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OF THE UNIVERSITY OF CALIFORNIA ○ LOS ALAMOS NEW MEXICO

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By Markus Lujan CIC-14 8-23-95

ASPEN

An Aerospace Plane With Nuclear Engines

(Title Unclassified)

LOS ALAMOS NATIONAL LABORATORY
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OF THE UNIVERSITY OF CALIFORNIA LOS ALAMOS NEW MEXICO

REPORT WRITTEN: May 1961

REPORT DISTRIBUTED: September 27, 1961

ASPEN

An Aerospace Plane With Nuclear Engines
(Title Unclassified)

by

R. W. Bussard

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ABSTRACT

An analysis is presented of the potential performance capabilities of a class of manned, single stage, aerospace planes utilizing nuclear rocket engines for final boost flight into orbit, and comparison is made with chemical-energy-powered vehicles proposed for the same mission. Such vehicles have the capability of multiple reuse from successful flights, and of complete recovery from any non-catastrophic (i.e., airframe structural failure) abort situation.

It is shown that the simplest nuclear-engined aerospace plane (ASPEN) considered is capable of placing 42.3% of its takeoff mass into a 300 n.mi. polar orbit, and that the most advanced versions may be able to carry 52.2% to polar orbit. Such figures imply the delivery of 3% to 17% of takeoff weight as useful payload, depending upon assumptions made for structural weight fraction and shielding.

The simplest ASPEN system uses conventional turbojet engines, hydrogen-fueled ramjet engines, and nuclear rocket engines at performance levels expected in the Rover program by 1965. The reactor is never operated below 100,000 ft altitude; thus the major shielding problem reduces to that of protection against decay radiation after reactor shutdown. This problem is explored and it is shown that shielding may be carried sufficient to allow the use of conventional air bases.

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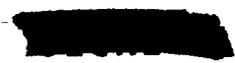
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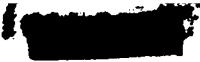
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I. Introduction and Summary

As further progress is made toward the era of manned space flight, the need becomes acute for a reliable, reusable, economical vehicle capable of ferry missions to low Earth satellite orbits. Unless some method is found to accomplish such missions more practically than by use of costly, one-shot, multistage rocket vehicles, as at present, the cost of space travel and transport will remain so high that its growth will be severely inhibited by economic considerations alone. While such considerations will always be of importance, their limiting effect could be greatly reduced if an orbital ferry system could be devised with most of the operational characteristics of present long-range jet transport aircraft. Past studies of methods of achieving this goal lead to the concept of the "orbital airplane", or "aerospace plane"; a vehicle with the following general properties:

1. Manned, piloted, winged, single-stage vehicle.
2. Horizontal ground takeoff from conventional air bases.
3. Flight to high speed within the atmosphere, using air-breathing propulsion systems.
4. Flight from (3) to orbit under rocket power.
5. Retrorocket and drag-glide return from orbit.
6. Horizontal landing at conventional air bases.



Operational flexibility for a workable vehicle with the properties outlined above includes:

- a. May be ferry-transported between bases on air-breathing engine power.
- b. Flight training is possible at any speed up to orbital speed.
- c. Use of multiple air-breathing engines for first phase of flight ensures complete vehicle recovery from any non-catastrophic abort situation during flight.
- d. Human crew allows in-flight judgement decisions to cope with minor emergency situations.
- e. Human crew allows choice of alternate missions during flight or modification of mission goals.
- f. Ready reuse without excessive maintenance.

Unfortunately, analysis (see ASP, 1960, and Sec. II C, following) of the payload performance potential of a variety of chemical-energy-powered aerospace plane (ASP) vehicles generally shows only marginal payload-to-orbit capability even for rather optimistic assumptions about system mass fractions and engine sub-component performance, so long as interest is restricted to the conditions (1) through (6) outlined above. If these conditions or ground rules are changed (e.g., by use of two stage vehicles rather than single stage), it may be possible to demonstrate greater performance potential, but only at the price of loss of vehicle operational flexibility. This is very undesirable because it is principally by virtue of the operational flexibility of a single-stage ASP



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vehicle that the cost of orbital ferry missions can be kept small relative to that for non-reusable systems. If reliability, recoverability, and reuse are lost, there is no economic incentive to consider the aerospace plane concept, since non-reusable multistage pure rocket vehicles can probably be built somewhat more cheaply than can non-reusable ASP vehicles.

In order to make the ASP concept attractive, some way must be found to provide greater performance potential than seems inherent from the use of chemical energy alone. In the body of this report we consider a method of reaching increased performance potential, by the use of nuclear energy in a nuclear rocket engine to power the ASP vehicle in the final phase of its flight to orbit. This aerospace plane with nuclear engine(s) (ASPEN) has all of the properties of the chemical-energy-powered vehicle, and retains all of the operational flexibility and advantages relative to other orbital ferry schemes. The use of nuclear engines does not introduce any difficult operational problems due to radiation hazards, because the nuclear engines are used only above 100,000 ft altitude for the final phase of ASPEN flight. Fission power is not used at takeoff or at landing, and considerable in-flight shielding can be built into the engine as an integral part of the reactor assembly, to reduce dose rates in the vicinity of the vehicle from decay gamma radiation after shutdown.

At takeoff and landing the ASPEN vehicle is essentially a chemically powered aircraft, and as such may be used from conventional air bases. The reactor pressure vessel, which is the installed flight

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shielding, can be made sufficiently thick and sturdy that dispersal of radioactive material is extremely unlikely in the event of a crash of the vehicle. Movable (liquid) shielding material may be added to the installed shield, following landing, to reduce decay gamma dose rates to laboratory tolerance levels on the shield surface, within the nuclear engine compartment, thus allowing rapid maintenance on all parts of the vehicle except the nuclear reactor itself. Reactor replacement may be accomplished at the base without the use of any remotely operable or shielded special equipment, since the reactor to be installed will not be radioactive, while that to be removed is unit-shielded to laboratory tolerance levels.

The performance of several propulsion system combinations is analyzed in this report. The "nominal" system chosen as of prime interest is the simplest one, making use of existing turbojet engines, and current state-of-the-art designs for chemical ramjet and nuclear rocket engines. The nominal vehicle is able to carry 42.3% of its takeoff mass into a 300 n.mi. high orbit, when launched on a polar trajectory. Equatorial eastward launching would yield 44.3% of gross mass into orbit. The nuclear rocket employed in the nominal vehicle produces propellant with specific impulse of 800 sec and is a natural product of the present Kiwi reactor development in the Rover program. A power level of 4900 Mw at a core power density of about 85 Mw/ft³ is adequate to propel an ASPEN vehicle of 500,000 lb takeoff weight. This sort of reactor performance could be achieved in static ground tests by 1965, by an extension of current

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programs, probably sooner than the rest of the vehicle could be made available. As for growth potential, the attainment of 1000 sec specific impulse from the nuclear rocket propellant would enable the placement of 50.3% of takeoff mass into orbit from an equatorial eastward launch, and 54.2% could be carried by use of hydrogen-fueled turbojets and reactor-preheated hydrogen in the ramjet engines as well.

Let us consider the nominal vehicle briefly. Making the reasonable assumptions of 25% of gross mass for airframe and tankage structure, and 5% for chemical engines, leaves 12.3% to 14.3% available for nuclear rocket engine, flight shielding, and payload, depending on our choice of polar or eastward launching. Without undue optimism we need allow no more than 2% of gross mass for the reactor and nuclear engine auxiliary equipment (pumps, valves, etc.), leaving 10.3% to 12.3% for shielding and payload-to-orbit. Using 6.3% of a 500,000 lb vehicle for shielding, we can carry to orbit 20,000 lb for polar launching and 30,000 lb for equatorial eastward launching. If 1000 sec specific impulse were attainable these figures would be 50,000 lb and 60,000 lb, respectively, with the nominal chemical engines, and 70,000 lb and 80,000 lb with advanced chemical engines. The amount of shielding assumed will reduce the dose rate to about 25 rad/hr at 100 ft from the reactor, one day after shutdown (which occurs in orbit). If 10.3% of gross mass is used for shielding, the corresponding dose rate would be about 1.6 rad/hr. The addition of roughly 21 cm (8.3 in) of Hg shielding to the installed flight shield, when the vehicle has landed, will reduce dose rates to laboratory tolerance levels (order of tens of millirad per hour) on the surface of the shield tank.



This range of vehicle performance indicates that ASPEN should be able to fulfill our desires for a practical means of transport to low Earth orbits, even with the simplest propulsion system combination. Expansion of effort should be undertaken toward this goal, with particular emphasis on the integration of Rover program reactor development and planning with requirements for the rest of the ASPEN vehicle. With continued success in reactor development and vigorous implementation of airframe and chemical engine development, it is not unreasonable to plan for flight testing of a complete ASPEN vehicle in the 1966-1968 period. Having such a vehicle at this time would allow economical and reliable performance of orbital rendezvous missions, and thus could greatly speed our long range efforts in manned space flight, as well as serve in a variety of useful and unique military and scientific capacities in near-Earth space.



II. Recoverable, Manned, Orbital Vehicles: Concepts, History, and Status(A) Introduction

Since it was first demonstrated (Goddard, 1936) that high performance rocket engines could be constructed and made to fly, man has speculated on means of employing this type of thrust system for propulsion of vehicles to carry him on extra-atmospheric journeys and return. Nearly all of the earliest thoughts along such lines involved the use of rocket vehicles and pure thrust-lifting flight to accomplish the purpose desired. Indeed, even today, the major man-in-space efforts of both the USSR and the USA are based principally on exploitation of the payload capability of pure rocket vehicles powered by chemical combustion energy. This approach, as we have seen from experience, requires rocket vehicles of at least two stages,* whose gross mass at liftoff must be the order of 20 to 30 times that of the payload delivered into low Earth satellite orbits. Larger fractional masses delivered to orbit by this method can be achieved only if striking (and presently totally unforeseen) advances are made in chemical rocket propellant specific combustion energy. It seems probable that the payload (excluding vehicle structures) delivery capability of practical chemical rockets will never exceed about 5% of gross mass for delivery from ground launching to low Earth orbits. To accomplish even this, two-stage vehicles of rather low structural

* For use of conventional liquid propellants of high density (e.g., Saturn-type vehicles). Use of oxygen/hydrogen propellant may allow payload to orbit fractions the order of half the value cited above, with refined design and construction. Nuclear rockets are inherently better than this for such application (see later discussion).

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fractional mass must be used. Unless methods are developed for the successful (both technically and economically) recovery and reuse of one and/or both of these relatively flimsy structures, new vehicles must be constructed and utilized for each ground-to-orbit transport mission. In this circumstance the cost of manned space flight will forever remain very, very high in comparison with the cost of other forms of human transport.

How much better it would be if we could devise a single-stage vehicle of sufficiently high performance to allow the transport of payload fractions of 10% to 15% (about the same as for present long range use of conventional commercial and military jet transport aircraft) under full control of a manned crew, and with sufficiently sturdy structure to assure reliable and inexpensive recovery and reuse capability. In short, we seek a vehicle with most of the characteristics of modern long-range jet transport aircraft, including the advantages attendant to the use of a human pilot, but capable of flying into a low Earth orbit, and returning to base for reuse. The purpose of this report is to demonstrate that this is not an idle goal; that such a vehicle is within reach of our present knowledge and technology, and does not require the development of exotic new propulsion schemes and equipment, or the construction of vehicles markedly less sturdy than those we are capable of building at present.

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(B) Early Studies

The first serious study of the problem of efficient manned flight at the borders of space was made nearly 20 years ago, by Eugen Sänger. His investigations of the basic principles of propulsion and flight of high speed rocket-powered aircraft were published in the book, "Raketenflugtechnik" (Sänger, 1933). He continued this work throughout the middle and late 1930's, and in the early 1940's together with his co-worker Irene Bredt. Under German Air Force sponsorship during World War II, he studied a specialization of the earlier general analyses to the case of a large, single stage, manned rocket bomber, capable of flight around the Earth and return to its launching base (Sänger and Bredt, 1944). Considerable experimental work on high performance rocket engines and supersonic aerodynamics was also carried out for the purpose of verifying assumptions or supplying input data for the study.

The vehicle of interest was not truly orbital or single-stage, but was sled launched at 500 m/sec, and climbed under rocket thrust to an altitude of 40 to 120 Km in 4 1/2 to 7 1/2 minutes depending on the specific impulse (I_{sp}) assumed for the propellant combination. Since I_{sp} was left as a parameter throughout the study, the payload capacity appears as a function of I_{sp} . Results from Sänger's work (Sänger and Bredt, 1944, p. 96) are shown in Figure 1, which gives the fractional payload carried to the velocity shown, for three different values of propellant specific impulse. We note from the figure that no payload at all can be carried to orbit unless I_{sp} is above about 400 sec

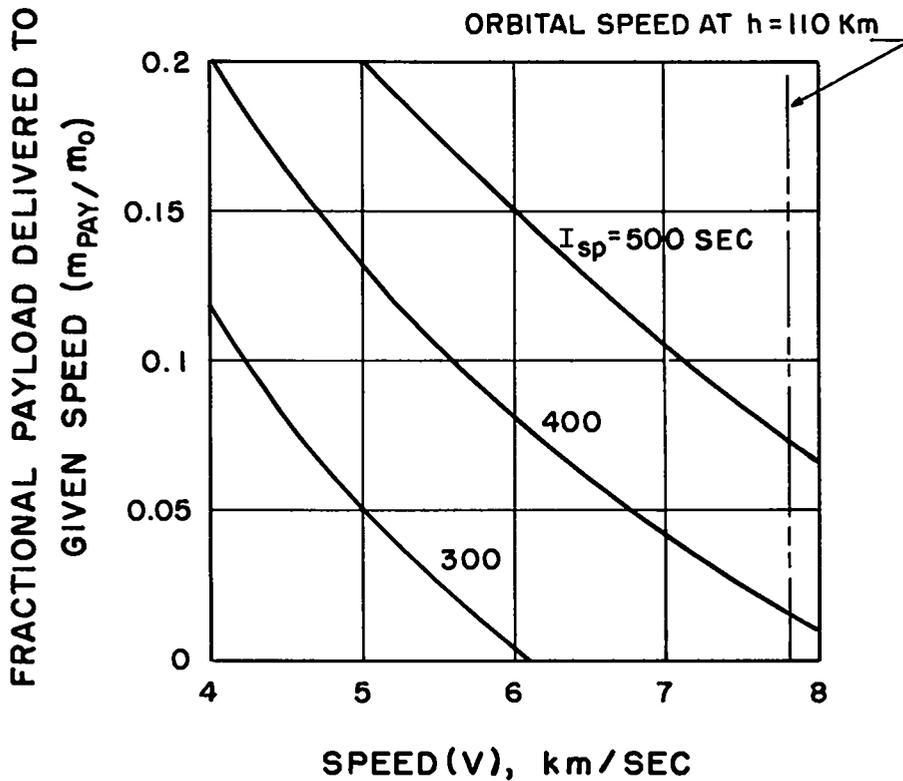
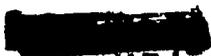
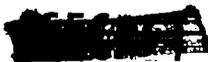


Fig. 1 Payload capability of Sanger-Bredt bomber.



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and that even 500 sec gives only $(m_{\text{pay}}/m_0) = \overline{m}_{\text{pay}} = 0.072$. These results coupled with the fact that the vehicle was allotted a dry weight (structure plus propulsion) of only 1/10 gross weight (0.07 airframe, 0.025 powerplant, 0.005 auxiliary equipment) make the concept of only historical interest for our purposes here, especially since it has not yet been found possible to attain I_{sp} much above 350 sec from the hydrocarbon-based fuels presumed in the rocket aircraft layout.*

However, these early studies were the first to recognize and employ the advantages of winged vehicles to provide atmospheric lift in place of thrust lift, and thus the first to attempt exploitation of the atmosphere in the flight of vehicles designed to carry man to orbital speeds at high altitude.

(C) The Aerospace Plane - 1960

In the years since Sanger's work, progress has been rapid in the development of air-breathing jet engines. Work begun in the late 1940's on both ramjet and turbojet powerplants (as well as a variety of ducted and hybrid systems) has produced engines of both types which are reliable and efficient. Even more important has been the understanding of high speed powerplant aerodynamics acquired by virtue of the painstaking research and development activity which has gone into this effort. Today we have production model turbojet engines capable of

* Sanger considered a wide variety of fuels, including liquid H_2 , Al metal, and B and Be metal additives to hydrocarbons, but sized his vehicle for a propellant mean density of about 68 lb/ft³.



operation up to $M \sim 3$ or so with conventional hydrocarbon-based jet fuels (e.g., G.E. J-93 with JP-6) and production model ramjet engines for flight up to $M \sim 5$ at high efficiency (e.g., Typhon and/or Bomarc engines). Tests of engine components, such as inlets, combustion chambers, and exit nozzles, have been made at considerably higher Mach numbers. Experience and knowledge gained over the years since 1950 leads to the general belief that further development of air-breathing (ramjet) engines for operation up to $M \sim 10$ or so can be carried out without essaying new fields of knowledge and will be a relatively straightforward extension of existing technology.

This feeling coupled with the observation that the on-board fuel specific impulse of air-breathing engines in their useful speed range inherently is larger than that of chemical rockets (which must carry their own oxidizer) prompted a revival of interest in manned ground/orbit transport vehicles, based upon use of vehicles which attain $1/3$ to $1/2$ of orbital speed by use of air-breathing propulsion systems. If these could be made to operate with very high fuel specific impulse, it was hoped that the payload-to-orbit might be made large enough to be interesting, even with vehicles of reasonable structural factors.

Results of studies toward this goal and proposals for further work were presented to the US Air Force on 1 December 1960 (ASP, 1960), by six companies who have been active in the field.* Since much of this

* A seventh company, McDonnell Aircraft, reported on 1 March 1961 on its work.



[REDACTED]

work is proprietary in nature, none of it is considered here in detail nor are the individual companies identified with specific items, concepts, or proposals.* However, in order to provide a basis for comparison with the present work we do review briefly, in the following, the vehicle and propulsion system types studied, general performance estimates for use of these, and some of the features of each basic system.

The principal types of vehicles, propulsion systems, and flight modes studied are summarized in Table 1, together with some comments on each system concept. The table is arranged in order of system complexity, the simplest system being considered first. We see from the table that the first system does not require the development of any truly new equipment or the solution of problems in major new areas of technology. Rather system (1) requires only the application of present-day technology and knowledge to yield hydrogen-fueled counterparts of existing (in component form and test laboratory stage) hydrocarbon-fueled air-breathing engines. System (2) requires an extension of existing knowledge into the area of supersonic combustion engine design, and involves a whole new area of materials research and vehicle structural design as well. This is because the vehicle of system (2) must attain orbital speed while still within sufficient atmosphere to provide oxidizer for the air-breathing engines, thus must live with the very high stagnation temperatures concordant with such speeds. System (3) is the most complex of all, not

* Those properly concerned with this work can make the identification from the referenced documents.

TABLE 1
CHARACTERISTICS OF CHEMICAL AEROSPACE PLANE SYSTEMS

System Concept	Propulsion Systems at Various Flight Speeds			Remarks
	M 0 to 2/4	M 2/4 to 7/11	M 7/11 to 26 (orbit)	
1	Hydrogen-fueled turbojet	Subsonic-burning hydrogen fueled ramjet	Liquid oxygen hydrogen rocket	Presumes substitution of hydrogen as fuel in existing engine technology.
2	Hydrogen-fueled turbojet	Subsonic-burning hydrogen fueled ramjet	Supersonic-burning hydrogen fueled ramjet (no rocket)	Presumes technology of system (1) and the development of technology of supersonic combustion above M ~ 7 or so. Vehicle must reach orbital speed while within the sensible atmosphere, thus new technology of ultra-high-speed, hot flight structures also required.
3	Air liquefaction and liquid air/hydrogen rocket (LACE)	Air liquefaction, storage and enrichment, and subsonic-burning hydrogen fueled ramjet (ACES)	Enriched liquid air/hydrogen rocket	Presumes technology of system (1), of H ₂ /air heat exchangers and of liquid air separators (O ₂ /N ₂) of very high performance and low mass.

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in terms of new areas of knowledge which must be explored and exploited (as for (2)) but in terms of mechanical gadgetry and power equipment. To operate at all (3) must employ heat exchangers to condense the atmosphere while in flight (must cool from air ram to condensation temperatures), by use of the on-board liquid hydrogen as a heat sink, must utilize ramjet engines fed with liquefied air as well as atmospheric ram air, and must use high efficiency separators to segregate liquefied O_2 from N_2 for later use (of the O_2 -enriched air) in the rocket phase of flight. None of this equipment exists at all, at present, nor does the technology for development of it to the degree of refinement (very light weight and very high efficiency) required for successful use in flight to orbit. In passing let us note one other feature inherent in system (3). This is that to be effective the air liquefaction and collection system must be used in both the initial boost phase of flight and also in the acceleration phase from boost to rocket takeover (or to as high speed as possible). Thus LACE and ACES must be used together with enriched-air-hydrogen rockets to achieve an efficient system. Use of either one alone is an inefficient use of mass carried by the vehicle. Performance estimates (see Table 2) show that the vehicle system (3) can not carry a payload for the cited mission unless the enrichment system (part of ACES) yields better than 70% O_2 from liquefied air.

Let us now consider the performance of vehicles based upon the three principal concepts listed in Table 1. Results of calculations presented in the reference studies (ASP, 1960) are shown in Table 2,

TABLE 2

PERFORMANCE OF CHEMICAL AEROSPACE PLANE SYSTEMS

<u>System Concept</u>	<u>Payload-In-Orbit Fraction of Gross Mass</u>	<u>System Mass Fractions</u>	<u>Remarks</u>
1	0/0.01	$f_{\text{struct}} = 0.20$ $f_{\text{propul}} = 0.06$	System performance marginal; virtually no payload delivered to orbit.
2	0.03/0.075	$f_{\text{struct}} = 0.25$ $f_{\text{propul}} = 0.13$	Structure factor is good, but performance is sensitive function of assumed propellant performance in supersonic burning flight.
3	0.01/0.06	$f_{\text{struct}} = 0.19/0.20$ $f_{\text{propul}} = 0.08/0.12$	Requires air enriched to 75% O ₂ /25% N ₂ . Higher payloads are only for lower propulsion system mass fractions.

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which lists the estimated payload delivered to a 300 n.mi. orbit by a single stage, manned vehicle weighing 500,000 lb at ground takeoff, together with the range of mass fractions used in the studies and some comments on the estimated system performance.

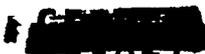
From the table we see that the simplest, hence probably most easily built and most reliable vehicle, system (1), is incapable of putting a useful payload into orbit, even with a structure factor considerably less than that attainable today (the DC-8 is about 0.33; B-70 would be about 0.25; both use high density fuel). In order to achieve payload-in-orbit, system (2) was forced to resort to exploitation of an as-yet-unknown body of knowledge, while system (3) assumed use of a rather large amount of complex equipment in order to exploit the idea of air condensation. In spite of the larger equipment load, structure factors assumed for system (3) were the same or less than that for system (1). Even then, payload capability appears marginal unless the lower values cited for propulsion system fraction can be attained, and if so a payload mass of about 0.06 of gross mass could be delivered to orbit. System (2) utilized structure factors closer to the present state-of-the-art, and presumed an equally sturdy propulsion system mass fraction, yet was still capable of placing up to 0.075 of gross mass into orbit as payload. Of course, this result is totally dependent upon being able to develop a new technological field (hypersonic combustion ramjets) and on being able to solve the myriad materials problems of a vehicle flying in air with maximum radiation equilibrium skin temperatures



the order of 5000°R. In general, none of these systems seems to offer a direct or straightforward development path to the desired goal of reliable, high performance, economic, ground/orbit transport.

(D) Recoverable Rocket Vehicles

Another approach which has received some attention is that of the use of pure rocket vehicles to perform the desired mission. As discussed earlier, it appears that overall payload fractions of 0.03 to 0.06 may be capable of transport to orbit by use of conventional ground launched rocket vehicles of about two stages (e.g., Saturn vehicles). The structure plus propulsion factor per stage runs around 0.06 for such vehicles. In order to effect economical payload delivery it is essential that the rocket vehicles used for boost as well as for upper stages be recovered cheaply and recovered in such a fashion that economical reuse is possible. Recovery in condition which requires costly refurbishment before reuse defeats the prime economic motivation for recovery itself. With the low structure plus propulsion factor of these vehicles it is not possible to carry a manned crew on each stage and thus allow them to be "flown" back to base. In order to do so additional structure and propellant would be required, thus reducing the payload capability. Even so, it seems doubtful that recovery of such tenuous vehicles would ever be as easy, cheap, or reliable as that of a manned, winged system with structure plus propulsion factor 5 to 7 times greater than that of the rocket.

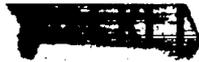


[REDACTED]

Other studies have been made for oxygen/hydrogen super-chemical rocket vehicles. These indicate that payload fractions of zero to 0.04 might be placed in orbit with single stage vehicles of, respectively, 0.12 to 0.08 structure plus propulsion factor. Again "beefing-up" to provide sturdier vehicles eliminates any hope of payload and again such results seem unencouraging from the standpoint of economic reuse.

Still better ground/orbit transport via rocket vehicle can be prognosticated with the use of nuclear rockets. In a recent study, Douglas Aircraft Company (RITA, 1961) analyzed the performance of a manned single stage, nuclear rocket vehicle with specific orientation toward the problem of recovery and reuse, and to crew survival in abort situations. Performance-wise such a vehicle would be capable of placing a payload fraction the order of 0.08 to 0.10 into a low Earth orbit, with a structure plus propulsion factor of about 0.12 to 0.10. In order to achieve such a low factor, very little shielding was allowed for the nuclear reactor powerplant, with the result that severe operational restrictions are required for use with manned crews. Post shutdown inspection and maintenance of the vehicle and its subsystems is completely impossible without use of extensive and massive shielded equipment and remote handling procedures. In the event of main propulsion system malfunction (one possible abort situation) the primary vehicle could not be recovered practically, although provision is made for the safe escape and recovery of the crew (by use of a chemical rocket propelled

[REDACTED]



escape capsule). It seems possible to devise ways to save the crew from nearly all abort situations, but not possible to recover the vehicle. This vehicle concept then, offers hope for recovery and reuse from successful flight operations, recovery of crew from all flights, but loss of vehicles from unsuccessful flights. Because of the very high radiation dose rates given the launch area at takeoff and at powered return to base, such a vehicle could not be operated from existing bases, but would require special bases and base facilities (e.g., the shielded maintenance equipment referred to above) for its employment.

Shape and aerodynamic characteristics of the RITA vehicle were chosen to ensure a high probability of successful recovery. This study is thus the first serious attempt toward the achievement of this goal by use of special features of the rocket vehicle configuration layout. Estimated potential performance is considerably better than that of any other rocket system for application to ground/orbit transport. Still, in spite of its relative performance advantages, the inability of the vehicle to recover from abort situations and the operational problems foreseen for use of nuclear propulsion as proposed reduce interest in this concept as a candidate for the versatile, reliable, low-cost, practical, manned, ground/orbit ferry we are seeking.

(E) Rover, Ducted Rocket Boost, and the Nuclear Rocket Aircraft

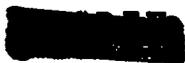
In this brief review of past studies we have seen that no simple system studied seems capable of fulfilling the desired mission goals, and that the complex systems evolved are only marginally capable





of doing so and presume the successful completion of extensive research and development programs in rather new areas of work. Little attention has been paid to the possible use of nuclear propulsion systems, except for the last-mentioned study in which nuclear energy is used in a "brute-force" manner via rocket application.

It is somewhat surprising that in all this work no consideration has been given to one of the simplest and yet oldest concepts applicable to the use of nuclear propulsion for the aero space plane mission. This is the idea of boosting a nuclear rocket to high speed by ducting air to the hydrogen exhaust from a nuclear rocket, burning it to achieve ramjet propulsion while in the atmosphere, and flying to completion of the mission on nuclear rocket power alone after ramjet shutdown. First proposed by H. T. Gittings in January, 1955, and studied by Gittings and R. W. Bussard (Gittings, 1955), this concept was employed in initial studies of nuclear rockets for ICBM missions (Aamodt, et al., 1955) and was instrumental in development of the vehicle concepts upon which the Rover program was begun. Use of such a ducted rocket for boost allowed the performance of the 1955 Atlas ICBM mission with a nuclear rocket weighing only 24,000 lb at time of duct drop-away (i.e., at $M \sim 3.5$ and 65,000 ft altitude). Shortly thereafter in the Fall of 1955, a brief investigation was carried out by R. W. Bussard of the use of ducted nuclear rockets for the propulsion of high speed aircraft for continuous flight in the atmosphere. This work was never published; partly because of the relatively uninteresting performance found for

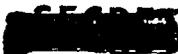




such a vehicle when used in competition with nuclear powered ICBM's, and partly because no thought was given to the aerospace plane mission of current interest.* Some results of this study are summarized in Appendix A, from which we see that vehicle performance (for the fuel/propellant assumed) drops markedly between $M \sim 4$ and $M \sim 5$. This drop is a result of the decrease in lift/drag ratio assumed in the study and the effect of increase in ram air temperature with increasing flight speeds. With hydrogen fuel this performance decrease would take place at about $M \sim 6$ to $M \sim 7$, other assumptions remaining the same.

The essential feature of both these concepts is that of making use of the "free" oxidizer supply from the atmosphere to burn with the on-board combustible fuel heated in a (monopropellant) nuclear rocket reactor. The aircraft concept in addition attempted to utilize the atmosphere for lift, thus eliminating the need for larger reactor power as required by thrust-lifting flight. The present proposal rests upon these two basic features: Use of (1) atmospheric air for lift, and (2) atmospheric oxygen for combustion.

* At this time Sputnik I was still two years in the future.



III. The ASPEN Concept: An Aerospace Plane with Nuclear Engines

(A) Basic Characteristics of the ASPEN System

The vehicle of interest to us here is an air-breathing-engine/boosted nuclear rocket powered aircraft. Three different stages of propulsion must be employed to carry payload from ground to orbit. These are: (1) An initial boost phase from horizontal ground takeoff to a speed sufficiently high that the air-breathing engines of the next phase can operate efficiently (typically $M \sim 2.5$); (2) an atmospheric acceleration phase where the vehicle is driven by air-breathing engines to as great a speed as is practical within the atmosphere, or until engine efficiency becomes low (typically $M \sim 8$ to 11); and (3) a final boost phase by use of pure rocket power to achieve orbital speed. To be specific we shall study flight to a circular orbit at 300 n.mi. altitude (1.824×10^6 ft altitude, orbital speed $\approx 24,800$ ft/sec) by means of powered propulsion to some 80 n.mi., coast to apogee and injection into orbit at 300 n.mi. Following completion of the orbital mission the vehicle is to retrorocket, re-enter, drag-glide down to $M \sim 2$ and be capable of flying home to horizontal landing at its base.

In the initial boost phase we consider two possibilities for propulsion: (1) Conventional turbojet engines fueled with standard (hydrocarbon based) JP-4 jet fuel, and (2) turbojet engines fueled with liquid hydrogen. For the atmospheric acceleration phase we limit interest to the use of hydrogen-fueled air-breathing ramjet engines, but with three variations: (1) Pure chemical with subsonic burning, to flight

speed of $M \sim 11$; (2) pure chemical with subsonic burning to $M \sim 8$ and supersonic burning to $M \sim 11$; and (3) mode (2) with nuclear reactor preheating of hydrogen fuel. For the final boost phase we consider only nuclear rockets for propulsion.

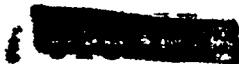
In all systems the nuclear reactor is to be shielded in order to reduce radiation dose rates a few days after shutdown sufficiently to allow ground maintenance of the vehicle including removal and replacement of the nuclear reactor system, without undue radiation hazard.

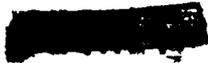
In carrying out the study we first choose a "nominal" vehicle, analyze its performance and then study the effects of alternate choices of powerplant and assumptions about performance as variations from the payload in orbit performance of the nominal vehicle. We choose the vehicle of prime interest as the simplest one; i.e., that vehicle which uses the simplest and most reliable set of propulsion systems from those outlined above. The nominal vehicle thus is powered in initial boost by conventional jet-fueled turbojets, in atmospheric acceleration by pure chemical, hydrogen-fueled, subsonic-burning ramjets, and in final boost by unit-shielded nuclear rocket engines. This system, as well as any of its more sophisticated alternates, is at least as simple as system (1) (Table 1) of the all-chemical aerospace plane studies, and is very much simpler than either system (2) or system (3), yet it retains all of the operational advantages of these.

At takeoff and landing it is effectively an all-chemical turbojet-powered airplane, which can be used from any conventional



airfield with runway length of about 6000 ft (takeoff run). Shorter runways can be used if JATO boosting is employed at takeoff. It can fly at subsonic or low supersonic speeds at modest altitudes on turbojet power, thus can be ferry-transported from base to base. In the event of abort due to failure of the ramjet or rocket propulsion systems, the vehicle can dead-stick glide down to turbojet operating speed and be flown home or to an alternate landing site; the vehicle is not lost in event of such a major propulsion system failure. Great flexibility results from the fact that it is a manned vehicle. The human pilot can make in-flight judgement decisions to cope with minor emergency situations, and because he is flying a fully controllable vehicle (both aerodynamically and propulsion-wise) can recover from these as readily as can the pilot of a commercial jet transport aircraft. Since the vehicle airframe is relatively sturdy (we will see later the relation between structure factor and payload for the various vehicles of interest), it should be capable of use over a long life, even though changes of turbojet, ramjet, or nuclear engines may be required from time to time, again as for conventional aircraft. With proper planning and operation, maintenance and replacement of the airframe and turbojet and ramjet engines is not essentially different from conventional practice, since the reactor is well shielded from personnel at all times. Reactor replacement may be carried out a few days after shutdown by removal from the aircraft of the complete, packaged, shielded assembly containing the reactor and part of its exhaust ducting, and replacement with a new assembly. Reactor





maintenance is done at a central remote-handling shop (e.g., typical facilities for such purposes are the ANP hot shop at Arco and the Rover MAD building at the Nevada Test Site); thus no radiation handling equipment is required at the vehicle's base. Reactor systems prepared for return to the maintenance center are packaged so as to allow shipment from base to shop by conventional means of transport.

Because of the superior payload capacity of ASPEN, relative to the all-chemical aerospace plane concepts, excess propellant can be carried to orbit, thus allowing a modest maneuvering capability of the complete vehicle in-orbit. Alternatively the larger payloads can be exploited in the form of small, manned, orbital shuttle vehicles of greater maneuvering capability than the parent ASPEN vehicle alone (because of its large inert mass). There are so many potential uses of such a vehicle that it is difficult to enumerate them. Some of the more obvious include: (1) Satellite intercept, inspection, and maintenance; (2) orbital altitude reconnaissance; (3) meteorological survey; (4) payload transport to and from orbit; (5) orbital bomb delivery; (6) missile detection and early warning; (7) real-time and delay-time communications relay, and many others. The presence of a manned crew allows full exploitation of the orbital payload capacity of ASPEN.

(B) ASPEN and ANP

We wish to emphasize at the outset that the nuclear aerospace plane and the nuclear airplane are totally unlike one another. The only similarity lies in the words "nuclear" and "plane" which appear



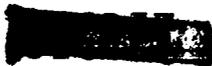


in the description of both concepts. It is perhaps unfortunate that they do, for ASPEN is in reality better described as an air-breathing-engine/boosted nuclear rocket, and the choice of names serves only to lead the uninitiated into imagining a (spurious) connection with ANP.

The rocket reactors required for ASPEN (and presently under successful development in the Rover program) operate at power densities from 10 to 30 times greater than those which were of interest for ANP; thus the reactor proper is much more readily shielded for use in manned vehicles than was the case for ANP reactors. This difference is a result of the fact that (direct-cycle) ANP reactors required cooling with a relatively large flow of air at rather low pressures, while the rocket reactor is cooled by a flow of hydrogen at very high pressure.* Another consequence of this is that coolant ducts will be much smaller, hence much more easily folded and shielded in rocket reactor systems than in nuclear air-breathing systems.

Still another difference arises in the mode of operation. In ASPEN the reactor is not brought to full power until the rocket phase of flight is reached; this will occur at altitudes greater than 100,000 ft , where air density is very low. Thus all of the problems are avoided of shielding a low-density "leaky" ANP reactor at full power operation at relatively low altitudes where air density is high and air-scattering of

* Problems of shielding against radiations from activated coolants arise in the indirect-cycle propulsion systems.



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radiation is severe. Furthermore, because ASPEN is a rocket it must carry a considerable quantity of hydrogen propellant, some of which may be exploited to do double-duty as partial shielding for a portion of the rocket-boosted flight, while ANP systems do not have this feature. In either case, any conventional jet fuel carried for chemical turbojet propulsion also can be used without difficulty for shielding mass, so long as adequate cooling is provided. And last, but not least, direct air-breathing reactors must operate fissioning fuel elements in the extremely reactive oxidizing environment of very hot air, while rocket reactor fuel elements run in the reducing atmosphere of hot hydrogen. Major reactor design and development problems, and the mode of application proposed herein to manned aero vehicles, are both quite different; on the net it appears that ANP experience is relatively inapplicable to the ASPEN system concept.

(C) ASPEN Performance

In this section we analyze the flight and estimate the payload-in-orbit performance of various ASPEN vehicles. The starting point for our studies is the nominal simplest ASPEN system described in an earlier section of this report. Such a vehicle might appear as shown in Figures 2 and 3 which portray a configuration which is a rough extrapolation from, and combination of features of the Sanger-Bredt bomber, Dyna-Soar, X-15, and the B-70. The figures are scaled to a vehicle of about 500,000 lb gross weight.

The outboard profile (Fig. 2) shows a flat-bottomed, lifting-body, with fineness ratio about 10:1, supported on swept stub wings

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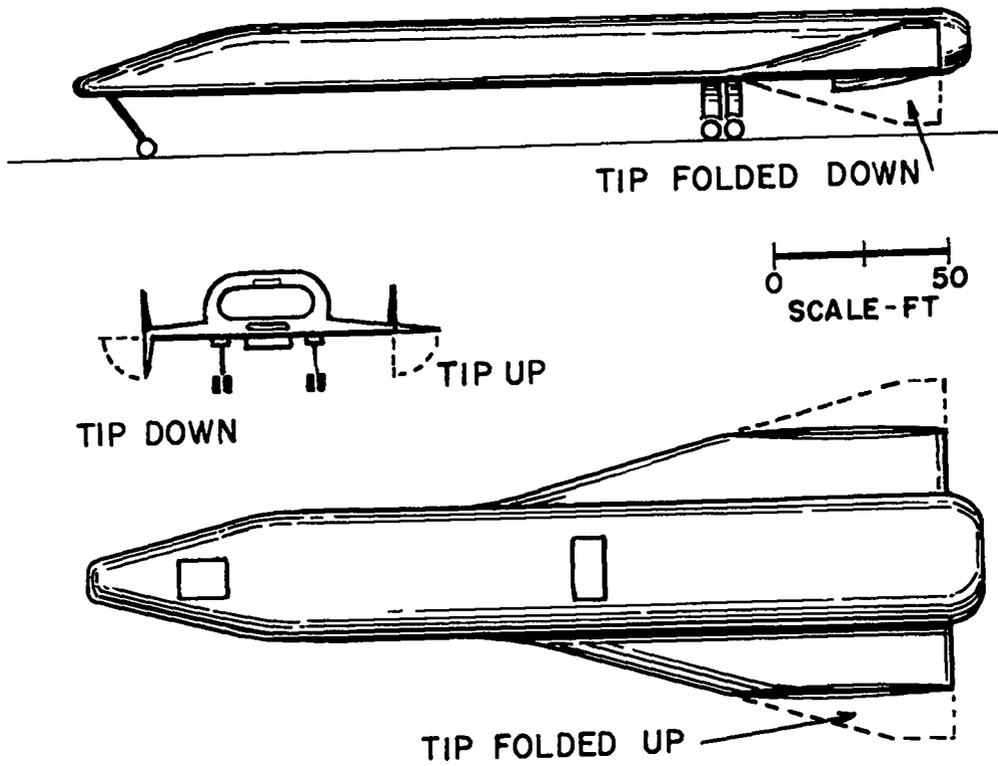


Fig. 2 Typical ASPEN outboard profile.

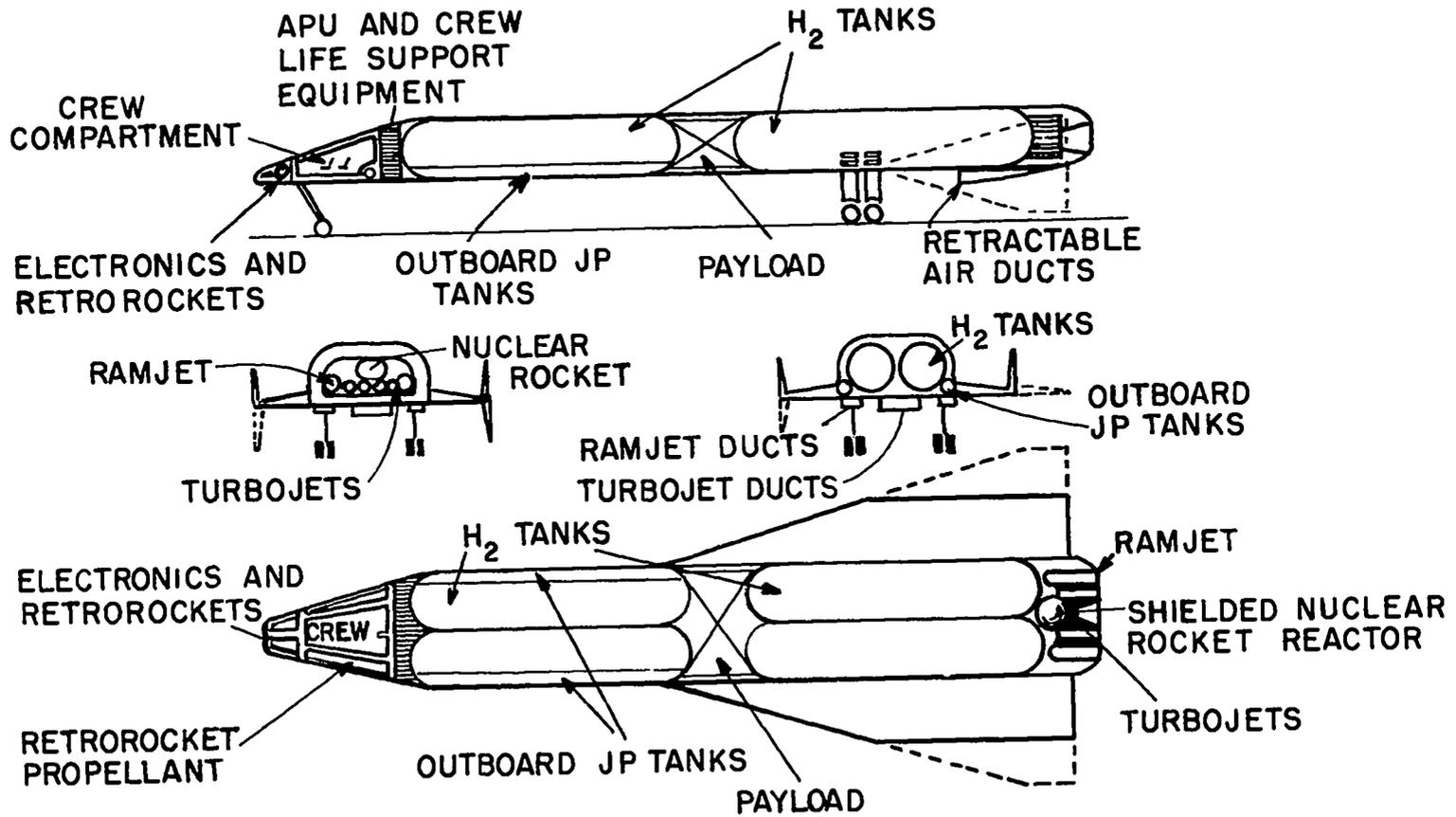
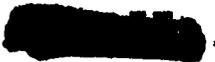


Fig. 3 Typical ASPEN inboard profile.



terminated by vertical tip plates whose lower half may be rotated between vertical and horizontal positions. The vertical position is used in lifting flight at hypersonic speeds in order to provide increased lateral aerodynamic control. For low speed flight adequate control surface is available from the upper half plate, and the horizontal position of the lower plate is used to provide increased lifting area. In spite of the increased drag at high speed with rounded surfaces, there are no extensive sharp-edged intersections (air inlets excepted). This is dictated by the desirability of minimizing local aerodynamic heating problems. Radii at intersections must be optimized between the requirements of good exit flight performance and surface structural integrity.

The inboard profile (Fig. 3) shows the air inlets for turbojet (initial boost phase propulsion) and ramjet (atmospheric acceleration phase) engines located beneath the wing-body lifting surface in order to exploit the under-body pressure field for higher efficiency inlet recovery in high speed flight. These are separate,* fully retractable, and variable in area. All engines are mounted internally at the rear of the vehicle. Most of the internal volume is devoted to liquid hydrogen storage; thus the fuselage section is lightly loaded and of low bulk density. Four main hydrogen tanks are shown, running longitudinally in pairs, with one pair forward and one pair aft of the payload compartment. A central cavity is left around the center of gravity of the vehicle to accommodate a payload volume $1/8$ that of the hydrogen tankage. Jet fuel required for initial

* One set of inlets could be used if both propulsion phases were to be combined by use of a "turboramjet" engine.



boost flight is stored in two longitudinal tanks located between the hydrogen tankage and the upper body external skin. An alternate arrangement could employ wing internal storage tanks, if desired. Additional jet fuel (available for use in landing) may be stored in a tank around the primary reactor shielding, to provide additional shielding while in orbit after reactor shutdown, or on the ground after landing if unused in the landing maneuver. A tricycle landing gear arrangement is shown, retracting into body space around the hydrogen tankage. Space is provided for a central crawlway in the forward half of the vehicle and two outboard crawlways in the aft half for access to the landing gear, payload compartment, and air-breathing engine auxiliary equipment in the vehicle rear. The crew compartment and all life support and communications equipment are in the vehicle nose.

(1) Vehicle trajectory

Let us consider an ASPEN vehicle flying in the atmosphere under the action of forces as indicated schematically in Figure 4. Here we have resolved the net aerodynamic forces into lift (L) and drag (D) acting at right angles to each other, through the center of gravity (c.g.) of the vehicle, and with thrust and drag axes aligned. In doing so we arbitrarily assume that any turning moment (counteracted by trim or tail surfaces) which will result from non-coincidence of lift center and c.g. has only a negligibly small effect on vehicle flight performance. In order that this assumption be satisfied, vehicle flight must be such that the c.g. does not shift significantly during operation above $M \sim 3$ or so. To ensure that this will be the case, propellant consumption must be programmed symmetrically about the initial c.g. of the vehicle at start



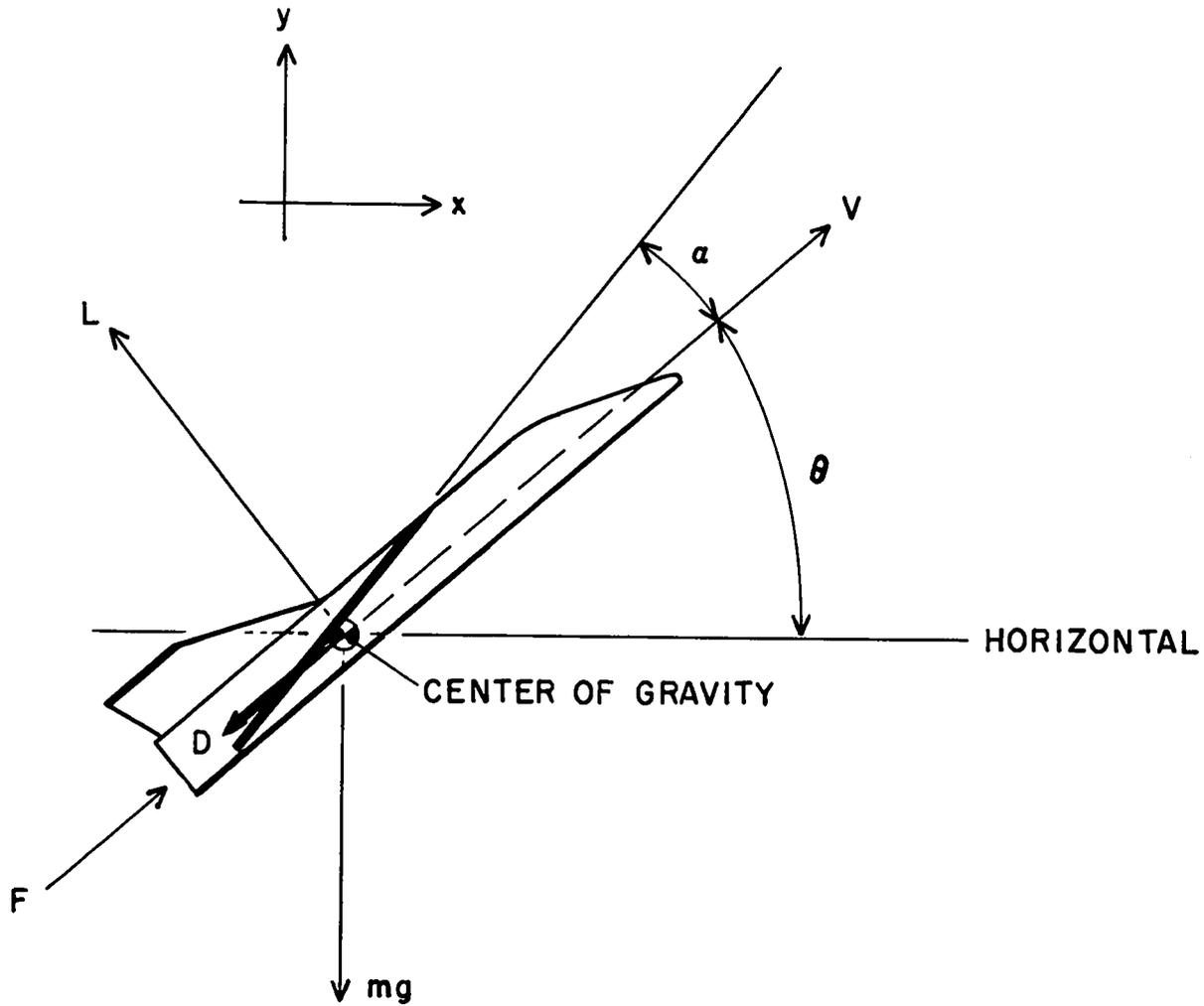


Fig. 4 Schematic force balance on ASPEN vehicle.

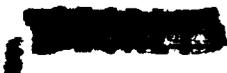


of the ramjet phase of supersonic flight, and the vehicle payload should be distributed symmetrically about this position, as indicated in Figure 3.

With the above restrictions we can write the equations of motion of the vehicle in two dimensions (a vertical x,y plane) as

$$\begin{aligned}
 m(t) \frac{d(v \cos \theta)}{dt} &= (F-D) \cos \theta - L \sin \theta = \\
 &= \left[\dot{m}_p(t) I_{sp}(v) - \frac{1}{2} \rho_a(y) v^2(t) A_b c_D(v, \alpha) \right] \cos \theta - \frac{1}{2} \rho_a(y) v^2(t) c_L(v, \alpha) \sin \theta \\
 (1) \\
 m(t) \frac{d(v \sin \theta)}{dt} &= (F-D) \sin \theta + L \cos \theta - mg = -m(t) g(t) + \\
 &+ \frac{1}{2} \rho_a(y) v^2(t) A_b c_L(v, \alpha) \cos \theta + \left[\dot{m}_p(t) I_{sp}(v) - \frac{1}{2} \rho_a(y) v^2(t) A_b c_D(v, \alpha) \right] \sin \theta
 \end{aligned}$$

Here the functional dependence of each variable has been indicated explicitly. Symbols are as follows: I_{sp} is the specific impulse based on on-board propellant consumption alone, ρ_a is air density, c_L and c_D are lift and drag coefficients, v is the instantaneous magnitude of vehicle velocity, θ is the angle between vector \bar{v} and the local horizontal plane, g is the local gravitational acceleration, m is instantaneous vehicle mass, \dot{m}_p is the rate of consumption of vehicle propellant, and α is the angle of attack of the prime lifting surface relative to the velocity axis. A_b is the effective area in the usual lift and drag formulae, $L = 1/2 \rho_a v^2 A_b c_L$; $D = 1/2 \rho_a v^2 A_b c_D$. In order to solve for the motion of ASPEN from eq. (1) we must know the



functional relations between interconnected parameters above; it is immediately clear that the problem of flight analysis and optimization is quite complex by this method. We do not propose to solve for vehicle motion from these formulae, since this is more readily done later by treatment of ASPEN as a type of rocket vehicle, but use them as a starting point for discussion of elements pertinent to the problem.

Let us consider the coefficients c_L and c_D of the lift and drag forces which appear in eq. (1). Accurate estimation of these for complete configurations over a wide Mach number range is difficult and tedious (e.g., Truitt, 1959, Chapter 8, or Nielsen, 1960, Chapters 1-3, 5, 7, 9) and will not be attempted here. It is sufficient for our purposes here to note only the simple relations for flat plate configurations in supersonic flow: $c_L = 4\alpha (M^2 - 1)^{-1/2}$ and $c_D = c_L \alpha$. These are valid for moderately supersonic flows, but become progressively less applicable as high hypersonic flight speeds are attained, becoming equal at $\alpha \sim 2/11$ at $M \sim 11$ to the values for Newtonian flow theory. For flight in the subsonic and low supersonic range it is reasonably satisfactory to assume* that the coefficients of α and α^2 in c_L and c_D above remain constant at roughly the values found for $M \sim 2.5$. Best choice of an actual shape (other than flat plate geometry) is complicated by restrictions imposed by materials temperature limitations and aerodynamic heating both in exit and re-entry flights. For re-entry with

* As was done in one of the reference studies (ASP, 1960).

least materials heating problems a high drag shape with low frontal area loading density is desirable (e.g., RITA, 1961), while for least propellant consumption during exit and highest payload delivered to orbit a low drag, high loading density configuration is best. Choice of an optimum shape for an arbitrarily fixed exit trajectory requires a detailed knowledge of the aerodynamic characteristics over the complete air-borne flight range. This information is attainable only by experimentation in high speed wind tunnels.

To minimize structural problems it is preferable that the net lift and drag forces not peak sharply anywhere in flight, but remain relatively constant over the air-breathing portion of flight. To achieve this state it is desirable that $\rho_a v^2$, c_L , and c_D vary only slowly or remain constant in this regime. From our previous discussion we see that both c_L and c_D decrease with increasing M , for fixed angle of attack. If we hold α constant we see that constant L and D can then be attained only by choosing a trajectory for which $\rho_a v^2$ varies as $\sqrt{M^2-1}$; i.e., increases almost linearly with M at high Mach number. This is unsatisfactory since the dynamic pressure or kinetic energy density seen by stagnation regions of the vehicle will also increase roughly linearly with flight speed. A better choice is that adopted in the majority of the reference studies (ASP, 1960) in which $\rho_a v^2$ is allowed to vary only slightly, decreasing slowly with Mach number above about $M = 3.5$. Now holding c_D constant by varying α as $(M^2-1)^{1/4}$, we find that such a trajectory has the property that c_L varies as $(M^2-1)^{-1/4}$; thus lift decreases slightly faster than

does drag. This results in a decreasing rate of climb of the vehicle as it accelerates, which is desirable in the first half of flight in order to keep it within sufficient air to allow operation of the air-breathing engines.

We note that in this portion of flight the vehicle gains altitude by virtue of the excess of upward force components (due principally to lift, but also including $(F-D) \sin \theta$, as in eq. 1) over downward forces for level flight. Thus altitude is gained through use of propulsive system thrust transformed (magnified) by the (L/D) ratio of the vehicle. In this way the propellant consumption required as "penalty" for gravitational losses is reduced from that which would obtain in pure thrust-lifting flight of a rocket, for example. In addition significant further reduction in relative propellant cost for gravity losses, as compared to that for flight of rocket-powered thrust-lifting vehicles, arises because of the use of atmospheric air as (the major) part of the total propulsive fluid flow stream, thus yielding considerably larger net on-board propellant specific impulse in the ASPEN case than for a simple rocket. The overall factor by which such losses can be reduced relative to rocket usage is the order of $(L/D)(H_c/\Delta H_r)$, where H_c is the combustion energy per unit propellant mass, and ΔH_r is the energy per unit mass supplied by the reactor in a pure rocket system. Typically, this factor may be the order of 15 to 40, thus allowing use of longer powered flight time for ASPEN than for a pure rocket vehicle, with correspondingly lower acceleration and hence smaller reactor power, at

the same time still reducing gravitational losses to the order of 1/5 to 1/10 those of a pure rocket (e.g., typically these would be about 4000 to 6000 ft/sec for nuclear rocket flight to $M \sim 11$, at the end of ramjet operation).

The band of trajectories employed in the reference studies (ASP, 1960) is shown in Figure 5, together with the mean path followed in this present study. The lower branch above $M \sim 12$ is essential to the supersonic-combustion ramjet concept (system (2) of Tables 1 and 2). The low altitudes needed for accelerating flight of such a system at large hypersonic Mach numbers force the undesirable corollary of very high vehicle external structure temperatures; in a sense such a scheme suffers from re-entry heating problems on the exit path as well. For the ASPEN vehicle it is no longer necessary for reasons of propulsion system thrust to remain within the sensible atmosphere following ramjet shutdown and switch over to nuclear rocket propulsion. In the first part of the rocket boost phase there is an optimum rate of climb which can be found by balancing the effect of more rapidly decreasing drag* with a steep trajectory against the advantage of retaining wing-lifting flight longer (to reduce the propellant cost of gravity losses) with a flatter trajectory. In the end it is aerodynamic heating and the ability of the ASPEN external structure to operate at high temperature which

* Note that closure of the ramjet air inlets at rocket takeover will yield a considerable reduction of total drag and corresponding increase in vehicle (L/D) with fixed α , thus allowing an increase in angle of attack for the same (L/D), hence increased aerodynamic lift and rate of climb, as is desirable for overall performance optimization.

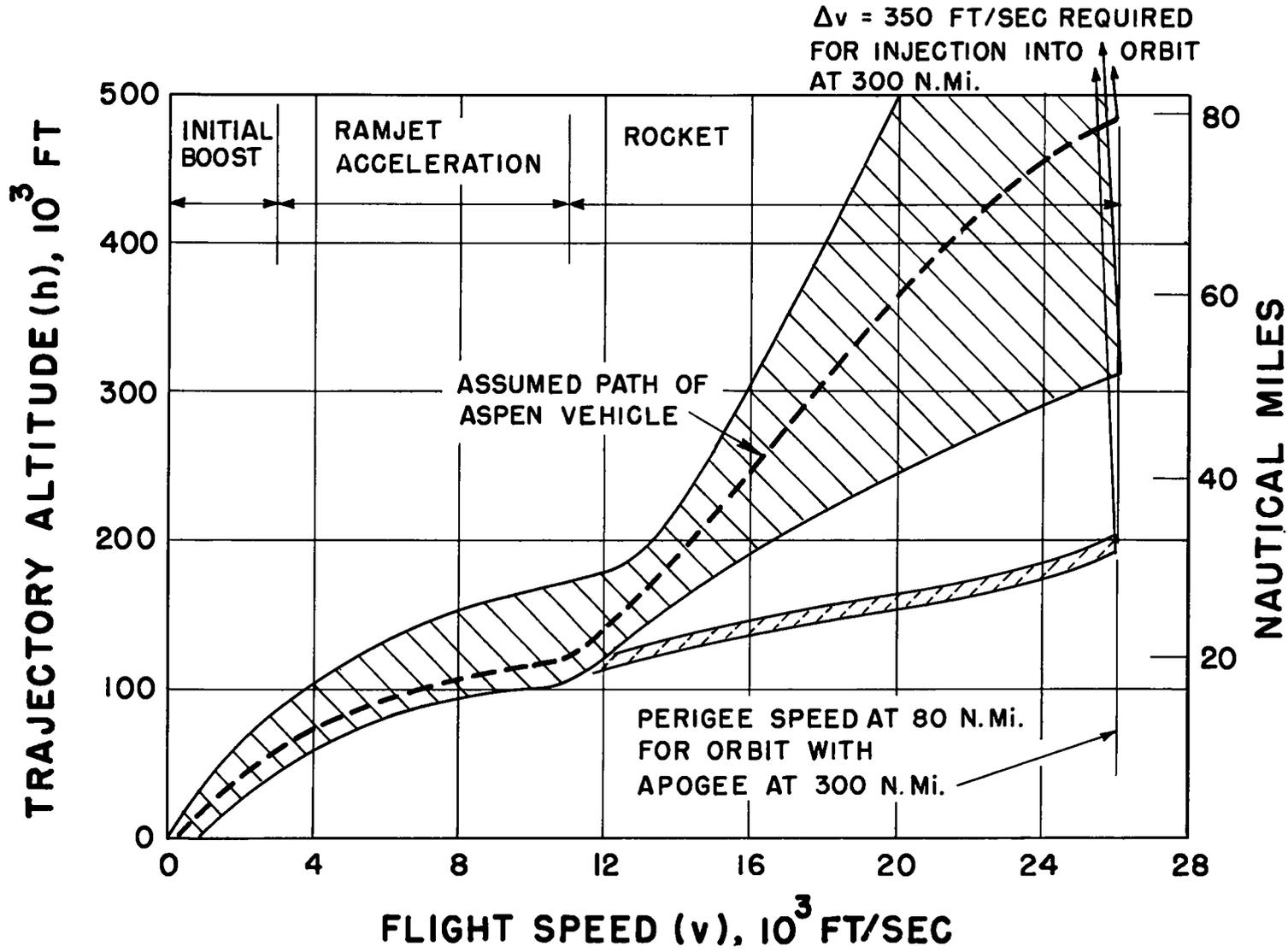
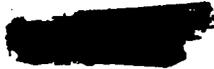


Fig. 5 Aerospace plane and ASPEN flight trajectories.



determines the limiting flight-speed/altitude envelope. Optimization of this portion of flight is very complex, involves a number of non-analytic variables, can be done only by recourse to machine numerical calculations, making use of considerable experimental aerodynamic and materials strength data, and is far beyond the scope of this report. Rather, we have chosen arbitrarily a mean flight path from those of the reference studies (see Fig. 5). Here the vehicle flies to $M \sim 2.5$ at 60,000 ft on turbojet power, from this point to $M \sim 11$ at 120,000 ft on ramjet power, and thence on rocket power to a velocity at rocket shut-down of 26,015 ft/sec at 485,000 ft (80 n.mi.) altitude. The vehicle then coasts upward in an elliptic orbit to a velocity of 24,750 ft/sec at the apogee at 300 n.mi. At this point rocket thrust is used to add an additional 350 ft/sec, required for conversion to a stable circular orbit at this altitude. The total velocity change required is 26,365 ft/sec, plus that for gravity losses in the rocket phase of flight. The rapid climb above $M \sim 11$ helps to minimize aerodynamic heating problems and thus lends confidence to the assumed successful use of modern conventional structural materials (e.g., Rene 41, Ti 150-A) for vehicle application. Another benefit obtained by exit along a relatively steep trajectory is from the increase in exhaust specific impulse with rapidly decreasing nozzle back pressure, allowing effective use of larger nozzle expansion ratios.



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(2) Vehicle and engine performance

Analysis of vehicle flight performance by use of the force balance eq. (1), written earlier, requires step-by-step computation of all lift, drag, and thrust forces both in magnitude and direction at each point along the flight path. Rather than attempt such a detailed calculation we can carry out performance analysis much more simply by use of momentum balance relations of the sort which lead to the well-known mass-ratio equation for rocket vehicle flight. Analysis of ASPEN flight along these lines is consistent with its nature as a continuously accelerating vehicle driven by propellant of varying specific impulse. For our purposes the momentum balance takes the form

$$(2) \quad -mg \, dv = g_0 I_s \, d(mg) + mg \sin \theta \, dt$$

where I_s is the net effective specific impulse based on on-board fuel consumption during powered flight and is related to the net engine thrust per unit fuel flow rate ($F/\dot{m}_p = I_{sp}$) by

$$(3) \quad I_s(v) = I_{sp}(v) \left[1 - \frac{D(v)}{F(v)} \right]$$

We have included the term $[mg \sin \theta \, dt]$ in eq. (2) in order to account explicitly for gravitational losses. As discussed previously, these are taken care of in the first phases of flight by aerodynamic lift, but some penalty must be paid for climbing in the final (rocket) boost phase of flight, when air density is so low that aerodynamic lift and drag are negligible. This is the case for ASPEN above about 200,000 ft (e.g., above $M \sim 14$). Thus, the range of integration of the gravitational loss

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term in eq. (2) should be limited to that part of the pure rocket powered flight in which aerodynamic lift is negligible. For simplicity, we extend this to cover the entire final boost phase, extending from time at rocket takeover, t_2 , to time at burnout, t_f . Dividing eq. (2) by mg and integrating we obtain the mass ratio equation as

$$(4) \quad m_f g_f = m_0 g_0 \exp \left[-\frac{1}{g_0} \int_0^{v_f} \frac{dv}{I_s(v)} - \frac{1}{g_0} \int_{t_2}^{t_b} \frac{g(t) \sin \theta(t) dt}{I_s(v)} \right]$$

where $(t_f - t_2) = t_b$ is the rocket engine operating time. Since all of the parameters except t itself vary only slowly in the last integral we can approximate this sufficiently well for our purposes as $-(g_f/g_0)(t_b \overline{\sin \theta / \overline{I_s}})$ where bars denote average values. It is more convenient to deal with vehicle velocity as a variable than with powered flight time, and we can eliminate t_b from the expression above by use of the identity $a(t) = dv/dt$. Integration of this from v_2 to v_f yields $(v_f - v_2) = \Delta v_r = \bar{a} t_b$ where Δv_r is the vehicle velocity change and \bar{a} is the average vehicle acceleration during final boost. This may be found from knowledge of the thrust/weight ratio as a function of vehicle velocity, according to

$$\bar{a} = g_0 \int_{v_2}^{v_f} (F/W) dv .$$

Combining all of these considerations the final

expression for vehicle mass at end of powered flight can be written as

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$$(5) \frac{m_f}{m_0} = \frac{g_0}{g_f} \exp \left[-\frac{1}{g_0} \int_0^{v_2} \frac{dv}{I_s(v)} - \frac{\Delta v_r}{g_0 \bar{I}_s} \left(1 + \frac{\overline{\sin \theta}}{\bar{a}/g_f} \right) \right]$$

where we have separated the exponential into an integral term concerned only with the air-breathing phases of flight, and another term pertaining only to the final boost (rocket) phase.

Estimation of vehicle performance is seen to require knowledge of I_{sp} and (D/F) as functions of v , to allow determination of $I_s(v)$ and \bar{I}_s ; and of (F/W) as a function of v in the final boost phase, to allow determination of \bar{a} . From these and the chosen trajectory an estimate can be made of $\overline{\sin \theta}$.

First we consider the specific impulse question. During the initial boost phase of flight, from ground takeoff to takeover of the ramjet at about $M \sim 2.5$, ASPEN relies on turbojet engines with variable area inlets for propulsion. Performance characteristics of these have been analyzed elsewhere (Bussard and Mills, 1955) and will not be repeated here; however the results of principal interest to us are shown in Figure 6 which plots net engine thrust per unit fuel flow rate (I_{sp}) as a function of flight speed, for hydrogen and JP-4 (hydrocarbon) used as fuels. These must be reduced in accordance with eq. (3) in order to obtain the effective $I_s(v)$ for the complete vehicle for the turbojet portion of flight.

NET THRUST PER UNIT FUEL WEIGHT FLOW RATE,
 PROPELLANT SPECIFIC IMPULSE (I_{sp}), 10^3 SEC

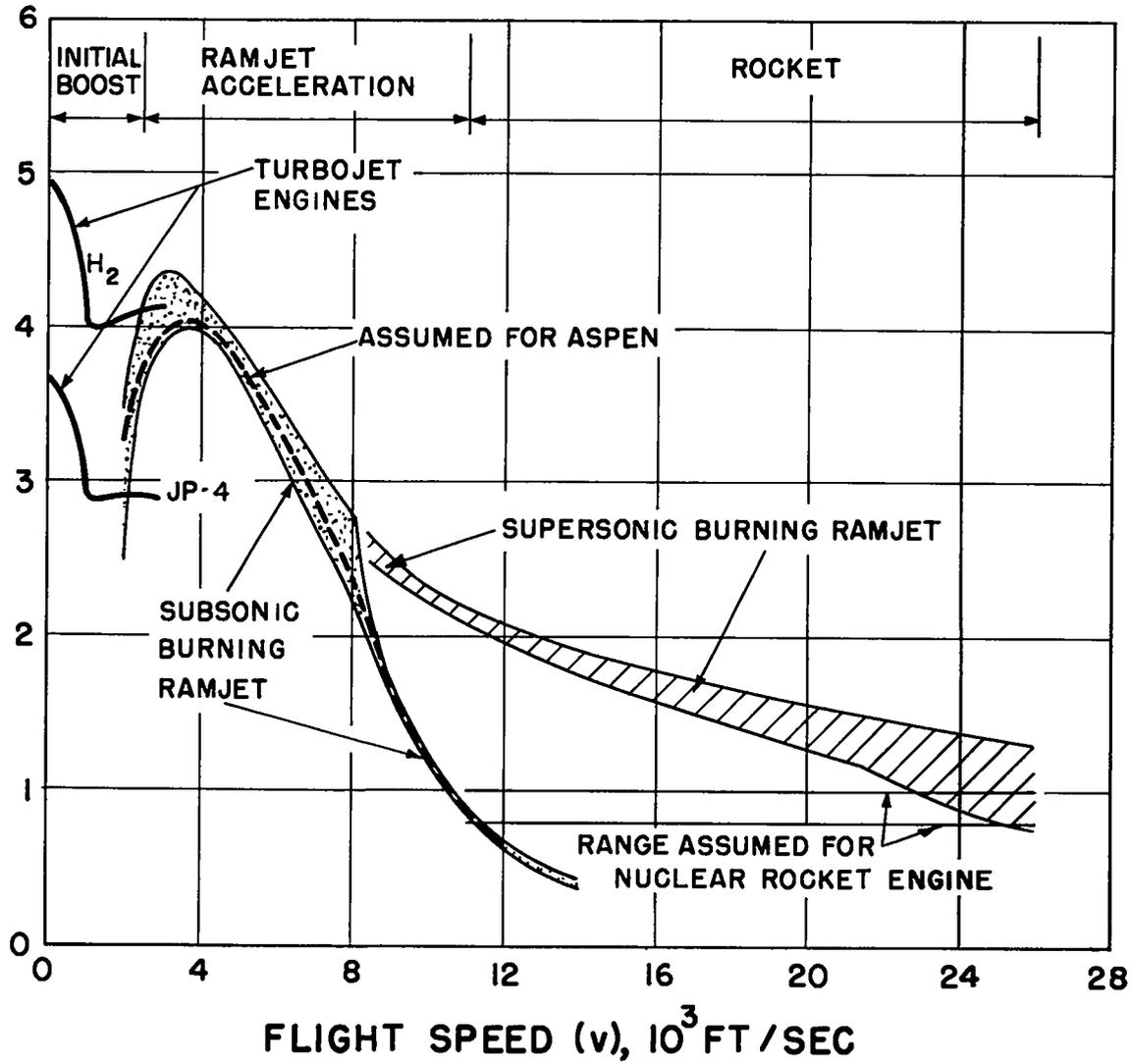


Fig. 6 ASPEN propulsion system performance.



Once the ramjet takeover speed of $M \sim 2.5$ is reached, the turbojet inlets are closed, thus reducing system drag, ramjet inlets are opened and ASPEN accelerates under ramjet thrust to an engine shutdown speed (for subsonic burning) of $M \sim 11$. The performance of such a hydrogen-fueled ramjet may be estimated on simple thermodynamic grounds with the aid of experimental information on inlet flow kinetic energy recovery. Results of such analyses, including consideration of real effects from wall cooling problems and combustion chamber pressure losses, are given in the reference reports (ASP, 1960). These and some results from independent work are portrayed in Figure 6 as a band bracketing the cited estimates,* together with a mean curve assumed for purposes of this study.

The figure shows ramjet engine thrust per unit flow rate of on-board propellant as a function of flight Mach number, for use of hydrogen combusted roughly stoichiometrically with air. The I_{sp} is seen to vary with flight speed, peaking to about 4000 sec at $M \sim 4.0$, and decreasing continuously thereafter. This behaviour is chiefly a result of decreasing expansion efficiency (lower pressure ratio) as speed drops below $M \sim 4$, and rapidly decreasing inlet pressure recovery and increasing internal losses, thus a less efficient engine cycle, as speed increases above $M \sim 4$. The relatively sharp decrease in I_{sp} shown above $M \sim 8$ for subsonic burning is the result of consideration of practical

*We neglect, for the moment, any energy input which would be due to reactor preheating of the hydrogen.



[REDACTED]

limitations on combustion chamber wall temperature. Inlet ram temperature increases with increasing Mach number; thus if net energy input (from combustion) per unit mass flow is held constant, the maximum gas temperature will rise. For a fixed combustion chamber geometry and size, such an increase must produce higher gas velocities (and larger pressure drop) in the combustion chamber, and changing physical properties with temperature lead to increased convective heat transfer coefficients. The net result is that it is not possible to keep wall temperatures to practical values with subsonic burning at flight speeds above about $M \sim 8$ by fuel cooling at stoichiometric air/hydrogen ratios, and resort must be made to use of excess fuel with resultant hydrogen-rich combustion gas mixtures.* Following this course of action, the I_{sp} drops to that attainable from a "conventional" nuclear rocket engine by the time flight speed of $M \sim 11$ has been reached. This is why $M \sim 11$ has been cited previously as the ramjet-shutdown/rocket-takeover speed. Although not practical at present, the figure also shows specific impulse values estimated for supersonic burning ramjets above $M \sim 8$.

As previously discussed, in order to use the curves of Figure 6 in assessing ASPEN performance we must know the (D/F) and (F/W) ratios as a function of flight speed. These may be obtained only from analyses of specific vehicle configurations, choice of engine size, duct inlet shape, etc., etc. Such studies are inherently nonanalytic in character, must involve a wealth of detail if done properly, and are beyond the scope of effort for this present study. Furthermore, a

*Some studies of this problem indicate that subsonic burning of near-stoichiometric mixtures may be possible without special wall cooling up to flight speeds of $M = 11$ to 12.

considerable number of such studies must be made if we are to find the optimum conditions which maximize payload-in-orbit.

If we choose air-breathing engines of small thrust and thus of small mass, the (D/F) ratio will be large, and net specific impulse will be reduced considerably from the values of Figure 6. The fraction of vehicle mass which reaches orbit will then be less than if a higher thrust level had been chosen for the atmospheric portion of flight. However, in the latter case a larger piece of the fractional mass delivered to orbit is taken up with engines, leaving less for payload. It is evident from this that optimum thrust levels and (D/F) ratios must exist which will maximize payload-in-orbit. Using the results of those reference studies (ASP, 1960) which are pertinent to the ASPEN concept,* as well as additional estimates of aerodynamic characteristics, based on the flat plate relations discussed earlier, an attempt was made to optimize performance in the air-breathing phases of flight, according to the discussion above. The results are shown in Figure 7, together with an estimate into the final boost phase of flight, above $M \sim 11$, on the assumption that initial acceleration at rocket takeover is between $g_0/2$ and g_0 (see following discussion). Using the drag/thrust ratio from Figure 7 to reduce the I_{sp} curve of Figure 6 assumed herein for subsonic burning, according to eq. (3), we obtain finally the net effective specific

* i.e., Those systems which are not penalized in drag by use of air liquefaction equipment.

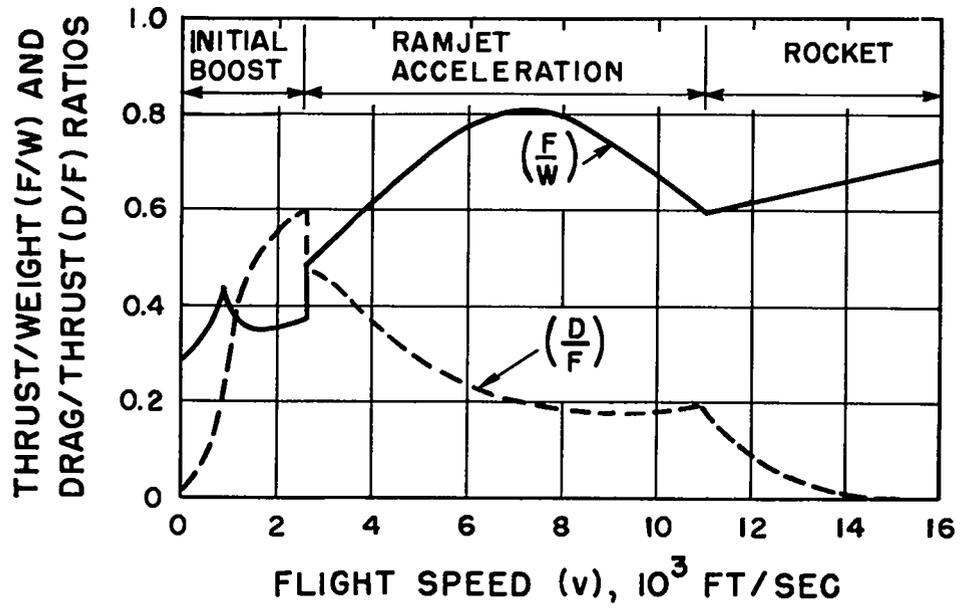


Fig. 7 ASPEN vehicle flight characteristics.

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impulse, including vehicle drag along the trajectory of Figure 5, as a function of vehicle flight speed up to orbital speed. This is shown in Figure 8.

Of course, the final boost phase acceleration is not really a free parameter; it is implicitly fixed by our ad hoc choice of trajectory, made earlier. We can use this assumed trajectory to work "backwards" and compute the acceleration required to yield the flight path. Results show that mean acceleration the order of g_0 is required. Some variation about this value is allowable without doing appreciable harm to vehicle performance, but accelerations markedly less than $(g_0/2)$ yield excessive gravitational losses (eq. 5) because the thrust must be used more and more to provide vehicle lift. This can be done only by progressively greater misalignment of the thrust and velocity vectors (assuming aerodynamic forces are small), thus reducing that available for acceleration parallel to the velocity. This effect as well as the increase in burning time required for generally lower acceleration flight both lead to a larger loss term ($t_b \overline{\sin \theta} \sim \Delta v_r \overline{\sin \theta} / \bar{a}$) in eq. (5). Calculations* show that this term is typically the order of 1100 $(\overline{W/F})$ ft/sec for the rocket part of the mean trajectory we have chosen in Figure 5, where $(\overline{F/W}) = \bar{a}$ is the mean acceleration during this phase. For $(\overline{F/W}) = 1/2$ (average acceleration of $g_0/2$) a penalty of about 2200 ft/sec equivalent additional burnout velocity must be assessed

* A detailed analysis of the flight trajectory above $M \sim 11$ is given in Appendix B.

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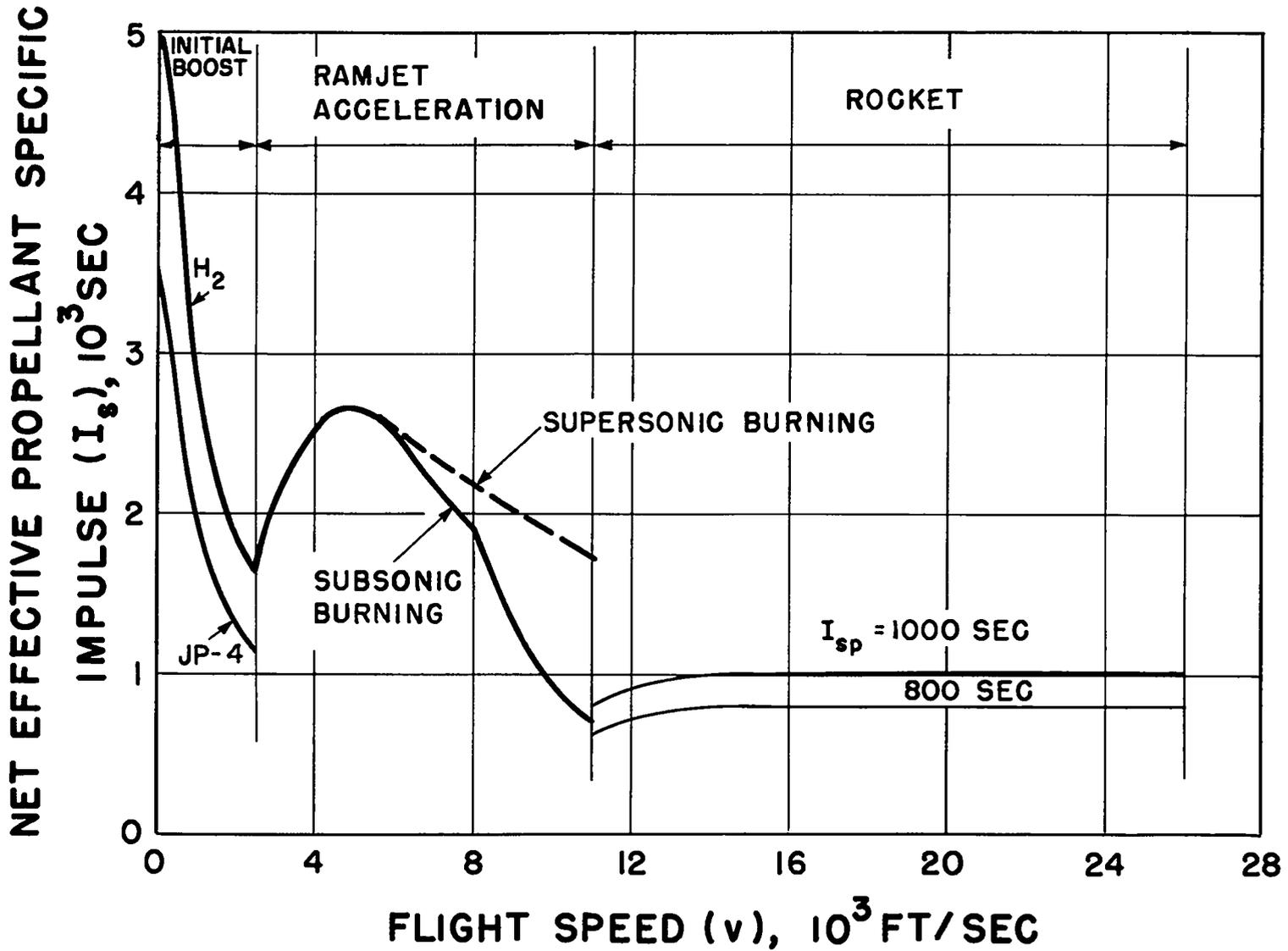


Fig. 8 Net thrust per unit propellant flow in ASPEN vehicle.

against the vehicle. Larger reactors will yield smaller velocity losses but will be heavier and require more shielding. The optimum reactor size (or optimum acceleration in rocket flight) which will yield maximum useful payload in orbit can be found only by numerical calculations using functional relations between ASPEN vehicle system component masses, operating power, and thrust output. If we assume that nuclear rocket engine mass (m_r) varies linearly with power output, and thus with acceleration at time of rocket takeover, according to $m_r = f_r m \bar{a}$, where m is mass of vehicle at this time, and \bar{a} is in units of g_0 , we find that optimum acceleration is given approximately by $50 f_r a_{opt}^2 \sim 1$. If $f_r = 0.06$ this yields $\bar{a}_{opt} \sim 1/\sqrt{3}$. We use $\bar{a} = 0.8$ throughout this report, which corresponds roughly to an initial rocket phase acceleration of $0.6 g_0$ and final acceleration of g_0 .

With this choice, the additional assumption that rocket propellant flow rate is held constant to burnout, and the curves of Figure 8, we can proceed with the evaluation of ASPEN flight performance by finite-interval calculations using eq. (5). Results of these calculations for the nominal ASPEN system are given in Figures 9 and 10, which show instantaneous velocity and fractional mass, respectively, as a function of flight time.

The fraction of gross mass at takeoff which actually reaches orbit* at 300 n.mi. is found to be 0.423. For comparison, system (3) of

*Note that all orbits considered herein are polar; i.e., they do not include any possible beneficial effects from eastward launching.

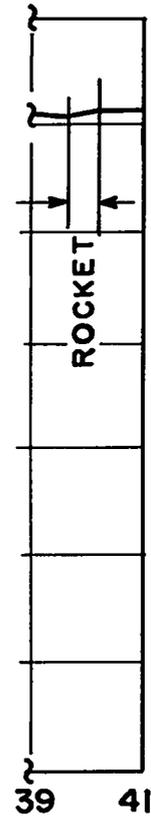
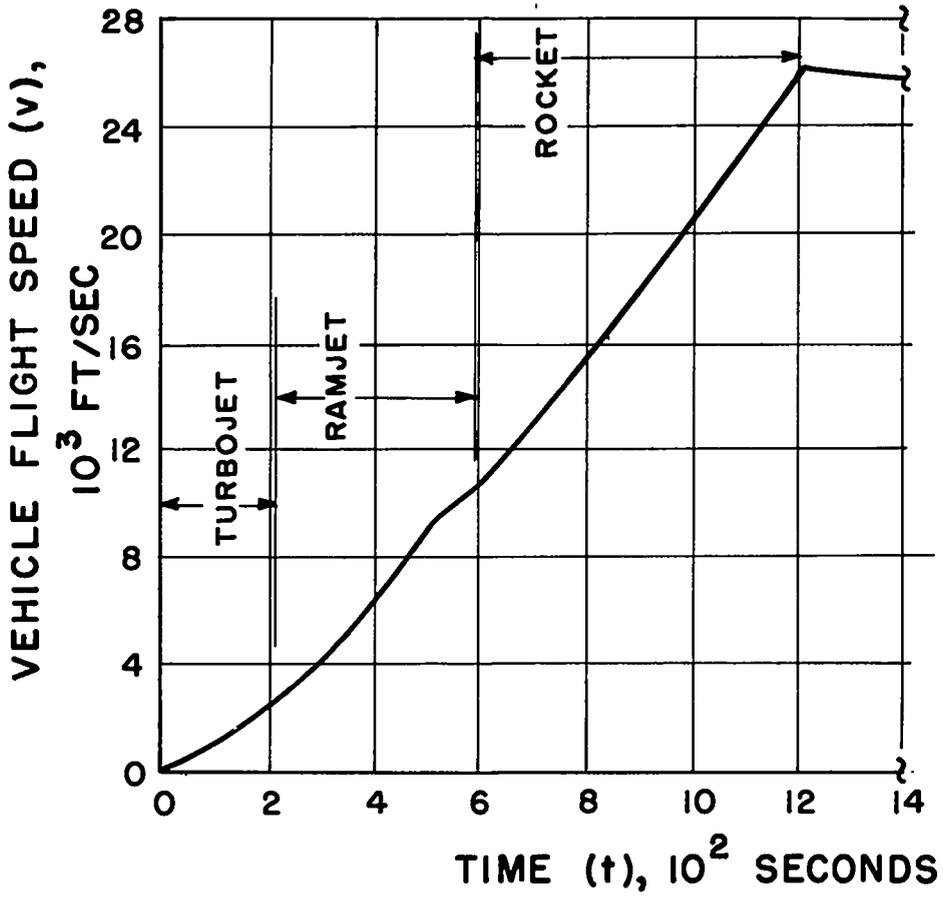


Fig. 9 ASPEN flight speed history.

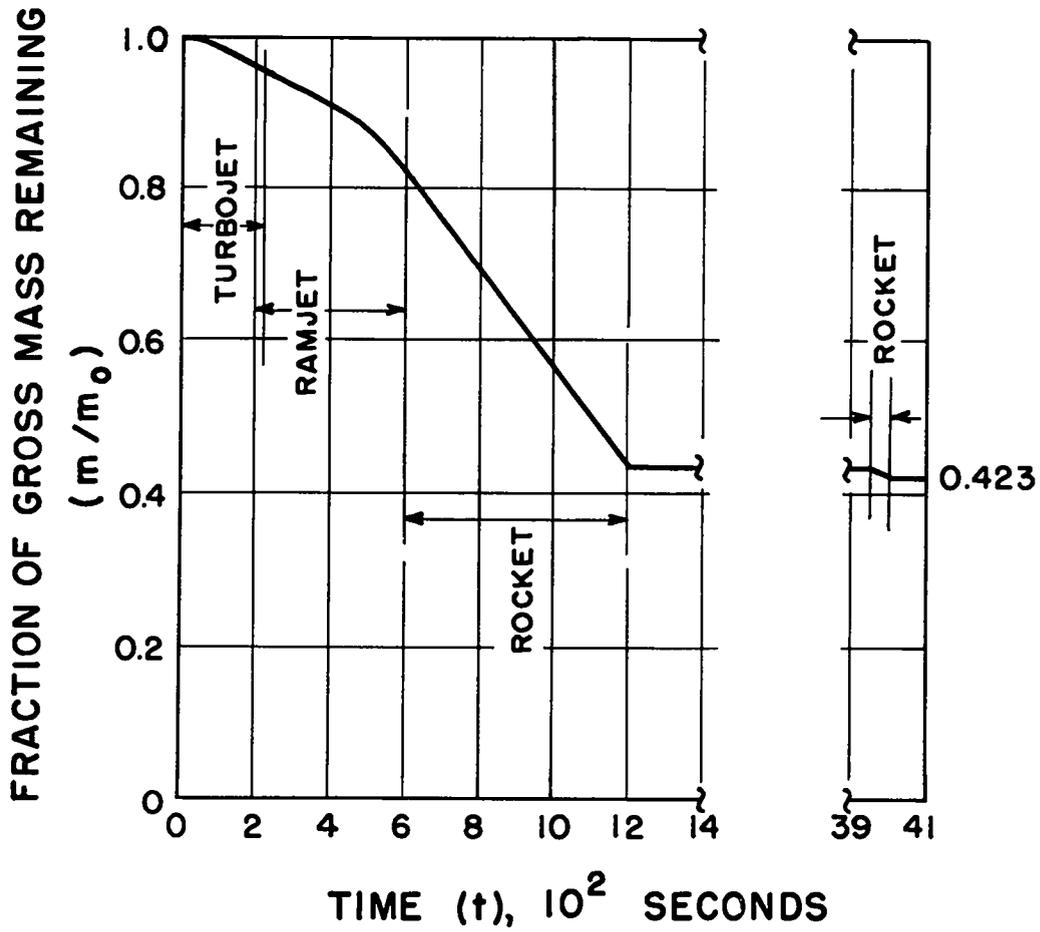


Fig. 10 ASPEN mass history; nominal vehicle.





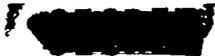
Table 2 will put only 0.35 to 0.38 of gross mass into orbit. For further comparison, if we take 0.20 for structure factor and 0.05 for the mass fraction devoted to non-nuclear propulsion equipment, then 0.173 of gross takeoff mass is left for payload, nuclear rocket reactor and shielding, crew compartment, and auxiliary crew equipment. This amounts to 87,000 lb of a vehicle weighing 500,000 lb at takeoff. If the reactor* plus shielding required to allow safe ground handling procedures 24 hours after shutdown weighs 47,000 lb, 40,000 lb are left as useful weight in orbit. This is to be compared with 5000 to 30,000 lb deliverable to orbit, according to Table 2, by chemical aerospace planes with structure and propulsion equipment mass fractions similar to those used above. From this it appears that even the most conservative and simple concept for ASPEN offers performance as good or better than that of its complex chemical competitors. Further, any improvement in performance beyond that assumed for the nominal ASPEN vehicle appears as large increases in payload capacity. The nominal system is good enough, but it does have the capacity for further growth without the introduction of extreme complication in propulsion systems.

(3) Possible improvement of performance

Examination of the performance curves for the nominal ASPEN vehicle shows that the greatest fraction of propellant is consumed during rocket operation, and relatively little is burned in the initial boost and ramjet acceleration phases of flight. This is a natural consequence of the fact that the average effective specific impulse is considerably

* Reactor power must be ~ 4900 Mw for initial rocket boost acceleration of $0.6 g_0$.





larger in these first two phases than in the last phase of flight. As a result it is clear that the most profitable place to seek improvement is in the rocket boost phase; little gains can be made without marked increases in performance in the other phases.

As previously discussed, the propulsion systems considered are as listed in Table 3 below.

TABLE 3
ASPEN PROPULSION SYSTEM POSSIBILITIES

Flight Phase	I Initial Boost	II Ramjet Acceleration	III Final Boost
Propulsion Systems	(a) Hydrocarbon-fueled turbojet	(a) Subsonic burning, hydrogen-fueled	(a) Conventional solid-core nuclear rocket engine ($I_{sp} = 800$ sec)
	(b) Hydrogen-fueled turbojet	(b) Supersonic burning, hydrogen-fueled	(b) Advanced solid-core nuclear rocket engine ($I_{sp} = 1000$ sec)
		(c) Nuclear preheated, supersonic burning hydrogen-fueled	

The combination IaIIaIIIa denotes the nominal vehicle previously analyzed. Analysis of the performance of systems using Ib and/or IIb and/or IIIb can be carried out using the data presented in



Figure 8 for effective specific impulse. However, the case of nuclear reactor preheating in phase II flight is not covered directly by these graphs. Let us consider this case briefly.

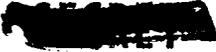
The addition of reactor energy with fixed air/hydrogen flow ratio simply yields higher gas temperatures in the combustion chamber. Since the engine performance is chamber-temperature limited anyway, this approach does not seem particularly useful. However, if air flow is increased (larger ducting), at given Mach number and with fixed hydrogen flow (system is now air-rich), so that the peak gas temperature remains the same in the combustion chamber as it was without reactor energy addition, the result is to increase the total thrust of the engine. If the hydrogen is heated to 4500°F (2500°C) in the reactor, energy equivalent to 33% of the energy of combustion is added to the gas and airflow must be increased by about 42% over stoichiometric to hold the temperature constant. Mean molecular weight increases slightly; and true specific impulse (based upon total mass flow) drops about 2%. The resulting propellant specific impulse is then, in theory, approximately 1.39 times that estimated for the non-nuclear air/hydrogen ramjet. This yields a peak $I_{sp} \sim 5600$ sec at $M \sim 4$. However, the attainment of this potential 39% increase in I_{sp} is hampered by problems of combustion chamber wall cooling, again, especially at the higher Mach numbers. Below about $M \sim 5$, increased air/hydrogen ratios appear allowable and the full theoretical gain seems possible. Above $M \sim 5$, with reactor power on, it becomes necessary to supply additional hydrogen for wall cooling, as


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was the case above $M \sim 8$ without reactor preheating.* If the primary mixture is air-rich, and transpiration cooling is used, coolant hydrogen will be able to react, thus releasing still more energy and defeating the purpose for which the coolant is used. If regenerative cooling is used, the reactor energy supplied per unit flow of hydrogen must decrease as the hydrogen is heated more and more by ramjet structure cooling, and less increase in performance is then possible. If the primary mixture is made stoichiometric at this point, again resulting in higher gas temperatures (i.e., those characteristic of $M \sim 6$ to 7 or so without reactor power) but without excess air, hydrogen wall injection cooling can be employed successfully. However, the potential gain in I_{sp} is now only $\sqrt{1.33} \sim 1.15$, and some of this is lost because of the use of wall coolant. As flight speed exceeds $M \sim 8$ the reactor energy is virtually unusable, if supplied to gases in the combustion chambers of a subsonic burning ramjet, because ram temperatures have become so high that combustion energy alone provides all the heat the walls and coolant system can handle.

The only way seen to make use of reactor energy over the complete speed range of ramjet operation is by the employment of supersonic combustion above about $M \sim 6$, allowing reaction of the combustible mixture to take place in a region of supersonic flow at low static pressure with consequent reduced heat transfer to the walls; thus wall cooling requirements are reduced somewhat. Because of this and the fact that the lower static temperature prevents any significant losses due to

* See footnote on p. 52.



energy-absorbing dissociation processes, the propellant specific impulse without reactor preheat can be held at the upper values shown in Figure 6, from $M \sim 8$ to $M \sim 11$ or higher. If reduced wall heating can be achieved as a result of supersonic burning, it should also be possible to obtain over this same range of flight speed the 39% increase in I_{sp} possible from reactor preheating. Beyond $M \sim 11$, the ramjet engines should be shut down and ASPEN flown on nuclear rocket power alone, in order to avoid the severe problems of airframe and skin heating which arise in atmospheric flight at higher Mach number.

Following the considerations outlined above, the flight performance was computed for ASPEN vehicles with a variety of propulsion system combinations, on the assumption (discussed in the next section) that the primary propulsion system equipment and reactor shielding weigh 0.15 of vehicle gross weight. Table 4 shows the results for five of the most interesting vehicles, for launching into polar orbits.*

These results show clearly the incentive to strive for high performance in the rocket part of the system, for increasing I_{sp} from 800 sec to 1000 sec increases the useful weight put in orbit by about 0.06 of the gross mass of any of the vehicles. If we consider a 500,000 lb vehicle this means that 30,000 lb more can be carried at 1000 sec than at 800 sec; about 150 lb additional load per sec of increase in specific impulse.

Vehicle (1) is the nominal vehicle, embodying the simplest system and most conservative set of assumptions used in this study.

* Equatorial eastward launching gives 0.02 additional weight fraction delivered to orbit.

TABLE 4
ESTIMATED PERFORMANCE OF FIVE ASPEN VEHICLES

Vehicle	Propulsion System	Mass-Fraction to Orbit	Useful* Weight in Orbit		Remarks
			$f_{\text{struct}} = 0.2$	$f_{\text{struct}} = 0.25$	
1	IaIIaIIIa	0.423	0.073	0.023	This is the nominal vehicle previously discussed.
2	IbIIbIIIa	0.445	0.095	0.045	Nominal vehicle converted to hydrogen turbojets and supersonic burning ramjets.
3	IaIIaIIIb	0.483	0.133	0.083	Nominal vehicle converted to advanced nuclear rocket.
4	IbIIbIIIb	0.507	0.157	0.107	Hydrogen turbojets, supersonic burning, and high I_{sp} rocket.
5	IbIIcIIIb	0.522	0.172	0.122	Above system with reactor preheat in ramjet phase.

*"Useful weight" includes payload, crew compartment, auxiliary crew equipment, and retrorocket system, and is estimated here on the assumption that 0.15 of gross mass is taken up with primary propulsion system equipment and reactor shielding.



Vehicle (2) differs from (1) only in its chemical propulsion systems, using hydrogen turbojets and supersonic burning in the ramjets from $M \sim 8$ to $M \sim 11$. Both of these improvements are probably within the capability of present day technology. The rocket reactor performance assumed is about that expected reliably before 1965 from graphite core reactors currently under development in the Rover program. Vehicle (3) uses the simpler set of chemical propulsion systems, but presumes use of a rocket reactor capable of yielding $I_{sp} = 1000$ sec from hydrogen propellant. At present such performance seems somewhat beyond the capabilities of graphite core reactors, but it may be attainable by use of fuel element base materials with higher melting points (e.g., metallic carbides) than that of uranium carbide in graphite. In view of the strong dependence of payload on rocket I_{sp} exhibited by the ASPEN vehicles, it would seem reasonable to pursue a program of reactor development oriented strongly toward the achievement of high gas temperature. Because of the present lack of information on reactor materials for ultra high temperature use, we can not now assert with confidence born of experiment that such a development lies within the framework of present day technology. Vehicle (4) combines both advanced chemical and advanced nuclear propulsion systems, while vehicle (5) has added nuclear preheat to the ramjet of (4). These latter two vehicles probably represent an upper limit on the performance attainable from any "practical"* sort of ASPEN system. Because of the greater complexity of the vehicle (5) system, the load advantage shown is probably illusory. If

* i.e., excluding such exotic notions as gaseous reactors and fusion propulsion.



[REDACTED]

this is the case there appears little reason to consider it further in comparison with vehicle (4).

An estimate of ultimate performance then seems to be that a vehicle of practical structure factor (B-70 has 0.25) should be able to put about 11% of gross mass into orbit as useful weight, while advanced, light construction ($f_{\text{struct}} \sim 0.20$) might yield a 16% useful weight capacity. In either case something like 51% of gross mass would arrive in orbit at rocket shutdown. For propulsion systems within the current state-of-the-art, about 42% to 45% of gross mass could reach orbit, corresponding to about 7% to 10% useful load for $f_{\text{struct}} = 0.20$ and 2% to 5% for the presently more practical value of $f_{\text{struct}} = 0.25$, always with 15% allowed for primary propulsion equipment and reactor shielding.

(4) Propulsion system and shield weights

The fraction of gross mass which must be allotted to turbojet and ramjet engines is determined by the thrust/weight ratios of the ASPEN vehicle and of the engines used. If $(F/W)_{\text{vehicle}} \sim 0.5$ (an average over the first 1/3 of ASPEN flight) and $(F/W)_{\text{engines}} \sim 20$, then the weight fraction devoted to engines must be 0.025 for each phase of air breathing flight, or 0.05 total aboard the vehicle. This simple estimate is consistent with those used in the reference studies (ASP, 1960).

The fraction of gross mass devoted to the rocket reactor is determined by its power/weight ratio and the thrust output per unit power. For operation yielding exhaust gas with $I_{\text{sp}} \sim 800$ sec, the thrust

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is roughly 50 lb/Mw of reactor power. Extrapolation to higher power of current reactor work indicates that a specific weight of 2 lb/Mw is probably attainable by 1965 from graphite core reactors at power the order of 4000 Mw. This gives $(F/W)_{\text{engine}} \sim 25$ and a rocket engine mass fraction of about 0.025 for initial rocket acceleration of $a_0 \sim 0.6 g_0$. Development of reactors of higher bulk power density seems possible to a practical limit the order of 300 Mw/ft³ of core volume (Bussard, 1953, 1955; Newman, Blevins, and Kirk, 1959). At this level of performance, reactor specific weight can be reduced to about 1 lb/Mw with a corresponding mass fraction of 0.0125 on board the vehicle.

Shielding is required to achieve our desired goal of low radiation dose rates outside the shield a day or so after shutdown from full power operation. Analysis of radiation leakage and accurate estimation of shield mass requirements is complicated by the problem of leakage through the propellant exit ducting. Reduction of this duct leakage may be accomplished in two ways. First, if we wish to avoid use of movable shielding, the duct may be "folded" to force multiple scattering of leakage photons, and the duct walls surrounded with shield material, as is the reactor proper, to reduce leakage losses of the scattering photons. Second, and inherently more straightforward in analysis, is the use of a movable shield "plug" to fill the exit duct cross-section and thus provide shielding all around the primary radiation source. This second concept may be invoked here because the shielding problem of principal interest is that of protection after reactor shutdown, to allow easy

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maintenance and permit use of existing air fields, not during reactor operation, which takes place only above 100,000 feet altitude.

Detailed analysis of the shielding problem, and further consideration of reactor and overall nuclear rocket engine system design are presently underway, and will be reported in a separate paper. Here we consider these from a simplified point of view in order to provide an estimate of the approximate numerical range of shield and reactor mass fractions. For thick shields (e.g., attenuation of 10^7 or greater) the error in such simple estimates is less than 20%.

The basic configuration we adopt is that of the movable plug shield assembly, discussed previously. For estimation of shield volume we assume that this configuration is equivalent to a cylindrical shell shield surrounding the reactor plus reflector, plus hemispherical end caps at each end of the cylindrical section. In analyzing shield attenuation for gamma photons we further assume a buildup factor of $(1 + \mu r)$ (conservative), an infinite cylindrical source (conservative), and a source diameter and density such that the source is many attenuation lengths thick (non-conservative). With these assumptions it is readily shown (Glasstone, 1955, Chapter X) that the radiation leakage flux at the outer surface of the shield at radius R_s is approximately related to that from the surface of the source (the reactor core) at radius R_0 after passage through a weakly attenuating reflector by

$$(6) \quad \frac{I_{\gamma}(R_s)}{I_0} = \sqrt{\frac{R_0}{R_s}} \left[E_1(\mu_s t_s) + e^{-\mu_s t_s} \right] \left[(1 + \mu_r t_r) e^{-\mu_r t_r} \right]$$

where μ_r and μ_s are attenuation coefficients and t_r and t_s are thicknesses of the reflector and shield material, respectively. $E_1(x) = \int_x^{\infty} e^{-y} dy/y$. The outgoing source leakage flux I_0 from a square cylindrical core (length = diameter) is given by

$$(7) \quad I_0 = P_{\gamma} / 4\pi \mu_0 R_0^3$$

where μ_0 is the attenuation coefficient and P_{γ} is the total gamma power of the source. This latter is related to the reactor operating power P_r , the length of operation t_o , and the time after shutdown t_s , approximately by the formula

$$(8) \quad \frac{P_{\gamma}}{P_r} = 0.006 f \left[t_o^{-0.2} - (t_o + t_s)^{-0.2} \right]$$

for time measured in hours. Here f denotes the fraction of all fission products generated during reactor operation which are retained within the core following shutdown. The leakage flux at any distance $R > R_s$ is just that from eq. (6) reduced by the ratio $(R_s/R)^2$. Making use of this fact, the conversion factors 1.6×10^{-19} Mw sec/Mev and 5.6×10^5 (Mev/sec cm^2)/(rad/hr), and combining eqs. (6) through (8) we obtain an expression for gamma dose rate at any position outside the shield of the ASPEN reactor, at any time after shutdown, as

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$$D_p(R) = 5.3 \times 10^9 f [t_0^{-0.2} - (t_b + t_0)^{-0.2}] [(1 + \mu_r t_r) e^{-\mu_r t_r}]^n$$

$$(9) \quad \times \left[E_1(\mu_s t_s) + e^{-\mu_s t_s} \right] \frac{(R_s/R_0)^{3/2}}{\mu_0 R_0 R^2} P_p$$

Here time is measured in hr, distance in cm, power in Mw, and dose rate in rad/hr. From this we see that smaller cores give larger dose rates, other parameters being equal. However, the mass of shielding around a small core is less for the same thickness than that around a large core; thus thicker shields may be used for the same shield mass around small cores than large ones. It is not possible to deduce by inspection of the above equation whether the mass of shielding for a fixed dose rate is greater or less for small core vs large core reactors. It is obvious, however, that the shield mass per unit power will be less for large cores than small if both systems operate at the same core power density, but that the shield mass per unit power can be made less for small cores at sufficiently high power density than for large cores at low power density, for the same total power output.

Some illustration of the situation is afforded by numerical examples for two different types and sizes of reactor. These are chosen to have the following characteristics:

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Reactor I: $P_r = 4900 \text{ Mw}$ (about 245,000 lb of thrust)

$R_o = 64 \text{ cm}$ (core power density of 85 Mw/ft^3)

Core consists of U-loaded graphite, with 35% void, and an average density of 1.20 gm/cm^3 .

Reactor II: $P_r = 700 \text{ Mw}$ (about 35,000 lb of thrust)

$R_o = 23.2 \text{ cm}$ (core power density of 255 Mw/ft^3)

Core consists of 0.2 UC, 0.2 ZrC, 0.10 Mo, 0.50 void, by volume, with an average density of 4.53 gm/cm^3 .

Both reactors: $t_r = 12 \text{ cm}$; material is Be, with 0.10 void for cooling holes; average density of 1.66 gm/cm^3 . Shield material is U^{238} , density 18.5 gm/cm^3 . Mass absorption coefficients are $0.030 \text{ cm}^2/\text{gm}$, $0.035 \text{ cm}^2/\text{gm}$, and $0.042 \text{ cm}^2/\text{gm}$ for Be, C, and U^{238} , respectively.

Core length is equal to core diameter, and core is reflected with equal thickness Be on top and sides.

$t_b = 0.278 \text{ hr}$ (1000 sec)

Results of calculations made for these conditions and materials are shown in Table 5, which gives values of the product $R^2 D_\gamma(R)$ and of the shield weight m_{sh} for several shield thicknesses. All values given for the former parameter are for retention of all fission products ($f = 1$) and for a time of one day after shutdown ($t_o = 24 \text{ hr}$). Figure 11 displays these results graphically, for two different source-receiver separation distances.

The power level of Reactor I is that required for propulsion of an ASPEN vehicle of about 500,000 lb weight at takeoff, and the power density assumed (by choice of core dimensions) is about that expected

[REDACTED]

TABLE 5

CHARACTERISTICS OF SOME REACTOR SHIELDS FOR ASPEN PROPULSION SYSTEMS

Thickness t_s (cm)	Reactor I		Reactor II	
	Attenuation Factor $R^2 D_p(R)$ (cm^2)(rad/hr)	Weight m_{sh} (lb)	Attenuation Factor $R^2 D_p(R)$ (cm^2)(rad/hr)	Weight m_{sh} (lb)
5	3.83×10^8	28,700	6.35×10^7	6,400
10	8.0×10^6	62,800	1.22×10^6	15,600
15	1.52×10^5	98,200	2.94×10^4	23,800
20	3.35×10^3	138,000	6.92×10^2	35,000
25	68	181,000	14.9	46,600

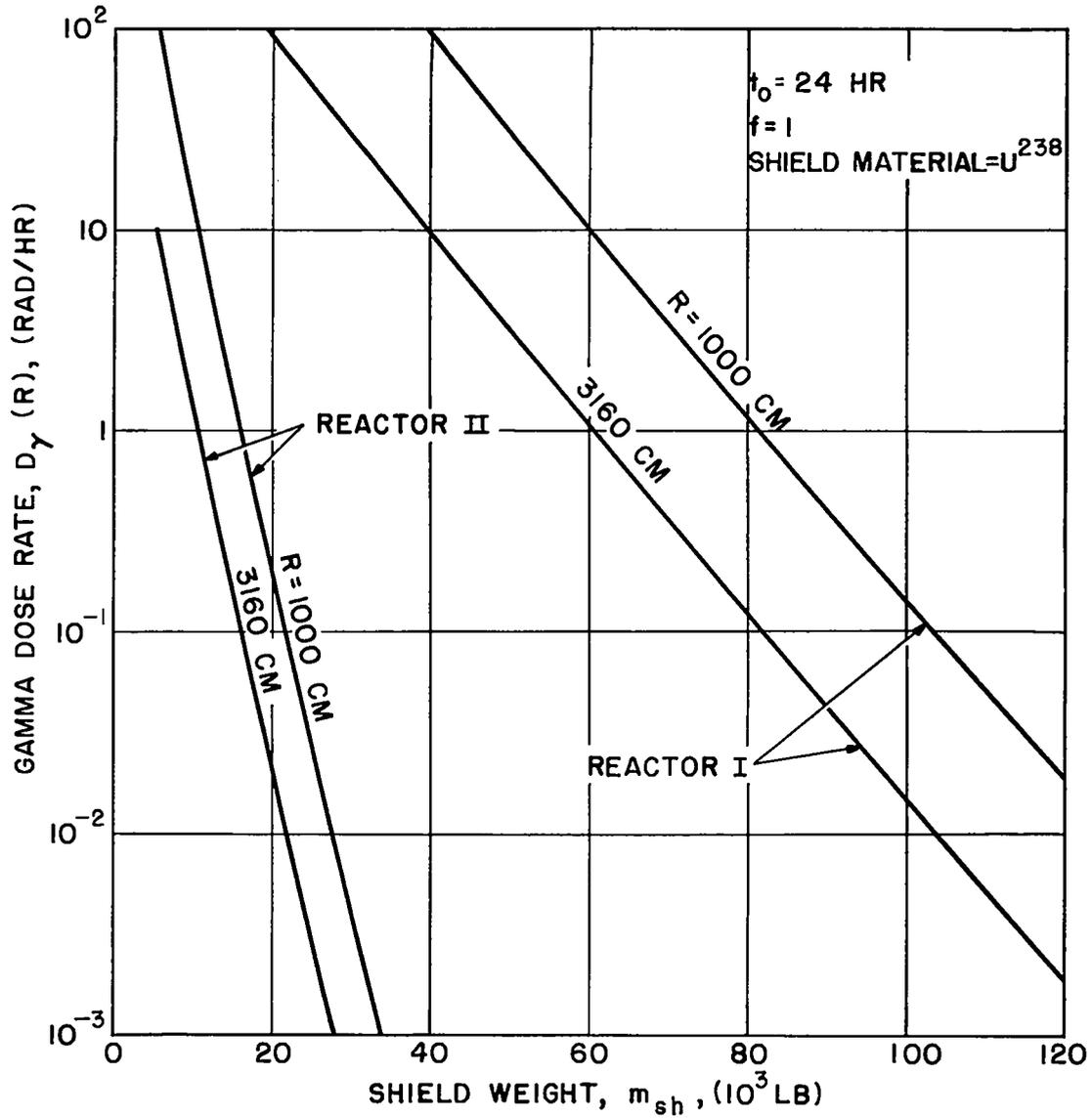
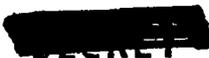
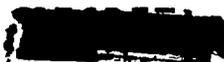


Fig. 11 Radiation dose rates one day after shutdown as a function of reactor shield mass.



from further development of the Kiwi-B Rover program reactors to the Phoebus power class. Direct calculation of the core plus reflector weight of this reactor gives 7640 lb. Assuming 10,000 lb for total engine weight (exclusive of reactor pressure shell, which is formed from the shield material itself) allows 2360 lb for nozzle, controls, turbo-pump plant, and miscellaneous piping. In order to keep within the 0.10 mass fraction assumed in Table 4 for the nuclear reactor powerplant plus shielding, it is then necessary to restrict shield weight to 40,000 lb. Inspection of Figure 11 shows that the dose rate one day after shutdown from 1000 sec of full power operation will be 9 to 90 rad/hr at distances of 3160 cm and 1000 cm, respectively, from the reactor. Five days after shutdown the dose rates would be reduced further by a factor of about 7. These dose rates are greater than is desirable for ease of handling and direct maintenance, yet are not so high that manned operations on the vehicle engines are rendered impossible without remotely operated equipment. Local shielding of U^{238} need be only about 8.9 cm (3.5 in) thick to reduce dose rates to 10^{-3} of the values cited above. Unfortunately, to achieve a similar reduction by use of thicker shielding built around the reactor is seen to require a shield weight of about 110,000 lb, very much larger than the nominal vehicle is capable of carrying to orbit and return. A significant improvement results if the core power density can be increased over that assumed for Reactor I. In particular, a power density 3 times that cited leads to a core radius of $R_0 = 44.4$ cm, and gives dose rates only 1/10 of those given above, so that the dose rate at



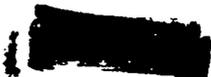
distances of 1000 cm and 3160 cm one week after shutdown would be only 0.9 rad/hr and 0.09 rad/hr, respectively. These values are small enough that ground handling should be relatively straightforward at later times. Of course, such long delays preclude frequent reuse of the vehicle, and thus defeat the purpose which motivates the desire for the vehicle in the first place. From the point of view of practical economics of an orbital ferry, we should direct our attention to the problems of a frequently-used vehicle and thus make shield mass estimates appropriate to safe handling at relatively short times (e.g., one day) following reactor shutdown.

Reactor II is chosen at a power level $1/7$ that of Reactor I, and has a power density 3 times larger. In concept it is a small, fast reactor, using metallic carbides, in contrast to Reactor I which is a moderated, epithermal, graphite system. In spite of its higher (assumed) power density, Figure 11 shows that the shield mass per unit power is larger for Reactor II than for I; a result due entirely to the smaller unit power size for the reactor itself. To attain the same power output as Reactor I, we must cluster 7 of the Reactor II engines; because of self-shielding inherent in the clustering, this will yield only about 4 times the dose rate from a single reactor. Again allowing 40,000 lb for shielding, the dose rate from a 7-reactor cluster is estimated from Figure 11 as 32 to 320 rad/hr at distances of 3160 cm and 1000 cm, respectively. These rates are about $3-1/2$ times larger than that from the single large reactor at lower power density, not sufficiently greater to make any qualitative difference in the ground shielding problem. To



attain the same dose rate, the shields must be increased to nearly 9000 lb/reactor, and the total shield weight will be roughly 60% greater for the Reactor II assembly than for Reactor I. The principal advantage in the use of Reactor II over Reactor I is that the unit size is smaller, thus allowing a range of vehicle sizes to be considered without also requiring consideration of a new reactor development program for each size vehicle. In addition, the use of clustered nuclear engines can increase the system reliability (up to some point of excessive clustering) in the event of failure of a single reactor, over that of a single Reactor I system where reactor failure always means failure to accomplish the mission. However, this additional reliability may prove to be slight, and may not be of any real interest in light of the abort recovery capability of the ASPEN vehicle by use of its chemical propulsion systems. In any event, whether propulsion is by clustered small engines or a single large engine, we see that post-shutdown dose rates are larger than desired if flyable shields are used in the system.

One obvious and straightforward way to solve this problem is by use of movable shielding added to the installed flight shielding while the vehicle is at its ground base. Here, as proposed some years ago in the ANP program, the simplest such shielding is that obtained by filling a tank around the nuclear reactor with a liquid gamma shield. The tank should add only a small weight to the propulsion system since it need not carry flight loads, and can be made from the material of the primary installed flight shielding, moved out to larger radius. It would be



: [REDACTED]

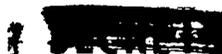
emptied of shielding prior to takeoff, but may be filled with jet fuel for the turbojet engines, leaving only the installed flight shielding as dead weight to be carried to orbit and return. It would be filled with liquid shielding shortly after ASPEN landing, reducing the external dose rate to a level set by the desires of the designer in sizing the thickness of the tank. For example, if Hg is used as liquid shield material, a tank thickness of 21.2 cm (8.34 in) will yield a dose reduction the order of 10^5 , giving a gamma dose rate of only 0.06 rad/hr on the surface of the shield tank at one day after shutdown. Dose rates at this level are the same order as those allowed for continuous exposure and do not represent any real radiation hazard. The weight of Hg shielding in the above tank would be 129,000 lb, and the weight of JP-4 which could be carried is 7900 lb. This amount of jet fuel is sufficient to provide a subsonic cruise for the order of 750 sec and 100 miles range at termination of the flight.

Additional flight shielding can be carried as improved performance is attained and ASPEN vehicles are developed with greater mass-in-orbit capability than that of the nominal vehicle. However, the nature of the variation of dose rate with shield weight is such that the reduction in post-shutdown dose rate by use of maximum possible additional shielding is probably not worth the loss in additional payload-to-orbit which would otherwise accrue to the more advanced vehicles. It seems likely that some movable ground shielding may always be employed profitably (though its use is not essential), and if so there is no reason to increase

[REDACTED]



the installed flight shielding beyond that required for in-orbit radiation safety and ease of initial handling at landing prior to the addition of ground shielding. From this point of view the shield requirements will be set in part by the manned operations desired while in orbit, where shadow shielding may suffice for protection. Although general analysis of the situation is not possible, preliminary estimates indicate that dose rates in the payload compartment can be kept below one rad/hr at one hour after shutdown with shield weight within the 40,000 lb taken as nominal for a 500,000 lb vehicle with a Reactor I propulsion system. Further detailed analysis of the shielding problem and its solutions is underway and will be reported separately.



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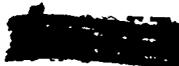
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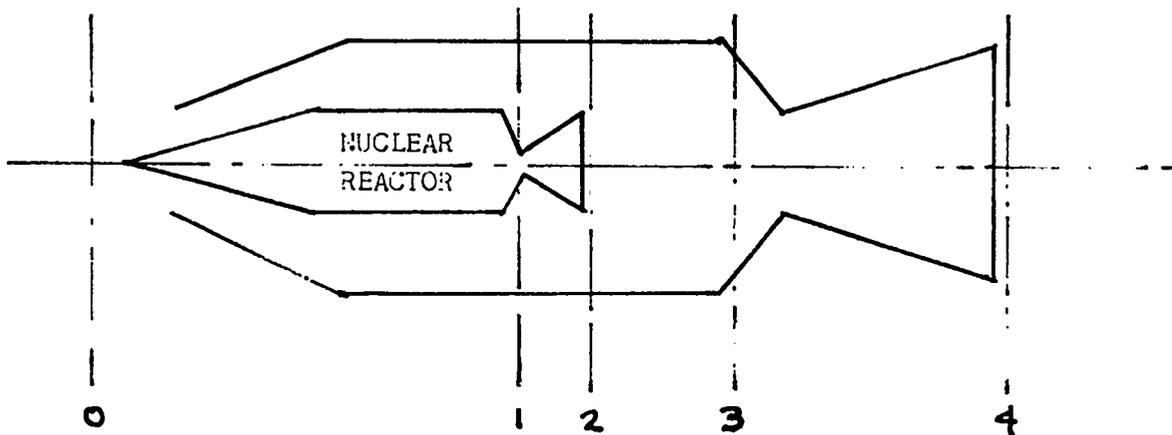


APPENDIX A

The Ducted Nuclear Rocket Aircraft

One of the potential applications considered for nuclear rocket reactors is as the prime power source for aircraft, manned or unmanned. This study considers the capability of nuclear rocket reactors used as ducted rockets ("ram-rockets") to propel aircraft.

A ducted nuclear rocket aircraft basically consists of an airframe, wings, fuselage, tail structure, etc., surrounding an air duct which contains a nuclear rocket motor. The air duct consists of a converging-diverging inlet diffuser section, a mixing, heating, and combustion section, and a converging-diverging exhaust gas exit section. The propulsion system is simply a ramjet engine with nuclear reactor heated gases used to provide static thrust and combustible fuel. Its geometry is sketched below.



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Unlike a conventional ramjet engine, such a system permits acceleration from zero velocity to design operating point.

It can be shown that the gross thrust of a ramjet engine per unit air flow is given by

$$(A-1) \quad \frac{F_{gr}}{w_a} = \frac{\eta_{nozz}}{g_0} \left[\frac{2g_0 r_{23}}{r_{23}-1} \frac{R_u}{(MW)_{23}} T_3 \left(1 - \frac{\phi_{M_0} (\eta_{diff})^{r_{01}-1/\gamma_{01}}}{\phi_{M_0} (\eta_{diff})^{r_{01}-1/\gamma_{01}}} \right) \right]^{1/2}$$

where

M_0 = flight Mach number

γ_m = ratio of specific heats at m^{th} station

$$\phi_{M_0} = \left[1 + \left(\frac{\gamma_{01}-1}{2} \right) M_0^2 \right]$$

R_u = universal gas constant = $1544 \frac{\text{lb}}{\text{ft}^2} \frac{\text{ft}^3}{\text{lb mol } ^\circ\text{R}}$

(MW) = molecular weight

T = gas temperature

η = component efficiency

The net specific thrust is given by

$$(A-2) \quad \frac{F_{net}}{w_a} = \frac{F_{gr}}{w_a} - \frac{M_0 v_0^*}{g_0}$$

where v_0^* = velocity of sound in free stream air

g_0 = acceleration of gravity at sea level = 32.2 ft/sec^2 .

The specific thrust per unit propellant flow rate, including rocket thrust, is the effective specific impulse and is given by

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$$(A-3) \quad \frac{F_{tot}}{w_a} = \frac{F_{net}}{w_a} \left(\frac{w_a}{w_f} \right) + \frac{F_r}{w_f} = I_s$$

Since the specific thrust depends strongly on the maximum bulk gas temperature T_3 , it is pertinent to analyze the temperature characteristics of a range of gas mixtures. The gas temperature after combustion is equal to the sum of the temperature of the duct gas at station 2, prior to combustion, the temperature rise due to combustion of duct gas with rocket gas, and the temperature rise due to mixing of the rocket gas with duct gas. Thus

$$(A-4) \quad T_3 = T_1 + \Delta T_{mix} + \Delta T_{comb}$$

For use of non-carbonaceous fuels (H_2O as only combustion product), the temperature rise due to combustion is given by

$$(A-5) \quad \Delta T_{comb} = \frac{H_c}{\left(\frac{w_a}{w_f} \right) c_{pa} + c_{p_{H_2O}} + \left(\frac{w_{a,comb}}{w_f} \right) (0.76 c_{p_{N_2}} + 0.24 c_{p_{H_2O}} - c_{pa})}$$

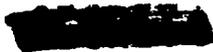
where c_p denotes heat capacity. Similar equations can be written for more complex fuels yielding a wider variety of mixed combustion gases.

As a first approximation the mixture temperature rise (or decrease) due to rocket and uncombusted duct gas mixing is assumed to be

$$(A-6) \quad \Delta T_{mix} = \frac{T_e - T_1}{1 + \left(\frac{w_a}{w_f} \right) \left(\frac{c_{pa}}{c_{pf}} \right)}$$

where T_e is propellant (fuel) gas temperature at exit from reactor nozzle.

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The rocket thrust per unit fuel flow rate is obtained from standard rocket gas flow equations as

$$(A-7) \quad \frac{F_r}{w_f} = \frac{\eta_{nozzle}}{g_0} \left[\frac{2g_0 \gamma}{\gamma-1} \frac{T_u}{(MW)_r} T_c \left(1 - \left(\frac{P_e}{P_c} \right)^{\gamma-1/\gamma} \right) \right]^{1/2}$$

where T_c is gas temperature and P_c is pressure at reactor core exit.

For proper rocket nozzle expansion the nozzle exit static pressure should be about equal to the ambient pressure within the duct. Thus

$$(A-8) \quad P_e \geq P_i$$

Standard nozzle equations also show that

$$(A-9) \quad T_e = T_c \left(\frac{P_e}{P_c} \right)^{\gamma-1/\gamma}$$

The stagnation temperature of the duct air is given by

$$(A-10) \quad T_i = T_0 \phi_{M_0}$$

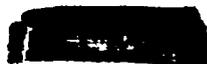
and the stagnation pressure after diffuser compression is

$$(A-11) \quad P_i = P_0 \eta_{diff} \phi_{M_0}^{\gamma_0/\gamma_0-1}$$

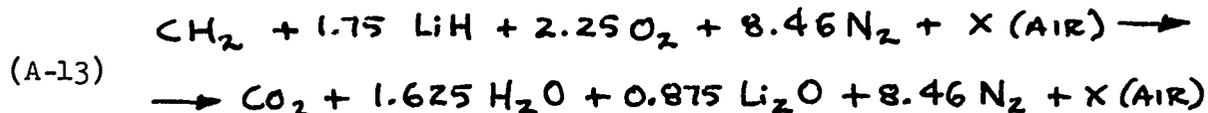
Detailed calculations were made with these equations for a propellant combination consisting of a 50/50 (by weight) mixture of LiH and CH_2 (JP-4). Using standard physical properties data it was found that the reactor power is related to the rocket thrust by

$$(A-12) \quad P_r (Mw) = 1.71 \frac{F_r (lb)}{\sqrt{T_c (^\circ R)}}$$

The available energy of combustion of such a mixture is 21,500 Btu/lb. Combustion of this mixture with air was taken to proceed according to



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The following conditions and properties also were assumed in order to carry through calculations:

$$\begin{aligned} T_o &= 400^\circ\text{R} \\ \eta_{\text{nozz}} &= 0.95 \\ T_3 &= T_c \end{aligned}$$

η_{diff}	0.91	0.68	0.35	0.15
M_o	2	3	4	5

The diffuser efficiencies given correspond roughly to single shock compression (NACA 1952, 1954 a,b) although higher efficiencies are possible, in theory, by use of multiple shock compression systems.

Using the above, and standard physical properties data where applicable, net engine performance was computed for flight over the speed range from $M = 2$ to $M = 5$. Results are shown in Figures (A-1) and (A-2) which show, respectively, net thrust per unit air flow, and air/fuel flow ratios as a function of maximum gas temperature $T_3 = T_c$ and flight Mach number.

To use these data with the foregoing equations to estimate aircraft performance it is necessary to relate aircraft gross weight to reactor power and ducted rocket engine performance, through functional relations connecting these primary engine parameters with vehicle component weights. Having done so, the range/payload capability of the aircraft

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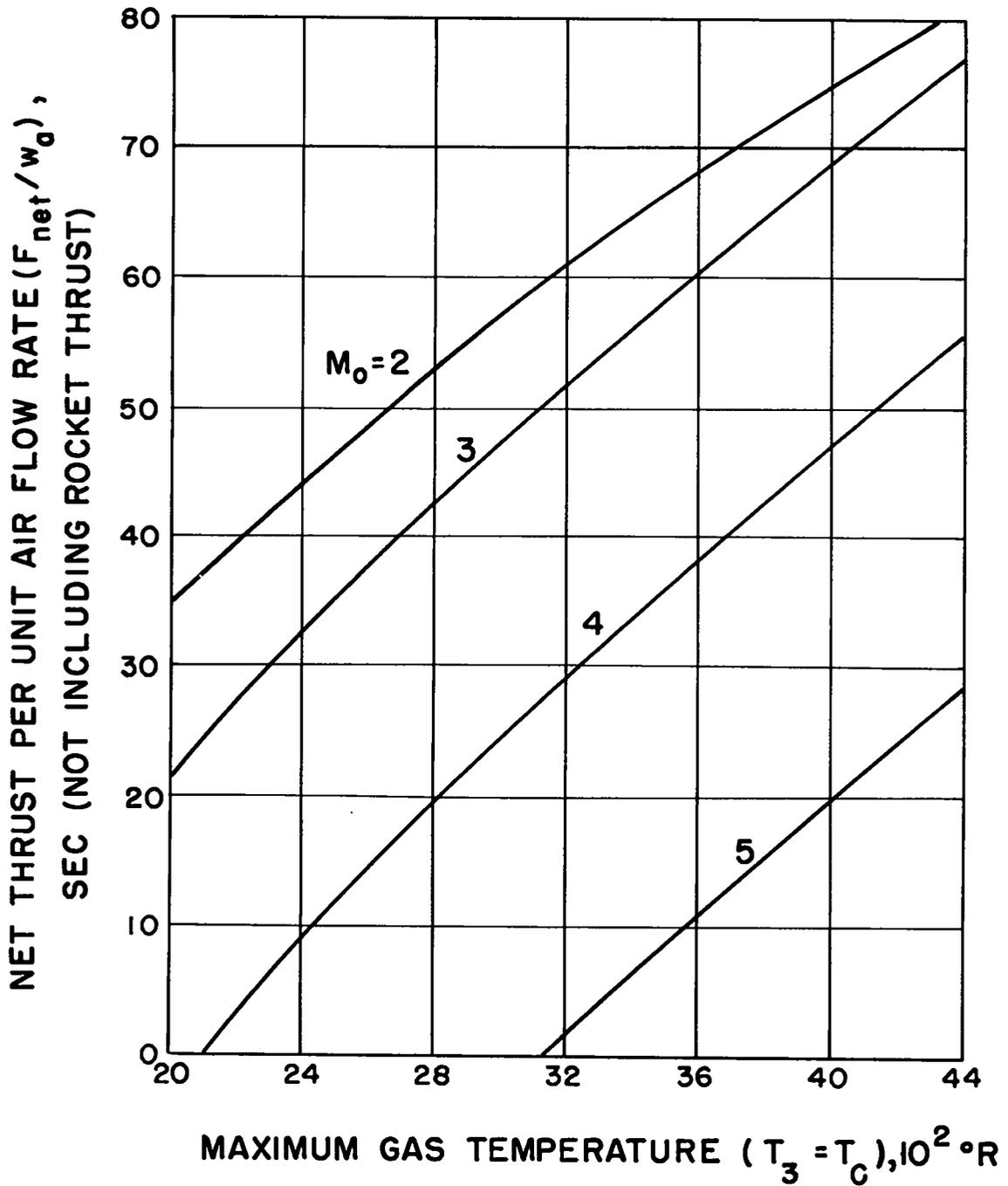
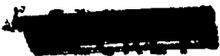
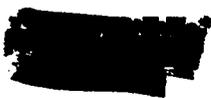


Fig. A-1 Ducted rocket ramjet performance.



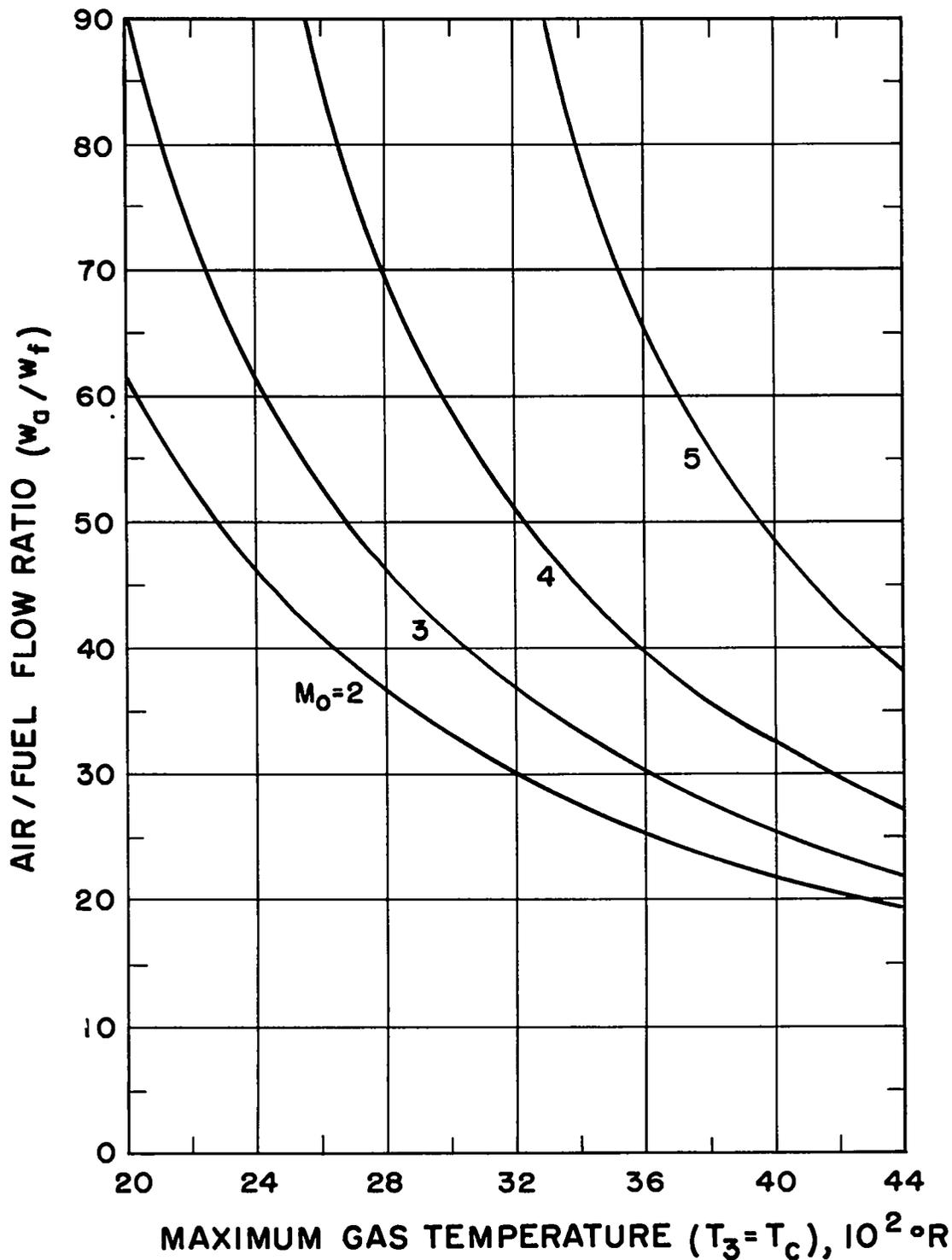


Fig. A-2 Ducted rocket ramjet performance.



can be computed for any specific variation with flight speed of vehicle (L/D) ratio. Calculations were made for an assumed (very conservative) variation of (L/D) according to

(L/D)	5.5	4.4	3.5	2.6
M_0	2	3	4	5

and a reactor specific weight of 10 lb/Mw.

Results of these are shown in Figure (A-3) which plots the ratio of dead load (payload plus shield weight) to gross weight at takeoff as a function of vehicle range for continuous powered flight at various Mach numbers.

We see from the figure that little payload can be carried beyond 1000 miles at the highest flight speed, and that even at the lowest speeds considered the payload/range capability shown is not striking. The dead load capacity is a rather sensitive function of (L/D) ratio; thus these conclusions are in part a natural consequence of the rather pessimistic (L/D) values assumed above, particularly for flight at high Mach number. For example, doubling the (L/D) ratio given for $M = 5$ gives a fractional dead load of about 0.38 at $R = 1300$ mi., and the $M = 5$ performance curve becomes very similar to that shown for $M = 2$ and 3. Another possibility for increased performance appears by choice of H_2 as propellant, rather than 50/50 (LiH/CH₂), for range is closely dependent on fuel combustion energy and heat capacity per unit mass.



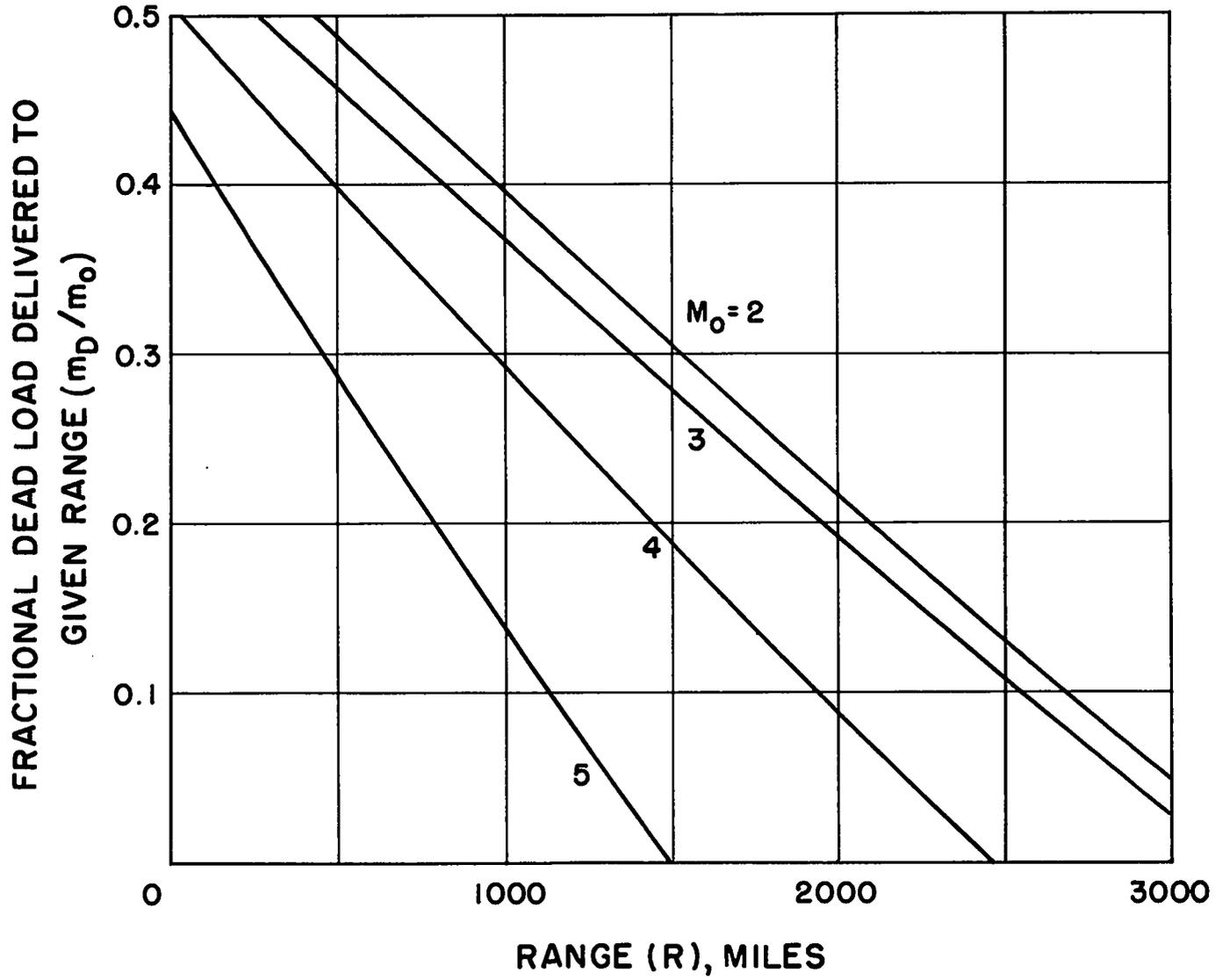


Fig. A-3 Performance capability of ducted nuclear rocket aircraft.





Using hydrogen it appears that the range might be increased as much as 80% over the values shown in Figure (A-3), for the same fractional dead load capacity. Even so, such a vehicle does not appear to offer performance comparable to that of the B-70 aircraft and introduces some of the operational problems of nuclear-powered aircraft (e.g., significant air-scattering of leakage radiation) which are avoided in the ASPEN vehicle concept and are totally absent from the B-70.



APPENDIX B

ASPEN Trajectory on Rocket Power

We consider here some of the details of motion of the vehicle following ramjet shutdown at $v = 11,000$ ft/sec ($M \sim 11$) and $h \sim 120,000$ ft. At this point the nuclear rocket reactor is brought to full power and operated at this condition until the desired altitude and flight speed are attained for subsequent coast to apogee height of 300 n.mi.

At ramjet shutdown the vertical component of vehicle velocity (the "rate-of-climb") is not large, because the vehicle has been flown as a lift-supported aircraft to this point. In order to minimize drag and other losses in rocket operation and to avoid the problem of excessive aerodynamic heating in ultra-high-speed flight, it is desirable to climb rapidly out of the sensible atmosphere. This can be done with least expenditure of rocket propellant by use of an aerodynamic maneuver, pulling the vehicle into a steep angle of attack, giving greatly increased lift force, and maintaining this attitude until a rate of climb has been attained sufficiently large to assure (together with vertical component of thrust) continued upward motion throughout the rest of the rocket operating time. Once the desired rate of climb has been reached the angle of attack is reduced to near-zero to give minimum total drag, and the rocket thrust is applied at the angle appropriate to maximize final vehicle speed tangential to the Earth's surface with minimum propellant expenditure.

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Analysis of this sequence of events is done most readily by use of Newton's Law ($F = ma$) equations of the form of eq. (1). The vertical acceleration in the non-lifting part of flight can be written as

$$(B-1) \quad a_y(t) = \frac{d^2y}{dt^2} = -g_0 \left[1 - \left(\frac{v(t)}{25,000} \right)^2 \right] + \bar{a} \overline{\sin \theta}$$

where \bar{a} is time-average rocket thrust acceleration and $\overline{\sin \theta}$ is time-average sine of the angle between the rocket thrust vector and the horizon plane. The instantaneous horizontal velocity is given by

$$(B-2) \quad v_x(t) = \frac{dx}{dt} = \int_0^t a_x(t) dt = v_1 + \bar{a} \overline{\cos \theta} t$$

where v_1 is horizontal velocity at end of the aerodynamic lifting phase of flight. Using eq. (B-2) we can integrate eq. (B-1) and obtain the vertical velocity as

$$(B-3) \quad u(t) = u_1 - g_0 t + \frac{g_0}{(25,000)^2} \left[\frac{(\bar{a} \overline{\cos \theta})^2 t^3}{3} + (\bar{a} \overline{\cos \theta}) t^2 v_1 + v_1^2 t \right] + \bar{a} \overline{\sin \theta} t$$

where u_1 is vertical velocity at end of the aerodynamic lifting phase of flight. Time zero in the above equations is taken at this point.

The lifting phase has a duration given by

$$(B-4) \quad t_L = u_1 / \bar{a}_{up}$$

where \bar{a}_{up} is the average net space-acceleration in the vertical direction during time t_L . This is not a free parameter but is related to the gain in altitude Δs during lift by

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$$(B-5) \quad u_1 = \sqrt{2 \bar{a}_{up} \Delta s}$$

The upward acceleration in lifting flight is also given in terms of the acceleration \bar{a}_L due to aerodynamic lift, the local gravitational field strength, and the upward component of rocket thrust by

$$(B-6) \quad \bar{a}_{up} = \bar{a}_L - g_0 \left[1 - \left(\frac{v_0}{25,800} \right)^2 \right] + \bar{a} \sin \theta$$

In writing this counterpart to eq. (B-1) we have ignored the (small) change in horizontal flight speed which will take place during time t_L . At end of lifting flight this speed is given by

$$(B-7) \quad v_1 = v_0 + (a_0 - \bar{a}_D) t_L \cos \theta$$

where a_0 is force acceleration due to rocket thrust at time of rocket takeover, and \bar{a}_D is the t_L -time average force acceleration due to aerodynamic drag. This expression can be rewritten in a form more convenient for our purposes by the introduction of some additional defining relations between variables. First we desire to replace a_0 , initial rocket thrust acceleration, by \bar{a} . Without proof here, but based upon numerical calculations of vehicle flight, we will use $a_0 = 0.8 \bar{a}$. Let us also define the average force acceleration due to lift as $\bar{a}_L = \bar{a}_D (L/D)$, where (L/D) is the lift/drag ratio during time t_L ; then eq. (B-7) becomes

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$$(B-8) \quad v_1 = v_0 + \left[0.8\bar{a} - \frac{\bar{a}_L}{(L/D)} \right] t_L \overline{\cos \theta}$$

In addition to horizontal and vertical speeds we desire to know the vertical position as a function of time. This can be found by integration of eq. (B-3) which yields

$$(B-9) \quad y(t) = y_0 + u_1 t - \frac{g_0 t^2}{2} + \frac{g_0}{(29,800)^2} \left[\frac{(\bar{a} \overline{\cos \theta})^2 t^4}{12} + \frac{(\bar{a} \overline{\cos \theta}) t^3}{3} v_1 + \frac{v_1^2 t^2}{2} \right] + (\bar{a} \overline{\sin \theta}) \frac{t^2}{2}$$

For convenience in use of eqs. (B-2), (B-3), and (B-9) we must change the time variable to $t \rightarrow (t-t_L)$ where the new time variable is now measured from zero at time of ramjet-shutdown/rocket-takeover. Two conditions must restrict the choice of parameters in our problem. One is that the horizontal velocity at end of rocket operation must be that at the perigee of a transfer orbit extending from rocket shutdown altitude to 300 n.mi.; and the other is that the rate-of-climb must be zero at this apogee point. From these we write the restrictive relations

$$(B-10) \quad (v_b - v_0) = (\bar{a} \overline{\cos \theta})(t_b - 0.2 t_L) - \frac{\bar{a}_L}{(L/D)} t_L$$

from eqs. (B-2) and (B-8), and

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$$\begin{aligned}
 (B-11) \quad u_1 \geq g_0 (t_b - t_L) - \frac{g_0}{(25,800)^2} & \left[\frac{(\bar{a} \cos \theta)^2 (t_b - t_L)^3}{3} + v_1^2 (t_b - t_L) + \right. \\
 & \left. + (\bar{a} \cos \theta) (t_b - t_L)^2 v_1 \right] - (\bar{a} \sin \theta) (t_b - t_L)
 \end{aligned}$$

from eq. (B-3). In addition to these it is of interest to know the total gain in altitude during rocket operation. This is found from eq. (B-9) as

$$\begin{aligned}
 (B-12) \quad \Delta h = y(t_b) - y_0 = -\frac{g_0}{2} (t_b - t_L)^2 + u_1 (t_b - t_L) + \\
 + \frac{g_0}{(25,800)^2} & \left[\frac{(\bar{a} \cos \theta)^2 (t_b - t_L)^4}{12} + \frac{(\bar{a} \cos \theta) (t_b - t_L)^3}{3} v_1 + \frac{(t_b - t_L)^2 v_1^2}{2} \right] + \\
 + (\bar{a} \sin \theta) & \frac{(t_b - t_L)^2}{2}
 \end{aligned}$$

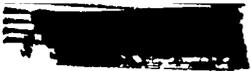
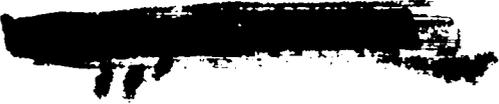
In these three equations t_L and v_1 must be taken from eqs. (B-4) and (B-6), and eq. (B-8), respectively.

For our case we desire that the altitude gain be $\Delta h = 3.65 \times 10^5$ ft, to reach 80 n.mi. from 120,000 ft; and that the final velocity at 80 n.mi. be $v_b = 26,015$ ft/sec when starting from $v_0 = 11,000$ ft/sec. Furthermore, calculation of vehicle lift as a function of altitude shows that aerodynamic support is not practical much above 180,000 ft. At this point the lift is only 5% of its value at 120,000 ft, for the same angle of attack of the lifting area. Some adjustment and maintenance of lift can be achieved by increasing angle of attack with increasing altitude, but

this increases drag due to lift and reduces the (L/D) ratio. If we choose 170,000 ft as cutoff altitude for aerodynamic lift, reducing angle of attack to the minimum drag condition at this point, eqs. (B-4) and (B-5) severely restrict the upward velocity and lift time possible with reasonable upward acceleration.

Choosing $\bar{a}_{up} = 3 g_0 = 96.6 \text{ ft/sec}^2$ and $\Delta s = 50,000 \text{ ft}$ we find that $t_L = 32.2 \text{ sec}$ and $u_1 = 3110 \text{ ft/sec}$. Next, assuming $\bar{a} = 0.8/g_0$, $\overline{\cos \theta} \sim 0.93$, and $\overline{\sin \theta} \sim 0.38$ (all to be verified later), we compute the lift acceleration from eq. (B-6) to be $\bar{a}_L = 115 \text{ ft/sec}^2 = 3.6 g_0$. Then using the values obtained above and assuming that $(L/D) \sim 6$ we find the rocket operating time from eq. (B-10) to be $t_b = 604 \text{ sec}$. With this and the above result for u_1 we can use eq. (B-11) to compute the average sine of the rocket thrust vector angle with the horizontal, required to yield zero vertical velocity at rocket shutdown. The result is $\overline{\sin \theta} = 0.38$, as assumed above; correspondingly $\overline{\cos \theta} = 0.93$. The equivalent velocity loss in non-lifting flight is then

$\delta v_2 = (v_b - v_0)(1 - \overline{\cos \theta}) = 1050 \text{ ft/sec}$. In making the foregoing calculation it was necessary to compute v_1 , from eq. (B-8). This was found to be $v_1 = 11,050 \text{ ft/sec}$. From this we can estimate the equivalent velocity loss during high lift flight as $\delta v_1 = v_0 + \bar{a}t_L - v_1 = 750 \text{ ft/sec}$; thus the total equivalent velocity loss due to drag and gravity is $\delta v = 1800 \text{ ft/sec}$. In the body of the text this was split into a gravitational loss component of roughly $1100 g_0/\bar{a} = 1400 \text{ ft/sec}$ and a drag loss component (through reduction of rocket I_{sp} by (D/F) ratio



according to eq. (3)) of about 400 ft/sec. The remaining point of interest is the gain in altitude during rocket flight. This was computed from eq. (B-12) and found to be $\Delta h \sim 150,000$ ft. This yields a rocket shutdown altitude of only about 45 n.mi., somewhat lower than assumed in the body of the text. Our choice of numerical values probably is not optimum. It seems likely that other values could be chosen which would give the desired altitude as well as satisfy the other conditions imposed on the problem; however, our purpose here has been to illustrate the simplified dynamics of ASPEN rocket flight, not to make a parametric study of interrelationships between variables.

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