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A Genesis Breakup and Burnup Analysis in Off-Nominal Earth Return and Atmospheric Entry

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The Genesis capsule returned to Earth on September 8, 2004, carrying samples of solar wind charged particles. The Genesis project conducted a detailed breakup/burnup analysis before the Earth return to determine if any spacecraft component could survive and reach the ground intact in case of an off-nominal entry. In addition, an independent JPL team was chartered with the responsibility of analyzing several definitive breakup scenarios to verify the official project analysis. This paper presents the analysis and results of this independent team.

INTRODUCTION

Genesis is a Solar Wind Sample Return mission managed by NASA's Jet Propulsion Laboratory (JPL) to collect samples of the solar wind particles from the L1 lagrangian point. In the last phase of the mission (Earth Return Phase) Genesis rendezvoused with planet Earth and then entered its atmosphere, beginning its descent to return its samples to Utah on September 8, 2004. It is customary in missions involving atmospheric entry/reentry (for the purpose of launch approval, aerospace nuclear safety, planetary protection, and safe sample return) to perform breakup and debris analyses to assess consequences of potential failures. In particular, sample return mission failures could lead to human casualty and the purpose of the analysis is to determine if any spacecraft component could reach the ground in case of entry failures to help in assessing this casualty. This paper provides the analysis and results of one of the two independent analyses conducted by the Genesis project to comply with NASA requirements. One analysis was conducted by LMSS, and this paper addresses the other analysis performed by JPL.

Nominally, after a couple of Trajectory Correction Maneuvers (TCM), the spacecraft (consisting of a bus and a Sample Return Capsule, SRC) is aligned to entry orientation about 6 hours before entry. The capsule is then released for entry, and the bus is diverted to an orbit around the sun. Four major cases (with different entry conditions) are examined for the failure scenarios in off-nominal entries in this analysis. These cases are meant to envelop all possible failure scenarios.

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Case 1: Spacecraft achieves SRC release attitude and spins up but fails to release SRC.

Case 2: Following the completion of EI minus 1 day final TCM, the spacecraft fails to achieve release attitude, and therefore the SRC is not released.

Case 3: Following the completion of EI minus 10 day TCM, a loss of control of the spacecraft occurs, resulting in entry of SRC-bus at incorrect attitude.

Case 4: The spacecraft releases the SRC but fails to accomplish the divert maneuver.

The task started with making the appropriate assumptions and acquiring entry conditions from the project and spacecraft data from LMSS. Engineering judgment and data from analyses for prior missions were used where necessary to supplement spacecraft data provided by LMSS. Genesis vehicle breakup and burnup analysis is basically a trajectory propagation, coupled with thermal and structural analyses to predict the progressive failure or break-off of various components to peel off these components. Ablation modeling is then considered to determine which components survive the reentry and reach the ground entirely or in part.

The analysis is constrained by the following high-level assumptions:

1. Limited failure scenarios are considered.
2. Only 10 major thermal/structural nodes are considered.
3. Engineering judgment is used to estimate some physical and material properties of these 10 nodes.
4. Ablation modeling is used only for three large components.

Overview of Breakup and Burnup Analysis

Breakup and burnup analyses are performed in missions involving atmospheric entry/reentry (accidental Earth reentry of spacecraft that carry nuclear fuel, Mars off-nominal entry, and Earth sample return missions). The analyses are required in accidental Earth reentry for launch approval and in Mars off-nominal entry for planetary protection. As a sample return mission, Genesis is required to determine if any spacecraft component could survive the atmospheric flight and reach the ground intact resulting in an unacceptably high risk of human casualty in the event of an off-nominal entry. Thus, the final product of breakup and burnup analyses is the prediction of debris dispersion patterns and the footprints of the components that survive the disintegration in the Earth atmosphere and reach the ground in part or entirely. As in any breakup and burnup analysis, a significant amount of navigation, aerodynamic and spacecraft data is required. The aerodynamic and spacecraft data are needed for the entire spacecraft as well as for specific components/nodes. These nodes are determined by the breakup/burnup analyst as a set of structurally and thermally weak points on the spacecraft. Depending on the condition of the off-nominal entry, the spacecraft follows a particular ballistic trajectory and may be relatively unprotected against aerodynamic heating and g-loads. Therefore, the Genesis vehicle breakup and burnup analysis is basically a trajectory propagation, coupled with thermal and structural analyses to predict the failure or break-off of various components in a progressive destruction analysis to peel off these components. At some point in this progressive process, the engineering judgment predicts that the entire spacecraft is decomposed into many components that are free-flying independently. The analyst, then, determines a number of components that have higher chance of surviving to reach the ground entirely or partially without full disintegration, and the analysis follows them to the ground to predict the impact footprints whenever applicable.

NASA’s Safety Standards

Reference (1) documents NASA’s procedures for limiting orbital debris as well as debris survivability requirements. As a guideline, the reference uses the heat of ablation as the indicator of a component’s ability to survive reentry (for instance Titanium, Ti 6A1-4V, has one of the highest heats of ablation, 1716421 J/kg). Specific heats of ablation for common spacecraft materials (such as Aluminum, Steel, Titanium, and Tungsten) are provided in that reference and used in this paper. It also specifies simple geometrical models of spacecraft components based on their basic geometry.

GENESIS EARTH RETURN PHASE

The Earth return phase of the Genesis mission began about two weeks before Earth reentry. Trajectory Correction Maneuvers (TCM) were executed to refine entry targeting, the Sample Return Capsule (SRC) was released, and the spacecraft bus was diverted to an orbit around the Sun with no return to Earth in the foreseeable future.

Nominal Mission Timeline

The following is the nominal mission timeline:

<u>Time</u>	<u>Event</u>
EI minus 10 day	TCM
EI minus 1 day	Final TCM
EI minus 6 hr	Align to entry orientation. Increase spin rate to 15 rpm.
EI minus 4 hr	Release capsule (Capsule will not be released if not correctly targeted).
EI minus N/E hr	Spacecraft performs divert maneuver to an orbit around the Sun.
EI	Nominal entry condition. Arrival date 9/8/2004.

Off-nominal Entry

A bus only or a bus/SRC combination entering the atmosphere with different entry flight path angle (FPA) and attitude constitute off-nominal entries.

FAILURE SCENARIOS

According to the nominal mission timeline above, a nominal entry constitutes a SRC release and entry with the nominal flight path angle and attitude, and a diverted bus with no return to Earth in the foreseeable future. However, an anomalous occurrence prior to entry could lead to failure of the bus/SRC separation, leading to atmospheric entry of the combination. A second scenario where the bus fails to execute the divert maneuver following a nominal bus/SRC separation could lead to the entry of the separated bus. These two failure entry configurations together with various attitudes and flight path angles result in a matrix of failure scenarios from which four cases are analyzed in this paper. These four cases represent an envelope of all possible scenarios.

The entry state for each case is provided in the J2000 BCI coordinate system and Earth-fixed, atmosphere-relative rotating frame as given in Table 2.

<u>Case 1</u>: Spacecraft achieves SRC release attitude and spins up but fails to release SRC.

The entry vehicle is the SRC and bus, attached, with the bus leading in the flying forward orientation. The vehicle is spinning at 15 rpm. Flying face-on, the solar arrays are in their fully deployed positions and may bend backwards due to dynamic pressure. The two solar arrays will most likely not fail simultaneously. After one panel fails, the unbalanced force from the remaining panel may cause the vehicle to start coning, where the spin axis becomes offset from the body x-axis. However, since the vehicle is spinning at the rate of 15 rpm, it is not expected to begin tumbling. The failure of the second solar array panel is expected to follow soon after. Therefore, it is assumed that the vehicle remains face-on to the flow as components fail or break off.

Case 2: Following the completion of EI minus 1 day final TCM, the spacecraft fails to achieve release attitude, and therefore the SRC is not released.

The entry vehicle is the SRC and bus, attached, with arbitrary attitude. Three orientations are analyzed: flying forward, backward, and tumbling face over back. It is assumed that orientations other than flying forward and backward, i.e. side-on, are not stable, and the vehicle will begin to rotate or tumble in an effort to trim itself. The vehicle does not spin up since the release attitude was not achieved. As the spacecraft starts to break up, it begins to trim/tumble due to the stabilizing/de-stabilizing moment.

Case 3: Following the completion of EI minus 10 day TCM, a loss of control of the spacecraft occurs.

The entry vehicle is the SRC and bus, attached, with arbitrary attitude as in Case 2. The vehicle does not spin up due to loss of control of spacecraft, and the entry conditions may differ slightly from those of Case 2.

Case 4: The spacecraft releases the SRC but fails to accomplish the divert maneuver.

At the time of the SRC release, the separation direction is along the spin axis, and there is a small separation speed. As the SRC and bus reach the entry altitude 4 hours later, it is assumed that the separation distance would have propagated sufficiently to avert collision between the SRC and bus during the reentry. The entry vehicles are the SRC and bus, separated, with the bus leading. Both vehicles are in the flying forward orientation, spinning at 15 rpm. It is assumed that there is an offset in the times of entry between the SRC and bus and the vehicles remain face-on to the flow as components fail or break off.

The entry conditions of the 4 cases are summarized in Table 1.

Table 1
Entry Conditions of Failure Scenarios

Entry Condition	Case 1	Case 2	Case 3	Case 4
Entry Configuration	SRC and bus, attached	SRC and bus, attached	SRC and bus, attached	SRC and bus, separated
Entry Orientation	Flying forward, bus leading	i. Flying forward, bus leading ii. Flying backwards, SRC Backshell leading iii. Tumbling	i. Flying forward, bus leading ii. Flying backwards, SRC Backshell leading iii. Tumbling	Both flying forward, bus ahead of SRC
Entry Attitude	Spinning about body x-axis	Non-spinning	Non-spinning	Both spinning about body x-axis

NAVIGATION, AERODYNAMIC AND SPACECRAFT INFORMATION

In any vehicle breakup analysis an extensive amount of navigation, aerodynamic and spacecraft data is required. The following is Genesis breakup and burnup data acquired for this analysis.

Entry State

The entry states are provided by the project as seen in Table 2.

Table 2
Entry State

Case	BCI	Earth Rotating Frame
1	Epoch: 9/8/2004 15:53:04.5 UTC X -1124.9566 Y 4507.9725 Z 4550.0696 XDOT -9.81009414 YDOT 0.41490108 ZDOT -5.05724836	Radius 6503.11285289468 km Latitude 44.3946480005722 deg Longitude 238.103987203131 deg Speed 10.7573556019088 km/s FPA -8.30489489621862 deg Azimuth 121.505240704692 deg
2	Epoch: 9/8/2004 15:52:49.7 UTC X -989.0243 Y 4489.8049 Z 4599.3742 XDOT -9.84667347 YDOT 0.49293247 ZDOT -4.97752328	Radius 6503.13463936279 km Latitude 45.0063741733155 deg Longitude 236.577486484490 deg Speed 10.7570076443916 km/s FPA -8.99872588900982 deg Azimuth 120.319326877875 deg
3	Epoch: 9/8/2004 15:54:24.5 UTC X -1176.1890 Y 4523.8112 Z 4521.3632 XDOT -9.80208683 YDOT 0.35637399 ZDOT -5.07595806	Radius 6503.15412113850 km Latitude 44.0411117310571 deg Longitude 238.331689903346 deg Speed 10.7557041212578 km/s FPA -8.06152523354083 deg Azimuth 121.781418169671 deg
4	Epoch: 9/8/2004 15:53:05.80 UTC X -1133.1291 Y 4509.2133 Z 4546.8581 XDOT -9.80788162 YDOT 0.40977075 ZDOT -5.06186699	Radius 6503.14575599374 km Latitude 44.3547331073772 deg Longitude 238.192524022304 deg Speed 10.7572898267093 km/s FPA -8.26328232343768 deg Azimuth 121.573214388153 deg

Aerodynamic Properties

The aerodynamic properties of the various possible spacecraft configurations while facing forward, backward, side-on, and tumbling are given in Table 3 through Table 6.

Note that the side-on aerodynamics were determined for the purpose of calculating the tumbling vehicle aerodynamic properties. The drag coefficients were obtained based on Newtonian theory for hypersonic flow.

Table 3
Flying Forward Spacecraft Aerodynamic Properties

Spacecraft Configuration	Mass (kg)	C_D	A_{ref} (m^2)	C_B (kg/m^2)
SRC + bus	596	1.94	7.64	40.3
SRC + bus – S/A	578	1.89	4.56	66.9
SRC + bus – S/A – tanks	459.7	2.0	4.07	56.4
SRC	205.17	1.09	1.77	106.5
Bus	390.83	1.94	7.64	26.4
Bus – S/A	372.83	1.89	4.56	43.18

Table 4
Flying Backward Spacecraft Aerodynamic Properties

Spacecraft Configuration	Mass (kg)	C_D	A_{ref} (m^2)	C_B (kg/m^2)
SRC + bus	596	2.16	7.64	36.1
SRC + bus – S/A	578	2.27	4.56	55.9
SRC + bus – S/A – tanks	459.7	2.42	4.07	46.6
SRC	205.17	1.37	1.77	84.8

Table 5
Side-On Spacecraft Aerodynamic Properties

Spacecraft Configuration	Mass (kg)	C_D	A_{ref} (m^2)	C_B (kg/m^2)
SRC + bus	596	0.34	7.64	229.5
SRC + bus – S/A	578	0.50	4.56	254.2
SRC + bus – S/A – tanks	459.7	0.50	4.07	226.7
SRC	205.17	0.37	1.77	313.8

Table 6
Tumbling Face-over-Back Spacecraft Aerodynamic Properties

Spacecraft Configuration	Mass (kg)	C_D	A_{ref} (m^2)	C_B (kg/m^2)
SRC + bus	596	1.19	7.64	65.3
SRC + bus – S/A	578	1.29	4.56	98.2
SRC + bus – S/A – tanks	459.7	1.35	4.07	83.4
SRC	205.17	0.8	1.77	145.1
Science canister	68.5	1.88	0.39	94.64

The coordinates of the SRC profile normalized to a maximum diameter of 1 meter are given in Table 7. The profile is input to the Newtonian Code to generate aerodynamic coefficients.

Table 7
Coordinates of the SRC Profile

x	R
0.0000	0.0000
0.0073	0.0610
0.0305	0.1220
0.0457	0.1524
0.2378	0.4878
0.2439	0.4939
0.2561	0.5000
0.2622	0.5000
0.2744	0.4939
0.2866	0.4878
0.4573	0.4268
0.4634	0.4238
0.6433	0.1037
0.6451	0.0000

Spacecraft Properties

The spacecraft components, modeled as lumped masses, or nodes (circled numbers in figures), for breakup analysis and debris footprint prediction are shown in Figure 1 through Figure 3.

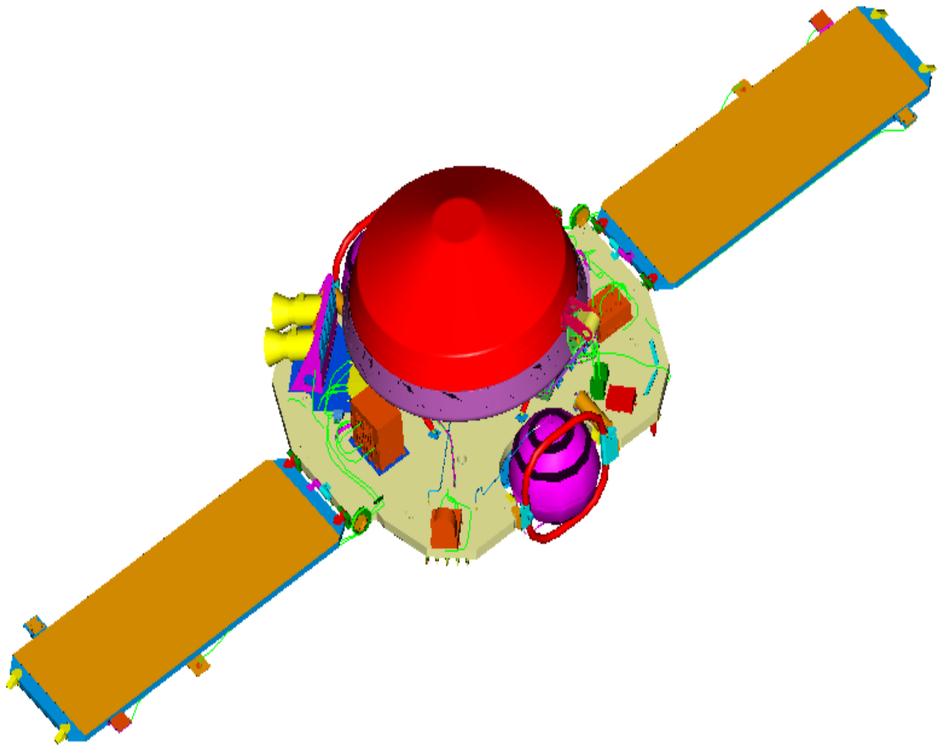

Figure 1 Thermal Nodes on SRC Side of Spacecraft

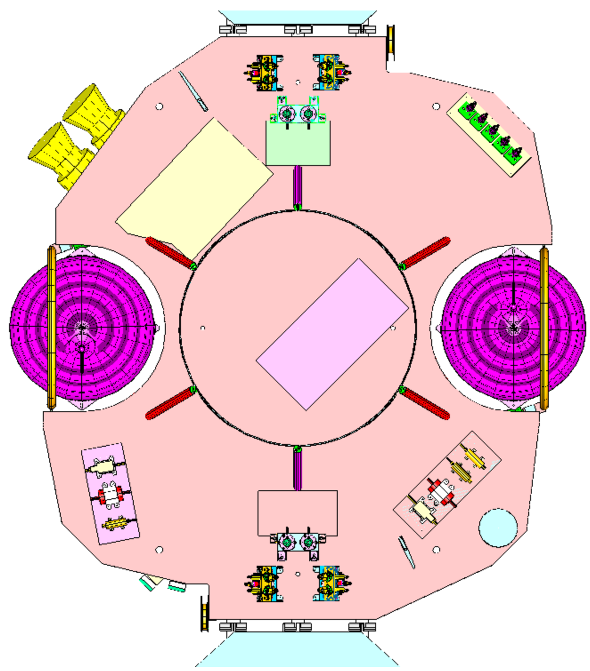

Figure 2 Thermal Nodes on LVA Side of Spacecraft

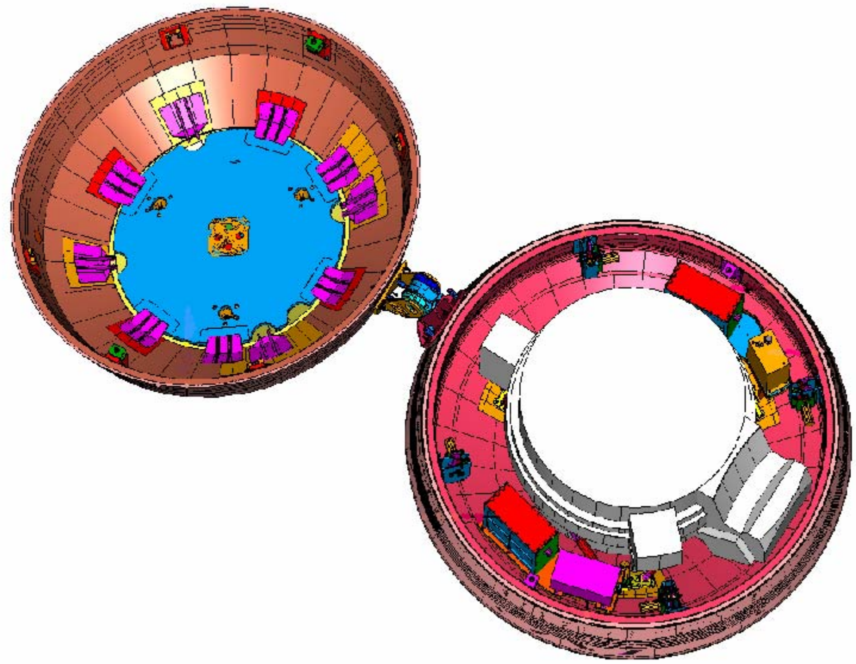

Figure 3 Thermal Nodes on SRC

Physical and material properties of the above nodes are obtained from spacecraft design documents and Reference (2). Failure criteria are obtained based on these properties and presented in Reference (3).

BREAKUP AND BURNUP ANALYSIS

The approach used in Genesis breakup and burnup analysis is to propagate the reentering trajectory coupled with thermal and structural analyses for all configurations, entry states and attitudes. Thermal and structural properties are required in this propagation to predict the failure or break-off of various spacecraft components in a progressive destruction analysis to peel off these components. After the break-off, another trajectory propagation with ablation prediction to maximize probability of ground impact of some tracked fragments and determine surviving mass on impact is performed. The choice of these fragments is based on their mass and material properties to maximize probability of impact. As shown in Figure 1 through Figure 3, ten major thermal/structural nodes are considered in the first breakup leg of the trajectory. Three large components are chosen for the analysis in the burnup leg of the trajectory. After one of these high temperature ablated component breaks-off from the vehicle due to thermal/structural loads, its trajectory continues to be followed to determine whether it disintegrates due to ablation or it may survive and reach the ground entirely or in part. In this analysis, the ablation nodes considered are the empty tank, a reference steel ball, and the graphite epoxy box. Reference (4) provides a more detailed analysis on the breakup/burnup of the STARDUST mission in its Earth Return Mission.

The software tool used in this analysis is the sample return mission version of VBA,Vehicle Breakup Analysis, (Reference 5). VBA is a software tool designed and built to analyze accidental Earth entry of vehicles that contain RTGs or LWRHUs for launch approval.

BREAKUP AND BURNUP RESULTS

The following is the results obtained for the cases studied in this analysis.

Case 1

The entry vehicle is the bus + SRC, with the bus leading. The following is the chronological order of failure after atmospheric entry:

1. The thermal blanket at 11 seconds and 108.4 km.
2. The solar arrays at 14 seconds and 104.1 km.
3. The equipment deck at 18 seconds and 98.5 km.
4. All components of the bus disintegrate.
5. The SRC heatshield facing the flow does not fail.
6. The SRC impact point is shown in Figure 4.

SRC impact point

Figure 4 Case 1 SRC Impact Point

Case 2

The entry vehicle is the bus + SRC, non-spinning. In all entry attitudes examined (flying forward, backward and tumbling) nothing survived to the ground.

Case 3

This case is similar to case 2 with the same attitudes but different entry conditions. Also, nothing survived to the ground.

Case 4

There are two entry vehicles: the bus and the SRC with the heatshield facing the flow. Both are spinning. The following is the chronological order of failure after atmospheric entry:

1. The thermal blanket at 11 seconds and 108.4 km.
2. The solar arrays at 14 seconds and 104.3 km.
3. The equipment deck at 18 seconds and 98.7 km.
4. All components of the bus disintegrate.
5. The SRC heatshield facing the flow does not fail.
6. The SRC impact point is shown in Figure 5

SRC impact point

Figure 5 Case 4 SRC Impact Point

CONCLUSIONS

The entry and breakup scenarios for the 4 cases are summarized below:

- **CASE 1**

  All components on the bus are disintegrated during the entry. The spinning SRC survives the entry and is intact upon ground impact at:

  Lat = 40.52° N

  Lon = 114.22 deg West

- **CASE 2**

  All components on the bus are disintegrated during the entry. The non-spinning SRC begins to tumble during the entry and eventually breaks up, beginning at 2 opposite ends of the shoulder. The content of the SRC, including the science canister, disintegrate after being released from the SRC breakup. The damaged Heatshield and Backshell also disintegrate due to high heat flux and g-load. Therefore, nothing survives to the ground.

- **CASE 3**

  Same as Case 2.

- **CASE 4**

  Same as Case 1. The SRC ground impact is at:

  Lat = 40.44° N

  Lon = 114.09 deg West

ACKNOWLEDGEMENT

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REFERENCES

1. NASA Safety Standards, "Guidelines and Assessment Procedures for Limiting Orbital Debris," NSS 1740.14, August 1995.
2. Sokolowski, W., "Thermo-Mechanical Property Data for MER Breakup Analysis", Thermal and Propulsion Engineering section, JPL, January 2004. (JPL internal document)
3. Ling, L., A. Salama, A. McRonald, "Genesis Return to Earth Breakup Analysis (Preliminary)" July 7, 2004. (JPL internal document)
4. Salama, A., "JPL Breakup & Burnup Analysis, STARDUST Project Earth Targeting and Entry safety Plan Review", 7 June 2005. (JPL internal document).
5. Vehicle Breakup Analysis (VBA) Software, User's Guide, Version 2.0, JPL/Raytheon, April 14, 2005. (JPL internal document)