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Entry Vehicles for Space Programs

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Introduction

ENTRY vehicles have been in operation for over a decade, and it is now possible to identify some trends which have developed in operational systems. In the case of landing on a planet that has a sensible atmosphere, advantage can be taken of the atmospheric drag to slow the entry of the landing vehicle, in accordance with Allen and Eggers' classic study.[1] The control of the aerodynamic deceleration loads and the protection of the payload from induced aerodynamic heating constitute the primary problems for the entry vehicle designer. Our objective is to review the major aspects of entry vehicles in terms of past and presently operational vehicles, and to indicate trends in terms of advanced mission requirements and studies.

Entry Mission Profiles and Vehicle Configurations

Operational requirements for recovery, propulsion system limitations, guidance system capabilities, and acceleration load limits constrain the flight path during entry.[1–7] These constraints, coupled with mission objectives, have led to the use of ballistic or near-ballistic trajectories for all existing operational spacecraft. The entry trajectory is first determined by the velocity V_E and angle γ_E at which the vehicle enters the atmosphere. The entry of Earth orbital vehicles is normally initiated by a propulsive maneuver which decreases the altitude of perigee of the orbit to the point of intersection of the trajectory with the atmosphere. This intersection can be controlled easily, and γ_E has normally been small, which

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allows a minimization of loads. For missions involving transfer between celestial bodies, such as lunar-return or planetary missions, γ_E is determined by guidance during midcourse maneuvers. The accuracy of the guidance system and the requirements for capture and for specific maximum inertial loads influence the choice of γ_E and the accuracy with which it can be determined. For missions involving transfer between celestial bodies, a variety of entry trajectories can occur for a given vehicle. Let us review mission profiles and configuration considerations for the entry vehicles in Fig. 1.

Unmanned Orbital—Biosatellite

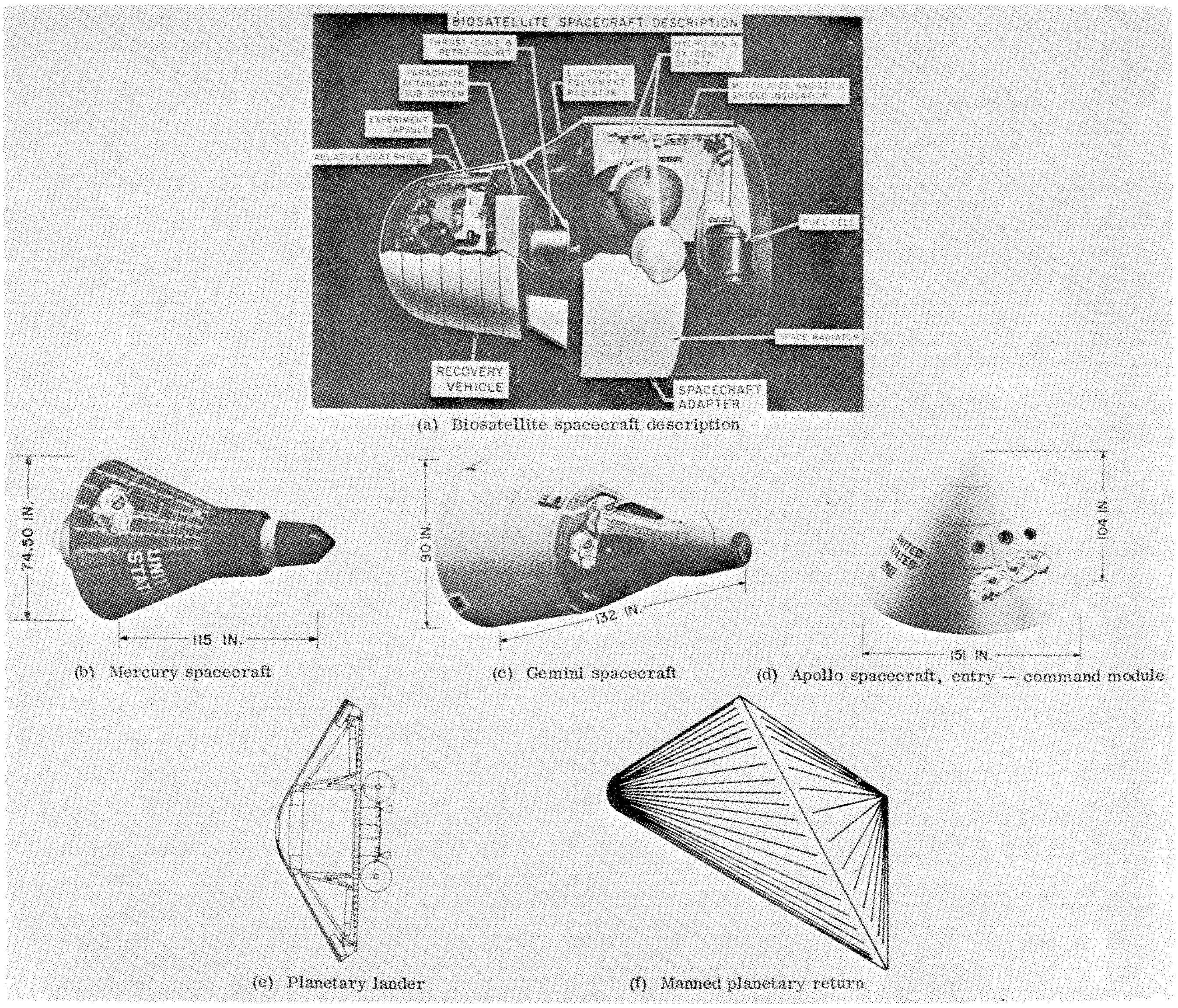
Unmanned entry vehicles which have been used to return scientific experiment data from Earth orbit include the Biosatellite (Fig. 1a). One of the objectives of this program is to determine the effect of the space environment (particularly weightlessness) on biological specimens. A key element in the design of the Biosatellite missions is the requirement of maximum experiment acceleration of 10^{-5} g for 95% of the time in orbit. Because of the resulting restrictions on angular rates, the uncertainties in initial orbit due to booster variations, and the orbit decay during the mission, a circular target orbit of 200 naut miles was chosen. This orbit imposed strong weight restrictions on the total spacecraft system.

The recovery dispersion limit of 400 naut miles is based on the requirement of air retrieval for quick transporation to Hawaii. The entry vehicle is placed in the proper attitude for deorbit by the spacecraft attitude-control system, with the use of two horizon sensors and a magnetic boom for attitude determination. The entry vehicle is spun up to approximately 60 rpm for gyroscopic stability during retrothrusting and then despun to about 10 rpm for entry. The thrust cone is ejected and, until parachute deployment, the entry vehicle is in free flight with the desired attitude being maintained through natural aerodynamic stability.

The aeroshell, a highly blunted 10° cone, orients and protects the payload during Earth entry and decelerates the payload such that the parachute can be deployed at high altitudes (approximately 50,000 ft) for aircraft acquisition and subsequent air snatch. For a maximum entry weight of 474 lb, this configuration attains proper orientation from an initial angle of attack of approximately 120° and decelerates to Mach 1 at approximately 50,000-ft altitude. This deorbit system was used on the "B" flight (3-day mission) in September 1967 and achieved a deorbit from a 160 naut mile orbit with impact less than 10 miles from the predicted impact point.

Manned Orbital and Lunar—Mercury, Gemini, and Apollo

Manned vehicles have progressed toward nonballistic trajectories. The Mercury capsule (Fig. 1b) had a lift-to-drag ratio (L/D) equal to zero and a ballistic coefficient β



(a) Biosatellite spacecraft description
(b) Mercury spacecraft
(c) Gemini spacecraft
(d) Apollo spacecraft, entry – command module
(e) Planetary lander
(f) Manned planetary return

Fig. 1 Entry vehicle configurations.

tailored to limit the entry load factor to a level compatible with human physiology. Control over the landing location was provided by timing the retromaneuver. The Gemini spacecraft (Fig. 1c) used an offset c.g. to achieve an L/D of approximately 0.1. This small L/D, coupled with an inertial guidance system, provided sufficient control of the entry trajectory to allow extremely accurate predictions of the landing site, but site selection was still dependent on the retromaneuver.

The technology of ballistic and near-ballistic orbital entry is approaching maturity. The most severe manned orbital entry trajectory for near-ballistic or ballistic vehicles is that which results from a minimum retromaneuver. Such entries are of long duration and result in high integrated heat loads. However, the associated heating rates are much lower than those associated with a nominal entry. The large effective radius of the Mercury-Gemini-type configuration minimizes the convective heating rate to the heat shield. Even for lunar-return velocities, the radiative heating to these blunt shapes is much lower than the convective rate.

The Apollo spacecraft (Fig. 1d), an aerodynamically symmetric body, is designed to provide an L/D of between 0.25 and 0.40 to satisfy two requirements: 1) the availability of negative lift provides assurance of capture after lunar-return at superorbital velocities with a higher overshoot boundary and 2) the maneuvering capability and the guidance system combine to allow arbitrary landing-site selection from 1500 to 2500 naut miles downrange from the entry interface location. Since the injection maneuver is executed several days prior to entry, entry maneuvering is necessary because the spacecraft must be able to avoid local bad weather conditions on a real time basis.

The maneuverable entry vehicle concept is so firmly entrenched, and the technology is so well proven that all future manned orbital entry vehicles will possess maneuvering capability. This tendency will probably result in the development of a true lifting body, or even a winged configuration, within a few spacecraft generations. Such spacecraft will have the aerodynamic and guidance capability to perform relatively steep entries using positive lift, and to fly at hypersonic velocities for extended periods of time during shallow prolonged entries, using negative lift, if necessary, to prevent skipping. The range of entry trajectories of these vehicles will be limited by the capability of the thermal protection system to sustain high heating rates, high heating loads, and combinations of these parameters. The task of defining the sequences of altitude-velocity combinations that are thermally permissible is one of great complexity, simply because the number of possibilities is large, and the task of displaying to the pilot the complex of alternatives open at each point in the trajectory will be very difficult.

Another factor in manned entry vehicle configuration selection is volumetric efficiency, since, in the first order, thermal protection system and structural weight are a direct function of vehicle area. The volumetric efficiencies of several entry vehicle configurations are compared in Fig. 2; the advantage of the Apollo type vehicle is obvious. Nevertheless, the

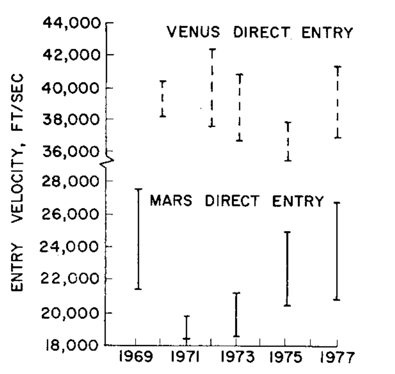


Fig. 2 Planetary entry velocities.

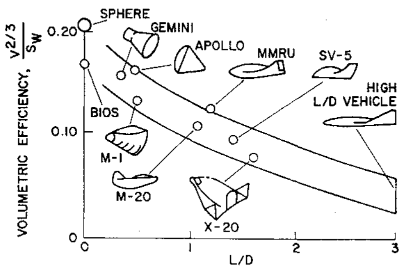


Fig. 3 Volumetric efficiency comparisons.

state of the art of thermal protection has advanced sufficiently to seriously consider the use of sharper-nosed vehicles for future manned Earth-orbit and lunar-return missions. An airplanelike, high-L/D configuration could be landed in a conventional manner at any desired location, thus minimizing recovery operations. As the basic cost of launching decreases with the development of more efficient launch vehicles, it is likely that a high-lift, truly flyable entry vehicle will be developed in the future for orbital missions.

Nuclear Fuel Abort—SNAP

The problem of radioisotope cask recovery leads to an extreme example of a highly indeterminate entry trajectory. Radioisotopes must be protected for atmospheric entry since aborted missions could result in entry of the fuel and may endanger human life. The SNAP-27 fuel cask used in the Apollo lunar landing mission may enter at orbital, suborbital, or lunar-return velocities during aborts initiated after trans-lunar injection. From Earth orbit, the entry angle may be as low as 0.06°, whereas at lunar-return velocities entry angles as high as 38° are likely, and 70° entries are possible.

Unmanned Planetary Lander—Mars, Venus

For direct-entry missions, the planetary entry velocity V_E is directly related to the approach velocity. Figure 3 shows V_E's for Mars and Venus. The velocity that is actually obtained in any particular mission depends on constraints such as launch energy, range safety conditions, relay or direct communication requirements, and lighting conditions, at the planet. On Mars direct-entry missions γ_E is constrained on the shallow side by the skipout limit which is of the order of 10°–15°, the exact value depending upon the atmospheric properties V_E and the aerodynamic characteristics of the entry vehicle. However, the constraint on γ_E must be set higher (to approximately 20°–30°) because of γ_E errors which result from uncertainties in aimpoint location and from errors in the deflection maneuver.

In out-of-orbit missions V_E is independent of the approach velocity and, therefore, is independent of the launch opportunity. Instead, the determining parameters for V_E are the orbit period and the manner in which the lander is separated from the orbiter. Typical values of out-of-orbit entry velocity are 14,000–16,000 ft/sec for Mars missions and 28,000–30,000 ft/sec for Venus missions. While a steep entry is typical for direct-entry missions, relatively shallow entry is the usual case in out-of-orbit missions.

Planetary Return

Manned planetary exploration missions will require the development of vehicles capable of entering the Earth atmosphere at velocities in excess of 40,000 ft/sec.[2–7] With present guidance system accuracies, vehicles of the Apollo class with with $L/D \approx 0.5$ can assure capture and physiologically acceptable inertial loads. Such vehicles are more probable choices for manned planetary missions than vehicles with $L/D > 1.5$ because thermal protection system technology for the latter is not well characterized. Since constant-altitude deceleration trajectories can result in extremely high heating rates ($\sim 10^4$ Btu/ft^2-sec), tailoring of the entry trajectory on the basis of

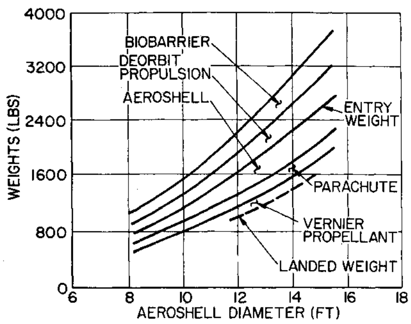


Fig. 4 Planetary out-of-orbit weight breakdown.

heating minimization has been studied (and shown to be feasible), but the analyses are complicated by the dependence of heating on vehicle configuration. One way to reduce V_E, as well as propulsion requirements, is to use swingby trip modes wherein a close approach to a third planet is made. However, the mission trajectories that alleviate the entry vehicle problem significantly extend the total mission time and thus significantly expand the requirements of other spacecraft systems, such as the life support systems.

Unmanned, Manned—Planetary

The problems of entry into the atmospheres of Mars and Venus present separate requirements which, after appropriate tradeoffs, result in surprisingly similar aerodynamic configurations. In the case of Mars entry, the basic problem is one of decelerating the vehicle sufficiently to allow for the tenuous nature of the atmosphere. For Venus entry, aerodynamic heating at the relatively high entry velocities (36,000–42,000 ft/sec) presents the major tradeoff parameter. As a result, the most significant vehicle requirement for ballistic entry is that it have a low β.

Guy[8] recently discussed the Mars deceleration requirements and the influence of the configuration upon the payload efficiency of the entry vehicle. A multistage deceleration system, consisting of aeroshell, aerodecelerator and retro-engine or impact attenuator may be used to maximize the landed payload within the constraints of the booster weight and shroud diameter limits. A typical planetary out-of-orbit weight breakdown is presented in Fig. 4. Since the vehicle is constrained in ballistic coefficient, the vehicle drag must be maximized. As a result, the aeroshell configurations have tended toward very blunt sphere cones with some consideration of the modified Apollo and tension shell shapes. Previous flight experience for blunted cones has been restricted to a cone angle of 52° or less, while cone angles of 60° or more provide a drag which is comparable to the segmented sphere (Apollo) or tension shell. Vehicle stability is also a consideration since the decision of whether or not to incorporate an attitude-control or rate-damping system must be made. In general, the blunt configurations have little dynamic damping, and measurements in ground tests are difficult. One configuration that is being developed is shown in Fig. 1e.

The Venus atmosphere is considerably more dense than that of Earth. As a result, low vehicle β is not necessary to achieve reasonable deployment conditions for aerodecelerators. The entry velocity of a Venus probe, however, is approximately 36,000–42,000 ft/sec and can produce considerable radiative heat transfer to the vehicle depending upon the vehicle β. In Ref. 9, however, it was reported that heating levels can be held to levels associated with Earth entry if a β less than approximately 25 lb/ft^2 and a cone angle less than 45° are used. Such a configuration is within the aerodynamic technological state of the art.

The effect of heat-shield ablation upon the vehicle dynamics has been investigated for small cone angles ($\theta_c = 10°$) with little bluntness but not for blunt-nose bodies. Since the technology for the projected Venus configuration is well advanced, the configuration tradeoffs reduce to an aeroshell optimization with perturbations within the range of the limiting parameters such as β, θ_c, and nose blunting.

Among the concepts being considered for Mars entry are: inflatable afterbodies for the basic aeroshell to provide for increased drag area and, thus, payload, while not exceeding booster shroud diameter limits[10]; modifications to the blunt sphere cone to provide lift and, thus, to increase the allowable ballistic coefficient and, therefore, payload weight.[10] The configurations being studied for manned planetary entry at Mars are similar to those previously discussed for unmanned planetary missions. The determining factor is the same: maximization of C_D. The opposite is true for Mars-return or Venus entries. For these cases, heating is a determining factor in vehicle configuration. Since manned vehicles are necessarily large, radiant heating becomes dominant for such entries if blunt vehicles are used. To avoid such heating, configurations such as Fig. 1f have been proposed for Mars return. Such sharp vehicles encounter convective heating problems due to the extremely high tip heating rates and shears and to possible turbulent heating to the conical forebody.

One result of a recent comparative study[11] of the cone and the Apollo shape for Mars return is shown in Fig. 5. This indicates that, when the uncertainties in heating technology are considered, no distinct advantage of either shape can be determined from a heating standpoint. The effects of vehicle shape on total vehicle weight, as a function of V_E, are shown[12] in Fig. 6. Again, up to velocities of 55,000 ft/sec, it is apparent that all geometries result in about the same total vehicle weight.

Continuing studies have attempted to define optimum configurations for entry vehicles from operational, heating, and other standpoints, and several new configurations have resulted.[13–15] However, unless new operational requirements, such as land recovery or refurbishment and reuse, can be identified, no obvious total advantage of any of these configurations over that of the Gemini-Apollo type entry vehicles has yet been shown, even for Mars-return entries. Thus, it is likely that at least the next one or two generations of manned entry vehicles for space programs will be of the same basic spherical segment-cone configuration as those used in the past.

System Design Considerations

Thermal Control Subsystem

The Biosatellite spacecraft has an environment control system (ECS) which consists of two circulating coolant loops coupled with an interloop heat exchanger. One loop provides coolant for temperature and humidity control of the payload capsule atmosphere and coolant for temperature control of the water and metabolic water storage tanks. The other loop

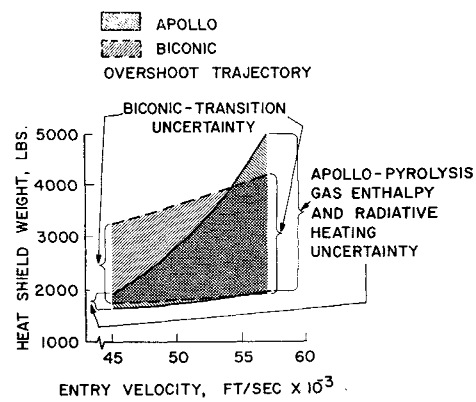


Fig. 5 Weight sensitivity for planetary return.

provides temperature control for the fuel-cell power source, cryogenic gases, electronic components, and the urine experiment package. A radiator for rejection of the spacecraft thermal energy and a regenerative heat exchanger are also parts of the second loop. The ECS vehicle thermal design loads a range from 400 to 800 Btu/hr for the 30-day mission and from 500 to 900 Btu/hr for the 21-day mission. Since the vehicle attitude during orbit is completely random, ECS must be capable of operating over a wide range of orbital thermal environments.

Through the Mercury, Gemini, and Apollo programs, increasingly more complex and sophisticated thermal control systems have evolved. The Apollo command module uses a combination of passive (insulation and thermal control coatings) and active systems to maintain a habitable environment throughout the Apollo mission from Earth-orbit insertion to lunar orbit and return. The primary active system consists of a continuous circulating mixture of water-glycol which provides a heat-transport-fluid loop. This flow is used to cool the cabin atmosphere, the pressure-suit circuit, the electronic equipment, and a portion of the potable water. It also serves as a heat source for the cabin, when required. All unwanted heat absorbed by the water-glycol is transported to the environmental control system space radiators located on the surface of the service module. Supplemental cooling takes place within the water-glycol evaporator where heat is rejected by the evaporation of waste water. The spacecraft also is provided with a completely independent, secondary coolant loop, which routes water-glycol to certain critical components that are absolutely necessary to complete a safe return to Earth. In addition, heaters are used to control specific components, such as the command module reaction control engines.

The selection and design of an internal thermal control system for a planetary entry vehicle are strongly influenced by external environments, mission duration, payload temperature requirements, power dissipation, and vehicle shape. Therefore, it is difficult to be completely general in discussing the thermal control of planetary vehicles. The methods of thermal control for the Mars and Venus landers are directly opposed, since the surface temperature of Mars may range from $-184°$ to $117°F$ with pressures ranging from 0 to 2 psia, while Venus surface temperatures can vary from 450° to 1050°F with pressures ranging from 10 to 120 atm. Therefore, systems for heating the payload for the Mars lander and removing thermal energy for the Venus lander are required.

A number of methods are available for providing the heat required to maintain Mars lander components at acceptable temperatures. For short stays (<10 days), a battery-powered system (e.g., insulated payload, heat of fusion thermal storage, and local electric heaters) would be practical. This approach is impractical for long stays, since the batteries and/or increased power supply capabilities become prohibitively heavy. For intermediate stays (~ 30 days), a fuel-cell power source may be desirable. The associated thermal-control methods are an active coolant circulation loop, electrical heaters, and insulation.

The most attractive method of thermal control for long-duration missions is the use of the thermal energy dissipated from the electrical power system when the power supply is a radioisotope thermoelectric generator (RTG). A liquid loop circulation system can be used to convect the thermal energy from the RTG unit to the payload as required to maintain the desired payload temperatures. On the surface of Mars, the excess thermal energy could be radiated directly from the outer surface of the RTG.

Since the external environmental temperatures of Venus are considerably above the desired operational temperatures of the equipment to be included within the lander, a means of cooling the payload must be devised. An expendable coolant system can provide the necessary cooling and has the advantage of very low power requirements, but such a system is limited by the amount of coolant which may be carried within the vehicle.

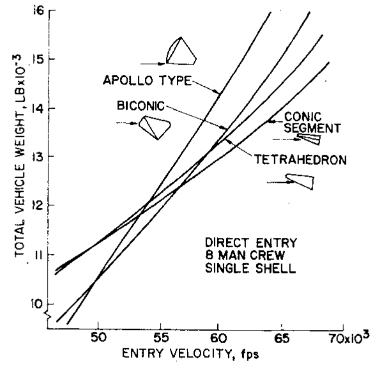


Fig. 6 Total vehicle weight comparison of several lifting geometries.

Detailed discussions of thermal control system design and analysis techniques are presented in Refs. 16–26.

Life Support System

In the Biosatellite, a controlled laboratory environment (i.e., 14.7 psia, O_2 and N_2 standard atmosphere) is maintained, within which the biological investigations are performed. The temperature and relative humidity of the air in the capsule is maintained at $75 \pm 5°F$ and 40–70%, respectively, to satisfy the biological requirements. A gas management assembly (GMA) is located in the payload capsule and controls the gaseous environment in the capsule to provide the controlled laboratory environment. The GMA controls the temperature and relative humidity of the air, the circulation and filtration of the atmosphere, the accumulation of trace gases and odors, the total pressure of the standard oxygen/nitrogen atmosphere, and the partial pressure of the oxygen. It also measures the total pressure, the oxygen partial pressure, the carbon dioxide partial pressure, the temperature, and the relative humidity. The gas management system receives its coolant supply from the ECS located in the adapter section. The oxygen and nitrogen makeup gases are supplied from cryogenic storage tanks in the adapter section.

The Gemini ECS was designed to support two men for a nominal 14-day flight with an atmospheric environment of 100% oxygen at 5.0 psia. The system consisted of a cabin circuit and a suit circuit. The cabin circuit was of the same design as that provided in the Mercury spacecraft, except that the heat exchanger was a liquid coolant fluid-to-gas design. The suit circuit contained the same processing and control components that were provided in the Mercury spacecraft suit circuit. The Apollo entry vehicle ECS provides a controlled environment for three astronauts during stays on the moon up to 14 days duration. This environment consists of a pressurized suit circuit for use during normal or emergency conditions and a pressurized shirt-sleeve cabin atmosphere (5 psia oxygen) used only when normal conditions exist. Metabolically, the system is responsible for supplying oxygen and hot and cold potable water as well as for removing CO_2, odors, water, and heat. Oxygen and potable water are supplied to the ECS from the electrical power system. The oxygen is stored at cryogenic temperatures; the potable water is a by-product of the fuel cell power system. The pressure-suit and cabin gases are processed for reuse by being routed through the suit circuit debris trap, the carbon dioxide odor-absorber filters, and the heat exchanger.

The primary penalty is generated by the atmosphere control system which must function for long periods of time, compensating for varying metabolic and equipment heat loads, resupplying the oxygen used, removing carbon dioxide and toxic substances, and controlling humidity. The storage of all food and potable water required by the crew and waste products is also a significant requirement. One requirement

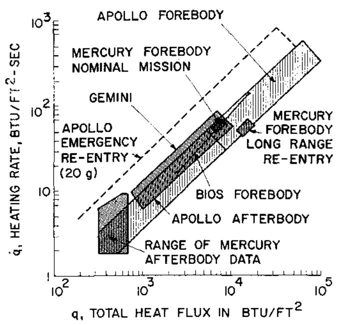


Fig. 7 Heat-transfer parameters.

peculiar to the crew which is more evident on the Apollo than on the Mercury and Gemini entry vehicles is the requirement for excess space in which the crew may exercise some freedom of movement. All these requirements impose significant penalties on the vehicle for longer mission durations.

Further information on the design of life control systems is presented in Refs. 27–31.

Crew Safety

Systems are frequently included in manned entry vehicles which are not required for normal crew operations exclusively to provide for crew safety. The launch pad abort system of the Gemini vehicle is a typical example. The Gemini crewmen sat in ejection seats behind large hatches. In case of an abort on the pad or during liftoff, the crew was ejected and parachuted a safe distance from the failing spacecraft and launch vehicle. The presence of a crew necessitates the incorporation of many systems and directly affects the design of other systems in manned entry vehicles. However, one of the most significant impacts on manned entry vehicle design is more subtle. Because a crew is carried, the reliability of the entry vehicle, and especially the reliability of intact recovery, must be extremely high. The present manned vehicles are designed to land on water, primarily to increase the reliability of intact recovery, since the structural and terminal deceleration systems requirements are minimized and since the probability of landing in the design environment is significantly increased. Failure of guidance and navigation systems, pilot errors, and times of mission abort are thus made somewhat less critical. All the systems in a manned vehicle which are considered necessary for crew survival are made redundant, and all single-point failures are eliminated. This is in sharp contrast to those systems in manned spacecraft which are not essential to crew survival, in which many single point failures are allowed. The most dramatic example of the impact of crew safety on entry vehicle design, testing, and operation is the extensive changes in the Apollo command module which resulted from the launch-pad fire which took the lives of three crewmen. The crew hatch has been completely redesigned. Interior materials that are most flammable in an oxygen-rich environment have been replaced. In addition, future manned entry vehicles may use two-gas atmospheres to decrease the propagation rate of fires and to allow the use of a wider range of materials in their construction.

For space stations, the requirements for crew safety include the provision that, throughout mission (which may extend for periods of a year or more), the crew shall have the capability to abandon the space station and return to Earth. This requirement necessitates the long-term storage in space f an entry vehicle in a quiescent state, either in the form of a logistics vehicle permanently docked to the station, or in the form of a lifeboat, a vehicle of minimum size to be used only in case of emergency. The problem associated with the storage of a logistics vehicle is that of the degrading effects of the space environment over extended periods of time. The lifeboat problem is unique in that its solution requires the development and launching of a vehicle which may never be used.

Communication

The general functions to be performed by the telecommunication system for unmanned vehicle include telemetry, tracking, and command (TT&C).[32–36] The particular requirements of the system are derived from the mission to be performed and from the physical limitations upon the system. An example of an Earth-orbital and entry telecommunication system is the Biosatellite vehicle. The two modes of operation, orbital and entry, generally imply different requirements of the TT&C functions. In the orbital case, all three functions are provided by separate VHF systems. A tape recorder is available to retain the information collected. During entry, a VHF telemetry transmitter provides for monitoring critical functions. Frequently, consideration must be given to the blackout problem, although this is not a concern for the Biosatellite mission. A beacon is on board for recovery purposes.

The telemetry system for Earth-centered missions is currently undergoing a shift in frequency from VHF to S band. The effect of the shift in frequency is that a higher transmitter power output is usually required. In addition, the S band antennas require multiple slots and hence are more complicated. Also, frequency or polarization diversity may be required to achieve the desired coverage during entry. The telecommunication system for a planetary probe can utilize either a direct link to Earth or a relay link to the spacecraft for subsequent transmission to Earth. In either case, the Earth link is constrained to S-band frequencies to be compatible with the deep-space network stations. The required transmitter powers are on the order of 20–50 w. At these high output powers, the high shock resistant travelling wave tube (TWT) has an efficiency approximately 2.5 times better than that currently available from solid state transmitters. Hence, a TWT power amplifier is considered the most likely candidate.

The selection of the frequency for a lander-to-spacecraft relay link is not constrained as is the direct link. Therefore, it can be optimized for the specific mission and vehicle configuration. The optimum frequency falls in the range of 100–500 MHz. Highly efficient solid state transmitters are available in this frequency range. The effective use of the spacecraft as a relay station is dependent upon the selected mission, the availability of the spacecraft, and the mutual visibility of the lander and the spacecraft. For long-life surface missions, the direct link is desirable to free the lander from dependence upon the spacecraft. However, the relay link is more efficient than the direct link and, therefore, is more suitable for transmission of data during entry and for the high data requirements of video information.

Power

The selection and design of power sources for entry vehicles[37–41] is determined by the power level and mission duration. For short, moderate-power missions, the silver-zinc primary battery provides excellent performance, high reliability, and low cost. The longer missions associated with planetary programs result in such total energy demands that solar cells, fuel cells, or RTG's must be used for prime power. In missions of 5–30 days duration, such as Biosatellite, a fuel cell provides weight savings and, with alternate tankage, weight flexibility. In the Biosatellite vehicle, where sterilization is not required, a primary battery has been used for the 3-day mission and a fuel cell supplemented by a battery for peak loading on the 21- and 30-day missions. The fuel cell is a hydrogen-oxygen ionic-membrane type which uses cryogenically stored gases. It will be required to furnish an average of 140 w during the missions.

Solar arrays have provided extended mission space power for a large number of orbiting, flyby, and lunar vehicles. For entry vehicles, the postlanding conditions place further constraints on the design. In some planetary flights (e.g., Venus), the unknown atmospheric conditions, cloud attenuations of sunlight, or surface dust storms could preclude the use of a solar cell. For a Mars hard landing, the environments are such that moderate mission durations of 3–90 days offer an opportunity for solar cell power to be used as a means for obtaining satisfactory operation beyond the short mission battery capability. For mission extensions beyond a year, the radioisotope thermoelectric generator may best meet requirements. A typical Venus probe would involve 12 months of storage and about 290 hr of actual operation. This can be performed with primary batteries supplemented by remotely activated thermal batteries for pulse loads. A Venus/Mercury swingby, a 14-month mission, would involve the use of solar cells for prime power in the space mode as well as the use of storage batteries for the entry mode. Most entry vehicles use thermal batteries for the high power levels encountered in the multiple pyrotechnic initiation associated with deorbit, separation, and entry functions.

Planetary quarantine requirements indicate the need for a power source that can be sterilized. Thermal, nickel-cadmium, and silver-zinc batteries definitely can be sterilized, and the RTG, by virtue of its normally high operating temperature, should have this capability. At the present time, fuel cell materials suitable for sterilization are being developed to meet the planetary quarantine requirements.

Impact System

The landing system must absorb the kinetic energy produced by the remnant of the capsule velocity subsequent either to its release from a drag device such as a parachute/ballute or at the termination cutoff of a retropropulsion system. The impact shock can be attenuated by crushable wraparound systems (balsa or honeycomb), pneumatic systems (gas bags), or hard-point systems. The choice of system would be dictated by the landing surface conditions, the touchdown velocity, the wind velocity, and the design inertial load level. For landing upon hard solid surfaces, the slope angles, protrusions, and yielding of surface must be considered. Wind direction and vehicle attitude will also affect the attenuator design. In addition, multidirectional and secondary rebounce capability, depending on such conditions as winds and stability, must be provided. For hard-point systems, such as legged devices and gas bags, reliable deployment and efficient packaging are added considerations. A great wealth of literature deals with the analysis of impact attenuation devices.[42–51]

Sterilization

Sterilization plays a major role in the definition of design and procedures for the scientific exploration of space.[52–54] In the case of Earth orbiters such as Biosatellite, the requirement is simply that the experiments not be compromised through the existence of undesired microbial growth. For Biosatellite in particular, each experimenter defined his individual specifications and performed the sterilization procedures so that his experimental results would not be invalidated. Sterilized subsystems concerned with the primate included a bloodstream injection subsystem and a urine collection subsystem. The bloodstream injection subsystem was sterilized to preclude infection of the animal; the urine collection subsystem was sterilized to preclude infection, to insure against degradation of the urinalysis results, and to prevent line clogging through precipitates caused by bacterial growth.

The accepted planetary entry vehicle sterilization specification states that the probability of releasing a single viable organism on the planet's surface must be less than 1 in 1000. Within this constraint, the equipment must operate reliably. A standard procedure has been designed which limits the growth of bacteria and contaminants while insuring operational reliability. It consists of three phases: 1) assembly of the equipment in clean rooms, 2) intermediate sterilization cycles prior to final assembly, and 3) terminal sterilization of the entire capsule/lander assembly together with its biological barrier, or canister, prior to interface with the launch vehicle. During entry into the planetary atmosphere, care must be taken to insure that neither the parent spacecraft nor any particle generated by the separation of the capsule from the spacecraft impacts the planet, since this would present a potential source of contamination. Sterilization may be accomplished by the application of dry heat at some particular temperature for some specified duration of time. Ethylene oxide may also be used as a surface decontaminant to reduce microbial population.

Entry Environment

Atmospheric braking is an efficient method for deceleration. Since significant entry heating is obtained from a small fraction of the dissipated energy, a rather thorough understanding of the vehicle-atmosphere interaction is required. The level and distribution of convective and radiative heating cannot be completely simulated in ground facilities. Although quite significant theoretical and experimental advances have been made, there is still much to be done in both research and engineering. Specifically, the definition of heating in orbital and lunar-return environments needs engineering refinement; the understanding of phenomena in hypervelocity Earth entry and planetary entry requires extensive research. Present experience is shown in Fig. 7.

The tradeoff between the desired engineering sophistication and the associated engineering investment must be considered; however, there exist many areas where significant improvements could be made. The description of the over-all flowfield around an entry vehicle is still a research problem, although this is the first prerequisite to establishing local environment and associated heating. Theoretical advances have been significant in the description of gas dynamic flow around an Earth-entry vehicle, in the basics of convective heating, in the convective heating around a vehicle, and in the radiation characteristics of air with the associated heating. However, the heating to surface irregularities and to separated flow regimes is not clearly defined.

Stagnation-Point Convective Heating

Several theoretical approaches to the solution of the stagnation-point heat transfer for entry into planetary atmospheres have been developed.[55–57] Experimental data were obtained[58–69] for a number of different gas mixtures, the principle constituents of which were CO_2, N_2, and Ar. Most of the experimental data agreed with the predictions of theory, but atmospheres with large Ar contents proved to be excep-

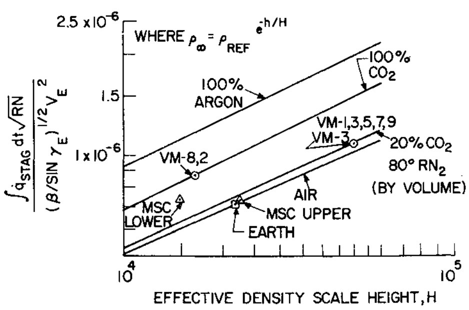


Fig. 8 Integrated heat transfer compared with gas composition.

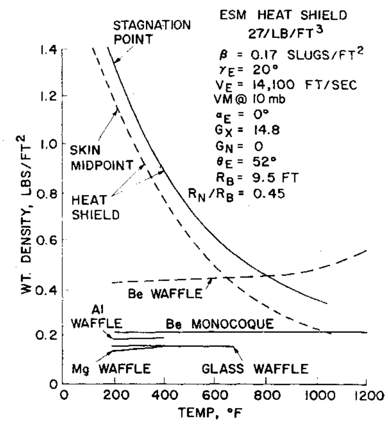


Fig. 9 Aeroshell shield-structure synthesis.

tions—the predictions of Marvin and Deiwert[55] appeared to underestimate the data by 65% at 35,000 ft/sec.

The effect of atmospheric model (both composition and density scale height) on the total-time integrated stagnation-point convective heating have been evaluated by combining the heating theory with the approximate entry analysis techniques of Allen and Eggers[1] (Fig. 8). Since the atmospheres of Mars and Venus are currently thought to contain 50–100% CO_2, the total uncertainty in the convective heating is small when compared to the other uncertainties that exist.

Boundary-Layer Transition

For ballistic entry into Mars and Venus, the nature of the boundary layer (laminar or turbulent) during the heat pulse depends primarily on the V_E, γ_F, and β. For a given V_E, as the γ_E and β are increased, the altitude at which peak heating occurs decreases, and there is a resultant increase in the Reynolds number and an eventual transition to a turbulent boundary layer. For lifting entry, the problem can be generalized to dependence on V_E and on pull-out altitude, dictated by L/D, V_E, W/C_LS (where C_L = lift coefficient and S = effective lift area), and the maximum inertial load. The effect of boundary-layer transition uncertainties on the heat shield requirements has been evaluated for hypervelocity planetary return by Coleman, Hearne, Lefferdo, Gallagher, and Vojvodich,[11] and for lifting entry return from Earth orbit by Masaki.[70]

Turbulent Heat Transfer

The prediction methods of heat-transfer levels in fully developed turbulent flow appear to be adequate for Earth-entry velocities up to 25,000 ft/sec. However, uncertainties in the actual turbulent boundary-layer flow mechanisms make it difficult to extrapolate outside the region of flow experience with any degree of confidence.

There are presently three major areas of concern: the behavior of the turbulent boundary layer at high local Mach numbers and Reynolds numbers, the effects of surface roughness and mass addition, and the role of wall temperature in effecting turbulent heat transfer. The effect of wall temperature on the heat-transfer coefficient h appears to be negligible, and it appears that all data correlate when the wall temperature is neglected in the calculation of the reference temperature by Eckert's method,[71,72] especially for nonblowing conditions.

The reduction in heat transfer and skin friction that results from mass addition into the compressible, turbulent boundary layer is of major concern in defining the heat protection potential of entry materials. The theory of turbulent heat transfer, while providing a reliable design tool for present energy and velocity levels in Earth entry, cannot be extended outside the range of presently available data without incurring considerable uncertainty. Studies of the effect of atmospheric composition on turbulent heat transfer need additional information that can be provided by experimental work and by correlation with existing theories for air.

Radiation Heat Transfer

Direct entry into Mars and Venus atmospheres at 18,000–45,000 ft/sec results in significant energy transfer because of shock-layer radiative emission, which can be equal to or greater than the magnitude of convective heat transfer. Hot gas radiation contains contributions from several sources. Most treatments of hot gas radiation for Earth or planetary entry do not consider all the sources; thus, they are in some cases very incomplete. Wolf and Spiegel's work[73] presents a comprehensive collection of experimental and theoretical predictions for several gas mixtures and indicates that the accuracy of prediction is limited to a factor of 2–4. However, they consider only the optically thin case and neglect the atomic line sources. Gruszczynski[74] has generated shock layer radiative intensities that result from molecular bands and the free-bound continuum of negative ions of oxygen and carbon for the gaseous mixture of 70% N_2, 30% CO_2 for flight velocities from 14,000 to 20,000 ft/sec. He includes molecular self-absorption for several optical thicknesses but does not include the contributions from atomic line sources. Woodward[75] has presented the radiative total intensity per particle and the number density for the molecular band systems of several gaseous mixtures. For shock temperatures above 11,000°K, the atomic line contributions dominate and must also be included. Total radiation including molecular, continuum, and lines has recently been reported by Menard, Thomas, and Helliwell[76] for a mixture of 30% CO_2, 40% N_2, and 30% Ar. Comparisons with theoretical prediction show that the total radiation is underpredicted by 30–50%.

Nonequilibrium effects present in the shock layer, particularly at the higher velocities, give rise to a radiative heat flux that exceeds the equilibrium portion determined by most calculations. Evaluation of the nonequilibrium radiation uses empirical correlations of the data[77–82] and truncation techniques. The reader's attention is directed to Refs. 89–105 for additional discussion of heat transfer in the entry environment.

Structural Design and Materials

Structural

The components of a vehicle which would require major structural design technology support are the following: the landing systems, such as the lander container or vessel; the entry systems, including an entry heat-protection shield and aeroshell or other decelerator; and the pre-entry systems, consisting of the biobarrier (including sterilization container), the meteroid and space radiation shield, and the spacecraft adapter and release system.

Aeroshell structural weight is governed by two factors: 1) aerodynamic pressure (peak value and distribution) and 2) deceleration (inertias in axial, lateral, and rotational directions). For a chosen mission mode and trajectory, the sum of heat-shield and structure weights must be minimized by, first, optimizing the interface temperature and, second, by optimizing the choice of materials (Fig. 9). Reradiative structures may also be compared with an ablative heat-shield structure composite.

The heat shield has three functions: ablative heat absorption, insulative heat protection, and structural integrity. Design is dictated primarily by aerothermodynamic considerations for heat protection of the capsule payload, and it is integrated with the structure for effects of thermal expansion, both for the temperature ranges during interplanetary phases of the flight and for the elevated temperatures during entry. Such factors as relative expansion, load strength at elevated temperatures, temperature gradient, and strength of charred and ablated shield are taken into consideration. Analytical methods are used for evaluating thermal stresses caused by radial gradients to satisfy critical requirements of compatibilities between thermal shield, the bond, and the substructure. Effects of the sterilization heating cycle on both the heat shield and the capsule must be considered and minimized.

Heat Shield and Materials

Improvements of booster-payload capabilities have resulted in a rapid development of spacecraft systems for extended exploration of space. With these developments, increasing demands have been made on entry thermal protection systems to sustain higher heating rates and larger heat loads at lower unit ablator weights. Although the Mercury and Biosatellite Programs relied upon the technology developed for early ballistic missile systems, current objectives of the space and ballistic missile programs result in a different set of thermal parameters and design criteria. This difference is exemplified by the Apollo command module, which enters the atmosphere at superorbital speeds with a modulated lift capability and a low β. However, developments in ballistic missiles have produced sharper nose cones and high β's and have resulted in entry environments which are characterized by high heating rates for short durations.

The primary thermal requirement for manned spacecraft heat-shield systems is that of low thermal conductivity in the virgin and char states. This requirement is a result of the low heating rate combined with a long-duration entry, in which the rate of thermal penetration in the ablator thickness exceeds the surface loss rate during the ablation process. To achieve low thermal conductivity values, ablator development for manned flight has tended toward low-density materials filled with ceramic fibers to maintain the physical integrity of the char state.

The development of entry spacecraft thermal protection systems requires a simultaneous activity in experimentation, theoretical analysis, and design while the ablation material and spacecraft are undergoing final formulation. Although the degree of accuracy of the data can be established, practical requirements dictate that a dual approach be taken which incorporates the design model formulation and the prediction model development. The formulation of a design model attempts to combine the various ablation phenomena into a gross or essential form (which is conservative in heat-shield sizing) to compensate for design uncertainty. After the heat shield is sized, its adequacy is verified through a series of ground and flight tests.

SUSPECTED CONTROLLING PHENOMENA	DESIGN MODEL
BOUNDARY LAYER	
CHEMICAL REACTIONS PRESSURE GRADIENT NON-SIMILAR VISCOUS FLOW MASS INJECTION	COLD WALL WIND TUNNEL DATA MODIFIED ECKERT BLOWING CURVE
CHAR LAYER	
HETROGENEOUS REACTIONS RE-RADIATION RADIATION MECHANICAL REMOVAL CONVECTION POROUS MEDIA FLOW TIME AND TEMPERATURE DEPENDANT PROPERTIES	SIMPLE TEMPERATURE DEPENDENT SURFACE RECESSION, GRAPHITE OXIDATION CONSTANT DENSITY TEMPERATURE DEPENDANT PROPERTIES
REACTION ZONE	
COMPLEX KINETIC REACTIONS POROUS MEDIA FLOW MECHANICAL CHANGES	ARRENHIOUS DECOMPOSITION FUNCTION TEMPERATURE-INTERPOLATED STATE AND PROPERTIES CONST. HEAT OF DECOMP.
VIRGIN MATERIAL	
PRESSURE AND TEMPERATURE DEPENDANT PROPERTIES POROUS MEDIA FLOW	TEMPERATURE DEPENDANT PROPERTIES CONSTANT DENSITY

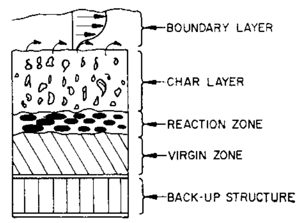


Fig. 10 Charring ablator modeling.

A prediction model must be developed to calculate the thermal response of a heat-shield system during entry. The development of such a model involves a detailed consideration of each phenomenon (Fig. 10) and its relation to the over-all material behavior. Although data collected from arc-heated facilities to support a design model formulation are used in the development of a prediction model, additional experimentation must be performed which uses the techniques of the physical sciences. This body of data is then combined into a numerical scheme to perform the necessary trajectory simulations. Certain numerical experimentation can also be performed to determine the sensitivities to data variation of the over-all thermal response. Comparisons of entry environments and vehicle weights are given in Table 1 for some of the entry spacecraft discussed in this paper.

The Biosatellite high-density phenolic nylon heat-shield system shown in Fig. 11 is an outgrowth of the original Discoverer shield system. The entry environment is predominantly laminar convective heating. Because of the rather mild heating levels and the oxygen depletion in the boundary layer (which results from pyrolysis gas injection), negligible char mass loss from oxidation is experienced, and the shield performs mainly in the insulation mode. A lighter weight heat shield could now be designed that performed primarily in the insulation mode (Fig. 12) and that used the foamed or syntactic silicones with a density of 30–41 lb/ft^3.

The Mercury and Gemini entries were characterized by a single heat pulse of a moderate heating rate level for time durations of approximately 500 sec. The Apollo entry from superorbital speeds is characterized by a double heat pulse; the first pulse of high heating rate levels occurs during deceleration at superorbital velocities and is followed by a second pulse of moderate heating rate level which occurs during suborbital deceleration. The total entry heating duration is approximately 1000 sec.

The Mercury ablator was composed of 40% phenolic resin and 60% fiberglass cloth which was wound with tape from the center at a bias angle of 20°. The ablator was then bonded to a phenolic structural laminate as shown in Fig. 11. The structural design temperature was 600°F at water impact. Because of the launch vehicle configuration, the ablator was protected from launch heating; however, during space operations, the heat shield is unprotected and experiences temperature extremes from -150° to +150°F.

Table 1. Entry environments for spacecraft and heat-shield properties

Entry environment	Bio-satellite	Mercury	Gemini	Apollo
β_E, lb/ft^2	82[a]		70.7	75.4
L/D	0	0	0.125	0.35
$\dot{q}_{max}$, Btu/ft^2-sec	134	63	42	230
$\dot{q}_{av}$, Btu/ft^2-sec	54.0	24	23	45.6
t, sec (400–50kft altitude)	348	390	550	980
$\int \dot{q} dt$, 10^3, Btu/ft^2	15.5	9.35	12.8	44.6
Q, 10^3, Btu/ft^2	142	210	...	7,760
Material (see Fig. 11):				
ρ (virgin), lb/ft^3	75	108	53	34
ρ (char), lb/ft^3	19	97	38	18–20
k (virgin), Btu/ft-hr-°F	0.14	0.198	...	0.07
Bondline design temperature, °F	450	600	500	600
Heat-shield thickness				
Blunt dish, in.	0.72–0.22	0.65	0.92 (av)	1.24 (av)
Afterbody, in.	0.072	...	...	...
Heat-shield unit weights				
Blunt dish, lb/ft^2	4.5	8.7	5.5 (av)	6.0
Afterbody, lb/ft^2	1.4	...	...	2.7
W_E, lb	490	2,600	5,050	12,000
W_{hs}, lb	48	315	345	1,650
W_{hs}/W_E	0.10	0.12	0.07	13.7
Q/W_{hs}, Btu/lb	3,000	666	...	5,470

[a] @ 200kft.

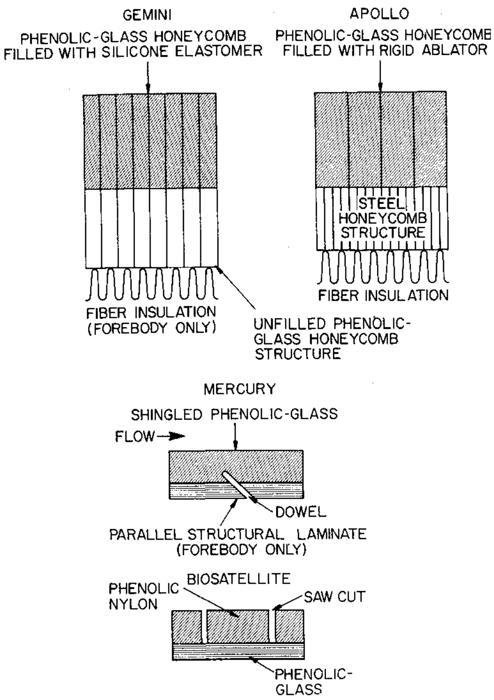


Fig. 11 Spacecraft thermal protection systems.

Improvements in ablation technology and materials were used in the Gemini Program to reduce heat-shield unit weights for heat loads and entry times which were greater than those encountered on the Mercury flights. The Gemini heat shield (Fig. 11) consisted of an elastomer, DC325, filled into an open phenolic honeycomb cell which was bonded to a phenolic honeycomb structural panel. The structural design temperature at water impact was 500°F, which was a structural limit determined by bondline and structural honeycomb panel strength. The Gemini heat shield was protected during launch and space operations by the retrorocket and equipment adapter sections in which the thermal environment varied from 0° to 150°F. The honeycomb cellwall provided additional physical integrity to the char and isolated char failure modes to a single cell.

The Apollo ablation material is an epoxy novalac resin system filled with phenolic micro balloons and quartz fibers. This material is gunned into an open cell phenolic honeycomb which is bonded to a stainless-steel structural panel. However, there are some important differences; first, the entire entry module (command module) is protected by an ablator; second, the heat shield contains a number of penetrations such as windows, access hatches, and utility lines which are connected to the module; and third, the heat shield is covered by a thermal control coating to limit temperature excursions to between −150° and +150°F.

High-speed entry into Martian and Venusian atmospheres as well as lunar or planetary return results in a combination

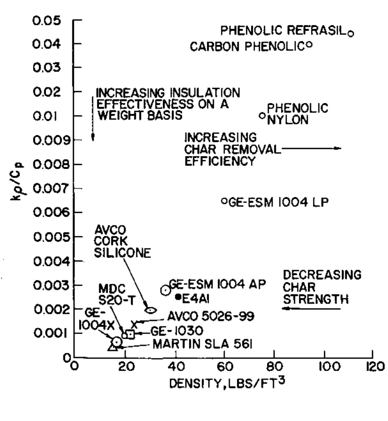


Fig. 12 Material insulation effectiveness.

of convective and radiative heating loads that are not encountered in normal Earth-ballistic entry flight tests. Several theoretical investigators[83,84] have predicted that significant radiation flux blockage is possible through mass addition. A limited amount of ground-based test data is available from which radiation flux blockage can be demonstrated in phenolic nylon and, by use of fluoride additives, in a low-density ablator.

Development and characterization of ablators for the out-of-orbit Mars entry has centered mostly on the lower-density (12–30 lb/ft^3) foamed and syntactic silicone and silicone epoxies, syntactic epoxies, and foamed Teflon.[85–88] These ablators, functioning primarily in the insulation mode, provide lighter shield systems and have successfully passed the ground tests simulating heat sterilization, vacuum exposure, Mars entry, and rf transparency in the charred state.

Selection of a shield material for Venus entry is more involved than for Mars because of the increased severity and complexity of the hypersonic environment which results primarily from the higher entry velocites. Nagler[85] has reviewed the applicability of several current material systems (including the Apollo rigid ablator, phenolic nylon, and phenolic carbon) across the range of entry conditions and ballistic coefficients currently of interest. Florence[9] summarized the configuration constraints imposed on the entry probe to hold the convective and radiative heating levels to values consistent with Earth-entry flight experience. If the effect of the boundary-layer chemistry in the CO_2 and N_2 atmospheres can be shown experimentally, correlations of ablator ground and flight test data in air that include the mechanical char removal effects can be extended for use in the planetary atmospheres. Since the entry velocity of a typical Venus probe may be only 36,000 ft/sec, the resulting radiative heating rates could be relatively moderate. Florence[9] also points out that Earth-entry flight tests have been flown successfully on high-β entry vehicles where the peak radiative heating rate was 600 Btu/ft^2-sec. Hence, it would appear that Venus entry at 36,000 ft/sec is not significantly outside the range of available Earth-entry flight test data. However, as the entry velocity is increased to 45,000 ft/sec, the peak radiative heating rates increase by a factor of $\sim$3 while the peak convective heating rates increase by a factor of $\sim$2. For material response characterization at these entry velocities, ground-based test simulation capability is the pacing item. Until material response to the simulated radiative/-convective environment is available, the entry velocity and ballistic coefficient must be kept as small as possible for the Venus probe.

The techniques presently in use can be readily extended to include foreign planet environments. The performance of ablation materials in these environments must be established by extensive studies similar to those which have led to present understanding of material performance in the Earth atmosphere. Space flight programs have resulted in the development of ablation materials, manufacturing techniques, and methods of analysis that are unique to entry technology. Thermal protection systems designs have evolved which are not only thermally efficient during entry but which maintain system integrity during launch and space operations prior to entry.

Entry Vehicle Tests

One of the most formidable and unique problems of entry vehicle development is realistic proof testing. The entry environment generates thermal energy densities up to 1 Gw/-ft^2, with heat-transfer rates that approach 10,000 Btu/ft^2-sec for the extreme trajectories. Most other significant portions of the environment encountered by spacecraft, such as vacuum, vibration, acoustic and inertial loads, solar radiation, the thermal sink of deep space, and even the mild heating encountered during launch, can be simulated in test facilities on complete spacecraft with sufficient accuracy to generate confidence in the vehicle design. However, the thermal performance ground testing of entry vehicles shield structures has been restricted to a collection of partial tests, most of which must be performed on small material samples or scale models and which are completely inadequate to lend any direct confidence in the entry vehicle design. Flight testing is therefore used extensively, not only to provide final qualification, but also to determine the basic performance of vehicle configurations and thermal protection systems.

Ground-based experimental facilities have come a long way in simulating many of the important parameters of the entry environment. Electric arc jets, shock tubes and tunnels, continuous flow facilities, and ballistic flight facilities have all advanced significantly in the past few years. The performance of these facilities will be enhanced even further, and conceptual new devices will appear. Flight experience with manned and unmanned entry vehicles provides a relatively firm base from which technology can grow. A large amount of flight data is available; however, weaknesses in the development of flight instrumentation have limited the types of available data. A greater technological investment in the area of both flight and ground instrumentation would be most fruitful.

Arc-heated wind-tunnel facilities are now operational which are capable of simultating the gas enthalpies, heat-transfer rates, and pressures encountered during entries at lunar-return velocities on models with size measured in feet. The capabilities of some of the newer arc-heated tunnel facilities are shown in Fig. 13. The current deficiencies in ground testing capability for thermal protection systems constitute somewhat of a paradox. One of the problems is the simulation of the low heating rates experienced by the afterbodies of Earth-orbital and lunar-return entry vehicles at realistic pressures and gas enthalpies. This problem is important to engineering attempts to design efficient or minimum weight thermal protection systems for such vehicles, since approximately $\frac{1}{2}$ of the heat-shield weight and $\frac{2}{3}$ of its area operates in such an environment. This problem in simulation exists simply because the convective heating rate varies inversely with the scale of the model. Thus, the innovation of new testing techniques which circumvent model scaling are required for such testing.

For hypervelocity entries, the primary heat-shield testing requirement is for either extremely high convective heating rates at high pressures and shears for sharp vehicles or for extremely high radiative heating simulation for blunt bodies. The model scale is again the nemesis since radiant flux is a direct function of model size. This problem presently defies a solution; an extensive research and development program will be required to meet this simulation requirement.

Experimental capability exists at several facilities to provide simultaneous simulation of radiative and convective heating loads. The current and near-future convective and radiative heating capabilities of these industry and NASA facilities is presented in Fig. 14. The Apollo flight corridor is included as well as the maximum heating rates experienced by an early ballistic entry vehicle flown with a phenolic nylon heat shield. Superimposed over the flight test experience and ground test capability are the predicted maximum heating rates to the vehicle stagnation point of a Venus entry vehicle. These data are for a representative range of nose radii and ballistic parameters for entry into the MSC model lower atmosphere at an entry velocity of 36,000 ft/sec. To fly within the range of available flight data or ground test capability requires use of a minimum value of the product $\beta \sin\gamma_E$. For a nominal Σ_E of 60° and $0.25 < R < 2$ ft, a β range $25 < \beta < 8$ lb/ft² is required.

Flight testing has been used in all major entry vehicle development programs. In some unmanned programs, repeated tests of prototype vehicles have been used in the development process, both with recoverable and nonrecoverable

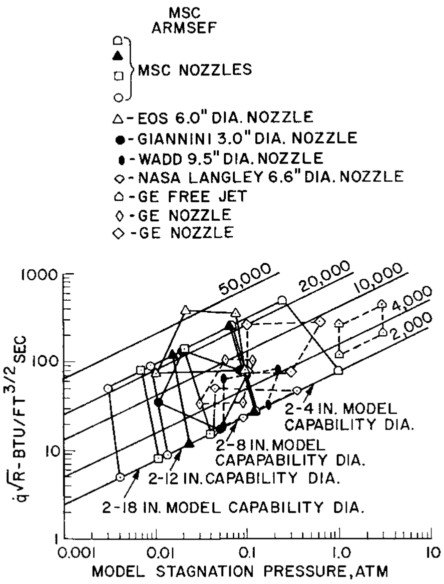


Fig. 13 Arc-heated tunnel capability.

payloads. The Apollo Program presented a unique problem in that prototype vehicles could not be tested at full lunar-return entry velocity until the Saturn V booster was operational, by which time the program dictated that the final entry vehicle was to be completely designed and built.

To meet the requirements for flight test data, the extensive test programs were performed for the Mercury, Gemini, and Apollo missions. The Apollo Program included small-scale tests (each for a specific objective) and lower-velocity tests of the complete system. These tests, however, did not yield data sufficient to answer the question of the performance of the full-scale vehicle during a full-velocity entry; they did suffice to dispel some of the concerns regarding radiative heating, turbulent flow ablator performance, and effects of protuberance on the heat shield. Quantitative data directly related to the Apollo could not be obtained from the small-scale tests because of the uncertain effects of scale on convective and radiative heating. For future missions, the entry vehicle flight problem will be alleviated by the availability of boosters capable of propelling essentially full-scale

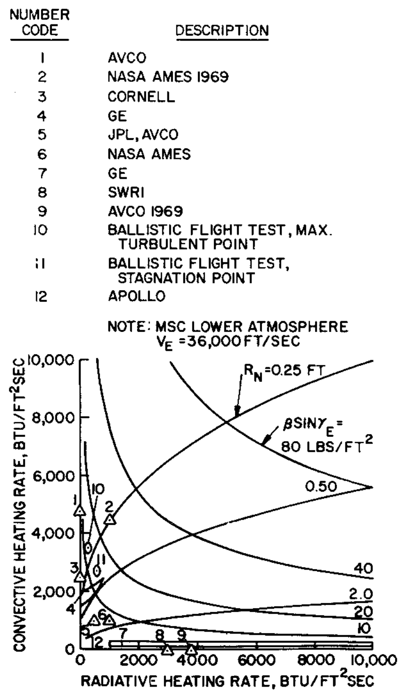


Fig. 14 Comparison of maximum stagnation-point heat flux to ground test capability and available flight test data.

test vehicles to the required entry velocities early enough in the program to provide meaningful inputs to entry vehicle development. In addition, it may be possible, with the more frequent operational missions anticipated in the next decade, to fly panels of candidate heatshield systems (on a noninterference basis) on some existing vehicles.

The most formidable flight test challenge faced at this time is that of peforming Earth-entry tests which realistically simulate entry into the atmosphere of other planets, such as Mars and Venus. Since the atmospheres of these planets are significantly different from that of the Earth, careful tailoring of the altitudes and velocities of these tests will be necessary to obtain the appropriate convective and radiant heating rates and dynamic pressures, even though full-scale vehicles can be used if desired. Studies[106–110] have been performed which indicate that reasonable simulation can be obtained by such tailoring and by adjustments in vehicle scale.

Conclusions

An extensive technological base has been developed for manned and unmanned entry from space at lunar-return velocities through programs such as the Biosatellite, Mercury, Gemini, and Apollo. The basic configurations have been blunt with a continuing trend toward the development of higher lift-to-drag ratios for greater flexibility and precision in recovery. It is anticipated that this trend will continue with a corresponding change in configuration. It does not appear that any major basic technological problems exist for the development of planetary entry systems for Venus or Mars, although engineering refinement in several technical disciplines would result in an appreciable improvement in the scientific-payload-to-entry-vehicle-weight ratio. As the entry velocity increases to those typical of planetary return missions, the uncertainties in the interaction of the environment with the entry vehicle system increase considerably, and a significant effort will be required to develop a sound engineering base for these systems. An extensive technical base has been developed in the environmental control systems and life support systems for manned and unmanned systems. These include single-gas and two-gas systems. Total spacecraft concept trends indicate that the entry vehicle may no longer serve as the full-time living quarters and command and control center for future space programs, although some mission modes will probably require the current approach.

Crew safety imposes stringent requirements on the entry vehicle/space vehicle system and will probably result in the need for a lifeboat system in the future. The shift from VHF to S band in telemetry systems will require higher transmitter power output, which could affect the power systems. For longer life space systems, the RTG power supply appears to offer significant advantages. Uncertainties in entry environment phenomena, such as radiant heat transfer at higher entry velocities and in boundary-layer transition, result in significant weight penalties to the entry system. In addition, the interaction of the environment with the materials, such as the ablation heat shield, need more extensive study to improve the system payload capability.

As a result of shortcomings in ground-based test simulation capability for evaluating shield-structure thermal performance, the need exists for effective entry flight simulation programs. In conjunction with this activity, improved sensors will be required to effectively measure the performance of the system.

In summary, it does not appear that any critical technological shortcomings exist to prevent early planetary lander system development although engineering refinements will be required to achieve scientific payload optimization. For higher-velocity entry such as planetary return, significant technological development will be required for effective development of these systems.

References

[1] Allen, H. J. and Eggers, A. J., "A Study of the Motion and Aerodynamic Heating of Missiles Entering the Earth's Atmosphere at High Supersonic Speeds," TN-4047, Oct. 1954, NASA.

[2] Space Flight Handbooks: a) *Orbital Flight Handbook*, Vol. I, NASA SP-33, 1963; b) *Lunar Flight Handbook*, Vol. II, NASA SP-34, 1963; c) *Planetary Flight Handbook*, Vol. III, NASA SP-35, 1963.

[3] Clarke, V. C., Jr., "A Summary of the Characteristics of Ballistic Interplanetary Trajectories," TR 32-209, Jan. 1962, Jet Propulsion Lab., Pasadena, Calif.

[4] Clarke, V. C., Jr. et al., "Design Parameters for Ballistic Interplanetary Trajectories," TR 32-77, Jan. 1963, Jet Propulsion Lab., Pasadena, Calif.

[5] Feitis, P. H., "Trajectory Design for Impulsive Earth-Mars-Earth Trajectories Launched in 1969 and 1971," TR 32-1007, Dec. 1966, Jet Propulsion Lab., Pasadena, Calif.

[6] Starnes, F. M., Jr., "Trajectory Analysis of an Earth-Venus-Mercury Mission in 1973," TR 32-1062, Jan. 1967, Jet Propulsion Lab., Pasadena, Calif.

[7] Repic, E. M., Boobar, M. G., and Chapil, F. G., "Aerobraking as a Potential Planetary Capture Mode," *Journal of Spacecraft and Rockets*, Vol. 5, No. 8, Aug. 1968, pp. 921–926.

[8] Guy, L. D., "Structural and Deceleration Design Options for Mars Entry," *Journal of Spacecraft and Rockets*, Vol. 6, No. 1, Jan. 1969, pp. 44–49.

[9] Florence, D. E., "Aerothermodynamic Considerations of a Venus Entry Probe," TIS 68SD438, April 1968, General Electric Co., Philadelphia, Pa.

[10] McLellan, C. H. and Pritchard, E. B., "Use of Lift to Increase Payload of Unmanned Martian Landers," *Journal of Spacecraft and Rockets*, Vol. 3, No. 9, Sept. 1966, pp. 1421–1425.

[11] Coleman, W. D. et al., "Effects of Environmental and Ablator Performance Uncertainties on Heat-Shielding Requirements for Hyperbolic Entry Vehicles," *Journal of Spacecraft and Rockets*, Vol. 5, No. 11, Nov. 1968, pp. 1260–1270.

[12] "Study of Manned Vehicles for Entering the Earth's Atmosphere at Hyperbolic Velocities, Final Report," LMSC-4-05-65-12, Nov. 1965, Lockheed Missiles and Space Co., Sunnyvale, Calif.

[13] Shapland, D. J. and Munroe, W. F., "A Comparative Design Analysis of Three Configurational Families for Manned Earth Entry at Hyperbolic Speeds," *Journal of Spacecraft and Rockets*, Vol. 4, No. 6, June 1967, pp. 732–740.

[14] McCarthy, J. F. and Hanley, G. F., "Manned Earth Entry at Hyperbolic Velocities," *Journal of Spacecraft and Rockets*, Vol. 5, No. 9, Sept. 1968, pp. 1009–1015.

[15] Love, E. S., "Factor Influencing Configuration and Performance of Multipurpose Manned Entry Vehicles," *Journal of Spacecraft and Rockets*, Vol. 1, No. 1, Jan. 1964, pp. 3–12.

[16] Baron, R., *Cryogenic Systems*, McGraw-Hill, New York, 1966.

[17] Barzelay, M. E., Tong, K. N., and Holloway, G. F., "Thermal Conductance of Contacts in Aircraft Joints," TN-3167, March 1954, NASA.

[18] Barzelay, M. E. and Holloway, G. F., "Interface Thermal Conductance of Twenty-Seven Riveted Aircraft Joints," TN-3991, July 1957, NASA.

[19] Fried, E. and Costello, F. J., "Interface Thermal Contact Resistance Problem in Space Vehicles," *ARS Journal*, Vol. 32, No. 2, Feb. 1962, pp. 237–243.

[20] Christensen, E. H., "Thermal Insulation of Space Vehicles," AZJ-55-005, Astronautics Div. of General Dynamics Corp., San Diego, Calif.

[21] Stevenson, J. A. and Grafton, J. C. "Radiation Heat Transfer Analysis for Space Vehicles," ASD TR 61-119, Oct. 1961, Space and Information Systems Div. of North American Aviation Inc., Downey, Calif.

[22] Mackay, D. B. and Bocha, C. P., "Analysis and Design of Space Radiators," ASD TR 61-30, Pt. 5, April 1961, Space and Information Systems Div. of North American Aviation Inc., Downey, Calif.

[23] Kilpatrick, B. K., Parks, C. H., and Watanabe, D., "Environmental Control System Selection for Unmanned Space Vehicles," ASD TR 61-164, Pt. 1, Space and Information Systems Div. of North American Aviation Inc., Downey, Calif.

[24] Glasser, S. P. and Stelzriede, M. E., "Integration and Optimization of Space Vehicle Environmental Control Systems," ASD TR 61-176, Pt. 1, Space and Information Systems Div. of North American Aviation Inc., Downey, Calif.

[25] Mason, J. L., Burriss, N. L., and Connally, T. J., "Vapor Cycle Cooling for Aircraft," WADC TR 53-338, Wright Air Development Center, Dayton, Ohio.

[26] Ginwala, K., ed., "Engineering Study of Vapor Cycle Cooling Equipment for Zero-Gravity Environment," WADD TR 60-776, Jan. 1961, Northern Research and Engineering Corp.

[27] Rousseau, J., "Atmospheric Control Systems for Space Vehicles," ASD TR 62-527, Pt. 1, March 1963, AiResearch Manufacturing Co., Los Angeles, Calif.

[28] Breeze, R. K., "Space Vehicle Environmental Control Requirements Based on Equipment and Physiological Criteria," TR 61-161, Pt. 1, Dec. 1961, North American Aviation Inc., Los Angeles, Calif.

[29] Samonski, F. H., Jr., "Technical History of the Environmental Control System for Project Mercury," TN D-4126, 1967, NASA.

[30] Shaffer, A., ed., "Analytical Methods for Space Vehicle Atmospheric Control Processes, ASD TR 61-162, Pt. 1, Oct. 1961, AiResearch Manufacturing Co., Los Angeles, Calif.

[31] Mayo, A. M., Thompson, A. B., and Whisenhunt, G. B., "Design Criteria for a Manned Space-Laboratory Environmental Control System," *Journal of Spacecraft and Rockets*, Vol. 1, No. 3, May-June 1964, pp. 296–302.

[32] Kinkead, W. K., "Implications on Antenna Design Resulting from Change to L&S Telemetry Bands," *ITC Proceedings of the International Foundation for Telemetering*, Tarzana, Calif., 1962.

[33] Shatas, J., "Fifty Watt Solid-State FM Transmitter for Deep Space Communications," TIS 66SD2002, Jan. 1966, General Electric Co., Philadelphia, Pa.

[34] Roberts, L. A. et al., "Some Recent Advances in Travelling Wave Tubes," *Telecommunications*, Vol. 2, No. 8, Aug. 1968.

[35] "Multiple Mission Telemetry System," *The Deep Space Network Space Programs Summary*, Vol. III, Jet Propulsion Lab., Pasadena, Calif., July 1967, pp. 37–46.

[36] Kinkead, W. K., "Antenna Breakdown in Planetary Atmosphere," TIS 65SD316, Sept. 1965, General Electric Co., Philadelphia, Pa.

[37] Iberall, A., "Study and Analysis of Satellite Power Systems Configurations for Maximum Utilization of Power," Rept. CR-898, Oct. 1967, NASA.

[38] MacKenzie, C. M., "Solar Power Systems for Satellites in Near Earth Orbits," TM X-55826, March 1967, NASA.

[39] Fono, P., "Special Design Considerations for the Apollo Spacecraft Electrical System," *IEEE Transactions on Aerospace*, Vol. 2, No. 2, April 1964, pp. 467–471.

[40] Dugan, E. E., "Electrical Power System Studies at MSFC," STAR N67-30593, 1967, NASA.

[41] Zeis, J. F., "Analysis of Prime Electrical Power Sources for Space Vehicles," TIS 64SD241, May 1964, General Electric Co., Philadelphia, Pa.

[42] McGehee, J. R., "Gas Bag Concept for a Stowable Omnidirectional, Multiple-Impact Landing System," *Journal of Spacecraft and Rockets*, Vol. 4, No. 10, Oct. 1967, pp. 1359–1362.

[43] Menkes, E. G., Kasakian, H., and Pyle, H., "Landing Dynamics of an Impacting Body," SM 8156-168, May 1966, General Electric Co., Philadelphia, Pa.

[44] Eagar, J. B., "Survey of Energy-Absorption Devices for Soft Landing of Space Vehicles," TN D-1308, June 1962, NASA.

[45] Conn, A. F., "Impact Energy Absorption Properties of Crushable Materials," Research Memo 315, Oct. 1966, Martin Co., Baltimore, Md.

[46] Bresie, D., "Practical Limits for Balsa Impact Limiters," TN D-3175, Jan. 1966, NASA.

[47] Stubbs, S. M., "Landing Characteristics of the Apollo Spacecraft with Deployed Heat-Shield Impact Attenuation Systems," TN D-3049, Jan. 1966, NASA.

[48] Knoell, A. C., "Environmental and Physical Effects on the Response of Bala Wood as an Energy Dissipator," TR 32-994, June 1966, Jet Propulsion Lab., Pasadena, Calif.

[49] Coppa, A. P., "Effect of Impact Velocity on the Performance of Several Energy Absorbing Materials," CR57591, Dec. 1964, NASA.

[50] Gerard, G., "Energy Dissipating Structures: A Structures-Materials-Design Synthesis," TR 286-1, Aug. 1964, Allied Research Associates Inc., Boston, Mass.

[51] Pohlen, J. C., "Planetary Lander Concepts Dictated by Touchdown Parameters," AIAA Paper 68-306, Palm Springs, Calif., 1968.

[52] Craven, C. W., "Planetary Quarantine Analysis," *Astronautics and Aeronautics*, Vol. 6, No. 8, Aug. 1968, pp. 20–24.

[53] Stern, J. A., "Current Concepts in Sterilization," *Astronautics and Aeronautics*, Vol. 6, No. 8, Aug. 1968, pp. 25–34.

[54] Ervin, G. F., "Sterilization Requirements, Operational Procedures Facilities and Hardware," *Astronautics and Aeronautics*, Aug. 1968, pp. 34–38.

[55] Marvin, G. F. and Deiwert, G. S., "Convective Heat Transfer in Planetary Gases," TR R-224, July 1965, NASA.

[56] Hoshizaki, H., "Heat Transfer in Planetary Atmospheres at Super-Satellite Speeds," *ARS Journal*, Vol. 32, No. 10, Oct. 1962, pp. 1544–1552.

[57] Scala, S. M. and Gilbert, L. M., "Theory of Hypersonic Laminar Stagnation Region Heat Transfer in Dissociating Gases," TIS R63SD40, Nov. 1964, General Electric Co., Philadelphia, Pa.

[58] Rutowski, R. W. and Chan, K. K., "Shock Tube Experiments Simulating Entry into Planetary Atmospheres," *General Research in Flight Sciences-Fluid Mechanics*, LMSD-288139, Vol. 1, Part II, Lockheed Aircraft Corp., Sunnyvale, Calif.

[59] Warren, W. R. and Gruszcyzynski, J. S., "Experimental Planetary Entry for Mars and Venus," TIS R62SD84, Oct. 1962, General Electric Co., Philadelphia, Pa.

[60] Yee, L., Bailey, H. E., and Woodward, H. T., "Ballistic Range Measurements of Stagnation Point Heat Transfer in Air and Carbon Dioxide at Velocities up to 18,000 ft/sec," TN D-777, March 1961, NASA.

[61] Gruszczynski, J. S., Warren, W. R., and Dioconis, N. S., "Laboratory Simulation of Hypervelocity Heat Transfer Problem During Planetary Entry," TIS R64SD73, Nov. 1964, General Electric Co., Philadelphia, Pa.

[62] Horton, T. E. and Babineaux, T. L., "Influence of Atmospheric Composition on Hypersonic Stagnation Point Convective Heating," *AIAA Journal*, Vol. 5, No. 1, Jan. 1967, pp. 36–43.

[63] Nerem, R. M., Morgan, C. J., and Graver, B. C., "Hypervelocity Stagnation Point Heat Transfer in a Carbon Dioxide Atmosphere," *AIAA Journal*, Vol. 1, No. 9, Sept. 1963, pp. 2173–2175.

[64] Reilly, J. P., "Stagnation Point Heating in Ionized Monatomic Gases," Publication 64-1 (AFOFR 5442), Jan. 1964, Fluid Mechanics Lab., Massachussets Institute of Technology, Boston, Mass.

[65] Rutowski, R. W. and Bershader, D., "Shock Tube Studies of Radiative Transport in an Argon Plasma," *The Physics of Fluids*, Vol. 7, No. 4, April 1964, pp. 568–577.

[66] Gruszczynski, J. S. and Warren, W. R., "Measurements of Hypervelocity Stagnation Point Heat Transfer in Simulated Planetary Atmospheres," R63SD29, March 1963, General Electric Co., Philadelphia, Pa.

[67] Collins, D. J. and Horton, T. E., "Experimental Convective Heat Transfer Measurements," *AIAA Journal*, Vol. 2, No. 11, Nov. 1964, pp. 2046–2047.

[68] Finson, M. L. and Kemp, N. H., "Theory of Stagnation Point Heat Transfer in Ionized Monatomic Gases," *The Physics of Fluids*, Vol. 8, No. 1, Jan. 1965, pp. 201–204.

[69] Marvin, J. G. and Pope, A. B., "Laminar Convective Heating and Ablation in the Mars Atmosphere," *AIAA Journal*, Vol. 5, No. 2, Feb. 1967, pp. 240–248.

[70] Masaki, M., "A Review of the Thermal Protection System Analyses for the Multipurpose Feasible Spacecraft Preliminary Design Effort," Rept. TDR-0158 (3520-64), Jan. 1968, Aerospace Corp., El Segundo, Calif.

[71] Eckert, E. R. G., "Survey of Heat Transfer at High Speed," TR 5470, 1954, Wright Air Development Center, Dayton, Ohio.

[72] Eckert, E. R. G., "Engineering Relations for Heat Transfer and Friction in High Velocity Laminar and Turbulent Boundary Layer Flow Over Surfaces with Constant Pressure and Temperature," *Transactions of the ASME*, Vol. 78, No. 8, Aug. 1956, pp. 1273–1283.

[73] Wolf, F. and Spiegel, J., "Status of Basic Shock Layer Radiation Information for Inner-Planet Atmospheric Entry," *Journal of Spacecraft and Rockets*, Vol. 4, No. 9, Sept. 1967, pp. 1166–1173.

[74] Gruszczynski, J. S. and Bradley, D. A., "Radiative Properties of High Temperature 30% CO_2-70% N_2 Gas Mixture, Part 2," Experimental Fluid Physics Technical Memo 207, June 1967, General Electric Co., Philadephia, Pa.

[75] Woodward, H. T., "Predictions of Shock-Layer Radiation from Molecular Band Systems in Proposed Planetary Atmospheres," TN D-3850, Feb. 1967, NASA.

[76] Menard, W. A., Thomas, G. M., and Helliwell, T. M., "Experimental and Theoretical Study of Molecular Continuum

and Line Radiation from Planetary Atmospheres," *AIAA Journal* Vol. 6, No. 6, April 1968, pp. 655–664.

[77] Gruszczynski, J. S. and Thomas, K., "Equilibrium and Nonequilibrium Radiation in Simulated Atmospheres," AIAA Paper 66-183, Monterey, Calif., 1966.

[78] Thomas, G. and Menard, W., "Experimental Measurements of Non-Equilibrium and Equilibrium Radiation from Planetary Atmospheres," *AIAA Publications CP-9*, AIAA Entry Technology Conference, Williamsburg and Hampton, Va., Oct. 1964.

[79] James, C., "Experimental Study of Radiative Transport from Hot Gases Simulating in Composition the Atmospheres of Mars and Venus," *AIAA Journal*, Vol. 2, No. 3, March 1964, pp. 470–475.

[80] Page, W., "Shock Layer Radiation of Blunt Bodies Traveling at Lunar Return Entry Velocities," Paper 63-41, Jan. 1963, IAS.

[81] James, C. and Smith, W., "Experimental Studies of Static Stability and Radiative Heating Associated with Mars and Venus Entry," Jan. 1963, IAS.

[82] Allen, Rose, and Camm, "Non-Equilibrium Radiation at Super-Satellite Re-entry Velocities," Paper 63-77, Jan. 1963, IAS.

[83] Macken, N. A. and Hartnett, J. P., "Interaction of Convection and Radiation in Axisymmetric and Two-Dimensional Stagnation Point Flow," Tr-E6, March 1967, Univ. of Illinois, Champaign Urbana, Ill.

[84] Hoshizaki, H. and Wilson, K. H., "Convective and Radiative Heat Transfer During Superorbital Entry," CR-584, Sept. 1966, NASA.

[85] Nagler, R. G., "Tailoring Polymers for Entry into the Atmospheres of Mars and Venus," *National Meeting of the ACS*, April 1968, San Francisco, Calif.

[86] Strauss, E. L., "Response of Superlight Ablators to Various Heat Pulses," AIAA Paper 68-301, Palm Springs, Calif., 1968.

[87] Hiltz, A. A., Florence, D. E., and Lowe, D. L., "Selection, Development and Characterization of a Thermal Protection System for a Mars Entry Vehicle," AIAA Paper 68-304, Palm Springs, Calif., 1968.

[88] Vosteen, L. F., "Heat Shield Materials Development for Voyager," *Stanford Research Institute Third International Symposium on High Temperature Technology*, Sept. 17–20, 1967, Asilomar, Pacific Grove, Calif.

[89] Spalding, D. B. and Chi, S. W., "The Drag of a Compressible Turbulent Boundary Layer on a Smooth Flat Plate with and without Heat Transfer," *Journal of Fluid Mechanics*, Vol. 18, Part 1, Jan. 1964, pp. 117–143.

[90] Winkler, E. M. and Cha, M. H., "Investigation of Flat Plate Hypersonic Turbulent Boundary Layers, with Heat Transfer at a Mach Number of 5.2," NAVORD Rept. 6631, Sept. 1959.

[91] Leont'ev, A. I., "Heat and Mass Transfer in Turbulent Boundary Layers," *Advances in Heat Transfer*, edited by Irvine and Hartnett, Academic Press, Vol. 3, New York, 1966.

[92] Spalding, D. B., Auslander, D. M., and Sundaram, T. R., "The Calculation of Heat and Mass Transfer Through the Turbulent Boundary Layer on a Flat Plate at High Mach Numbers, With and Without Chemical Reaction," Northern Research and Engineering Corp., Cambridge, Mass.

[93] Walker, G. K., "A Particular Integral of the Turbulent Boundary Layer Equations," *Journal of the Aerospace Sciences*, Vol. 27, No. 9, Sept. 1960, pp. 715–716.

[94] Wilson, D. M., "Measurements of Hypersonic Turbulent Heat Transfer on a Highly Cooled Cone," NOLTR 67-24, Aerodynamics Research Rept. 283, July 1967, U.S. Naval Ordnance Lab., White Oak, Md.

[95] Danberg, J. E., "Characteristics of the Turbulent Boundary Layer with Heat and Mass Transfer at M 6.7," NOLTR 64-99, Oct. 1964, U.S. Naval Ordnance Lab., White Oak, Md.

[96] Bartz, D. R., "Turbulent Boundary Layer Heat Transfer from Rapidly Accelerating Flow of Rocket Combustion Gases and of Heated Air," *Advances in Heat Transfer*, edited by J. P. Hartnett and T. F. Irvine, Vol. 2, Academic Press, New York, 1965.

[97] Bertram, M. H. and Neal, L., Jr., "Recent Experiment in Hypersonic Turbulent Boundary Layers," AGARD Specialists Meeting on Recent Developments in Boundary Layer Research, May 10–14, 1965, Naples, Italy.

[98] Diaconis, N. S., Shaw, T. E., and Warren, W. R., Jr., "Final Report for Contract NAS 9-4771," Feb. 1967, General Electric Co., Philadelphia, Pa.

[99] Diaconis, N. S., Shaw, T. E., and Warren, W. R., Jr., "The Hypervelocity Heat Protection Problem," TIS-R65SD60, Sept. 1965, General Electric Co., Philadelphia, Pa.

[100] "Arc-Heater Characterization of Ablative Plastics and Composites," Quarterly Progress Report No. 6, AVSSD-0023, 68-CR, Feb. 1968, AVCO, Boston, Mass.

[101] Flügge-Lotz, I. and Blottner, F. G., "Computation of the Compressible Laminar Boundary Layer Flow Including Displacement of Thickness Interaction Using Finite-Difference Methods," Division of Engineering Mechanics, TR 131, Jan. 1962, Stanford University, Palo Alto, Calif.

[102] Scala, S. M. and Gilbert, L. M., "The Sublimation of Graphite at Hypersonic Speeds," *1964 AIAA Re-Entry Technology Proceedings*, AIAA, New York, 1964, pp. 239–258.

[103] Dorrance, W. H. and Dore, E. J., "The Effect of Mass Transfer on the Compressible Turbulent Boundary-Layer Skin Friction and Heat Transfer," *Journal of the Aeronautical Sciences*, Vol. 21, No. 6, June 1954, pp. 404–410.

[104] Wakefield, R. M., Lundell, J. H., and Dickey, R. R., "The Effect of Oxygen Depletion in Gas-Phase Chemical Reactions on the Surface Recession of Charring Ablators," AIAA Paper 68-303, Palm Springs, Calif., 1968.

[105] Kindall, R. M., Rindal, R. A., and Bartlett, F. P., "Thermochemical Ablation," *AIAA Journal*, Vol. 5, No. 6, June 1967, pp. 1063–71.

[106] Hiester, N. K. and Clark, C. F., "Feasibility of Standard Evaluation Procedures for Ablating Materials," CR-379, Feb. 1966, NASA.

[107] Schaefer, J. and Flood, D. T., "High Enthalpy Re-Entry Simulation for Planetary Return Missions," AIAA Paper 68-381, San Francisco, Calif., 1968.

[108] Gowen, F. E., Lundell, J. H., and Wick, B. H., Annual Institute of Environmental Sciences Technical Meeting and Equipment Exposition, Washington, D.C., April 10-12, 1967.

[109] Hiester, N. K. and Clark, C. F., "Relative Operating Capabilities of Selected Electric Arc Re-entry Environment Simulators," CR-99, Sept. 1964, NASA.

[110] Boatright, W. B. et al., "Review of Testing Techniques and Flow Calibration Results for Hypersonic Arc Tunnels," AIAA Paper 68-379, San Francisco, Calif., 1968.