Advanced Space Mission Capabilities of Nuclear Rockets

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In this paper possible trends in the application of the solid-core, heat-exchanger type of nuclear rocket propulsion are indicated. Engine performance parameters, such as the propellant feed cycles, specific impulse, nozzle length, area ratio, and engine flow rate, are discussed briefly. Potential space missions are covered on the basis of the typical ranges of velocity (from orbital launch) required. Results are presented parametrically, permitting the reader to determine engine thrust, I_{sp}, duration, and startup weight or delivered payload for any specific mission or velocity requirement. Performance data are presented in terms of hyperbolic excess velocity and actual velocity above escape. The results suggest that the best thrust levels for nuclear propulsion used for third and second stages on a Saturn V vehicle are in the ranges 0.2–0.3 and 0.8–1.0 million lb, respectively. These conclusions are based on both engine duration and payload achievable for a broad range of missions.

Introduction

THE Space Nuclear Propulsion Office (SNPO) was established in 1960 to direct nuclear rocket propulsion activities for the Atomic Energy Commission and NASA. The government soon decided to select an industrial engine contractor to direct and control, under SNPO guidance, the development of a useful and reliable flight propulsion system. This program started in July 1961 with the award of the NERVA (Nuclear Engine for Rocket Application) contract to the Aerojet-General Corporation, and with the Astro-Nuclear Laboratory of the Westinghouse Electric Corporation (WANL) as the principal subcontractor responsible primarily for the flight reactor development. The early nuclear vehicle stage would be a third stage of the Saturn V.

Figure 1 shows the mockup of the prototype NERVA, and Fig. 2 is a schematic of its propellant (hydrogen) flow path. The major subsystems are the propellant feed system, the nuclear system, the thrust chamber assembly (including thrust structures, the pressure vessel, the gimbal for thrust vector control, and the nozzle), the engine control system, and the pneumatic system. The cycle shown by Fig. 2 is the "hot-bleed cycle," which derives its name from the point (and hence the temperature) at which hydrogen is extracted to drive the turbopump. Liquid hydrogen is first pumped through the cooling passages of the main thrust nozzle, then passes up through the reflector and shield, and down through the reactor core. At this point, the hot bleed (a small percentage) is drawn off, diluted, and cooled with liquid hydrogen prior to ducting it to the turbine. The turbine discharge gas is ducted overboard through suitable nozzles to provide vehicle roll control. The main flow from the reactor is, of course, discharged through the propulsive nozzle.

The primary objective of the NERVA program is to demonstrate the capability of the engine to perform safely and reliably in the space environment. Operational mission influences have been considered of secondary importance, because they might slow the initial development by placing undue emphasis on performance and operational flexibility. However, the ultimate justification for the development of nuclear rocket propulsion must be related to the national space objectives. Quoting Finger,[1] the manager of SNPO, ". . . (it is) when we consider the possibilities of extensive exploration of the moon, of bases and scientific outposts on the moon, and manned missions to the near planets, that the full nuclear-rocket potential becomes apparent." The purpose of this paper is to indicate possible trends in the application of nuclear rocket propulsion systems of the solid-core heat-exchanger type in the years ahead.

Engine Performance Parameters

Three mission categories of immediate interest are: lunar requirements, manned and unmanned missions to the nearer planets, and deep-space probes. Included in the lunar missions are the logistics requirement, perhaps a backup for the Apollo/LEM mission, and, ultimately, a direct-landing vehicle. The possibility of accomplishing such missions by various modes of operation with the nuclear rocket now in development which uses the Saturn V booster is being investigated. The selected modes will indicate propulsion system requirements, such as number of restarts, duration of each thrust period, coast times, accelerations, thrust termination requirements, orbital as compared with suborbital starts, staging sequences, and man-rating requirements. The first step is to define operational requirements for a NERVA-class rocket for these applications. Several arbitrarily selected NERVA modifications are being studied as possible operational propulsion systems. The performance capabilities of each is assessed by trajectory and mission analyses, and propulsion system specifications are prepared for the particular configuration that provides the performance for a given mission (or missions). If such mission-dictated requirements can be incorporated into the on-going engine development, then the engine provided for initial flight testing will also be capable of fulfilling selected mission applications at an early date.

Preliminary analyses have indicated that one of the NERVA modifications, used with Saturn V, can provide useful payloads for all of the initial missions of interest. For the lunar missions, for example, using the least complex mode of operation, the nuclear stage on Saturn V will place at least 30% more payload into the lunar trajectory than will its chemical counterpart. Even more significant gains can be realized for Mars or Venus flyby and orbiting missions. Mars orbital payloads of the order of 25,000 lb are obtainable.

It is to be expected that still greater payloads of a more advanced nature will be required in the future, using nuclear propulsion in conjunction with the Saturn V and NOVA-class boosters. This further growth could take several courses, but two in particular require examination. First, it would be reasonable to consider exploiting the NERVA/Kiwi technology to the fullest and to postulate a second-generation

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system as a growth version of NERVA, say, a NERVA II. This could well be the most logical, straightforward, and economical approach. This second course would require the examination of alternate reactor concepts that may offer possible system performance improvements. They must be considered for possible future propulsion system applications, particularly if performance and economic advantages can be realized. Possibilities in this advanced area include the thermal reactor (of the type used in NERVA), the fast neutron solid reactor, the fluid core reactors, and possibly other advanced concepts. The thermal reactor may have the fissionable material mixed directly with the moderating material and may be operated at as high a temperature as possible, commensurate with the physical properties of the materials. This concept has been employed in the Kiwi reactor, and considerable experience has been gained in its use. Knowledge of the material properties and characteristics is extensive.

For the various reactor concepts, several alternates with regard to the propellant feed system are available (see Figs. 2 and 3). The design of the propellant feed system is significant, because it influences engine performance, over-all engine configuration, major subsystem configurations, and many of the individual component specifications. The "topping cycle" (Fig. 3c) is characterized by a series flow of the propellant from the tank through the pump, through the coolant passages of the nozzle and the reactor reflector, into the low-pressure-ratio turbine, and finally into the reactor core. Highest specific impulse is achievable with this cycle for any given reactor-core exit temperature. A complication exists, however, because special reactor design features must be provided to raise the temperature of the turbine working fluid to a level where the pump power requirements can be satisfied. In the bleed cycles (Figs. 2, 3a, and 3b), a small portion of the hydrogen propellant flow is used as the turbine working fluid; the location at which this bleed is effected determines the temperature of the working fluid and, hence, the turbine performance and over-all engine performance.

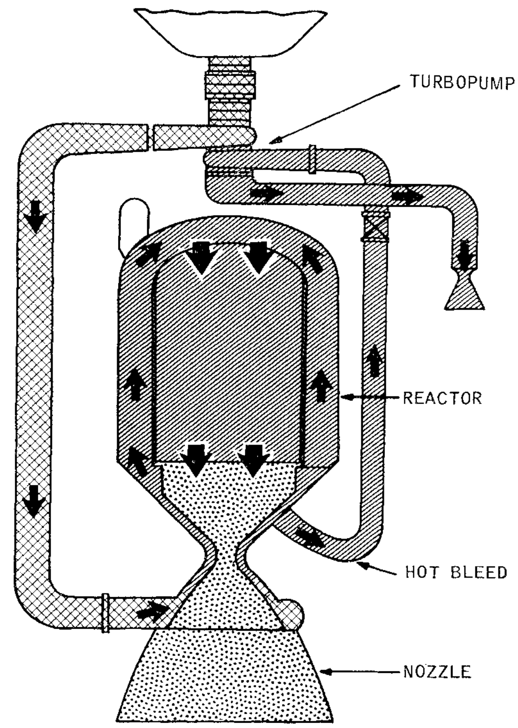

Fig. 2 NERVA engine flow schematic.

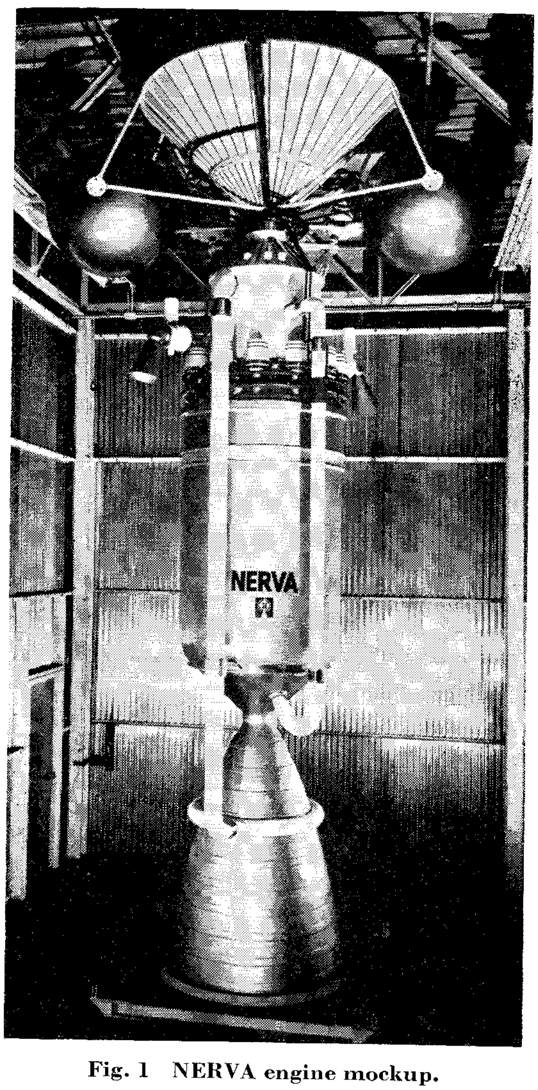

Fig. 1 NERVA engine mockup.

Because it is rational to assume that the second-generation nuclear rocket propulsion system will be based on the NERVA/Kiwi technology, an advanced solid-core graphite thermal reactor is postulated for NERVA II, and (because of the performance gains that can be realized) the topping cycle concept should be strongly considered. With this tentative selection made, it is necessary to define the size and performance characteristics that may be of interest. Computed engine performance and other data are used in the development of mission, economic, and schedule tradeoff analyses; then, from an evaluation of the mission studies, objective and performance requirements are established, and preliminary design begins. Iterations of the mission performance and design activity produce increasingly refined definitions and specifications. For these purposes, parametric performance data for a rather wide range of engine sizes have been prepared.

A second concept is to cluster smaller nuclear engines to provide performance equivalent to that of a larger single

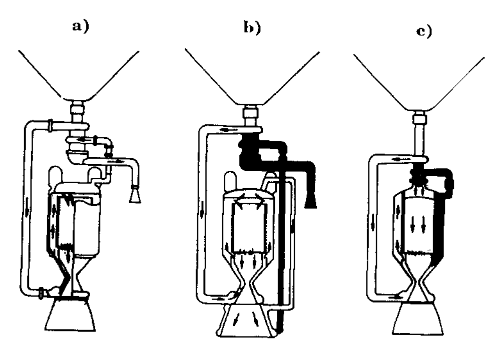

Fig. 3 Nuclear engine propellant feed cycles: a) cold-bleed cycle, b) heated-bleed cycle, and c) topping cycle.

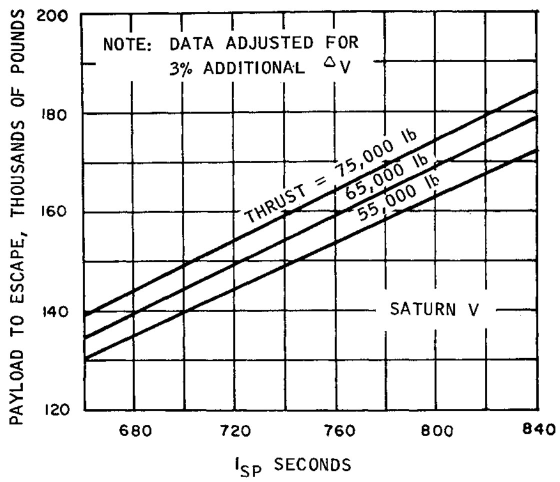

Fig. 4 Effects of nuclear engine I_{sp} and thrust on escape payload.

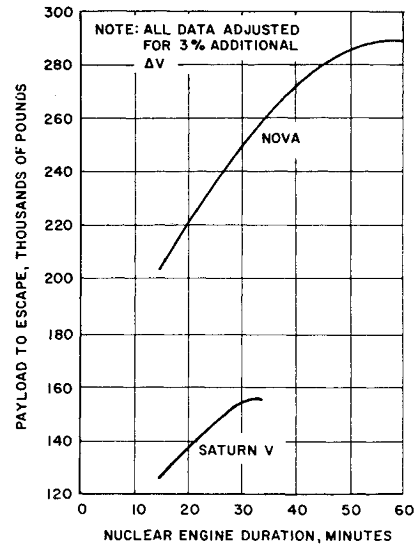

Fig. 5 Effects of nuclear engine duration on escape payload.

engine. The clustering of the NERVA engine (in development) can provide increased operational capability at a much earlier date than any other approach. However, the problems associated with engine clustering are far more difficult with nuclear engines than with chemical rockets, because the influences of the neutronic and radiation effects and interactions must be determined through extensive analysis. The relatively straightforward structural problem is also complicated by the unique combination of environments. The effects of engine clustering on radiation-induced heating rates in engine components and propellant tank must be evaluated, and the difficult problems associated with test facilities for clustered engines require serious attention.

Those responsible for planning space missions are always interested in placing heavier payloads into lunar and planetary trajectories, completing missions in shorter trip times, and minimizing the cost per pound of payload delivered. Two broad guidelines can be postulated within this framework: for missions in which the nuclear stage is started in Earth orbit, improved nuclear engine specific impulse is of primary significance; for suborbital start, greater gains can be realized by increasing thrust, with improved I_{sp} as a secondary consideration. For any given engine cycle, performance in terms of thrust and I_{sp} can be improved by increasing the rocket-nozzle exit-area ratio. Other parameters remaining the same, there is a tradeoff between performance on the one hand and there is nozzle weight on the other. Performance improvements become rather small for area ratios greater than, say, 50. Changing the area ratio from 10 to 50 results in about a 7% improvement in I_{sp}, but from 50 to 100, the additional increase is 2 to 3%. Since I_{sp} is generally proportional to the square root of exit gas temperature, other elements remaining constant, any development that allows an increase in the temperature of the gas leaving the nuclear core, such as the successful development of higher temperature materials or improvements in the temperature properties of the existing materials, is important. Thrust is directly proportional to engine propellant flow rate, if other parameters remain the same. To increase the flow rate, pumps, lines, valves, cooling passages, and appropriate methods for handling fluid flow and heat transfer must be designed and developed. Thrust also can be improved by increases in chamber pressure with attendant increases in engine structural weight. One might postulate nuclear rockets with as much as ten to twenty times the thrust levels of currently planned engines.

The foregoing discussion implied the isolation and identification of individual influences, but this rarely happens in practice. A change in one parameter can affect the configuration and characteristics of other engine components and subsystems. The system planner must judge whether or not the particular improvement in performance, in terms of development time and cost, is warranted.

Table 1 Payload gains with nuclear (NERVA) vs chemical (J-2) third stage

	Saturn V[a]		Nova[b]	
Third stage	W_{pl}, 10^3 lb	Gain, %	W_{pl}, 10^3 lb	Gain, %
Direct to escape				
Chemical	99	...	189	...
Nuclear	159	61	289	52
Escape via 100-naut-mile orbit				
Chemical	97	...	185	...
Nuclear	156	61	265	43
Escape with third-stage ignition in 100-naut-mile orbit				
Chemical	90	...	...	...
Nuclear	125	39	...	...
Two-stage to 100-naut-mile orbit	250	...	468	...

[a] Five F-1's plus five J-2's plus third stage.
[b] Eight F-1's plus four M-1's plus third stage; current design specifications call for W_{pl} = 1.0 Mlb in orbit.

Mission Capabilities

The first application of a nuclear engine will be in conjunction with the Saturn V vehicle in place of the S-IV stage. Its prime purpose will be to lift a payload to escape velocity. Other advanced missions can be foreseen where the payload capability of the chemical/nuclear propelled system exceeds that of an all-chemical system by more than 50%. A study of the performance, operational cost, and reliability of the Saturn V and NOVA launch vehicles with nuclear stages has permitted comparison with present and future all-chemical systems. Beyond the first ballistic test of the NERVA engine in a flight test program lies a spectrum of operational applications for the present nuclear engine and its successors. A converging two-pronged analytical approach is being taken at present: 1) evaluation of the currently anticipated capability of the NERVA-class engine, as installed in a suitable and available launch vehicle, and 2) prediction of the needs for advanced space nuclear engines and launch vehicles in a suitable time era. These two factors are combined to define the need and capability for nuclear engines.

Table 2 Mission velocity requirements

Destination	Burnout velocity, fps	Trip time, days
Moon	35,500	3.0
	35,550	2.5
	35,750	2.0
	36,100	1.5

	Hyperbolic velocity, fps		
Destination	Minimum energy[4]	NASA[5]	Calendar year
Mars	10,000	14,700	1971
	14,000	20,000	1975
	13,500	18,000	1979
Venus	10,500	...	1971
	9,000	...	1975
	11,000	14,500	1979

Saturn and NOVA systems utilizing chemical vs nuclear propulsion for the third stage have been compared with respect to payload capability, performance, cost, stage and mission reliability, and schedule. Only the performance comparisons are present in this study. The basic configurations studied were the Saturn V with five F-1's, five J-2's, and one J-2, and the liquid NOVA with eight of 14 F-1's, four M-1's, and one J-2. Recent NASA literature was surveyed for potential NASA launch-vehicle configurations, and a number of potential modifications for Saturn V and NOVA were also postulated. Since Saturn upper-stage designs are in the early phase of evaluation, suggestions as to potential changes in stage thrust levels were considered appropriate.[2]

The dependency of payload on nuclear thrust and specific impulse I_{sp} is illustrated in Fig. 4. The effect of varying the exhaust temperature (and hence the I_{sp}, flow rate, and net thrust) on the payload-to-escape can easily be seen. The following performance calculations were based on a typical I_{sp} of 800 sec. This chart shows that an increase of 100 sec in I_{sp} yields a payload increase of 25,000 lb, whereas a thrust increase of 20,000 lb gives a payload gain of 8000 lb.

A performance summary comparing all chemical and chemical/nuclear Saturn V and NOVA launch vehicles is shown in Table 1, which lists the payload capabilities for the following three modes of operation: 1) direct-to-escape with suborbital start of the third stage, 2) direct-to-escape with the third stage "flying" through a 90° attitude or orbital position during its run to escape velocity, and 3) two-stage chemical burn to a 100-naut-mile orbit from which the final stage is started. These calculations were obtained from a two-dimensional point-mass trajectory program (see Appendix). The penalty for the last case is a 10% reduction in payload compared to direct flight condition. The Saturn V chemical/nuclear direct flight yields approximately a 61% payload improvement over the all-chemical Saturn V; for NOVA, the increase is smaller (52%), primarily because of the reduced thrust of the nuclear stage relative to the first-stage vehicle. Structural weights for the latter vehicle were based upon conservative estimates for the time period of interest. Weights analyzed show propellant fractions ranging between 0.91 and 0.93 for the chemical stages. Payloads of 0.25 Mlb (millions lb) for the Saturn V and 0.47 Mlb for the NOVA will be the basic weights in orbit to be used for later comparisons of payload delivered to escape, as based upon the use of families of nuclear engines ranging in thrust from 0.05 to 2.0 Mlb.

The effect of nuclear engine duration on payload-to-escape is illustrated in Fig. 5 for Saturn V and NOVA vehicles. Each point on these curves represents a three-stage propellant optimization program to obtain nearly optimum payload for that specific nuclear duration. (It is assumed that optimum thrust level can be achieved at the assumed I_{sp} = 800 sec in all cases; this is an oversimplification, of course.) The best duration for Saturn V is about 32 min, whereas NOVA, because of its increased second-stage burnout weight, requires about 55-min duration in order to yield peak payload to escape. The payload penalties paid for shorter nuclear durations must be measured against development cost and time required to obtain longer (near-optimum) durations.

The various advanced operational modes that could utilize nuclear propulsion on either the Saturn V or NOVA launch vehicles include two- and three-stage launchings to orbit, escape, and beyond. No attempt is made to define required payloads (future studies will determine them in light of specified national space objectives), and for the present purpose, fixed payloads of 0.15 and 0.3 Mlb are considered. This payload range should permit an evaluation of a system capability to accomplish Venus and Mars round trips. With a fixed payload as the base, the total mission velocity requirement can be varied and then the needed payload rendezvous in low Earth orbit can be estimated, from which the total launching, support facility, and program cost requirements can be estimated.[3]

The results from the orbital-launch portion of the study are used in conjunction with those from the Earth-launch portion. The payloads delivered to a 100-naut-mile orbit are used to obtain payload delivered to escape velocity with a variety of potential nuclear engines. Each combination of nuclear engine and orbital launch weight is flown to various V_∞ and ΔV_T, and corresponding propellant weights and payloads are obtained.

The term V_∞ may be explained as follows. A vehicle launched from Earth orbit shuts down at some velocity and altitude; as it coasts, the Earth's gravitational field causes a small but finite slowdown. V_∞ is defined as the velocity of the vehicle (relative to Earth orbit) *after* an infinite length of coast. ΔV_T is defined as the actual velocity increment above the vehicle's initial escape speed at cutoff. V_∞ is independent of altitude and relates to the various potential space missions: 1) $V_\infty = 0$ is a lunar mission, 2) V_∞ = 10–20 kft/sec is a Venus or Mars mission, and 3) V_∞ = 30–50 kft/sec is a deep-space probe. A more specific mission velocity requirement table based on calendar year of operation and data obtained from Refs. 4 and 5 is shown in Table 2. The relationship between the ratios of propellant weight to startup weight and thrust to startup weight are shown in Fig. 6 for various V_∞'s. Using this figure in conjunction with an estimate of propellant mass fraction for the stage, one may find the payload and propellant requirements at a V_∞ for any combination of engine thrust and orbit startup weight. The relatively small penalty imposed by a low thrust-to-weight ratio can readily be seen; a change from F/W_0 = 0.25–0.05 causes only an 18% decrease in payload for a lunar mission ($V_\infty = 0$).

The comparative payload capabilities of the J-2 engine and a family of potential nuclear engines started up in a 300-naut-

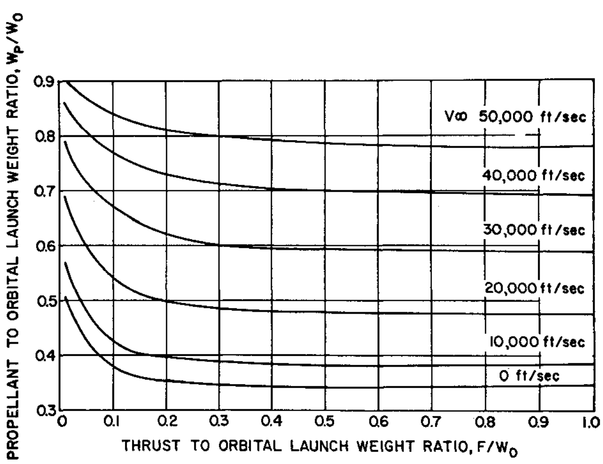

Fig. 6 Effects of nuclear thrust-to-weight ratio on propellant requirements.

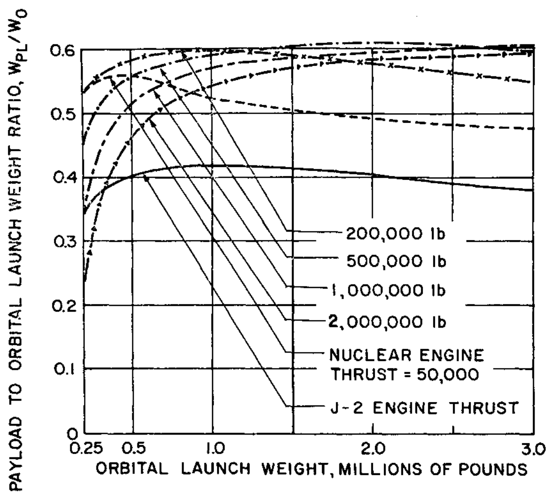

Fig. 7 Payload vs initial weight in orbit for various nuclear engine thrust levels.

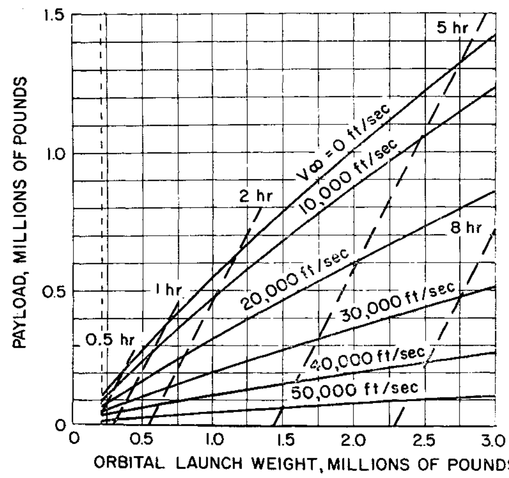

Fig. 9 Payloads for orbitally launched nuclear vehicles (thrust = 50,000 lb).

mile orbit and flown to escape velocity are shown in Fig. 7. Although the earliest operational application would be the use of NERVA on Saturn V at 0.05-Mlb thrust, greater efficiency and versatility could be obtained from the development of a nuclear engine in the 0.2-Mlb thrust range. The NERVA-class engine superiority over the J-2 shows a drop ranging from 55 to 25% as orbital weight is increased from 0.2 to 1.0 Mlb, but the larger (0.2-Mlb thrust) nuclear engine maintains its 50% payload superiority throughout the range. The still larger thrust engines give somewhat better (10%) payload capability over the 0.2-Mlb thrust engine for launch weights near 3.0 Mlb, but they are outside of present NERVA growth capability and would cost more to develop than assumed mission requirements permit.

Figure 8 illustrates the mission capabilities in terms of V_∞ of the various engines for a 0.15-Mlb payload, and Figs. 9 and 10 present a means of finding approximate operation times required for the NERVA-class vehicle to reach some desired V_∞ (or ΔV_T). A comparison of Figs. 9 and 10 shows the added advantages of increasing the nuclear engine thrust level to 200,000 lb. These figures can be used to quickly evaluate mission capability from a low orbital altitude starting condition. Using Table 2 to obtain a required V_∞ and assuming a weight in orbit, the nuclear duration and payload delivered to the desired V_∞ can be obtained quickly. In Fig. 11, such results are cross plotted as payload-to-launch weight ratio vs nuclear engine thrust level; groups of curves are presented for V_∞'s of 0, 20, and 40 kft/sec to cover the three basic space missions; lunar, Mars/Venus, and space probes. Assuming a mission velocity and a weight in orbit, the best engine thrust to achieve the peak payload can be determined. The payload penalty also can be quickly obtained for thrust-level changes at any value of weight in orbit. Figure 12 summarizes the payload vs thrust results for various Saturn V flight modes. The two left-hand curves are for Saturn V with a third nuclear stage; curve A is for direct-to-escape with suborbital start, and curve B is for escape from a 300-naut-mile orbit. The three right-hand curves are for a two-stage (Saturn V booster and nuclear stage) launch vehicle. Curve C yields payload to a 300-naut-mile orbit with a thrust range of second-stage nuclear engine from 0.45 to 1.6 Mlb, and curves D and E are for direct-to-escape with suborbital and orbital start, respectively. It can be seen that for two-stage applications, the nuclear thrust level must be at least 0.45 Mlb and that the greatest incremental payoffs are obtained by jumping to the 0.8–1.0-Mlb range.

Figure 13 presents relations between nuclear engine duration and thrust level for each of three operational modes.

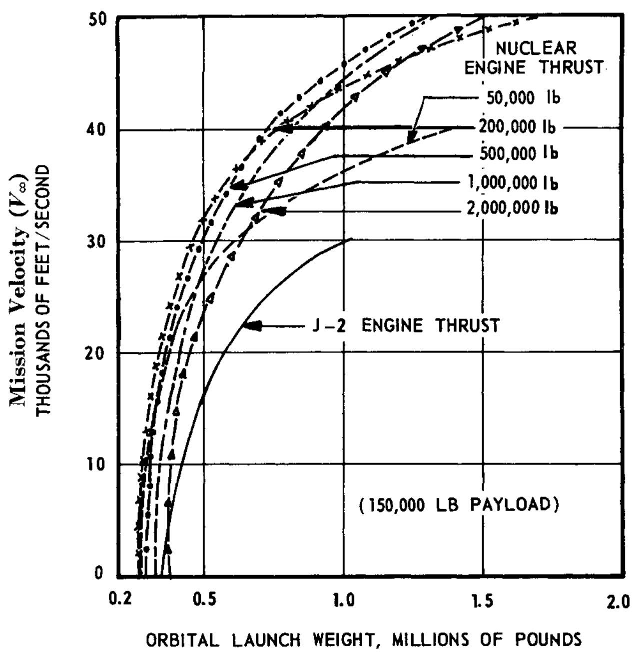

Fig. 8 Initial weight in orbit to deliver vs mission velocity for various nuclear engine thrust levels.

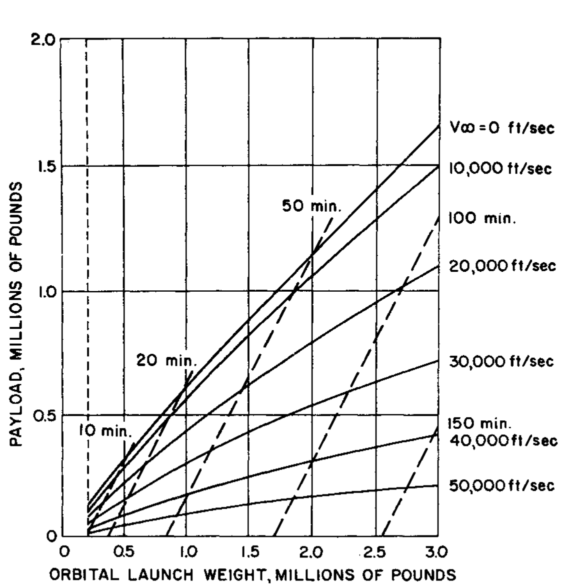

Fig. 10 Payload for orbitally launched nuclear vehicles (thrust = 200,000 lb).

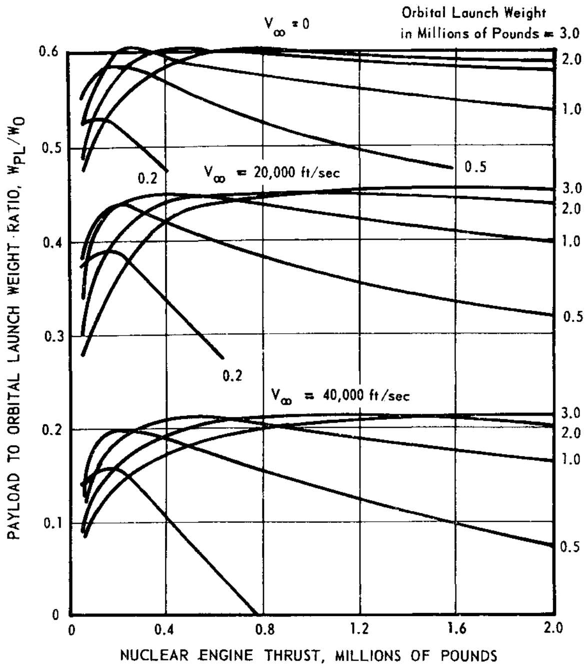

Fig. 11 Optimum thrust for orbitally launched vehicles.

The bandwidths represent the duration variation for operational velocity goals of $V_\infty = 0$ to $V_\infty = 40$ kft/sec. At very low nuclear thrust levels launched with a two-stage chemical system, the nuclear engine duration required lies between 20 and 50 min. A rapid decrease in required duration is noted as the thrust increases to 0.2 Mlb. From this thrust level onward, the time reduction becomes less and less significant. This results from a boot-strap type of flight in which the engine weight and associated tankage and extra propellant requirements offset the gain obtained from the increased thrust. The orbital startup shown on band 3 can utilize a nuclear engine of 0.1 to 0.2 Mlb; there is little, if any, gain obtained with nuclear engines above 0.3-Mlb thrust. Conclusions from this duration chart are: 1) for best reduction in third-stage duration, a nuclear engine in the 0.1- to 0.3-Mlb range is best, and 2) the two-stage launch vehicle system has a desirable duration level at 0.8- to 1.0-Mlb thrust.

Ranges of nuclear engine thrust levels which can be employed to obtain maximum payload capability are illustrated

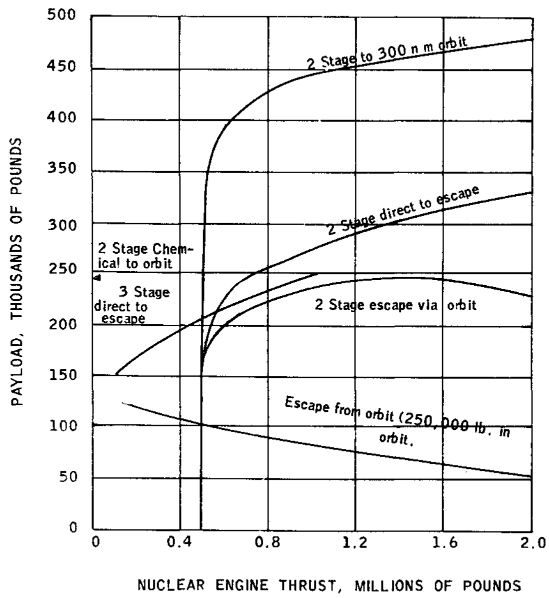

Fig. 12 Payload capabilities of Saturn V for various nuclear engine thrust levels.

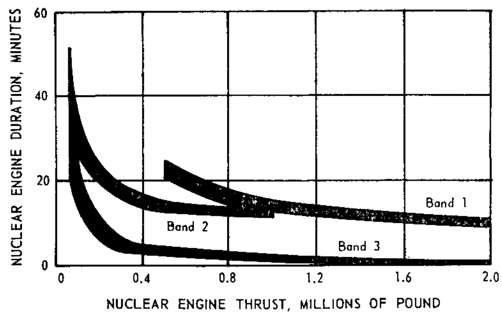

Fig. 13 Nuclear engine duration requirements. Band 1: First-stage chemical, second-stage nuclear to $V_\infty = 0$, 20,000, and 40,000 ft-sec, both direct and via orbit. Band 2: First- and second-stage chemical, third-stage nuclear to $V_\infty = 0$, 20,000, and 40,000 ft-sec, direct. Band 3: Same as band 2 with nuclear engine starting up in orbit.

in Fig. 14; for each of the combinations of configuration and mission described on the left, the thrust level yielding the maximum payload is found. The information is obtained from data similar to that in Figs. 9–12, and it is presented in Fig. 14 by the solid triangles. The thrust ranges (band lengths) about these triangles yield payloads within 10% of the maximum. This then presents a tradeoff peak payload range for a large number of missions plotted against nuclear engine thrust. Bandwidths again illustrate V_∞ range, as shown by the lower inset in Fig. 14. This figure shows even more clearly that the two thrust levels that could satisfy the greatest number of potential missions are 0.2 to 0.3 Mlb and 0.8 to 1.0 Mlb. The latter could be obtained by either a new single-chamber engine or a cluster of the earlier engines.

Appendix: Two-Dimensional Trajectory Program

The trajectory program is based on a mathematical model of a point-mass, two-dimensional trajectory, which computes the forces tangential and normal to the velocity vector. It is programed on the IBM 7094 to determine optimum payload into orbit or to various hyperbolic velocities.

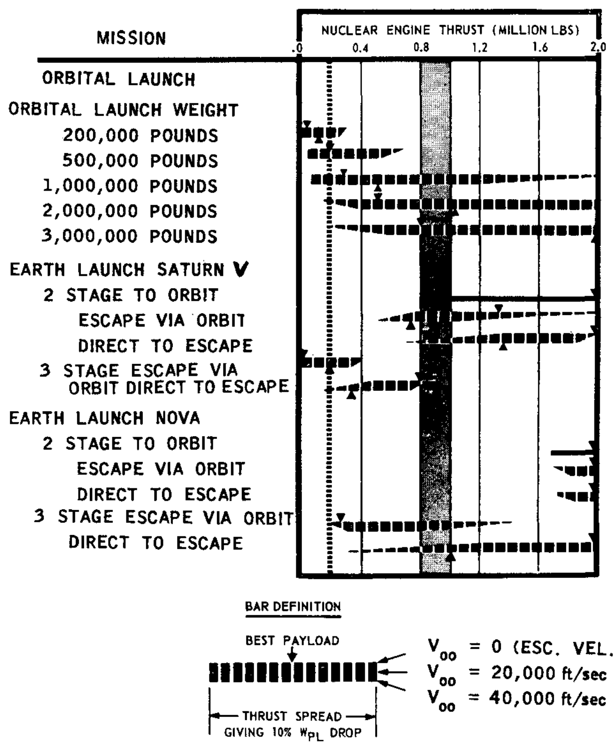

Fig. 14. Nuclear engine thrusts for optimum payload.

The technique of integrating the entire acceleration is known as Cowell's method,[6] and it is the preferred method for powered flight. The parameters are integrated by the Runge-Kutta[7] method, using formulas accurate to the fourth order of the time interval of integration. This method is used to integrate the four functions: gamma (the angle of flight path measured from the local vertical to the vehicle velocity vector), the beta (the angle subtending the length of arc on the surface of earth from launch vertical to the local vertical), altitude, and the instantaneous velocity of the vehicle. It is assumed that the Earth is a nonrotating spheroid incorporating the inverse square law of gravity relationship. The effect of the Earth's rotation on orbital and escape mission performance is approximated by vectorally incrementing the missile velocity with the appropriate components of Earth's rotational velocity at a preselected point above the atmosphere. Vehicle input characteristics are startup weight, number of stages, and the I_{sp} and all engine constants for each stage. Thrust, drag, and lift forces are computed in the conventional manner. Atmospheric parameters are incorporated in the program.

The angle program for each trajectory may employ options of the following types: constant angle of attack, constant attitude, constant pitch rate, and instantaneous attitude change or "kick." The angle programs can start or stop on selected values of problem variables, such as time, velocity, altitude, and angle of attack. The iterative feature of the program allows for varying the magnitude of an angle program, burning time of a stage, and coast time to achieve flight-path angle, altitude, circular orbit, velocity, or dynamic pressure at burnout of a stage.

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