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PROPULSION ISSUES, OPTIONS,& TRADES

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ABSTRACT

This paper briefly discusses several different types of propulsion concepts: (1) pulsed fission, (2) continuous nuclear fission, (3) chemical, and (4) chemical boost with advanced upper stage concept. Some of the key characteristics of each type are provided, and typical concepts of each are shown.

COMPARSION OF ADVANCED PROPULSION CONCEPTS

Considerable confusion exists concerning the relative attributes of various advanced propulsion concepts. Figure 1 shows a relative performance comparison of propulsion concepts with respect to important vehicle design parameters.

In general, propulsion concepts to the left of the dashed line result in unsatisfactory trip times for a manned MARS mission because of insufficient vehicle acceleration. However, these advanced propulsion concepts could become feasible if combined with a nuclear or chemical boost from LEO, or if the vehicle starts from a Lunar libration point or GEO, thus reducing Earth escape spiral time. For Mars missions there is little advantage for low thrust if it is necessary to boost to escape from LEO. The libration points or GEO options are mission design options beyond the scope of this paper. The discussion herein is therefore restricted to: (1) pulsed fission, (2) continuous nuclear fission, (3) chemical, and (4) chemical boost with advanced upper stage concept.

NUCLEAR FISSION PULSE PROPULSION

Nuclear fission pulse propulsion was studied extensively as a space transportation device from 1958 until 1965 under project Orion. An illustration of the NASA Orion vehicle, sized for compatibility with the Saturn V launch vehicle, is shown in Figure 2. This vehicle, according to reference 1, would be capable of completing a manned Mars surface-excursion mission from a single Earth launch, using a Saturn first stage. For this mission, the nuclear pulse propulsion would begin at suborbital velocity, starting at an altitude greater than 100 km (50 n mi). The vehicle shown has an estimated specific impulse of 2500 sec, a dry mass

Figure 1. Advanced Propulsion System Performance

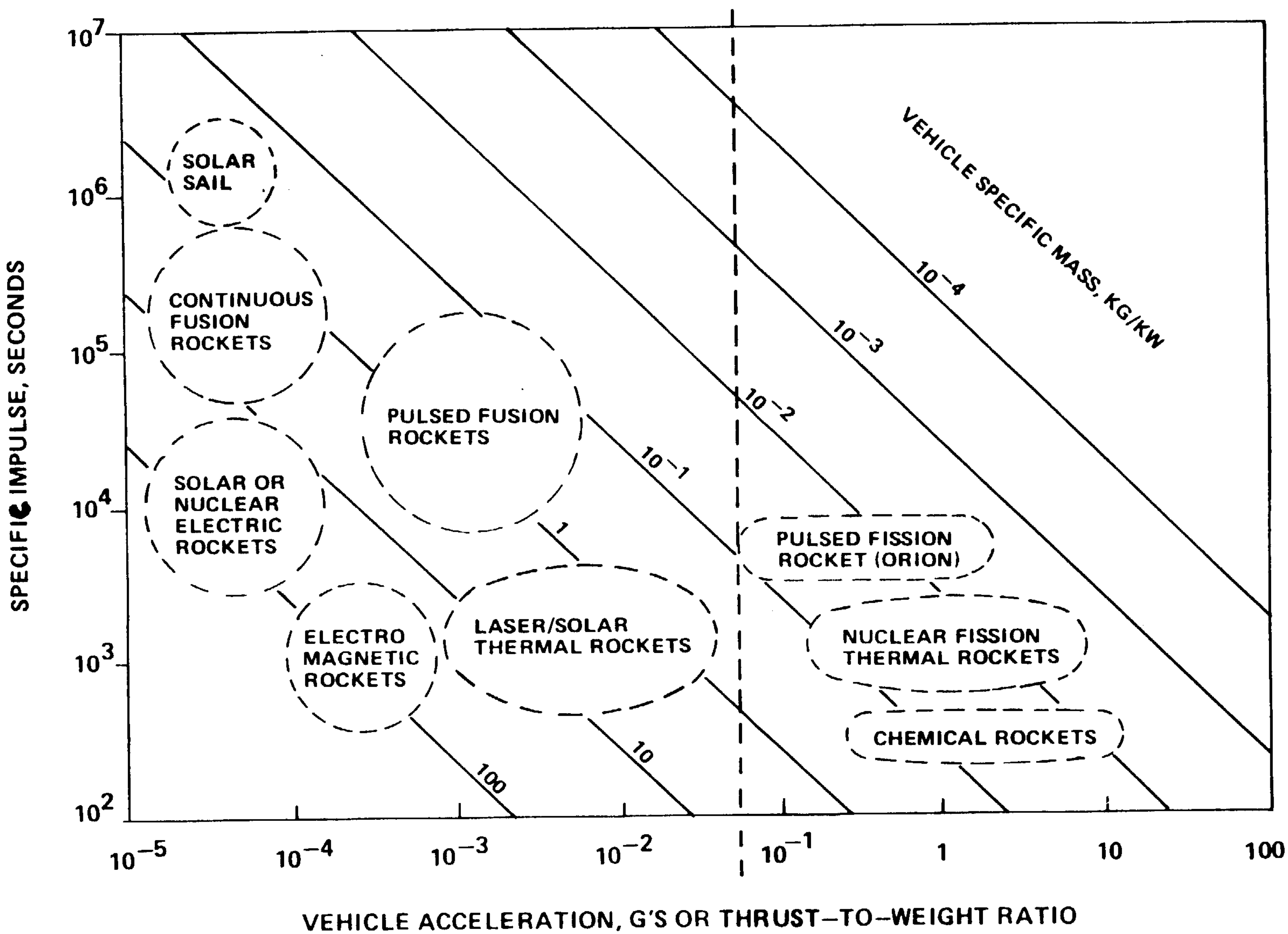
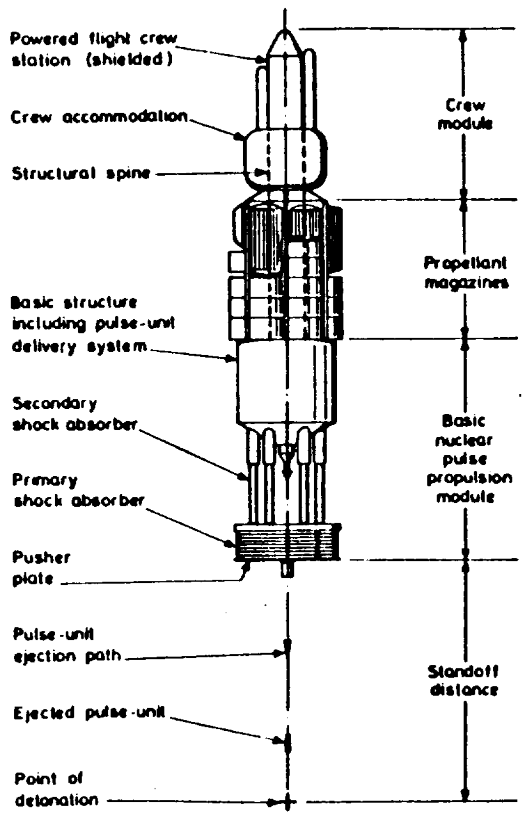


Figure 2. Summary of Nuclear Fission Pulsed Rocket (Orion) Characteristics



PERFORMANCE	• I_{sp} = 2000-6000 SEC • ENGINE T/W~4	DEPENDS ON DIAMETER OF PUSHER PLATE
OPERATIONAL CAPABILITY	• VERY LONG LIFETIME • CAN BE MAINTAINED • OPERATION BELOW GEO QUESTIONABLE (PROJECT STARFISH)	
RISK/FEASIBILITY ISSUES	• DATA BASE BELIEVED GOOD (CLASSIFIED) • NUCLEAR TEST BAN TREATY PROHIBITS TESTING • HIGH DEVELOPMENT COST MODERATE RADIATION HAZARD	

RECOMMENDATION

THIS CONCEPT HAD THE BEST ALL-AROUND PERFORMANCE OF ANY CONCEPT EXAMINED, BUT FEASIBILITY FOR NEAR EARTH MISSIONS NEEDS FURTHER INVESTIGATION. THIS CONCEPT RECOMMENDED FOR FUTURE STUDY CONSIDERATION

Figure 3. Summary of Nuclear Fission Thermodynamic Rocket Characteristics

REACTOR TYPE	PERFORMANCE	OPERATIONAL CAPABILITY	RISK/FEASIBILITY ISSUES
SOLID CORE REACTOR	800 - 1000 SECS ENGINE T/W ≃ 3	LIMITED LIFETIME POOR MAINTAINABILITY	RADIATION HAZARD FROM USED ENGINE; MUCH DESIGN DATA AVAILABLE.
ROTATING BED REACTOR	1000 - 1200 SECS ENGINE T/W ≃ 6	LIMITED LIFETIME CAN BE SERVICED	RADIATION HAZARD MODERATED BY CORE REMOVAL; DESIGN LEVEL TECHNOLOGY AVAILABLE.
LIQUID CORE REACTOR	1400-1600 SECS ENGINE T/W ≃ 1	VERY SHORT LIFETIME ONE SHOT MISSIONS	NO CONTAINMENT OF FISSION PRODUCTS: VERY LIMITED DATA BASE.
OPEN-CYCLE GAS-CORE REACTOR	1500 - 2000 SECS ENGINE T/W ≃ 1	LONG LIFETIME, BUT MUST BE REFUELED EVERY BURN	NO CONTAINMENT OF FISSION PRODUCTS: GOOD DATA BASE BUT FEASIBILITY NOT PROVEN.
CLOSED-CYCLE GAS CORE REACTOR	1500 - 2000 SECS ENGINE T/W ≃ 1	LIFETIME UNKNOWN CAN BE SERVICED	"LIGHTBULB" EXTREMELY HIGH RISK GOOD DATA BASE BUT FEASIBILITY NOT PROVEN.

RECOMMENDATION

SOLID CORE AND ROTATING BED REACTORS SHOULD BE CARRIED INTO TASK 2.

of 90,000 kg (200,000 lb), and an effective thrust level of 3,470,000 N (780,000 lbf).

Unfortunately, the same grounds used in 1965 to terminate the original Orion project are still valid today. For instance: (a) The large size and power of the vehicle made full-scale tests difficult and very expensive (final testing in space required); and (2) The 1963 nuclear-test-ban treaty specifically excluded nuclear explosions in the atmosphere or in space.

NUCLEAR FISSION THERMODYNAMIC ROCKET

The characteristics of five types of nuclear fission thermodynamic rockets are summarized in Figure 3. Much work was expended on these concepts prior to abandonment of the U.S. nuclear rocket program in 1973 and, for most of these concepts, the data base is quite good. Of the five concepts, the solid-core and rotating-bed rockets are recommended for vehicle-level assessment. The liquid-core reactor was dropped for not being reusable, the open-cycle gas-core reactor was dropped for being too large and too expensive to operate in near-term applications, and the closed-cycle gas-core or "light bulb" reactor was dropped because of feasibility issues concerning the light bulb.

CHEMICAL PROPULSION

Space vehicle design work at MSFC in 1985 has centered primarily on the cryogenic system, utilizing liquid oxygen/liquid hydrogen as propellants. Advanced engine candidates include the STME 625 (SSME derivative) for Stage 1 engines and the advanced expander cycle engine (RL-10 derivative) for Stage 2 and Stage 3 engines.

The storable propellant option utilizing nitrogen tetroxide/monomethyl hydrazine as propellants has been pursued to alleviate the cryogenic propellant boil-off problem; however, the storable propellant option has a significant vehicle weight penalty compared to the cryogenic. Figure 4 depicts typical chemical propulsion engine concepts.

MULTIPLE ENGINES SIMPLIFY ATTITUDE CONTROL

Consider the space vehicle of reference 2 as depicted in Figure 5. Note that Stage 2 and Stage 3 have single main engines. If these engines were replaced with two or more smaller engines with gimballing capability, the outbound midcourse correction system, the inbound midcourse

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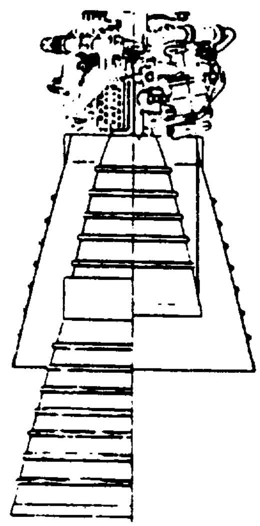
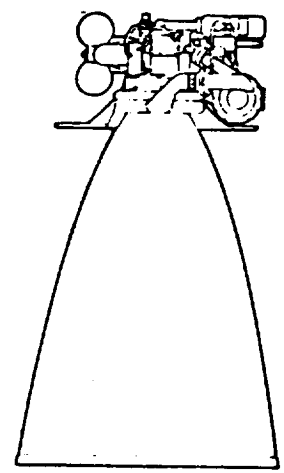
SYSTEM	CHEMICAL	
	CRYOGENIC	STORABLE
SPECIFIC IMPULSE	461–482	327–346
ENGINE THRUST–TO–WEIGHT	38–78	29–41
PROPELLANT	LOX/LH_2	N_2O_4/MMH
		

Figure 4. Summary of Chemical Propulsion Characteristics

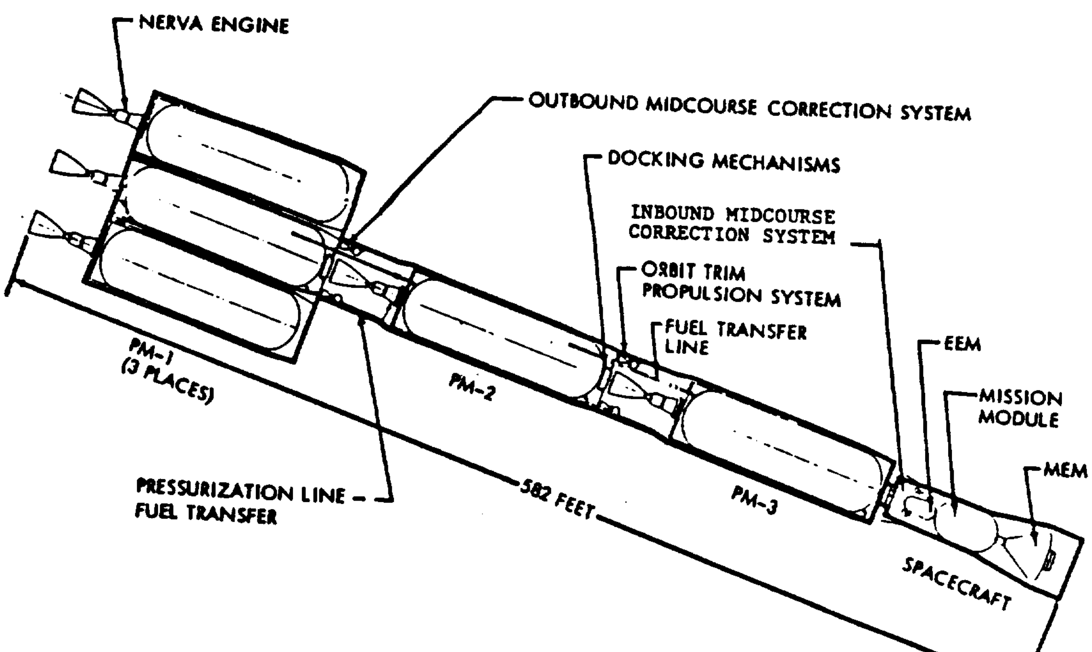


Figure 5. Space Vehicle Design of 1968.

correction system, and the orbit trim propulsion system could be eliminated.

The resulting vehicle design would be much simpler, with three fewer propulsion systems as well as an "engine-out" capability. A small chemical attitude control system could be incorporated to handle small correction maneuvers, rather than restarting the reactor.

MARS EXCURSION MODULE (MEM) DESCENT/ASCENT ENGINE OPTIONS

Early studies (ref.3) investigated the trade-offs between plug nozzle engines and bell nozzle engines. The envelopes of these engine types is shown in Figure 6. Propellant combinations evaluated were OF_2/MMH, FLOX/CH_4, and LO_2/LH_2.* Plug nozzle engines were baselined at that time in order to fit the MEM envelope.

MSFC studies in 1985 have centered around engine types and propellant combinations which are closer to state-of-the-art. Two engine designs were evaluated, both utilizing two-position nozzles. A summary of the performance characteristics of these engines is shown in Figure 7.

*These formulae and acronyms denote:

OF_2	oxygen difluoride
MMH	monomethyl hydrazine
FLOX	a mixture of liquid fluorine and liquid oxygen
CH_4	methane
LO_2	liquid oxygen
LH_2	liquid hydrogen

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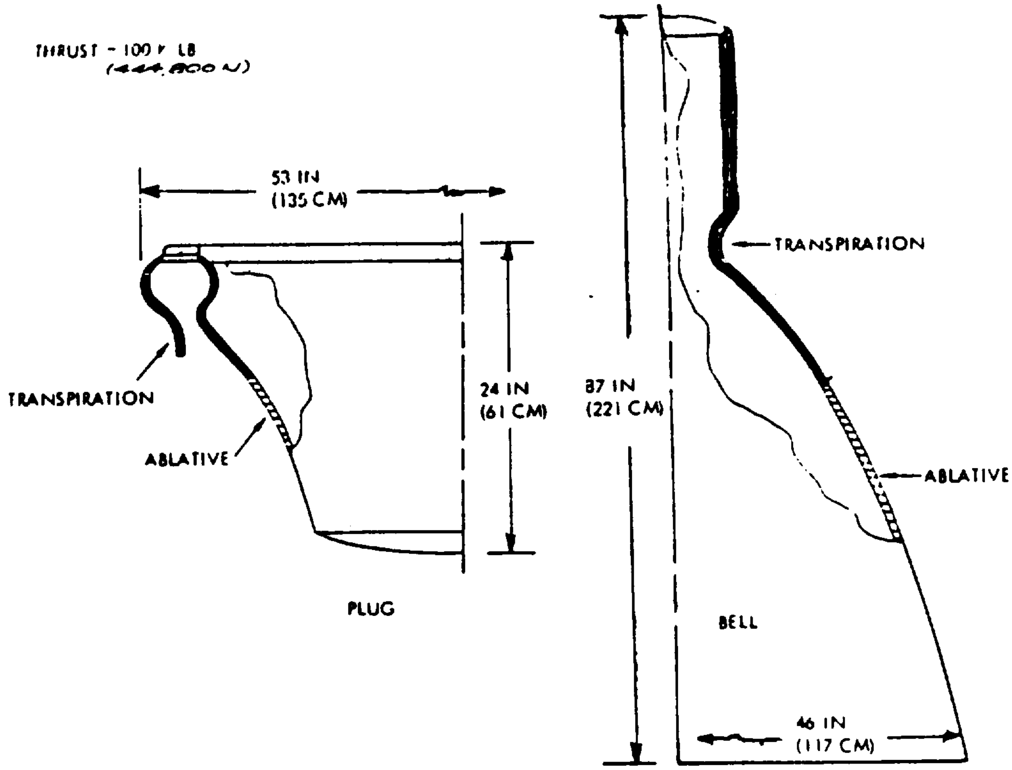

Figure 6. Plug Nozzle and Bell Nozzle Envelope

MEM DESCENT/ASCENT ENGINES

	BASELINE	OPTION
• PROPELLANTS	LOX/MMH	N_2O_4/MMH
• NOZZLE AREA RATIO (FIXED/EXTENDED)	30/75	30/75
• VACUUM THRUST (LBF)	40K	40K
• CHAMBER PRESSURE (PSIA)	1430	1430
• MIXTURE RATIO (O/F)	1.7	2.0
• DEL ISP VAC (SEC)	360.5	328.6
• LENGTH (IN)	52.6/76.8	53.7/78.4
• DIAMETER (IN)	37.6	38.5
• DRY WEIGHT (LBM)	555	573

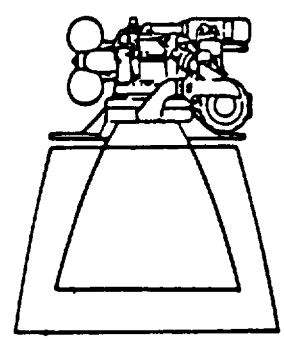

Figure 7. Two-Position Nozzle Designs

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1. "Advanced Propulsion Systems for Orbital Transfer", D180-26680-1, NAS8-33935,1981.
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3. "Definition of Experamental Tests for a Manned MARS Excursion Module", SD67-755-2, NAS9-6464, 12 Jan. 1968.