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COMPOSITE ENGINES FOR APPLICATION TO A SINGLE-STAGE-TO-ORBIT VEHICLE

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THE MARQUARDT COMPANY
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16. Abstract

Seven composite engines were designed for application to a reusable single-stage-to-orbit vehicle. The engine designs were variations of the Supercharged Ejector Ramjet engine. The resulting performance, weight, and drawings of each engine form a data base for establishing the potential of this class of composite engine to various missions, including the single-stage-to-orbit application. The impact of advanced technology in the design of the critical fan turbine was established.

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PREFACE

The "Composite Engines for Application to a Single-Stage-to-Orbit Vehicle" program was performed for the National Aeronautics and Space Administration/Langley Research Center under Contract NAS1-13304. The basic objectives of this program were: (1) Design seven composite engines for application to a single-stage-to-orbit reusable launch vehicle and (2) Characterize these engines in terms of performance, weight and geometry.

Special acknowledgement is given to William R. Hammill who prepared the engine drawings and estimated the weights for all engines and to Jeanette A. Yocham who typed this report.

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COMPOSITE ENGINES

FOR

APPLICATION TO A REUSABLE SINGLE-STAGE-TO-ORBIT VEHICLE

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SUMMARY

Seven composite engines were designed for application to a reusable single-stage-to-orbit vehicle. Six of the engines were airturborocket variations of the Supercharged Ejector Ramjet engine in which the airbreathing gas generator was replaced with a bipropellant gas generator to reduce engine weight. One of the engines was an ejector ramjet. Variables perturbed were fan pressure ratio, fan removable or fixed, relative ejector subsystem thrust, and fuel.

The basis for engine design was to maximize specific impulse for the Supercharged Ejector Ramjet operating mode at sea level static conditions. Engine performance was then estimated for a series of flight trajectory conditions. Four additional operating modes considered were Fan Ramjet, Ejector Ramjet, Ramjet, and Rocket. A weight statement and outboard profile and installation drawings were prepared for each engine.

The results of this study form a data base for establishing the potential of this class of composite engine to various missions, including the single-stage-to-orbit application. An examination of possible improvements indicated that the Fan Ramjet specific impulse can be increased significantly if a multi-stage turbine is used. A study of ground test facilities indicated reduced size engines should be considered to minimize facility improvement requirements.

INTRODUCTION

A number of satellite launch vehicles have been successfully developed by the United States and other nations. Good examples are the Thor-Delta, Titan III, Scout and Saturn launchers. These programs have demonstrated man's ability to transport large payloads into earth orbit. However, these vehicles are not recoverable and, therefore, the cost per unit payload in orbit is very high. The primary objective of the current Space Shuttle program is to significantly reduce the cost of putting large payloads into orbit through recovery and reuse of the costly orbiter vehicle and the solid rocket motor cases. However, launch vehicles in which all components are largely reusable must eventually be developed if truly low-cost space operations are to be achieved. The airbreathing launch vehicle has inherent features which make it a prime candidate for future second generation shuttle systems.

The rocket engine is characterized by high thrust-to-weight ratio but at low specific impulse values. In contrast, true airbreathing engines are characterized by high specific impulse performance, but engine weights are heavy. The composite engine combines the best features of rocket and airbreathing engines into simple integrated, highly flexible propulsion systems. These propulsion systems feature multimodal operation capability with cycle process interactions between engine components. Increased engine performance results from this synergistic design approach. Examples of composite engines are the Ejector Ramjet (ERJ) and the Supercharged Ejector Ramjet (SERJ). This class of engine is also frequently referred to as mixed cycle engines, rocket ramjet engines, and less frequently, as compound cycle engines.

Under an earlier NASA contract (NAS7-377, references 1 and 2 and appendix A), the potential of several composite engines, when applied to the first stage of a two-stage manned advanced reusable launch vehicle, was evaluated. Payload in orbit was the prime evaluation criteria. Composite engines were shown to be competitive to an advanced liquid rocket engine. In particular, the payload performance of the SERJ engine was promising.

Recently, NASA and industry design studies have shown the potential of reusable single-stage-to-orbit vehicles for the 1995 time period. The composite engine is one candidate propulsion system since earlier work indicated its potential. The current study was initiated to design and characterize composite engines similar to the SERJ cycle to provide a data base for evaluating such engines for single-stage-to-orbit and other vehicle concepts. In this study, extensive use was made of the SERJ/ERJ technology developed under Air Force/Navy sponsored programs (ref. 3, 4, 5, 6, 7, 8 and 9).

SYMBOLS

The data were obtained in the U. S. Customary Units but are presented in both International Units (SI) and U. S. Customary Units.

A	Area
a	Speed of sound
ATR	Airturborocket
C_p	Specific heat at constant pressure
D	Diameter; mixer divergence area ratio, $A_5/(A_4 + A_{4_p})$
ERJ	Ejector ramjet
F	Thrust
FRJ	Fan ramjet
F/W	Thrust to weight ratio
g	Gravitational constant
H	Enthalpy
h	Altitude
I_{sp}	Propellant specific impulse
L	Length
M	Mach number
O/F	Oxidizer/fuel flow ratio
P	Pressure
R	Gas constant
R_c	Fan total pressure ratio
RJ	Ramjet
SERJ	Supercharged ejector ramjet
SFC	Specific fuel consumption
SLS	Sea level static conditions
T	Temperature
V	Velocity
W	Weight flow
W_s/W_p	Ejector secondary to primary flow ratio
γ	Ratio of specific heats
η	Component process efficiency
ρ	Density

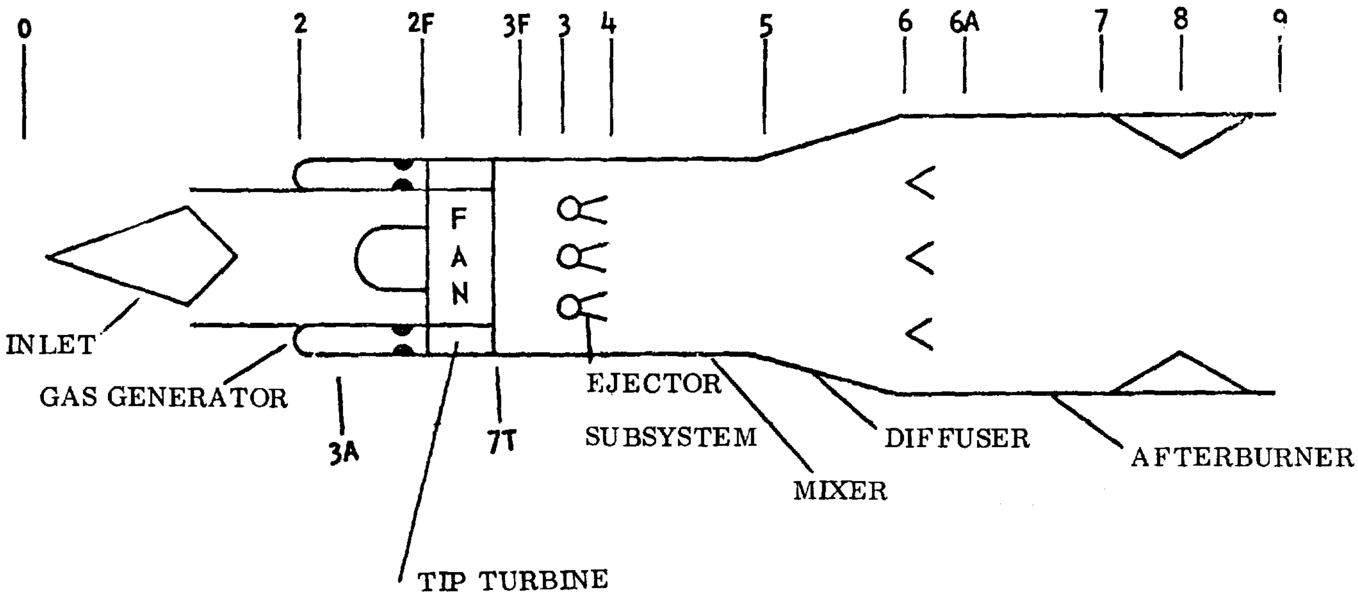
ϕ	Fuel equivalence ratio
θ_{T_2}	Corrected inlet total temperature, $\frac{T_{T_2}\,(°R)}{519}$ or $\frac{T_{T_2}\,(°K)}{288}$
δ_{T_2}	Corrected inlet total pressure, $\frac{P_{T_2}\,(psia)}{14.7}$ or $\frac{P_{T_2}\,(Newton's/meter^2)}{1.0135 \times 10^5}$

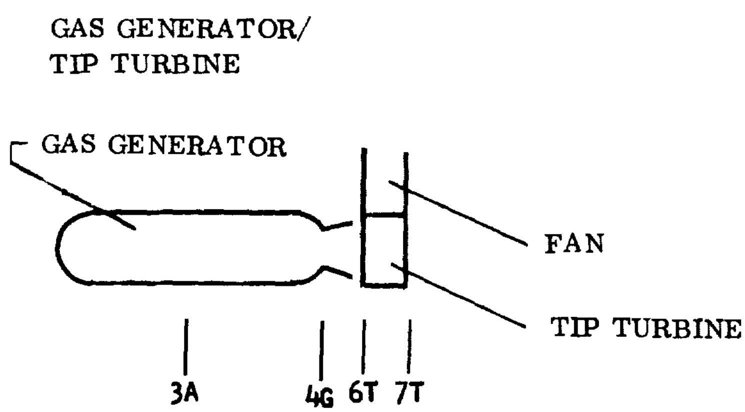
Subscripts

a	Air
A/B	Afterburner
C	Combustion, cowl
f	Fuel
g	Gas
m	Mass
N	net
NJ	Net jet
o	Oxidizer
P	Primary exit, primary stream
s	Secondary stream
T	Total condition

Engine Station Subscripts

0	Freestream
2	Engine inlet
2F	Fan inlet
3F	Fan exit
3	Fan/gas generator exhaust (mixed flow)
4	Mixer inlet
5	Mixer exit/diffuser inlet
6	Afterburner inlet
6A	Afterburner station after flameholders
7	Afterburner exit
8	Nozzle throat
9	Nozzle exit
3A	Gas generator chamber
4G	Gas generator nozzle throat
6T	Fan turbine inlet
7T	Fan turbine exit
2P	Ejector chamber
3P	Ejector throat
4P	Ejector exit





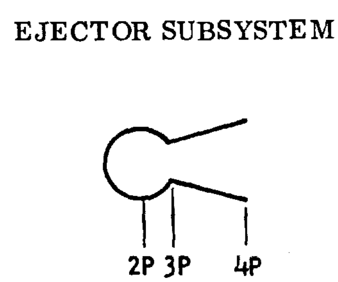


Figure 1. Engine Station Nomenclature for Supercharged Ejector Ramjet Engine, Airturborocket Cycle Variation

APPROACH

Supercharged Ejector Ramjet Engine

The basic arrangement and salient features of the Supercharged Ejector Ramjet (SERJ) engine as developed in references 1 and 2 are presented in Figures 2(a). In this engine, a low pressure ratio fan driven by an airbreathing gas generator (i.e. a turbojet), is integrated with rocket-like ejector primaries and an afterburner/ramjet combustor. The propulsion system features multimodal operation capability; six distinct engine operating modes can be used to maximize mission performance.

The airbreathing gas generator of the basic SERJ cycle has a relatively low fuel consumption but is a considerable portion of the engine weight, especially at high fan pressure ratios (ref. 2). A variation of the basic SERJ cycle in which the airbreathing gas generator is replaced with a bipropellant gas generator is shown in Figure 2(b) and is designated the airturborocket SERJ engine (ATR-SERJ). The bipropellant gas generator is lighter than the airbreathing gas generator, but the consumption of fuel and on-board oxydizer is greater. Since preliminary NASA mission studies indicate that the ATR-SERJ is capable of greater mission performance than the basic SERJ engine, the ATR-SERJ was evaluated in this study.

Engine Operating Modes

The multimode operation capability of the ATR-SERJ engine is the key to its operating versatility. Figure 3 shows the five major operating modes in which this engine functions during a single-stage-to-orbit mission.

The concurrent operation of all subsystems, designated the Supercharged Ejector Ramjet (SERJ) mode, is the engine's maximum thrust mode in the lower speed regime ($M_0 < 3.0$). In this operating mode, the fan is driven by the gas generator, the ejector operates at full or partial power and the ramjet functions as a stoichiometric fuel/air ratio afterburner, in combination with the two preceding stages of compression. The Fan Ramjet (FRJ) mode which has higher specific impulse performance, albeit significantly lower thrust than the SERJ mode, is accomplished by shutting down the ejector primary subsystem. Thermodynamically, this mode functions as an augmented plenum burning turbofan. As the vehicle accelerates to approximately Mach three, the gas generator is shut down and the fan is either mechanically removed from the airstream or allowed to autorotate (windmill). This is the Ramjet (RJ) mode. Engine specific impulse is very high in this operating mode. During ramjet operation, if an increase in thrust is required, then the Ejector Ramjet (ERJ) mode, as distinguished from the SERJ mode, is accomplished by turning on the ejector. This mode can be characterized as a high thrust, air augmented rocket engine cycle. Finally, at high Mach number and high altitude flight conditions ($M_0 = 4.5 - 8.0$), the rocket operating mode is used. In this engine mode, the inlet is closed off, the ejector is turned on and the ramjet combustor is shut down. This operating mode is continued until orbital speed is achieved. A fan mode can also be used for low thrust applications, but this mode was not analyzed in this study.

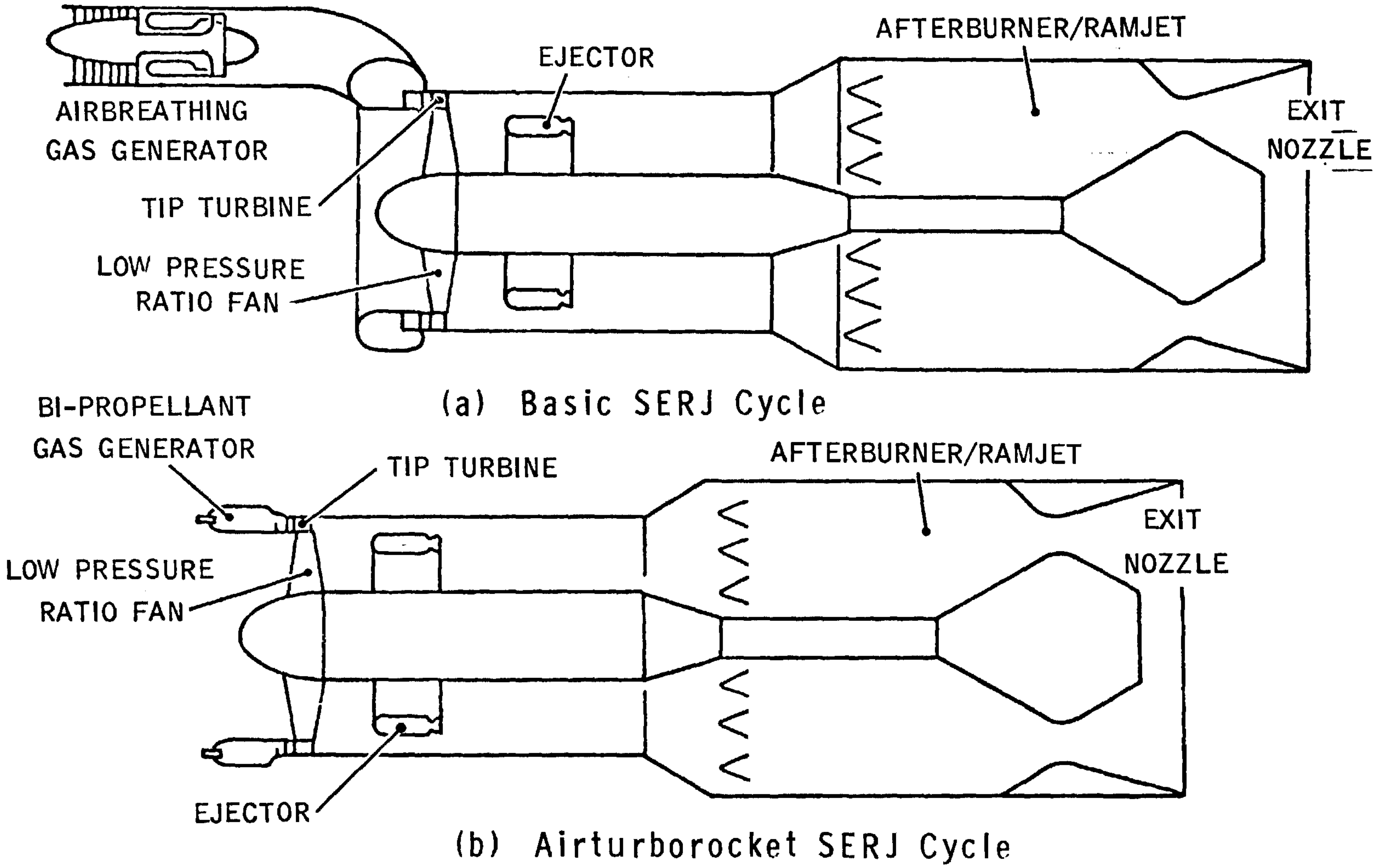


Figure 2. Supercharged Ejector Ramjet Cycle Variations. External Gas Generator.

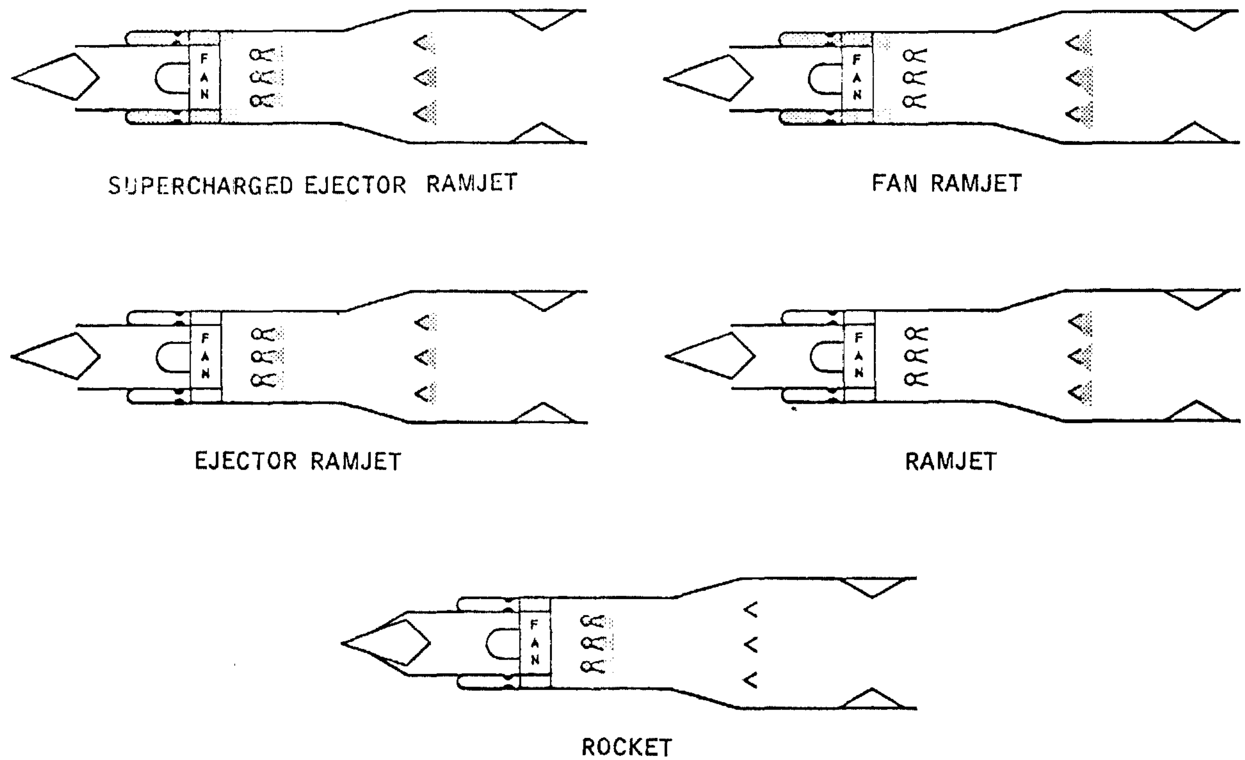


Figure 3. ATR-SERJ Operating Modes

Performance Computer Programs

A digital computer program was developed to calculate steady-state design and off-design performance for the Supercharged Ejector Ramjet (SERJ) engine. The writing of this program was initiated with Marquardt funds and completed under contract N00019-69-C-0541, a jointly funded Navy/Air Force program (Ref. 7). This program incorporates a number of computational features, some of which are unique to cycle analysis computer programs in general, which provide for the capability of conducting the following analyses:

1. Design and off-design performance
2. Standard and non-standard atmosphere performance
3. Propulsion modes
4. Fan/gas generator cycle arrangements
5. Component performance tradeoffs
6. Fan/gas generator cooling bleed air requirements
7. Fan/gas generator matching studies
8. Handling various fuels and propellants
9. Control system studies.

While this program was developed specifically for the SERJ engine, with modifications, it is applicable to other types of composite engines as well as to other airbreathing propulsion cycles. For this study, the airbreathing gas generator code was removed, and a new gas generator subroutine was written which computed the ATR-SERJ performance for either a bipropellant or monopropellant gas generator. The new bi/monopropellant gas generator subroutine was written to include chemical equilibrium/frozen flow performance. All propellants within a C-H-N-O system can be analyzed. The inputs to the gas generator subroutine are similar to those for the ejector subsystem with the exception that gas generator outlet/fan turbine inlet total temperature is used as a control parameter. Figure 4 presents the modified program logic to compute airturborocket performance. Station nomenclature was retained where possible with the appropriate interfacing station retained. In addition, the SERJ program mixer subroutine was changed to allow consideration of a divergent area ratio mixer. Several engines optimized in this program have a divergent mixer.

To reduce the computer time required to optimize a specific engine design, a modification was made to the basic SERJ program to allow changing any component downstream of the fan/mixer without recomputing the performance of the components upstream of the component being optimized. Additional improvement in running time was accomplished by modifying equilibrium chemistry subroutine tolerances. Running time on the IBM 360-65 is 2 to 4 minutes per point, and the storage capacity requirement is 210, 000 bytes.

The engine technology and component performance used to design and estimate performance for the six study engines is presented in Table I and Figures 5 through 10. The key engine design technology assumptions were:

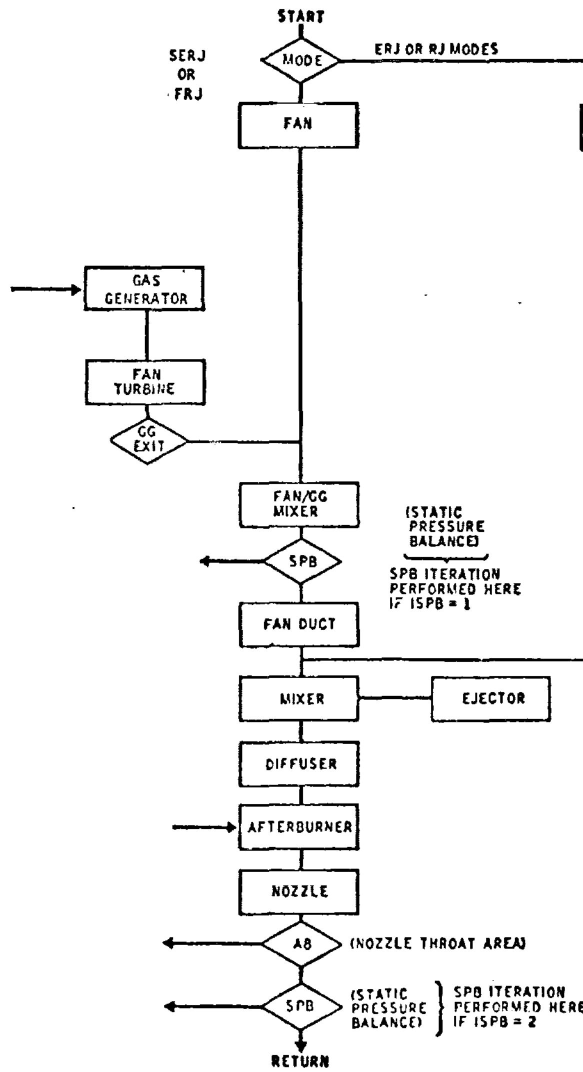


Figure 4. Airturborocket SERJ Engine Performance Computer Program Flow Diagram

1. Use of single stage tip turbine on single stage fan.
2. Design fan turbine pressure ratio = 3.50.
3. Maximum fan turbine inlet temperature = 1367°K (2460°R).
4. Ejector primary and afterburner/ramjet stoichiometric combustion.
5. External gas generator.
6. In estimating inlet performance, the lower surface of the vehicle is assumed to be at 0° angle of attack.

In principal, the modifications made to the SERJ engine performance computer program to compute airturborocket performance were relatively straightforward. However, considerable difficulty was experienced in interfacing these changes with the basic SERJ program. As a consequence of this delay, the performance of engines No. 1, 2 and 3 were computed with the composite engine program developed under Contract NAS7-377. The performance of engines no. 4, 5, 6 and 7 were computed with the modified SERJ program.

Rocket Mode Performance

The analyses techniques developed to estimate rocket operating mode performance are belived to be unique and, therefore, are presented in Appendix B of this report. It should be noted that two distinctly different rocket operating modes can exist. These operating modes are:

1. Subsonic Solution - In the subsonic solution, the ejector (rocket) flow shocks down to subsonic velocities in the mixer, diffuser and afterburner. The flow then accelerates to Mach 1 at engine exit nozzle throat and expands to supersonic velocities in the divergent section of the engine exit nozzle.

2. Supersonic Solution - In the supersonic solution, the ejector flow remains supersonic through the mixer, diffuser, afterburner and engine exit nozzle.

Which rocket operating mode will exist is largely determined by the maximum flow area of the engine exit nozzle throat. The supersonic solution, which is the higher performance operating mode, requires a considerably larger exit nozzle throat area than with the subsonic solution. One of the basic reasons for incorporating the translating ring variable geometry exit nozzle into all the engine designs was its ability to vary throat area over a very wide range. This nozzle design satisfies the throat area requirements for the supersonic solution rocket operating mode. However, it must be pointed out that there are large flow deflections in passing through the translating ring nozzle. Whether these deflections could cause choking is unknown and would have to be experimentally verified.

Engine Optimization

The basis for engine design was to maximize specific impulse for the SERJ operating mode at sea level static conditions. This optimization process proceeded as follows. A fan diameter was arbitrarily selected. The fan hub to tip ratio was taken as 0.45. With the fan total pressure ratio specified, the fan discharge flow area and Mach number are established. Based on prior composite engine design experience, the ratio of the afterburner flow area to the fan discharge area (mixer entrance area) was taken as ~ 2.2. The engine incorporates a variable geometry exit nozzle, therefore, engine specific impulse is maximized through variation of mixer geometry. The result of this optimization is presented in Figure 11 for engine 1. A mixer divergence ratio, $(A_4 + A_{4P})/A_5$, of 0.68 was selected as optimum. With this geometry, the engine specific impulse is 426.8 seconds. The resulting afterburner flow velocities were examined and judged acceptable. Engine thrust was computed. The arbitrarily selected fan/engine flow areas were then scaled to produce the specified thrust of 1.8×10^6 Newtons (404,656 lbs).

Figure 12 shows the optimization for engine 2, and engine 3 is the same in this respect. A mixer divergence ratio of .884 was selected as optimum. For engine 4, the optimization led to a mixer exit Mach number that approached sonic, as shown on Figure 13. Based on prior experience, the mixer exit Mach number was limited to 0.7 to avoid choking. As shown on Figure 14, engine 5 was similarly limited. Based on the results for engine 4, engines 6 and 7 were assumed to be optimum with the mixer exit Mach number limited to 0.7.

TABLE I. ASSUMED ENGINE TECHNOLOGY AND COMPONENT PERFORMANCE

Inlet (Air Induction System)	
Total Pressure Recovery, P_{T_2}/P_{T_0}	- Figure 5
Capture Area Ratio, A_0/A_2	- Variable Geometry Inlet
Fan	
Total Pressure Ratio (R_C = 1.3)	- Figure 6
Total Pressure Ratio (R_C = 1.8)	- Figure 7
Corrected Airflow, $W_a \sqrt{\theta_{T_2}} / P_{T_2} A$	- Figure 8
Adiabatic Efficiency, η_F	- Figure 9
Windmilling Fan Performance	- Figure 10
Bipropellant Gas Generator	
Maximum Outlet Temperature, $T_{T_{3A}}$	= 1367° K (2460° R)
Combustion Efficiency, $\eta_{C_{GG}}$	= 0.95
Propellants	
Engines No. 1, 2, 3 and 4	- H_2/O_2
Engines No. 6 and 7	- $JP\text{-}4/O_2$
Fan Turbine	
Maximum Turbine Inlet Temperature, $T_{T_{4G_{max.}}}$	= 1367° K (2460° R)
Single Stage Tip Turbine	
Total Pressure Ratio. $P_{T_{6T}}/P_{T_{7T}}$	= 3.50
Adiabatic Efficiency, η_T	= 0.85
Ejector (Primary)	
Design Chamber Pressure, $P_{T_{3A}}$	= 10.3×10^6 N/m^2 (1494 lb/in^2)
Combustion Efficiency, η_{C_P}	= 0.975
Stream Thrust Nozzle Coefficient, $C_{F_{NP}}$	= 0.984
Propellants	
Engines No. 1, 2, 3, 4, 5 and 7	- H_2/O_2
Engine No. 6	- $JP\text{-}4/O_2$

TABLE I . Continued

Mixer	
Mixing Efficiency, η_M	= 0.985
Mixer Drag Coefficient, C_{D_M}	= 0.0
Diffuser	
Diffuser Efficiency, η_D	= 0.90
Afterburner/Ramjet	
Combustion Efficiency, $\eta_{C_{A/B}}$	= 0.95
Burner Drag Coefficient, $C_{D_{A/B}}$	= 0.0
Propellants	
Engines No. 1, 2, 3, 4 and 5	- H_2
Engines No. 6 and 7	- JP-4
Exit Nozzle	
Nozzle Area Ratio, A_9/A_8	Variable Geometry Exit Nozzle
Stream Thrust Nozzle Coefficient, $C_{F_{N_{A/B}}}$	= 0.98

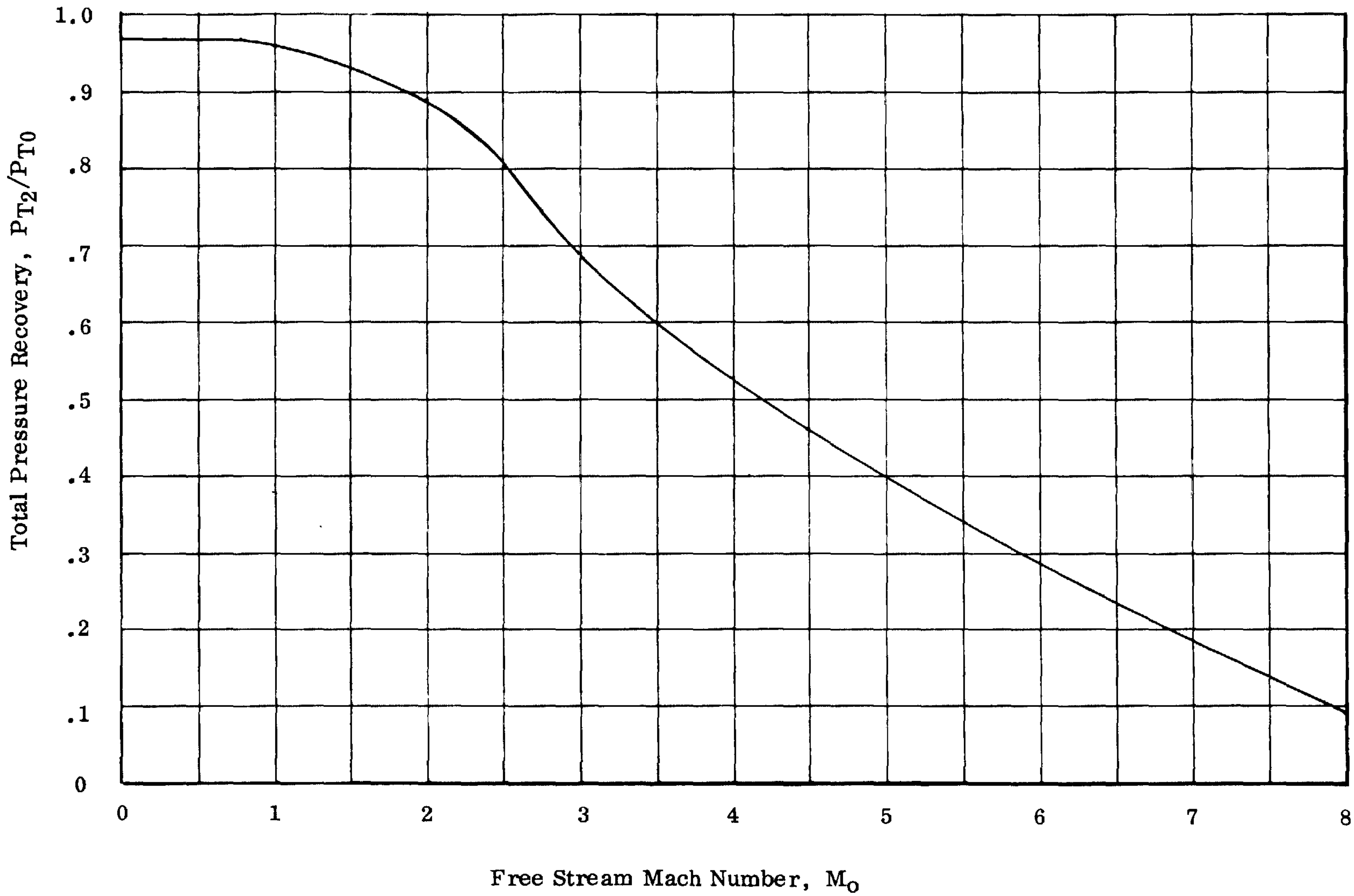


Figure 5. Inlet Total Pressure Recovery

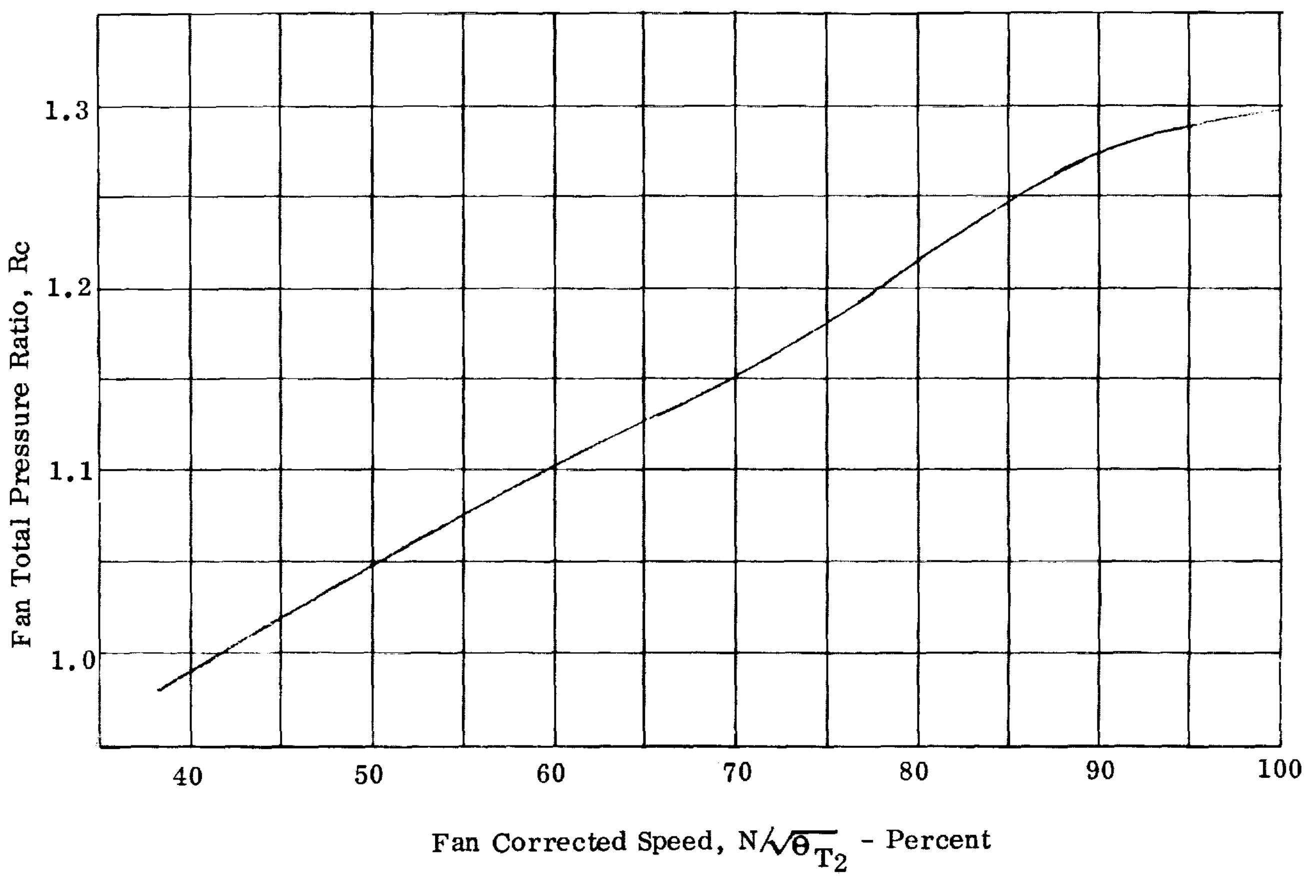


Figure 6 . Fan Pressure Ratio - 1.3 Total Pressure Ratio Design

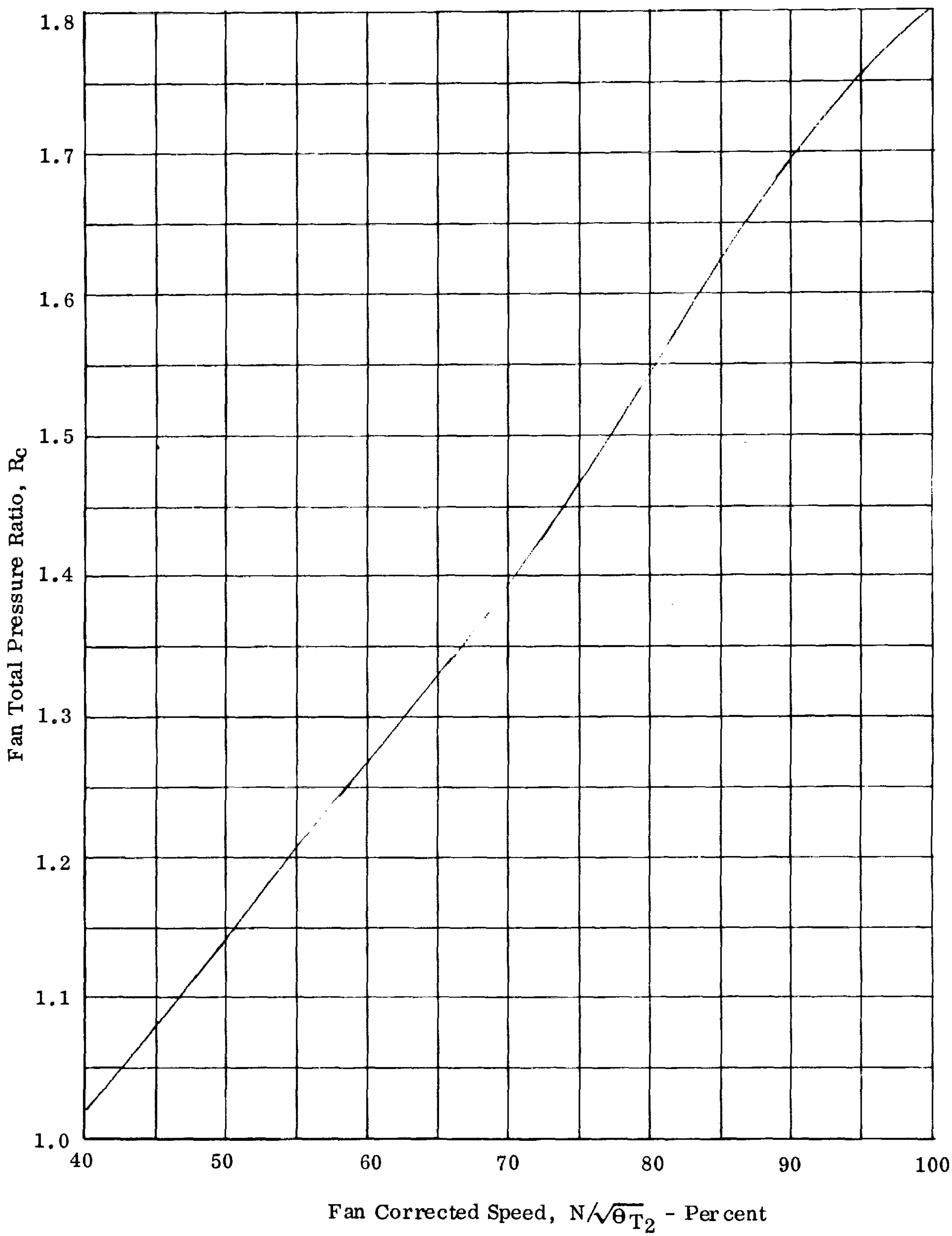


Figure 7 . Fan Pressure Ratio - 1.8 Total Pressure Ratio Design

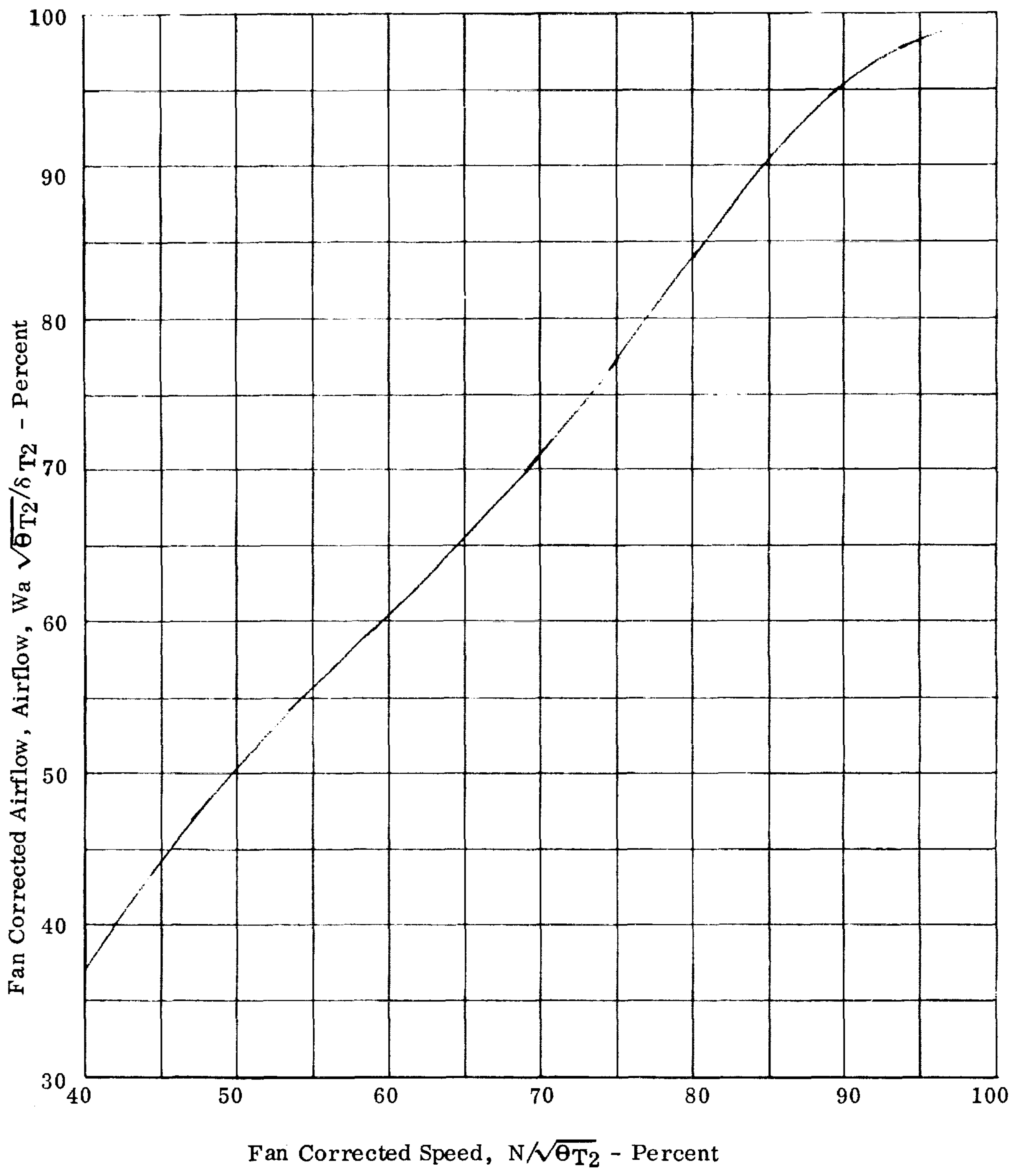


Figure 8 . Fan Corrected Airflow

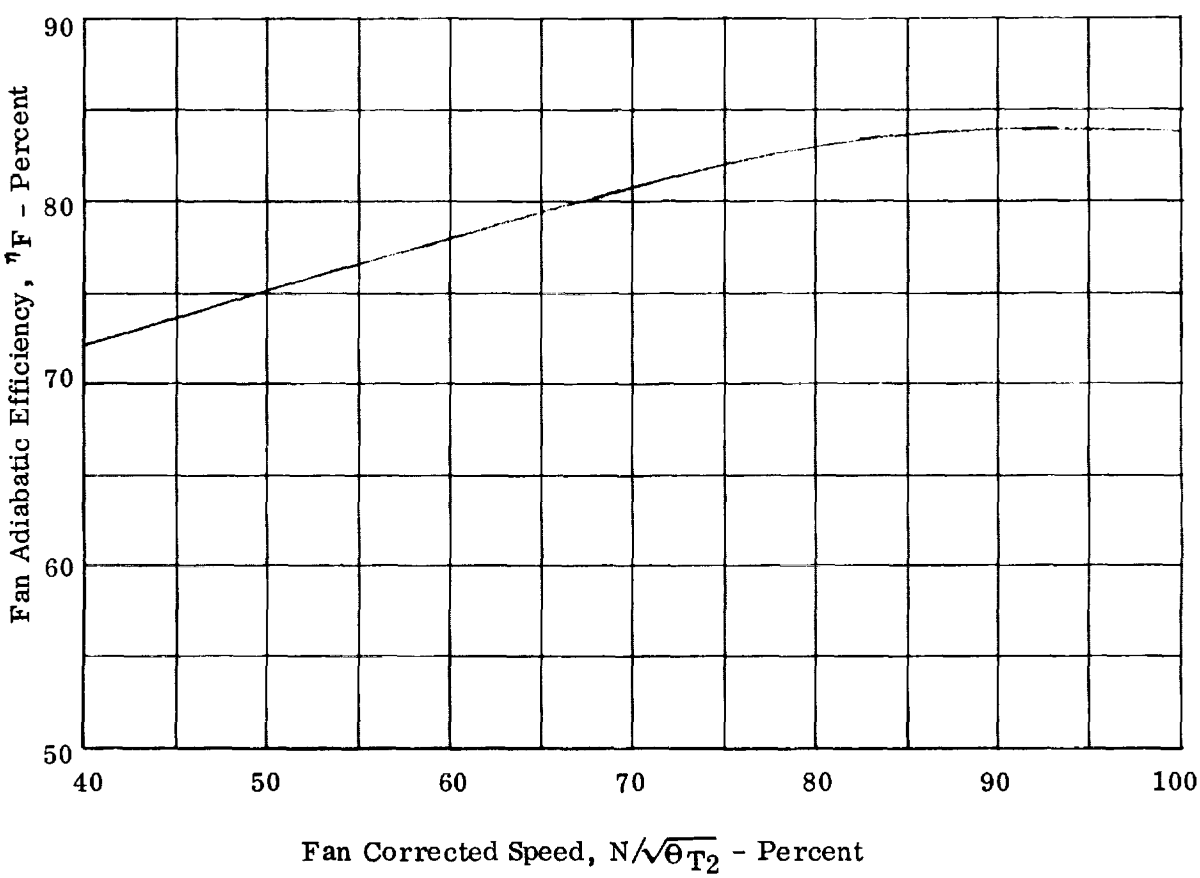


Figure 9. Fan Adiabatic Efficiency

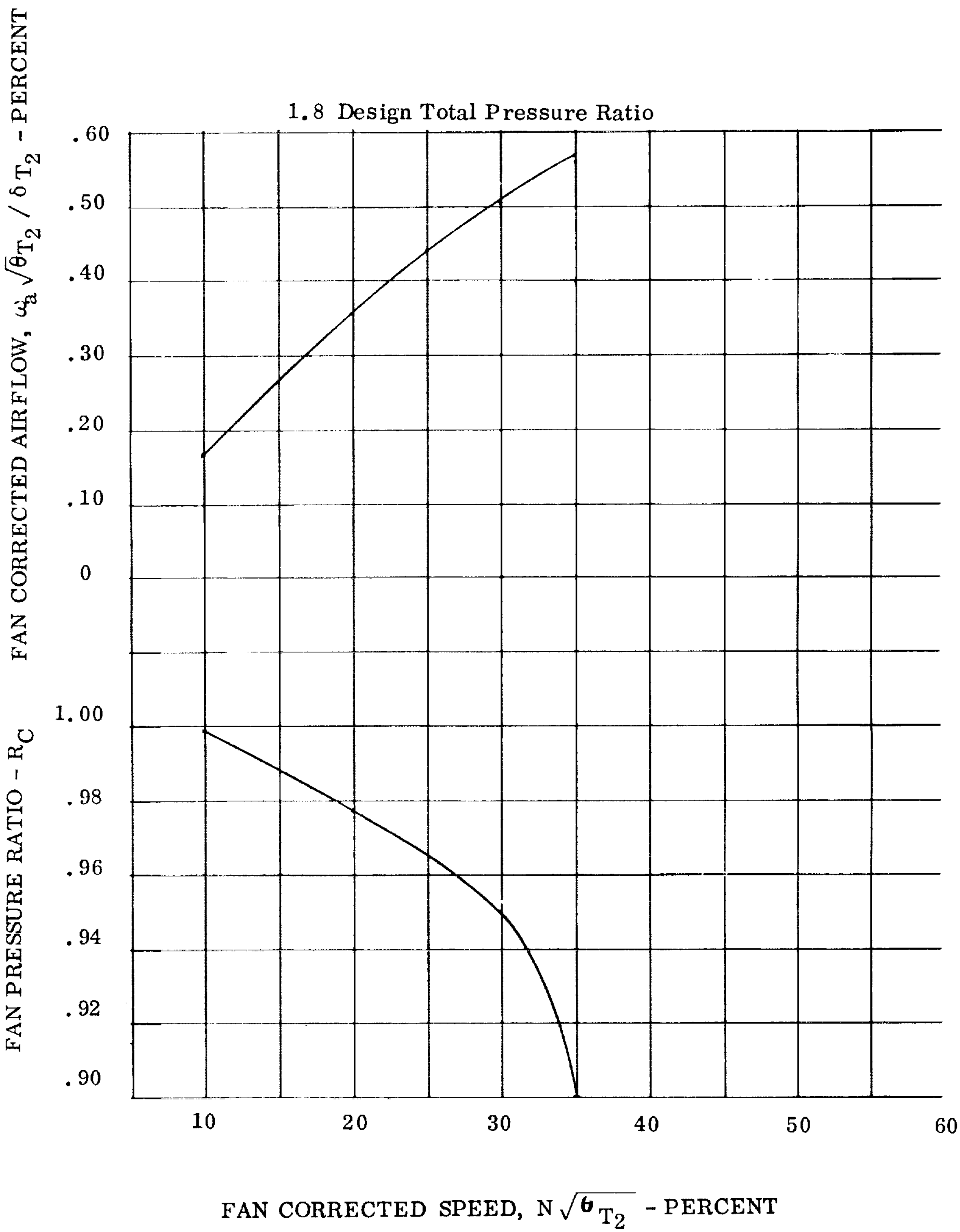


Figure 10 . Windmilling Fan Performance

Supercharged Ejector Ramjet Operating Mode
Airturborocket Engine/1.3 Fan Total Pressure Ratio
Sea Level Static Conditions

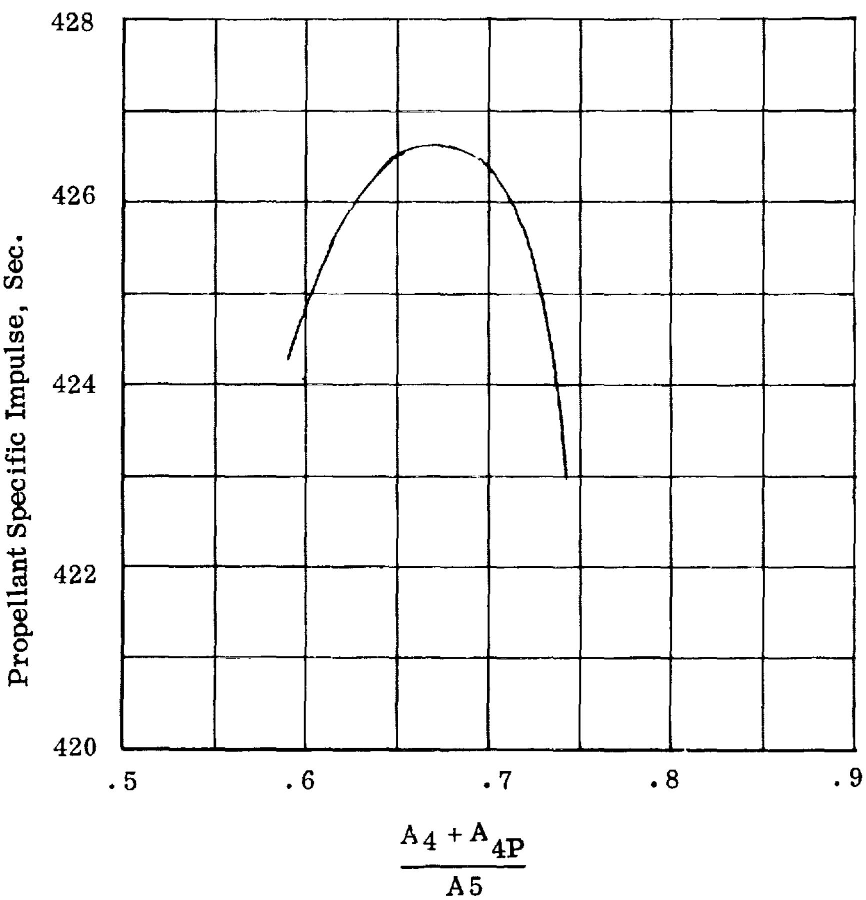


Figure 11. Engine No. 1 - Mixer Geometry Optimization

Supercharged Ejector Ramjet Operating Mode
Airturborocket Engine/1.8 Fan Total Pressure Ratio

Sea Level Static Conditions

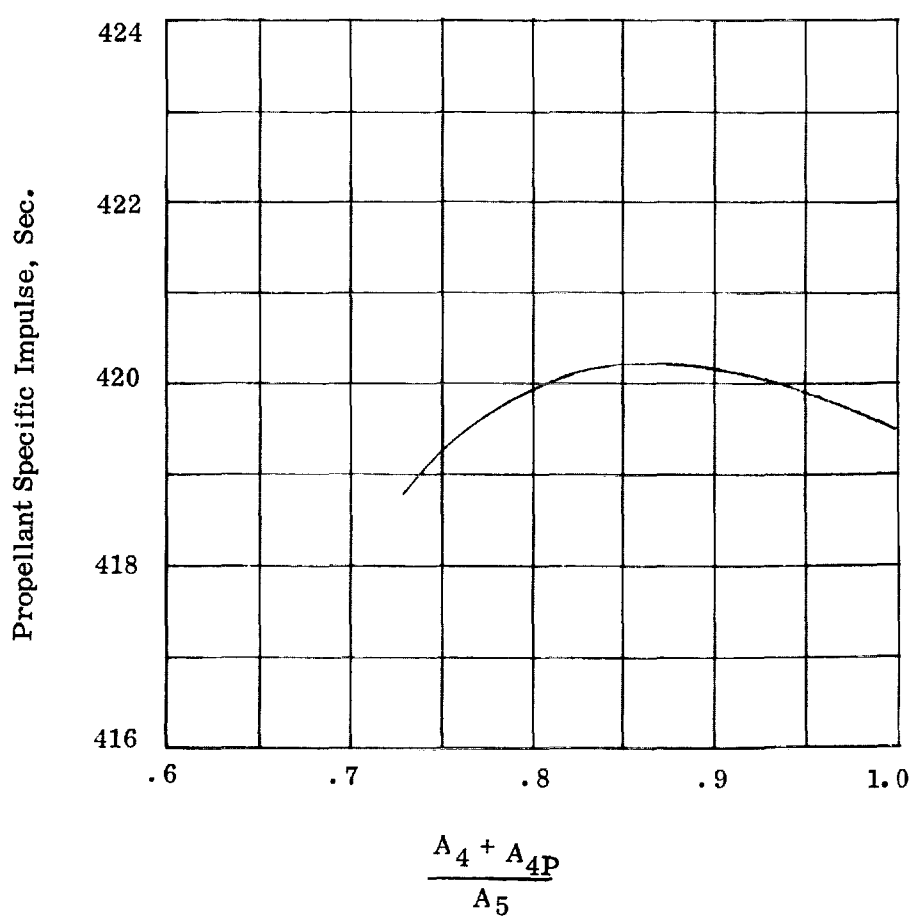


Figure 12. Engine No. 2 - Mixer Geometry Optimization

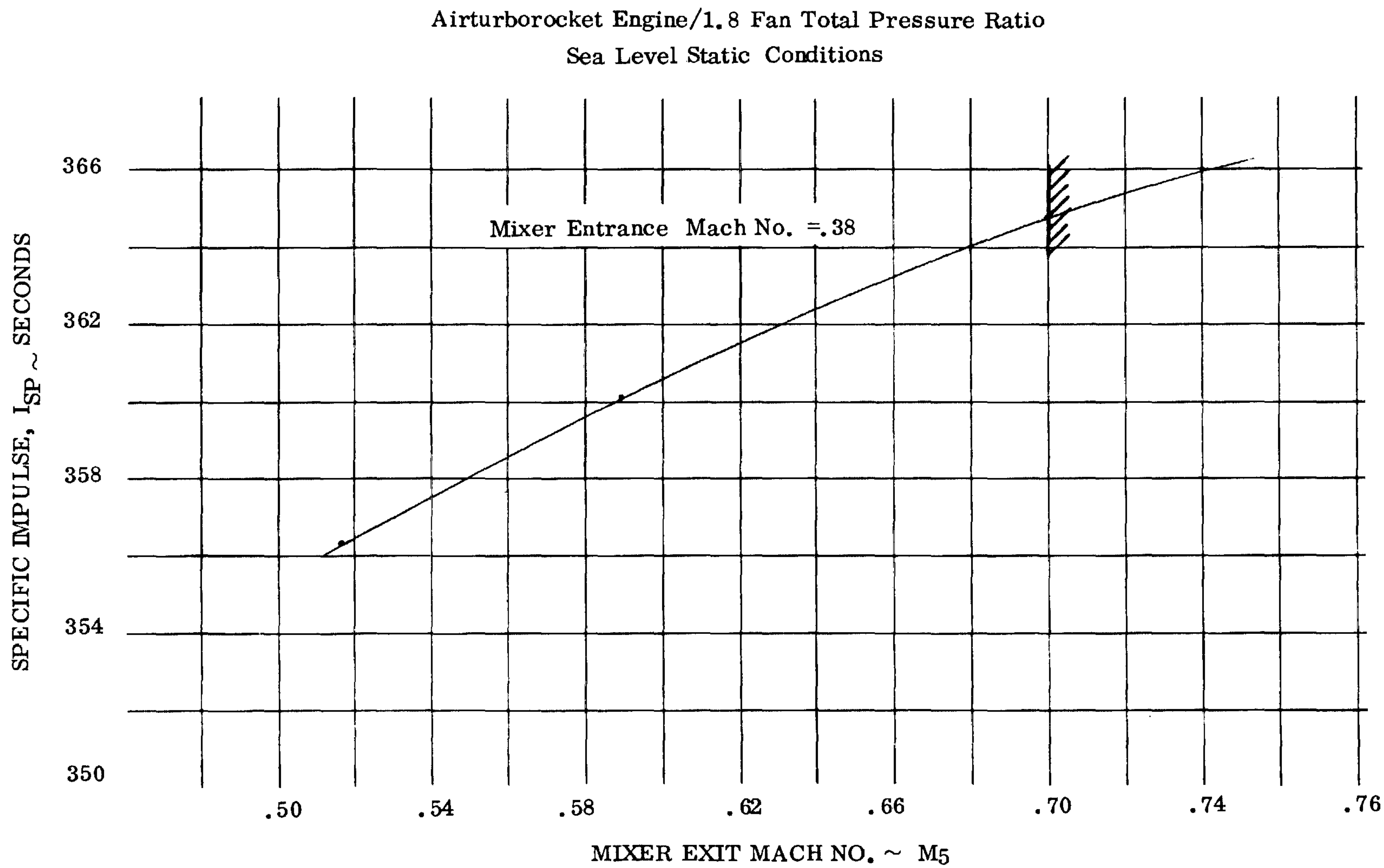


Figure 13 . Engine No. 4 - Mixer Geometry Optimization

EJECTOR RAMJET ENGINE
SEA LEVEL STATIC CONDITIONS

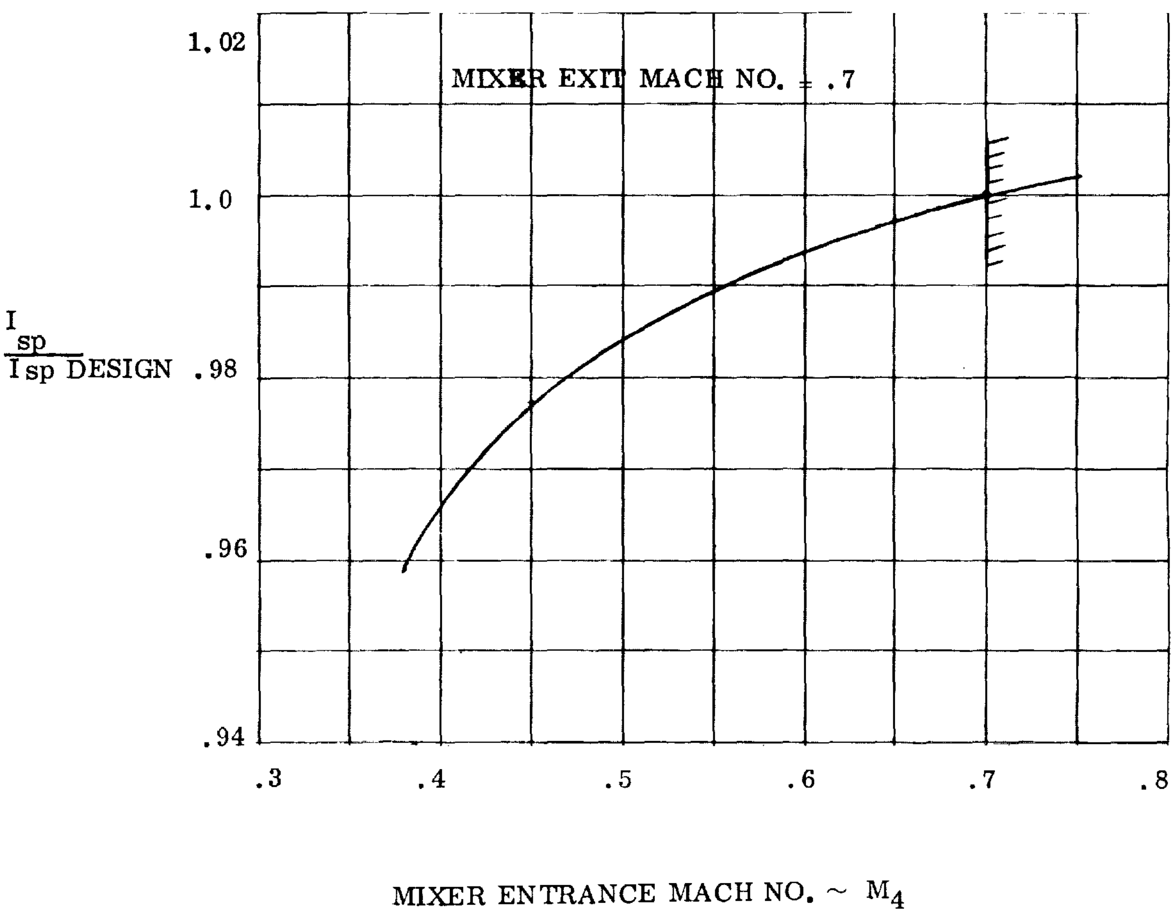


Figure 14 . Engine No. 5 - Mixer Geometry Optimization

DESIGN SPECIFICATIONS

The design specifications were selected to provide incremental effects from the SERJ engine of the previous composite engine launch vehicle study (Ref. 2).

The size of each engine was selected as follows:

Engines 1, 2, and 3	1.8×10^6N (404, 656 lb) sea level static thrust
Engines 4, 6, and 7	Same fan as engine 3
Engine 5	Same primary as engine 4

The matrix of design parameters was as follows:

TABLE II. ENGINE DESIGN SPECIFICATIONS

Study Engine	Engine Cycle	Design Fan Pressure Ratio	Design Ejector Secondary to Primary Flow Rate	Gas Generator Propellants Fuel	Gas Generator Propellants Oxidizer	Ejector (rocket) Propellants Fuel	Ejector (rocket) Propellants Oxidizer	A/B Ramjet Fuel	Fan Disposition
1	ATR	1.3	3.3	H_2	O_2	H_2	O_2	H_2	Removable
2	ATR	1.8	3.3	H_2	O_2	H_2	O_2	H_2	Removable
3	ATR	1.8	3.3	H_2	O_2	H_2	O_2	H_2	Fixed
4	ATR	1.8	1.0	H_2	O_2	H_2	O_2	H_2	Fixed
5	ERJ	-	1.0	-	-	H_2	O_2	H_2	-
6	ATR	1.8	1.0	JP-4	O_2	JP-4	O_2	JP-4	Fixed
7	ATR	1.8	1.0	JP-4	O_2	H_2	O_2	JP-4	Fixed

RESULTS

The results for each engine are presented in tables and drawings as follows:

Engine	Performance	Propellant Flow Rates	Weights	Engine Drawing	Installation Drawing
1	Table III	Table IV	Table V	Fig. 15	Fig. 16
2	Table VI	Table VII	Table VIII	Fig. 16	Fig. 18
3	Table IX	Table X	Table XI	Fig. 19	Fig. 20
4	Table XII	Table XIII	Table XIV	Fig. 21	Fig. 22
5	Table XV	Table XVI	Table XVII	Fig. 23	Fig. 22
6	Table XVIII	Table XIX	-	-	-
7	Table XX	Table XXI	Table XXII	Fig. 24	Fig. 22

The maximum airbreathing Mach number is believed to be 8.0 for engine 1, 2 and 5 and 4.5 for the remaining engines which have fixed fans. The Mach number and altitude combinations were selected to give performance data near an ascent flight path limited to 90,000 N/m^2 (1880 lb/ft^2). The performance and propellant flow rates data were output from the performance computer programs. The weights were scaled from work accomplished under previous programs already referenced.

The major components and salient features of the engine designs are presented in the engine drawings. The use of multiple ejector nozzles is assumed. Based on test experience, the mixer length/diameter was taken as 1.0. To minimize engine length and, therefore, weight, a wide angle ($\sim 20°$) diffuser is used to reduce the high mixer exit velocities to suitable afterburner/ramjet combustor entrance velocities. Some form of boundary layer control, such as vortex generators or a ribbed diffuser is required to avoid diffuser separation. As discussed earlier, a translating ring variable geometry exit nozzle is incorporated in this engine design. This type of exit nozzle was also used in Contract NAS7-377. The maximum exit nozzle flow area was largely established by experience. Improved high flight speed performance versus exit nozzle weight must be traded off to define exit nozzle size.

The installation drawings show the two dimensional variable geometry inlet designed by Lockheed under Contract NAS7-377 and used in this study. For engines 1, 2 and 5, the maximum engine capture area was established at a Mach number of 3 on the ramjet mode and the inlet was scaled to meet that capture area requirement.

The inlet for engine 2 was also assumed for engine 3, although the windmilling fan would restrict the flow thru the engine at a Mach number of 3. In contrast, the inlet for engine 3 would restrict the flow at a Mach number of 4.5. The inlets for engines 4, 6, and 7 were sized for ramjet mode operation at a Mach number of 3 with no windmilling fan losses assumed.

Some data were obtained with the primary ejectors throttled to 50 percent of the design flow rate in the SERJ mode. For engine 1, the data were computed for ejectors throttled by reducing the chamber pressure and for ejectors throttled by shutting down part of the system, thus maintaining chamber pressure at reduced flow rates. For engines 2 and 3, only the latter approach was used.

Engine 6 was analyzed only at the sea level static, SERJ mode condition. Since the hydrocarbon ejectors would have to be larger than hydrogen ejectors, the mixer would be larger and heavier. This fact and the lack of a mission for which this engine appeared to be attractive led to the conclusion to stop further work on this engine.

TABLE III
ENGINE NO. 1/PERFORMANCE
AIRTURBOROCKET ENGINE/1.3 FAN TOTAL PRESSURE RATIO

M_o	Altitude		Operating Mode	Thrust		Isp		Capture Area A_0	
	ft.	m		lbs.	Newtons	sec.	Newton-sec / k_g	ft^2	m^2
0	0		SERJ-100% W_p	404,656	1.8×10^6	426.8	4.184×10^3	-	-
0.3	0	0		401,767	1.787×10^6	415.5	4.075×10^3	91.84	8.532
0.8	1,969	600		477,274	2.132×10^6	457.8	4.508×10^3	45.59	4.235
0.8	11,811	3,600		436,436	1.941×10^6	462.2	4.553×10^3	46.33	4.304
1.0	2,953	900		535,585	2.382×10^6	489.4	4.799×10^3	42.84	3.980
1.0	14,764	4,500		468,378	2.083×10^6	486.2	4.767×10^3	43.76	4.065
1.3	9,843	3,000		610,849	2.717×10^6	543.5	5.330×10^3	44.09	4.096
1.3	19,685	6,000		530,603	2.360×10^6	529.8	5.195×10^3	43.94	4.082
2.6	42,500	12,954		763.552	3.396×10^6	684.5	6.712×10^3	82.41	7.656
0.8*	1,969	600	SERJ-W_p=50%	332,664	1.480×10^6	486.6	4.772×10^3	45.59	4.235
0.8**	1,969	600		304,548	1.355×10^6	445.5	4.368×10^3	45.59	4.235
0	0	0	FRJ	87,884***	$.391 \times 10^6$	438.5	4.301×10^3	-	-
0.8	1,969	600		195,560	$.870 \times 10^6$	602.4	5.908×10^3	45.59	4.235
1.0	2,953	900		263,964	1.174×10^6	701.3	6.876×10^3	42.84	3.980
1.0	14,764	4,500		176,173	$.784 \times 10^6$	718.1	7.045×10^3	43.76	4.065
1.3	9,843	3,000		336,559	1.497×10^6	829.1	8.131×10^3	44.09	4.096
2.0	29,528	9,000		437,582	1.946×10^6	1050.8	1.030×10^4	61.00	5.667
2.0	39,370	12,000		297,003	1.321×10^6	1065.8	1.045×10^4	63.08	5.860
2.6	42,500	12,954		462,157	2.056×10^6	1102.6	1.140×10^4	82.41	7.656
3.0	49,213	15,000		437,436	1.946×10^6	1195.5	1.172×10^4	93.80	8.714
2.6	42,500	12,954	RJ	417,738	1.858×10^6	3739.8	3.480×10^4	91.00	8.454
3.0	49,213	15,000		480,135	2.136×10^6	3745.0	3.673×10^4	124.50	11.566
5.0	68,898	21,000		311,579	1.386×10^6	3705.3	3.635×10^4	124.50	11.566
5.0	78,740	24,000		193,520	$.861 \times 10^6$	3690.1	3.619×10^4	124.50	11.566
8.0	103,340	31,500		65,270	$.290 \times 10^6$	2446.9	2.451×10^4	124.50	11.566
-	Subsonic Solution		Rocket $A_9 = 22.56\ m^2 = 242.82\ ft^2$	295,012	1.312×10^6	410.9	4.029×10^3	O/F	= 7.936
				312,933	1.392×10^6	435.8	4.275×10^3		= 5.25
				310,865	1.383×10^6	433.0	4.247×10^3		= 3.50
	Supersonic Solution			315,690	1.404×10^6	439.7	4.311×10^3		= 7.936
				329,490	1.466×10^6	458.9	4.502×10^3		= 5.25
				321,894	1.432×10^6	448.3	4.397×10^3		= 3.50

* Primary Pressure = 10.3×10^6 N/m^2 (1494 psi); Primary Flow Area Reduced 50%

** Primary Pressure = 5.15×10^6 N/m^2 (747 psi); Original Primary Flow Area

*** ϕ_{AB} = .434

TABLE IV
ENGINE NO. 1/COMPONENT PROPELLANT FLOW RATES
AIRTURBOROCKET ENGINE/1.3 FAN TOTAL PRESSURE RATIO

M_0	Altitude		Operating Mode	Gas Generator				Primary				Afterburner		Propellant Flow Rates			
				O_2		H_2		O_2		H_2		H_2		O_2		H_2	
	ft.	m		lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec
0	0		SERJ-100% W_p	159.75	72.46	2.35	1.07	637.59	289.21	80.33	36.44	68.53	31.08	797.34	361.67	151.21	68.59
0.3	0	0		173.20	78.59	2.55	1.15					73.12	33.17	810.85	367.80	156.00	70.76
0.8	1,969	600		230.12	104.38	3.38	1.53					91.13	41.34	867.71	393.59	174.84	79.31
0.8	11,811	3,600		159.75	72.46	2.35	1.06					59.77	27.11	797.34	361.67	142.45	64.61
1.0	2,953	900		268.69	121.87	3.95	1.79					103.77	47.07	906.28	411.08	188.05	85.30
1.0	14,764	4,500		172.75	78.36	2.54	1.15					70.05	31.77	810.34	367.57	152.92	69.36
1.3	9,843	3,000		291.73	132.32	4.29	1.95					109.90	49.85	929.32	421.53	194.52	88.23
1.3	19,685	6,000		202.13	91.68	2.97	1.35					78.53	35.62	839.72	380.89	161.83	73.41
2.6	42,500	12,954		284.46	129.03	4.18	1.90					108.87	49.38	922.05	418.24	193.38	87.72
0.8*	1,969	600	SERJ-W_p=50%	230.13	104.39	3.38	1.53	318.79	144.60	40.17	18.22	91.12	41.33	548.92	248.99	134.67	61.09
0.8**	1,969	600		230.13	104.39	3.38	1.53	318.79	144.60	40.17	18.22	91.12	41.33	548.92	248.99	134.67	61.09
0	0	0	FRJ	163.70	74.25	2.40	1.09	-	-	-	-	34.52*	15.57	163.70	74.25	36.72	16.66
0.3	1,969	600		230.13	104.39	3.38	1.53	-	-	-	-	91.13	41.34	230.13	104.39	94.51	42.87
1.0	2,953	900		268.69	121.88	3.95	1.79	-	-	-	-	103.77	47.07	268.69	121.88	107.72	48.86
1.0	14,764	4,500		172.75	78.36	2.54	1.15	-	-	-	-	70.05	31.78	172.75	78.36	72.59	32.93
1.3	9,843	3,000		291.72	132.32	4.29	1.95	-	-	-	-	109.90	49.85	291.72	132.32	114.19	51.80
2.0	29,528	9,000		300.70	136.40	4.42	2.00	-	-	-	-	111.30	50.49	300.70	136.40	115.72	52.49
2.0	39,370	12,000		201.09	91.21	2.95	1.34	-	-	-	-	74.64	33.85	201.09	91.21	77.59	35.19
2.6	42,500	12,954		284.46	129.03	4.18	1.90	-	-	-	-	108.87	49.37	284.46	129.03	113.05	51.27
3.0	49,213	15,000		258.74	117.36	3.80	1.72	-	-	-	-	103.37	46.90	258.74	117.36	107.17	48.62
2.6	42,500	12,954	RJ	-	-	-	-	-	-	-	-	117.70	53.39	-	-	117.70	53.39
3.0	49,213	15,000		-	-	-	-	-	-	-	-	128.21	58.16	-	-	128.21	58.16
5.0	68,898	21,000		-	-	-	-	-	-	-	-	84.07	38.13	-	-	84.07	38.13
5.0	78,740	24,000		-	-	-	-	-	-	-	-	52.44	23.79	-	-	52.44	23.79
8.0	103,340	31,500		-	-	-	-	-	-	-	-	26.67	12.10	-	-	26.67	11.63
-	Subsonic Solution		Rocket	-	-	-	-	637.59	289.21	80.33	36.44	-	-	637.59	289.21	80.33	36.44
			A_e = 22.56 m^2	-	-	-	-	703.05	273.54	114.87	52.10	-	-	603.05	273.54	114.87	52.10
			= 242.82 ft^2	-	-	-	-	558.38	253.28	159.54	72.37	-	-	558.38	253.28	159.54	72.37
	Supersonic Solution			-	-	-	-	637.59	289.21	80.33	36.44	-	-	637.59	289.21	80.33	36.44
				-	-	-	-	603.05	273.54	114.87	52.10	-	-	603.05	273.54	114.87	52.10
				-	-	-	-	558.38	253.28	159.54	72.37	-	-	558.38	253.28	159.54	72.37

*ϕ_{AB} = .434

TABLE V

ENGINE NO. 1 /WEIGHT STATEMENT
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

Engine Component	Weight	
	kg	lb
Fan Assembly	1,136	2,504
Fan Housing	1,560	3,440
Primary Rockets	1,635	3,604
Mixer/Afterburner	2,527	5,570
Exit Nozzle	3,482	7,676
Controls	276	608
Total Engine Weight	10,616	23,402
Inlet Weight	6,777	14,940
Installed Engine Weight	17,393	38,342

$\text{Thrust/Weight*}_{\text{Engine}}$ = 17.3

$\text{Thrust/Weight*}_{\text{Installed Engine}}$ = 10.6

*SERJ Operating Mode/Sea Level Static Conditions

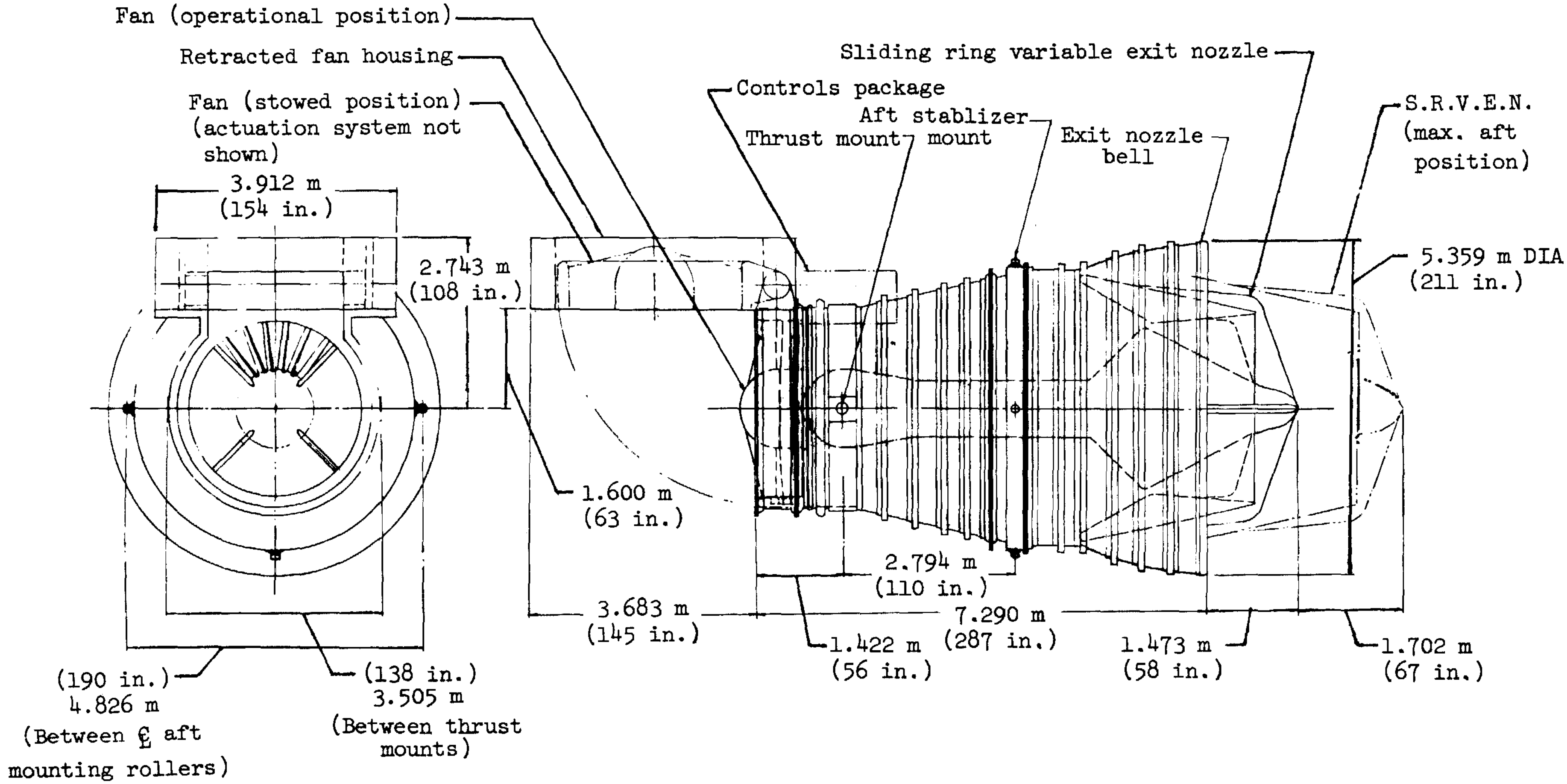


Figure 15.- Engine 1 outboard profile - airturborocket SERJ cycle.

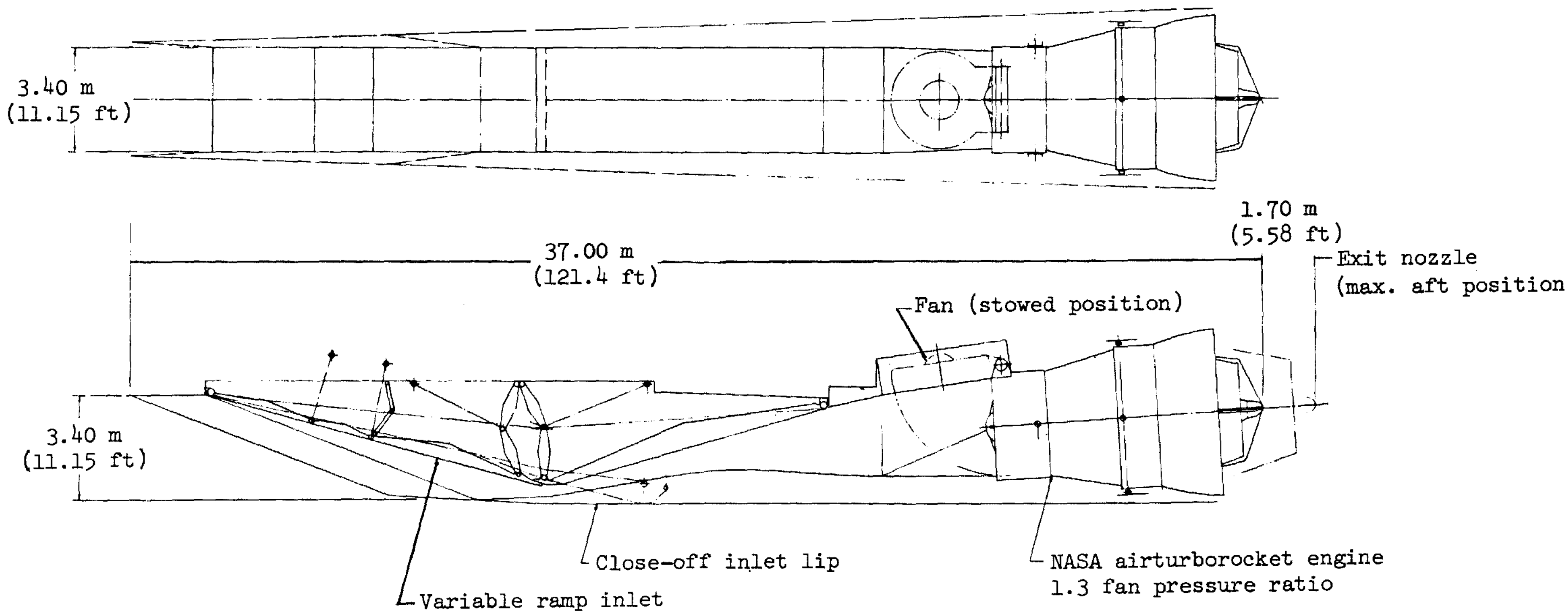


Figure 16.- Engine 1 installation - airturborocket SERJ cycle.

TABLE VI

ENGINE 2. /PERFORMANCE

AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude ft	Altitude m	Operating Mode	Thrust lbs	Thrust Newtons	Isp sec.	Isp $\frac{\text{Newton-sec}}{\text{kg}}$	Capture Area A_o ft^2	Capture Area A_o m^2
0	0	0	SERJ-100% W_p	404,656	1.8×10^6	420.7	4.126×10^3	-	-
0.3	0	0		401,475	1.786×10^6	410.4	4.025×10^3	71.26	6.620
0.8	1,969	600		472,165	2.100×10^6	429.7	4.218×10^3	35.37	3.266
0.8	11,811	3,600		412,352	1.834×10^6	432.9	4.223×10^3	35.91	3.336
1.0	2,953	900		523,731	2.330×10^6	445.5	4.370×10^3	33.24	3.088
1.0	14,764	4,500		438,575	1.951×10^6	447.7	4.391×10^3	33.89	3.148
1.3	9,843	3,000		578,070	2.571×10^6	474.3	4.651×10^3	34.21	3.178
1.3	19,085	6,000	↓	490,231	2.181×10^6	473.3	4.642×10^3	35.03	3.254
0.8	1,969	600	SERJ-50% W_p	360,152	1.602×10^6	454.0	4.460×10^3	35.37	3.286
0	0	0	FRJ	182,931	$.814 \times 10^6$	518.0	5.081×10^3	-	-
0.3	0	0		182,721	$.813 \times 10^6$	494.5	4.786×10^3	71.26	6.620
0.8	11,811	3,600		184,562	$.821 \times 10^6$	536.9	5.265×10^3	35.91	3.336
0.8	1,969	600		252,267	1.122×10^6	515.9	5.059×10^3	35.37	3.286
1.0	2,953	900		302,613	1.346×10^6	533.6	5.235×10^3	33.24	3.088
1.0	14,764	4,500		207,973	$.925 \times 10^6$	560.8	5.499×10^3	33.89	3.148
1.3	9,843	3,000		348,611	1.551×10^6	571.5	5.606×10^3	34.21	3.178
2.0	29,528	9,000		410,879	1.828×10^6	647.3	6.349×10^3	47.44	4.407
2.0	39,370	12,000		280,452	1.248×10^6	663.4	6.506×10^3	49.21	4.572
2.5	29,528	9,000		655,422	2.915×10^6	658.2	6.454×10^3	59.02	5.483
2.5	39,370	12,000		451,712	2.009×10^6	668.6	6.555×10^3	61.57	5.720
2.5	49,213	15,000		282,480	1.256×10^6	669.7	6.519×10^3	61.57	5,720
2.6	42,500	12,954		421,940	1.877×10^6	667.8	6.550×10^3	64.15	5,959
3.0	49,213	15,000	↓	395,127	1.758×10^6	656.3	6.437×10^3	72.78	6,761
2.6	42,500	12,954	RJ	356,240	1.585×10^6	3739.9	3.669×10^4	77.60	7.209
3.0	49,213	15,000		408,276	1.816×10^6	3747.4	3.675×10^4	105.80	9.829
4.5	49,213	15,000		626,280	2.786×10^6	3832.3	3.758×10^4	105.60	9.629
	68,893	21,000		246,493	1.096×10^6	3833.4	3.757×10^4	105.80	9.829
5.0	68,898	21,000		264,723	1.178×10^6	3705.2	3.625×10^4	105.80	9.829
5.0	78,740	24,000		164,450	$.732 \times 10^6$	3690.1	3.620×10^4	105.80	9.829
8.0	103,340	31,500	↓	55,468	$.247 \times 10^6$	2446.9	2.403×10^4	105.80	9.829
-	Subsonic Solution		Rocket (Vacuum)	248,964	1.107×10^6	409.0	3.533×10^3	O/F = 7.936	
	↓			264,121	1.175×10^6	433.9	3.750×10^3	= 5.25	
			$A_9 = 19.59\ m^2$	262,356	1.167×10^6	431.0	3.725×10^3	= 3.5	
	Supersonic Solution		$= 210.84\ ft^2$	257,121	1.144×10^6	422.4	3.651×10^3	= 7.936	
	↓			276,905	1.232×10^6	455.0	3.932×10^3	= 5.25	
				257,730	1.146×10^6	423.4	3.658×10^3	= 3.50	

TABLE VII

ENGINE NO. 2/COMPONENT PROPELLANT FLOW RATES
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude ft.	Altitude m	Operating Mode	Gas Generator O_2 lb/sec	Gas Generator O_2 k_g/sec	Gas Generator H_2 lb/sec	Gas Generator H_2 k_g/sec	Primary O_2 lb/sec	Primary O_2 k_g/sec	Primary H_2 lb/sec	Primary H_2 k_g/sec	Afterburner H_2 lb/sec	Afterburner H_2 k_g/sec	Total Flow Rates O_2 lb/sec	Total Flow Rates O_2 k_g/sec	Total Flow Rates H_2 lb/sec	Total Flow Rates H_2 k_g/sec
0	0	0	SERJ-100% W_p	291.31	132.14	4.28	1.94	540.60	245.21	68.11	30.89	57.57	26.12	631.91	377.35	129.97	55.95
0.3	0	0		305.56	138.60	4.49	2.04					59.47	26.98	846.16	383.81	132.07	59.91
0.8	1,969	600		406.36	184.32	5.97	2.71					76.66	34.77	946.96	429.54	150.74	68.27
0.8	11,811	3,600		284.67	129.13	4.18	1.90					59.94	27.19	825.27	374.34	132.24	59.95
1.0	2,953	900		472.46	214.31	6.94	3.15					87.42	39.66	1,013.06	459.52	162.47	73.70
1.0	14,764	4,500		307.55	139.51	4.52	2.05					58.78	26.67	848.15	384.72	131.41	59.61
1.3	9,843	3,000		509.80	231.25	7.49	3.40					92.68	42.04	1,050.40	476.46	165.28	76.33
1.3	19,685	6,000		355.67	161.33	5.23	2.37					66.13	30.00	896.27	406.54	139.47	63.26
0.8	1,969	600	SERJ-50% W_p	406.36	184.32	5.97	2.71	270.30	122.61	34.06	15.45	76.66	34.77	676.66	306.93	116.69	52.93
0	0	0	FRJ	291.31	132.14	4.28	1.94	-	-	-	-	57.57	26.11	291.31	132.14	61.85	28.05
0.3	0	0		305.55	138.60	4.49	2.03	-	-	-	-	59.47	26.98	305.55	138.60	63.96	29.01
0.8	11,811	3,600		284.67	129.12	4.18	1.90	-	-	-	-	54.94	24.92	284.67	129.12	59.12	26.82
0.8	1,969	600		406.36	184.32	5.97	2.71	-	-	-	-	76.66	34.77	406.36	184.32	82.63	37.48
1.0	2,953	900		472.46	214.30	6.94	3.15	-	-	-	-	87.42	39.65	472.46	214.31	94.36	42.80
1.0	14,764	4,500		301.54	136.78	4.52	2.05	-	-	-	-	58.78	26.66	307.54	139.50	63.30	28.71
1.3	9,843	3,000		509.80	231.24	7.49	3.40	-	-	-	-	92.68	42.04	509.80	231.24	100.17	45.44
2.0	29,528	9,000		532.60	241.58	7.82	3.55	-	-	-	-	94.36	42.80	532.60	241.58	102.19	46.35
2.0	39,370	12,000		354.14	160.64	5.20	2.36	-	-	-	-	63.38	28.75	354.14	160.64	69.59	31.11
2.5	29,528	9,000		836.56	379.46	12.29	5.57	-	-	-	-	146.96	66.66	836.56	379.46	159.25	72.23
2.5	39,370	12,000		567.79	257.55	8.34	3.78	-	-	-	-	99.49	45.13	567.79	257.55	107.83	48.91
2.5	49,213	15,000		354.46	160.78	5.21	2.36	-	-	-	-	62.11	28.18	354.46	162.14	67.32	30.54
2.6	42,500	12,954		531.16	240.93	7.80	3.54	-	-	-	-	92.84	42.11	531.16	240.93	100.65	45.65
3.0	49,213	15,000		506.43	229.71	7.44	3.38	-	-	-	-	88.19	40.00	506.43	229.71	95.63	43.38
2.6	42,500	12,954	RJ	-	-	-	-	-	-	-	-	95.25	53.20	-	-	95.25	43.20
3.0	49,213	15,000		-	-	-	-	-	-	-	-	108.95	49.42	-	-	108.95	49.42
4.5	49,213	15,000		-	-	-	-	-	-	-	-	163.42	74.13	-	-	163.42	74.13
4.5	68,898	21,000		-	-	-	-	-	-	-	-	64.30	29.17	-	-	64.30	29.17
5.0	68,898	21,000		-	-	-	-	-	-	-	-	71.45	32.41	-	-	71.45	32.41
5.6	78,740	24,000		-	-	-	-	-	-	-	-	44.57	20.22	-	-	44.57	20.22
6.0	103,340	31,500		-	-	-	-	-	-	-	-	22.67	10.28	-	-	22.67	10.28
-	Subsonic Solution		Rocket (Vacuum)					540.60	245.21	68.11	30.89	-	-	540.60	245.21	68.11	30.89
			A_9 = 19.59 m^2					511.32	231.93	97.39	44.18	-	-	511.32	231.93	97.39	44.18
			210.84 ft^2					473.44	214.75	135.27	61.36	-	-	473.44	214.75	135.27	61.36
	Supersonic Solution							540.60	245.21	68.11	30.89	-	-	540.60	245.21	68.11	30.89
								511.32	231.93	97.39	44.18	-	-	511.32	231.93	97.39	44.18
								473.44	214.75	135.27	61.36	-	-	473.44	214.75	135.27	61.36

TABLE VIII

ENGINE NO. 2 /WEIGHT STATEMENT
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

Engine Component	Weight	
	kg	lb
Fan Assembly	2,272	5,008
Fan Housing	1,326	2,924
Primary Rockets	1,549	3,414
Mixer/Afterburner	2,255	4,972
Exit Nozzle	2,960	6,525
Controls	276	608
Total Engine Weight	10,638	23,451
Inlet Weight	5,759	12,696
Installed Engine Weight	16,397	36,147

$\text{Thrust/Weight*}_{\text{Engine}}$ = 17.3

$\text{Thrust/Weight*}_{\text{Installed Engine}}$ = 11.2

*SERJ Operating Mode/Sea Level Static Conditions

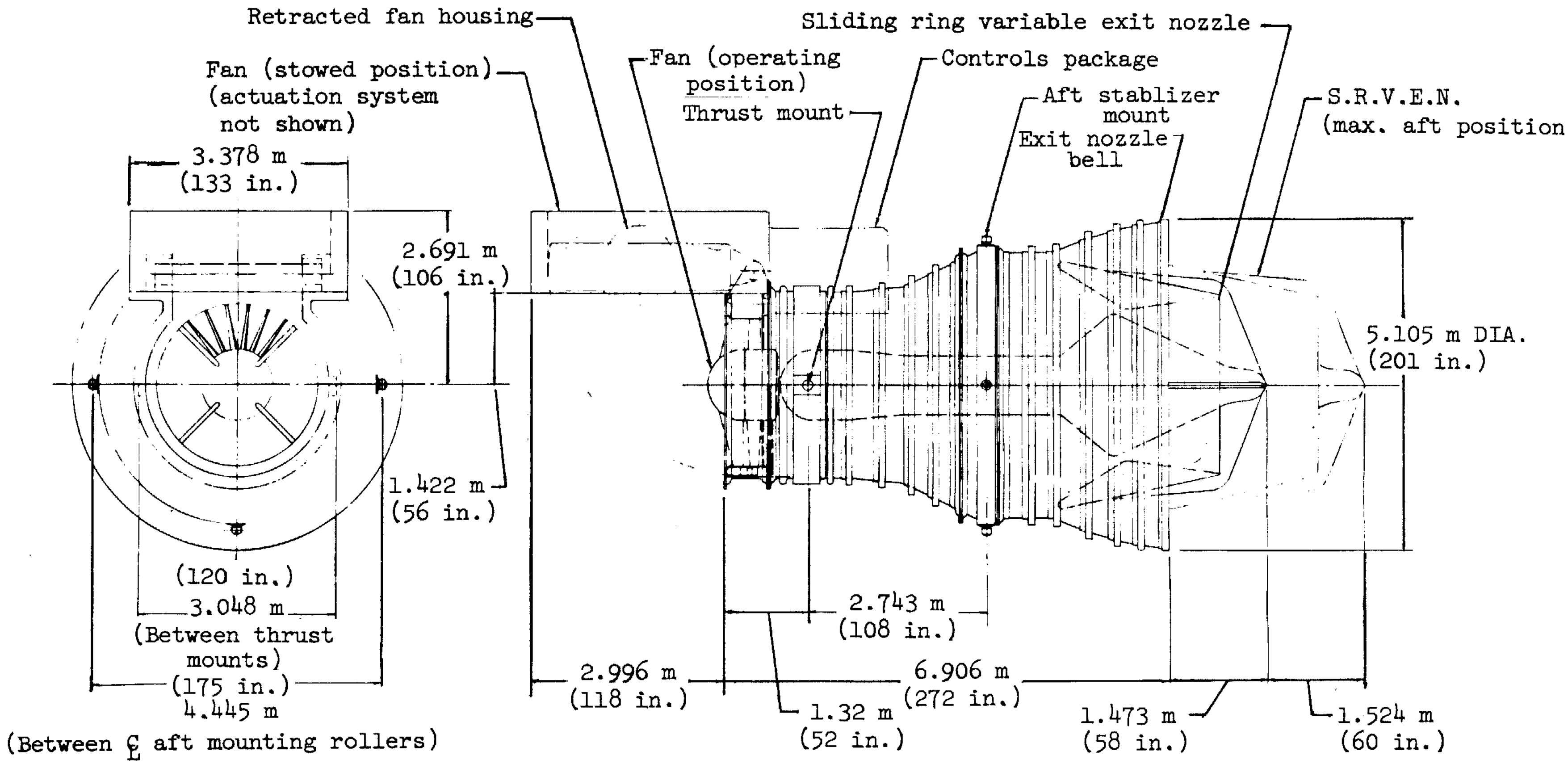


Figure 17.- Engine 2 outbouard profiles - airturborocket SERJ cycle.

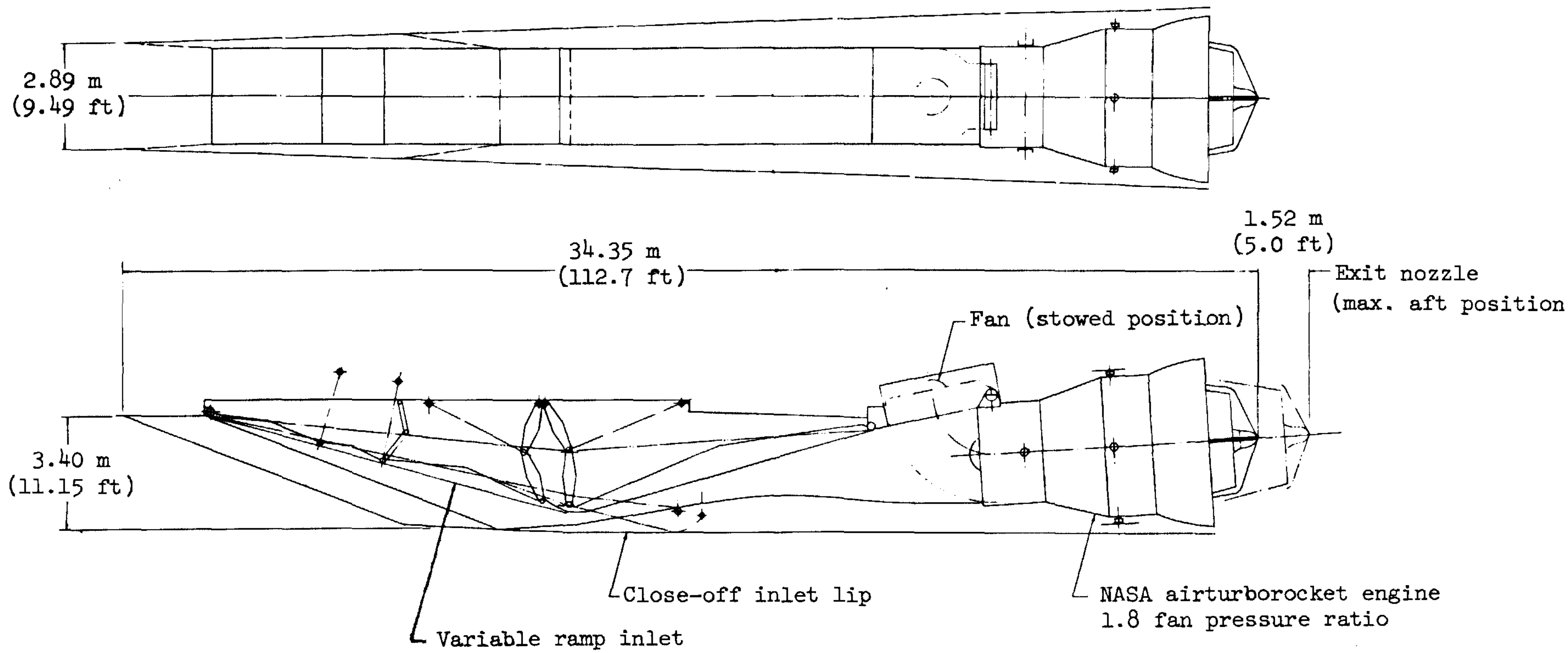


Figure 18.- Engine 2 installation - airturborocket SERJ cycle.

TABLE IX

ENGINE NO. 3/PERFORMANCE
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude ft.	Altitude m	Operating Mode	Thrust lbs'	Thrust Newtons	Isp sec.	Isp Newton-sec/kg	Capture Area A_0 ft²	Capture Area A_0 m²
0	0	0	SERJ-100% W_p	404,656	1.8×10^6	420.7	4.126×10^3	-	-
0.3	0	0	↓	401,475	1.786×10^6	410.4	4.025×10^3	71.26	6.620
0.8	1,969	600	↓	472,165	2.100×10^6	429.7	4.218×10^3	35.37	3.286
0.8	11,811	3,600	↓	412,362	$1.834 \times 10^{[illegible]}$	432.9	4.223×10^3	35.91	3.336
1.0	2,953	900	↓	523,731	2.330×10^6	445.5	4.370×10^3	33.24	3.088
1.0	14,764	4,500	↓	438,575	1.951×10^6	447.7	4.391×10^3	33.89	3.148
1.3	9,843	3,000	↓	578,070	2.571×10^6	474.3	4.651×10^3	34.21	3.178
1.3	19,685	6,000	↓	490,231	2.181×10^6	473.3	4.642×10^3	35.03	3.254
0.8	1,969	600	SERJ-50% W_p	360,182	1.602×10^6	454.0	4.480×10^3	35.37	3.286
0	0	0	FRJ	182,931	$.814 \times 10^6$	518.0	5.081×10^3	-	-
0.3	0	0	↓	182,721	$.813 \times 10^6$	494.5	4.786×10^3	71.26	6.620
0.8	11,811	3,600	↓	184,582	$.821 \times 10^6$	536.9	5.265×10^3	35.91	3.336
0.8	1,969	600	↓	252,267	1.122×10^6	515.9	5.059×10^3	35.37	3.286
1.0	2,953	900	↓	302,613	1.346×10^6	533.8	5.235×10^3	33.24	3.088
1.0	14,764	4,500	↓	207,973	$.925 \times 10^6$	560.8	5.499×10^3	33.89	3.148
1.3	9,843	3,000	↓	348,611	1.551×10^6	571.5	5.606×10^3	34.21	3.178
2.0	29,528	9,000	↓	410,879	1.828×10^6	647.3	6.349×10^3	47.44	4.407
2.0	39,370	12,000	↓	280,452	1.248×10^6	663.4	6.508×10^3	49.21	4.572
2.5	29,528	9,000	↓	655,422	2.915×10^6	658.2	6.454×10^3	59.02	5.483
2.5	39,370	12,000	↓	451,712	2.009×10^6	668.6	6.555×10^3	61.57	5.720
2.5	49,213	15,000	↓	282,480	1.256×10^6	669.7	6.519×10^3	61.57	5.720
2.6	42,500	12,954	↓	421,940	1.[illegible]77 x 10[illegible]	667.8	6.560×10^3	64.15	5.959
3.0	49,213	15,000	↓	395,127	1.758×10^6	656.3	6.437×10^3	72.78	6.761
2.0	29,528	9,000	RJ	167,047	7.431×10^5	3240.9	3.178×10^4	30.43	2,827
2.0	39,370	12,000	↓	109,416	4.867×10^5	3282.1	3.219×10^4	30.43	2.827
2.5	29,528	9,000	↓	335,566	1.493×10^6	3618.2	3.549×10^4	43.80	4.069
2.5	39,370	12,000	↓	220,181	9.794×10^5	3670.5	3.599×10^4	43.80	4.069
2.5	49,213	15,000	↓	138,104	6.143×10^5	3673.9	3.603×10^4	43.80	4.069
3.0	49,213	15,000	↓	234,444	1.043×10^6	3845.4	3.771×10^4	59.20	5.500
4.5	49,213	15,000	↓	623,921	2.775×10^6	3817.8	3.743×10^4	105.80	9.829
4.5	68,898	21,000	↓	245,559	1.092×10^6	3818.9	3.744×10^4	105.80	9,829
-	Subsonic Solution		Rocket (Vacuum) $A_9 = 19.59$ m² 210.84 ft²	248,964	1.107×10^6	409.0	3.533×10^3	O/F = 7.936	
	↓			264,121	1.175×10^6	433.9	3.750×10^3	= 5.25	
	↓			262,356	1.167×10^6	431.0	3.725×10^3	= 3.5	
	Supersonic Solution		↓	257,121	1.144×10^6	422.4	3.651×10^3	= 7.936	
	↓		↓	276,965	1.232×10^6	455.0	$3,932 \times 10^3$	= 5.25	
	↓		↓	257,730	1.146×10^6	423.4	3.658×10^3	= 3.50	

TABLE X

ENGINE NO. 3/COMPONENT PROPELLANT FLOW RATES
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude		Operating Mode	Gas Generator				Primary				Afterburner		Total Flow Rates			
				O_2		H_2		O_2		H_2		H_2		O_2		H_2	
	ft.	m		lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec
0	0	0	SERJ-100% W_p	291.31	132.14	4.28	1.94	540.60	245.21	68.11	30.89	57.57	26.12	831.91	377.35	129.97	58.95
0.3	0	0		305.56	138.60	4.49	2.04					59.47	26.98	846.16	383.81	132.07	59.91
0.8	1,969	600		406.36	184.32	5.97	2.71					76.66	34.77	946.96	429.54	150.74	68.37
0.8	11,811	3,600		284.67	129.13	4.18	1.90					59.94	27.19	825.27	374.34	132.24	59.98
1.0	2,953	900		472.46	214.31	6.94	3.15					87.42	39.66	1,013.06	459.52	162.47	73.70
1.0	14,764	4,500		307.55	139.51	4.52	2.05					58.78	26.67	848.15	384.72	131.41	59.61
1.3	9,843	3,000		509.80	231.25	7.49	3.40					92.68	42.04	1,050.40	476.46	168.28	76.33
1.3	19,685	6,000		355.67	161.33	5.23	2.37					66.13	30.00	896.27	406.54	139.47	63.26
0.8	1,969	600	SERJ-60% W_p	406.36	184.32	5.97	2.71	270.30	122.61	34.06	15.45	76.66	34.77	676.66	306.93	116.69	52.93
0	0	0	FRJ	291.31	132.14	4.28	1.94	—	—	—	—	57.57	26.11	291.31	132.14	61.85	28.06
0.3	0	0		305.55	138.60	4.49	2.03	—	—	—	—	59.47	26.98	305.55	138.60	63.96	29.01
0.8	11,811	3,600		284.67	129.12	4.18	1.90	—	—	—	—	54.94	24.92	284.67	129.12	59.12	26.82
0.8	1,969	600		406.36	184.32	5.97	2.71	—	—	—	—	76.66	34.77	406.36	184.32	82.63	37.48
1.0	2,953	900		472.46	214.30	6.94	3.15	—	—	—	—	87.42	39.65	472.46	214.31	94.36	42.80
1.0	14,764	4,500		301.54	136.78	4.52	2.05	—	—	—	—	58.78	26.66	307.54	139.50	63.30	28.71
1.3	9,843	3,000		500.80	231.24	7.49	3.40	—	—	—	—	92.68	42.04	509.80	231.24	100.17	45.44
2.0	29,528	9,000		532.60	241.58	7.82	3.55	—	—	—	—	94.36	42.80	532.60	241.58	102.19	46.35
2.0	39,370	12,000		354.14	160.64	5.20	2.36	—	—	—	—	63.38	28.75	254.14	160.64	68.59	31.11
2.5	29,528	9,000		836.56	379.46	12.29	5.57	—	—	—	—	146.96	66.66	836.56	379.46	159.25	72.23
2.5	39,370	12,000		567.79	257.55	8.34	3.78	—	—	—	—	99.49	45.13	567.79	257.55	107.83	48.91
2.5	49,213	15,000		354.46	160.78	5.21	2.36	—	—	—	—	62.11	28.18	354.46	162.14	67.32	30.54
2.6	42,500	12,354		531.16	240.93	7.80	3.54	—	—	—	—	92.84	42.11	531.16	240.93	100.65	45.65
3.0	49,213	15,000		506.43	229.71	7.44	3.38	—	—	—	—	88.19	40.00	506.43	229.71	95.63	43.38
2.0	29,528	9,000	RJ	—	—	—	—	—	—	—	—	51.54	23.38	-	-	51.54	23.38
2.0	39,370	12,000		—	—	—	—	—	—	—	—	33.34	15.12	-	-	33.34	15.12
2.5	29,528	9,000		—	—	—	—	—	—	—	—	92.75	42.07	-	-	92.75	42.07
2.5	39,370	12,000		—	—	—	—	—	—	—	—	59.99	27.21	-	-	59.99	27.21
2.5	49,213	15,000		—	—	—	—	—	—	—	—	37.59	17.06	-	-	37.59	17.05
3.0	49,213	15,000		—	—	—	—	—	—	—	—	60.97	27.66	-	-	60.97	27.66
4.5	49,213	15,000		—	—	—	—	—	—	—	—	163.42	74.13	-	-	163.42	74.13
4.5	68,898	21,000		—	—	—	—	—	—	—	—	63.30	28.71	-	-	63.30	29.17
-	Subsonic Solution		Rocket (Vacuum) A_9 = 19.59 m^2 210.84 ft^2	—	—	—	—	540.60	245.21	68.11	30.89	—	—	540.60	245.21	68.11	30.89
				—	—	—	—	511.32	231.93	97.39	44.18	—	—	511.32	231.93	97.39	44.18
				—	—	—	—	473.44	214.75	135.27	61.36	—	—	473.44	214.75	135.27	61.36
	Supersonic Solution			—	—	—	—	540.60	245.21	68.11	30.89	—	—	540.60	245.21	68.11	30.89
				—	—	—	—	511.32	231.93	97.39	44.18	—	—	511.32	231.93	97.39	44.18
				—	—	—	—	473.44	214.75	135.27	61.36	—	—	473.44	214.75	135.27	61.36

TABLE XI

ENGINE NO. 3 /WEIGHT STATEMENT
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

Engine Component	Weight	
	kg	lb
Fan Assembly	2,272	5,008
Fan Housing	565	1,246
Primary Rockets	1,549	3,414
Mixer/Afterburner	2,255	4,972
Exit Nozzle	2,960	6,525
Controls	276	608
Total Engine Weight	9,877	21,773
Inlet Weight	5,759	12,696
Installed Engine Weight	15,636	34,469

$\text{Thrust/Weight*}_{\text{Engine}}$ = 18.6

$\text{Thrust/Weight*}_{\text{Installed Engine}}$ = 11.7

*SERJ Operating Mode/Sea Level Static Conditions

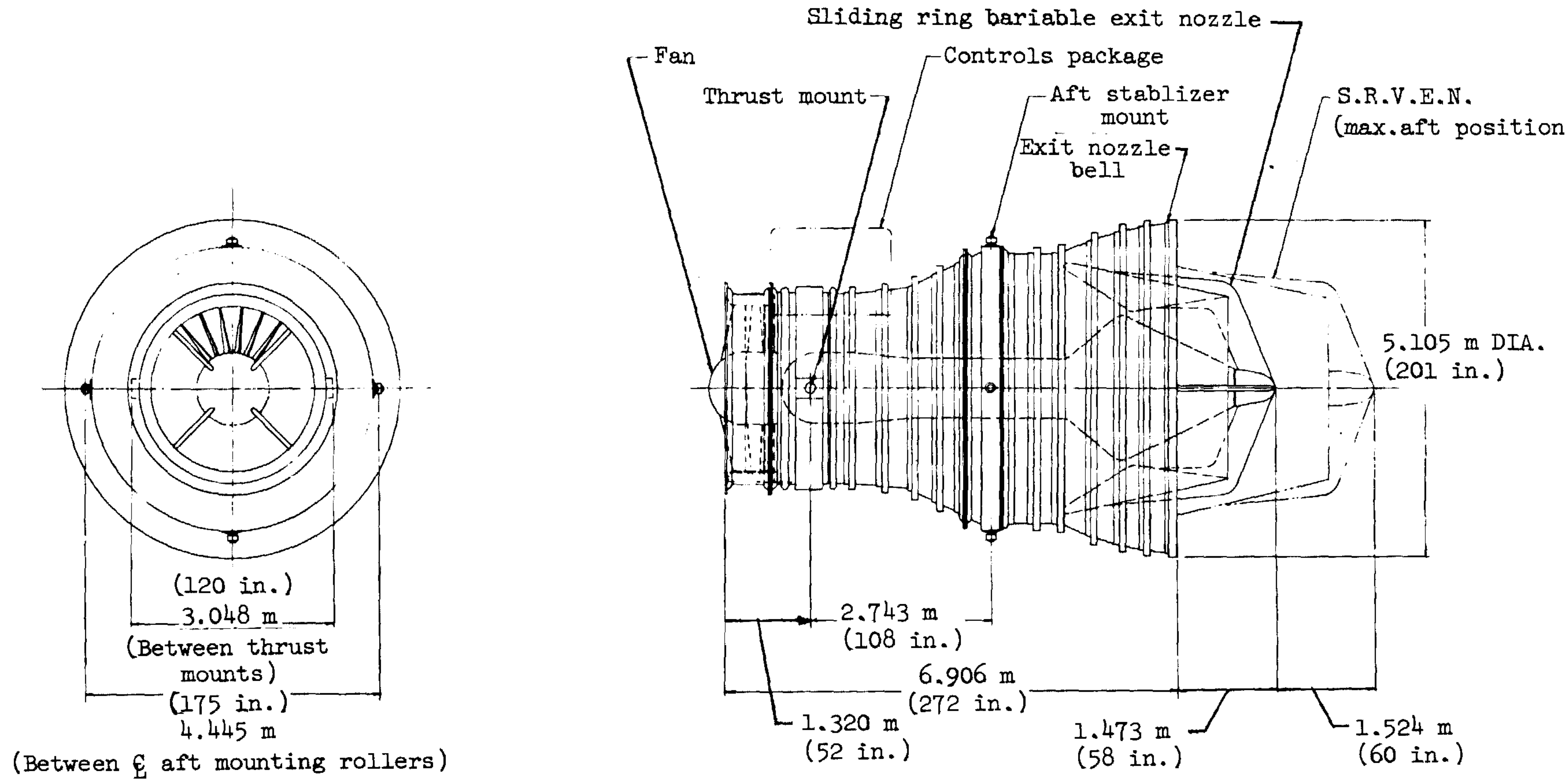


Figure 19.- Engine 3 outboard profile - airburborocket SERJ cycle.

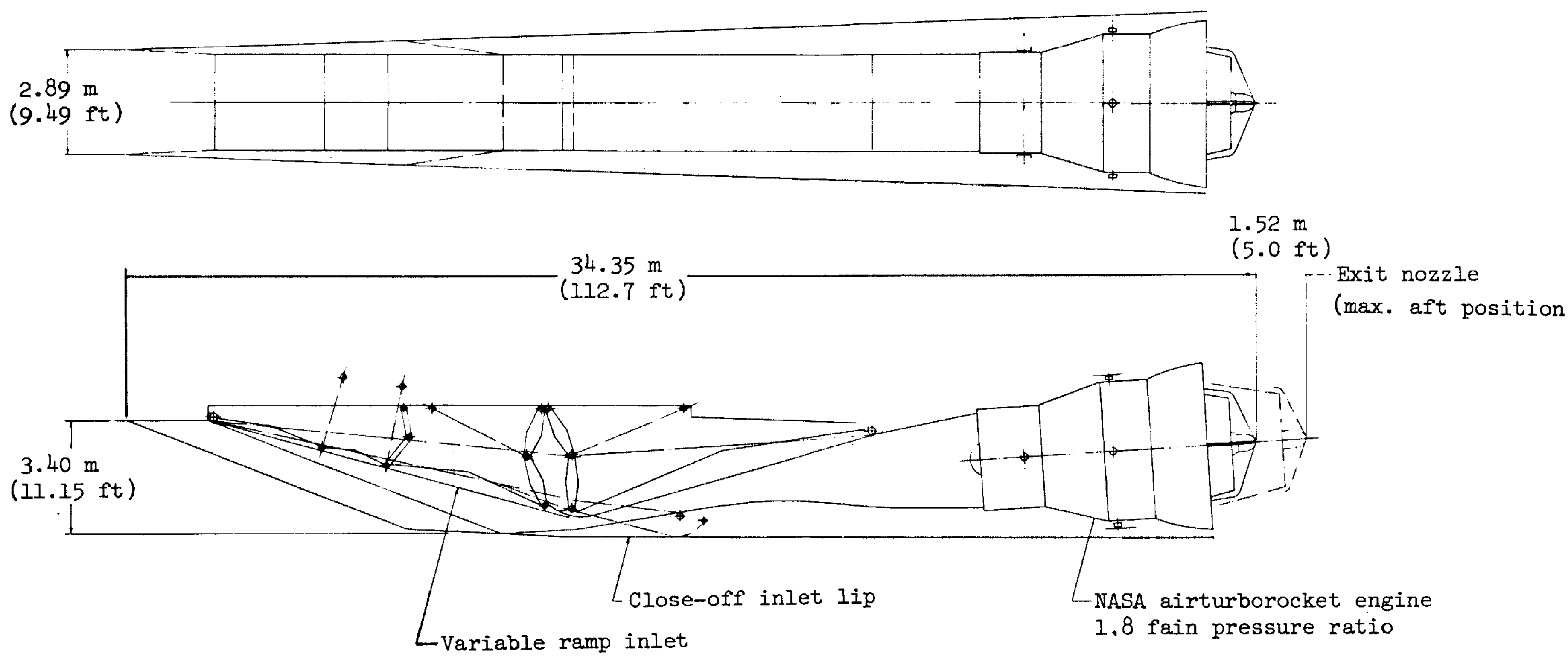


Figure 20.- Engine 3 installation - airturborocket SERJ cycle.

TABLE XII

ENGINE NO. 4/PERFORMANCE
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude ft	Altitude m	Operating Mode	Thrust lbs	Thrust Newtons	Isp sec.	Isp Newton-sec/kg	Capture Area A_o ft^2	Capture Area A_o m^2
0	0	0	SERJ - 100% W_p	873,647	3.886 x10^6	364.9	3.578 x10^3	-	-
.3	0	0	↓	864,376	3.844 x10^6	357.8	3.509 x10^3	70.15	6.517
.8	1,969	600	↓	960,611	4.273 x10^6	379.5	3.722 x10^3	34.95	3.247
.8	11,811	3,600	↓	887,748	3.948 x10^6	373.3	3.661 x10^3	35.53	3.301
1.0	14,764	4,500	↓	925,338	4.116 x10^6	384.5	3.771 x10^3	33.51	3.113
1.3	9,843	3,000	↓	1,090,278	4.849 x10^6	409.9	4.020 x10^3	33.88	3.148
1.3	19,685	6,000	↓	997,192	4.435 x10^6	404.8	3.970 x10^3	34.54	3.209
2.0	29,528	9,000	↓	1,240,539	5.518 x10^6	460.6	4.517 x10^3	46.93	4.360
0	0	0	FRJ	197,844	0.880 x10^6	517.9	5.078 x10^3	-	-
.8	1,969	600	↓	290,892	1.293 x10^6	560.1	5.492 x10^3	34.95	3.247
.8	11,811	3,600	↓	212,094	0.943 x10^6	577.4	5.662 x10^3	35.53	3.301
1.0	14,764	4,500	↓	240,573	1.070 x10^6	609.3	5.975 x10^3	33.51	3.113
1.3	9,843	3,000	↓	419,087	1.864 x10^6	645.4	6.329 x10^3	33.88	3.148
1.3	19,685	6,000	↓	300,230	1.335 x10^6	664.2	6.513 x10^3	34.54	3.209
2.0	29,528	9,000	↓	520,653	2.315 x10^6	763.7	7.489 x10^3	46.93	4.360
2.0	39,370	12,000	↓	349,286	1.553 x10^6	777.9	7.628 x10^3	48.38	4.495
3.0	49,212	15,000	↓	567,103	2.522 x10^6	768.3	7.534 x10^3	74.36	6.908
3.0	49,212	15,000	RJ	243,323	1.082 x10^6	3986.6	3.910 x10^4	59.26	5.505
4.5	49,213	15,000	↓	642,308	2.857 x10^6	3983.6	3.907 x10^4	104.37	9.696
4.5	68,898	21,000	↓	249,507	1.109 x10^6	3971.8	3.895 x10^4	104.37	9.696
4.5	49,213	15,000	ERJ	1,410,909	6.276 x10^6	649.3	6.367 x10^3	104.37	9.696
Subsonic Solution			Rocket (Vacuum) A_9 = 19.59 m^2 210.84 ft^2	820,721	3.651 x10^6	408.0	4.001 x10^3	O/F = 7.936	
↓				809,053	3.599 x10^6	402.2	3.944 x10^3	= 5.25	
↓				745,488	3.316 x10^6	370.6	3.634 x10^3	= 3.50	
Supersonic Solution			↓	884,487	3.934 x10^6	439.7	4.312 x10^3	= 7.936	
↓			↓	905,609	4.028 x10^6	450.2	4.415 x10^3	= 5.25	
↓			↓	876,642	3.899 x10^6	435.8	4.274 x10^3	= 3.50	

TABLE XIII

ENGINE NO. 4/COMPONENT PROPELLANT FLOW RATES
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude		Operating Mode	Gas Generator				Primary				After Burner		Total Propellant			
				O_2		H_2		O_2		H_2		H_2		O_2		H_2	
	ft	m		lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec
0	0	0	SERJ-100% W_p	295.27	133.93	4.50	2.04	1786.46	810.32	225.11	102.11	82.63	37.48	2081.73	944.25	312.24	141.63
.3	0	0		312.57	141.78	4.76	2.16					87.04	39.48	2099.03	952.10	316.91	143.75
.8	1,969	600		403.29	182.93	6.14	2.79					109.95	49.87	2189.75	993.25	341.20	154.77
.8	11,811	3,600		283.51	128.60	4.32	1.96					78.74	35.72	2069.97	938.92	308.17	294.56
1.0	14,764	4,500		306.11	138.85	4.66	2.11					84.09	38.14	2092.57	949.17	313.86	188.31
1.3	9,843	3,000		505.84	229.44	7.70	3.49					134.66	61.08	2292.30	1039.76	367.47	166.68
1.3	19,685	6,000		351.69	159.52	5.36	2.43					95.00	43.09	213c.15	969.84	325.47	147.63
2.0	29,528	9,000		534.82	242.59	8.15	3.70					138.83	62.97	2321.28	1052.91	233.26	105.81
0	0	0	FRJ	294.92	133.77	4.49	2.04	-	-	-	-	82.58	37.46	294.92	133.77	87.07	39.50
.8	1,969	600		403.29	182.93	6.14	2.79	-	-	-	-	109.95	49.87	403.29	182.93	116.09	52.66
.8	11,811	3,600		284.21	128.92	4.33	1.96	-	-	-	-	78.82	35.75	284.21	128.92	83.15	37.71
1.0	14,764	4,500		306.10	138.84	4.66	2.11	-	-	-	-	84.09	38.14	306.10	138.84	88.75	40.25
1.3	9,843	3,000		506.83	229.89	7.72	3.50	-	-	-	-	134.77	61.13	506.83	229.89	142.49	64.63
1.3	19,685	6,000		351.67	159.51	5.36	2.43	-	-	-	-	94.99	43.09	351.67	159.51	100.35	45.52
2.0	29,528	9,000		534.79	242.58	8.15	3.70	-	-	-	-	138.82	62.97	534.79	242.58	146.97	66.67
2.0	39,370	12,000		351.49	159.43	5.35	2.43	-	-	-	-	92.15	41.80	351.49	159.43	97.50	44.23
3.0	49,212	15,000		587.54	266.50	8.95	4.06	-	-	-	-	141.68	64.26	587.54	266.50	150.63	68.32
3.0	49,212	15,000	RJ	-	-	-	-	-	-	-	-	61.04	27.69	-	-	61.04	27.69
4.5	49,213	15,000		-	-	-	-	-	-	-	-	161.24	73.14	-	-	161.24	73.14
4.5	68,898	21,000		-	-	-	-	-	-	-	-	62.82	28.49	-	-	62.82	28.49
4.5	49,213	15,000	ERJ	-	-	-	-	1786.46	810.32	225.11	102.11	161.24	73.14	1486.46	810.32	386.35	175.25
Subsonic Solution			Rocket (Vacuum)	-	-	-	-	1786.46	810.32	225.11	102.11	-	-	1786.46	810.32	225.11	102.11
			A_s = 19.59 m^2	-	-	-	-	1689.72	766.44	321.85	145.99	-	-	1689.72	766.44	321.85	145.99
			210.84 ft^2	-	-	-	-	1564.55	709.67	447.02	202.76	-	-	1564.55	709.67	447.02	202.76
Supersonic Solution				-	-	-	-	1789.72	810.32	225.11	102.11	-	-	1786.46	810.32	225.11	102.11
				-	-	-	-	1689.72	766.44	321.85	145.99	-	-	1689.72	766.44	321.85	145.99
								1564.55	709.67	447.02	202.76	-	-	1564.55	709.67	447.02	202.76

TABLE XIV

ENGINE NO. 4 /WEIGHT STATEMENT
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

Engine Component	Weight	
	kg	lb
Fan Assembly	2,272	5,008
Fan Housing	565	1,246
Primary Rockets	3,737	8,239
Mixer/Afterburner	2,681	5,911
Exit Nozzle	2,960	6,525
Controls	276	608
Total Engine Weight	12,491	27,537
Inlet Weight	5,681	12,524
Installed Engine Weight	18,172	40,061

$\text{Thrust/Weight*}_{\text{Engine}}$ = 31.7

$\text{Thrust/Weight*}_{\text{Installed Engine}}$ = 21.8

* ERJ Operating Mode/Sea Level Static Conditions

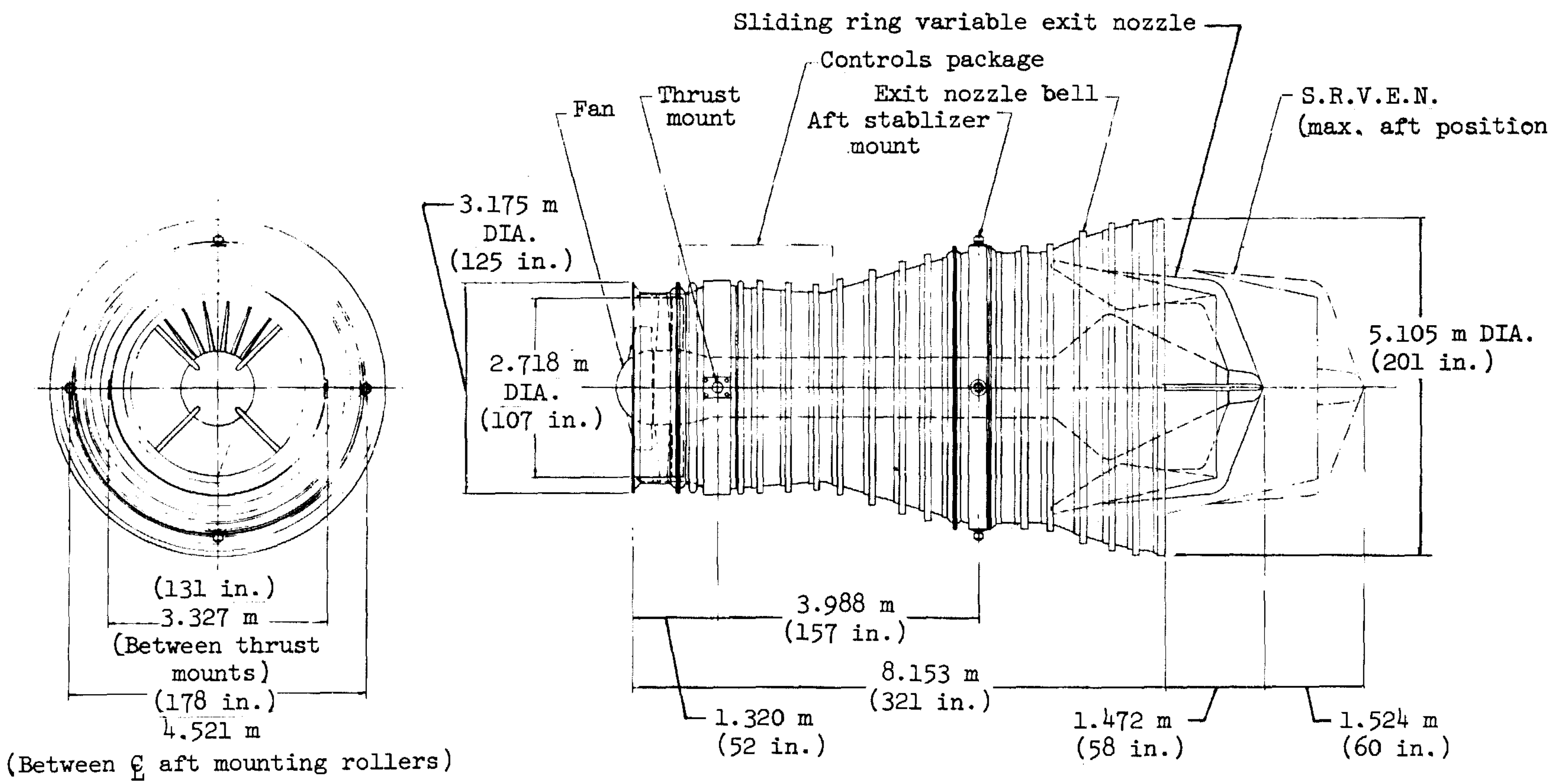


Figure 21.- Engine 4 outboard profile - airturborocket SERJ cycle.

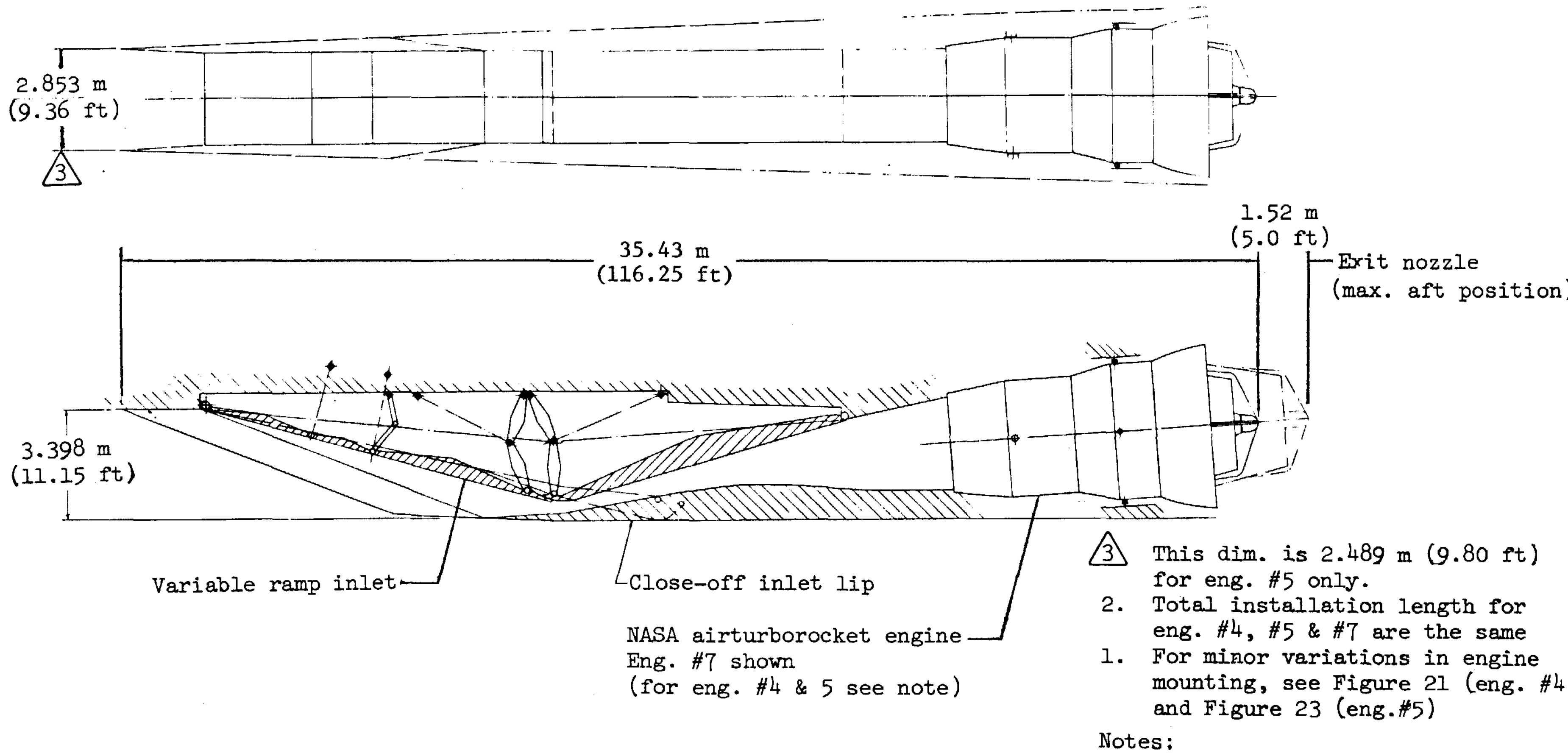


Figure 22.- Engine 4, 5 and 7 installation. Airturborocket SERJ/ejector ramjet cycles.

TABLE XV

ENGINE NO. 5 /PERFORMANCE
EJECTOR RAMJET ENGINE

M_o	Altitude		Operating Mode	Thrust		Isp		Capture Area A_o	
	ft	m		lbs	Newtons	sec.	$\frac{\text{Newton-sec}}{\text{kg}}$	ft^2	m^2
0	0	0	ERJ-100% W_p	776,539	3.454×10^6	375.2	3.679×10^3	-	-
.3	0	0		770,187	3.426×10^6	371.6	3.644×10^3	82.75	7.688
.8	1,969	600		772,620	3.437×10^6	372.3	3.651×10^3	34.40	3.196
.8	11,811	3,600		808,994	3.599×10^6	391.4	3.838×10^3	42.08	3.909
1.0	14,764	4,500		840,120	3.737×10^6	405.7	3.978×10^3	40.04	3.720
1.3	9,843	3,000		880,728	3.917×10^6	421.3	4.132×10^3	34.32	3.188
1.3	19,685	6,000		903,427	4.018×10^6	434.5	4.261×10^3	41.92	3.894
2.0	29,528	9,000		1,030,795	4.585×10^6	491.3	4.818×10^3	51.33	4.769
2.0	39,370	12,000		990,652	4.407×10^6	476.3	4.671×10^3	62.53	5.809
3.0	49,213	15,000		1,126,056	5.009×10^6	531.5	5.212×10^3	104.37	9.696
4.5	49,213	15,000		1,354,301	6.024×10^6	623.5	6.114×10^3	104.37	9.696
3.0	49,213	15,000	RJ	412,597	1.835×10^6	3838.4	3.764×10^4	104.37	9.696
4.5	49,213	15,000		644,434	2.867×10^6	3996.7	3.919×10^4	104.37	9.696
4.5	68,898	21,000		250,307	1.113×10^6	3984.6	3.908×10^4	104.37	9.696
8.0	103,340	31,500		69,707	0.113×10^6	3154.7	3.094×10^4	104.37	9.696
Subsonic Solution			Rocket (Vacuum)	747,087	3.323×10^6	371.5	3.643×10^3	O/F = 7.936	
			$A_9 = 19.59\ m^2$	810,835	3.607×10^6	403.2	3.954×10^3	= 5.25	
			$210.84\ ft^2$	824,309	3.667×10^6	409.9	4.020×10^3	= 3.50	
Supersonic Solution				882,226	3.924×10^6	438.7	4.302×10^3	= 7.936	
				910,179	4.049×10^6	452.6	4.438×10^3	= 5.25	
				891,879	3.967×10^6	443.5	4.349×10^3	= 3.50	

TABLE XVI

ENGINE NO. 5/COMPONENT PROPELLANT FLOW RATES

M_o	Altitude ft	Altitude m	Operating Mode	Gas Generator O_2 lb/sec	Gas Generator O_2 k_g/sec	Gas Generator H_2 lb/sec	Gas Generator H_2 k_g/sec	Primary O_2 lb/sec	Primary O_2 k_g/sec	Primary H_2 lb/sec	Primary H_2 k_g/sec	After Burner H_2 lb/sec	After Burner H_2 k_g/sec	Total Propellant O_2 lb/sec	Total Propellant O_2 k_g/sec	Total Propellant H_2 lb/sec	Total Propellant H_2 k_g/sec
0	0	0	ERJ-100% W_p	-	-	-	-	1785.95	810.09	225.05	102.08	58.64	26.60	1785.95	810.09	283.69	128.68
.3	0	0		-	-	-	-					61.82	28.04			286.97	130.12
.8	1,969	600		-	-	-	-					64.24	29.14			289.29	131.22
.8	11,811	3,600		-	-	-	-					56.06	25.43			281.11	127.51
1.0	14,764	4,500		-	-	-	-					59.96	27.20			285.01	129.28
1.3	9,843	3,000		-	-	-	-					79.64	36.12			304.69	138.20
1.3	19,685	6,000		-	-	-	-					68.02	30.85			293.07	132.93
2.0	29,528	9,000		-	-	-	-					87.04	39.48			312.09	141.56
2.0	39,370	12,000		-	-	-	-					68.78	31.20			293.83	133.28
3.0	49,213	15,000		-	-	-	-					107.49	48.76			332.54	150.84
4.5	49,213	15,000		-	-	-	-					161.24	73.14			386.29	175.22
3.0	49,213	15,000	RJ	-	-	-	-	-	-	-	-	107.49	48.76			107.49	48.76
4.5	49,213	15,000		-	-	-	-	-	-	-	-	161.24	73.14	-	-	161.24	73.14
4.5	68,898	21,000		-	-	-	-	-	-	-	-	62.82	28.49	-	-	62.82	28.49
8.0	103,340	31,500		-	-	-	-	-	-	-	-	22.10	10.02	-	-	22.10	10.02
Subsonic Solution			Rocket (Vacuum)	-	-	-	-	1785.95	810.09	225.05	102.08	-	-	1785.95	810.09	225.05	102.08
			$A_9 = 19.59\ m^2$	-	-	-	-	1639.24	766.23	321.76	145.95	-	-	1689.24	766.23	321.76	145.95
			$210.84\ ft^2$	-	-	-	-	1564.11	709.47	446.89	202.71	-	-	1564.11	709.47	446.89	202.71
Supersonic Solution				-	-	-	-	1785.95	810.09	225.05	102.08	-	-	1785.95	810.09	225.05	102.08
				-	-	-	-	1639.24	766.23	321.76	145.95	-	-	1689.24	766.23	321.76	145.95
				-	-	-	-	1564.11	709.47	446.89	202.71	-	-	1564.11	709.47	446.89	202.71

TABLE XVII

ENGINE NO. 5/WEIGHT STATEMENT
EJECTOR RAMJET ENGINE

Engine Component	Weight	
	kg	lb
Fan Assembly	-	-
Fan Housing	-	-
Primary Rockets	3,760	8,289
Mixer/Afterburner	2,876	6,341
Exit Nozzle	2,960	6,525
Controls	138	304
Total Engine Weight	9,734	21,459
Inlet Weight	5,681	12,524
Installed Engine Weight	15,415	33,983

$\text{Thrust/Weight*}_{\text{Engine}}$ = 36.2

$\text{Thrust/Weight*}_{\text{Installed Engine}}$ = 22.9

*ERJ Operating Mode/Sea Level Static Conditions

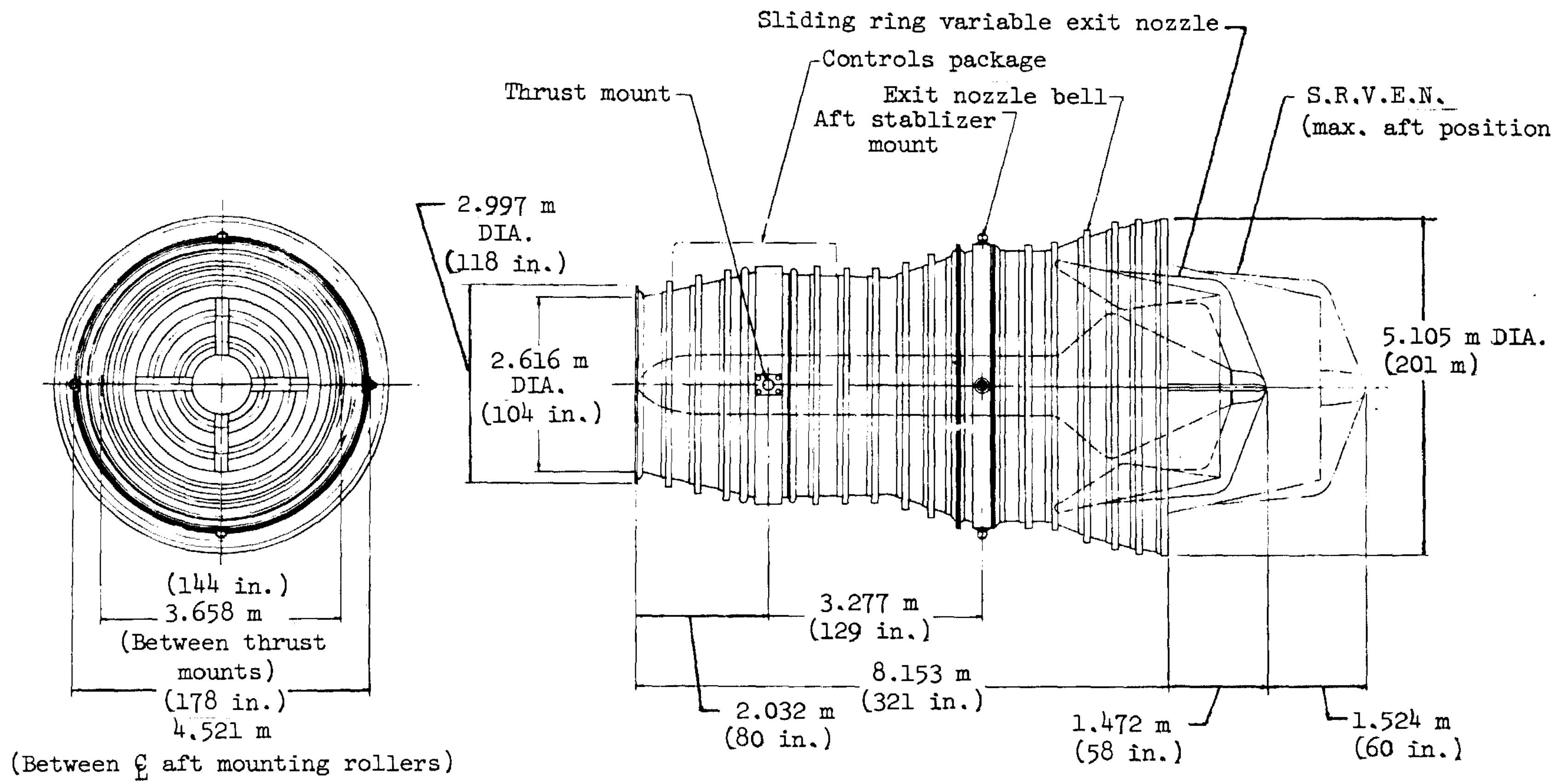


Figure 23.- Engine 5 outboard profile - ejector ramjet cycle.

TABLE XVIII

ENGINE NO. 6/PERFORMANCE
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude		Operating Mode	Thrust		Isp		Capture Area A_o	
	ft	m		lbs	Newtons	sec.	$\frac{\text{Newton-sec}}{\text{kg}}$	ft^2	m^2
0	0	0	SERJ	672,405	2.991×10^6	263.2	2.580×10^3	-	-

TABLE XIX

ENGINE NO. 6/COMPONENT PROPELLANT FLOW RATES
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude		Operating Mode	Gas Generator				Primary				After Burner		Total Propellant			
				O_2		H_2		O_2		H_2		H_2		O_2		H_2	
	ft	m		lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec
0	0	0	SERJ	309.37	140.32	17.77	8.06	1578.58	716.02	460.36	208.81	188.17	85.35	1887.95	856.34	666.30	302.22

TABLE XX

ENGINE NO. 7 /PERFORMANCE
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude ft	Altitude m	Operating Mode	Thrust lbs	Thrust Newtons	Isp sec.	Isp Newton-sec/kg	Capture Area A_o ft^2	Capture Area A_o m^2
0	0	0	SERJ	870,723	3.873×10^6	340.9	3.343×10^3	-	-
.3	0	0		872,461	3.880×10^6	337.7	3.311×10^3	70.15	6.517
.8	1,969	600		953,878	4.243×10^6	348.6	3.418×10^3	34.95	3.247
.8	11,811	3,600		898,555	3.996×10^6	354.7	3.478×10^3	35.53	3.301
1.0	14,764	4,500		934,495	4.156×10^6	363.7	3.566×10^3	33.51	3.113
1.3	9,843	3,000		1,089,708	4.847×10^6	374.9	3.676×10^3	33.88	3.148
1.3	19,685	6,000		1,003,839	4.465×10^6	379.6	3.722×10^3	34.54	3.209
0	0	0	FRJ	192,194	0.854×10^6	372.8	3.655×10^3	-	-
.8	1,969	600		275,057	1.223×10^6	394.6	3.869×10^3	34.95	3.247
.8	11,811	3,600		199,761	0.888×10^6	404.0	3.961×10^3	35.53	3.301
1.0	14,764	4,500		226,500	1.007×10^6	426.7	4.184×10^3	33.51	3.113
1.3	9,843	3,000		391,616	1.741×10^6	451.2	4.424×10^3	33.88	3.148
1.3	19,685	6,000		282,120	1.251×10^6	466.0	4.569×10^3	34.54	3.209
2.0	29,528	9,000		484,530	2.155×10^6	535.3	5.249×10^3	46.93	4.360
2.9	39,370	12,000		325,254	1.446×10^6	544.1	5.335×10^3	48.38	4.495
3.0	49,213	15,000		527,269	2.345×10^6	549.1	5.384×10^3	74.36	6.906
3.0	49,212	15,000	RJ	223,226	0.993×10^6	1577.6	7.018×10^3	159.26	5.505
4.5	49,213	15,000		613,089	2.727×10^6	1566.1	1.526×10^4	109.31	10.155
4.5	68,898	21,000		237,943	1.058×10^6	1560.0	1.530×10^4	109.31	10.155
4.5	49,213	15,000	ERJ	1,403,520	6.243×10^6	577.5	5.663×10^3	109.31	10.155
Subsonic Solution			Rocket (Vacuum)	755,631	3.361×10^6	370.6	3.634×10^3	O/F = 7.936	
			$A_9 = 19.59\ m^2$	820,062	3.648×10^6	402.2	3.944×10^3	= 5.25	
			$210.84\ ft^2$	831,888	3.700×10^6	408.0	4.001×10^3	= 3.50	
Supersonic Solution				888,570	3.953×10^6	435.8	4.274×10^3	= 7.936	
				917,931	4.083×10^6	450.2	4.415×10^3	= 5.25	
				896,522	3.988×10^6	439.7	4.312×10^3	= 3.50	

TABLE XXI

ENGINE NO. 7/COMPONENT PROPELLANT FLOW RATES
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

M_o	Altitude		Operating Mode	Gas Generator				Primary				After Burner		Total Propellant					
				O_2		JP-4		O_2		H_2		JP-4		O_2		JP-4		H_2	
	ft	m		lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec	lb/sec	k_g/sec
0	0	0	SERJ-100% W_p	309.37	140.33	17.77	8.06	1810.77	821.35	228.17	103.50	188.17	85.35	2120.14	961.68	205.94	93.41	228.17	103.50
.3	0	0	↓	327.35	148.48	18.80	8.53	↓	↓	↓	↓	198.17	89.89	2138.12	969.83	216.97	98.42	↓	↓
.8	1,969	600		422.56	191.67	24.26	11.00					250.28	113.53	2233.33	1013.02	274.54	124.53		
.8	11,811	3,600		297.84	135.10	17.10	7.76					179.48	81.41	2108.61	956.45	196.58	89.17		
1.0	14,764	4,500		320.86	145.54	18.43	8.36					191.48	86.85	2131.63	966.89	209.91	95.21		
1.3	9,843	3,000		530.91	240.82	30.49	13.83					306.61	139.04	2341.69	1062.17	337.10	152.91		
1.3	19,685	6,000		368.20	167.01	21.14	9.59					216.14	98.04	2178.97	988.36	237.28	107.63		
0	0	0	FRJ	309.42	140.35	17.77	8.06	-	-	-	-	188.19	85.36	309.42	140.35	205.96	93.42	-	-
.8	1,969	600		422.45	191.62	24.26	11.00	-	-	-	-	250.25	113.51	422.45	191.62	274.51	124.51	-	-
.8	11,811	3,600		297.84	135.10	17.10	7.76	-	-	-	-	179.48	81.41	297.84	135.10	196.58	89.17	-	-
1.0	14,764	4,500		320.87	145.54	18.43	8.36	-	-	-	-	191.48	86.85	320.87	145.54	209.91	95.21	-	-
1.3	9,843	3,000		530.93	240.83	30.49	13.83	-	-	-	-	306.61	139.08	530.93	240.83	337.10	152.91	-	-
1.3	19,685	6,000		368.17	167.00	21.14	9.59	-	-	-	-	216.13	98.03	368.17	167.00	237.27	107.62	-	-
2.0	29,529	9,000		558.02	253.11	32.04	14.53	-	-	-	-	315.17	142.96	558.02	253.11	347.21	157.49	-	-
2.0	39,370	12,000		367.33	166.62	21.09	9.57	-	-	-	-	209.39	94.98	367.33	166.62	230.48	104.55	-	-
3.0	49,213	15,000		606.95	275.31	34.85	15.81	-	-	-	-	319.70	145.01	606.95	275.31	354.55	160.62	-	-
3.0	49,213	15,000	RJ	-	-	-	-	-	-	-	-	141.49	64.18	-	-	141.49	64.18	-	-
4.5	49,213	15,000		-	-	-	-	-	-	-	-	391.48	177.57	-	-	391.48	177.57	-	-
4.5	65,893	21,000		-	-	-	-	-	-	-	-	152.52	69.18	-	-	152.52	69.18	-	-
4.5	49,213	15,000	ERJ	-	-	-	-	1810.77	821.35	228.17	103.50	391.48	177.57	1810.77	821.35	-	-	228.17	103.50
Subsonic Solution			Rocket (Vacuum)	-	-	-	-	1810.77	821.35	228.17	103.50	-	-	1810.77	821.35	-	-	228.17	103.50
↓			$A_9 = 19.59\ m^2$	-	-	-	-	1712.71	776.87	326.23	147.98	-	-	1712.71	776.87	-	—	326.23	147.98
			$210.84\ ft^2$	-	-	-	-	1585.84	719.32	453.10	205.52	-	-	1585.84	719.32	-	-	453.10	205.52
Supersonic Solution			↓	-	-	-	-	1810.77	821.35	228.17	103.50	-	-	1810.77	821.35	-	-	228.17	103.50
↓				-	-	-	-	1712.71	776.87	326.23	147.98	-	-	1712.71	776.87	-	-	326.23	147.98
				-	-	-	-	1585.84	719.32	453.10	205.52	-	-	1585.84	719.32	-	-	453.10	205.52

TABLE XXII

ENGINE NO. 7 /WEIGHT STATEMENT
AIRTURBOROCKET ENGINE/1.8 FAN TOTAL PRESSURE RATIO

Engine Component	Weight	
	kg	lb
Fan Assembly	2,272	5,008
Fan Housing	565	1,246
Primary Rockets	3,778	8,330
Mixer/Afterburner	2,691	5,932
Exit Nozzle	2,960	6,525
Controls	276	608
Total Engine Weight	12,542	27,649
Inlet Weight	5,950	13,117
Installed Engine Weight	18,492	40,766

$\text{Thrust/Weight*}_{\text{Engine}}$ = 31.5

$\text{Thrust/Weight*}_{\text{Installed Engine}}$ = 21.4

*SERJ Operating Mode/Sea Level Static Conditions

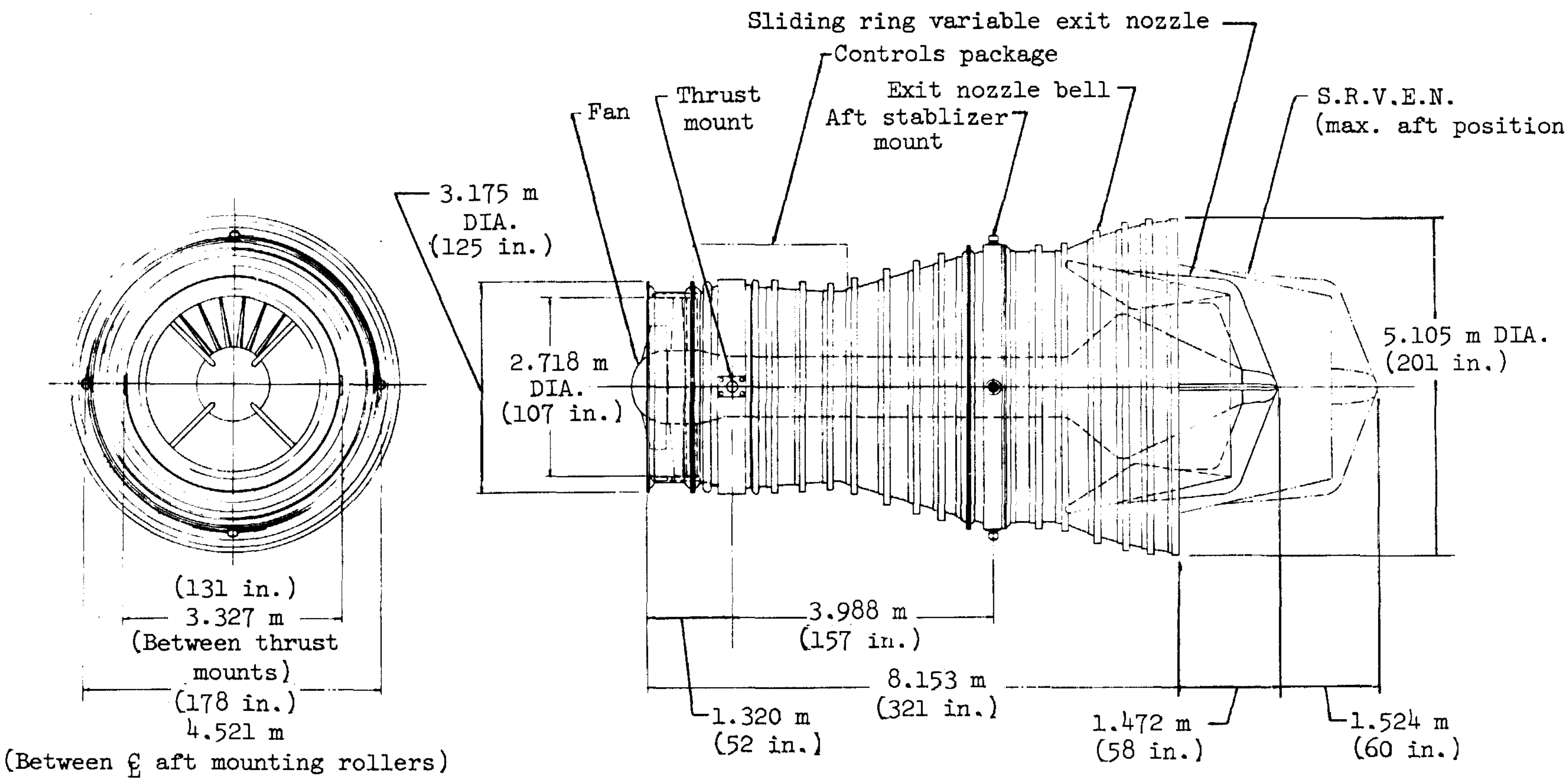


Figure 24.- Engine 7 outboard profile - airturborocket SERJ cycle.

DISCUSSIONS

Comparison Between Engines

Some significant data for the seven engines analyzed are summarized in Table XXIII. Engine 1 has considerably better fan ramjet specific impulse than engine 2. The difference is entirely due to the increased work per unit airflow required from the gas generator to power the higher pressure ratio (1.8) fan. Some possibilities for decreasing the gas generator propellant flows are presented later in this section.

Engine 3 is lighter than engine 2 because the fan is fixed but the ramjet mode is limited to a Mach number of 4.5. The ramjet thrust at a Mach number of 3.0, is also reduced (compare Table VI with Table IX).

Engine 4 has increased thrust compared to engines 1 to 3 because the ejector thrust has been increased. The engine size (Figure 19 and 21) has not changed. The SERJ mode specific impulse was degraded 56 seconds.

Engine 5, which does not have a fan subsystem, is lighter than engine 4. The specific impulse, however, never greatly exceeds that of a rocket except in the ramjet mode.

Engines 6 and 7 use hydrocarbon fuel, so the specific impulse is less than engine 4. The tankage required for a corresponding total impulse is less for the hydrocrabon fuel, so there may be advantages to using it early in a mission.

Advanced Technology Payoffs

Five key design/technology assumptions were made in conducting this study. These key assumptions were:

1. Single stage fan
2. External gas generator
3. Single stage tip turbine
4. Fan turbine design total pressure ratio = 3.50
5. Maximum fan turbine inlet temperature = 1367°K (2460°R).

An objective of this program were to assess the payoff of incorporating advanced engine technology into the engines designed in this study. Specifically, the effects on performance of increased fan turbine design pressure ratio and fan maximum turbine inlet temperature was assessed for engine No. 4 and are discussed below.

Increased Fan Turbine Pressure Ratio/Internal Gas Generator. - The airturbo-rocket engines studied in this program assumed the use of a single stage turbine located on the periphery of a single stage fan. A turbine design pressure ratio of 3.50 was assumed. Fan mechanical speed was held constant for all SERJ and Fan Ramjet flight conditions. The fan turbine inlet total temperature was also held constant. Thus the fan turbine's corrected speed is constant; therefore, the turbine always operates at its design point.

TABLE XXIII. COMPOSITE ENGINE PERFORMANCE COMPARISON

Flight Condition	Sea Level Static			M_o = 2.0 @ 9000 m (29528 ft)		M_o = 4.5 @ 21000 m (68898 ft)		Vacuum-O/F=5.25	
Eng. No. \ Eng. Operating Mode	Supercharged Ejector Ramjet			Fan Ramjet		Ramjet		Rocket	
	Thrust N (lb)	I_{sp} Sec.	Thrust / Weight	Thrust N (lb)	I_{sp} Sec.	Thrust N (lb)	I_{sp} Sec	Thrust N (lb)	I_{sp} Sec
1	1.8×10^6 (404, 700)	427	17.3	1.946×10^6 (437, 600)	1051	1.386×10^6 (311, 600)	3705	1.466×10^6 (329, 500)	459
2	1.8×10^6 (404, 700)	421	17.3	1.828×10^6 (410, 900)	647	1.096×10^6 (246, 500)	3833	1.232×10^6 (276, 900)	455
3	1.8×10^6 (404, 700)	421	18 6	1.828×10^6 (410, 900)	647	1.092×10^6 (245, 600)	3819	1.232×10^6 (276, 900)	455
4	3.886×10^6 (873, 600)	365	31.7	2.315×10^6 (520, 700)	764	1.109×10^6 (249, 500)	3972	4.028×10^6 (905, 600)	450
5	3.454×10^6 (776, 500)	375	36.2	4.585×10^6 (1, 030, 800)	491	1.113×10^6 (250, 307)	3985	4.049×10^6 (910, 200)	453
6	2.991×10^6 (672, 405)	263	-	- -	- -	- -	- -	- -	- -
7	3.873×10^6 (870, 800)	341	31.5	2.155×10^6 (484, 500)	535	1.058×10^6 (238, 000)	1560	4.083×10^6 (917, 900)	450

Sea level static Supercharged Ejector Ramjet and Fan Ramjet performance was computed for engine No. 4 over a range of turbine design total pressure ratios. The following turbine pressure ratio, adiabatic efficiency, and number of turbine stages relationship was assumed in this analysis:

Stages	Total Pressure Ratio	Adiabatic Efficiency
1	3.5	.85
2	7.0	.86
3	14.0	.87

The effect of turbine pressure ratio on engine specific impulse and thrust is presented in Figures 25, 26 and Table XXIV. The improvement in SERJ mode specific impulse is modest. The large ejector thrust/propellant flow rate dominate this engine operating mode. However, Fan Ramjet performance is dramatically improved with increased fan turbine pressure ratio.

Clever design may permit the use of two tip turbine stages on the periphery of the single stage fan. If this is not technically feasible, a more conventional internal gas generator/fan-turbine drive configuration would be required (See Figure 27). In any case, the use of a three stage turbine may be be accomplished with the conventional fan/turbine drive arrangement.

Increasing the number of turbine stages to improve engine performance will also increase engine weight. It should be recognized that one of the attractive features of the airturborocket engine is its light weight, albeit the airbreathing gas generator SERJ engine has better engine performance. A number of turbine stages/ engine performance/engine weight design tradeoff study is clearly indicated. Such a study was beyond the scope of this contract.

Increased Fan Turbine Inlet Temperature. - The effect of increased fan turbine inlet total temperature on engine No. 4 sea level static performance is presented in Figures 28, 29 and Table XXIV. Supercharged Ejector and Fan Ramjet specific impulse and thrust performance is shown. Specific impulse , for both operating modes, linearly increases with temperature. Better materials and/or active turbine blade cooling is probably required for turbine temperatures much in excess of 1367°K (2460°R). Active blade cooling with a tip turbine configuration appears to be a formidable design problem. A turbine design study is required to define the operating limits of current or near term materials. Such a study was beyond the scope of this program.

Engine Ground Test Facility Requirements/Limitations

Successful development of a high speed airbreathing engine may require a considerable amount of engine ground testing. Engine development as well as performance/structural documentation testing will be required. In addition, a flight test program is required to evaluate potential engine/vehicle dynamic interaction problems and to demonstrate total system performance.

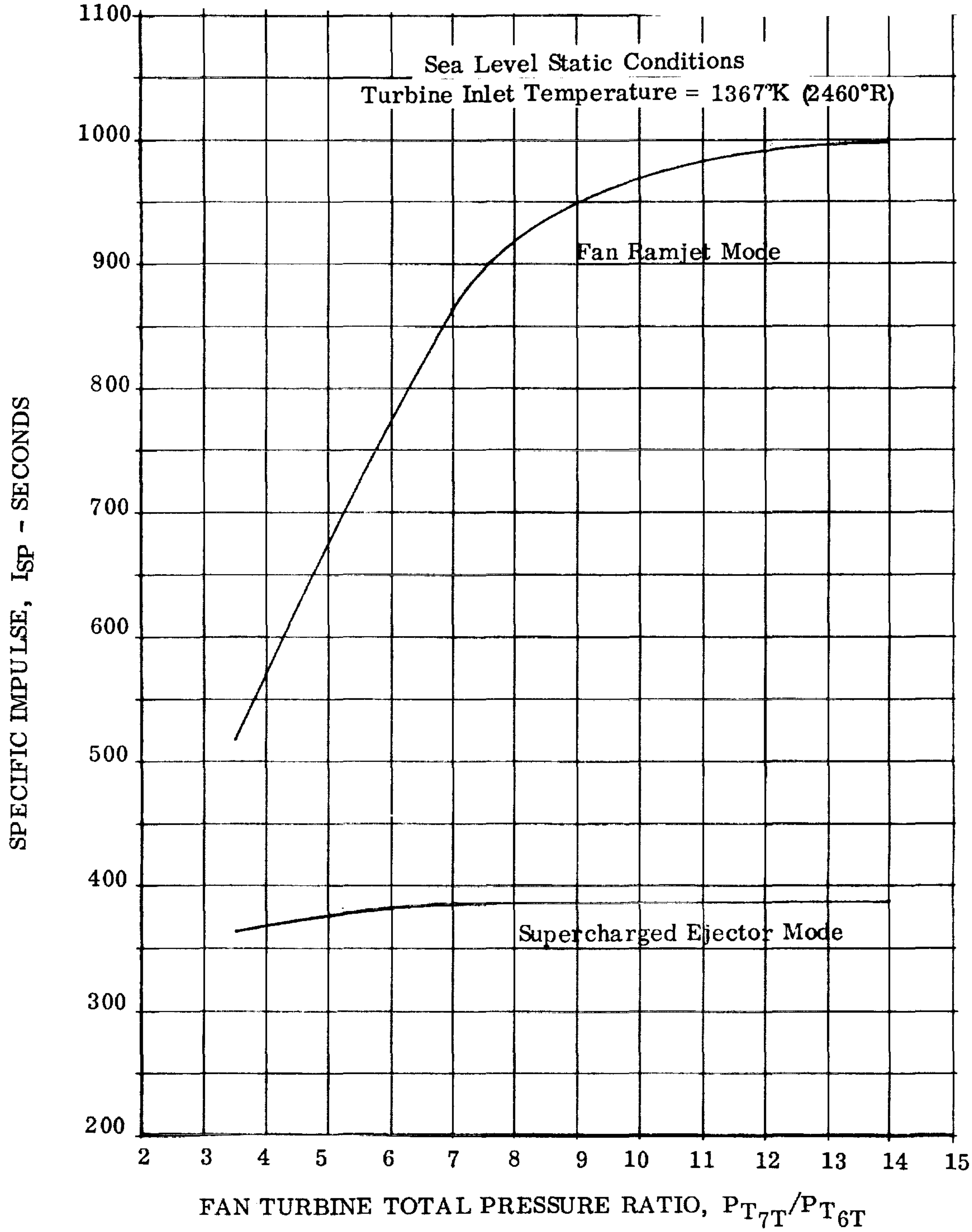


Figure 25 . Effect of Fan Turbine Pressure Ratio on Specific Impulse

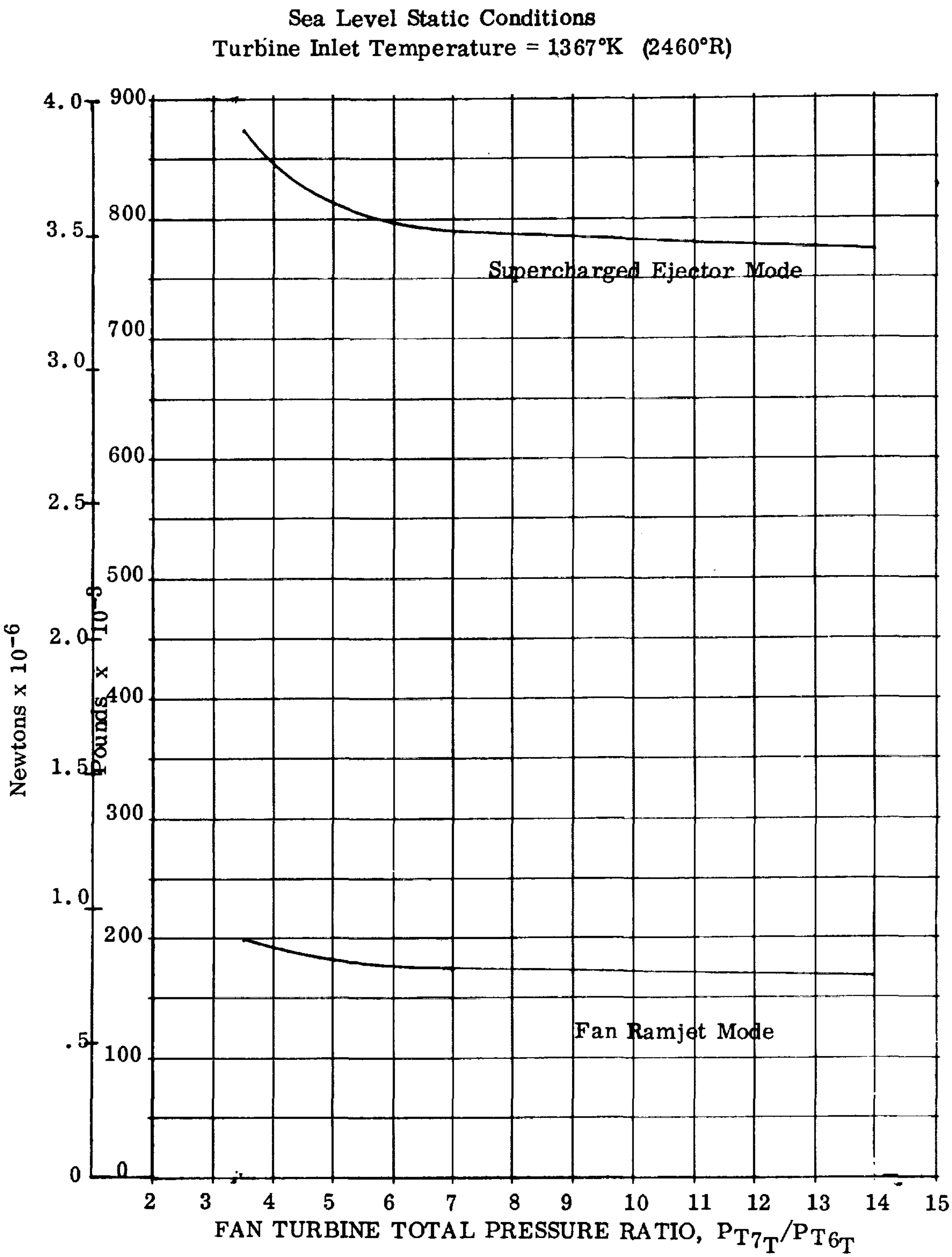


Figure 26. Effect of Fan Turbine Pressure Ratio on Engine Thrust

TABLE XXIV

ADVANCED ENGINE TECHNOLOGY/COMPONENT PROPELLANT FLOW RATES
ENGINE NO. 4 - SEA LEVEL STATIC

Fan Turbine Performance		Operating Mode	Gas Generator				Primary				After Burner		Total Propellant			
Inlet Total Temperature °K/°R	Total Pressure Ratio		O_2		H_2		O_2		H_2		H_2		O_2		H_2	
			lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec	lb/sec	kg/sec
1367/2460	3.5	SERJ	295.27	133.93	4.50	2.04	1786.46	810.32	225.11	102.11	82.63	37.48	2081.73	944.25	312.24	141.63
1367/2460	3.5	FRJ	294.92	133.77	4.49	2.04	-	-	-	-	82.58	37.46	294.92	133.77	87.07	39.50
1506/2710	3.5	SERJ	263.74	119.63	4.53	2.05	1758.49	797.63	221.58	100.51	78.62	35.66	2022.23	917.26	304.73	138.22
1506/2710	3.5	FRJ	263.60	119.57	4.53	2.05	-	-	-	-	78.60	35.65	263.60	119.57	83.13	37.70
1644/2960	3.5	SERJ	237.01	107.51	4.57	2.07	1734.78	786.88	218.60	99.15	75.21	34.11	1971.79	894.39	298.38	135.33
1644/2960	3.5	FRJ	237.03	107.51	4.57	2.07	-	-	-	-	75.21	34.11	237.03	107.51	79.78	36.18
1367/2460	7.0	SERJ	134.37	60.95	3.21	1.46	1645.67	746.46	206.96	93.87	63.63	28.86	1780.04	807.41	273.80	124.19
1367/2460	7.0	FRJ	134.28	60.91	3.21	1.46	-	-	-	-	63.63	28.86	134.28	60.91	66.84	30.02
1367/2460	14.0	SERJ	106.01	48.09	2.53	1.15	1616.63	733.29	203.71	92.40	60.74	27.55	1722.64	781.38	266.98	121.10
1367/2460	14.0	FRJ	105.96	48.06	2.53	1.15	-	-	-	-	60.74	27.55	105.96	48.06	63.27	28.70

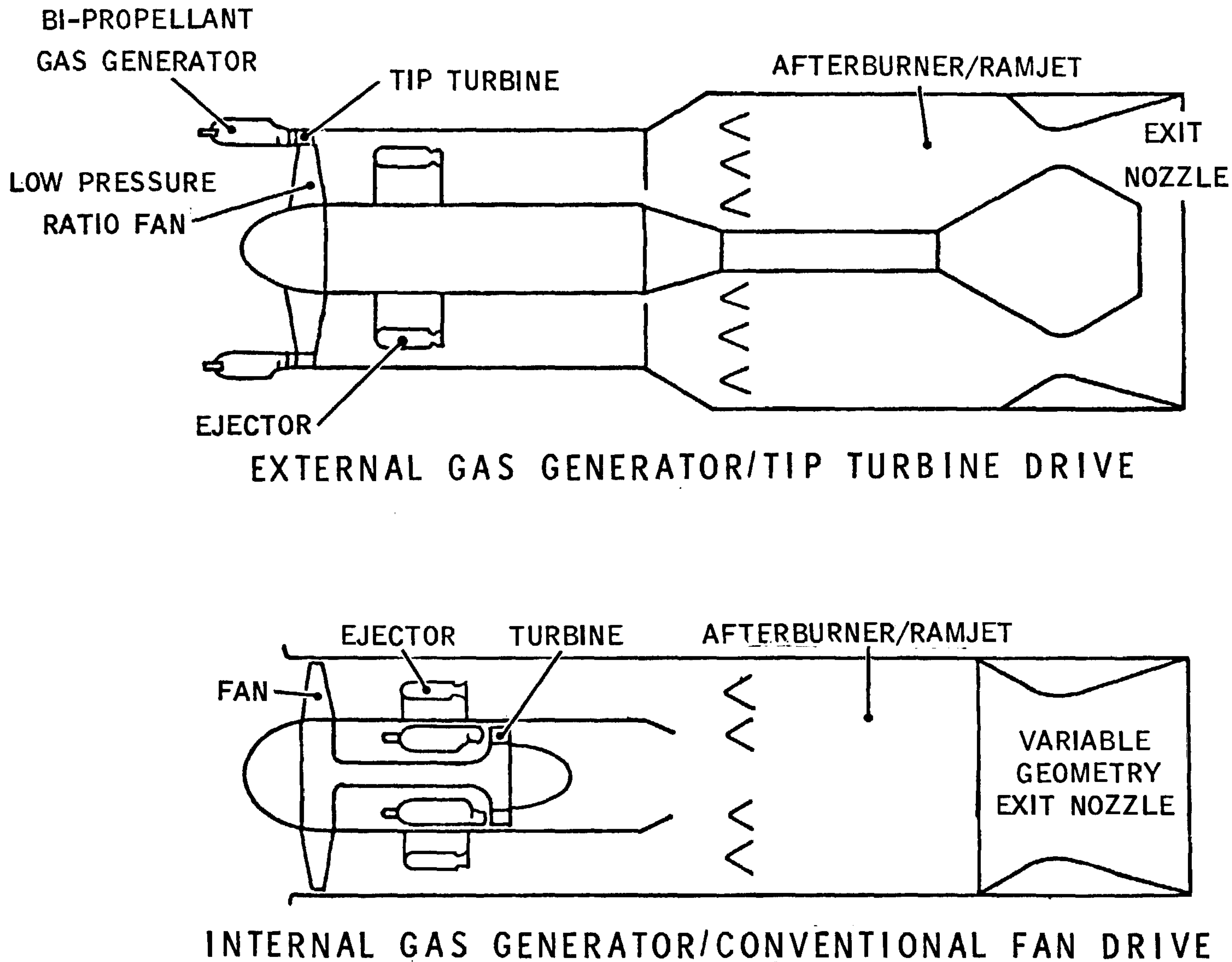


Figure 27. Airturborocket Engine Cycle Variations

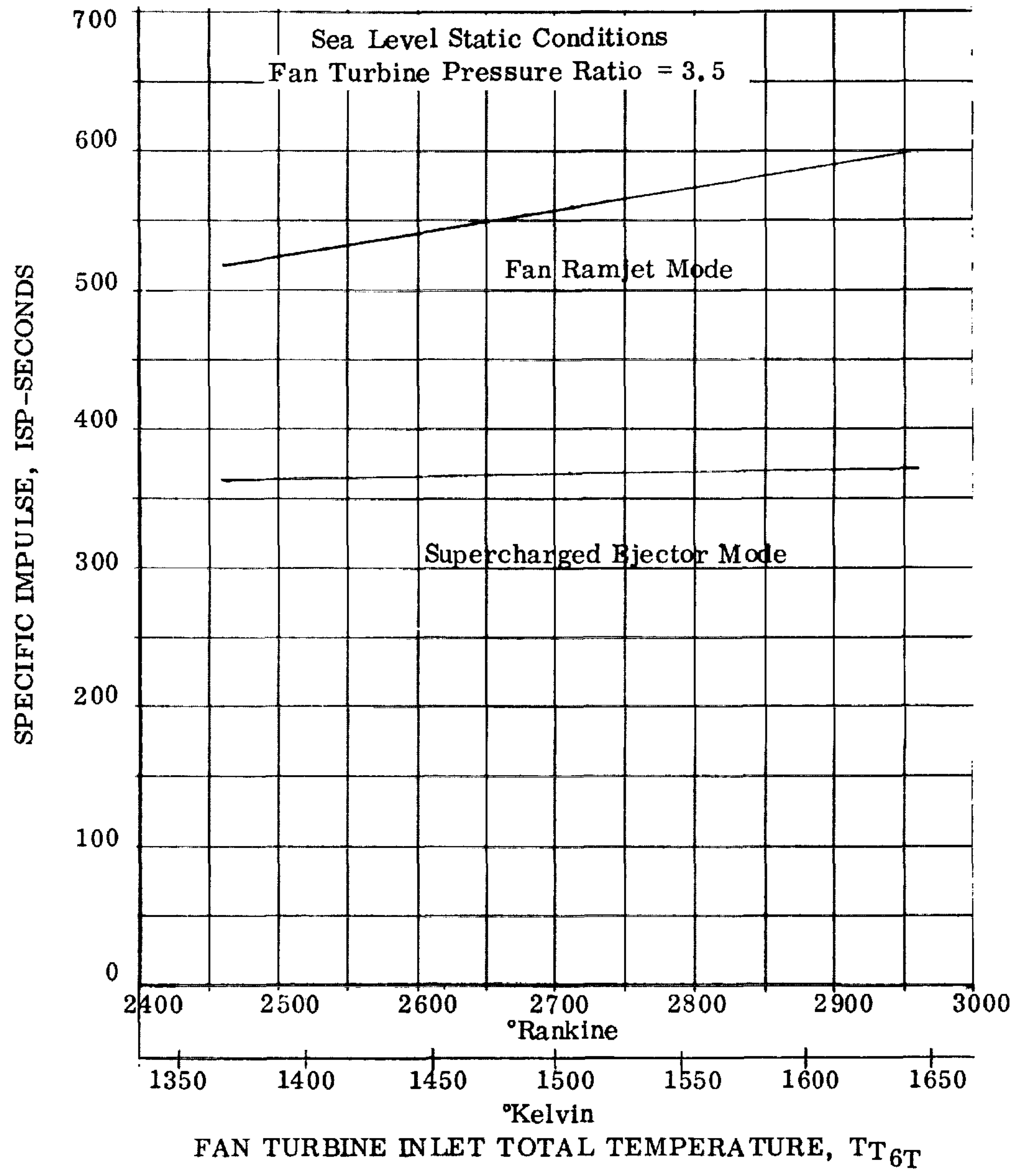


Figure 28. Effect of Turbine Inlet Temperature on Specific Impulse

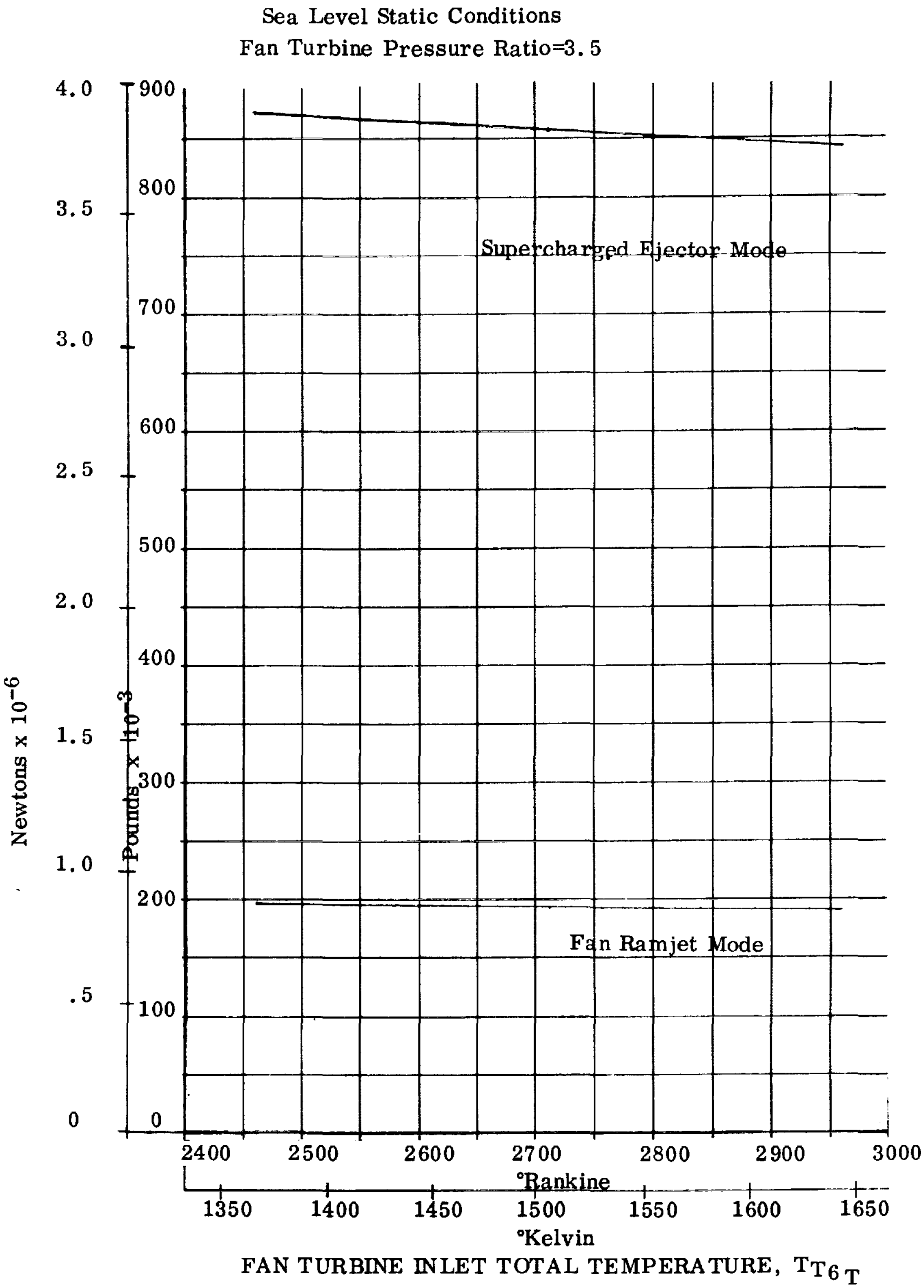


Figure 29. Effect of Fan Turbine Inlet Temperature on Engine Thrust

Engine ground test facility simulation requirements for engine No. 4 are presented in Table XXV . These requirements are typical for the engines studied in this program. There are a limited number of major airbreathing engine altitude test cells in the United States. These facilities are described in Table XXVI. Generally speaking, many of these test cells can simulate pressure and temperature conditions up to the Mach 3 condition. The AEDC APTU facility (when operational) and Marquardt's Cell 8 can simulate pressure and temperature requirements up to and including the Mach 4.5 flight condition. However, none of these facilities come close to meeting engine airflow requirements. In addition, the large airflow cells are blown down facilities, therefore, run time is limited.

To develop the engines characterized in this program, four program plan options exists:

1. Build a new facility or significantly modify an existing facility to meet engine simulation requirements.

2. Design and develop a reduced size engine - then install a large number of engines on the launch vehicle.

3. Design and develop a reduced size engine - then scale this engine to full size and proceed directly into the flight test program.

4. Design and develop the full size engine using only sea level static tests - then proceed directly into the flight test program.

Options 3 and 4 appear to be high risk programs; some reduction in engine size (option 2) and modification of an existing facility (option 1) appear to be an attractive approach.

TABLE XXV

ENGINE TEST FACILITY SIMULATION REQUIREMENTS
ENGINE NO. 4/HIGH "q" TRAJECTORY

Mach Number	Altitude Feet	Altitude Meters	Operating Mode	Freestream Total Temperature °R	Freestream Total Temperature °K	Fan Inlet Total Pressure psia	Fan Inlet Total Pressure Newtons/m^2	Engine Airflow #/Sec	Engine Airflow kg/sec	Exit Nozzle Throat Static Pressure psia	Exit Nozzle Throat Static Pressure Newtons/m^2
0	0	0	SERJ	519	288	14.7	$.101 \times 10^6$	1712	777	50.0	$.345 \times 10^6$
0.3	0	0	SERJ	528	293	15.2	$.105 \times 10^6$	1797	815	50.4	$.347 \times 10^6$
0.8	1,969	600	SERJ	577	321	20.1	$.139 \times 10^6$	2238	1015	51.8	$.400 \times 10^6$
1.0	14,764	4,500	SERJ	559	311	15.1	$.104 \times 10^6$	1721	781	50.5	$.348 \times 10^6$
1.3	9,843	3,000	SERJ	646	359	26.4	$.182 \times 10^6$	2696	1223	54.0	$.372 \times 10^6$
1.3	9,843	3,000	FRJ	646	359	26.4	$.182 \times 10^6$	2696	1223	23.7	$.163 \times 10^6$
2.0	29,528	9,000	FRJ	742	412	30.9	$.213 \times 10^6$	2729	1238	26.6	$.183 \times 10^6$
3.0	49,213	15,000	FRJ	1080	600	44.8	$.309 \times 10^6$	2626	1191	34.5	$.278 \times 10^6$
3.0	49,213	15,000	RJ	1080	600	44.8	$.309 \times 10^6$	2626	1191	21.9	$.151 \times 10^6$
4.5	68,898	21,000	RJ	1886	1048	95.1	$.656 \times 10^6$	2154	977	52.2	$.360 \times 10^6$

TABLE XXVI

MAJOR U.S. AIRBREATHING ENGINE ALTITUDE TEST CELLS

Organization Location Facility Name	Type of Facility	Test Section Size (Dimensions in feet)	Mach Range	Thrust Measuring Capacity (lb_f)	Total Temp. (°R)	Mass Flow Rate (lb/sec)
NASA Lewis Cleveland, Ohio Altitude Test Cell No. 1	Altitude	14 diam. x 24L (Test cell)	Not Available	Not Available	To 1060	To 450
Naval Air Propulsion Test Center Trenton, New Jersey Engine Altitude Test Cell Facility No. 3E	Altitude	17 diam. x 30L (Test cell)	To 3.0	50,000	395-1660	To 550
Naval Air Propulsion Test Center Trenton, New Jersey Engine Multipurpose Test Cell Facility No. 3W	Altitude	33Wx33Hx196L	Not Available	Not Available	395-680	To 350
AEDC Arnold Air Force Station, Tennessee Propulsion Wind Tunnel - 16T	Closed-circuit, single-return, variable density, continuous flow	16Wx16Hx40L	.2-1.6	Not Available	410-620	Not Available
AEDC Arnold Air Force Station, Tennessee Propulsion Wind Tunnel - 16S	Closed-circuit, single-return, variable density, continuous flow	16Wx16Hx40L or 16Wx16Hx20L	1.5-4.75	Not Available	560-1110	Not Available
NASA Lewis Cleveland, Ohio 10-Foot by 10-Foot Supersonic Wind Tunnel	Closed or open circuit, variable density, continuous flow	10x10x40L	2-3.5	Not Available	500-785	Not Available
NASA Lewis Cleveland, Ohio 8-Foot by 6-Foot Supersonic Wind Tunnel	Closed or open circuit, continuous flow	8Wx6Hx39L	.8-2.1	Not Available	600-700	Not Available
NASA Lewis Plumbrook Station; Sandusky, Ohio Hypersonic Tunnel Facility	High temperature, blowdown (Free jet or direct connect)	3.5 diam. free jet nozzles	5,6,67	10,000 (Free jet) 20,000 (Direct connect)	To 4800	To 220
Ordnance Aerophysics Laboratory Daingerfield, Texas Altitude Test Cell No. 6	Altitude, continuous or blowdown, (Free jet or direct connect)	15 diam. x 7½L (4.19 diam. free jet nozzle max. or 4 diam. bellmouth diam. max.)	1-5	60,000	to 3960	50-1600
AEDC Arnold Air Force Station, Tennessee Airbreathing Propulsion Test Unit	Altitude, blowdown (Free jet or direct connect)	Not Available	Not Available	Not Available	Not Available	Not Available
AEDC Arnold Air Force Station, Tennessee Altitude Test Cell T-2	Altitude (Free jet or direct connect)	Not Available	Not Available	Not Available	Not Available	Not Available
AEDC Arnold Air Force Station, Tennessee Altitude Test Cell T-4	Altitude (Free jet or direct connect)	Not Available	Not Available	Not Available	Not Available	Not Available
AEDC Arnold Air Force Station, Tennessee Altitude Test Cell J-1	Altitude (Free jet or direct connect)	Not Available	Not Available	Not Available	Not Available	Not Available
AEDC Arnold Air Force Station, Tennessee Altitude Test Cell J-2	Altitude (Free jet or direct connect)	Not Available	Not Available	Not Available	Not Available	Not Available
AEDC Arnold Air Force Station, Tennessee Propulsion Research Test Bed R-2B	Altitude (Free jet or direct connect)	Not Available	Not Available	Not Available	Not Available	Not Available
Allison Indianapolis, Indiana Altitude Test Cell No. 1	Altitude (direct connect)	18 diam. x 60L (Test cell)	Not Available	30,000	395 to Ambient	To 450
General Electric Co. Evendale, Ohio Altitude Test Cell No. 43	Altitude (direct connect)	17 diam. x 56L (Test cell)	To 3	100,000 (Thrust frame capacity)	410-1110	1000 (ram only) 400 (ram exhaust)
United Aircraft Corp. E. Hartford, Conn. Altitude Engine Test Stand No. X-210	Altitude (direct connect)	22.5L (Test cell)	To 3	25,000	425-1135	To 384
United Aircraft Corp. West Palm Beach, Fla. Altitude Test Cell No. C-4	Altitude (direct connect)	6.5 diam. x 20.7L (Test cell)	To 3.2	Not Available	To 1160	To 480
Air Force-Marquardt Jet Laboratory Van Nuys, California Full Scale Altitude Cell No. 8	Altitude (Free jet or direct connect)	16 diam. x 80L (Bellmouth diameters to 2.67 ft)	.8-4.2	100,000	To 3000	To 800

CONCLUDING REMARKS

Seven composite engines were designed for application to a resuable single-stage-to-orbit vehicle. The engines were variations of the Supercharged Ejector Ramjet engine. The results form a data base for evaluation of this class of composite engine to various missions. An examination of possible improvements indicated that the Fan Ramjet specific impulse can be increased significantly if a multi-stage turbine is used. A study of ground test facilities indicated reduced size engines should be considered to minimize facility improvement requirements.

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APPENDIX A

COMPOSITE PROPULSION SYSTEMS FOR AN ADVANCED REUSABLE LAUNCH VEHICLE APPLICATION

by

J. G. Bendot

The Marquardt Company, USA

Paper presented at the 2nd International Symposium on Air Breathing Engines held in March 1974 at Sheffield.

COMPOSITE PROPULSION SYSTEMS FOR AN ADVANCED REUSABLE LAUNCH VEHICLE APPLICATION

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ABSTRACT

The composite engine combines the best features of airbreathing and rocket engines into a simple, integrated, highly flexible propulsion system. These propulsion systems feature multimodal operation capability with cycle process interactions between engine components. Increased engine performance results from this synergistic design approach. Examples of composite engines are the Ejector Ramjet (ERJ), Supercharged Ejector Ramjet (SERJ), SCRAMLACE and Ejector SCRAMJET. This class of engine is also frequently referred to as mixed cycle engines, rocket ramjet engines, and less frequently compound cycle engines.

This paper will summarize a detailed engineering study conducted to evaluate the potential of composite engines when applied to the first stage of a two-stage manned advanced reusable launch vehicle. Briefly, the launch vehicle mission/ design constraints were as follows:

- Reusable vehicle, passenger/light cargo payload
- Two stage to 262 nautical mile (485 kilometer) orbit
- Horizontal takeoff and landing
- Hydrogen/Oxygen (rocket engine only) propellants
- One million pound (453600 kilogram) vehicle takeoff gross weight
- Full mission profile, liftoff to landing with 3 "g" acceleration limit (manned application).

Payload in orbit was the prime evaluation criteria.

A total of 36 composite engines were evaluated for this mission/application. In all cases considered, composite engines were clearly shown to be superior when compared to an advanced liquid rocket engine. Payload in orbit results are presented for the more attractive composite engines. In addition, vehicle payload performance as a function of vehicle staging velocity is presented. Typical staging velocities are Mach 8 to 10.

INTRODUCTION

Launch vehicles in which all components are fully reusable must eventually be developed if truly low cost space launch operations are to be achieved. The horizontal takeoff and landing airbreathing launch vehicle has inherent features which make it a prime candidate for future second generation shuttle systems.

The rocket engine is characterized by high thrust to weight ratio but at low specific impulse values. In contrast, true airbreathing engines are characterized by high specific impulse performance but engine weights are heavy, The composite engine combines the best features of rocket and airbreathing engines into simple integrated, highly flexible propulsion systems.

As suggested in Figure 1, a fully reusable launch vehicle (two stage to orbit mission), powered by composite engines, may offer major improvements in payload in orbit performance. This paper summarizes a detailed engineering study conducted to evaluate the potential of the composite engine.*

COMPOSITE ENGINE SYNTHESIS, DESIGN, AND OPERATION

The elemental propulsion systems which provide the basic building blocks for synthesizing composite engines are the familiar rocket and airbreathing systems, which are symbolically illustrated in Figure 2.

If it is desired to incorporate the features of both elements (rocket and airbreather) in a single vehicle, two approaches are obvious: the elements may be installed either separately or integrally. The former may be termed a combination propulsion system. Thus, to illustrate the contrast, combination propulsion systems incorporate two or more elemental engine types in a nonintegrated installation, i.e., with little or no direct physical or process interaction between engine types within the vehicle's propulsion complement.

If, however, the elements are physically integrated into a single propulsion system, having multimodal operation capabilities, with cycle process interactions between elements, the result is a composite propulsion system. Increased engine performance results from this synergistic design approach.

Marquardt approached the composite engine from the standpoint of expanding and developing the functions and operation of the basic ramjet cycle. Thus a clear initial goal was to provide the ramjet with low speed thrust capability such that it could accelerate under its own power to ramjet cruise conditions. The Ejector Ramjet is thus the most simple composite engine which integrates the rocket function and ramjet function into a single integrated engine (see Figure 3). The Ejector Ramjet engine has two operating modes: (1) Ejector Mode and (2) Ramjet Mode (high flight speed operation).

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*This study was sponsored by the National Aeronautics and Space Administration (NASA) under Contract NAS7-377. Marquardt was supported in this study by the Lockheed-California Company and Rocketdyne.

Manned high supersonic/hypersonic speed aircraft will cruise on ramjet power. However, such aircraft also require low speed cruise/loiter capability at low specific fuel consumption. The Supercharged Ejector Ramjet (SERJ) engine meets these requirements. In this engine, a low pressure ratio fan/gas generator is integrated with the ejector primaries and ramjet/afterburner. The fan provides an additional pressure rise for the high thrust/acceleration Ejector Ramjet operating mode, or Fan Ramjet operation (i.e. augmented turbofan) for intermediate speed acceleration, or low speed cruise/loiter capability with the ejector and ramjet components inoperative, (Fan Mode). During ramjet operation, the fan may be allowed to windmill or be removed from the airstream. The windmilling technique has been experimentally demonstrated.

The Ejector Ramjet and SERJ engines can be designed for cryogenic (i.e. liquid hydrogen, liquid methane) or storable propellants. Advanced high performance composite propulsion systems which operate over a very wide flight speed range have been established by combining ramjet, Ejector Ramjet, SERJ, supersonic combustion and/or LACE engine technology. The Ejector SCRAMJET, RAMLACE, and SCRAMLACE engines are shown in Figure 4. Figure 5 illustrates the Supercharged Ejector SCRAMJET, Supercharged RAMLACE and Supercharged SCRAMLACE engines.

The basic LACE engine cycle uses the large cooling capacity of liquid hydrogen to liquefy air. Therefore, this basic engine cycle and its derivatives are limited to the use of liquid hydrogen fuel. The Supercharged Ejector SCRAMJET and Ejector SCRAMJET engines can be designed to operate on cryogenic fuels or storable propellants. The basic technology to develop these advanced engines has largely been demonstrated; however, development of these engines will be more costly and will require a longer development period than for either the Ejector Ramjet or Supercharged Ejector Ramjet engine.

Figures 4 and 5 described several composite engines which use the basic LACE engine as the ejector primary/rocket subsystem. These engines are RAMLACE, SCRAMLACE, Supercharged RAMLACE and Supercharged SCRAMLACE. The performance of these engines can be significantly improved if the cooling capacity of liquid hydrogen can be increased.' One demonstrated approach is the use of slush hydrogen (super cooling) which lowers the hydrogen boiling point from 36°R (20°K) to approximately 25°R (14°K). The performance potential of this technique as applied to the RAMLACE engine is illustrated in Figure 6. In this composite engine application study, this cooling/improved performance concept was evaluated for the following engine designs:

Recylcled RAMLACE
Recycled SCRAMLACE
Recycled Supercharged RAMLACE
Recycled Supercharged SCRAMLACE

The ejector mixing and pumping (jet compression) offered by the ejector primary/rocket subsystem forms the heart of the Marquardt developed composite engine cycles. The kinetic energy of high pressure ejectors is used to induce airflow at low speed conditions and at all flight speeds to raise the total pressure level of the mixed air/primary system. As with all airbreathing engines, thrust and cycle efficiency increase as the cycle pressure ratio is increased.

The continuity, momentum and energy equations relate the aerothermodynamic properties of the fully mixed air-primary (i.e. mixer exit) to mixer entrance flow conditions. Experimental mixing data are in close agreement with predictions. However, the length required to achieve full mixing cannot be analytically predicted and, therefore, must be experimentally correlated.

Representative ejector/mixer performance in terms of total pressure ratio is presented in Figure 7 as a function of secondary/primary mass flow ratio or more simply engine airflow/ejector propellant flow ratio. Achievable ejector/mixer total pressure ratios are modest, therefore, the resultant improvement in engine performance is maximum at lower flight speeds.

Marquardt has conducted an intensive jet compression research program. Initial ejector primary propellants (fluids) included heated air, hydrogen/air and hydrogen/oxygen. Later tests included hydrocarbon fuels and hydrogen peroxide as the oxidizer. As a result of this work, the required mixer length was correlated as a function of the number of primary nozzles, the primary exit Mach number, air/primary flow rate ratio, primary/secondary total temperature ratio and mixer/primary geometry (see Figure 7). Efficient jet compression with short mixer lengths has been demonstrated.

The Ejector Ramjet engine cycle has been successfully demonstrated in several engine test programs. Initial small scale demonstrations were accomplished with hydrogen/air and hydrogen/oxygen ejector primary subsystems. Later, two 18-inch diameter Ejector Ramjet engine demonstration programs were conducted using hydrocarbon fuel/hydrogen peroxide propellants. A photograph of the second test engine is presented in Figure 8. Briefly these test programs demonstrated the following:

- Experimental thrust stand performance agreed within ± 5% of predicted performance.
- Both engine operating modes were demonstrated.
- Mode transitions were demonstrated.
- Ejector primary throttling was demonstrated.
- Afterburner throttling was demonstrated.
- Static and high flight speed operation were demonstrated.

Thus, the composite engine has been convincingly demonstrated in scale engine test programs.

The composite engine is characterized by its multi-operating mode capability and resultant mission flexibility. The flexibility of this class of engine is illustrated in Figure 9. Specifically, this figure describes the operating modes of the Supercharged Ejector SCRAMJET engine when used to power an advanced reusable launch vehicle. A single stage to orbit option is apparent.

Studies conducted by Marquardt and several major airframe companies have shown that while composite engine multimode operating flexibility is a valuable asset, the mission optimization process is complex and requires more effort than for single operating mode engines.

ADVANCED LAUNCH VEHICLE APPLICATION

Marquardt evaluated the performance potential of composite engines when applied to an advanced reusable launch vehicle. Briefly, the launch vehicle/mission design constraints were as follows:

- Reusable vehicle, passenger/light cargo payload
- Two-stage to 262 nautical mile (485 kilometer) orbit
- Horizontal takeoff and landing
- Hydrogen/oxygen (rocket engine only) propellants
- One million pound (453600 kilogram) vehicle takeoff gross weight
- Full mission profile, liftoff to landing and 3"g" acceleration limit.

Payload in orbit was the prime evaluation criteria.

A total of 36 composite engines were evaluated for this application/mission; however, the engines of primary interest have been reviewed in this paper. Study results were compared to "Very Advanced" rocket and Turboramjet engine performance.

The baseline composite engine fully reusable launch system is a two-stage, horizontal takeoff and landing, nested lifting body configuration. The first stage provides an aerodynamic pressure field for the inlets of the integrated propulsion system. The all-rocket second stage vehicle is fully recoverable and reusable and was established in a previous study program.* The design of this vehicle was not perturbed; rather, it was scaled in accordance with first stage capabilities.

An orbital launch system was defined for each composite propulsion system. Figure 10 describes a representative system. Specifically, the first stage of this launch vehicle is powered by SCRAMLACE engines. Figure 11 presents an artist's rendering of this vehicle.

In general terms, composite engines operate on ejector (primary rocket) mode from Sea Level Static to as high as Mach 2, where ramjet mode transition occurs. For engines employing the subsonic combustion mode, the maximum airbreathing Mach number is 8. For engines employing the SCRAMJET mode, the transition from subsonic to supersonic combustion occurs at Mach 6 and this high speed mode nominally continues to Mach 12. Specifically, engine operating modes/transition speeds were optimized in terms of minimum propellant plus engine weight for each composite propulsion system. Figure 12 illustrates the broad range of composite engine performance during ejector mode operation. Corresponding Sea Level Static Engine thrust to weight ratios are also presented.

*NASA Contract NAS8-11463

A stated objective of this study was to compare composite engine mission results with "Very Advanced" rocket engine propulsion. For this study, "Very Advanced" rocket engines were defined as follows:

Propellants	Liquid Hydrogen/Oxygen
Cycle	Fuel Rich Tap Off
Oxidizer/Fuel Ratio	6.5
Chamber Pressure	2000 psi (141 kg/cm^2)
Specific Impulse, Sea Level	375 sec.
Thrust/Weight Ratio	179

The results of the mission analysis study are presented in Figure 13 in terms of payload in orbit and system total dry/gross payload weight ratio. System cost effectiveness was specifically not an objective of this study. However, system total dry/gross payload weight ratio is a rough indicator of total system cost.

With these results, two composite engines were selected for further detailed study. An objective of this study was to conduct an assessment and evaluation of engine technology requirements. This resulted in the selection of the two composite engines; (1) the first engine should provide attractive payload in orbit performance yet only require near term technology for its successful development while (2) the second engine should provide near maximum payload in orbit performance recognizing major advances in engine technology would be required for its successful development. The selected engines were: (1) near term technology - Supercharged Ejector Ramjet engine and (2) advanced engine technology - SCRAMLACE engine.

These two engines and comparison "Very Advanced" rocket and Turboramjet engines were intensely studied. For example, optimum* launch vehicle staging velocity was defined (see Figure 14). In addition, study of the "Very Advanced" rocket powered vehicle was expanded. Specifically, three takeoff modes were evaluated:

1. Horizontal takeoff and landing (internal landing gear)
2. Horizontal takeoff and landing (rocket sled assisted takeoff)
3. Vertical takeoff.

Optimum staging velocities for these vehicles are also presented in Figure 14.

In the same format as previously presented, the results of this detailed study are presented in Figure 15. This figure compares the following engines:

- "Very Advanced" Rocket
- Supercharged Ejector Ramjet (SERJ)
- SCRAMLACE
- Turboramjet.

It should be noted that the payload in orbit values are the maximum from Figure 14.

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*In terms of payload in orbit performance

These results indicate a more favorable position for the "Very Advanced" rocket engine particularly with the alternate takeoff modes. These results notwithstanding, the payload in orbit potential of the composite engine is clearly shown. The Turboramjet engine shows excellent growth potential; however, development of the Supercharged Ejector Ramjet engine could be accomplished for a fraction of the development cycle and cost of the Turboramjet engine.

Although the Ejector Ramjet engine was not evaluated in the detailed engineering study, its high performance/low development cost potential should be recognized. If first stage loiter is not established as a requirement, this conclusion becomes stronger. With this engine, development of rotating machinery is not required and this engine cycle has been convincingly demonstrated in several engine test programs.

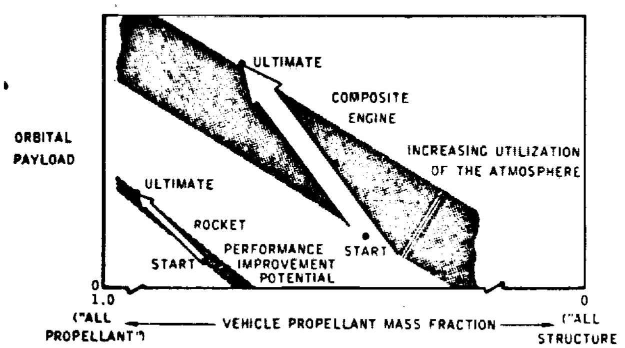


Figure 1. Reusable Launch Vehicle Performance - Composite Engine VS Rocket

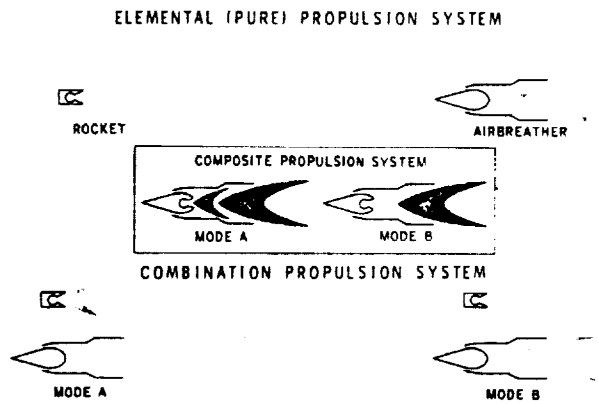


Figure 2. Propulsion Classes

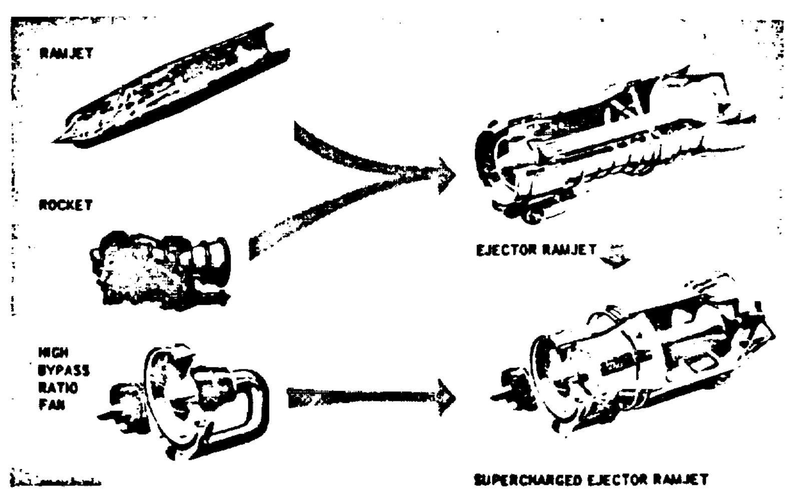


Figure 3. Gensis of The ERJ and SERJ Composite Engines

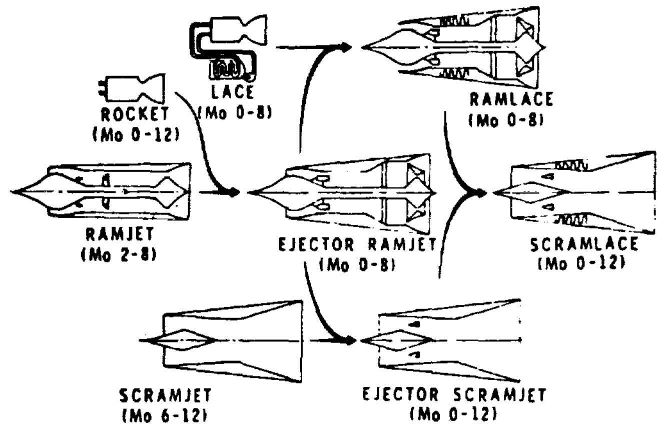


Figure 4. Evolution of Advanced Propulsion Systems

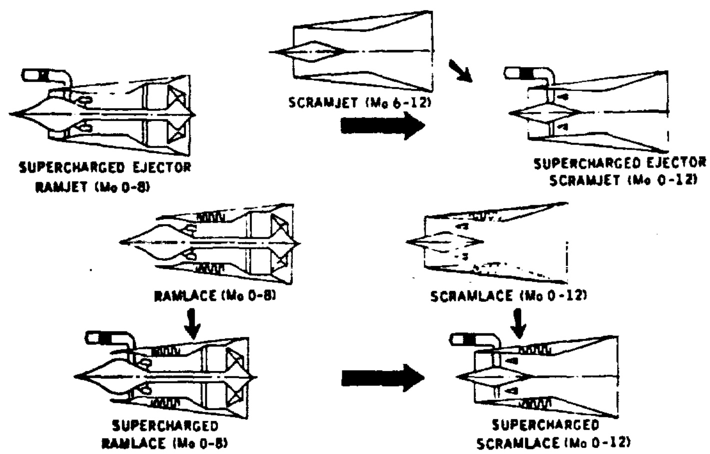


Figure 5. Evolution of Advanced Propulsion Systems (Continued)

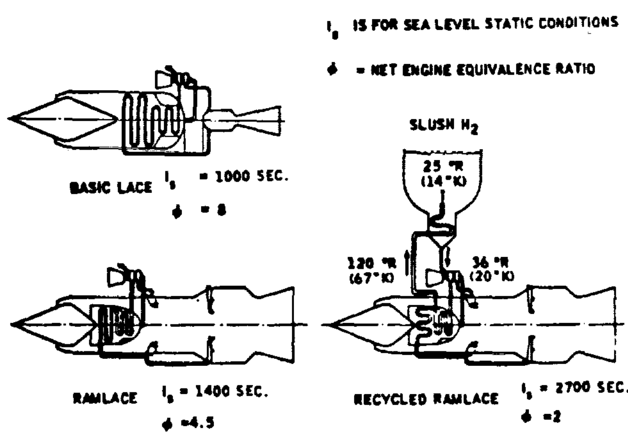


Figure 6. Air Liquefaction Cycles

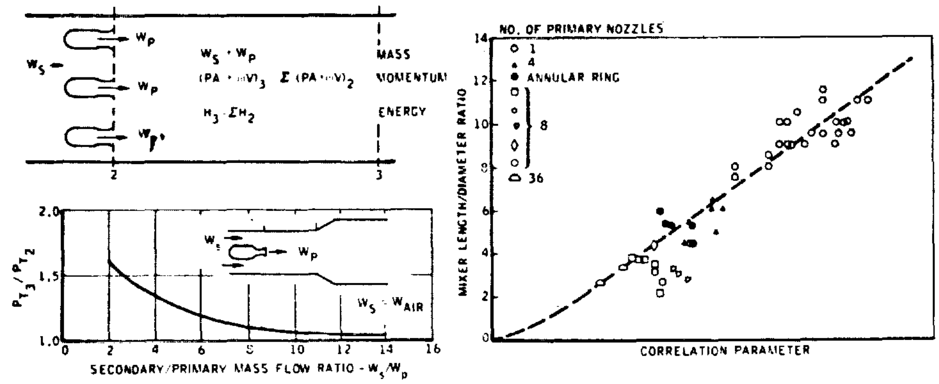


Figure 7. Ejector Mixing And Pumping

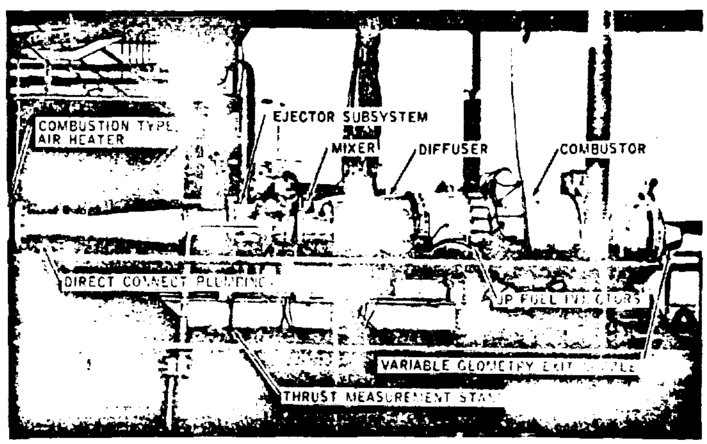


Figure 8. Ejector Ramjet 18 Inch Engine Installation

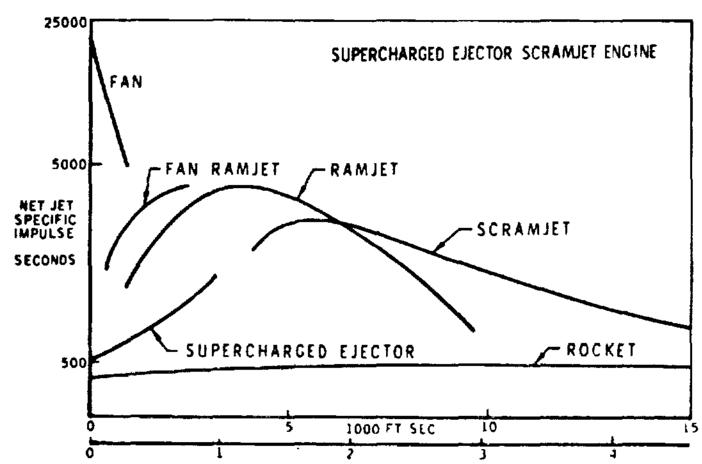


Figure 9. Composite Engine Multimode Operation

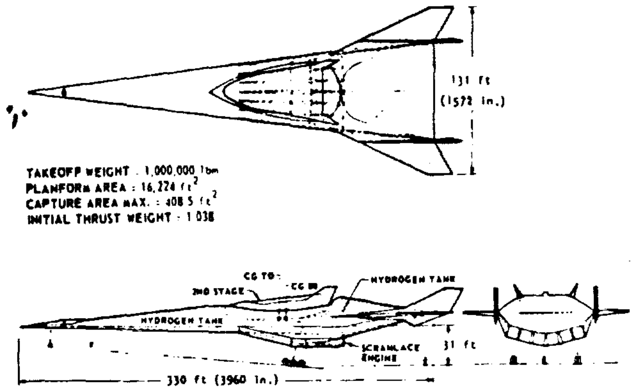
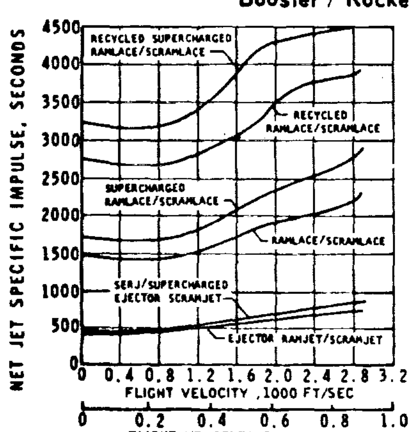


Figure 10. Baseline Reusable Launch Vehicle

Figure 11. Airbreathing Launch Vehicle - Composite Engine Powered Booster / Rocket Powered Orbiter



ENGINE	THRUST TO WEIGHT RATIO*
EJECTOR RAMJET	35.0
EJECTOR SCRAMJET	37.8
SERJ	24.9
SUPERCHARGED EJECTOR SCRAMJET	26.6
RAMLACE	18.7
SCRAMLACE	19.8
SUPERCHARGED RAMLACE	15.8
SUPERCHARGED SCRAMLACE	17.0
RECYCLED RAMLACE	14.2
RECYCLED SCRAMLACE	15.0
RECYCLED SUPERCHARGED RAMLACE	14.2
RECYCLED SUPERCHARGED SCRAMLACE	15.0

*SEA LEVEL STATIC CONDITIONS
THRUST= 250000 LB (113400 KG)

Figure 12. Engine Performance Comparison - Ejector Mode Operation

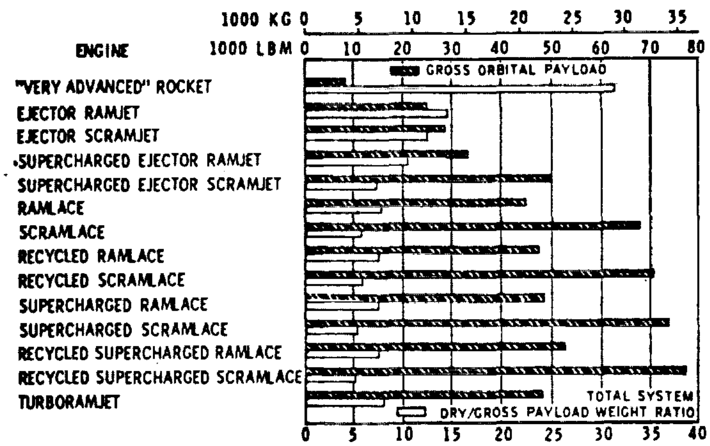


Figure 13. Payload in Orbit Performance Summary

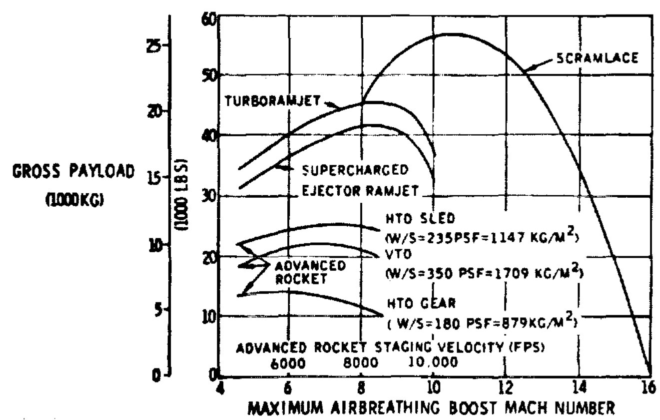


Figure 14. Payload in Orbit Vs. Launch Vehicle Staging Velocity

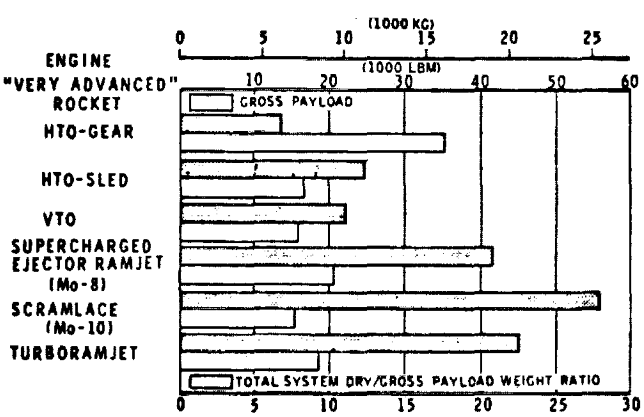


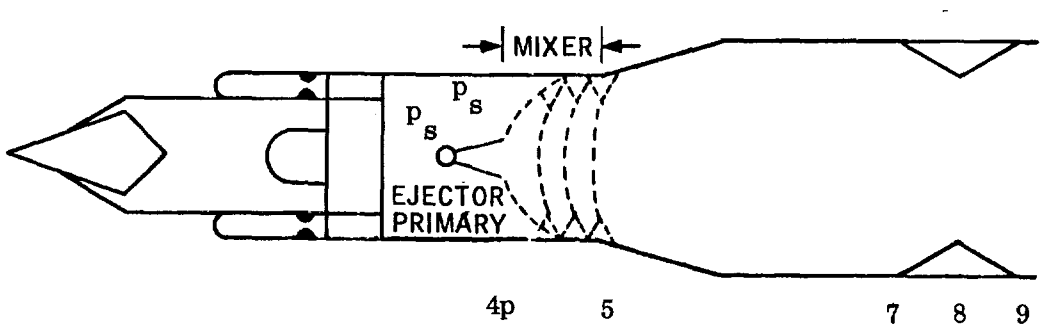
Figure 15. Payload in Orbit Performance Summary

APPENDIX B

SUPERCHARGED EJECTOR RAMJET ENGINE

ROCKET OPERATING MODE (VACUUM)

(A) Operating Mode Schematic



(B) Major assumptions:

1. The inlet is blocked off and sealed.
2. Static pressure p_s is sufficiently low that primary nozzle remains choked.
3. For structural design reasons, mixer exit static pressure, p_5, is not to exceed 150 psia.

(C) General comments:

1. Conditions at Station 5 are defined by throat area A_8. If A_8 is small, Station 5 is subsonic; if A_8 is sufficiently large, Station 5 and downstream of that station is supersonic. Minimum engine back pressure, p_s, occurs with supersonic solution.
2. As A_8 is reduced, M_5 is reduced and p_s and p_5 are increased.
3. A limiting condition exists when p_5 = 150 psi (engine structural design limit).
4. If the pressure at Station 5 were increased sufficiently, the ejector primary nozzle would unchoke and the primary flow rate would decrease.

(D) Subsonic solution - method of approach:

1. Assume Mach number M_5 and that flow fills mixer area A_5.

2. Define total pressure at Station 5, assuming adiabatic flow.

$$w_p = k_{3p} S_{3p} P_{T_p} = k_5 A_5 P_{T_5} \qquad \text{where } k = f(M);\ M_{3p} = 1.0$$

$$\frac{P_{T_5}}{P_{T_p}} = \frac{k_{3p} A_{3p}}{k_5 A_5} \qquad \text{where } P_{T_p} = 1494 \text{ psia}$$

Note: k_{3p} and k_5 are defined from equilibrium chemistry data.

P_{T_5} vs M_5 plotted in Figure 1 attached. Note that M_5 is limited to $\geqq$ .051.

3. Size throat area A_8

$$w_p = k_5 A_5 P_{T_5} = k_8 A_8 P_{T_8}$$

assume $P_{T_5} = P_{T_8}$; $T_{T_5} = T_{T_8}$

$$\text{than } A_8 = \frac{k_5 A_5}{k_8} = \frac{k_5 A_5}{k_{3p}}$$

4. Define nozzle expansion ratio A_9/A_8 where A_9 = physical max. area for engine exit nozzle. Note for engine No. 1, this area ratio $\cong$ 41.0 for a range of O/F ratios.

5. Determine Isp from chemical equilibrium solutions. (Note: constant γ cannot be used to compute M_9 and stream thrust). A typical chemical equilibrium solution at $\phi = 1$ (O/F = 7.936) is shown in Figure 2 for a range of chamber pressures. At an area ratio A_9/A_8 of 41 for a chamber pressure of 150 psia the ideal vacuum specific impulse is 428 seconds. This impulse can be increased to 454 seconds by use of an O/F ratio of $\sim$ 4.75 as shown in Figure 3 .

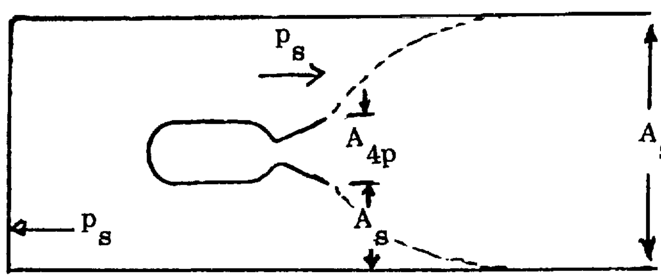
At this point, it is necessary to make a note about exit nozzle variable geometry requirements. The A_9/A_8 value of 41.0, which corresponds to a chamber pressure of 150 psia, is larger than the (A_9/A_8) value corresponding to ramjet operation at Mach 8.0 (i.e. $A_9/A_8 \cong 27.0$).

If the minimum value of A_8 is assumed established by the Mach 8 ramjet requirement, the chamber pressure drops from 150 psia to 114 psia and the ideal vacuum impulse also drops slightly. This effect is shown in Figure 3 for comparison. There is no limit to the minimum A_8 of the translating ring nozzle, and therefore the higher impulse levels of Figure 3 are believed attainable.

6. The ideal specific impulse values presented in Figure 3 were reduced 4% to account for real nozzle effects (See Table III).

7. Check nozzle back pressure. The above solution is valid provided that the primary nozzle remains choked. This requires an interactive type solution as follows:

   (a) Assume primary nozzle is flowing full for starting point



continuity: $\dfrac{p_5}{p_{4p}} = \dfrac{\mathring{m}_{4p}\, A_{4p}}{\mathring{m}_5\, A_5}$ i.e. $T_{Tp} = T_{T5}$

momentum: $\dfrac{p_s}{p_{4p}}\;\dfrac{A_s}{A_{4p}} + (f/p)_{4p} = \dfrac{\mathring{m}_{4p}}{\mathring{m}_5}\;(f/p)_5$

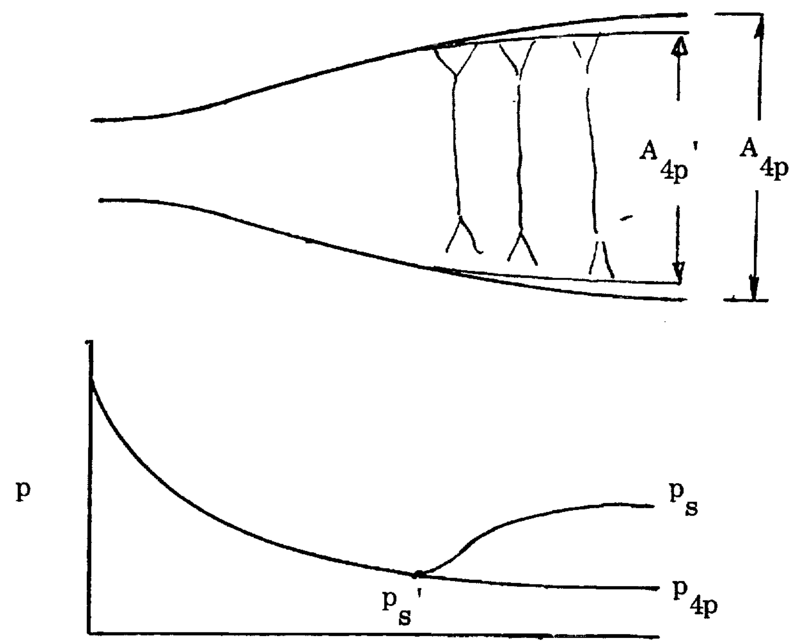
geometry: $A_s + A_{4p} = A_5$

definition: $\mathring{m} = g\sqrt{\dfrac{\gamma}{R}}\; M \left[1 + \dfrac{\gamma-1}{2}\; m^2\right]^{1/2}$

for given M_5, $\dfrac{p_5}{p_{4p}}$ is defined from continuity equation

$\dfrac{p_s}{p_{4p}}$ then defined from momentum equation

Note: if $\dfrac{p_s}{p_{4p}} > 2.0$, the primary nozzle will separate, i.e.



PRIMARY NOZZLE STATION

Thus if $\frac{p_s}{p_{4p}} > 2$, the full flow solution cannot be used,

and an effective primary flow area A_{4p}', must be used.

(b) Method of defining effective primary flow area

defining $p_s' = \frac{p_s}{2}$ as in above sketch

then $\frac{p_s'}{P_{T_p}} = \frac{p_s}{2\ P_{T_p}}$

A

then $\frac{3p'}{A_{4p'}}$ given in Figure 4 (Chemical Equilibrium Data)

and $\Delta A_{4p} = A_{4p} - A_{4p'}$

from continuity define new Mach number at primary exit, i.e.

$$\dot{m}_s = \frac{P_{3p}}{P_{T_p}} \frac{P_{T_p}}{p_s} \dot{m}_{3p} \frac{A_{3p}}{A_{4p'}}$$

$$M_s = f\,(\dot{m}_s)$$

$$(f/p)_s = 1 + \gamma M_s^{\,2}$$

new momentum equation:

$$\frac{A_s + \Delta A_{4p}}{A_{4p}} + \frac{A_{4p}}{A_{4p}} (f/p)_s = \frac{p_5}{p_s} \frac{A_5}{A_{4p}} (f/p)_5$$

$$\text{where } \frac{p_5}{p_s} = \frac{A_{4p'}}{A_{4p}} \frac{A_{4p}}{A_5} \frac{\dot{m}_s}{\dot{m}_5}$$

(c) For a given M_5, the solution proceeds by assuming various values of p_s until agreement is reached between momentum and continuity equations.

(d) Subsonic solution for back pressure p_s is shown in Figure 5. . For a mixer Mach number less than 0.02, it can be expected that the primary nozzle would unchoke. For the selected operating point (i.e. $M_5 = .051$ resulting in a chamber pressure of 150 psia) the flow in the primary nozzle is supersonic although separated. In terms of absolute units, the back pressure is

$$p_s = \frac{p_s}{p_{4p}} \frac{p_{4p}}{P_{T_p}} P_{T_p} \cong 10.3 \times \frac{1}{92.93} \times 1494 = 165.6 \text{ psia}$$

(E) Supersonic solution - method of approach

1. Assume flow fills mixer area A_5 and there is no local choke point downstream. (A_8 sufficiently large not to choke).

2. Continuity: $\dfrac{p_5}{p_{4p}} = \dfrac{\dot{m}_{4p}}{\dot{m}_5} \dfrac{A_{4p}}{A_5}$

   Momentum: $\left(\dfrac{f}{p}\right)_{4p} + \dfrac{p_s}{p_{4p}} \dfrac{A_s}{A_{4p}} = \dfrac{p_5}{p_{4p}} \left(\dfrac{A_s}{A_{4p}} + 1\right)\left(\dfrac{f}{p}\right)_5$

   Condition: $p_s \equiv p_5$ This is required according to NACA RME 5IEOI.

3. Assume M_5 defining $\dot{m}_5$, $(f/p)_5$, $\dfrac{p_5}{p_{4p}}$, k_5

   Check momentum equation; vary M_5 until agreement reached.

4. $$\frac{P_{T_5}}{PT_p} = \frac{k_{3p} A_{3p}}{k_5 A_5}$$

5. For M_5 of step 3 determine an effective $(A/A^*)_5$ from chemical equilibrium data. Then assuming neither temperature nor pressure losses between mixer station 5 and exit station 9.

$$(A/A^*)_9 = \left(\frac{A_9}{A_5}\right) \left(\frac{A}{A^*}\right)_5$$

6. From chemical equilibrium determine ideal specific impulse for the area ratio of step 5.

7. Specific impulse performance presented below and in Figure 6 :

Ideal Isp = 458 sec @ O/F = 7.936; PT_5 = 612; p_s = 1.59 psi

= 478 sec @ O/F = 5.25; PT_5 = 852; p_s = .98 psi

= 467 sec @ O/F = 3.50; PT_5 = 721; p_s = .75 psi

8. The above performance were reduced 4% to account for real nozzle effects (See Table III).

9. A word of caution is necessary for the supersonic solution. It is necessary that the minimum flow area A_8 be large enough not to choke. For the variable exit nozzle geometry configuration established for this engine, the throat area A_8 can be as large as the combustor area A_7 and there is, therefore, no geometric area contraction. However, there are large flow deflections in passing through the exit nozzle. Whether this would cause choking is unknown and would have to be experimentally determined.

Rocket Operating Mode

Subsonic Solution

$\phi = 1.0$

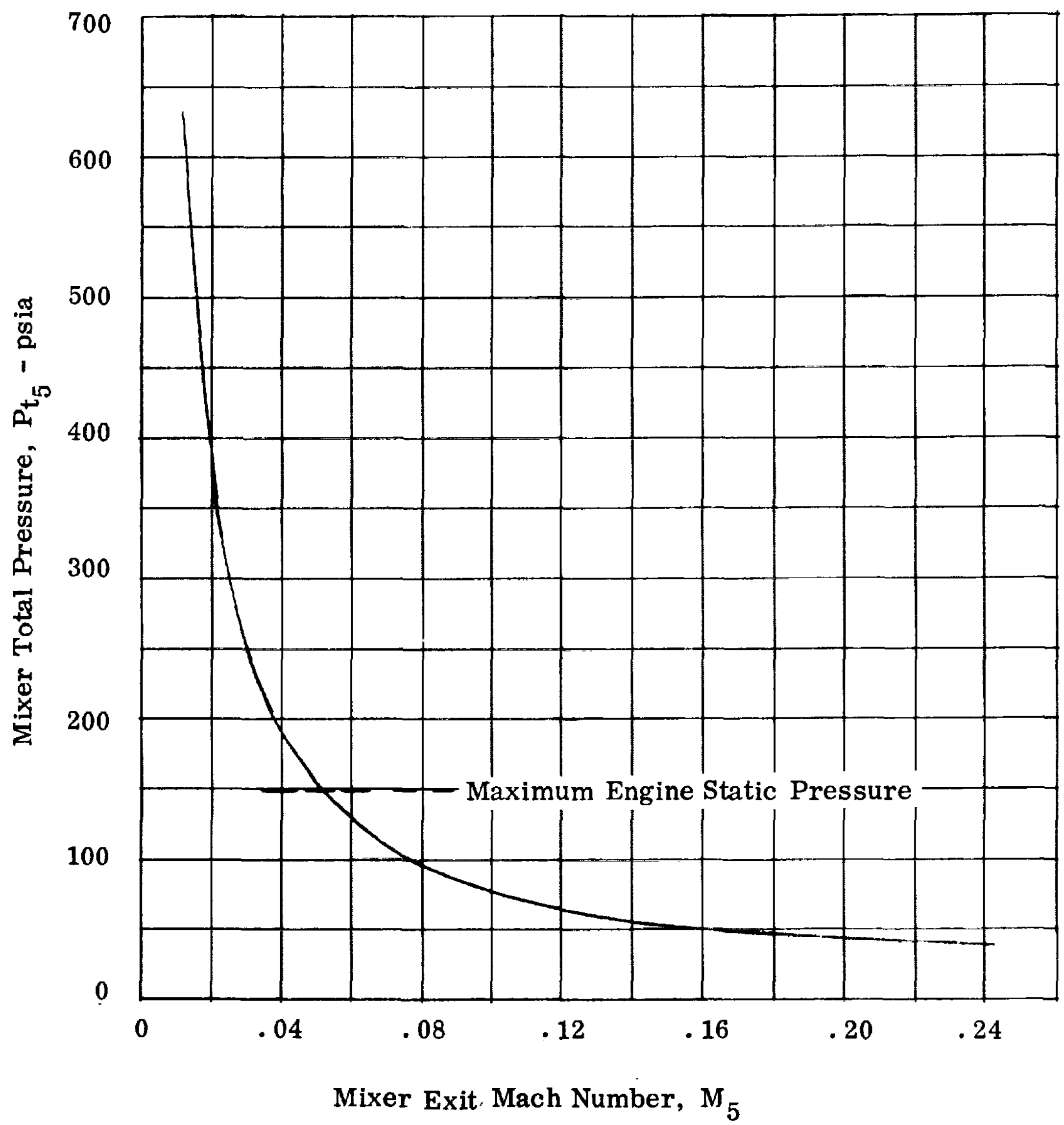


Figure 1 . Engine No. 1 - Mixer Total Pressure

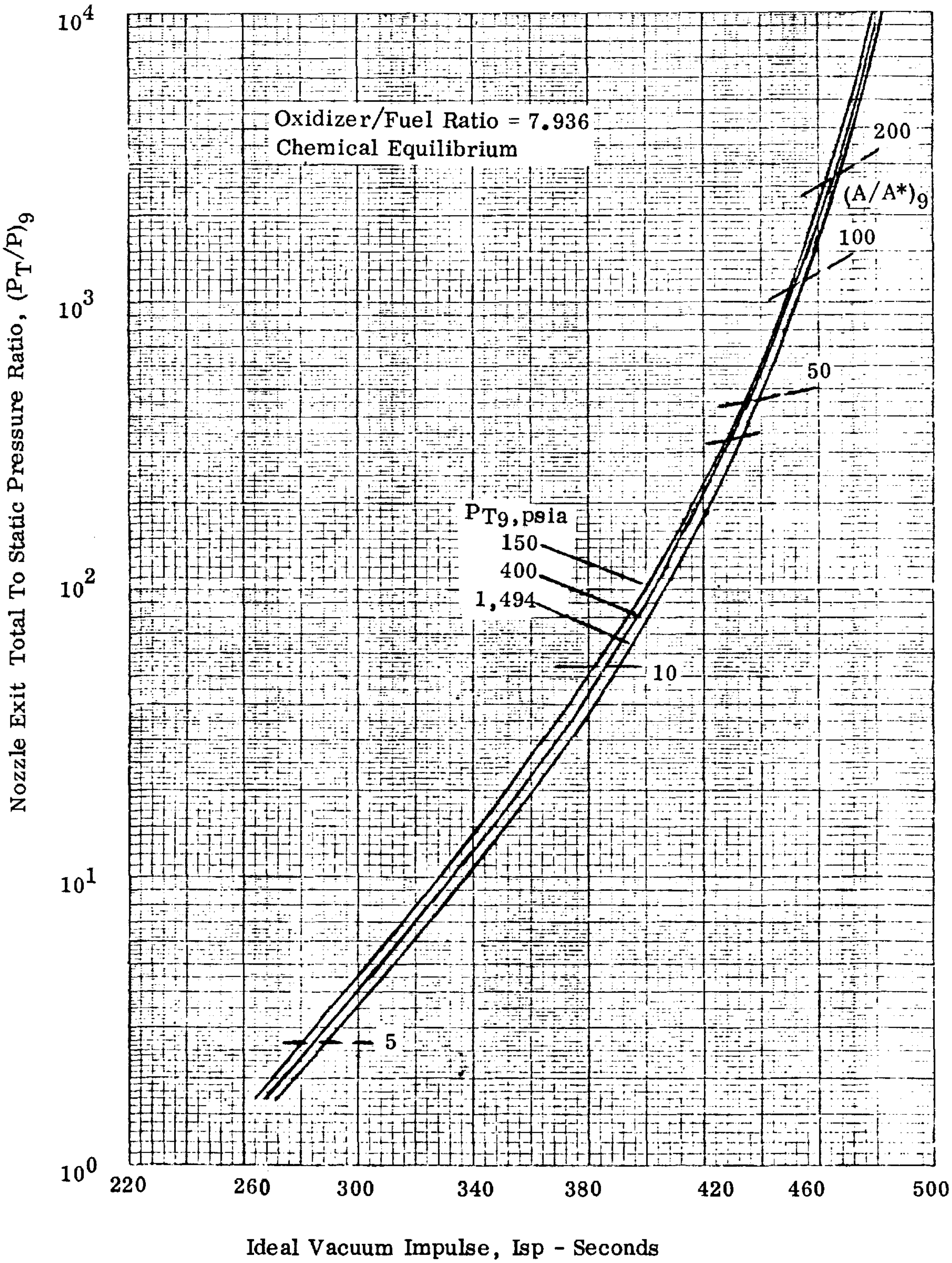


Figure 2 . Parametric Rocket Operating Mode Performance

Subsonic Flow Solution

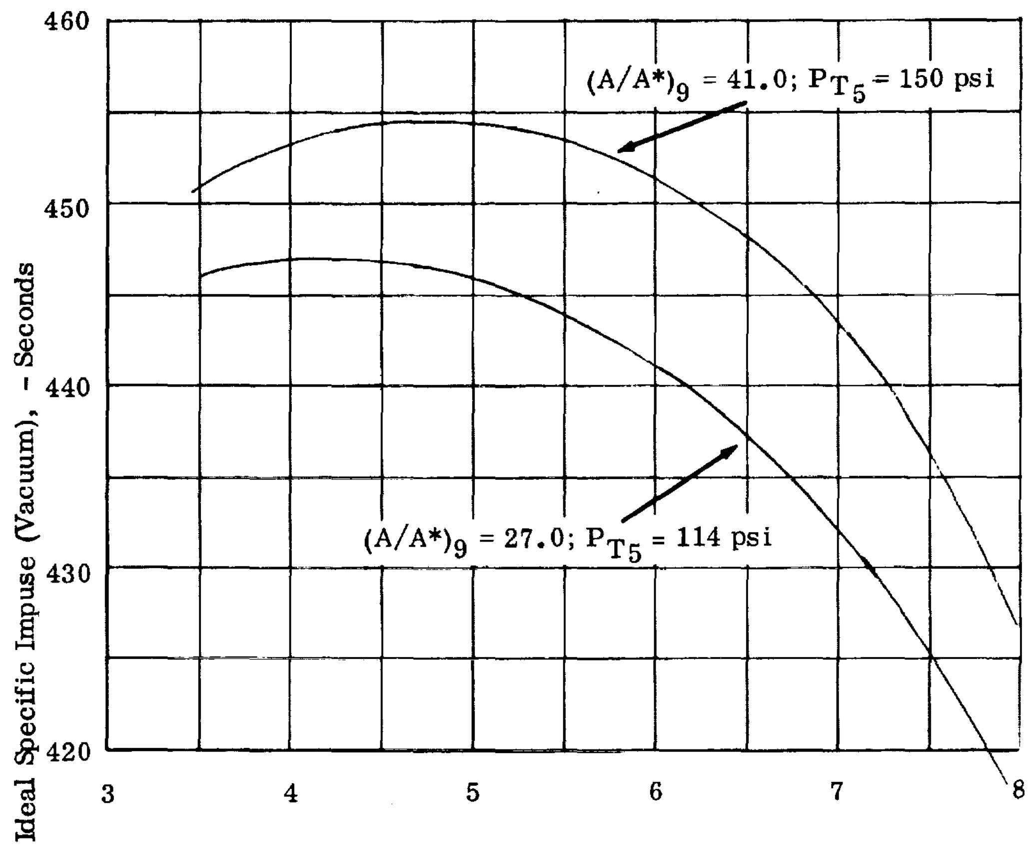


Figure 3 . Engine No. 1 - Rocket Operating Mode Performance

HYDROGEN/OXYGEN PROPELLANTS
OXIDER/FUEL RATIO = 7.936 CHEMICAL EQUILIBRIUM

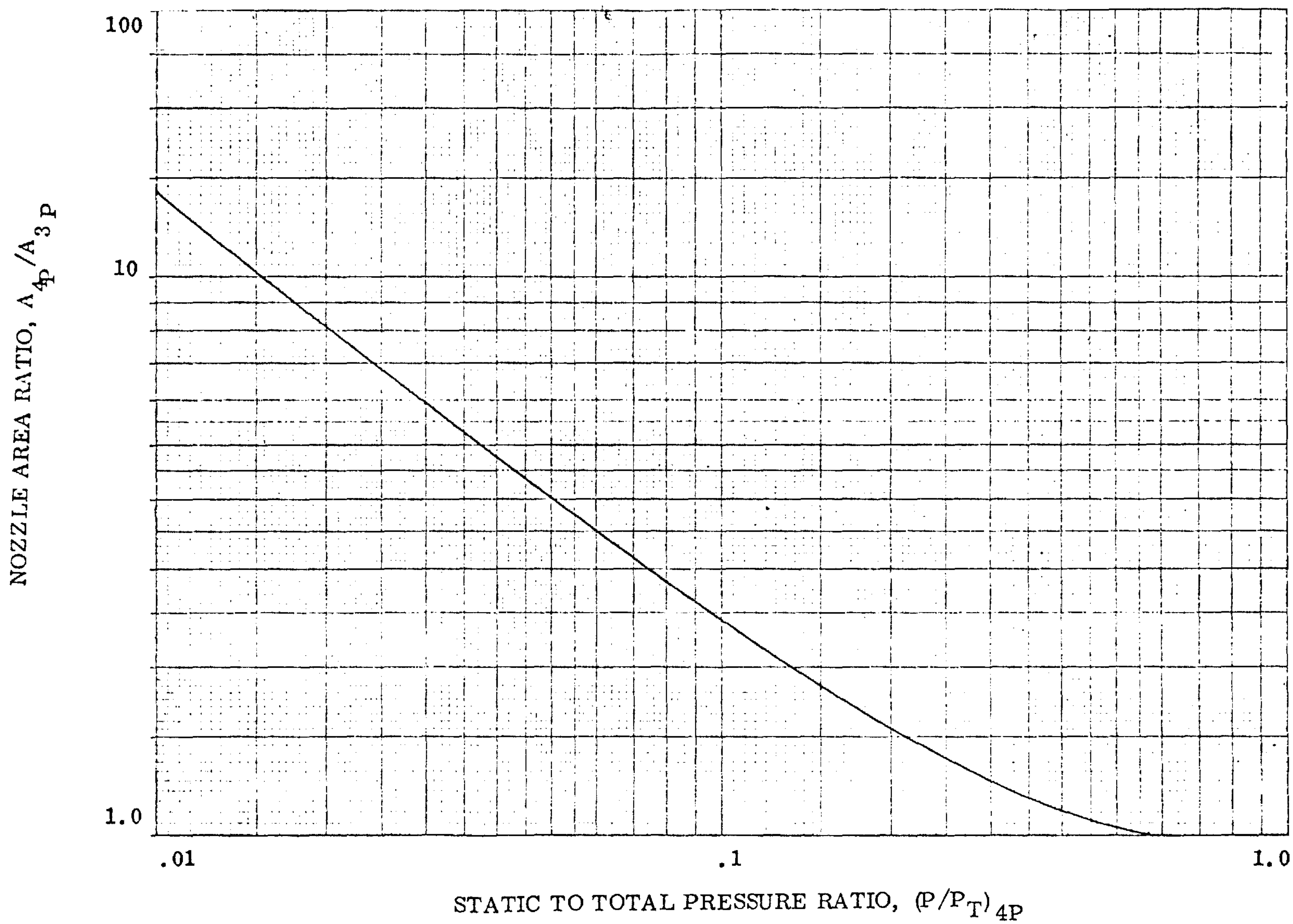


Figure .4 . Ejector Nozzle Pressure Ratio

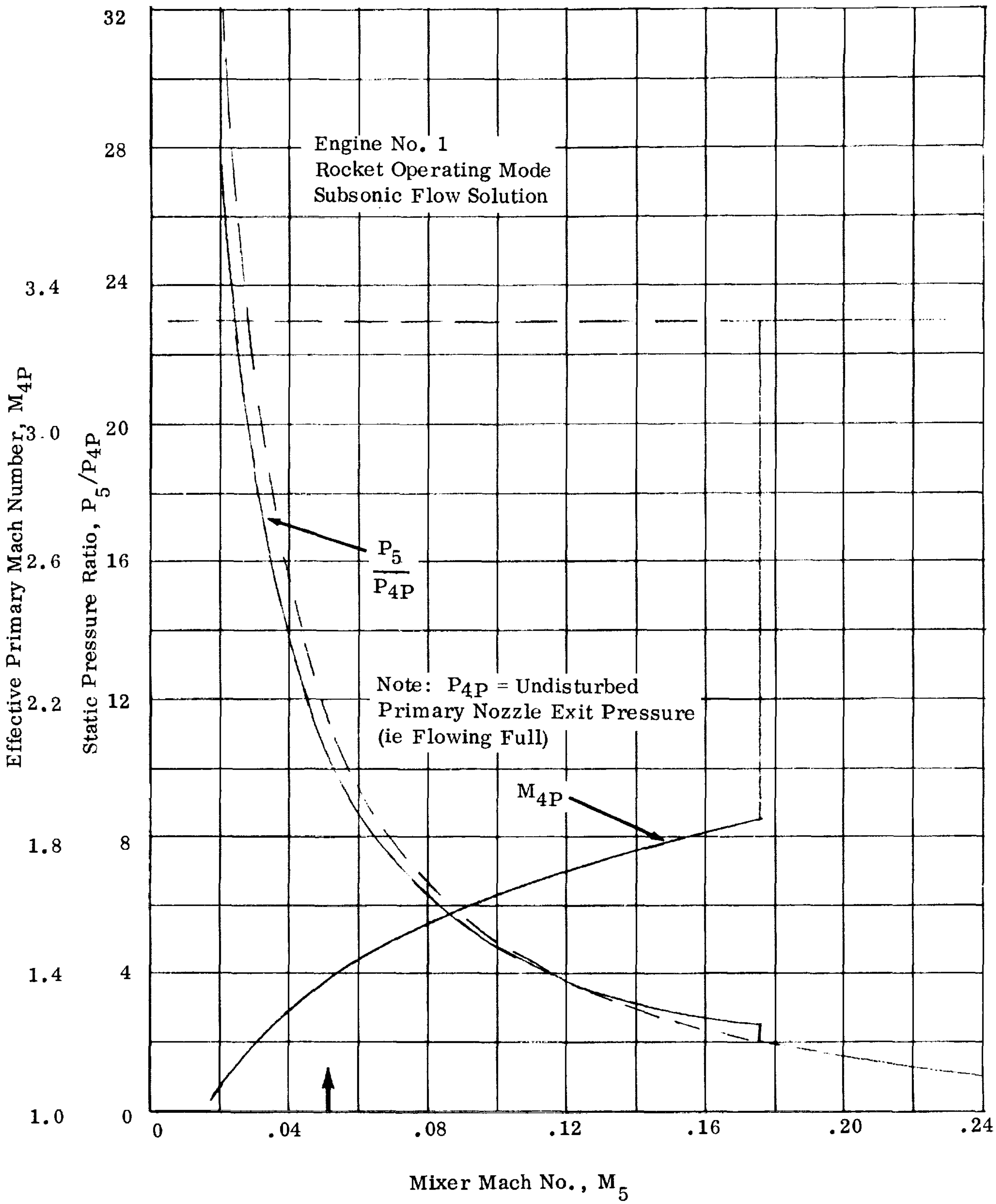


Figure 5. Effect of Mixer Back Pressure on Ejector Nozzle Performance

Supersonic Flow Solution

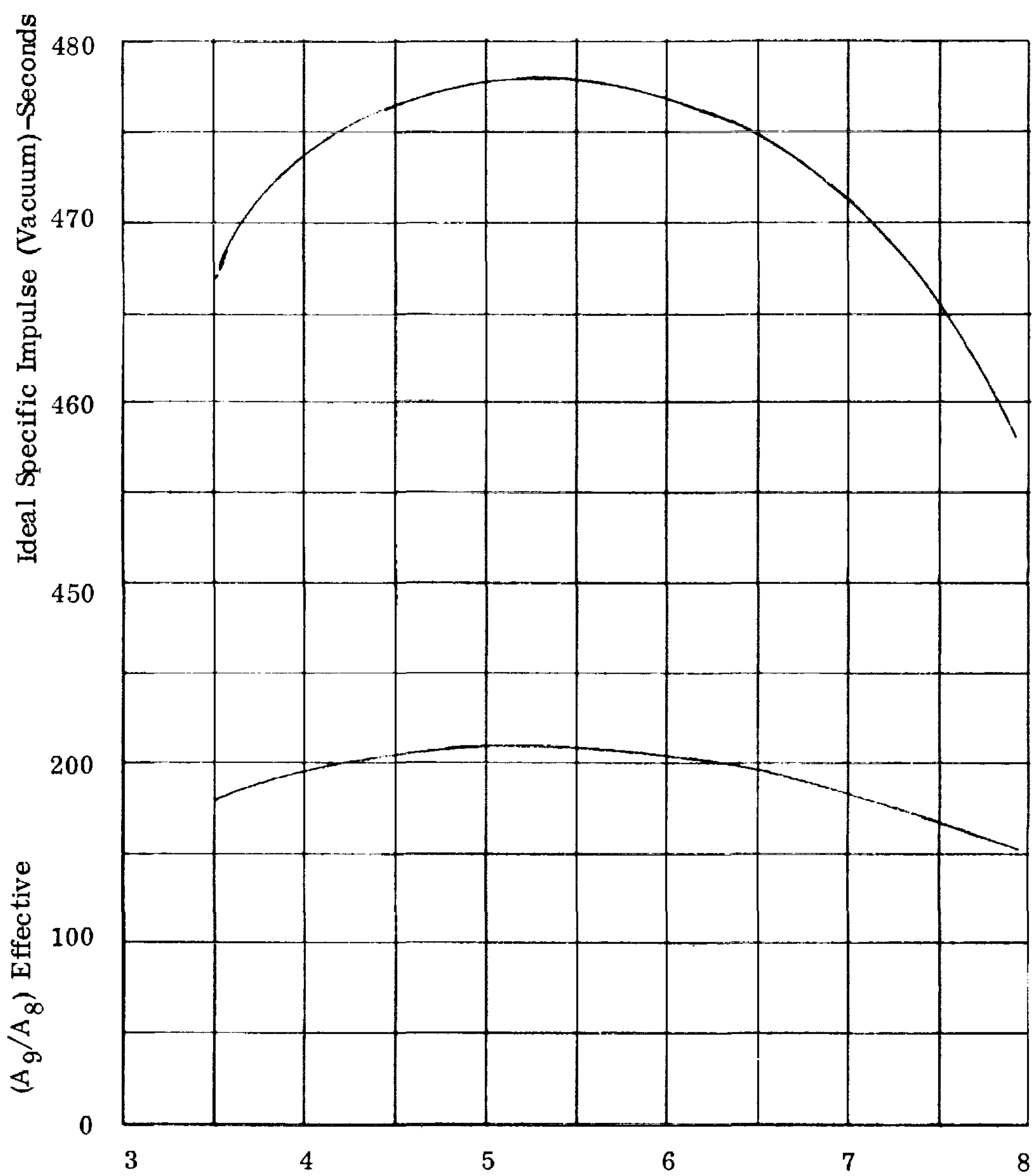


Figure 6. Engine No. 1 - Rocket Operating Mode Performance

NASA-Langley, 1975