N71-22616

NASA TECHNICAL MEMORANDUM

NASA TM X-2256

NASA TM X-2256

CASE FILE COPY

MISSION PERFORMANCE POTENTIAL OF REGENERATIVELY COOLED GAS-CORE NUCLEAR ROCKETS

by Laurence H. Fishbach

Lewis Research Center

Cleveland, Ohio 44135

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • APRIL 1971

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TM X-2256		
4. Title and Subtitle MISSION PERFORMANCE POTENTIAL OF REGENERATIVELY COOLED GAS-CORE NUCLEAR ROCKETS		5. Report Date April 1971
		6. Performing Organization Code
7. Author(s) by Laurence H. Fishbach		8. Performing Organization Report No. E-6092
		10. Work Unit No. 124-08
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135		11. Contract or Grant No.
		13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

The performance of regeneratively cooled gas-core nuclear rockets is compared to that of solid-core nuclear rockets for four missions: a reusable lunar ferry; a "slingshot," in which an orbiting rocket boosts payloads to various hyperbolic excess velocities and then returns to orbit; and manned round trips to Mars and to Jupiter. The results indicate that gas-core nuclear rockets may offer a significant capability improvement for performing these missions, requiring only 50 to 75 percent of the initial mass in Earth orbit needed by the solid-core rocket. In addition, high specific impulse, even with a corresponding increase in engine weight, seems desirable.

17. Key Words (Suggested by Author(s))		18. Distribution Statement	
Missions; Nuclear Propulsion; Performance; Nuclear rockets; Gas-core; Gas-core nuclear rockets; Lunar ferry; Mars; Jupiter; Mission analysis; Booster		Unclassified - unlimited	
19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	19	$3.00

*For sale by the National Technical Information Service, Springfield, Virginia 22151

MISSION PERFORMANCE POTENTIAL OF REGENERATIVELY COOLED GAS-CORE NUCLEAR ROCKETS

by Laurence H. Fishbach

Lewis Research Center

SUMMARY

The performance of regeneratively cooled gas-core nuclear rockets is compared to that of solid-core nuclear rockets for four missions: a reusable lunar ferry; a "sling-shot," in which an orbiting rocket boosts payloads to various hyperbolic excess velocities and then returns to orbit; and manned round trips to Mars and to Jupiter.

The results indicate that gas-core nuclear rockets may offer a significant capability improvement for performing these missions, requiring only 50 to 75 percent of the initial mass in Earth orbit needed by the solid-core rocket. In addition, high specific impulse, even with a corresponding increase in engine weight, seems desirable.

INTRODUCTION

The low specific impulse I_{sp} of chemical rockets severely restricts mission capability. For missions more difficult than lunar exploration and one-way probes to the planets, high impulsive-velocity requirements or ΔV's result in extremely large initial masses in Earth orbit. To embark on an ambitious program of space exploration it becomes immediately apparent that higher specific impulses are necessary.

Solid-core nuclear rockets (SCNR) develop relatively high thrusts at about twice the specific impulse of the best chemical rockets (825 sec as opposed to 400+ sec). Even this increase in I_{sp} is inadequate for very energetic, very high payload missions. At the opposite end of the advanced-engine spectrum, electric rockets can develop extremely high specific impulse. On the other hand, their very low thrust implies long propulsion times and, in many cases, undesirably long mission times.

This report investigates the performance of a device - the gas-core nuclear rocket - which is midway between solid-core rockets and electric rockets in terms of both thrust and specific impulse. Gas-core nuclear rockets (GCNR) have been studied in the past (refs. 1 to 5). References 1 and 2 developed relations between the design parameters of GCNR and its weight. References 3 to 5 estimated GCNR capabilities for some missions, but these used off-optimum thrust levels and/or considered only near-Earth missions. The present work investigates the GCNR performance potential on the basis of optimized thrust levels for a more representative mission sample. Its purpose is to indicate whether (or under what circumstances) the GCNR engines as described in references 1 and 2 would lead to a sufficiently improved mission capability to warrant further, more detailed study.

A regeneratively cooled GCNR is shown schematically in figure 1. Uranium is injected into the reactor core, where it vaporizes. Hydrogen after being circulated through the nozzle, moderator and pressure shell for cooling is injected into the core, heated by the fissioning uranium, and expelled through the nozzle to produce thrust. Since this cycle is "open" (i.e., the uranium is in direct contact with the hydrogen), some uranium will also be expelled through the nozzle. Specific impulses of the rocket are in the range 1000 to 3000 seconds, the upper limit being determined by the cooling capability of the incoming liquid hydrogen. Thus GCNR's might have at least seven times the I_{sp} of chemical rockets and four times that of SCNR's.

Four missions representing various degrees of energy requirements were chosen for this study. They were a reusable lunar ferry (from, and return to, low Earth orbit); a "slingshot," in which an orbiting rocket boosts payloads to various hyperbolic excess velocities and then returns to Earth orbit; and manned Mars and Jupiter round trips. The criterion of merit in all cases is the initial mass of the space vehicle starting in Earth orbit.

ANALYSIS

Engine Mass Relations

The mass of a gas-core nuclear rocket can be expressed as the sum of four component masses: moderator, pressure shell, turbopump, and nozzle. (All symbols are defined in the appendix.)

$$M_e = M_{mod} + M_{ps} + M_{tp} + M_n \tag{1}$$

References 1 and 2 developed engine mass relations for the regeneratively cooled gas-core nuclear rocket. The relations were derived by assuming that the fraction of the core volume filled by uranium gas was a function of the hydrogen- to uranium-flow-rate ratio. More recent experiments (unpublished data by Robert Ragsdale, Lewis Research Center) have yielded data from which has arisen a better representation of this function:

$$\frac{V_U}{V_c} = 0.4\left(\frac{\dot{M}_{H_2}}{\dot{M}_U}\right)^{-0.0926} \tag{2}$$

where $\dot{M}_{H_2}$ and $\dot{M}_U$ are the flow rates out of the nozzle for hydrogen and uranium, respectively; and V_U and V_c are the volume of uranium gas and the volume of the core, respectively. The equations in references 1 and 2 were then rederived using this more accurate relation.

The moderator is assumed to be 0.762 meter thick, so that the mass of the moderator can be determined from

$$M_{mod} = \frac{\pi}{6}\left[(D + 1.524)^3 - D^3\right]\rho_{mod} \tag{3}$$

The pressure in the core is a function of critical mass, thrust level, specific impulse, flow-rate ratio, and engine diameter.

$$P = \left[\frac{0.25617 \times M_{cr} \times (F \times I_{sp})^{0.277}}{D^{3.277}} \times \left(\frac{\dot{M}_{H_2}}{\dot{M}_U}\right)^{0.1012}\right]^{1.383} \tag{4}$$

The temperature of the hydrogen in the core is given by

$$T = 938.6\left(\frac{P \times F \times I_{sp}}{D}\right)^{0.1565} \times \left(\frac{\dot{M}_{H_2}}{\dot{M}_U}\right)^{0.004831} \tag{5}$$

This temperature is not used in the calculation of engine mass, but is of interest to the rocket designer.

The thickness of the pressure shell t is a function of the pressure, diameter, and strength of the material

$$t = P \times \frac{D + 1.524}{4\sigma} \tag{6}$$

and the mass of the pressure shell is obtained from

$$M_{ps} = \frac{\pi}{6}\left[(D + 1.524 + 2t)^3 - (D + 1.524)^3\right]\rho_{ps} \tag{7}$$

This mass increases with the pressure and, hence, with the specific impulse of the rocket. The pressure shell and the moderator are the two heaviest parts of the engine, so that engine mass increases with specific impulse.

The hydrogen flow rate is determined from the thrust and specific impulse

$$\dot{M}_{H_2} = \frac{F}{I_{sp} \times g} \tag{8}$$

and the mass of the turbopump is determined from

$$M_{tp} = 48.07 \times \dot{M}_{H_2} \times \frac{(1.5 \times P)^{2/3}}{\rho_{H_2}} \tag{9}$$

The mass of the nozzle is

$$M_n = 0.001739 \times \epsilon \times \frac{F}{P} \tag{10}$$

where ϵ is the area ratio of the nozzle.

Thus, all the terms necessary to determine the mass of the four components for equation (1) have been found.

For all cases studied in this report, unless otherwise specified, the diameter of the engine is 3.66 meters, the critical mass is 48 kilograms, and the hydrogen- to uranium-flow-rate ratio is 100. The following constants were used:

(1) Allowable stress in pressure shell, $\sigma = 13\,600$ atm

(2) Density of pressure shell material, $\rho_{ps} = 8000\ \text{kg/m}^3$

(3) Density of hydrogen, $\rho_{H_2} = 72\ \text{kg/m}^3$

(4) Area ratio of nozzle, $\epsilon = 300$

(5) Density of moderator material, $\rho_{mod} = 1150\ \text{kg/m}^3$

In addition, the thrust level was optimized for all cases.

Impulsive Velocity Requirements

To account for gravity losses for finite-burn-time rockets, the ratio of actual to ideal impulsive ΔV's was calculated as a function of acceleration level, parking orbit eccentricity, and final hyperbolic excess velocity. These data were pretabulated into another computer code in the form of a table with interpolation.

The ideal ΔV's were calculated for the particular mission and then these correction factors were used to correct the ΔV's to "actual" values. Since the thrust level was being optimized and hence the acceleration level changing, these correction factors varied while the optimization was progressing.

The required hyperbolic excess velocities from which the ideal ΔV's can be calculated are shown in table I.

Vehicles and Mission Profiles

Vehicle weight relations. - Vehicle weights were calculated by using the classic rocket equation. Propellant weight for each maneuver was calculated from

$$\left(M_p\right)_i = \left(M_0\right)_i \left(1 - \frac{e^{-\Delta V_{actual}}}{I_{sp} g}\right) \tag{11}$$

The initial mass for the next maneuver was then

$$\left(M_0\right)_{i+1} = \left(M_0\right)_i - \left(M_p\right)_i - \left(M_{pstr}\right)_i - M_{jettison} - \left(M_{is}\right)_i \tag{12}$$

The final maneuver requires that the initial mass equal

$$\left(M_0\right)_{i_{max}} = \left(M_p\right)_{i_{max}} + M_{pstr} + M_{ts} + M_{pay} + M_e \tag{13}$$

$$M_{ts} = 0.002 \times F \tag{14}$$

$$\left(M_{pstr}\right)_i = 0.20 \times \left(M_p\right)_i \tag{15}$$

$$\left(M_{is}\right)_i = 0.01 \times \left(M_0\right)_i \tag{16}$$

and, for solid-core nuclear rockets,

$$M_e = 0.035 \times F \tag{17}$$

For Mars and Jupiter missions,

$$M_{jettison} = 136\ 100\ kg \tag{18}$$

For the lunar ferry and slingshot missions the payload is actually $M_{jettison}$ for the second maneuver and the first maneuver, respectively. In addition, 50 000 kilograms of payload was returned to Earth orbit for each of these two missions.

Lunar ferry. - There are four missions studied in this report. The first of these and its associated vehicle the lunar ferry, are shown in figure 2. In this mission, the GCNR, which was in a parking orbit about Earth, follows a Hohmann transfer to deliver various amounts of payload into a lunar orbit. The vehicle then returns on a Hohmann transfer to Earth and into the initial parking orbit. There it would be refueled, pick up another payload, and depart for lunar orbit. The capability to deliver large payloads to the moon will be a necessity if and when lunar colonies are formed.

Slingshot. - The second mission studied was the "slingshot." This mission and its associated vehicle are shown in figure 3. The vehicle is initially in an Earth parking orbit, then boosts out of orbit to a given V_{∞} and separates, with the payload continuing along the initial path. The gas-core rocket then retrofires, returning to low Earth orbit where an additional impulse places the vehicle back into a circular parking orbit. This mission is analogous to the reusable launch vehicle; the "slingshot" is ready to boost another payload as soon as it has been refueled.

Manned interplanetary. - The third and fourth missions studied are manned interplanetary round trips to Mars and Jupiter. These missions are shown in figure 4. For these missions it was assumed that 136 100 kilograms of payload was delivered into a 0.9-eccentricity planetary parking ellipse with periapsis at 1.1 planet radii. Additional payload of 90 700 kilograms which includes the Earth reentry vehicle was returned to Earth. Deceleration at Earth return was accomplished with atmospheric braking; no limit on entry velocity was set.

RESULTS AND DISCUSSION

Lunar Ferry Mission

The previously chosen values of diameter (3.66 m), critical mass (48 kg), and hydrogen- to uranium-flow-rate ratio (100) were picked as being reasonable estimates. Using these values and a specific impulse of 2000 seconds and optimizing the thrust, a comparison between the performance of gas-core and solid-core nuclear rockets was made for the lunar ferry mission. This comparison is shown in figure 5. Two 825-second-I_{sp}, solid-core rockets are indicated: a state-of-the-art, NERVA-II-type engine and an optimistic SCNR (zero engine weight). The gas-core rocket requires less initial mass in Earth orbit to deliver a given payload; in fact, the payload ratio at a payload of 500 000 kilograms to lunar orbit and the additional 50 000 kilograms back to Earth orbit approaches 50 percent.

Initial mass in Earth orbit (IMEO) is only one criterion for evaluating a mission of this nature. Presumably, the vehicle will be used many times, since it is functioning as a ferry. Thus, although for the first trip the entire initial mass must be boosted into orbit, for subsequent trips all that need be boosted would be the fuel for a round trip. The fuel requirements as a function of payload mass are shown in figure 6. The gas-core rocket performance is even more impressive when viewed in this manner since, for example, at a payload of 500 000 kilograms only 50 percent as much fuel as for the optimistic SCNR must be delivered to Earth orbit. On the basis of IMEO, the GCNR required 75 percent as much mass.

The effect of the four independent variables describing the gas-core rocket on the initial mass requirements for the 500 000-kilogram-payload ferry mission is shown in figures 7 and 8.

Raising the specific impulse from 2000 seconds to 3000 seconds results in a decrease of 11 percent in initial mass requirements (fig. 7). This is, of course, a result of lower propellant requirements at the higher impulse. The effect of diameter and of critical mass is shown in figure 8(a). The values of M_{cr} for these cases were supplied by Robert Ragsdale of the Lewis Research Center and are indicated on the figure. Increasing the diameter increases the initial mass required for the "best guess" and "low" estimates of critical mass. This is largely a result of having to spread an essentially constant-thickness moderator over an increasing surface as diameter is increased, and thus increasing the engine mass. The curve representing "high" critical mass goes through a minimum in initial mass requirements. Although here also the moderator mass increases with diameter, another factor at times dominates. With a high fuel loading, as diameter of the engine decreases, the temperature and pressure rise so significantly in the core that the increase in pressure-shell mass dominates the decrease in moderator mass, and this results in the minimum.

In figure 8(b), it is shown that initial mass is relatively insensitive to hydrogen- to uranium-flow-rate ratio. It increases slightly with flow-rate ratio since pressure-shell weight increases (recall eq. (4)); this penalty then must be compared with the benefits of lower uranium consumption for the mission.

Slingshot Mission

The next mission studied was the slingshot (fig. 3) where 500 000 kilograms of payload mass was brought to various hyperbolic excess velocities by a gas-core rocket -

which then returned itself to low Earth orbit. The results of this study are shown in figure 9. Initial mass in Earth orbit is seen to increase with increasing V_∞ and to decrease with increasing I_{sp}. Doubling the gas-core I_{sp} from 1500 to 3000 seconds results in savings of 20 percent at a V_∞ of 0 to 30 percent at a V_∞ of 5 kilometers per second. Indicated on this figure is the IMEO for a solid-core nuclear rocket. It is interesting to note that a V_∞ of 5.5 kilometers per second, which is about the requirement for a 300-day Mars round trip, cannot be delivered by a single-stage solid-core rocket regardless of the IMEO, while it can be delivered by all the single-stage GCNR's.

Manned Interplanetary Missions

Shown next in figure 10(a) are the results of the Mars round-trip study. Both these results and those for Jupiter were calculated by assuming circular coplanar orbits, so that the results represent the average expected performance regardless of synodic period.

The IMEO is seen to decrease with increasing travel time and I_{sp}. The importance of I_{sp} can be shown first by noting that if the IMEO is fixed at 10^6 kilograms for all cases, the 3000-second-I_{sp} GCNR engine can do the mission in 225 days while the 1500-second-I_{sp} GCNR engine requires 360 days and the 825-second NERVA-II-type SCNR requires 430 days. At short trip times, such as 200 days, the 3000-second engine requires one-fifth of the mass of the 1500-second engine. Since trip time for manned round trip missions may well be a prime consideration, this time savings can be very important. In figure 10(b) it can be seen that the optimum thrust levels are in the range of 400 000 to 5 million newtons. It should be recalled that for a given thrust, engine weight increases with I_{sp}. To compensate, the optimum thrust decreases; however, the final result is that the optimum engine weight increases with I_{sp}.

In figure 11(a) are illustrated the results of the study of round trips to Jupiter. In reality, the data points shown are discrete and should not be connected by lines; that is, the trip can be done in 1020 days or in 1420 days or in 580 days but not at intermediate values of time. At these times, the planets are in opposition or in conjunction at midstay. The mission ΔV requirements between these times rise steeply. (For a further discussion of this rise in ΔV as opposed to a continuous curve for Mars ΔV's the reader is referred to ref. 6.) For the GCNR's, since solutions are possible at each trip time for each of the specific impulses, higher impulse can only lower IMEO, not shorten trip time as for the Mars mission. They can, however, perform the mission faster (in 580 days) than SCNR's since the SCNR becomes energy limited at that trip time. For the 1420-day trip, the 3000-second-I_{sp} GCNR requires only 15 percent as much IMEO as the 825-second-I_{sp} SCNR and 60 percent as much IMEO as the 1500-second GCNR.

From figure 11(b) it can be seen that the optimum thrust levels for this mission are in the 10^6- to 10^7-newton range, or about twice that of the Mars mission.

CONCLUDING REMARKS

It has been shown that the regeneratively cooled gas-core nuclear rocket (GCNR) appears to be significantly better than the solid-core, 825-second-specific-impulse (I_{sp}) nuclear rocket engine for the four missions studied herein. For the cases studied, savings of between 25 and 50 percent in the initial mass required in Earth orbit were obtained. Improvements occur even if the I_{sp} of the GCNR has only a 1500-second specific impulse. It was shown, however, that the really significant improvements occur as the I_{sp} approaches 3000 seconds, the approximate upper limit for this type engine. This is true even though the engine weight is higher for a 3000- than for a 1500-second-I_{sp} engine. A more reasonable engine of 2000-second I_{sp} still gives fair-sized performance gains. The results imply that an I_{sp} greater than 3000 seconds, even if accompanied by a major increase in engine specific weight, would lead to even better mission performance. Future efforts should consider the feasibility of building a very high impulse GCNR and should evaluate the weight or thrust penalties that are involved.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, January 13, 1971,
124-08.

APPENDIX - SYMBOLS

D	outside diameter of core (i.d. of moderator), m
F	thrust, N
g	standard value of gravity, 9.80665 m/sec^2
H_2/U	hydrogen- to uranium-flow-rate ratio
I_{sp}	specific impulse, sec
M_{cr}	critical mass in reactor, kg
M_e	mass of engine, kg
M_{is}	interstage structure mass, kg
$M_{jettison}$	mass jettisoned, such as Mars lander, kg
M_{mod}	moderator mass, kg
M_n	nozzle mass, kg
M_p	propellant mass, kg
M_{pay}	payload mass, kg
M_{ps}	pressure-shell mass, kg
M_{pstr}	propellant-structure mass, kg
M_{tp}	turbopump mass, kg
M_{ts}	thrust-structure mass, kg
M_0	mass at beginning of maneuver, kg
$\dot{M}$	flow rate, kg/sec
P	pressure, atm
T	temperature
t	thickness of pressure shell, m
V	volume, m^3
V_∞	hyperbolic excess velocity, km/sec
ΔV	impulsive velocity increment, km/sec
ϵ	area ratio of nozzle, 300
ρ_{H_2}	density of hydrogen, 72 kg/m^3
ρ_{mod}	density of moderator material, 1150 kg/m^3
ρ_{ps}	density of pressure-shell material, 8000 kg/m^3
σ	allowable stress in pressure shell, 13 600 atm (1.378×10^9 N/m^2)

Subscripts:

c	core
H_2	hydrogen
i	ith maneuver
i_{max}	last maneuver
U	uranium

REFERENCES

1. Ragsdale, Robert G.: Relationship Between Engine Parameters and the Fuel Mass Contained in an Open-Cycle Gas-Core Reactor. Transactions of the Symposium at the University of Florida. NASA SP-236, 1971.

2. Ragsdale, Robert G.: Some Fuel Loss Rate and Weight Estimates of an Open-Cycle Gas-Core Nuclear Rocket Engine. Nasa TM X-52775, 1970.

3. Hesse, J.: Orbital Mission Analysis - Nuclear Light Bulb Propulsion. Rep. D-110280-1, United Aircraft Research Lab., Nov. 24, 1965.

4. Carstens, J. P.: Mission Capabilities of Advanced Nuclear Rockets. Rep. D-110280-2, United Aircraft Research Lab., Dec. 13, 1965.

5. Titus, R. R.; and Hesse, J. E.: Performance and Economic Implications of the Gaseous-Core Nuclear Rocket for Near-Earth Operation. Rep. UAR-J119, United Aircraft Research Lab., May 20, 1970.

6. Fishbach, Laurence H.; Giventer, Lawrence L.; and Willis, Edward A., Jr.: Approximate Trajectory Data for Missions to the Major Planets. NASA TN D-6141, 1971.

TABLE I. - HYPERBOLIC VELOCITY

	Hyperbolic excess velocity, V_∞, km/sec			
	Earth escape	Arrival	Departure	Earth return
Lunar ferry	0	1.414	1.414	0
Slingshot	V_{∞_1}	V_{∞_1} + 3.22	------	---
Mars:				
200 days	8.212	11.691	20.224	0
300 days	5.223	8.710	12.137	
400 days	3.477	5.392	6.705	
500 days	2.615	2.510	5.506	
600 days	6.344	3.537	6.055	
Jupiter:				
580 days	19.23	28.95	31.69	
1020 days	11.25	14.91	14.91	
1420 days	9.207	18.52	18.52	

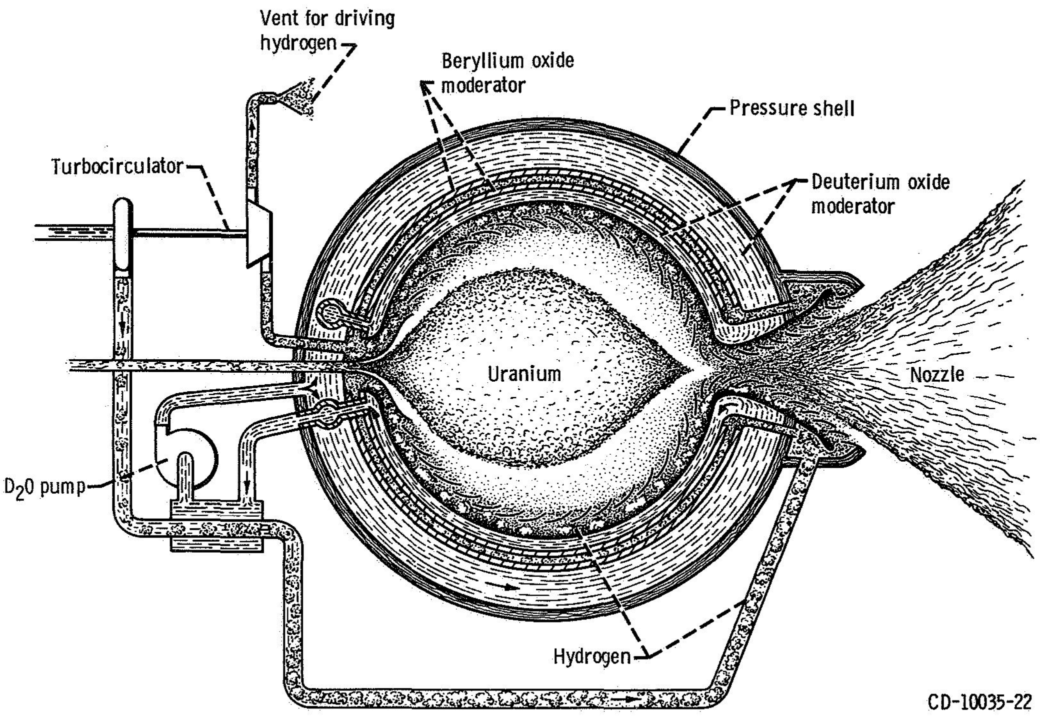


Figure 1. - Regeneratively cooled gas-core nuclear rocket.

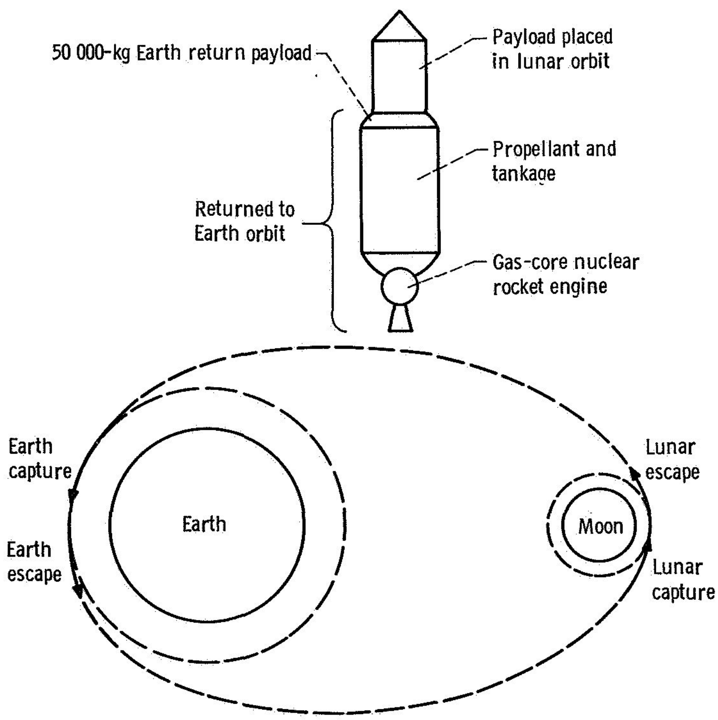


Figure 2. - Lunar ferry mission.

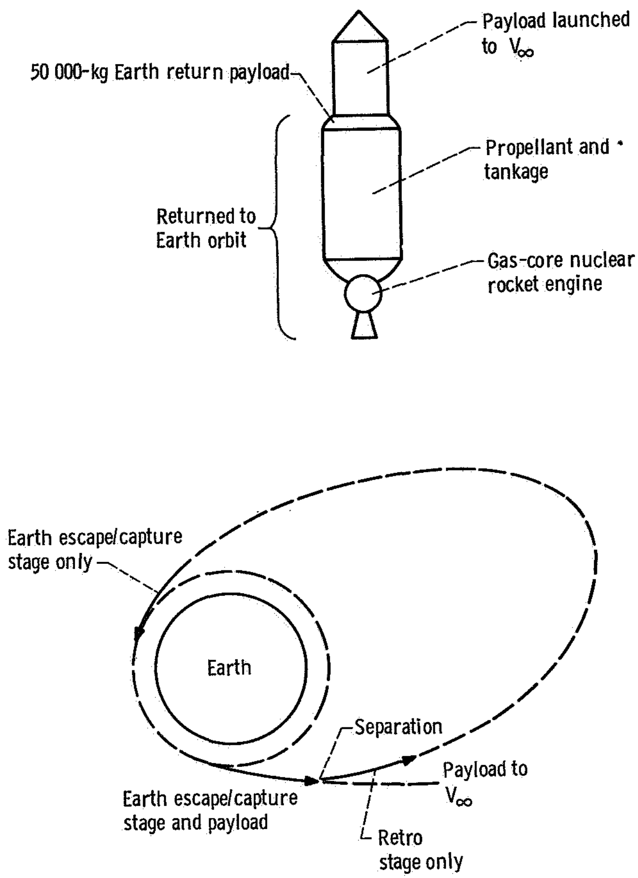


Figure 3. - Slingshot mission.

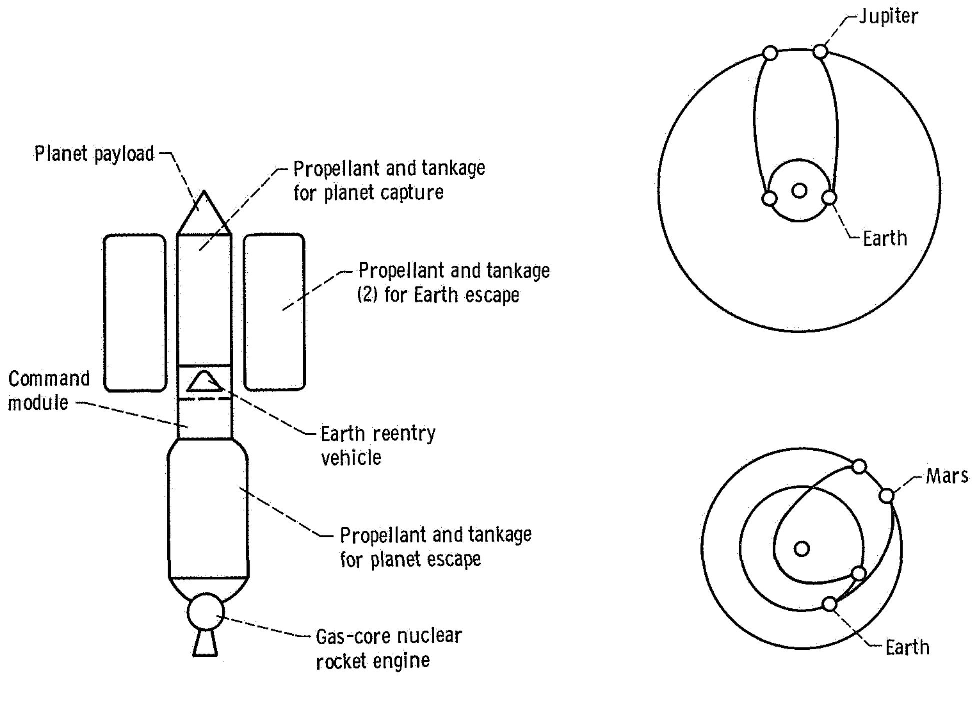


Figure 4. - Manned interplanetary missions.

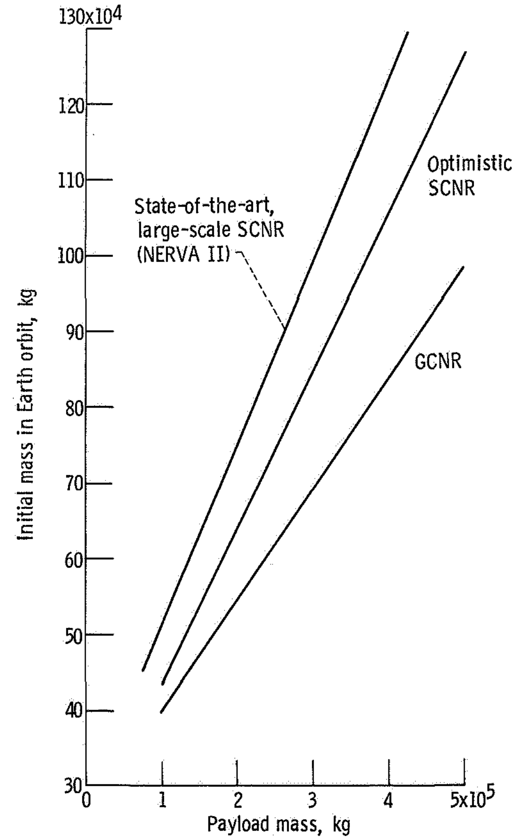


Figure 5. - Effect of payload mass on lunar ferry mission. Gas-core-nuclear-rocket diameter, 3.66 meters; critical mass, 48 kilograms; hydrogen- to uranium-flow-rate ratio, 100; specific impulse: GCNR, 2000 seconds; SCNR, 825 seconds.

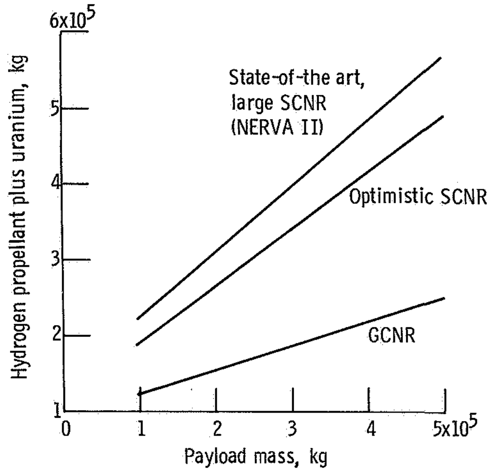


Figure 6. - Propellant requirements for lunar ferry mission. Gas-core rocket diameter, 3.66 meters; critical mass, 48 kilograms; hydrogen- to uranium-flow-rate ratio, 100.

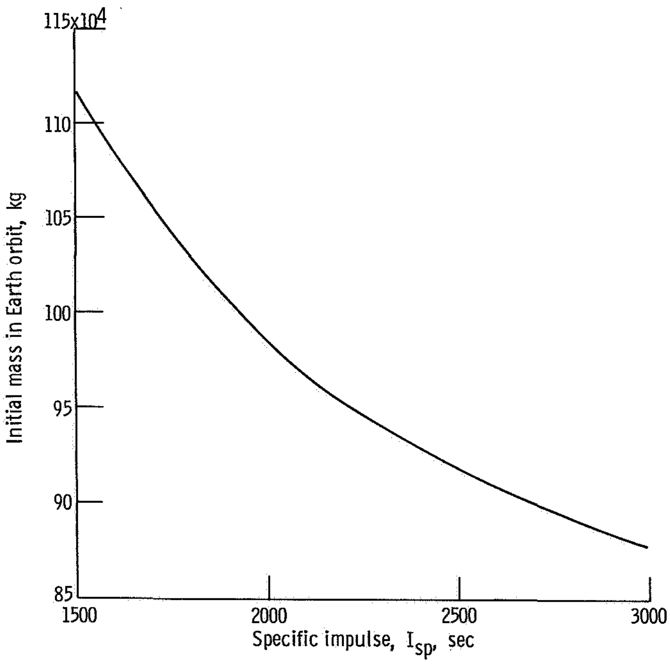
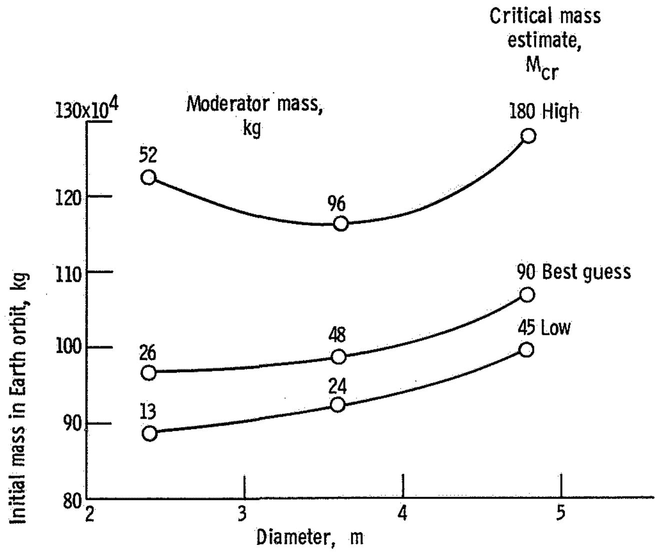
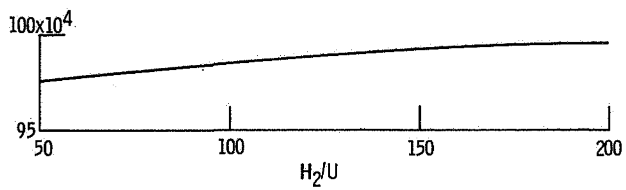


Figure 7. - Effect of specific impulse on lunar ferry mission. Gas-core nuclear rocket: diameter, 3.66 meters; critical mass, 48 kilograms; hydrogen- to uranium-flow-rate ratio, 100; payload, 500 000 kilograms.



(a) Effect of diameter, H_2/U = 100.



(b) Effect of flow-rate ratio. Gas-core engine diameter, 3.66 meters.

Figure 8. - Effect of core diameter and hydrogen- to uranium-flow-rate ratio on lunar ferry mission. Specific impulse, 2000 seconds; payload, 500 000 kilograms.

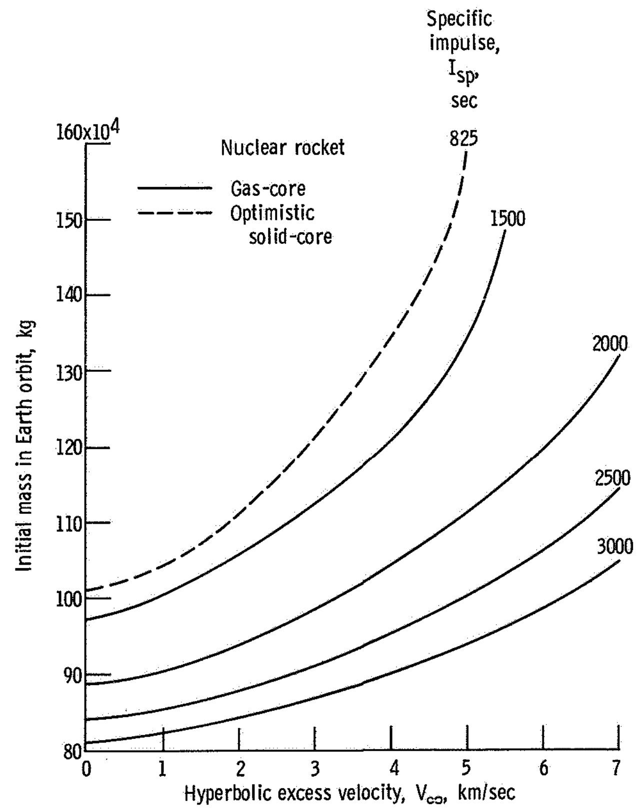
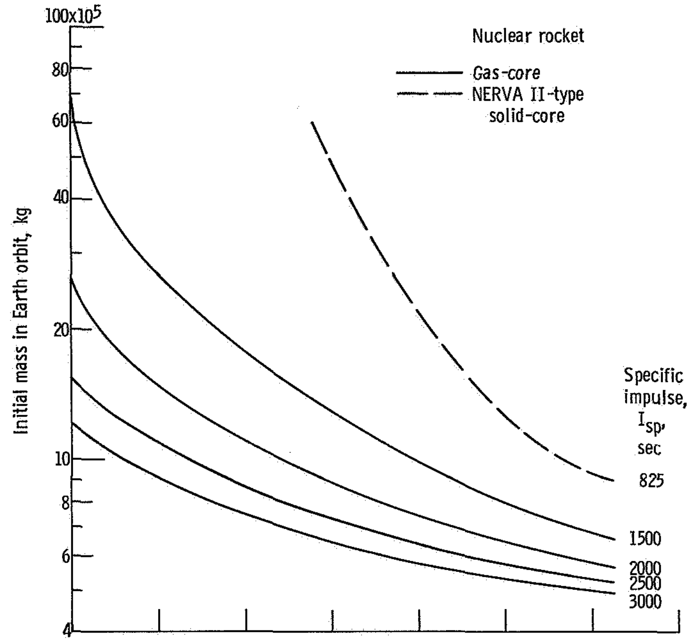
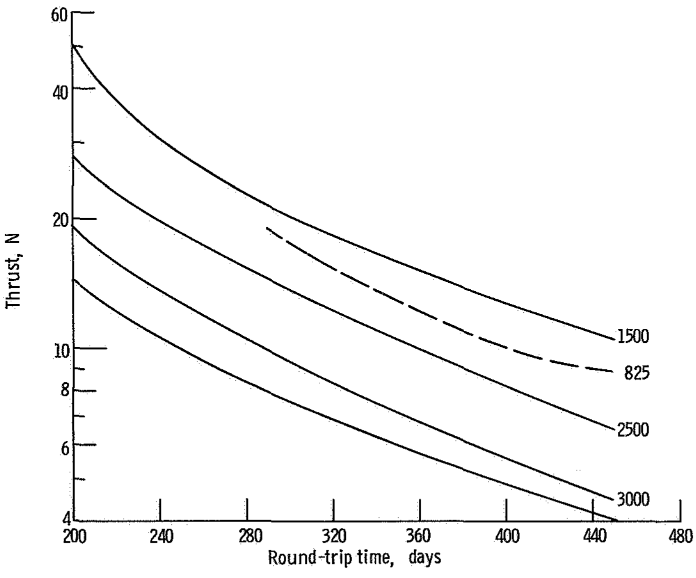


Figure 9. - Slingshot mission to various hyperbolic excess velocities. Gas-core-nuclear-rocket diameter, 3.66 meters; hydrogen- to uranium-flow-rate ratio, 100; critical mass, 48 kilograms.

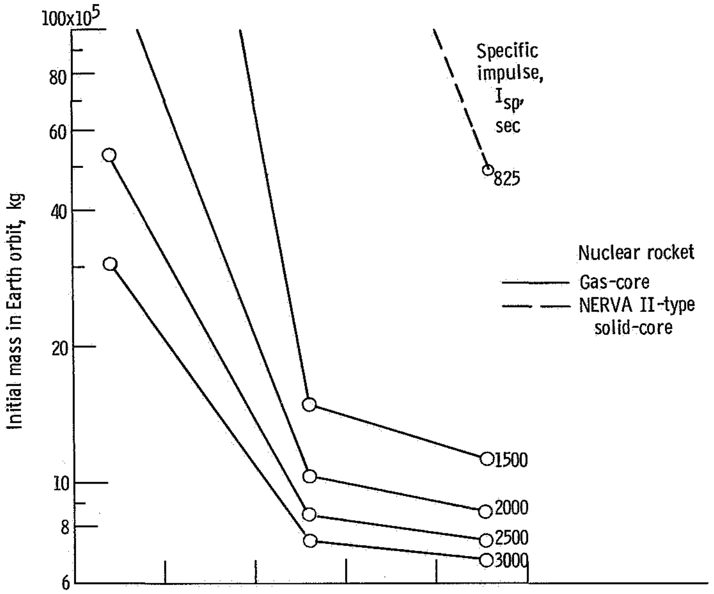


(a) Initial mass in Earth orbit as function of trip time.

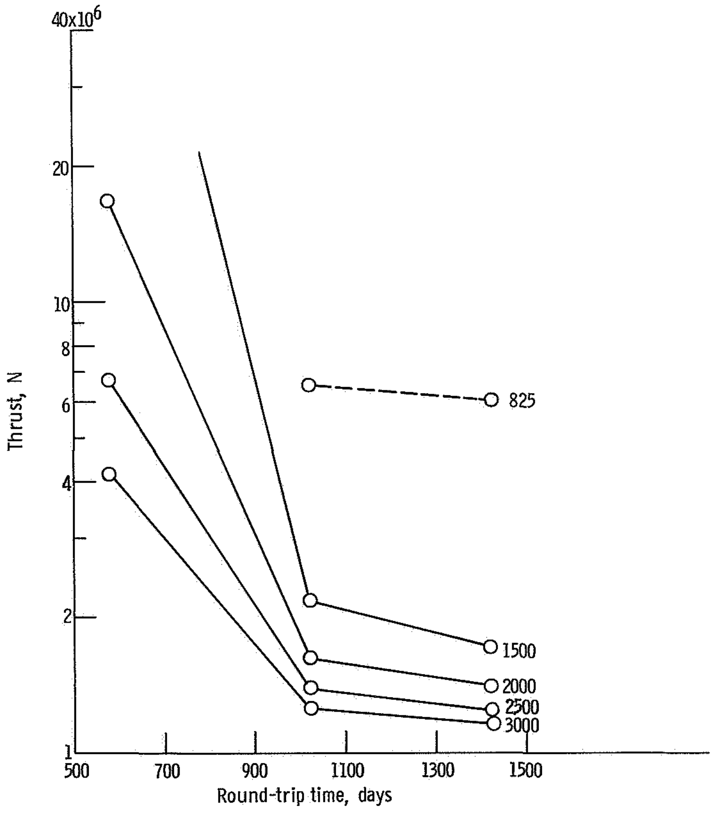


(b) Thrust as function of trip time.

Figure 10. - Manned round trip to Mars. Two-dimensional circular coplanar orbits; 1.1-Earth-radii circular parking orbit to 0.9-eccentricity parking orbit at Mars (periapsis at 1.1 Mars radii); stay time, 40 days. Gas-core nuclear rocket: diameter, 3.66 meters; critical mass, 48 kilograms; hydrogen- to uranium-flow-rate ratio, 100; payload, 136 100 kilograms to Mars and 90 700 kilograms more back to Earth.



(a) Initial mass in Earth orbit as function of trip time.



(b) Thrust as function of trip time.

Figure 11. - Manned round trip to Jupiter. Two-dimensional circular coplanar orbits; 1.1-Earth-radii circular parking orbit to 0.9-eccentricity parking orbit at Jupiter (periapsis at 1.1 Jupiter radii); stay time, 100 days. Gas-core nuclear rocket: diameter, 3.66 meters; critical mass, 48 kilograms; hydrogen- to uranium-flow-rate ratio, 100; payload, 136 100 kilograms to Jupiter and 90 700 kilograms more back to Earth.

NASA-Langley, 1971 —— 22 E-6092

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

— NATIONAL AERONAUTICS AND SPACE ACT OF 1958

NASA SCIENTIFIC AND TECHNICAL PUBLICATIONS

TECHNICAL REPORTS: Scientific and technical information considered important, complete, and a lasting contribution to existing knowledge.

TECHNICAL NOTES: Information less broad in scope but nevertheless of importance as a contribution to existing knowledge.

TECHNICAL MEMORANDUMS: Information receiving limited distribution because of preliminary data, security classification, or other reasons.

CONTRACTOR REPORTS: Scientific and technical information generated under a NASA contract or grant and considered an important contribution to existing knowledge.

TECHNICAL TRANSLATIONS: Information published in a foreign language considered to merit NASA distribution in English.

SPECIAL PUBLICATIONS: Information derived from or of value to NASA activities. Publications include conference proceedings, monographs, data compilations, handbooks, sourcebooks, and special bibliographies.

TECHNOLOGY UTILIZATION PUBLICATIONS: Information on technology used by NASA that may be of particular interest in commercial and other non-aerospace applications. Publications include Tech Briefs, Technology Utilization Reports and Technology Surveys.

Details on the availability of these publications may be obtained from:

SCIENTIFIC AND TECHNICAL INFORMATION OFFICE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D.C. 20546