N91-14364

A GNM Mission and System Design Proposal

A Position Paper in Response to the Session B of the Mars Global Network Workshop, Feb 6-7, 1990

Introduction

After having attended the Mars Global Network Mission (GNM) Workshop, and upon some reflection, I have put together a mission and system design option for the GNM which I believe is complementary to the 2001 Sample Return Mission (SRM). In this paper I take an advocacy position for the proposed mission; it is not intended to be an objective review, although both pros and cons are presented in summary. This work represents my own opinions and judgements, and is not an SRM policy statement, nor is it supported by any systematic analysis. These ideas are an expansion and elaboration of the design proposed by Al Friedlander of SAIC in the Session B discussion of the GNM workshop.

In arriving at the proposed design I used the following criteria, in order of priority, for evaluation:

1) Maximize Science Value
2) Keep Costs Low
3) Maximize Heritage (both from previous missions and heritage to be provided to future exploration missions, particularly the SRM)
4) Design to fly in the earliest possible opportunity
5) Make it "Innovative"

The Elements of the proposed mission are:

1) Aeroshelled Landers
2) Communication Orbiter(s)

Mission Scenario

The mission consists of launches from earth in the '96, '98, and '01 opportunities on Delta-class launch vehicles (~1000 Kg injected to Mars in 8 to 10 ft diameter shroud). The trans Mars boost stage injects a stack of small independent, aeroshelled spacecraft. The stack separates from the boost stage and each rigid (as opposed to deployable) aeroshell flies to Mars on its own, performing midcourse maneuvers as necessary. On-board GN&C systems provide precision pointing (via torque wheels) and burn execution. Each spacecraft flies a unique trajectory which is targeted to achieve approach atmospheric interface at the desired latitude and lighting conditions; arrival times may vary by a month or more. A direct entry is performed, there is no propulsive orbit capture. The aeroshelled rough-landers are targeted to achieve a desired attitude and entry flight path angle, and then follow a passive ballistic trajectory until terminal descent. Based on sensed acceleration (integrated to deduce altitude), the aft aeroshell skirt is jettisoned, a short time later a supersonic parachute is deployed. The ballistic coefficient of the parachute is sized to achieve terminal velocity at about 8 Km. However the parachute is not deployed until a few Km above the surface to minimize wind-induced drift. This relatively short period on the parachute is possible because of the low ballistic coefficient of the aeroshell, and allows surface sites up to 6 Km above the mean surface level to be visited. The nose cap (weighted by the no longer required torque wheel assembly) is jettisoned and descent imaging begins, a laser altimeter also measures true altitude. (Depending on what altitude descent imaging is first required, the nose cap may be jettisoned prior to the aeroshell skirt jettison.) Based on range and range rate to the surface, the parachute is jettisoned and the lander uses descent engines to achieve touchdown velocity. (Note: if the ballistic coefficient of the aeroshell is sufficiently low, a parachute is not required, the ballistic terminal velocity provided by the aeroshell would be low enough that a propulsive descent could be performed directly). A contact sensor shuts down the motors to avoid cratering, and the lander rough-lands at less than 5 m/sec. The remaining aeroshell and a deployable bladder attenuate landing loads and minimize the possibility of tip over. Science instruments are deployed and activated, and the network is established.

See the appendix of figures which illustrate the mission and spacecraft designs.

Shared Communications Infrastructure

In this scenario, the communications relay orbiter(s) are provided as infrastructure for both the GNM and the SRM. In the reference GNM and SRM scenarios, each mission provides its own communications system. These systems are a part of the carriers which are captured into deployment and (in the case of the SRM) retrieval orbits; these orbits are not the preferred ones from a communications standpoint, and may in fact be far from optimum. Because of the successive nature of these missions, commonality between the communications system requirements should be explored. Because of the stated commitment to planetary exploration, consideration should include the use of this system to backup or augment future, higher capability Mars communications systems.

Deployment from the Trans Mars Boost Stage Contrasted to the Reference GNM Mission Scenario

Another key feature of this design proposal is the lack of a centralized carrier vehicle which propulsively captures into Mars orbit and performs deployment of landers from that orbit. In the proposed approach, the aeroshells are separated for the boost structure via a simple sequencer. They then become independent spacecraft, each targeted and tracked on a unique trajectory.

In contrast, the reference GNM mission designs involve a combination of deployment from orbit and deployment on approach.

Although an orbit design exists which satisfies lighting conditions over a wide range of latitudes, including polar (re. "A Polar Orbit Mission for the Mars Global Network Mission", Philip Knocke, JPL), it comes at some expense. The 1/5 sol polar orbit requires a higher capture Delta-V than a more elliptic orbit.

A 160 day wait (for the 1998 launch opportunity) is also required to achieve the correct orbital conditions before entry vehicle deployment may begin. The opportunity available then sweeps from the south pole to the north over a 180 day period - thus to get full latitude coverage and emplace the full network would take almost a year from the time of arrival at Mars.

The carrier deployment strategies discussed in the workshop considered the deployment of aeroshells with no active GN&C system. In this scenario, the carrier would provide pre-deployment pointing and would deploy the aeroshell in such a way that tip off rates were negligible; the aeroshell would then simply execute a fixed delta-V burn. This "point-and-shoot" strategy for aeroshell deployment on approach variety has the virtue of simplicity, but at the expense of landing accuracy (especially for low entry flight path angles). Of course this accuracy can be improved by putting a GN&C system on the aeroshell. Also execution accuracy for approach can be improved by a combination of steep entry flight path angle and simply delaying approach deployment until the last "minute" (2 days outs, 1 day out, hours...?). Waiting however, incurs .a Delta-V penalty.

The design choice of putting an on-board GN&C system then leads one to the scenario proposed here. That is to deploy aeroshells on approach, but that deployment may begin immediately after the Trans Mars Injection (TMI) burn. In this way the aeroshells are independently guided to entry interface from post TMI separation from the boost stage. Since the aeroshells all perform direct entry they are all of the same design (ie. there are no disparities between having to design both orbital deployed and approach deployed aeroshells).

The Development of a Spacecraft Bus

Whether or not an existing bus such as the Mars Observer bus can be used, there is significant development, integration, testing and certification to go through prior to launch. A closer look is warranted to compare the costs of developing a large central bus with separate aeroshells as contrasted to developing a simple aeroshell deployment mechanism and many small, independent spacecraft. The development of many smaller and simpler appears to have great potential to lower the overall costs of the GNM mission, and may help moderate costs for the following sample return mission by providing valuable infrastucture and heritage for the SRM program.

This leverage would be provided by the design of a single small aeroshelled lander which could have broader application than the currently proposed penetrator concept. Once a kick stage has provided the necessary trans-Mars Delta-V, only attitude maintenance and periodic midcourse corrections up to the point of entry interface are required for the proposed spacecraft. The GN&C heritage to solving this problem is vast, and an off-the-shelf solution requiring little more than integration is possible, given the current trend towards miniature satellites. A spacecraft required to do orbital insertion and orbital deployment is in my opinion an unnecessary complication. Each aeroshell would simply maintain course and attitude until entry interface, and from there follow a passive ballistic trajectory (no aeromaneuvering) up to terminal descent.

Mission Strategy

There is a possibility that a vigorous, aggressive development schedule could produce a '96 launch. This is possible because of the strong heritage that exists from previous and current engineering and development efforts. In any case, the science objectives and program enhancing opportunities available from this proposal argue for launch in multiple opportunities. For instance, if the scheduled launch dates were in successive years (say '96, '98, and '01), a unique strategy for mission reliability exists. If a first attempt at an attractive site fails, PLAN on trying again later instead of sacrificing global placement for a strategy of sending two landers to every site in order to achieve redundancy. Or, if every thing works on the first try and the network is satisfactorily established - stop, you're finished, no extra launch or set of launches is required.

The Advantage of Smaller Independent Spacecraft

The idea of simple independent carriers has a number of other advantages:

1) It allows smaller, simpler launch vehicles like the Delta or Atlas to be used (while still allowing the launch of a GREATER number of landers from a Titan IV than currently planned), which translates both into costs savings for the agency and much greater launch flexibility.

2) The mission is adaptable at modest cost. The global network can be sustained, added to, or evolved incrementally as questions arise, objectives evolve, and instrumentation improves.

3) The payload bay is reconfigurable (more so as compared to a penetrator fore/aft body design). The science equipment bay on the proposed lander is reconfigurable to accommodate 20 Kg of science instruments specific to latitudes or science objectives.

4) The design is reusable and provides heritage to the SRM. There may be tremendous design leverage to be found in the SRM if the sites selected for the SRM can be visited by simple landers (either carried piggy back and deployed on approach or launched separately), that provide exact terrain knowledge at the site and establish navigation aids that lead the lander to a landing area verified to be safe per lander design. Using GNM heritage, this could be done at a fraction of the cost of a comparable imaging orbiter mission. The Human Exploration Vehicles could use these "throwaway" landers in a similar fashion, and to conduct specific surface experiments related to site selection.

5) The design may be suitable for micro-rover ("Ant") deployment.

6) The aeroshells may be placed with relatively high accuracy by employing radiometric approach navigation via the communications orbiter(s). This would provide a flight demonstration for this navigation technique for the SRM while enhancing the GNM. A high factor of safety for the GNM is retained since earth based navigation would probably be the primary method.

7) Engineering heritage for future possible missions. A modified aeroshell bus (without the aeroshell skirt) could be used as a flying testbed for various L/D configurations by modifying the aft aeroshell skirt. The testbed could be used to evaluate various GN&C algorithms and would as a bonus extend our operational understanding of the variabilities of the Martian atmosphere. This kind of testbed may the the most cost effective method of getting operational aerocapture experience at Mars. The aeroshell bus will also fit inside very small launchers such as the Orbital Sciences Pegasus or Taurus, or the General Dynamics Atlas-E. A deployable aeroshell skirt could be developed (which could have a much lower achievable ballistic coefficient), with a modified bus used in flight test and operations. This has the additional advantage of sending a large number of probes through the martian atmosphere thus building the engineering knowledge database of Mars atmospheric flight prior to launch of a Human exploration mission.

Technology

I believe this proposal can be accomplished with minimal technology risk. This may not qualify it as "technologically innovative", but I see no need to invent technology where it obstructs timely, cost effective execution of the mission. The possibility of pressing for a '96 launch should be investigated. However, for serious consideration of a '96 launch, funding for concept studies needs to be provided now.

For the mission proposed here, the program risk that I believe exists for early launch of high G designs is mitigated. This is a simple mission, with a single simple spacecraft to design (excluding the comm orbiter which has even greater heritage working for it). I am sure that no show stoppers exist for penetrators, but there seem to be significant development costs and schedule risks associated with them. The fact is that none of the instruments, with rare exception, have been developed and tested for the very high G environments, and I am not aware that INTEGRATION of this number and variety of high G instruments has ever before been attempted (CRAF penetrator is the nearest data point that I am aware of, but the G loads there are considerably lower than those considered for the Mars penetrators, especially the aft body G loads). The combination of designing for the intense thermal flux, radiation, and G load environments, have probably not been predominate considerations for the majority of past high G development programs.

In this proposal, the strategy was to provide a relatively generous 20 Kg science payload capability with an ample 10 watt constant power supply augmented with rechargeable batteries. Several types of science payloads can be envisioned, each tailored of objectives which vary with latitude and the required number of a particular experiment type. As far as satisfying the requirements which lead to penetrator designs (subsurface sampling, placement of seismic geophones) a number of proposals emerged in the Session B workshop for satisfying these requirements. For instance a flexible, cable driven drill for acquiring subsurface regolith samples to a depth of up to 3 meters should be quite possible to incorporate into one such payload type. Geophones may be placed away from the lander on teathers to reduce the chance of interference, or they may be driven into the surface with a pyrotechnic device. I believe that the consensus at the meeting was clearly that engineering solutions could be found to satisfy science objectives, whether the surface device was a soft, rough, or hard lander. For the proposed rough lander design, risk and cost are mitigated by the using current expertise in developing, integrating, and testing moderate G instruments (10's of G's instead of 100's or 1000's).

Mass Guess-timates

Subsystem or Component	Mass (Kg)
Science Payload (including atmosphere profiling)	20
Structure (primary and secondary)	45
Power	
RTG's (2)	5
Batteries	5
Communications	10
GN&C/Propulsion	
---Avionics	10
---Torque Wheel Assembly	10
---Attitude Control System (Spin/Despin) and RCS Hardware	10
---Tanks & Fuel	35
Thermal Protection	30
Thermal Control	5
Parachute Assembly	15
Miscellaneous	5
Total	205

Notes:

1) This breakdown was used to get a rough estimate of the total mass. The numbers here represent only an educated guess, actual mass may vary, perhaps significantly, from these based on a detailed requirements analysis of the Global Network mission, and a comprehensive mass assessment.

2) For an aeroshell diameter of 2.44 m (8 ft) the ballistic coefficient would be about 44 Kg/m^2, for a diameter of 3.05 m (10 ft), all other things being equal, the ballistic coefficient is about 28 Kg/m^2. Lower ballistic coefficient translates into higher entry G-loads and heating rates, but also into steeper achievable entry flight angles which improve landing accuracy and provide the ability to achieve higher (polar) latitudes; lower ballistic coefficient also means lower mach numbers, or subsonic conditions, at parachute deployment. Exactly what latitudes are achievable should be the subject of future study.

3) Usable payload volume is about 50 cubic centimeters (1.8 cubic feet).

Challenges...

This proposed design is certainly not without technical and programmatic challenge. I encourage others to critique this proposal, but some difficulties I can see with the design are:

1) Navigating a fleet of vehicles to Mars simultaneously. This may saturate an already oversubscribed DSN. An integrated DSN upgrade or an alternate communications and navigation approach or system may have to be pursued.

2) A systematic injected mass study may show that some of my estimates are significantly in error. For instance, a stellar sensor such as the Ball CS-203 is required to provide inertial attitude reference, but even the CS-203 at 5.5 Kg, 6 watts, and 9 arcsec accuracy is not as small or precise as desired; a lightweight, low power Canopus tracker is assumed to be available. A total target weight of less than 200 Kg is attractive, I believe that lower weight (and thus lower ballistic coefficient, higher achievable latitudes, and higher landing accuracy) is attainable given the current trend towards micro-spacecraft. In any case, using off the shelf miniaturized components and technology is key to the success of the proposed design approach. (Is this technologically innovative?)

3) Science objectives best accomplished from orbit will require another orbiter (Son of Mars Observer?), or perhaps science payloads could be piggy backed on the (separate) communications orbiter.

4) Establishing a shared communications infrastructure may be a challenge. The communications and operations requirements of the missions need to be analysed together to determine what the best approach is to solving both problems. The placement of GNM landers at the poles, for instance implies the need for highly inclined relay orbits, while a sample return operation may best be satisfied with an aerosynchronous relay orbit.

5) Achieving the desired (steep) entry flight path angles from approach velocity may be problematic. Heating rates and total heat load are of special concern. The proposed approach would rely heavily on the heritage of Shuttle, AFE, and the High Energy Aerobrake work currently underway for Thermal protection materials, heat resistant substructure, and insulation materials and techniques.

6) Mission planning to achieve the desired distribution of landers at preselected longitudes and latitudes at the proper lighting conditions for descent imaging may be constrained by orbital mechanics and the launch dates, combined with the achievable entry flight path angles (function of ballistic coefficient, G loads, heating - requires further analysis). It may be necessary to relax the lighting condition requirement for descent imaging for some of the sites.

7) Achieving a '96 launch date would require an immediate commitment to GNM concept studies, and an innovative approach to contracting, developing, managing, and administering the program.

Conclusions

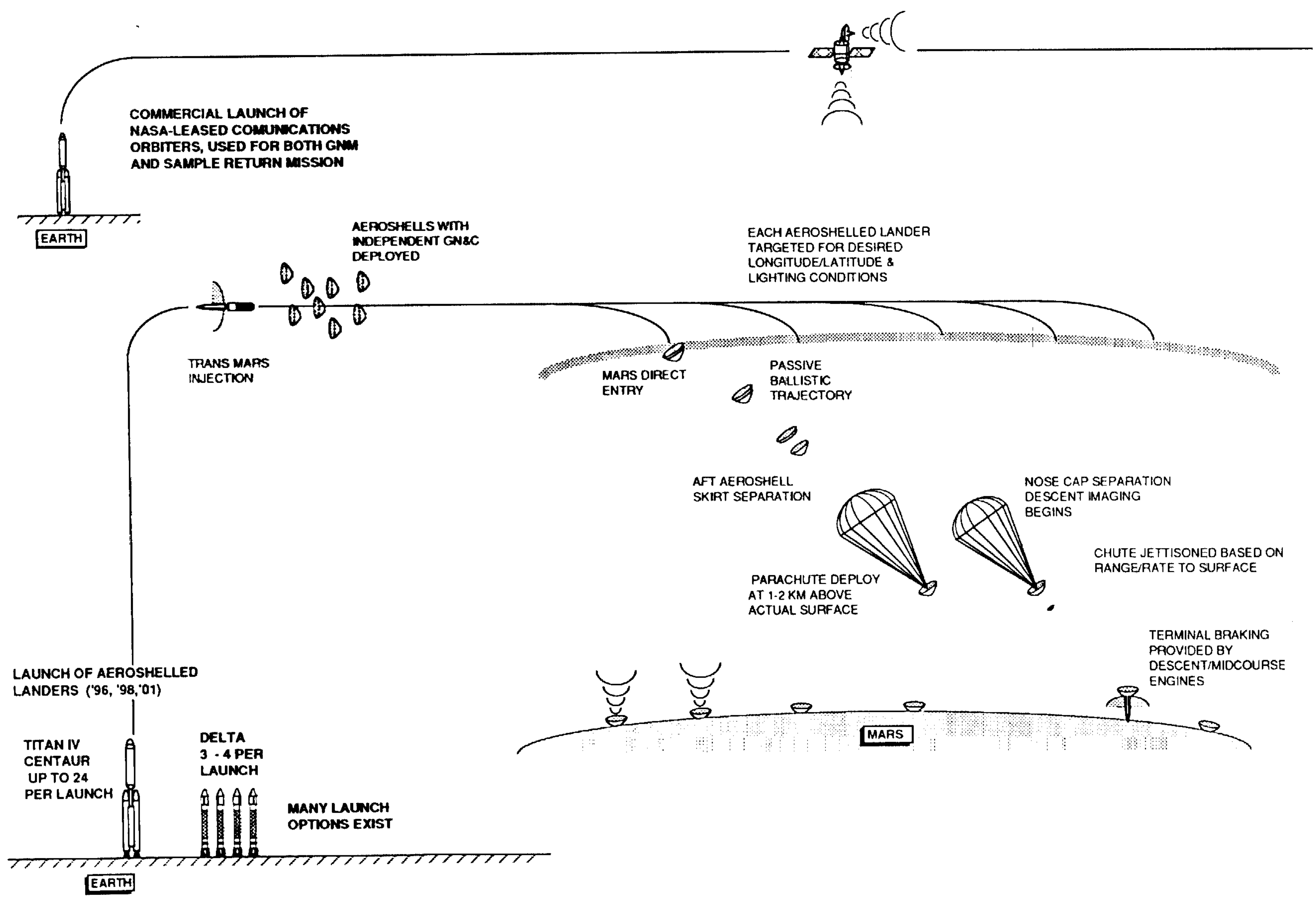
An approach such as the one proposed by Al Friedlander, which I have elaborated on here, has great promise in terms of reduction of cost and risk, increased flexibility, heritage and commonality, and I believe can reap substantial political dividends as well. However, a system engineering and cost estimation effort is needed to ascertain what the payoff of such a proposal might be. For a serious investigation of the possibility of a '96 launch of any description, it is imperative that funding of these important concept studies be swiftly provided.

While there is much refinement and analysis needed for this proposal, it has attributes which I hope will receive serious attention. My hope is that this and other proposals can generate the kind of discussions which will lead to a well balanced Robotic Exploration Program and Human Exploration Initiative.

I ask the readers of this proposal who have become hardened by the decade long neglect of planetary exploration to try to suspend doubt in a sustained exploration program. Consider the GNM in a broader context of planetary exploration that has a new commitment behind it. If there is significant gold to be found in getting science value and the taxpayers money's worth in this program, it is in looking beyond the event horizon of the next mission.

I believe the GNM work shop was very productive and I look forward to future discussion of this and other promising mission and system design options for the Robotic Exploration Program.

Proposed GNM Mission Scenario



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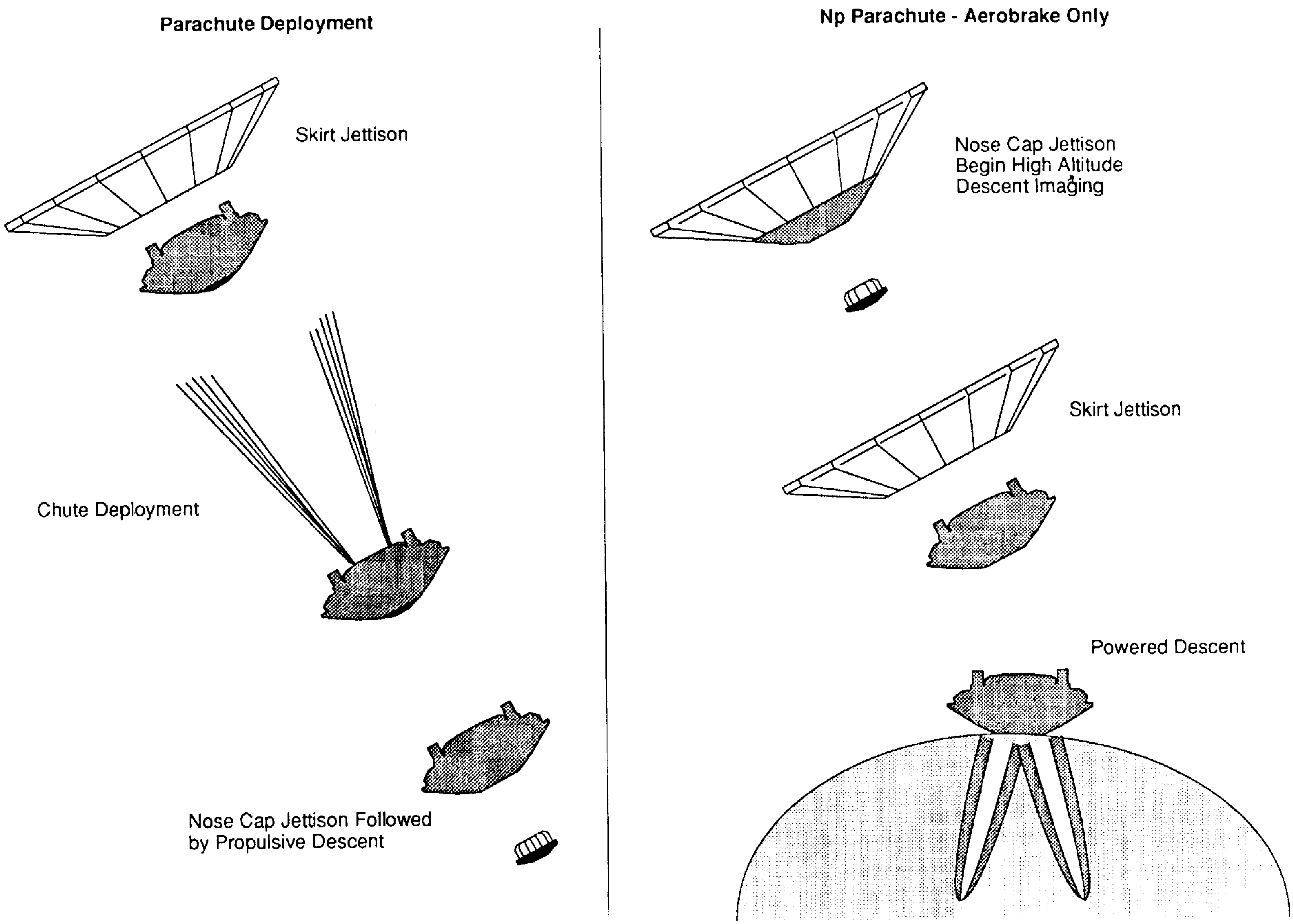
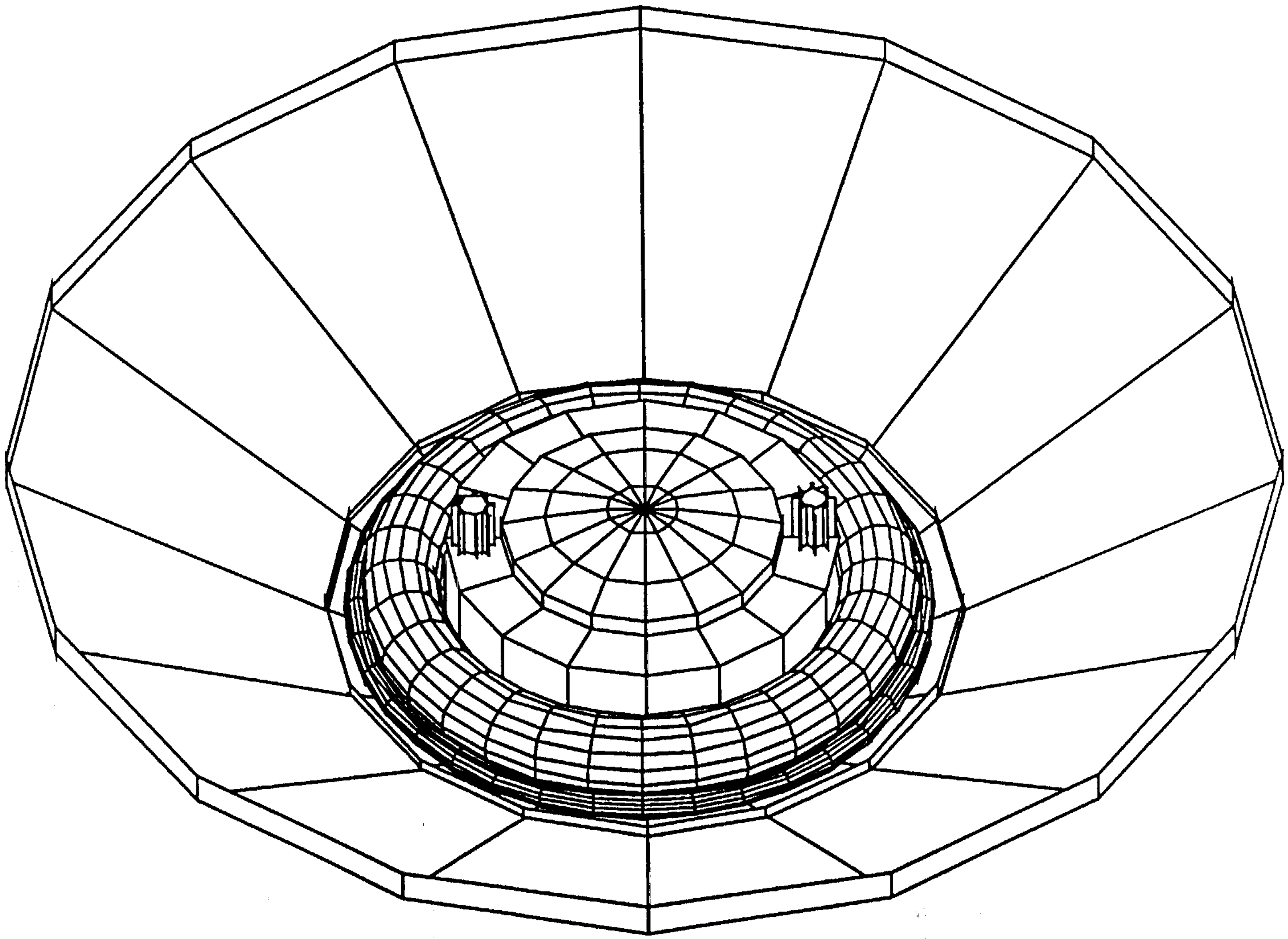
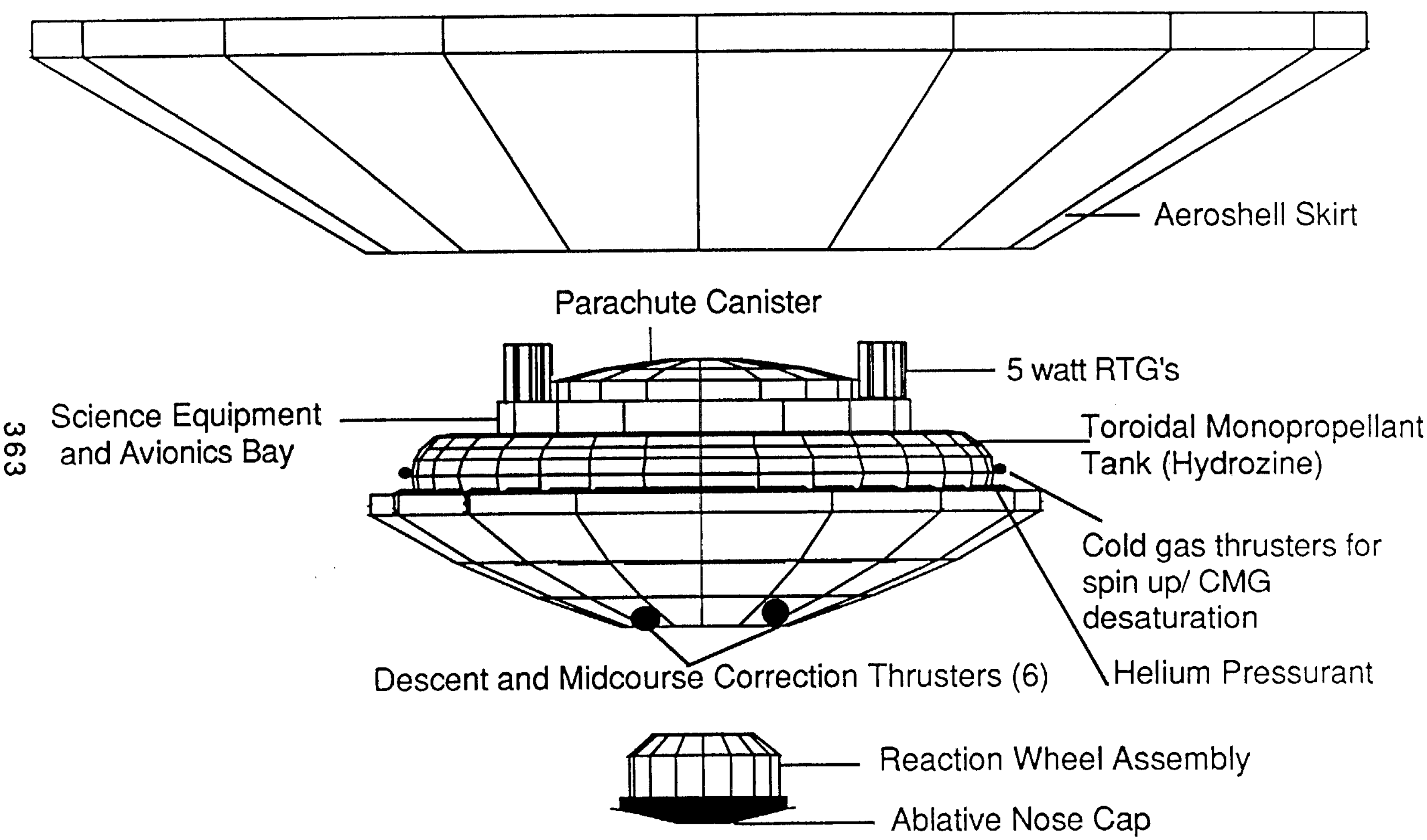
Terminal Descent Options



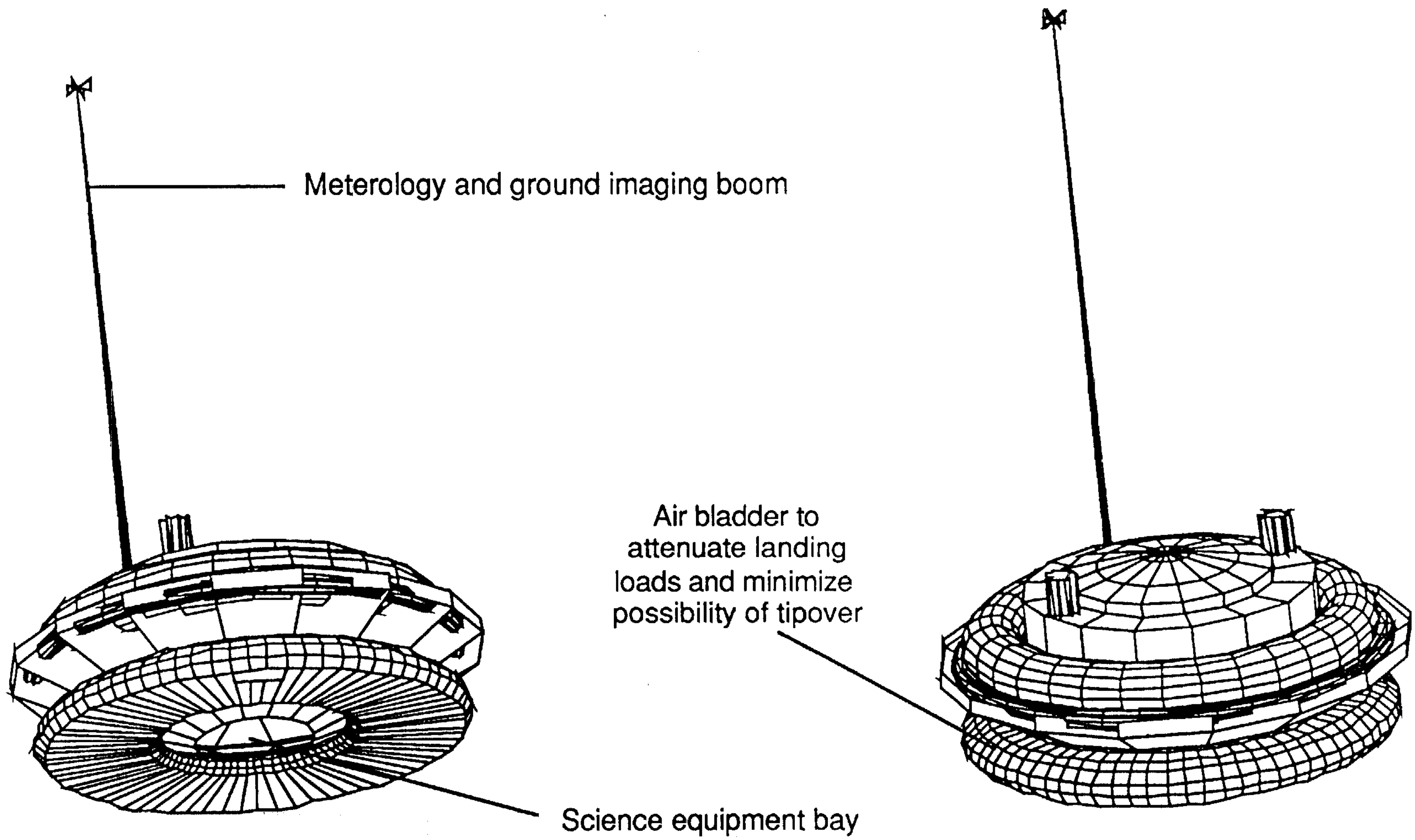
Aeroshelled Lander (Perspective)



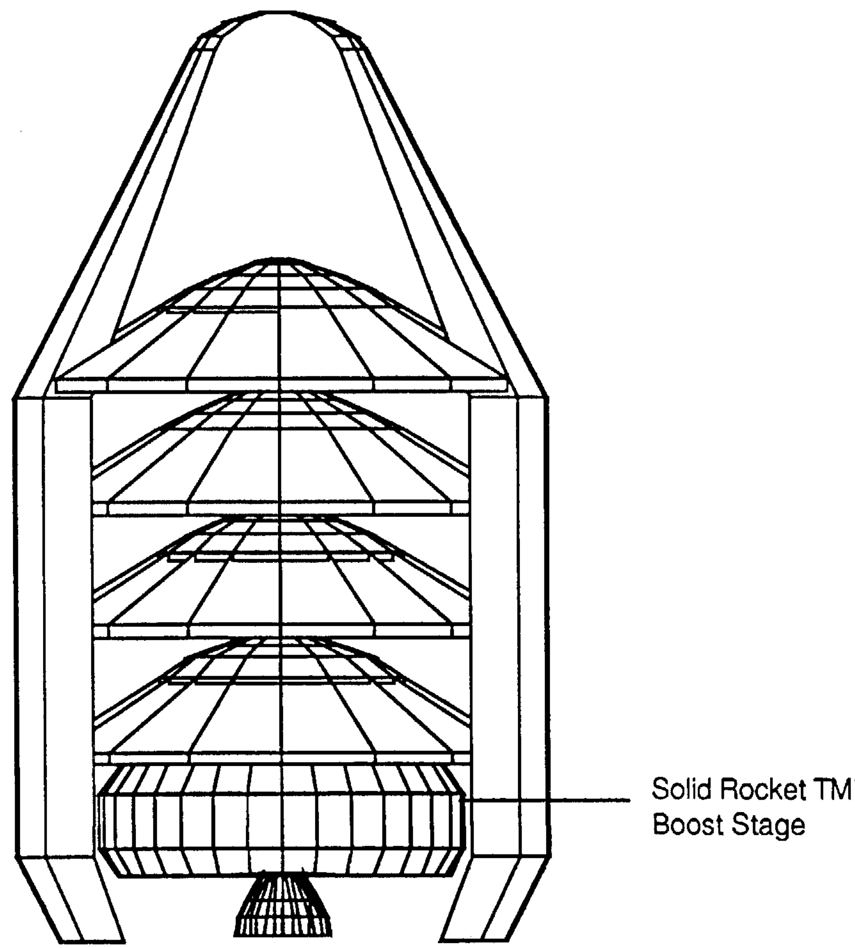
Simple Aeroshelled Lander Concept



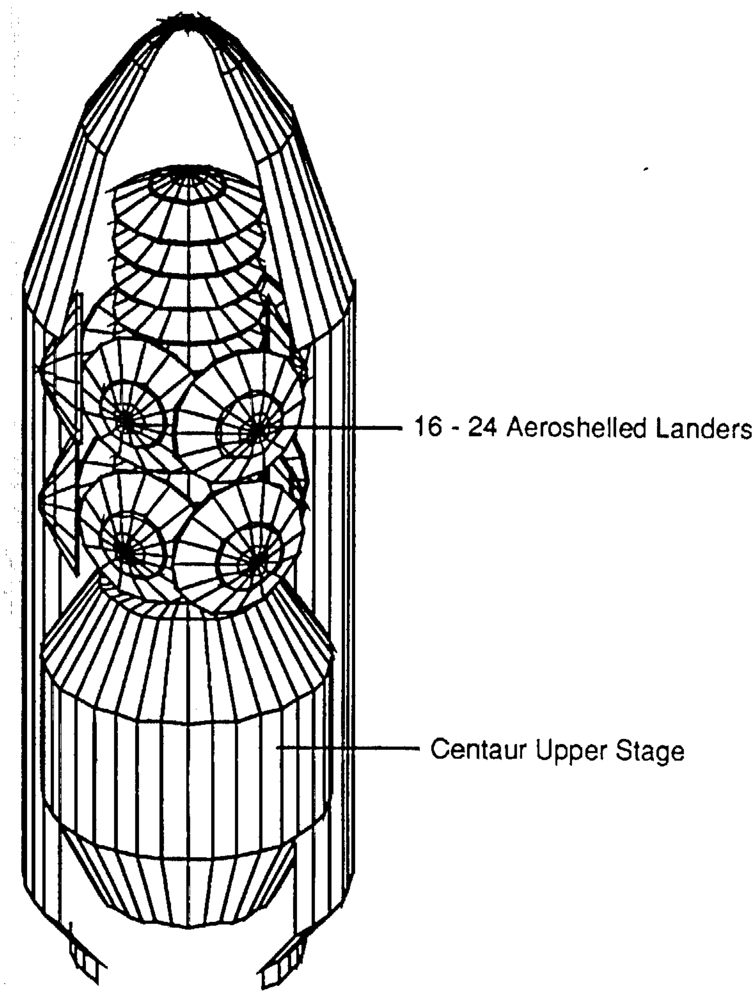
Lander Operational Configuration



Possible Delta Launch Configuration
3-4 Aeroshells per launch



Possible Titan IV Launch Configuration
15 - 24 Aeroshells per launch

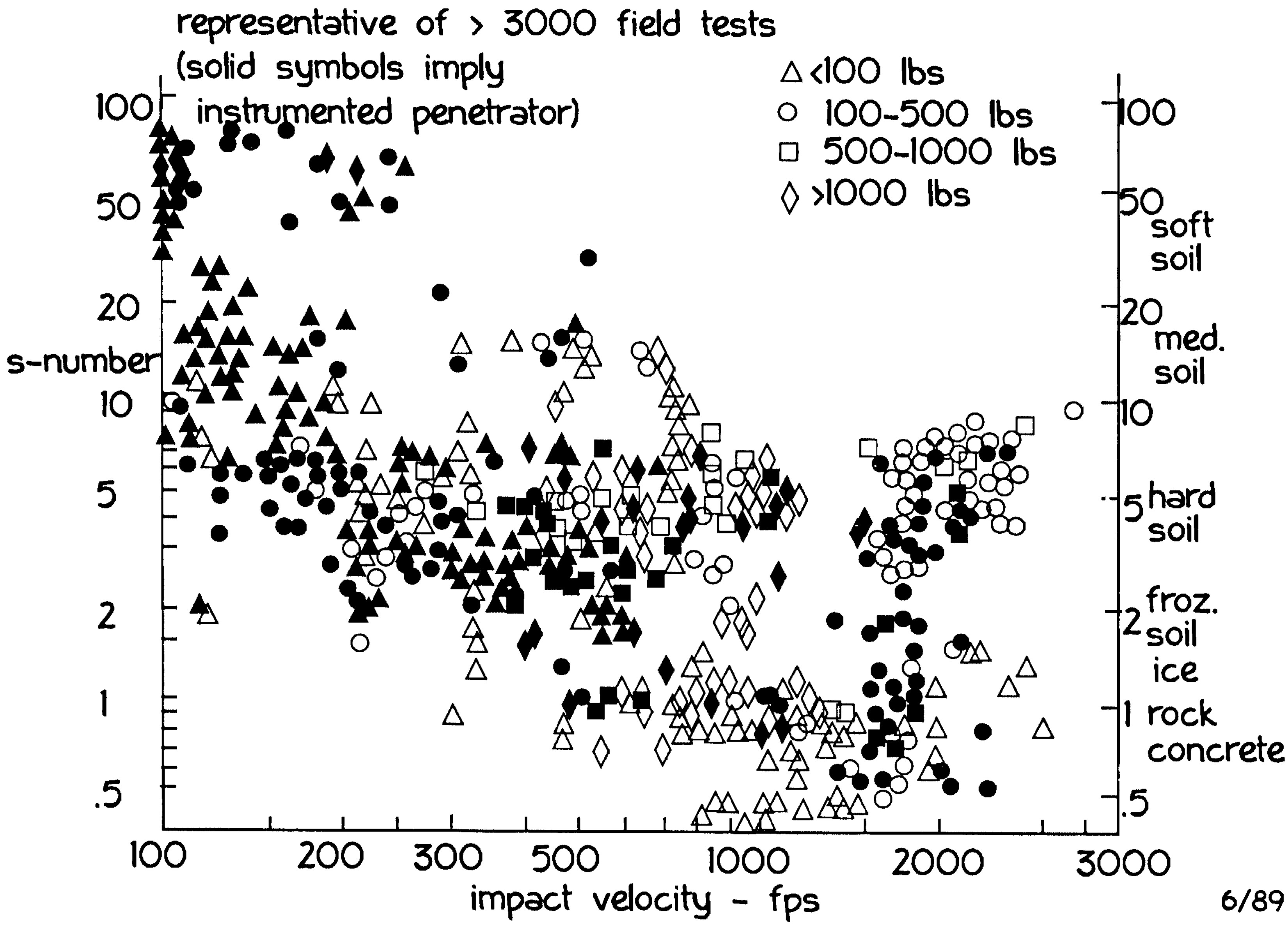


6.3 SESSION C SUBMITTALS

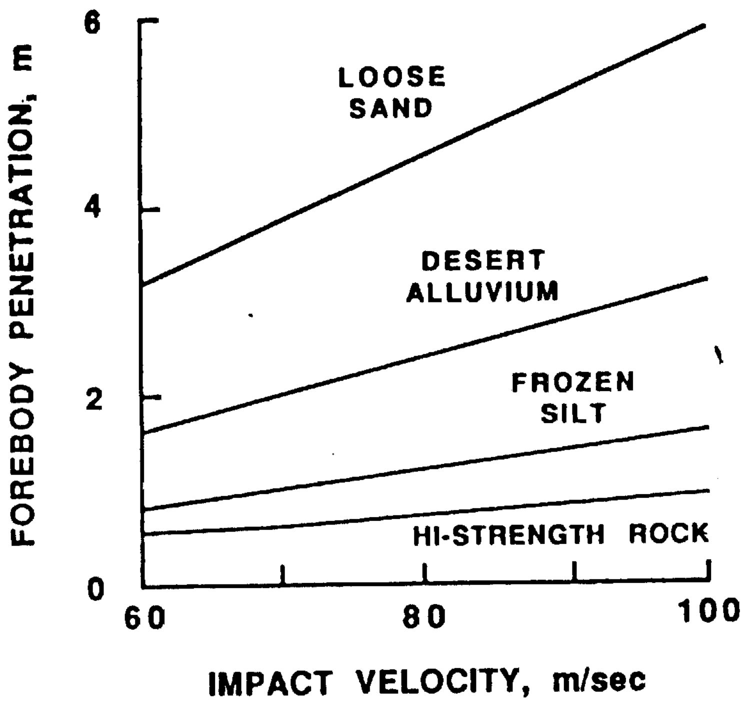
Session C, Submittal No. 1

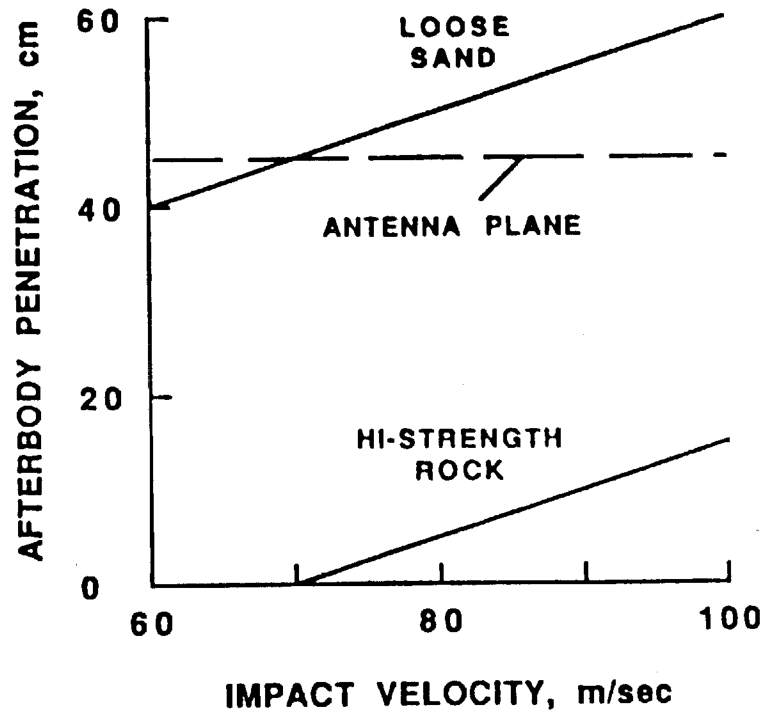
C. Wayne Young
Sandia National Laboratories

Sandia experimental earth penetration program

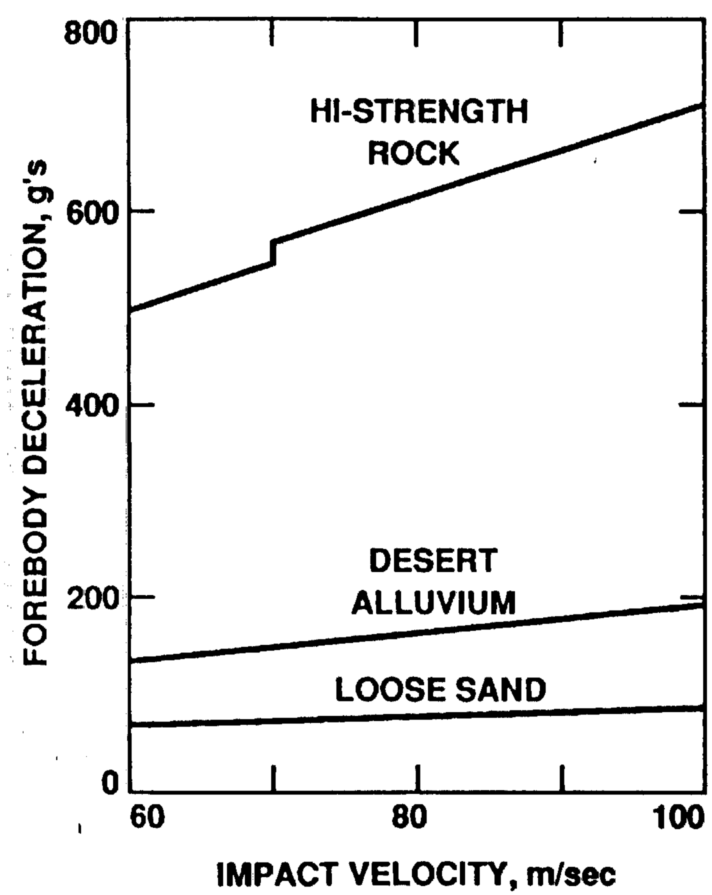


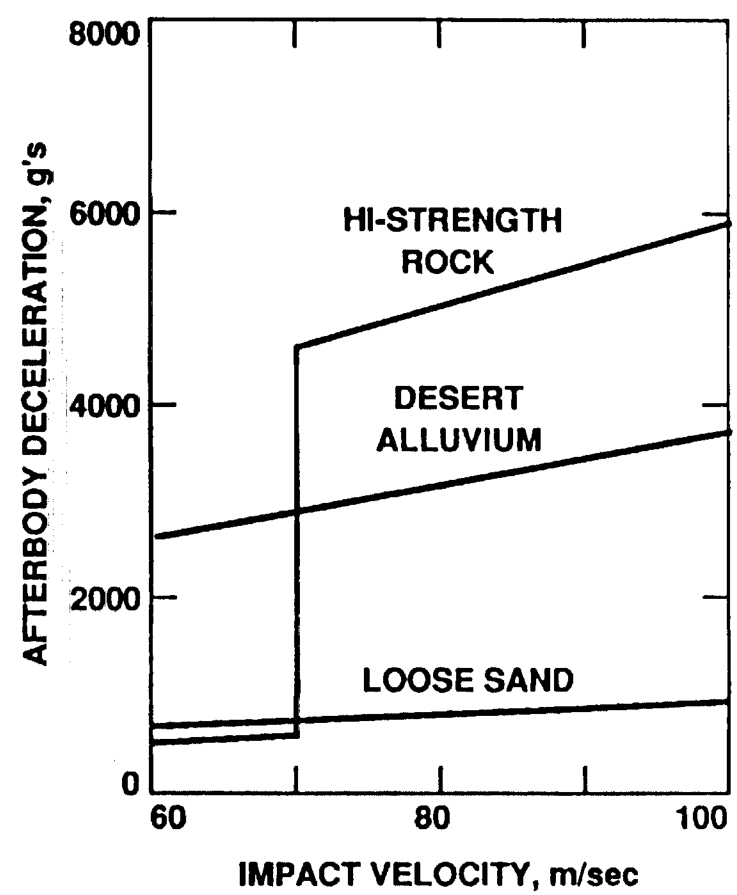
PENETRATION DEPTH





PENETRATION LOADS





Soil Penetrability

Unfrozen (use soil equation)

<u>Very hard</u> (dense and/or cemented sand, caliche, damp stiff silt or clay): $S = 5 \pm 2$

<u>Medium hard</u> (medium to loose sand, moist silt and clay): $S = 9 \pm 2$

<u>Soft</u> (topsoil, wet silt or clay): $S = 15 \pm 4$

Frozen (use ice equation)

Most moist to wet soils: $S = 2 \pm 0.25$

Dry soils: ?

Analytical Methods Recommended

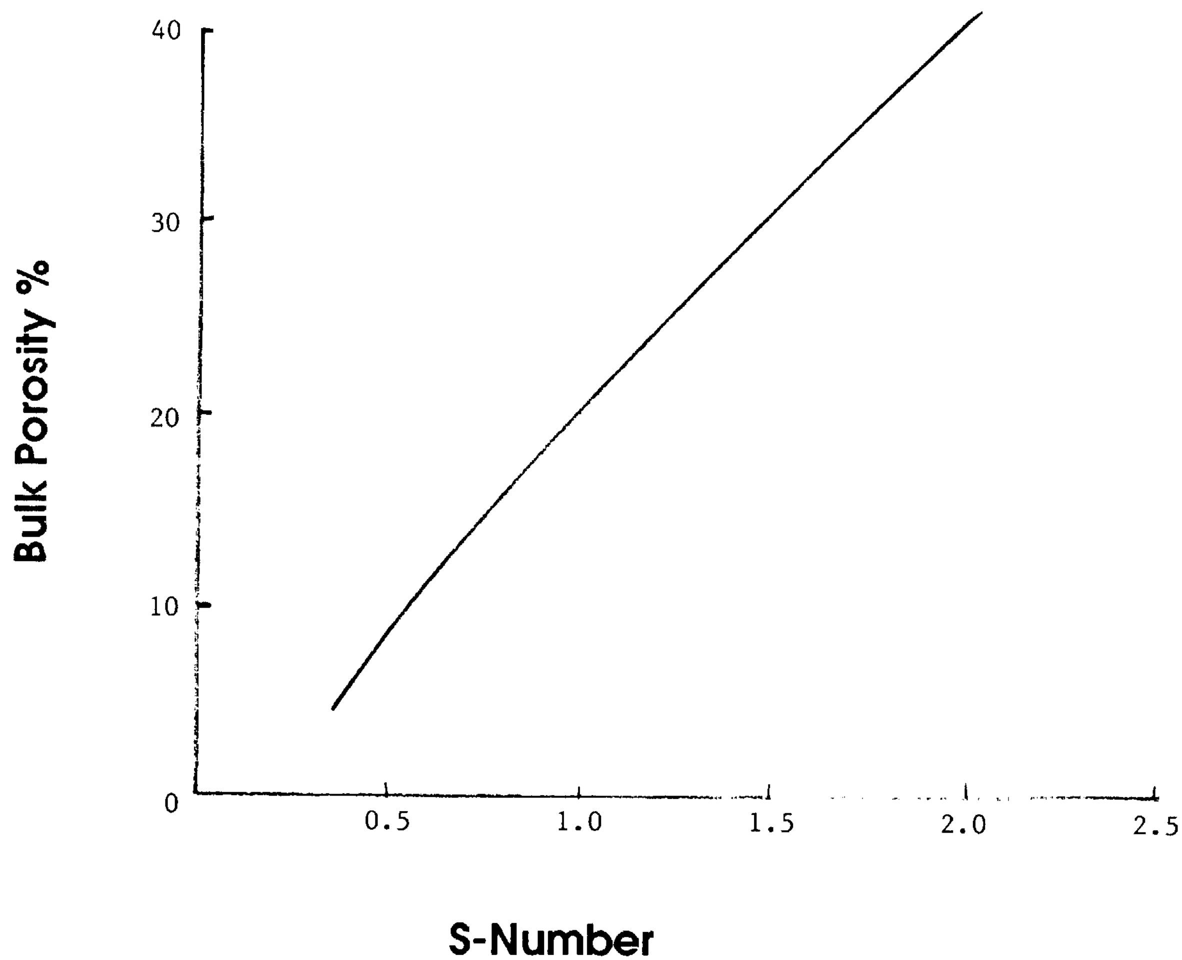
Axial Loading

- Sandia empirical equations
- Wavecodes are available, but perhaps not the most appropriate method

Lateral Plus Axial Loading

- SAMPLL - engineering tool
- Hydrocodes (PRONTO, DYNA, HULL, etc.) - more limited use, but necessary for coupled calculation with structural response

Rock Penetrability



Note: This is a useful guide, but not the only consideration.

Lateral Loading

NOTE: This will be critical element in penetrator design and component loading.

- Impact Angle
- Angle of Attack

} SAMPLL code & Hydrocodes

- Target nonhomogeneity (borders, etc.)
  - SAMPLL, with approximations
  - Mostly experiments with instrumentation

Penetration Technology

Issues requiring early effort:

- Target description in geologic terms
- Determine effect of boulders, etc.
- Evaluate effectiveness/utility of shock mitigation
- Suitability of Titanium for penetrator

Conclusion: The technology exists to develop a penetrator with suitable structural and penetration performance. As with previous penetrating systems, this effort will require proper combination of experience, analysis, and experiment.

Session C, Submittal No. 2

David E. Ryerson
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Sandia National Laboratories
Telemetry Department
High Shock Penetrator Instrumentation Program

D. E. Ryerson
Division 5144
February 2, 1990

Sandia National Laboratories Telemetry Department has been building high shock instrumentation systems for penetration studies for over twenty years. The instrumentation systems are digital stored data acquisition systems used to gather data during the penetration event and then recovered for data readout. The systems are powered by batteries, which are presently Eagle Picher LTC-7PST thionyl chloride batteries.

The shock loads that these systems are designed for are:

20,000 g for 1 millisecond
8,000 g for 10 milliseconds
3,000 g for 20 milliseconds
1,000 g for 50 milliseconds

Sandia has been fielding an average of sixty instrumented penetrator tests per year for the last five years. Attached is a plot of a sample penetrator test acceleration record.

To make our electronics survive high shock, we constrain all of the components very tightly in the penetrator package. We use selected components and encapsulate them in hard potting per the attached "Rules for Building High-g Electronics." Our temperature environment is typically between 0 and 50 degrees Celsius, so we can use components that would not survive standard military temperature ranges.

We normally try not to use shock attenuation to protect the electronic components. An analysis of shock attenuation is given on an attached page. It shows that to get shock attenuation, one must let electronics move a much larger distance than the penetrator housing, which is impossible.

We have used material to remove high frequency components of a shock pulse to protect such devices as accelerometers which can be broken by high-amplitude high-frequency inputs. The disadvantage of this shock material is that it may distort the accelerometer response and in certain cases, actually amplify certain frequencies of the shock pulse. In our work, we stay away from shock attenuators if at all possible.

HIGH-G PENETRATOR INSTRUMENTATION

Sandia National Laboratories
Telemetry Department

1. Digital Stored Data Acquisition System

2. Shock Load Design Levels

   20,000 g for 1 millisecond

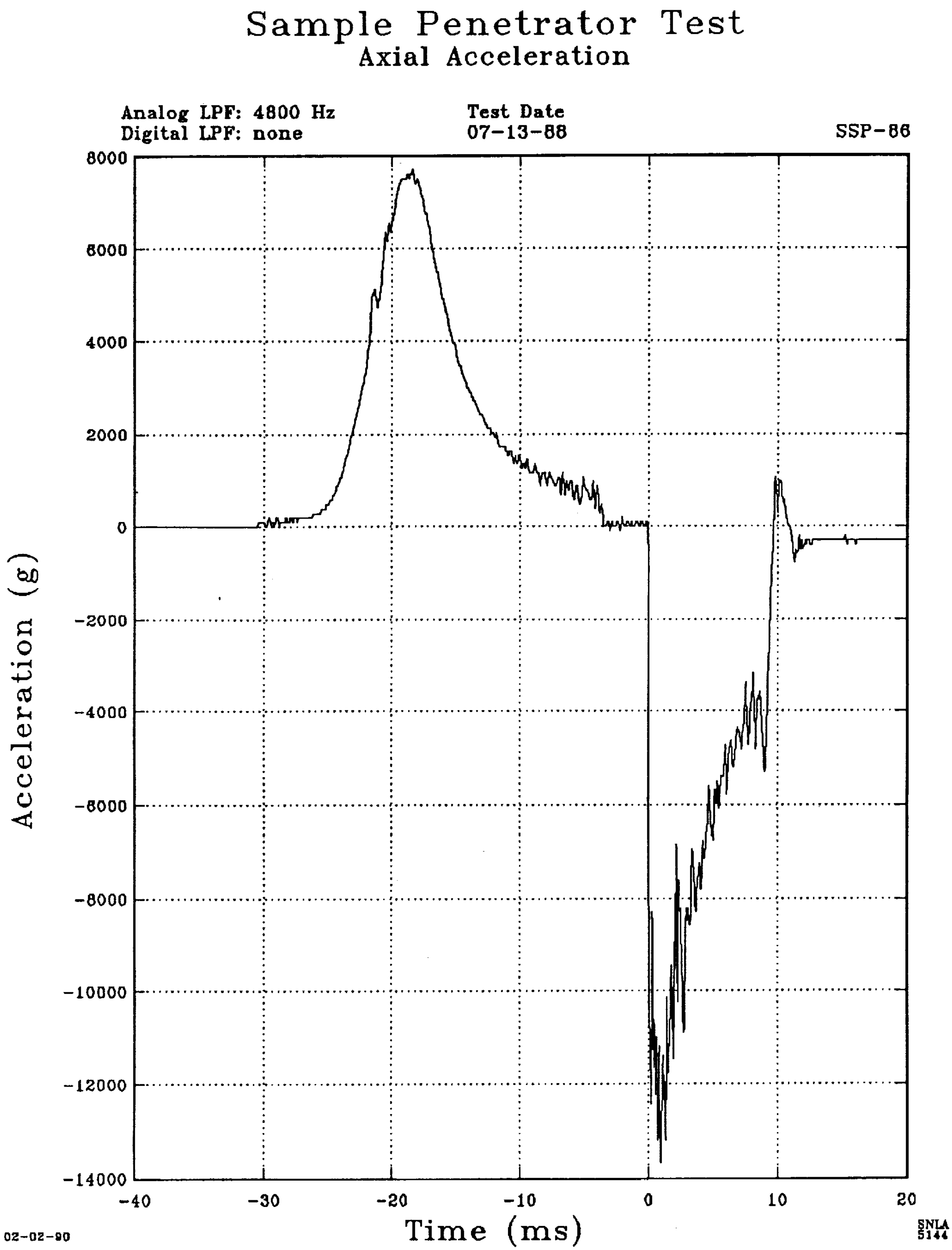
   8,000 g for 10 milliseconds

   3,000 g for 20 milliseconds

   1,000 g for 50 milliseconds

3. Components Tightly Constrained

4. Shock Attenuators Not Used


Sample Penetrator Test
Axial Acceleration
Analog LPF: 4800 Hz
Digital LPF: none
Test Date
07-13-88
SSP-86
8000
6000
4000
2000
0
-2000
-4000
-6000
-8000
-10000
-12000
-14000
Acceleration (g)
-40
-30
-20
-10
0
10
20
Time (ms)
02-02-90
SNLA
5144

RULES FOR BUILDING HIGH-G ELECTRONICS

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February 2, 1990

1. **Constrain the PC Boards and Components in Hard Potting** - Hard potting is required to keep components from moving during shock. Typical potting is epoxy filled with glass micro-balloons. Make sure electronics and potting material are compatible with temperature ranges that the system will see in curing of potting and system operation.

2. **Cover the Components with a Thin Layer of Soft Potting** - Soft potting protects components during the hard potting curing process. It also gives a slight cushion to the component. Typical potting used is polysulfide rubber. Some silicone-type materials will not work because they act like mold release and will not let the hard potting adhere to the boards.

3. **Use as Small a PC Board as Possible** - The smaller a board is the less likely it is going to flex and break.

4. **Mount Small Components such as Resistors and Diodes Away from PC Board** - Small components can be broken by a board which flexes, especially if the board has raised solder mounds or lands under the component.

5. **Mount Shims Between Integrated Circuits and PC Boards** - Potting will typically not flow under an IC and a void will be left. Voids or air pockets allow components to move and break.

6. **Interconnect PC Boards with Fixed Wires or Spring Sockets and Beryllium Wire** - Normal connectors are prone to break.

7. **Use Plastic Integrated Circuits** - Plastic integrated circuits have the wires running from the IC pins to the die encapsulated. Ceramic IC's leave a cavity for the wires and die. The small wires will often move and short out during shock in a ceramic IC.

8. **Do Not Use Large Electrolytic Capacitors - Use Only Ceramic Capacitors** - Many large electrolytic capacitors cannot take shock. Solid electrolytic capacitors such as Kemet parts may work. Avoid large capacitors if possible. If that cannot be done, test components under shock to determine survivability.

9. **Use Small Known High-Shock Batteries** - Large batteries typically have internal construction which will not survive shock. Test battery types under shock to determine survivability.

10. **Do Not Overcharge Batteries** - When batteries are overcharged or charged too fast, they will expand and crack the case or potting which holds the cells.

11. **Keep Power Consumption Low** - Keep the system power consumption low to keep battery size down.

12. **Preload Package when Mounting in Hardware**

13. **Present Major Shock Perpendicular to PC Board Instead of Along Board**

RULES FOR BUILDING HIGH-G ELECTRONICS

Sandia National Laboratories
Telemetry Department

1. Constrain PC Boards and Components in Hard Potting
2. Cover Components with a Thin Layer of Soft Potting
3. Use Small PC Boards
4. Mount Discrete Components Away from PC Board
5. Mount Shims between IC and PC Board
6. Minimize Connector Use
7. Use Plastic ICs
8. Don't Use Large Electrolytic Capacitors
9. Use Small Batteries
10. Don't Overcharge Batteries
11. Keep Power Consumption Low
12. Preload Package in Mounting Hardware
13. Shock PC Boards Perpendicularly

Shock Attenuation

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February 2, 1990

The purpose of a shock attenuator is to reduce the amplitude of a deceleration pulse. Assume a deceleration pulse of constant amplitude A for time T. Calculate the motion parameters as follows:

acceleration $= a = A$ for time T

velocity $= v = -V_o + \int a\,dt = -V_o + A\,t \quad , \quad 0 < t < T$

$$V_o = A\,T \quad \text{to force} \quad v = 0 \quad \text{at} \quad t = T$$

$$v = A\,(t - T) \quad , \quad 0 < t < T$$

depth $= d = -\int v\,dt = -A\,(^1/_2\,t^2 - T\,t) \quad , \quad 0 < t < T$

$$d = {}^1/_2\,A\,T^2 \quad , \quad t = T$$

A shock attenuator would reduce the deceleration by slowing the body over a longer time interval. Let's calculate the energy in the shock pulse and hold it constant as follows:

energy = E = force * distance = mass * acceleration * distance

$$E = m\,A\,{}^1/_2\,A\,T^2 = {}^1/_2\,m\,(A\,T)^2$$

let $E_2 = E_1 \quad \Rightarrow \quad \boxed{A_2\,T_2 = A_1\,T_1}$

since $d = {}^1/_2\,A\,T^2$ and $(A_2\,T_2)^2 = (A_1\,T_1)^2$

$$\Rightarrow \quad \boxed{A_2\,d_2 = A_1\,d_1}$$

Therefore, the time of the deceleration pulse is inversely proportional to the amplitude of the pulse to keep the energy in the pulse constant and the depth of penetration is also inversely proportional to the deceleration amplitude.

Summary

A shock attenuator must allow the device being decelerated to travel over a longer distance to get any shock attenuation. If the device is being stopped in centimeters, it may be possible to double the stop distance to halve the deceleration. If the device is being stopped in meters, it probably is not possible to double this stop distance.

In penetrator work at Sandia, we have found that shock attenuators do not work to protect our electronics. We have found that in some cases an elastic medium has been useful in removing the high frequency components or fast rise times of the deceleration pulse. If one is not careful, it is possible that such elastic media will become shock amplifiers at certain frequencies (resonances) rather than shock attenuators.

DECELERATION SHOCK ATTENUATION

Sandia National Laboratories
Telemetry Department

1. Assume Constant Deceleration of Amplitude (A)
   for Time (T) seconds
   for Depth (D) meters

2. For Constant Energy in Shock Pulse
   Amplitude (A) * Time (T) = constant
   Amplitude (A) * Depth (D) = constant

3. To Significantly Reduce Shock Amplitude, Depth Must Be Significantly Increased

4. Shock Attenuators Have Not Proven Feasible in Sandia's Penetrator Program

David E. Ryerson

Supervisor of Telemetry Technology Development Division 5144 at Sandia National Laboratories, Albuquerque, New Mexico.

BS in Electrical Engineering, Iowa State University, 1965.
MS in Electrical Engineering, University of New Mexico, 1967.

Worked at Sandia from 1965 to the present in telemetry, data acquisition, and control systems. Designed real-time aircraft computer-controlled systems for target tracking and rocket-launch computer systems for Sandia's Kauai test range. Developed long-life (1 to 3 years) ocean-floor seismic systems and underwater acoustic telemetry for data recovery. Presently directing the designing and fielding of ultra-high shock (up to 20,000 times gravity) penetrator data acquisitions systems, rocket and reentry vehicle instrumentation, and specialized data acquisition systems.

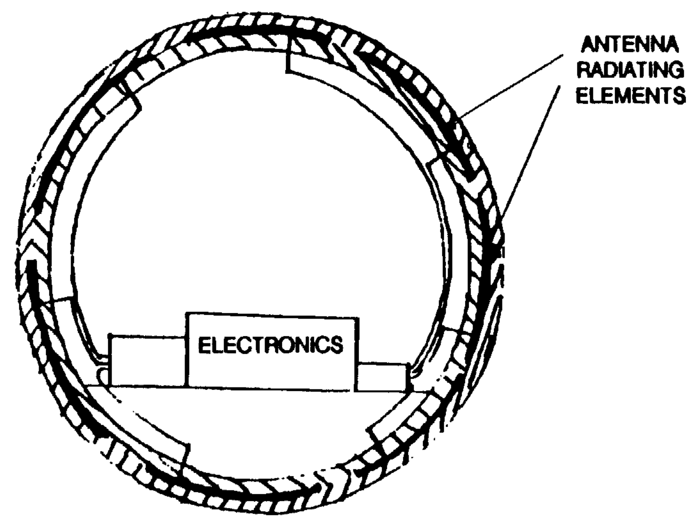
Session C, Submittal No. 3

Tomas A. Komarek
Jet Propulsion Laboratory/California Institute of Technology

PENETRATOR RF HARDWARE CONCERNS

- PACKAGING METHODOLOGY FOR SURVIVAL OF SHORT-DURATION, HIGH - IMPACT DECELERATIONS (TO 10,000 *g's*).
- THERMAL DESIGN OF RF/ELECTRONIC CIRCUITS FOR A WIDE RANGE OF SURFACE TEMPERATURES.
- LONG LIFE TIME (3 – 5 YEARS) FOR MANY DEEP TEMPERATURE CYCLES AFTER HIGH-IMPACT SHOCK.
- RELIABLE RF CIRCUIT PERFORMANCE WITH STABILITY FOR THE ANTICIPATED DIVERSE ENVIRONMENTS.
- CRYSTAL OSCILLATORS (RECEIVER, EXCITER, CDU, TMU).
    - CRYSTAL FILTERS FOR NARROW - BAND RECEIVERS.
    - FILTERS/DIPLEXERS (RF SYSTEM).
    - RF SWITCHES FOR REDUNDANCY SWITCHING.
    - RECEIVERS, EXCITERS, TRANSMITTERS, ANTENNAS AND RF COMPONENTS.
- VOLTAGE?
- TRANSMITTERS.
- CONCLUSION: ATD NEEDED TO EVALUATE AND IMPROVE DESIGNS.

HIGH-IMPACT SPHERICAL SHELL ARRAY ANTENNA



- SURVIVE UP TO 10,000 g IMPACT.
- BEAM SWITCHING OR BEAM SCANNING
- GRAVITY-ASSISTED DIRECTION FINDING.
- UHF.
- APPROXIMATELY 10-dB GAIN.
- APPROXIMATELY 14 RADIATING ELEMENTS.
- APPROXIMATELY 12-IN. IN DIAMETER.

KEW 2/8/90

Session C, Submittal No. 4

Farley Palmer
Hughes Aircraft Company

Session C - Subsystem Technology Focal Questions

HUGHES AIRCRAFT COMPANY

1) What technology will help achieve 10 year lifetimes?

2) What technology will help survival of high-g landings?

3) Are RTGs a workable power subsystem (size, location on the lander)?

Achieving 10 Year Lifetimes

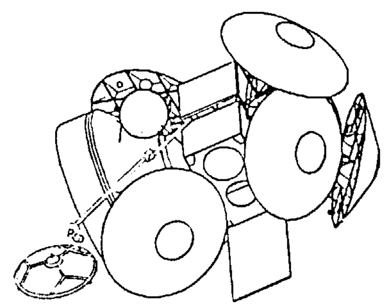
HUGHES AIRCRAFT COMPANY

Terrestrial communications satellites designed for >10 years today

- o Life = 14 yrs
- o Mean Mission Duration = 12.6 yrs at 0.74 reliability

UHF FOLLOW-ON

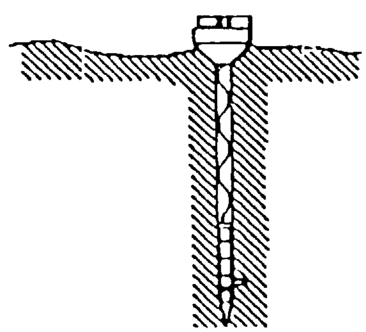
Direct application of contemporary satellite design practice



CARRIER

- o Identify requirements
- o Design
  - o Materials/parts
  - o Reliability analysis
  - o Redundancy
- o Qualification test

Cross-application of contemporary satellite design practice



PROBE

ISSUES

- o Requirements?
- o Which objectives require long life?
- o Can they be separated?
- o High-g survivability
- o Unique Martian environmental effects

Mars Global Network Mission
Science & HEI Functions

HUGHES AIRCRAFT COMPANY

Science or Function	Location	g-Hard Requirement	Life Requirement
Chemistry	Top-side/Sub	Possible	Short
Entry	Inbound	No	Ultra short
Hydrology	Sub-surface	Yes	Short
Imagery	Inbound/Top	No	Short->Medium
Impact	Sub-surface	Yes	Ultra short
Meteorology	Inbound/Top	No	Medium->Long
Mineralogy	Top-side/Sub	Yes & No	Short
Navigation beacon	Top-side	No	Very long
Seismometry	Sub-surface	Yes	Long
Sub-surface experiments	Sub-surface	Yes	Short
Surface experiments	Top-side	No	Short->Long

Probe Issues

HUGHES AIRCRAFT COMPANY

Science goals

Human Exploration Initiative (HEI) support & infrastructure goals

System optimization

Single-design vs multi-design
- Penetrator vs Penetrator & surface lander
- Long life vs Long & short life probes
- Large mass vs medium & low mass probes

Surface lander impacts vs soft-lands

Aeroshell vs propulsive deceleration

Trades Associated with Life
Power System Example

HUGHES AIRCRAFT COMPANY

LIFE, Yrs

	10^{-5}	10^{-4}	10^{-3}	10^{-2}	10^{-1}	10^{0}	10^{1}
Class:	G	F	E	D	C	B	A
Science:	Atmos-pheric entry			Site Imagery & Chemistry		Meteorology	Seismol-ogy

- Do all science objectives require integration in a single probe?
- What power availability is required?
  - Guaranteed vs Intermittant-but-statistically-probable
- Is the requirement "baseload," "bi-modal," or "transient periodic/aperiodic?"

Candidate Power System Approa- ches

HUGHES

Class	Type	Availability	Burst Power	Example
A1	Bi-modal	Guaranteed	Oversized primary with or without secondary storage	• Primary battery • RTG + 2ary batte- ry or flywheel
A2	Bi-modal	Intermittant but stat- istically probable	Oversized primary with or without secondary storage	• Solar panel or wi- nd generator + 2ary battery or flywheel
B1	Baseload	Guaranteed	N/A	• Primary battery • RTG
B2	Baseload	Intermittant but stat- istically probable	N/A	• Solar panel • Wind generator
C1	Transient	Guaranteed	Possible secondary storage	• Primary battery • RTG + 2ary batt- ery or flywheel
C2	Transient	Intermittant but stat- istically probable	Possible secondary storage	• Solar panel or wi- nd generator + 2ary battery or flywheel

High-g Survival

HUGHES
AIRCRAFT COMPANY

SDI Electromagnetically Launched Guided Projectiles

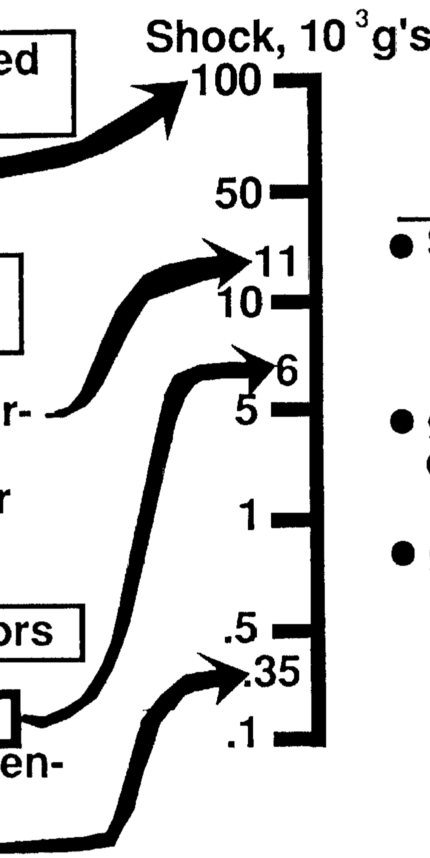
- (LEAP)

Artillery-Tube-Launched Guided Projectiles

- 155 mm Laser homing (Copper-head)
- 81 mm Infrared homing mortar bomb

Celestial Body Probes & Impactors

- Mars Global Network Mission
- European asteroid or comet pen-etrator
- Galileo Probe to Jupiter



High-g Technology

- Super-scale integra-tion to reduce mass and size
- g-hard subsystem designs
- g-hard packaging

Summary and Conclusions

HUGHES

- Achieving 10-year-life spacecraft is feasible
- Life concerns believed exclusive to probe
  - Power system (eg: RTG)
- Precedent exists for g-hard probe design
  - "Brilliant pebbles"-like guided missiles
  - Requires strong system integrator role

Session C, Submittal No. 5

Michael Shirbacheh
Jet Propulsion Laboratory/California Institute of Technology

(Information provided by Wayne M. Brittain,
Teledyne Energy Systems)

SPECIAL APPLICATIONS LOW-POWER RTG

DEVELOPMENT PROGRAM SUMMARY

January 1990

TELEDYNE ENERGY SYSTEMS

SPECIAL APPLICATIONS LOW-POWER RTG DEVELOPMENT PROGRAM SUMMARY

A. Background

The Special Applications RTG Development Program was initiated at Teledyne Energy Systems (TES) in September 1983 under DOE Contract DE-AC01-83NE32115. The development effort was performed under this contract through September 1988. After this time the program was continued as the Two-Watt Special Applications RTG Program (DOE Contract DE-AC01-88NE32142) with the objective of fueling a prototype RTG unit. Present activities at TES include fabrication, assembly and test of the electrically-heated prototype RTG which will be delivered to EG&G/Mound in June 1990 for fueling in December 1990.

Development of a sealed, 3-layer fuel capsule for use in the Two-Watt RTG is being performed for DOE in a joint effort by TES, EG&G/Mound and LANL. The capsule design is based on an upsizing of the Milliwatt RTG and Navy One-Half Watt RTG terrestrial 3-layer capsule technology.

B. Introduction

The primary objectives of the Special Applications RTG Development Program are to:

(1) develop a low-power (2 to 5W) relatively high voltage (5 to 12V) thermoelectric module using proven PbTe/TAGS thermoelectric materials. This materials technology has been applied to both NASA SNAP-19 space RTGs (Pioneer 10 and 11 Jupiter Fly-by spacecraft and Viking 1 and 2 Mars Landers), and terrestrial RTGs delivered to DOE

for subsea applications. Demonstrated thermoelectric module technology for low-power terrestrial RTGs at the initiation of the development program was limited to bismuth telluride with a typical RTG system efficiency of 3.5 to 4.0%. The goal for the development program was to increase this efficiency by 50%.

(2) develop a sealed heat source intended for terrestrial applications to contain the helium gas generated by the Pu-238 fuel decay. Available RTG heat source technologies for the anticipated thermal inventory requirement were all vented designs which result in increased parasitic heat losses with operating time due to the introduction of helium into the thermal insulation. The goal was to contain this helium within the capsule.

(3) design, fabricate, assemble, fuel and test a prototype terrestrial RTG system to demonstrate the developed technology. The selected terrestrial RTG design would consider potential near-term applications of low-power RTGs.

Although the hardware development for the Special Applications RTG has been oriented towards terrestrial applications, the thermoelectric module technology is generic and may be adapted to both space and terrestrial missions which require a low-power RTG power source. The radioisotopic heat source for space applications can be selected from available, qualified space hardware (such as the GPHS technology) or possibly be specifically designed and qualified for the mission requirements.

C. Thermoelectric Module Technology Description

The Special Applications thermoelectric module has evolved during the development program from using a couple with an all-PbTe N-leg and TAGS P-leg to one with Bi_2Te_3 cold segments on both the N and P-legs. The Bi_2Te_3 cold segments were added to the latest generation of thermoelectric modules to enhance the thermoelectric conversion efficiency for terrestrial applications where the RTG would be exposed directly to the cold subsea environment. These cold segments would not be beneficial for space applications and would not be included in the thermoelectric couple design.

1. Viewgraph 1

Viewgraph 1 shows Special Applications PbTe/TAGS minicouples which exemplify a configuration which could be considered for space application. The couple design is basically a miniaturization of the proven SNAP-19 space RTG thermoelectric technology. The couple has iron hot and cold shoes and copper pins to provide for electrically interconnecting the couples within a module. The Special Applications module uses a printed circuit board at the cold side to complete the interconnects between the couples. For the couple shown the individual legs are 0.102 in. sq. by 0.625 lg.

2. Viewgraph 2

Viewgraph 2 shows the typical internal construction of a Special Applications RTG. The configuration shown is that for the subsea prototype RTG now being fabricated at TES. The 30-pound weight shown is almost all in the BeCu pressure housing, with less than 5 pounds attributable to the RTG internal components (thermoelectric module, heat source, heat distribution cup, thermal insulation and preload springs). For a space RTG configuration, particularly for a penetrator mission with high shock loading, the internal configuration would probably vary somewhat from that shown to satisfy mission vibration/shock requirements. For example, the heat source could have a support system independent of the thermoelectric module to minimize dynamic loads on the module.

3. Viewgraph 3

Viewgraph 3 shows a typical Special Applications thermoelectric module containing 68 couples. The module is approximately 3 inches in diameter by 0.8 inch thick. The cold side printed circuit board provides the basic structure for the module. Powdered Min-K thermal insulation is vacuum-impregnated between the couples to minimize heat loss. A thermoelectric module similar to that shown has been successfully tested to a 100g axial, 50g lateral (both applied simultaneously) shock loading to simulate impact deployment of an RTG.

4. Viewgraph 4

Viewgraph 4 shows the typical performance for the subsea RTG design shown on Viewgraph 2. The BOL in-water power output is approximately 5W with a system efficiency approaching 7%. Note that the hot junction temperature of the terrestrial RTG is limited by the 3-layer capsule technology which has a long-term operating limit of approximately 1100^{O}F. For a space application the hot junction would probably be increased to the 950^{O}F range to take advantage of the high temperature heat source.

5. Viewgraphs 5, 6 & 7

Viewgraphs 5, 6 and 7 depict an alternate module configuration developed on the Special Applications program called a "Close-Packed-Array" or CPA. These viewgraphs show the configuration and performance of a 30-couple module rated at approximately 1.2W power output at 2.4V load voltage.

6. Viewgraphs 8, 9 and 10

Viewgraphs 8, 9 and 10 depict a module with a construction similar to that of the 30-couple module previously shown rated at 4.2W power output at approximately 6V load voltage.

7. Viewgraphs 11, 12 and 13

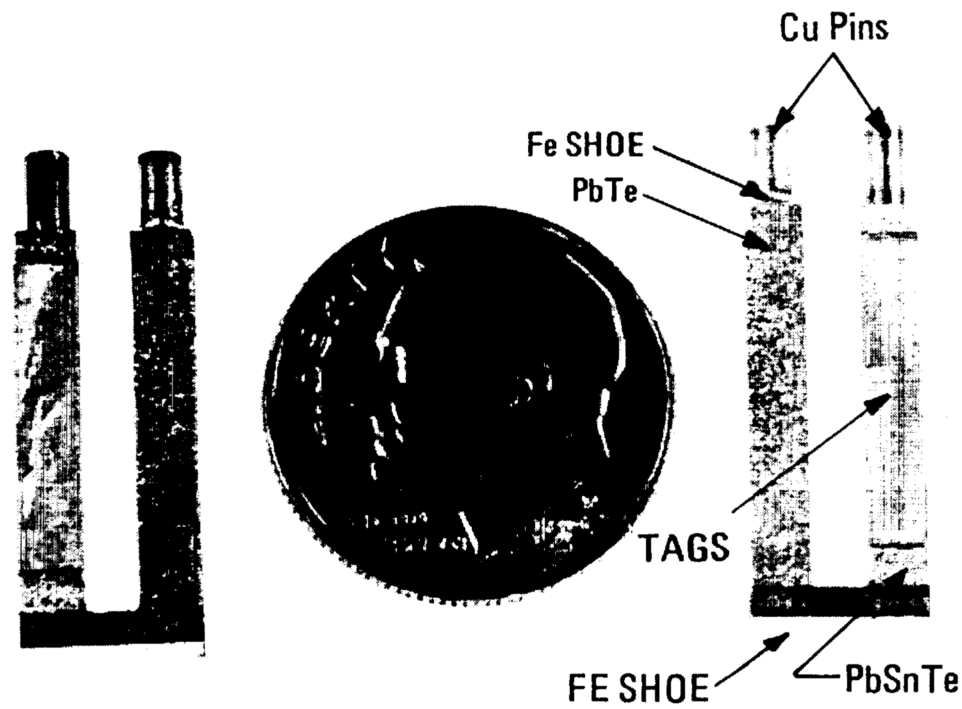
Viewgraphs 11, 12 and 13 show a 5W level module at approximately 9V load voltage. The module has 126 couples.

8. Viewgraph 14

Viewgraph 14 shows the conceptual design for a 10-15 W (at 9-12V) space RTG generated for a potential DOD space application using minicouples in conjunction with a 250W thermal GPHS heat source module. This concept uses the conventional SNAP-19 spring/piston cold end hardware arrangement to individually spring-load each leg of the thermoelectric couple. This arrangement is an alternate to that shown in Viewgraph 2 where the thermoelectric module is loaded as a unit with preload springs.

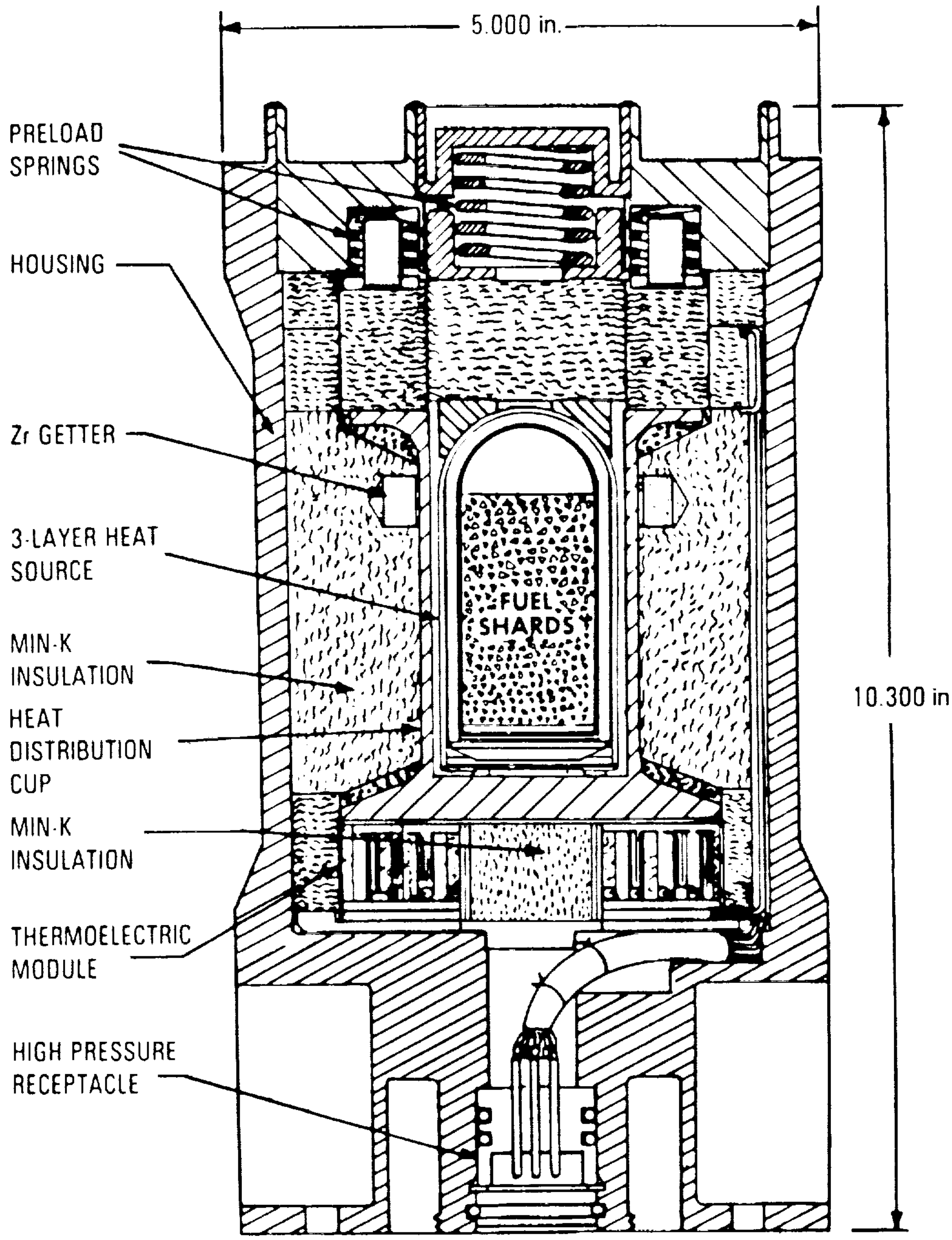
In summary, the Special Applications thermoelectric module technology is flexible both in its configuration and power level, permitting its adaptation to both space and terrestrial RTG missions requiring low-power RTGs. The RTG configuration and internal component support structure design would depend on the specific mission requirements.

SPECIAL APPLICATIONS THERMOELECTRIC MINICOUPLE



Viewgraph 1

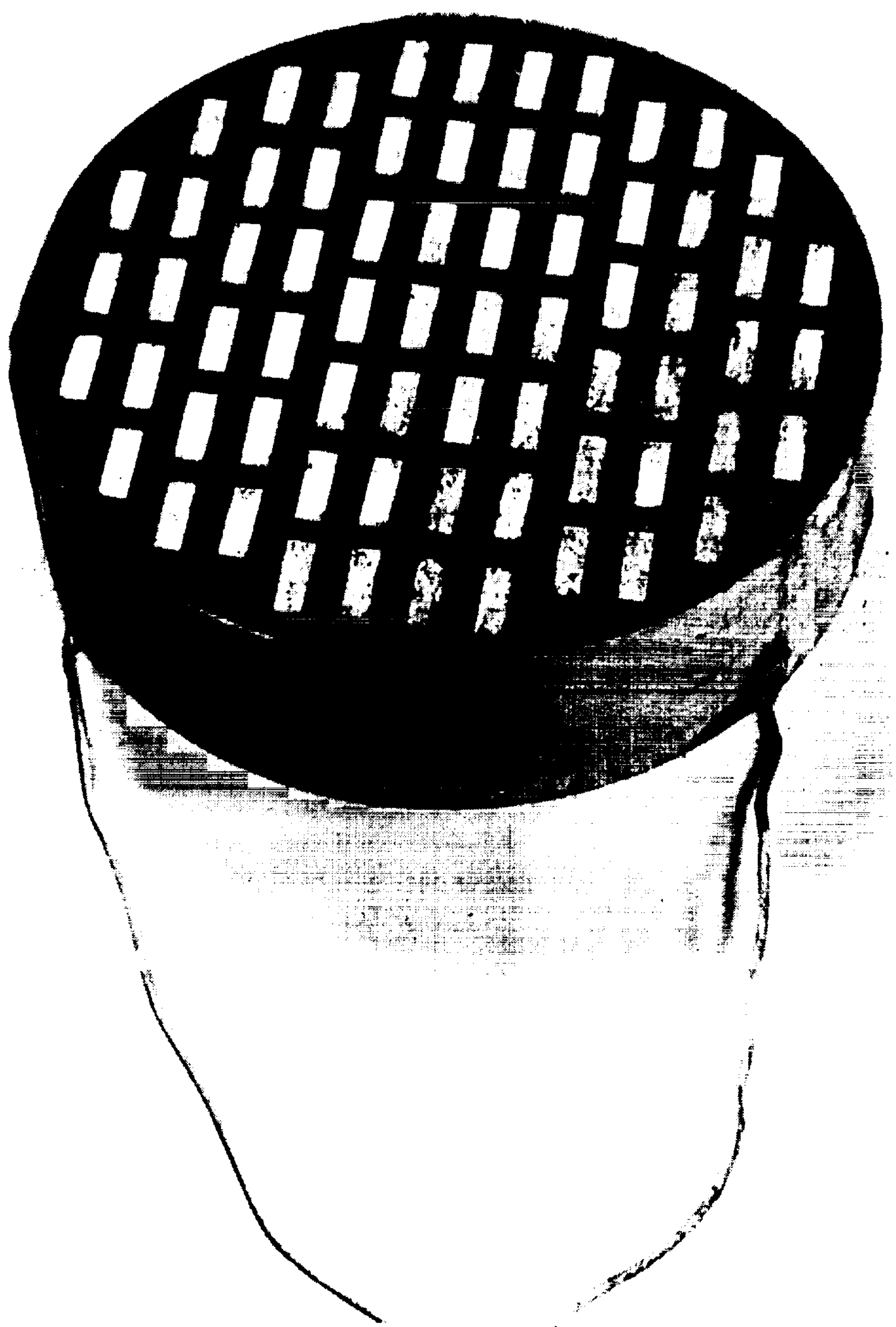
TWO-WATT SPECIAL APPLICATIONS TERRESTRIAL RTG
(BeCu Housing Design For 10,000 Psi External Pressure)



BOL POWER OUTPUT = 5.0 W(e)
THERMAL INVENTORY = 72 W(t)
APPROXIMATE WEIGHT = 30 LBS.

Viewgraph 2

SPECIAL APPLICATIONS MODULE
(HOT SIDE VIEW)



Viewgraph 3

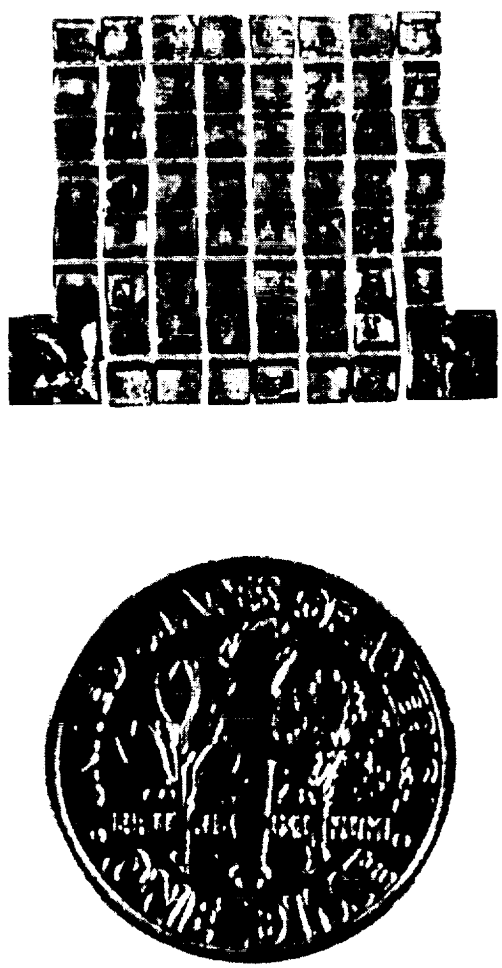
TWO-WATT SPECIAL APPLICATIONS RTG PERFORMANCE PREDICTION SUMMARY

	Worst-Case In-Air (BOL)	In-Water (BOL)	In-Water (10 Yrs.)
Power Output (W(e))	3.95	4.92	3.37 (2.6 @ .99 to 0.999 rel.)
Fuel Inventory (W(t))	70.8	70.8	65.4
T/E Efficiency (%)	7.74	9.36	6.89
Thermal Efficiency (%)	73.2	75.1	74.8
System Efficiency (%)	5.58	6.94	5.15
Hot Junction Temperature (^{O}F)	814	676	641
Cold Junction Temperature (^{O}F)	214	50	50
Ambient Temperature (^{O}F)	113	40	40

NOTES: (1) T/E element dimensions: 0.450 in. lg. x 0.103 in. sq.
(2) Number T/E couples: 68
(3) RTG fill gas: 100% xenon

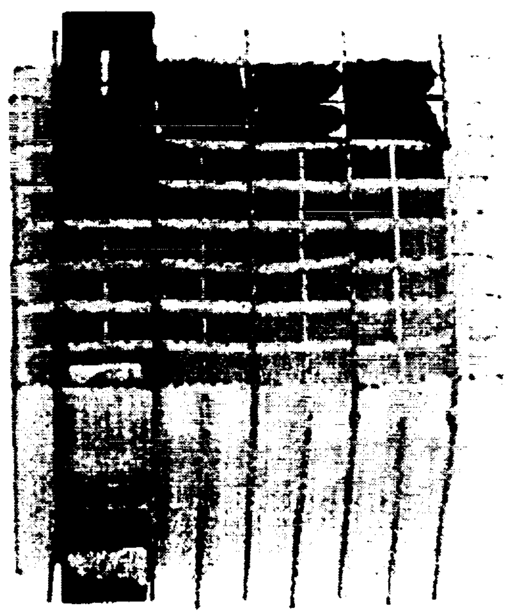
Viewgraph 4

30-COUPLE (.067" SQ. ELEMENTS) DEVELOPMENT MODULE QUADRANT (COLD SIDE VIEW)



Viewgraph 5

30-COUPLE (.067" SQ. ELEMENTS) DEVELOPMENT MODULE QUADRANT (PARTIALLY "STUFFED" WITH COUPLES)



Viewgraph 6

SPECIAL APPLICATIONS PROGRAM
30-COUPLE MODULE PERFORMANCE
(ELEMENT SIZE = .067" SQ. X .483" LG.)

	PREDICTION	QUADRANT #3 (11/11/84)
POWER OUTPUT (WATTS(e))	1.27	1.28
POWER INPUT (WATTS(t))	16.3 (1)	15.8 (2)
HOT JUNCTION (°F)	925	925 (3)
COLD JUNCTION (°F)	160	160
OPEN CIRCUIT VOLTAGE (VDC)	4.80	4.79
LOAD VOLTAGE (VDC)	2.40	2.41
INTERNAL RESISTANCE (Ω)	4.55 (4)	4.48
EXTRANEOUS RESISTANCE (%)	0	-1.5
THERMOELECTRIC EFFICIENCY (%)	9.3	-

(1) INCLUDES: $Q_{T/E}$ (14.1 W) + $Q_{SEPARATORS}$ (2.2 W)

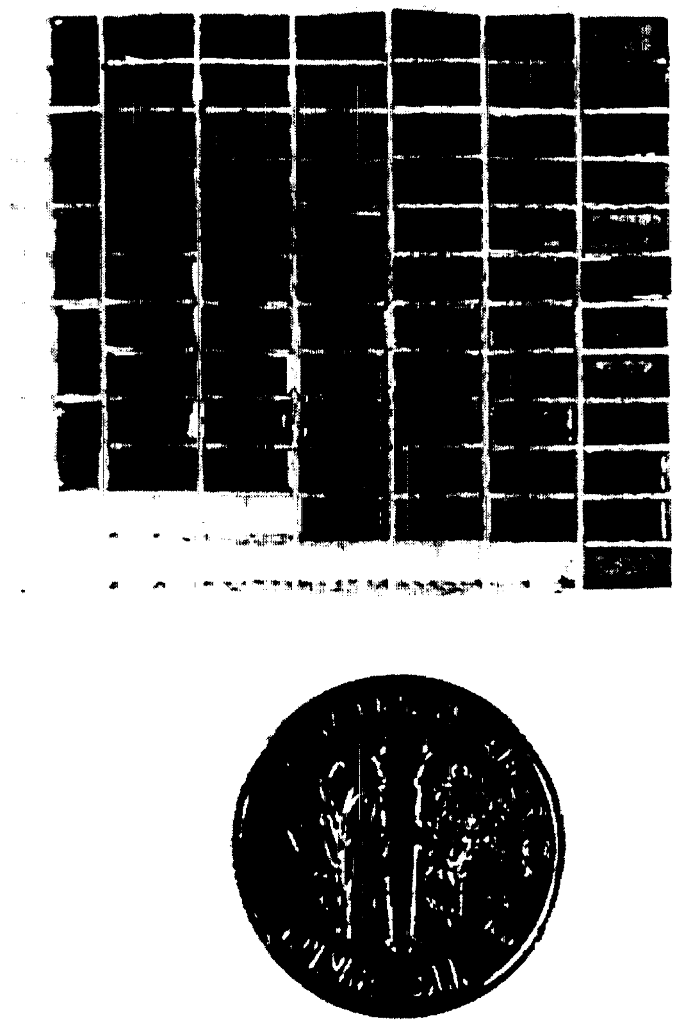
(2) MEASURED POWER INPUT LESS TEST FIXTURE TARE LOSSES.

(3) INFERRED TEMPERATURE BASED ON POWER INPUT AND OPEN CIRCUIT VOLTAGE.

(4) INCLUDES $R_{T/E}$ (4.38 Ω) + R_{STRAPS} (.07 Ω) + R_{LEADS} (.10 Ω).

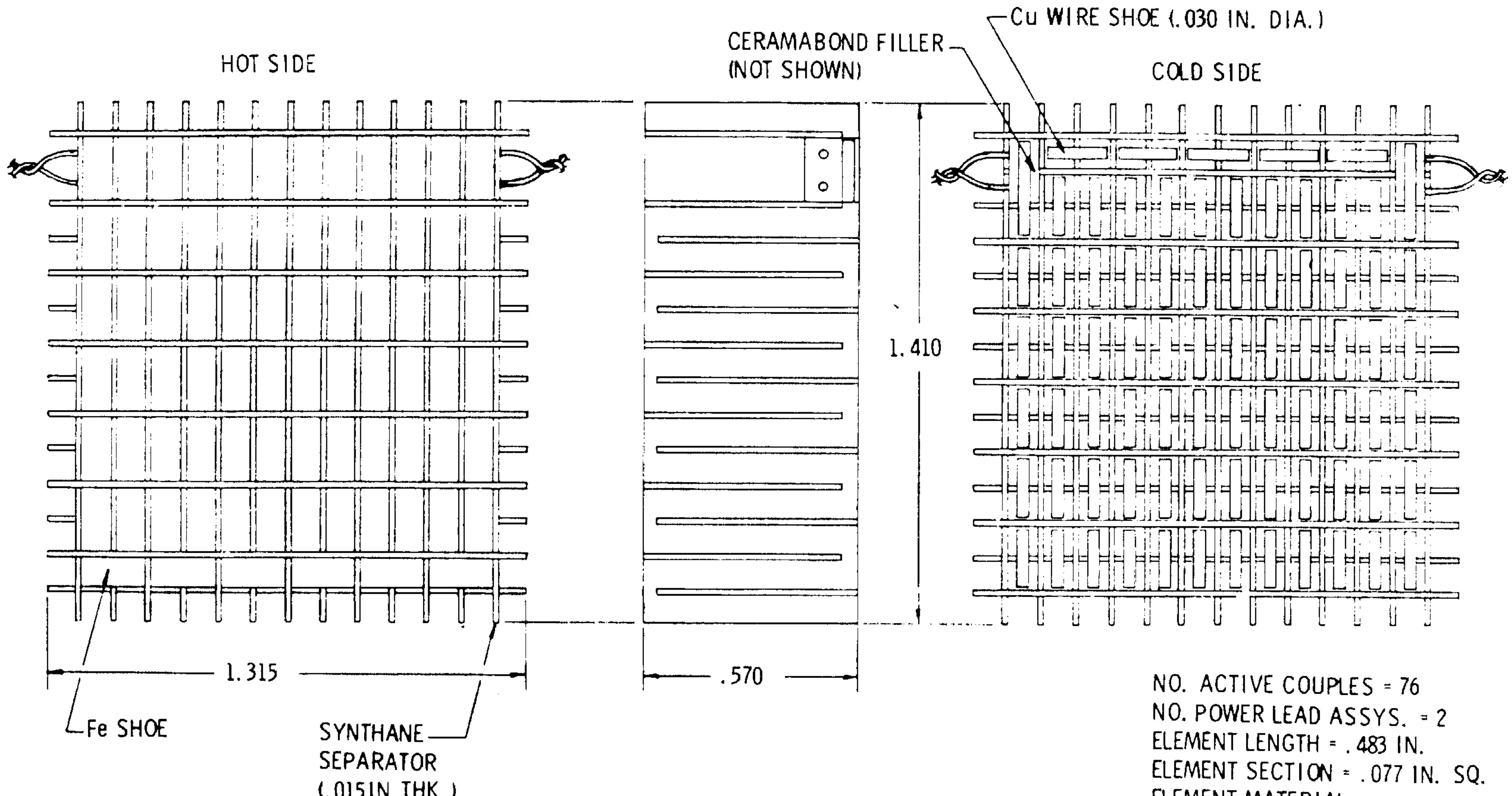
Viewgraph 7

SPECIAL APPLICATIONS TECHNOLOGY PARTIALLY ASSEMBLED MODULE ASSEMBLY



Viewgraph 8

SPECIAL APPLICATIONS PROGRAM 76-COUPLE DEVELOPMENT MODULE



NO. ACTIVE COUPLES = 76
NO. POWER LEAD ASSYS. = 2
ELEMENT LENGTH = .483 IN.
ELEMENT SECTION = .077 IN. SQ.
ELEMENT MATERIAL
N LEGS = TE1006
P LEGS = TAGS 85

Viewgraph 9

SPECIAL APPLICATIONS PROGRAM
76-COUPLE MODULE PERFORMANCE
(ELEMENT SIZE = .077" SQ. X .483" LG.)

	PREDICTION	MODULE S/N 6 (11/3/84)
POWER OUTPUT (W(e))	4.25	4.26
POWER INPUT (W(t))	56.1 (1)	56.2 (2)
HOT JUNCTION (°F)	925	925 (3)
COLD JUNCTION (°F)	160	160
OPEN CIRCUIT VOLTAGE (V)	12.16	12.14
LOAD VOLTAGE (V)	6.10	6.10
INTERNAL RESISTANCE (Ω)	8.69 (4)	8.65
EXTRANEOUS RESISTANCE (%)	0	-0.5
THERMOELECTRIC EFFICIENCY (%)	9.3	–

(1) INCLUDES: $Q_{T/E}$ (47.3 W) + $Q_{SEPARATORS}$ (8.8 W).

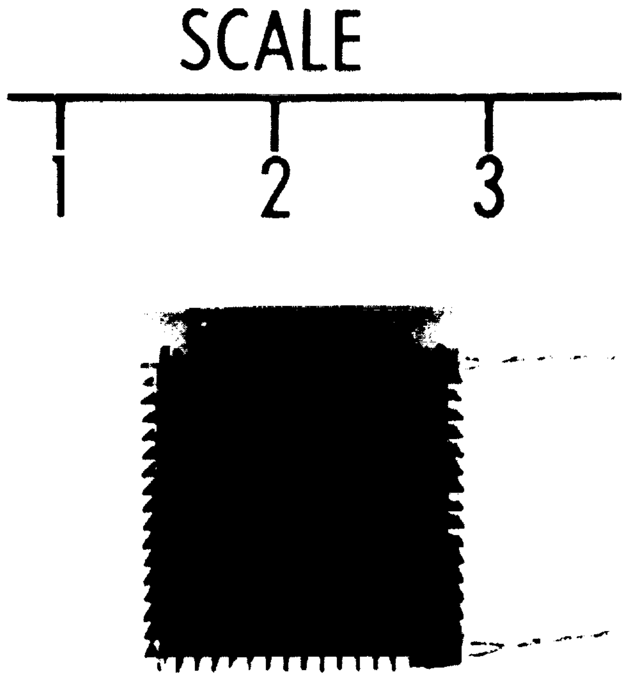
(2) MEASURED POWER INPUT LESS TEST FIXTURE TARE LOSSES.

(3) INFERRED TEMPERATURE BASED ON POWER INPUT AND OPEN CIRCUIT VOLTAGE.

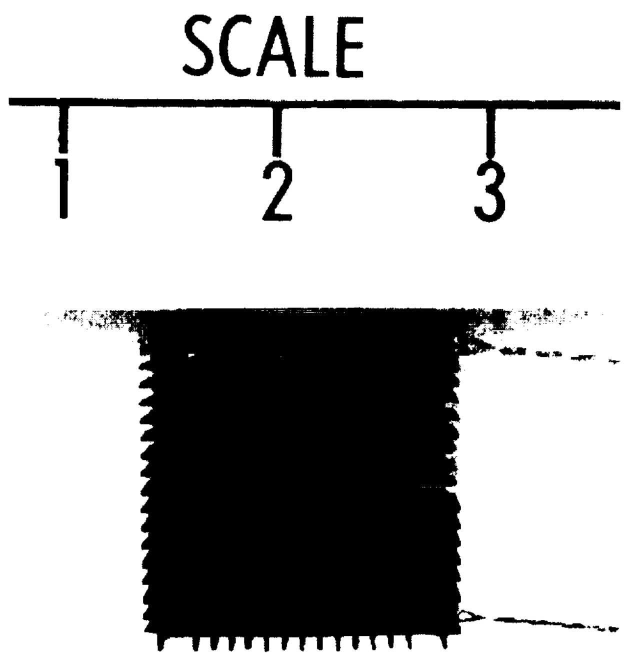
(4) INCLUDES: $R_{T/E}$ (8.41 Ω) + R_{STRAPS} (.18 Ω) + R_{LEADS} (.10 Ω).

Viewgraph 10

FIVE-WATT DEVELOPMENT THERMOELECTRIC MODULE



COLD SIDE OF COMPLETED MODULE

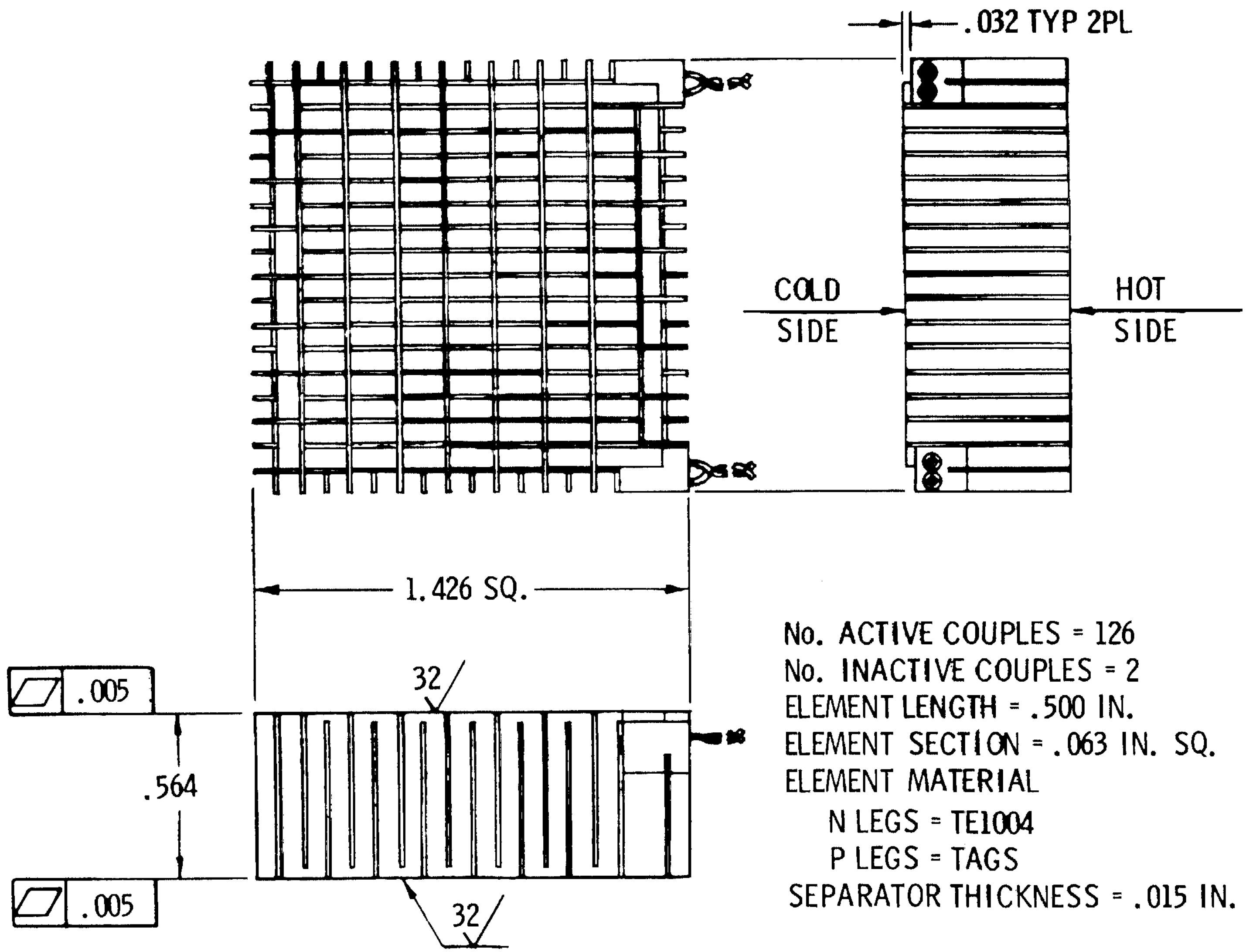


HOT SIDE OF COMPLETED MODULE

Viewgraph 11

ORIGINAL PAGE IS OF POOR QUALITY

THERMOELECTRIC MODULE ARRANGEMENT



No. ACTIVE COUPLES = 126
No. INACTIVE COUPLES = 2
ELEMENT LENGTH = .500 IN.
ELEMENT SECTION = .063 IN. SQ.
ELEMENT MATERIAL
N LEGS = TE1004
P LEGS = TAGS
SEPARATOR THICKNESS = .015 IN.

Viewgraph 12

SPECIAL APPLICATIONS PROGRAM
126-COUPLE MODULE PERFORMANCE
(ELEMENT SIZE = .061" X .466" LG.)

	PREDICTION	MODULE S/N 5 (10/12/84)
POWER OUTPUT (WATTS(e))	4.51	4.47
POWER INPUT (WATTS(t))	59.7 (1)	60.0 (2)
HOT JUNCTION (°F)	925	925 (3)
COLD JUNCTION (°F)	160	160
OPEN CIRCUIT VOLTAGE (VDC)	19.27	19.30
LOAD VOLTAGE (VDC)	9.63	9.72
INTERNAL RESISTANCE (Ω)	20.56 (4)	20.83
EXTRANEOUS RESISTANCE (%)	0	1.3
THERMOELECTRIC EFFICIENCY (%)	9.2	–

---

(1) INCLUDES: $Q_{T/E}$ (49.4 W) + $Q_{SEPARATORS}$ (8.2 W) + $Q_{INERT\ COUPLES(5)}$ (2.1 W).

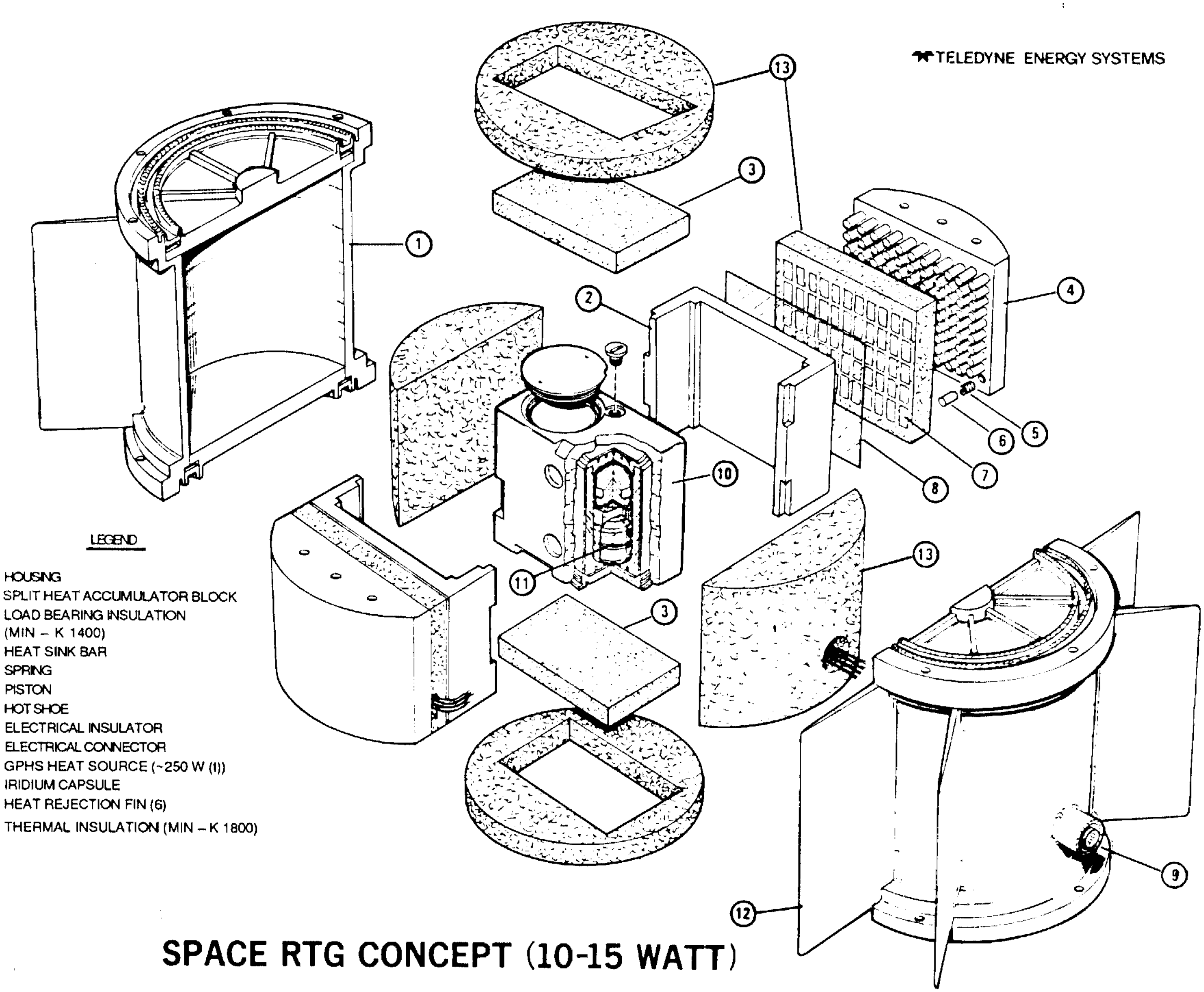
(2) MEASURED POWER INPUT LESS TEST FIXTURE TARE LOSSES.

(3) INFERRED TEMPERATURE BASED ON POWER INPUT AND OPEN CIRCUIT VOLTAGE.

(4) INCLUDES $R_{T/E}$ (20.40 Ω) + R_{STRAPS} (.06 Ω) + R_{LEADS} (.10 Ω).

Viewgraph 13

TELEDYNE ENERGY SYSTEMS



LEGEND

1 HOUSING
2 SPLIT HEAT ACCUMULATOR BLOCK
3 LOAD BEARING INSULATION (MIN – K 1400)
4 HEAT SINK BAR
5 SPRING
6 PISTON
7 HOT SHOE
8 ELECTRICAL INSULATOR
9 ELECTRICAL CONNECTOR
10 GPHS HEAT SOURCE (~250 W (t))
11 IRIDIUM CAPSULE
12 HEAT REJECTION FIN (6)
13 THERMAL INSULATION (MIN – K 1800)

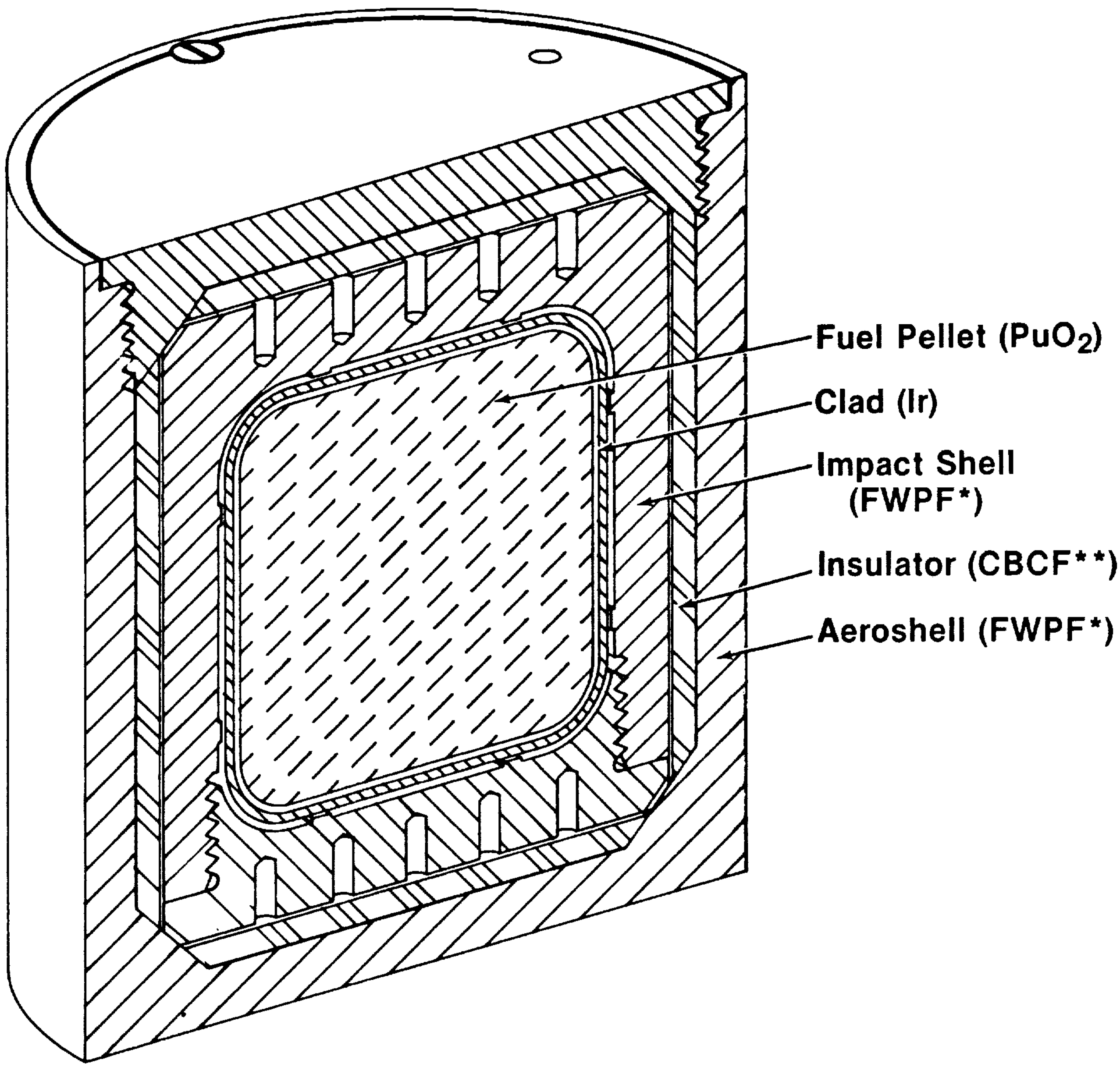
SPACE RTG CONCEPT (10-15 WATT)

Viewgraph 14

Session C, Submittal No. 6

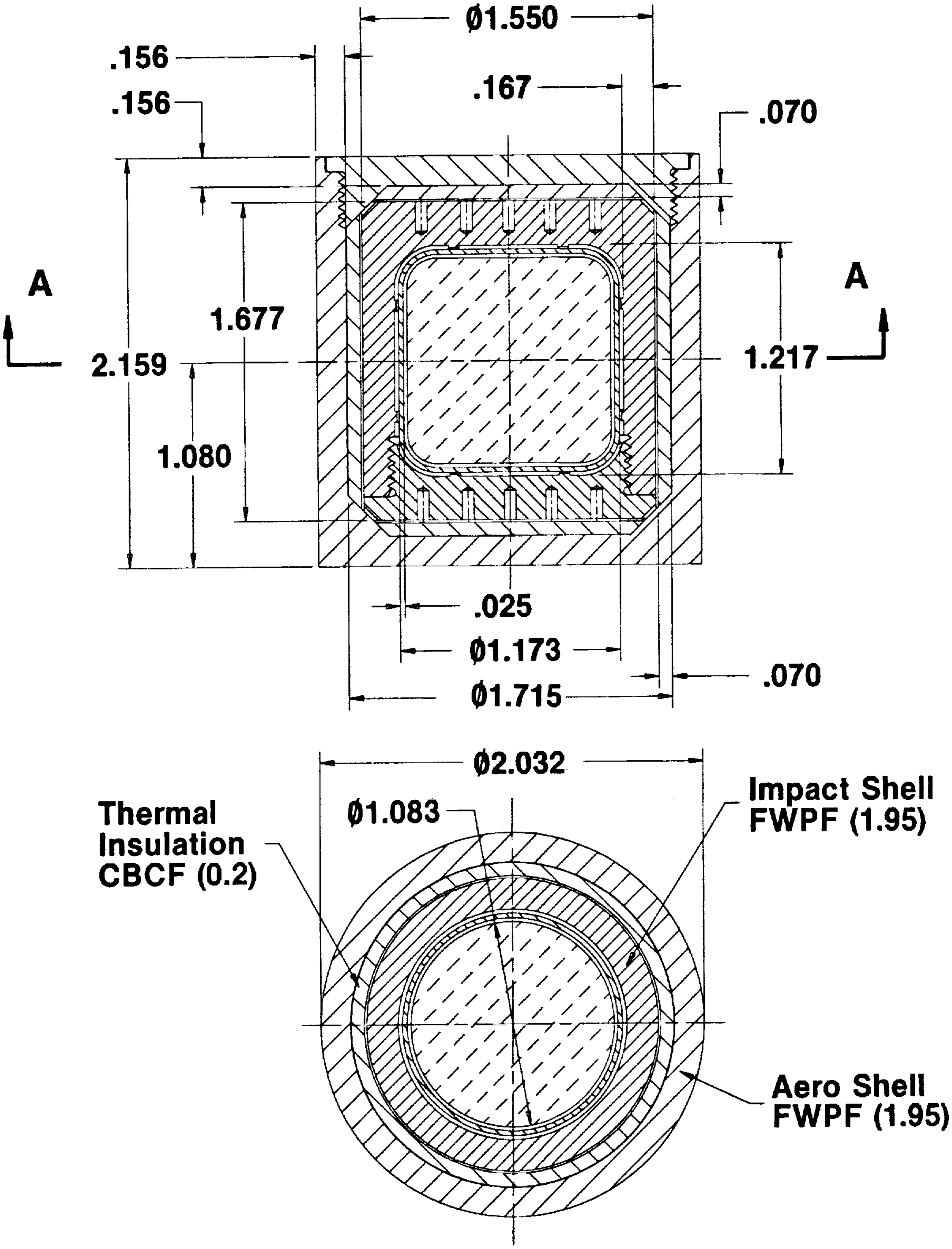
Alfred Schock
Fairchild Space Company

RADIOISOTOPE HEAT SOURCE



*Fine-Weave Pierced Fabric)
**Carbon-Bonded Carbon Fibers

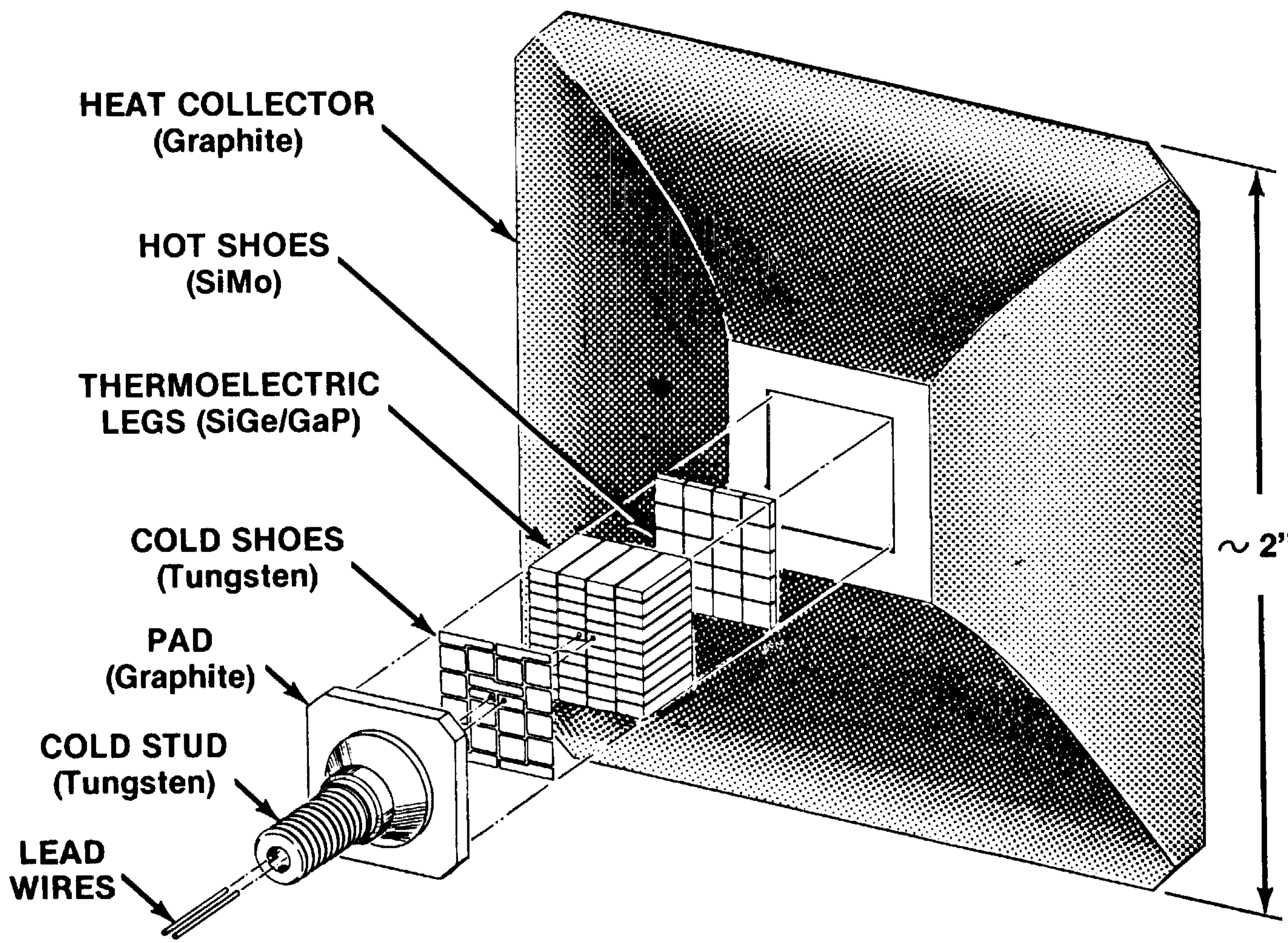
HEAT SOURCE CROSS-SECTIONS



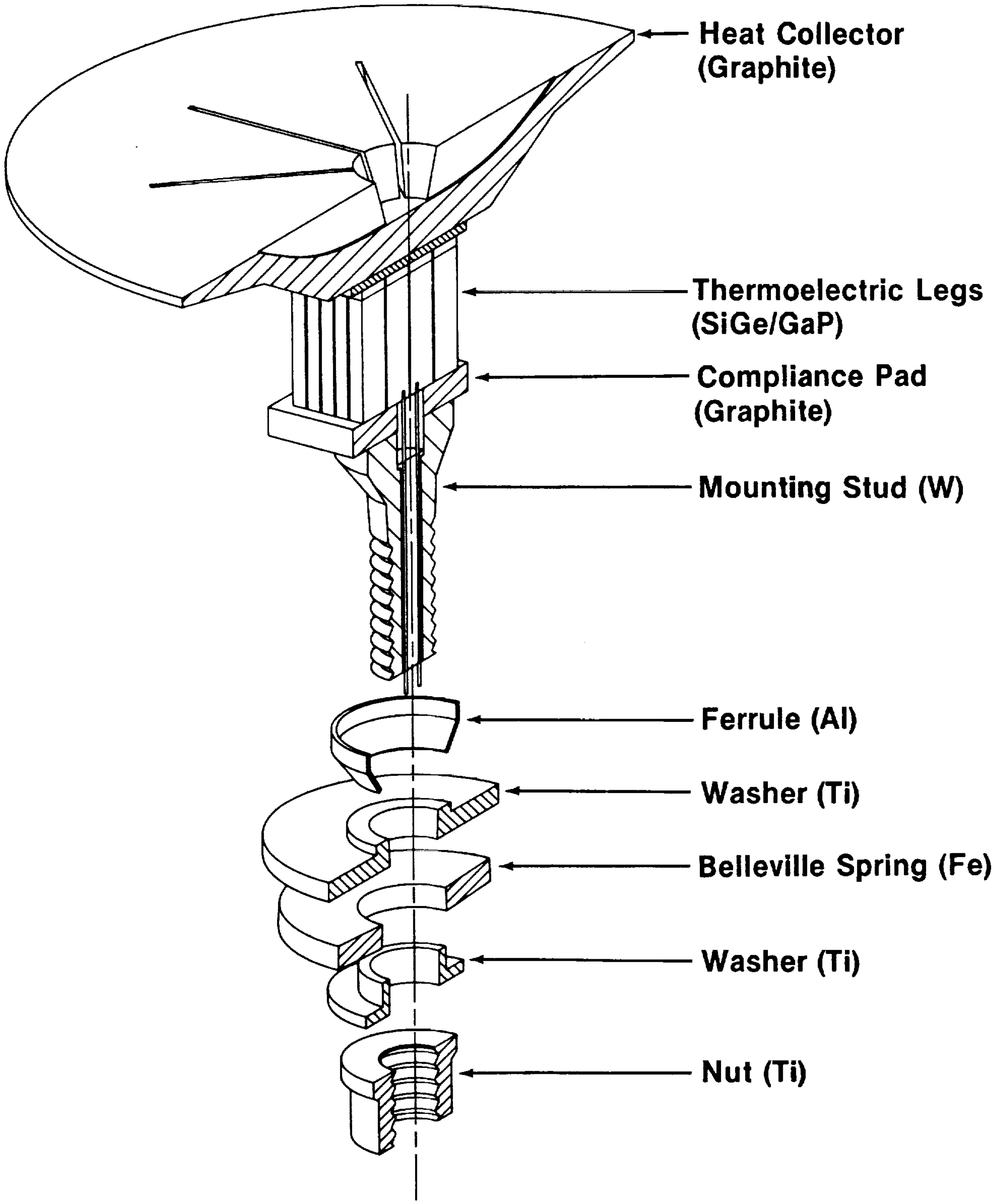
SECTION A - A

Mass - 1.346 LB = 0.611 kg

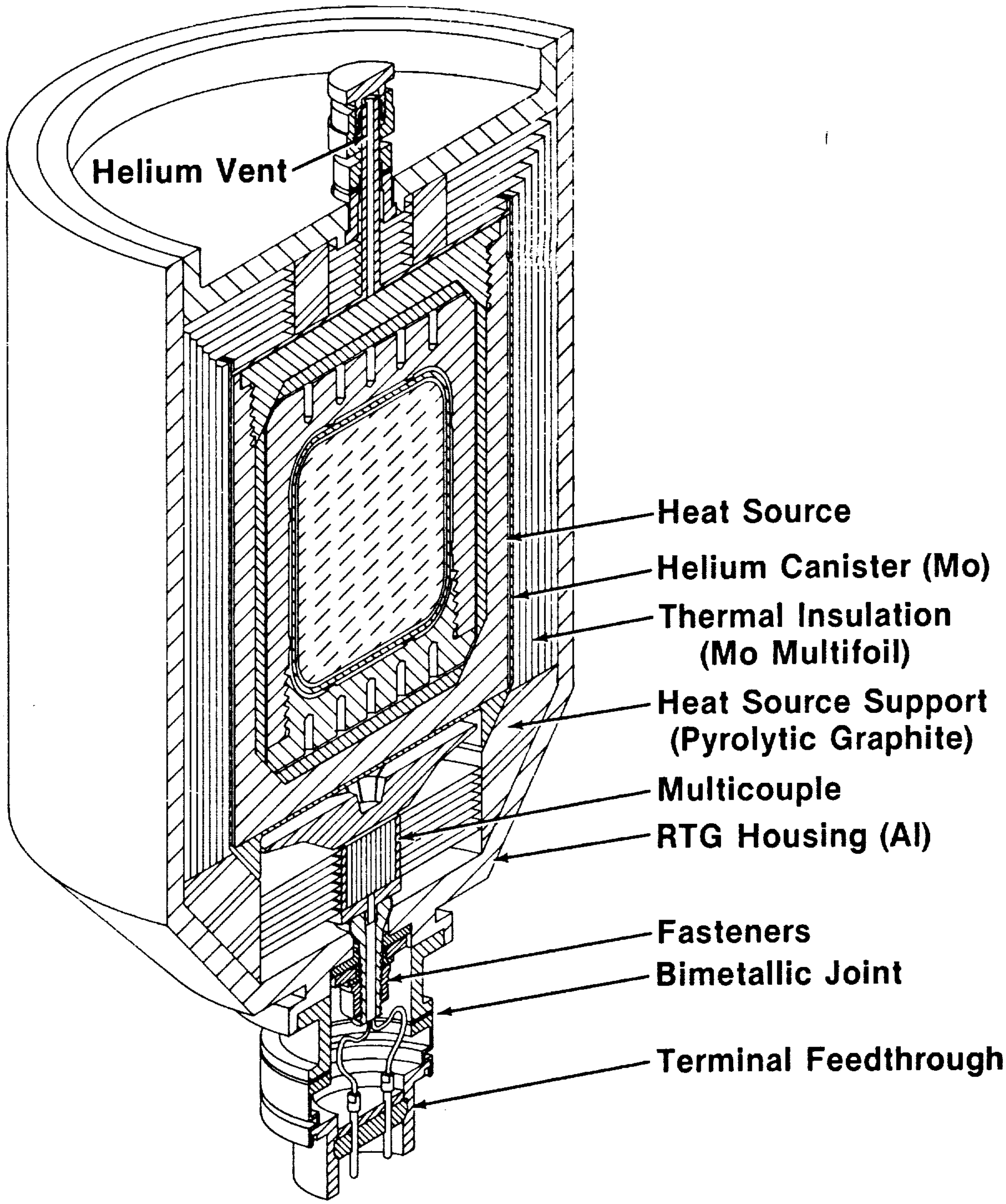
EXPLODED VIEW OF MULTICOUPLE (2.6 Watt, 3.5 Volt)



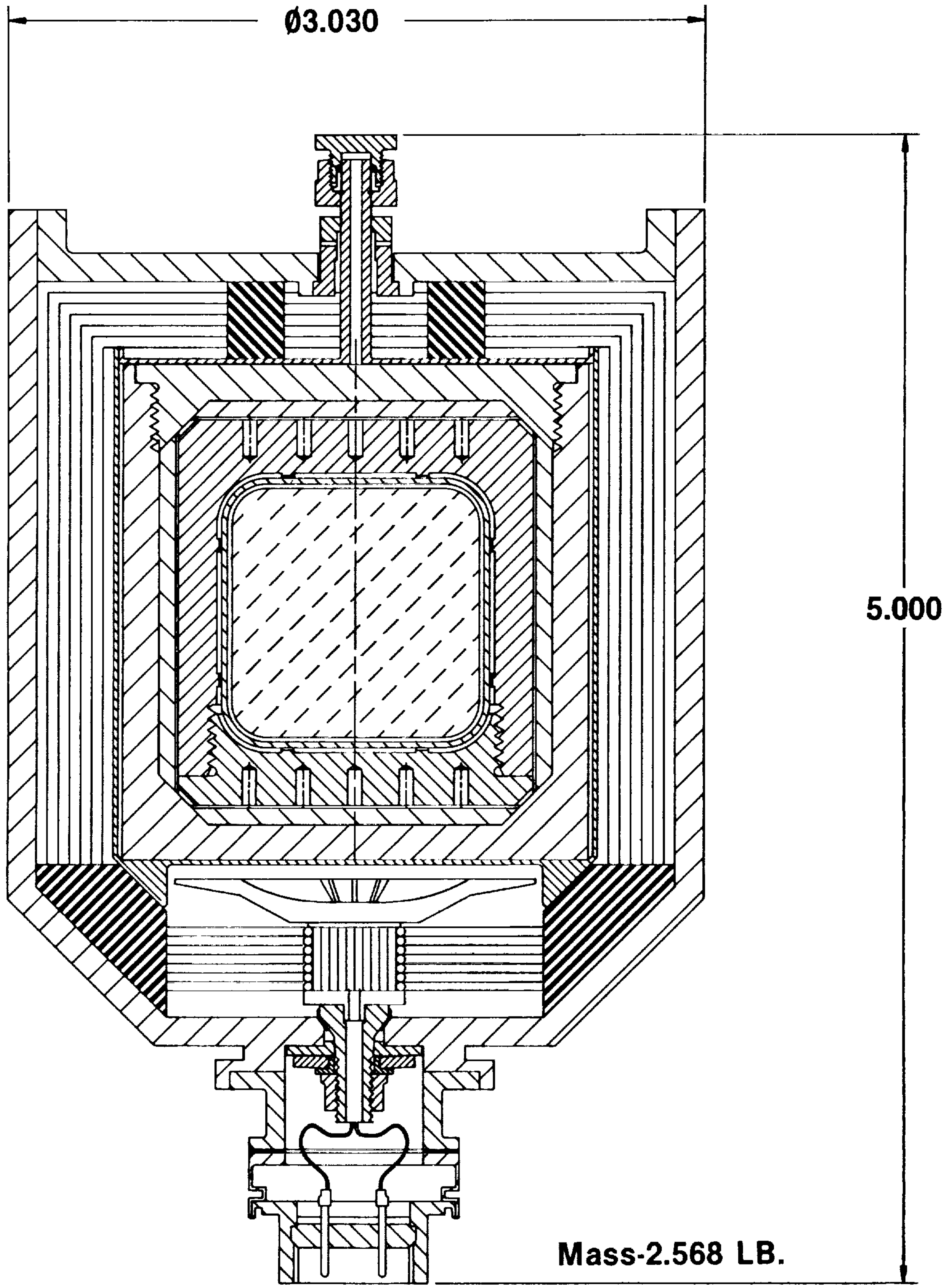
MULTICOUPLE AND FASTENERS



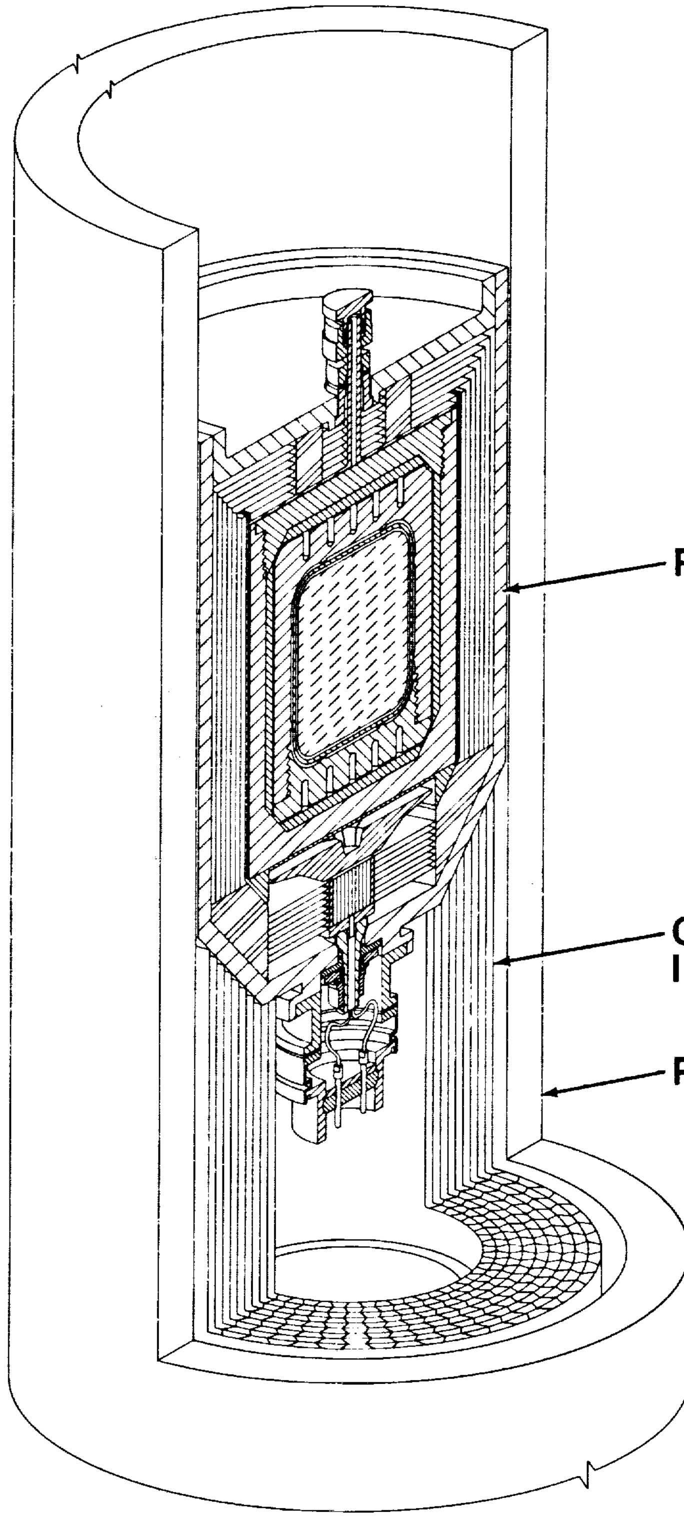
RADIOISOTOPE THERMOELECTRIC GENERATOR (RTG)



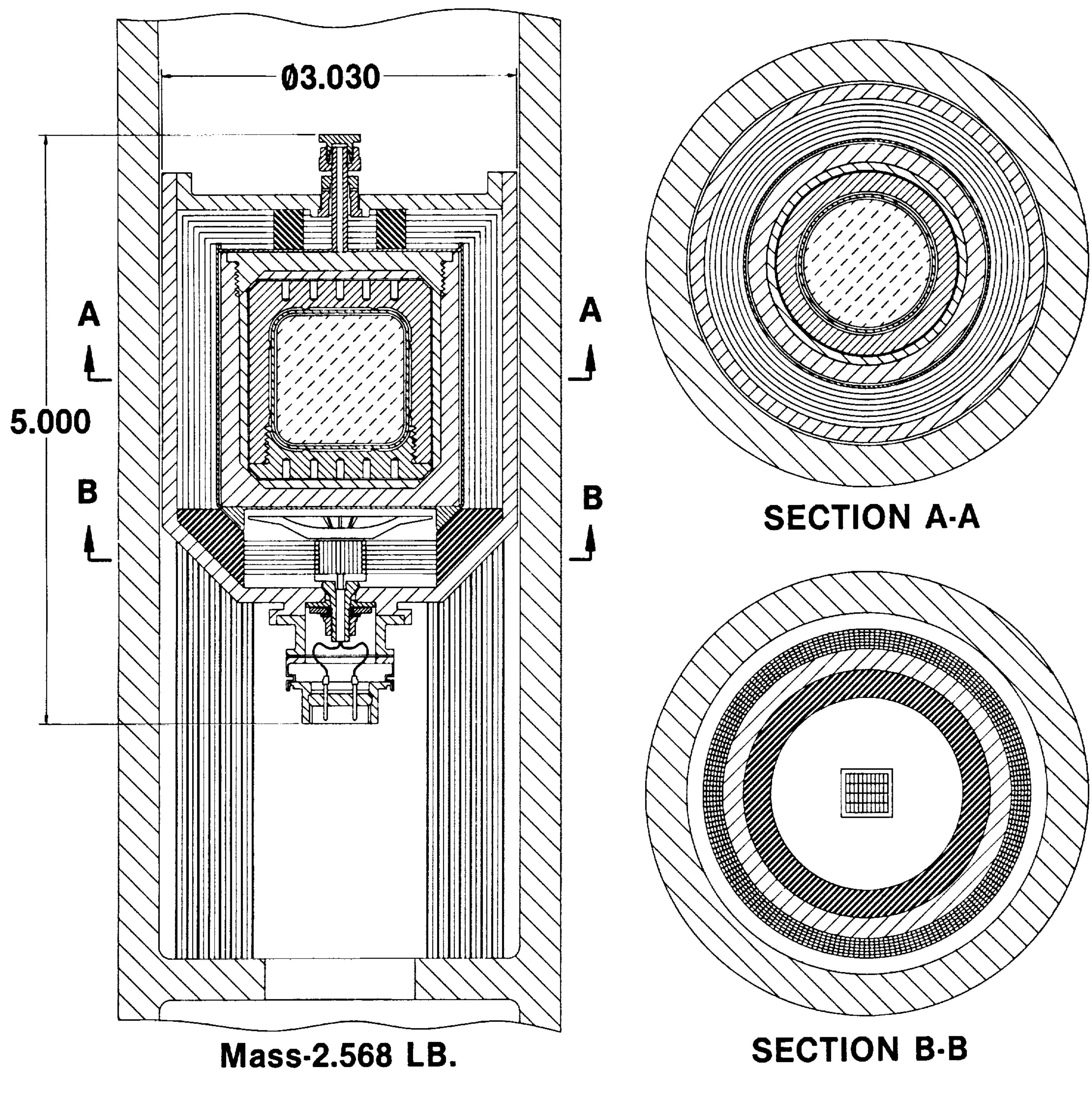
RTG CROSS-SECTIONS



RTG IN PENETRATOR



RTG IN PENETRATOR CROSS-SECTIONS



Mass-2.568 LB.

SECTION A-A

SECTION B-B

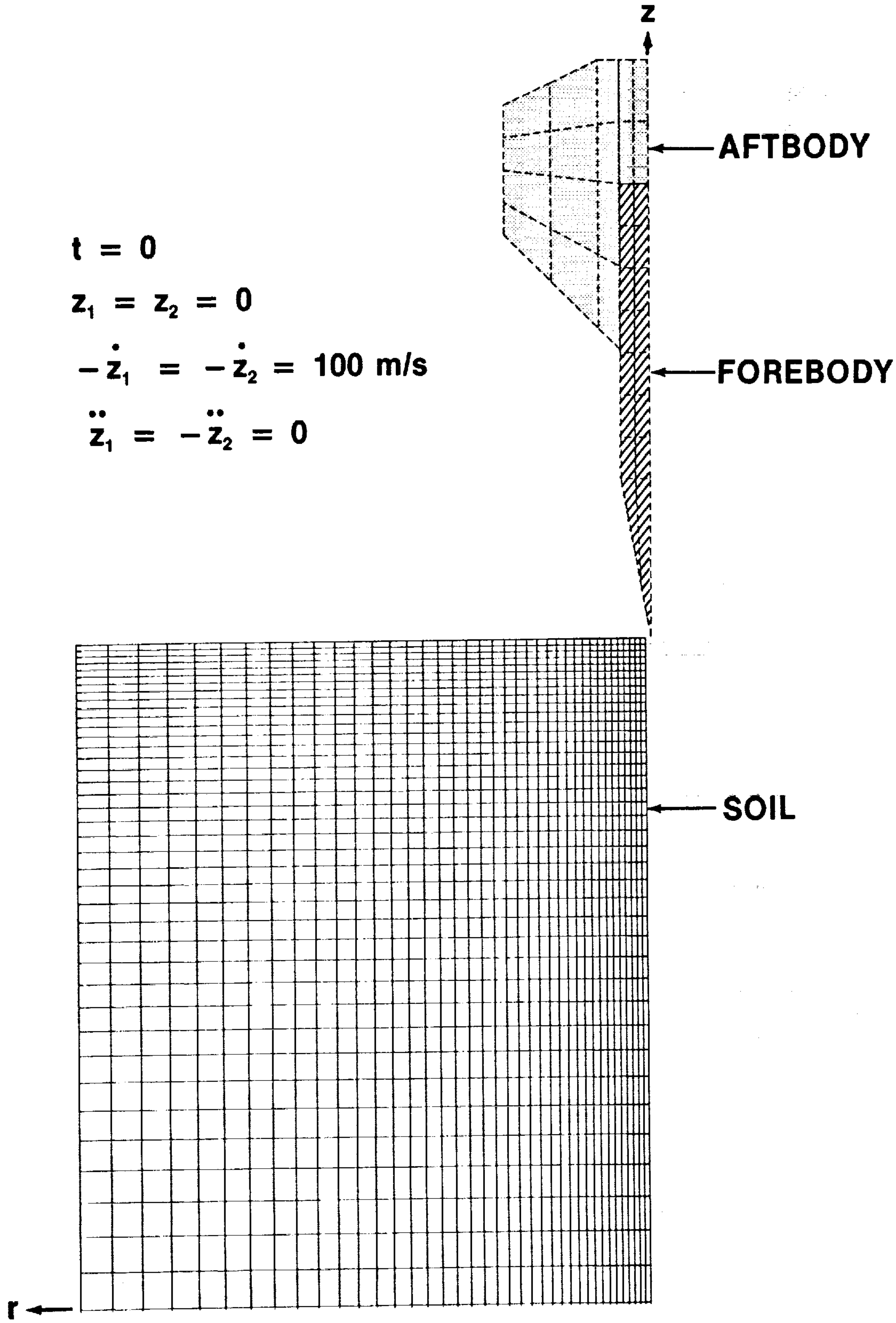
PENETRATOR DECELERATION, PAGE 1

$t = 0$

$z_1 = z_2 = 0$

$-\dot{z}_1 = -\dot{z}_2 = 100$ m/s

$\ddot{z}_1 = -\ddot{z}_2 = 0$



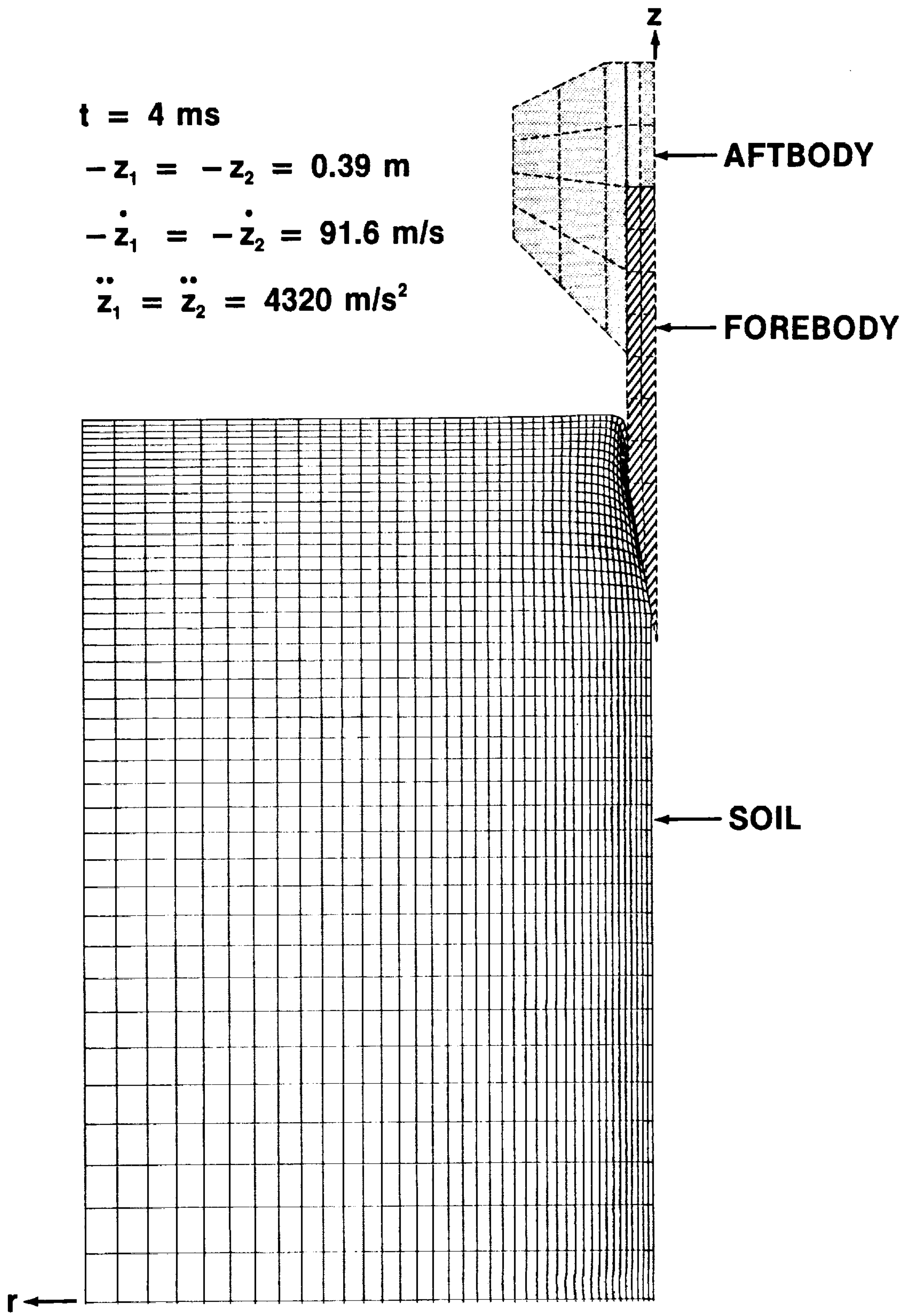
PENETRATOR DECELERATION, PAGE 2

$t = 4$ ms

$-z_1 = -z_2 = 0.39$ m

$-\dot{z}_1 = -\dot{z}_2 = 91.6$ m/s

$\ddot{z}_1 = \ddot{z}_2 = 4320$ m/s^2



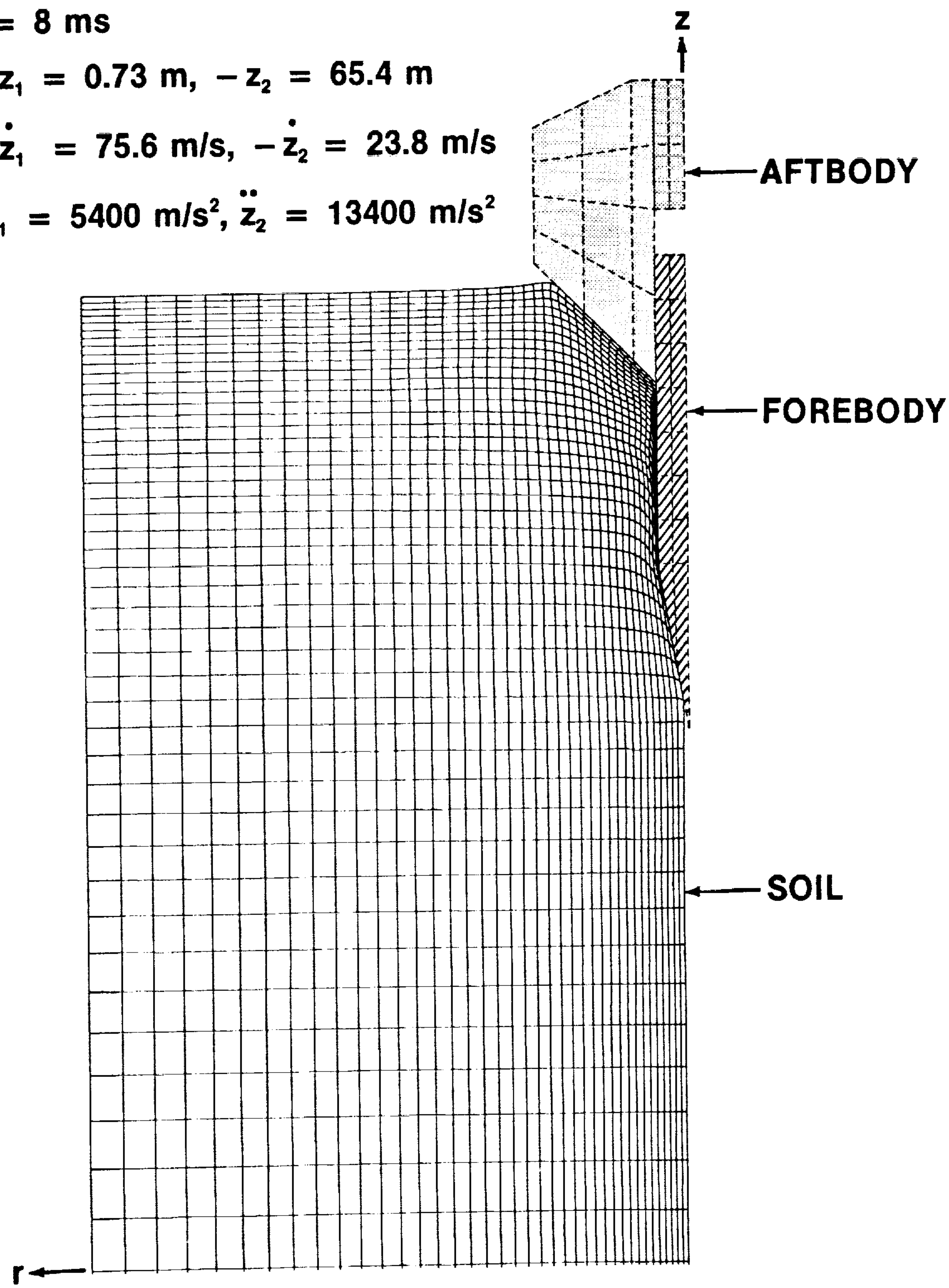
PENETRATOR DECELERATION, PAGE 3

$t = 8$ ms

$-z_1 = 0.73$ m, $-z_2 = 65.4$ m

$-\dot{z}_1 = 75.6$ m/s, $-\dot{z}_2 = 23.8$ m/s

$\ddot{z}_1 = 5400$ m/s^2, $\ddot{z}_2 = 13400$ m/s^2



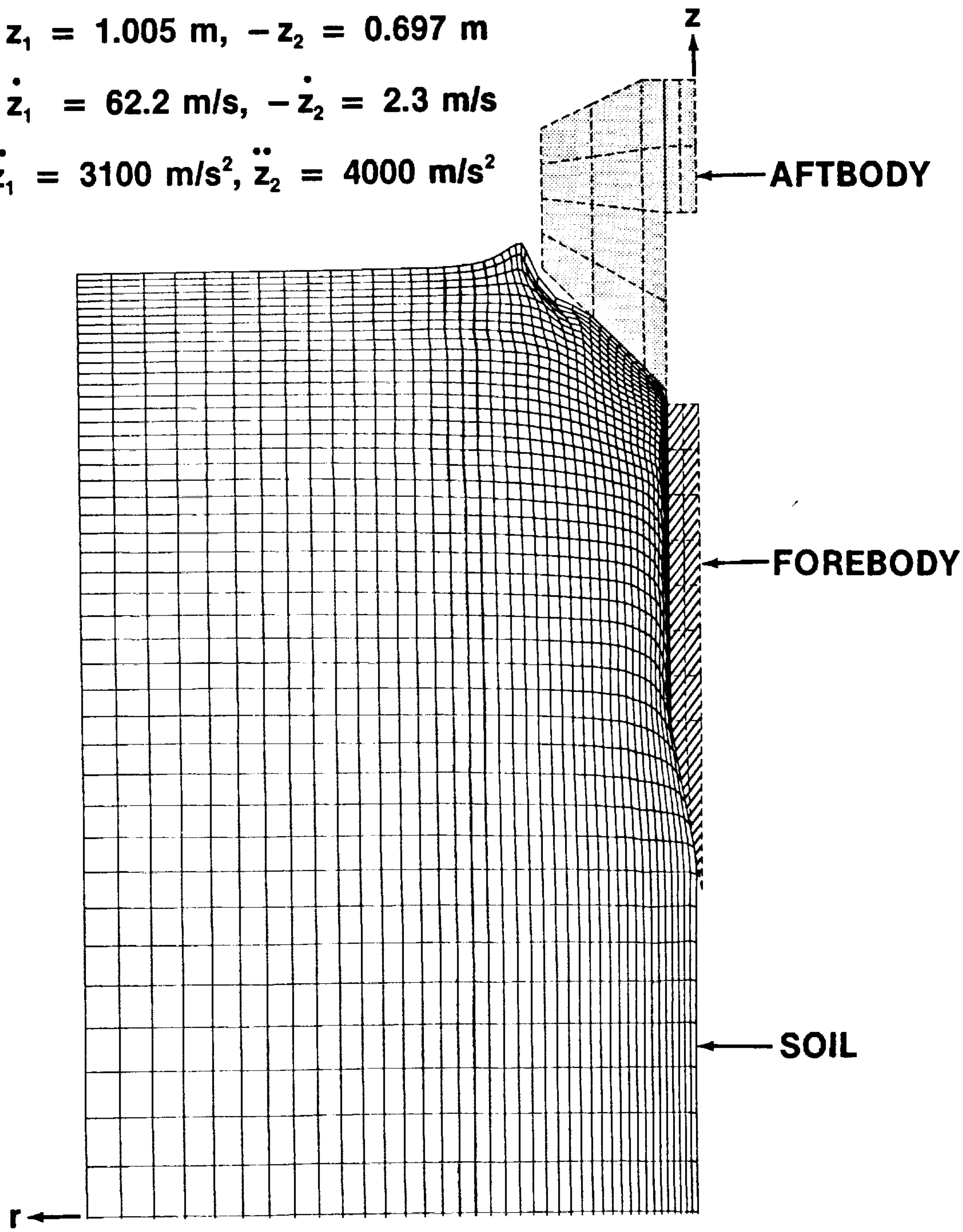
PENETRATOR DECELERATION, PAGE 4

$t = 12$ ms

$-z_1 = 1.005$ m, $-z_2 = 0.697$ m

$-\dot{z}_1 = 62.2$ m/s, $-\dot{z}_2 = 2.3$ m/s

$\ddot{z}_1 = 3100$ m/s^2, $\ddot{z}_2 = 4000$ m/s^2



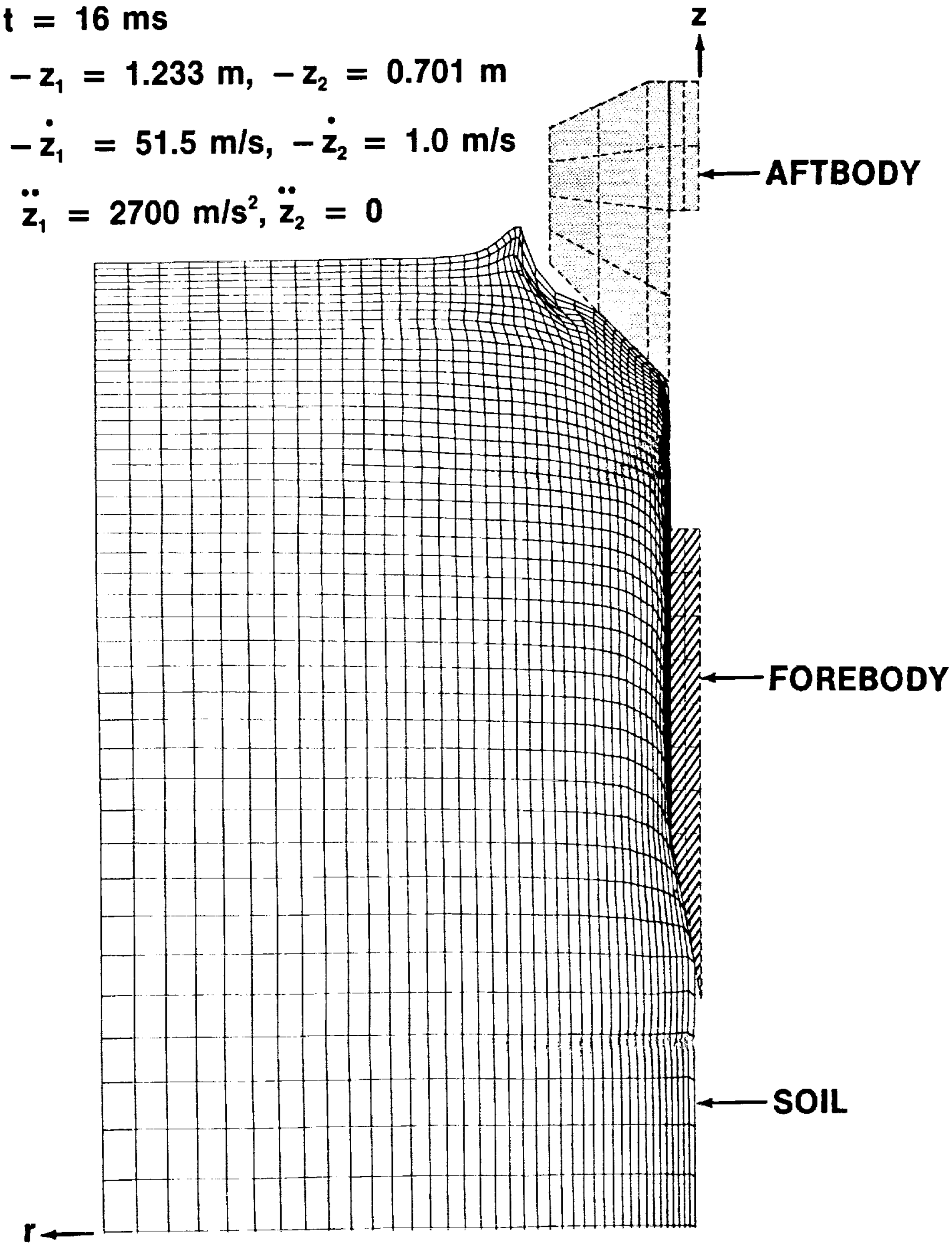
PENETRATOR DECELERATION, PAGE 5

$t = 16$ ms

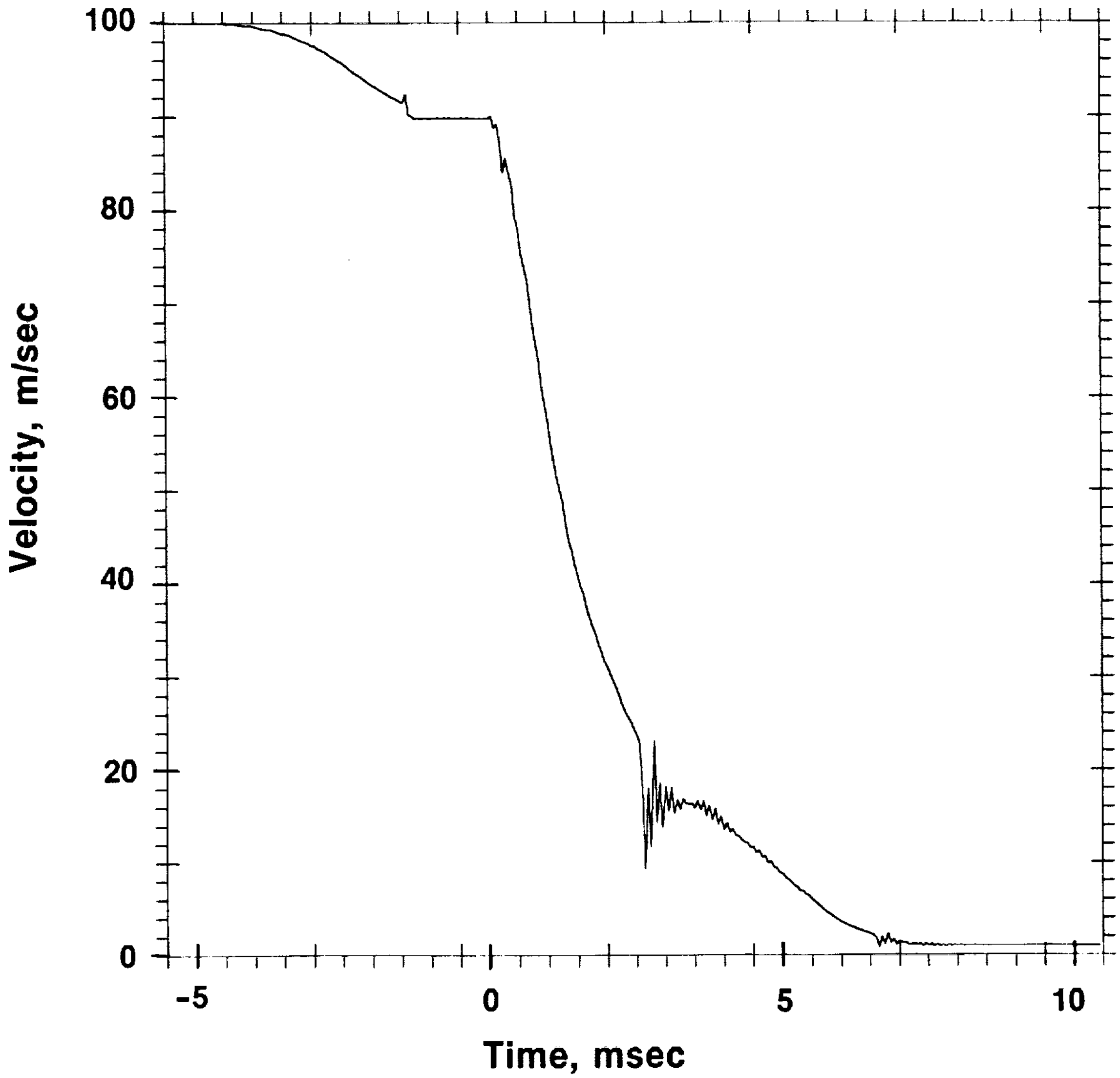
$-z_1 = 1.233$ m, $-z_2 = 0.701$ m

$-\dot{z}_1 = 51.5$ m/s, $-\dot{z}_2 = 1.0$ m/s

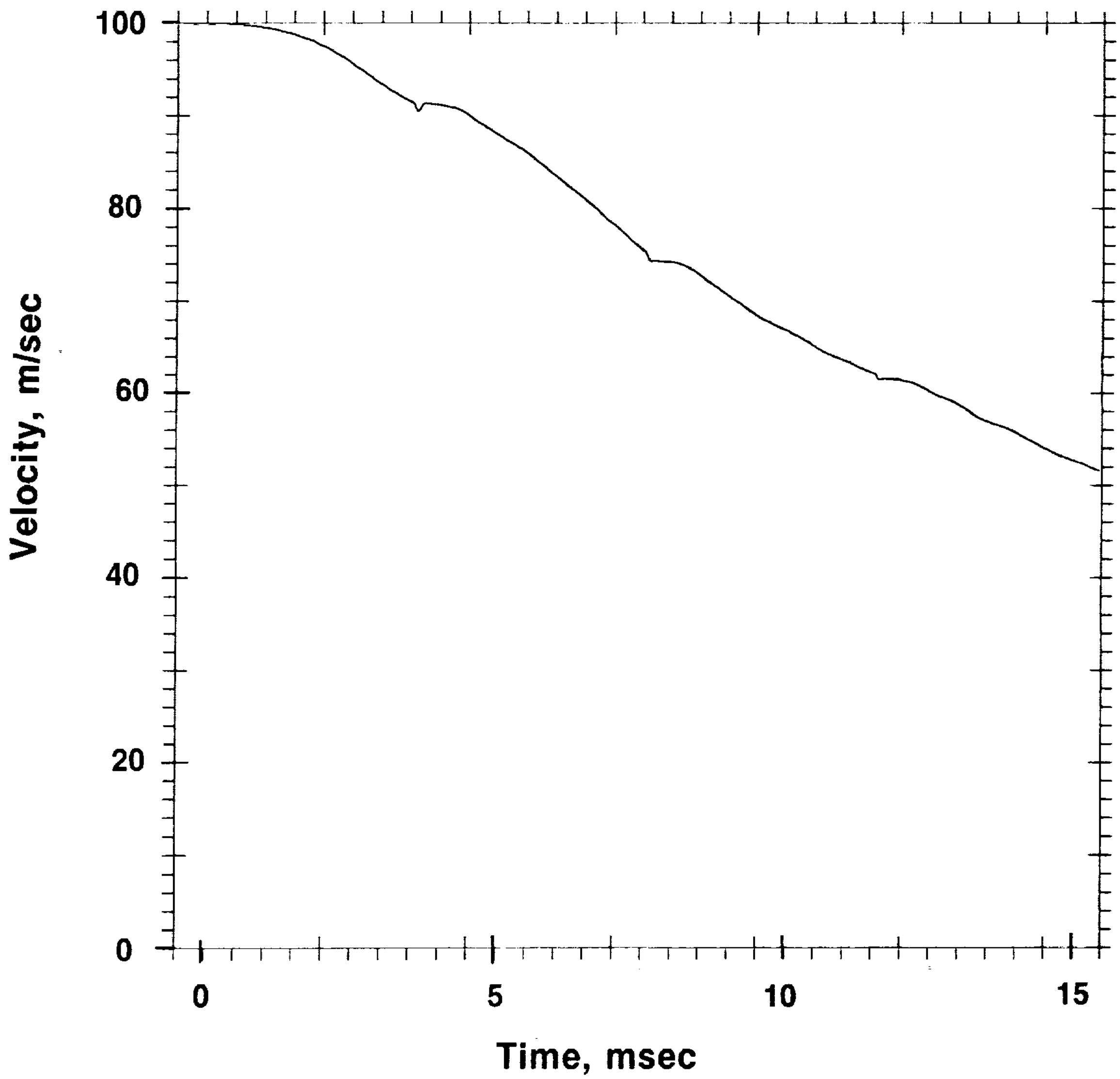
$\ddot{z}_1 = 2700$ m/s^2, $\ddot{z}_2 = 0$



DECELERATION OF AFT BODY, VELOCITY VERSUS TIME AFTER IMPACT



DECELERATION OF FOREBODY, VELOCITY VERSUS TIME AFTER IMPACT



DECELERATION OF FOREBODY, VELOCITY VERSUS PENETRATION