Navigation and Guidance for a Mars Aerocapture Mission: A System Design Challenge

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Current NASA plans for the exploration of Mars call for a variety of unmanned spacecraft to be sent to explore the planet, both from orbit and on its surface, before the initiation of manned missions. Some of these unmanned precursors will be the largest and most complex interplanetary space probes ever flown. Aerocapture, a concept for performing orbit insertion using a controlled flight through a planetary atmosphere to reduce a spacecraft's velocity, has received a great deal of attention as a means for delivering much larger payloads into Mars orbit than could be accomplished using solely propulsive means. This article presents a brief survey of some of the important issues in aerocapture navigation and guidance system design and development. It quickly becomes apparent that this design cannot be performed independently; it must be viewed as but one element of an aerocapture delivery system, whose other major elements are the aerocapture vehicle itself and the Deep Space Network (DSN) radio navigation system. Since these precursor missions will likely be very costly, another important issue in aerocapture system design will be ensuring a very high probability of mission success without the benefit of a test flight program. Some methods used by the aerospace industry to develop complex, highly reliable systems are reviewed to assess their applicability to aerocapture navigation and guidance systems, and some suggestions are made for the design and testing of these systems, which may yield very high reliability levels.

I. Introduction

All of the previous unmanned space probes that have been sent into orbit around other planets have used rocket propulsion systems to accomplish orbit insertion. The rocket engine and its fuel are often a significant portion of the total mass that must be injected into the interplanetary trajectory needed to reach the target planet. Launch vehicles, both those used in the past and those presently in service, generally have very limited payload capabilities for interplanetary missions, with the exception of the Soviet Energia booster. Thus, launch vehicle limitations have provided strong motivation in the search to find other ways of performing orbit insertion.

The idea of using aerodynamic drag in a controlled manner to modify the orbit of a spacecraft originated in the early 1960s [1]. Aerocapture is just one of several aerodynamic-maneuver concepts that have been proposed as a means of altering the size, shape, and orientation of an orbit. It is a trajectory-control scheme in which a spacecraft, using some type of aerodynamic shell that allows it to execute controlled flight in an atmosphere at hypersonic speeds, makes one or more passes through a target planet's atmosphere. This passage (or passages) removes enough energy through aerodynamic drag to "capture" the vehicle into an elliptical orbit. After the aerocapture is completed, a propulsive maneuver must be executed to raise the periapsis (closest approach) altitude above the atmosphere, so that the vehicle does not undergo reentry on subsequent orbits. Aerocapture is sometimes referred to as aerobraking, a term which also encompasses the larger family of related aerodynamic-maneuver schemes.

Preliminary studies of Mars Rover/Sample Return missions indicate that large vehicle masses of ~3000 kg will be needed. It is highly desirable to complete these missions with as few launches as possible, since launch costs can often be a substantial portion of the total cost of a mission. At present, aerocapture is considered to be the preferred method for performing Mars orbit insertion for these missions, since it allows increases of 30 to 50 percent in payload mass over that achievable with a propulsive orbit insertion [2]. The amount of payload increase is not constant, but depends upon the Mars approach trajectory used by the spacecraft which, by necessity, is different for each Mars launch opportunity. Aerocapture is considered to be a "mission-enabling" capability for some types of Mars missions, but not for others. Manned missions to Mars using propulsive orbit insertion may require a great number of launches at great cost; hence, aerocapture may be a necessity for carrying out these missions. Unmanned Mars Sample Return (MSR) missions using current U.S. launch vehicles will probably also have payload masses large enough to require the use of aerocapture [3–5]. Unmanned Mars Rover missions, though, using new launch vehicles such as the Titan IV/Centaur G-Prime, could probably be carried out in a single launch using propulsive orbit insertion at Mars [2].

II. Aerocapture System Overview

Orbit insertion using aerocapture consists of basically three phases. The first is the approach phase, in which the spacecraft is guided along a predetermined hyperbolic orbit toward its entry point into the target planet's atmosphere. Next is the atmospheric flight phase, during which the vehicle uses the target planet's atmosphere to reduce its velocity and modify its trajectory. The third and final phase is atmospheric exit, which occurs when the vehicle has depleted sufficient energy to achieve the desired closed orbit upon exiting the atmosphere [6]. The overall flight profile is depicted in Fig. 1.

There are actually two navigation/guidance systems needed for aerocapture. The first system is needed in the approach phase, when the spacecraft has not yet reached the atmosphere. In this phase, navigation may be accomplished using either the Deep Space Network (DSN) radio navigation system, some type of onboard system, or possibly a combination of both. Guidance during the approach phase must be accomplished using propulsive maneuvers to control the trajectory. During the atmospheric flight phase, navigation and guidance must be performed by an onboard system designed especially for an aerodynamic vehicle due to the speed at which these functions must be performed. The aerocapture onboard navigation/guidance system gets its initial conditions from the approach navigation system, which must deliver the vehicle to the desired atmospheric entry point with sufficient accuracy to ensure that the guidance system will be able to execute a successful atmospheric passage and achieve insertion into the desired orbit.

Atmospheric flight is accomplished using a maneuverable entry vehicle, or aeroshell, which contains the spacecraft and a thermal-protection system. While in the atmosphere, the onboard guidance system uses an inertial or aided-inertial navigation system to determine the trajectory and executes course corrections by controlling the lift and drag generated by the aeroshell. It also monitors the accumulated energy loss experienced by the vehicle due to drag and, once the accumulated loss has reached some predetermined level, it initiates a pull-out maneuver in which the aerocapture vehicle generates the maximum possible lift to quickly exit the atmosphere. Once out, the vehicle will be in an elliptical orbit about the target planet. As mentioned earlier, the spacecraft may execute multiple passes through the atmosphere to further reduce its orbital energy. Shortly after leaving the atmosphere for the final time, the spacecraft sheds the aeroshell and reorients itself. Some time later, usually at the apoapsis (peak altitude) of its interim elliptical orbit, it will execute a propulsive maneuver to raise the periapsis altitude above the atmosphere.

The protective aeroshell serves two purposes. First, it is a heat shield which protects the spacecraft from aerodynamic heating during the atmospheric passage. Secondly, the aeroshell is a maneuverable lifting body which gives the vehicle the capability to control its flight path. Figure 2 shows several shapes that are being considered for aerocapture and aeromaneuvering vehicles, using the two parameters that most commonly describe the aerodynamic characteristics of these vehicles: lift-to-drag ratio (L/D), and hypersonic ballistic coefficient [5]. In Eqs. (1) and (2), the ballistic coefficient is defined as

$$\beta = M/C_D A \quad (1)$$

where

$$\beta \triangleq \text{hypersonic ballistic coefficient}$$

$$M = \text{total vehicle mass}$$

$$C_D \triangleq \text{drag coefficient, } D/qA \quad (2)$$

and

D = aerodynamic drag force

$q \triangleq$ dynamic pressure, $1/2\rho V^2$

ρ = atmospheric density (at current altitude)

V = vehicle speed relative to surrounding atmosphere

A = vehicle reference surface area

For the purpose of conducting preliminary evaluations of aerocapture vehicle aerodynamic characteristics, the ballistic coefficient is generally considered to be a constant, although it actually changes with the drag coefficient. This means it can vary with the vehicle's speed and angle of attack (the angle between the vehicle's longitudinal axis and its velocity in the surrounding atmosphere). The drag coefficient of a given shape characterizes its ability to generate drag, while the ballistic coefficient indexes this ability against the vehicle's mass. As seen in Fig. 2, larger ballistic coefficients are generally associated with streamlined, low-drag shapes that begin to resemble high-speed aircraft, while vehicles with more blunt shapes, such as the space capsules used in manned programs, have smaller values of β.

Lift-to-drag ratio is most easily expressed as the ratio of lift and drag coefficients

$$L/D = C_L/C_D \quad (3)$$

where

$$C_L \triangleq \text{lift coefficient, } L/qA$$

In Eq. (3), the dynamic pressure q, and reference surface area A, have the same meaning as in Eq. (2). An aerocapture vehicle's L/D (which for design purposes is usually quoted as a single number that is either a maximum value or the value achieved in some nominal flight scenario, even though it also varies with flight conditions) is an indicator of its overall maneuverability, which translates into flight path control capability. Aerodynamic vehicles such as aircraft maneuver primarily by controlling the magnitude and direction of lift. Vehicles that can achieve large L/D ratios are generally capable of greater flight path control than those that only achieve smaller values of L/D. Together, the ballistic coefficient and the reference (nominal or maximum) L/D ratio characterize the ability of an aerocapture vehicle to reduce its speed and follow a desired flight path.

Studies of aerocapture vehicle requirements for Mars missions have explored vehicles with nominal L/D ratios ranging from 0.6 up to 2.0 and ballistic coefficients of 120 to 800 kg/m^2 [4–7]. Figure 3 depicts a hypothetical Mars Sample Return aerocapture vehicle (aeroshell/spacecraft) with a mass of about 5000 kg, L/D of 1.5, and a ballistic coefficient of about 550 kg/m^2 [5]. Although they possess greater maneuverability than more blunt shapes, vehicles with L/D of 1.5–2.0 encounter problems with the packaging of the spacecraft within the aeroshell. As seen in Fig. 3, the narrow, streamlined shape needed to achieve an L/D of 1.5 makes it very difficult to utilize the aeroshell's available volume efficiently.

Aerocapture at Mars entails greater risk than at planets with denser atmospheres, such as Earth and Venus. Mars has a very thin, tenuous atmosphere: surface atmospheric pressure is about 5–10 mbar, in contrast to Earth's mean sea level pressure of 1000 mbar [8]. The flight path within the martian atmosphere must, by necessity, approach the planet's surface, possibly to altitudes below 20 km, depending upon the aerocapture vehicle configuration and guidance algorithm employed. While vehicles with larger L/D ratios are more maneuverable than those with smaller L/D values, they must penetrate deeper into the atmosphere to develop sufficient drag and to take advantage of their maneuver capabilities. At Mars, this consideration becomes especially important, since the planet has many natural features which rise from 10 to 26 km above the surrounding terrain [9]. In addition to the problem of terrain clearance, larger L/D aerocapture vehicles must, in general, withstand relatively greater structural loads and aerodynamic heating levels than smaller L/D vehicles, which do not need to penetrate as deeply into Mars' atmosphere.

Figure 4 shows an approximate indication of the payload mass fraction (payload mass/total vehicle mass) achieved for Mars orbit insertion with representative aerocapture and propulsive systems. The data shown in Fig. 4 were developed by Hoffman [10]. He computed these estimates for aerocapture using approach trajectory parameters representing an "average" Mars approach orbit and an aerocapture vehicle configuration similar to that shown in Fig. 3. The propulsive system model used for comparison was representative of the performance of rocket engines carried by previous interplanetary space probes.

The model assumed a liquid-fuel engine that used hypergolic propellants, producing a specific impulse of about 290 sec. As shown in Fig. 4, the payload mass increase obtainable with aerocapture varies significantly with the size of the desired final orbit. Even though Hoffman obtained his results for a single Mars approach trajectory, he noted that both aerocapture and propulsive orbit insertion performance also vary over different approach trajectories. Orbit insertion performance comparisons similar to Fig. 4 were also developed for representative missions to Venus, Earth, Saturn, Titan, Uranus, and Neptune [10]. The results indicated that the performance increases realized with aerocapture, if any, varied significantly, but that orbit insertion using aerocapture could deliver larger payloads than propulsive systems in most of these cases.

III. Aerocapture Navigation/Guidance System Design Issues

As with many other aerospace systems, the design of a Mars aerocapture system involves a number of trade-offs balancing the requirements and needs of various subsystems against one another. Performance characteristics of the aerocapture vehicle and the approach trajectory chosen for atmospheric entry at Mars can have significant impact on the navigation/guidance system design. Navigation and guidance capability are, in turn, important factors influencing the design of the Mars approach trajectory and aerocapture vehicle. In addition to cost and performance considerations, the reliability of competing system designs needs to be addressed in the early stages of conceptual design, not after this stage has been completed.

Aerocapture navigation and guidance will be accomplished using three major elements—the Mars approach navigation system, the aerocapture onboard navigation/guidance system, and the aerocapture vehicle itself. Table 1 presents a simple layout of the trade space between these three elements and three important design parameters: the asymptotic approach velocity, V_∞, and two of the primary vehicle parameters, L/D and β. The table also lists the relative magnitudes of the most desirable design parameters for each system element. The asymptotic approach velocity, V_∞, is the magnitude of the spacecraft's velocity relative to Mars as it begins its approach to the planet from a great distance away. It largely determines what the spacecraft's velocity will be when it enters the atmosphere. The values of L/D and β, discussed in the previous section, characterize the maneuver capability of the aerocapture vehicle.

Table 1 illustrates the primary trade-off that must be addressed for aerocapture systems: the conflict between approach and aerocapture navigation/guidance system requirements and aerocapture vehicle requirements. A spacecraft with relatively large values of L/D and β is capable of tolerating larger navigational errors during the approach phase and during atmospheric flight, and will also deliver the vehicle into the desired closed orbit more accurately than a vehicle with a relatively smaller L/D. Also, the density profile of Mars' atmosphere is not very well known, and an aerocapture vehicle with a large navigation/guidance safety margin would be more able to deal with the large uncertainties in the dynamic environment which may result. On the other hand, these vehicle characteristics, while desirable for navigation and guidance, lead to a design for a spacecraft that is very costly and difficult to build. The problems of spacecraft packaging within the aeroshell, structural design, and heat-shield design all become progressively more difficult as L/D is increased. In addition, spacecraft subsystem reliability levels generally become more difficult to maintain as vehicle performance and complexity are increased. Thus, spacecraft cost and reliability considerations point towards vehicle designs that possess the minimum L/D needed to perform aerocapture.

Navigation of the spacecraft during its approach to Mars is perhaps the most critical task which must be performed for aerocapture. The spacecraft's flight path at atmospheric entry must be controlled to a tolerance level that allows the aerocapture guidance system to correct for navigational errors. The most critical approach targeting parameter is the initial flight-path angle at the point of entry into Mars' atmosphere. If the entry angle is too shallow, the spacecraft may "skip out" of the atmosphere without achieving a closed orbit upon exit. If entry is at too steep an angle, the guidance system may be unable to prevent the vehicle from crashing [11]. In addition to the obvious risk of impact, an unexpectedly large entry angle may impose structural loads and heating rates on the aerocapture vehicle that it cannot withstand.

The navigation of deep-space probes in the past has been done primarily using Earth-based radio tracking data from the DSN. Modern DSN radio metric measurements include Doppler, range and Delta Very Long Baseline Interferometry (ΔVLBI). In planetary approach navigation, the accuracy of the DSN radio navigation system is limited primarily by uncertainty in the position and velocity of the target planet. Since the mid-1970s, optical navigation measurements made with spacecraft onboard imaging systems have been brought into use because they provide direct observations of target-relative position. The exploration of Mars with unmanned spacecraft may comprise a series of missions, which may allow radio metric measurements to be made between an approaching spacecraft and an earlier spacecraft that is either in orbit or on the surface of the planet. These types of measurements also provide direct target-relative information. Proposals have been made for placing a radio beacon, or even an entire network of radio beacons, on Mars as navigation aids for spacecraft and for performing science experiments [12].

Table 2 contains a tentative assessment of the navigational performance and risk of failure for five approach navigation system options which are presently being weighed against one another. This table is considered tentative because it is based on the author's interpretation of the work that has been performed to date in this area [7,13]. The author's conclusion is that, from a reliability standpoint, Mars approach navigation should be carried out using Earth-based DSN radio metric data for as much of the approach as possible. Deep Space Network-only navigation is the most proven and reliable option shown in Table 2. It is not yet clear whether the DSN navigation system will be capable of delivering the performance needed without any additional measurement sources. However, the DSN navigation system will undoubtedly play a significant role during the approach phase, whether its purpose is to perform the entire approach navigation function or merely to initialize an onboard navigation system which would guide the vehicle during final approach to atmospheric entry.

Evaluation of the other options becomes even more entangled with spacecraft design considerations. All of the system options shown in Table 2, which make use of spacecraft onboard measurements, are capable of meeting preaerocapture navigation requirements for the aerocapture vehicles considered to date. Each one of these options, though, may impose added requirements on the spacecraft in terms of additional hardware and/or software. The accompanying increase in system complexity implies an increase in the risk of failure. Many of the planned Mars exploration spacecraft for which aerocapture is being considered do not currently need imaging systems. If it were later determined that onboard optical navigation measurements were needed, imaging equipment—with its associated mass, power, and cost requirements—would have to be added solely for navigation purposes. Using onboard ranging and Doppler measurements to a Mars orbiter or surface beacon, if deemed essential, implies that the lead spacecraft in Mars exploration would not be able to use aerocapture for Mars orbit insertion, since it would have no "pathfinder" spacecraft with which to make navigational measurements.

Onboard optical, Doppler, and range measurements can either be transmitted back to Earth for ground processing or processed onboard the spacecraft. An onboard data processing capability places added computing hardware and software requirements on the spacecraft, but provides an advantage in that the spacecraft can quickly process radio and optical measurements for navigation. An onboard system takes much less time than transmitting the data back to Earth, processing it using ground-based software, and transmitting the resulting navigational information back to the spacecraft. This is an important consideration, since optical and radio measurements may need to be taken when the spacecraft is only a few hours away from atmospheric entry in order to attain the needed approach navigational accuracies. Even if optical data were to be processed on the ground, the number of optical frames needed to meet navigational accuracy requirements may make it necessary to implement an onboard optical image compression capability to reduce data volume and meet telemetry constraints.

The trade space shown in Table 1 also contains some implications regarding the design of the Mars approach trajectory. It is desirable to make V_∞ as small as possible for several reasons. Larger values of V_∞ mean greater structural loads and aerodynamic heating levels for the aerocapture vehicle. In addition, the determination of the flight-path angle at atmospheric entry becomes progressively more difficult as V_∞ increases. This fact can be illustrated in a fairly straightforward manner. The entry flight-path angle is a function of the approach trajectory parameters, the entry altitude, and the mass of Mars [6]

$$\cos\gamma_\epsilon \;=\; (B/r_\epsilon)[1 + 2\mu/(r_\epsilon V_\infty^2)]^{-1/2} \tag{4}$$

where

$B \triangleq$ impact parameter

$r_\epsilon =$ distance from center of Mars to atmospheric entry point

$\mu \triangleq$ gravitational parameter of Mars, GM

$G =$ Universal Gravitational Constant

$M =$ mass of Mars

The impact parameter, B, is commonly used by orbit determination analysts to describe the targeting coordinates of planetary flyby trajectories. A diagram of the B-plane coordinate system is shown in Fig. 5. The B-plane is perpendicular to the asymptote of the hyperbolic approach trajectory. The origin of the B-plane coordinate system is the center of mass of the target planet. The B-vector, whose magnitude is B, lies within the B-plane and points from the center of the target planet to the intersection of the approach asymptote and the B-plane. As shown in Fig. 5, the B-plane coordinate system is defined by the orthogonal axes S, R, and T. The S-axis is parallel to the approach asymptote, while T lies in the ecliptic plane, the plane defined by the Earth's orbit; R completes the triad. Orbit determination accuracy for approach trajectories is typically expressed in terms of the one-sigma uncertainty in knowledge of B and the individual components of the B-vector.

Equation (4) can be used to obtain an approximate expression for the uncertainty in B as a function of the uncertainty in flight-path angle

$$\sigma_B \approx (\partial B/\partial \gamma_\epsilon)\sigma_{\gamma_\epsilon} = r_\epsilon \sin\gamma_\epsilon[1 + 2\mu/(r_\epsilon V_\infty^2)]^{1/2}\sigma_{\gamma_\epsilon} \tag{5}$$

Previous studies have indicated that an aerocapture vehicle with L/D of about 1.5 would require an entry flight-path angle error of about one degree or less [3–6]. Recent studies of a simpler aerocapture vehicle configuration, with an L/D of 0.6–0.7, have indicated that this vehicle would require a flight-path angle error of 0.5 deg or less [7]. Equation (5) is plotted in Fig. 6, showing the approximate accuracy needed to meet the flight-path angle requirement for L/Ds of 0.7 and 1.5, as a function of V_∞ values spanning the range of likely Mars approach trajectories. Note that the required accuracy can vary by almost a factor of two over the V_∞ range shown.

Computer simulations of Mars approach navigation using DSN radio metric data only, performed for a hypothetical MSR mission launched in 1999, have indicated that the navigational accuracy seen in Fig. 6 could only be met with current DSN capability for the L/D = 1.5 vehicle, and only for V_∞ values of 3.0 km/sec or less. With projected improvements in DSN tracking capability over the next ten years, the approach navigation requirements for a vehicle with L/D of 1.0–1.5 could probably be met by DSN radio navigation over most of the V_∞ range shown in Fig. 6 [13]. The question of whether or not future DSN improvements would enable DSN-only navigation for a vehicle with an L/D of 0.7 is still subject to debate.

The remaining major aerocapture system element is the onboard navigation/guidance system for the atmospheric flight phase. This system will be some type of inertial or aided-inertial guidance system containing gyroscopes and accelerometers and, possibly, other external sensors. The inertial instrument technology needed for aerocapture has already seen service in a variety of operational aerospace vehicles, both military and civilian. The most recent example of a space vehicle which has successfully used an aided-inertial guidance system for controlled atmospheric flight is the space shuttle [14,15]. Several navigation and guidance algorithms have been developed which appear to be capable of flying aerocapture vehicles with L/D values in the 0.6–1.5 range [7,16,17]. Computer simulations of aerocapture guidance have been conducted which established the navigation-error envelope that can be tolerated by some, but not all, of these algorithms [7]. At the present time, there do not appear to be any insurmountable technical problems preventing the design and development of a suitable onboard navigation/guidance system, but development work to date is still only preliminary. While a sizable body of theory exists in the area of aerocapture navigation/guidance algorithms, very little of this work has been subjected to any kind of thorough and systematic evaluation. There is still a great deal of work that needs to be done to develop a set of candidate onboard guidance system designs which are well matched to the range of aerocapture vehicle and approach navigation system options.

IV. Aerocapture Navigation/Guidance System Reliability Issues

The high cost of proposed Mars Rover and Mars Sample Return missions makes it very likely that flight testing of the aerocapture system will not be performed, since test flights often consume a significant fraction of the total resources allocated to complete these missions. Aerocapture systems are a good example of what Rechtin [18] has dubbed "ultraquality systems," which are very complex and must be highly reliable, yet because of cost constraints or impracticality, cannot be rigorously tested to establish their failure rates to a high degree of confidence. Put another way, how does one establish that a Mars aerocapture navigation/guidance system does indeed have a failure rate of 1 in 10,000 if only five missions are planned?

Several methods have been developed by the aerospace industry to design systems that must meet demanding reliability requirements. Some of the most successful for engineering interplanetary space probes are progressive redesign, failure modes and effects analysis (FMEA), and fault-tolerance system design methods. What follows is a discussion of the role these and other methods might play in developing highly reliable systems for these missions.

Progressive redesign is a cyclical process of identifying faults or problems, finding their sources and correcting them, observing the effects of the fixes, and then repeating the identify-fix-observe process over and over again. An important element of this cycle is the detailed documentation of the faults uncovered, the fixes applied to them, and the results stemming from the modifications made. For systems that are built in large enough numbers (e.g., automobiles and airplanes) and given enough time, the reliability achieved through progressive redesign usually increases gradually, to the point where it begins approaching some peak success rate asymptotically.

Some good examples of progressive redesign in aerospace systems are expendable launch vehicles and unmanned space probes such as the Mariner, Viking and Voyager series of missions. Over the past 30 years, launch vehicle success rates have risen from about 0.5 (1 failure in 2 launches) to a current level of about 0.94 (6 failures in 100 launches)[18]. The two Voyager spacecraft, launched in 1977, are good examples of the benefits of cumulative redesign efforts in interplanetary space vehicles. They are both still in operation after 13 years in space and a total of six planetary encounters. These two spacecraft can trace their design lineage back through the seven missions of the Mariner program and the two orbiters built for the Viking mission.

While there is some commonality between Mars aerocapture navigation/guidance and similar problems, such as space shuttle reentry guidance, there are substantial differences as well. Many of these are due to the differences between the atmospheres of Earth and Mars. In addition, the unmanned Mars precursor missions that might use aerocapture will probably be very few in number. Aerocapture vehicles will be built using some subsystems that may have benefitted from the use of progressive redesign in previous spacecraft, but it will probably not be an acceptable method for increasing the overall reliability of aerocapture systems.

Failure modes and effects analysis is a technique which has been used extensively by NASA and the Department of Defense in spacecraft design [18]. It involves detailed, exhaustive analysis of all possible system failures and the effects of these failures. It is typically employed in an iterative manner, in which an FMEA is constructed, failure modes that are subsequently identified are eliminated through design changes, and a new FMEA is constructed. This process is repeated until a sound design has been achieved. Table 2 is an example of one aspect of FMEA, establishing the relative failure risk of system elements. Although the estimation of risk is often judgmental, a performance-risk matrix such as that in Table 2 can be a useful aid in making critical decisions during the early stages of system design. At JPL, the use of FMEA can be traced back to the Ranger missions from 1960 to 1964, and it has been employed in the design of every spacecraft built at JPL since then. It is a method well suited for aerocapture systems, and will likely be an important part of any spacecraft/aerocapture system design effort.

Fault tolerance, in the form of redundancy and the management of redundant components, is a method commonly used to guard against catastrophic failures. Redundancy, in particular, has been frequently employed in critical spacecraft subsystems to eliminate single-point failures. It is generally most effective when the redundant system elements are independent; that is, they accomplish the same task, but in a different way. Redundancy may be of little value in preventing single-point failures if the redundant elements backing up some primary system element are identical to the primary element, including the primary element's flaws. For aerocapture, an approach navigation system employing independent redundancy could be a combination of the DSN radio navigation system and some type of onboard system. For reasons of cost, it may be desirable to perform aerocapture with a relatively simple vehicle ($L/D < 1.0$), instead of a more complex vehicle which is capable of better performance ($L/D \approx 1.1 - 1.5$). In this situation, an onboard optical or radio approach navigation system may be designated as the primary system for the final approach navigation, since onboard systems are generally capable of greater accuracy than Earth-based systems. It may be wise to choose a vehicle design that also allows using the DSN radio navigation system as a backup (even if the DSN is only marginally capable of meeting the requirements) in the event of a failure or other problem with the onboard system.

While the reliability of hardware is of great importance in any spacecraft design, aerocapture vehicles will also include extensive software elements. The onboard navigation/guidance software, in particular, is a critical part of an onboard aerocapture guidance system. If it does not work properly, the aerocapture attempt will probably fail. Monte Carlo analysis is one method that can be used to study guidance software performance over a wide variety of aerocapture trajectories. Although it has not been used extensively in the past, Monte Carlo analysis has been used to evaluate the performance and reliability of some spacecraft navigation and guidance systems. It is carried out by running hundreds, often thousands, of computer simulations of the navigation/guidance system controlling a simulated vehicle in a simulated environment. Within the simulation environment, the navigation/guidance system software can be fed flight scenarios representing extreme conditions to rigorously establish the system's performance envelope.

Monte Carlo analysis could be used to evaluate the onboard navigation/guidance system using a "hardware-in-the-loop" simulation. The actual navigation/guidance system hardware and software, with the exception of inertial instruments, can be connected to a computer to provide a realistic simulation of the aerocapture vehicle flying in the Martian atmosphere. The computer supplies the onboard guidance system software with signals that simulate the outputs of its gyroscopes, accelerometers, and other sensors, generated by a sophisticated mathematical model of these instruments. The navigation algorithm then uses these input signals to compute the current trajectory of the vehicle, just as it would in actual flight. The guidance algorithm uses the navigation data to compute guidance commands to control the vehicle, which are sent back to the computer. The computer uses its model of the aerocapture vehicle dynamics to simulate the vehicle's response to the guidance commands. The computer's dynamic and instrument models generate new instrument readings that are sent to the flight computer, beginning a new guidance cycle. Using Monte Carlo analysis, the onboard navigation/guidance system can be made to "fly" literally thousands of aerocapture orbit insertions, all within the confines of a laboratory.

Although it is a powerful system evaluation tool, Monte Carlo analysis has some limitations in what it can and cannot predict. By its nature, it is based on assumed models for the physical processes being simulated. If the models are substantially incorrect, or the statistical assumptions regarding random processes that may be present in the system are incorrect, then the reliability predictions obtained are not valid. The obvious conclusion is that Monte Carlo analysis works best for systems that are well modeled and fairly well understood. Fortunately, there exists a sizable experience base in the modeling of spacecraft dynamics and, to a lesser extent, the dynamics of high-speed aerodynamic vehicles.

V. Summary and Conclusions

Aerocapture is a concept for performing orbit insertion about a target planet in which a spacecraft, contained within a protective aeroshell which can maneuver aerodynamically, reduces its velocity by passing through the atmosphere of the target planet, becoming "captured" into a closed orbit. Aerocapture navigation and guidance must be viewed in the context of an overall aerocapture system, whose other major elements are the DSN radio navigation system, the aerocapture vehicle, and the onboard navigation/guidance system. The benefits of orbit insertion using aerocapture versus rocket propulsion are dependent upon the Mars approach trajectory and the size of the final orbit which will be assumed by the spacecraft. System design trade-offs between approach navigation accuracy, spacecraft cost, and aerocapture vehicle configuration were addressed. In addition, some ways of developing highly reliable aerocapture system designs were discussed.

Mars preaerocapture approach navigation is an area of some uncertainty. The work to date on this subject is of a preliminary nature and has not yet resolved some important questions, such as the capability of the DSN to meet the approach navigation requirements of smaller L/D aerocapture vehicles. Since approach navigation requirements are a function of aerocapture guidance system and vehicle characteristics, the design of the approach navigation system, guidance system, and aerocapture vehicle must be an iterative process, in which the needs of each are carefully evaluated in terms of the cost, performance, and reliability of the complete aerocapture system.

Aerocapture onboard navigation and guidance technology is another area in which some important questions remain, such as the ability of different guidance algorithms to function in the presence of large flight environment uncertainties. At the present time, it appears that there are no insurmountable technological barriers in the way of developing guidance systems for unmanned Mars exploration spacecraft, but there is still a great of deal of work to be done in order to bring that technology to a level of maturity that can support the design of guidance systems for actual missions. Although reliability is a significant challenge for Mars aerocapture missions, it also does not appear to be an insurmountable obstacle. Design and testing practices such as independent redundancy, exhaustive computer simulation (Monte Carlo analysis), and FMEA can be used to develop aerocapture systems that meet the reliability levels which will be demanded of these missions. Most of these methods have been used in the past in the development of similar systems, such as the reentry guidance system for the space shuttle.

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Table 1. Aerocapture system design trade space

Parameter	Mars approach navigation system	Aerocapture onboard navigation/guidance system	Aerocapture vehicle
	Element		
V_∞	Small	—	Small
L/D	Large	Large	Small
β	Large	Large	Small

Table 2. Tentative approach navigation options–performance matrix

Approach system option	Relative performance	Relative risk of failure	Remarks
Earth-based DSN radiometric data only	Marginal[a]	Very low	Mature, reliable
Earth-based radiometric data and onboard optical data (ground processing)	Good	Low	May require onboard optical data compression
Onboard optical data (onboard processing)	Good	Moderate	Need onboard computer assets
Earth-based radiometric data and onboard radiometric data with Mars orbiter/beacon (ground processing)	Excellent	Low	Best possible accuracy; need orbiter/beacon though
Onboard radiometric data with Mars orbiter/beacon (onboard processing)	Excellent	Moderate	Need onboard computer assets

[a] Based on current DSN capability; may improve to "good" during the time frame proposed for Mars precursor missions (2000–2005).

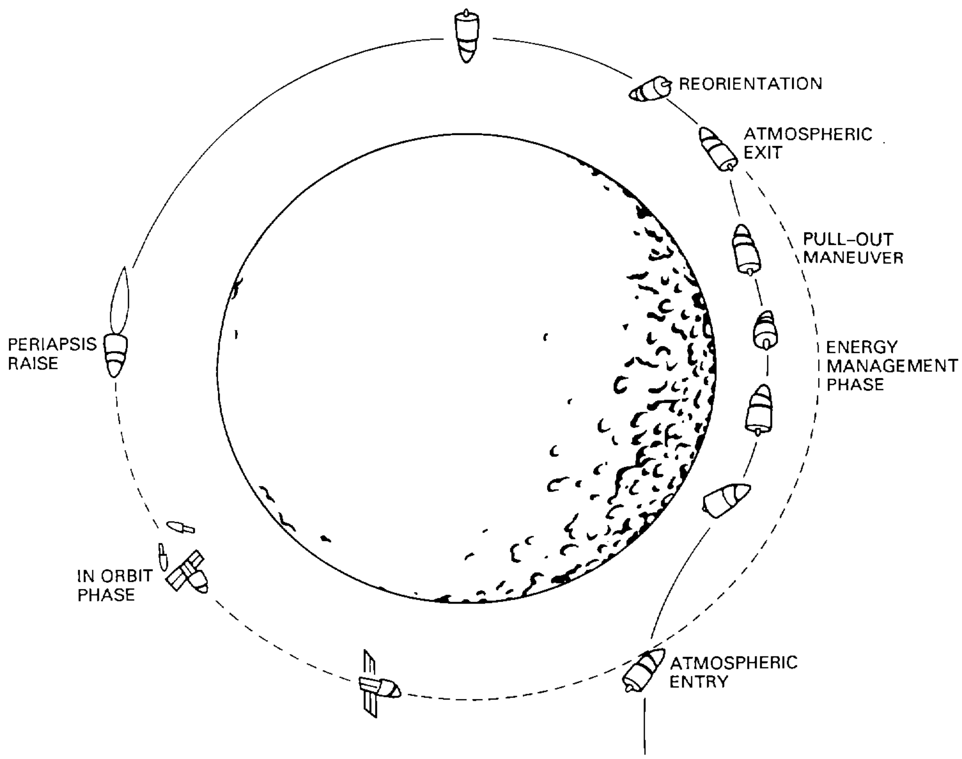

Fig. 1. Aerocapture mission flight profile.

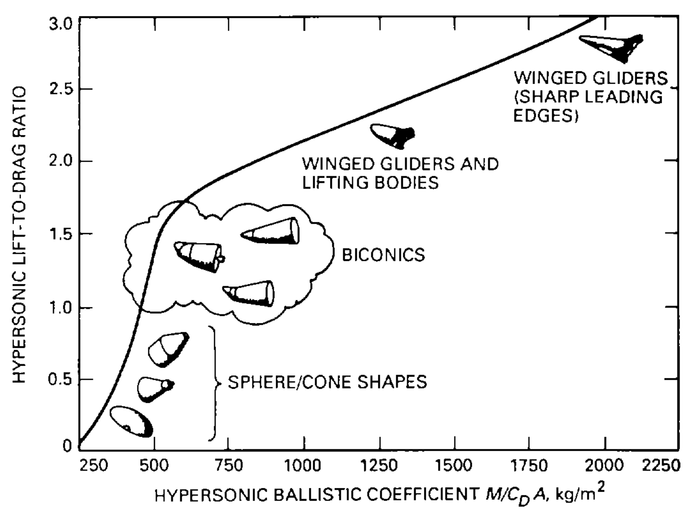

Fig. 2. Aeroshell shapes for aerocapture vehicles.

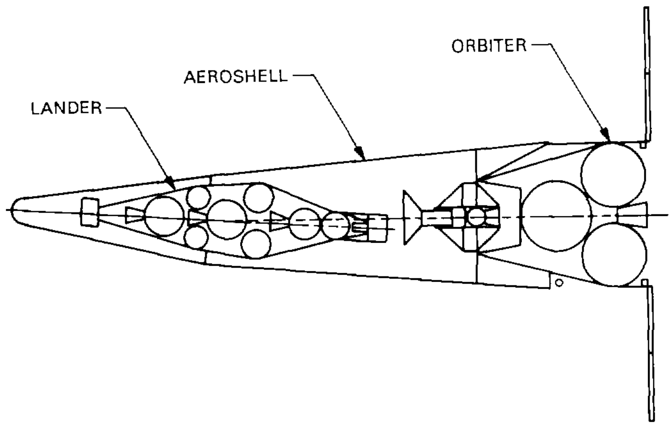

Fig. 3. Side view of a proposed spacecraft configuration for a Mars Sample Return Mission.

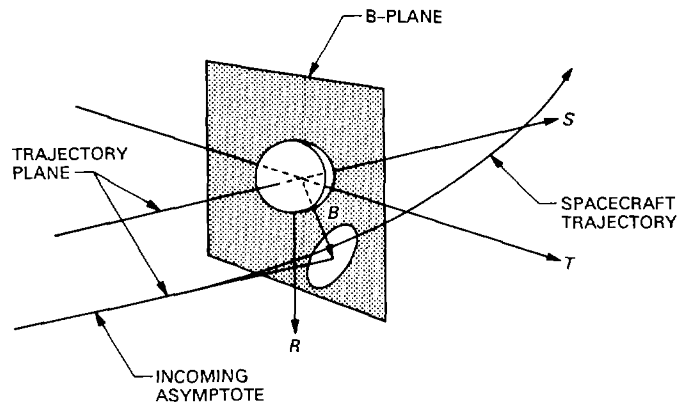

Fig. 5. The B-plane target body–centered coordinate system.

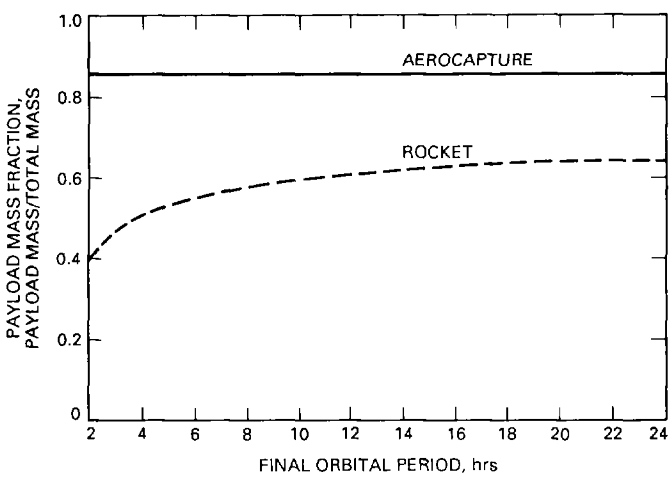

Fig. 4. Payload mass fraction for aerocapture and propulsive orbit insertion at Mars.

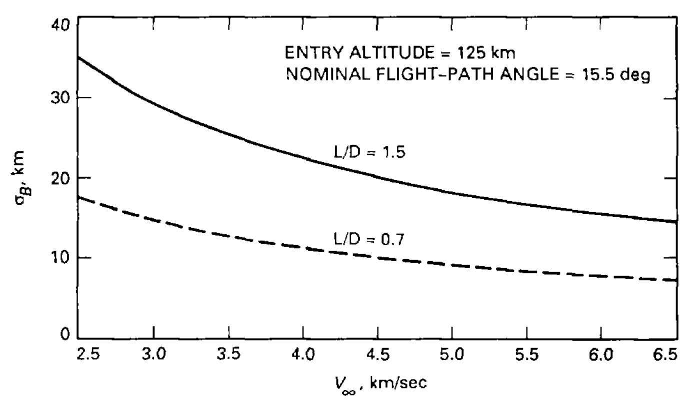

Fig. 6. Required approach navigation accuracy versus V_∞.