A Survey of Aeroassisted Orbit Transfer

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Introduction

IN 1961, Howard London presented a paper[2] at the 29th Annual Meeting of the Institute of the Aeronautical Sciences in which he demonstated the possibility of using aerodynamic forces to produce an orbital plane change with an expenditure of energy significantly smaller than that associated with an extra-atmospheric propulsive maneuver. Since that time, numerous studies of aeroassisted orbit transfer have been carried out for a wide variety of Earth-orbital and planetary missions. Almost any mission that involves changes in orbital altitude and/or inclination in the vicinity of an atmosphere-bearing planet is a candidate for aeroassist. For ease of discussion these many possible mission applications may be categorized as 1) synergetic plane change, 2) planetary mission applications, and 3) orbital transfer vehicle applications.

The term, synergetic plane change, is commonly used to denote a maneuver in which a change in orbit inclination is accomplished through a combination of aerodynamic and propulsive forces rather than through propulsion alone. While the maneuver is applicable to any planet with a suitable atmosphere, practically all studies to date have dealt with low Earth orbit missions.

As illustrated in Fig. 1, a retrorocket burn causes the vehicle to leave its initial low Earth orbit (LEO) and dip into the atmosphere. During the atmospheric pass, the vehicle is banked so that the lift vector is in a lateral direction, thus producing an aerodynamic turn. As will be discussed subsequently, this turn is often accomplished with continuous thrusting sufficient to balance the aerodynamic drag. Following the aerodynamic turn, the vehicle is reboosted to the desired orbital altitude and a rocket burn is effected to circularize the orbit (shown by the dashed low Earth orbit in Fig. 1).

The typical orbital transfer vehicle (OTV) maneuver is in many ways similar to the synergetic plane change. In the OTV case, however, the vehicle is initially in a high Earth orbit (HEO)—geosynchronous in most studies—and must undergo a sizeable velocity decrement in its transfer to LEO as illustrated in Fig. 1. This velocity decrement is more efficiently achieved through a combination of aerodynamic drag and propulsion than through propulsion alone. The OTV mission usually involves some change in orbital plane—for instance, the return from geosynchronous orbit to a typical Shuttle orbit involves a plane change of 28.5 deg. If the OTV is a lifting configuration, some of the required plane change can be achieved by banking the vehicle as in the synergetic plane change maneuver.

In planetary mission applications, aeroassist is again used to produce a velocity decrement through aerodynamic drag. For these missions two distinctly different applications of aeroassist have been studied: 1) aerocapture in which the velocity decrement achieved in a single deep atmospheric pass is sufficient to transfer the vehicle from its hyperbolic approach trajectory to a target orbit about the planet (i.e., the vehicle is captured by the planet's gravitational field); and 2) multipass aerobraking in which the vehicle is transferred from its approach trajectory to a highly elliptic orbit about the planet by a rocket burn and then the orbit is circularized by many high-altitude atmospheric passes, each followed by small corrective rocket burns at apoapsis to maintain the periapsis altitude low enough to produce the desired drag decrement but high enough to avoid excessive aerodynamic heating. The first type of planetary application, as illustrated in Fig. 1, is similar to the OTV maneuver except that it usually has a larger velocity excess and no plane change. The second type of planetary application is illustrated in Fig. 2. As is also illustrated in Fig. 2, this type of multipass orbit lowering has also been studied for the GEO-to-LEO OTV mission.

In the intervening years since London presented his paper in 1961, many studies have been carried out and a rich literature has developed. In the process of researching the present paper, an extensive bibliography was compiled. It lists the relevant publications in chronological order for each of the three vehicle classes considered. While certainly incomplete (new and additional references keep turning up), this collection of papers is believed to present an accurate overall picture of the evolution of aeroassist technology. Examination of this bibliography shows that the various aeroassist mission applications received their most intensive study in different time periods. As was mentioned previously, the serious study of synergetic plane change maneuvers began with London's paper in 1961. Actually, there were studies published prior to 1961 (e.g., Ref. 1) but London's paper appears to be the first to convincingly demonstrate a significant performance gain. During the 1960s, synergetic plane change was studied extensively with the peak activity occurring around 1967. The initial

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motivation for this work was the recognition that orbital plane changes required very large characteristic velocities (ΔV). As pointed out by Caudra and Arthur,[8] the ΔV required for a single-impulse 60 deg plane change is as large as that required to place the satellite in low Earth orbit in the first place. As these studies progressed, it became evident that the same type of high-L/D (lift-to-drag ratio) vehicle required for the synergetic plane change maneuver could also perform Earth-return missions with large cross-range capability, perform evasive maneuvers, and have considerable military importance.

These synergetic plane change vehicles tended to be slender, high L/D, blended wing-bodies with refractory metal thermal protection systems for reusability. While deployable air-breathing engines and canards were sometimes used to provide subsonic loiter capability and improved landing characteristics, most recent design concepts[34] have been unpowered gliders and have achieved acceptable landing performance without canards. Of course, there are civilian as well as military uses for synergetic plane change. For example, the Space Shuttle Orbiter has significant synergetic plane change capability.[31] Since 1970, work on synergetic plane change vehicles has continued at a relatively low level, but there are signs that interest in this area may be building again along with interest in "transatmospheric" vehicles.

The second type of aeroassist mission to be studied in depth was planetary aerocapture, for which there have been two periods of rather high interest—1964-1968 and 1979-1982. Work in the 1964-1968 time frame was aimed primarily at the manned Mars mission proposed by NASA as a follow-on to Apollo. When it became evident around 1969 that there would be no manned Mars mission, interest in the United States waned as attention was forcused on unmanned planetary missions and the Space Shuttle. In fact, almost all the publications on this subject in 1969-1978 are European (mostly Russian). Beginning in the late 1970s, there as a strong rekindling of interest in aeroassist for application to unmanned planetary missions. This activity included studies of the application of both aerocapture (as described previously and illustrated in Fig. 1) and multipass aerobraking (Fig. 2) to a wide range of planetary missions. As described previously, aerocapture involves a single pass deep into the planetary atmosphere and hence requires a smooth aerodynamic shape. Flight control requirements are satisfied by an L/D in the range of 1.0-1.5 with bank angle modulation for trajectory shaping. Because of the high levels of aerodynamic heating, ablative heat shields are required.

While aerocapture missions involve deep atmospheric penetration and continuum flow, multipass aerobraking is usually carried out at such high altitudes that near-free-molecule flow is experienced. Accordingly, an aerobraking spacecraft may have no aerodynamic surfaces other than one or more deployable drag brakes that provides the required frontal area and shield the other parts of the spacecraft from the rarefield flow. Because of the low levels of aerodynamic heating, the aerobrakes can be made of metallic, ceramic, or carbonaceous materials that will survive numerous atmospheric passes. Because of the low levels of aerodynamic forces, multipass aerobrakers can be aerodynamically stabilized or spin stabilized. Since many shallow atmospheric passes are involved in an aerobraking mission, the vehicle can literally "feel its way" into an unknown planetary atmosphere by measuring decelerations and surface temperatures and continually adjusting its periapsis altitude to insure acceptably low heating. The great disadvantage of multipass aerobraking is that it takes a long time. For instance, an aerobraking version of the Venus Orbiting Imaging Radar mission required 44 days.

The final mission category involves the application of aeroassist to orbital transfer vehicles. Interest in this area began around 1970 when a multipass aerobraking version of the Space Tug (the initially proposed upper stage for the Space Transportation System) was studied.[70] While the aerobraking Tug achieved significant payload gains, the multiple atmospheric passes severely inhibited operational flexibility. Hence, one- and two-pass aeroassisted OTVs (AOTVs) were considered and this led to a rather extensive series of studies of the AMOOS (Aeromaneuvering Orbit-to-Orbit Shuttle) concept. AMOOS[79] was an elipsoidal cylinder sized to fit the Shuttle cargo bay with a raked-off nose and an aft-mounted trim flap. It flew at relatively high angles of attack (approximately 35 deg), was capable of L/Ds in the range of 0.5-1.0 and was controlled through bank-angle modulation. Because of its lifting capability, AMOOS could achieve 7 deg of the required 28.5 deg GEO-LEO orbit plane change aerodynamical-

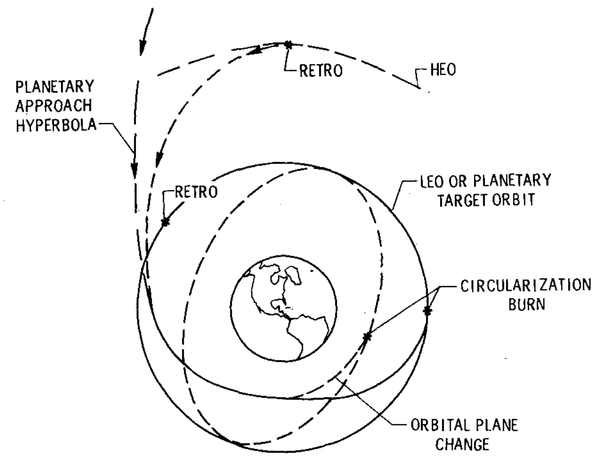


Fig. 1 Aeroassisted maneuvers for synergetic plane change, orbital transfer vehicle, and planetary aerocapture missions.

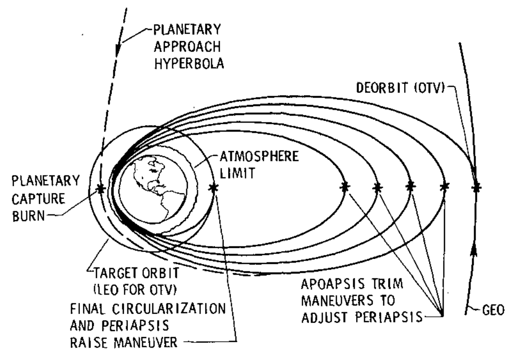


Fig. 2 Multipass aeroassisted maneuvers for planetary aerobraking and orbital transfer vehicle missions.

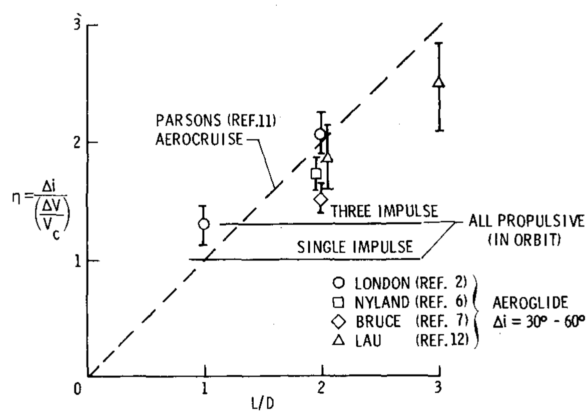


Fig. 3 Dependence of synergetic efficiency η on vehicle lift-to-drag ratio.

ly. It was not truly reusable since it had an ablative nose heat shield. The design concept called for AMOOS to be retrieved by the Shuttle after its LEO-GEO round-trip mission and returned to Earth for refurbishment. Numerous studies in the 1970-1977 time frame indicated a round-trip LEO-GEO payload capability of about twice that of a comparable all-propulsive vehicle. However, one paid the price of significantly increased vehicle complexity and sophistication.

Around 1980, there was a resurgence of interest in AOTVs sparked by the OTV concept definition studies.[84,89] Again, the vehicles were Shuttle-compatible, but in these studies some new and innovative approaches focused attention on low-L/D, low-ballistic-coefficient concepts.. Once again factor-of-two payload gains over all-propulsive vehicles were indicated. As a result of the AMOOS and OTV concept definition studies, there is great current interest in aeroassisted OTVs.

From the preceding discussion, it can be seen that aeroassist is hardly a new technology. The literature is extensive and contains numerous convincing demonstrations of significant performance gains due to aeroassist. Yet, aeroassist has seldom been employed in actual missions. In the subsequent sections of the present paper, the various vehicle concepts will be examined in greater detail and an attempt will be made to identify the technology barriers that have, thus far, prevented aeroassisted orbit transfer from achieving practical implementation.

Synergetic Plane Change

As was mentioned in the introduction, serious study of the synergetic plane change maneuver began around 1961 and was pursued vigorously throughout the 1960s. Many of the early studies were essentially point-mass trajectory analyses in which the emphasis was on defining the theoretically achievable plane changes and corresponding ΔV requirements rather than on the design details of the vehicle. The results of several such analyses are summarized in Fig. 3.

The synergetic efficiency η is defined as the ratio of the orbital plane change achieved (expressed in radians) to the required ΔV (normalized by satellite velocity V_c). For a single-impulse, all-propulsive maneuver performed outside the atmosphere, η is approximately unity. In Ref. 11, Parsons showed that, under certain simplifying assumptions, the maximum achievable value for η is nearly equal to L/D (indicated by the dashed line in Fig. 3). The symbols with bars represent ranges of results obtained by various authors[2,6,7,12] for orbital plane changes from 30 to 60 deg. Two of the main conclusions drawn by the early investigators are illustrated in Fig. 3. First of all, it appears to be desirable to maximize vehicle L/D. Secondly, vehicle L/D must be significantly higher than 1.0 if the synergetic maneuver is to be superior to an all-propulsive maneuver. Actually, a three-impulse propulsive maneuver (in which the vehicle is first put in a highly elliptical orbit and then the plane change is produced by a rocket burn at apogee followed by recircularization to LEO) can achieve η significantly greater than 1.0; hence, for practical purposes, synergetic plane change vehicles should have L/D of at least 2.0. The terms "aeroglide" and "aerocruise" shown in Fig. 3 refer to two widely studied types of synergetic maneuvers. In the first (aeroglide), the plane change is accomplished by a gliding turn. In the second (aerocruise), continuous thrust sufficient to cancel the aerodynamic drag is used. The merits of these two approaches will be discussed subsequently.

Not only did the early investigators agree that high L/D's were desirable, they also agreed that the maneuver should be carried out at $(L/D)_{max}$. Hence, the early concept of a synergetic plane change vehicle was a long, slender, high-L/D, low-wing-loading configuration that performed a gradual, high-altitude turn at $(L/D)_{max}$. Unfortunately, there are practical difficulties with such a concept.

First of all, high-L/D vehicles are heavy. Figure 4 presents the variation of vehicle weight (normalized by the weight of a ballistic configuration W_0) with increasing L/D. The results presented in Fig. 4 were compiled by Love and Pritchard[4] from a series of manned entry vehicle design studies. While these results are for Earth entry vehicles (entering at satellite velocity), not synergetic plane change vehicles, the configurations and the variations of W/W_0 with L/D would be expected to be very similar for these two vehicle types. Secondly, vehicles flying at $(L/D)_{max}$ experience very high integrated heat loads and slender, high-L/D configurations have small nose radii and, hence, high-stagnation-point heating rates. The high integrated heat loads are illustrated in Fig. 5, which contrasts the heating environments for entries at $(L/D)_{max}$ and $C_{L\max}$.

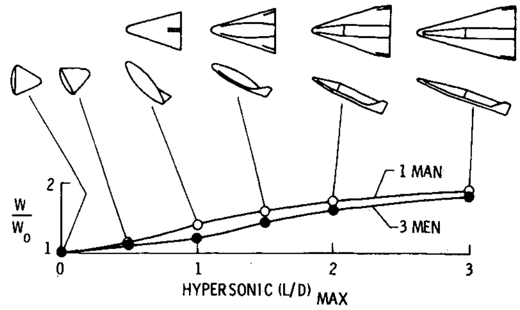
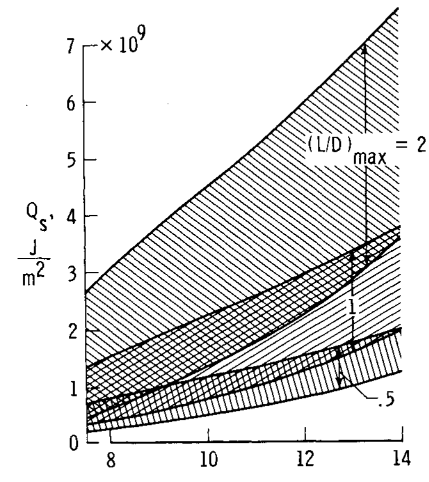
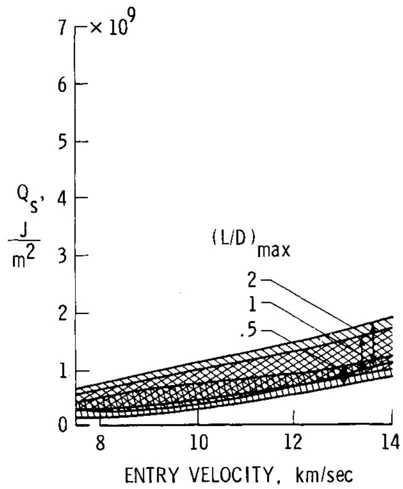


Fig. 4 Increase in vehicle weight with increasing maximum lift/drag ratio.[4]



a) Entry at L/D_{max}.



b) Entry at $C_{L_{max}}$.

Fig. 5 Integrated heat load as a function of entry velocity.[4]

Figure 5 is also taken from Ref. 4. Again, it is for entry vehicles, but it accurately depicts the trends for synergetic plane change vehicles and suggests that a large reduction in the heat load can be achieved by carrying out a short-duration, high-angle-of-attack (i.e., $C_{L\max}$) turn rather than the gradual $(L/D)_{\max}$ turn that was identified as most efficient by the early investigators. Of course, turns performed near $C_{L\max}$ (i.e., at high angle of attack) will produce higher acceleration loads, which must also be analyzed to make sure that vehicle and pilot tolerances are not exceeded. A third constraint that was not evaluated initially is the necessity of completing the turn in a relatively short time. This is illustrated in Fig. 6, which shows that a turn performed at the orbit node (i.e., at the equator) is fully effective in producing a change in orbit inclination, whereas a turn performed at the orbit apex actually produces no change in orbit inclination. Instead it produces a change only in the location of the orbit nodal crossing. Hence, if the turn lasts a long time (half an orbit, for instance), a significant part of the turn (when the vehicle is near the apex of the original orbit) is essentially wasted from the standpoint of changing the orbit inclination. When these several constraints (heating, accelerations, range, etc.) are considered together, the maneuvering envelope of the vehicle is markedly reduced. This is illustrated by Fig. 7, which was originally published by Lau.[12]

The range angle R shown in Fig. 7 is defined as the included angle between two radial lines originating at the center of the Earth and passing respectively through the points where the vehicle initially departs from its original orbit and subsequently achieves its new orbit after the synergetic turn. Lau considers $R = 100$ to be approximately the maximum practical range angle. Hence, the hatched line in Fig. 7 indicates the lower boundary of the maneuvering region. The upper boundary is set by two constraints: a surface temperature limit of 1944 K (3500°R) that Lau[12] considered the maximum operational temperature for future reradiative thermal protection systems and the $\Delta V = 2.44$ km/s line representing a propellant mass fraction of 54%. Actually, 1944 K (3500°R) is beyond the practical capability of present reradiative systems. A more realistic number would be 1667 K (3000°R), which would yield a surface temperature boundary even lower than that shown in Fig. 7. The left- and right-hand boundaries in Fig. 7 are indicated by the heavy lines that represent the respective characteristic velocity requirements for an all-propulsive (exoatmospheric) maneuver and a minimum-energy [i.e., $(L/D)_{\max}$] synergetic turn. Actually, the right-hand boundary does not go all the way to the minimum energy turn line, but is defined by the intersection of the $R = 100$ and $\Delta V = 2.44$ km/s lines. Lines of peak acceleration (4g, 6g, etc.) are also presented in Fig. 7, but Lau concluded that the heating rather than the acceleration constraint would usually fix the upper boundary of the maneuvering region. The heading change $\Delta\theta$ shown in Fig. 7 is essentially equal to the change in orbit inclination. Hence, Fig. 7 indicates that plane changes of less than about 15 deg would be better carried out propulsively, that the maximum plane change capability for the vehicle studied ($L/D = 3.0$, $W/A = 146$ kg/m^2) would be about 45 deg, and that this would require about half the vehicle weight to be taken up by fuel. Figure 7 also indicates that near-minimum energy turns in the 20-30 deg range are not practically achievable because of excessive range requirements.

Another unrealistic assumption made in the early analyses was that the hypersonic maximum L/D for the vehicle was constant. For instance, all the results presented in Figs. 3 and 7 were based on the assumption of constant L/D. Actually, the $(L/D)_{\max}$ degrades with increasing altitude because of viscous effects. This was pointed out by Maslan,[18] who carried out the first analysis of synergetic plane change using variable $(L/D)_{\max}$. Maslan estimated the variation of $(L/D)_{\max}$ by using approximate generalized expressions for C_L and C_D that depended on a bluntness parameter and a similarity variable (which contained viscous effects) and yielded simple expressions for the decrease in $(L/D)_{\max}$ with decreasing atmospheric density (and hence, increasing altitude). With the $(L/D)_{\max}$ variation thus described, Maslan essentially repeated Parson's analysis[12] for a powered synergetic turn (aerocruise) and showed significant reductions in plane change capability due to the effective reduction in L/D. While the trends predicted by Maslan are correct, it now appears that his approximate expressions significantly overpredicted the reduction in L/D.

Figure 8 presents the variation of $(L/D)_{\max}$ with altitude for the Space Shuttle as presented in the Shuttle Aero Data Book[30] and as predicted by Maslan's approximate expressions from Ref. 18. The solid curve represents a best fit to the extensive wind tunnel data base developed for the Shuttle Orbiter and has been shown to agree with the Orbiter flight data to within + 3%. Comparison of the solid curve with the hatched band shows that Maslan's expressions underpredict $(L/D)_{\max}$ by a factor of over three at an altitude of 90 km. Hence, the high-altitude degradation of $(L/D)_{\max}$ is important and should be taken into account, but its impact may have been overestimated in many prior analyses.

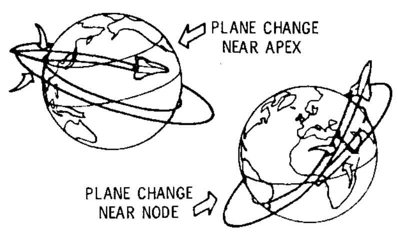


Fig. 6 Effect of performing plane change at orbit node and apex.[33]

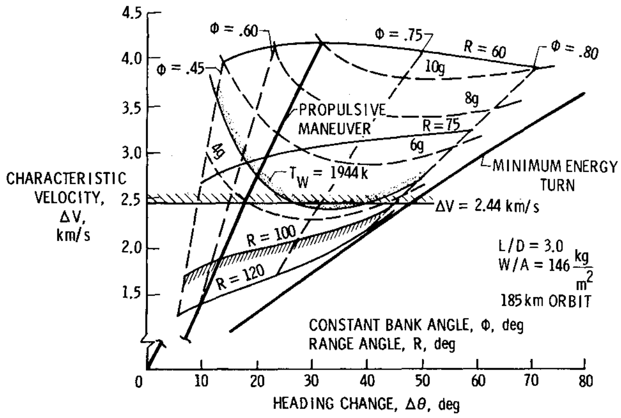


Fig. 7 Effects of thermal, dynamic, characteristic velocity, and range constraints of achievable orbital plane change.[12]

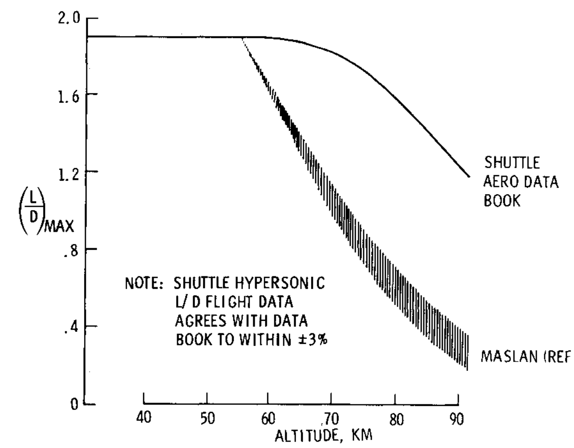


Fig. 8 Degradation of $(L/D)_{\max}$ due to high-altitude viscous effects.

As has been mentioned previously, numerous investigators have studied both gliding and powered turns. From the standpoint of theoretical synergetic efficiency, the two techniques are comparable (see Fig. 3). However, when more detailed vehicle designs are considered, the thrusting turn (aerocruise) has been shown to have significant advantages. This is illustrated by Fig. 9, which is from an analysis by Claus and Yeatman.[13]

In Fig. 9, the ratio of the vehicle's weight after the orbital plane change (W_2) to its initial weight (W_1) is presented as a function of the plane change (Δi) for aerocruise, aeroglide, and all-propulsive vehicles. Note that the aerocruise vehicle is markedly superior to the aeroglide one, especially when lower wall temperature limits are imposed. The explanation for these results is that as an aeroglide vehicle carries out its gliding turn, it is continuously decelerating and descending through the atmosphere and, hence, will encounter the peak heating region corresponding to the "knee" of the velocity-altitude curve typical of a re-entry vehicle. In this peak heating region, the aeroglide vehicle must fly at reduced bank angles to avoid exceeding the surface temperature limit and hence has less turning capability. Only after the vehicle has decelerated through the peak heating region can it resume high bank angles and complete the turn. The lower the surface temperature limit, the more the vehicle must decelerate and descend deep into the atmosphere before completing the turn. As a result, more ΔV (and hence propellant weight) is required for reboost to LEO. On the other hand, the aerocruise vehicle employs continuous thrusting sufficient to balance aerodynamic drag and maintains an altitude high enough to allow optimum bank angles throughout the turn without exceeding surface temperature limits.

The most recent studies of synergetic plane change vehicles are typified by Ref. 33, which analyzes the MRRV (maneuverable re-entry research vehicle). The MRRV configuration and aerodynamic characteristics are presented in Fig. 10 and time histories describing the synergetic plane change maneuver are presented in Fig. 11. As shown in Fig. 10, the MRRV is a blended wing/body configuration which has a maximum hypersonic L/D of about 2.4. The vehicle flies a powered (aerocruise) turn at a nearly constant altitude of approximately 73 km (240,000 ft). During the turn, the bank angle is modulated about 70 deg and the angle of attack varies from 26 to 19 deg. The maneuver was optimized with respect to the aerodynamic heating constraint limiting the maximum stagnation point temperature to 2778 K (5000°R) and the maximum temperature downstream of the nose cap ($x/L>0.02$) to 2667 K (3000°R). These temperatures are consistent with a carbon-carbon nose cap and a metallic reradiative heat shield over the rest of the vehicle. Note that the maneuver is carried out at angles of attack considerably higher than that corresponding to $(L/D)_{max}$ (compare Figs. 11a and 10). This is done to provide a quicker turn, which can be carried out near the orbit node and which significantly reduces the integrated aerodynamic heat load (see Fig. 5). This point is further illustrated by the plot of the orbit inclination vs the vehicle weight presented in Fig. 11b. Comparison of the two solid curves shows the superiority of high α-ϕ_B maneuver to an $(L/D)_{max}$ maneuver carried out at a constant angle of attack of 14 deg. This plot also shows that the weight of propellant expended during the maneuver is equal to about half the initial vehicle weight and that an orbit inclination change of about 20 deg is achieved. A comparable all-propulsive maneuver would produce an inclination change of about 15 deg. These results are typical of the present state-of-the-art in synergetic plane change vehicles. Overall, the capabilities of these vehicles are much smaller than those predicted in the early 1960s. As Maslan[18] concluded, "...the results indicate that the synergetic mode can be more desirable than in-orbit propulsion only if the aerodynamic performance must be provided for other reasons...".

However, any entry vehicle that has sizable cross-range capability possesses the potential for significant synergetic plane change. For example, Joosten and Pierson[31] recently assessed the synergetic plane change capability of the Space Shuttle Orbiter. Their results suggest that significant plane changes could be produced by relatively small additional ΔV. A 15 deg synergetic plane change would require an additional ΔV of approximately 1.2 km/s (4000 ft/s) which could be produced by four Orbital Maneuvering System (OMS) kits. One can envision a future space-based, high L/D vehicle that would be capable of carrying out high-cross-range entries (i.e., for space rescue) and that could perform synergetic maneuvers to transfer men and equipment between orbits of various inclination.

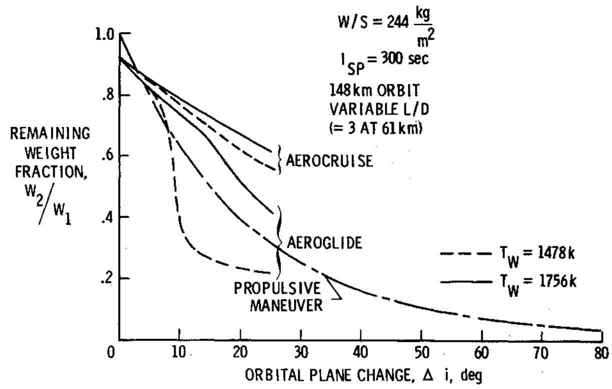


Fig. 9 Comparative synergetic plane change performance for gliding (aeroglide) and powered (aerocruise) turns.[13]

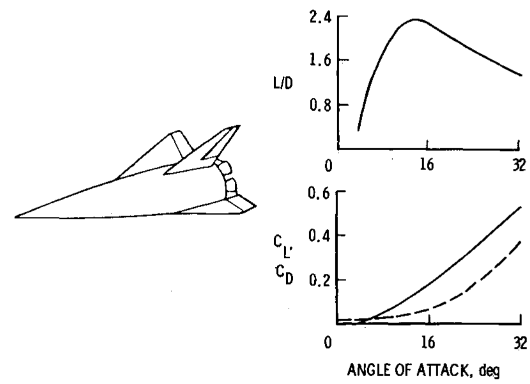


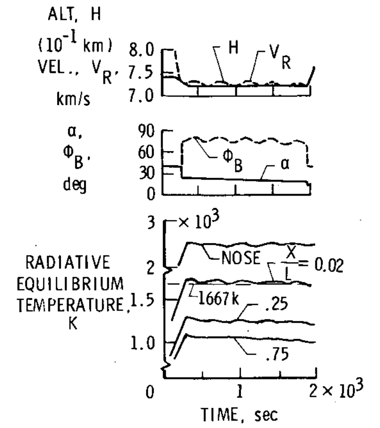
Fig. 10 Configuration and aerodynamic characteristics of the maneuverable re-entry reseach vehicle (MRRV).[33]

Planetary Missions

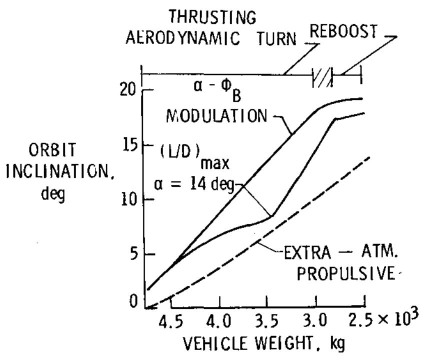
Aerocapture

In the 1964-1968 time frame, NASA was seriously proposing a manned mission to Mars as the next major space initiative following the Apollo program. Minimizing the mass of the vehicle placed in Martian orbit is very important for such missions, since and increase of 1 kg in the mass of the Martian orbiter requires an increase of approximately 10 kg in the mass of the spacecraft that must be placed in low Earth orbit. Accordingly, the use of aerodynamic drag to provide the velocity decrement necessary for capture into Martian orbit was studied and found to yield significant performance gains. A typical vehicle and the performance gains for both a Mars orbiter and a Mars lander mission (according to Boobar et al.[42]) are presented in Figs. 12 and 13.

The vehicle is an on-axis biconic configuration with a circular cross section and a spherically blunted nose. As shown by the plot of L/D vs C_DA, practical configurations of this type can have maximum L/D in the range of 0.8-1.6. The configuration studied in Ref. 42 had a half-cone angle, $\theta_1=16$ deg, which produced $(L/D)_{max}\simeq 1.3$. As the vehicle passed through the Martian atmosphere, it flew at a constant angle of



a) Altitude, velocity, angle-of-attack, bank angle, and surface temperature time histories.



b) Orbit inclination change.

Fig. 11 Synergetic plane change maneuvers for the MRRV.[33]

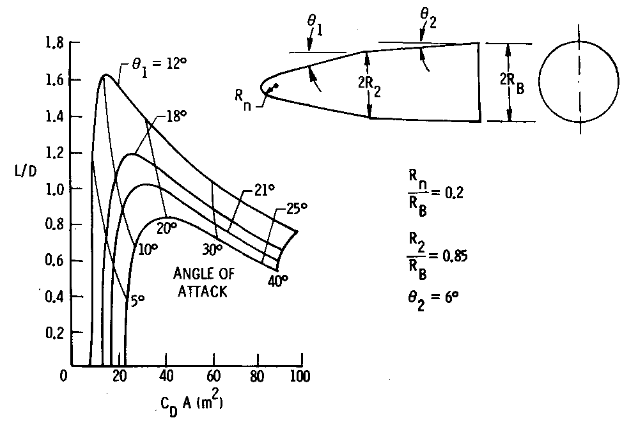


Fig. 12 Planetary aerocapture configuration proposed for a manned mission to Mars.[42]

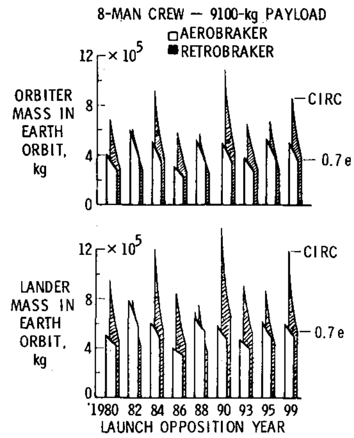


Fig. 13 Comparative performance of aeroassist (aerobraker) and all-propulsive (retrobraker) spacecraft for both orbiter and lander missions to Mars.[42]

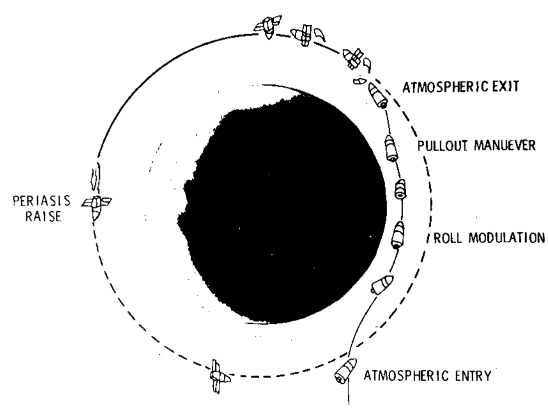


Fig. 14 Planetary aerocapture maneuver.[51]

attack and was controlled through bank angle modulation. The guidance and control concept used was essentially the same as that developed for the Apollo mission. An ablative heat shield provided thermal protection. A measure of the optimism of the late 1960s is provided by the fact that the vehicle analyzed carried an eight-man crew and a 9100-kg payload to Mars. Figure 13 presents the comparative masses that would have to be placed in Earth orbit for aeroassist (aerobraker) and all-propulsive (retrobraker) vehicles for various launch opposition years. The peak of each bar (labeled CIRC) represents the case where the vehicle is placed in a circular orbit about Mars. The point labeled 0.7e indicates the mass requirements if the vehicle is placed in an elliptical orbit with an eccentricity of 0.7. As Fig. 13 shows, mass reductions of over a factor of two are not uncommon. These benefits are so large that, had there been a manned Mars mission, it probably would have employed aeroassist. However, the manned Mars mission was not to be and, around 1970, interest in planetary aeroassist missions declined as emphasis was shifted to unmanned planetary missions and the Space Shuttle. Throughout the early and mid-1970s, research on aeroassisted planetary missions continued at a low level, with most of the work being carried out in Europe. Then, around 1978, there was a strong resurgence of interest, this time associated with unmanned missions.[51-62] The aerocapture maneuver and vehicle concept proposed in these recent studies is similar to that studied earlier for the manned Mars mission (see Fig. 12). The aerocapture maneuver is illustrated in Fig. 14. The vehicle enters the planetary atmosphere with a significant hyperbolic excess velocity and flies a roll-modulated, constant-drag trajectory that produces the required velocity decrement and achieves the atmospheric exit conditions required to place it in the desired orbit about the planet. A relatively small exoatmospheric rocket burn is required to circularize the final orbit.

The approach trajectory correction maneuvers outside the atmosphere can be controlled by commands computed on Earth. However, once the vehicle enters the atmosphere, trajetory maneuvers must be computed and commanded on board by a closed-loop guidance navigation and control system using adaptive guidance algorithms that can correct the flight path to meet the required atmospheric exit boundary conditions. In order to successfully perform this maneuver, the vehicle must have lifting capability. Studies by Cruz[51] and others show that the accuracy of the aerocapture maneuver improves with increasing L/D until it reaches a maximum around $L/D = 1.5$. This is illustrated in Fig. 15, where the 3σ

accuracy of the vehicle speed at atmospheric exit and the period of the final orbit is presented as a function of L/D for a Mars mission.

In Fig. 16, typical L/D and ballistic coefficients (m/C_DA) are displayed for a range of generic configurations. As can be seen from Fig. 16, L/D greater than 1.5 requires lifting bodies and blended wing/body configurations that have relatively poor volumetric efficiency and higher ballistic coefficients (based on cross-sectional area). Accordingly, most studies of planetary aerocapture missions have recommended on-axis or bent biconic configurations. This is true of the manned missions studied in the late 1960s, as well as the unmanned missions of current interest.

Since planetary aerocapture vehicles must dissipate considerable energy during the atmospheric pass, heating and thermal protection systems are significant design drivers. In Fig. 17 (which is based on data presented in Ref. 60), the heating enevironment is shown for various planetary missions. The Mars and Uranus missions indicated in Fig. 17 have mission scenarios similar to those illustrated in Fig. 14. The designation S02P-Titan, however, refers to the Saturn Orbiter dual-probe mission in which the spacecraft is captured into an orbit about Saturn as a result of a pass through the atmosphere of Titan, one of the Saturnian moons. As shown by Fig. 17, aerocapture maneuvers produce relatively high values for both the heat-transfer rate and the heating time (and, hence, integrated heat load). This is in contrast to conventional entry vehicles that generally experience either low heating rates and high integrated heat loads or high heating rates and low integrated heat loads (as shown on Fig. 17 for the Pioneer Venus entry probe and strategic military entry vehicles). As is also shown by Fig. 17, Martian and low-entry-velocity Titan aerocapture vehicles can use low-density ablators, whereas Uranus and high-entry-velocity Titan vehicles will require high-performance ablators (carbon-phenolic or carbon-carbon) backed by efficient insulators over the forward portions of the aeroshell.

As an illustration of a typical recently studied aerocapture configuration, the Saturn, Orbiter Dual Probe (S02P) vehicle (developed in Ref. 60) is presented in Figs. 18 and 19. In Fig. 18, the vehicle is shown in its atmospheric flight configuration with the radioisotope thermoelectric generator (RTG) and science booms and the high-gain antenna retracted and enclosed within the aeroshell. In Fig. 19, the spacecraft is shown in its fully deployed condition after having completed the aerocapture maneuver at Titan. During the S02P mission, entry probes are sent into the atmosphere of Titan and Saturn, and the core spacecraft with its imaging and rf science experiments is placed in orbit about Saturn and carries out a series of fly-by encounters with the Saturnian satellites. The Titan probe makes up the nose portion of the Titan aerocapture configuration as shown in Figs. 18 and 19. The Saturn probe is packaged between the Titan probe and the core spacecraft (not shown in Fig. 18). One of the most serious design challenges for planetary aeroassist vehicles is illustrated in Figs. 18 and 19. For an all-propulsive mission, the spacecraft would be in its deployed condition (as shown in Fig. 19) throughout its interplanetary transit and during its approach to the target planet. The RTGs (with their radioactivity and heat) would be deployed on their booms away from the science payload, the high-gain antenna would be deployed and available for tracking and data transmission, and the spacecraft could maintain thermal control by radiating excess heat to space. For an aeroassisted spacecraft, the enclosing aeroshell tends to trap the heat generated by the RTGs and the spacecraft instrumentation. Hence, radiation shielding is required for the science instruments and other spacecraft subsystems, an additional expendable surface-mounted antenna is required for tracking and navigation, and an active cooling system is required to maintain spacecraft thermal control. Of course, there is the option of opening doors in the aeroshell to

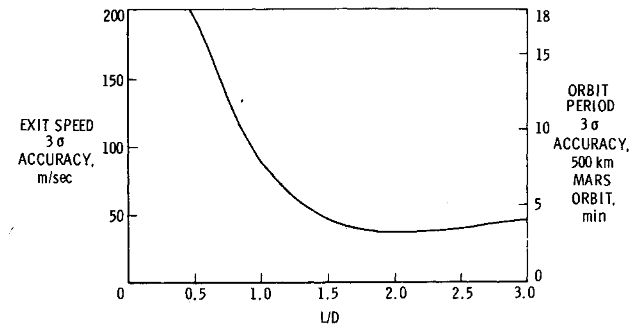


Fig. 15 Effect of L/D on aerocapture accuracy (Mars mission, vehicle entry mass = 5000 kg).[51]

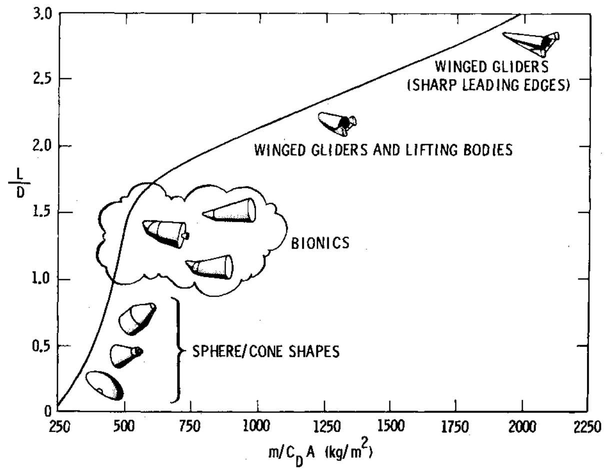


Fig. 16 Lift/drag ratios and ballistic coefficient values for typical lifting vehicles (entry mass = 5000 kg, cross-sectional area = 14 m²).[51]

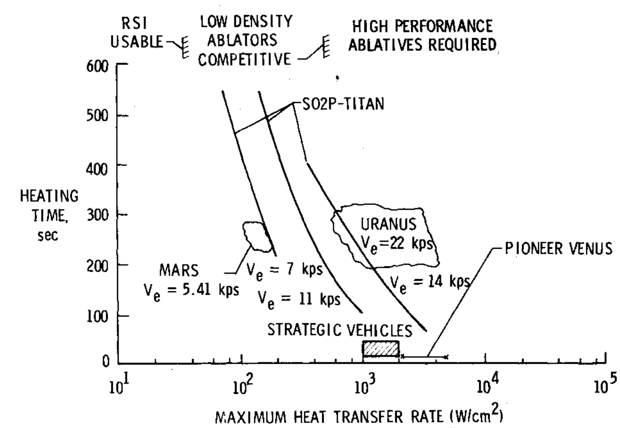


Fig. 17 Entry heating for planetary aerocapture vehicles.[60]

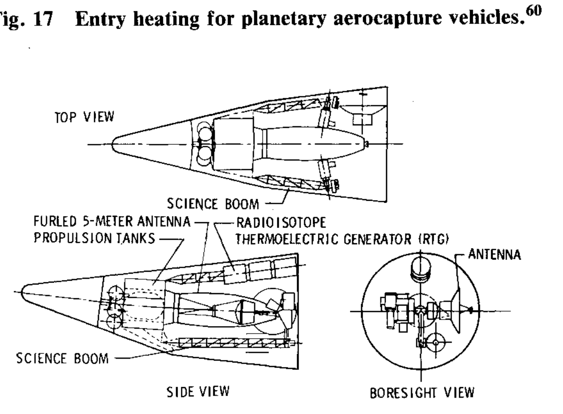


Fig. 18 Saturn Orbiter Dual Probe (S02P) spacecraft in interplanetary and entry configuration.[60]

allow the deployment of spacecraft subsystems during interplanetary transit, but this introduces questions about the integrity of the thermal protection system during the atmospheric pass. In any event, the use of aeroassist significantly complicates spacecraft design.

On the other hand, the use of aeroassist results in large payload gains. For the S02P mission just described, an aeroassisted vehicle can place approximately twice as much payload into Saturn orbit as an all-propulsive vehicle.[55] Furthermore, aeroassist has been shown to yield similar payload gains for missions to other planets.

Figure 20 shows aerocapture payload gains for the Venus Orbiting Imaging Radar (VOIR) and Mars Sample Return (MSR) missions.[55] The results shown in Fig. 20 were calculated assuming that the launch vehicle was the Shuttle with a two-stage Inertial Upper Stage (IUS). As shown by the dashed horizontal lines, the VOIR mission is just barely possible with an all-propulsive spacecraft in 1984. Otherwise, aeroassist is required to meet the science payload requirements of either mission.

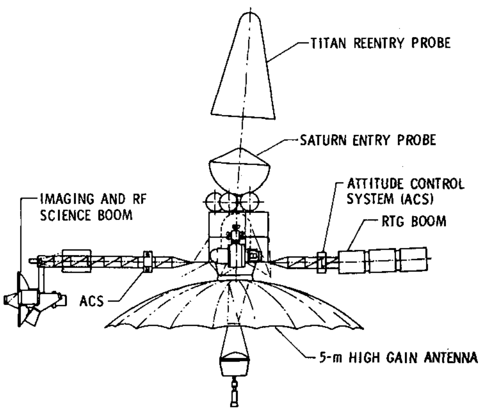


Fig. 19 S02P spacecraft in fully deployed configuration.[60]

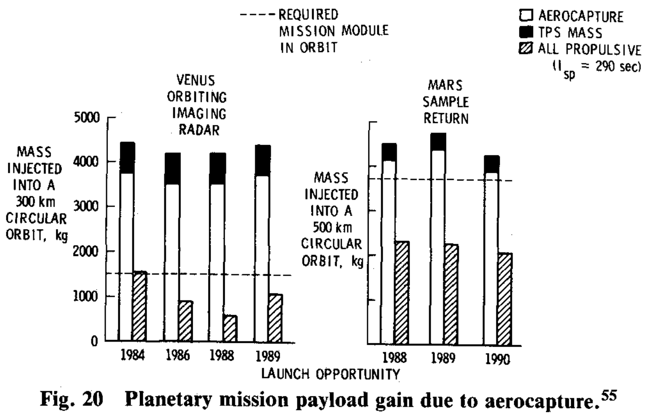


Fig. 20 Planetary mission payload gain due to aerocapture.[55]

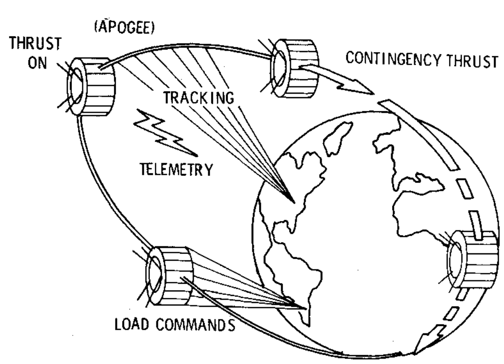


Fig. 21 Use of multipass aerobraking for the Atmospheric Explorer mission.[48]

Multipass Aerobraking

The second type of aeroassisted planetary mission that has been studied involves an initial propulsive maneuver that places the spacecraft in a highly eccentric orbit about the target planet followed by many passes through the outer fringes of the planetary atmosphere.[54,55,63-65] Each atmospheric pass produces a small velocity decrement and a consequent lowering of the orbit apoapses, eventually producing a circularization of the orbit. Small rocket burns, carried out at apoapsis, maintain the periapsis altitude in a range where aerodynamic heating and loads are acceptably low. Since many atmospheric passes are carried out, it is possible to design such a mission to be "fail-safe." The initial periapsis altitude may be set high enough to preclude excessive heating and tracking data obtained during the early high-altitude orbits can be used to provide estimates of atmospheric density even for a previously unknown atmosphere. Then the periapsis altitude may be lowered gradually, while onboard measurements of deceleration and spacecraft surface temperatures are made. Following each periapsis pass, these data may be used to determine the subsequent apoapsis rocket burns required.

This technique of orbit circularization was demonstrated by the Atmospheric Explorer mission flown to investigate the outer reaches of the Earth's atmosphere.[48] Figure 21 presents a schematic representation of this mission and shows how, after each perigee pass, new orbital parameters were calculated and commands sent to the spacecraft to carry out the required propulsive maneuver at apogee. Following the rocket burn at apogee, tracking data were used to determine an updated orbit and, if necessary, a contingency thrusting maneuver was carried out to produce the desired perigee. In Fig. 22, actual flight histories of apogee and perigee altitude are presented to show how the technique was successful in lowering the apogee altitude from 4.3×10^3 km to 2×10^3 km and hence circularizing the orbit.

While the application of multipass aerobraking to planetary missions involves the same basic principles illustrated in Figs. 21 and 22, the much longer communication distances (and times) involved with planetary missions place a lower limit on orbital periods for which the vehicle can be commanded from Earth. Also, deployable drag brakes must be used in order to produce significant payload gains as compared to an all-propulsive mission. As indicated in Fig. 21, the Atmospheric Explorer spacecraft was not specially configured to take advantage of aerobraking. For that mission, the perigee altitude

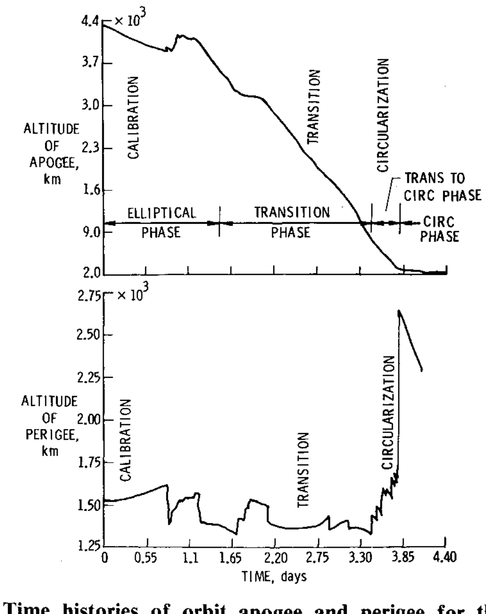


Fig. 22 Time histories of orbit apogee and perigee for the Atmospheric Explorer mission.[48]

was kept high enough that aerodynamic heating was insignificant. If such an approach were employed for a planetary mission, the time required to achieve orbit circularization would be unacceptably long. When vehicles are specially configured to take advantage of aerobraking, however, significant payload gains can result.

An aerobraking spacecraft was proposed for the Venus Orbiting Imaging Radar (VOIR) mission. In this instance, the basic spacecraft, which was originally designed for an all-propulsive mission, was modified by the addition of two deployable drag brakes. The motivation for this modification was that the baseline (all-propulsive) spacecraft had become too heavy to be launched by the Shuttle and a two-stage IUS. Rather than reduce the mission science payload, it was proposed to reduce the vehicle launch mass by replacing part of the propulsion capability with aerobraking. The resulting reduction, which brings the mission within the Shuttle IUS capability, in shown in Fig. 23. Note that while the aerobraking and orbit circularization propulsion add a small amount of mass, this is more than offset by the large reduction in the propulsion needed for orbit insertion. One of the advantages of multipass aerobraking is that the entire spacecraft need not be completely enclosed in an aeroshell. Hence, many of the design complexities mentioned for the previously discussed aerocapture vehicles are avoided.

All in all, multipass aerobraking is feasible within the present state of technology. However, advanced technologies could make it even more attractive. Onboard, adaptive guidance and control algorithms could remove the present limitation on minimum orbital period and improved rarefied flow analysis techniques (such as direct-simulation Monte Carlo methods) would allow the vehicles to be confidently designed for lower periapsis altitude where the flow is in the transitional regime rather than free molecule regime. These low-altitude, short-period final orbits would significantly increase the potential payload gains.

Orbital Transfer Vehicles

The most recently studied application of aeroassist technology is directed toward increasing the payload of orbital transfer vehicles. As pointed out previously, interest in AOTVs began in the early 1970s and was recently given a significant boost by activities related to the OTV concept Definition Studies[84,89] in 1980. Since 1981, there has been a veritable flood of publications. Except for some fairly recent moderate and high *L/D* concepts that will be discussed subsequently, the most widely studied AOTV concepts are illustrated in Fig. 24.

The Aerobraking Tug, studied in the early 1970s, was an aeroassisted version of the Space Tug, which was the originally proposed upper stage for the Space Transportation System. The Aeromaneuvering Orbit-to-Orbit Shuttle (AMOOS) was a lifting descendant of the Aerobraking Tug and was studied intensively in the middle and later 1970s. The lifting brake and the inflatable ballute were developed as part of the Orbital Transfer Vehicle Concept Definition Studies. While these four vehicle concepts represent radically different approaches, they have all been shown to provide significant payload gains over all-propulsive OTVs and it is this fact that accounts for the present high interest in this type of vehicle. While the OTV concepts studied to date are distinctly different from the previously discussed synergetic plane change and planetary vehicles, there are distinct similarities in mission requirements. Like the synergetic plane change missions, most OTV applications involve significant orbital plane changes. For instance, the widely studied geosynchronous-to-Shuttle-orbit transfer requires a 28.5 deg plane change. Like a typical aeroassisted planetary vehicle, the OTV returning from GEO to LEO possesses significant excess velocity. Hence, much of what has been discussed for synergetic plane change and planetary vehicles also applies to OTVs.

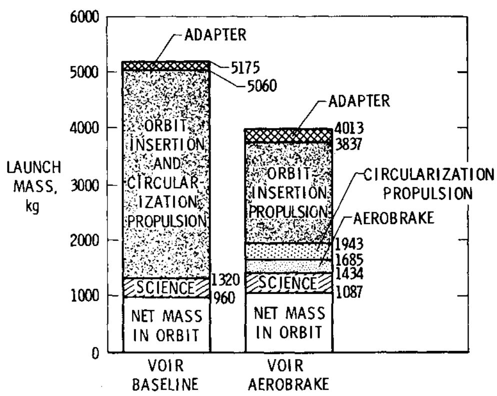


Fig. 23 Reduction in VOIR launch mass due to aerobraking.[55]

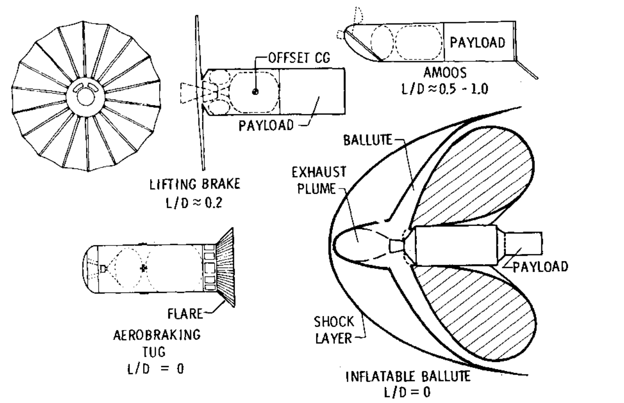


Fig. 24 Aeroassisted OTVs: Aerobraking Tug,[70,71], AMOOS,[79,81] lifting brake,[89] ballute.[84]

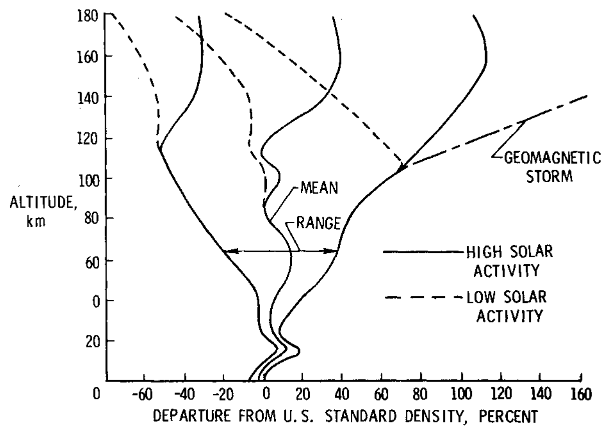


Fig. 25 Atmospheric density dispersions based on the 1962 U.S. Standard Atmosphere, summer.

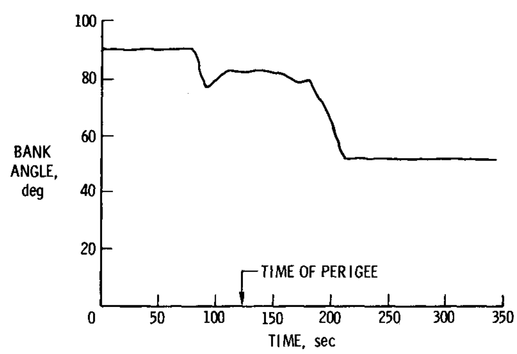


Fig. 26 Bank angle time history for AMOOS.[78]

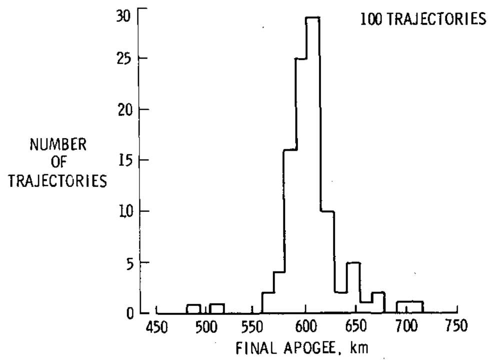


Fig. 27 Histogram illustrating success of AMOOS guidance law (nominal apogee = 600 km).[78]

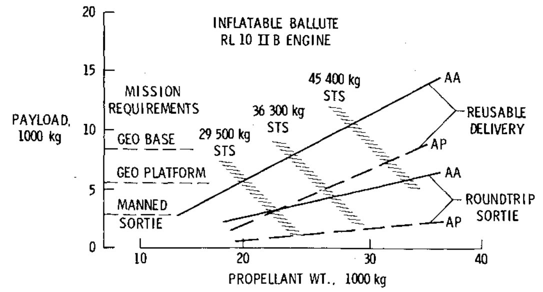


Fig. 28 OTV payload gains due to aeroassist.[84]

Aerobraking Tug

As shown in Fig. 24, the aerobraking Tug[70,71] was a nonlifting configuration with a cylindrical body, an ellipsoidal nose, and an aerodynamically stabilizing, drag-producing flare. The vehicle was originally proposed as a multipass aerobraker and various flare geometries and mission scenarios were studied. It was found that the maximum geosynchronous return payload was obtained with a relatively small flare (similar to that shown in Fig. 24) and a mission profile that involved approximately 30 passes through the Earth's upper atmosphere. This approach, which allowed the use of a reradiative metallic heat shield, maximized the payload but had serious operational drawbacks. For instance, such a GEO-to-LEO transfer required 4-7 days. In an effort to gain operational flexibility, a fewer number of passes, deeper into the Earth's atmosphere, were considered. This led to the development of a two-pass configuration that could complete the GEO-to-LEO transfer in 1 day, but required a larger flare and an ablative heat shield and had a reduced payload capability. While the Aerobraking Tug could carry considerably more payload than a comparable all-propulsive vehicle, it had several serious drawbacks. As has already been mentioned, the 30-pass vehicle suffered from a lack of operational flexibility and the 2-pass vehicle required refurbishment of its ablative heat shield after each mission. Perhaps the vehicle's most serious shortcoming, however, was its lack of control authority. There was no way to modulate drag in order to compensate for unpredictable variations in the density of the Earth's upper atmosphere. The magnitude of atmospheric dispersions that must be dealt with is indicated in Fig. 25, which presents estimates of the mean and range of density over an 11 year period expressed as a relative difference (percent) from the 1962 Standard Atmosphere. The mean constitutes a systematic variation that is predictable for a given month and solar activity. The range constitutes a random variability that is not predictable. The geomagnetic storm variation is for an extreme event that occurs once or twice per solar cycle and is not predictable a few days in advance. It usually occurs during periods of medium or high solar activity and the effects do not propagate into the lower atmosphere. Since the Aerobraking Tug achieves most of the required velocity decrement in the altitude range of 70-90 km, the ability to compensate for density variations of ±50% is required. Since there was no way to modulate drag for the Aerobraking Tug, the only safe mission scenario for this vehicle involved initial high-altitude passes to "sample" the atmosphere followed by deep aerobraking passes to achieve the required velocity decrement. This, of course, severely decreased the vehicle's operational flexibility. The requirement to accommodate unpredictable density variations, such as those shown in Fig. 25, is one of the principal design drivers for an aeroassisted OTV and is one of the main motivations for considering lifting vehicles such as the AMOOS.

AMOOS

The Aeromaneuvering Orbit-to-Orbit Shuttle (AMOOS),[73,75,76,79,81] illustrated in Fig. 24, was a direct descendent of the Aeromaneuvering Tug. It is also cylindrical, but has a slightly elliptical cross section and a raked-off nose and, at moderate angles of attack, is capable of hypersonic L/D of 0.5-1.0. Through bank angle modulation, this lifting capability provides adequate trajectory shaping to respond to density variations such as shown in Fig. 25. The aerobraking maneuver for the AMOOS is very similar to that described previously for planetary aerocapture vehicles (see Fig. 14). Several investigators studied the AMOOS in the 1970-1980 time period and concluded that a control system can be designed that will allow the vehicle to respond successfully to unexpected atmospheric density variations.[77,78,80] Typical results obtained by Rehder[78] are presented in Figs. 26 and 27.

Rehder considered a simple linear feedback control system that commanded the bank angle according to the guidance law,

$$\phi_B = \phi_{B_{nom}} + K(E_{act} - E_{nom})$$

where $\phi_{B_{nom}}$ was a constant of 90 deg, E_{act} and E_{nom} are, respectively, the actual and desired vehicle energies, and K is a gain constant. Rehder actually designed successful control laws by using both simple heuristic procedures and modern control theory. A sample bank-angle history for an AMOOS GEO-to-LEO transfer maneuver is presented in Fig. 26. Note that the bank angle is between 80 and 90 deg throughout half the atmospheric pass and then is reduced to 60 deg to produce enough positive lift to execute the skip-out maneuver that sends the vehicle up to its new apogee altitude. Hence, AMOOS not only achieves the required velocity decrement, but also executes a synergetic turn. In fact, the AMOOS vehicle obtains 7 deg of the 28.5-deg change plane required for a GEO-LEO transfer during the atmospheric pass. The remaining 21.5 deg is produced propulsively as part of the deorbit burn at GEO. To test the ability of his control law to compensate for atmospheric variations, Rehder carried out 100 simulated aeroassisted trajectories in which a statistical model of the atmosphere was used to impose random atmospheric variations. The results are presenteed in Fig. 27. The target apogee was 600 km and any apogee between 500 and 720 km was deemed acceptable. Figure 27 shows that the guidance law is successful even for extreme density variations. Not only was the AMOOS capable of successfully performing the aeroassist maneuver, it also had impressive payload capability. The LEO-GEO round-trip payload for AMOOS was approximately twice that of a comparable all propulsive OTV; however, the vehicle also had drawbacks.

With its relatively low L/D and relatively high M/C_DA, AMOOS was required to descend deep into the atmosphere to achieve the aerodynamic forces required for the aeroassist maneuver. Hence, aerodynamic heating for this vehicle is relatively high and an ablative heat shield is required. Hence, the AMOOS suffers from many of the design problems discussed previously for the planetary aerocapture vehicles.

The payload and all vehicle systems must be enclosed within a protective aeroshell and the ablative heat shield must be refurbished after each mission—hence, operational utility and flexibility are reduced. Also, since the vehicle requires a special protective aeroshell, it cannot be part of an evolutionary OTV development that starts with a state-of-the-art all-propulsive vehicle and evolves to greater payload capability by the addition of an aeroassist "kit." This means that a project manager must be willing to pay the extra price (both in dollars and in design risk) associated with aeroassist in order to gain an increased payload capability that might not be needed for many years. In the mid to late 1970s, NASA project managers were not willing to pay this price.

Lifting Brake and Inflatable Ballute

Two AOTV concepts that have been the subject of much recent study are the lifting brake[82,89] and the inflatable ballute,[84-86,88] both of which were developed as part of the OTV Concept Definition Studies. Interestingly, the initial guidelines for these studies called for the baseline vehicle to be all-propulsive. However, such large payload gains were shown for aeroassist that, by the conclusion of the studies, both contractors proposed aeroassisted vehicles as the baseline design. Figure 28 illustrates the payload gains that result from employing aeroassist. This figure presents results for the ballute, but both vehicles are predicted to have comparable payload capabilities. The labels AB and AP refer to aeroassist and all-propulsive vehicles, respectively. Payload capabilities are shown for both a round-trip sortie mission (where payload is both carried to and returned from a high Earth orbit) and a delivery-only mission in which the OTV returns empty and is reused. The shaded bands indicate the OTV capability when launched by Space Transportation Systems (STS) ranging from the present Shuttle (29,500 kg to LEO) to heavy-lift derivative vehicles (45,400 kg to LEO). Note that when the present Shuttle is used, aeroassist more than doubles the payload capability for both delivery and round-trip missions and enables the manned sortie and GEO platform missions that are not feasible with comparable all-propulsive OTVs.

The inflatable ballute surrounds the basic OTV and, when inflated, produces an ellipsoidal nose shape with the OTV rocket nozzle at its apex. As originally proposed, this vehicle used thrust modulation to achieve a wide range of drag coefficient. During the atmospheric pass, the rocket engine is run at a reduced thrust level and fires forward, producing a thick shock layer with a sizable separated region near the rocket nozzle. This is illustrated schematically in Fig. 24. Drag modulation is achieved by varying the thrust level of the rocket engine. Figure 29 presents wind tunnel test data, obtained by W.C. Woods at the NASA Langley Research Center, showing how the drag coefficient varies with the jet momentum flux ratio.

When the engine is run at tank-head idle (momentum flux ratio less than ≈ 0.003), the drag is only slightly less than for the non-thrust case. When the thrust is increased to the pumped idle case (momentum flux ratio ≈ 0.02), the drag is decreased by an order of magnitude. This ability to modulate drag by changing the engine thrust level could produce the necessary control authority for this vehicle. Unfortunately, separated shock layers such as those produced by the forward-firing rocket may become unstable under certain flow conditions (the shaded region in Fig. 29 indicates such an area of unsteadiness) and their characteristics can be difficult to predict. Because of this, the baseline ballute AOTV concept now achieves drag modulations by changing internal pressure rather than by using the forward-firing rocket plume. Whereas rocket thrust modulation can produce a nearly 10 to 1 change in drag coefficient, internal pressure changes will produce a change of only about 3 to 1. This reduction in control authority places greater demands on the control system algorithm, as will be discussed below. The idea of using a ballute to produce a low m/C_DA OTV is not new. A towed ballute was considered for the Aerobraking Tug. However, the idea of converting a basic all-propulsive OTV into an aeroassisted OTV by inflating a compactly stowed, lightweight ballute to surround the core vehicle is new and promising. The ballute AOTV has one great virtue: studies consistently show it to be among the lightest of AOTV concepts.

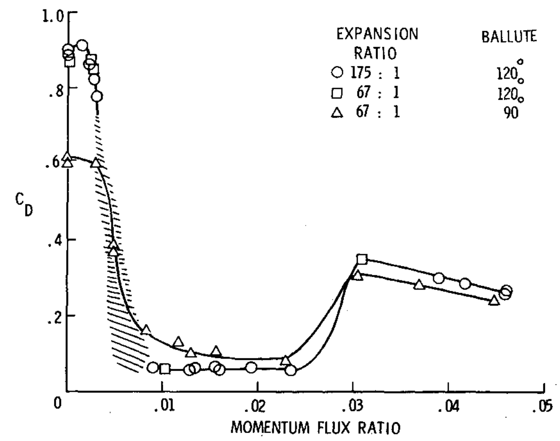


Fig. 29 Variation of drag coefficient with rocket plume momentum flux for ballute OTV.

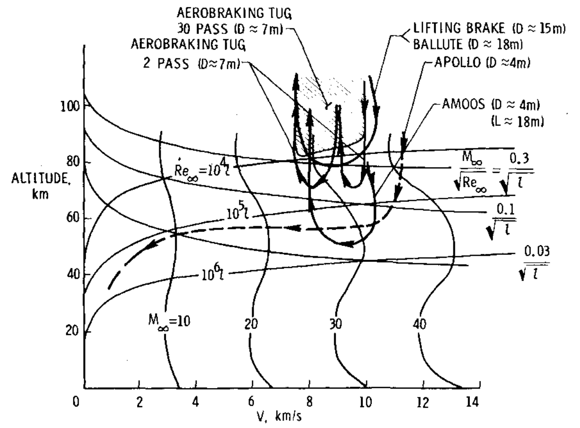


Fig. 30 Flight regimes for aeroassisted OTVs.

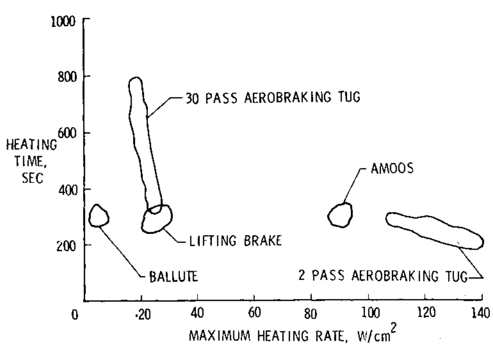


Fig. 31 Aerodynamic heating for aeroassisted OTVs.

The lifting brake, which was also developed as part of the OTV Concept Definition Studies, is a very promising combination of state-of-the-art technologies. Aerodynamically, the vehicle is very similar to the Apollo command module for which there is a wealth of flight experience. Like the Apollo Earth return configuration, the lifting brake is blunt with a large effective nose radius, has an offset center of gravity to produce an L/D of approximately 0.25, and achieves trajectory shaping through roll modulation during the atmospheric pass. The original lifting brake concept was a basic, all-propulsive OTV with a deployable drag brake that was structurally similar to a wrapped-rib antenna. The brake was packaged in an annular housing surrounding the rocket nozzle. It was deployed for the aerodynamic braking maneuver and was then retracted and was available for reuse. The combination of large drag area, large effective nose radius, and low weight (and hence low ballistic coefficient) allows the aerobraking maneuver to be carried out at high altitudes with convective heating rates low enough to permit the use of relatively low-temperature (<1700K) materials (such as ceramic-fiber cloth) for the surface of the drag brake. Unlike the ballute, the baseline lifting brake OTV was a reusable vehicle, whereas the ballute was discarded after the atmospheric pass and replaced as part of the vehicle's refurbishment between missions. Even though the ballute and the lifting brake employ very different design approaches, they both represent a trend away from high L/D toward low L/D and low m/C_DA. This is illustated in Fig. 30 where typical aeroassist OTV trajectories are presented along with contours of Mach number, Reynolds number, and viscous interaction parameter $M_\infty/\sqrt{Re_\infty}$. Note that the AMOOS, with its relatively high L/D and m/C_DA penetrates much deeper into the atmosphere than either the ballute or the lifting brake. In fact, by employing innovative design approaches, both the lifting brake and ballute obtain the advantages of a high-altitude flight regime (comparable to that of the 30-pas Aerobraking Tug) while retaining the operational flexibility of a one- or two-pass aeroassist maneuver. This high-altitude flight regime has two significant aerothermodynamic implications—low heating and rarefied flow.

The aerodynamic heating environment for the various aeroassisted OTVs discussed thus far is shown in Fig. 31 where the significantly lower heating rates of the high-altitude vehicles (lifting brake, 30-pass Aerobraking Tug, ballute) are obvious. It is also interesting to contrast the OTV heating environments shown in Fig. 31 with those discussed previously for synergetic plane-change vehicles and planetary aerocapture vehicles. From such a comparison, it is seen that the high-altitude OTV vehicles and the downstream regions of the synergetic plane-change vehicles have similar heating rates, while the planetary aerocapture vehicles and the nose and leading-edge regions of the synergetic plane-change vehicles experience heating rates that are significantly higher. Heating times are comparable for OTVs and planetary aerocapture vehicles and are much longer (by factors of 2-5) for synergetic plane change vehicles. The results presented in Fig. 31 are for convective heating to a fully catalytic surface. However, recent studies have shown finite wall catalycity and nonequilibrium radiative heating to be of great potential importance in determining AOTV heating levels. These studies are discussed in the following section of this paper.

The rarefied flow environment of the ballute and lifting brake are illustrated by the $M_\infty/\sqrt{Re_\infty}$ contours in Fig. 31. If diameter is taken as the characteristic dimension, ℓ, for the ballute and lifting brake and overall length is used for the AMOOS, it is seen that the viscous interaction parameter is an order of magnitude larger for the high-altitude vehicles and is large enough (≈ 0.06) to indicate significant rarefied flow effects even at perigee. This trend toward rarefied flight conditions has significant research implications, since low-density flowfields are difficult to analyze and simulate in ground test facilities.

Recent AOTV Studies

As shown by the Bibliography, the AOTV literature has more than tripled in size in the past three years. As a result of this intensive study, some significant new vehicle concepts have been put forward and some significant new technology issues have been identified.

Many recent studies[120-124,140,142] have provided more insight into the characteristics and capabilities of the ballute, lifting

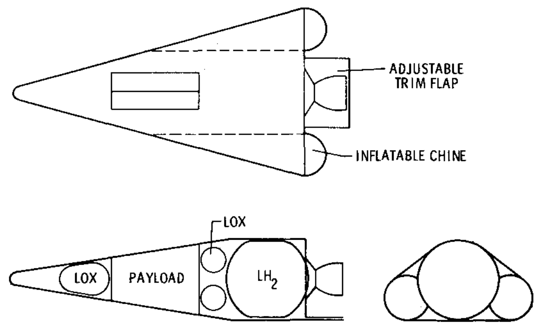


Fig. 32 AOTV with high L/D capability.[124]

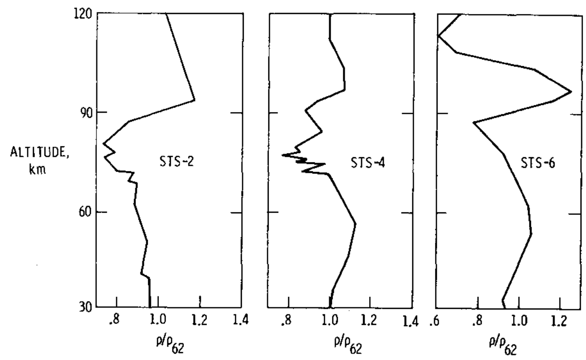


Fig. 33 Shuttle-derived densities compared to the 1962 U.S. Standard Atmosphere.[124]

brake, and AMOOS and some significant refinements have resulted. As mentioned previously, the internal pressure variation has now been adopted as the baseline approach to providing drag modulation for the ballute.[142] Whereas the early lifting brake concepts featured an aerosurface that was a nearly flat disk (see Fig. 24), more recent concepts[142] have tended toward a wide-angle cone configuration with rounded outer edges to reduce edge heating. Experimental studies[104,120] have underlined the importance of edge heating and afterbody heating (which significantly reduces the usable angle-of-attack range and/or afterbody length) for these configurations.

Most prominent among the new vehicle concepts are moderate-L/D biconic configurations[94,107,114] and high-L/D delta wing configurations[115,117,121-124] with greater orbital plane change capabilities. The biconic vehicles have configurations similar to the previously discussed planetary aerocapture concepts (see Figs. 12, 16, and 18) and benefit from an extensive aerothermodynamic data base.[95,96,102,103,139] Recent studies have shown advantages for short, low-fineness-ratio moderate-L/D bioconic vehicles that utilize external expendable drop tanks for delivery-only missions. In this application, the AOTV returning to low-Earth orbit contains the OTV engines and the avionics subsystems. It is essentially a high-orbit analog to the propulsion/avionics module used on Shuttle-derived heavy-lift launch vehicles, i.e., its main function is to facilitate the reuse of the expensive engines and avionics components. The high-L/D delta wing vehicles are typified by the configuration shown in Fig. 32. These vehicles have hypersonic L/D in the range of 1.8-2.0 and are, therefore, capable of producing relatively large changes in orbit inclination through aerodynamic turns. For example, Talay et al.[124] considered the GEO return payload and aerodynamic plane-change capabilities of representative low, moderate, and high-L/D AOTVs. The low-L/D vehicle was a lifting brake ($L/D \approx 0.2$), the moderate-L/D vehicle was an AMOOS configuration ($L/D \approx 0.8$), and the high-L/D vehicle was the configuration shown in Fig. 32 ($L/D \approx 1.9$). While the high-L/D vehicle was capable of producing much larger aerodynamic plane changes (26 deg compared to 10 deg for the AMOOS and 2 deg for the lifting brake), its return payload capability was only slightly greater than that of the other vehicles (7705 kg compared to 7436 kg for the AMOOS and 7181 kg for the lifting brake). And this increased plane-change capability was obtained at the cost of significantly higher aerodynamic heating and structural weight (as illustrated earlier in Figs. 4 and 5). Hence, high-L/D vehicles do not appear promising for routine cargo missions. However, they may be advantageous for missions (such as survelilance missions) that benefit greatly from aerodynamic maneuvering capability.

Recently, several investigators[100,119,130-132] have studied AOTV concepts that are intended to be assembled and based at a space station. Hence, these vehicles are freed of the constraint that they must fit into the Space Shuttle cargo bay. As a result, these vehicles can employ large, lightweight fixed aero surfaces rather than deployable concepts. The vehicles studied thus far are of the lifting brake type with wide-angle, axisymmetric cone or raked-off cone geometries and are predicted to have light structural weights and high performance.

Two technical issues that have recently received intensive study are guidance and control system design and nonequilibrium aerodynamic heating. The central issue in guidance and control system design is the amount of aerodynamic control authority required to deal with unpredictable variations in atmospheric density. Previous studies have considered density variation that could be characterized as multiples of the 1962 Standard Atmosphere density profile (see, for instance, the discussion accompanying Figs. 25-27). However, data from recent Shuttle flights have shown the existence of so-called "potholes in the sky" as illustrated in Fig. 33 where ratios of the derived densities to the 1962 Standard Atmosphere are presented for Shuttle flights 2, 4, and 6. These large density excursions, which occur over relatively narrow altitude ranges, pose a significant challenge to AOTV flight control systems. The question is, how much drag variation (in the case of $L/D=0$ vehicles such as the ballute) or L/D capability (in the case of lifting vehicles) is required to compensate for these density variations? In spite of the large number of recent studies,[109-113,122,138,142] there is no consensus answer to this question. Present indications are that fairly sophisticated adaptive control laws may be required for vehicles such as the ballute (with its relatively small drag modulation capability) and the lifting brake (with L/D on the order of 0.2).

The central issue in aerodynamic heating has to do with the effects of nonequilibrium on both convective and radiative heating. Until recently, most analyses assumed chemical and thermodynamic equilibrium both in the shock layer and at the vehicle surface. Now there is strong evidence[97,99,130] that finite wall catalycity can significantly reduce the AOTV convective heating rates, perhaps to values as low as one-half their equilibrium values. The question here is whether or not thermal protection materials can be developed that will maintain low levels of catalycity under actual flight conditions for multiple missions. There is also strong evidence that nonequilibrium shock-layer radiation cannot be ignored.[99-101,108,118,131] These investigations show that analyses can be developed that are reasonably consistent with shock-tube and flight data and predict significant nonequilibrium radiation for the AOTV flight regime. If this nonequilibrium radiation occurs, it could increase peak heating rates up to twice the equilibrium convective values, which would effectively prohibit the use of many low-temperature heat shield materials. On the other hand, other investigators[130,136] examine the same data and argue persuasively that nonequilibrium radiation may not be significant. This may be the most important single question for aeroassisted OTVs and it is possible that a definitive answer can be obtained only through a carefully designed flight experiment.

Conclusions

This survey of aeroassisted orbit transfer has revealed a rich literature and well-developed technology base that clearly shows high-performance gains in comparison with comparable all-propulsive vehicles. In some areas such as multipass aerobraking, it appears that the technology is essentially ready. What is needed is a mission. In most areas, however, it appears that further work is needed to bring the design risk down to an acceptable level. One thing is clear, however. There are enormous opportunities for mission enhancement through aeroassist and there are some intriguing new ideas that are just now coming under study. A strong research effort in this area certainly seems justified. If this research is pursued virgorously, practial aeroassisted vehicles should be operational within the next 10 years.

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In this paper, relevant publications are presented in a bibliography rather than a conventional reference list. So that the reader may more easily follow the historical evolution of the technology, items are listed in chronological order for each vehicle class.

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