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AN EVALUATION OF COMPOSITE PROPULSION FOR SINGLE-STAGE-TO-ORBIT VEHICLES DESIGNED FOR HORIZONTAL TAKE-OFF

James A. Martin

Langley Research Center
Hampton, Va. 23665

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16. Abstract

Composite propulsion has been analyzed for single-stage-to-orbit vehicles designed for horizontal take-off. Six engines previously studied have been evaluated. Trajectory, geometric, and mass analyses were performed to establish the orbital payload capability. This study augmented other studies of advanced Earth-to-orbit vehicles using rocket propulsion.

The results indicate that none of the engines has performed adequately to deliver payloads to orbit as analyzed. The single-stage turbine and oxidizer-rich gas generator result in a low engine specific impulse, and the performance increment of the ejector subsystem is less than that of a separate rocket system with a high combustion pressure. There is a benefit from incorporating a fan into the engine, and removal of the fan from the airstream during the ramjet mode increases the orbital payload capability.

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James A. Martin
Langley Research Center

SUMMARY

Composite propulsion has been analyzed for single-stage vehicles designed for transportation from Earth to low Earth orbit using horizontal take-off. Six engines previously studied have been evaluated based on assumptions for gross mass, aerodynamics, mission, geometric, and mass relationships. Trajectory, geometric, and mass analyses were performed to establish the orbital payload capability of vehicles using the composite engines. This study augmented other studies of advanced Earth-to-orbit vehicles using rocket propulsion.

The results of this study indicate that none of the engines has performed adequately to deliver payloads to orbit as analyzed. The single-stage turbine and oxidizer-rich gas generator result in a low engine specific impulse, and the performance increment of the ejector subsystem is less than that of a separate rocket system with a high combustion pressure. There is a benefit from incorporating a fan into the engine, and removal of the fan from the airstream during the ramjet mode increases the orbital payload capability.

INTRODUCTION

The National Aeronautics and Space Administration is currently developing the space shuttle, a partially reusable vehicle system for transportation from the surface of the Earth to low Earth orbit. Advanced vehicles are now being studied to evaluate technology requirements for vehicles which might augment or replace the space shuttle in the future (refs. 1 to 5); in most of these advanced vehicle studies, only rocket propulsion is considered. Since previous studies (refs. 6 and 7) had shown the favorable potential of composite engines for staged Earth-to-orbit transportation systems, a study was initiated to evaluate the use of composite engines in single-stage vehicles. The term composite engine defines an engine that includes both air-breathing and rocket components.

The first part of the overall study of composite engines for single-stage-to-orbit vehicles was the generation of the performance, size, and weight of seven composite engines. Six of these were analyzed in considerable detail, and the results are given in reference 8. The second part of the study, reported herein, consisted of evaluating the potential of these engines for single-stage-to-orbit vehicles. The present analysis involved trajectory, geometric, and mass calculations of vehicle configurations designed for horizontal take-off and landing. In addition, analysis of the engine data is included to determine what engine modifications may be of interest. Another purpose of the study was to determine in which areas of technology improvements are needed or useful for this class of engine.

The trajectory analysis was for the portion of the flight from lift-off to insertion into low Earth orbit. The payload capability was calculated for the six engines and various combinations of operating modes and installed thrust levels for a fixed gross mass. The resulting payload capabilities were used as a primary criterion for evaluating the engine parameters.

SYMBOLS

Values in this report are given in the International System of Units (SI). Some of the calculations were made in the U.S. Customary Units.

b	wing span, m
C_D	drag coefficient, $\frac{\text{Drag force}}{qS_{th}}$
C_L	lift coefficient, $\frac{\text{Lift force}}{qS_{th}}$
C_{L_α}	lift-curve slope, $\frac{\partial C_L}{\partial \alpha}$, per degree
g	reference gravitational acceleration, 9.807 m/s^2
h	body height, m
l	body length, m
m_e	vehicle entry mass, kg
m_H	mass of hydrogen, kg
m_{JP-4}	mass of JP-4, kg
m_l	mass at landing, kg
m_O	mass of oxygen, kg
n	ultimate load factor, maximum acceleration multiplied by margin of safety and divided by g
q	dynamic pressure, Pa

r maximum wing root thickness divided by structural span of exposed wing

S_b body wetted area, m^2

S_e exposed wing area, m^2

S_{th} theoretical total wing planform area, m^2

S_w vehicle wetted area, m^2

T thrust, N

V_b body volume, m^3

α angle of attack, deg

ρ_H density of hydrogen, 70.5 kg/m^3

ρ_{JP-4} density of JP-4, 809 kg/m^3

ρ_O density of oxygen, 1137 kg/m^3

Abbreviations:

ATR air turborocket

ERJ ejector ramjet

STATEMENT OF THE PROBLEM

The problem considered was that of evaluating several composite engines in terms of the payload achievable in single-stage-to-orbit vehicles designed for horizontal take-off and landing. The composite engines were those described in detail in reference 8; a summary of the engine design parameters is given in table 1. Figure 1 illustrates the various components of the engines. The operating modes are named for the components operating. The gas generator is operated only when the fan is being driven. For example, the fan-ramjet mode implies that the gas generator, fan, and ramjet combustor are operating and the ejector (rocket) is not operating.

The mission specifications for which the vehicle was designed are described in table 2 and were selected to be nearly consistent with space shuttle and other current advanced vehicle programs. The maximum dynamic pressure limit, 90 kPa, is considerably higher than typical rocket vehicle dynamic pressure limits, but reducing the limit degrades performance significantly. The maximum acceleration limit, 1.7g, is quite low compared with the space shuttle value of 3.0g, but increasing the limit does not significantly improve the performance.

METHOD OF ANALYSIS

The analysis included examining the calculated engine data, synthesizing a vehicle aerodynamic design, calculating the trajectory performance, sizing, and estimating mass properties. The examination of the engine data involved some comparisons with possible engine modifications and is discussed in subsequent sections.

Aerodynamic Design

The aerodynamic configuration for this investigation is shown in figure 2. The cross section, forebody, wing, and vertical tail were initially selected to be compatible with current advanced rocket vehicle studies. The afterbody was selected as a reasonable compromise between aerodynamics and structural design. After the initial selection, the configuration was modified to achieve hypersonic and subsonic trim and stability. The optimal design integration (ODIN) system of computer programs (ref. 9) was used to evaluate the trim and stability characteristics and to locate the wing longitudinally. The forebody planform and the body camber were modified parametrically to achieve acceptable aerodynamic characteristics.

The lift and drag characteristics of the configuration were estimated from comparisons with other existing data for use in the trajectory analysis. Although there is no experimental basis for the aerodynamics used, the characteristics are believed to be representative for this class of vehicle. Figure 3 shows some of the data used, and all the data are given in table 3. The drag coefficients are given in table 3 for the vehicle design shown in figure 2. For other vehicle designs, the drag was increased approximately in proportion to the vehicle wetted area. The correction factor varied from 1.00 to 1.15.

Trajectory Analysis

For each engine and vehicle combination considered, a trajectory was integrated numerically from lift-off to orbit insertion. The program to optimize simulated trajectories (POST), discussed in reference 10, was used.

A typical trajectory is shown in figure 4. Lift-off occurs at a Mach number of 0.38. The maximum dynamic pressure limit is intersected at a Mach number of 1.8; this condition was selected because for many thrust options the rocket thrust could be reduced at a Mach number of about 1.8. The pitch rate for the early portion of the trajectory was varied to meet the dynamic pressure limit at a Mach number of 1.8. The trajectory was determined by holding the dynamic pressure at its limit until the end of the air-breathing segment.

At the end of the air-breathing segment, a pull-up to lower dynamic pressures was initiated by using one optimized angle-of-attack rate. The remaining portion of the trajectory was controlled with five optimized pitch rates. The engines were assumed to be capable of throttling to meet the acceleration limit of 1.7g. The orbital insertion conditions of altitudes of 50 n. mi., flight-path angle of 0°, final velocity of 7464 m/s, and maximum insertion mass were achieved by varying the engine cut-off time and the six optimization variables.

The installed thrust level of a vehicle was varied by using a noninteger number of engines. The same total thrust would be achieved in an actual vehicle by selecting an appropriate number of engines and sizing the engines to provide the required thrust per engine. The engine characteristics (propellant mass flow and engine mass per unit thrust) are believed to be reasonably independent of thrust level.

The engines of this study have a considerable degree of flexibility in the manner in which they are used. In table 4, the performance data for one of the engines has been reproduced. The operational schedules of fan, ramjet burner, and rocket considered in this study form seven thrust options which are presented in figure 5 as thrust options A to G. For each option, the ramjet burner was used from lift-off to the maximum Mach number for the air-breathing segment, also indicated in figure 5.

The fan was used from lift-off to a Mach number of 2.5 for options A and B; these options were used with engines 1 and 2 which had removable fans. For the engines having fixed fans (engines 3, 4, and 7), the fan was operated up to a Mach number of 3.5. For these engines, the ramjet thrust was still relatively low below a Mach number of 3.5. Options F and G were used with engine 5 which had no fan (ejector ramjet).

The rocket thrust level was varied from no thrust to full thrust by assuming that none, one-third, two-thirds, or all of the engines had the rocket subsystem on. In all cases, the full rocket thrust is used at lift-off and also after the maximum Mach number for the air-breathing segment is reached (until throttling is required). The thrust options chosen ranged from continuous use of all rockets throughout the trajectory to no rocket use during some period. The points along the trajectory at which the rocket thrust level changed were optimized on some preliminary runs which indicated that within a range there was little effect on the payload. Therefore, these points were not optimized for each case.

The wing area used for most of the present study was 700 m^2. This area was selected for two reasons: It provided sufficient lift initially at an angle of attack of 17°, and it avoided the limit on the product of angle of attack and dynamic pressure during the high-dynamic-pressure portion of the flight.

Geometry Analysis

The ODIN system (ref. 9) of computer programs was used to calculate the volume of the input body shape. Then a small program added to the ODIN system was entered to calculate the body length required to house the propellants. The required mass of the propellants was found from the trajectory analysis. The equation for body volume that was satisfied was

$$V_b = V_p + 1.81(m_O/\rho_O + m_H/\rho_H + m_{JP-4}/\rho_{JP-4})$$

where V_p is the volume allowed for the payload and equals 300 m^3. The 1.81 factor increases the propellant volume to include volume for orbit maneuver propellant, reserves, residuals, ullage, tank utilization, and packaging of tanks within the body. The fact that the body volume was proportional to the body length cubed was used to resize the vehicle.

When the proper body length was found, the ODIN programs were used to calculate the geometric properties needed for the mass analysis. For example, the theoretical total wing area of 700 m^2 was input, and the calculations provided exposed wing wetted area, thickness of the root chord, and so forth.

Mass Analysis

The ODIN system computer program WAATS (ref. 11) was used to calculate the vehicle component masses for an assumed payload. The relationships and constants are given in the appendix. Auxiliary landing gear for take-off was assumed to be dropped at lift-off. Then, after the initial execution of WAATS, a small program added to the ODIN system was entered to check the payload assumption. Payload adjustments were then made until the trajectory requirements were met.

RESULTS AND DISCUSSION

The results of this overall study of composite engines for single-stage-to-orbit vehicles fall into four categories. The first category covers the detailed estimated performance and mass data for each engine. The second category covers the results of analyzing the engine data alone. The third category covers the results of analyzing a single vehicle design, and the fourth category covers results which determine the effects of various engine parameters. The effects of various engine parameters (the fourth category) are shown in terms of payload achievable in figures 6 to 11. The last three categories are discussed in the following sections.

Single-Stage Turbine and Oxidizer-Rich Gas Generator

The engines considered, with the exception of engine 5, use a bipropellant gas generator to provide high-pressure gas to drive a turbine which in turn drives the fan (fig. 1). The mixture ratio of the gas generator must be far from stoichiometric to avoid high temperatures which would destroy the turbine blades. An oxidizer-rich mixture ratio for the gas generator and a single-stage turbine with the pressure ratio limited to 3.5 were chosen in the generation of the engine designs.

The combination of a low turbine pressure ratio and an oxidizer-rich gas generator caused the engines to have a low specific impulse in the fan modes. For example, the sea-level static specific impulse of engine 4 in the fan ramjet mode was 517.9 seconds (table 4). A method of increasing the specific impulse by reducing the gas generator flow is to increase the turbine pressure ratio. A turbine pressure ratio of about 14, which might require a three-stage turbine, would increase the sea-level specific impulse to about 1000 seconds in the fan-ramjet mode (ref. 8). An additional method of increasing the specific impulse is to use a fuel-rich gas generator. The fuel-rich gas generator flow could then be burned in the ramjet combustion chamber. A fuel-rich gas generator may not be permissible with the ejector subsystem operating. If it can be used, the oxygen flow in the gas generator could be reduced from 48 kg/s for engine 4 with a turbine pressure ratio of 14 (ref. 8) to about 9 kg/s, the amount actually burned to produce the needed energy. The resulting specific impulse in the fan-ramjet mode would be about 2030 seconds.

The limitations imposed on the gas generator and turbine combination caused the benefit of air-breathing propulsion to be reduced from what it might have been. For this reason, the results tend to favor minimizing the air-breathing thrust. Two engine modifications appear promising. The first modification is increasing the turbine pressure ratio, which decreases the gas generator mass flow requirement at the expense of additional stages. The second is eliminating the ejector subsystem and adding a separate rocket engine integrated into the nozzle, which would allow the gas generator to operate in the fuel-rich mode.

Ejector Rocket Subsystem

The ejector principle is that momentum from the rocket exhaust is transferred to the airflow. Then the airflow can be diffused to a static pressure that is higher than free-stream total pressure, and fuel can be added and burned. Higher combustion pressure in the ramjet combustion chamber results in a more efficient cycle thermodynamically. Another advantage of the ejector concept is that the total expansion ratio for the rocket mode can be large, since the entire engine nozzle is used. The disadvantages of the ejector concept are that the ejector combustion-chamber pressure must be relatively low and that with the fan engines the gas generator may need to be oxidizer-rich. For an engine with no fan, such as engine 5, the ejector concept is useful, but when a fan with a moderate pressure ratio is used, such as engine 4, the ejector disadvantages may outweigh the advantages. An analysis of the relative merits of the ejector for engine 4 follows.

As shown in table 4, the sea-level static thrust of engine 4 is 3.886 MN in the fan-ejector-ramjet mode and 0.880 MN in the fan-ramjet mode, so the thrust difference between the two modes is 3.006 MN. The specific impulse based on thrust difference and the ejector propellant flow of 912.4 kg/s is 335.9 seconds. The space shuttle main engine provides a sea-level specific impulse of 363.2 seconds (ref. 12). The best vacuum specific impulse for engine 4 is 450.2 seconds, whereas the space shuttle main engine delivers 455.2 seconds. At a Mach number of 2.0 and an altitude of 9 km, the thrust difference between fan-ejector-ramjet and fan-ramjet modes is 3.203 MN, so the specific impulse is 358.0 seconds. The space shuttle main engine at that condition would deliver a specific impulse of 427.9 seconds. The conclusion is that engine 4 without the ejector subsystem plus a separate rocket would provide higher specific impulse at all conditions than engine 4 would with the ejector subsystem operating.

There may be a mass advantage of the ejector subsystem over the separate rocket. If the maximum vacuum thrust is divided by the product of g and the entry for "primary rockets" (i.e., ejector subsystem) in the mass breakdown in reference 8, the result is 110; this value is higher than 74, which is the comparable value for the space shuttle main engine (ref. 12). The total mass for the ejector subsystem, however, may be considerably higher than the mass of the primary rockets. The elimination of the mixer and the reduction in size of the ramjet combustion chamber and nozzle would considerably reduce the engine mass if the ejectors were removed. A more detailed mass analysis would be required to show the total effect of removing the ejector rocket system.

Typical Vehicle Design

The results of the analysis of one engine and vehicle combination are shown in figure 2 and table 5. The vehicle is sketched in figure 2 to show the relative size of the wing, tail, body, and inlets and the body shape assumed.

The component mass breakdown, presented in table 5, shows that the payload mass is negative. With the given gross mass and the trajectory performance calculated, the mass at entry is insufficient to allow for all the components as calculated. The difference has been accounted for by allowing the payload to have a negative mass. Although a vehicle with a negative payload mass cannot be built, the calculation is still useful as a tool in determining the relative merits of vehicle and engine combinations.

Three other items are noteworthy in the component mass breakdown (table 5). First, the ascent propellant is a large part of the gross mass; the ratio of gross mass to burnout mass is 7.45, which is as high as the corresponding mass ratios estimated for a single-stage rocket vehicle in reference 5. The second item is the combined mass of the engines and inlets, which is 23 percent of the dry mass. For most of the engine and vehicle combinations considered, the engines and inlets were an even greater fraction of the dry mass. The third item is the combined mass of the basic structure, fuel tanks, and thermal protection system, which totals 42 percent of the dry mass. Much of this mass is required to cover the large hydrogen volume, 1901 m^3.

Effect of Thrust Option and Number of Engines

Figure 6 shows the payload achieved with three different thrust options as a function of the number of engines. When the number of engines is less than the optimum there can be a rapid drop in payload. This drop occurs because the thrust at some point in the flight is barely sufficient for acceleration. With 1.5 engines, the wing area had to be increased from 700 m^2 to 750 m^2 to achieve a trajectory that reached orbit, and the resulting payload was greatly reduced from that for the case with 1.6 engines.

The three thrust options shown in figure 6 represent the complete range of rocket throttling between Mach numbers of 1.8 and 3.5. Option C (fig. 5) uses no rocket thrust in this range, and option E uses full rocket thrust throughout the air-breathing portion of the trajectory. With option C, the air-breathing thrust dominates the air-breathing portion of the flight, and the rocket thrust is used only as needed to achieve satisfactory acceleration. With option E, the air-breathing thrust simply adds to the rocket thrust. Option D is a compromise between the two extremes. The results shown in figure 6 indicate that for more than 2.1 engines option C may be superior to option E, but option E has a somewhat better payload capability when each option is compared at the optimum number of engines. The reduced propellant consumption from throttling the rocket subsystem could be worthwhile if sufficient thrust were available, but the engine mass reduction achievable with option E is considerably more important.

The effect of thrust option is also shown in figures 7, 10, and 11 (other results of figs. 7, 10, and 11 are discussed subsequently). Figures 7 and 11 show the same trend as figure 6. The engine 1 data in figure 10 show a trend opposite to that described previously; that is, using rocket thrust continuously did not improve the payload. The difference is that the low rocket thrust level of engine 1 was already limiting the acceleration potential of the vehicle using engine 1.

Effect of Choice of Fuel

Figure 7 shows the comparison of results obtained with engines 4 and 7. The difference between these engines is the ramjet burner and gas generator fuel; engine 7 uses JP-4 (kerosene), and engine 4 uses hydrogen. The results indicate that the use of hydrogen yields a better payload. The JP-4 vehicles are smaller and lighter, but the fuel mass required is greater.

Effect of Ratio of Rocket Thrust Level to Air-Breathing Thrust Level

The comparison in figure 8 of the results obtained with engines 3 and 4 shows the effect of the ratio of rocket thrust level to air-breathing thrust level. The engines were nearly equivalent in air-breathing thrust level, but the rocket thrust level of engine 4 was nearly three times that of engine 3.

The results shown in figure 8 indicate that the rocket thrust level for engine 3 was too low. About twice as many engines were required with engine 3 as with engine 4, and the fuel consumption was not improved as would have been expected from comparisons of other trajectory analyses. Examination of the trajectory parameters showed that at Mach 3.5, the engine 3 vehicle with 4.2 engines had a mass of 732 000 kg compared with 704 000 kg for the engine 4 vehicle with 2.1 engines. This advantage was lost in the rocket phase, however, because of the low rocket thrust level of engine 3. The injected mass was about 143 000 kg for both vehicles. The difference in payload shown in figure 8 is essentially caused by the difference in engine weight. Increasing the rocket thrust level of an engine allows a reduction in the number of engines and hence the sum of the weight of the air-breathing components in all the engines.

Effect of Fan Removal

The effect on payload of removing the fan from the airstream during the ramjet mode is shown in figure 9 for engines 2 and 3. An increase in payload mass of about 15 Mg because of the removable fan of engine 2 is shown. The fixed fan is lighter, but two limitations are avoided with the removable fan. The maximum Mach number for the air-breathing segment is limited to 4.5 with the fixed fan, whereas the removable fan allows air breathing to a Mach number of 8.0. The thrust at the lower end of the ramjet regime is reduced with the fixed fan because the fan restricts the airflow. During the actual removal of the fan from the airstream, the inlet flow may be disturbed. The resulting problems would have to be considered in developing the technology for a removable fan.

Effect of Fan Pressure Ratio

The effect of the design fan pressure ratio is shown in figures 10 and 11. Figure 10 shows the results for engines 1 and 2, which are designed for the same parameters except fan pressure ratio. The pressure ratio is 1.3 for engine 1 and 1.8 for engine 2. The results indicate that the lower pressure ratio is better; however, there are possible qualifications. One of the reasons the higher fan pressure ratio yielded less payload is the high gas generator flow, which could be reduced by using a turbine with more than one stage. Another reason is that the rocket thrust level was reduced, which was shown to be undesirable in the discussion of effect of thrust-level ratio. A third reason is that the fan mass doubled although all other engine component mass items decreased.

The results shown in figure 11 compare engine 4, which has a fan pressure ratio of 1.8, with engine 5, which has no fan. Since the fan for engine 4 was not removable, the engine 4 vehicle was limited to a maximum Mach number of 4.5 for the air-breathing segment. The engine 5 vehicle could operate up to a maximum Mach number of 8.0 for the air-breathing segment. The results indicate that there is a benefit from incorporating a fan into the engine. Other results indicate that the benefit would be even greater for a removable fan having a lower pressure ratio. The results are nearly equal for the modes which maximize the use of air-breathing thrust and separate when the rocket is not throttled. The inclusion of the fan also results in a more flexible engine. Engine 5 has only three modes of operation: ramjet, ejector ramjet, and rocket. Engine 4 has, in addition to these three modes, a fan-ejector-ramjet mode, a fan-ramjet mode, and a low-thrust fan-only mode.

CONCLUSIONS

Composite propulsion has been analyzed for single-stage-to-orbit vehicles designed for horizontal take-off and landing. Six engines previously studied have been evaluated based on assumptions for gross mass, aerodynamics, mission, geometric, and mass relationships. The results of this evaluation led to the following conclusions:

1. None of the engines has performed adequately to deliver payloads to orbit as analyzed.

2. The single-stage turbine and oxidizer-rich gas generator of the engines with fans result in a low engine specific impulse.

3. The specific impulse based on the incremental thrust and propellant flow of the ejector subsystem is less than that of a separate rocket system with a high combustion pressure.

4. There is a benefit from incorporating a fan into the engine.

5. Removal of the fan from the airstream during the ramjet mode increases the orbital payload capability.

Langley Research Center
National Aeronautics and Space Administration
Hampton, VA 23665
September 14, 1977

APPENDIX

MASS CALCULATIONS

The following list gives the relationships and constants used in the component mass calculations:

Element	Relationship or constant	Units
Wing	$(0.044\ \text{kg})\left(\frac{m_l n S_e}{r}\right)^{0.584}$	m_l, kg; S_e, m^2
Tail	$(1.21\ \text{kg})(0.15 S_{th})^{1.1}$	S_{th}, m^2
Body basic structure	$(0.523\ \text{kg})\left(\frac{l}{h} n\right)^{0.15} q^{0.16} S_b^{1.05}$	q, Pa; S_b, m^2; l, m; h, m
Thrust structure	$0.015 \frac{T}{g}$	T, N; g, m/s^2
Oxygen tanks	$(27.0\ \text{kg})\left(\frac{m_O}{\rho_O}\right)^{0.843}$	m_O, kg; ρ_O, kg/m^3
Hydrogen tanks	$(32.3\ \text{kg})\left(\frac{m_H}{\rho_H}\right)^{0.795}$	m_H, kg; ρ_H, kg/m^3
JP-4 tanks	$(0.0305\ \text{kg})\left(\frac{m_{JP-4}}{\rho_{JP-4}}\right)^{0.824}$	m_{JP-4}, kg; ρ_{JP-4}, kg/m^3
Thermal protection	$(7.81\ kg/m^2) S_w$	S_w, m^2
Landing gear	$(0.00676\ \text{kg}) m_l^{1.124}$	m_l, kg
Auxiliary propulsion	$0.01375 m_e$	m_e, kg
Prime power	1774 kg	
Electrical conversion and distribution	$(0.135\ \text{kg}) m_e^{0.7213} l^{0.3606}$	m_e, kg; l, m
Ascent oxygen, m_O	From trajectory	m_O, kg
Ascent hydrogen, m_H	From trajectory	m_H, kg
Ascent JP-4, m_{JP-4}	From trajectory	m_{JP-4}, kg
Lift-off mass	1 000 000 kg, sum of preceding items and payload	
Runway propellant	30 000 kg	
Gross mass	1 030 000 kg	

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TABLE 1.- ENGINE DESIGN PARAMETERS

Study engine	Engine cycle	Design fan pressure ratio	Design flow-rate ratio (airflow to ejector flow)	Propellant				Ramjet fuel	Fan
				Gas Generator		Ejector (rocket)			
				Fuel	Oxidizer	Fuel	Oxidizer		
[a]1	ATR	1.3	3.3	H_2	O_2	H_2	O_2	H_2	Removable
[a]2	ATR	1.8	3.3	H_2	O_2	H_2	O_2	H_2	Removable
[a]3	ATR	1.8	3.3	H_2	O_2	H_2	O_2	H_2	Fixed
[b]4	ATR	1.8	1.0	H_2	O_2	H_2	O_2	H_2	Fixed
[c]5	ERJ		1.0			H_2	O_2	H_2	
b [d]6	ATR	1.8	1.0	JP-4	O_2	JP-4	O_2	JP-4	Fixed
[b]7	ATR	1.8	1.0	JP-4	O_2	H_2	O_2	JP-4	Fixed

[a]Criterion for sizing: 1.8 MN sea-level static thrust.
[b]Criterion for sizing: same fan as engine 3.
[c]Criterion for sizing: same ejector flow rate as engine 4.
[d]Only sea-level static performance calculated for engine 6.

TABLE 2.- MISSION DESIGN SPECIFICATIONS

Initial latitude, deg	28.5
Initial heading	East
Maximum dynamic pressure, kPa	90
Maximum acceleration, g units	1.7
Maximum product of dynamic pressure and angle of attack, kPa-deg	450
Insertion orbit perigee, n. mi.	50
Insertion orbit apogee, n. mi.	100
Orbit maneuver velocity increment, m/s	140
Propellant reserves, m/s	60
Margin of safety	1.40
Payload diameter, m	4.57
Payload length, m	18.29

TABLE 3.- AERODYNAMIC CHARACTERISTICS USED IN TRAJECTORY ANALYSIS

Mach number	C_{L_α}, per deg	C_D at α, deg, of -					
		0	2	5	8	18	20
0.0	0.047	0.0150	------	0.0260	0.0537	0.2115	------
.6	.047	.0150	------	.0260	.0537	.2115	------
.8	.044	.0170	------	.0271	.0545	.2115	------
.9	.040	.0200	------	.0289	.0555	.2115	------
1.0	.045	.0400	0.0411	.0471	------	------	0.1534
1.2	.045	.0350	.0366	.0451	------	------	.1970
1.5	.043	.0300	.0322	.0439	------	------	.2519
2.0	.037	.0250	.0272	.0387	------	------	.2440
3.0	.026	.0200	.0215	.0295	------	------	.1714
4.0	.021	.0160	.0172	.0237	------	------	.1395
6.0	.019	.0150	.0161	.0219	------	------	.1262
30.0	.019	.0150	.0161	.0219	------	------	.1262

TABLE 4.- ENGINE 4 PERFORMANCE

Operating mode	Mach number	Altitude, km	Thrust, MN	Ejector flow, kg/s	Specific impulse, s
Fan ejector ramjet	0.0	0.0	3.886	912.4	364.9
	.3	.0	3.844		357.8
	.8	.6	4.273		379.5
	.8	3.6	3.948		373.3
	1.0	4.5	4.116		384.5
	1.3	3.0	4.849		409.9
	1.3	6.0	4.435		404.8
	2.0	9.0	5.518		460.6
Fan ramjet	.0	.0	.880	.0	517.9
	.8	.6	1.293		560.1
	.8	3.6	.943		577.4
	1.0	4.5	1.070		609.3
	1.3	3.0	1.864		645.4
	1.3	6.0	1.335		664.2
	2.0	9.0	2.315		763.7
	2.0	12.0	1.553		777.9
	3.0	15.0	2.522		768.3
Ramjet	3.0	15.0	1.082		3986.6
	4.5	15.0	2.857		3983.6
	4.5	21.0	1.109		3971.8
Ejector ramjet	4.5	15.0	6.276	912.4	649.3
Rocket	(Vacuum)	(Vacuum)	4.028	912.4	450.2

TABLE 5.- COMPONENT MASS BREAKDOWN. ENGINE 4, THRUST OPTION E, 1.6 ENGINES

	Mass, kg
Wing	8 458
Tail	2 772
Body basic structure	15 898
Thrust structure	1 720
Oxygen tanks	6 547
Fuel tanks	12 736
Thermal protection system	24 650
Landing gear	3 660
Engines, uninstalled	19 986
Inlets	9 090
Subsystems	13 061
Margin	8 954
Dry mass	127 532
Personnel	705
Payload	-8 556
Reaction control propellant and reserves	4 604
Ascent propellant residuals	6 284
Entry mass	130 569
Orbit maneuver propellant	4 189
Reserves and in-flight losses	3 584
Ascent oxygen	736 721
Ascent hydrogen	124 937
Lift-off mass	1 000 000
Runway propellant	30 000
Gross mass	1 030 000

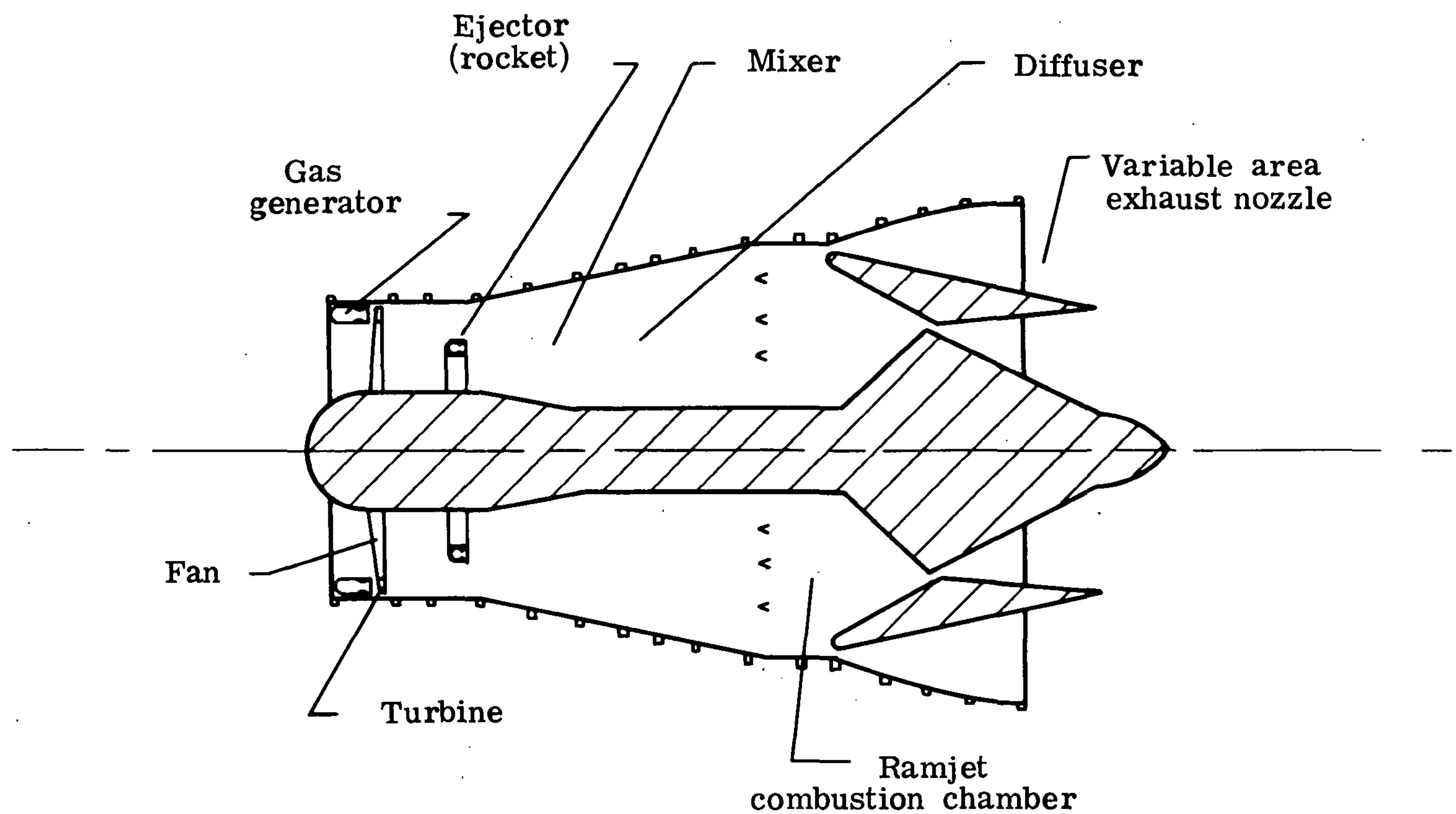

Figure 1.- Typical engine schematic.

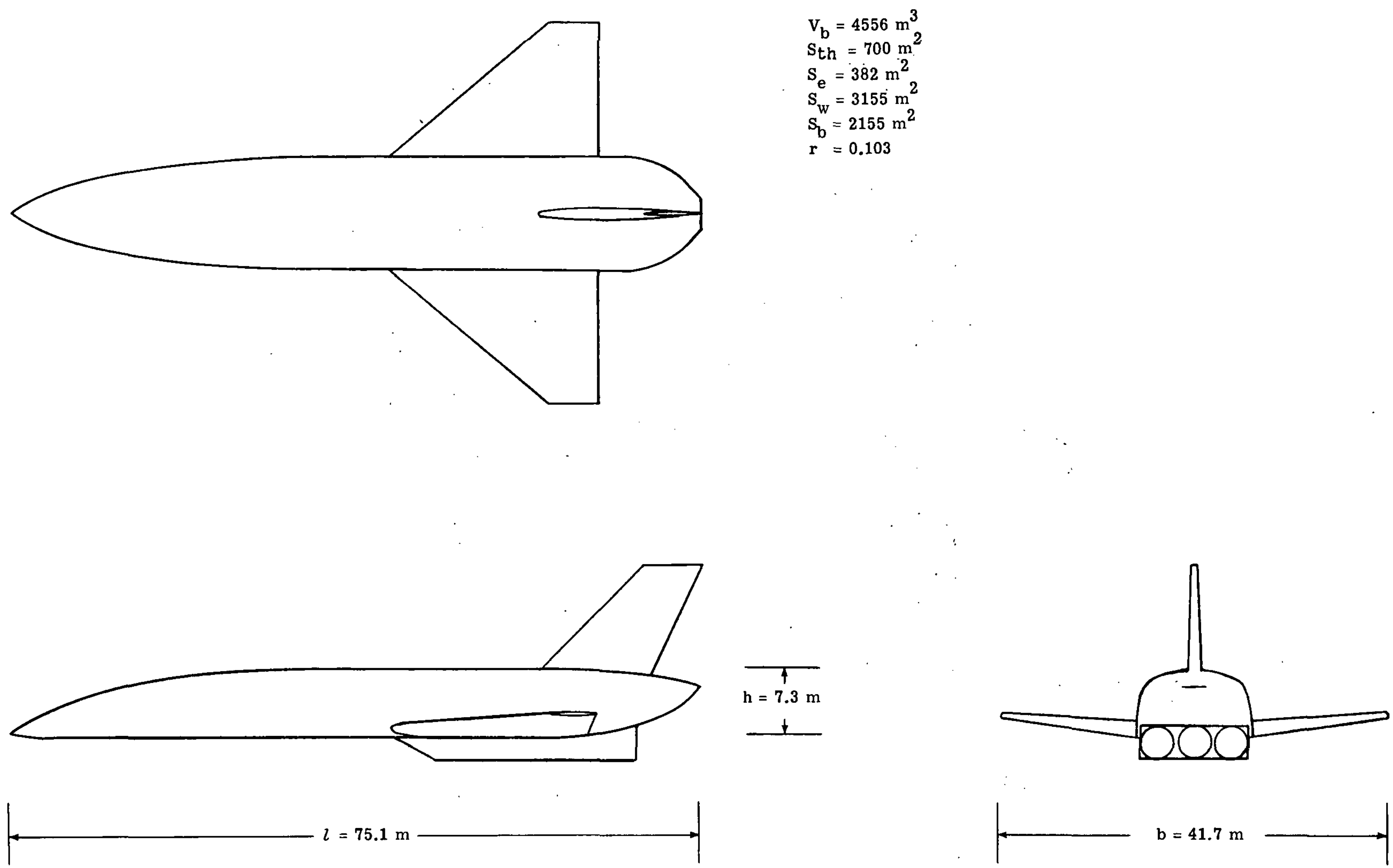

Figure 2.- Typical vehicle configuration. Engine 4; thrust option E; 1.6 engines (shown with 3 scaled smaller engines).

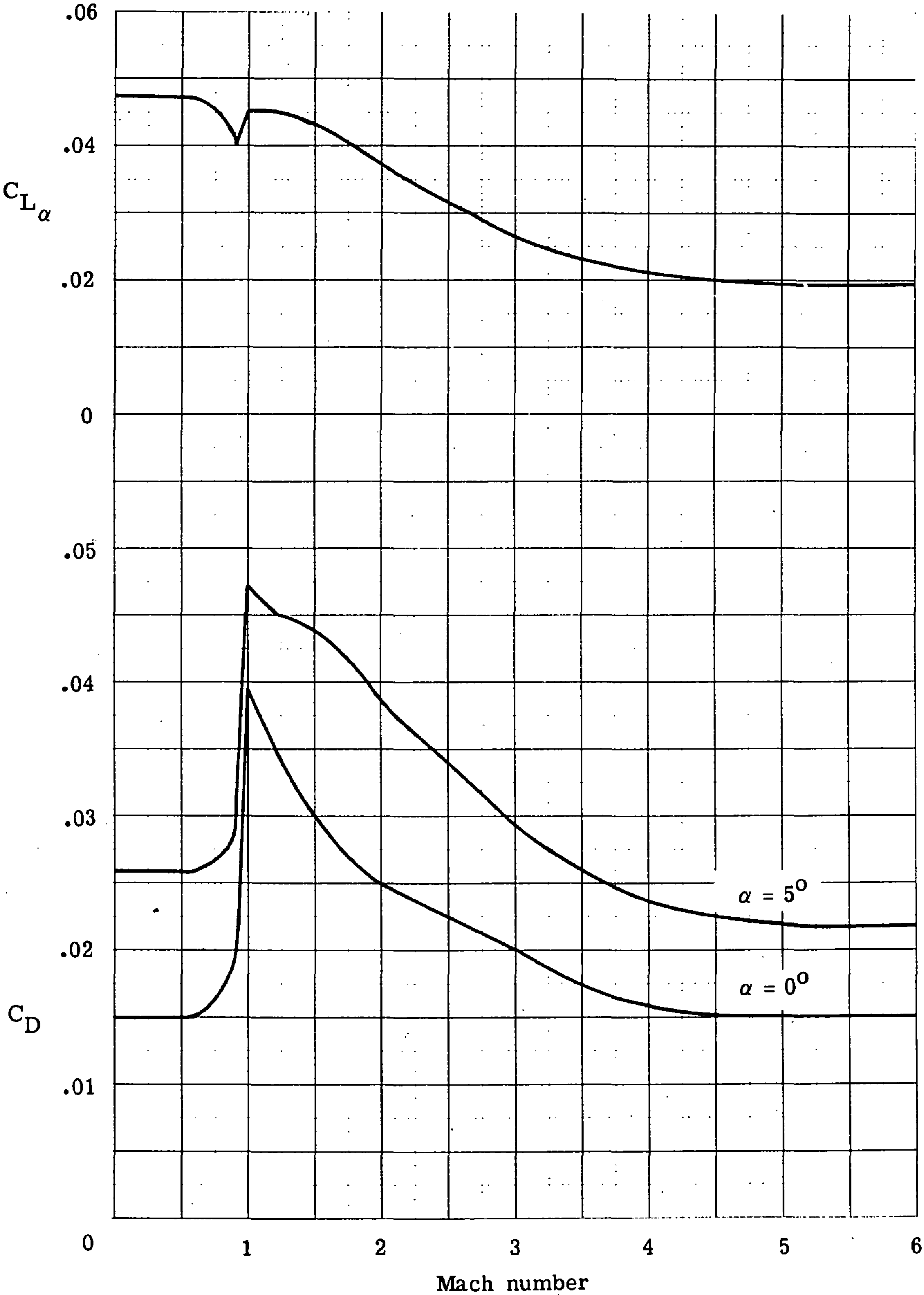

Figure 3.- Representative sample of aerodynamic coefficients used.

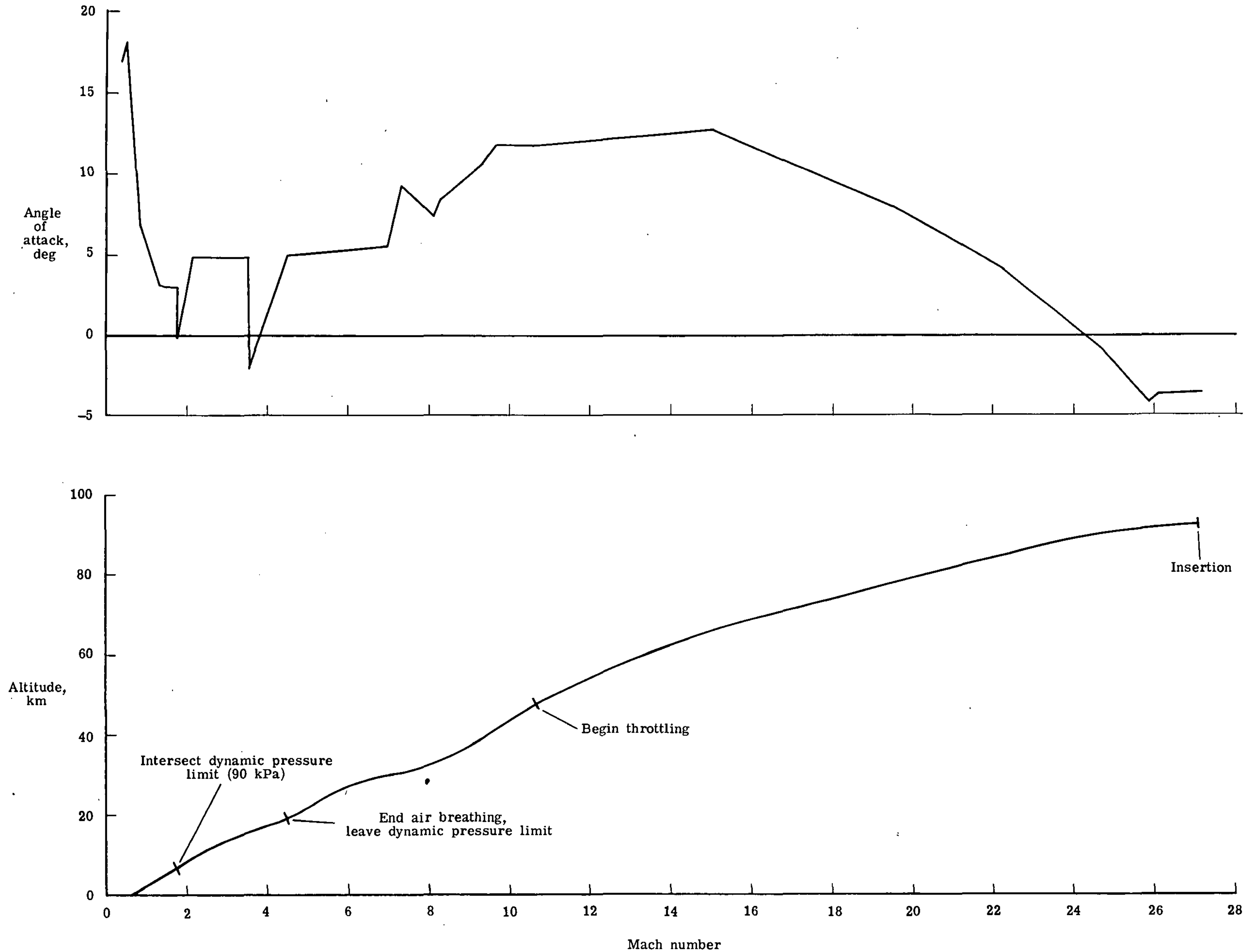

Figure 4.- Ascent trajectory from lift-off to insertion. Engine 4; thrust option E; 1.6 engines.

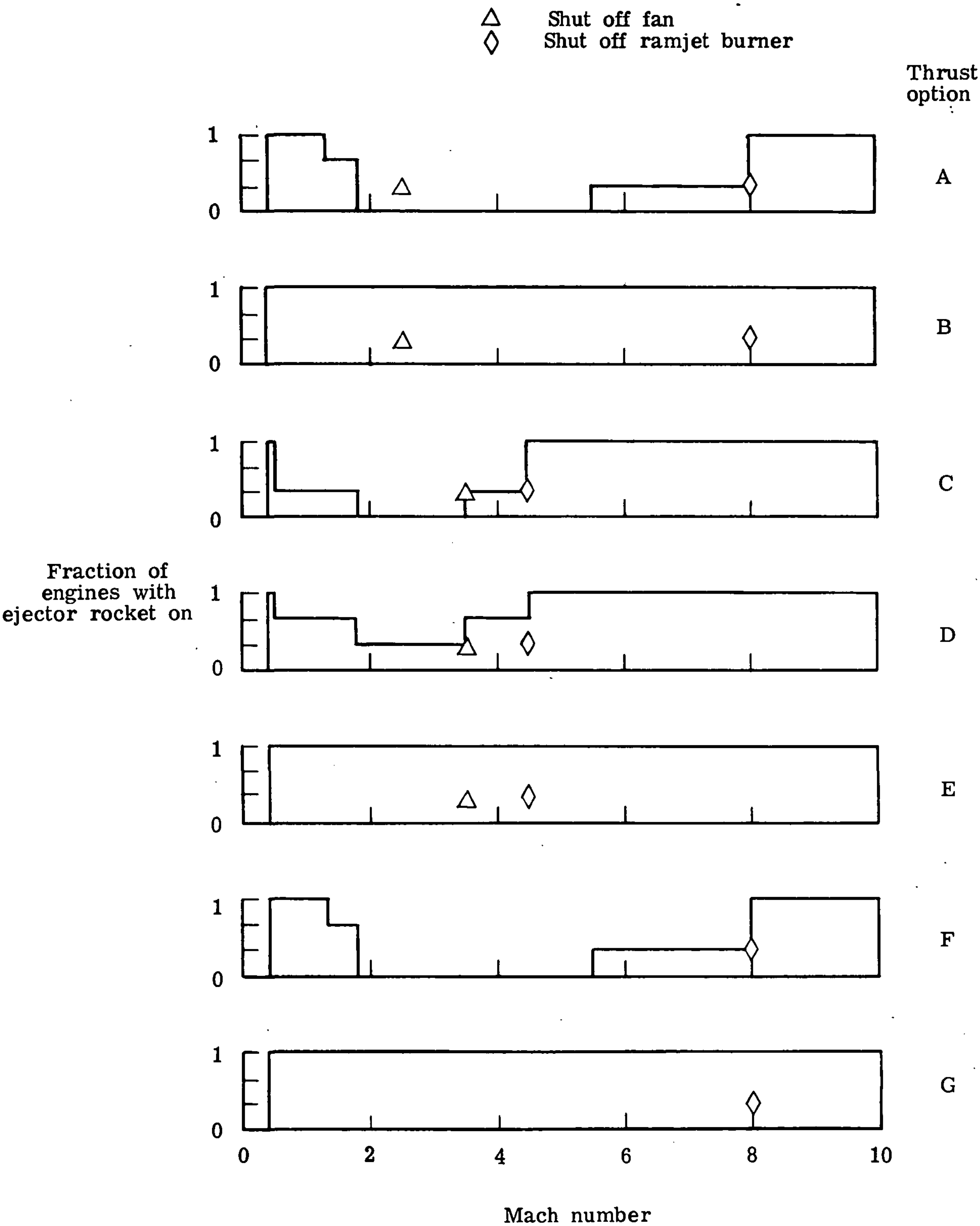

Figure 5.- Operational schedules of fan; ramjet burner and rocket use for each thrust option.

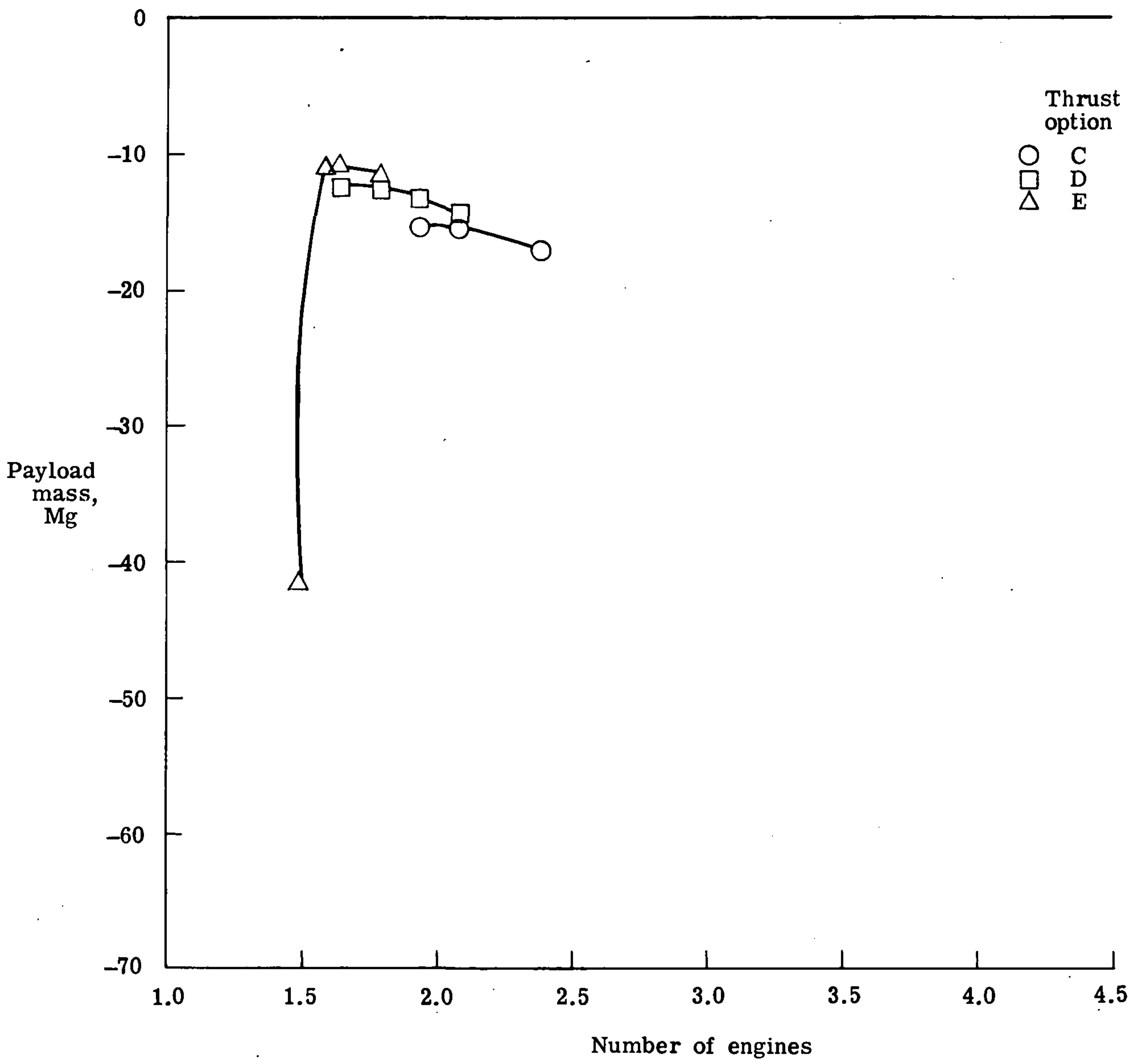

Figure 6.- Effect of thrust option. Engine 7.

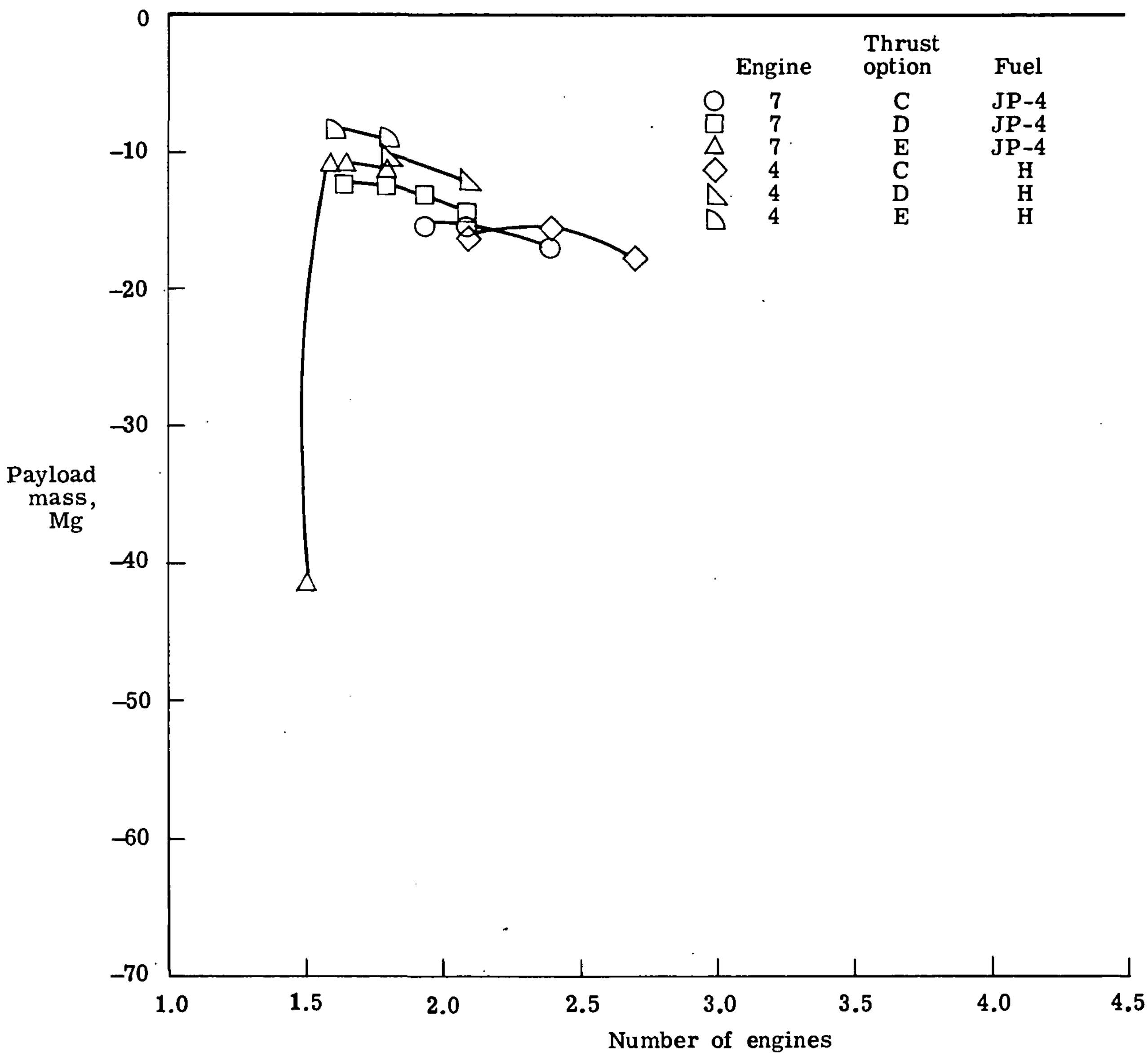

Figure 7.- Effect of choice of fuel.

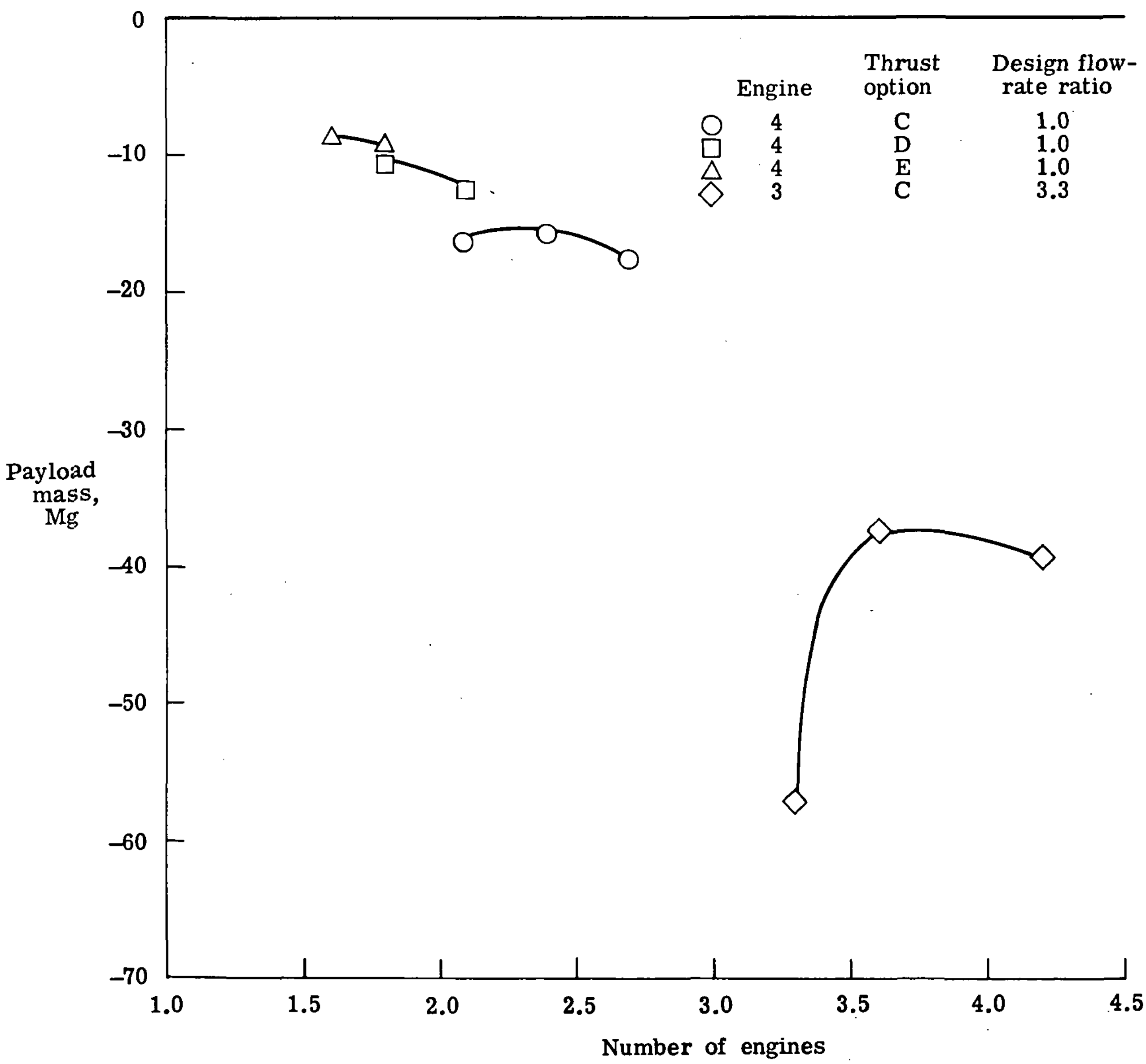

Figure 8.- Effect of ratio of rocket thrust level to air-breathing thrust level (increased design flow-rate ratio represents decreased rocket thrust). Fixed fan.

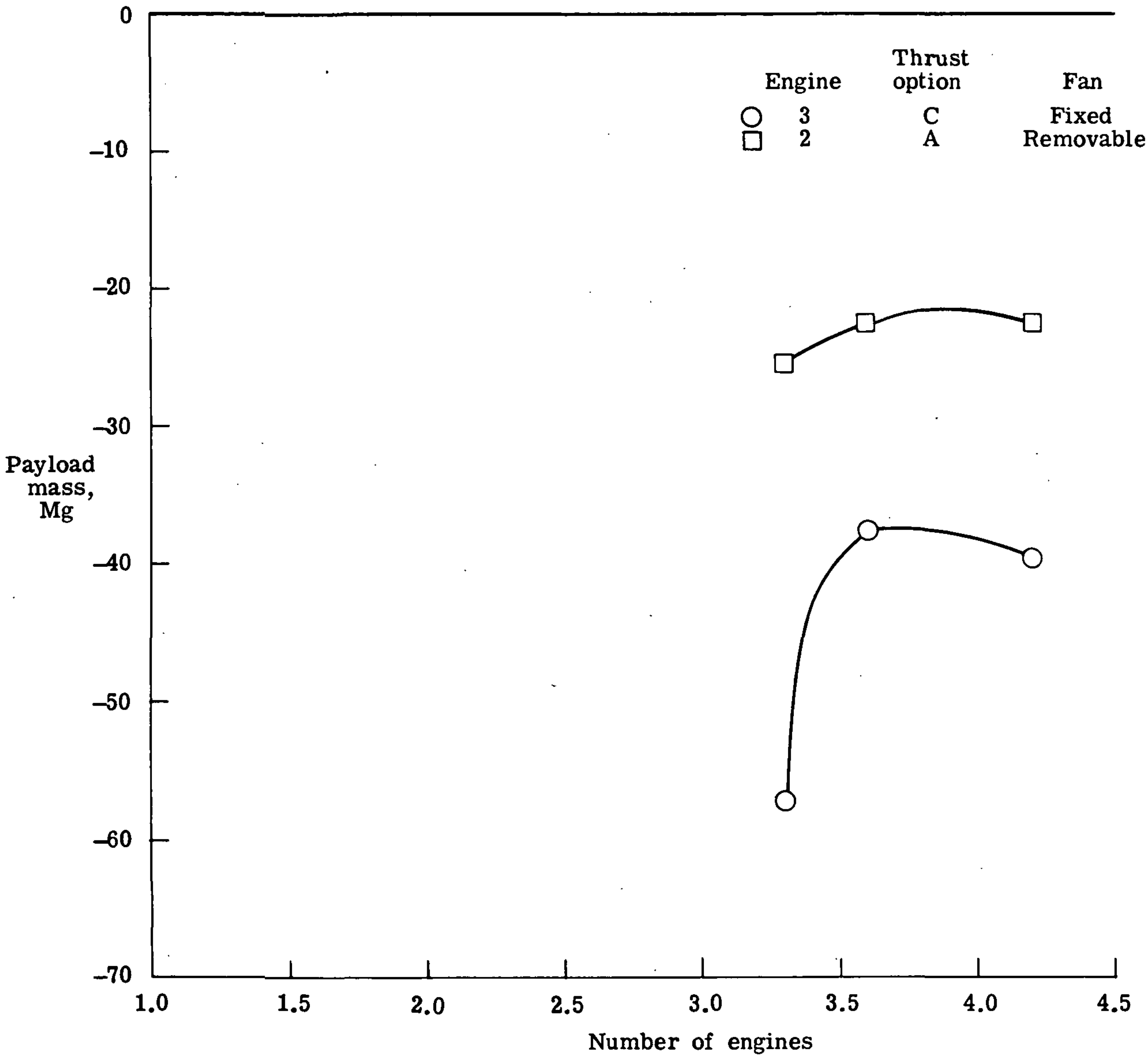

Figure 9.- Effect of fan removal from airstream.

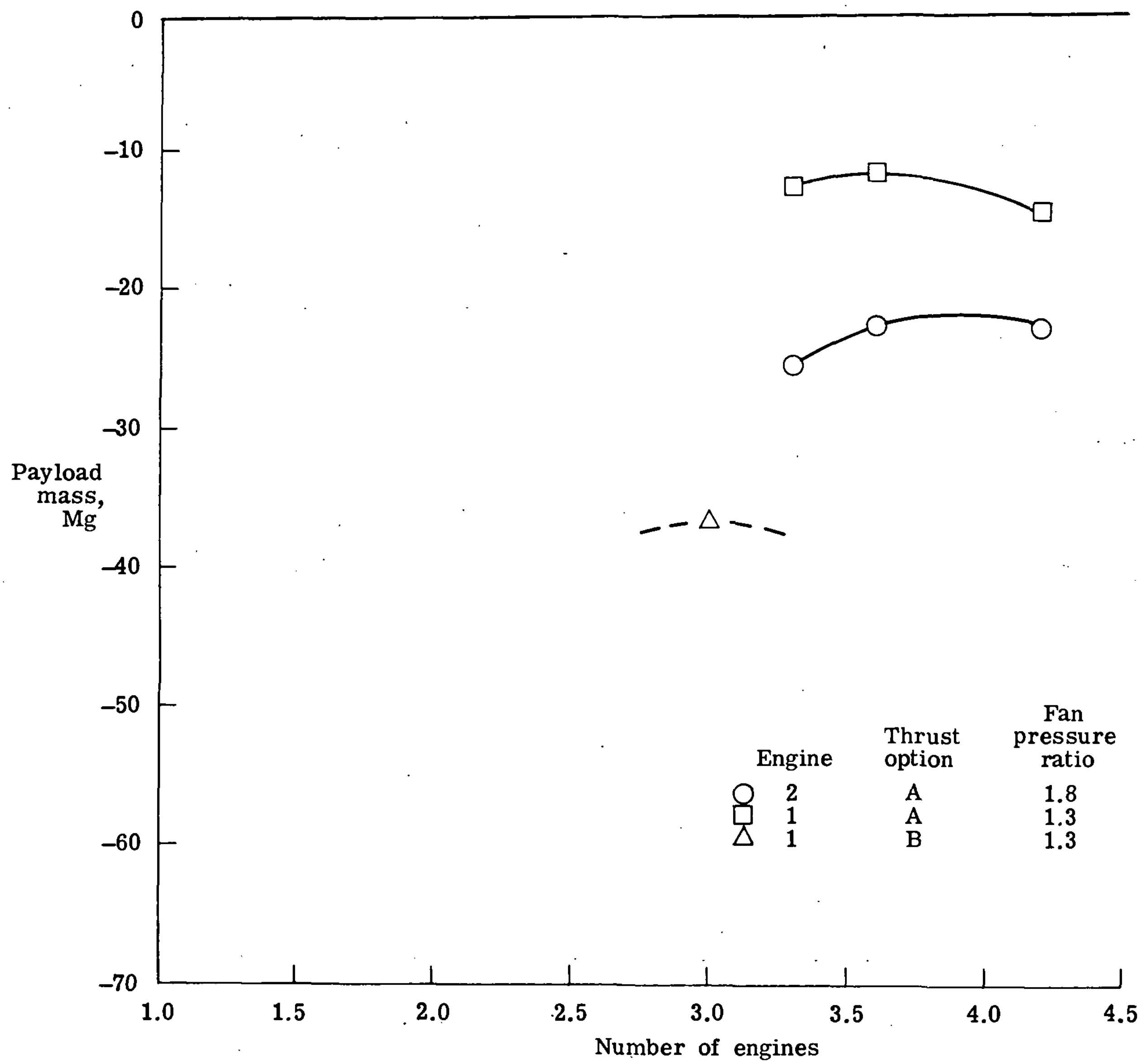

Figure 10.- Effect of fan pressure ratio.

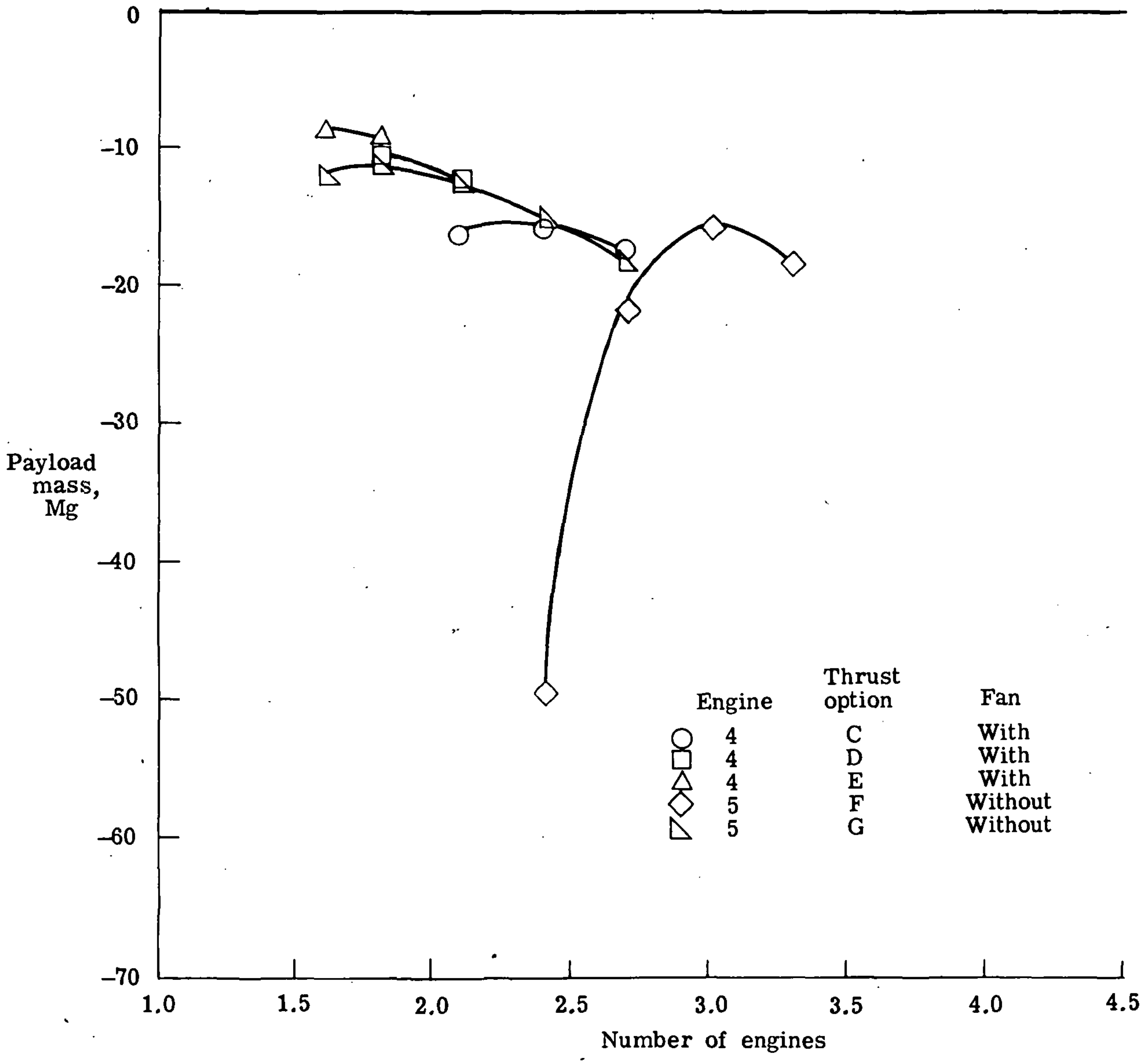

Figure 11.- Effect of use of fan.

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