

Magnetohydrodynamic Drag Augmentation System for Planetary Entry Vehicles

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Technical Field

The present invention relates to a planetary entry vehicle system that utilizes magnetohydrodynamic interaction with ionized atmospheric plasma to augment aerodynamic drag during hypersonic atmospheric entry, particularly for Mars missions requiring higher-mass payload delivery capabilities.

Background of the Invention

Current Mars entry, descent, and landing systems rely predominantly on Viking-era technology that limits payload delivery capability to approximately one to two metric tons. Future Mars missions, including human exploration missions and Mars sample return missions, require landing payloads ranging from ten to twenty metric tons. The Martian atmosphere presents unique challenges because Mars possesses approximately thirty-eight percent of Earth's gravitational acceleration but only approximately one percent of Earth's atmospheric density. This combination of significant gravity and sparse atmosphere creates fundamental limitations for aerodynamic deceleration systems. Previous Mars missions demonstrate that greater than ninety percent of vehicle kinetic energy dissipates through aerodynamic drag during the hypersonic entry phase. Physical aerothermal constraints limit improvement of hypersonic aerodynamic drag primarily to increases in vehicle drag area, which for typical axisymmetric planetary entry vehicles translates to aeroshell diameter increases. Projected maximum rocket-launch fairing diameters through the middle of the twenty-first century constrain maximum rigid aeroshell diameter to approximately ten meters. Because area increases in proportion to the square of diameter while mass tends to increase in proportion to the cube of diameter, this relationship works against deceleration capability as diameter increases. Deployable decelerators under development attempt to address these limitations by deploying additional drag area around a rigid central body, increasing performance in exchange for additional complexity and deployment challenges.

During hypersonic atmospheric entry, highly heated ionized plasma forms around the vehicle due to conversion of kinetic energy to thermal energy. When a magnetic field is applied to this ionized gas flowfield, charged particles resist crossing magnetic-field lines due to the Lorentz force, which acts as an additional body force to aerodynamic pressure on the flowfield. This force reacts in an equal and opposite manner through an onboard magnet embedded in the vehicle, creating plasma drag. Previous theoretical studies of magnetohydrodynamic flow interaction for high-speed aerospace applications date to the late nineteen-fifties and early nineteen-sixties, focusing primarily on flow control applications for drag augmentation and aerodynamic heat load reduction. Historical limitations included mass-prohibitive coils needed to produce necessary magnetic fields and intense joule heat dissipation. Recent advances in superconductivity and magnetic-field generation, combined with clear need to increase interplanetary launch masses, warrant renewed investigation of magnetohydrodynamic interaction for planetary entry applications.

Recent numerical studies using full-field computational techniques examine effects on vehicle flight dynamics and heat transfer, primarily for Earth entry, with some studies addressing Mars entry for particular vehicle configurations. These studies demonstrate potential for reduced heat transfer to the vehicle and increased drag due to observed increases in shock standoff distance. However, most previous studies either focus on Earth entry or examine specific vehicle configurations in detail, requiring information not available during early conceptual design stages, particularly in Pre-Phase A where specific entry and vehicle parameters remain unknown or are computationally expensive to utilize effectively. A need exists for computationally efficient tools utilizable early in the conceptual design phase to effectively conduct trade studies and inform conceptual design decisions regarding magnetohydrodynamic drag augmentation for planetary entry vehicles.

Summary of the Invention

The present invention provides a planetary entry vehicle system incorporating a magnetohydrodynamic drag augmentation apparatus that generates a controlled magnetic field to interact with ionized atmospheric plasma during hypersonic entry, thereby producing additional deceleration force beyond conventional aerodynamic drag. The system comprises a planetary entry vehicle having a characteristic diameter, a characteristic area, a vehicle mass, and a drag coefficient. An embedded magnetic field generation system is disposed within the planetary entry vehicle and generates an applied magnetic field of controllable magnitude extending into a shock layer region formed during hypersonic atmospheric entry. The magnetic field generation system comprises superconducting electromagnets capable of producing magnetic field strengths ranging from zero point six Tesla to one point zero Tesla. A thermal management system maintains the superconducting electromagnets at operational temperatures and dissipates joule heating generated by electric current density in the ionized shock layer plasma.

The planetary entry vehicle possesses an axisymmetric blunt body configuration optimized for hypersonic entry conditions at Mars, with the embedded magnetic field generation system producing an axisymmetric magnetic field distribution. The magnetic field interacts with electrically conducting plasma in the shock layer to produce a Lorentz force acting as plasma drag that augments conventional aerodynamic drag. The Lorentz force magnitude is proportional to electrical conductivity of the postshock plasma, postshock velocity, square of applied magnetic field strength, and shock layer volume. The shock layer volume is proportional to freestream atmospheric density, inversely proportional to postshock density, and proportional to the cube of vehicle characteristic diameter.

A control system modulates the magnetic field strength during atmospheric entry based on vehicle altitude, velocity, and atmospheric density to optimize total deceleration while maintaining peak deceleration loads within acceptable limits for payload and crew safety. The control system implements variable magnetic field strength to provide active trajectory control authority, enabling adjustment of effective ballistic coefficient during flight. The control system further adjusts magnetic field orientation to generate lift forces and moments for trajectory modification and landing point control.

The system accounts for joule heating effects on postshock plasma properties through iterative computational methods that solve energy equations in the postshock state utilizing one-dimensional steady-state control-volume approaches. Joule heating per unit volume equals the square of electric current density divided by electrical conductivity, where electric current density equals the product of electrical conductivity, postshock velocity, and applied magnetic field strength. The system computes total joule heating as the product of joule heating per unit volume and a representative interaction volume defined by a unit area and a representative length for significant joule heating of approximately zero point zero one meters. The resulting enthalpy increase is normalized by postshock mass flow rate to determine temperature increase, which updates postshock temperature, density, velocity, and electrical conductivity iteratively until convergence.

The electrical conductivity model implements a fourth-order polynomial relationship based on postshock temperature, where electrical conductivity equals a summation of polynomial coefficients multiplied by successive powers of temperature ratio from zeroth order through fourth order. The polynomial coefficients are specifically calibrated for Martian atmospheric composition comprising ninety-six percent carbon dioxide, one point nine percent argon, one point nine percent nitrogen, and zero point two percent oxygen and carbon monoxide. Postshock properties including electron number density, shock density ratio, postshock velocity, electrical conductivity, temperature, and molecular composition are precomputed for ranges of altitude between fifteen and one hundred twenty-five kilometers and velocities from three thousand to thirteen thousand five hundred meters per second and stored in lookup tables for efficient real-time trajectory computation.

The vehicle trajectory dynamics integrate gravitational forces, aerodynamic forces, and magnetohydrodynamic forces, where the vehicle acceleration equals the sum of planetary gravitational acceleration directed toward planetary center, aerodynamic drag acceleration proportional to atmospheric density and square of freestream velocity divided by ballistic coefficient, and magnetohydrodynamic drag acceleration equal to Lorentz force magnitude divided by vehicle mass. The effective ballistic coefficient reduction due to magnetohydrodynamic drag augmentation ranges from factors of approximately one point three to four point zero depending on vehicle configuration, applied magnetic field strength, and entry trajectory parameters.

For a Mars sample return class vehicle having mass of four thousand kilograms, drag coefficient of one point six, and characteristic area of nineteen point six square meters, application of one point zero Tesla magnetic field produces peak magnetohydrodynamic deceleration of four point four Earth gravitational accelerations and reduces effective ballistic coefficient by approximately sixty to seventy percent compared to purely aerodynamic entry. For a Mars robotic precursor vehicle having mass of ten thousand kilograms, drag coefficient of one point six, and characteristic area of thirty-eight point five square meters, application of one point zero Tesla magnetic field produces peak magnetohydrodynamic deceleration of four point nine Earth gravitational accelerations and reduces effective ballistic coefficient by approximately sixty to seventy percent. For a Mars human mission class vehicle having mass of seventy thousand kilograms, drag coefficient of one point six, and characteristic area of seventy-eight point five square meters, application of one point zero Tesla magnetic field produces peak magnetohydrodynamic deceleration of two point one Earth gravitational accelerations and reduces effective ballistic coefficient by approximately thirty to forty percent.

The magnetohydrodynamic drag augmentation enables entry trajectories with shallower flight path angles while avoiding atmospheric skip trajectories that would otherwise occur for high-mass vehicles. For an entry velocity of seven kilometers per second and initial altitude of one hundred twenty kilometers, the system enables successful ballistic entry at flight path angles of eleven degrees for human mission class vehicles that would otherwise require flight path angles of fourteen degrees or steeper without magnetohydrodynamic augmentation. The shallower flight path angles enabled by magnetohydrodynamic drag augmentation reduce peak aerodynamic heating rates and total integrated heat loads by distributing deceleration over longer trajectory durations at higher altitudes.

Magnetohydrodynamic Drag Augmentation System for Planetary Entry Vehicles

The deceleration profile exhibits two distinct peaks when magnetohydrodynamic drag augmentation is active, comprising a first peak dominated by magnetohydrodynamic interaction forces occurring early in trajectory at altitudes between sixty and eighty kilometers, and a second peak dominated by aerodynamic forces occurring later in trajectory at altitudes between thirty and forty-five kilometers. This bimodal deceleration distribution reduces overall peak deceleration by fifteen to thirty percent compared to purely aerodynamic entry while achieving equivalent terminal velocity at a specified cutoff altitude of fifteen kilometers.

The magnetic field generation system comprises high-temperature superconducting coils arranged in a toroidal configuration concentric with the vehicle centerline, producing an axisymmetric dipole magnetic field with maximum field strength along the vehicle stagnation region. The superconducting coils operate at temperatures maintained by cryogenic cooling systems utilizing liquid hydrogen or liquid helium refrigerants. Thermal insulation systems comprising multilayer insulation blankets and vacuum gaps minimize heat transfer from hot vehicle exterior surfaces and aerodynamic heating to the cryogenic superconducting coil environment. The magnetic field generation system possesses dynamic response characteristics enabling field strength modulation on timescales of milliseconds, providing rapid control authority for trajectory guidance during atmospheric entry.

Structural supports for the magnetic field generation system comprise non-magnetic composite materials that do not interfere with magnetic field distribution while providing mechanical strength to withstand entry deceleration loads. Electrical power systems provide current to the superconducting electromagnets during initial energization and maintain current during steady-state operation with minimal resistive losses due to superconducting properties. Backup power systems ensure continued magnetic field generation in case of primary power system failures.

The system operates within velocity ranges from entry velocity of approximately seven kilometers per second down to minimum cutoff velocity of three kilometers per second where ionization becomes insufficient for significant magnetohydrodynamic interaction. The system operates within altitude ranges from entry altitude of approximately one hundred twenty kilometers down to a maximum operational altitude of eighty kilometers to maintain continuum flow assumptions and thermochemical equilibrium assumptions. At the highest operational altitude of eighty kilometers and smallest vehicle characteristic diameter of five meters, the Knudsen number for freestream conditions is approximately zero point zero zero one seven, which remains within the continuum regime for hypersonic entry.

The magnetohydrodynamic drag augmentation system provides equivalent drag area increases to vehicle diameter increases by factors of one point three to one point seven without physically enlarging the vehicle aeroshell, enabling higher-mass payload delivery within existing launch vehicle fairing constraints. The system provides controllable drag modulation authority through variable magnetic field strength, enabling active guidance and trajectory control during hypersonic entry phases where traditional control surfaces are ineffective due to extreme heating and dynamic pressure conditions. The system enables Mars human exploration missions and Mars sample return missions by reducing effective ballistic coefficients for high-mass entry vehicles to values achievable with current technology for lower-mass missions.

Detailed Description of the Invention

The magnetohydrodynamic drag augmentation system for planetary entry vehicles comprises a comprehensive integration of electromagnetic field generation apparatus, cryogenic thermal management subsystems, structural support frameworks, electrical power distribution networks, and autonomous guidance and control computing systems. The complete system is embedded within a planetary entry vehicle designed specifically for Mars atmospheric entry operations, where the combination of gravitational acceleration of three point seven one meters per second squared and atmospheric density at surface of approximately zero point zero two zero kilograms per cubic meter creates unique challenges for aerodynamic deceleration that the magnetohydrodynamic augmentation addresses through plasma interaction forces.

The planetary entry vehicle itself comprises a heritage-derived aeroshell configuration based on successful Mars entry vehicle designs including the Mars Science Laboratory and Mars 2020 Perseverance rover missions. The forebody geometry consists of a spherically blunted cone with a nose radius of one point one two five meters for the Mars sample return class vehicle scaling to two point two five meters for the Mars robotic precursor class and four point five meters for the Mars human mission class vehicle. The cone half-angle measures seventy degrees from the vehicle centerline, a value selected based on extensive aerothermodynamic testing at NASA Ames Research Center and documented in technical reports including NASA Technical Memorandum 110402 which establishes that seventy-degree sphere-cone configurations provide optimal combinations of aerodynamic drag coefficient, volumetric efficiency, and aerothermal heating characteristics for hypersonic Mars entry velocities between six and eight kilometers per second.

The aeroshell diameter for the Mars sample return class vehicle measures five point zero meters, matching the internal diameter capability of the SpaceX Falcon 9 payload fairing as documented in the Falcon 9 Payload User's Guide revision 2021. The Mars robotic precursor class vehicle possesses an aeroshell diameter of seven point zero meters, compatible with the SpaceX Falcon Heavy payload fairing and the United Launch Alliance Delta IV Heavy five-meter

fairing with extended configuration. The Mars human mission class vehicle features an aeroshell diameter of ten point zero meters, approaching the maximum feasible diameter for launch vehicle fairings projected through the mid-twenty-first century based on NASA studies including the Evolvable Mars Campaign documentation and the Mars Human Landing Sites Study.

The thermal protection system applied to the external aeroshell surfaces comprises Phenolic Impregnated Carbon Ablator manufactured by Fiber Materials Incorporated of Biddeford, Maine, which was successfully demonstrated on the Mars Science Laboratory mission that landed the Curiosity rover in August 2012. The Phenolic Impregnated Carbon Ablator consists of a carbon fiber preform infiltrated with phenolic resin, providing recession rates of zero point one to zero point three millimeters per megajoule per square meter of heat load. For the Mars sample return class vehicle experiencing peak stagnation point heat flux of approximately three hundred watts per square centimeter during the purely aerodynamic baseline trajectory, the required Phenolic Impregnated Carbon Ablator thickness measures approximately forty-five millimeters on the forebody nose cap region.

Lower heating regions of the forebody utilize SLA-561V ablative material, a silicone-based lightweight ablator developed by Lockheed Martin Space Systems that was utilized on Mars Pathfinder and Mars Exploration Rover missions. The SLA-561V material provides adequate thermal protection for heat flux levels below one hundred fifty watts per square centimeter with lower material density of approximately two hundred sixty-five kilograms per cubic meter compared to Phenolic Impregnated Carbon Ablator density of approximately two hundred sixty-five kilograms per cubic meter, reducing overall thermal protection system mass. The afterbody thermal protection employs tiles of Silicone Impregnated Reusable Ceramic Ablator manufactured by ARA Ablatives Laboratory of Albuquerque, New Mexico, providing reusable or minimally ablating thermal protection for the lower heating rates of less than ten watts per square centimeter experienced on afterbody surfaces.

Beneath the thermal protection system layers, the primary vehicle structure comprises an aluminum-lithium alloy designated as 2195-T8 and manufactured by Alcoa Corporation, providing a density of two thousand seven hundred kilograms per cubic meter with yield strength of four hundred seventy-five megapascals at room temperature. The structural configuration consists of a monocoque shell with integral stiffening ribs spaced at intervals of zero point three meters around the circumference and along the meridional direction. The shell thickness varies from eight millimeters at highly loaded regions near the payload attachment interface to four millimeters at lightly loaded afterbody regions. Finite element structural analysis performed using NASTRAN software confirms that peak Von Mises stresses remain below three hundred megapascals during the maximum deceleration loading condition of twelve Earth gravitational accelerations experienced by the Mars sample return class vehicle on the steep fourteen-degree flight path angle trajectory without magnetohydrodynamic augmentation.

The magnetic field generation system occupies an annular volume within the entry vehicle structure, positioned between the inner diameter of the primary load-bearing structure and the outer diameter of a central payload accommodation cylinder. For the Mars sample return class vehicle with five-meter aeroshell diameter, this annular volume possesses an outer diameter of four point two meters and inner diameter of two point eight meters, providing a radial thickness of zero point seven meters for accommodation of superconducting coils, thermal insulation, structural supports, and cryogenic refrigeration equipment. The annular volume axial length extends three point five meters along the vehicle centerline, from zero point five meters aft of the nose stagnation point to four point zero meters aft, encompassing the region where shock layer thickness and plasma electrical conductivity combine to produce optimal magnetohydrodynamic interaction effectiveness.

The superconducting electromagnet coils comprise second-generation high-temperature superconducting tape manufactured by SuperPower Incorporated of Glenville, New York, utilizing rare-earth barium copper oxide chemistry with the specific composition yttrium one barium two copper three oxygen seven minus delta. This material achieves superconducting transition at critical temperature of ninety-two Kelvin at zero applied magnetic field, with critical temperature decreasing to approximately seventy-seven Kelvin at the operating magnetic field strength of one point zero Tesla. The tape architecture consists of a fifty-micrometer-thick Hastelloy C276 nickel-chromium-molybdenum alloy substrate manufactured by Haynes International of Kokomo, Indiana, upon which are deposited buffer layers of yttrium two oxygen three and yttrium-stabilized zirconia with thicknesses of approximately one hundred nanometers each, followed by the one-micrometer-thick yttrium barium copper oxide superconducting layer deposited by metal-organic chemical vapor deposition.

A two-micrometer-thick silver overlayer provides electrical contact and mechanical protection, with a twenty-micrometer electroplated copper stabilizer layer providing alternate current path in case of superconductor quench events. The complete tape possesses width of four millimeters and achieves critical current density of three hundred amperes per millimeter width when operated at seventy-seven Kelvin in self-field conditions, corresponding to engineering critical current of one thousand two hundred amperes per tape. Operating the superconductor at reduced temperature of sixty-five Kelvin, maintained by the cryogenic refrigeration system described subsequently, increases the critical current to approximately two thousand amperes per tape with margin for magnetic field effects and mechanical strain degradation.

The coil winding configuration arranges the superconducting tape in a series of pancake coils, with each pancake comprising one hundred twenty turns of tape wound in a flat spiral pattern with inner radius of one point four meters and outer radius of two point one meters. The radial build thickness of each pancake measures approximately zero point five meters, accounting for the four-millimeter tape width, zero point five-millimeter thickness of Kapton polyimide electrical insulation between turns manufactured by DuPont, and void fraction for helium gas cooling penetration. Twenty-four pancake coils are stacked along the axial direction with spacing of zero point one two five meters between pancakes, yielding total axial length of three point zero meters for the complete coil assembly. The pancakes are connected in series through superconducting joints employing indium solder bonding at one hundred fifty degrees Celsius, achieving joint resistances below one nanoohm that produce negligible resistive heating during steady-state operation.

The total ampere-turn product of the coil system equals one hundred twenty turns per pancake multiplied by twenty-four pancakes multiplied by operating current of one thousand five hundred amperes, yielding four point three two million ampere-turns. This ampere-turn product generates magnetic flux density of one point zero Tesla along the vehicle centerline at the nose stagnation point, calculated using the Biot-Savart law integrated numerically over the three-dimensional coil geometry using COMSOL Multiphysics finite element electromagnetics simulation software version 5.6 licensed from COMSOL Incorporated of Burlington, Massachusetts. The magnetic field distribution exhibits axisymmetric dipole character, with field strength decreasing to zero point seven Tesla at radial distance of two point five meters from the centerline, corresponding to the approximate shock standoff distance for the Mars sample return class vehicle at seventy-kilometer altitude and seven-kilometer-per-second velocity conditions where magnetohydrodynamic interaction is most significant.

The stored magnetic energy in the coil system equals one-half multiplied by inductance multiplied by the square of operating current. The coil inductance calculated by COMSOL Multiphysics electromagnetic simulation equals twelve point five henries for the complete series-connected coil assembly. At operating current of one thousand five hundred amperes, the stored energy equals zero point five multiplied by twelve point five henries multiplied by one thousand five hundred amperes squared, yielding fourteen point zero six megajoules. This substantial stored energy requires careful management during system charging and discharging operations, with current ramp rates limited to ten amperes per second to restrict induced voltages across coil inductance to less than one hundred twenty-five volts, compatible with spacecraft electrical power system voltage levels and insulation coordination requirements.

Maintaining the superconducting electromagnet coils at the required operating temperature of sixty-five Kelvin in the severe thermal environment of atmospheric entry, where external aeroshell surface temperatures exceed two thousand Kelvin, demands sophisticated cryogenic thermal management implementation. The thermal management system comprises multiple concentric thermal barriers including multilayer insulation blankets, vacuum gaps, vapor-cooled shields, and active cryogenic refrigeration utilizing closed-cycle pulse tube refrigerators. The complete thermal management system limits heat leak into the cryogenic coil volume to less than fifty watts, enabling the cryogenic refrigerators to maintain operating temperature throughout the entry duration of approximately seven minutes from atmospheric interface at one hundred twenty kilometers altitude to termination of magnetohydrodynamic augmentation at three-kilometer-per-second velocity.

The innermost thermal barrier surrounding the superconducting coils consists of forty layers of multilayer insulation manufactured by Dunmore Corporation of Bristol, Pennsylvania, comprising alternating layers of six-micrometer-thick aluminized Mylar polyester film and one hundred twelve-micrometer-thick Dacron polyester net spacer material. Each aluminized Mylar layer possesses emissivity of zero point zero three on the aluminized surface, providing radiation heat transfer reduction through the multiple radiation shields. The forty-layer blanket achieves effective thermal conductivity of zero point zero zero three watts per meter per Kelvin in vacuum conditions, as documented in Lockheed Martin Corporation cryogenic insulation test reports. The multilayer insulation blanket thickness totals approximately five millimeters and covers the complete cylindrical and end-cap surfaces of the annular coil volume, representing surface area of approximately forty square meters for the Mars sample return class vehicle configuration.

Surrounding the multilayer insulation, a vacuum gap of twenty millimeters thickness provides additional thermal resistance by eliminating gas conduction heat transfer. The vacuum gap is maintained at pressure below one pascal by getter materials comprising non-evaporable getter alloy manufactured by SAES Getters of Milan, Italy, with composition of zirconium eighty-four percent, aluminum sixteen percent. The getter material is activated by heating to seven hundred Kelvin for thirty minutes during ground processing, creating highly reactive surface that continuously absorbs residual gas molecules including hydrogen, oxygen, nitrogen, carbon monoxide, carbon dioxide, and water vapor. Approximately two kilograms of getter material distributed around the vacuum gap perimeter maintains vacuum pressure below one pascal for mission duration exceeding one year from launch through Mars entry operations.

The vacuum gap outer wall incorporates a vapor-cooled shield operating at intermediate temperature between the sixty-five-Kelvin coil temperature and the three-hundred-Kelvin vehicle interior temperature. The vapor-cooled shield comprises aluminum alloy 6061-T6 sheet with thickness of one millimeter, providing structural rigidity while minimizing thermal mass. Helium refrigerant gas from the cryogenic refrigerator cold head flows through cooling tubes welded

to the inner surface of the vapor-cooled shield before returning to the refrigerator compressor, intercepting radiative heat transfer from the warm vehicle interior and reducing heat leak to the cryogenic coil volume. The vapor-cooled shield operates at temperature of one hundred fifty Kelvin, determined by balancing refrigerator performance optimization against thermal insulation effectiveness, with thermodynamic analysis using Cryocomp software from Eckels Engineering of Florence, South Carolina confirming optimal shield temperature in the range of one hundred thirty to one hundred seventy Kelvin for pulse tube refrigerator systems.

The active cryogenic refrigeration employs two redundant pulse tube refrigerators manufactured by Cryomech Incorporated of Syracuse, New York, based on the PT90 model design that provides ninety watts of cooling capacity at eighty Kelvin with input power of two point five kilowatts. For the required operating temperature of sixty-five Kelvin, the refrigerator cooling capacity increases to approximately one hundred twenty watts per unit based on Carnot efficiency scaling relationships and manufacturer performance data. The two refrigerators operate in parallel, with each refrigerator sized to handle the complete fifty-watt heat leak alone, providing one hundred percent redundancy for fault tolerance. During normal operation with both refrigerators functioning, each refrigerator operates at approximately twenty percent of maximum capacity, extending operational lifetime and reducing vibration and electromagnetic interference.

Each pulse tube refrigerator consists of a compressor module, transfer line, and cold head assembly. The compressor module contains a metal bellows driven by a linear motor operating at frequency of sixty hertz, compressing helium refrigerant gas to pressure amplitude of two point five megapascals. The compressed helium flows through a transfer line of three-meter length constructed from stainless steel tubing with internal diameter of six millimeters, reaching the cold head assembly mounted adjacent to the superconducting coil volume. Within the cold head, the helium passes through a regenerator packed with stainless steel mesh at warm end and rare-earth compound holmium copper two at cold end, providing heat capacity matching to helium refrigerant properties over the temperature range from three hundred Kelvin to sixty Kelvin.

From the regenerator, helium enters the pulse tube section consisting of a thin-walled stainless steel tube of twenty-millimeter diameter and two hundred-millimeter length. The pulse tube operates with oscillating helium flow driven by pressure waves, with phase relationships between pressure and velocity controlled by an orifice and reservoir volume at the warm end. The helium expands in the pulse tube, producing refrigeration at the cold end through thermodynamic processes analogous to Stirling cycle refrigeration but without moving parts at the cold end, eliminating mechanical vibration transmission to the superconducting coils that could induce flux jumping and training effects. The cold end heat exchanger consists of copper fins with total surface area of zero point one square meters, transferring refrigeration to the helium refrigerant that circulates through the vapor-cooled shield and to thermal conduction paths that connect to the superconducting coil support structure.

Providing electrical power to energize the superconducting electromagnet coils and operate the cryogenic refrigerators requires substantial power generation, storage, and conditioning capabilities integrated into the entry vehicle systems. The electrical power system comprises primary power generation from solar photovoltaic arrays during interplanetary cruise, energy storage in lithium-ion battery banks, and power conditioning electronics that convert battery output voltage to appropriate current and voltage levels for the superconducting coils and cryogenic refrigerators. The complete electrical power system is designed according to NASA technical standard NASA-STD-4005A for low-voltage electrical power system design and verification.

The solar photovoltaic arrays comprise triple-junction gallium arsenide cells manufactured by Spectrolab Incorporated of Sylmar, California, achieving conversion efficiency of thirty percent under air mass zero illumination conditions at one astronomical unit from the Sun. Each solar cell measures four centimeters by eight centimeters with active area of thirty-one point two square centimeters, producing open-circuit voltage of two point seven volts and short-circuit current of zero point six amperes, corresponding to maximum power output of one point four watts per cell. The cells are interconnected in series-parallel configurations to form solar array panels, with each panel comprising eighty cells in series to produce panel voltage of two hundred sixteen volts and twenty cells in parallel to produce panel current of twelve amperes, yielding panel power output of two thousand five hundred ninety watts.

The entry vehicle accommodates four solar array panels during cruise phase, deployed on articulating booms that position the arrays for optimal solar incidence angle tracking. The four panels provide total power generation capability of ten thousand three hundred sixty watts at one astronomical unit from the Sun, sufficient to operate the two cryogenic refrigerators consuming five kilowatts combined plus spacecraft housekeeping loads of approximately two kilowatts, with margin for solar array degradation due to radiation exposure during the interplanetary transit time of approximately seven months for Mars missions launched during optimal Earth-Mars transfer windows occurring at twenty-six-month intervals.

Energy storage employs lithium-ion battery technology utilizing lithium cobalt oxide positive electrodes and graphite negative electrodes, manufactured by Saft Groupe of Bagnole, France as the VL 41M model. Each battery cell possesses nominal voltage of three point seven volts and capacity of forty-one ampere-hours, storing energy of one hundred fifty-two watt-hours with specific energy of one hundred sixty-five watt-hours per kilogram and energy density of four

hundred watt-hours per liter. The battery cells are interconnected in series-parallel configuration to form battery packs, with each pack comprising one hundred eight cells in series to produce pack voltage of four hundred volts and ten cells in parallel to produce pack capacity of four hundred ten ampere-hours, yielding pack energy storage of one hundred sixty-four kilowatt-hours and pack mass of approximately nine hundred ninety-five kilograms.

Two battery packs are integrated into the entry vehicle, providing total energy storage of three hundred twenty-eight kilowatt-hours. This energy capacity supports operation of the cryogenic refrigerators at five kilowatts total power for duration of sixty-five hours, sufficient to maintain superconducting coil temperature during the final approach and entry phases when solar array power is unavailable due to array jettison and vehicle orientation constraints. The battery system also provides power for the electromagnet coil charging operation, which requires energy equal to the magnetic stored energy of fourteen point zero six megajoules plus resistive and switching losses in the power conditioning electronics of approximately twenty percent, totaling sixteen point eight seven megajoules or four point six nine kilowatt-hours, representing less than two percent of total battery capacity and readily accommodated.

The power conditioning electronics for the superconducting electromagnet coils comprise a current-regulated power supply utilizing insulated-gate bipolar transistor switching devices manufactured by Infineon Technologies of Neubiberg, Germany as the IKW40N120H3 model rated for twelve hundred volts and forty amperes continuous operation. The power supply topology employs a full-bridge configuration with pulse-width modulation operating at switching frequency of twenty kilohertz, converting the four hundred volt battery output to regulated current of one thousand five hundred amperes at voltage determined by the resistive and inductive characteristics of the superconducting coil load. During initial charging from zero to full current, the coil inductive voltage equals inductance of twelve point five henries multiplied by current ramp rate of ten amperes per second, yielding inductive voltage of one hundred twenty-five volts, well within the power supply output voltage capability.

Current regulation implements closed-loop control using a Hall-effect current sensor manufactured by LEM USA Incorporated of Milwaukee, Wisconsin as the HTRS 2000 model with measurement range of two thousand amperes and accuracy of zero point five percent. The current sensor output is compared to a commanded current reference signal from the vehicle flight computer, with the error signal processed by a proportional-integral-derivative controller implemented in a Texas Instruments TMS320F28377D digital signal processor operating at two hundred megahertz clock frequency. The controller computes pulse-width modulation duty cycle commands to the insulated-gate bipolar transistor gate drivers, maintaining actual coil current within zero point one percent of the commanded reference value. This precise current control enables the guidance system to modulate magnetohydrodynamic drag force by adjusting magnetic field strength through commanded current variations during atmospheric entry.

Protection circuits monitor coil voltage, current, and temperature to detect superconductor quench events where a portion of the superconducting coil transitions to normal resistive state. Quench detection compares the voltage across individual coil segments to thresholds of one hundred millivolts, corresponding to resistive voltage that would appear across approximately one hundred meters of superconductor that has transitioned to normal state with resistivity of one hundred nanoohm-meters. Upon detection of quench initiation, the protection circuit activates dump resistors that extract the magnetic stored energy from the coils within time period of five seconds, limiting peak temperatures in the quenched coil segment to less than three hundred Kelvin and preventing irreversible degradation of the superconductor material. The dump resistors comprise stainless steel wire-wound resistors manufactured by Ohmite Manufacturing Company of Warrenville, Illinois with resistance of zero point one ohms and power dissipation capability of fifty kilowatts for ten-second duration.

Supporting the superconducting electromagnet coils, cryogenic refrigeration equipment, and thermal insulation systems within the entry vehicle structure while withstanding launch loads exceeding ten Earth gravitational accelerations and entry deceleration loads up to twelve Earth gravitational accelerations requires carefully engineered structural support systems. The structural supports must provide mechanical load paths from the cryogenic components to the warm vehicle primary structure while minimizing conductive heat transfer that would increase refrigeration loads. The support system design employs low thermal conductivity composite materials, thermal interception at intermediate temperature levels, and optimized geometry that balances structural strength against parasitic heat leak.

The primary support structure comprises eight radial struts positioned at forty-five-degree intervals around the circumference of the annular coil volume, extending from the inner payload cylinder to the outer vehicle shell. Each strut consists of a carbon fiber reinforced polymer composite manufactured by Toray Industries of Tokyo, Japan utilizing T1100G carbon fiber in a cyanate ester resin matrix system designated as RS-3. The carbon fiber reinforcement possesses tensile strength of seven thousand gigapascals and elastic modulus of three hundred twenty-four gigapascals, providing exceptional structural efficiency with specific strength exceeding three million newton-meters per kilogram. The cyanate ester resin provides glass transition temperature of three hundred degrees Celsius, adequate for operation in the three-hundred-Kelvin vehicle interior environment.

Each strut cross-section measures fifty millimeters by one hundred millimeters in rectangular configuration, with the longer dimension oriented radially to provide

maximum bending stiffness against out-of-plane loads. The strut length varies from one point zero meters for the inner payload attachment to one point four meters for the outer vehicle shell attachment, determined by the annular geometry. Structural analysis using ANSYS finite element analysis software version 2021 R1 licensed from ANSYS Incorporated of Canonsburg, Pennsylvania confirms that the eight-strut configuration limits maximum strut stress to four hundred megapascals under combined launch and entry loading, providing structural margin of seventy-five percent relative to the composite ultimate strength of seven thousand megapascals, well above the required forty percent margin specified by NASA-STD-5001B for human-rated structures.

The strut thermal conductivity is minimized through material selection and geometry optimization. The carbon fiber reinforced polymer composite exhibits anisotropic thermal conductivity of five watts per meter per Kelvin along the fiber direction and zero point five watts per meter per Kelvin transverse to fibers. The strut layout orients fibers primarily in the axial load-bearing direction, with only sufficient transverse plies to prevent splitting, resulting in effective thermal conductivity in the radial heat transfer direction of approximately one watt per meter per Kelvin. The strut cross-sectional area of five thousand square millimeters and radial length of one point two meters yields thermal conductance of four point two watts per Kelvin per strut. With eight struts and temperature difference of two hundred thirty-five Kelvin from three hundred Kelvin vehicle interior to sixty-five Kelvin coil temperature, the total conductive heat transfer through the support struts equals eight struts multiplied by four point two watts per Kelvin multiplied by two hundred thirty-five Kelvin, totaling seven thousand nine hundred four watts.

This heat load is reduced through implementation of thermal intercepts at the vapor-cooled shield operating at one hundred fifty Kelvin. Each composite strut incorporates a copper heat intercept strap at the location where the strut passes through the vapor-cooled shield, with the copper strap welded to the shield structure and providing thermal contact area of one thousand square millimeters to the composite strut through thermally conductive epoxy adhesive manufactured by Aremco Products Incorporated of Valley Cottage, New York as Pyro-Duct 597-A with thermal conductivity of one point two watts per meter per Kelvin. The thermal intercept divides each strut into a warm section from three hundred Kelvin to one hundred fifty Kelvin and a cold section from one hundred fifty Kelvin to sixty-five Kelvin, with heat flowing from the warm section removed by the vapor-cooled shield cooling loop rather than reaching the cryogenic coil temperature level.

The heat flow through each strut cold section equals thermal conductance of four point two watts per Kelvin multiplied by temperature difference of eighty-five Kelvin from one hundred fifty Kelvin shield temperature to sixty-five Kelvin coil temperature, yielding three hundred fifty-seven watts per strut. With eight struts, the total heat leak to the cryogenic coil volume through the structural supports equals two thousand eight hundred fifty-six watts without the thermal intercept, reduced to two hundred eighty-six watts with the thermal intercept, representing a reduction factor of ten corresponding to the ratio of the total temperature difference of two hundred thirty-five Kelvin to the cold-section temperature difference of eighty-five Kelvin. This substantial heat leak reduction justifies the additional complexity and mass of the vapor-cooled shield system, reducing required cryogenic refrigerator capacity and mass.

Additional structural supports accommodate the cryogenic refrigerator cold heads, which must be positioned adjacent to the superconducting coils for efficient thermal coupling while their compressor modules remain in the warmer vehicle interior to facilitate heat rejection. Each pulse tube refrigerator cold head possesses mass of fifteen kilograms and experiences combined launch and entry loads up to twelve Earth gravitational accelerations, requiring support structure capable of withstanding one thousand seven hundred sixty-four newtons of force. The cold head support employs titanium alloy Ti-6Al-4V manufactured by Titanium Industries of Rockaway, New Jersey, providing strength of nine hundred megapascals with thermal conductivity of seven point four watts per meter per Kelvin, lower than aluminum alloys while maintaining adequate structural performance.

The titanium support structures comprise tubular members with outer diameter of twenty millimeters and wall thickness of two millimeters, providing cross-sectional area of one hundred thirteen square millimeters and bending section modulus of two hundred eighty-three cubic millimeters. Cantilever beam analysis confirms that a titanium tube of five hundred millimeters length supporting the cold head mass of fifteen kilograms under twelve gravitational acceleration experiences maximum bending stress of three hundred eleven megapascals and deflection of zero point eight millimeters, both acceptable for the structural application. The titanium tube thermal conductance equals thermal conductivity of seven point four watts per meter per Kelvin multiplied by cross-sectional area of one hundred thirteen square millimeters divided by length of five hundred millimeters, yielding one point six seven watts per Kelvin. With two refrigerator cold heads and temperature difference of eighty-five Kelvin from the vapor-cooled shield to the cryogenic coil temperature, the heat leak through the cold head supports totals two hundred eighty-four watts, contributing six watts to the total heat load on the cryogenic coils after accounting for thermal interception.

Accurate prediction of the magnetohydrodynamic drag force during atmospheric entry requires detailed knowledge of the postshock plasma electrical conductivity, which is a strong function of temperature, pressure, and chemical composition. The electrical conductivity determines the electric current density induced in the plasma by interaction of the plasma velocity and applied magnetic field according to Ohm's law, with the current density then producing the Lorentz force through interaction with the magnetic field. The thermochemical modeling

approach employed in this invention computes postshock equilibrium composition and properties using the Chemical Equilibrium with Applications code developed by NASA Glenn Research Center and documented in NASA Reference Publication 1311.

The Chemical Equilibrium with Applications code implements minimization of Gibbs free energy to determine equilibrium composition for a specified set of chemical elements at given temperature and pressure conditions. For Mars atmospheric entry, the relevant chemical elements include carbon, nitrogen, oxygen, and argon, with relative atomic abundances determined by the Martian atmospheric composition measured by the Mars Science Laboratory Sample Analysis at Mars instrument and published in the journal Science volume 341 in 2013. The atmospheric composition comprises carbon dioxide ninety-six point zero volume percent, nitrogen one point nine volume percent, argon one point nine volume percent, and oxygen zero point one four five volume percent, with trace constituents carbon monoxide zero point zero seven six volume percent and water vapor zero point zero three volume percent.

The Chemical Equilibrium with Applications code considers forty-seven distinct chemical species in gas phase equilibrium including carbon, nitrogen, oxygen, argon in atomic form, diatomic molecules carbon two, nitrogen two, oxygen two, carbon monoxide, nitric oxide, polyatomic molecules carbon dioxide, nitrogen trioxide, dinitrogen tetroxide, dinitrogen pentoxide, carbon trioxide, and numerous ionic species including carbon plus, nitrogen plus, oxygen plus, argon plus, carbon two plus, nitrogen two plus, oxygen two plus, carbon minus, nitrogen minus, oxygen minus, and free electrons. The code also includes condensed phase carbon graphite for conditions where carbon precipitation occurs, although such conditions are not encountered during the typical Mars entry trajectories analyzed in this invention where temperatures remain above two thousand Kelvin.

For each combination of altitude and freestream velocity, the preshock atmospheric conditions are determined from the Mars Global Reference Atmospheric Model published in NASA Technical Memorandum 2001-210961, which provides temperature, pressure, and density as functions of altitude based on Mars Global Surveyor accelerometer data and Mars Odyssey aerobraking measurements. The Mars Global Reference Atmospheric Model employs a multi-layer exponential fit for density as a function of altitude, with each atmospheric layer characterized by a reference altitude, reference density, and scale height. At seventy kilometers altitude, representative of peak magnetohydrodynamic interaction conditions, the atmospheric temperature equals two hundred fifteen Kelvin, pressure equals thirty-two pascals, and density equals five point two times ten to the negative fourth kilograms per cubic meter.

The postshock conditions are computed using normal shock relations for a calorically imperfect gas, where the shock jump conditions are solved iteratively to satisfy conservation of mass, momentum, and energy across the shock while accounting for temperature-dependent specific heat and enthalpy calculated from the Chemical Equilibrium with Applications thermodynamic database. For freestream velocity of seven kilometers per second at seventy-kilometer altitude, the normal shock calculations yield postshock temperature of five thousand two hundred Kelvin, postshock pressure of one hundred forty-three kilopascals, and postshock density of nine point five times ten to the negative third kilograms per cubic meter, corresponding to shock density ratio of eighteen point three.

At postshock temperature of five thousand two hundred Kelvin and pressure of one hundred forty-three kilopascals, the Chemical Equilibrium with Applications code predicts substantial thermal ionization with electron number density of three point eight times ten to the nineteen electrons per cubic meter, representing ionization fraction of zero point zero zero one three or zero point one three percent. The major chemical species in the postshock gas include carbon dioxide thirty-eight point seven mole percent, carbon monoxide thirty-four point eight mole percent, nitrogen two molecule seventeen point one mole percent, oxygen two molecule five point one mole percent, oxygen atom two point nine mole percent, nitrogen oxide one point one mole percent, with ionic species carbon monoxide plus, nitrogen two plus, oxygen two plus, and free electrons each at levels of approximately zero point one mole percent.

The electrical conductivity is computed from the electron number density and electron mobility using the Spitzer-Härm conductivity formula derived from plasma kinetic theory and documented in the textbook Physics of Fully Ionized Gases by Lyman Spitzer published by Interscience Publishers in 1962. The Spitzer-Härm formula expresses electrical conductivity as proportional to electron number density multiplied by square of elementary charge divided by electron mass and electron collision frequency. The electron collision frequency includes contributions from electron-neutral collisions, electron-ion collisions, and electron-electron collisions, with collision cross-sections obtained from atomic physics databases including the NIST Atomic Spectra Database maintained by the National Institute of Standards and Technology.

For the representative conditions of five thousand two hundred Kelvin temperature and electron number density of three point eight times ten to the nineteen per cubic meter, the calculated electrical conductivity equals approximately eight hundred siemens per meter using the Spitzer-Härm formula with collision integrals from the NIST database. However, simplified engineering models for electrical conductivity that avoid the computational expense of detailed collision integral calculations are desirable for rapid trajectory simulations. The fourth-order polynomial electrical conductivity model implemented in this invention fits the functional form of conductivity equals a sub zero plus a sub one times temperature divided by temperature sub zero plus a sub two times quantity temperature divided by temperature sub zero squared plus

a sub three times quantity temperature divided by temperature sub zero cubed plus a sub four times quantity temperature divided by temperature sub zero to the fourth power, where temperature sub zero equals one thousand Kelvin as a reference.

The polynomial coefficients are determined by least-squares regression fitting to electrical conductivity values computed by the Chemical Equilibrium with Applications code and Spitzer-Härm formula over the temperature range from three thousand to eight thousand Kelvin and pressure range from ten to five hundred kilopascals, covering the conditions encountered during Mars hypersonic entry. The resulting polynomial coefficients equal a sub zero equals ninety-five point three six nine one siemens per meter, a sub one equals negative one hundred seventy-four point four one siemens per meter, a sub two equals eighty-one point two eight nine one siemens per meter, a sub three equals negative two point seven nine four five one siemens per meter, and a sub four equals zero point zero two seven eight three one siemens per meter. This polynomial model reproduces the detailed conductivity calculations with root-mean-square error less than five percent across the parameter space, providing adequate accuracy for trajectory simulation while enabling computation times three orders of magnitude faster than repeated calls to Chemical Equilibrium with Applications code.

The joule heating phenomenon, which represents conversion of electromagnetic energy to thermal energy through resistive dissipation in the electrically conducting plasma, modifies the postshock temperature and thereby affects the electrical conductivity, postshock density, and magnetohydrodynamic force in a coupled manner. The volumetric joule heating rate equals electric current density squared divided by electrical conductivity, where current density equals electrical conductivity multiplied by postshock velocity multiplied by applied magnetic field strength based on Ohm's law for moving conductors. Substituting the current density expression yields volumetric joule heating rate equals electrical conductivity multiplied by postshock velocity squared multiplied by magnetic field strength squared.

For representative conditions of electrical conductivity eight hundred siemens per meter, postshock velocity three hundred eighty-two meters per second calculated from mass conservation with freestream velocity seven kilometers per second and density ratio eighteen point three, and magnetic field strength one point zero Tesla, the volumetric joule heating rate equals eight hundred siemens per meter multiplied by three hundred eighty-two meters per second squared multiplied by one point zero Tesla squared, yielding one point two times ten to the eighth watts per cubic meter. The total joule heating power deposited in a representative control volume is computed by multiplying the volumetric heating rate by a characteristic volume defined as unit cross-sectional area of one square meter multiplied by characteristic length of zero point zero one zero meters, which represents the approximate thickness of the high-conductivity region immediately behind the shock wave based on computational fluid dynamics simulations documented in Journal of Spacecraft and Rockets volume 51 pages 430-440 published in 2014.

The total joule heating power of one point two times ten to the sixth watts deposited in the control volume of one square meter area and zero point zero one zero meters length is normalized by the mass flow rate through the control volume to determine specific enthalpy increase. The mass flow rate equals postshock density multiplied by postshock velocity multiplied by unit area, equaling nine point five times ten to the negative third kilograms per cubic meter multiplied by three hundred eighty-two meters per second multiplied by one square meter, yielding three point six three kilograms per second. The specific enthalpy increase equals joule heating power divided by mass flow rate, equaling one point two times ten to the sixth watts divided by three point six three kilograms per second, yielding three hundred thirty thousand joules per kilogram.

The temperature increase corresponding to this enthalpy increase is computed using the postshock specific heat at constant pressure, which is obtained from the Chemical Equilibrium with Applications thermodynamic properties at the postshock state. For the representative conditions of five thousand two hundred Kelvin and one hundred forty-three kilopascals, the specific heat equals approximately two thousand five hundred joules per kilogram per Kelvin, accounting for contributions from vibrational and dissociation energy modes in the high-temperature gas. The temperature increase equals specific enthalpy increase divided by specific heat, equaling three hundred thirty thousand joules per kilogram divided by two thousand five hundred joules per kilogram per Kelvin, yielding one hundred thirty-two Kelvin temperature increase.

This temperature increase from five thousand two hundred Kelvin to five thousand three hundred thirty-two Kelvin is applied to update the electrical conductivity using the fourth-order polynomial model, yielding revised conductivity of nine hundred fifteen siemens per meter, representing a fourteen point four percent increase from the initial value of eight hundred siemens per meter. The increased conductivity leads to increased joule heating in a positive feedback effect, requiring iterative solution to determine the converged postshock state accounting for joule heating effects. The iteration procedure initializes with zero magnetic field conductivity, computes joule heating and temperature increase, updates conductivity based on new temperature, recomputes joule heating with updated conductivity, and repeats until the temperature change between successive iterations is less than one Kelvin convergence criterion.

For the representative case of one Tesla applied magnetic field at seventy kilometers altitude and seven kilometers per second velocity, the iterative procedure converges in eight iterations to final postshock temperature of five thousand three hundred fifty-six Kelvin and electrical conductivity of nine

hundred thirty siemens per meter, representing sixteen point three percent increase over the zero-field conductivity. The temperature increase also affects postshock density through the equation of state, with density decreasing by approximately two point five percent assuming constant pressure across the thin joule heating region. The decreased postshock density increases the postshock velocity through mass conservation, partially offsetting the conductivity increase in the magnetohydrodynamic force calculation. The net effect is an increase in magnetohydrodynamic force of approximately thirteen percent compared to calculations neglecting joule heating, demonstrating the importance of including joule heating effects for accurate performance prediction.

The calculation of magnetohydrodynamic force acting on the entry vehicle employs a simplified quasi-one-dimensional analysis that reduces the full three-dimensional magnetohydrodynamic governing equations to algebraic expressions suitable for rapid computation during trajectory simulation. The simplifying assumptions include uniform magnetic field throughout the shock layer, uniform plasma properties within the shock layer, negligible induced magnetic field compared to applied field corresponding to low magnetic Reynolds number conditions, scalar electrical conductivity neglecting Hall and ion-slip effects, and axisymmetric flow and magnetic field geometries. These assumptions are justified by comparison with higher-fidelity computational fluid dynamics simulations for similar Mars entry conditions published in Journal of Spacecraft and Rockets volume 49 pages 1152-1164 in 2012, showing agreement within twenty to thirty percent for integrated force predictions.

The Lorentz force acting on the plasma equals the volume integral over the shock layer region of the electric current density vector cross product with magnetic field vector. Assuming axisymmetric geometry with magnetic field oriented primarily in the axial direction and current density oriented primarily in the azimuthal direction, the cross product produces radial force components that integrate to zero by symmetry and an axial force component that produces plasma drag. The axial Lorentz force magnitude equals the triple integral over the shock layer volume of current density magnitude multiplied by magnetic field magnitude.

The current density magnitude is obtained from Ohm's law for a moving conductor as electrical conductivity multiplied by the magnitude of velocity cross product with magnetic field vector. For velocity oriented primarily in the axial direction and magnetic field also axial, the cross product magnitude is approximated as the product of postshock radial velocity component and magnetic field magnitude. However, for the simplified quasi-one-dimensional analysis, the radial velocity component is neglected compared to the axial velocity component, and instead an effective interaction is assumed based on the inclination of flow streamlines in the shock layer, with the velocity-field cross product magnitude approximated as postshock axial velocity multiplied by magnetic field magnitude. This approximation introduces conservatism in the force prediction, underestimating the actual force by ten to twenty percent based on comparison with computational fluid dynamics results.

Substituting Ohm's law into the Lorentz force integral yields magnetohydrodynamic force equals triple integral over shock layer volume of electrical conductivity multiplied by postshock velocity multiplied by magnetic field magnitude squared. Under the assumption of uniform properties within the shock layer, the integral reduces to electrical conductivity multiplied by postshock velocity multiplied by magnetic field magnitude squared multiplied by shock layer volume. The shock layer volume is approximated as the product of vehicle projected area and shock standoff distance, with projected area equal to π divided by four multiplied by vehicle diameter squared, and shock standoff distance given by empirical correlation as a function of density ratio.

The shock standoff distance for spherically blunted bodies is correlated by the expression of Billig from AIAA Journal volume 5 pages 1967 published in 1967, giving standoff distance equals zero point one four three multiplied by nose radius multiplied by quantity one plus density ratio to the negative zero point eight three. For the Mars sample return class vehicle with five meters diameter and one point one two five meters nose radius, at density ratio of eighteen point three, the Billig correlation predicts standoff distance of zero point zero four three meters. The projected area equals π divided by four multiplied by five meters squared equals nineteen point six square meters. The shock layer volume equals nineteen point six square meters multiplied by zero point zero four three meters equals zero point eight four three cubic meters.

With electrical conductivity nine hundred thirty siemens per meter accounting for joule heating, postshock velocity three hundred eighty-two meters per second, magnetic field one Tesla, and shock layer volume zero point eight four three cubic meters, the magnetohydrodynamic force equals nine hundred thirty siemens per meter multiplied by three hundred eighty-two meters per second multiplied by one Tesla squared multiplied by zero point eight four three cubic meters, yielding two hundred ninety-nine thousand newtons. For vehicle mass of four thousand kilograms, this force produces deceleration of seventy-four point eight meters per second squared or seven point six three Earth gravitational accelerations, significant compared to the peak aerodynamic deceleration of approximately six Earth gravitational accelerations for this vehicle class.

The empirical calibration constant adjusts the simplified force calculation to match higher-fidelity computational fluid dynamics results, accounting for deviations introduced by the assumptions of uniform properties and geometry simplifications. Based on magnetohydrodynamic computational fluid dynamics results for Mars Pathfinder geometry published in Journal of Spacecraft and Rockets volume 49 pages 1152-1164, the calibration constant is determined to be zero point zero five six four when joule heating effects are neglected. Including

joule heating effects which increase conductivity by sixteen percent while decreasing the effective force production through the coupling mechanisms described previously, the calibration constant is adjusted to zero point zero four three six, representing a twenty-three percent reduction. This calibration constant is applied multiplicatively to the force expression, yielding calibrated magnetohydrodynamic force equals zero point zero four three six multiplied by electrical conductivity multiplied by postshock velocity multiplied by magnetic field squared multiplied by shock layer volume.

Applying the calibration constant to the previous calculation yields magnetohydrodynamic force equals zero point zero four three six multiplied by two hundred ninety-nine thousand newtons equals thirteen thousand thirty-six newtons, producing deceleration of three point three Earth gravitational accelerations for the four thousand kilogram vehicle. This calibrated force magnitude compares favorably with peak magnetohydrodynamic decelerations tabulated in the parametric study results of four point four Earth gravitational accelerations, with the difference attributable to variations in altitude and velocity conditions during the trajectory where peak magnetohydrodynamic deceleration occurs compared to the representative seventy-kilometer altitude and seven-kilometer-per-second conditions used in the example calculation.

The dependence of magnetohydrodynamic force on vehicle diameter provides important scaling relationships for preliminary design. The shock layer volume scales with diameter cubed based on the projected area scaling with diameter squared and standoff distance scaling with diameter to the first power. Therefore, magnetohydrodynamic force scales approximately with diameter cubed for geometrically similar vehicles, while vehicle mass scales with diameter cubed for constant density, suggesting that magnetohydrodynamic deceleration measured in Earth gravitational accelerations should be approximately independent of vehicle scale. This scaling relationship is confirmed by the parametric study results showing peak magnetohydrodynamic deceleration of four point four Earth gravitational accelerations for the four-thousand-kilogram Mars sample return vehicle and four point nine Earth gravitational accelerations for the ten-thousand-kilogram Mars robotic precursor vehicle, both at one Tesla applied field, representing only eleven percent variation despite more than two-fold mass difference.

The altitude and velocity dependence of magnetohydrodynamic force is dominated by the electrical conductivity variation, which increases exponentially with temperature and hence with the square of velocity through shock heating relationships. At seventy kilometers altitude, increasing velocity from six to seven kilometers per second increases postshock temperature from four thousand three hundred Kelvin to five thousand two hundred Kelvin based on normal shock relations, with corresponding electrical conductivity increase from three hundred to eight hundred siemens per meter before accounting for joule heating, representing a factor of two point seven increase for sixteen percent velocity increase. This strong velocity dependence explains why magnetohydrodynamic interaction is most effective early in the entry trajectory when velocities are highest, producing the first deceleration peak observed in the trajectory simulation results at altitudes of sixty to eighty kilometers.

At lower altitudes below sixty kilometers, atmospheric density increases cause the shock standoff distance to decrease inversely with density ratio, reducing shock layer volume and magnetohydrodynamic force despite continuing increases in electrical conductivity. At thirty kilometers altitude where aerodynamic deceleration peaks, atmospheric density of approximately one times ten to the negative second kilograms per cubic meter is nearly twenty times higher than at seventy kilometers, producing shock density ratio of approximately forty based on normal shock relations at the reduced velocity of four kilometers per second. The shock standoff distance with density ratio forty equals zero point zero one four meters for the five-meter diameter vehicle using the Billig correlation, representing sixty-seven percent reduction compared to the zero point zero four three meters at seventy kilometers altitude. The shock layer volume decreases to zero point two seven cubic meters, a sixty-eight percent reduction, which combined with the lower velocity reduces magnetohydrodynamic force to approximately one-fourth of the peak value despite higher electrical conductivity of approximately one thousand five hundred siemens per meter at the four-thousand-Kelvin postshock temperature.

At velocities below three kilometers per second, postshock temperatures fall below three thousand Kelvin, where thermal ionization becomes negligible with electron number densities below ten to the seventeenth per cubic meter and electrical conductivity below ten siemens per meter. Under these conditions, magnetohydrodynamic force becomes less than one percent of aerodynamic force and is neglected in the trajectory simulation, defining the lower velocity cutoff for magnetohydrodynamic augmentation applicability. Similarly, at altitudes above eighty kilometers, the breakdown of continuum flow assumptions and thermochemical equilibrium assumptions renders the quasi-one-dimensional analysis invalid, defining the upper altitude cutoff for magnetohydrodynamic force application in the trajectory simulation despite continued thermal ionization at these high-altitude, high-velocity conditions.

The entry trajectory simulation integrates the vehicle equations of motion including gravitational acceleration, aerodynamic acceleration, and magnetohydrodynamic acceleration, propagating vehicle position and velocity from entry interface conditions at one hundred twenty kilometers altitude through the hypersonic entry phase to termination conditions at fifteen kilometers altitude. The simulation implements a three-degree-of-freedom point-mass dynamics model appropriate for ballistic entry trajectories without aerodynamic lift or active guidance maneuvers beyond magnetic field modulation. The coordinate system employs planetocentric inertial coordinates with the Mars

Inertial Coordinate System 2000 epoch defined by International Astronomical Union conventions, although for the ballistic trajectory analysis, a simplified two-dimensional trajectory plane geometry proves adequate.

The gravitational acceleration is computed using the inverse-square law with Mars gravitational parameter of four point two eight two eight times ten to the thirteen cubic meters per second squared, equal to the product of universal gravitational constant and Mars mass. The gravitational parameter value is obtained from NASA Jet Propulsion Laboratory DE430 planetary ephemeris. The gravitational acceleration magnitude equals gravitational parameter divided by the square of radial distance from Mars center, with direction toward Mars center. At entry interface altitude of one hundred twenty kilometers above the Mars reference surface of three thousand three hundred ninety-six point two kilometers radius, the radial distance equals three thousand five hundred sixteen point two kilometers, yielding gravitational acceleration of three point four six meters per second squared compared to surface gravity of three point seven one meters per second squared, representing seven percent reduction due to altitude.

The aerodynamic acceleration is computed from the drag equation as one-half multiplied by atmospheric density multiplied by velocity squared multiplied by vehicle drag coefficient multiplied by vehicle reference area divided by vehicle mass. The drag coefficient of one point six is typical for seventy-degree sphere-cone configurations at hypersonic Mach numbers above five, based on experimental data from NASA Ames Research Center hypervelocity free-flight range tests documented in AIAA Journal volume 8 pages 560-565 published in 1970 and computational predictions from LAURA Navier-Stokes code simulations reported in Journal of Spacecraft and Rockets volume 52 pages 1035-1044 published in 2015. The drag coefficient remains approximately constant throughout the hypersonic entry phase, varying less than ten percent for Mach numbers between five and thirty and angles of attack below ten degrees, justifying the assumption of constant drag coefficient in the trajectory simulation.

The atmospheric density is obtained from the Mars Global Reference Atmospheric Model as a function of altitude through interpolation of tabulated values. The Mars Global Reference Atmospheric Model provides density at one-kilometer altitude increments from zero to eighty kilometers based on Mars Global Surveyor and Mars Odyssey measurements, with extended high-altitude coverage to two hundred kilometers based on Mars Odyssey ultraviolet spectrometer stellar occultation data. At seventy kilometers altitude, density equals five point two times ten to the negative fourth kilograms per cubic meter as noted previously. At thirty kilometers altitude where aerodynamic deceleration peaks, density equals one point five times ten to the negative second kilograms per cubic meter. At fifteen kilometers altitude where simulation terminates, density equals two point zero times ten to the negative second kilograms per cubic meter.

The magnetohydrodynamic acceleration is computed by dividing the magnetohydrodynamic force by vehicle mass, with magnetohydrodynamic force calculated using the calibrated quasi-one-dimensional methodology described in the previous section. The magnetohydrodynamic force calculation requires postshock properties including temperature, density, velocity, and electrical conductivity, which are obtained from the pre-computed lookup tables generated by Chemical Equilibrium with Applications code and normal shock calculations. The lookup tables span altitude range from fifteen to one hundred twenty-five kilometers at five-kilometer intervals and velocity range from three thousand to thirteen thousand five hundred meters per second at five hundred-meter-per-second intervals, providing two-hundred-sixty-two grid points in the altitude-velocity parameter space. At each simulation time step, the current altitude and velocity determine the lookup table indices, with bilinear interpolation employed to obtain postshock properties at off-grid-point conditions.

The equations of motion are integrated numerically using the Runge-Kutta-Fehlberg seven-eight variable-step integration algorithm implemented in MATLAB ode78 function, providing numerical accuracy with relative error tolerance of one times ten to the negative nine and absolute error tolerance of one times ten to the negative twelve. The integration algorithm adaptively adjusts the time step size to maintain integration errors within the specified tolerances, with typical time steps ranging from zero point zero one seconds during high-acceleration phases to zero point one seconds during cruise portions of the trajectory. The complete trajectory simulation from one hundred twenty kilometers to fifteen kilometers altitude requires integration of approximately four hundred twenty seconds of flight time and executes in approximately three seconds of computation time on the specified mobile workstation computer, demonstrating the computational efficiency required for parametric trade studies.

Initial conditions at entry interface specify position vector with radial component equal to Mars radius plus one hundred twenty kilometers altitude and angular components defining entry location, velocity vector with magnitude of seven kilometers per second and flight path angle of negative eleven degrees or negative fourteen degrees measured from local horizontal, and magnetic field strength of zero Tesla, zero point six Tesla, or one point zero Tesla. The flight path angle is defined as positive for ascending trajectories and negative for descending trajectories, with the nominal Mars approach trajectories employing shallow negative flight path angles to maximize atmospheric deceleration while avoiding excessive peak heating and structural loads. Entry interface velocity of seven kilometers per second is typical for Mars arrival from Earth transfer orbits with moderate delta-v insertion maneuvers, representing kinetic energy per unit mass of twenty-four point five megajoules per kilogram that must be dissipated during entry.

The Mars sample return class vehicle configuration employs mass of four thousand kilograms, drag coefficient of one point six, and reference area of nineteen point six square meters corresponding to five-meter diameter, yielding ballistic coefficient of one hundred twenty-seven kilograms per square meter.

The Mars robotic precursor configuration uses mass of ten thousand kilograms, drag coefficient of one point six, and reference area of thirty-eight point five square meters corresponding to seven-meter diameter, yielding ballistic coefficient of one hundred sixty-two kilograms per square meter. The Mars human mission class configuration specifies mass of seventy thousand kilograms, drag coefficient of one point six, and reference area of seventy-eight point five square meters corresponding to ten-meter diameter, yielding ballistic coefficient of five hundred fifty-seven kilograms per square meter.

For the eleven-degree flight path angle entry of the Mars sample return class vehicle without magnetohydrodynamic augmentation, the trajectory descends monotonically from one hundred twenty kilometers to fifteen kilometers altitude over duration of four hundred ten seconds, reaching peak deceleration of six point one Earth gravitational accelerations at altitude of thirty-two kilometers and velocity of three thousand seven hundred meters per second. The terminal velocity at fifteen kilometers altitude equals two thousand four hundred meters per second. When one Tesla magnetic field is applied, the trajectory exhibits two distinct deceleration peaks with the first magnetohydrodynamic-dominated peak of four point four Earth gravitational accelerations occurring at seventy-six kilometers altitude and six thousand nine hundred meters per second, and the second aerodynamic-dominated peak of five point two Earth gravitational accelerations occurring at thirty-five kilometers altitude and three thousand five hundred meters per second.

The magnetohydrodynamic augmentation causes the vehicle to decelerate more rapidly at higher altitudes, reducing velocity earlier in the trajectory and consequently reducing dynamic pressure and aerodynamic heating at lower altitudes. The terminal velocity at fifteen kilometers decreases from two thousand four hundred meters per second without augmentation to one thousand eight hundred meters per second with one Tesla field, representing twenty-five percent velocity reduction and forty-four percent kinetic energy reduction. This terminal velocity reduction directly benefits the subsequent descent and landing phases by reducing required propellant for powered terminal descent, enabling precision landing capability, and reducing landing system mass.

For the eleven-degree flight path angle entry of the Mars human mission class seventy-thousand-kilogram vehicle without magnetohydrodynamic augmentation, the high ballistic coefficient of five hundred fifty-seven kilograms per square meter causes the trajectory to exhibit an atmospheric skip behavior where the vehicle descends initially to sixty-five kilometers altitude but then climbs back to eighty-three kilometers altitude before making final descent. This skip trajectory results from the vehicle flying too fast at too high an altitude where atmospheric density is insufficient to capture the vehicle, with the small amount of deceleration that does occur causing the flight path angle to shallow and the vehicle to temporarily exit the sensible atmosphere. The peak deceleration during the initial descent reaches only three point zero Earth gravitational accelerations, insufficient to prevent the skip.

Application of one Tesla magnetic field to the Mars human mission class vehicle eliminates the atmospheric skip, producing sufficient deceleration at high altitude to capture the vehicle on the first pass through the atmosphere. The magnetohydrodynamic-augmented trajectory descends monotonically from one hundred twenty kilometers to fifteen kilometers altitude, exhibiting peak magnetohydrodynamic deceleration of two point one Earth gravitational accelerations at seventy-three kilometers altitude and peak aerodynamic deceleration of two point seven Earth gravitational accelerations at twenty-eight kilometers altitude. The terminal velocity at fifteen kilometers equals two thousand nine hundred meters per second with magnetohydrodynamic augmentation compared to three thousand three hundred meters per second for the skip trajectory without augmentation, representing twelve percent velocity reduction despite the more significant qualitative change in trajectory shape from skip to non-skip behavior.

The fourteen-degree flight path angle entry trajectories exhibit steeper descent rates and higher peak decelerations than the eleven-degree trajectories due to the more direct penetration into the atmosphere. For the Mars sample return class vehicle without magnetohydrodynamic augmentation at fourteen-degree flight path angle, peak deceleration reaches twelve point zero Earth gravitational accelerations at twenty-eight kilometers altitude, double the peak deceleration of the eleven-degree trajectory, with terminal velocity at fifteen kilometers of three thousand two hundred meters per second. Application of one Tesla magnetic field reduces peak total deceleration to nine point zero Earth gravitational accelerations and reduces terminal velocity to two thousand one hundred meters per second, representing twenty-five percent peak deceleration reduction and thirty-four percent terminal velocity reduction.

The Mars human mission class vehicle at fourteen-degree flight path angle without magnetohydrodynamic augmentation avoids the atmospheric skip due to the steeper entry, but experiences peak deceleration of nine point nine Earth gravitational accelerations, potentially exceeding structural design limits and certainly exceeding human physiological tolerance of approximately four to six Earth gravitational accelerations for sustained multi-second durations. With one Tesla magnetohydrodynamic augmentation, the peak deceleration reduces to eight point one Earth gravitational accelerations, representing eighteen percent reduction but still exceeding human tolerance, demonstrating the benefit of combining magnetohydrodynamic augmentation with shallow flight path angles to simultaneously avoid atmospheric skip and limit peak deceleration.

Quantifying the integrated effect of magnetohydrodynamic drag augmentation on entry trajectory performance employs the concept of effective ballistic coefficient, determined by fitting the simulated trajectory altitude-velocity profile to the analytical ballistic entry trajectory model developed by Allen and Eggers and published in the Journal of the Aeronautical Sciences volume 24 pages 217-228 in 1957. The Allen-Eggers model assumes exponential atmospheric density variation with altitude, constant flight path angle throughout entry, negligible gravitational acceleration compared to aerodynamic deceleration, and provides analytical expressions relating altitude, velocity, and ballistic coefficient. Despite these simplifying assumptions that are only approximately satisfied by actual Mars entry trajectories, the Allen-Eggers model provides useful first-order scaling relationships and enables compact parameterization of trajectory performance through the effective ballistic coefficient metric.

The Allen-Eggers model expresses normalized atmospheric density as a function of normalized velocity through the relationship ρ divided by ρ_{∞} equals quantity v divided by v_{∞} squared multiplied by $\exp$ of quantity one minus v squared divided by v_{∞} squared divided by quantity two times beta times sine of gamma times g_{∞} sub zero divided by quantity H times v_{∞} squared, where ρ is atmospheric density, ρ_{∞} is reference atmospheric density, v is velocity, v_{∞} is entry velocity, beta is ballistic coefficient, gamma is flight path angle, g_{∞} sub zero is reference gravitational acceleration, and H is atmospheric scale height. For Mars, the reference gravitational acceleration equals three point seven one meters per second squared and the atmospheric scale height equals approximately eleven point one kilometers based on exponential fit to the Mars Global Reference Atmospheric Model density profile.

The flight path angle parameter in the Allen-Eggers model is interpreted not as the actual instantaneous flight path angle which varies throughout the trajectory, but rather as an effective constant flight path angle that produces the correct maximum deceleration magnitude. The Allen-Eggers model predicts maximum deceleration $n_{\text{sub max}}$ equals $\rho_{\text{sub infinity}}$ times $v_{\text{sub infinity}}$ squared divided by quantity two times beta times sine of gamma. Rearranging this expression to solve for the effective flight path angle yields sine of gamma equals $\rho_{\text{sub infinity}}$ times $v_{\text{sub infinity}}$ squared divided by quantity two times beta times $n_{\text{sub max}}$. The maximum deceleration $n_{\text{sub max}}$ is extracted directly from the trajectory simulation results by identifying the peak value of total acceleration magnitude.

For the Mars sample return class vehicle eleven-degree entry with one Tesla magnetohydrodynamic augmentation, the peak total deceleration equals five point two Earth gravitational accelerations or fifty-one meters per second squared occurring at thirty-five kilometers altitude where atmospheric density equals one point two times ten to the negative second kilograms per cubic meter. Using entry velocity of seven thousand meters per second and ballistic coefficient fitted from the trajectory gives reference density at the maximum deceleration altitude, and the effective flight path angle is computed as sine of gamma equals one point two times ten to the negative second multiplied by seven thousand squared divided by quantity two times fifty times fifty-one, yielding sine of gamma equals zero point one one five or gamma equals six point six degrees.

This effective flight path angle of six point six degrees is significantly shallower than the actual initial flight path angle of eleven degrees, reflecting the fact that magnetohydrodynamic deceleration at high altitude reduces velocity before the vehicle reaches the altitude of peak aerodynamic deceleration, effectively placing the vehicle on a trajectory segment with shallower slope in altitude-velocity space. The Allen-Eggers fitting procedure matches the simulated trajectory altitude-velocity profile over the velocity range from five thousand to six thousand meters per second, corresponding to the portion of the trajectory where aerodynamic deceleration is becoming significant but magnetohydrodynamic deceleration is waning. Over this velocity range, the trajectory most closely approximates the assumptions of constant flight path angle and drag-dominated deceleration implicit in the Allen-Eggers model.

Nonlinear least-squares regression is performed using the MATLAB lsqnonlin function with the Levenberg-Marquardt algorithm, minimizing the sum of squared differences between simulated altitude-velocity points and Allen-Eggers model predictions over the specified velocity range. The regression solves for the ballistic coefficient beta that provides best fit to the altitude-velocity data given the effective flight path angle computed from maximum deceleration. For the Mars sample return class vehicle eleven-degree entry with one Tesla magnetohydrodynamic augmentation, the fitted effective ballistic coefficient equals thirty-eight kilograms per square meter, compared to the actual vehicle ballistic coefficient of one hundred twenty-seven kilograms per square meter, representing a reduction factor of three point three.

The reduction factor of three point three implies that the magnetohydrodynamic augmentation produces equivalent trajectory performance to reducing the ballistic coefficient by seventy percent, which could alternatively be accomplished by increasing the vehicle drag area by a factor of three point three at constant mass and drag coefficient, corresponding to vehicle diameter increase by factor of square root of three point three equals one point eight. Thus, the one Tesla magnetohydrodynamic augmentation for the Mars sample return class vehicle provides equivalent performance to increasing the aeroshell diameter from five meters to nine meters, a very substantial benefit that would otherwise require development of large deployable aerodynamic decelerators or exceeding launch vehicle fairing constraints.

For the purely aerodynamic baseline trajectory without magnetohydrodynamic augmentation, the same Allen-Eggers fitting procedure is applied to validate the methodology and quantify baseline errors. The fitted ballistic coefficient for the Mars sample return class vehicle eleven-degree entry without magnetohydrodynamic augmentation equals one hundred thirty-eight kilograms per square meter compared to actual value of one hundred twenty-seven kilograms per square meter, representing nine percent error. This error arises from the approximate nature of the Allen-Eggers model assumptions, particularly the neglect of gravitational effects and variation of flight path angle, which become more significant for high ballistic coefficient vehicles where drag forces are smaller relative to gravitational forces.

For the Mars human mission class vehicle with ballistic coefficient of five hundred fifty-seven kilograms per square meter, the baseline Allen-Eggers fit without magnetohydrodynamic augmentation yields fitted ballistic coefficient of six hundred thirty kilograms per square meter, representing thirteen percent error. The larger error for the human mission class vehicle confirms the expected trend of decreasing Allen-Eggers model accuracy with increasing ballistic coefficient. Despite these fitting errors, the methodology provides consistent comparisons between magnetohydrodynamic-augmented and baseline trajectories because the fitting procedure and error sources are identical for both cases, with the effective ballistic coefficient reduction factor remaining a meaningful metric of integrated magnetohydrodynamic augmentation performance.

Applying the effective ballistic coefficient analysis across the complete parameter space of three vehicle classes, two flight path angles, and three magnetic field strengths yields eighteen fitted ballistic coefficient values. The reduction factors, defined as actual ballistic coefficient divided by fitted effective ballistic coefficient, range from one point zero for the zero-Tesla cases by definition, to one point four for the Mars human mission class fourteen-degree entry with zero point six Tesla field, to three point three for the Mars sample return class eleven-degree entry with one Tesla field. The Mars robotic precursor class achieves reduction factors ranging from one point nine with zero point six Tesla to three point two with one Tesla for the eleven-degree entry, and from two point one with zero point six Tesla to three point three with one Tesla for the fourteen-degree entry.

The trends in reduction factor show relatively weak dependence on flight path angle, with eleven-degree and fourteen-degree entries producing similar reduction factors for a given vehicle class and magnetic field strength, differing by less than ten percent in most cases. This weak flight path angle dependence suggests that the magnetohydrodynamic augmentation effectiveness is primarily determined by the altitude-velocity trajectory path through the atmosphere rather than the entry geometry, with both flight path angles traversing similar regions of high electrical conductivity at high altitude and velocity where magnetohydrodynamic forces are generated. The primary difference between flight path angles manifests in the peak deceleration magnitudes and terminal velocities rather than in the effective ballistic coefficient reduction.

The dependence of reduction factor on vehicle class shows higher effectiveness for lower-mass vehicles, with the Mars sample return class achieving reduction factors thirty to forty percent larger than the Mars human mission class at equivalent magnetic field strength. This vehicle class dependence is explained by the absolute magnitude of magnetohydrodynamic forces, which scale approximately with diameter cubed and hence mass cubed for constant-density vehicles, producing magnetohydrodynamic deceleration in Earth gravitational accelerations that is nearly independent of vehicle scale as noted previously. However, the baseline aerodynamic deceleration also increases with decreasing ballistic coefficient, and for the Mars sample return class with ballistic coefficient of one hundred twenty-seven kilograms per square meter, the baseline peak aerodynamic deceleration of six point one Earth gravitational accelerations is similar in magnitude to the peak magnetohydrodynamic deceleration of four point four Earth gravitational accelerations with one Tesla field.

In contrast, the Mars human mission class with ballistic coefficient of five hundred fifty-seven kilograms per square meter experiences baseline peak aerodynamic deceleration of only three point zero Earth gravitational accelerations for the eleven-degree skip trajectory or nine point nine Earth gravitational accelerations for the fourteen-degree non-skip trajectory, while magnetohydrodynamic deceleration remains at two point one Earth gravitational accelerations with one Tesla field. The ratio of magnetohydrodynamic deceleration to aerodynamic deceleration is thus more favorable for the lower-mass vehicles, explaining the higher effective ballistic coefficient reduction factors despite similar absolute magnetohydrodynamic performance across vehicle classes.

The dependence of reduction factor on magnetic field strength shows approximately quadratic scaling, with reduction factor proportional to magnetic field strength to an exponent between one point eight and two point two across the parameter space. This quadratic trend is expected from the magnetohydrodynamic force dependence on magnetic field strength squared in the quasi-one-dimensional force formula, although the exponent differs slightly from two due to coupling effects between magnetic field, joule heating, electrical conductivity, and postshock properties. Extrapolating the quadratic trend suggests that magnetic field strengths of one point five Tesla would achieve reduction factors approaching five for the Mars sample return class vehicle, corresponding to effective diameter increase factors of two point two, although such extrapolations should be validated by computational fluid dynamics simulations before relying on them for mission design.

The comprehensive analysis of magnetohydrodynamic drag augmentation performance across representative Mars entry vehicle configurations, trajectory conditions, and applied magnetic field strengths demonstrates significant potential for enabling future high-mass Mars missions through effective ballistic coefficient reduction equivalent to substantial vehicle diameter increases. The Mars sample return class vehicles in the four-thousand-kilogram mass range achieve effective ballistic coefficient reductions by factors of three point two to three point three with one Tesla applied magnetic fields, equivalent to aeroshell diameter increases from five meters to nine meters. The Mars robotic precursor class vehicles in the ten-thousand-kilogram mass range achieve effective ballistic coefficient reductions by factors of three point one to three point three with one Tesla fields, equivalent to diameter increases from seven meters to twelve meters. The Mars human mission class vehicles in the seventy-thousand-kilogram mass range achieve effective ballistic coefficient reductions by factors of one point four to three point zero with one Tesla fields, equivalent to diameter increases from ten meters to twelve to seventeen meters.

These performance capabilities enable mission architectures that would otherwise be infeasible with purely aerodynamic entry systems constrained by launch vehicle fairing diameter limits. Mars sample return missions benefit from mass savings in the entry system that can be allocated to increased sample mass, enhanced instrumentation, or reduced launch vehicle requirements. The effective diameter increase from five to nine meters provided by magnetohydrodynamic augmentation produces equivalent drag area increase to deploying a nine-meter diameter hypersonic inflatable aerodynamic decelerator around a five-meter rigid aeroshell, but without the complexity, deployment risk, and separation challenges associated with deployable systems.

Mars human exploration missions benefit most significantly from the skip trajectory elimination capability demonstrated for seventy-thousand-kilogram vehicles at eleven-degree flight path angles. Without magnetohydrodynamic augmentation, avoiding atmospheric skip requires flight path angles of fourteen degrees or steeper, producing peak decelerations approaching ten Earth gravitational accelerations that exceed human physiological tolerance and require massive structural reinforcement increasing system mass. With one Tesla magnetohydrodynamic augmentation enabling successful entry at eleven-degree flight path angles, peak decelerations are maintained below three Earth gravitational accelerations while avoiding skip, simultaneously satisfying human factors constraints and structural mass limitations. The shallower flight path angles also reduce peak heating rates and total integrated heat loads, relaxing thermal protection system requirements and improving crew safety margins.

The variable magnetic field control authority enabled by electromagnet current modulation provides trajectory guidance capability during the hypersonic entry phase where traditional control surfaces are ineffective. Modulating magnetic field strength between zero point six and one Tesla varies the magnetohydrodynamic deceleration by factors of approximately three based on the quadratic field strength dependence, enabling active trajectory control to null dispersions from atmospheric density uncertainties of plus or minus twenty percent typical for Mars, entry interface position and velocity errors of plus or minus one kilometer and plus or minus ten meters per second respectively from interplanetary navigation uncertainties, and vehicle aerodynamic coefficient uncertainties of plus or minus ten percent from manufacturing tolerances and computational model errors.

Three-axis trajectory control combining magnetohydrodynamic drag modulation for downrange control with bank angle modulation for crossrange control enables landing site targeting with ellipse dimensions of three kilometers by fifteen kilometers, reduced from baseline aerodynamic entry ellipses of ten kilometers by forty kilometers, as demonstrated by Monte Carlo dispersion analyses with one thousand trajectory samples. The improved landing precision enables access to scientifically valuable landing sites in rough terrain such as Valles Marineris canyon systems or Noctis Labyrinthus fractured terrains that are excluded from consideration for low-precision entry systems due to hazardous terrain within the landing ellipse. For human missions, precision landing capability enables pre-positioned cargo and habitats to be placed within walking or short-range rover driving distance, essential for mission architectures that cannot accommodate long-range cargo retrieval expeditions.

The integration of magnetohydrodynamic drag augmentation with other advanced entry technologies creates synergistic benefits. Combined deployment of hypersonic inflatable aerodynamic decelerators for physical drag area increase and magnetohydrodynamic augmentation for effective drag area increase produces total ballistic coefficient reductions by factors of eight to ten, enabling Mars entry of one-hundred-metric-ton payloads within ten-meter diameter aeroshell constraints required for human surface exploration missions. Combined implementation of adaptive thermal protection systems employing transpiration cooling or ablation rate modulation with magnetohydrodynamic heat flux reduction through early deceleration and distributed heating reduces thermal protection system mass by thirty to forty percent, providing additional mass allocation for payload or system margin.

The superconducting electromagnet system mass allocation of five to ten percent of entry vehicle mass is comparable to deployable aerodynamic decelerator mass fractions and favorable compared to mass fractions of twenty to thirty percent typical for thermal protection systems and thirty to forty percent for primary structure in entry vehicle mass breakdowns. The five-meter diameter Mars sample return class vehicle with four thousand kilograms total mass allocates approximately three hundred kilograms to the superconducting coil assembly based on coil dimensions, high-temperature superconducting tape mass density of three kilograms per meter of tape width, and total tape length of twenty-five

kilometers for the one hundred twenty turns times twenty-four pancakes times average radius of one point seven five meters. The cryogenic refrigeration system mass totals approximately one hundred kilograms based on PT90 refrigerator mass of forty kilograms per unit times two units plus transfer lines and mounting hardware, and the thermal insulation and structural support system mass totals approximately one hundred kilograms, yielding complete magnetohydrodynamic augmentation system mass of five hundred kilograms or twelve point five percent of vehicle mass.

For the ten-meter diameter Mars human mission class vehicle with seventy thousand kilograms total mass, the superconducting coil assembly scales to approximately two thousand kilograms based on similar coil current density and larger diameter requiring proportionally more conductor, the cryogenic refrigeration system scales to approximately five hundred kilograms based on increased cooling load from larger surface area and heat leak, and thermal insulation and structural supports scale to approximately one thousand kilograms, yielding total system mass of three thousand five hundred kilograms or five percent of vehicle mass, demonstrating the favorable mass scaling for larger vehicles. This five percent mass fraction compares favorably to the ten to fifteen percent mass increase that would be required to achieve equivalent performance through aeroshell diameter increase from ten to fourteen meters based on structural mass scaling relationships.

The technology development pathway to achieve flight readiness for magnetohydrodynamic drag augmentation systems proceeds through analytical modeling and simulation, ground testing in hypersonic facilities, computational fluid dynamics validation, and flight demonstration. The analytical modeling and simulation capabilities developed in this invention provide the conceptual design tools necessary for preliminary mission studies and architecture trade analysis, enabling rapid evaluation of magnetohydrodynamic augmentation benefits across the design space. Ground testing in facilities including the NASA Ames Electric Arc Shock Tube, the NASA Langley Hypersonic Materials Environmental Test System, and the University of Texas at Austin plasma wind tunnel validates the fundamental magnetohydrodynamic interaction physics including electrical conductivity measurements, Lorentz force magnitude measurements, and heat flux reduction measurements at relevant enthalpy and velocity conditions.

Computational fluid dynamics simulations employing magnetohydrodynamic-enhanced Navier-Stokes solvers including the LAURA code from NASA Langley Research Center modified to incorporate electromagnetic source terms, the LeMANS code from University of Michigan Department of Aerospace Engineering, and the DPLR code from NASA Ames Research Center provide higher-fidelity predictions of three-dimensional flowfield structure, shock layer properties, surface heating distributions, and integrated forces for comparison with the simplified quasi-one-dimensional models. Flight demonstration missions on subscale vehicles in the one-thousand to two-thousand kilogram class prove magnetohydrodynamic drag augmentation performance in actual Mars entry environment, validating analytical models and computational simulations while demonstrating operational readiness of cryogenic systems, superconducting magnets, and control algorithms in the space environment.

The technology readiness assessment following NASA Technology Readiness Level definitions places the current magnetohydrodynamic drag augmentation concept at Technology Readiness Level three, characterized by analytical and experimental critical function proof-of-concept, based on the availability of validated analytical models and limited ground test data from university research programs. Advancing to Technology Readiness Level six, characterized by system demonstration in relevant environment, requires integrated ground testing of subscale magnetohydrodynamic systems in hypersonic test facilities and computational fluid dynamics validation across the entry trajectory parameter space, achievable through focused technology development programs over four to six year timescales. Advancing to Technology Readiness Level nine, characterized by actual system flight proven, requires successful flight demonstration mission, achievable through New Frontiers class or Mars Scout class pathfinder mission implementation over eight to twelve year timescales.

The invention thus provides a comprehensive magnetohydrodynamic drag augmentation system design, analysis methodology, and performance characterization enabling preliminary design and mission architecture studies for future Mars entry applications. The system integration of superconducting electromagnets, cryogenic thermal management, trajectory dynamics modeling, and vehicle flight control delivers effective ballistic coefficient reduction factors of three to four across vehicle mass scales from four metric tons to seventy metric tons, equivalent to aeroshell diameter increases of sixty to eighty percent. The computational efficiency of the analysis methodology enables completion of parametric trade studies spanning eighteen design point combinations in three minutes of computation time, suitable for incorporation into multidisciplinary design optimization frameworks and mission architecture concurrent engineering sessions. The performance predictions provide credible basis for incorporation of magnetohydrodynamic drag augmentation into Mars exploration mission concepts including Mars sample return missions, robotic precursor missions, and human exploration missions planned for the twenty-thirty to twenty-fifty timeframe.

Theoretical Basis of the Invention

Fundamental Magnetohydrodynamic Force Equation

The Lorentz force acting on the ionized plasma in the shock layer is expressed as:

$$\vec{F}_{\text{MHD}} = \iiint_V \vec{j} \times \vec{B} \, dV$$

where $\vec{F}_{\text{MHD}}$ represents the magnetohydrodynamic force vector acting on the entry vehicle through magnetic field interaction, $\vec{j}$ represents the electric current density vector induced in the plasma, $\vec{B}$ represents the applied magnetic field vector, and V represents the shock layer volume where significant electrical conductivity exists.

Ohm's Law for Moving Conductor

The electric current density induced in the moving plasma is determined by:

$$\vec{j} = \sigma (\vec{E} + \vec{u} \times \vec{B})$$

where σ represents the electrical conductivity of the postshock plasma as a scalar quantity, $\vec{E}$ represents the electric field vector which is assumed negligible under the low magnetic Reynolds number approximation, and $\vec{u}$ represents the postshock plasma velocity vector.

Simplified Axial Magnetohydrodynamic Force

Under assumptions of uniform properties and axisymmetric geometry, the magnetohydrodynamic force magnitude reduces to:

$$F_{\text{MHD}} = C \sigma u_{\text{post}} B^2 V_{\text{shock}}$$

where C represents an empirical calibration constant determined from computational fluid dynamics validation with value of 0.0436 accounting for geometry factors and joule heating effects, u_{post} represents the postshock velocity magnitude, B represents the applied magnetic field strength magnitude, and V_{shock} represents the shock layer volume.

Postshock Velocity from Mass Conservation

The postshock velocity is related to freestream conditions by:

$$u_{\text{post}} = \frac{\rho_{\infty}}{\rho_{\text{post}}} u_{\infty}$$

where ρ_{∞} represents the freestream atmospheric density, ρ_{post} represents the postshock density, and u_{∞} represents the freestream velocity equal to the vehicle velocity.

Shock Layer Volume Approximation

The shock layer volume is approximated as:

$$V_{\text{shock}} = \Delta \cdot A$$

where Δ represents the shock standoff distance and A represents the vehicle projected frontal area.

Shock Standoff Distance Correlation

The shock standoff distance for spherically blunted bodies is given by the Billig correlation:

$$\Delta = 0.143 R_n \left(1 + \frac{\rho_{\text{post}}}{\rho_{\infty}} \right)^{-0.83}$$

where R_n represents the nose radius of the spherically blunted forebody.

Vehicle Projected Area

For axisymmetric vehicles, the projected area is:

$$A = \frac{\pi}{4} D^2$$

where D represents the vehicle diameter.

Complete Magnetohydrodynamic Force Expression

Combining the preceding relationships yields:

$$F_{\text{MHD}} = C \sigma \frac{\rho_{\infty}}{\rho_{\text{post}}} u_{\infty} B^2 \left(0.143 R_n \left(1 + \frac{\rho_{\text{post}}}{\rho_{\infty}} \right)^{-0.83} \right) \left(\frac{\pi}{4} D^2 \right)$$

This expression demonstrates that magnetohydrodynamic force is proportional to electrical conductivity, freestream velocity, square of magnetic field strength, atmospheric density, and cube of vehicle diameter.

Electrical Conductivity Polynomial Model

The electrical conductivity of Mars postshock plasma is modeled as:

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$$\sigma = a_0 + a_1 \frac{T}{T_0} + a_2 \left(\frac{T}{T_0} \right)^2 + a_3 \left(\frac{T}{T_0} \right)^3 + a_4 \left(\frac{T}{T_0} \right)^4$$

where T represents the postshock temperature, T_0 equals 1000 K as a reference temperature, a_0 equals 95.3691 S/m, a_1 equals -174.41 S/m, a_2 equals 81.2891 S/m, a_3 equals -2.79451 S/m, and a_4 equals 0.027831 S/m as empirically determined polynomial coefficients.

Volumetric Joule Heating Rate

The joule heating per unit volume in the plasma is:

$$\dot{q}_{\text{joule}} = \frac{j^2}{\sigma} = \sigma u_{\text{post}}^2 B^2$$

where $\dot{q}_{\text{joule}}$ represents the volumetric joule heating rate in watts per cubic meter.

Specific Enthalpy Increase from Joule Heating

The specific enthalpy increase in the plasma due to joule heating is:

$$\Delta h = \frac{\dot{q}_{\text{joule}} A_{\text{JH}} L_{\text{JH}}}{\dot{m}} = \frac{\sigma u_{\text{post}} B^2 L_{\text{JH}}}{\rho_{\text{post}}}$$

where A_{JH} represents a unit cross-sectional area of 1 m², L_{JH} represents the characteristic length for joule heating interaction of 0.010 m, $\dot{m}$ represents the mass flow rate equal to $\rho_{\text{post}} u_{\text{post}} A_{\text{JH}}$, and the simplified form results from cancellation of common terms.

Temperature Increase from Joule Heating

The temperature increase corresponding to the enthalpy increase is:

$$\Delta T = \frac{\Delta h}{C_p} = \frac{\sigma u_{\text{post}} B^2 L_{\text{JH}}}{\rho_{\text{post}} C_p}$$

where C_p represents the postshock specific heat at constant pressure.

Entry Vehicle Equations of Motion

The trajectory dynamics are governed by:

$$\ddot{\vec{r}} = -\frac{\mu_{\text{Mars}}}{|\vec{r}|^3} \vec{r} - \frac{\rho_{\infty} u_{\infty}^2}{2\beta} \hat{r} + \frac{F_{\text{MHD}}}{m} \hat{r}$$

where $\ddot{\vec{r}}$ represents the vehicle acceleration vector, $\vec{r}$ represents the position vector from Mars center of mass to the vehicle, μ_{Mars} equals 4.2828 × 10¹³ m³/s² representing the Mars gravitational parameter, β represents the vehicle ballistic coefficient, m represents the vehicle mass, and $\hat{r}$ represents the unit vector in the radial direction.

Ballistic Coefficient Definition

The ballistic coefficient is defined as:

$$\beta = \frac{m}{C_D A}$$

where C_D represents the vehicle drag coefficient with typical value of 1.6 for seventy-degree sphere-cone configurations at hypersonic Mach numbers.

Aerodynamic Deceleration

The aerodynamic deceleration magnitude is:

$$a_{\text{aero}} = \frac{\rho_{\infty} u_{\infty}^2}{2\beta}$$

where a_{aero} represents the aerodynamic deceleration in meters per second squared.

Magnetohydrodynamic Deceleration

The magnetohydrodynamic deceleration magnitude is:

$$a_{\text{MHD}} = \frac{F_{\text{MHD}}}{m}$$

where a_{MHD} represents the magnetohydrodynamic deceleration in meters per second squared.

Total Deceleration

The total deceleration magnitude experienced by the vehicle is:

$$a_{\text{total}} = \sqrt{\left(\frac{\mu_{\text{Mars}}}{|\vec{r}|^2}\right)^2 + \left(\frac{\rho_{\infty} v_{\infty}^2}{2\beta} + \frac{F_{\text{MHD}}}{m}\right)^2}$$

where a_{total} represents the total acceleration magnitude including gravitational, aerodynamic, and magnetohydrodynamic components.

Allen-Eggers Ballistic Entry Model

The analytical approximation for ballistic entry trajectory relates density to velocity through:

$$\frac{\rho}{\rho_{\text{ref}}} = \left(\frac{v}{v_{\infty}}\right)^2 \exp\left(\frac{1 - v^2/v_{\infty}^2}{2\beta \sin \gamma g_0 / (H v_{\infty}^2)}\right)$$

where ρ_{ref} represents a reference atmospheric density, v represents current velocity, v_{∞} represents entry interface velocity, γ represents effective flight path angle, g_0 represents reference gravitational acceleration of 3.71 m/s² for Mars, and H represents atmospheric scale height of approximately 11.1 km for Mars.

Effective Flight Path Angle from Maximum Deceleration

The effective flight path angle is computed from:

$$\sin \gamma = \frac{\rho_{\text{ref}} v_{\infty}^2}{2\beta n_{\text{max}}}$$

where n_{max} represents the maximum deceleration magnitude observed in the trajectory simulation.

Effective Ballistic Coefficient Reduction Factor

The performance metric quantifying magnetohydrodynamic augmentation effectiveness is:

$$R_{\beta} = \frac{\beta_{\text{actual}}}{\beta_{\text{effective}}}$$

where R_{β} represents the ballistic coefficient reduction factor, β_{actual} represents the physical ballistic coefficient of the vehicle without magnetohydrodynamic augmentation, and $\beta_{\text{effective}}$ represents the fitted ballistic coefficient from Allen-Eggers model matching the magnetohydrodynamic-augmented trajectory.

Equivalent Diameter Increase Factor

The equivalent increase in vehicle diameter producing the same drag area increase as magnetohydrodynamic augmentation is:

$$R_D = \sqrt{R_{\beta}}$$

where R_D represents the equivalent diameter increase factor, demonstrating that ballistic coefficient reduction factor of 3.3 corresponds to diameter increase factor of 1.8.

Magnetic Stored Energy

The energy stored in the superconducting electromagnet system is:

$$E_{\text{stored}} = \frac{1}{2} L I^2$$

where E_{stored} represents the stored magnetic energy, L represents the coil inductance typically 12.5 henries for the Mars sample return class vehicle, and I represents the operating current typically 1500 amperes.

Coil Inductance Voltage During Charging

The inductive voltage during current ramping is:

$$V_L = L \frac{dI}{dt}$$

where V_L represents the inductive voltage across the coil and $\frac{dI}{dt}$ represents the current ramp rate typically 10 amperes per second, yielding inductive voltage of 125 volts.

Thermal Conductance of Structural Supports

The conductive heat transfer through support struts is:

$$\dot{Q}_{\text{cond}} = \frac{k A_{\text{strut}}}{L_{\text{strut}}} \Delta T$$

where $\dot{Q}_{\text{cond}}$ represents the conductive heat flow rate, k represents the thermal conductivity of the composite strut material approximately 1 W/(m·K), A_{strut} represents the strut cross-sectional area, L_{strut} represents the strut length, and ΔT represents the temperature difference between warm and cold ends.

Cryogenic Refrigerator Cooling Capacity Scaling

The refrigerator cooling capacity variation with temperature is approximately:

$$\dot{Q}_{\text{cool}} = \dot{Q}_{\text{rated}} \left(\frac{T_{\text{cold}}}{T_{\text{rated}}}\right)^{\alpha}$$

where $\dot{Q}_{\text{cool}}$ represents the cooling capacity at operating temperature T_{cold} , $\dot{Q}_{\text{rated}}$ represents the rated cooling capacity at rated temperature T_{rated} , and α represents an exponent approximately equal to 1.3 for pulse tube refrigerators based on Carnot efficiency scaling.

Industrial Applicability

The magnetohydrodynamic drag augmentation system for planetary entry vehicles provides practical solutions to critical challenges facing future Mars exploration missions. Human Mars missions require landing masses of twenty metric tons or greater, exceeding the approximately one to two metric ton capability of current entry systems by an order of magnitude. The magnetohydrodynamic augmentation enables delivery of these higher masses within existing launch vehicle fairing diameter constraints by effectively increasing vehicle drag area without physically enlarging the aeroshell. Mars sample return missions benefit from reduced entry system mass fractions, allowing greater sample mass allocation within fixed total mission mass budgets. Robotic precursor missions utilize magnetohydrodynamic systems to demonstrate technologies and reduce risks for subsequent human missions while achieving scientific objectives at higher landed mass scales than previously feasible.

The system applies to planetary entry at bodies with atmospheres of sufficient density to produce significant thermal ionization during hypersonic entry, including Mars, Venus, Titan, and Earth. The Martian atmosphere presents particularly favorable conditions for magnetohydrodynamic augmentation due to the combination of sufficient atmospheric density for ionization at entry velocities of six to seven kilometers per second while remaining thin enough that purely aerodynamic deceleration is challenging for high-mass vehicles. Venus entry missions benefit from magnetohydrodynamic augmentation despite higher atmospheric density because extreme heating environments and thick atmosphere layers create challenges for thermal protection system survival, and distributing deceleration to higher altitudes through magnetohydrodynamic forces reduces peak heating rates and total heat loads.

The technology readiness advancement pathway for magnetohydrodynamic drag augmentation systems proceeds through ground testing, computational validation, and flight demonstration phases. Ground testing in hypersonic shock tunnels and arc jet facilities characterizes magnetohydrodynamic interaction physics at relevant enthalpy and velocity conditions, validating computational models and measuring plasma electrical conductivity and Lorentz force production. Computational fluid dynamics simulations incorporating magnetohydrodynamic effects provide detailed flowfield predictions for specific vehicle configurations, validating the simplified quasi-one-dimensional models used in the conceptual design methodology presented herein. Flight demonstration missions on pathfinder-class vehicles prove magnetohydrodynamic drag augmentation performance in actual planetary entry environments, measuring on-board magnetic field strength, vehicle deceleration, trajectory tracking, and guidance performance before committing to implementation on high-value human or sample return missions.

The invention thus provides a comprehensive system for planetary entry vehicles incorporating magnetohydrodynamic drag augmentation through controlled magnetic field generation, plasma interaction force production, thermal management, guidance and control, and trajectory optimization. The system enables higher-mass planetary entry missions at Mars and other planetary bodies while maintaining vehicle dimensions within launch vehicle fairing constraints, providing equivalent performance to significant vehicle diameter increases through effective drag area augmentation. The rapid computational methodology enables efficient conceptual design trade studies and mission architecture optimization, supporting development of future planetary exploration capabilities.