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A PERFORMANCE STUDY FOR THE APPLICATION OF THE SATURN V TO HIGH ENERGY EARTH ESCAPE MISSIONS

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ABSTRACT

The results of a performance survey for the application of the product-improved Saturn V launch vehicles to various escape energies are presented. Two upper stage (S-II/S-IVB) propulsion systems (J-2 and J-2S) were investigated. Exchange ratios of payload with respect to vehicle parameters versus C_3 (twice the energy per unit mass) are presented for a C_3 range of 0 to 125 km^2/sec^2. The effect of booster variations and proposed vehicle improvement for different missions can be mapped into the payload through the judicious use of the exchange ratios. These data are primarily for use as a guide to payload planning for various earth escape and interplanetary missions. The results of this performance survey are presented graphically.

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RESEARCH AND DEVELOPMENT OPERATIONS

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DEFINITION OF SYMBOLS

Symbol	Definition
WO_1	lift-off weight at ground ignition
F_1	first stage total sea level thrust
ISP_1	first stage sea level specific impulse
WD_1	first stage dead weight
F_2	second stage total vacuum thrust
ISP_2	second stage vacuum specific impulse
WD_2	second stage dead weight
F_3	third stage vacuum thrust
ISP_3	third stage vacuum specific impulse
WD_3	third stage dead weight
IU	instrument unit
T/WO_1	thrust-to-weight ratio
J-2S	J-2 simplified propulsion system
A_z	launch azimuth
C_3	twice the energy per unit mass or hyperbolic excess speed squared
PLD	net payload
C_{AT}	axial force coefficient

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A PERFORMANCE STUDY FOR THE APPLICATION OF THE SATURN V TO HIGH ENERGY EARTH ESCAPE MISSIONS

SUMMARY

The results of a performance survey for application of a product-improved Saturn V launch vehicle to various escape energies are presented in this report. Two upper stage (S-II/S-IVB) propulsion systems (J-2 and J-2S) were investigated. The nominal flight profile was S-IC/S-II/S-IVB to a one-hundred-nautical-mile parking orbit with reignition of the S-IVB stage to inject the payload to the desired energy value. Because the first-burn propellant of the S-IVB stage optimized to zero for high energy values, special flight profiles were investigated to extend the payload capability to higher energy values. These are S-II injection into a higher circular orbit than nominal, and a single burn S-IVB stage out of orbit.

Exchange ratios of payload with respect to vehicle parameters versus C_3 (twice the energy per unit mass) are presented for a C_3 range of 0 to 125 km^2/sec^2. The effect of booster variations and proposed vehicle improvements for different missions can be mapped into the payload through judicious use of the exchange ratios. These data are primarily for use as a guide to payload planning for various earth escape and interplanetary missions. The results of this performance survey are presented graphically.

I. INTRODUCTION

During the past decade, the major goal of the national space program has been the exploration of near-earth space with a prime goal of landing two men on the moon and returning them to earth by 1970. As the Apollo program comes nearer this goal, the interest of NASA planners to send large payloads beyond the earth/moon system has greatly increased. A question often asked is "What is the payload capability of the Saturn V launch vehicle to high energy missions?" This investigation indicates that the Saturn V launch vehicle, developed for the Apollo program, has the capability with minimum modification (slosh baffles in the S-IVB stage) of placing space probes to the outer reaches of the solar system.

A product-improved Saturn V launch vehicle, designated as SA-516, has been defined for two configurations. One configuration, the Apollo geometry configuration, is a man-rated vehicle; i.e., the Launch Escape System is available for booster aborts. The second configuration, the MSFC Nose Cone configuration, is an unmanned flight version. The nose cone is jettisoned in orbit for the performance data presented, and no cylindrical payload shroud section is defined. When a mission is defined, the effect of the payload shroud upon the injection payload can be calculated from the exchange ratios contained in Appendix B, as can other vehicle perturbations.

Two upper stage (S-II/S-IVB) propulsion systems were investigated for each configuration, the first being the standard J-2 engine propulsion as presently defined for the Apollo program. The second system investigated is the J-2 simplified engine (J-2S) which displays a gain in specific impulse while reducing the respective S-II and S-IVB stage dead weights. The performance characteristics of the respective propulsion systems are given in Appendix C.

These performance data are applicable to single launch Saturn V's.

II. ASSUMPTIONS USED FOR PERFORMANCE CALCULATIONS

The following items list the assumptions used for the nominal performance calculations:

(1) Configuration aerodynamic data are obtained from references 1 and 2 and are contained in Appendix C.

(2) Vehicle weight data are taken from reference 3 and are contained in tables 2 through 5 of Appendix C.

(3) Launch from Kennedy Space Center (KSC), Pad 34, geodetic latitude = 28°31'17.5064", and geodetic longitude = -80°33'40.8869". Firing azimuth = 70° measured from north to south over east.

(4) All stages are filled to propellant capacity and T/WO_1 and trajectory shaping optimized to yield maximum payload as a function of mission energy.

(5) The vehicle lifted off with a vertical rise of twelve seconds. A constant pitch rate is initiated and executed until thirty-five seconds of flight when total angle of attack is set to zero for the remainder of the first stage flight.

(6) The first stage exercised an engine shutdown sequence of one - four with a four-second interval.

(7) Three and eight-tenths seconds coast is allowed between the first stage final cut-off and second stage ignition. The atmosphere is dropped from the calculations at second stage ignition.

(8) A programmed mixture ratio is used during the second stage burn to increase performance and use more propellant tank volume.

(9) The large S-IC/S-II interstage is dropped thirty seconds after S-IC final cut-off on both configurations.

(10) The Launch Escape System is jettisoned thirty-five seconds after S-IC final cut-off for the Apollo configuration only.

(11) The nose cone for the MSFC Nose Cone configuration was jettisoned in parking orbit.

(12) Parking orbit altitude equals one hundred nautical miles except for the special trajectory profiles where the S-II stage places a fully loaded S-IVB/IU/Payload into an optimum altitude parking orbit as a function of C_3. The optimum altitude is defined as that which will yield the maximum payload while depleting the S-IVB stage propellants minus reserves to reach the specified energy level.

(13) Boil-off and attitude control losses, calculated for four and one-half hour parking orbit coast, remained constant for all energy levels as listed in tables 2 through 5, Appendix C.

(14) Upper stage thrust angles were optimized via the steepest ascent technique over a rotating 1960 Fischer Ellipsoid earth model with the fourth-order gravity function.

(15) Flight performance reserves were calculated equal to .75 percent of the total vehicle characteristic velocity for $C_3 = 0$ (local escape) to 1 percent of the total vehicle characteristic velocity at $C_3 = 125$ km^2/sec^2. This variation was calculated by using a 3σ launch vehicle error analysis at various values of C_3.

(16) Flight geometry reserves were calculated equal to sixty meters per second.

(17) Net payload is defined as the weight forward of the instrument unit (IU) at final injection.

III. DESCRIPTION - EXPLANATION OF RESULTS

The results of this study are presented in graphical form. The figures are self-explanatory but care is to be used in extracting data, as explained in Section IV.

Figures A-1 and A-2 are drawings of the Apollo configuration and the MSFC Nose Cone configuration, respectively. It will be noticed that total height of the MSFC Nose Cone configuration has not been specified because this will depend on the payload packaging procedure used. For the purpose of this study, no cylindrical payload section was assumed; i.e., the nose cone is attached directly to the instrument unit and is jettisoned in parking orbit. The effect on payload of the requirement of a cylindrical payload fairing will be discussed in Section IV. Figures A-3 and A-4 display the net payload at injection for the Apollo and MSFC Nose Cone configurations, respectively, as a function of C_3 for the two types of upper stage propulsion systems investigated. The solid payload curves of the graphs denote mission profiles where the S-IVB stage is suborbitally burned for injection into orbit and reignited to reach the desired C_3 value. The dashed portion of each curve denotes profiles where the S-II stage injects into an optimum altitude circular parking orbit and the S-IVB performs a single burn to obtain the final C_3 value. Figures A-5 and A-6 are plots of optimum T/WO_1 at liftoff versus C_3 for both configurations with J-2 and J-2S propulsion, respectively. Figure A-7 displays the ratio of S-IVB first burn into parking orbit propellants to the total stage propellant capacity.

Figure A-8 is a plot of optimum orbit altitude versus C_3 for values of C_3 greater than 105 km^2/sec^2 for J-2S propulsion and 125 km^2/sec^2 for standard J-2 propulsion systems. The weight savings resulting from removing the requirement for restart capability from the S-IVB stage have not been added to the payloads shown, but any S-IVB stage weight reduction is directly additive to payload.

IV. EXCHANGE RATIOS

Exchange ratios of payload with respect to various vehicle parameters are presented in figures B-1 to B-12. These exchange ratios are applicable only to flight profiles where the S-IVB stage is suborbitally burned into parking orbit and reignited to reach the desired C_3 value.

Figure A-7 shows that the upper C_3 limit of the exchange ratios for configurations with J-2S propulsion is C_3 = 105 km^2/sec^2 and C_3 = 125 km^2/sec^2 for the configurations using the standard J-2 propulsion system. The payload effects of the various exchange ratios are additive within the specified limits and are applicable to both configurations with either J-2 or J-2S propulsion systems. The exception, naturally, is the exchange ratios and effects of the nose cone and payload shroud. Figures B-9 and B-10 are not applicable to the Apollo configuration.

The exchange ratios for jettisoning the nose cone and shroud weights (where desired) are special cases where much care is to be used. If the nose cone is jettisoned at an altitude before orbit insertion, the payload gain can be calculated by multiplying the percentage of payload gain for the given C_3 value from figure B-9 and the nominal payload from figure A-4. If an additional weight greater than the nominal 2700-pound nose cone is to be jettisoned, the effect can be calculated by multiplying the difference in this weight and the 2700-pound nose cone weight and the respective exchange ratio from figure B-10 and subtracting this from the previously calculated payload. For an increased weight jettisoned in parking orbit, the payload loss due to shroud weight greater than 2700 pounds is then to be subtracted from the nominal payload from figure A-4. These types of calculations, when applicable, must be performed before applying the other exchange ratios. When exercising an increment in launch azimuth (figure B-12), all effects of vehicle perturbations (propulsion, weights, etc.) must be exercised before calculating the payload increment due to a launch azimuth variation.

V. CONCLUSIONS

The results of this study show the Saturn V three-stage launch vehicle capable of handling sizeable payloads for high energy escape missions. The defined vehicle displays a capability of injecting approximately 25,000 pounds into a Jupiter probe transfer mission (C_3 = 80 km^2/sec^2) using the standard J-2 propulsion systems. The J-2S propulsion system for this energy results in a payload gain of approximately 3,400 pounds.

All mission profiles displayed will probably require a relocation of the slosh baffles in the S-IVB stage because of the two-burn propellant split variation as displayed in figure A-7. Mission energies which require a fully loaded S-IVB stage burned from parking orbit are feasible by injecting the fully loaded S-IVB/IU/Payload into an optimum altitude orbit with the S-II stage. This type of profile will require only a single burn S-IVB, and the weight saved by removing the restart from the S-IVB can be converted into payload.

The exchange ratios display a need for greater thrust levels and higher specific impulse values for all energies. The most worthy candidate for an increased thrust is the S-II stage, while the best candidate for a specific impulse increase is the S-IC stage.

The J-2S propulsion system (now being investigated by NASA planners) shows a good possibility of increasing the Saturn V payload capability. When applied to the S-II and S-IVB stages, payload gains range from 6 percent at $C_3 = 0$ (local escape) to 17 percent at $C_3 = 100\ km^2/sec^2$.

APPENDIX A

Performance Results

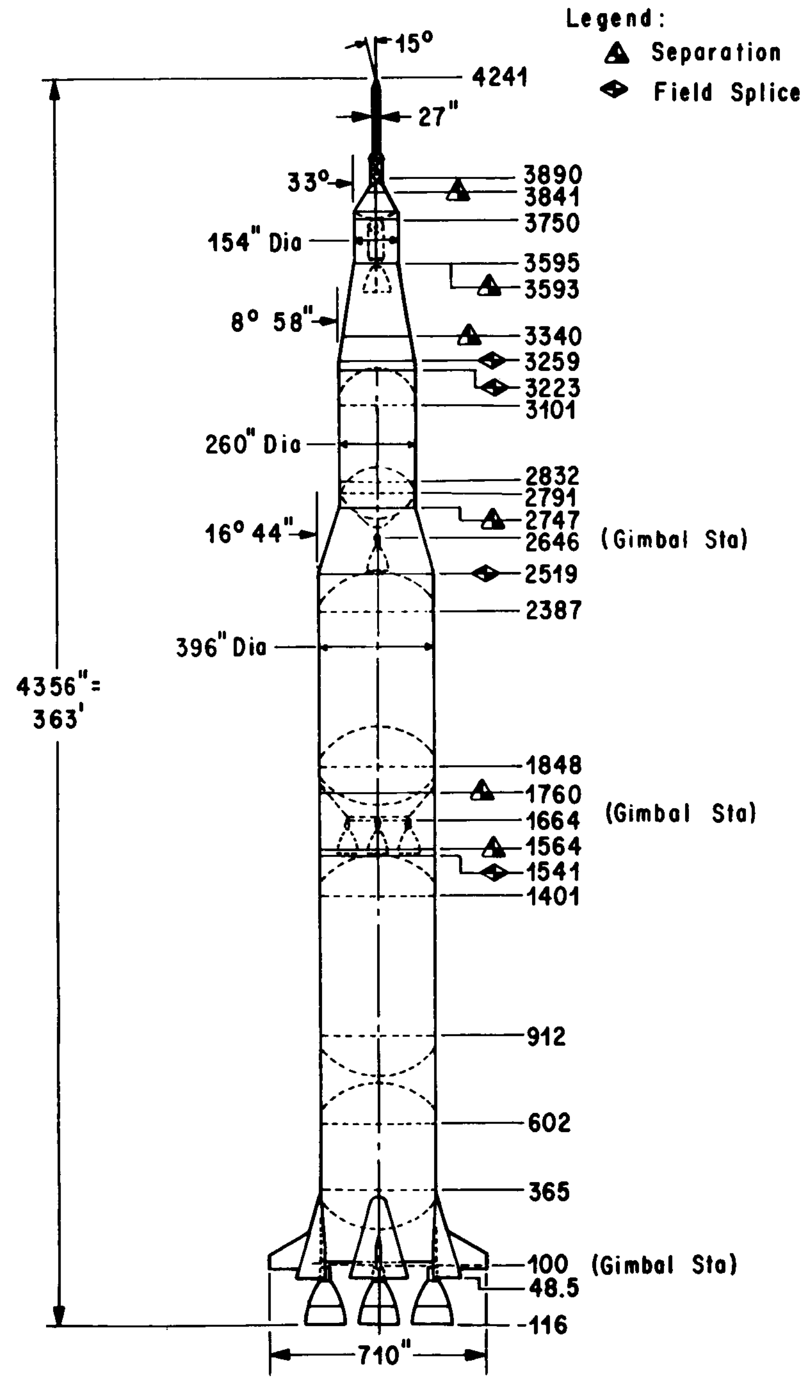

FIG. A-1. SATURN V LAUNCH VEHICLE, APOLLO CONFIGURATION

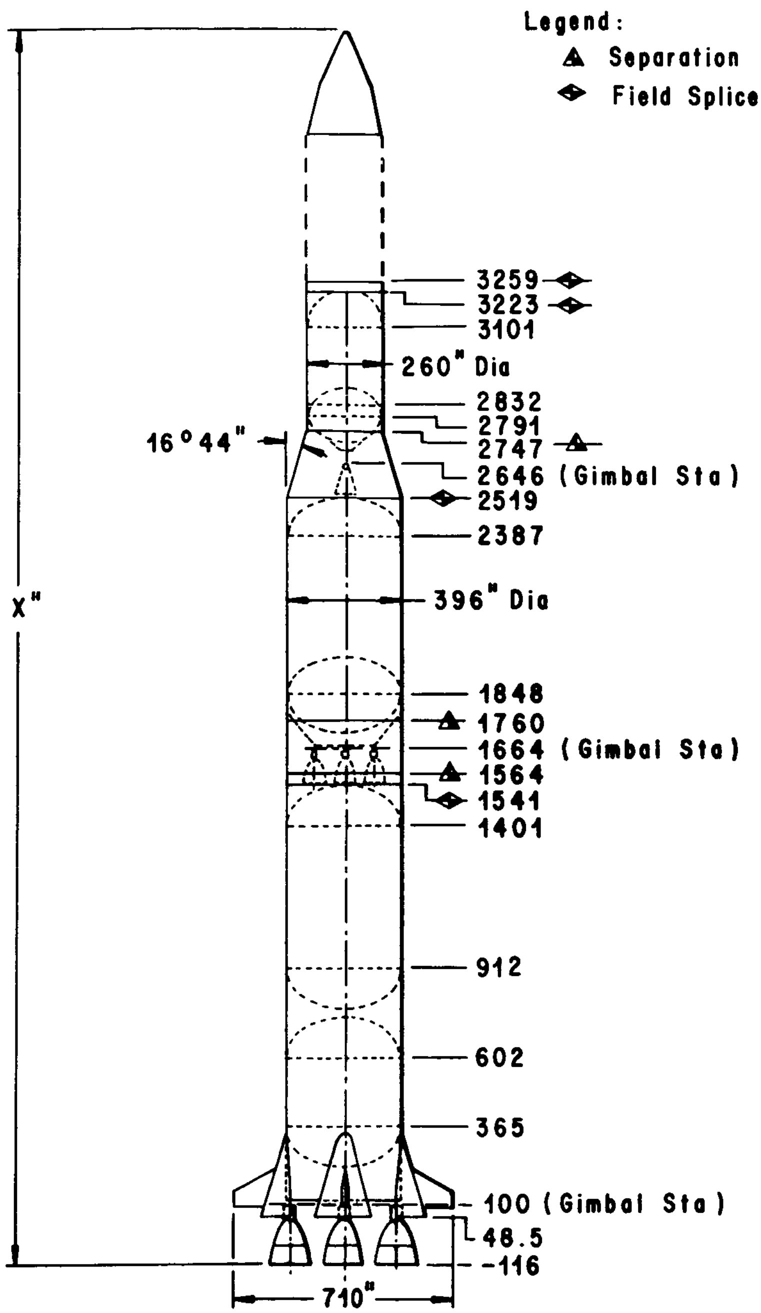

FIG. A-2. SATURN V LAUNCH VEHICLE
MSFC NOSE CONE CONFIGURATION

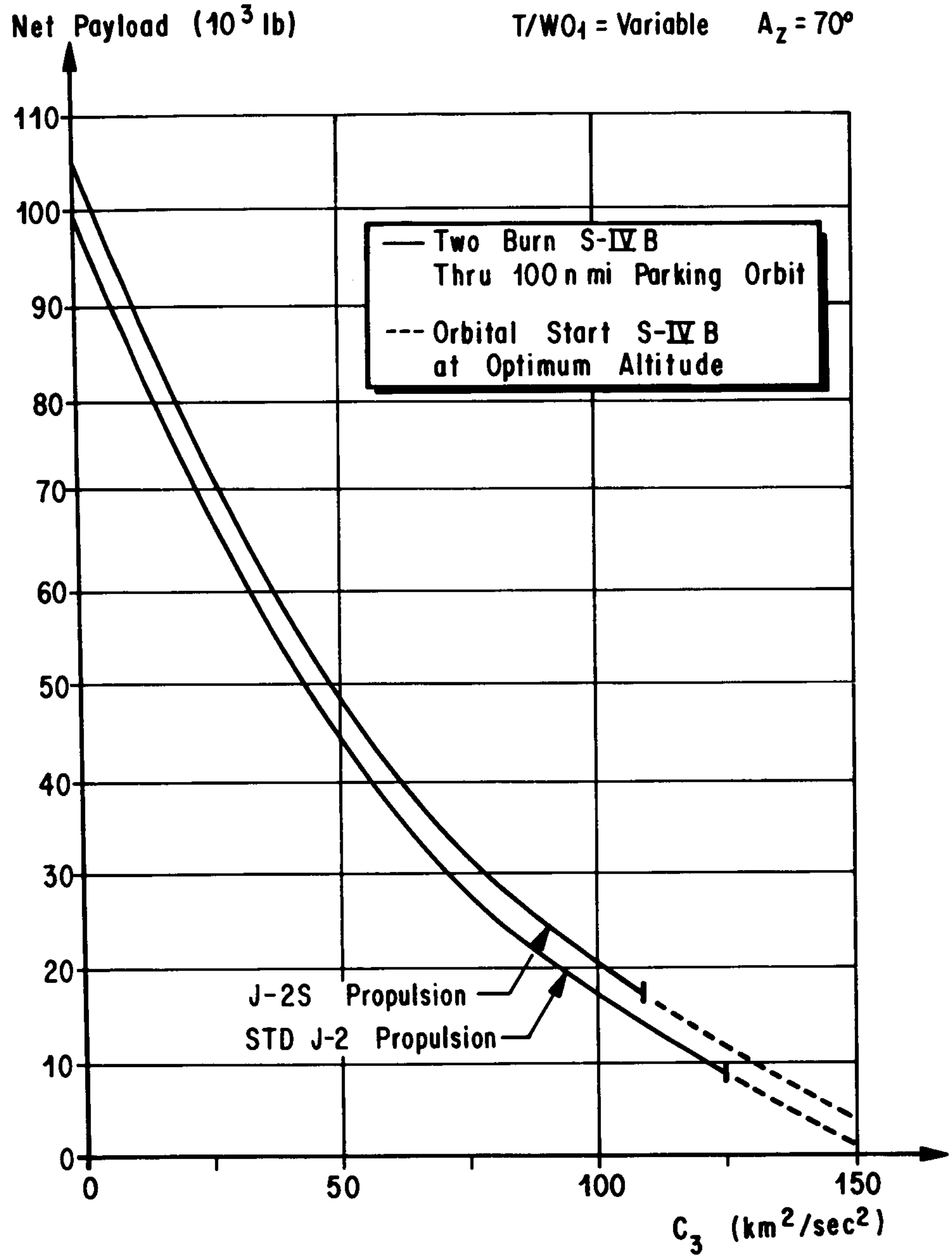

FIG. A-3. LAUNCH VEHICLE PERFORMANCE FOR SA-516 APOLLO CONFIGURATION VS C_3

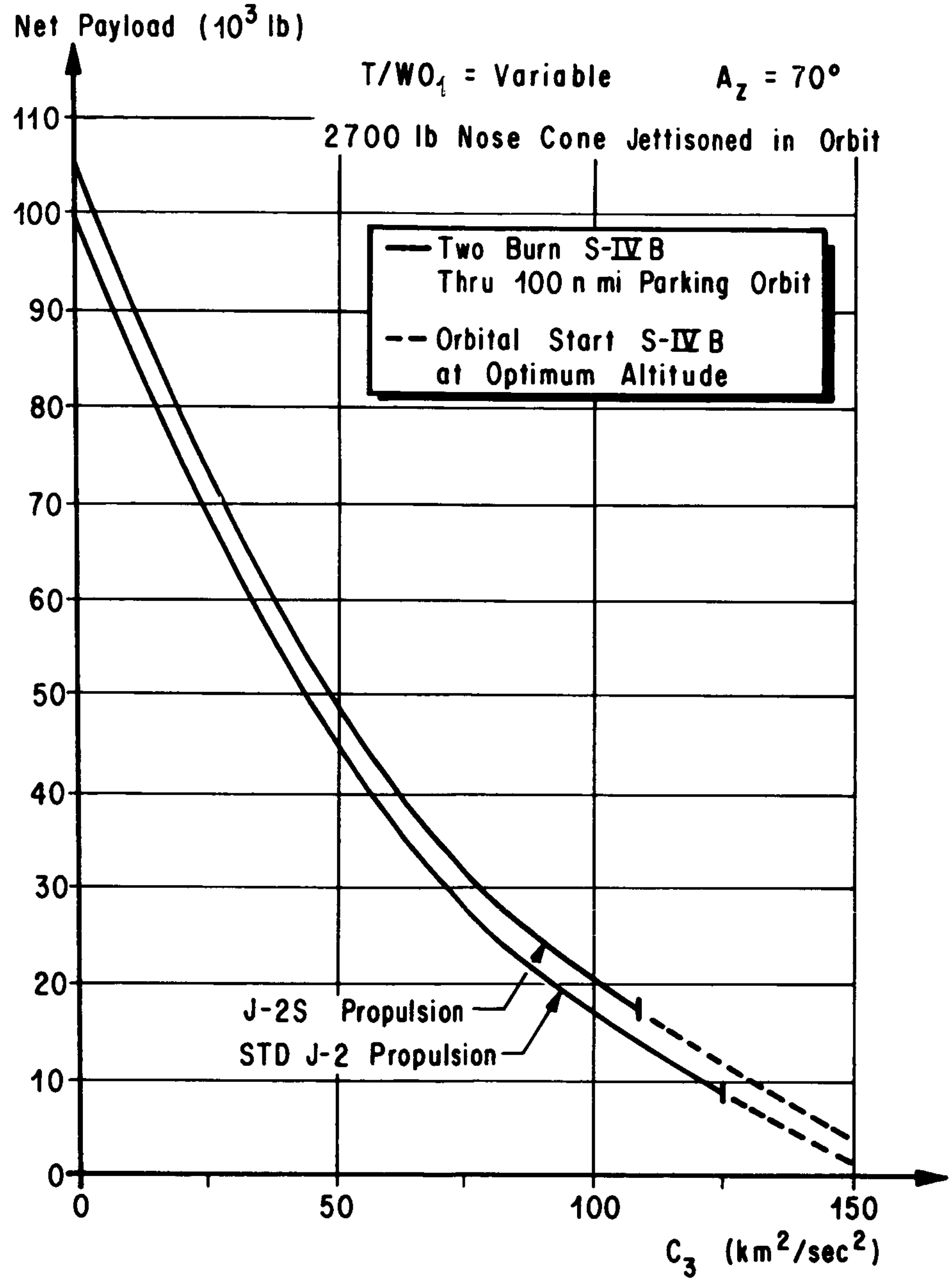

FIG. A-4. LAUNCH VEHICLE PERFORMANCE FOR SA-516 MSFC NOSE CONE CONFIGURATION VERSUS C_3

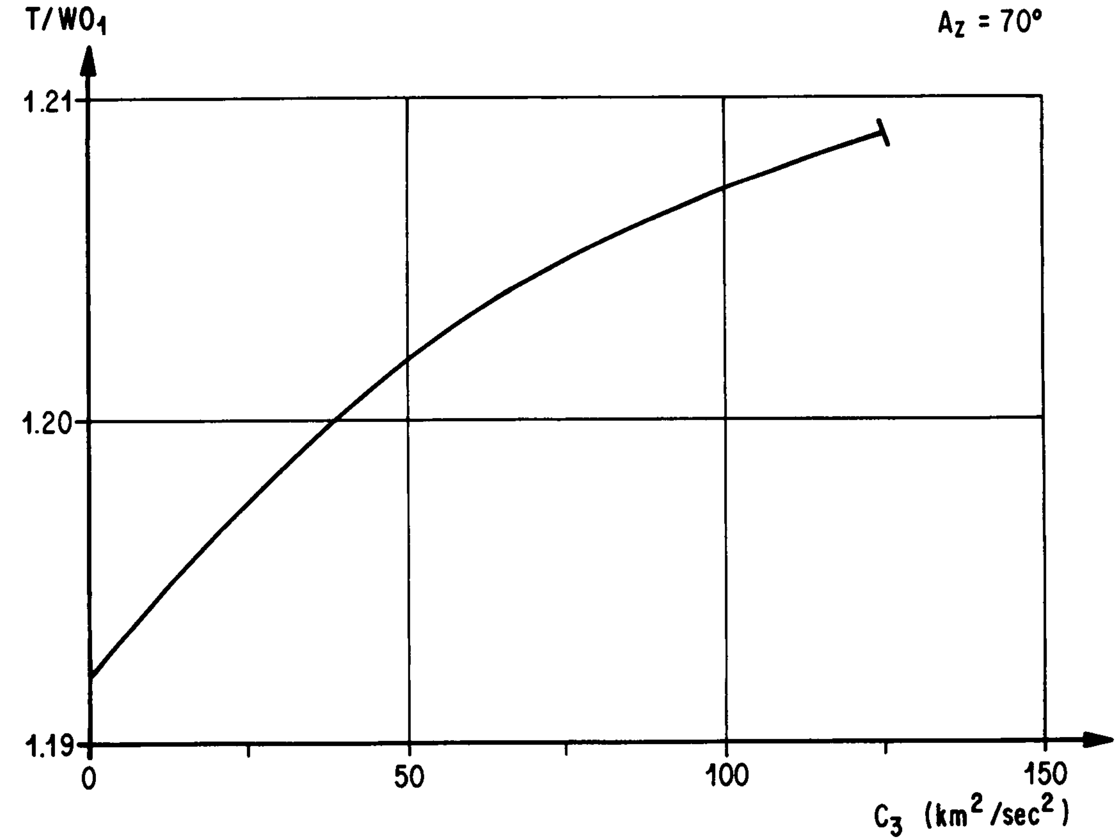

FIG. A-5. OPTIMAL THRUST TO WEIGHT AT LIFTOFF VERSUS C_3 FOR BOTH CONFIGURATIONS USING STANDARD J-2 PROPULSION IN S-II / S-IVB STAGES

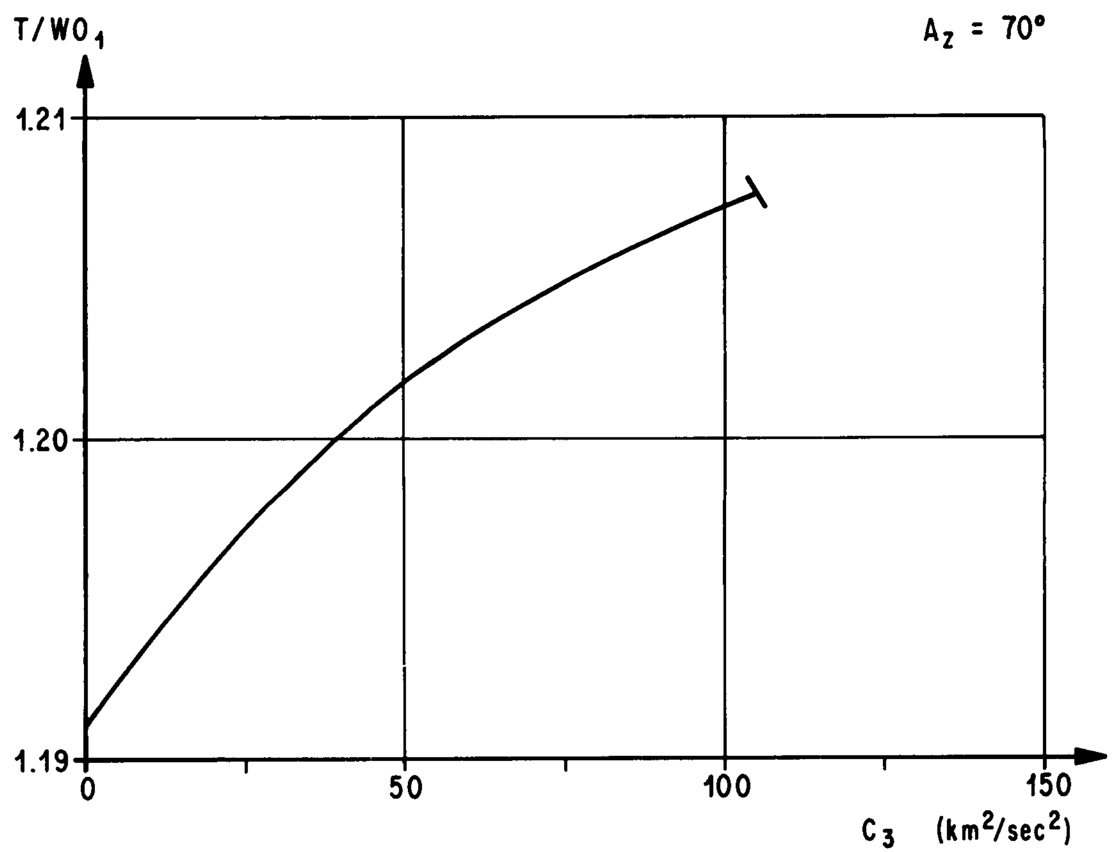

FIG. A-6. OPTIMAL THRUST TO WEIGHT AT LIFTOFF VERSUS C_3 FOR BOTH CONFIGURATIONS USING J-2S PROPULSION IN S-II/S-IVB STAGES

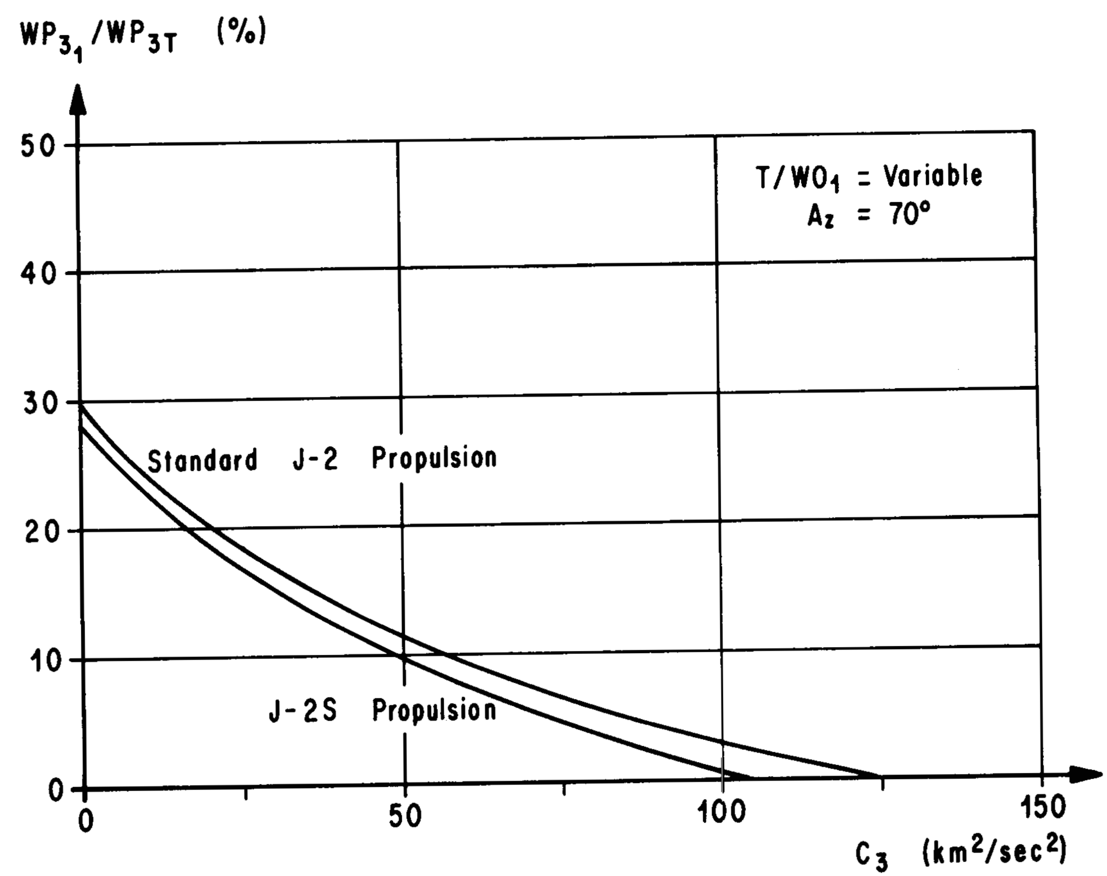

FIG. A-7. PERCENTAGE RATIO OF S-IVB FIRST BURN PROPELLANTS INTO ORBIT TO TOTAL S-IVB CAPACITY VERSUS C_3

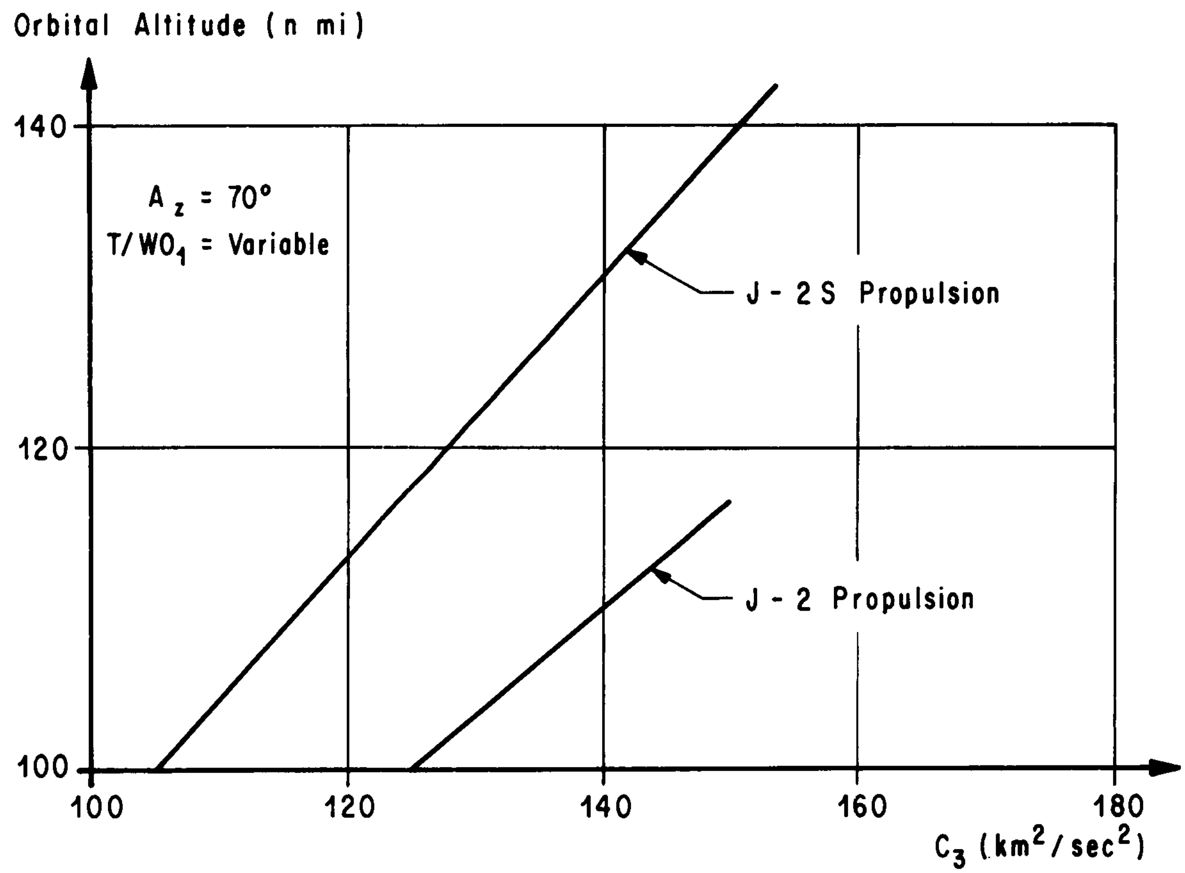

FIG. A-8. OPTIMUM PARKING ORBIT ALTITUDE FOR S-II INSERTION INTO PARKING ORBIT VERSUS C_3

APPENDIX B

Exchange Ratios

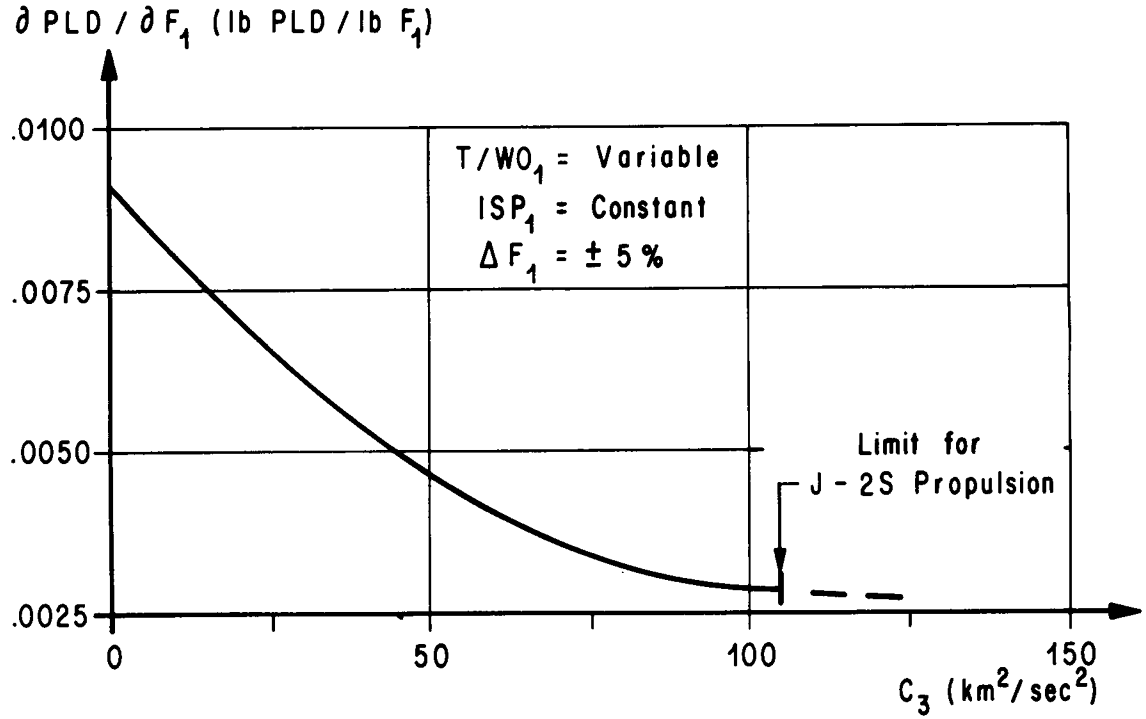

FIG. B-1. EFFECT OF TOTAL FIRST STAGE THRUST ON PAYLOAD VERSUS C_3

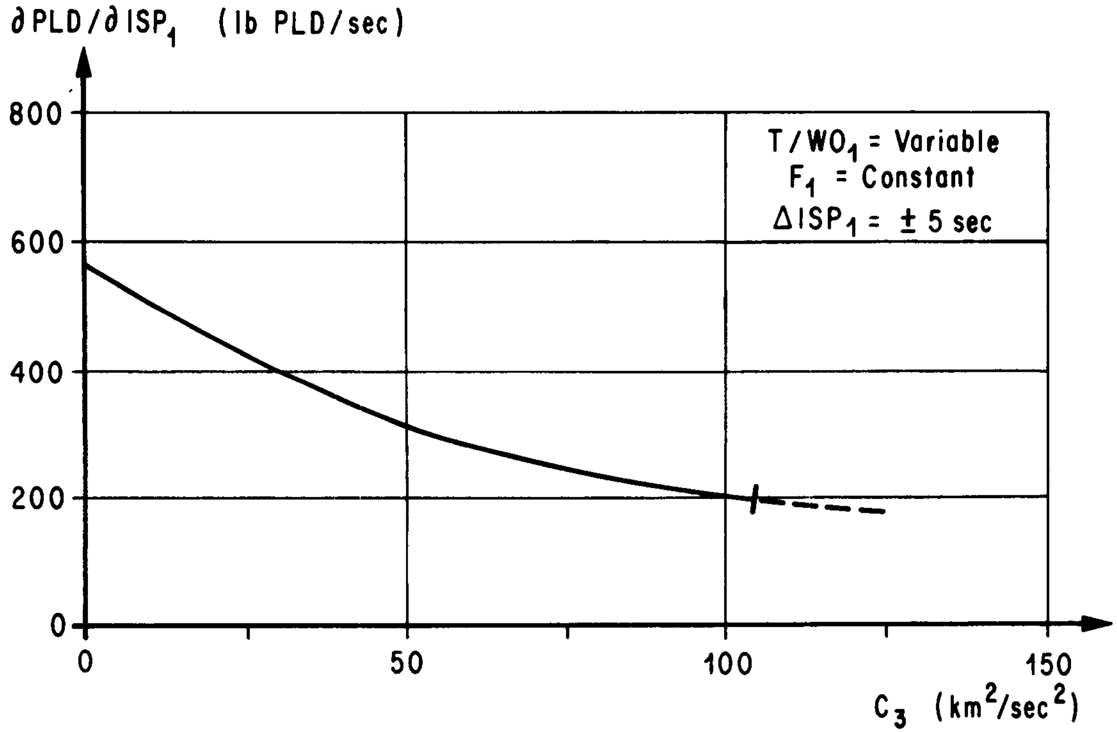

FIG. B-2. EFFECT OF FIRST STAGE SPECIFIC IMPULSE ON PAYLOAD VERSUS C_3

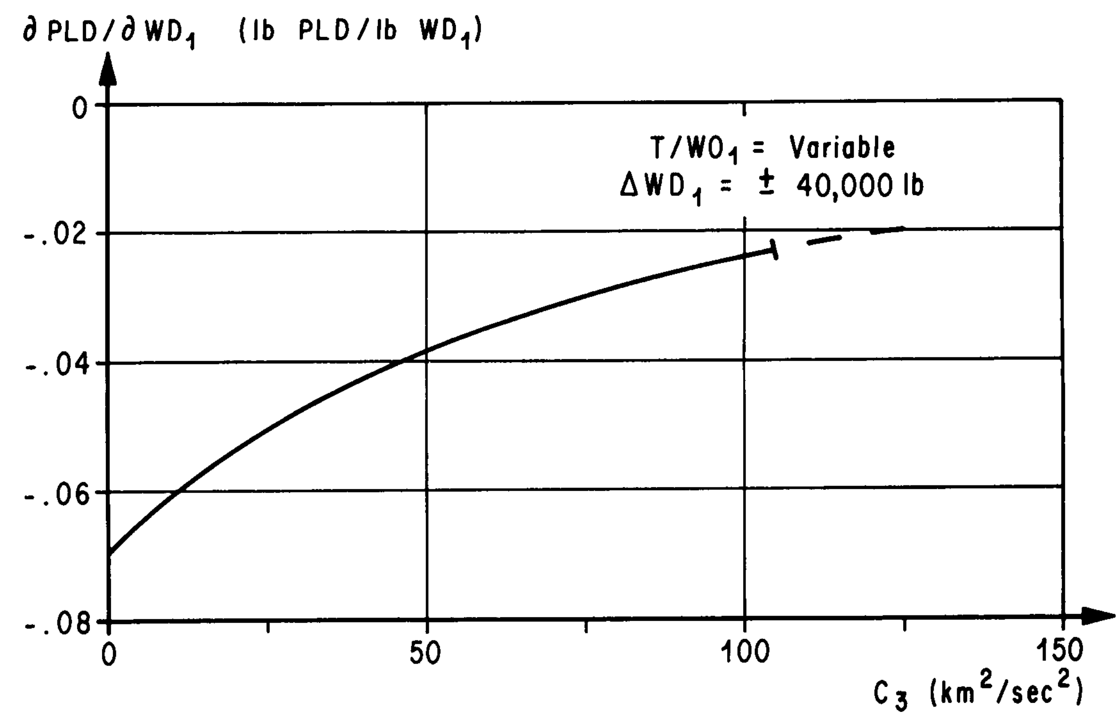

FIG. B-3. EFFECT OF FIRST STAGE DEAD WEIGHT ON PAYLOAD VS C_3

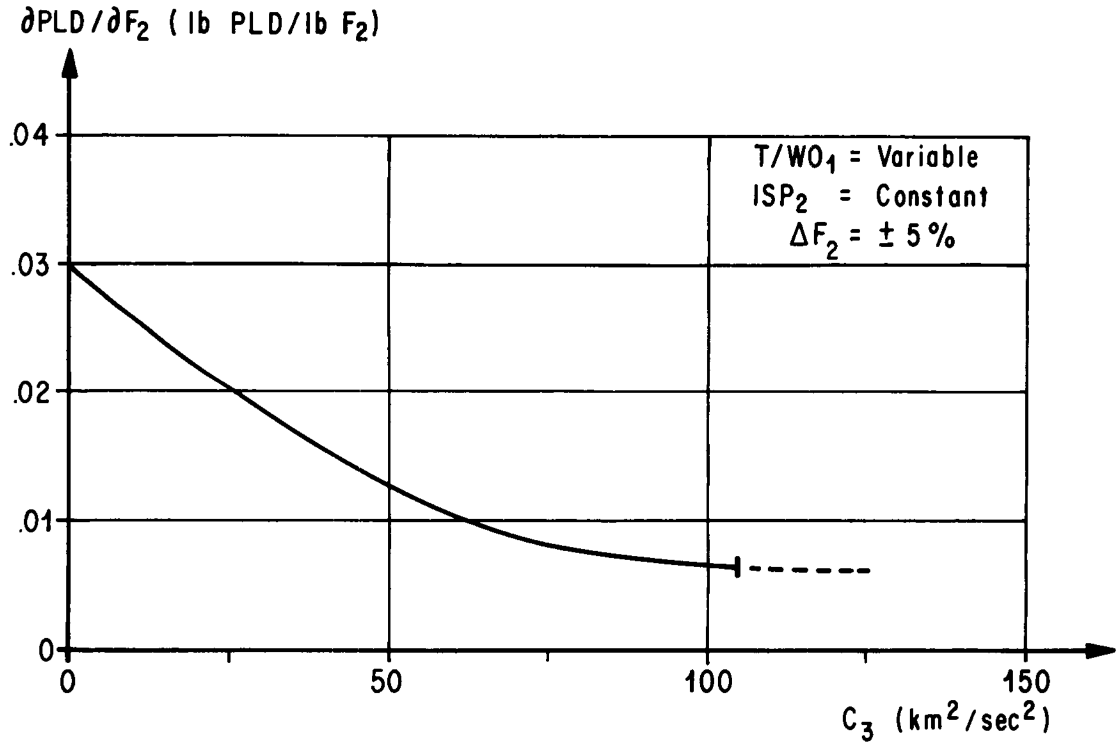

FIG. B-4. EFFECT OF TOTAL SECOND STAGE THRUST ON PAYLOAD VS C_3

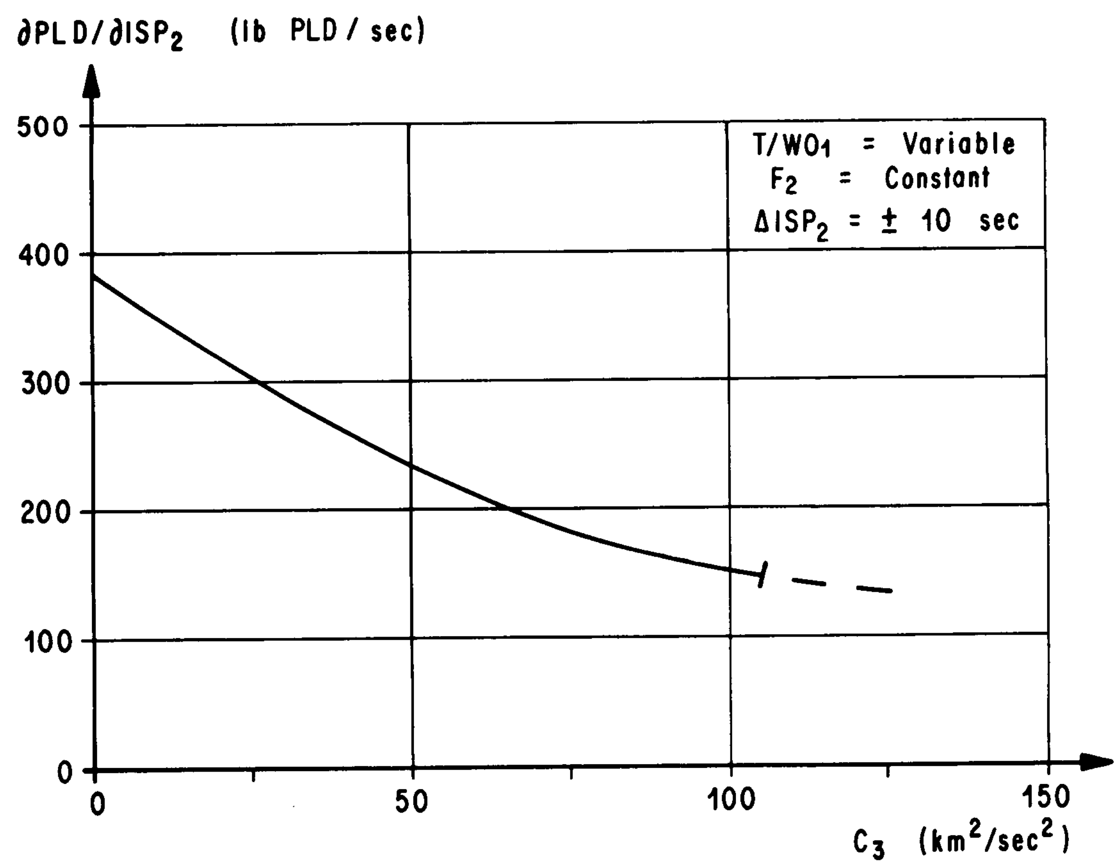

FIG. B-5. EFFECT OF SECOND STAGE SPECIFIC IMPULSE ON PAYLOAD VERSUS C_3

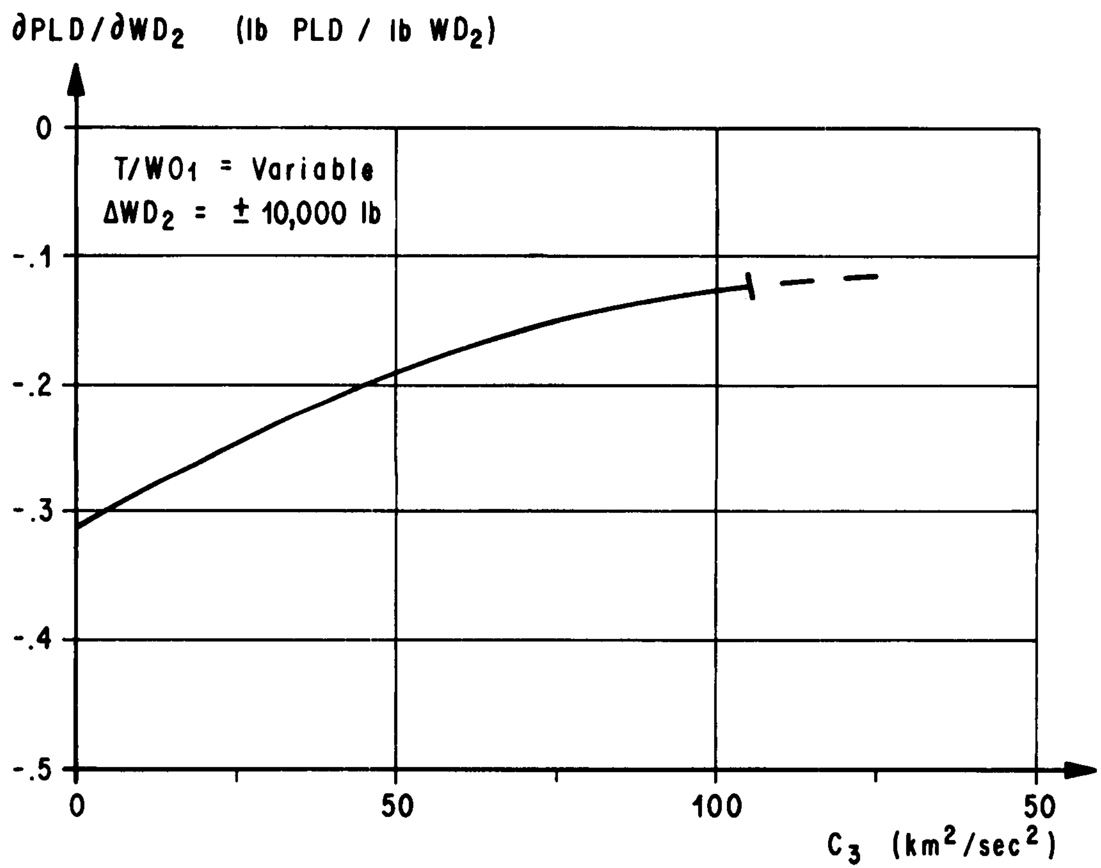

FIG. B-6. EFFECT OF SECOND STAGE DEAD WEIGHT ON PAYLOAD VERSUS C_3

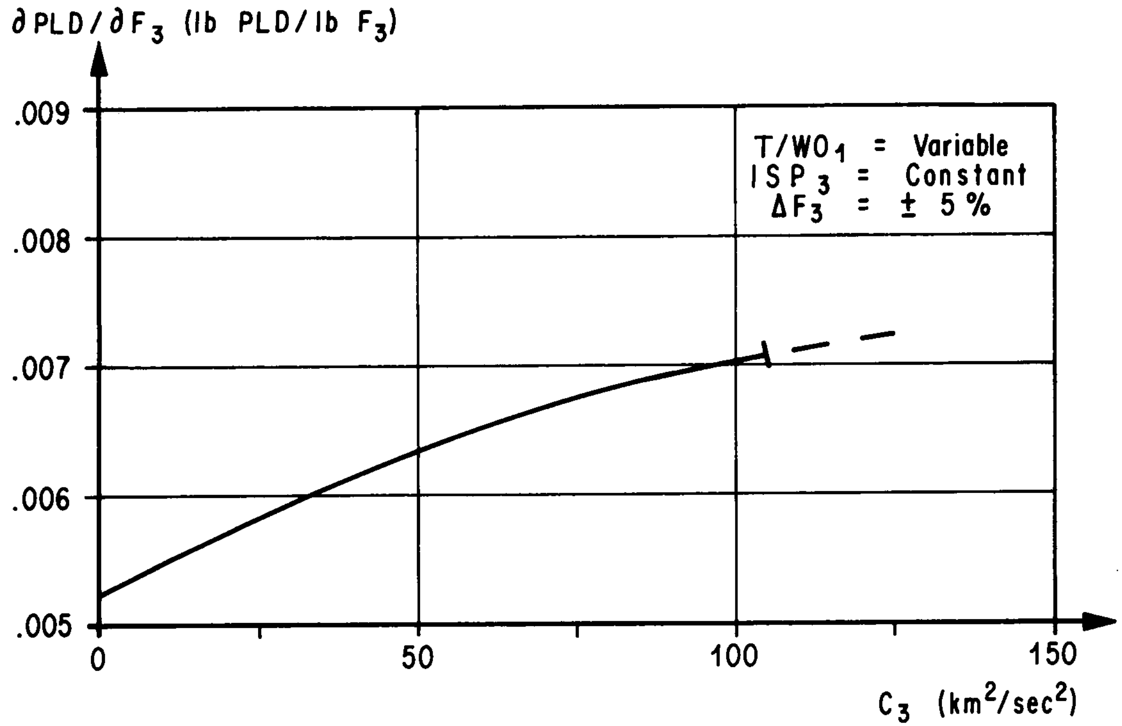

FIG. B-7. EFFECT OF THIRD STAGE THRUST ON PAYLOAD VS C_3

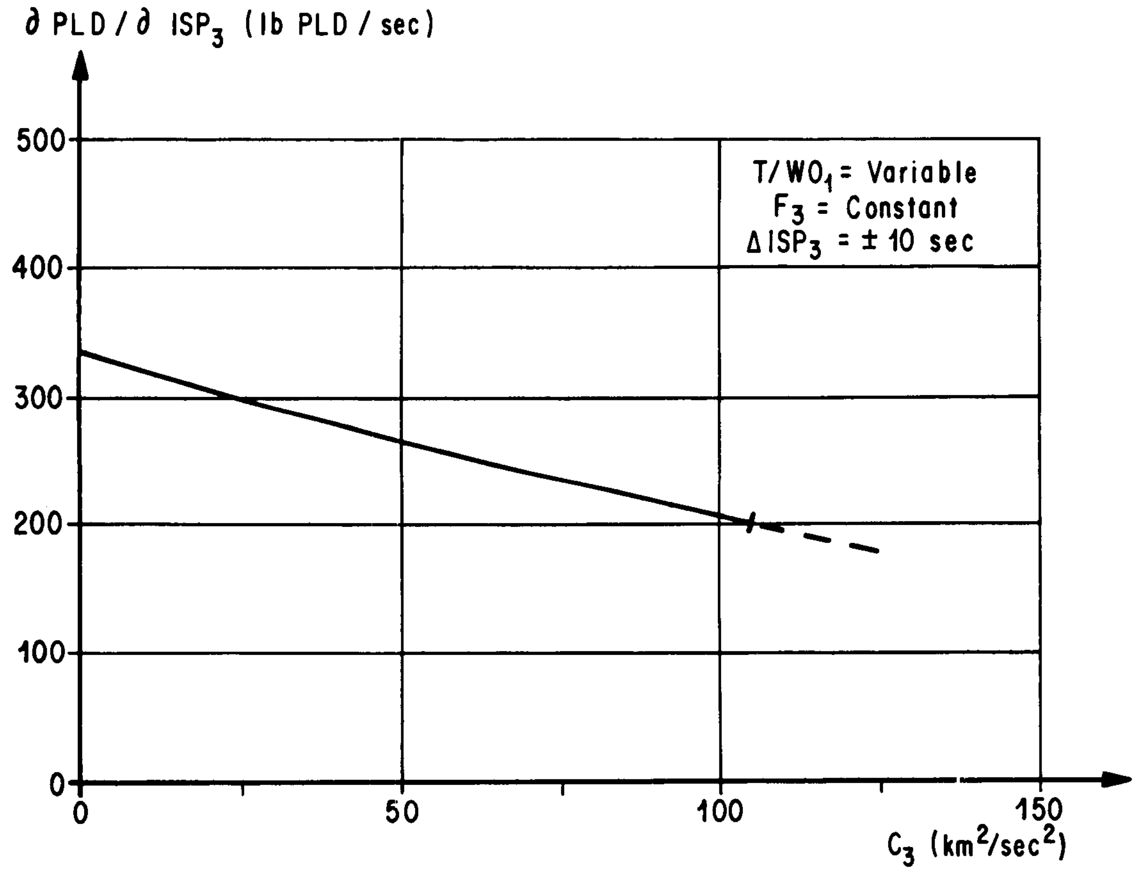

FIG. B-8. EFFECT OF THIRD STAGE SPECIFIC IMPULSE ON PAYLOAD VS C_3

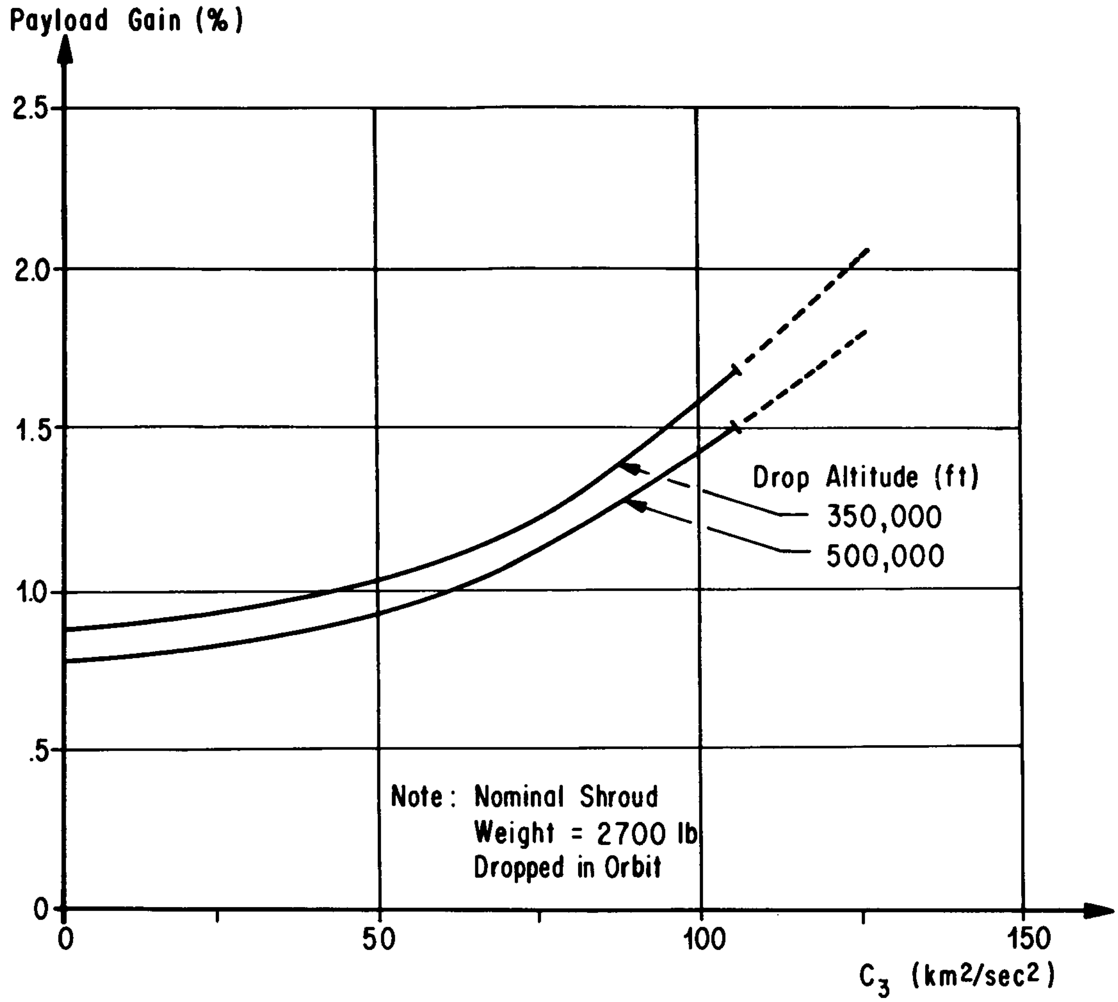

FIG. B-9. EFFECT OF NOSE CONE DROPPED AT EARLY ALTITUDE VERSUS C_3 FOR MSFC NOSE CONE CONFIGURATION

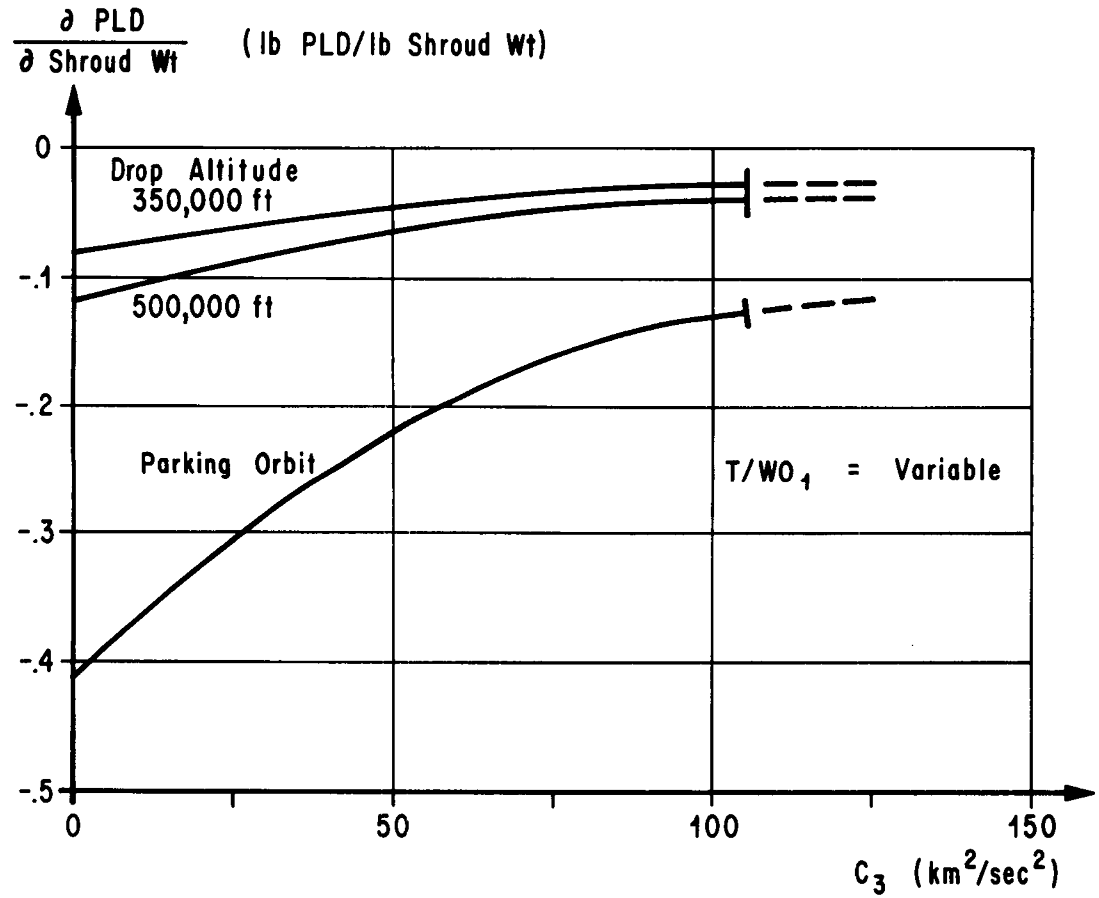

FIG. B-10. EFFECT OF SHROUD/NOSE CONE DROPPED AT VARIOUS ALTITUDES VERSUS C_3 FOR MSFC NOSE CONE CONFIGURATION

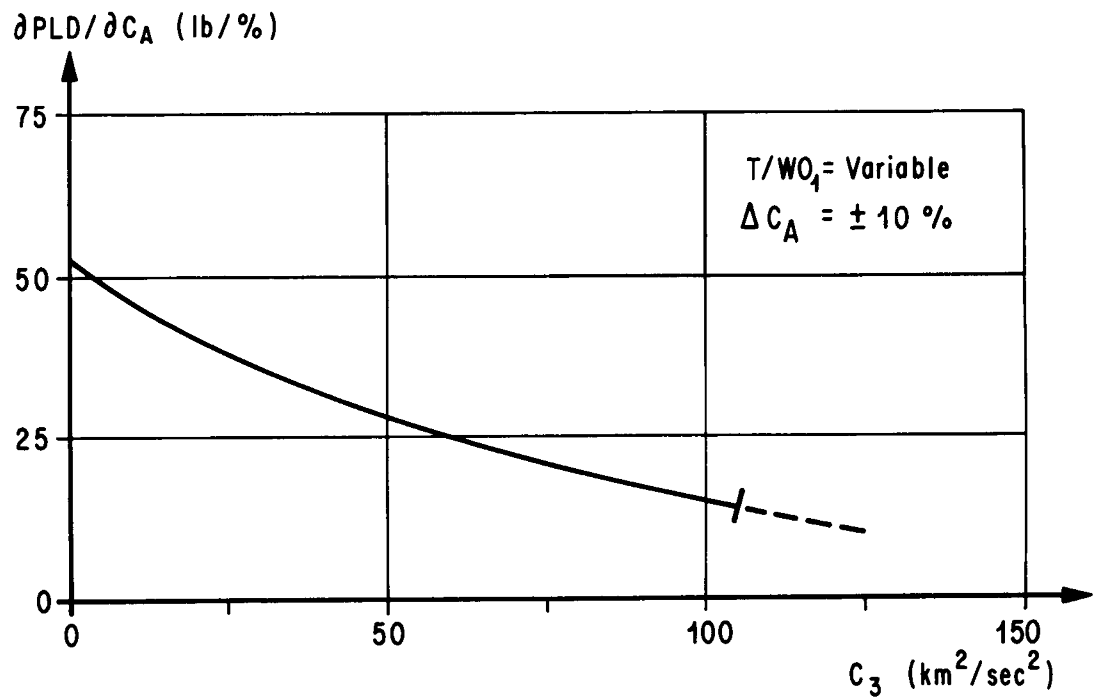

FIG. B-11. EFFECT OF AXIAL FORCE COEFFICIENT ON PAYLOAD VS C_3

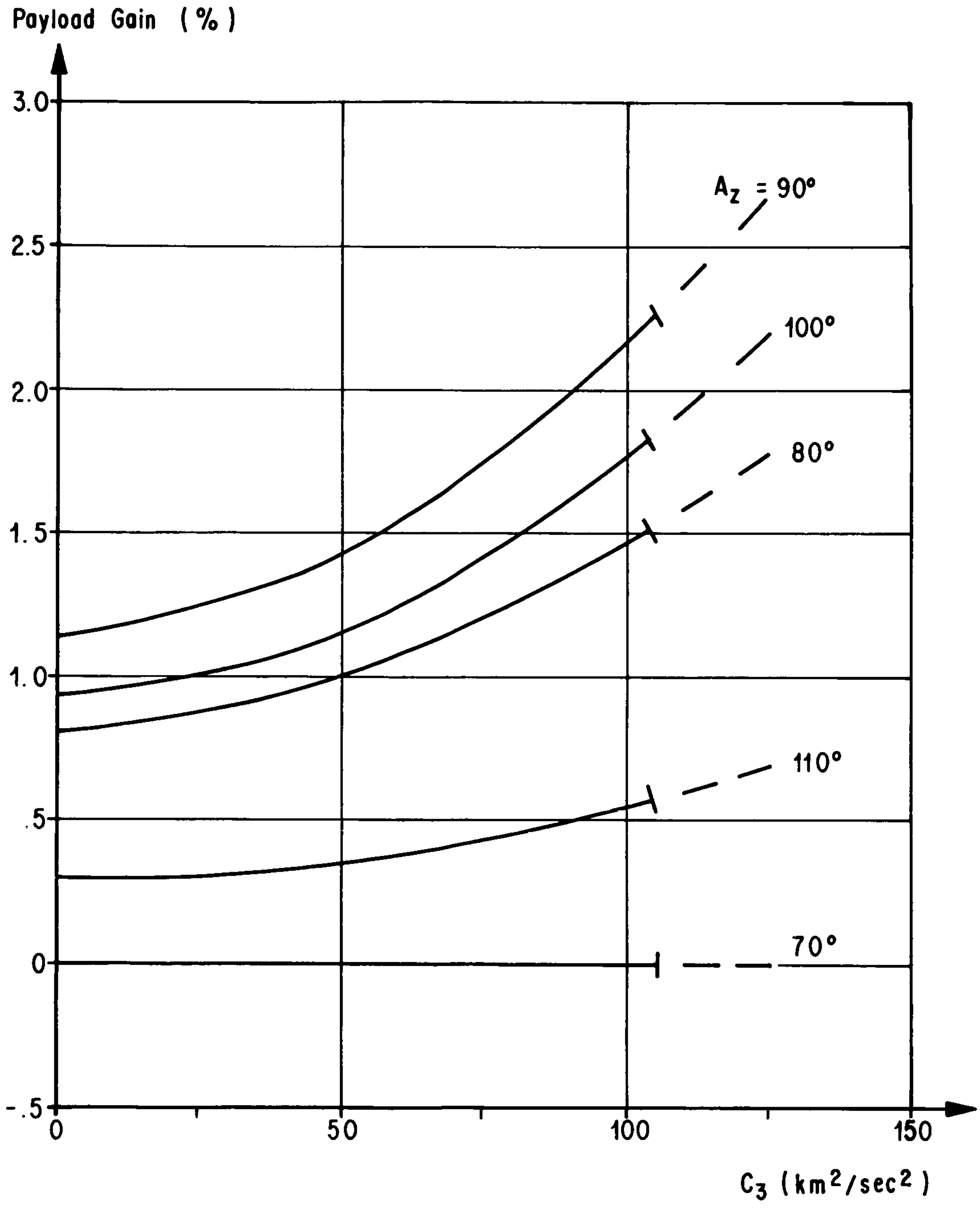

FIG. B-12. EFFECT OF LAUNCH AZIMUTH ON PAYLOAD VS C_3

APPENDIX C

Vehicle Data

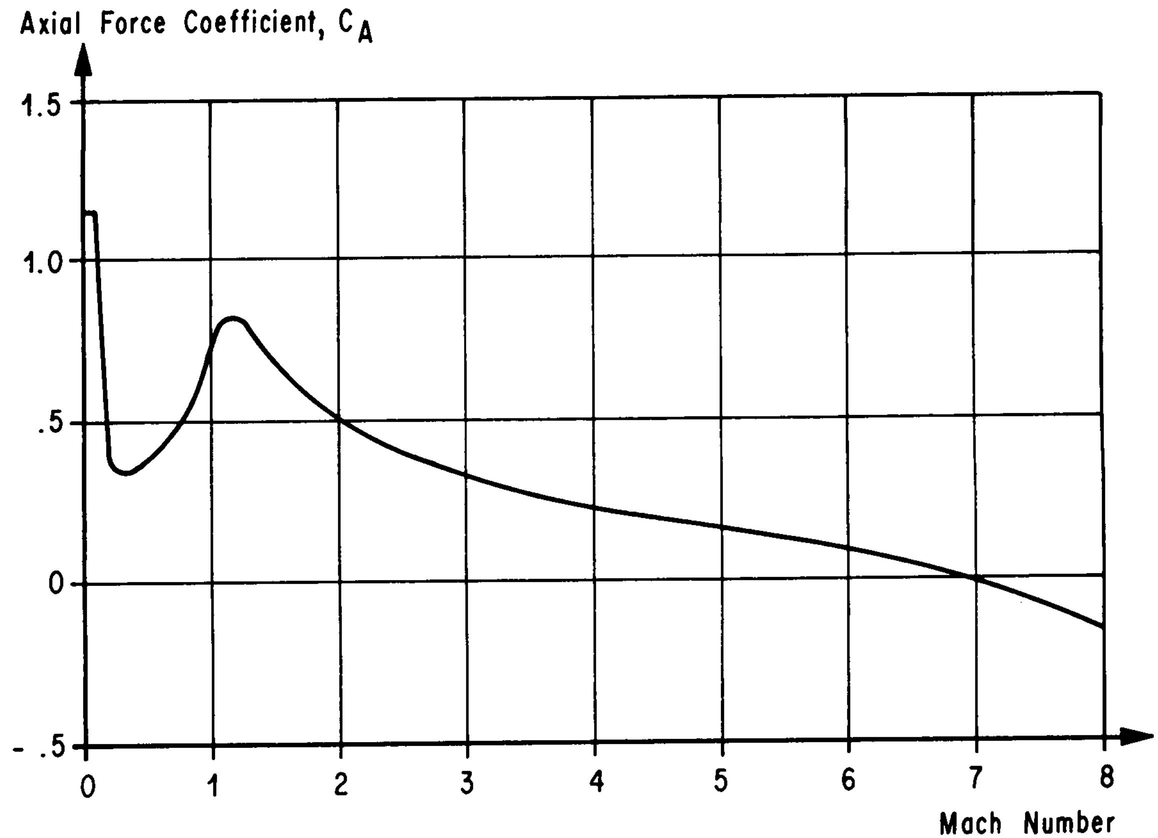

FIG. C-1. NOMINAL VARIATION OF AXIAL FORCE COEFFICIENT WITH MACH NUMBER FOR APOLLO SATURN V CONFIGURATION

Ref : NASA TMX - 53517

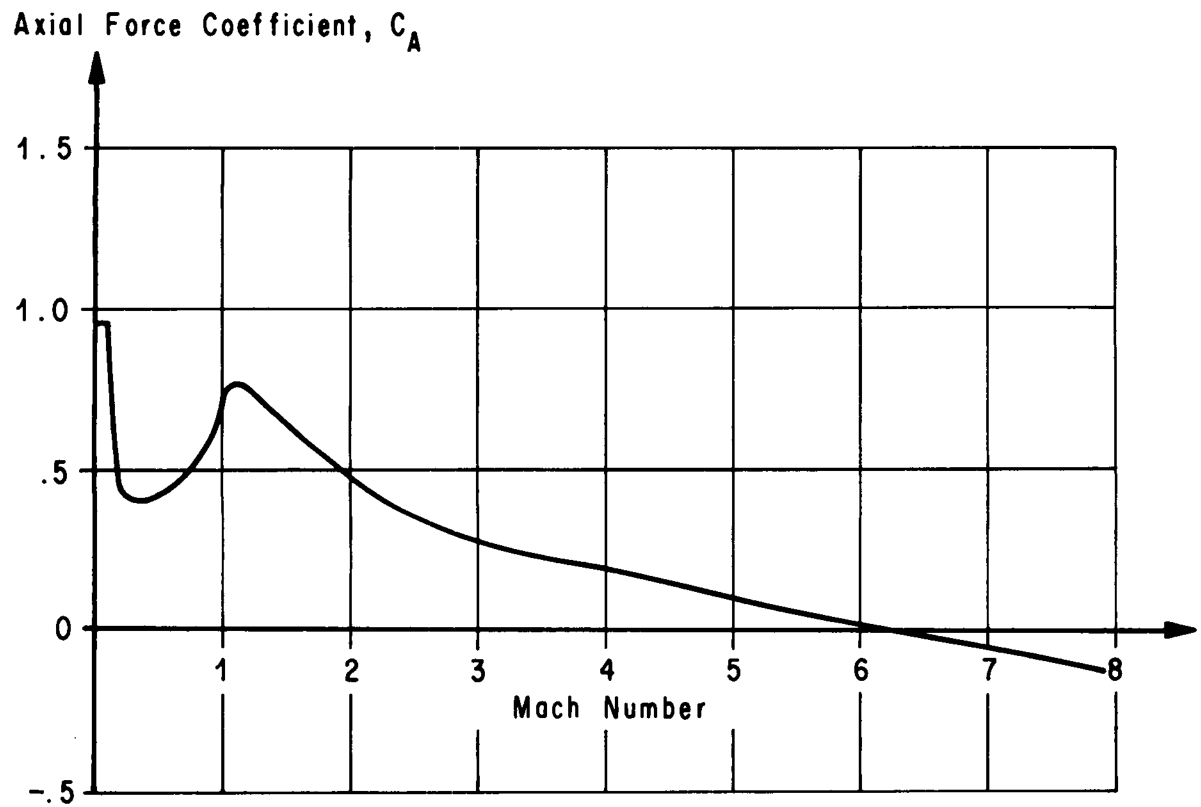

FIG. C-2. NOMINAL VARIATION OF AXIAL FORCE COEFFICIENT WITH MACH NUMBER FOR SATURN V MSFC NOSE CONE CONFIGURATION

Ref: R - AERO - AD - 66 - 59

TABLE 1

Propulsion Characteristics for Saturn Launch Vehicle SA-516

Stage	Number of Engines	Engine Designation	Thrust/Engine (lb)	Specific Impulse (sec)	Mixture Ratio W_f/W_o
S-IC	5	F-1	1,522,000*	264.5*	2.27:1
S-II	5	J-2	204,080	425	5.0:1
			228,915	422.1	5.5:1
			189,113	426.47	4.7:1
S-IVB	1	J-2	205,000	427	5.0:1
S-II	5	J-2S	204,080	430.5	5.0:1
			228,915	427.6	5.5:1
			189,113	431.97	4.7:1
S-IVB	1	J-2S	205,000	432.5	5.0:1

*Denotes Sea Level Values.

TABLE 2

Weight Summary for Saturn Launch Vehicle SA-516
Apollo Configuration with Standard J-2 Engines in S-II/S-IVB Stages

Stage	Item	Weight
S-IC	Mainstage Propellant Capacity	4,598,260
	N_2 Purge (Liftoff to Cutoff)	32
	S-II Insulation Purge Gas	120
	Frost (Total)	1,400
	Inboard Engine Thrust Decay Propellant	1,770
	Outboard Engines Thrust Decay Propellant	6,760
	Stage at Separation	325,013
	S-IC/S-II Interstage (Small)	1,400
S-II	Ullage Rocket Propellant	2,720
	Thrust Buildup Propellant	1,836
	S-IC/S-II Interstage (Large)	9,220
	Launch Escape System	8,300
	Mainstage Propellant Capacity W/PMR	970,000
	Thrust Decay Propellant	360
	Stage at Separation	93,031
	S-II/S-IVB Interstage	7,682
	S-IVB Aft Frame (Separated with Interstage)	48
S-IVB	Ullage Propellant	122
	H_2 in Start Tank	4
First	Thrust Buildup Propellant	360
Burn	Ullage Rocket Gases	127
	APS Propellant - Power Roll (First Burn)	18
	Thrust Decay Propellant	94
Orbital	Propellant Below Engine Valve	39
Coast	H_2 + H_e Vented in Orbit	3,016
	APS Propellant Used in Orbit	438
	LOX/Hydrogen Burner Propellant	16
	Oxidizer Vented in Orbit	130
Second	H_2 in Start Tank	6
Burn	Thrust Buildup Propellant	360
	Thrust Decay Propellant	94
	Total Mainstage Capacity (Incl. Reserves*)	230,000
	Stage at Separation	26,108
	Instrument Unit	4,050

*Reserves calculated as function of mission profile.

TABLE 3

Weight Summary for Saturn Launch Vehicle SA-516
Apollo Configuration with J-2S Engines in S-II/S-IVB Stages

Stage	Item	Weight
S-IC	Mainstage Propellant Capacity	4,598,260
	N_2 Purge (Liftoff to Cutoff)	32
	S-II Insulation Purge Gas	120
	Frost (Total)	1,400
	Inboard Engine Thrust Decay Propellant	1,770
	Outboard Engines Thrust Decay Propellant	6,760
	Stage at Separation	325,013
	S-IC/S-II Interstage (Small)	1,400
S-II	Ullage Rocket Propellant	2,720
	Thrust Buildup Propellant	1,836
	S-IC/S-II Interstage (Large)	9,220
	Launch Escape System	8,300
	Mainstage Propellant Capacity W/PMR	970,000
	Thrust Decay Propellant	360
	Stage at Separation	89,931
	S-II/S-IVB Interstage	7,682
	S-IVB Aft Frame (Separated with Interstage)	48
S-IVB	Ullage Propellant	122
	H_2 in start Tank	4
First Burn	Thrust Buildup Propellant	360
	Ullage Rocket Cases	127
	APS Propellant-Power Roll (First Burn)	18
	Thrust Decay Propellant	94
Orbital Coast	Propellant Below Engine Valve	39
	$H_2 + H_e$ Vented in Orbit	3,016
	APS Propellant Used in Orbit	438
	LOX/Hydrogen Burner Propellant	16
	Oxidizer Vented in Orbit	130
Second Burn	H_2 in Start Tank	6
	Thrust Buildup Propellant	360
	Thrust Decay Propellant	94
	Total Mainstage Capacity (Incl. Reserves*)	230,000
	Stage at Separation	25,208
	Instrument Unit	4,050

*Reserves calculated as function of mission profile.

TABLE 4

Weight Summary for Saturn Launch Vehicle SA-516
MSFC Nose Cone Configuration with Standard J-2 Engines
in S-II/S-IVB Stages

Stage	Item	Weight
S-IC	Mainstage Propellant Capacity	4,598,260
	N_2 Purge (Liftoff to Cutoff)	32
	S-II Insulation Purge Gas	120
	Frost (Total)	1,400
	Inboard Engine Thrust Decay Propellant	1,770
	Outboard Engines Thrust Decay Propellant	6,760
	Stage at Separation	325,013
	S-IC/S-II Interstage (Small)	1,400
S-II	Ullage Rocket Propellant	2,720
	Thrust Buildup Propellant	1,836
	S-IC/S-II Interstage (Large)	9,220
	Mainstage Propellant Capacity W/PMR	970,000
	Thrust Decay Propellant	360
	Stage at Separation	93,031
	S-II/S-IVB Interstage	7,682
	S-IVB Aft Frame (Separated with Interstage)	48
S-IVB	Ullage Propellant	122
	H_2 in Start Tank	4
First Burn	Thrust Buildup Propellant	360
	Ullage Rocket Cases	127
	APS Propellant - Power Roll (First Burn)	18
	Thrust Decay Propellant	94
Orbital Coast	Propellant Below Engine Valve	39
	H_2 + H_e Vented in Orbit	3,016
	APS Propellant Used in Orbit	438
	LOX/Hydrogen Burner Propellant	16
	Oxidizer Vented in Orbit	130
	Payload Fairing	2,700
Second Burn	H_2 in Start Tank	6
	Thrust Buildup Propellant	360
	Thrust Decay Propellant	94
	Total Mainstage Capacity (Incl. Reserves*)	230,000
	Stage at Separation	26,108
	Instrument Unit	4,050

*Reserves calculated as function of mission profile.

TABLE 5

Weight Summary for Saturn Launch Vehicle SA-516
MSFC Nose Cone Configuration with J-2S Engines in S-II/S-IVB Stages

Stage	Item	Weight
S-IC	Mainstage Propellant Capacity	4,598,260
	N_2 Purge (Liftoff to Cutoff)	32
	S-II Insulation Purge Gas	120
	Frost (Total)	1,400
	Inboard Engine Thrust Decay Propellant	1,770
	Outboard Engines Thrust Decay Propellant	6,760
	Stage at Separation	325,013
	S-IC/S-II Interstage (Small)	1,400
S-II	Ullage Rocket Propellant	2,720
	Thrust Buildup Propellant	1,836
	S-IC/S-II Interstage (Large)	9,220
	Mainstage Propellant Capacity W/PMR	970,000
	Thrust Decay Propellant	360
	Stage at Separation	89,931
	S-II/S-IVB Interstage	7,682
	S-IVB Aft Frame (Separated with Interstage)	48
S-IVB	Ullage Propellant	122
	H_2 in Start Tank	4
First Burn	Thrust Buildup Propellant	360
	Ullage Rocket Cases	127
	APS Propellant - Power Roll (First Burn)	18
	Thrust Decay Propellant	94
Orbital Coast	Propellant Below Engine Valve	39
	H_2 + H_e Vented in Orbit	3,016
	APS Propellant Used in Orbit	428
	LOX/Hydrogen Burner Propellant	16
	Oxidizer Vented in Orbit	130
	Payload Fairing	2,700
Second Burn	H_2 in Start Tank	6
	Thrust Buildup Propellant	360
	Thrust Decay Propellant	94
	Total Mainstage Capacity (Incl. Reserves*)	230,000
	Stage at Separation	25,208
	Instrument Unit	4,050

*Reserves calculated as function of mission profile.

REFERENCES

1. "Static Aerodynamic Characteristics of the Apollo-Saturn V Vehicle," NASA TM X-53517, Marshall Space Flight Center, September 16, 1966.

2. "Static Aerodynamic Characteristics of the Saturn V Voyager (45' Shroud Length)," MSFC, R-AERO-AD-66-59, December 20, 1966.

3. "Jettison Weight Summaries for Saturn SA-516 Vehicle for Various Configurations," MSFC, R-P&VE-AV-66-193, November 10, 1966.

4. Battle, C. Tucker and Robert G. Gottlieb, "Optimization of Multistage Three-Dimensional Boost Trajectories," Raytheon Reports BR-2758-1 and BR-2758-2, May 1, 1964.

5. Gottlieb, Robert G., Malcolm Frazier and Richard Moroney, "Updating and Checkout of the BOP-01 Computer Program," Raytheon Report FR-66-87-2, March 11, 1966.

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A PERFORMANCE STUDY FOR THE APPLICATION OF THE SATURN V TO HIGH ENERGY EARTH ESCAPE MISSIONS

Ronald G. Toelle

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

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