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Nuclear Pulsed Space Propulsion Systems (U)

UNITED STATES
ATOMIC ENERGY COMMISSION
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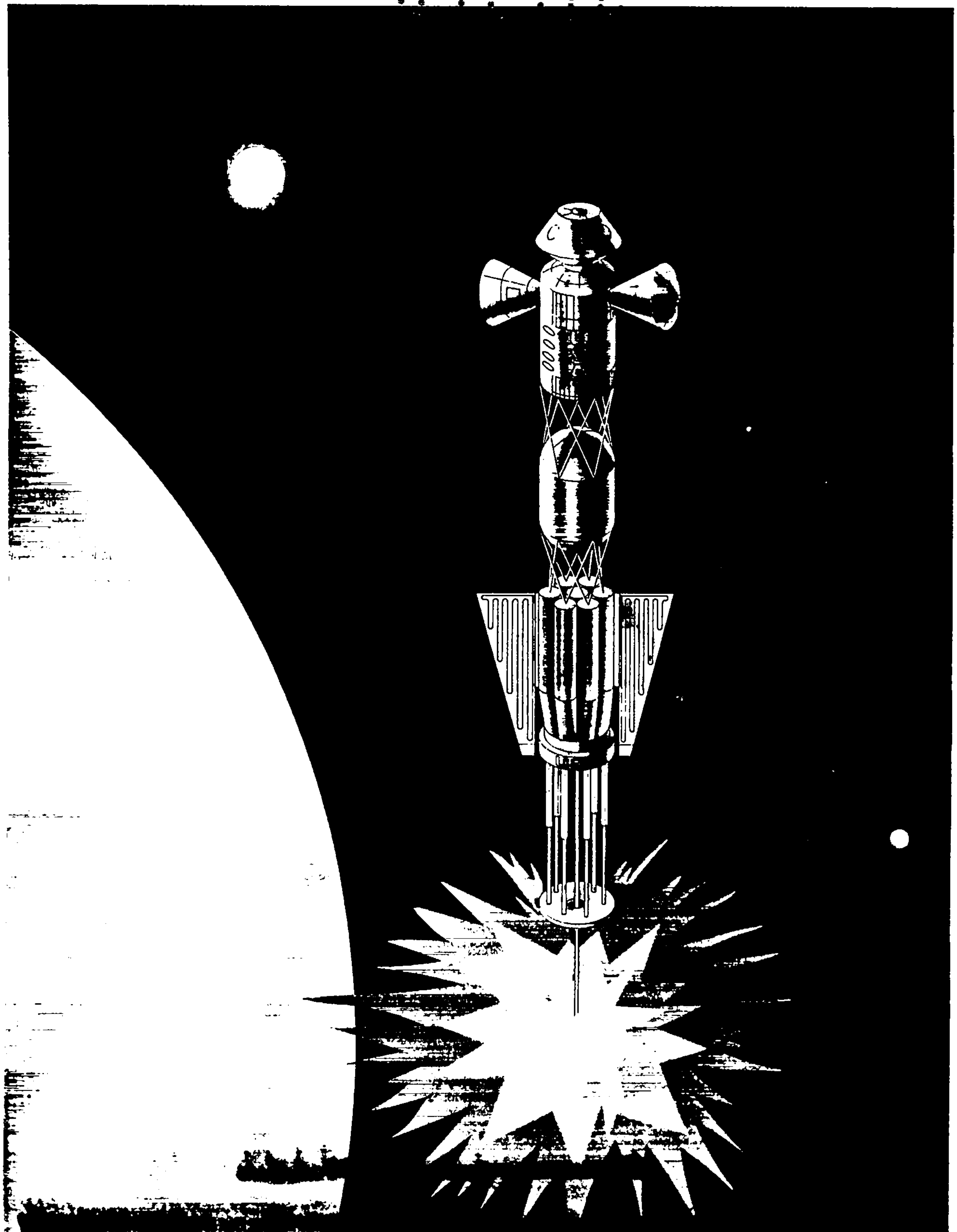

Artist's concept of nuclear-pulsed space propulsion vehicle.

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Appendices

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NOMENCLATURE

a_a	Pusher-plate acceleration, at end of mission, relative to the spacecraft, m/sec^2
a_o	Spacecraft acceleration at end of mission, relative to an inertial frame of reference, m/sec^2
b	$\sqrt{M_e/M_p}$, dimensionless
g_o	Standard acceleration of gravity, 9.8 m/sec^2
I	Impulse delivered to the spacecraft by one pulse unit, N-sec
I_{sp}	Specific impulse, $(I_{sp} = V_e/g_o)$, sec
m	Pulse-unit mass, kg
M	Total spacecraft initial mass, kg
M_a	Momentum-absorber mass $(M_a = M_e + M_p)$, kg
M_b	Initial spacecraft propellant mass, kg, consisting of n pulse units of mass m, $(M_b = nm)$
M_e	Momentum-conditioner mass, kg
M_o	Payload mass, kg. Any mass not designated otherwise; includes structure (other than M_a) and the laser unit, $(M_o = M - M_a - M_b)$.
M_p	Pusher-plate mass, kg
n	Number of pulse units, dimensionless
q	Pulse-unit energy release, J
U	Initial velocity of the pusher plate (just after the pulse-unit interaction) relative to the spacecraft, m/sec
V_e	Effective velocity of the propellant exhaust, m/sec
V_i	Average velocity of propellant impinging against the pusher-plate surface, m/sec
W	Material strength-weight constant defined by Eq. (6), m/sec
X_a	M_a/M, dimensionless
X_o	M_o/M, dimensionless
α	Dimensionless efficiency term defined by Eq. (5) for internal system
α'	Dimensionless efficiency term defined by Eq. (5) for external system
β	Collimation factor defined by Eq. (17), dimensionless
Δt	Time interval between pulse-unit detonations, sec
Δx	Relative distance of travel, or stroke, of the pusher plate relative to the spacecraft, m
ΔV	Mission-equivalent free-space velocity change, m/sec
η	Efficiency term defined by Eq. (2), dimensionless
μ	Dimensionless parameter for an energy scaling law, defined by Eq. (8)
μ'	Dimensionless parameter for an impulse scaling law, defined by Eq. (15)

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NUCLEAR PULSED SPACE PROPULSION SYSTEMS

by

J. D. Balcomb
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ABSTRACT

Initial considerations are presented for advanced space-propulsion concepts that are based on energy production by laser-driven thermonuclear pulses. Preliminary design concepts are compared in which an individual pulse unit, located either internally or externally to the system, imparts an impulse to the spacecraft. Many pulse units are sequentially discharged and initiated until the desired spacecraft velocity is reached. Various means of shock absorption and shielding against fast neutrons generated by the thermonuclear reactions are investigated. Indications are that the maximum specific impulse for an internal system would be ~ 2500 sec, whereas that for an external system might attain ~ 7500 sec.

I. INTRODUCTION

Studies at Los Alamos Scientific Laboratory (LASL) and at laboratories throughout the world have indicated the possibility of initiating low-yield thermonuclear reactions by the use of an intense laser pulse to heat to ignition a very small pellet of fusionable material. If a sufficiently high temperature and density can be achieved in the pellet, thermonuclear reactions will occur releasing substantial energy. Presumably, this energy release would be much lower (0.1 to 100 equivalent tons of TNT) than the minimum practical yields presently available from fission-implosion systems. Possible applications of such small, clean energy releases include terrestrial electric power generation and the object of this report: space propulsion.

Activities at LASL related to laser-driven thermonuclear reactions have been under way for over one year, including thermonuclear burning studies and laser development. In June 1970, the Advanced Propulsion Concepts task group was formed with the specific objective of investigating advanced propulsion concepts; as part of this investigation this group began to study the application of such energy pulses for space propulsion. Various LASL groups have initiated supporting studies and experiments, and are particularly active in investigations of laser-induced thermonuclear reactions.

The propulsion potential of nuclear-pulsed systems is extraordinary: It offers the possibility of approaching the optimal utilization of the ultimate known source of energy with minimal adverse side effects.

The use of nuclear explosions for propulsion was first proposed by Ulam[1] and has been investigated in detail between 1958 and 1965 under Project Orion.[2] In the final concept, fission explosions of one to 12 kilotons (equivalent TNT explosive energy release) were to be detonated behind a spacecraft to accelerate propellant material that would impart momentum to a massive pusher plate; this momentum, in turn, was to be transmitted more gradually from the pusher plate to the spacecraft through a pneumatic spring system. The study was terminated because of the inherent huge size of the resulting spacecraft (4000 tons), because of the limitations imposed by the

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nuclear test-ban treaty, and because of the difficulty of testing the system. It was, however, generally acknowledged that the concept was technically sound and might have achieved the very high specific impulse (I_{sp}) of ~ 2500 sec.

The possibility of producing small, clean nuclear explosions removes the two primary obstacles in the Orion concept--large size and space pollution. Spacecraft can now be envisioned that may be even smaller than those currently proposed for the nuclear space shuttle, yet be able to greatly outperform even that high-performance system. The system can also be meaningfully tested on or below ground. Although a laser-driven pulsed propulsion system, as any other advanced system, would require a fairly long development time (perhaps until 1985-1990), it should be actively pursued because it would make manned exploration of the solar system as feasible as present-day lunar exploration, and travel into earth orbit inexpensive and routine. These, in fact, are the characteristics of the system identified by NASA as necessary in a long-range space-exploration program.

The high potential performance of a nuclear-pulsed propulsion system is due to (1) high specific energy and (2) short interaction times. The specific impulse increases as the square root of the specific energy. Efficient conversion of chemical reaction energies would result in an I_{sp} of only ~ 460 sec. On the other hand, if a material were used which had 100 times the specific energy of TNT and this energy were converted to kinetic energy, velocities corresponding to an I_{sp} of ~ 3090 sec could be reached. If a mixture of $D+{}^3He$ were totally fused and the resulting energy totally converted into backward momentum of the reaction products, the I_{sp} would be as high as 2.2×10^6 sec. Clearly, particle velocities associated with the latter example could not be tolerated; this value is cited only to indicate the upper limit of fusion propulsion.

Nuclear propulsion concepts, in general, are limited by the ability of the materials used to withstand extreme temperatures rather than by the specific energy available. This limit is especially constraining in solid-core nuclear rockets where the temperature of the fuel elements must exceed that of the propellant without the elements deteriorating. Nuclear-pulsed propulsion systems are similarly limited, but, because of the extremely short interaction time of the pulses, at much higher temperatures. The interaction times, of the order of milliseconds, are simply too short to cause much destructive damage, especially if ablative materials are used to further protect the exposed surfaces from high temperatures.

Although, at present, it is assumed that the generation of laser-driven thermonuclear reactions applicable to space propulsion is feasible, i.e., that the large efforts at LASL and elsewhere will yield the expected results, it is nevertheless desirable to obtain an early working knowledge of the theories and practices of high-energy pulsed lasers, of thermonuclear burning, and of the thermodynamics of dense plasmas. This information is needed to attack the propulsion problem, to identify the products of such energy releases and their interaction with the surroundings, and to investigate means of tailoring the energy releases to specific propulsion applications.

This report discusses some aspects of using nuclear reactions (either fission or fusion) for pulsed propulsion; compares preliminary design concepts, including pulse units, and their performance limitations; presents mission considerations; and gives preliminary performance estimates.

Although a detailed reference design has not been prepared, early publication of these preliminary considerations seemed desirable to disseminate as quickly as possible the information generated and to indicate promising areas of research.

Thoughts have been given to naming the proposed Nuclear Pulsed Propulsion Concept, should further consideration lead to a full-fledged project, and the name Sirius* was chosen.

*Sirius, the brightest star in the heavens, is the principal star in the constellation Canis Major, the Great Dog (né Rover). Canis Major follows in the sky close to the heels of Orion, the great hunter, who by one mythological story was a "foolhardy, heaven-daring rebel who was chained to the sky for his impiety." The name Sirius is from the Greek adjective σειρος, meaning scorching, or perhaps from the Arabic siraj, the glittering one. The Egyptian venerated Sirius, regarding it as a token of the rising of the Nile and a subsequent good harvest.

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II. NUCLEAR REACTIONS FOR PULSED PROPULSION

Fusion

The isotopes of hydrogen, helium, and lithium would fuse if a sufficient quantity of material were heated to a sufficient temperature and held together for a sufficient length of time. In the laser-driven fusion concept, the necessary conditions are achieved by coupling the output from a laser to a small, effectively confined region of fusionable isotopes. Only a small quantity, of the order of one gram, is required. For example, one gram of deuterium plus tritium (D+T), totally burned, would produce an energy equivalent to exploding 80 tons of TNT.

In the propulsion concept discussed herein, the amount of energy released in what seems at the present time a plausible size for a laser-driven fusion reaction is clearly far below the amount of energy that is desired for each pulse, i.e., a few to many tens of tons TNT equivalent. It does not seem necessary to dwell here on how one might proceed from the initial small amount to the required large amount; it is simply assumed that a solution to the problem will be available at the time it is needed.

The products of the basic reactions will be high-energy charged particles and neutrons. The following basic fusion reactions are being considered:

D+D	→	^{3}He (0.82 MeV)	+ n (2.45 MeV)
D+D	→	T (1.01 MeV)	+ H (3.02 MeV)
D+T	→	^{4}He (3.5 MeV)	+ n (14.1 MeV)
D+^{3}He	→	^{4}He (3.6 MeV)	+ H (14.7 MeV)

The easiest to initiate is the D+T reaction which has a reaction cross section at low temperatures more than two orders of magnitude greater than that of the other three pairs. However, its usefulness may be impaired because 80% of the fusion energy is carried off by the 14.1-MeV neutrons, which have a long mean free path. Even in burning deuterium, a substantial fraction of the total fusion energy is carried off by 14.1-MeV neutrons from the D+T reaction, which will normally follow the tritium-forming branch of the D+D reaction; another portion of the energy is dissipated by 2.45-MeV neutrons. Burning an equal atomic mixture of D+^{3}He will also produce some 14.1-MeV neutrons due to the competing D+D reaction and the following D+T reaction.

These high-energy neutrons require shielding of the spacecraft. Most efficiently, this shielding should be placed near the source of the neutrons, particularly because other considerations also suggest the desirability of diluting the initial energy release with a surrounding or adjacent mass. The source particles, moving at near relativistic velocities, are simply too energetic to handle: their stagnation would cause excessive ablation of any surface, and they would penetrate a protective magnetic field. Also, per unit of energy ($E = 1/2\ mV^2$), these particles are carrying only a relatively small amount of momentum ($P = mV$). Because the momentum per unit energy (P/E) is equal to $2/V$, it would be wasteful of energy to impart momentum into high velocities. Some nonreacting mass, much larger than the fusioning mass, must be used to dilute the energy into a more usable form. This mass, called the propellant, might also provide the desired shielding function.

Fission

In a very analogous manner, fissionable isotopes can be compressed to the point where a very small quantity becomes supercritical. Such a fissioning system may produce fewer net neutrons per unit of usable energy released, but would form radioactive fission products--an undesirable byproduct. However, release of this small amount of radioactivity may be tolerable in the already hostile environment of space because the vast majority of the debris will have velocities sufficient to escape the solar system.

Laser Initiation of Nuclear Reactions

The following characteristics make a laser eminently suitable for achieving extreme compression and heating of small pellets:

- High energy density. Energy stored as excited molecular states of a suitable lasing material can be swept out of a large volume and focused on a small area.
- Short pulse duration. Pulse widths in the range of 10^{-10} to 10^{-7} sec can be achieved.
- Controllability. Laser pulses can be shaped and selected at the desired time in a low-energy stage and then amplified in successive high-energy stages.

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Present estimates of the laser energy required to heat and compress D+T in milligram quantities, and to achieve a net energy gain, range from 2×10^4 to 1×10^6 J (joule) delivered in a period varying from 10^{-10} to 10^{-7} sec. A laser of this magnitude may require a large volume of lasing medium for the final stage (of the order of 1 m^3), which may be expended after each laser pulse. Although glass and gaseous lasers are presently the most advanced, laser systems utilizing chemical reaction energy to achieve the required molecular excited states are more promising for space propulsion. Chemical lasers have a relatively high energy density and offer the possibility of expending the unused energy together with the spent reaction products. An attractive possibility is to assemble the final high-energy laser stage and the mirror-focusing system with the individual pulse units so that they could be deployed as one package just prior to pulse-unit initiation; the preliminary, low-energy laser stages would remain in the spacecraft. Thus, positioning of the pulse unit aft of the spacecraft would be not nearly as critical as had been expected.

A more complete description of laser concepts and considerations relevant to pulse propulsion are presented in Appendix A.

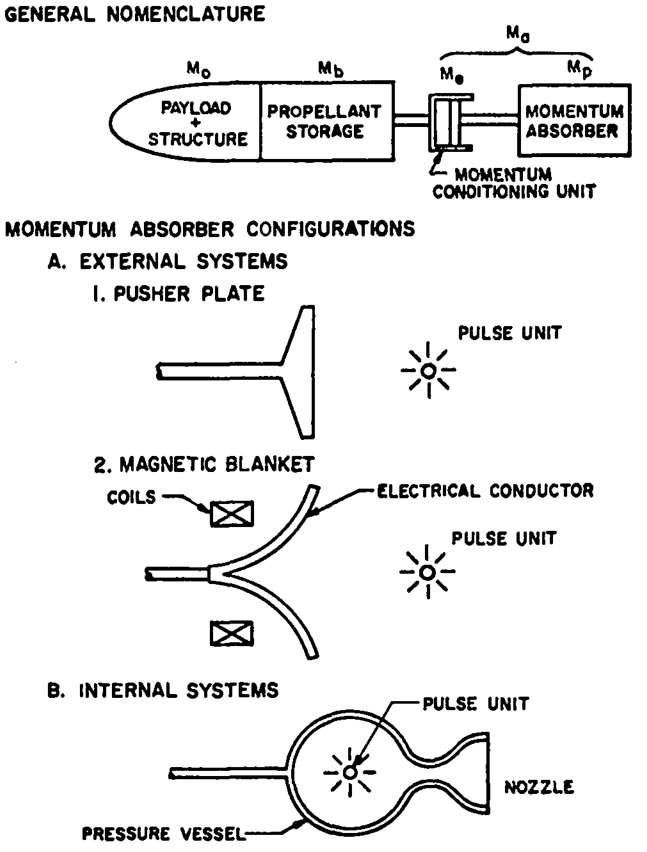

Fig. 1. Nuclear pulsed propulsion concepts.

III. PRELIMINARY DESIGN CONCEPTS

General

Three designs for converting the energy from an explosion into propulsive thrust are shown in Fig. 1. In each case it is supposed that an individual pulse unit is properly located and then initiated by a laser pulse. In the resulting nuclear explosion a quantity of propellant is heated by the released energy and expands as a high-energy plasma. A portion of this expanding plasma interacts with the space vehicle, thereby imparting momentum to it. Many pulse units are sequentially discharged and initiated, probably at equal intervals, resulting in the desired spacecraft velocity change.

The spacecraft mass is considered to consist of a payload plus supporting structure, propellant storage, and a momentum absorber, which smoothes out the shock or impulse resulting from the explosion of the pulse unit. Various momentum-absorber configurations are being cataloged according to whether the pulse unit is exploded externally, as depicted in Fig. 1A, or internally, as depicted in Fig. 1B. It will be shown in Section IV that, at least to a first-order approximation, the mass of the momentum absorber is proportional to the impulse delivered to the pusher plate (in an external system) or to the pulse-unit energy release (in an internal system).

Because a significant mass fraction of the spacecraft will be allocated to the momentum absorber, it will be desirable to select the proper value for the impulse delivered (or a pulse-unit energy release) which is appropriate for the minimum feasible pulse-unit mass. If too high an impulse is delivered or too much energy released, the increased mass of the momentum absorber will more than offset the gain in performance; conversely, if too low an impulse is delivered or too little energy released, the loss in performance will more than offset the

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reduction in momentum-absorber mass. In both cases (external and internal system) an optimum momentum-absorber mass fraction of the spacecraft corresponds to a maximum payload delivered which, in turn, corresponds to an optimum specific impulse and determines the impulse delivered (external system) or the pulse-unit energy release (internal system).

For the same pulse-unit mass, the external system will greatly outperform the internal system, simply because the momentum-absorber mass in the internal system must be larger to contain the pulse energy, rather than only to absorb the impulse as in the external system.

Concept A-1, External System with Pusher Plate

The pusher-plate concept (Fig. 1A-1) was developed in the Orion studies and, at this juncture, appears to be the most promising for the smaller laser-driven nuclear-pulse propulsion systems. The pulse-unit energy release occurs at some distance from the spacecraft, and a strong, probably flat metal structure, called a pusher plate, absorbs the shock of the explosion. A momentum-conditioning unit is required for a gradual momentum transfer between pulses and for return of the pusher plate to the proper location for the next pulse.

The magnitude of the kinetic energy stored in the pusher plate after the impulse requires that the momentum conditioner must approximate a conservative system, that is, a system in which kinetic energy is converted to a potential energy and vice-versa, without appreciable energy dissipation. (In a dissipative system quite unrealistic quantities of heat energy would have to be radiated to space or otherwise removed.) Thus the pusher plate and the momentum conditioner undergo a cyclic interaction.

At the beginning of a typical cycle the pusher plate is assumed to be positioned at its maximum distance from the spacecraft and moving toward the craft. As the pusher plate approaches the spacecraft it is decelerated by the momentum-conditioning unit (with a nearly constant force), which brings the plate to rest with respect to the spacecraft. At this point all the original kinetic energy of the pusher plate is stored in potential energy of the momentum-conditioning unit. Over the next part of the cycle the pusher plate is accelerated away from the spacecraft, attaining a final velocity equal to the initial velocity but opposite in direction. At this time, a pulse unit is fired converting the propellant into a plasma which acts at high pressure over the surface of the pusher plate for a very short time interval. The net effect is an impulse delivered to the pusher plate reversing its direction. The cycle is then repeated.

Functionally, the momentum conditioner acts as a long coil spring separating the spacecraft from the pusher plate. If the spring is greatly compressed so that the motion over a cycle is only a small fraction of the total compression, then the force exerted on the spacecraft is nearly constant, approximating the ideal condition described above.

The total mass of the momentum-absorber system is the sum of the masses of the pusher plate and of the energy-storage or momentum-conditioning unit. It will be demonstrated in Section IV that the total mass should be divided equally between the momentum-conditioning unit and the pusher plate if the total mass is to be minimized.

Concept A-1 can be designed for a high specific impulse because the pusher-plate surface is exposed to the propellant in very short pulses and the propellant energy density can be therefore quite high.

As the propellant material piles up against the pusher-plate surface, the temperature increases due to stagnation and much of the original kinetic energy is radiated away to space. Ablative material covering the plate vaporizes to form a thin film further protecting the subsurface. Orion studies verified that common materials, e.g., aluminum or steel, can withstand surface temperatures exceeding 80,000°K under such conditions with only nominal ablation.[3]

The specific impulse of an external system is the integral of the change in the axial component of the momentum of the propellant as a result of its interaction with the pusher plate.* This specific impulse is approximately equal to the product of the propellant impingement velocity times the fraction

*The specific impulse, so defined, should be divided by g_0, the standard acceleration due to gravity (9.8 m/sec^2), in order to obtain the usually accepted unit of seconds.

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of the pulse-unit mass which strikes the pusher plate. Initial performance estimates, presented in Section IV, indicate that the desired propellant impingement velocities are within the constraint of 150 km/sec imposed during the Project Orion study; further investigations, however, may disclose that plate-surface energy density will eventually limit the performance of this concept. Present specific-impulse estimates for the pusher-plate concept indicate an upper limit of ~ 7500 sec.

Another constraint which may limit the performance of the pusher-plate concept is the total pressure of the propellant acting against the surface of the plate. Excessive pressures will cause spallation if the resulting internal tensile stresses exceed the strength of the plate material when the internal pressure wave is reflected from either surface.

Design considerations for propellant-pusher plate interactions are presented in Appendix B.

Concept A-2, External System with Magnetic Field and Pusher Plate

The limitations imposed on Concept A-1 by ablation and spallation of the pusher plate might be overcome with a magnetic-field "blanket" to protect the plate surface from the high-energy propellant plasma. Figure 1B-2 shows a cusp-shaped, electrically conductive pusher plate and a superconductive coil to generate strong magnetic field lines parallel to the pusher-plate surface. As the dense plasma from the explosion expands it pushes the magnetic field lines against the conductor, increasing the field strength by inducing a circular current in the conductor. The increased magnetic pressure ($B^2/8\pi$) slows down the plasma, turns it, and accelerates it away from the pusher plate. The impulse is transferred to the pusher plate by magnetic interactions which spread out the force in space and time and protect the surface of the plate from particle impingement. Because the propellant particle velocities can be higher than for a plain pusher plate, the specific impulse of this concept can also presumably be higher. Whether the specific impulse derived by the optimization procedures will imply the need for magnetic fields to protect the pusher-plate surface will be determined in future studies. If so, it will also be necessary to determine whether the improved performance will offset the additional mass and complexity of the superconductive coils, etc.

Concept B, Internal System

In this configuration the energy is released inside a pressure vessel that is equipped with a conventional rocket nozzle for the discharge of the heated propellant (Fig. 1B). Various possibilities for controlling the process have emerged. In one concept liquid hydrogen is fed into the pressure vessel radially through the wall and acts as a coolant before uniformly filling the vessel. The energy from the pulse unit is released at the center of the tank after the vessel has been recharged to the full propellant mass. A shock wave is propagated through the hydrogen until it reaches the walls where it is reflected back toward the center. Because the stagnation pressure at the wall is higher than the frontal shock pressure by an order of magnitude, this delivers a sharp initial impulse to the wall. The wave is subsequently reflected back and forth in the vessel, continually increasing the internal energy of the hydrogen until equilibrium is established. By this time, which is of the order of milliseconds, the hydrogen has reached an average pressure and an isothermal condition due to the transfer of shock-wave kinetic energy to hydrogen internal energy. The hot gas is then expanded through a nozzle while the tank is being refilled with propellant. The expansion process is continued until the previous initial conditions in the tank are attained, then the cycle is repeated.

Other possible internal design concepts have been considered but no design work has been initiated. One of these is a pressure vessel lined with a thick layer of lithium hydride (LiH). The internal energy release vaporizes some of the LiH, which collects as a hot gas in the cavity, and then flows out of the pressure vessel through a nozzle. Thus the pressure-vessel lining is the propellant, which is gradually vaporized by successive explosions (similar to core burning in a solid-fueled rocket motor).

Although the internal concept initially seemed attractive, so many inherent drawbacks have emerged, as compared with the external system, that further

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work does not seem warranted, at least not in the near future. Basic problems are:

- For the same pulse-unit mass, the performance of the internal system is at least an order of magnitude poorer than that of the external system.
- The specific-impulse limit is about 2500 sec, corresponding to excessive radiative heat transfer to the pressure vessel and nozzle from a hydrogen propellant.
- There is no apparent way to solve the shielding problem. Isotropic shielding will be required to protect the pressure vessel.
- Positioning and initiating the pulse unit within the gas-filled pressure vessel may be much more difficult than in the vacuum aft of the spacecraft.

The only significant advantage of the concept is the minimal requirement for momentum-conditioning; this function is predominantly performed by the pressure vessel. Although the thrust from the nozzle may vary by an order of magnitude over a propulsion cycle, the mass requirements for smoothing these variations out in the spacecraft are small compared with the pressure-vessel mass.

Pulse Unit Design Concepts

To minimize the mass of the momentum absorber, the mass of the pulse unit should be as small as possible. The factors which tend to increase the size of the pulse unit are the requirements for spacecraft shielding, the desire to limit the number of laser pulses, and the maintenance of a high average thrust within the limitation of laser-pulse recycle time. As mentioned earlier, spacecraft shielding is of particular concern because of the 14-MeV neutrons produced to a greater or lesser degree by the fusion reactions utilized. In any case, the pulse unit will consist predominantly of propellant material which will serve the dual function of (1) diluting the energy density thereby reducing the specific impulse to acceptable levels and increasing the impulse obtained per unit of energy; and (2) providing primary shielding for the spacecraft. Pulse-unit masses in the range of 1 to 100 kg are thought to be appropriate.

In an internal system, spherical geometries are generally indicated, which offer little latitude for most effective placing of the propellant material: it would simply form a globe around the propulsion pellet. Calculations show that 20 kg of a good shielding material, e.g., lithium-6 hydride (^6LiH), would remove only 45% of the 14-MeV neutrons generated in the fusion reactions, and that hundreds of kilograms would be required for effective shielding from a D+T pulse.

In addition to shielding, the major consideration in the design of the pulse unit for an external system is proper collimation of the expanding propellant so that a major fraction impinges on the spacecraft.

With respect to shielding, the external system benefits from an inherent advantage because pulse unit and spacecraft are physically separated. This will significantly decrease spacecraft radiation doses, mainly for the following reasons.

- The spacecraft subtends a small solid angle from the pulse unit. Because radiations will emanate isotropically from the detonation, only a small fraction must be dealt with. For example, at a distance of 8.5 m from a point, a 4.72-m-diam plate subtends a 30° included-angle cone representing only 1.7% of the total solid angle around the point.
- Propellant material can be positioned to form a shadow shield between the radiation source (which is nearly a point source) and the spacecraft, i.e., shielding mass is required only in the solid angle subtended by the spacecraft. From a shielding standpoint, the logical geometry is a cone of propellant material with the energy source located at the apex of the cone and with the cone axis oriented along the spacecraft axis.
- Single fast-neutron scatterings will remove most of the neutrons within the cone angle from further consideration. For small angles, the probability of a scattering neutron collision resulting in a new path still remaining within the cone angle is exceedingly small. The probability for multiple scatterings back into the original cone angle is even smaller. Thus the

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neutron shielding problem is reduced from the usual requirement of slowing down and capturing the neutrons to one of simply causing single scattering events. Calculations have confirmed that, even for large shielding factors, most neutrons striking the spacecraft will be uncollided.

Monte Carlo calculations, presented in Appendix C, indicate that 3.8 kg of beryllium in the shape of a 30° included-angle cone, 30 cm high, will absorb 99.0% of all 14-MeV neutrons originally directed within the cone angle toward the spacecraft. Thus, in this example, only 0.010 x 0.017 = 0.017% of all neutrons emanating from the source will travel toward the craft.

For an external system, a sophisticated hydrodynamic design of the pulse unit would favor a layered flat plate rather than a cone with the pellet at its apex; compromise geometries may eventually emerge. In general, if the pulse directs one element of mass toward the pusher plate, an equal momentum must be carried by another element of mass away from the spacecraft. Also, the fraction of total energy carried by the first element is equal to the fraction of total mass carried in the second element and vice-versa. One maximizes the momentum carried off, per unit of energy expended, by equalizing the masses of the two elements, which, however, may not be a desirable goal.

If the laser-driven fusion concept can be made to work, it can probably be made to give any desired energy, and the cost might be nearly independent of the energy level. This reasoning may lead to designs that are wasteful of energy in order to save propellant.

If the pusher-plate impingement velocity is fixed, the maximum impulse transferred to the pusher plate, per unit of total mass expended (i.e., the specific impulse), can be clearly obtained by letting the majority of the mass impinge on the pusher plate at this maximum velocity. The remaining mass carries the majority of the energy off in the opposite direction at higher velocities. Selection of materials for the propellant will depend on the effective neutron scattering that can be achieved from a given mass. Neutrons scattering out of the propellant cone impart some of their momentum to the propellant atoms, implying some direct propellant heating. In one concept this is supposed to be the dominant source of propellant energy, and the design reduces to the determination of geometries leading to good collimation of volumetrically heated materials, somewhat similar to the design of shaped charges for the formation of explosive jets. One possibility is to surround the neutron-heated propellant with a pressure vessel-and-nozzle configuration of thin, heavy material which will direct the propellant toward the pusher plate in a supersonic stream.

Another, more attractive, possibility is to design for utilization of the energy carried by the charged products of the fusion reactions. This energy could be absorbed in a shell of high-density material surrounding the pellet which would then confine this energy to a channel around the propellant-cone surface where it would be absorbed in a thin layer on the surface of the propellant cone. This heated surface would blow off at high velocity propelling the remaining cone at lower velocities toward the pusher plate.

Other geometries, e.g., layered plates of various materials, are also being considered.

<u>Design Implications of Payload Shielding</u>

Payload shielding is a major design consideration for all spacecraft components particularly because manned missions are assumed. The spacecraft are therefore to be designed so as to provide the most favorable shielding. They will be of elongated geometries, with as much distance and spacecraft material between the energy source and the payload as possible. Monte Carlo neutron-shielding calculations have been performed for a reference case with the assumption that the primary energy source is the D+T reaction; detailed results are presented in Appendix C. Conclusions are that some attenuating medium, in addition to the propellant, is required to protect the payload if the D+T reaction is utilized; but calculations show that the shielding masses are acceptable: a maximum of 15,000 to 40,000 kg if the mass is at the payload location, or 2000 to 5000 kg if the mass is more favorably deployed nearer the energy source, but still within the spacecraft. These masses can be included within a basic total spacecraft mass of 100,000 kg, either

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as structural components, stored pulse units, or life-support supplies. Thus, payload shielding considerations, although to be recognized in the design from the beginning, apparently will not invalidate the entire nuclear-pulsed propulsion concept even in the unlikely event that the D+T reaction is the primary energy source.

IV. MISSION CONSIDERATIONS AND PERFORMANCE ESTIMATES

General

In the normal rocket equation, it is assumed that the spacecraft consists of two components--a payload, M_o, and a propellant that is expended over a period of time at a constant exhaust velocity, V_e. Given the mission-equivalent free-space velocity change, ΔV, the required propellant fraction is determined by the rocket equation: $M_o/M = \exp(-\Delta V/V_e)$, where M is the total initial mass of the spacecraft. In any propulsion system there will be an additional mass in the spacecraft, associated with the propulsion system, which scales in some manner with the engine characteristics. This mass must be included in M. For example, in an electric propulsion system, an extra power-supply mass must be included in the spacecraft as determined by the required jet power and the efficiency of the system. The inclusion of this engine mass usually results in some design flexibility in choosing the fraction of the total spacecraft mass allocated to this engine mass and in a resulting variation in engine performance, V_e. Frequently there is an optimum engine mass fraction leading to a maximum spacecraft payload fraction. As is well known, this is the case for the preceding example of electric propulsion--there exists an optimum fraction of the spacecraft mass which should be allocated to power supply and a corresponding optimum specific impulse depending on the thruster and power-supply characteristics and on the mission. In other words, the inclusion of the engine mass in the spacecraft provides an apparent extra degree of freedom in the design, which, however, is usually only defined during optimization of the system.

The above generalizations are also true for pulsed-propulsion systems. In this case, the extra spacecraft masses consist of a pusher plate or pressure vessel, of the momentum-conditioning system, and of the laser system. The size of these components will depend on the size of the propulsion pulse. Although the details of this scaling are not yet well understood, optimizations for two possible scaling laws have been analyzed and are presented below.

Consider a spacecraft of total initial mass, M, consisting of components of three kinds: propellant, of mass M_b; a momentum absorber, of mass M_a; and the remainder (i.e., payload and other structures) of mass M_o. Thus

$$M = M_o + M_a + M_b \ . \qquad (1)$$

In this allocation it is appropriate to ascribe to M_a any mass that scales with the pulse size.

Assume that the propellant consists of n identical pulse units each of mass m. Although the use of nonuniform pulse units may be desirable in some circumstances, this generalization is deferred. Only a minor fraction of the pulse unit consists of fuel. The main mass is a suitable arrangement of propellant materials designed to maximize the total impulse of the explosion while diluting the fuel energy density to a tolerable level at the momentum absorber and shielding the spacecraft from radiations originating in the fuel. All ancillary expended material is also included in the pulse-unit mass.

As a result of the explosion of the pulse unit and its interaction with the spacecraft, a total impulse, I, will be imparted to the spacecraft along its axis. In a specific case this impulse can be determined by integrating the axial component of the velocity change for each component of the pulse mass as a result of its interaction with the spacecraft. It is convenient to represent the impulse as a fraction of the total impulse which could be realized if all the energy, q, were converted into kinetic energy, uniformly distributed in the pulse-unit mass, and wholly directed opposite to the spacecraft motion. Thus

$$I = \eta \sqrt{2\,mq} \qquad (2)$$

where the coefficient η is the efficiency for converting energy into impulse and $0 \le \eta \le 1$.

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It is convenient to characterize the spacecraft performance in terms of the payload delivered through a total velocity increment, ΔV, in a rectilinear flight without external forces. The overall requirement of an actual space mission, which will involve several distinct propulsion intervals in planetary and solar gravitational fields, must then be expressed as the total rectilinear free-flight velocity change, ΔV, to which it is dynamically equivalent. The total ΔV will be made up of n velocity increments which increase slightly as the total mass of the spacecraft decreases. Rather than to express the total ΔV as a sum it is convenient to approximate it by an integral. The differential effect of one impulse is to change the velocity of the spacecraft mass by dV where:

$$M dV = I = \frac{I}{m}(-dM). \qquad (3)$$

The mass ratio for the total mission is obtained by integrating this equation to obtain

$$\Delta V = (I/m) \ln \left[M/(M-M_b)\right] . \qquad (4)$$

This approximate procedure will overestimate the correct value of ΔV by a fractional amount of order m/M.

The pulse units cannot practically be made so small and so numerous as to simultaneously achieve an interesting velocity increment and maintain the peak vehicle acceleration at an acceptable level. Thus the momentum-absorber system performs the main function of accepting the short-duration impulses and delivering them to the spacecraft with a nearly continuous force over a longer time. This is true for either an external (or impingement) system or an internal (or containment) system as described in the previous section and as illustrated in Fig. 1.

Scaling Laws

Two scaling laws, energy scaling and impulse scaling, have been considered for the total mass of the momentum absorber.

Energy Scaling

It is assumed that M_a is proportional to the energy release per pulse, q. This type of scaling should be appropriate, for example, in an internal system where the energy must be contained. A convenient expression of this proportionality is

$$M_a = \alpha q/W^2. \qquad (5)$$

The quantity, W, which has units of velocity, characterizes the specific strength of the pressure-vessel material, and is defined by

$$W = \sqrt{\frac{\text{allowed stress}}{\text{mean density}}} . \qquad (6)$$

The quantity, W, is very nearly the maximum peripheral velocity of a flywheel made of the pressure-vessel material. The quantity α is a dimensionless factor with a magnitude approximately equal to unity. The precise value of α will depend on design details, on the allowance needed for shock-loading of the pressure vessel, on the safety factor desired, and on the equation of state of the heated propellant gas.

Elimination of the quantities M_b, I, and q from Eqs. (1), (2), (4), and (5) leads to the expression

$$M_o/M = \exp\left[-2/(\mu\sqrt{M_a/M})\right] - M_a/M \qquad (7)$$

where the parameter μ is defined as

$$\mu = (\sqrt{M/m})(W/\Delta V)(\eta\sqrt{8/\alpha}) . \qquad (8)$$

Before discussing the implications of Eqs. (7) and (8), equivalent equations for impulse scaling will be derived.

Impulse Scaling

It is assumed that M_a is proportional to the total impulse, I, delivered to the momentum absorber per pulse. A convenient expression of the proportionality is

$$M_a = \frac{I}{W}\sqrt{\frac{\alpha'}{2}} . \qquad (9)$$

It will be seen that this scaling may be appropriate to an external concept.

The total mass of the momentum-absorber system in an external configuration is the sum of the masses of the pusher plate, M_p, and the energy storage or momentum-conditioning system, M_e. The cyclic nature of the pusher plate-momentum conditioner systems has been described in Section III. In a typical cycle the change in velocity is twice the initial velocity and the initial kinetic energy is $I^2/8 M_p$. This energy must be stored when the pusher plate is brought to rest.

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It can be shown that Eq. (5) is a very general relation between the energy and the minimum mass of a system required to store it. It applies regardless of the manner in which the energy is stored, i.e., as compressed gas, as a rotating mass, or in a magnetic field. Only the quantity α varies and only over a range of about a factor of two. However, the requirements of the momentum conditioner exceed just the raw requirement of energy storage. The momentum conditioner must also be able to cope nondestructively with a failure of the pulse to fire at the correct time. If it is further required that the acceleration of the spacecraft be relatively uniform it will become necessary to increase the mass of the momentum conditioner above the minimum by an order of magnitude. Cooling systems to remove the dissipative fraction of the energy, lubrication systems, and other components will ultimately increase the required α to well above the minimum value of unity.

The mass of the momentum-conditioning unit can be related to the energy term in a manner analogous to Eq. (5),

$$M_e = \frac{\alpha' \; (I^2/8 \; M_p)}{W^2} \tag{10}$$

where the prime denotes the momentum-conditioning unit. The total mass of the momentum absorber is then

$$M_a = M_p + M_e = M_p + \alpha' \; I^2/8 \; M_p W^2 \; . \tag{11}$$

M_a will have a minimum value when M_p is appropriately chosen as

$$M_p = M_e = I \sqrt{\alpha'/8} \; /W \tag{12}$$

so that

$$M_a = I \sqrt{\alpha'/2} \; /W . \tag{13}$$

Elimination of M_b and I from Eqs. (1), (4), and (13) leads to the expression

$$M_o/M = \exp(-M/\mu' \; M_a) \; -M_a/M \tag{14}$$

where μ' is defined as

$$\mu' = (M/m) \; (W/\Delta V) \; \sqrt{2/\alpha'} \; . \tag{15}$$

Optimization Procedure

Expressions have been obtained for two possible ways in which the momentum-absorber mass, M_a, scales with the pulse size. As a first consideration, it is clearly desirable to minimize the pulse size in order to minimize the corresponding M_a. Several factors, however, will limit the minimum size of the pulse unit.

- Shielding requirements. For the internal system this requirement may impose a large lower limit, of the order of 100 kg. The limit for an external system may be much lower--of the order of five kilograms.
- Wastage of laser reactants. A fixed quantity of gas may be expended after each pulse and the amount will probably be independent of pulse size. It can be shown that this leads to a desire to select a pulse-unit mass which would be of the same order as the laser reactant mass expended.
- Minimum time interval between pulses. To avoid inefficient propulsion, the net average thrust should be comparable to the local gravitational attraction. The average thrust is equal to the impulse times the pulse rate. For a fixed maximum pulse rate, this will tend to set a minimum impulse.

These considerations--in particular the last two--may lead to an optimum pulse-unit mass; however, in the present discussion it will be assumed that the minimum pulse-unit mass is fixed. In each of the expressions for μ and μ' given in Eqs. (8) and (15), the quantity M/m appears as a ratio which, in a given case, will be determined. The quantity W is determined by the choice of materials. The mission ΔV will be given and the quantities $\eta\sqrt{8/\alpha}$ in Eq. (8) and $\sqrt{2/\alpha'}$ in Eq. (15) will be determined by the sophistication of the system design. Thus a maximum μ or μ' will be determined.

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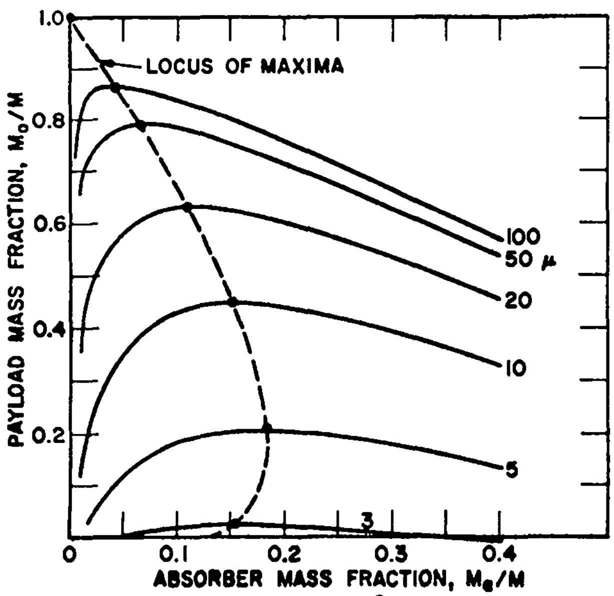
Plots of Eqs. (7) and (14) are presented in Figs. 2 and 3, respectively, for constant values of μ and μ'. It can be observed that there is in each case a maximum payload fraction, M_o/M, and a corresponding optimum momentum-absorber fraction, M_a/M. Clearly, large values of μ and μ' are desirable. The locus of the maxima is indicated on each figure, and the corresponding optimum allocations of the spacecraft mass between payload, propellant and momentum absorber are plotted in Figs. 4 and 5 as functions of μ and μ', respectively. It can be seen that there is a limiting minimum value of $\mu = \mu' = e$ for which the payload vanishes and the mission cannot be performed.

Comparison of External and Internal System Concepts

The scaling considerations discussed above permit a direct performance comparison of the external and internal systems. The parameters μ (for energy scaling) and μ' (for impulse scaling) have intentionally been defined as roughly comparable. In each case the minimum usable value is e. Larger values result in comparable maximum payload fractions, M_o/M, as can be demonstrated by reference to Figs. 4 and 5. The correspondence is not exact; for example, at $\mu = \mu' = 100$, the maximum payload fraction for an energy scaling law is 0.86, whereas for an impulse scaling law it is 0.80. However, for values of μ and μ' in the range from 2.7 up to 10.0 the correspondence is within 3%. Thus the performance of the two types of systems can be compared directly by comparing μ and μ'. The ratio, μ'/μ, can be obtained directly from Eqs. (8) and (15) as

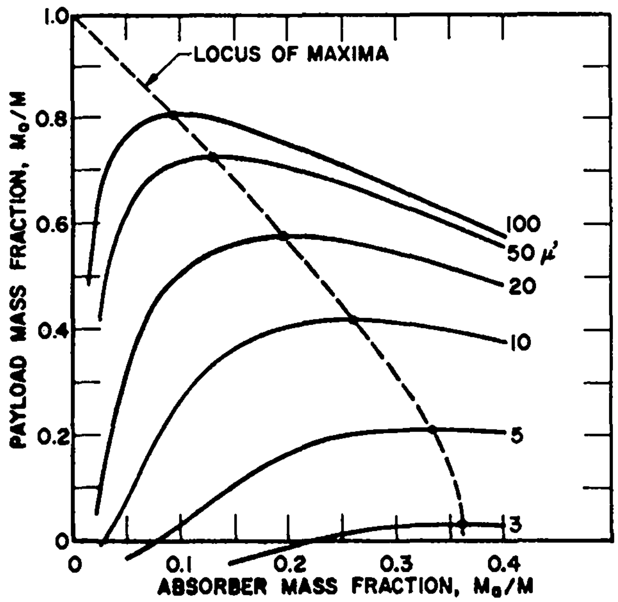
$$\mu'/\mu = \sqrt{(M/m)(\alpha/\alpha')} \;/2\eta \; . \qquad (16)$$

From this expression it is clear that μ' is significantly greater than μ in any practical case. An example, intentionally chosen to favor the internal system, is: $M = 100{,}000$ kg, $m = 20$ kg, $\eta = 0.9$, $\alpha = 2.0$, $\alpha' = 47$. From Eq. (16) we obtain $\mu'/\mu = 8.1$. Because μ and μ' appear in Eqs. (7) and (14) as dimensionless specific impulse, it follows that the optimum specific impulse of an external system is eight times higher than that of an internal system for the same spacecraft/pulse-unit mass ratio. In fact, larger pulse-unit masses may be required for an internal system than for an external system, providing another argument against an internal design.



$$X_o = e^{-\frac{2}{\mu\sqrt{X_a}}} - X_a$$

Fig. 2. Payload fraction for an energy scaling law.



$$X_o = e^{-\frac{1}{\mu X_a}} - X_a$$

Fig. 3. Payload fraction for an impulse scaling law.

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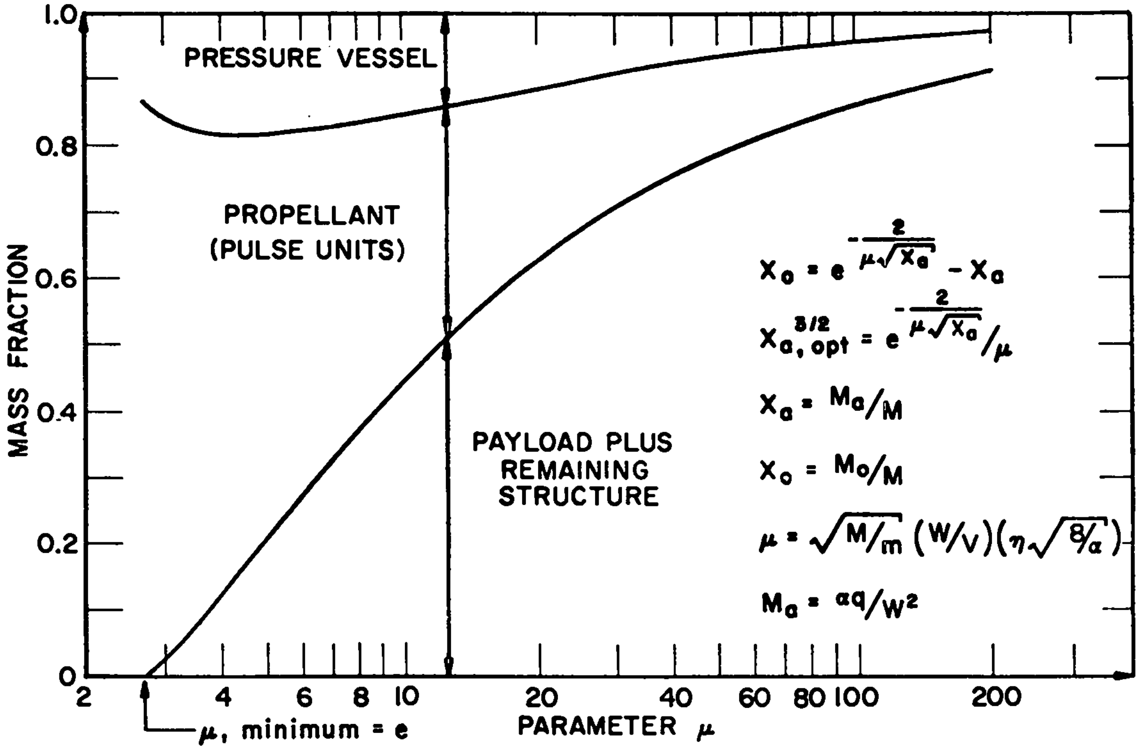

Fig. 4. Optimum allocation of spacecraft mass for energy scaling law.

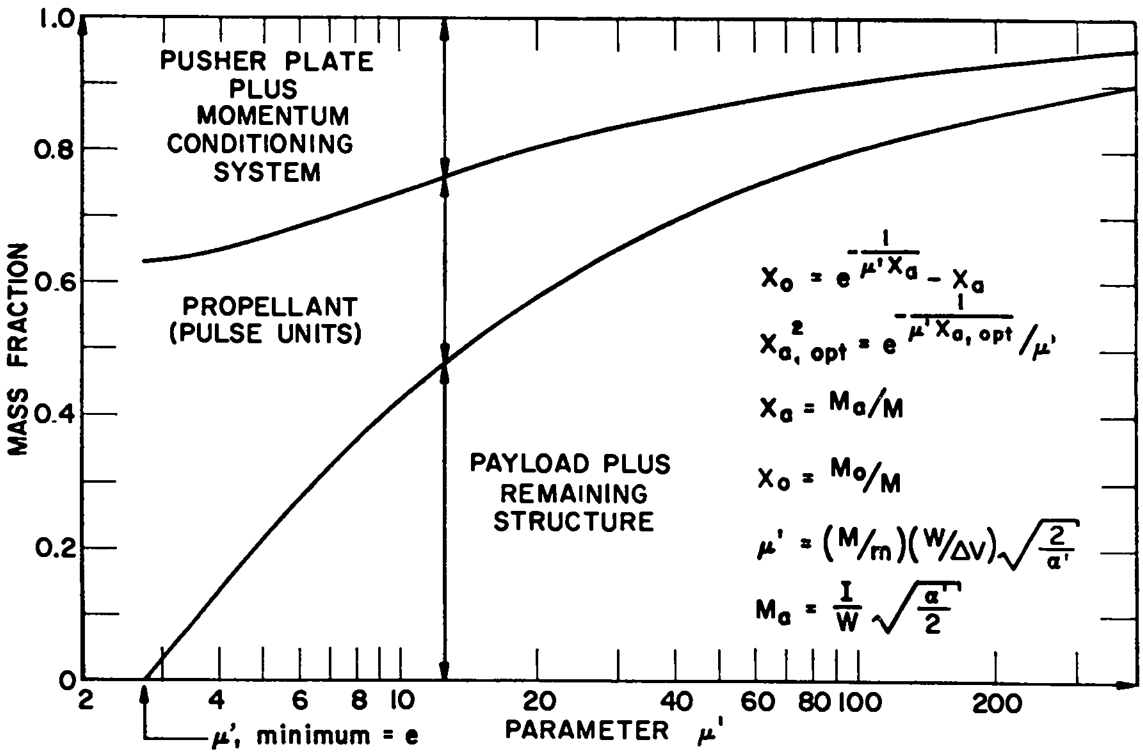

Fig. 5. Optimum allocation of spacecraft mass for impulse scaling law.

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Performance Limitations

The basic performance limitation, that of minimum pulse-unit mass, has been discussed. However, other design limitations may prevent the selection of the optimum pulse-unit impulse or energy.

Specific Impulse Limitation

This limitation is manifested, both mathematically and physically, in different ways for the energy scaling law and the impulse scaling law. These two cases are therefore treated separately.

Impulse Scaling. In the Orion system studies, the impingement velocity of the propellant on the pusher plate was limited to 150 km/sec. The limit was not based upon physical limits, which were unknown, but rather on a breakdown of the validity of the models which were used to predict ablation of the pusher-plate surface. In any case, surface ablation will impose some limit on the impingement velocity, V_i. The net impulse can be written

$$I = \beta m V_i, \tag{17}$$

which serves as a definition of β. In the case where the propellant impinges on the pusher plate axially and does not recoil with appreciable velocity, β is equal to the fraction of the pulse-unit mass which impinges on the pusher plate. Elimination of I between Eqs. (4), (14), and (17) leads to the following expression:

$$\beta V_i = \Delta V \mu' (M_a/M) . \tag{18}$$

If the optimization procedure described previously leads to a value of $\Delta V \mu' (M_a/M)$ greater than the allowable value of βV_i, then the system is limited to a maximum $I_{sp} = \beta V_i/g_o$ and the payload plus momentum-absorber fraction is simply

$$(M_o + M_a)/M = \exp(-\Delta V/\beta V_i). \tag{19}$$

The impulse is given by Eq. (17) and the design becomes one of minimizing M_a.

Energy Scaling. The final temperature of the heated propellant filling the pressure vessel (probably hydrogen) is limited by the acceptable level of heat transfer to the cooled pressure-vessel and nozzle walls. This imposes a limit on the maximum value of the energy density, q/m, and sets a limit on the product $\mu\sqrt{M_a/M}$ through the relationship

$$\mu\sqrt{M_a/M} = \eta\sqrt{8\,(q/m)}\;/\Delta V .$$

The corresponding maximum specific impulse is

$$I_{sp} = \eta\sqrt{2\,(q/m)}\;/g_o .$$

For hydrogen at 8333°K (15,000°R), $q/m = 3.5 \times 10^8$ J/kg, corresponding to an I_{sp} of 2430 sec. At this point the hydrogen is totally dissociated and is only slightly ionized. Radiative heat transfer would become excessive at higher temperatures.

Momentum Conditioner Limitations

In the momentum-conditioner concept described above, the pusher plate exerts a constant force, F, on the spacecraft throughout a cycle of duration Δt. A force balance at a time when the propellant is just expended can be written as:

$$a_o (M_o + M_e) = M_a (a_a - a_o) \tag{20}$$

where:

a_o = spacecraft acceleration, at the end of mission, relative to an inertial frame of reference.

a_a = pusher-plate acceleration, at the end of mission, relative to the spacecraft.

The distance traveled by the pusher plate, relative to the spacecraft, is the stroke, Δx. In terms of acceleration and time interval between pulses, it is

$$\Delta x = a_a \,\Delta t^2/8. \tag{21}$$

The initial velocity of the pusher plate, relative to the spacecraft is

$$U = \sqrt{2\, a_a \Delta x} = \frac{I}{2 M_p} . \tag{22}$$

Elimination of a_a and I from Eqs. (10), (21), and (22) results in an expression for the mass ratio, $b^2 \equiv M_e/M_p$:

$$b^2 = M_e/M_p = 8\,\alpha' (\Delta x/W\,\Delta t)^2 . \tag{23}$$

Limitations may be imposed on the maximum stroke, Δx, and on the minimum time between pulses, Δt. To minimize the total momentum-absorber mass, $M_a = M_p + M_e$, it is desirable to split the mass equally so that $M_p = M_e$ and $b = 1$. This can be shown to be the case by differentiating Eq. (11) with respect to M_p. However, if the maximum permitted stroke, Δx, and the minimum permitted time interval, Δt, are such that b as given by Eq. (23) is less than unity, then the minimum M_a cannot be used and Eq. (15) for μ' must be modified as follows:

$$\mu' = (M/m)\,(W/\Delta V)\,\sqrt{2/\alpha'}\,\left[2/(b + 1/b)\right] .$$

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The final factor, $2/(b + 1/b)$, will be less than unity if the permitted stroke and time interval combine to limit the design flexibility. The net result is a decreased parameter μ' and, consequently, a decreased payload. As with most smooth maxima, deviations from an optimum parameter result in only minor performance reductions. For example, if $M_p = 2 M_e$, then $b = \sqrt{1/2}$ and μ' is reduced by only 6%.

Performance Estimates

External System

To estimate the performance of an external system, example parameters are chosen as follows:

M = 100,000 kg. This initial spacecraft mass is comparable to that of the largest spacecraft contemplated in earth orbit in the time period 1970-85.

m = 20 kg. This pulse-unit mass should provide roughly a factor-of-600 shielding against 14-MeV neutrons if only half the mass is beryllium configured in a 15° (half-angle) cone between the source and the spacecraft.

W = 300 m/sec. This quantity corresponds to the use of steel with a 100,000-psi yield strength.

ΔV = 20 km/sec (65,616 ft/sec). This free-space velocity change would occur in a very advanced mission. It would provide for a fast round trip to Mars from low earth orbit.

α' = 47. This efficiency value is taken from the Orion study for an actual momentum-conditioner design. To some degree, this high value can be attributed to the complex two-stage design employed and to the large number of parts which serve no energy-storage role.

The resulting value of μ' is 15.57 (from Eq. 15), and corresponds to a maximum payload fraction of 0.525. Corresponding spacecraft parameters are:

Optimum specific impulse = 6927 sec
Number of pulses = 1275 (for M_b = 25,515 kg)
Energy per pulse = 44 equivalent tons of TNT (assuming 50% of propellant impinges on pusher plate)
Pusher-plate mass (M_p) = 10,970 kg
Momentum-conditioner mass (M_e) = 10,970 kg
Payload plus other structures (M_o) = 52,542 kg
Payload acceleration = 19.6 m/sec^2 (2 g), end of mission
Pusher-plate stroke (Δx) = 14.4 m
Interval between pulses (Δt) = 0.93 sec
Total propulsion time = 20 min/mission.

These values are based on an optimum allocation of masses from a mission point of view; no corresponding detailed design exists. Clearly, some portion of mass M_o will be required for the laser, for pulse-unit conveyance and storage mechanism, shielding, etc. (Neutron environmental considerations for such an external system are presented in Appendix C.)

Each of the parameters affecting μ' has been varied to study the effect on system performance; the results are presented in Table I. The first example is the base case presented above. The parameters which have been modified from this base case are underlined. Examples of the effects of various constraints are listed in the last five cases. Table I clearly illustrates the extraordinary performance which can be expected from an external pulsed propulsion system.

In the base case presented, the specific impulse desired could be achieved if one half the propellant impinged on the pusher plate at a velocity of 135 km/sec. This is barely within the impingement velocity constraint of 150 km/sec imposed during Project Orion. It is not presently known whether this impingement fraction can be achieved within the geometric constraints governed by shielding considerations.

Internal System

If the internal system is to deliver a positive payload, the value of μ in Eq. (8) must exceed e. If the pellet is a major source of neutrons that require shielding, then the implied mass is very high--of the order of hundreds of kilograms. Two additional parameter estimates, for α and for η, are required to obtain a numerical estimate of μ (other parameters are defined in the previous discussion of external systems):

α = 2. This selection is based on the observation that the mass in an internal design is very nearly just that of a simple pressure vessel. It can be demonstrated that the mass of the

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TABLE I

EXTERNAL NUCLEAR PULSED PROPULSION SYSTEM PARAMETER STUDY

M, kg	m, kg	W, m/sec	ΔV, m/sec	α'	μ'	b	I_{sp}, sec	M_o, kg	M_a, kg	M_b, kg	M_p, kg	M_e, kg	a_o, g	Δ_x, m	Δ_t, sec	q, tons*
100000	20.0	300	20000	47.00	15.47	1.000	6927	52542	21941	25515	10970	10970	2.0	14.39	.930	44.0
100000	5.0	300	20000	47.00	61.88	1.000	14998	75402	11875	12721	5937	5937	2.0	6.64	.429	51.6
100000	10.0	300	20000	47.00	30.94	1.000	10279	65714	16278	18006	8139	8139	2.0	9.69	.626	48.5
100000	50.0	300	20000	47.00	6.18	1.000	3910	28377	30966	40655	15483	15483	2.0	25.49	1.647	35.1
100000	100.0	300	20000	47.00	3.09	1.000	2306	4754	36524	58721	18262	18262	2.0	43.22	2.793	24.4
100000	20.0	150	20000	47.00	7.73	1.000	4531	35035	28705	36258	14352	14352	2.0	5.49	.710	18.8
100000	20.0	600	20000	47.00	30.94	1.000	10279	65714	16278	18006	8139	8139	2.0	38.79	1.253	97.0
100000	20.0	300	10000	47.00	30.94	1.000	5139	65714	16278	18006	8139	8139	2.0	9.69	.626	24.2
100000	20.0	300	40000	47.00	7.73	1.000	9063	35035	28705	36258	14352	14352	2.0	21.99	1.421	75.4
100000	20.0	300	100000	47.00	3.09	1.000	11532	4754	36524	58721	18262	18262	2.0	43.22	2.793	122.1
100000	20.0	300	20000	2.00	75.00	1.000	16621	77586	10859	11554	5429	5429	2.0	140.94	1.879	253.7
100000	20.0	300	20000	5.00	47.43	1.000	12994	72042	13423	14534	6711	6711	2.0	72.11	1.520	155.0
100000	20.0	300	20000	20.00	23.71	1.000	8857	61121	18209	20579	9149	9149	2.0	26.45	1.115	72.0
100000	20.0	300	20000	100.00	10.60	1.000	5525	43592	25527	30880	12763	12763	2.0	8.47	.799	28.0
Constrained																
100000	20.0	300	20000	47.00	15.47	1.000	5102	59873	16158	32967	8079	8079	2.0	11.77	.761	23.9
100000	20.0	300	20000	47.00	15.47	1.000	6927	52542	21941	25515	10970	10970	5.7	5.00	.323	44.0
100000	20.0	300	20000	47.00	15.47	1.000	6927	52542	21941	25515	10970	10970	.9	30.94	2.000	44.0
100000	20.0	300	20000	47.00	4.87	.161	3309	20694	33282	46022	32435	846	.6	4.99	2.000	10.0
100000	20.0	300	20000	47.00	14.10	.646	6561	50477	22790	26732	16074	6715	1.7	9.99	1.000	39.5

*Minimum kinetic energy carried by pulse-unit mass, in equivalent tons of TNT (4183 MJ/ton). This assumes that 50% of the propellant mass impinges on the pusher plate carrying 50% of the energy.

momentum-conditioning system is very small--of the order of ten times the pulse-unit mass for a conservative design. The value of α for an actual internal design which has been studied is 1.2. If supporting structures, parts in the pressure vessel, pumps, a nozzle, and other requirements amount to 2/3rd the mass of the bare vessel, α will increase to 2.0.

η = 0.9. This value is taken from an internal system that has been analyzed. In no case can η be less than 0.75.

With these values, $\mu = 0.85$ (from Eq. 8) indicating that the chosen mission exceeds the capability of an internal design system.

The maximum pulse-unit mass which can be used for these given parameters can be obtained by setting $\mu = e$. In this case m = 49 kg. The spacecraft mass is apportioned as M_b = 86,470 kg (pulse units or propellant) and M_a = 13,530 kg (pressure vessel). The corresponding energy release per pulse unit is $q = M_a W^2/\alpha$ = 609 MJ or 0.19 equivalent tons of TNT. Comparison of these values with those for the external system example used previously illustrates the disadvantage of the internal system: In the external system an energy release of 44 equivalent tons of TNT required a momentum-absorber mass of only 21,941 kg--a much smaller mass per unit of energy or per unit of impulse imparted than in the internal system.

An internal system with a reasonable performance can be envisioned only by ignoring the shielding requirement and choosing an acceptably small pulse-unit mass. This may be practical if suitable fusion or fission reactions can be used as the energy source.

One such internal design has been partially studied. A pressure-vessel diameter of 3.658 m (12 ft) was chosen arbitrarily and hydrogen was assumed to be the propellant. The cyclic operation of the system has been described in Section III. The hydrogen stagnation conditions in the vessel, after the shock wave has decayed and before expansion through the nozzle has begun, were chosen to

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Temperature = 8333°K (15,000°R)
Pressure = 689 N/cm^2 (1000 psia).

At this temperature hydrogen is almost completely disassociated, and ionization is insignificant. At higher temperatures, ionization increases, the hydrogen becomes opaque, and radiation heat transfer from the opaque hydrogen to the walls probably becomes excessive.

With the above assumptions the mass of hydrogen in the vessel is 2.53 kg and the corresponding value of μ from Eq. (8) is 5.37. A maximum payload of 23,550 kg can be delivered if the optimum M_a and energy release are selected as 18,230 kg and 0.196 equivalent tons of TNT, respectively. This is about equal to the energy required to heat the hydrogen to the postulated conditions.

A pressure vessel has been designed for this example by applying some criteria that had been used in the Helios system.* The result is a steel pressure vessel with a wall thickness of 3.66 cm (1.44 in.) and a mass of 12,800 kg. Detailed hydrodynamic calculations have been performed to study the effect of shock waves generated in the hydrogen and the peak pressures which result from their reflection off the pressure-vessel wall. The calculated peak pressure of the radial pressure wave in the wall is 2.5 kilobars (36,250 psi), which should not cause failure.

The difference of 5430 kg between M_a and the pressure-vessel mass is reserved for nozzle, momentum conditioner, and related structures. In addition, some part of the payload mass must be reserved for the laser system, other structures, and perhaps shielding.

*Helios: An internal concept studied in parallel with the Project Orion.[4] Conventional fission explosions were utilized with a yield ranging from 1 to 40 equivalent tons of TNT and a repetition rate of about 10 sec or longer. A plug in the nozzle was used to allow recharging of the vessel with the propellant (hydrogen). These constraints made the Helios system very large.

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APPENDIX A

LASER CONCEPTS AND CONSIDERATIONS

Introduction

The essential characteristics of an energy pulse required for thermonuclear ignition are high power and short pulse width. In pulsed fusion reactions a sufficient quantity of material is ionized, confined, and heated to such high levels that nuclear fusion will occur during thermal collision of the ions. Because a laser energy pulse can be concentrated in both space and time, it is believed that a light pulse can be generated which will satisfy the specific requirements for initiating a fusion reaction.

General Features of Laser Pulse

Calculations indicate that the minimum laser energy required to burn D+T in milligram quantities ranges from 2×10^4 to 1×10^6 J. These values are being revised as more optimal means for confining and heating the fusionable material are examined. The D+T reaction is usually considered because its fusion threshold is lower than that of the D+D or D+^{3}He reactions. The optimum time width of the laser pulse is thought to be between 10^{-10} and 10^{-7} sec, to take advantage of the resonant absorption which occurs near the plasma frequency of the medium. An alternative approach consists of using longer pulse widths for direct coupling of the field to the ions, thereby predominantly heating the ions.

Laser Systems Under Study

The laser system must possess some basic characteristics if the requirements for the light pulse are to be met. Because the energy needed is high, a reasonable efficiency in converting input energy into laser energy is essential. In attempting to minimize the energy required to trigger the fusion reaction, pulse-shaping is critical. Pulse-shaping and timing are performed in an initial, low-power stage (oscillator) before the pulse is transmitted through one or more successive amplifier stages where its desired high power is produced.

Three types of lasers are under consideration: (1) optically pumped Nd/glass, (2) electrically pumped CO_2/N_2, and (3) chemically pumped lasers using various gas reactants. Laser pumping is the method of providing the input energy required to excite the material to its upper lasing level.

The Nd/glass laser transition is centered at a wavelength of 1.06 μ (1.06×10^{-6} m). The state of the art of these lasers for producing high-power, ultrashort pulses is quite advanced. These systems are optically pumped via xenon flash tubes and at present can produce 200 J in a few picoseconds. However, the possibility of considerably increasing the output appears limited. It is anticipated that the maximum output obtainable will be of the order of 2000 J in a few picoseconds because of the energy density limitation in the glass coupled with the limitation of uniformly pumping the glass volume. Also, the energy input for the required high-power pulses would be excessive because the overall efficiency of these systems is only $\sim 0.3\%$. Thus it is not anticipated that a Nd/glass system could produce more fusion yield energy than that required to power the system. Neodymium/glass lasers will therefore be used primarily in experiments to study the interaction of high-energy laser pulses with matter, and to verify the correctness of the mathematical models used to calculate laser-initiated fusion. To date, neutron production has been observed in experiments where laser pulses from Nd/glass systems have been focused on fusion pellets.

The CO_2/N_2 laser utilizes a resonant energy transfer between the first vibrational state in N_2 and the upper energy coincident laser state in CO_2. The laser transition is centered at a wavelength of 10.6 μ (1.06×10^{-5} m). Because the energy is distributed among a large number of rotational states, it is necessary to promote transitions among very many of these states to obtain high efficiencies with short pulses. This can be done by operating at high pressures (1 to 10 atm) to collision-broaden the lines, and then mode-locking the laser output so that all rotational lines will lase simultaneously. Commercial CO_2/N_2 lasers usually operate at pressures of only 10^{-2} atm.

The pumping scheme under study for the high-pressure CO_2/N_2 laser is an electrical discharge in which the ionization is produced by a high-energy electron beam, while the discharge-electron temperature and energy pumping are provided by an applied

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electric field. The overall efficiency of this laser should be ~ 10% and the maximum output is expected to be 10^5 J in several tenths of a nanosecond. The output per volume of CO_2/N_2 operating at a pressure between 1 and 10 atm is expected to be 10^5 J/m^3.

The chemical laser is self-pumped, utilizing the energy output of the chemical reaction of the materials which form the laser medium. The requirements of the chemical reaction are that it be very fast (explosive) and that the resulting medium have a large excited population in the upper level of a possible laser transition. The chemical reaction can be initiated either by a light pulse, an electrical discharge, or an electron beam propagating through the active medium. The energy in the upper laser-vibration level is distributed among a number of rotational states (though not as many as in CO_2), and it is therefore advantageous to operate at higher pressures (1 atm) in order to extract maximum energy by the lasing action. Research attention in chemical lasers has been directed mostly to reactions of hydrogen with the halogens, specifically F_2, Cl_2, and $(CN)_2$. To increase the energy density within the medium, the chain-branching reactions, e.g., $N_2F_4 + H_2$, seem quite promising. The HF laser transition is centered at a wavelength of 2.7 μ. The overall efficiency of chemical lasers is expected to be at least 10%, and outputs of 10^6 to 10^7 J in a fraction of a nanosecond appear to be possible. One promising concept for a chemical laser amplifier involves the traveling-wave ignition of the chemical reaction, followed immediately by the laser pulse to sweep the cavity. The unit output of a typical chemical system operating at 1 atm is expected to be 2×10^6 J/m^3.

Laser Characteristics Relating to Propulsion Applications

Of prime importance for any laser-pulsed thermonuclear space propulsion system are weight and power requirements. Because none of the laser systems being considered is sufficiently advanced for use in this application, detailed planning for incorporating the laser into a spacecraft must await future laser developments. However, certain fundamental questions should be considered now. The laser could either be expended with each explosion or could remain on board the spacecraft for continued reuse.

For an expendable system, the problems of focusing the laser beam onto the fuel pellet would be essentially eliminated because the laser, the fuel pellet, and the propellant material could be constructed as one unit. Such an approach appears essential for internal pulse-propulsion systems, where propellant material completely surrounds the energy device. Also, from a propulsion viewpoint, it would be much more economical to impart high velocity to any not reusable laser medium (as in a chemical laser) before expelling it (because the explosive source is essentially energy unlimited) rather than to expel the material with negligible propulsion contribution. The generation of sufficient electrical power within the unit and cost reduction to a level where the disposal of laser components becomes economically feasible are some of the problems to be encountered.

For a non-expendable laser, the principal problem is the focusing of the laser energy onto the fuel pellet and the protection of the laser from the ensuing explosion. Lightweight expendable mirrors could solve both problems for an external pulse-propulsion system, but the problems would be more acute for an internal system because the reaction must take place in an enclosed volume that is filled with propellant material.

Of the two laser systems presently under development, the chemical laser appears to be more suitable for a space-vehicle application than the electrically pumped CO_2/N_2 system. It represents the most compact energy source in both volume and weight. The chemical laser is also inherently simpler, requiring little electrical power and storing the required energy within the atoms of the chemical elements. In comparison, the electrically pumped CO_2/N_2 laser requires all of its input energy in the form of electrical power, which is difficult to provide in a space vehicle. Even if large amounts of electrical energy were stored in capacitor banks supplied from a modest power source over a period of time, the mass of such a storage bank for supplying 10^7 J of input energy would be prohibitive. In addition, the overall efficiency of converting the

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energy input to laser energy output should include the large loss incurred by converting thermal energy into electrical energy in the power supply. Thus, a thermally pumped CO_2/N_2, applying the thermal energy directly into laser excitation, might have a better overall efficiency for a space-vehicle laser than the electrically pumped system and, additionally, would obviate the demand for large capacitor storage banks. A singular advantage of either the electrically-pumped or thermally-pumped CO_2/N_2 laser system is the fact that the laser medium is, in principle, reusable after the excess energy residing in the system has been removed.

APPENDIX B

DESIGN CONSIDERATIONS FOR PUSHER-PROPELLANT INTERACTION

General

The design of a pusher-and-pulse system is constrained by the characteristics of the momentum-exchange interaction of the expanding propellant wave with the pusher plate, the design criterion being a maximum momentum exchange with no destructive effects on the pusher. In addition, the optimum mass of the pusher is determined by mission considerations.

Pusher Shape

The maximum momentum exchange between an expanding propellant and a solid pusher plate is obtained when the shape of the plate mirrors the surface of the propellant cloud. Assuming that the propellant expansion is divergent, the shape of the pusher surface should be spheroidal; and the useful momentum exchange at any point on the plate is a result of only the axial component of the local propellant velocity, i.e., the direction of vehicle motion. If the mass of the plate is uniform, the local plate velocity varies and bending occurs. The plate can be designed to react at uniform velocity if its mass distribution is made to vary radially in order to match the axial component of the local propellant impulse. The cross-section of such a plate is then a crescent. However, the stagnation-pressure buildup on the plate surface develops hoop stresses from the edge moments because of the lack of radial restraint. To prevent yielding of the plate material from hoop stresses, the plate must be thick enough at the edge and must increase in thickness toward the center to maintain a uniform axial plate velocity. These opposing requirements demand an unnecessary accumulation of material at the center of the plate to satisfy the yield-strength criterion alone, because the tangential tensile stresses, i.e., hoop stresses if the plate were a sphere, are less than the hoop stresses from the edge moments.

The momentum exchange between the expanding propellant and a flat, circular pusher plate differs from that of the shaped plate in that hoop stresses do not develop. As with the shaped plate, the mass distribution of the disc should correspond to the radial impulse distribution in the axial direction across the surface of the disc. This would prevent bending of the plate. Radial stresses are developed only from the viscous shear of the radial propellant-velocity component as the wave spreads radially across the disc. These radial stresses are much less than the tangential tensile stresses in the shaped plate.

Although the momentum exchange with a shaped pusher is greater than that with a flat disc, the extra weight of the shaped pusher to compensate for hoop stresses may be a more severe penalty than the decreased performance of the flat disc if both subtend the same solid angle of the propellant expansion.

Pusher Stresses

Although the hoop and radial tensile stresses are not a design consideration for the disc pusher, both plane and radial strain waves will be transmitted through the material from the shock of the propellant interaction. If the acoustic thickness (plate thickness/speed of sound of material) is much less than the propellant pulse width (interaction time for propellant stagnation), the plane wave will reverberate repeatedly and dissipate as the stagnation pressure builds to maximum. Therefore, the strength of the material must be sufficient to prevent the "blowing-off" of the pusher back side when rarefaction occurs from the reflection. If the radial waves traverse the plate in

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times exceeding the pulse width, stress concentrations will be produced at the center. However, the radial waves will be weaker than the plane waves, and the stress-concentration level may therefore be less than the yield strength. This stress concentration from radial strain waves must be carefully analyzed for any pusher-plate design.

Pusher Ablation

The energy release from the stagnating propellant at the pusher surface causes the pusher surface to heat with attendant ablation of pusher material. To prevent this loss, the plate surface should be coated with an ablative material of high heat capacity, low thermal conductivity, and large Rosseland opacity at the stagnation conditions. Preferably, this material would adhere to the plate and be of sufficient thickness to last throughout the firing pulses for the mission. If such a material is unavailable, an ablative film could be deposited on the pusher surface between each pulse or after a group of successive pulses.

The ablation phenomena, although complex, are predictable for stagnating plasmas with incident energy fluxes typical of plasmas driven by high explosives. An analytical approximation developed in the Orion program[2] successfully predicts ablation rates for experiments with propellant velocities up to 4×10^6 cm/sec. In applying this expression for energy fluxes from a nuclear-pulse-driven plasma, the ablation rates become unpredictable at propellant velocities greater than 1.5×10^7 cm/sec.

The ablation phenomena occur in stages. During the early phases, the phenomenon is predicted by a kinematic model, involving particle-to-particle collision transfer of kinetic energy and indicating an exponential increase in the ablation rate. As the propellant plasma, due to compression at the pusher surface, heats to temperatures in the eV range, radiation transport predominates, resulting in a net decrease in the ablation rate. Next, as the ablating material increases in temperature, the ablation rate is determined by diffusion of the ablating material into the propellant. For the Orion configuration, the bulk of the mass was calculated to be ablated by the diffusion process. With the Orion analytical technique, at velocities greater than 1.5×10^7 cm/sec, convective-flow instabilities are calculated, thus rendering the diffusion model invalid.

Other Engineering Aspects

Because of the inherent high acceleration of the pusher, a shock-absorber system must be designed to reduce accelerations of the payload mass to tolerable levels for manned flight. This should be accomplished with a minimum dissipation of energy. The attachment of the shock-absorber system to the pusher plate requires a minimum edge thickness, which, in turn, is a factor in determining the pusher mass. The kinematics of the shock-absorber system is strongly dependent upon the pulse period, one of the free parameters of a pulsed-propulsion system. Because the pusher acceleration is inversely proportional to the pusher mass, a minimum pusher mass will be determined by the maximum acceleration that the pusher may be subjected to in order to ensure the structural integrity of the shock-absorber system.

Although hoop and tensile stresses theoretically are not developed, a mismatch of the propellant wave and the pusher from pulse-system positioning and nuclear-yield tolerances will result in bending stresses. These criteria probably will not affect the pusher design, but must be analyzed to determine the positioning and yield tolerances.

Relevant Information from Project Orion

A wealth of information is available about the interaction of propellant and pusher from the design studies of the Orion project. Unfortunately, much of the information is empirical and relevant only to the reference vehicle designs, and therefore may not be applicable to pulsed systems in general. However, some calculational techniques, e.g., the kinematics of the shock-absorber system and the strain propagation in the pusher, should be applicable.

The information from Orion ablation experiments is relevant only for verifying the analytical technique, and is largely empirical. The lack of experimental data for propellant velocities of interest ($> 10^5$ m/sec) resulted in a conservative constraint on propellant velocity (1.5×10^5 m/sec) for designs of Orion vehicles.

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APPENDIX C

NEUTRON ENVIRONMENTAL CONSIDERATIONS FOR THE EXTERNAL SYSTEM

This appendix presents some general conclusions regarding the effect of neutron environmental considerations upon the performance and overall system configuration of the external pulsed-propulsion concept. Results from initial calculational studies leading to these conclusions are also given.

The calculations generally estimate relative quantities (e.g., neutron reaction rates, attenuation factors, and weight-effectiveness per source neutron). An absolute source intensity must be assumed, however, to estimate neutron effects (e.g., temperature rise, radiation damage, and dose rate). A reference mission was therefore selected consisting of 1275 (identical) pulses over a thrust period of 20 min, with 44 tons (TNT equivalent) of energy transferred to the propellant in each pulse. This energy was further assumed to include all non-neutron energy from the pulse and no neutron energy.

The D+T reaction was taken as the energy source, which introduces the most severe neutron environmental problems of all possible fusion sources. The results therefore are indicative of the upper-limit, or worst, case, and will provide reference values for comparison with calculations based upon other fusion reactions. Furthermore, a D+T burn is the easiest to achieve, and this choice could be, therefore, realistic.

The implied total energy for each pulse in the assumed mission is 220 tons/pulse (80% in 14.1-MeV neutrons and 20% in non-neutron energy). The following source terms result.

Neutron energy/pulse $= (0.8)(220) = 176$ tons

or $(176)(4183) = 7.36 \times 10^5$ MW-sec

or $= 7.36 \times 10^{11}$ J

or $\frac{(7.36)(10^{11})}{(1.6)\ (10^{-13})} = 4.60 \times 10^{24}$ MeV.

Neutrons/pulse $= \frac{(4.60)(10^{24})}{(14.1)} = 3.26 \times 10^{23}$ neutrons/pulse.

Total neutron energy $= (1275)(176) = 2.24 \times 10^5$ tons

$= 9.39 \times 10^8$ MW-sec

$= 9.39 \times 10^{14}$ J

$= 5.87 \times 10^{27}$ MeV.

Total neutron source $= 4.16 \times 10^{26}$ neutrons.

These magnitudes are so large that "attenuation" in the usual sense of slowing down and/or capturing the neutrons implies non-survival of the attenuating medium. For example, the (uncooled) pusher plate in the aforementioned reference case can survive a total heat load of $\sim 10^9$ J, or some six orders of magnitude less than the neutron energy from the source. An unshielded manned payload located at ten meters from the source would receive a total dose of $\sim 10^{12}$ rem, which is 11 to 12 orders of magnitude higher than could be allowed. Furthermore, capture γ-ray sources resulting from the absorption of even a minute fraction of these generated neutrons (perhaps as few as 10^{-6} to 10^{-4}) would produce heating problems at most locations throughout the spacecraft.

Two basic approaches toward alleviating these problems are apparent: (1) An expendable attenuating material can be interposed between the source and the pusher plate for each pulse, and (2) the pulse unit-spacecraft configuration can be stretched out so that critical parts subtend relatively small solid angles from the pulse location. Expendable shielding mass will reduce the system specific impulse directly as this mass is increased; the stretching-out implies payload losses due to added structure, and, more importantly, a possible pulse-unit design problem since it may be more difficult to collimate the pulse-unit mass (to utilize effectively all non-neutron energy from the pulse as assumed earlier) than if the system were more compact.

There are other fundamental considerations in such a propulsion system which also require both expendable mass at the energy source and elongation of the spacecraft. For example, a propellant mass must be expended to provide the impulsive force which drives the spaceship, and the shock-absorber stroke must be long to keep the payload acceleration within acceptable limits. Presumably, a major

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fraction of this propellant mass can also be used as shielding. In addition, typical pulse-pusher plate separation distances and overall spacecraft dimensions implied by other considerations are such as to provide important geometrical attenuation of the neutron fluxes that impinge upon the pusher plate and upon the payload.

Some basic overall considerations thus begin to emerge. We are led to consider a point source of neutrons on the axis of a circular pusher plate, at a distance, a, from the plate as shown in Fig. C-1. The plate subtends a relatively small solid angle from the source, as defined by the angle θ. For maximum shielding effectiveness per unit mass (this will be elaborated upon later), the pulse-unit mass or some portion of it is placed in a cone of height, ℓ, with its apex at the source.* A payload region is located at a distance, b, from the position of the pusher plate at the time the fusion pulse occurs, or at a distance a + b from the neutron source. The intervening space is occupied by a shock-absorber system, the spacecraft structure, propellant storage, etc.

The principal neutron attenuators for each main component can be identified as follows:

Component	Attenuators
Pusher plate	The "propellant"
	The separation distance, a
Shock absorber	The above
	The pusher plate
Main spacecraft (including unused pulse units)	The above
	The shock-absorber system
	The distance c
Payload	The above
	Main spacecraft structure
	Unused pulse units
	The distance d

If the pertinent solid angles are kept small enough, material attenuation between the source and any position in the system (as measured along the x-axis) will be primarily via scattering-out of the solid angle subtended from the source by the component in question; i.e., all neutrons which

*In the remainder of this appendix, "propellant" will mean only that portion of the pulse unit which is in the indicated shielding cone.

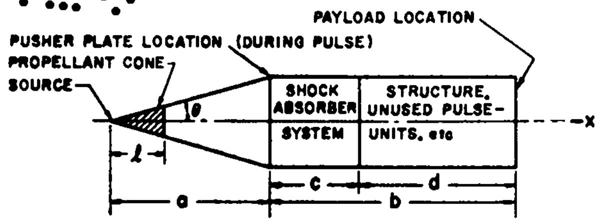

Fig. C-1. Basic spacecraft schematic.

have undergone one, or at most a few, scattering events escape into space and can no longer intercept the target area. Ideally, every scattering event would, with high probability, remove that neutron from consideration and the neutron fields in the system would then be produced predominantly by uncollided 14-MeV neutrons.* As we shall see later, this ideal can be approached closely with a rather straightforward, albeit elongated, system configuration.

The neutron environmental problem areas as currently envisioned for such a spacecraft can be classified as follows:**

1. Neutron heating of the pusher plate.
2. Neutron heating and radiation damage in the remainder of the spacecraft (the stored pulse units may pose a particular problem).
3. Payload doses (e.g., crew shielding).
4. Neutron capture γ-ray effects. (This may be an important factor in Items 2 and 3.)
5. Neutron activation levels, particularly from the standpoint of long-term crew doses. (This is closely related to Item 4).

Although these areas are clearly not independent of one another, they are being emphasized in the order given. Most calculations to date have been directed toward Item 1; the remainder of this appendix will deal mainly with these results and with some results for Item 3.

In addition to the single-scattering neutronic criterion mentioned earlier, smallness of the solid angle defined by the angle θ might also be determined on the basis of geometry required to minimize

*Indeed, this is the principal neutronics criterion by which the smallness of the solid angle can be judged.

**This listing is in order of decreasing expected seriousness (on a rather intuitive basis) from the standpoint of overall mission performance and conceptual feasibility.

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TABLE C-I

PROPELLANT ATTENUATION CHARACTERISTICS

Cone half-angle (θ), deg	15	15	15	30	30	15	15	15	15	15	15	15
Propellant material	H_2O	H_2O	H_2O	H_2O	H_2O	CH_2	CH_2	^6LiH	^6LiH	^6LiH	^6LiH	Be
Propellant density (ρ), g/cm^3	1.0	1.0	1.0	1.0	1.0	0.91	0.91	0.65	0.65	0.65	0.65	1.87
Propellant cone length (ℓ), cm	10	20	30	10	20	20	30	20	30	50	70	30
Propellant attenuation factor (A)	2.5	6.2	15.9	2.2	5.0	7.4	19.8	8.0	22.8	167	1339	102
Fractional solid angle subtended by pusher plate, %	1.7	1.7	1.7	6.7	6.7	1.7	1.7	1.7	1.7	1.7	1.7	1.7
Overall attenuation factor	144	365	935	33	74	433	1165	469	1339	9860	78,800	6000
Fraction of neutrons at pusher plate that are essentially uncollided	0.99	0.98	0.80	0.89	0.78	0.94	0.90	0.95	0.89	0.74	0.66	0.72
$\Sigma - (\ell n\ A)/\ell$, cm^{-1}	0.0896	0.0912	0.0922	0.0779	0.0805	0.1001	0.0995	0.1040	0.1042	0.1024	0.1029	0.1542
Total (14 MeV) macroscopic x-section of propellant (Σ_t), cm^{-1}	0.0962	0.0962	0.0962	0.0962	0.0962	0.1051	0.1051	0.1156	0.1156	0.1156	0.1156	0.1863
Σ/Σ_t	0.93	0.95	0.96	0.81	0.84	0.95	0.95	0.90	0.90	0.89	0.89	0.83
Propellant mass (m), g	75	602	2030	349	2790	548	1848	391	1320	6110	16,770	3797
$m/(\ell n\ A)^3$	104	99	96	712	670	68	69	43	43	46	45	38

the propellant mass needed to achieve a given neutron attenuation. A corollary question arises as to the sensitivity of the propellant mass to propellant shape, for a given overall attenuation. To address this question, equations were derived which relate both total propellant mass and the radial variation of pusher-plate neutron current to propellant shape (e.g., convex or concave cone face) and to θ. These geometrical relationships become complex, and overall optimization is not straightforward. However, some principal conclusions are apparent: (1) the propellant mass is insensitive to variations in the shape of the cone's face (thus, the simple cone is very nearly optimum), and (2) the radial distribution of pusher-plate heating is relatively flat and also insensitive to the cone shape, as well as to θ. This is true for all values of θ considered, up to $\theta = 30°$.

We can arrive via these overall considerations at some useful simplified relationships. If the neutrons arriving at the pusher plate are essentially uncollided, the propellant attenuation factor is very nearly just

$$A = e^{\Sigma \ell} \qquad \text{(C-1)}$$

where Σ is a constant appropriate to the propellant material, and ℓ is shown in Fig. C-1.

As we shall see later for several cases of interest, Σ is (almost) independent of ℓ and is approximately equal to the macroscopic total (14-MeV) neutron cross-section of the propellant, Σ_t. Further, if the propellant is a simple cone, its mass is given by

$$m = \frac{\pi\rho}{3} (\tan\theta)^2 \ell^3 \qquad \text{(C-2)}$$

$$= \frac{\pi\rho}{3\Sigma^3} (\tan\theta)^2 (\ell n\ A)^3$$

where ρ is propellant density and A is the propellant attenuation, defined here as (pusher-plate neutron-energy current with no propellant cone present) ÷ (pusher-plate neutron-energy current with the propellant cone in place). Note that, for a given propellant attenuation factor, the propellant mass depends very simply upon θ, i.e., $m \propto (\tan\theta)^2$, because ρ, Σ, and A are independent of θ.

Some calculated quantitative results may help to clarify the above discussions.* Table C-I shows pertinent results for some typical configurations (see Fig. C-1).

*Because of the diversity and multidimensional character of the geometries of interest, and because of the rather substantial attenuation factors likely to be encountered, Monte Carlo has been selected as the calculational method of choice.

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From Table C-I we see the following:

1. Σ is generally within 10% of Σ_t (at least for $\theta \sim 15°$).
2. Between 70 and 100% of the neutrons incident upon the pusher plate are uncollided, depending upon the propellant attenuation factor.
3. The relative shielding performance per unit mass of the various materials is evident, e.g.,

$$H_2O: \quad m \doteq 100\ (\ell n\ A)^3, \quad \theta = 15°$$
$$\doteq 700\ (\ell n\ A)^3, \quad \theta = 30°$$
$$CH_2: \quad m \doteq 70\ (\ell n\ A)^3, \quad \theta = 15°$$
$$LiH: \quad m \doteq 45\ (\ell n\ A)^3, \quad \theta = 15°$$
$$Be: \quad m \doteq 40\ (\ell n\ A)^3, \quad \theta = 15°$$

The approximate validity of Eqs. (C-1) and (C-2) for the configurations of Table C-I allows us to derive a convenient figure of merit with which to screen potential propellant shielding materials. For this purpose we assume

$$\Sigma \doteq \Sigma_t\ (14.1\ \text{MeV})$$
$$= \frac{0.6023\ \rho}{M}\ \sigma_t$$

where M is the propellant molecular weight and σ_t is the microscopic total cross-section for 14.1-MeV neutrons.

Thus,

$$A = e^{\Sigma \ell} = e^{(0.6023\ \rho\ \sigma_t/M)\ell}$$

or

$$\ell = \frac{M}{0.6023\ \rho\ \sigma_t}\ \ell n\ A.$$

Also,

$$m = \frac{\pi\rho}{3}\ (\tan\ \theta)^2\ \ell^3 = 0.0752\ \rho\ \ell^3,$$

for $\theta = 15°$.

Thus,

$$m = \frac{0.0752\ \rho}{(0.6023)^3 (\rho\ \sigma_t/M)^3}\ (\ell n\ A)^3$$
$$= 0.344\ (\ell n\ A)^3\ \frac{\rho}{(\rho\ \sigma_t/M)^3}\ .$$

For two propellants, 1 and 2, and a given attenuation, A, we have

$$\frac{m_1}{m_2} = \frac{[\rho/(\rho\ \sigma_t/M)^3]_1}{[\rho/(\rho\ \sigma_t/M)^3]_2}\ .$$

The quantity $\frac{\rho}{(\rho\ \sigma_t/M)^3}$ is a measure of the propellant shielding mass required to achieve the given attenuation factor, A. Table C-II gives values of this parameter for several representative materials.

TABLE C-II

FIGURE-OF-MERIT VALUES FOR VARIOUS SHIELDING MATERIALS

Material	Molecular Weight (m)	Density (ρ), g/cm^3	Cross Section σ_t(14.1 MeV), b	$\rho/(\rho\ \sigma_t/m)^3$
LH_2	2	0.069	1.4	612
^6LiH	7	0.65	2.08	90
^7LiH	8	0.75	2.16	80
Be	9.0	1.86	1.49	64
B	10.8	2.54	1.34	80
C	12	2.2	1.30	163
CH_2	14	0.91	2.7	168
H_2O	18	1.0	2.89	242
Al	27	2.7	1.71	540
Ti	47.9	4.5	2.26	470
Cr	52	7.1	2.42	197
Fe	55.9	7.9	3.14	90
Ni	58.7	8.9	2.68	133
Cu	63.5	8.9	2.96	125
Zr	91.2	6.4	4.04	280
Nb	92.9	8.4	2.1	1227
Mo	96	10.2	4.03	130
Cd	112	8.6	4,4	223
Ta	181	16.6	5.07	165
W	184	19.3	5.3	112
Pb	207	11.4	5.32	453
U	238	18.7	6.23	159

Since Eq. C-3 is approximate and, more importantly, because the relative importance of (n,2n) reactions varies considerably from species to species, these values are not an exact measure of relative propellant effectiveness. However, they are consistent with the calculated relative masses in Table C-I, with Be being the predicted best propellant shielding material on a mass basis. However, detailed calculations indicate that Be is only about 5 to 10% better than LiH.

As an approach to examining a more realistic pulse unit, attenuation curves* have been calculated for the configuration of Fig. C-2. The variation

*Here, the propellant attenuation factor is defined as (pusher-plate heating with no propellant cone present) ÷ (pusher-plate heating with the propellant cone). Note the difference from the previous definition based upon neutron current

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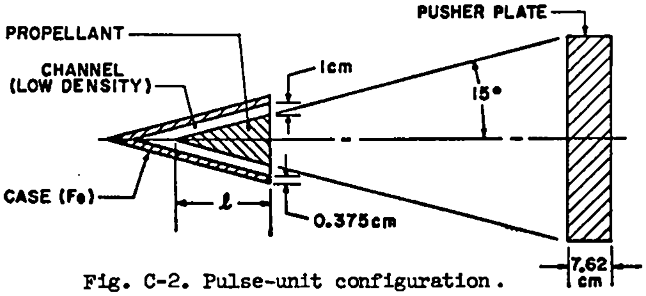

Fig. C-2. Pulse-unit configuration.

of Σ, m_{prop}, and m_{case}, with ℓ is shown in Fig. C-3 for a LiH propellant. Using these curves and the neutron source intensity presented earlier (appropriate to a D-T energy source and the given reference case), the pusher-plate heating curve of Fig. C-4 can be plotted. (Figure C-4 also shows one calculation using Be as the propellant.)

The resulting pusher-plate heat load is too high to be tolerable. It has been estimated that the pusher plate of Fig. C-2 (uncooled) can survive a neutron heating load of $\sim 10^9$ J during the course of the reference mission. Thus, even if we assume perfect hydrodynamic collimation, so that the entire 20 kg of pulse-unit mass in the reference case can be configured as in Fig. C-2, the pusher-plate heating from Fig. C-4 is still too large by a factor of ~ 25.

The implications are fairly obvious. Either the pulse-unit shielding mass must be substantially increased (introducing a severe performance penalty), or the pusher plate must have a cooling mechanism that acts during the course of the propulsion period (which might seriously compromise its structural integrity), or the pulse energy must be obtained from a fusion reaction other than D+T (e.g., D+^{3}He). Roughly 40 kg of pulse-unit shielding mass, or $\sim 10^{10}$ to 10^{11} J of pusher-plate cooling, or some appropriate combination of these approaches is required if D+T fusion is to be the energy source. Certainly, if the primary energy comes from the D+^{3}He reaction, or even from D+D, the pusher-plate heating problem is greatly relieved; however, no quantitative results are as yet available for such systems.

In concluding this discussion, a few comments will be made regarding the payload (crew) neutron dose to be expected in the D+T driven system. Some initial calculational results are also available.

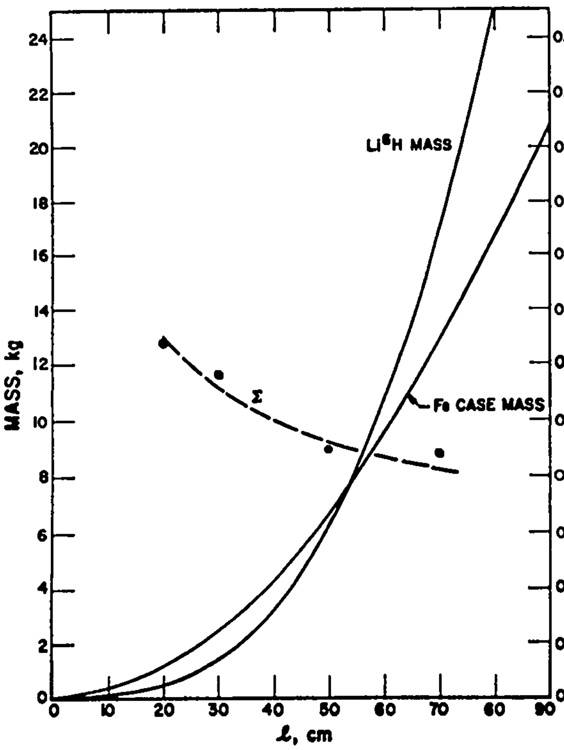

Fig. C-3. Variation with ℓ of masses and Σ for configuration of Fig. C-2.

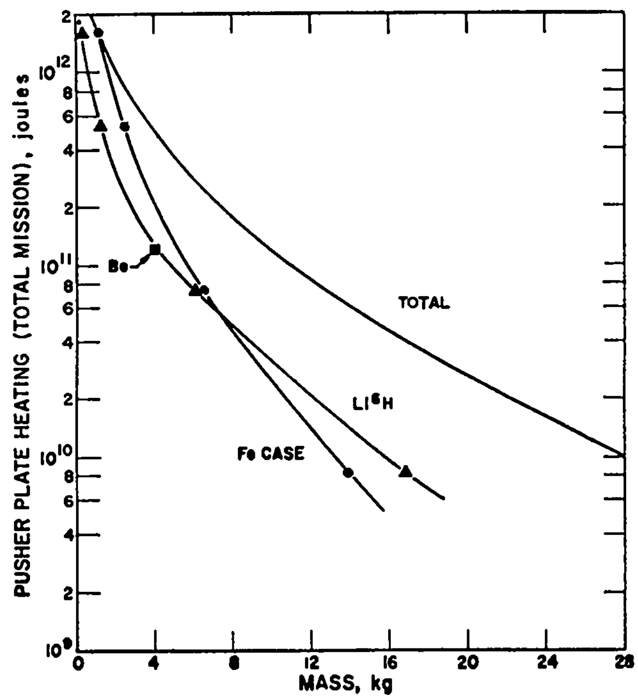

Fig. C-4. Neutron heating in pusher plate vs masses of components, in configuration of Fig. C-2.

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TABLE C-III

PAYLOAD BIOLOGICAL DOSE CALCULATIONS

Propellant Material	Length (ℓ), cm	Propellant Mass, kg	Unshielded Payload Dose, rem	Shield Material	Shield Thickness (X), cm	"Shield" Masses, kg Type (1)	Type (2)
LiH	20	0.39	7.8×10^{8}	LiH	205	21,800	2790
LiH	20	0.39	7.8×10^{8}	Be	145	44,300	5540
LiH	50	6.11	4.09×10^{7}	LiH	175	18,600	2350
LiH	50	6.11	4.09×10^{7}	Be	125	38,200	4736
Be	30	3.80	7.02×10^{7}	LiH	181	19,200	2440
Be	30	3.80	7.02×10^{7}	Be	129	39,400	4900
Be	60	30.40	1.58×10^{6}	LiH	143	15,200	1900
Be	60	30.40	1.58×10^{6}	Be	102	31,100	3830

Once again it is evident that elongation of the spacecraft is advantageous. Indeed, this problem is quite analogous to the pusher-plate problem in that the required attenuation can best be achieved through a small solid angle (subtended by the payload) and through material attenuation via scattering-out of this solid angle.

However, there are fundamental differences between these two problems. Much larger attenuation factors must be attained for payload protection, and the resulting implications are quite different from those for the pusher plate. Any shielding mass that must be added for payload protection does not affect performance by diminishing the factor μ. Rather, it simply replaces (roughly pound-for-pound) "useful" payload that can be carried. Furthermore, intervening mass (e.g., stored pulse units) that must be carried anyway will substantially reduce (or eliminate) the payload shielding mass that must be carried.

To help put these considerations in perspective, we consider the "generalized" spacecraft configuration of Fig. C-1, with the pulse-unit and pusher-plate configuration of Fig. C-2. We assume, consistent with the aforementioned reference case,

$\theta = 15°$

Pusher plate diameter = 4.572 m

Payload diameter = 4.572 m (area = 1.63×10^{5} cm^2)

a = 8.5 m

b = 60 m

Pusher-plate thickness = 7.62 cm

Source = D+T neutron source (described earlier)

Flux-to-dose conversion factor (~ 14-MeV neutrons) $= 4.2 \times 10^{-8}$ rem/(neutron/cm^2)

ℓ = variable.

The fractional solid angle subtended by the payload from the source is 2.77×10^{-4}, and for a total dose of ~ 1 rem, the additional material attenuation required is

$$A_o = \frac{(4.16)(10^{26})(2.77)(10^{-4})(4.2)(10^{-8})}{(1.63)(10^{5})} = 2.97 \times 10^{10}.$$

This must be provided by the combined propellant-pusher plate-spacecraft material, plus any necessary additional shielding.

The actual material attenuation that might be available cannot be calculated without assuming many of the spacecraft design details. However, the attenuation available in a propellant and pusher plate can be estimated, which gives an indication of how much shielding must be accomplished by the remainder of the system. To do this, a vacuum between the pusher plate and the payload is assumed, and the resulting (upper limit) payload doses are calculated. Table C-III shows such results using the configuration shown in Fig. C-2. Also given in Table C-III are two estimated shielding masses. Shield mass Type (1) is the estimated mass of the given shielding material that would be required at the payload (diameter, 4.572 m) to reduce the dose to 1 rem. Shield

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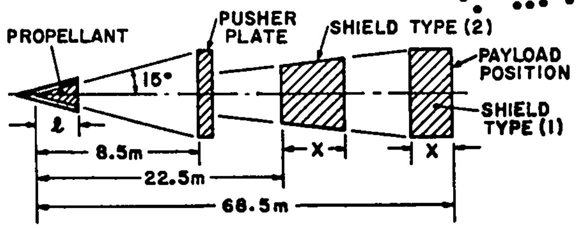

Fig. C-5. Payload shield configurations.

mass Type (2) is the corresponding requirement in a shadow shield placed 15 m from the pusher-plate position at the time of the pulse (see Fig. C-5). Note that Type (1) is indicative of the maximum shield mass that could be implied, whereas Type (2) is a lower limit on such mass. Recall, however, that these shielding masses assume that no other mass is located between the pusher plate and the payload. This is, of course, a gross overestimate for the shielding requirements. Indeed, it seems likely that a much more serious neutron environmental problem will be that of protecting the structure and equipment (including the initial 25,000 kg of stored pulse units) between the pusher plate and the payload location, and that little or no additional shielding mass will be required to protect the payload.

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