Precursor Orbital Guidance

W. C. Hailey* and P. Mosner†
Avco Corporation, Wilmington, Mass.
W. E. Vander Velde‡
Massachusetts Institute of Technology, Cambridge, Mass.

A technique is presented which may facilitate the employment of atmospheric braking to attain an orbit about a planet as the terminal phase of an interplanetary trajectory. The proposed system comprises a set of small aerodynamic precursor vehicles, scaled to the same M/C_DA as the mothercraft, which will be launched at various angles in the planetary approach trajectory plane to acquire the atmospheric data needed by the mothercraft. Data will be telemetered from each precursor, which will contain its own small rocket to allow a trajectory change. Results of a preliminary feasibility study show that the precursor-atmospheric braking technique could provide Martian-orbital payload factors of 4 to 5 compared to a straight retropropulsion system.

Introduction

THE establishment of spacecraft orbits around the near planets is an objective of high priority in our space exploration program. However, at the termination of the heliocentric transfer orbit the hyperbolic excess velocity with respect to the target planet is significant, and a substantial amount of retropropulsion would be required to reduce the energy to that of a low-altitude orbit most suitable for experimental purposes such as mapping, etc. The payload weight of the spacecraft in orbit is thus severely limited if the orbit is established purely through propulsive braking. An impressive gain in payload can be realized if the atmosphere of the planet can be used to dissipate some of the excess energy. The major difficulty in the attempt to use atmospheric braking is the uncertainty in the characteristics of the atmospheres of the near planets. One can readily show that for any reasonable estimate of the range of uncertainty in the density vs altitude characteristic for Mars or Venus, a vehicle making a pass through such an atmosphere would have very little probability of leaving the atmosphere with an energy suitable for efficient establishment of a prescribed orbit. Even if the vehicle were controllable during the atmospheric flight, the density uncertainty is so great as to yield only a moderate probability of successful mission accomplishment. Clearly what is needed to realize the advantage of atmospheric braking is timely information about the nature of the atmosphere that the spacecraft will encounter.

A solution to this problem is proffered by a precursor system, in which a number of very small entry vehicles, or precursors, are deployed by the spacecraft during its approach to the planet. Scaled to the same M/C_DA as the spacecraft, they pass through the atmosphere (impact the planet or skip out) prior to the spacecraft entry, with a controlled range of entry conditions. A communication link relays information derived from an accelerometer on each precursor back to the spacecraft, and from this record of the experience of the several precursors in the actual planetary atmosphere, a computer on board the spacecraft can determine an appropriate adjustment to the spacecraft velocity vector to give favorable initial conditions upon entry into the atmosphere shortly thereafter.

This paper describes such a precursor system in the context of its operational environment. Following a brief description of the role of aerodynamic braking in attainment of a planetocentric orbit, the precursor system is described in detail. The required initial condition information, the means for deployment, the type and amount of information relayed back to the spacecraft, and the means of processing this data into the spacecraft velocity correction are discussed. A preliminary estimate is given of the space and weight requirements and the propulsion requirement for the establishment of a prescribed orbit after the atmospheric pass. Results of an error analysis for the system are reported.

Aerodynamic Braking

The use of aerodynamic braking to attain an orbit about a planet was apparently first suggested by Hohmann[1] and later extended by others, notably Chapman.[2] The intent of this section is to discuss cursorily certain aspects of aerodynamic braking which relate to a precursor system. Two cases are considered; the lifting overshoot, and the undershoot (Fig. 1). In the overshoot case, the planetary-atmospheric entry angle is such that, with the lift vector directed toward the planet (negative lift), the vehicle experiences a sufficient velocity decrement to leave the atmosphere with a state

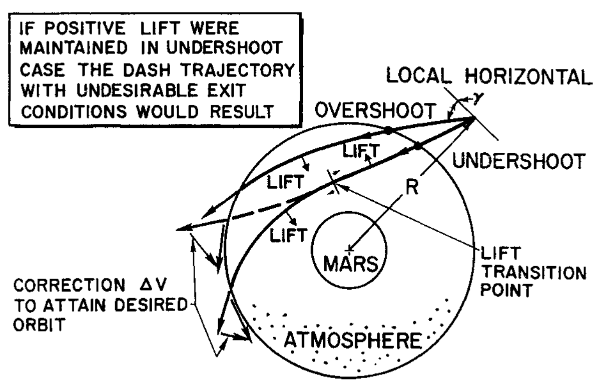

Fig. 1 Entry corridor flight geometry.

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* Project Engineer, Advanced Space Systems Directorate, Research and Advanced Development Division. Associate Member AIAA.

† Senior Staff Scientist, Guidance and Control Department, Research and Advanced Development Division.

‡ Consultant and Associate Professor of Aeronautics and Astronautics. Member AIAA.

Table 1 Martian atmosphere

Alt, kft	Density, slug/ft³	Alt, kft	Density slug/ft³
0	19×10^{-5}	440	61×10^{-9}
40	12×10^{-5}	480	49×10^{-9}
80	73×10^{-6}	520	41×10^{-9}
120	33×10^{-6}	560	34×10^{-9}
160	14×10^{-6}	600	28×10^{-9}
200	54×10^{-7}	640	22×10^{-9}
240	16×10^{-7}	680	16×10^{-9}
280	44×10^{-8}	720	10×10^{-9}
320	18×10^{-8}	760	73×10^{-10}
360	11×10^{-8}	800	53×10^{-10}
400	76×10^{-9}	...	...

which permits a transfer into the desired planetocentric orbit with a prescribed propulsive ΔV. In the undershoot case, the entry angle is sufficiently steep that the lifting vehicle just succeeds in leaving the atmosphere with a state which permits a subsequent transfer into the desired orbit with the prescribed propulsive ΔV. The lift vector is initially directed away (positive lift) from the planet to prevent impact; however, at a suitable point the direction of the lift vector is reversed to effect those atmospheric exiting conditions which will minimize the required correction ΔV necessary to attain the desired vacuum orbit. The technique used to calculate these optimally controlled lifting trajectories is reported in Ref. 3.

The orbit entry corridor is defined as the difference between corresponding trajectory parameters for the extreme overshoot and undershoot cases, e.g., the difference in atmospheric entry angles or virtual periapses for given conditions. The atmospheric model used is presented in Table 1. Figure 2 defines the orbit entry corridors for Mars as a function of propulsion system ΔV capability for various values of M/C_DA, $L/D_{\max} = 0$ and 0.5, and an entry velocity of 25,000 fps at 800 kft alt. A circular orbit of 200-mile alt is the end condition. The increase in corridor width with available propulsion capability is impressive; however, it is essentially restricted to the undershoot region and reaches a limit slightly beyond a $\Delta V = 3000$ fps. Further increases in propulsion capability are ineffectual, because the vehicle lift capability can no longer effect atmospheric exit. By comparison, the ballistic ($L/D = 0$) corridor is quite small. In terms of virtual periapses it is approximately 25,000 ft wide; in terms of flight-path angle it is approximately 0.4° wide. It is interesting to note that the available corridor width for a drag-modulated vehicle may also be deduced from Fig. 2. If we consider a 20:1 modulation ratio ($0.5 < M/C_DA < 10$ slug/ft²), then the available corridor ranges from $-14°$ to $-16.5°$ for a width of approximately 175,000 ft or 2.5°.

Since there is an equivalence between M/C_DA and constant percentage density changes, Fig. 2 may also be considered as applicable to $M/C_DA = 1$ slug/ft² (constant) and the ordinate now represents density profiles that vary from nominal by a constant percentage: $M/C_DA = 5$ slug/ft² corresponds to 0.2 nominal density, etc. If flight were to continue to impact, then an interpretation of this nature perhaps would be unjustifiable. However, the altitude region for significant flight dynamics is high (150 to 250 kft), and this fact, coupled with the present conjectural knowledge of the Martian atmosphere, suggests that such interpretation is reasonable.

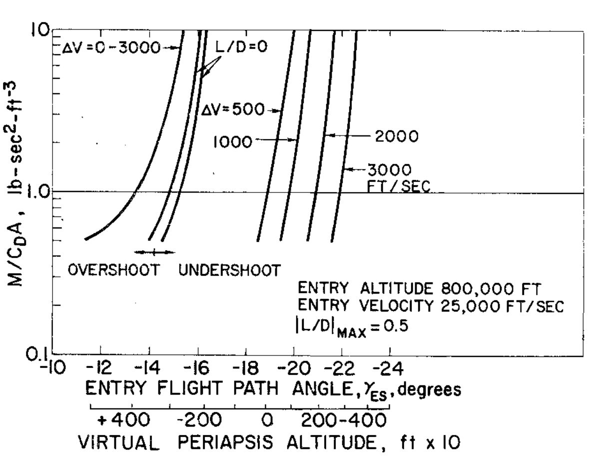

Fig. 2 Thrust-augmented aero orbit corridor: Mars.

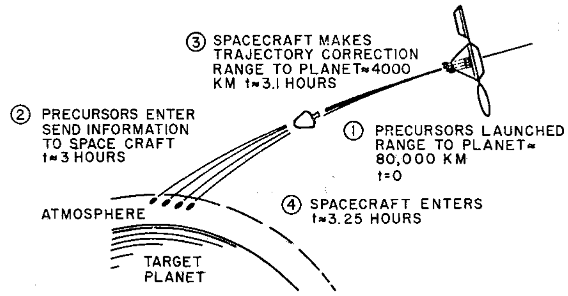

Fig. 3 Precursor concept.

Precursor System

The proposed system is depicted in Fig. 3. In sequential detail, operation is as follows. At a predetermined range from the target planet, the spacecraft is aligned to some suitable reference attitude. The precursor vehicles§ (Fig. 4) are spun up in launch fixtures to 150–250 rpm and are gently¶ ejected at various angles from the spacecraft in the trajectory plane, i.e., the plane defined by the planet and the spacecraft velocity vector. After a suitable time delay, a solid rocket attached to each precursor is fired to increase the precursor velocity. The precursors fan out in the trajectory plane, and their resultant flight paths are essentially perturbations of the mothercraft trajectory. The position and velocity of the spacecraft relative to the target planet at precursor launch are known. Knowledge of the ΔV applied to the precursors permits inference of their vacuum trajectories *relative* to the spacecraft trajectory. Since some error can exist in the knowledge of these quantities, the spacecraft continues its navigation function as it approaches the planet, in order to improve the definition of the state conditions at the time the precursors were launched. The velocity differentials possessed by the precursors relative to the spacecraft permit them to precede it to the planet, enter** (or miss) the plane-

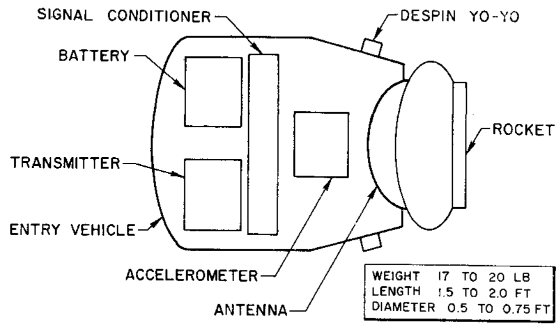

Fig. 4 Precursor vehicle.

§ The precursors are atmospheric-entry vehicles which possess the same aerodynamic similarity to the spacecraft as ballistic missile decoys do to the ballistic missiles they simulate.

¶ The purpose of gentle ejection is to minimize spacecraft attitude perturbations and tipoff errors.

** Shortly prior to atmospheric entry, a yo-yo despin system reduces precursor spin rate to 5 to 10 rpm to permit angle-of attack-convergence.

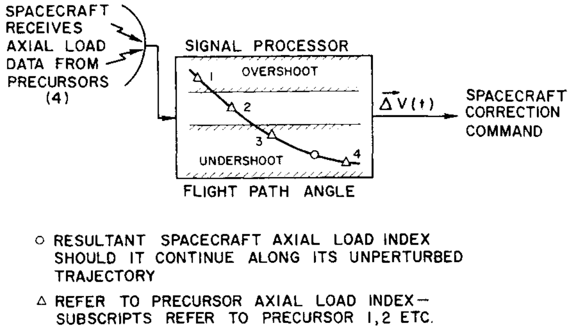

Fig. 5 Signal processor.

tary atmosphere, and pass through peak deceleration well before spacecraft atmospheric entry.

Each precursor is instrumented with a longitudinal accelerometer and a transmitter located such that transmission of certain characteristics of the sensed specific force is directed out of the base of the precursor to the spacecraft. Figure 5 is a symbolic representation of a possible data processing scheme. The quantities with subscripts refer to indices of the total velocity change which each precursor has experienced due to aerodynamic forces. (It will be recalled that each will have a different virtual periapsis or entry angle.) Those precursors lying in the cross-hatched undershoot region will land, because their velocity changes are too great for them to orbit or escape. The precursor in the overshoot region has not lost enough energy to be captured by the planet; therefore, it escapes. Precursor no. 2 has been captured by the planet and will orbit before landing. Through extrapolation or interpolation of these load indices and the knowledge of where the mothercraft load index would lie were it to continue on its unaltered course, the mothercraft, still several thousand miles away from atmospheric entry, can determine and effect that flight-path angle change most suitable for its atmospheric entry.

Launch Considerations

The major factor influencing precursor design for launch and vacuum flight is the required maintenance, with respect to an inertial reference, of the precursor attitude during its thrust phase such that the desired orientation of the velocity perturbation is achieved. No significant precursor attitude errors are expected to result from soft ejection of the spinning precursors from their launching tubes. More significant errors can be expected from thrust offset and principal axis misalignment of the precursor during the thrust phase. Reference 4 establishes a dynamic equivalence between principal axis misalignment and thrust offset which facilitates analysis of this problem. Judicious manipulation of the parameters involved, such as thrust level, burning time, spin rate, and roll moment of inertia, coupled with precision requirements on nozzle

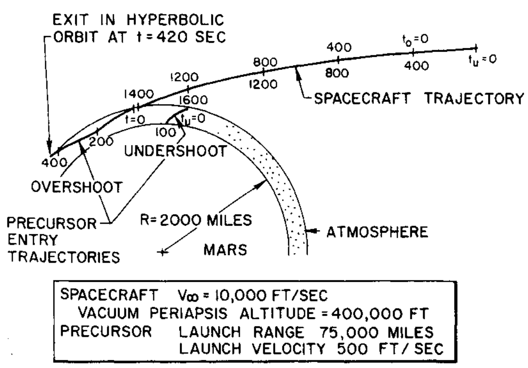

Fig. 6 Precursor: spacecraft terminal phase geometry.

alignment and vehicle design, may be expected to result in an accuracy of about 0.1° for the orientation of the vector ΔV achieved by the precursor.

Propulsion requirements were investigated to the extent that a high degree of confidence was established with respect to the attainment of precursor requirements. A preliminary rocket design obtained from Atlantic Research Corporation indicated that weight, envelope, and performance requirements are well within the realm of practicality.

Aerodynamics and Flight-Path Geometry

The aerodynamic criterion associated with the simulation of a space vehicle entering either Mars or Venus is the accurate prediction of the M/C_DA for both the space vehicle and the precursor. Factors which affect this performance parameter are:

1) Drag coefficient
   a) Mach number, compressibility effect
   b) Reynolds number, viscous effect
   c) Ablation, shape variation and also induced pressure drag
2) Mass variations due to ablation
3) Reference area variation due to ablation
4) Vehicle dynamics (increased drag due to angle of attack and also effect of lift introduced during angle-of-attack flight)

The factors affecting the drag coefficient are very closely coupled, but they can be handled by test and analytical techniques. Considerable experience has been achieved at Avco in terms of simulating re-entry vehicles by means of subscale flight testing and decoys. In these cases simulation is not restricted solely to trajectory history but also to vehicle observables. Experience indicates that the precursor aerodynamic design and the criteria for satisfactory simulation can be established from the presently available technology.

Figure 6 depicts, to scale, the flight paths of two precursors entering the Martian atmosphere. These precursors represent typical ballistic ($L/D = 0$) overshoot and undershoot extremes; although they were launched simultaneously, their trajectories are different and they enter at different times. The time at precursor entry, however, is taken as zero for both cases, and these time scales indicated on the mothercraft trajectory. The undershoot precursor reaches the planet approximately 400 sec prior to the overshoot precursor; the spaceship is at the position marked $t_u = 0$ when the undershoot precursor enters and is at the position marked $t_0 = 0$ when the overshoot precursor enters.

There has been no attempt to optimize these trajectories with respect to any of the numerous parameters involved, yet the geometry of communication, as well as the available flight time for the spacecraft to process data and effect a correction, are seen to be favorable. Should the vehicle that impacts experience a period of blackout, there remains ample time in low-speed flight prior to impact for the communication of data. For the conditions selected, the flight time remaining to the mothercraft, after signal reception from the last precursor, is at least 1000 sec and may be as high as 1300 sec or 15–20 min. This seems to be more than adequate time to process data, effect mothercraft trajectory correction, and rig for entry.

Data Acquisition and Communications

Typical deceleration curves for precursors entering the atmosphere of Mars are shown in Fig. 7. An accelerometer could sense this deceleration and relay this information to the mothercraft. However, to maintain simplicity, it appears desirable to limit the amount of information required from the precursor. The transmission of peak deceleration, the peak rate of change of deceleration, or the integral of deceleration for a given time may prove to be sufficient. The minimum amount of data required from the precursor remains to be determined; once this is known, exact design of the precursor instrumentation can be resolved. The blackout of transmission does not appear to be a problem at present; however, should this become a problem, the time of tranmission will have to be chosen to insure the data acquisition by the mothercraft. Hence the type of data available may be affected.

The instrumentation and transmission system for the precursor vehicle was examined to establish the weight required. A transmitting frequency of 1500 Mc was selected to optimize the over-all system gain. The transmitting antenna on the base of the precursor vehicle has been limited to 6 in. to minimize the precursor vehicle size and weight. An estimate of the size of the receiving antenna on the spacecraft was made; a 2-ft diameter appears quite feasible. A detailed analysis of this communication link indicates that a 1 w (input power) transmitter on the precursor driving the antenna, which has a 90° half-power beam width, will yield a satisfactory S/N ratio. The total weight estimate for transmitter, antenna, signal conditioner, and battery for 2-hr operation is 3 lb.

Information Processing

One can imagine the use of a single precursor, launched toward a steep entry condition in order to realize a sufficiently high probability for entry and traversal of all altitudes, from which complete deceleration and altitude-velocity histories could be derived. From these the density vs altitude curve for the planet at the local position and time could be constructed. This extreme, however, presents an imposing requirement for data sensing, communication, and processing. The other extreme would be the use of a large number of precursors and the recovery of just a single word of information, such as the integrated longitudinal specific force, from each. If precursors were deployed through a wide range, with a fine spacing between the entry conditions for adjacent vehicles, several of them would yield useful values of integrated specific force. By interpolation, the desired spacecraft entry condition could be determined. Note further that if the initial trajectory of the spacecraft is inaccurately known, this inaccuracy will influence the calculated entry conditions for both the precursors and the spacecraft in nearly the same way, so the first-order effect of inaccurate trajectory data is compensated.

The most favorable system will lie between these extremes. The work completed to date has been directed primarily toward finding parameters that are both convenient to measure and indicative in a simple way of the important effects of the atmosphere on the vehicle. A very desirable characteristic of such measured parameters is that they have quantitatively useful values for trajectories that impact, as well as for those which leave the atmosphere. One such parameter is the maximum deceleration experienced by the precursor. Its usefulness was indicated by a study of a simplified situation—the establishment of an orbit around Mars. Atmospheric uncertainties were embodied in four different density-altitude models, which differed by as much as a factor of 100 in density in the altitude range at which significant deceleration is experienced. One of these models was treated as the nominal or expected atmosphere, the others as perturbed actual atmospheres. From the values of maximum deceleration recovered from the precursors, the desired entry-flight-path angle, and thus the velocity correction, for the spacecraft was calculated using formulas based on influence coefficients linearized about the nominal situation. The resulting entry-flight-path angles for the spacecraft into the three perturbed atmospheres differed from the most desirable entry angle in each case by less than 0.4°, which is a small enough deviation to be accommodated by a modest amount of lifting control during the atmospheric pass. A total of nine precursors was required to span the range of atmospheric uncertainty in this example.

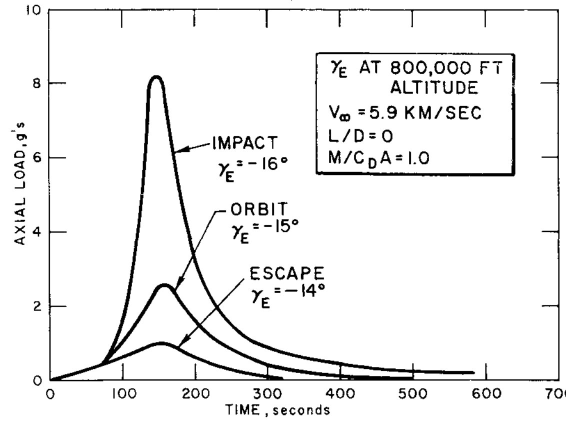

Fig. 7 Precursor deceleration histories for Martian atmosphere.

This test is only a first step. Additional information can be derived from each precursor to reduce the number of precursors required, and other parameters should be studied. However, this example demonstrates the significant information content of simple parameters such as maximum deceleration.

Error Analysis

A digital computer program[5] was written to carry out a complete planar (error) analysis to determine: 1) the influence coefficients, 2) desirable precursor launch and spacecraft positional fix conditions, and 3) uncertainties in the spacecraft flight-path angle relative to the precursor flight-path angle at a reference altitude approximating the top of the sensible atmosphere. The program is based on the parametric evaluation of partial derivatives (derived analytically) of a sequence of equations which are equivalent to a set which might be used for guidance. The primary outputs are the quantities in the equation determining the uncertainty of the spaceship flight-path angle $\Delta\gamma_{ES}$ relative to the precursor flight-path angle at the reference range. This equation is

$$(\Delta\gamma_{ES})^2 = \sum_i (\Delta G_i \partial\gamma_{ES}/\partial G_i)^2$$

where the various parameters represented by G_i and the corresponding uncertainty ΔG are as follows: R_M, V_{MS}, γ_{MS} are the spaceship range, velocity, and flight-path angle relative to the planet at the time of the last measurement used for retrocomputation of precursor launch parameters; ϕ is the central angle subtended at the planet by the portion of the spaceship trajectory from precursor launch to the last measurement range; V_{OP} is the velocity perturbation applied to a precursor at launch; θ_{OP} is the application angle (relative to the spaceship velocity vector) of the velocity perturbation applied to a precursor at launch; V_{CS} is the velocity correction added to the spaceship velocity vector at the correction point to change its flight-path angle; and θ_{CS} is the application angle (relative to the spaceship velocity vector) of the velocity correction applied to the spaceship at the correction point to change its flight-path angle.

In Fig. 8, the root sum square (RSS) value of the entry angle uncertainty is plotted as a function of the entry angle for a launch range of 50,000 miles and a last measurement range of 10,000 miles. In this case, the planetary approach velocity to Mars of the spacecraft is 20,000 fps, with a periapsis equal to the planet radius, and the precursors are launched with a velocity perturbation of 1000 fps relative to the spacecraft. The upper curve gives the RSS value of the uncertainty in the absolute value of the spacecraft entry angle. In the region of interest, this varies from 0.03 to 0.018 rad (1.7° to 1.0°). The lower shows the RSS value of the uncertainty of the spacecraft entry angle relative to the precursor

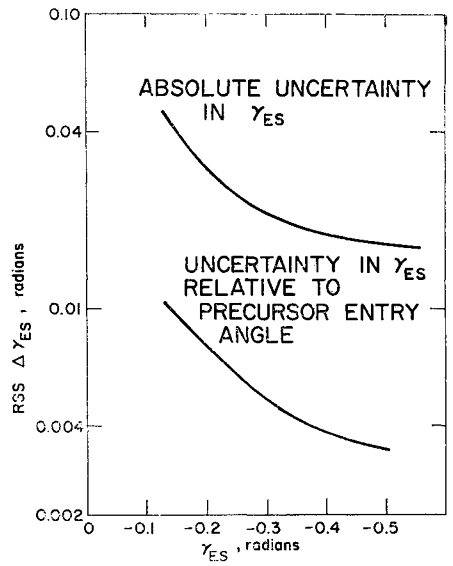

Fig. 8 Uncertainty in spacecraft entry angle, γ_{ES}.

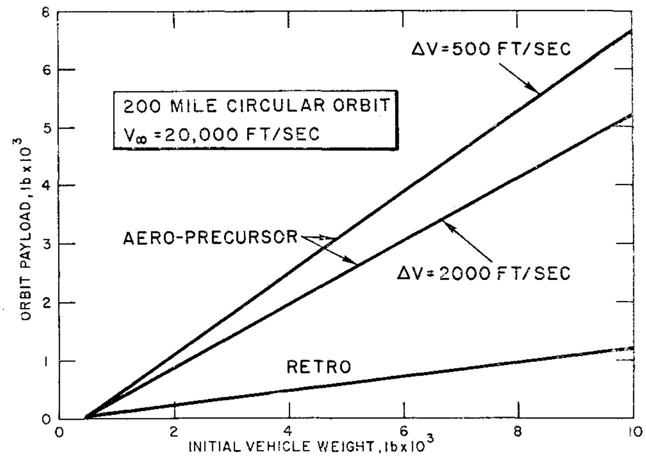

Fig. 9 Orbit payload weight comparison of aero-precursor vs retropropulsion braking to establish 200-mile circular orbit around Mars.

entry angle. In the region of interest, this varies from 0.0072 to 0.0039 rad (0.41° to 0.22°). Table 2 gives the influence coefficients for a typical point on the lower curve showing the relative importance of the various parameters. For a particular set of launch and fix-state conditions, a comparison of the uncertainty in spacecraft flight-path angle relative to the precursor flight-path angle at the reference altitude with the available entry corridor will indicate the feasibility of the precursor guidance technique. (As the uncertainty increases, compared with the available entry corridor, the data supplied the spacecraft by the precursors begin to lose significance for the spacecraft). For $M/C_DA = 1$ slug/ft², $L/D = 0.5$, $\Delta V = 1000$ fps, Fig. 3 shows that the corridor width is approximately 7° centered around −16.5°. At this entry angle, Fig. 8 gives the 3σ value of the uncertainty in the spacecraft entry angle *relative* to the precursor as 1.0°. The spacecraft is considered, then, to have been placed within the actual corridor to this accuracy irrespective of the uncertainty in the absolute entry angle.

Atmospheric Flight Guidance

Depending on the results of an error analysis for the entire aerodynamic braking system, either a lifting or nonlifting aerodynamic configuration may be indicated. The nonlifting vehicle poses no guidance or control requirements during its atmospheric flight, but the ability of such a system to tolerate errors is small. With a lifting configuration, the entry corridor is considerably broadened and the control of the aerodynamic trajectory can be used to relieve the specifications on other parts of the system. The control of a high-velocity lifting vehicle is a problem that has been studied at length by many organizations, including Avco.[6] There is no need for a complete discussion of this matter here; however, since control of the spacecraft during its pass through the planetary atmosphere may be an essential part of a system utilizing aerodynamic braking, it is pertinent to point out that a feasible solution to this problem is in hand. Studies including computer tests of approximate formulations of the spacecraft trajectory equations and of specific numerical procedures for integrating these equations have demonstrated that a computer-predictor final state control philosophy is entirely feasible in this application. An arithmetic-type digital computer of modest performance can perform the navigation function for the spacecraft and provide predicted trajectories at a sufficiently high frequency.

Table 2 Relative importance of error components for $\gamma_{ES} = 0.275$ rad (≈15.75°)

Influence coefficient[a]	Error, rad
$\sigma_R \partial\gamma_{ES}/\partial R_M$	1.08×10^{-7}
$\partial_V \partial\gamma_{ES}/\partial V_{MS}$	7.13×10^{-9}
$\sigma_\gamma \partial\gamma_{ES}/\partial\gamma_{MS}$	3.77×10^{-7}
$\sigma_\phi \partial\gamma_{ES}/\partial\phi$	1.78×10^{-7}
$\sigma_{VOP} \partial\gamma_{ES}/\partial V_{OP}$	1.16×10^{-3}
$\sigma\theta_{OP} \partial\gamma_{ES}/\partial\theta_{OP}$	3.79×10^{-3}
$\sigma_{VCS} \partial\gamma_{ES}/\partial V_{CS}$	1.21×10^{-3}
$\sigma\theta_{CS} \partial\gamma_{ES}/\partial\theta_{CS}$	2.26×10^{-1}

[a] The one-sigma values used were: $\sigma_R = 2 \times 10^5$ ft; $\sigma_V = 25$ fps; $\sigma_\gamma = \sigma_\phi = \sigma\theta_{OP} = 0.1°$; $\sigma/V_{OP} = 10$ fps; $\sigma\theta_{CS} = 0.05°$; and $\sigma/V_{CS} = 0.2$ fps.

Payload Performance

Results are presented in Fig. 9 comparing the efficacy of aerodynamic braking vs pure retropropulsion in placing a payload in a 200-mile circular orbit around Mars from hyperbolic approach ($V_\infty = 20{,}000$ fps) conditions. It can be seen that for any given initial vehicle weight, the use of atmospheric braking based on precursor information can result in payload factors of 4 to 5 compared to retropropulsive braking. The magnitude of the payload with atmospheric braking depends on the velocity increment required to achieve a stable orbit after leaving the Martian atmosphere; the curves for $\Delta V = 2000$ and 500 fps are considered to bracket this requirement. For both the aeroprecursor system and the retrosystem, a propellant specific impulse of 310 sec and a propellant mass fraction of 0.865 for the rockets were assumed. In addition,

Table 3 Weight parameters used for aeroprecursor system

Parameters	Value
Ratio of spacecraft structure and heat shield weight to initial spacecraft weight	0.25
Precursor weight breakdown	
Accelerometer	0.5
Communications	3.0
Structure and heat shield	5.3
Rocket	6.7
Miscellaneous (yo-yo despin weights, etc.)	2.0
Total	17.5 lb
Allowance for 6 precursors in Fig. 4 at nominal 17.5 lb each	105 lb
Miscellaneous equipment weight attributed to 6 precursors: spinup motors, power, launch tubes, etc.	130 lb

the weight parameters, shown in Table 3, were used for the aeroprecursor system.

Figure 9 suggests an answer to the question whether precursors are necessary if "adequate atmospheric information is available." It is not difficult to postulate a future time period in which uncertainties in our knowledge of the Martian atmosphere are possibly reduced to the point where the atmosphere can be used for braking purposes without data augmentation. However, the desirability, as opposed to the necessity, of incorporating precursors should be considered. Uncertainties in the knowledge of the atmosphere will still remain, and the propulsive requirements to attain the prescribed orbit after the atmospheric flight are a strong function of these uncertainties. Significant increases in payload can also result from reductions in these propulsive requirements, as may be inferred from Fig. 9.

Conclusions

The use of precursor vehicles to probe planetary atmospheres prior to the entry of a spacecraft appears to make possible the use of atmospheric braking in spite of substantial uncertainties in knowledge of the atmospheric characteristics. A feasibility study of the complete system indicates a very attractive potential for the concept; a more detailed design study seems warranted.

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Hall-Current Accelerator Utilizing Surface Contact Ionization

EDWARD A. PINSLEY,* CLYDE O. BROWN,† AND CONRAD M. BANAS‡
United Aircraft Corporation, East Hartford, Conn.

Analyses and experiments were performed on a Hall-current accelerator in which ion acceleration occurs in an annular region filled with a neutral, fully ionized, nonequilibrium cesium plasma. Cesium ions are formed by both surface contact and volume ionization, and electrons are introduced by a downstream cathode. Acceleration is achieved by interaction of a radial magnetic field and an impressed axial electric field producing azimuthal electron drifts. Magnetic field strength is adjusted such that $\omega_e\tau_e \gg 1$, and the ion cyclotron radius is large compared with accelerator length. Accordingly, axial electron currents are inhibited by the magnetic field, whereas ion currents are unaffected. Constraints on accelerator performance and design imposed by the onset of a two-stream instability are considered. Losses due to this instability can be avoided by properly choosing operating variables and accelerator size. Efficient, high-thrust-density performance is theoretically achievable at moderate specific impulse with large accelerators and high thrust levels, provided that electron backflow can be held to tolerable limits. Potential advantages include high tolerance to neutral efflux and low ionizer heater requirements. Preliminary tests have been conducted on an experimental accelerator and ion source under conditions of exhaust beam current neutrality.

Introduction

USE of a low-density Hall-current accelerator to circumvent the space-charge-limited current operation of conventional ion accelerators has been suggested[1] as a means of obtaining high-performance electric propulsion thrustors in the moderate (2000–5000 sec) specific impulse regime. Neutralization of the ion space charge in this device is achieved by the introduction of an equal number density of electrons in an annular accelerating region. Electrons are constrained to azimuthal Hall-current drift motions by a radial magnetic field and an impressed axial electric field that also provides a means of adding directed kinetic energy to the ions. The Hall-current Lorentz force couples the ion reaction force to the accelerator structure. The accelerating region is filled with a low-density, collisionless ($\omega_e\tau_e \gg 1$) plasma, in which the ion cyclotron radius is large and the electron cyclotron radius is small with respect to accelerator dimensions. This so-called "EM-region" plasma, in principle, permits electrostatic acceleration of ions while electrons are constrained to $\mathbf{E} \times \mathbf{B}$ drift motions.

Previous experimental investigations[2–5] have been conducted on devices in which argon ions are formed by volume ionization in the accelerating region. To sustain the discharge it has been necessary to operate such accelerators at rather high neutral gas pressures (over 10^{-3} mm Hg). The presence of background neutrals at these pressures can de-

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* Chief, Electrical Propulsion, Research Laboratories. Associate Fellow Member AIAA.

† Associate Research Engineer, Electrical Propulsion, Research Laboratories. Member AIAA.

‡ Research Engineer, Electrical Propulsion, Research Laboratories. Member AIAA.