

## Integrated OLAP Cubes Database driven Aerothermodynamic Trajectory Analysis for Planetary Probes

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### Abstract

A novel approach to perform trajectory and aerothermodynamic analysis using the state-of-the-art OLAP multi-dimensional database technology will be presented in this publication. An Online Analytical Processing (OLAP) Database Management System integrated to a trajectory code has been developed. The database contains a list of planetary probe architectures and includes vehicle dimensions, trajectory and aero-thermal data. Material properties for several thermal protection systems are also populated into the OLAP database. The user interface allows for a selection from a list of existing planetary probe configurations. In addition, probe design specifications can be changed, including, trajectory, and entry conditions. Vehicle design parameters such as the geometric configuration, flight path angle, entry velocity, entry mass and ballistic coefficient can be varied through the interface.

The specified design is dynamically linked to the trajectory code. The trajectory is computed and the database is populated back for the selected specifications of the chosen vehicle architecture. Once a trajectory is run, the data is analyzed from within the OLAP DB using a drill-down approach and can be plotted for comparative data analysis. The framework employs the fourth order Runge-Kutta integration for trajectory calculations. To study the aerodynamic heating, Fay-Riddell and Tauber-Sutton empirical correlations have been modeled for the stagnation point heat transfer computations. In addition, the OLAP database (DB) provides dynamic links to compressible flow solvers (CFD++, GASP, CFD-RC, LAURA, SPARTA) and allows for aerothermodynamic CFD modeling.

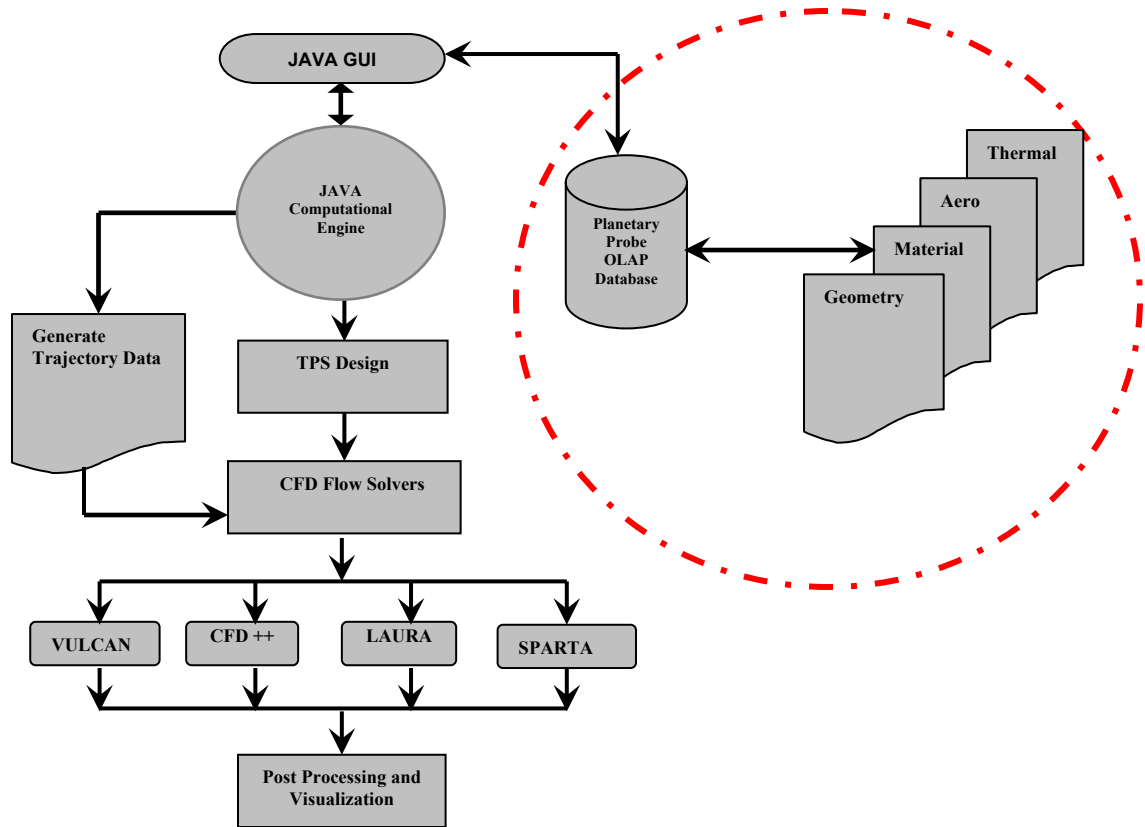
### I. Introduction

A complete computational design framework is necessary that automates the calculations for stagnation point heating, TPS design so that better probes can be built with this analysis. An integrated planetary probe design framework that generates trajectory data for planetary probes given vehicle geometry and several different entry trajectory flight conditions is presented in this paper.

SPARTA is a platform independent Graphical User Interface (GUI) based, relational database-driven JAVA tool that accurately predicts aerodynamic and heating entry environments. The aero-heating environment depends on the trajectory flown, size and shape of the vehicle. Figure 1 shows the software architecture of SPARTA. A comprehensive database of different ballistic reentry vehicles has been developed using the Planetary Mission Entry Vehicles manual <sup>[1]</sup>.

This includes vehicle dimensions and trajectory data for all the capsules that has been flown in the past. The SPARTA GUI provides the capability to choose from a list of flight vehicle geometric information and entry trajectories. From the GUI, it is possible to choose the atmospheric model and the empirical convective and radiative stagnation point analysis. An approach is presented for dynamic TPS design. Materials properties like Carbon, Silicon and Carbon-Phenolic based Ablators have been obtained from the NASA Ames Thermal Protection Materials and System Branch TPSX Internet Database and modeled into the local OLAP database and integrated with the user interface. Any previously flown planetary probe can be chosen from the user interface and tied to an Ablator from the list in the interface and a trajectory case can be run for a planet that's available in the database. This populates the trajectory data back into the database.

Figure 1: SPARTA Software architecture.



## II. Software Architecture

The overall software architecture is shown in Figure 1. The front GUI allows the user to provide inputs for trajectory, geometry and grid generation computations. It is connected to an extensive planetary probe database and a Java computational engine, which is used to generate flight trajectory data. Based on the flight conditions and probe specifications, the geometry engine constructs the vehicle configuration. With the emerging Information Technology growth, heterogeneous systems are rapidly developing and this presents a need for distributed analysis system. So, with this in mind a platform independent tool has been developed. This architecture neutral code has the capability to execute in any platform for analysis without having to recompile.

Two different Ablators can be chosen for the same probe and the tool can run two different trajectories and the aerodynamic heating is computed for comparative data analysis. Two different ablators from two different categories like Carbon Ablator and Silicon based ablators can also be chosen for TPS Sizing analysis. The JAVA graphical user interface is shown in figure (5). The purpose of this research is to investigate the aerothermodynamics environment of atmospheric entry probes and to generate an automatic computational trajectory database populated by OLAP cubes and to provide flow field analysis for such probes.

### III. OLAP Database Cubes and Multidimensional Analysis

An OLAP cube has been constructed for multidimensional trajectory analysis. A cube is a store of multidimensional data; it is defined by dimensions and measures. Different dimensions are populated in the database. Several measures are populated in the database for each of these dimensions. These variables that are characterized as measures: Effective Nose Radius ( $R_{eff}$ ), Nose radius ( $R_n$ ), Shoulder radius ( $R_s$ ), Corner radius ( $R_c$ ), vehicle's mass ( $m$ ), initial flight path angle

( $\gamma_e$ ) and initial entry velocity ( $V_e$ ). Once these dimensions are specified as different values in the cube and the 3-DOF trajectory engine is run, it populates the database with all these different permutations and combinations of the design parameters and the data can be fetched from the database either by specifying a query or by choosing the appropriate design parameter for a set of flight conditions.

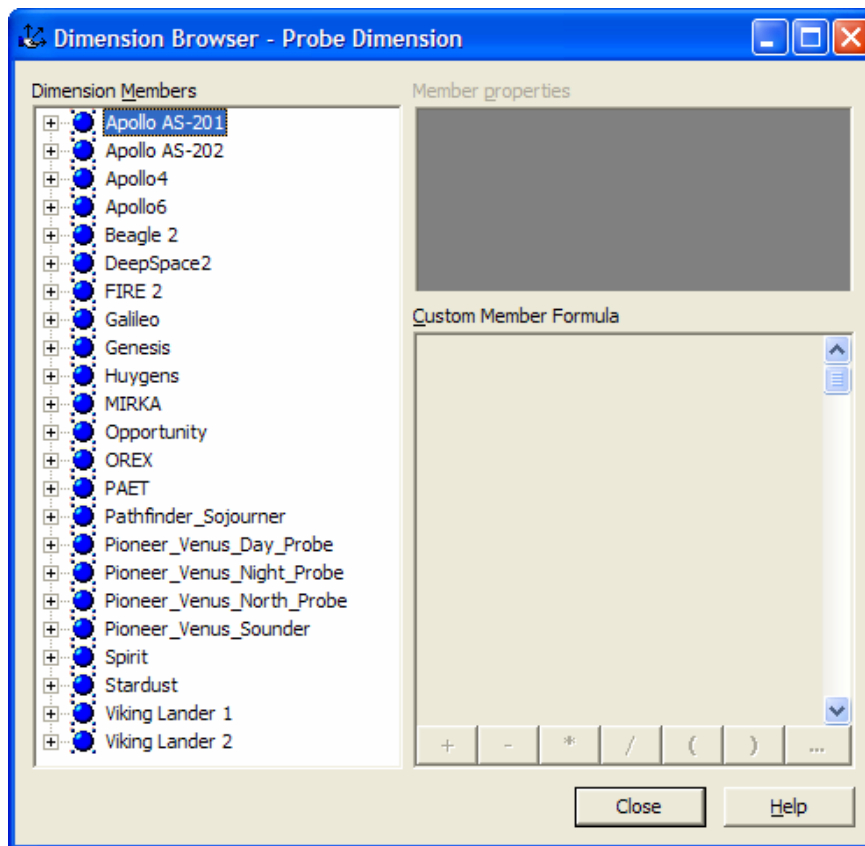


Figure 2: Planetary probes modeled in OLAP DB

OLAP cubes are constructed as dimensions and measures. Measures are the design parameters that are modeled in the OLAP database. These design parameters can be changed and forms the basis for multi dimensional data analysis capability. Figure 2 shows 23 different probes that are modeled as the dimension members in the cubes. Once the design parameters for these probes are modeled in the database as measures, from the trajectory graphical user interface, the flight vehicle is chosen, the

appropriate dimensions and the measures from the OLAP cubes are loaded as inputs for the trajectory. Once the trajectory is run, data is generated from the 3-DOF trajectory engine and populated back into the database. From this point, it's data mining to retrieve the data for the specified set of flight conditions, thus for a change in the initial entry velocity for instance, a new run is not necessary, it's just a fetch from the OLAP cubes by browsing the dimensions data.

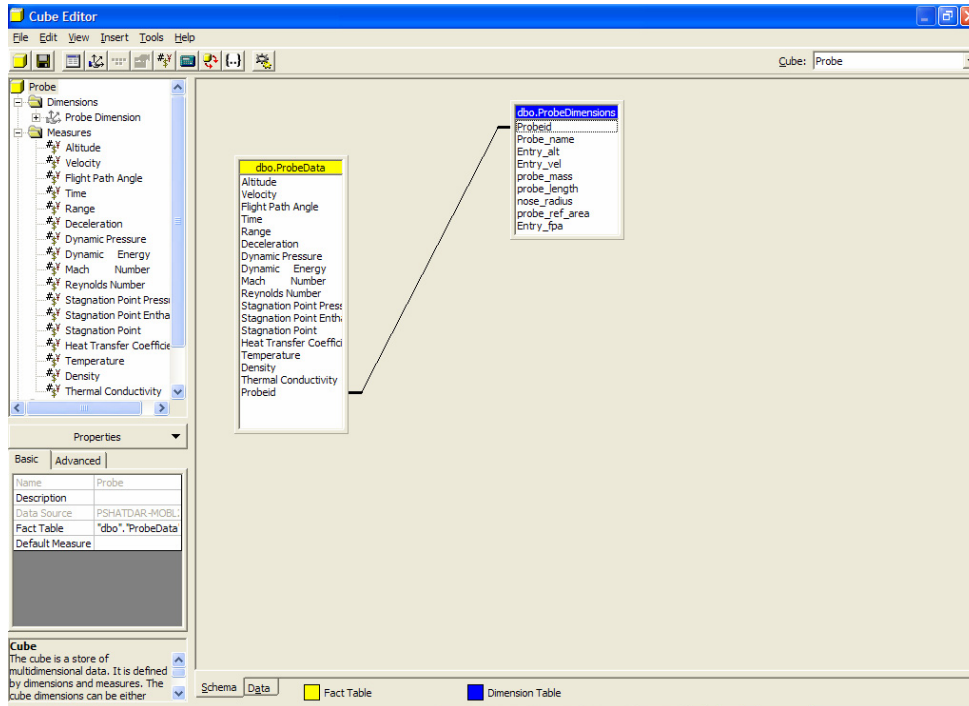


Figure 3: Measures (design parameters) modeled in OLAP Cube for planetary probes

Cubes are the logical storage medium for an OLAP database. The cube is an OLAP database what the table is to an RDBMS. A cube presents to the outside world, users and front-end applications, a potential intersection point, i.e. a cell, for every member from every dimension with every member of every other dimension. Cubes give the ability to model multiple loops in the trajectory code without having to write loops. Several values for three dimensions are changed in SPARTA cubes which are linked to the trajectory code. They also provide a capability to change up to 64 design parameters. The trajectory code is run once and the data-warehouse is populated by varying these design parameters. Data mining techniques are employed to drill-down the data and queried for specific flight conditions or entry design values from the database.

SPARTA Cubes have multiple values for these three dimensions: (a) initial flight path angles, (b) Initial entry velocity and (c) Nose radius. The term cube implies *three dimensions*. In fact OLAP server cubes can contain up to 64 dimensions. The Trajectory engine takes a little longer if more dimensions are included in the simulation. Each variable can have a maximum of up to five values. If each variable has five different values, then in this case each variable forms a separate loop and the trajectory engine is run within these three loops populating the data for 125 permutations and

combinations of these design variables for each single point in the altitude. This whole setup is populated for the entire altitude integration in the database. Hence a drill-down approach to fetch the data from the OLAP database becomes necessary if several dimensions are specified for the trajectory simulation run.

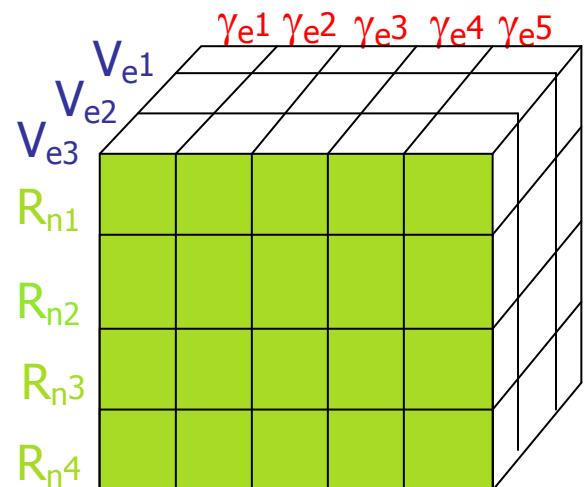


Figure 4: SPARTA OLAP Cube as modeled in the database

Table 1 is the data queried (drill-down approach) from the OLAP Cubes database that has been populated by varying five different values for the entry angles, entry velocities and nose radii (three

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of the 64 dimensions that can be modeled). Also, minimum g-forces that is chosen in the SPARTA cubes have can fetch the best trajectory optimization section from the graphical user based on either the minimum heating loads or the interface.

**Table 1: OLAP driven Trajectory Analysis at different Entry Angles, Velocities and Nose radii**

Apollo 4, Entry angle: -12.0°, Entry Velocity: 22500 ft/s, Nose Radius: 4.69m						
Altitude (ft)	Velocity (ft/sec)	Flight Path Angle $\gamma$ (deg)	Mach Number	Reynolds Number	Stag Point Pressure (lbf/ft <sup>2</sup> )	Stag Point Heat Transfer rate $\dot{q}_{conv}$ (W/cm <sup>2</sup> )
250000.00	22513.69	12.00	23.83	65486.33	30.70	391.30
249000.00	22512.32	12.00	23.80	68432.20	32.16	395.82
248000.00	22510.96	12.01	23.77	71501.97	33.69	400.44

Apollo 4, Entry angle: -6.93°, Entry Velocity: 36545 ft/s, Nose Radius: 5.69m						
Altitude (ft)	Velocity (ft/sec)	Flight Path Angle $\gamma$ (deg)	Mach Number	Reynolds Number	Stag Point Pressure (lbf/ft <sup>2</sup> )	Stag Point Heat Transfer rate $\dot{q}_{conv}$ (W/cm <sup>2</sup> )
250000.00	36545.00	6.93	33.75	106364.35	81.02	482.53
249000.00	36544.30	6.92	33.75	111144.56	84.87	482.90
248000.00	36544.00	6.91	33.70	116125.59	88.89	483.01

Data in table 1 has been fetched (or queried) from a user specified query in the OLAP Analysis Manager and can be specifically tailored to suit one of the entry conditions that has been used in the OLAP cubes to populate the database. So the idea is to run the trajectory engine only once for several different design conditions and populate the database and later retrieve the data for specified flight and design conditions so that the database has all the different permutations and combinations of trajectory data pertinent to a flight vehicle. The initial entry flight conditions in the OLAP Cubes for the Apollo vehicles were taken from Apollo Experience Report <sup>[2]</sup>

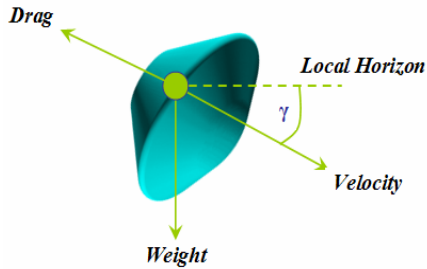


**Figure 5: JAVA based GUI for Aerothermodynamic Trajectory Analysis**

#### IV. Trajectory Optimization Analysis

The probe model developed for this trajectory is a point-mass model with two translations and one

rotation (3-DOF) around a spherical planet. It integrates the equations of motions of a vehicle on a ballistic entry trajectory so that no lift is generated and the body acts only on gravity. Figure 6 shows the various aerodynamic forces acting on the body. The vehicle model is build from a number of parameters defining the geometry of the probe including body diameter, cone half-angle, nose and shoulder radius. The aerodynamic properties of the probe are subsequently derived from the geometry of the vehicle as shown in figure 6. The graphical user interface is tied to the OLAP database at the backend. When a flight vehicle is chosen, appropriate geometry and initial trajectory values are fetched from the database and populated in the GUI. The user has the flexibility to change it or use the values from the database,



**Figure 6: Aerodynamic forces acting on the atmospheric vehicle**

Based on the capsule's entry interface, flight path angle and the vehicle shape and geometry, the trajectory engine is able to do multiple simulations involving undershoot, overshoot and finally recommend the optimal trajectory to reduce either the maximum convective heat or deceleration loads based on the optimization factor that is chosen during the simulation. The critical aerothermal conditions in the velocity-altitude plane are identified, in terms of the highest temperature surface points that correspond to the Thermal Protection System material that is chosen for modeling. The code adjusts in the trajectory automatically when it encounters a state when the peak heat exceeds the maximum allowable stagnation point temperature of the material that is chosen thus stabilizing the trajectory to a "low-risk trajectory".

During the reentry process, the aerodynamic parameters are first characterized by free molecular flow and then the trajectory engine allows the simulation to follow a gradual transition to continuum flow as the capsule descends into the denser parts of the atmosphere. Thus, despite being a 3-DOF model, the code can reliably simulate trajectories for reentry vehicles having predictable body dynamics. Modified Newton Flow Theory (NFT) is used to evaluate the pressure coefficient

$C_p$  around the body and derive the drag coefficient  $C_d$  of the configuration. The Modified NFT states that

$$C_p = C_{pmax} \sin^2\theta \quad (1)$$

where  $\theta$  is the angle between the flow and the body surface and  $C_{pmax}$  is evaluated as the maximum pressure coefficient found behind a normal shockwave at the stagnation point. The Ballistic Coefficient ( $\beta$ ) of the probe is derived from the aerodynamic model as per equation (2)

$$\beta = m/(C_d)(A) \quad (2)$$

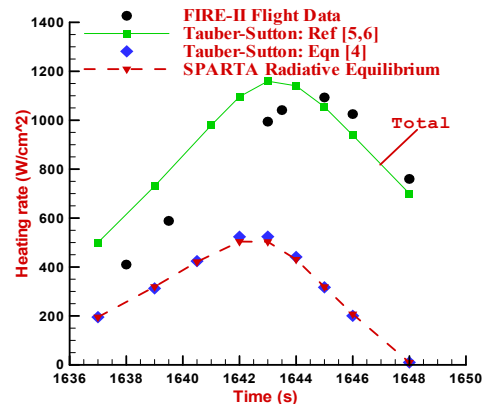
where  $C_d$ ,  $m$  and  $A$  are respectively the capsule coefficient of drag, mass of the vehicle an average cross-sectional area in the direction of the air flux.

Introducing the average specific heat [3], constant Lewis and Prandtl-Number ( $Le=1$  and  $Pr=0.71$ ), Fay and Riddell proposed an empirical correlation to define the stagnation point heat flux (*heating rate per unit area*) towards a fully catalytic, hot wall. The convective heat transfer to the stagnation point along the trajectories of the reentry capsule for Earth has been modeled using Sutton-Grave and correlation [3] Fay-Riddell theory [4] as described in equation (3) which gives the heat flux as

$$\dot{q}_{conv} = k \sqrt{\frac{\rho}{R_n}} \left( \frac{V_\infty}{1000} \right)^3 \quad (3)$$

where  $\dot{q}_{conv}$  is the convective heat-transfer rate into the flight body, per unit area,  $k$  is a constant based on the planetary atmosphere,  $\rho$  is the freestream density, and  $V_\infty$  is the flight velocity. The heating rate for  $\dot{q}_{conv}$  is in  $W/cm^2$  if the velocity is given in  $m/s$  and the density in  $kg/m^3$ . Radiative heat transfer is computed using the Tauber-Sutton Radiative heating correlation for Earth and Mars [5] as given in equation (4)

$$\dot{q}_{rad} = Cr_n^a \rho^b f(V) \quad (4)$$



**Figure 7: Radiative stagnation point heating of FIRE II data at a reentry velocity of 11.37 km/s**



The plots in figure 7 show the comparison of calculated Radiative heating with Fire II flight data and the engineering correlations described by Tauber in [5] and [6]. The vehicle configuration for FIRE II is an axisymmetric, 66° sphere cone and truncated spherical forebody, with a small corner radius. This trajectory point corresponds to the first FIRE II heatshield, with a nose radius of 0.935 m.

High temperatures in high-speed atmospheric flight require that the properties behind a shock wave must be obtained using an equilibrium chemically reacting air model instead of a calorically perfect air model. The temperature and density ratios in the shock layer along some trajectory points were obtained from the results of Huber [7] that give the variations of shock temperature and density with velocity and altitude. The equilibrium composition of air at the corresponding temperature and density of the shock layer was computed using the Chemical Equilibrium with Applications (CEA) code [8]. Briefly, this code solves for chemical equilibrium using the minimization of the Gibbs free energy, which is the equivalent of finding the point where the maximum net work obtainable has been extracted from the system.

## **V. Ablative TPS Sizing and Materials Property Database**

The objective of the convective and Radiative heating point calculation is to finally do the Ablative thermal protection system sizing. Currently, the heat shield materials used on a space vehicle such as Space Shuttle can withstand a total heat flux of the order of 50 W/cm<sup>2</sup> without causing ablation of the material. Advanced heat shield materials are believed to be able to withstand up to about 70 W/cm<sup>2</sup>. When the sum of the convective and radiative heat fluxes exceeds such a maximum allowed heat flux value, the heat shield material ablates. In the ablation process, the heat energy is converted into the latent energy of vaporization.

For a hypersonic reentry vehicle entering the Earth's atmosphere on return from a planetary mission, its entry velocity is hyperbolic, that is greater than the escape velocity of 11.3 km/sec. From the Apollo flights experience, we know that the hyperbolic entries will produce a massively ablating environment. For a Martian entry, for which the escape velocity is 5.1 km/sec, a massively ablating environment may occur if the entry speed is greater than 8km/sec, and if the nose radius of the vehicle is large of the order of 3 m or larger. Thus, it is imperative to accurately model the ablation temperature for TPS materials.

Vehicle thermal response is calculated by an approximate method that uses heat of ablation data to estimate heat shield recession during entry. This analysis is coupled to a one-dimensional finite-difference calculation that determines in-depth thermal response. The in-depth solution accounts for material decomposition, but does not account for pyrolysis gas energy absorption through the material. As inputs, the method relies on trajectory data, including relative velocity, atmospheric density, pressure, and convective heat rate as a function of time.

The tool calculates radiative heating, recovery enthalpy, wall enthalpy, surface pressure, and heat transfer coefficient. Ultimately, the tool determines recession thickness, total thickness, and heat shield area mass based on thermal response at the stagnation-point. A uniform thickness heatshield is modeled. In the existing atmospheric reentry RDBMS, a material database has been constructed and added for common ablative thermal protection and substructure materials. User-defined materials can be easily added to the database without having to modify the thermal protection systems subsection code of the Reentry tool.

Stored constants for ablative materials include the decomposition kinetic constants used in the Arrhenius formulation for density decomposition, the resin volume fraction, the heats of formation, thermal conductivity, specific heat, emissivity, and heat of ablation curve fit constants. There are two components to the approximate technique presented here. The first component makes use of a steady state ablation assumption and employs the heat of ablation, or  $Q^*$ , to estimate recession during entry. The second component involves calculating the in-depth temperature response to predict the amount of material required as insulation to keep the bond-line temperature below a specified limit.

Calculating the in-depth temperature response is accomplished using a finite-difference formulation for the in-depth conduction through the material. Using the heat of ablation, the recession rate at any instant in time can be calculated by equation (5). The total recession is then found by integrating the recession rate over the entire trajectory. This formulation is conservative and will generally over predict recession rate.

$$\dot{S} = \frac{\dot{Q}_{hw}}{\rho Q^*} \quad (5)$$

where  $\dot{S}$  is the recession rate,  $\dot{Q}_{hw}$  is the hot wall heat flux,  $\rho$  is the material density and  $Q^*$  is the heat of ablation. The one dimensional heat conduction equation can be written along with the surface energy balance as shown in the following equations (6) and (7).

$$\rho c_p \frac{\partial T}{\partial t} = \frac{1}{A} \frac{\partial}{\partial x} \left( \kappa A \frac{\partial T}{\partial x} \right) \quad (6)$$

$$\dot{q}_{conv} + \alpha \dot{q}_{rad} - \dot{q}_{cond} - \epsilon \sigma T_w^4 = 0 \quad (7)$$

where  $T$  is the temperature,  $T_w$  is the surface temperature,  $\kappa$  is the thermal conductivity,  $x$  is

measured from TPS surface,  $\rho$  is the instantaneous material density,  $C_p$  is the material specific heat,  $\dot{q}_{conv}$  is the convective heat flux,  $\dot{q}_{rad}$  is the radiative heat flux,  $\epsilon$  is the material emissivity,  $\alpha$  is the material absorptivity and  $\sigma$  is the Stephen-Boltzmann's constant.

Table 2 shows the thermal and material properties for Avcoat-5026-39 and RCC. Because of the high thermal conductivity ( $\kappa$ ) of the nose cap wall and the thinness of the wall when compared to the radius of the nose cap, the Biot number is small and the nose cap wall is effectively modeled as a "lumped-mass" structure with constant temperature throughout.

**Table 2: Material and Thermal Properties of Avcoat-5026-39 and RCC**

Material	Density ( $\rho$ ) (kg/m <sup>3</sup> )	Thermal Conductivity ( $\kappa$ ) (watt/m-°k)	Specific Heat ( $C_p$ ) (J/kg-°k)	Heat of Ablation (H) MJ/kg	Ablation Temperature ( $T_a$ ) °k
<b>Avcoat-5026-39</b>	512.6	25.0	1110.0	47	835
<b>RCC</b>	1650.2	120.0	1312.0		

The wall thickness is defined as the depth of the ablative layer and the RCC thickness and the average head capacity of the wall is the thickness weighted heat capacity of the ablative and RCC wall layers. The "lumped-mass" temperature approximation for this wall segment can be written as

$$\dot{q}_{in} - \dot{q}_{out} = [\rho_{Avcoat} C_p t_{Avcoat} + \rho_{RCC} C_p t_{RCC}] \dot{T}_w \quad [8]$$

where  $\dot{q}_{in}$  is the heat flux into the wall,  $\dot{q}_{out}$  is the heat flux out of the wall,  $\dot{T}_w$  is the time rate of change of the wall temperature and the remaining terms in the brackets is the average heat capacity of the wall segment. The only to significantly vary the heat capacity is to change the material thickness

since the values of specific heat and density are thermal properties of the material.

## V. Benchmarking SPARTA

SPARTA has been compared and evaluated against the industry standard tools like POST [9]: Program to Optimize Simulated Trajectories and TrajOpt: Trajectory Optimization tool. POST uses more traditional direct shooting approach that calculates the state variables as a function of time throughout the entire trajectory and this guarantees that the physics of the problem are accurate at all times during the simulation. This makes POST a perfect candidate for benchmarking against SPARTA.

**Table 3: Benchmarking Analysis for Apollo 6 at an Entry Angle Of -5.9°**

Codes	Velocity at Peak heating (km/s)	Altitude at Peak heating (km)	Maximum Deceleration [g's]	Mach Number
<b>POST</b>	8.45	24.01	12.15	26.40
<b>TrajOpt</b>	7.98	23.00	11.16	24.93
<b>SPARTA</b>	8.32	23.16	11.96	26.00

The flight vehicle configuration that was evaluated in POST, TrajOpt and SPARTA for the

benchmark analysis was Apollo 6 reentry profile. Velocity at peak heat was found to be 8.32 km/s in



SPARTA calculations and around 8 km/s in the other trajectory tools that were used for benchmarking.

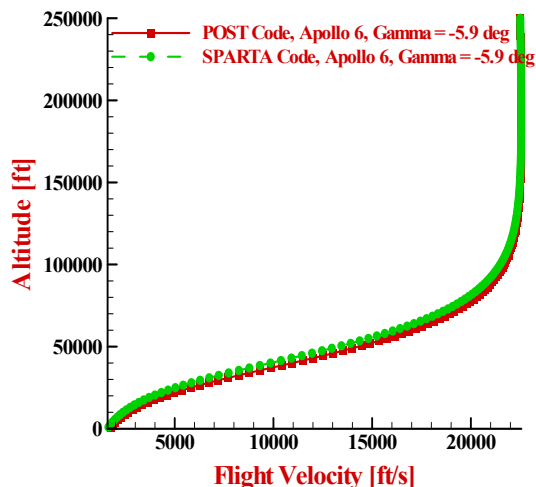


Figure 8: SPARTA Vs POST: Reentry Flight Velocity Vs Altitude

### Conclusions

A platform-independent engineering level analysis, OLAP database-driven JAVA GUI based application has been developed for preliminary investigation of reentry vehicles. The code calculates all flight conditions along the trajectories including aerodynamic and Aerothermodynamic stagnation point quantities and populates the data back into OLAP database for comparative analysis capability. A fourth order Runge-Kutta integration is employed for trajectory calculations. In addition, a comprehensive database of existing planetary probe designs has been developed and integrated with the 3-DOF GUI tool. An algorithm to optimize trajectories is embedded in the code for undershoot, overshoot and optimal low-risk trajectory.

### References

- [1]. C. Davies, "Planetary Mission Entry Vehicles," Quick Reference Guide, Ver. 2.1, NASA-ARC, 2002.
- [2]. J. E. Pavlosky and Leslie G. St. Leger, "Apollo Experience Report – Thermal Protection Subsystem", NASA TN D 7564
- [3]. K. Sutton, R.A Graves, "A General Stagnation Point Convective Heating Equation for Arbitrary Gas Mixture", NASA TR-376, 1971

- [4]. J. Fay and F. Riddell, "Theory of Stagnation Point Heat Transfer in Dissociated Air" *Journal of Aeronautical Sciences* 25 (2), Feb 1958.
- [5]. Tauber, M. E., Sutton, K., "Stagnation Point Radiative Heating Relations for Earth and Mars Entries", *Journal of Spacecraft and Rockets*, Vol 28, No 1, 1991, pp.40-42
- [6]. Tauber, M. E., "A Review of High-Speed, Convective Heat Transfer Computation Methods", NASA TP-2914, July 1989.
- [7]. P. W. Huber, "Hypersonic Shock-Heated Flow Parameters for Velocities to 46,000 Feet per Second and Altitudes to 320,000 Feet, NASA TR-163, December 1963.
- [8]. S. Gordon, B. J. McBride, Computer Program for Calculations of Complex Chemical Equilibrium Compositions and Applications Analysis, NASA RP-1311, October, 1994
- [9]. G. L. Brauer, D. E. Cornick, and R. Stevenson, "Capabilities and Applications of the Program to Optimize Simulated Trajectories (POST)," NASA CR-2770, Feb. 1977.