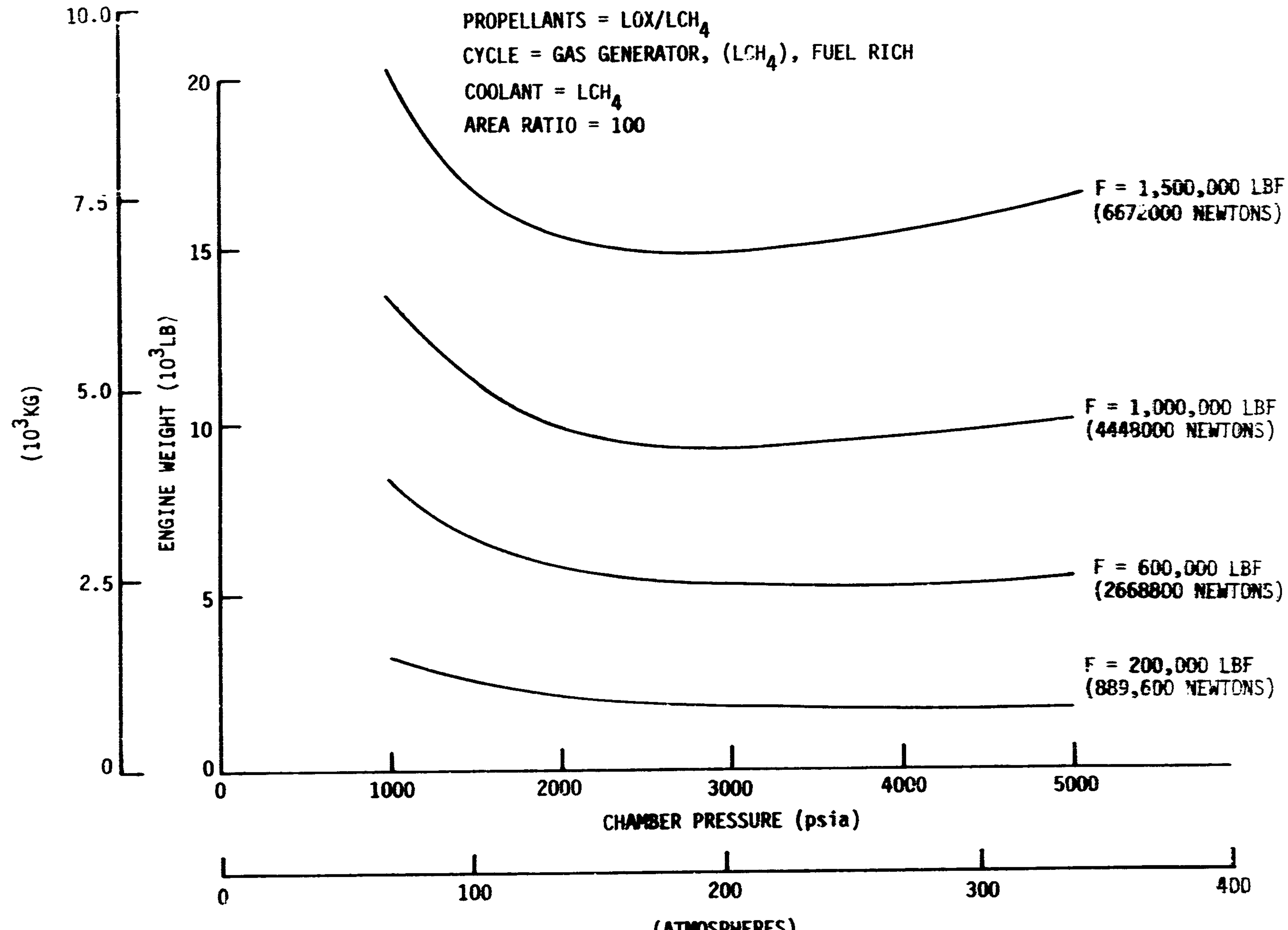

Figure 24. LOX/CH_4 Engine Weight Versus Chamber Pressure (ε = 100)

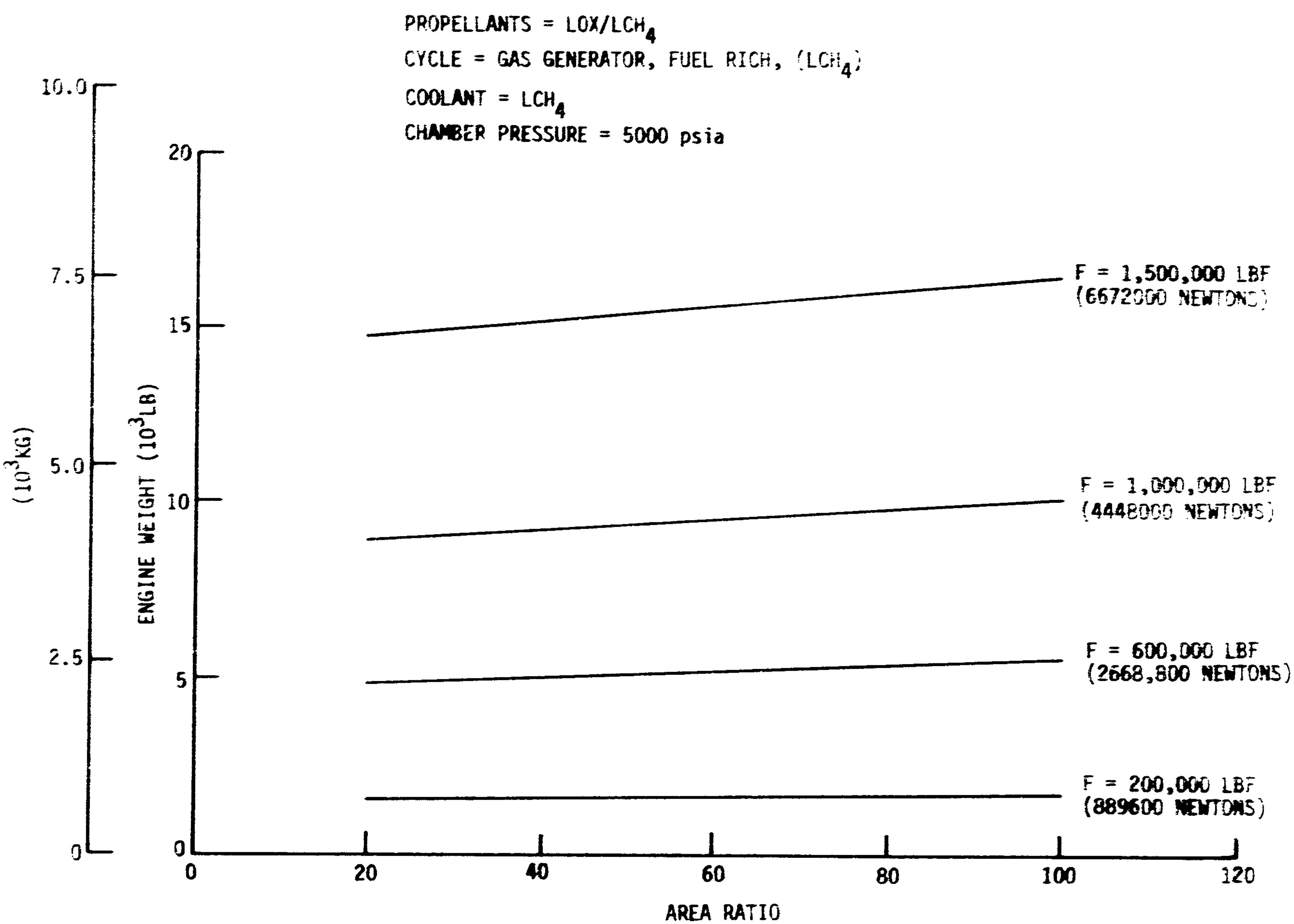

Figure 25. LOX/CH_4 Engine Weight Versus Area Ratio (Pc = 5000)

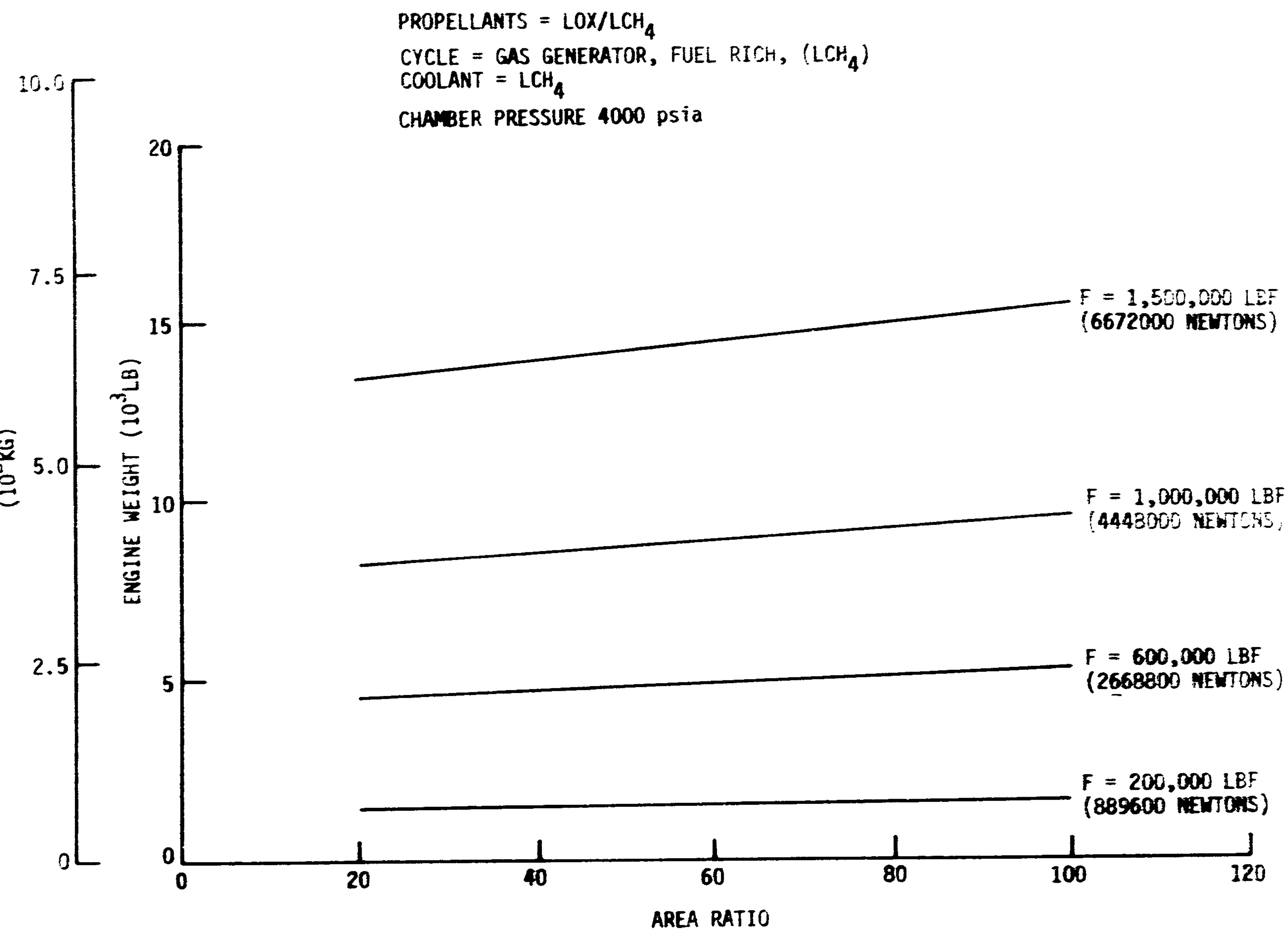

Figure 26. LOX/CH$_4$ Engine Weight Versus Area Ratio (Pc = 4000)

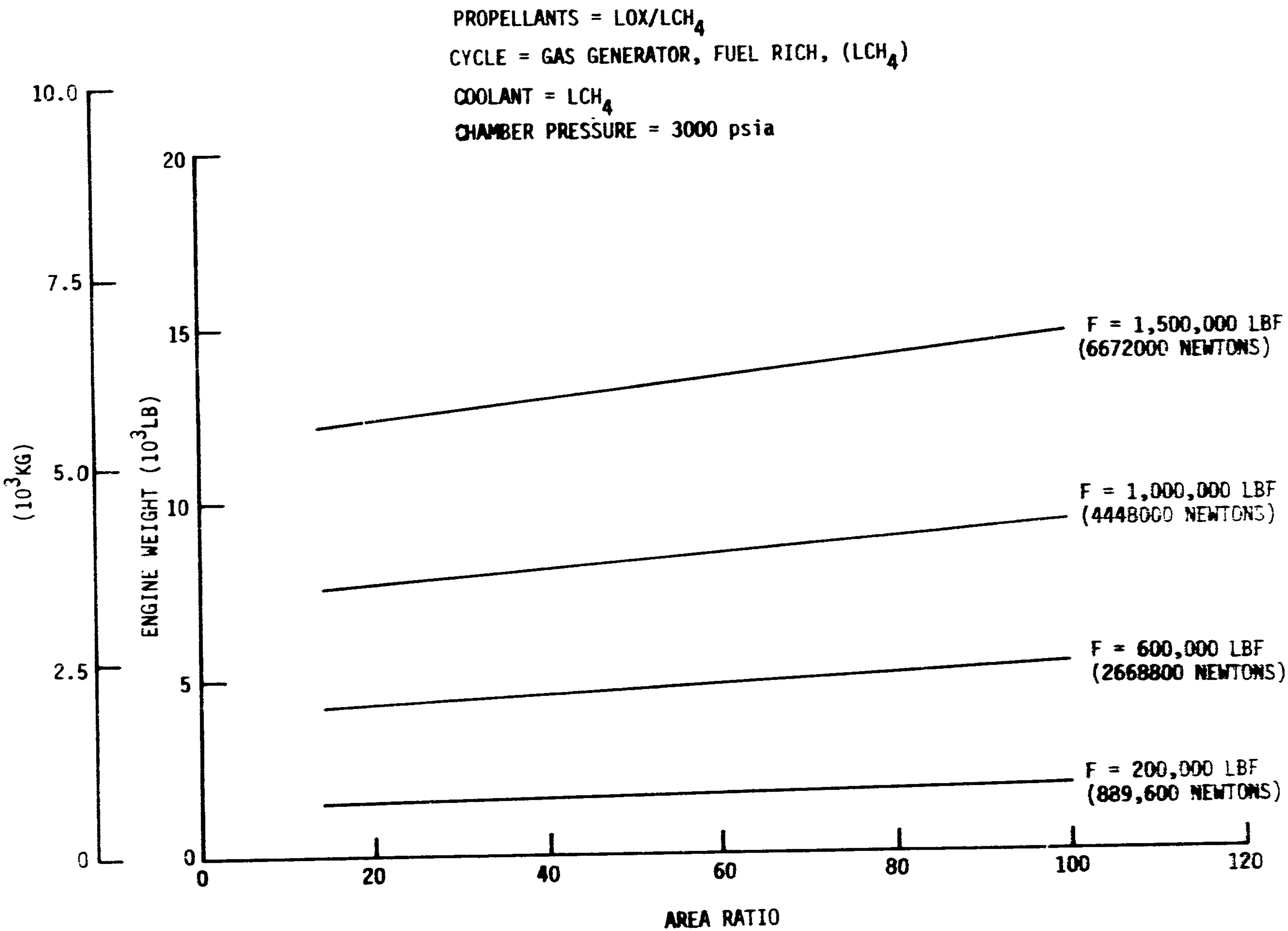

Figure 27. LOX/CH_4 Engine Weight Versus Area Ratio (Pc = 3000)

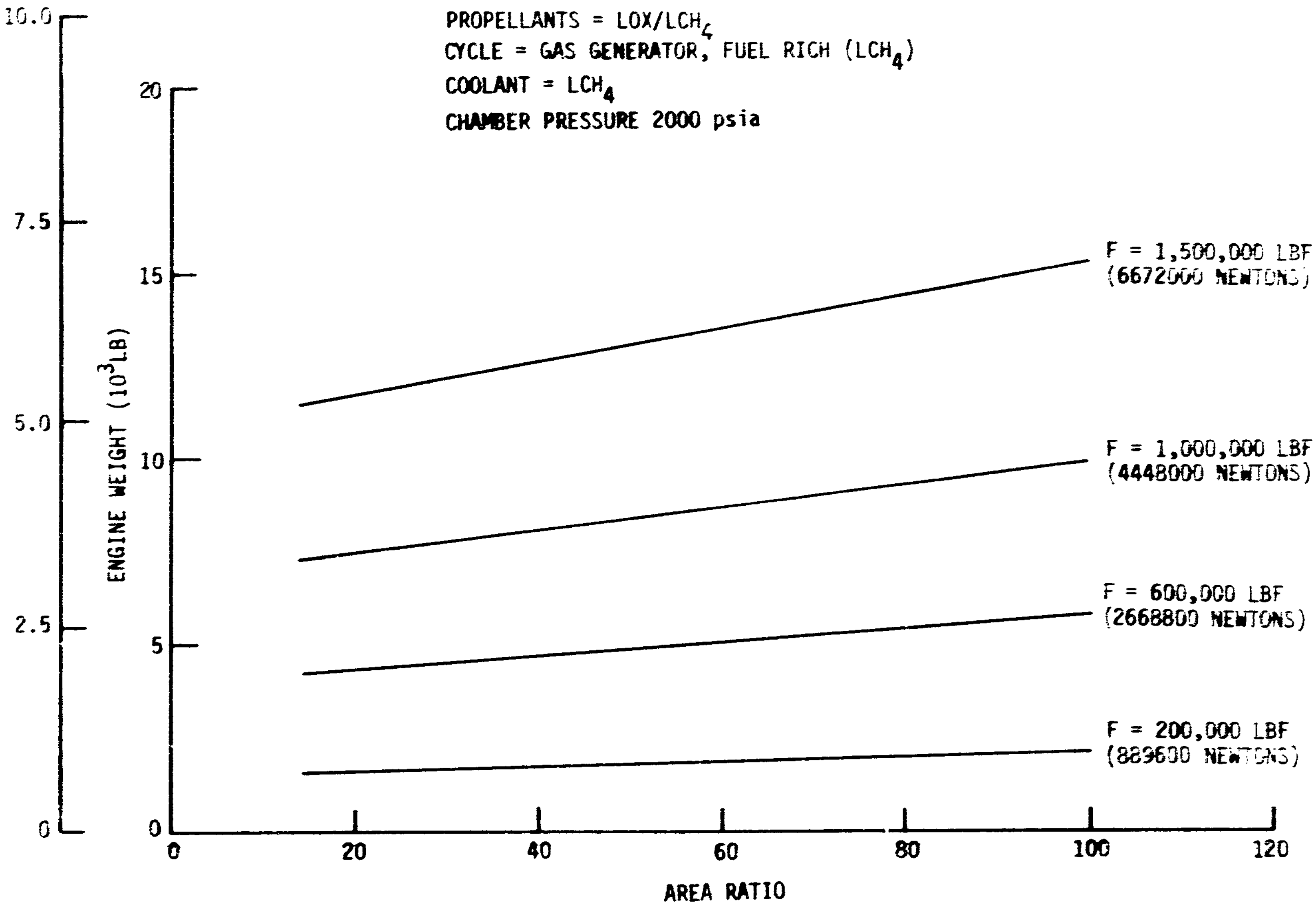

Figure 28. LOX/CH_4 Engine Weight Versus Area Ratio (Pc = 2000)

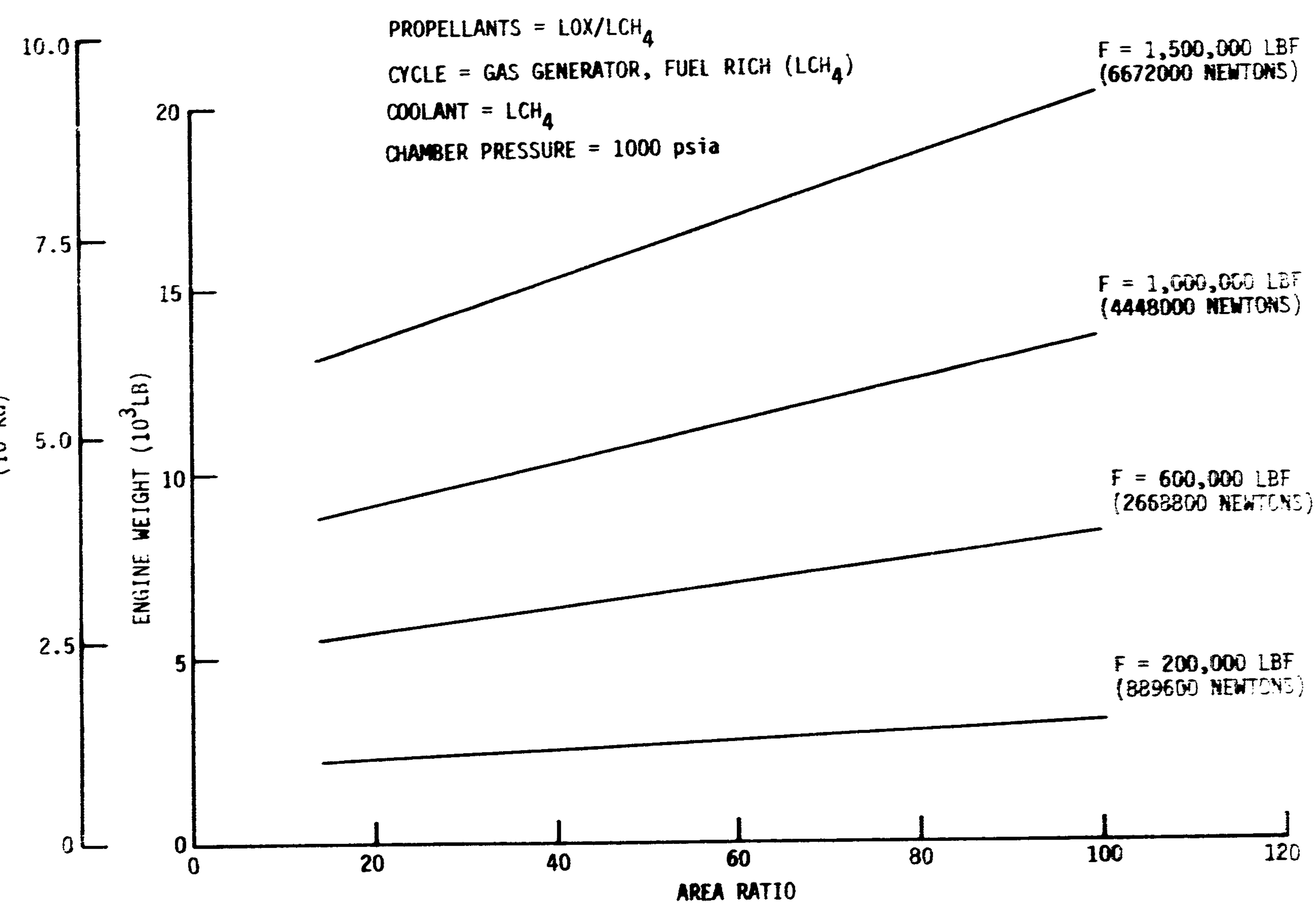

Figure 29. LOX/CH_4 Engine Weight Versus Area Ratio (Pc = 1000)

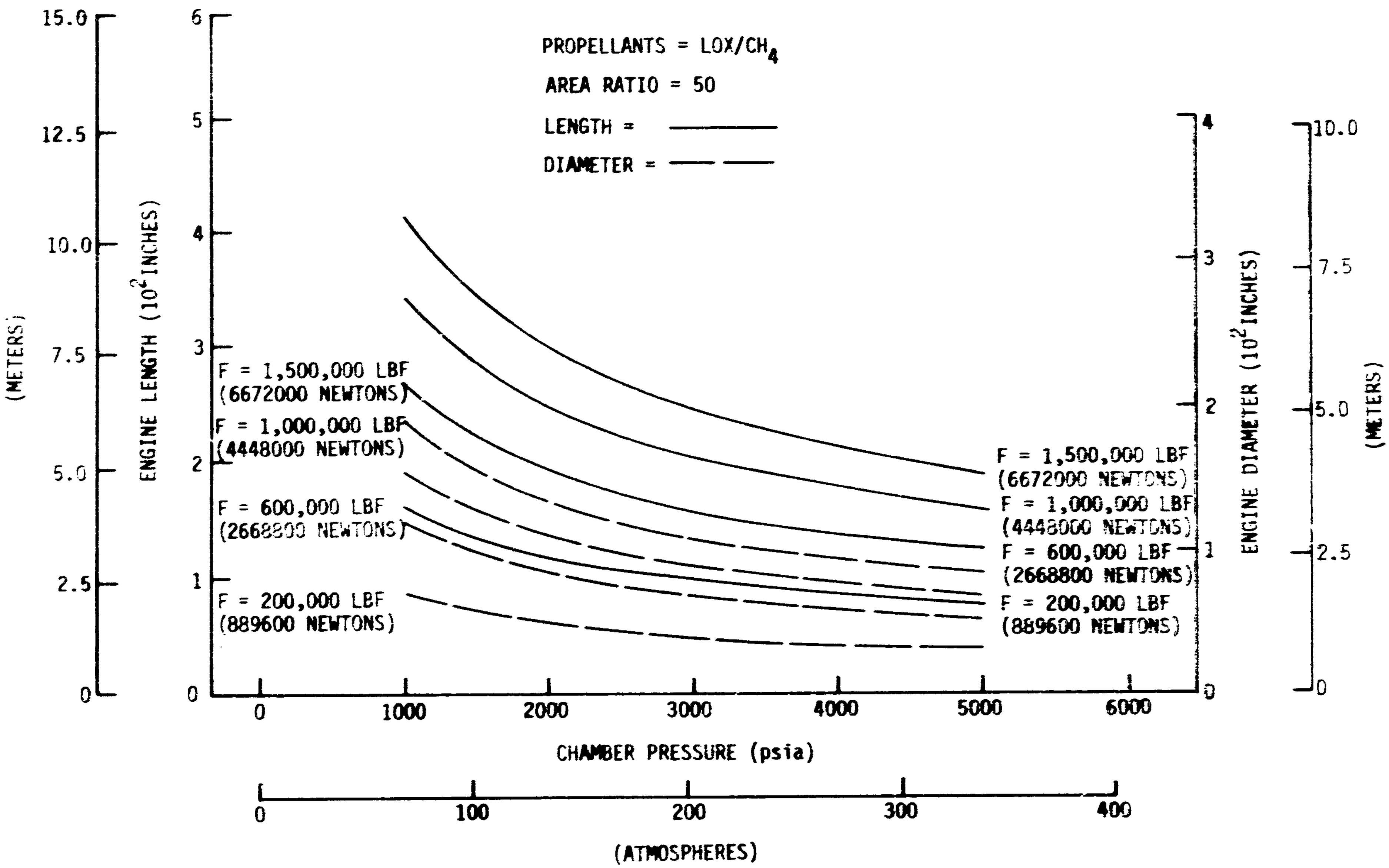

Figure 30. LOX/CH_4 Engine Envelope Parameters Versus Chamber Pressure

TABLE VII

LOX/HDF BASELINE ENGINE WEIGHT BREAKDOWN

	LOX/RP-1		LOX/CH_4	
	STAGED COMBUSTION	GAS GENERATOR	STAGED COMBUSTION	GAS GENERATOR
F_B (Thrust, lb)	600,000	600,000	600,000	600,000
P_{C_B} (Chamber Pressure, psia)	4000	4000	4000	4000
ϵ_B (Area Ratio)	50:1	50:1	50:1	50:1
ϵ_{ATTB} (Attached Area Ratio)	8:1	8:1	8:1	8:1
A_{TB} (Throat Area, in.^{2})	85.66	85.66	86.14	86.14
(All Weights in lbs)				
WGB (Gimbal	207	207	207	207
WMISCB (Miscellaneous)	296	296	296	296
WINJB (Injector)	656	656	656	656
WTCNB (Nozzle)	420	420	422	422
WCCB (Thrust Chamber)	226	226	227	227
WPBOB (Ox Rich Preburner)	224	-	224	-
WPBFB (Fuel Rich Preburner)	181	50	181	51
WVOB (Oxidizer Valves & Actuators)	325	325	331	331
WVFB (Fuel Valves & Actuators)	82	82	131	131
WBPOB (Oxidizer Boost Pump)	307	307	313	313
WBPFB (Fuel Boost Pump)	52	52	83	83
WMPOB (Main Oxidizer Pump)	862	623	878	638
WMPFB (Main Fuel Pump)	327	366	521	567
WLPLB (Low Pressure Lines)	201	201	243	243
WHPLB (High Pressure Lines)	268	268	324	324
WPSSB (Pressurization System)	133	133	133	133
WHGMB (Hot Gas Manifold)	207	207	207	207
WIGNB (Igniters)	60	60	60	60
WCNTRB (Controller)	130	130	130	130
TOTAL	5164	4609	5567	5019

TABLE VIII

LOX/RP-1 ENGINE ENVELOPE

F, lbf (Newtons)	Pc, psia (atm)	ε = 20 L, in. (meters)	ε = 20 D, in. (meters)	ε = 80 L, in. (meters)	ε = 80 D, in. (meters)	ε = 100 L, in. (meters)	ε = 100 D, in. (meters)
200,000	1000 (68.03)	107.2 (2.7)	53.9 (1.4)	197.8 (5.0)	107.9 (2.7)	219.2 (5.6)	120.6 (3.1)
(889600)	2000 (136.05)	79.6 (2.0)	38.1 (1.0)	143.6 (3.6)	76.3 (1.9)	158.7 (4.0)	85.3 (2.2)
	3000 (204.08)	67.2 (1.7)	31.1 (.8)	119.5 (3.0)	62.3 (1.6)	131.8 (3.3)	69.6 (1.8)
	4000 (272.11)	59.9 (1.5)	27.0 (.7)	105.2 (2.7)	53.9 (1.4)	115.9 (2.9)	60.3 (1.5)
	5000 (340.14)	54.6 (1.4)	24.1 (.6)	95.1 (2.4)	48.2 (1.2)	104.7 (2.7)	53.9 (1.4)
1,000,000	1000 (68.03)	223.2 (5.7)	120.6 (3.1)	425.8(10.8)	241.2 (6.1)	473.6 (12.0)	269.6 (6.8)
(4448000)	2000 (136.05)	161.6 (4.1)	85.3 (2.2)	304.8 (7.7)	170.5 (4.3)	338.6 (8.6)	190.7 (4.8)
	3000 (204.08)	134.2 (3.4)	69.6 (1.8)	251.1 (6.4)	139.2 (3.5)	278.7 (7.1)	155.7 (4.0)
	4000 (272.11)	117.9 (3.0)	60.3 (1.5)	219.2 (5.6)	120.6 (3.1)	243.1 (6.2)	134.8 (3.4)
	5000 (340.14)	106.5 (2.7)	53.9 (1.4)	197.0 (5.0)	107.9 (2.7)	218.4 (5.5)	120.6 (3.1)
1,500,000	1000 (68.03)	270.4 (6.9)	147.7 (3.8)	518.5(13.2)	295.4 (7.5)	577.0 (14.7)	330.3 (8.4)
	2000 (136.05)	194.9 (5.0)	104.4 (2.7)	370.3 (9.4)	208.9 (5.3)	411.7 (10.4)	233.5 (5.9)
	3000 (204.08)	161.4 (4.1)	85.3 (2.2)	304.6 (7.7)	170.5 (4.3)	338.4 (8.6)	190.7 (4.8)
	4000 (272.11)	141.5 (3.6)	73.8 (1.9)	265.5 (6.7)	147.7 (3.8)	294.8 (7.5)	165.1 (4.2)
	5000 (304.14)	127.6 (3.2)	66.1 (1.7)	238.5 (6.1)	132.1 (3.4)	264.7 (6.7)	147.7 (3.8)

TABLE IX

LOX/LCH$_4$ ENGINE ENVELOPE

F, lbf (Newtons)	Pc, psia (atm)	ε = 20 L, in. (meters)	ε = 20 D, in. (meters)	ε = 80 L, in. (meters)	ε = 80 D, in. (meters)	ε = 100 L, in. (meters)	ε = 100 D, in. (meters)
200,000 (889600)	1000 (68.03)	107.4 (2.7)	54.1 (1.4)	198.3 (5.0)	108.2 (2.7)	219.3 (5.6)	120.9 (3.1)
	2000 (136.05)	79.7 (2.0)	38.2 (1.0)	144.0 (3.7)	76.5 (1.9)	159.1 (4.0)	85.5 (2.2)
	3000 (204.08)	67.3 (1.7)	31.2 (.8)	119.8 (3.0)	62.4 (1.6)	132.2 (3.4)	69.8 (1.8)
	4000 (272.11)	60.1 (1.5)	27.0 (.7)	105.5 (2.7)	54.1 (1.4)	116.2 (3.0)	60.5 (1.5)
	5000 (340.14)	54.7 (1.4)	24.2 (.6)	95.3 (2.4)	48.4 (1.2)	104.9 (2.7)	54.1 (1.4)
1,000,000 (4448000)	1000 (68.03)	223.8 (5.7)	120.9 (3.1)	426.9 (10.8)	241.9 (6.1)	474.9 (12.1)	270.4 (6.9)
	2000 (136.05)	162.0 (4.1)	85.5 (2.2)	305.6 (7.8)	171.0 (4.3)	339.5 (8.6)	191.2 (4.9)
	3000 (704.08)	134.5 (3.4)	69.8 (1.8)	251.8 (6.4)	139.6 (3.5)	279.4 (7.1)	155.1 (4.0)
	4000 (272.11)	118.2 (3.0)	60.5 (1.5)	219.8 (5.6)	120.9 (3.1)	243.7 (6.2)	135.2 (3.4)
	5000 (340.14)	106.7 (2.7)	54.1 (1.4)	197.6 (5.0)	108.2 (2.7)	219.0 (5.6)	120.9 (3.1)
1,500,000 (6672000)	1000 (68.03)	271.1 (6.9)	148.1 (3.8)	519.9 (13.2)	296.2 (7.5)	572.6 (14.7)	331.2 (8.4)
	2000 (136.05)	195.5 (5.0)	104.7 (2.7)	371.3 (9.4)	209.5 (5.3)	412.9 (10.5)	234.2 (5.9)
	3000 (204.08)	161.8 (4.1)	85.5 (2.2)	305.4 (7.8)	171.0 (4.3)	339.3 (8.6)	191.2 (4.9)
	4000 (272.11)	141.9 (3.6)	74.1 (1.9)	266.2 (6.8)	148.1 (3.8)	295.6 (7.5)	165.6 (4.2)
	5000 (340.14)	127.9 (3.2)	66.2 (1.7)	239.1 (6.1)	132.5 (3.4)	265.4 (6.7)	148.1 (3.8)

inlet temperature for oxidizer rich turbines be re-examined in an effort to establish the technology requirements to make some of the marginal cycles competitive.

III. CURRENT PROBLEMS

Completion of the heat transfer effort six weeks behind schedule is not anticipated to cause a slip in the overall schedule for completion of Task I and the interim program review scheduled for mid-June.

IV. WORK PLANNED

A. TASK I

Complete the engine cycle power balance, define the component design requirements, establish optimum engine operating conditions, and establish an engine cycle rating system. Document the heat transfer results.

B. TASK II

Complete documentation of the work accomplished in this task.

C. TASK III

Set up trajectory performance models for the most promising engine cycles, determine vehicle performance for the various engine cycles, and establish an engine cycle ranking based upon the mission payload results.

D. TASK IV

No scheduled activity.

IV. Work Planned (cont.)

E. TASK V

Prepare for the Task I, II and III program review.